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AERODYNAMIC DESIGN AND ANALYSIS SYSTEM FOR SUPERSONIC AIRCRAFT

Part 2 - User's Manual

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Prepared by
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16. Abstract

An integrated system of computer programs has been developed for the design and analysis of supersonic configurations. The system uses linearized theory methods for the calculation of surface pressures and supersonic area rule concepts in combination with linearized theory for calculation of aerodynamic force coefficients. Interactive graphics are optional at the user's request.

The description of the design and analysis system is broken into three parts:

Part 1--General Description and Theoretical Development
Part 2--User's Manual
Part 3--Computer Program Description

This part contains a description of the system, an explanation of its usage, the input definition, and example output.

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AERODYNAMIC DESIGN AND ANALYSIS SYSTEM FOR SUPERSONIC AIRCRAFT

PART 2 - USER'S MANUAL

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Boeing Commercial Airplane Company

1.0 SUMMARY

An integrated system of computer programs has been developed for the design and analysis of supersonic configurations.

The system consists of an executive driver and seven basic computer programs including a plot module, which are used to build up the force coefficients of a selected configuration. Documentation of the system has been broken into 3 parts:

Part 1 - General Description and Theoretical Development
Part 2 - User's Manual
Part 3 - Computer Program Description

This part, the user's manual, contains a description of the system, an explanation of its usage, the input definition, and example output.

Interactive graphics for use with the system are optional, employing the NASA-LRC CRT display and associated software. A description of the interactive graphics portion of the system is given in Appendix A.

The computer program is written in FORTRAN IV for a SCOPE 3.0 or KRONOS 2.0 operating system and library file. It is designed for the CDC 6000 series of computers and executes in OVERLAY mode. The system requires approximately 110000_8 (octal) central memory words and uses seven peripheral disc files in addition to the input and output files.

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2.0 INTRODUCTION

A series of individual computer programs for design or analysis of supersonic configurations has been linked together into a single system. The system, because of built-in communication between the programs, is substantially simpler to input and use than the individual programs operating in a stand-alone mode. In addition, a common geometry format, based on the NASA-LRC configuration plotting program, has been adopted to standardize the input requirements of the basic programs.

Interactive graphics have been included in the system, to display or edit input and to permit monitoring and read-out of program results. The graphics arrangement is tailored specifically to the NASA-LRC CDC 250 cathode ray tube and associated software. However, all graphics applications have been subroutined to the main programs and could be easily converted to a different graphics set-up.

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3.0 DISCUSSION

A schematic of the design and analysis system is shown in figure 3.0-1. The system consists of an executive "driver" and seven basic computer programs including a plot program and a geometry input module, which are used to build up the force coefficients of a selected configuration as shown in figure 3.0-2. The system may be used with or without interactive graphics.

The complete design and analysis system is a single overlaid computer program, with the executive driver as the main overlay and the basic programs as primary overlays. The basic programs manipulate input (geometry module), draw a picture of the configuration (plot module), or perform design or analysis calculations.

Aerodynamic force coefficients for a selected configuration are built up through superposition. The individual modules of the system provide data for the force coefficient build-up as follows:

- Skin friction is computed using flat plate turbulent theory.
- Wave drag is calculated from either near-field (surface pressure integration) or far-field (supersonic area rule) methods. The near-field method is used primarily as an analysis tool, where detailed pressure distributions are of interest. The far-field method is used for wave drag coefficient calculations and for fuselage optimization according to area rule concepts.
- Drag-due-to-lift is computed from the lift analysis program, which breaks arbitrary wing/fuselage/canard/nacelles/horizontal tail configurations into a mosaic of "Mach-box" rectilinear elements which are employed in linear theory solutions. A complementary wing design and optimization program, also using the Mach-box approach, solves for the wing shape required to support an optimized pressure distribution at a specified flight condition.

3.1 System Communications

Communication between the executive and the different basic modules is performed by disc files and limited common block storage.

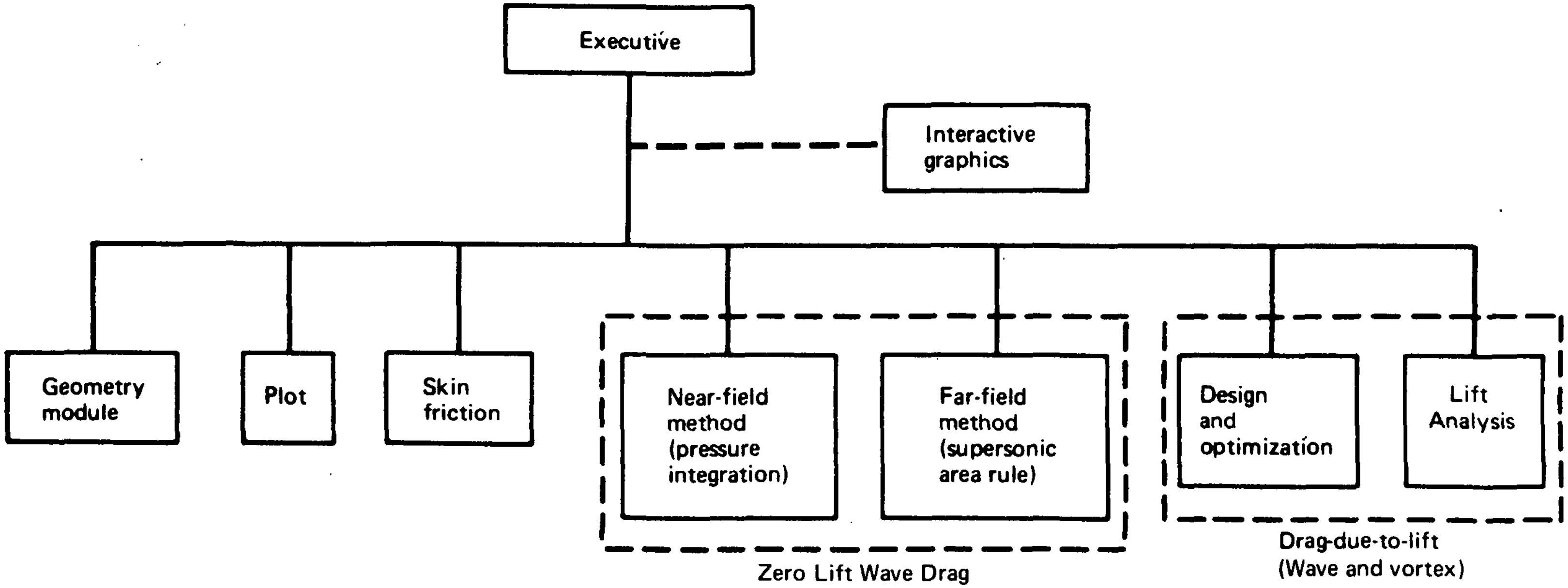


FIGURE 3.0-1.—INTEGRATED SUPERSONIC DESIGN AND ANALYSIS SYSTEM

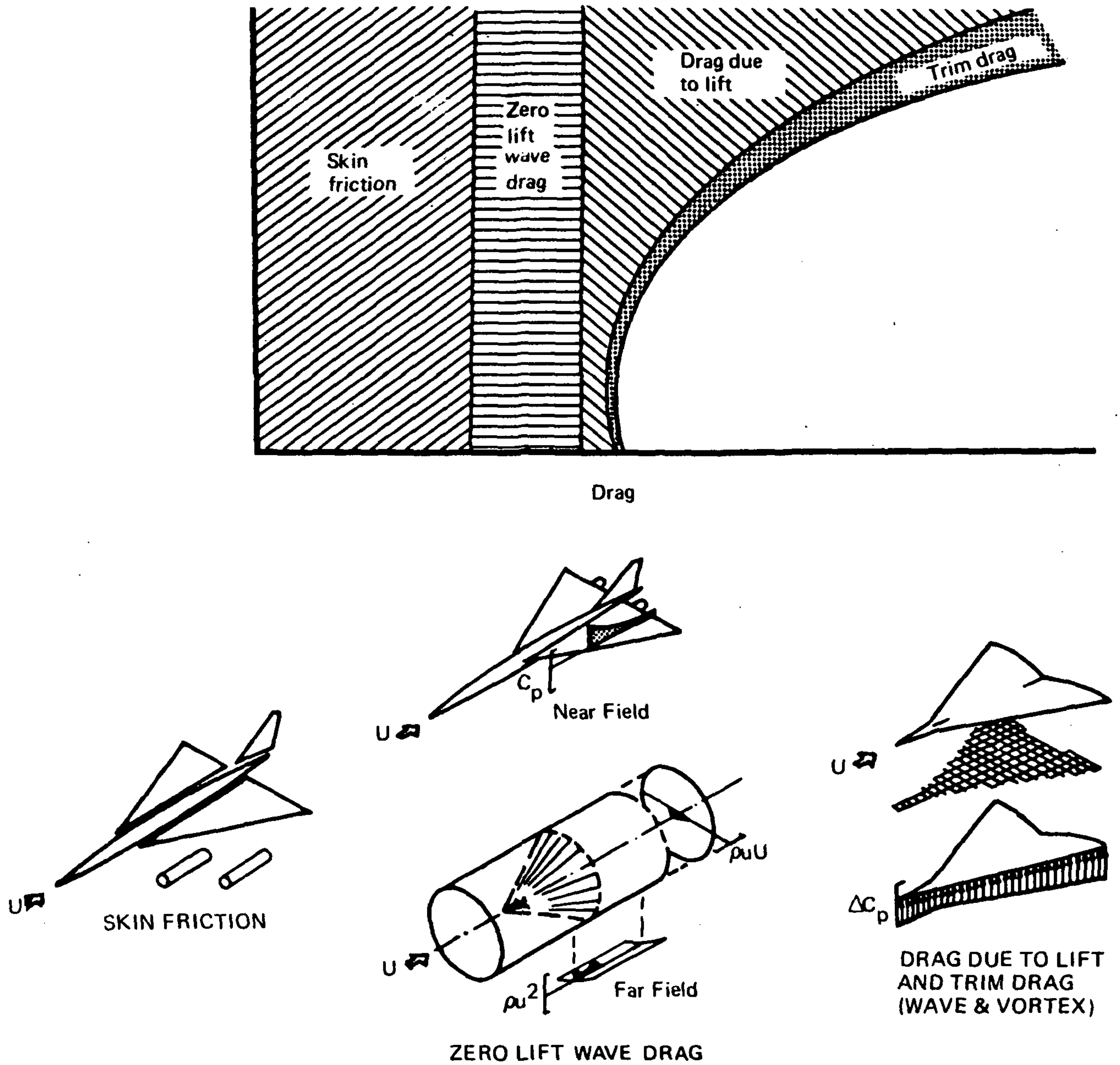


FIGURE 3.0-2.—DRAG BUILDUP

1) Input

All input to the basic modules is handled through the common geometry module and its associated interfaces. A fundamental consideration in the setup of the system has been that input to the basic modules would not be changed by their incorporation into the overall system. However, to minimize and simplify system input requirements, a special geometry module has been created to read all input, and then sort and structure the input needs of the basic programs.

2) Program Sequencing

Program execution is ordered by means of special identification cards, read in the executive, which initiate a specific operation; for instance:

GEØM
This card instructs the executive to have the geometry module read configuration geometry.

PLØT
This card orders a plot of the configuration to be drawn, according to size and view requirements which will be supplied.

SKFR
Compute skin friction for the configuration.

Other similar cards control the other basic modules. The configuration that is to be plotted, or analyzed, need not be the complete configuration that has been input. Also, the geometry definition may be updated without complete replacement of the geometry input.

A summary of the executive control cards is given in Section 4.

For each basic program, there are some inputs that are not geometry. (e.g., Mach number, number of longitudinal cuts in analysis, etc.) These inputs are given immediately after the program calling card and are read in the proper interface routine in the geometry module.

3) Program Answers

A limited amount of common storage between the different programs is used to preserve answers and transfer data between modules. The lift analysis module is the largest single program in the system. Therefore, some common blocks used in the lift analysis program are carried also in the executive level without increasing total system size. These data blocks include:

- Wing camber surface definition
- Wing thickness pressures
- Fuselage upwash bouyancy pressures
- Nacelle pressure field
- Asymmetric fuselage buoyancy field (non mid-wing configurations)

Another data block transfers the optimized fuselage area distribution, based on wave drag considerations, to the geometry module for updating.

3.2 Geometry Module

The function of the geometry module is to read system geometry input, update it if required, and arrange it as needed for the individual programs of the system. A schematic of the geometry module is shown in figure 3.2-1.

The geometry module is accessed by the executive control cards GEØM NEW (input new configuration) or GEOM (addition or replacement of components). The geometry module is also called to update the fuselage or wing camber surface definitions if the executive cards FSUP or WGUP are read.

In addition, the geometry module is called by the executive as an intermediate step in the execution of any of the basic programs. This requires the proper interface routine to be entered, the system geometry to be put into the correct form for the program to be executed, and any special (non-geometric) data required to be read. This is all stacked in the proper order, whereupon the executive then calls the basic program.

In order to minimize core storage requirements of the input data, both the basic system geometry and the transferred input (from the geometry module to another program) are stored on tape (or disk). The basic system geometry is preserved on a tape when the geometry module is not in core, and the input "stack" for a given program is written on a tape to be read by the programs when called by the executive. The input tape created by the geometry module thus merely replaces the usual input tape written from cards.

The format of the system geometry input is the same as that of the NASA-LRC plot program (reference 2). Some optional geometry has been added, however. This includes provisions for fuselage perimeters to be input (if needed by the skin friction program), and provisions for wing camber surface input at planform spanwise stations other than those specified for the system geometry. This camber surface definition, called WZØRD, is data in the form

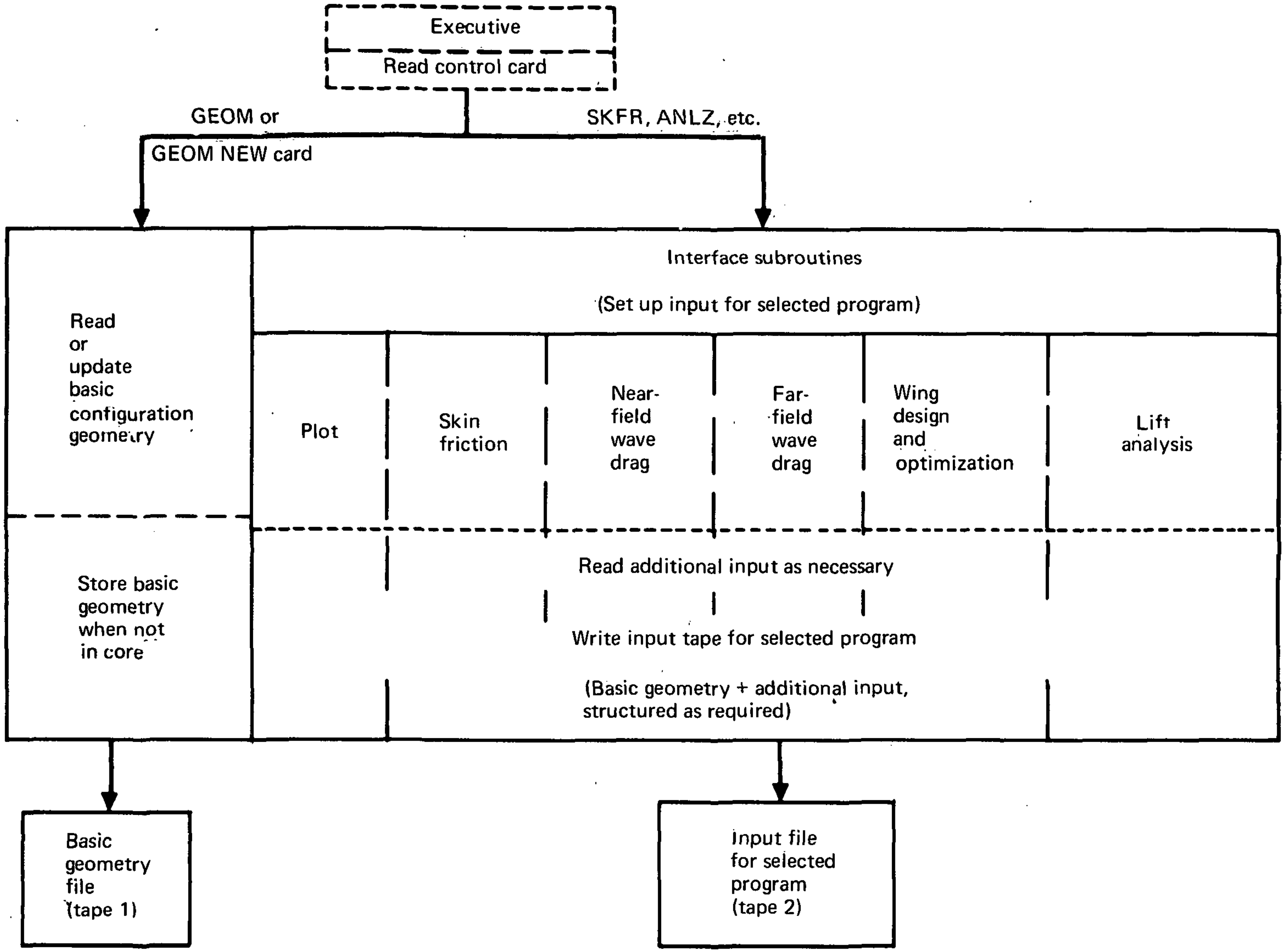


FIGURE 3.2-1.—SCHEMATIC OF GEOMETRY MODULE

normally generated or used by the wing design and analysis programs. Also, nacelles may be located either in the Z coordinate system of the basic geometry, or relative to the local wing surface, whichever is more convenient.

3.3 Plot

The plot module generates the necessary instructions for drawings of the input configuration, either in hard-copy form (Cal Comp) or on the cathode ray tube. Various view options are available. The view option and drawing size are controlled by program inputs.

The plot program was developed at NASA-LRC and has been incorporated into the system with minimum change. Documentation of the program is presented in reference 2.

A typical configuration drawing generated by the plot program is shown in figure 3.3-1.

3.4 Skin Friction

Skin friction drag for a configuration is computed by separating the airplane into its components, then calculating wetted area and the corresponding turbulent skin friction drag for each component. The wing, tail and/or canard (components which may have large variations in chord length) are strip-integrated to obtain an accurate average skin friction coefficient. Skin friction coefficients are computed from the method of reference 1.

Flight conditions for skin friction calculations may be input either as Mach number/altitude, or Reynolds number per foot and total temperature. If the user wishes to input wetted areas for the different components, rather than have the program generate the wetted areas from the system geometry, several special input options are provided.

A schematic of the skin friction program is shown in figure 3.4-1.

3.5 Far-Field Wave Drag Program

This program computes the zero-lift wave drag of an arbitrary configuration by means of the supersonic area rule. The program was originally developed at the Boeing Company, and has been documented (reference 3) and updated by NASA-LRC. The version of the program used in the design and analysis system is that of LRC.

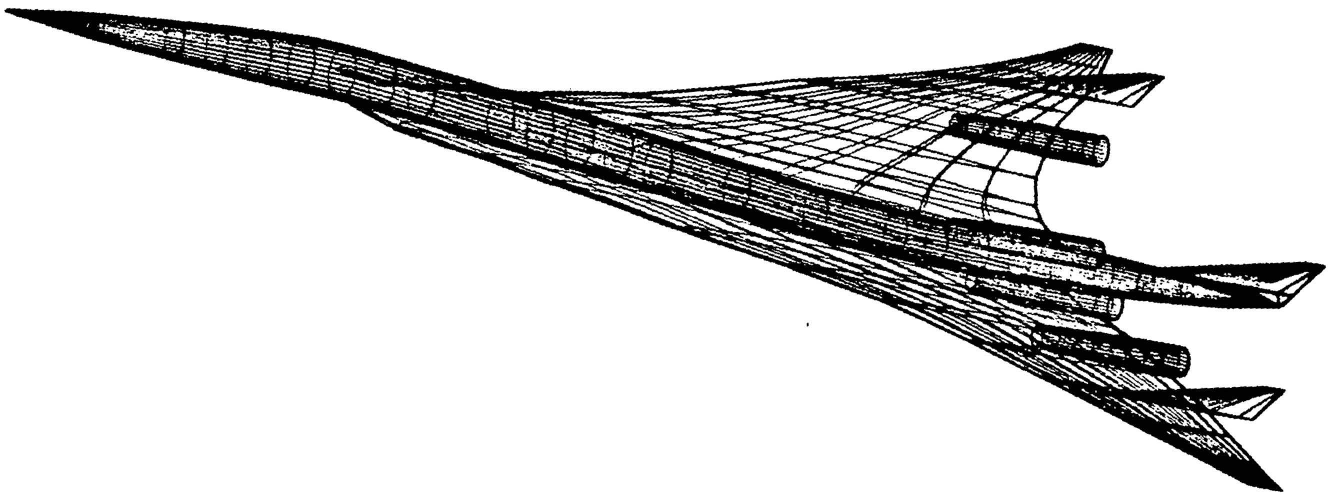


FIGURE 3.3-1.—TYPICAL PLOT PROGRAM DRAWING

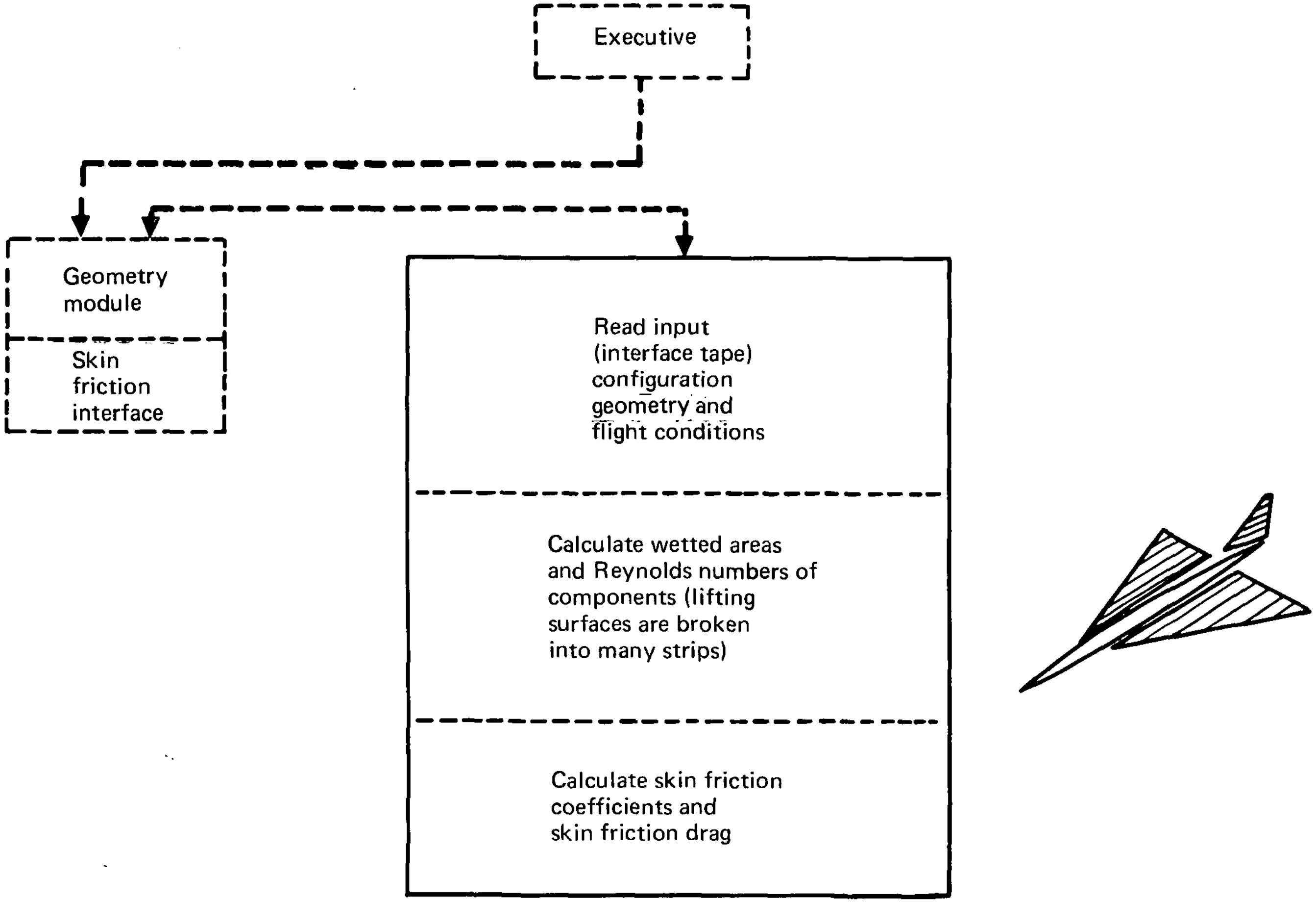


FIGURE 3.4-1.—SCHEMATIC OF SKIN FRICTION PROGRAM

The far-field wave drag program is extremely versatile, and includes a fuselage area optimization feature which is very useful. The fuselage optimization is accomplished by requiring the program to optimize the overall area distribution of wing-nacelles-tail, etc., subject to a few fuselage area control points or "restraints". The program then "fills-in" the non-restrained fuselage area distribution in an optimum fashion for minimum wave drag.

In the design and analysis system, a fuselage area distribution may be optimized by initially defining it in the basic geometry, optimizing the definition in the far-field wave drag program, and then transferring the optimized definition to the geometry module for use in further design or analysis cycles. The actual transfer of the optimized fuselage geometry is performed by use of the executive card FSUP, as described in Section 4.

3.6 Near-Field Wave Drag

The near-field wave drag program computes zero-lift thickness pressure distributions for an arbitrary wing-body-nacelle configuration. The pressure distributions are integrated over the cross-sectional areas of the configuration to obtain the resultant drag force. This force may or may not correspond directly to the drag computed by the far-field method, depending upon the degree of "transparency" specified for the near-field pressure integrations.

By transparency is meant the assumption of the far-field method that pressure fields from all components "pass through" and interact with all other components, regardless of possible physical barriers imposed by in-between components.

Typical pressure data from the near-field program is presented in figure 3.6-1. A wave drag coefficient summary from the program is shown in figure 3.6-2.

The near-field program has three principal uses:

1) As an analysis tool for studying the zero-lift drag forces associated with the interacting pressure fields of different configuration components. In this respect, the near-field program has an advantage over the far-field wave drag method in that there need be no assumption of transparency.

2) As a source of loads data for structural design and analysis.

3) As a source of thickness pressure fields for use in the pressure limiting options of the wing design and lift

$C_{D_{wing}} = 0.00113$ $\quad C_{D_{wing\text{-}on\text{-}body}} = 0.00008$ $\quad \Sigma C_{D_{wing\text{-}body}} = 0.00153$

$C_{D_{body}} = 0.00032$ $\quad C_{D_{body\text{-}on\text{-}wing}} = -.000003$

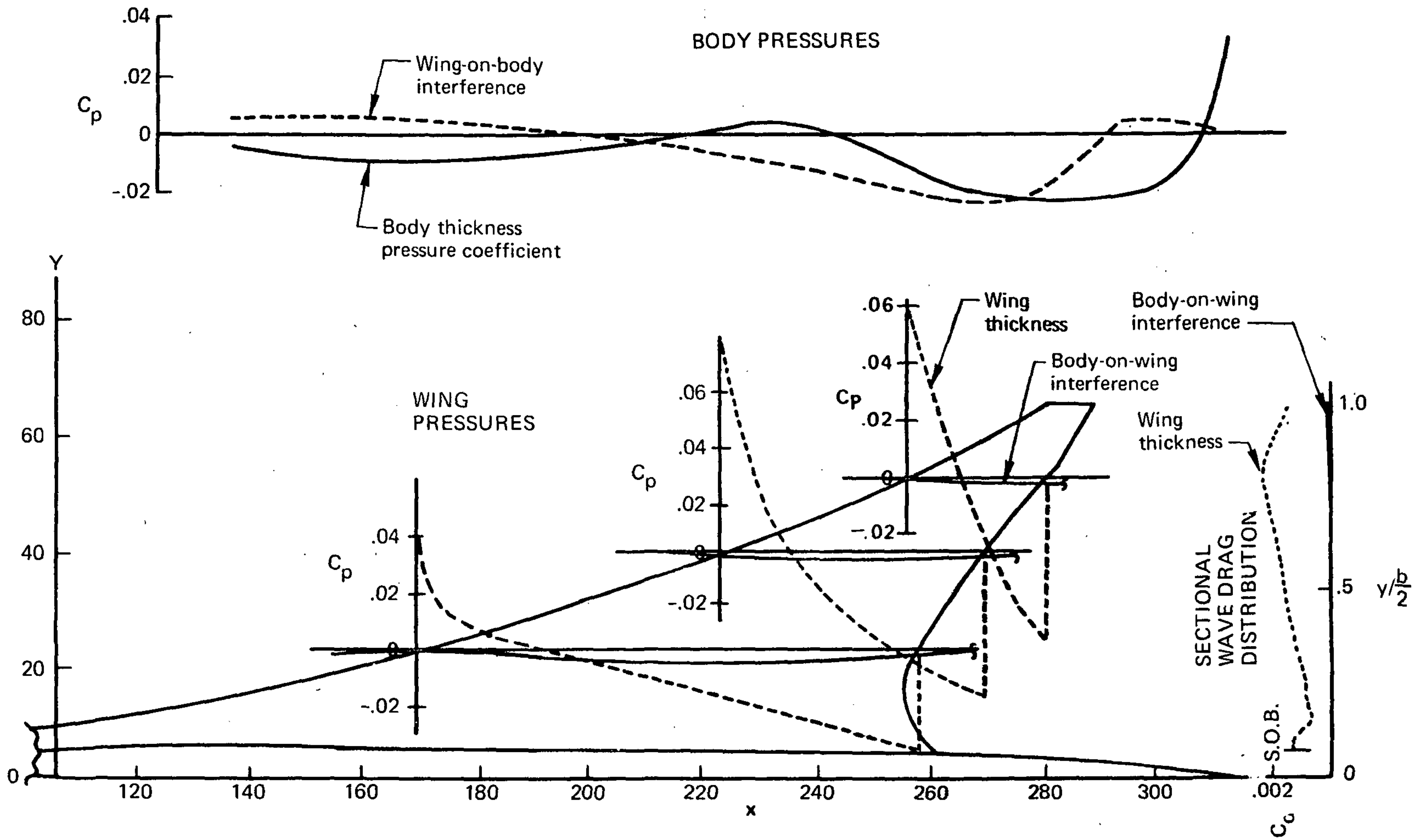
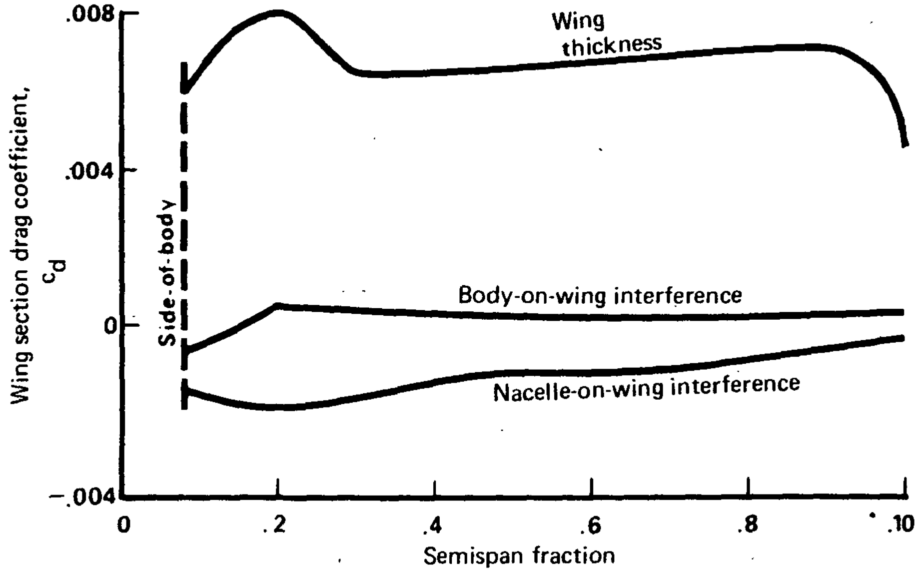


FIGURE 3.6-1.— WING-BODY SOLUTION, M = 2.6



Wing-Body Terms

$C_{D_{wing}} = 0.00639$ $\quad C_{D_{\text{wing-on-body interference}}} = -0.00013$

$C_{D_{body}} = 0.00072$ $\quad C_{D_{\text{body-on-wing interference}}} = 0.00013$

$\Sigma = 0.00711$

Nacelle Terms

	Inboard	Outboard
Isolated $C_{D_{wave}}$	0.00075	0.00075
Body-on-nacelle interference	-0.00002	0.00000
Nacelle-on-body interference	0.00005	0.00010
Nacelle-on-nacelle interference		
Direct	0.00034	0.00023
Image	0.00054	0.00046
Wing-on-nacelle interference	-0.00043	-0.00058
Nacelle-on-wing interference	-0.00156	

$\Sigma C_{D_{nac}} = 0.00064$

Σ Wing-body-nacelle $C_{D_{wave}} = 0.00775$

FIGURE 3.6-2.—TYPICAL WAVE DRAG COEFFICIENT SUMMARY NEAR-FIELD PROGRAM (M = 1.1) analysis programs. (This option is described in section 3.7, but basically requires that the total surface pressure coefficient on the wing, i.e., thickness+lift, cannot be less than some specified fraction of vacuum pressure coefficient.)

If the wing thickness pressures are to be used by the wing design or lift analysis programs in pressure limiting options, then the near-field program must first be run. During program execution, the thickness pressures are loaded into a system common block and are then available where needed.

Nacelle pressure field options. - The near-field program allows for up to 3 pairs of nacelles located external to the wing-fuselage (or 2 pairs plus a single nacelle at Y=0). The nacelles may be either above or below the wing (or both).

The nacelle pressure field is the pressure field imposed on the surface of the wing by the nacelles. A feature of the near-field program is the choice of "wrap" or "glance" solutions for the nacelle pressure field, as shown in figure 3.6-3. (The far-field wave drag program uses essentially the "wrap" solution).

Available experimental data do not make it clear whether a "wrap" or "glance" solution is more correct. Since the nacelle-on-wing interference term is substantial, both solutions are available in the program (controlled by an input code).

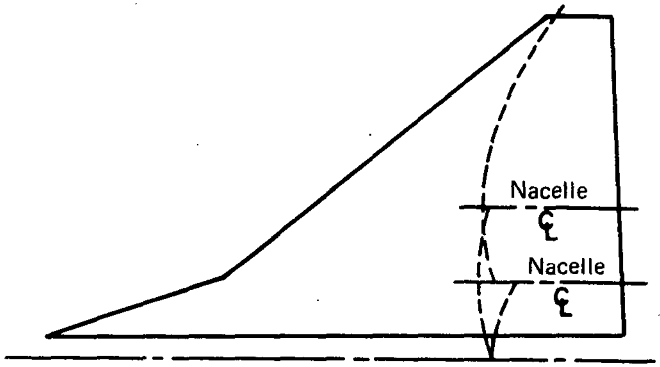
3.7 Wing Design and Lift Analysis

The wing design and lift analysis programs are separate lifting surface methods which solve the direct or inverse problem of:

- Design - to define the wing camber surface shape required to produce a selected lifting pressure distribution. The wing design program includes methods for defining an optimum pressure distribution.

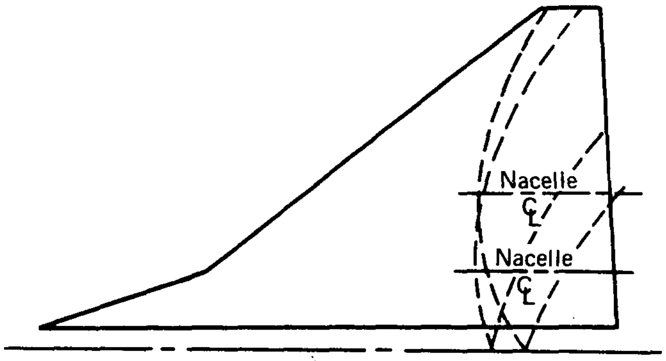
- Lift analysis - to define the lifting pressure distribution acting on a given wing camber surface shape, and calculate the associated force coefficients.

The lift analysis program contains solutions for the effect of fuselage, nacelles, canard and/or horizontal tail, and wing trailing edge flaps or incremental wing twist. Using superposition, the program solves for drag-due-to-lift, lift curve slope, and pitching moment characteristics of a given configuration through a range of angles of attack at a selected Mach number.



PRESSURES "GLANCE" AWAY FROM WING AT ADJACENT NACELLES

The nacelle pressure field and accompanying shock waves "glance" away from the wing when encountering adjacent nacelles. In application, the nacelle generated pressure field is terminated on encountering another nacelle.



PRESSURES "WRAP" AROUND ADJACENT NACELLE

The nacelle pressure fields and accompanying shock waves "wrap" around adjacent nacelles. In application, the nacelle generated pressure field is allowed to pass through another nacelle as if it were transparent.

FIGURE 3.6-3.—NACELLE PRESSURE FIELD CONCEPTS

The wing design program is more limited in scope, since it is used to solve for the wing shape required to support a design pressure distribution at a specified flight condition. The program also contains, however, a number of optional features for identifying the design pressure distribution. This is a demanding solution, because it requires that:

- Drag-due-to-lift of the wing be minimized at a given total lift, subject to an optional pitching moment constraint.
- Constraints be applied to the design pressure distribution to provide physical realism.
- Effects of fuselage upwash, nacelle pressure field, etc., be reflected in the design solution.

Wing Design and Optimization

Given a wing planform and flight condition, the wing design program solves for an optimum (least drag) pressure distribution and the corresponding wing shape, subject to specified constraints on total lift, pitching moment and/or allowable pressure coefficients.

Basically, the method of the wing design program is that of references 4 and 5. For use in the integrated design and analysis system, however, the program has been substantially expanded to provide the following capability:

- Use of any combination (or all) of ten basic lifting pressure loadings, in an optimum fashion.
- Optional imposition of pressure constraints on the wing upper surface, to prevent occurrence of unrealistically low pressure coefficients.
- Optional consideration of three configuration-dependent loadings (fuselage upwash and buoyancy, and nacelle pressure field).
- Optional consideration of three wing camber-induced loadings which are proportional to the three configuration-dependent loadings. This introduces camber-related terms to modulate the configuration related loadings (Example: trailing edge reflex for nacelle buoyancy loading).
- Optional identification of a small planform region (e.g., trailing edge flap) for special incremental loading.

The presentation of the wing design results, for selection of an optimum pressure distribution, is in the form of drag-due-to-lift versus zero-lift pitching moment (C_{mo}). A typical presentation is shown in figure 3.7-1, illustrating the effect of increasing the number of design loadings and adding the nacelle-buoyancy loading. Selecting a drag-due-to-lift, C_L and C_{mo} combination for the wing defines a corresponding pressure distribution which may then be used to generate the associated wing camber surface shape.

Pressure constraints. - The use of a large number of basic wing loadings permits great flexibility in identifying a theoretically optimum lifting pressure distribution. Such an optimum may be physically unrealistic, however. Linear theory contains no limitations on allowable surface pressures, and "optimum" pressure distributions may well involve upper surface pressure coefficients lower than vacuum C_p. To avoid this possibility, a pressure constraint formulation has been added to the solution. This functions by limiting the total upper surface wing pressure coefficient to be equal to or greater than an input C_p.

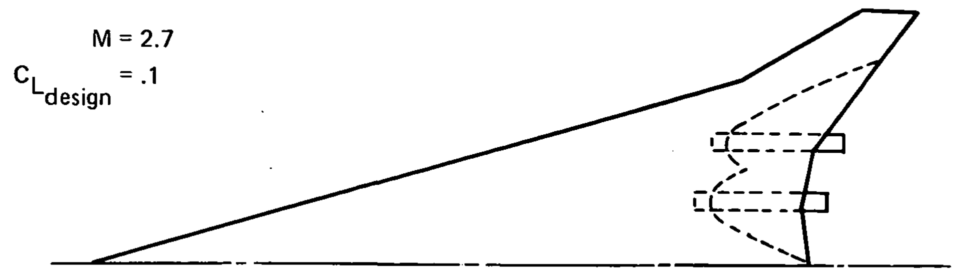
By superposition, the total upper surface pressure coefficient is the sum of wing thickness pressure (from the near-field wave drag program, as noted in Section 3.6), fuselage pressure field, and the upper surface lifting pressure.

The effect of constraining the allowable design pressure distribution to a limit of .7 vacuum is illustrated in figure 3.7-2. For a given planform and set of loadings, the program cycles to find an optimum pressure distribution subject to the pressure limit (with C_{mo} constraint optional). First, an optimum loading combination is found, then the corresponding peak pressure is located. If it violates the pressure limit, a new optimum loading combination is found with a pressure constraint applied at the location of the peak pressure.

This operation is repeated until the wing pressure distribution everywhere satisfies the pressure limit. In the example case shown in figure 3.7-2, the sequence of peak pressure locations is shown, together with the effect of the final constrained solution on drag-due-to-lift.

Loading definitions. - A tabulation of the pressure loadings available within the design program is given in Table I on page 23. The configuration dependent loadings may be used both as an independent effect and also as a definition of a loading which may be varied (by wing camber) in the optimization process.

- As an independent effect, the configuration-dependent loading acts upon the wing in the optimization process, but cannot be varied (loadings 15-17).



Note:

At the design points denoted by circular symbols,

$$C_{p_{\substack{upper\\surface}}} \geqslant 0.7\, C_{p_{vacuum}}$$

Wing thickness pressures included

Two and three loading combinations are the first two and first three loadings in Table 1

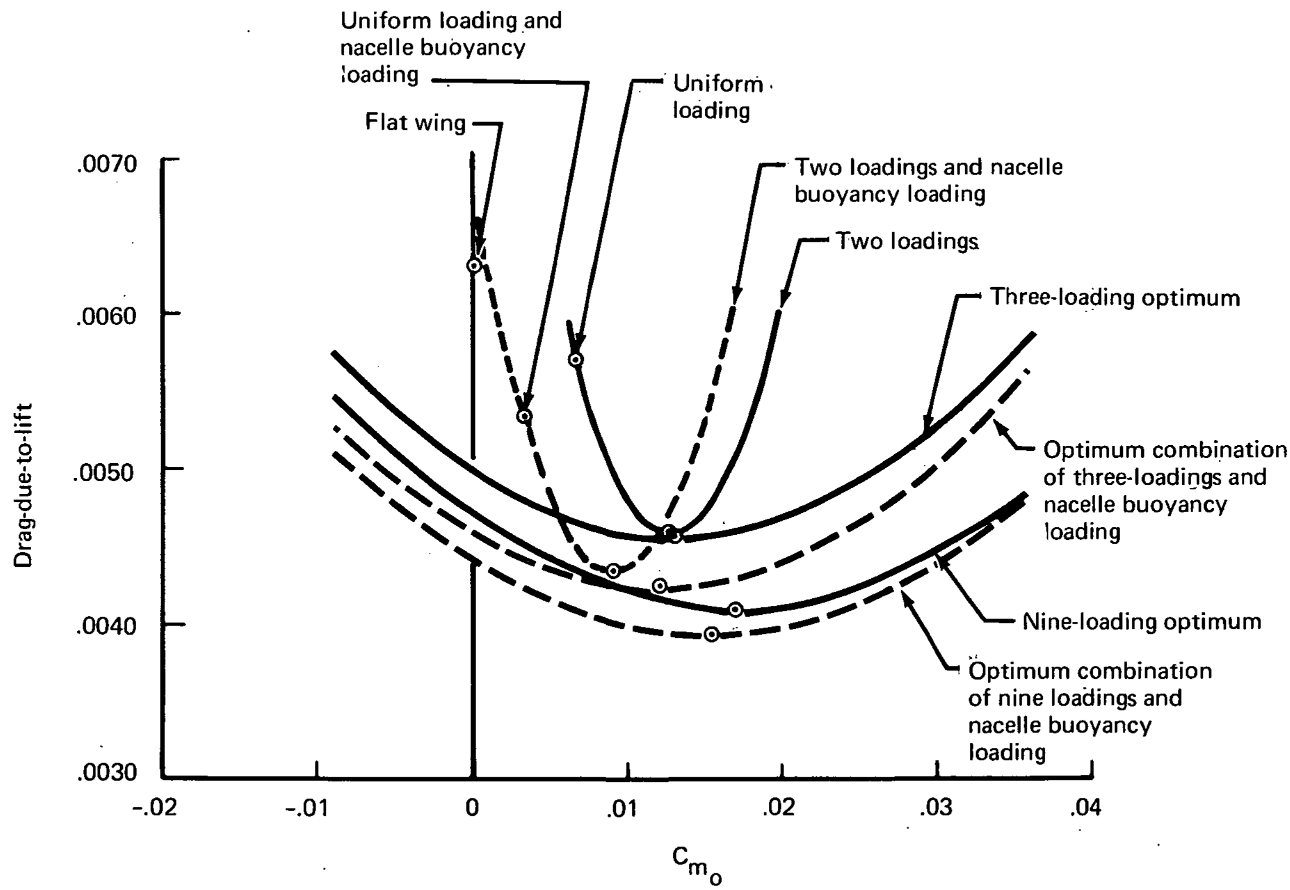
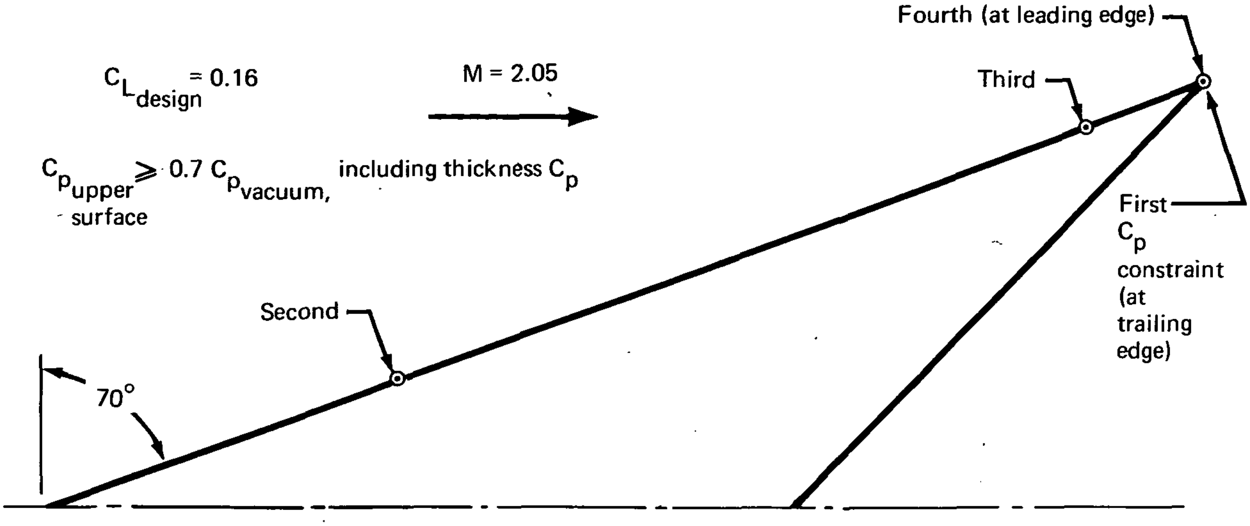


FIGURE 3.7-1.—EFFECT OF NUMBER OF LOADINGS ON WING DESIGN



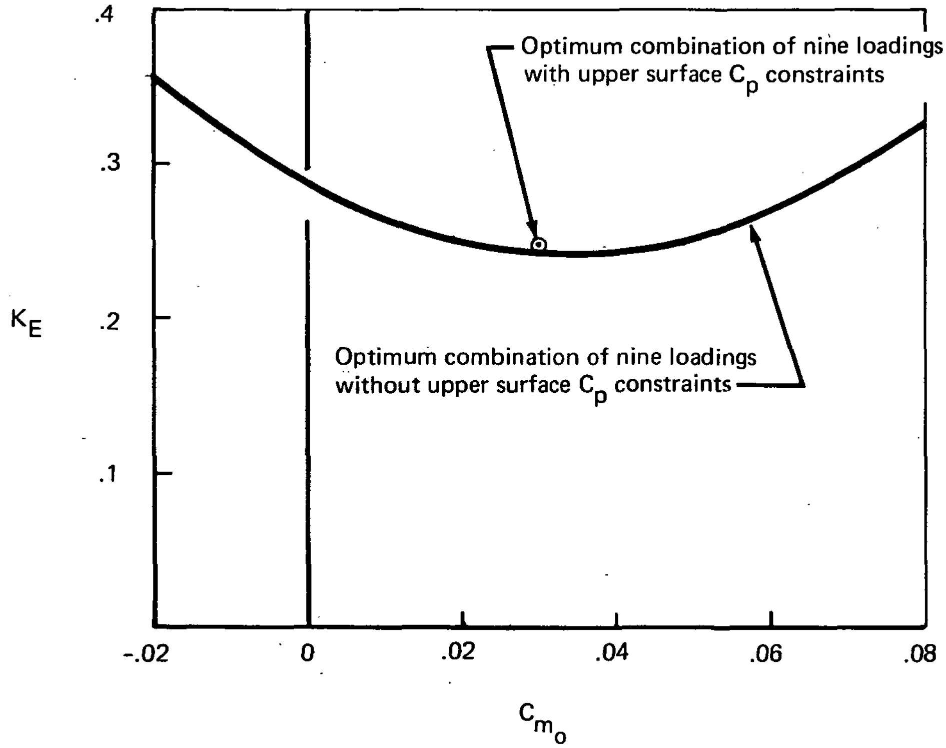


FIGURE 3.7-2.—EFFECT OF PRESSURE CONSTRAINTS ON WING DESIGN

TABLE I

DESCRIPTION OF WING LOADING TERMS

Loading Number	Definition
1.	Uniform
2.	Proportional to x, the distance from the leading edge
3.	Proportional to y, the distance from the wing centerline
4.	Proportional to y^2
5.	Proportional to x^2
6.	Proportional to $x(c - x)$, where c is local chord
7.	Proportional to $x^2\ (1.5\,c - x)$
8.	Proportional to $2\ (1 + 15\ \frac{x}{c})^{-0.5}$
9.	Proportional to $(c - x)^{0.5}$
10.	Elliptical spanwise, proportional to $\sqrt{(1 - y/\frac{b}{2})}$
11.	Proportional to x, the distance from the leading edge of an arbitrarily defined region
12	A camber-induced loading proportional to the body bouyancy loading
13.	A camber-induced loading proportional to the body upwash loading
14.	A camber-induced loading proportional to the nacelle buoyancy loading
15.	The body bouyancy loading
16.	The body upwash loading
17.	The nacelle buoyancy loading

- As a loading definition (12-14), a configuration-dependent loading may be introduced in addition to its independent effect. The optimization then could cancel the lift of the independent effect with this loading, if that were the optimum solution.

A configuration-dependent loading may not be used as the source of a variable loading without also using it as an independent loading.

Use of configuration-dependent loadings. - In designing a wing in the presence of the nacelle pressure field, the design solution includes both the effect of the nacelles on wing lift and drag, and also the effect of the wing lift on nacelle drag. An example of the inclusion of the nacelle influence on the wing design solution is shown in figure 3.7-3. The wing trailing edge is bent upward, or "reflexed", to take advantage of positive pressure coefficients from the nacelle pressure field.

The loadings due to the fuselage include both lift caused by upwash from fuselage incidence, and also lift due to asymmetric distribution of fuselage volume above and below the wing (if any).

As a special case, the asymmetric fuselage buoyancy loading (number 15), can be used even if its net lift is zero; this feature permits the inclusion of fuselage thickness pressures in the pressure limiting case for any wing-fuselage arrangement. However, if the fuselage buoyancy lift is zero, the use of the wing camber loading proportional to the fuselage buoyancy loading (number 12) cannot be used, since it would cause the optimization solution to fail.

Optimization of the wing design considering influence of the fuselage upwash field is performed iteratively, using both the wing design and lift analysis modules. A fuselage shape and incidence is first assumed, the corresponding upwash field is calculated by the analysis program, and the design solution is performed. The resulting camber surface is incorrect in the inboard region (the part covered by the fuselage), both because of the usual linear theory root difficulties and because the design solution does not include the wing-on-fuselage term in the optimization. The camber surface may, however, be cut off at the side of the fuselage and run in the analysis program to obtain a complete solution including the fuselage at all lift coefficients.

The fuselage incidence may then be varied and the cycle repeated. The sequence of events and the corresponding executive control cards (see Section 4) is as follows:

Event	Executive Card
Define fuselage	GEØM
Calculate fuselage upwash	ANLZ (WHUP=1.0)

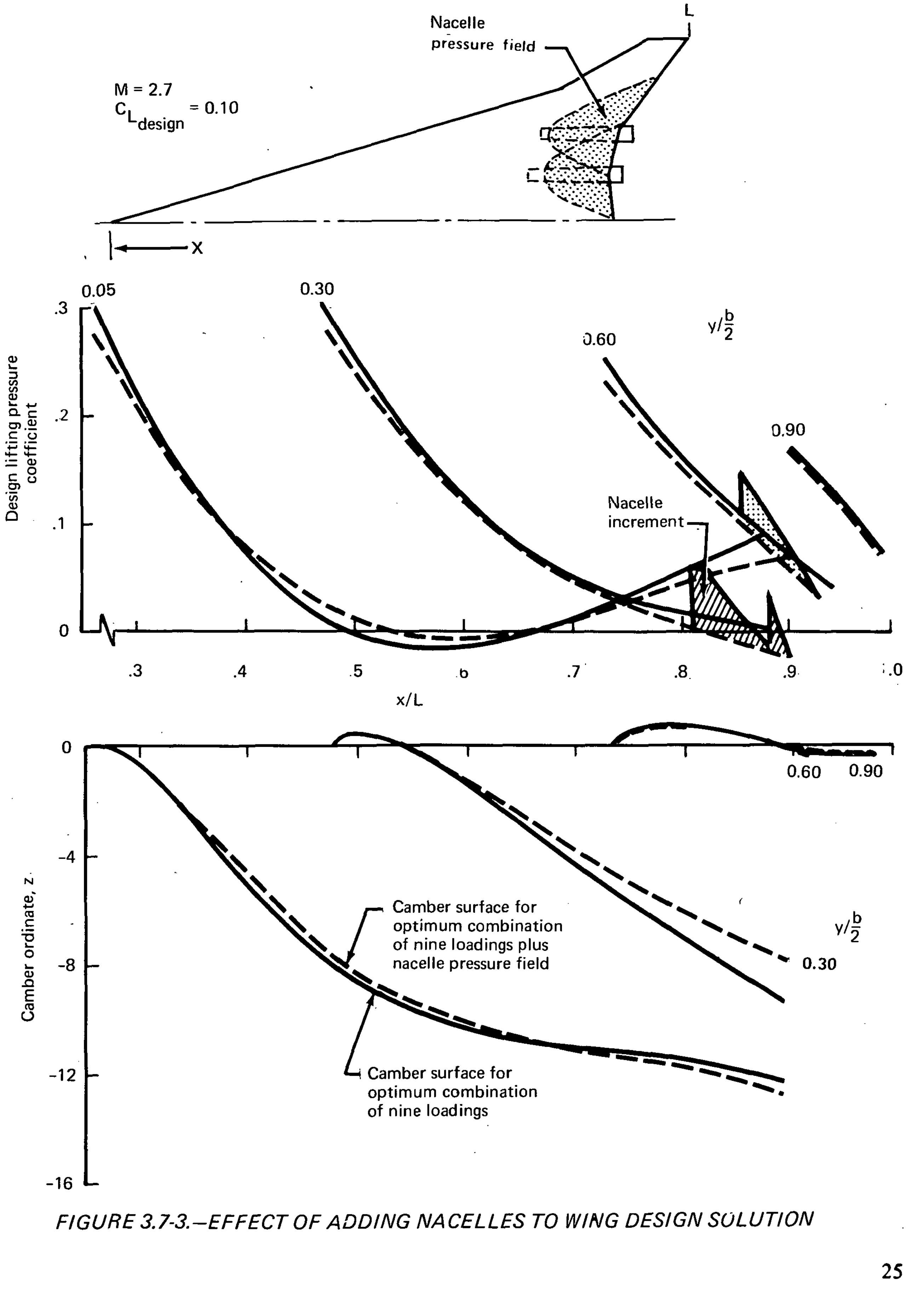


FIGURE 3.7-3.—EFFECT OF ADDING NACELLES TO WING DESIGN SOLUTION

Wing design solution	WDEZ	
Analyze configuration	ANLZ	(TIFZC=3.0)
Redefine fuselage	GEØM	
Etc.		

When the wing camber surface is finalized, it may be transferred into the basic geometry by the executive control card WGUP. (With the interactive graphics attached, the design wing shape may also be viewed and edited between design and analysis programs).

Small planform region option. - Since there may be small regions of the wing (such as a trailing edge flap) that could be relatively highly loaded to good advantage, a program option allows the definition of such a region and a corresponding loading (no. 11 in Table I).

An example of the use of the planform region option is shown in figure 3.7-4. Inclusion of the region and loading 11 results in a small improvement in drag-due-to-lift, especially as C_{mo} is increased.

A condition imposed upon the planform region option is that the region cannot be re-entrant in the spanwise direction, relative to the forward end. The region is input starting at the most inboard span station (which will be at the wing trailing edge), and successive span stations must increase monotonically.

Loading 11 and the small planform region are only used in combination with each other.

Input considerations. - The wing design program principally requires the specification of a set of loadings, a design point, and the definition of four basic control parameters. The control parameters (on card 7 of the design program input) govern the type and extent of the solution.

The design point solution may be obtained with constraints on:

- C_L only
- C_L and C_{mo}
- C_L and upper surface pressure
- C_L, C_{mo}, and upper surface pressure

The four types of solutions are not completely independent. If the C_L and upper surface pressure solution is requested, then the program must first generate the C_L only solution. Similarly, if the C_L, C_{mo}, and upper surface pressure solution is requested, then the program must first generate the C_L and C_{mo} solution. Thus, if the upper surface pressure constraint condition is requested, the program performs the corresponding no pressure constraint solution whether it was requested or not.

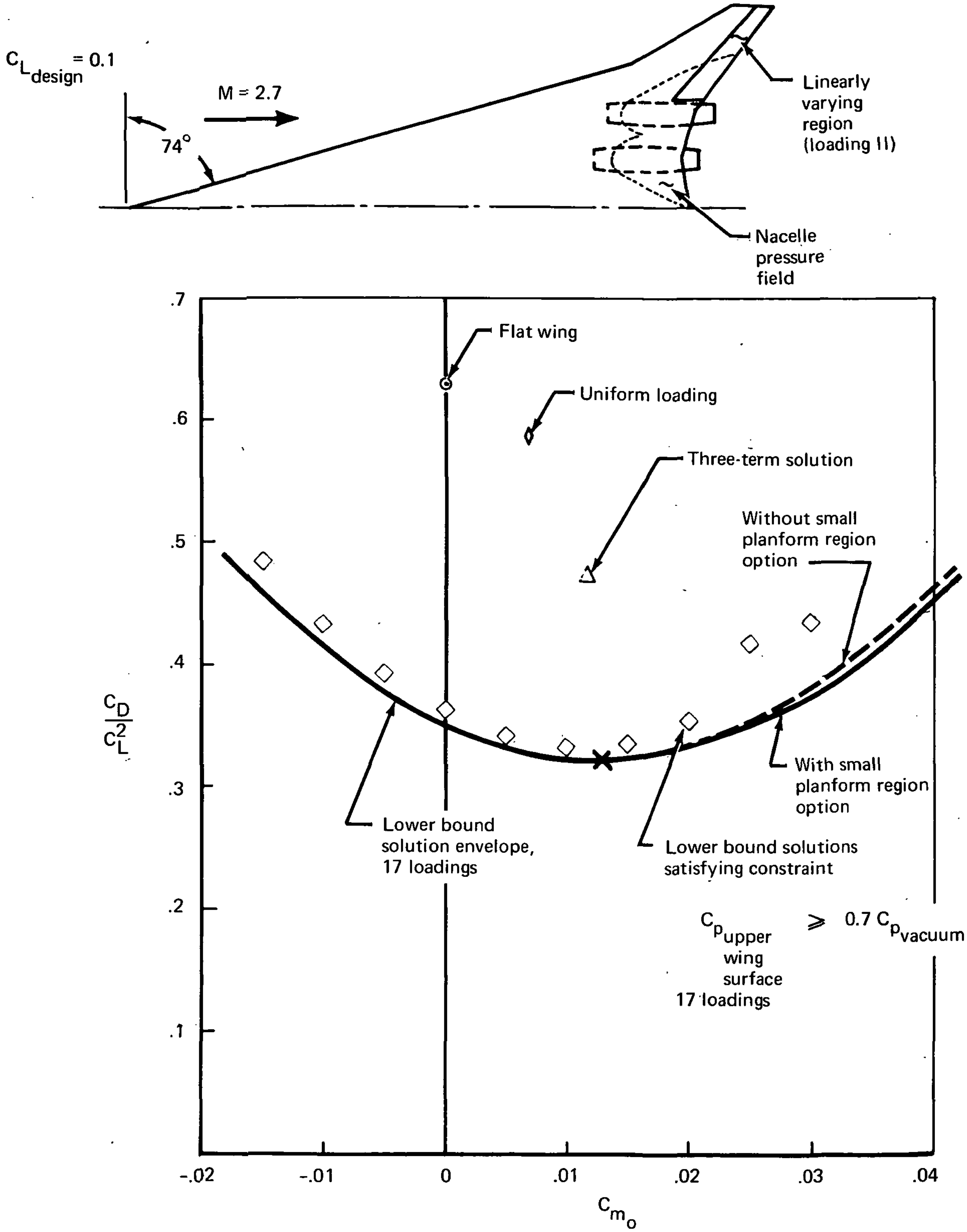


FIGURE 3.7-4.—PLANFORM REGION OPTION

It is not necessary to calculate the camber surface shape corresponding to a specific design point (lift coefficient, pitching moment coefficient, constraint condition) in order to obtain the drag-due-to-lift versus C_{mo} plot. Also, if the design camber surface is requested, it may be only printed out, or may be also punched into cards (for later input into the lift analysis program).

Loading selection. - Experience with the wing design program has shown that the principal loadings of interest in a typical design case are the uniform, linear spanwise, linear chordwise, and quadratic spanwise loadings, plus also the configuration dependent loadings due to fuselage and nacelles (if applicable). The remaining loadings in Table 1 are of diminishing importance in obtaining an optimized solution, although useful if pressure-limiting is requested, or if a substantial C_{mo} is to be provided by wing camber and twist.

Evaluation of the resulting wing design by the lift analysis program is required to obtain the associated configuration force coefficients. This is necessary because the root region of the wing (within the fuselage cross-section) is incorrectly loaded by the design program, and because the wing design program does not consider the drag of the fuselage in isolation or the effect of the wing on the fuselage. In addition, the wing design program normally calculates a camber surface shape which includes "kinks" aft of leading edge breaks (e.g., wing apex) which must be lofted out. The lift analysis program is then used to evaluate the resultant wing shape.

Restart option. - A "restart" option has been provided in the program to minimize computer time on runs involving the same planform and Mach number. (i.e., different design points in terms of C_L, C_{mo}, or pressure constraints). The restart option works as follows: For a given wing planform, Mach number, and set of loadings, most of the computer time is used in calculating the force coefficients and interference coefficients associated with all the component loadings. The calculations involving the solution of an optimum combination of loadings, with or without constraints, are relatively quick (a few seconds). However, it may be desirable to look at a number of different optimization or constraint solutions. Therefore, on successive cases involving the same basic loadings, it is possible to bypass the component loadings solution and go directly to the optimization routines. This is done by setting RESTART= -1. in the program input for cases 2 and on.

If the program cases are to be input at a later time, the component loadings data may be punched into cards and read back in to the computer through use of RESTART= 2. The RESTART=2. data deck includes, as well, the definition of any configuration-dependent loadings that were present in the wing design program at the time the data deck was punched.

RESTART=3.0 is a special provision in which the restart data is read onto a tape, which may later be reread in the wing design program. This feature is useful in cases where the lift analysis program may be run between successive wing designs. (RESTART=3.0 actually functions the same as RESTART= -1., but RESTART= -1. can only be used on successive wing design cases without exiting the wing design program).

The restart option also will work in the case of a decreased number of loadings. E.g., if a maximum (17) loading case were run, then the force and interference loading terms for certain lesser combinations of loadings are available. Successive cases could then be run with different loading combinations to check the design sensitivity to certain loadings, without repeating the basic loadings calculations. The combinations that can be run are only those for a lesser number of loadings and for which the loading numbering order is preserved. However, this latter condition is not as restrictive as it perhaps sounds, since the loadings in Table I can be numbered in arbitrary order in the original input by using card set 10 in the design module input data.

Planform considerations and spanwise integration. - The wing design program is a direct type solution, i.e., a wing shape is calculated from a known pressure distribution. It is not necessary to calculate the wing shape at all spanwise stations in the grid system used to represent the wing; only a representative set of spanwise stations is used. The lift, drag and pitching moment coefficients are then computed from a spanwise integration of the characteristics obtained at the selected spanwise stations.

In the program input, the camber surface calculations are performed at 11 stations (every 10 percent semi-span) unless otherwise specified. If the planform is irregular, particularly along the leading edge, additional spanwise stations in the vicinity of these irregularities should be input to improve the solution accuracy. (This is done through inputs TJBYMX and TJBYS, as described in Section 4.)

In addition, it has been found that the wing root singularity and the corresponding root camber line can often be moderated by substituting a parabolic apex for the sharp apex common to supersonic wing planforms. This will be performed automatically in the program if the input YSNØØT is not zero. The program then fits a parabola tangent to the wing leading edge at YSNØØT, with symmetry about Y=0.

Because the computed camber surface slopes tend to exhibit some irregularity near the leading edge (due to the sawtooth nature of the grid system), a smoothing option is provided in the program. This is activated by the code SMØØTH in the program input. The smoothing technique involves averaging the computed surface slopes of each grid element with the slopes of adjacent elements, which suppresses any erratic slopes of individual elements.

Lift Analysis

Given a wing planform, camber shape, and Mach number, the lift analysis program solves for the lifting pressure distribution and force coefficients for a range of angles of attack. As options, the program will also include the effects of:

- Fuselage (nominally circular in cross-section, arbitrary camber and incidence)
- Nacelles
- Canard and/or horizontal tail
- Wing trailing edge flaps and/or incremental wing twist

Fuselage solutions. - Fuselage effects are obtained by calculating the isolated fuselage upwash field, then calculating the wing solution in the presence of the fuselage upwash field, then calculating the fuselage forces in the wing flow field, and combining the solutions by superposition.

The fuselage upwash field is calculated from slender body theory. The input area distribution of the fuselage is considered to be circular in cross-section. If a digitized fuselage cross-section is input into the basic geometry, the area and centroid of each section is computed and used to define the area and meanline distribution for the analysis program.

The lift analysis program contains a wing-fuselage intersection option. This feature tracks each wing percent chord line out through the side of the fuselage (again considered circular in cross-section), and breaks the wing solution into the proper exposed and carry-over type lifting pressure calculations. Alternatively, the side-of-fuselage span station may be input either as a constant or as a table of values to override the wing-fuselage intersection option.

The local fuselage upwash angle is strongly affected by span station and wing height on the side of the fuselage. The side-of-fuselage span station must be carefully input to avoid exposing any wing area to the upwash field that is actually inside the fuselage.

The lift analysis program contains an option to calculate the buoyancy field due to unequal fuselage area growth above and below the wing. This pressure distribution, termed asymmetric fuselage buoyancy, is calculated by splitting the fuselage area into pieces above and below the wing and adding the resultant area growth onto the fuselage forebody area distribution. (The fuselage is again considered circular, and the side-of-fuselage Z value is used to define the above-wing and below-wing area pieces). The asymmetric fuselage term is zero, of course, in the case of a mid-wing arrangement.

The asymmetric buoyancy calculation is requested by input SYMM (value greater than zero). For a fuselage significantly non-circular in cross-section, use may be made of two special options to define the above-wing and below wing area distributions and the corresponding wing-fuselage intersection:

- SYMM = 2.0 requires input of the above wing and below-wing areas.

- ANYBOD = -10. allows input of definition of the wing-fuselage intersection.

Both of these options require input of the data at the same per cent chords used in the camber surface definition.

Nacelles. - The nacelle calculations are very similar to the solution used in the near-field wave drag program. The pressure fields imposed by the nacelles on the wing, and wing-on-nacelles, are computed and their combined effect on the lifting solution obtained through superposition. The effect of the nacelles on the wing drag-due-to-lift can be substantial because of lift contributed by the nacelle pressure field. Both "wrap" and "glance" solutions for the nacelle pressure field are available, as described in Section 3.6.

Canard and horizontal tail. - Canard and horizontal tail lifting pressure distributions and force coefficients are calculated as for the wing case. The program assumes that a canard is located forward of the wing and a horizontal tail aft of the wing. The effects of downwash from upstream lifting surfaces (if any) are included in the solution.

Experimental comparisons. - Theoretical calculations for a typical supersonic transport configuration are compared with corresponding wind tunnel data in figures 3.7-5 and 3.7-6 (wing-fuselage-nacelles) and figures 3.7-7 and 3.7-8 (incremental effects of horizontal tail). The theoretical buildup of the zero-lift drag coefficient is given in figure 3.7-5.

The lift analysis program contains an optional pressure limiting feature for the wing surface pressures which operates somewhat different from the one in the design program. In the design case, the local wing angle of attack is not allowed to exceed the value

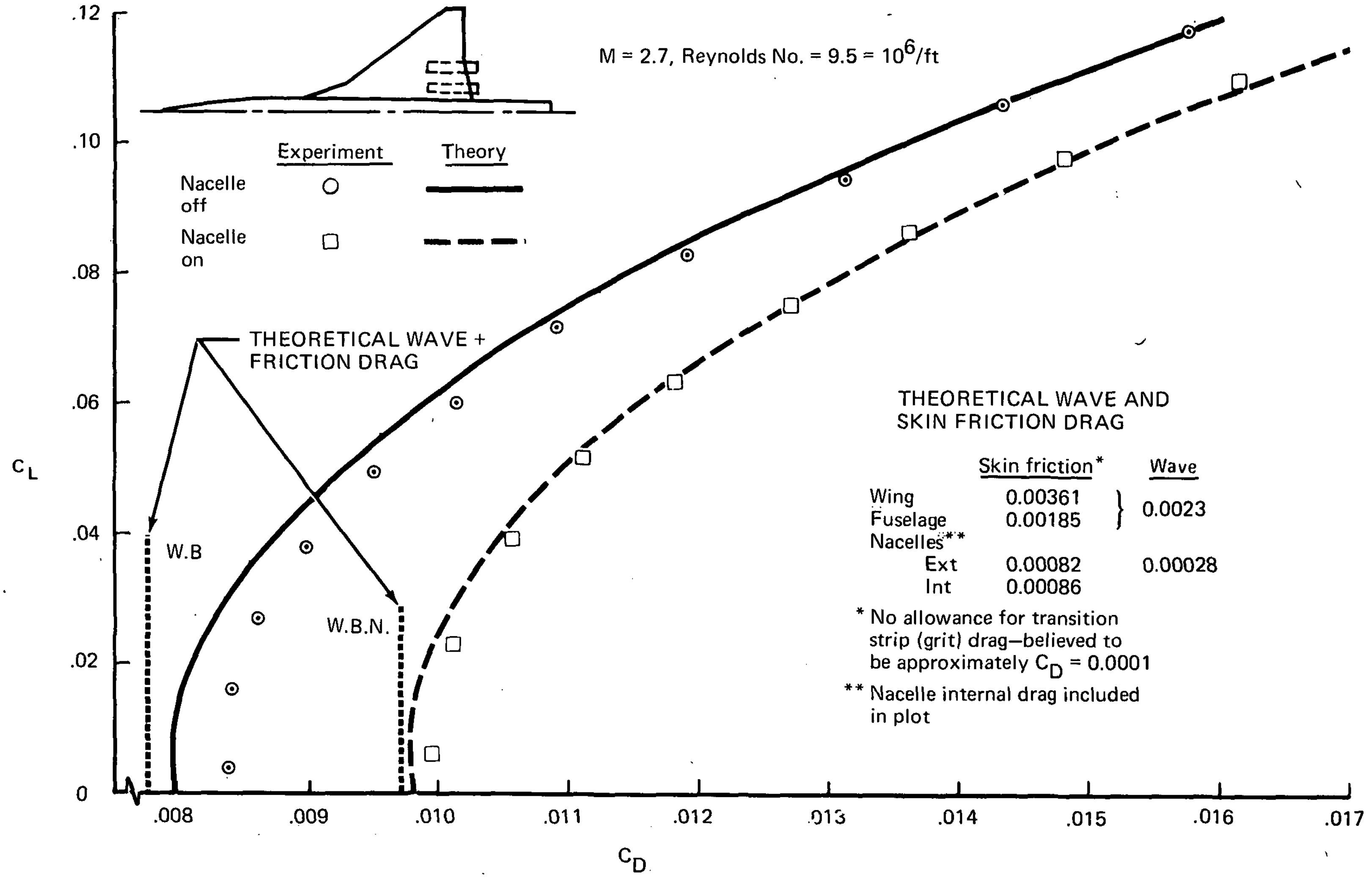


FIGURE 3.7-5.—DRAG POLAR COMPARISON

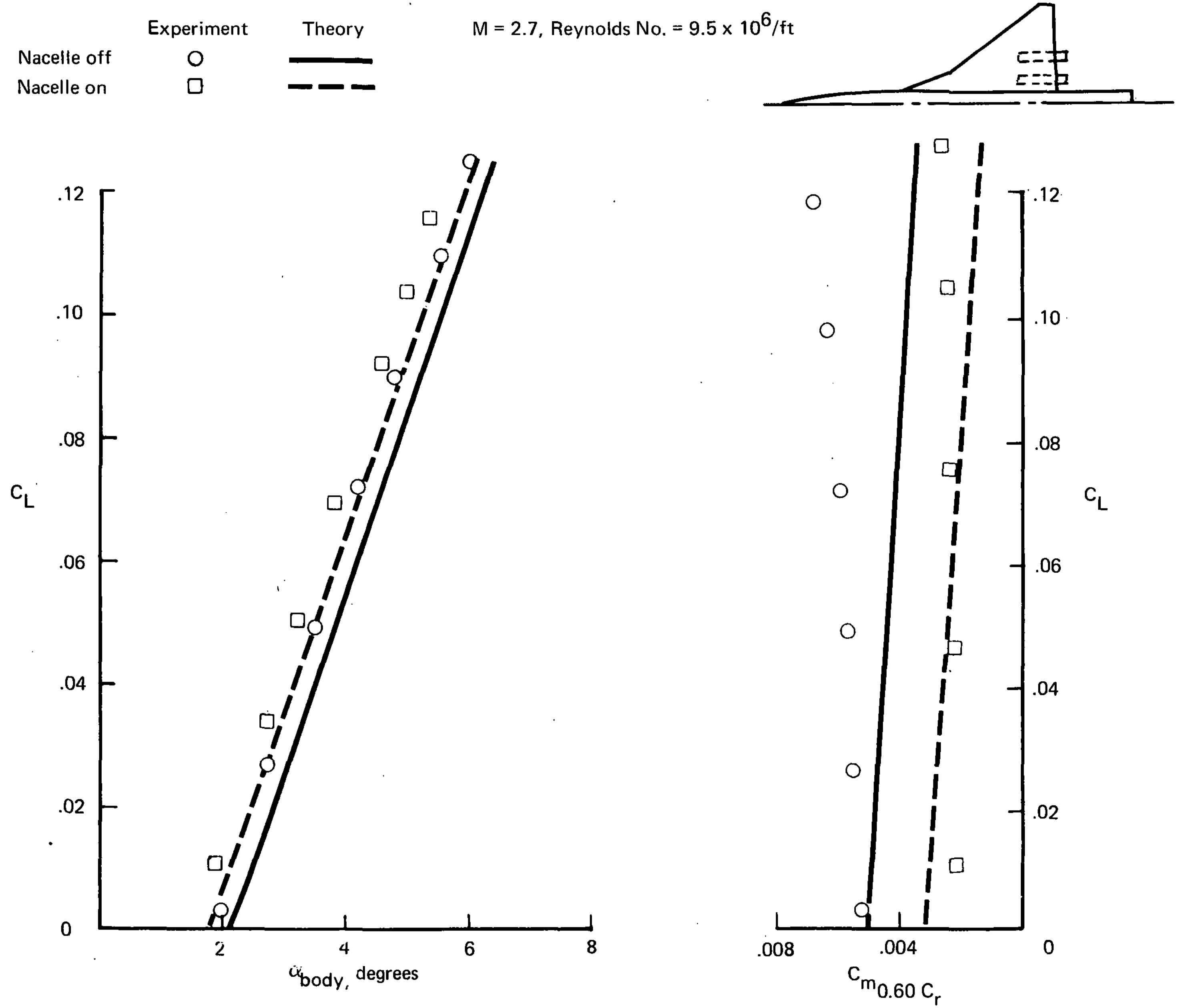


FIGURE 3.7-6.—FORCE COEFFICIENT COMPARISON

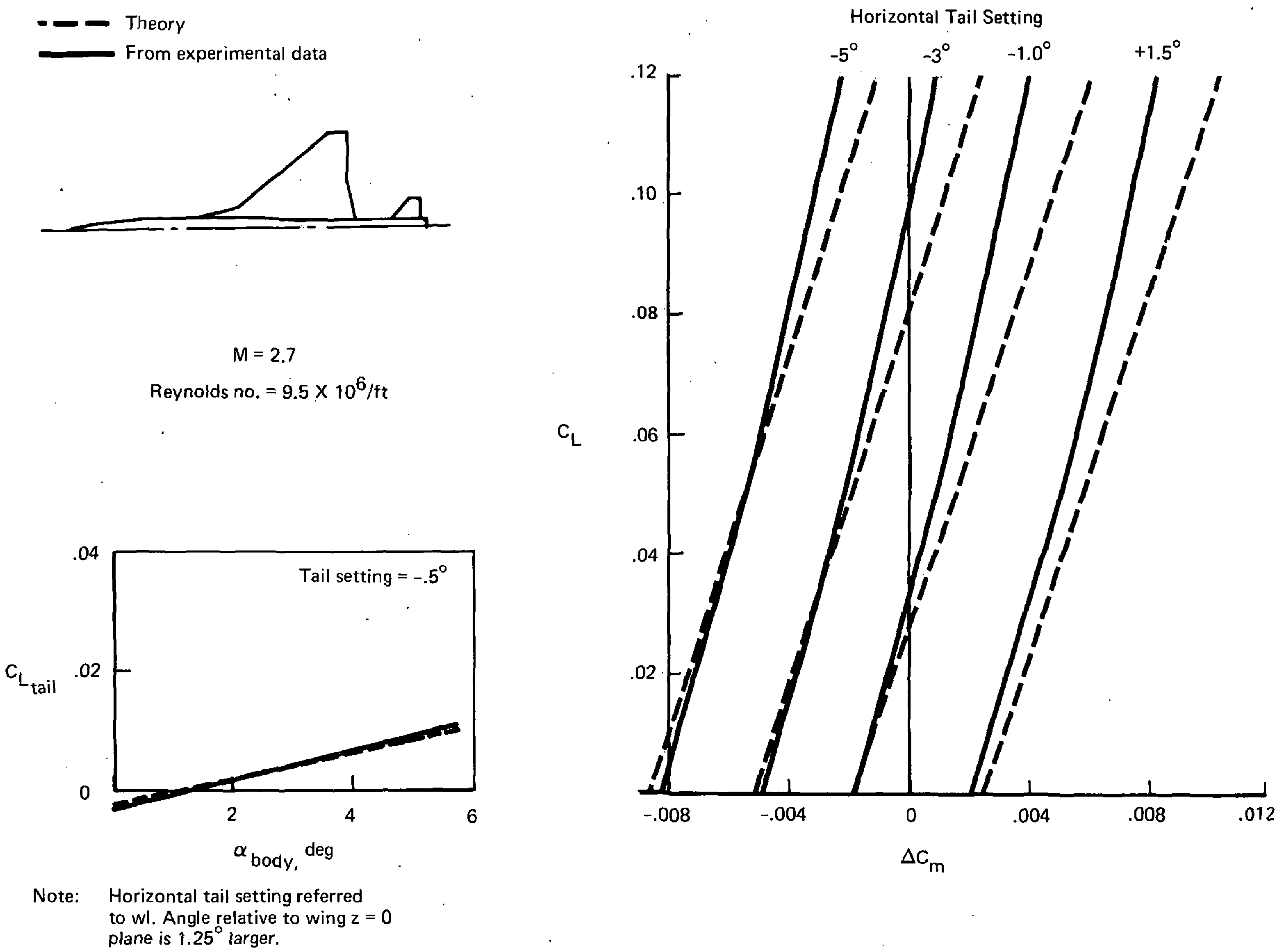


FIGURE 3.7-7.—HORIZONTAL TAIL EFFECTS

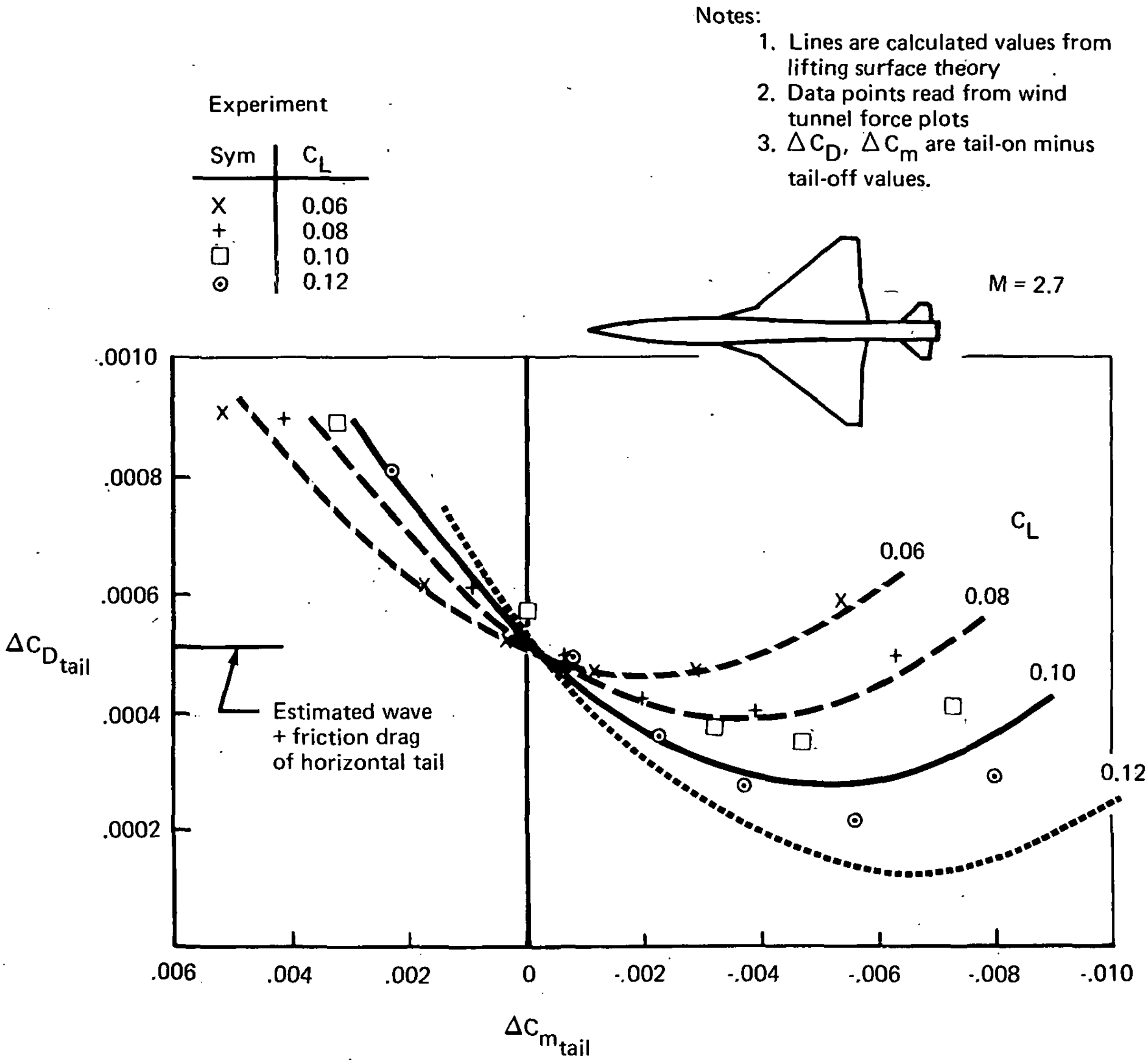


FIGURE 3.7-8.—HORIZONTAL TAIL EFFECTS

associated with a pressure limit condition. In the analysis case, the pressure coefficient limit is imposed, but the local wing incidence may greatly exceed the value at which a limit is first encountered.

When the pressure limiting option is used, a set of configuration angles of attack for the solution must be provided, and the configuration thickness pressures from the near-field program must be provided to permit limiting of the total surface pressure. A solution for a typical wing through an angle of attack series using the pressure limiting feature is shown in figures 3.7-9 and 3.7-10. The limiting feature greatly improves the linear theory representation of the wing pressure distribution as angle of attack is increased.

Configuration-dependent loadings. One mode of lift analysis program usage is to generate configuration-dependent data for the wing design program. These data are produced as follows:

DATA	DESCRIPTION	REQUIREMENTS
Nacelle pressure field	Pressure field caused by nacelles on wing.	Call for nacelles (AJ3=1.0)
Fuselage upwash field	Pressure field induced on wing by fuselage upwash.	Calculate fuselage effects on wing
Fuselage buoyancy field	Pressure field induced on wing by unequal fuselage volume above and below wing.	SYMM=1.0

Upon execution, the program then loads the pressure fields into the proper system common blocks.

If the fuselage buoyancy field is not requested (i.e., SYMM = 0.), the program computes the pressure field due to a mid-wing arrangement. This is done so that a thickness pressure field due to the fuselage will be available for pressure limiting calculations, if desired.

In calculating the fuselage upwash or buoyancy fields, it is important to remember the powerful influence of wing height on the side of the fuselage. This strongly affects both the local upwash angles, and the above-and-below wing area distributions.

Calculation of the fuselage upwash field may be done in either of two ways: the principal condition is that the resultant pressure field is that due to upwash only. In the computer program, this is handled by inputting a camber surface having approximately the correct wing-fuselage relationship (wing height, etc.), but then

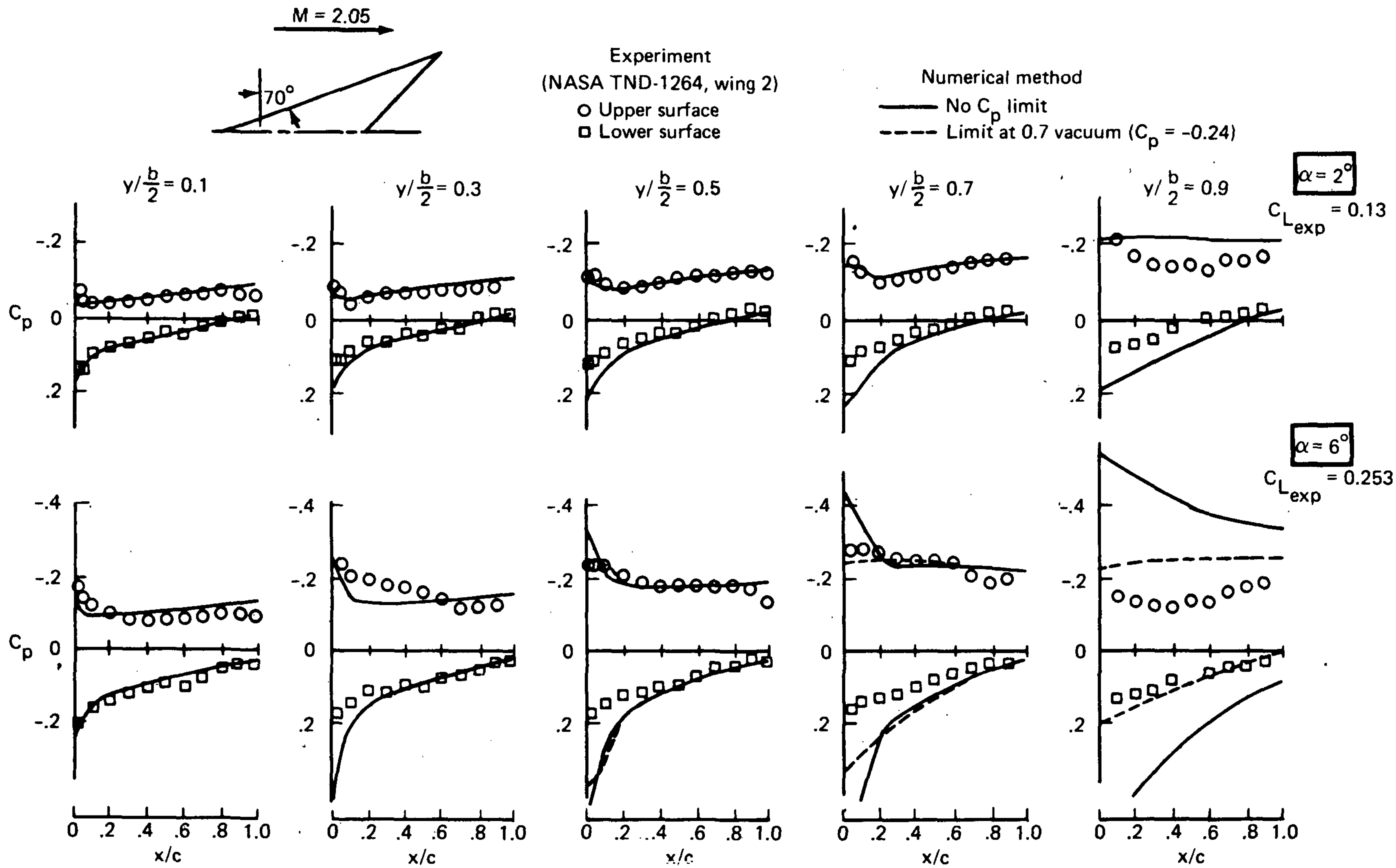


FIGURE 3.7-9.—PRESSURE COEFFICIENT COMPARISON—WING 2 (TWISTED AND CAMBERED WING, C_L = 0.08)

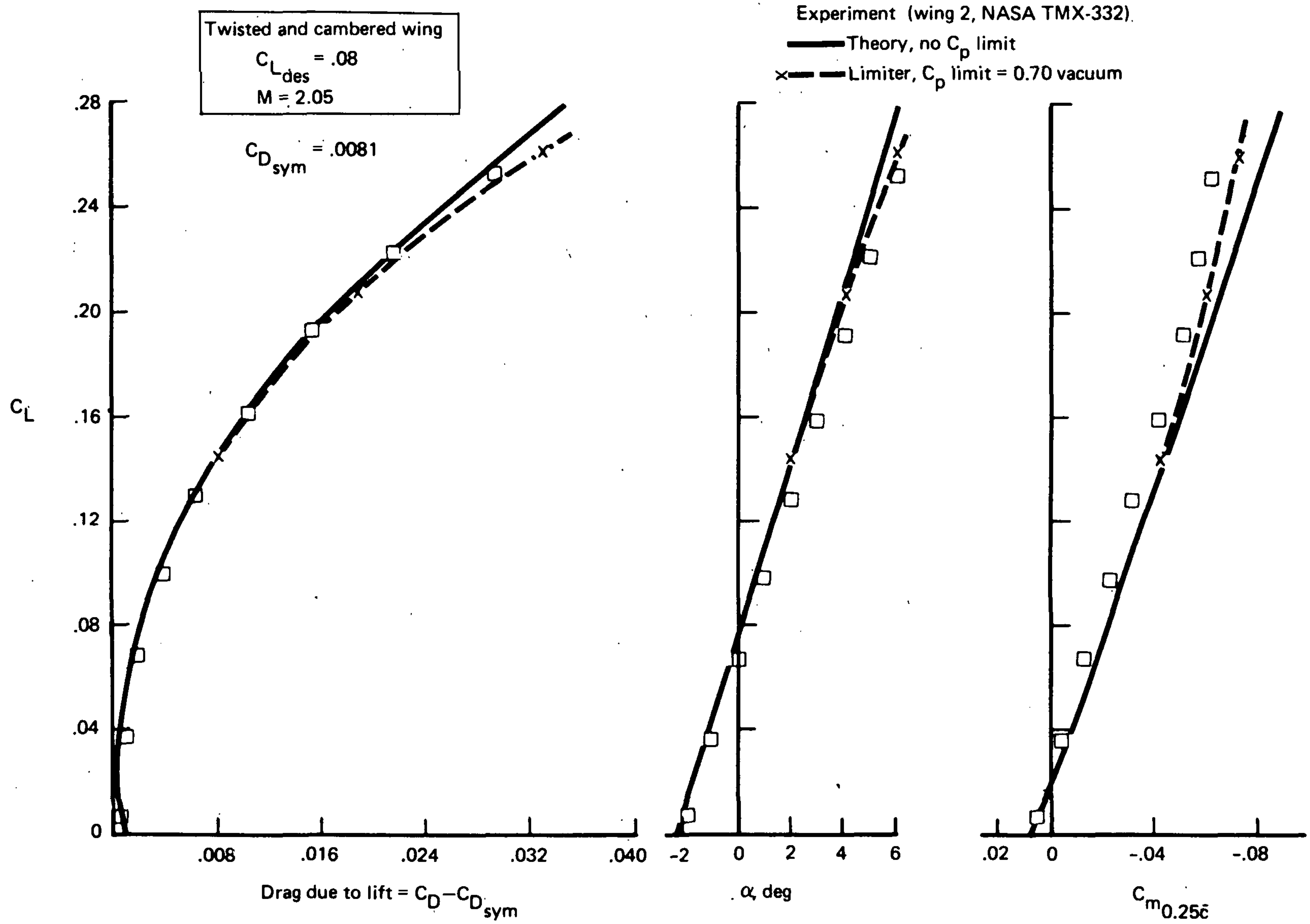


FIGURE 3.7-10.—TEST—THEORY COMPARISON, WING 2
($C_{L_{des}}$ = 0.08)

zeroing the wing slopes in the camber surface calculations (by setting WHUP=1.0). In iterative cycles, the wing camber surface and fuselage relationship can be refined.

Alternatively, as a crude starting point in the fuselage upwash calculation, the flat wing option can be used. By setting TIFZC=2.0, the wing slopes are automatically zeroed and the wing height relative to the fuselage will be controlled by the fuselage meanline input and the wing leading edge Z definition (ZLED and Z FUS in the basic geometry).

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4.0 INPUT FORMAT

Input requirements for the system are given in this section and consist of:

- Executive control card summary
- Basic geometry definition
- Additional data input for programs of system

The usual input format is 10 field - 7 digit, punched with decimals to the left in the card fields. Some data (particularly the control codes in the basic geometry) are input in integer form, without decimal, to the right in the card field. The formats are identified in all cases.

To provide design or analysis flexibility, there are numerous program options that are controlled by input codes. Where there is a "normal" way of handling the option, the code is defaulted to zero (i.e., if the field contains a zero or is blank, the "normal" solution will be calculated).

NOTE

The interface tape writes input to the program in F10.4 format, so that only four places to the right of the decimal point get transferred from the interface to the programs, regardless of the number of places originally input on cards.

4.1 Executive Control Card Summary

Configuration input and program execution are ordered by means of control cards read at the executive level.

The control cards consist of a few alphanumeric characters starting in column 1.

Geometry input. - The configuration geometry is read and manipulated in the geometry module. Geometry may be input as all-new, or as a replacement or addition to existing geometry. The control cards for geometry input are:

GEØM NEW	All-new configuration description follows, and any previous geometry is purged. (Leave one column space between GEØM and NEW).
GEØM	Input geometry is added to (or replaces) existing description.

Geometry update. - The basic geometry description contained in the geometry module may be updated using data contained in 0, 0 level common blocks. This applies to a new fuselage definition (i.e., optimized fuselage from the far-field wave drag program) or a new wing camber surface definition. The control cards are:

FSUP	Fuselage will be updated to definition contained in /ØPBØD/. The /ØPBØD/ definition is created each time the far-field wave drag program executes the optimum-fuselage-with-restraints case.

If the fuselage update is requested, a second card, telling how to perform the update, is required. Punch (starting in column 1) the following code:

-1. Fuselage is to be redefined at same x stations as previous definition.

1. Fuselage is to be defined at 50 equally spaced stations.

WGUP	Wing camber surface will be updated to the definition contained in /CAMBER/. The /CAMBER/ definition is created each time the wing design program executes, produces a camber surface for a specified set of conditions.

The user must remember that the update for fuselage or camber surface will require that the /ØPBØD/ or /CAMBER/ definition be current. These common blocks will contain the last definition produced by the far-field wave drag or wing design programs.

Program execution. - Execution of the programs in the system is ordered by the following cards:

PLØT	plot program
SKFR	skin friction program
FFWD	far-field wave drag program
NFWD	near-field wave drag program
ANLZ	lift analysis program
WDEZ	wing design program

The control card for program execution is the first card of the set describing the program data input. Individual program inputs are given on the following pages.

Multiple case execution with the basic programs of the system is possible, as in the stand-alone versions of the programs. The data for successive cases are stacked as described in the program input description. At the end of the data stack, an END card is required to terminate the program. The END card is not needed for the geometry module, however.

Interactive graphics. - The graphics subroutines in the system are activated by the executive card CRT (punched in first three card columns). The CRT card may be placed anywhere in the data deck that an executive card may be read. If no CRT card is included, the system will execute without accessing any of the graphics programs.

A description of the interactive graphics part of the design and analysis system is presented in Appendix A.

4.2 Geometry Program

The geometry program stores the basic geometry data, and stacks it as required by the individual programs of the system.

Access to the geometry program, to store or alter the configuration description, is through the GEØM or GEØM NEW control card (see executive control card summary).

The format of the geometry input uses both integer (control cards) and floating point numbers. All integers are punched right justified in their fields on the cards, without decimals. All floating point numbers are punched, with decimals, to the left of the field in 10 field -7 digit format. The program logic uses the component control codes (J1, J2, etc.) on card 3 as follows:

Value	Use
0	Component will not be input. However, if the component has previously been input (and not purged by a GEØM-NEW card) the 0 is interpreted as a 2.
2	Previously input component is left as is.
Other	New input for this component replaces previous input.

The logic of treating a 0 as a 2 for existing components is to protect data on the geometry file from inadvertent loss. Then, if it is desired to add or change a configuration component on successive runs, only the new component need be addressed.

A control code other than 0 or 2 instructs the program to completely replace the previous component description with a new one. It is not possible to add a fin or nacelle to a previous fin or nacelle; the new description must be complete in itself.

Deletion of a component is possible only through purging the entire configuration, using the GEØM NEW card.

Card Number	Card Column	Decimal Required	Variable Name	Description
1				GEØM or GEØM NEW
	1-4			GEØM = geometry addition
	1-8			GEØM NEW = all-new geometry
2	1-70			Any desired title information.
3	1-3	NO	J0	Reference geometry code. 0 = Reference geometry not required (plot program) 1 = Read reference area, $\bar{c}$, x_{cg} 2 = Reference geometry same as previous case.
3	4-6	NO	J1	Wing input code -1 = Read uncambered wing 0 = No wing 1 = Read cambered wing 2 = Wing same as previous case.
3	7-9	NO	J2	Fuselage input code -1 = Read circular fuselage 0 = No fuselage 1 = Read arbitrarily shaped (digitized) fuselage 2 = Fuselage same as previous case 3 = Read circular fuselage and perimeter values.
3	10-12	NO	J3	Nacelle input code 0 = No nacelles 1 = Read nacelles 2 = Nacelles same as previous case.

Card Number	Card Column	Decimal Required	Variable Name	Description
3	13-15	NO	J4	Fin input code 0 = No fin 1 = Read fin data 2 = Fin data same as previous case.
3	16-18	NO	J5	Canard (or horizontal tail) input code 0 = No canards 1 = Read canard data 2 = Canards same as previous case.
3	19-21	NO	J6	Fuselage Simplification code -1 = Uncambered circular fuselage 0 = Cambered circular or arbitrary fuselage. 1 = Complete configuration is symmetrical with respect to X-Y plane, which implies uncambered circular fuselage if there is a fuselage.
3	22-24	NO	NWAF	Number of airfoils describing wing. $2 \leq NWAF \leq 20$.
3	25-27	NO	NWAFOR	Number of ordinates defining each airfoil section. $3 \leq NWAFOR \leq 20$.
3	28-30	NO	NFUS	Number of fuselage segments. $0 \leq NFUS \leq 4$.
3	31-33	NO	NRADX(1)	Number of points defining half section of first fuselage segment. If fuselage is circular, the program calculates the indicated number of Y and Z ordinates. $3 \leq NRADX(1) \leq 30$.
3	34-36	NO	NFORX(1)	Number of stations for first fuselage segment. $4 \leq NFORX(1) \leq 21$.
3	37-39	NO	NRADX(2)	Same as above for segment 2.
3	40-42	NO	NFORX(2)	Same as above for segment 2.

Card Number	Card Column	Decimal Required	Variable Name	Description
3	43-45	NO	NRADX(3)	Same as above for segment 3.
3	46-48	NO	NFORX(3)	Same as above for segment 3.
3	49-51	NO	NRADX(4)	Same as above for segment 4.
3	52-54	NO	NFORX(4)	Same as above for segment 4.
3	55-57	NO	NP	Number of nacelles to read. NP ≤ 3.
3	58-60	NO	NPODOR	Number of stations at which nacelle radii are specified. 4 ≤ NPODOR ≤ 20.
3	61-63	NO	NF	Number of fins to read. NF ≤ 6.
3	64-66	NO	NFINOR	Number of ordinates defining each fin airfoil section. 3 ≤ NFINOR ≤ 10.
3	67-69	NO	NCAN	Number of canards to read. NCAN ≤ 2.
3	70-72	NO	NCANOR	Number of ordinates defining each canard airfoil section. 3 ≤ NCANOR ≤ 10. If negative, airfoils are non-symmetric.
4	1-7	YES	REFA	Wing reference area
4	8-14	YES	CBAR	Pitching moment reference length. (Required for ANLZ and WDEZ only)
4	15-21	YES	XBARIN	X value of pitching moment center (Required for ANLZ and WDEZ only)

Note: Omit this card if J0 (Card 3) is 0 or 2.

Card Number	Card Column	Decimal Required	Variable Name	Description
Omit card sets 5, 6, 7, 8 and 9 if J1 is 0 or 2.				
5	1-70	YES	XAF	Array of percent chords at which wing airfoil ordinates will be specified.
6	1-7	YES	XLED	X coordinate of airfoil leading edge.
6	8-14	YES	YLED	Y coordinate of airfoil leading edge.
6	15-21	YES	ZLED	Z coordinate of airfoil leading edge.
6	22-28	YES	CLED	Airfoil chord length
	Note:	This card is repeated for each airfoil, ordered inboard to outboard.		
7	1-70	YES	TZORD	Array of camber Z values referenced to Z coordinate of airfoil leading edge, ordered leading edge to trailing edge.
	Note:	This card is repeated for each airfoil, ordered inboard to outboard. Omit card set 7 if wing not cambered.		
8	1-70	YES	WAFORD	Array of airfoil upper surface half thickness ordinates expressed in percent chord, ordered leading edge to trailing edge.
	Note:	Repeat Card Set 8 for each airfoil, ordered from inboard to outboard.		
	Note:	Card Set 9, an option in the plot program input to define the lower surface airfoil for an asymmetric airfoil shape, was deleted from the basic geometry to reduce core size.		

Fuselage Description

Omit card sets 10-15 if J2 is 0 or 2. The fuselage is input in segments. Complete input for each segment before going on to next segment. A segment may contain ≤21 defining stations.

If there is more than one fuselage segment, the first station of a segment repeats the definition of the last station of the preceding segment (i.e., cross-section is again defined at the same X station). Otherwise, a gap in the fuselage description will occur between the last station of one segment and the first station of the following segment.

Card Number	Card Column	Decimal Required	Variable Name	Description
10	1-70	YES	ZFUS	Array of fuselage X stations
11	1-70	YES	ZFUS	Array of Z coordinates defining fuselage centerline.
	Note:	Omit card set 11 if J6 ≠ 0 or if J2 = 1.		
12	1-70	YES	FUSARD	Array of fuselage cross sectional areas.
	Note:	Omit card set 12 if J2 not equal to -1 or 3.		
13	1-70	YES	FUSPER	Array of fuselage perimeters.
	Note:	Omit card set 13 if J2 not equal to 3		
14	1-70	YES	SFUS	Array of Y coordinates defining first station half section, ordered bottom to top.
15	1-70	YES	SFUS	Array of Z coordinates defining first station half section, ordered bottom to top.

Note: Repeat card sets 14 and 15 for each station in segment 1. Omit card sets 14 and 15 if J2 is not equal to 1.

Note: For each fuselage segment, repeat card sets 10 thru 15.

Nacelle Description

Card Number	Card Column	Decimal Required	Variable Name	Description
Omit card sets 16, 17 and 18 if J3 is 0 or 2.				
16	1-7	YES	PODORX	X coordinate of origin of first nacelle
16	8-14	YES	PODORY	Y coordinate of origin of first nacelle
16	15-21	YES	PODORZ	Z coordinate of origin of first nacelle
16	22-28	YES	PODZW	Z coordinate of origin of first nacelle, referenced to local wing surface. 0., *program will calculate from* PODORZ +D, nacelle is located D units above local wing surface -D, nacelle is located D units below local wing surface
	Note: If PODZW $\neq$ 0., PODORZ is not required.			
17	1-70	YES	XPOD	Array of X coordinates, referenced to nacelle origin, at which nacelle radii will be specified.
18	1-70	YES	RPOD	Array of nacelle radii.

Note: For each nacelle, repeat card sets 16 thru 18.
If PODORY is non-zero, a duplicate nacelle is located symmetrically to the X-Z plane.

Fin Description

Card Number	Card Column	Decimal Required	Variable Name	Description

Omit card sets 19, 20 and 21 if J4 is 0 or 2.

19	1-7	YES		X coordinate of lower fin airfoil leading edge.
19	8-14	YES		Y coordinate of lower fin airfoil leading edge.
19	15-21	YES		Z coordinate of lower fin airfoil leading edge.
19	22-28	YES		Chord length of lower airfoil leading edge.
19	29-35	YES		X coordinate of upper fin airfoil leading edge.
19	36-42	YES		Y coordinate of upper fin airfoil leading edge.
19	43-49	YES		Z coordinate of upper fin airfoil leading edge.
19	50-56	YES		Chord length of upper airfoil.
20	1-70	YES	XFIN	Array of percent chords, ordered leading edge to trailing edge, at which fin airfoil ordinates will be specified.
21	1-70	YES	FINORD	Array of fin airfoil half thickness ordinates expressed as percent chord.

Note: Repeat card sets 19 thru 21 for each fin.

Canard (Or Horizontal Tail) Description

Card Number	Card Column	Decimal Required	Variable Name	Description
				Program identifies horizontal tail or canard by location relative to wing. Omit card sets 22-25 if J5 is 0 or 2.
22	1-7	YES		X coordinate of inboard canard airfoil leading edge.
22	8-14	YES		Y coordinate of inboard canard airfoil leading edge.
22	15-21	YES		Z coordinate of inboard canard airfoil leading edge.
22	22-28	YES		Chord length of inboard canard airfoil.
22	29-35	YES		X coordinate of outboard canard airfoil leading edge.
22	36-42	YES		Y coordinate of outboard canard airfoil leading edge.
22	43-49	YES		Z coordinate of outboard canard airfoil leading edge.
22	50-56	YES		Chord length of outboard canard airfoil
23	1-70	YES	XCAN	Array of percent chords, ordered leading edge to trailing edge, at which canard airfoil ordinates will be specified.
24	1-70	YES	CANORD	Array of canard airfoil upper surface half-thickness ordinates expressed as percent chord ordered leading edge to trailing edge.
25	1-70	YES	CANOR1	Same as above for lower canard airfoil

Note: If canard is symmetric, omit card set 25.

Note: For each canard, repeat card sets 22 thru 25.

4.3 Plot Program

This program draws a picture of the configuration defined in the basic geometry, as requested by the codes on card 3.

Views of the configuration are controlled by the inputs on card 4. There will be as many drawings of the configuration as there are cards 4.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			PLØT
2	1-80			Any desired title information.
3	1-7	YES	AJ1	Wing input code. 0. = Ignore wing definition. 1. = Include wing definition.
3	8-14	YES	AJ2	Fuselage input code. 0. = Ignore fuselage definition. 1. = Include fuselage definition.
3	15-21	YES	AJ3	Nacelle input code. 0. = Ignore nacelle definitions. 1. = Include nacelle definitions.
3	22-28	YES	AJ4	Fin input code. 0. = Ignore fin definitions. 1. = Include fin definitions.
3	29-35	YES	AJ5	Canard input code. 0. = Ignore canard definitions. 1. = Include canard definitions.
4	1		HORZ	X, Y, Z for horizontal axis.
4	3		VERT	X, Y, or Z for vertical axis.
4	5-7		TEST1	ØUT if deletion of hidden lines required; otherwise blank.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	8-12	YES	PHI	Roll angle in degrees.
4	13-17	YES	THETA	Pitch angle in degrees.
4	18-22	YES	PSI	Yaw angle in degrees.
4	48-52	YES	PLOTSZ	Length in inches of maximum configuration dimension.
4	53-55			Punch ØRT in these columns

Note: For each additional plot desired, card 4 will be repeated at this position in the data deck.

5	1-3			END

4.4 Skin Friction Program

Codes on card 3 control inclusion of basic geometry as requested. Where additional input is required (e.g., fuselage perimeter option), input areas or lengths in units consistent with the basic geometry definition.

The skin friction coefficient subroutine in the program requires lengths in feet. The input lengths are converted to feet, if necessary, using the factor SCAMOD on card 5 or 6.

Inputs on cards 3 and 4 are integers, and must be right-justified in the field, without decimal. The other input are 10 field -7 digit format, with decimals.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			SKFR
2	1-70			Any desired TITLE information

Card Number	Card Column	Decimal Required	Variable Name	Description
3	1-3	NO	J1	Wing input code -1 = Wing defined in basic geometry. Make no correction for wing-fuselage joint. 0 = No wing defined. 1 = Wing defined in basic geometry. Subtract wing root area from fuselage wetted area. 2 = Wing same as preceding case.
3	4-6	NO	J2	Fuselage input code -1 = Wetted area and reference length will be input. 0 = No fuselage defined. 1 = Fuselage defined in basic geometry. 2 = Fuselage same as preceding case.
3	7-9	NO	J3	Nacelle input code -1 = Wetted area and reference length will be input. 0 = No nacelles defined. 1 = Nacelles defined in basic geometry. 2 = Nacelles same as preceding cases.
3	10-12	NO	J4	Fin input code -1 = Fins defined in basic geometry. Make no correction for fin-fuselage joint. 0 = No fins defined. 1 = Fins defined in basic geometry. Subtract fin root area from fuselage wetted area. 2 = Fins same as preceding case.

Card Number	Card Column	Decimal Required	Variable Name	Description
3	13-15	NO	J5	Canard (or horizontal tail) input code -1 = Canards defined in basic geometry. Make no correction for canard-fuselage joint. 0 = No canards defined. 1 = Canards defined in basic geometry. Subtract canard root area from fuselage wetted area. 2 = Canards same as preceding case.
4	1-4	NO	K1	Mach number-altitude combination code. -K1 = Combination same as preceding case. 0 = Use Mach number-Reynolds combinations. K1 = Number of Mach-altitude combinations. $K1 \leq 20$
4	5-8	NO	K4	Mach number-Reynolds combination code. -K4 = Combinations same as preceding case. 0 = Use Mach number-altitude combinations. K4 = Number of Mach-Reynolds combinations. $K4 \leq 20$
4	9-12	NO	NXTPT	Miscellaneous components code. -NXTPT = Same components as preceding case. 0 = No miscellaneous components defined. NXTPT = Number of miscellaneous components. $NXTPT \leq 10$.
4	13-19	YES	POVLP	Total overlap area for nacelles Subtract from wing wetted area.
5	1-7	YES	AM	Mach number
5	8-14	YES	AL	Altitude (feet/1000.)
5	15-21	YES	DELT	Temperature deviation from standard day (^{o}F)

Card Number	Card Column	Decimal Required	Variable Name	Description
5	22-28	YES	SCAMOD	Scale factor to convert input dimensions to feet.
	Note:	There will be K1 of these cards. Omit card set 5 if K1 is 0 or negative.		
6	1-7	YES	AM	Mach number
6	8-14	YES	RNPFL	Reynolds Number per foot length x $10.^{-6}$
6	15-21	YES	SCAMOD	Scale factor to convert input dimensions to feet.
6	22-28	YES	TOTEM	Total temperature (^{o}R)
	Note:	There will be K4 of these cards. Omit card set 6 if K4 is 0 or negative.		
7	1-7	YES	SWETRB	Fuselage wetted area
7	8-14	YES	FUSL	Fuselage reference length.
	Note:	Omit card 7 if J2 is 0, 1 or 2.		
8	1-7	YES	SWETNA	Total nacelle wetted area
8	8-14	YES	TODL	Nacelle reference length.
	Note:	Omit card 8 if J3 is 0, 1 or 2.		
9	1-7	YES	SWETXP	Wetted area of miscellaneous component.
9	8-14	YES	RXLP	Reference length of miscellaneous component.
9	15-24		PTITLE	Any desired title information.
	Note:	There will be NXTPT of these cards. Omit card set 9 if NXTPT is 0 or negative.		

For each new case, add Cards 2 through 9 at this position in the data deck.

10	1-3			END

4.5 Far-Field Wave Drag Program

Codes on card 3 control inclusion of basic geometry data as requested. The case number in first field of card 4 is an integer, and must be right justified in the field, without decimal. Other input are in 10 field, -7 digit format.

If the fuselage restraint feature is used, the resulting fuselage definition for the last case will be stored and can be used to update the basic geometry (see executive control card summary, FSUP).

Multiple cases involving a given configuration description (e.g., various Mach numbers) may be run by a card 4 series. If the geometry is to be changed, an END card must be input and the program re-entered by an FFWD or GEOM and FFWD set-up.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			FFWD
2	1-80			Any desired title information.
3	1-7	YES	AJ1	Wing input code. 0. = Ignore wing definition. 1. = Include wing definition.
3	8-14	YES	AJ2	Fuselage input code. 0. = Ignore fuselage definition. 1. = Include fuselage definition.
3	15-21	YES	AJ3	Nacelle input code 0. = Ignore nacelle definitions. 1. = Include nacelle definitions.
3	22-28	YES	AJ4	Fin input code. 0. = Ignore fin definitions. 1. = Include fin definitions.
3	29-35	YES	AJ5	Canard (or horizontal tail) input code. 0. = Ignore canard definitions. 1. = Include canard definitions.

Case Cards

Cards 4 input a series of cases of different Mach number, cut or theta variables, and/or fuselage restraints. The solution is performed with the fuselage as input, and also for an optimum fuselage shape (subject to restraint points at which the fuselage shape must be as input). If no fuselage restraint is specified (NREST = 0.), one will be assumed at the station of maximum overall area. If NREST>0., a restraint card (card 5) will follow the case card, and that restraint condition will apply for subsequent cases if NREST is not changed.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	1-4	NO	NCASE	Case identification (right-justified)
4	8-14	YES	XMACH	Mach number
4	15-21	YES	NX	Number of equal intervals into which the portion of the X-axis, XA to XB for each roll angle, is to be divided. NX ≤ 100. and an even number.
4	22-28	YES	NTHETA	Number of equal intervals into which the domain of theta (-90° to 90°) is to be divided. If the area distribution at only theta = -90 is desired, then NTHETA ≤ 36 and a multiple of four.
4	29-35	YES	NREST	Number of X stations for fuselage restraint points (≤ 10.), used for all subsequent cases if NREST does not change. If NREST = 0., program assumes restraint points at nose, base, and station of maximum overall area.
5	1-70	YES	XREST	Array of fuselage stations, (including nose and base) at which computed minimum drag curve will be restrained to input area.

Note: Repeat card 4 for each new case. Only 1 card 5 may be input, after first card 4 with NREST ≠ 0.

6	1-3			END

4.6 Near-Field Wave Drag Program

Two options are provided for fairing the wing section shape at a given spanwise station: linear or second order, controlled by TNOPCT on card 4.

The code ANYBOD (on card 5) identifies the span station of the inboard end of the wing for calculating wing thickness pressures and wave drag. This is the y value of the wing-fuselage intersection if there is a fuselage.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			NFWD
2	1-72			Any desired TITLE information.
3	1-7	YES	AJ2	Fuselage input code. 0. = Ignore fuselage definition. 1. = Include fuselage definition.
3	8-14	YES	AJ3	Nacelle input code. 0. = Ignore nacelle definitions. 1. = Include nacelle definitions.
4	1-7	YES	TNOPCT	Fairing code. -1. = Linear chordwise fairing. 0. = Second order fairing.
4	8-14	YES	XM	Basic Mach number for this case.
4	15-21	YES	TNOM	Number of additional Mach numbers. TNOM $\leq$ 5.
4	22-28	YES	DONT	Wing data printout code. 0. = Minimal printout. 2. = Thickness pressure coefficients at each grid element in the wing calculations will be printed. 101. = Velocity potential will also be printed.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	29-35	YES	TNON	Number of semi-span element rows in wing calculations. TNON ≤ 40. If blank, TNON set to 40.
4	36-42	YES	TJBYMX	Number of spanwise stations at which wing thickness pressures are calculated. TJBYMX ≤ 24. Leave blank if TNON not specified.
4	43-49	YES	TNCUT	Number of body stations at which pressure coefficients are calculated (≤ 60). If blank, TNCUT set to 50.
5	1-7	YES	ANYBOD	Wing Y dimension at inboard edge. If negative, program will solve for wing-fuselage intersection.
5	8-14	YES	WRAP	Nacelle pressure field code. -1. = Wrap solution for nacelle pressure field is desired. 1. = Glance solution is performed.
5	15-21	YES	DLT2	Interference printout code. -1. = Summary table printout only. 1. = Details of nacelle/fuselage interference calculations will be printed.
5	22-28	YES	BCUT	Number of divisions of nacelles used to define nacelle pressures and Whitham F(Y) function. BCUT ≤ 40. If blank, BCUT set to 40.
6	1-35	YES	TXM	Array of additional Mach numbers. Solution will be performed for these Mach numbers after the solution for XM.
		Note:		There will be a total of TNOM values on the card. Omit this card if TNOM = 0.
7	1-70	YES	TYB2	Array of semi-span values of element row at which wing thickness pressures are calculated.
		Note:		These values should be whole numbers beginning with 0. and ending with TNON. Up to ten values per card. Up to three cards. Omit these cards if TJBYMX was not specified.
8	1-3			END

4.7 Wing Design Program

The wing design program principally requires a wing planform (supplied from the basic geometry), a description of the loadings to be used in optimizing the wing shape, and specification of the design point and constraints to be applied to the solution.

Punch all data, with decimals, to the left in the card columns (10 field -7 digit format).

Default options are provided to help keep input simple. These include:

● TLOADS	This is the number of loadings to be used in finding an optimum loading combination. If input as a positive number, the specified number of loadings will be taken, in order, from the table on page 62. (A negative sign requires the user to list the loading numbers to be used.)
● XOCNUM	This is the number of percent chords used in printing the camber surface output. If input as -12.0, standard percent chords are used.
● TJBYMX	Standard semi-span stations are provided if TJBYMX = 0.

If program options are used that require wing thickness pressures, nacelle buoyancy field, fuselage upwash loading, or asymmetric fuselage loading, it is necessary to have previously run the near-field wave drag or lift analysis programs to load the proper tables. This is done as follows:

● Nacelle buoyancy loading	May be calculated by either wing analysis program or near-field wave drag program.
● Wing thickness pressures	Obtained from near-field wave drag program.
● Fuselage upwash loading	Obtained by running lift analysis program with wing slopes zeroed (WHUP = 1.0).
● Asymmetric fuselage loading	Obtained from lift analysis program with SYMM = 1.0.

The most efficient way to obtain all of the configuration dependent data is to first run the near-field wave drag program,

WING DESIGN LOADINGS

Loading Number	Definition
1.	Uniform
2.	Proportional to x, the distance from the leading edge
3.	Proportional to y, the distance from the wing centerline
4.	Proportional to y^2
5.	Proportional to x^2
6.	Proportional to $x(c - x)$, where c is local chord
7.	Proportional to $x^2\ (1.5\,c - x)$
8.	Proportional to $2\ (1 + 15\ \frac{x}{c})^{-0.5}$
9.	Proportional to $(c - x)^{0.5}$
10.	Elliptical spanwise, proportional to $\sqrt{(1 - y/\frac{b}{2})}$
11.	Proportional to x, the distance from the leading edge of an arbitrarily defined region
12.	A camber-induced loading proportional to the body bouyancy loading
13.	A camber-induced loading proportional to the body upwash loading
14.	A camber-induced loading proportional to the nacelle buoyancy loading
15.	The body bouyancy loading
16.	The body upwash loading
17.	The nacelle buoyancy loading

without nacelles, to get the wing thickness pressures. Then run the lift analysis program, with nacelles, and with the zero slope option (WHUP = 1.0) and asymmetric fuselage option (SYMM ≠0.).

The fuselage upwash loading will be that obtained with the fuselage at input incidence. If the upwash fields corresponding to a series of fuselage angles of attack are desired, it will be necessary to change the fuselage definition and rerun the lift analysis program to produce each upwash pressure loading.

CAUTION

The loading options must be used with some care. Loadings 12-14 cannot be used without also using loadings 15-17. Loading 11 cannot be used without specifying a corresponding planform region (ANOARB>0). If all loadings are requested, the resultant optimum combination of loadings (and camber shape) may be physically unrealistic if no constraints on upper surface pressure coefficient are imposed.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			WDEZ
2	1-70			Any desired TITLE information
3	1-7	YES	AJ3	Nacelle input code 0. = Ignore nacelle definition 1. = Include nacelle definition (required if loading 17 is used)
3	8-14	YES	TNON	Numbers of semispan elements in wing grid system. 2. ≤ TNON ≤ 50. If blank, TNON set to 40.
3	15-21	YES	TJBYMX	Number of semispan stations at which camber surface is calculated. 2. ≤ TJBYMX ≤ 25.
3	22-28	YES	TIFAF	Flat plate calculation code -1. = Use data from previous case. 0. = Flat plate calculation will be made. 1. = Flat plate calculations will not be made. (Card 9 must be input).

Card Number	Card Column	Decimal Required	Variable Name	Description
4	1-7	YES	APRINT	Printed output code. -2. = Summary output printed. -1. = Input data (except large tables) and summary output printed. 0. = Input data, output summary and camber shapes at design condition, if requested, are printed. 1. = Same as APRINT = 0., plus some diagnostic data. 2. = All input, output and diagnostic data printed.
4	8-14	YES	SMOOTH	Code to determine smoothing procedure applied to camber surface longitudinal slope at each span station. 0. = No smoothing performed. 1. = Smooth-as-you-go technique used. 3. = Three point smoothing technique used.
4	15-21	YES	RESTART	Code to determine disposition of force and moment coefficients for component and interference loadings. -1. = Data from previous case will be used. 0. = Data will be calculated by program for use in current case and subsequent cases. 1. = Data will be calculated, and also punched on cards. 2. = Data are read from card sets 17 through 19. 3. = Data are read from tape 3 (written by previous case)
4	22-28	YES	YSNOOT	Y value for parabolic apex tangent to wing leading edge. (Leave blank if not used.)
5	1-7	YES	XM	Basic Mach number
5	8-14	YES	CMO	Design value of pitching moment coefficient at zero lift.
5	15-21	YES	CLDZIN	Value of design lift coefficient. If blank or zero, CLDZIN set to 1.0.

Card Number	Card Column	Decimal Required	Variable Name	Description
5	22-28	YES	TLOADS	Number of loadings to be combined $2. \leq$ TLOADS ≤ 17 TLOADS < 0.= Loading numbers will be input on card(s) 10. Loading numbers will be taken from table on page 62. TLOADS > 0. = Loadings will be in the order tabulated on page 62. E.g., if TLOADS = 3.0, first 3 loadings from page 62 will be used.
5	29-35	YES	XOCNUM	Number of chordwise locations at which camber ordinates will be printed, corresponding to options selected on card 7. (XOCNUM) $\leq$ 20. -12. = Default locations of 0., 5., 10., 20., 30., ... 90., 100. as used. Omit card 11. + = Values in percent of local chord will be input (card 11).
5	36-42	YES	ANOARB	Numbers of points on cards 12 and 13 used to define the arbitrary region of the wing planform for loading number 11. ANOARB $\leq$ 20. If blank, cards 12 and 13 not read.
6	1-7	YES	AXCPLIM	Number of chordwise locations (card set 14) used to specify wing upper surface limiting pressures. AXCPLIM $\leq$ 15. - = Use values from previous case if /AXCPLIM/ same as previous case. 0. = Card sets 14, 15 and 16 not read. +. = Card set 14, 15 and 16 are read.
6	8-14	YES	AYCPLIM	Number of spanwise stations (card set 16) used to specify wing upper surface limiting pressures. Needed only if AXCPLIM > 0. AYCPLIM $\leq$ 15.
6	15-21	YES	TXCPT	Code to request use of wing thickness pressures in pressure limiting calculations. 0. = Wing thickness pressures not used. 1. = Wing thickness pressures used.

Solution and Constraint Options

Card 7 contains four inputs which control the extent of the solution and the constraints to be applied. Each of the 4 inputs may take on 4 different values, as follows:

0. No solution of this type desired.

1. Calculate pressure distribution, drag, and pitching moment for optimum combination of loadings.

2. Same as 1, plus also calculate the wing shape required to support the optimum pressure distribution.

3. Same as 2, plus also punch the wing shape on cards. Order is percent chords for ordinates, percent span stations, and then the ordinates in percent chord. 10F7.3 format. (May be input directly into wing analysis program with TIFZC = 1.0).

Card Number	Card Column	Decimal Required	Variable Name	Description
7	1-7	YES	CONSTR(1)	Obtain solution for minimum drag with constraint on C_L only.
7	8-14	YES	CONSTR(2)	Obtain solution with constraints on C_L and C_{mo} (requires C_{mo} value on card 5).
7	15-21	YES	CONSTR(3)	Obtain solution with constraint on C_L and pressure limiting on wing upper surface.
7	22-28	YES	CONSTR(4)	Obtain solution with constraint on C_L and C_{mo}, plus pressure limiting on wing upper surface.

Card Number	Card Column	Decimal Required	Variable Name	Description
8	1-70	YES	TJBYS	Array of semispan stations at which the camber surface is calculated.
	Note:	Up to ten values per card. There will be a total of TJBYMX whole numbers which must begin with 0.0 and end with TNON. If TJBYMX was blank, the following values are used: 0., 4., 8., 12., 16., 20., 24., 38., 32., 36., 40.		
9	1-7	YES	XF	X coordinate of wing aerodynamic center.
9	8-14	YES	SCL9	Flat wing lift-curve slope (per degree), based on the reference area for force and moment coefficients.
9	15-21	YES	DF	Flat wing lift-dependent drag factor.
9	22-28	YES	AREA9	Planform area in program units.
	Note:	Omit this card if TIFAF (card 4) ≤ 0. The data on card 9 would normally be calculated by a previous run of the same planform at the same Mach number.		
10	1-70	YES	TLOAD	Loading numbers for use in pressure optimization. Integer numbers from 1.0 to 17.0, TLOADS (see card 5) in number, and in arbitrary order. Up to 10 values per card. Omit card(s) 10 if TLOADS > 0.
11	1-70	YES	TPCT	Array of X/C (percent of local chord) values will be interpolated at each span station. Omit card(s) 11 if XOCNUM = -12.
12	1-70	YES	YARB	Array of Y coordinates which define an arbitrary planform region for loading number 11.
	Note:	Up to ten values per card. Up to two cards. There will be a total of ANOARB values. If ANOARB (card 5) is blank, omit card set 12.		
13	1-70	YES	XARB	Array of X coordinates which define an arbitrary planform region for loading number 11.
	Note:	Up to ten values per card. Up to two cards. There will be a total of ANOARB values. If ANOARB (card 5) is blank, omit card set 13.		

Card Number	Card Column	Decimal Required	Variable Name	Description
14	1-70	YES	XCPLIM	Array of chordwise locations (percent of local chord) used to define the wing upper surface limiting pressure coefficient. Needed if CONSTR(3) or (4) is $\neq 0$ on card 7.

Note: Up to ten values per card.
There will be a total of AXCPLIM values starting with 0. and ending with 100. If AXCPLIM is not positive, omit card set 14.

Card Number	Card Column	Decimal Required	Variable Name	Description
15	1-70	YES	YCPLIM	Array of spanwise locations (percent or semispan) used to define the wing upper surface limiting pressure coefficient.

Note: Up to ten values per card.
There will be a total of AYCPLIM values starting with 0. and ending with 100.
If AXCPLIM is not positive, omit card set 15.

Card Number	Card Column	Decimal Required	Variable Name	Description
16	1-70	YES	CPLIMIT	Array of limiting pressure coefficients on the wing upper surface. All coefficients at a given semispan are input in the same order as XCPLIM. Begin each semispan set on a new card and in the same order as YCPLIM.

Note: Up to ten values per card.
There will be a total of AXCPLIM X AYCPLIM values.
If AXCPLIM is not positive, omit card set 16.

Card Number	Card Column	Decimal Required	Variable Name	Description
*17	1-80		TITLE	Title card of RESTART data.
*18	1-80	YES	RESTRT	Array of force and moment coefficients for component and interference loading, as punched from a previous run, for restarting program execution.
*19	1-60	NO	IRESTRT	Array of elements per semispan station used in integration process as punched from previous run for restarting program execution.

Note: Omit cards 17-19 if RESTART (card 4) is not equal to 2.0.

Card Number	Card Column	Decimal Required	Variable Name	Description
20	1-3			END

*The restart card sets 17-19 are printed on the Output file and identified by the statement: RESTART DATA PUNCHED, DECK IMAGE FOLLOWS.

4.8 Lift Analysis Program

Codes on cards 3 and 4 control the inclusion of basic geometry data as requested. Input is in 10 field -7 digit format.

Note that the wing camber surface may be defined in several ways, controlled by input TIFZC on card 4:

TIFZC	
0. or 1.	Input to lift analysis program on cards
2.	Flat wing (Z = 0 everywhere)
3.	As defined by wing design program (which must have been run previously).
4.	As defined in basic geometry.

The wing camber surface input to the lift analysis program will automatically be used to update the basic geometry definition if TIFZC = 0. or 1.

By definition, a canard is required to be located forward of the wing, and a horizontal tail aft of the wing. One each is allowed, and they may both be input at the same time.

If the pressure limiting feature (controlled by FLIMIT on card 4) is used, it requires the wing thickness pressures from the near-field wave drag program, which must have been run previously at the same Mach number.

All angles are input to the program in degrees.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			ANLZ
2	1-70			Any desired TITLE information.
3	1-7	YES	AJ2	Fuselage input code. 0. = Ignore fuselage definition. 1. = Include fuselage definition.

Card Number	Card Column	Decimal Required	Variable Name	Description
3	8-14	YES	AJ3	Nacelle input code. 0. = Ignore nacelle definitions. 1. = Include nacelle definitions.
3	15-21	YES	AJ5	Canard input code. 0. = Ignore canard definition. 1. = Include canard definition.
3	22-28	YES	AJ7	Horizontal tail input code. 0. = Ignore horizontal tail definition. 1. = Include horizontal tail definition.
4	1-7	YES	TJBYMX	Number of spanwise stations defining camber surface. TJBYMX ≤ 20.
4	8-14	YES	TNOPCT	Number of percent chords defining each spanwise station. TNOPCT ≤ 20.
4	15-21	YES	TIFZC	Code for camber surface ordinate. 0. = Z is input. 1. = Z/C (percent) is input. 2. = Flat wing option (Z = 0). 3. = Camber surface is defined in common block /CAMBER/. 4. = Use definition contained in basic geometry.
	Note:	If TIFZC is 2., 3., or 4., inputs TJBYMX, TNOPCT, TPCT, TYB2 and WZORD are not required.		
4	22-28	YES	TNOM	Number of Mach numbers in addition to basic Mach number XM. TNOM ≤ 5.
4	29-35	YES	FNON	Number of semi-span rows in wing grid system. FNON ≤ 40. If left blank, will be set to 40.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	36-42	YES	FLIMIT	Limiting pressure feature code. 0 = feature not desired. FLIMIT = number of configuration angles of attack for solution using pressure limiting.
5	1-7	YES	TNFLAP	Number of trailing edge flaps on right hand wing. TNFLAP ≤ 5.
5	8-14	YES	TNTWST	Number of values (Y in percent, and angle) to define wing twist. Relative to input wing shape. TNTWST ≤ 40.
5	15-21	YES	TNALP	Number of canard angles of attack (≤ 5). Not required if AJ5 = 0.
5	22-28	YES	WRAP	Code for nacelle pressure field solution -1. = wrap 1. = glance
5	29-35	YES	OXML	Mach number input code for nacelle pressure field calculations. 0. = Free stream Mach number used. 1. = Mach number input on card 19.
5	36-42	YES	DLT2	Nacelle pressure field calculation printout code. -1. = summary only 1. = detailed printout
5	43-49	YES	BCUT	Number of cuts used to define pressure signature from nacelles. If blank, will be set to 40. BCUT ≤ 40.
6	1-7	YES	ANYBOD	Wing/fuselage intersection Y value. If negative, solve for intersection. If ANYBOD = -10.0, intersection will be input on Card Sets 14-16.
6	8-14	YES	THALP	Number of horizontal tail angles of attack. THALP ≤ 10. Not required if AJ7=0.

Card Number	Card Column	Decimal Required	Variable Name	Description
6	15-21	YES	SYMM	Asymmetric body volume term calculation code. 0. = Do not calculate 1. = Calculate 2. = Calculate using area distribution input on Card Sets 17 and 18.
6	22-28	YES	SMOGO	Smoothing code. 0. = Use 9 term smoothing 1. = Use smoothing-as-computed pressure calculation.
6	29-35	YES	WHUP	Wing slope control code. 0. = Wing slopes calculated from input camber surface. 1. = Wing slopes = 0. (used for fuselage upwash field).
7	1-7	YES	XM	Basic Mach number for case.
7	8-14	YES	TZSKAL	Scale factor for input Z ordinates. If blank, no scaling performed.
7	15-21	YES	CLIN(1)	Number of lift coefficients input for first Mach number (XM) at which the combined flat plate and camber pressure coefficients will be computed. (CLIN(1) ≤5.)
7	22-28	YES	CLIN(4)	Same as CLIN(1) for fourth Mach number (TMACH(3)).
7	29-35	YES	CLIN(5)	Same as CLIN(1) but for fifth Mach number (TMACH(4)).
7	36-42	YES	CLIN(6)	Same as CLIN(1) but for sixth Mach number (TMACH(5)).
7	43-49	YES	CLIN(5)	Same as CLIN(1) but for fifth Mach number (TMACH(4)).
7	50-56	YES	CLIN(6)	Same as CLIN(1) but for sixth Mach number (TMACH(5)).
8	1-35	YES	TMACH	Array of additional Mach numbers for this case. TNOM values. Omit this card if TNOM = 0.

Wing Camber Surface Definition

Card Number	Card Column	Decimal Required	Variable Name	Description

Omit card sets 9, 10 and 11 if TIFZC = 2., 3., or 4.

Card Number	Card Column	Decimal Required	Variable Name	Description
9	1-70	YES	TPCT	Array of chord percentages at which Z (or Z/C) ordinates are input and pressure coefficients are evaluated and output.

Note: Up to ten values per card. Up to two cards. There will be a total of TNOPCT values from 0. through 100.

Card Number	Card Column	Decimal Required	Variable Name	Description
10	1-70	YES	TYB2	Array of semi-span percentages at which Z (or Z/C) ordinates are input.

Note: Up to ten values per card. There will be a total of TJBYMX values from 0. through 100.

Card Number	Card Column	Decimal Required	Variable Name	Description
11	1-70	YES	WZORD	Array of Z (or Z/C) ordinates of the right hand wing camber definition. All ordinates at a given semi-span are input in the same order as TPCT. Begin each semi-span percent on a new card and in the same order as TYB2.

Note: Up to ten values per card. There will be a total of TPCT x TYB2 values.

Wing Twist Definition

Omit cards 12 and 13 if TNTWST = 0.

Card Number	Card Column	Decimal Required	Variable Name	Description
12	1-70	YES	YTWIST	Array of semi-span percentages at which wing twist angles are input.

Note: Up to ten values per card. Up to four cards. TNTWST values.

Card Number	Card Column	Decimal Required	Variable Name	Description
13	1-70	YES	ATWIST	Array of twist angles, in degrees, corresponding to YTWIST. A positive angle means an increase in local angle of attack. Linear interpolation is used for points between input points.

Note: Up to ten values per card. Up to four cards. TNTWST values.

Wing-Fuselage Intersection

Omit cards 14-16 if ANYBOD $\neq$ -10.

Card Number	Card Column	Decimal Required	Variable Name	Description
14	1-70	YES	WX	X values
15	1-70	YES	WY	Y values
16	1-70	YES	WZ	Z values

Input X array defining wing-fuselage intersection, then Y and Z. Start each array on a new card. Values are input at the percent chords of the camber surface definition (Card 9), or basic geometry definition (if WZORD not input).

Asymmetric Fuselage Area Input

Omit cards 17 and 18 if SYMM $\neq$ 2.0

Card Number	Card Column	Decimal Required	Variable Name	Description
17	1-70	YES	AOVR	above-wing area
18	1-70	YES	AUND	under-wing area

Input area distribution above wing, then below. Start each array on a new card. Values are input at the percent chords of the camber surface definition (Card 9), or basic geometry definition (if WZORD not input).

Alternate Mach Nos. For Nacelle Pressure Field Calculations

Omit card 19 if OXML = 0.

Card Number	Card Column	Decimal Required	Variable Name	Description
19	1-42	YES	TMLOC	Array of local Mach numbers for nacelle pressure field calculations. First value corresponds to XM, successive values correspond to TMACH (if included).

Note: Up to six values on the card.
There will be a total of TNOM + 1. values.

Wing Flap Definition

Omit cards 20 if TNFLAP = 0.

Card Number	Card Column	Decimal Required	Variable Name	Description
20	1-7	YES	X1	Inboard X value of flap leading edge.
20	8-14	YES	Y1	Inboard Y value of flap leading edge.
20	15-21	YES	XO	Outboard X value of flap leading edge.
20	22-28	YES	YO	Outboard Y value of flap leading edge.
20	29-35	YES	DEFLAP	Flap deflection in degrees. A positive angle means the flap trailing edge is deflected downward.

Note: There will be a total of TNFLAP cards, one for each flap.

Card Number	Card Column	Decimal Required	Variable Name	Description
21	1-35	YES	TCA	Array of canard angles of attack. A positive angle means the leading edge is rotated upward.

Note: There will be a total of TNALP values on the card. Omit this card if TNALP = 0. or AJ5 = 0.

Card Number	Card Column	Decimal Required	Variable Name	Description
22	1-63	YES	THA	Array of horizontal tail angles of attack. A positive angle means the leading edge is rotated upward.

Note: There will be a total of THALP values on the card. Omit this card if THALP = 0. or AJ7 = 0.

Card Number	Card Column	Decimal Required	Variable Name	Description
23	1-7	YES	VACFR	Fraction of vacuum pressure coefficient for pressure limiting.
24	1-35	YES	TLALP	Array of α's for limiting pressure coefficient.

Note: There will be a total of FLIMIT values on the card. Omit cards 23 and 24 if FLIMIT = 0.

Card Number	Card Column	Decimal Required	Variable Name	Description
25	1-35	YES	CLINP	Arrays of lift coefficients for the input Mach numbers (XM and TMACH) at which the combined flat plate and camber pressure coefficients are computed. C_L's for each Mach number begin on a new card.

Note: Up to five values per card. Up to six cards.
The number of values on each card will correspond with CLIN(I) on card 7.
If CLIN(I) = 0, omit the Ith card.

For a new case, input cards 2 through 25 at this place in the data deck.

26	1-3			END

5.0 TYPICAL CASE AND PROGRAM OUTPUT

A typical design and analysis case and associated program output are presented in this section. Given a configuration consisting of wing, fuselage, nacelles and horizontal tail, the following are obtained:

- Wing design at Mach number = 2.7 for C_L = .10 and C_{mo} = .015, in presence of fuselage and nacelles with pressure limiting at .7 vacuum.
- Analysis of configuration drag-due-to-lift for a series of horizontal tail settings.
- Skin friction drag
- Far-field and near-field wave drag analyses
- Drawing of configuration.

The input card listing for this case is shown on page 83.

The program output has been edited to reduce page count while illustrating output format.

The output begins with a listing of the basic geometry, separated into components (wing, fuselage, etc). An uncambered wing was specified in the basic geometry, since the camber surface will be defined by the wing design program.

Configuration-Dependent Loadings

Since the wing design case is to be performed with pressure limiting, and in the presence of fuselage and nacelles, the corresponding pressure arrays must be computed. The near-field wave drag program is run first, to generate the wing thickness pressure data (page 90). Only the wing geometry is required for this calculation; output for the complete wing-fuselage-nacelle configuration from the near-field program is illustrated later (page 81).

The lift analysis program is executed next, to calculate the nacelle pressure field and the fuselage upwash pressure field. To obtain an approximate orientation between the fuselage and wing for the upwash field calculations, a previously defined camber surface was input using the TIFØC = 1.0 option. The ANLØ interface program inserts this definition into the basic geometry and prints it (page 91). The lift analysis program then computes the wing upwash field (page 96), the nacelle pressure field (page 97), and the loading on the wing due to the fuselage upwash field (page 101). The wing upwash loading is that for the basic wing angle of attack with all wing slopes zeroed, i.e., as computed with input WHUP = 1.0.

Wing Design Solution

Much diagnostic output is available from the wing design module. However, print controls are used in the program (input APRINT) to provide output flexibility. In the typical case shown, the print control was set at +1.0, to illustrate output format. The design case shown utilized seven total loadings, including those due to fuselage upwash and nacelles.

The wing design program first prints the input data and checks the design and constraint options (the card 7 inputs) for consistency. The semi-span stations, in program units, at which the camber surface will be calculated is next printed, followed by a listing of the component loadings to be used and the chordwise locations at which the camber surface will be interpolated. Tables of the configuration dependent loadings are also output.

The program next computes and prints the flat wing solution (page 112). This includes lift and drag coefficients, the lengthwise center of pressure position (as a fraction of overall wing length), the pitching moment derivative (dC_m/dC_L), and the drag-due-to-lift factor.

The program then cycles through all the component loadings. For each, a table giving spanwise distributions of lift, drag, and pitching moment coefficients is printed. This is followed by the integrated values of lift coefficient, drag coefficient, center of pressure position, drag-due-to-lift factor, the ratio of input reference area to gross planform area (S_{ref}/S_{prog}), the pitching moment slope with design C_L, and the C_{mo} associated with the component C_L. This is followed by the interference drag of the component loading on the nacelle area distribution (if nacelles were input), and a tabulation of the interference drag coefficients associated with all other component loadings. The camber surface for the selected loading is not printed, but it would have been had APRINT = 2.0 been input.

The program next summarizes the force and interference drag coefficients of all the component loadings (page 120), and writes the RESTART data deck (if requested). Only a portion of the RESTART listing is shown since it is quite long, consisting of the matrix values and all configuration-dependent pressure arrays.

With all component loading data defined, the program then solves for the wing designs requested on card 7. The solution constraints are identified in the title block, followed by (if APRINT is 1.0 or 2.0) the optimization matrix. A check of the solution accuracy is made by multiplying the solution matrix by the left-hand side matrix (which should equal the right-hand side matrix).

The solution matrix tables are followed by a table of the component loadings combination, i.e., the product of A_iC_{Li} for the different basic loadings. These products give the contributions to total wing C_L for each of the component loadings. The solution C_{mo}, lift coefficient, and drag coefficient are also printed. The resultant wing upper surface pressure distribution (C_{pu}) is then searched for the minimum difference between C_{pu} and the limit C_p, and this minimum is printed, together with its planform location.

The bucket plot of drag-due-to-lift factor (K_E) versus C_{mo} for the optimum wing designs of the component loadings is next printed (page 123). Finally, if the camber surface shape corresponding to the selected design point was requested, the camber surface is printed as Z/C (in per cent) versus X/C at the solution spanwise stations.

This process is repeated for all other design options, except for the bucket plot, which is printed only once. If a pressure constraint condition is encountered which required another term to be added to the solution matrix, a note to this effect is printed and the optimization is performed again. (If the pressure constraint condition cannot be satisfied with a maximum size matrix, the solution stops and a failure message is printed.)

For the test case shown, one constraint on pressure coefficient was required to satisfy the design solution, with or without the C_{mo} constraint. (The pressure limit was exceeded at 70 per cent semi-span and 0 per cent chord, as noted). The force coefficients and camber surface for the design point wing were calculated and printed, as requested, on page 127.

Wing Camber Surface Update

In the illustrative case, the final camber surface design was used to update the basic geometry by means of the executive card WGUP. The updated definition is printed on page 130.

Lift Analysis

Given the basic geometry definition and the camber surface obtained by the design program, the lift analysis program was used to calculate the lifting pressure solutions for the complete configuration, both tail-off and tail-on at a series of horizontal tail settings.

The lift analysis program output consists of the input, the wing-fuselage intersection definition, fuselage upwash data (upwash in degrees), fuselage buoyancy field, the nacelle pressure field definition, camber surface data and the wing lifting pressure coefficients. These are summed over the configuration to obtain lift, drag, and pitching moment data. The fuselage force coefficients are printed both with and without wing downwash effects included (page 149).

The force coefficient summary, tail-off, is shown on page 150. The program first prints a table of lift, drag, and pitching moment coefficients for the wing at the input incidence, and also per degree angle of attack (FP at 1 degree). The increments due to the nacelles are also printed. This table is then repeated with the fuselage contribution added. The drag terms are then combined into two equations (nacelles on and off), and drag and pitching moment coefficients tabulated for a series of lift coefficients.

The configuration streamwise lift distribution is next summed and printed and further broken into separate summations for wing-fuselage-canard, nacelles, and horizontal tail. These summations are cumulative and are divided by the total lift of the configuration.

The force coefficient and streamwise lift distribution data are repeated for each tail angle of attack, together with the contributions due to the horizontal tail.

The spanwise lift distribution is printed last (page 158). This tabulation is for the wing-canard-nacelles combination only (excluding fuselage or horizontal tail).

If the limiting pressure option of the lift analysis program is requested, the output is the same except for two alterations:

1. The data at the configuration basic angle of attack become data at a specified angle of attack.

2. Notes are printed to call attention to the pressure limiting option.

Addition of a canard to the configuration produces an additional set of force coefficient summary data, i.e., data is printed both with and without the direct canard contribution.

Skin-Friction

The skin friction program prints input, then a table of wetted areas, drag/dynamic pressure (D/q), and drag coefficient, for each input flight condition (page 161).

Far-Field Wave Drag

The far-field wave drag program prints an enriched area distribution for the fuselage (page 162), then the area distribution for different configuration component buildups at a series of theta (cutting plane inclination) values. The program next identifies and prints the area restraint points corresponding to the case restraint condition, followed by configuration data for the input configuration and one optimized subject to the restraint points. An optimized fuselage area distribution corresponding to the restraint case is then calculated and printed, followed by a drag summary for the configuration as-input and with the optimized fuselage (page 168).

Near-Field Wave Drag

The near-field wave drag module, for wing-fuselage-nacelles, was executed next. The program input is first printed, followed by the wing fuselage intersection. (The Z values of this intersection are relative to the fuselage centerline, rather than the overall coordinate system.)

The nacelle terms are next printed. First the nacelle pressure field acting on the wing is output (edited out in this case since it is the same as previously illustrated in the lift analysis program output). The interference pressure signatures associated with the nacelles and fuselage acting on one another are next calculated and printed, including the "image" signatures associated with reflections off the wing surface.

The buoyancy field of the fuselage acting on the wing is then summarized, followed by the wing definition and isolated thickness pressure solutions.

The isolated fuselage pressure distribution and the wing-on-fuselage signature is next tabulated (page 179), together with a running summation of the drag associated with these pressures. Each of these sums is divided by the total corresponding drag value.

The final drag summary (page 183) consists of wing section data, tabulated fuselage and nacelle drag coefficients, total drag and wetted areas.

The wing section data, at the solution spanwise stations, consist of the isolated wing section drag coefficient (CDW/C= drag of the element row divided by chord), interference drag of fuselage on wing section (CDBØW/C), interference drag of nacelles acting on the section (CDNØW/C), the sum of those section coefficients (SUM CD/C), and the fraction of the total wing wave drag for the section.

Drag of the wing-fuselage combination is next printed, including the isolated wing (CDW), isolated fuselage (CDB), fuselage-on-wing interference (CDB/W), wing-on-fuselage interference (CDW/B), and the total of those (CD WING-BODY).

A table of nacelle drag terms is then printed, giving the isolated wave drag and the interference terms for the nacelles at each input origin.

The total wave drag for the configuration is printed as TØTAL CD.

Plot Program

The plot program prints the program input and view data. A typical drawing from the program is presented on page 12.

INPUT DECK LISTING

```
GEOM NEW                                                                                                  1
969-500A GEOMETRY DECK FOR SYSTEM CHECK CASE.                                                2/13/74      2
 1 -1 -1  1  0  1  0  8 13  1  9 19  0  0  0  0  0  0  2  7  0  0  1                                     33
9898.0  106.41  187.                                                                                      4
0.0       2.5       5.0       10.0      20.0      30.0      40.0      50.0      60.0      70.0            5-1
80.0      90.0      100.0                                                                                 5-2
76.59     4.757     0.0       165.93                                                                      6-1
93.104    6.625     0.        160.133                                                                     6-2
93.165    9.51      0.0       149.79                                                                      6-3
115.96    16.333    0.0       125.35                                                                      6-4
168.38    31.25     0.0       77.295                                                                      6-5
225.81    47.544    0.0       32.581                                                                      6-6
225.31    47.545    0.0       32.581                                                                      6-7
258.21    66.25     0.0       14.445                                                                      6-8
0.0       0.57      0.714     0.972     1.05      1.145     1.2       1.23      1.249     1.17            8-1-1
0.937     0.545     0.0                                                                                   8-1-2
0.0       0.57      0.714     0.872     1.05      1.145     1.2       1.23      1.249     1.17            8-2-1
0.937     0.546     0.0                                                                                   8-2-2
0.0       0.55      0.712     0.972     1.054     1.156     1.213     1.235     1.237     1.127           8-3-1
0.883     0.507     0.0                                                                                   8-3-2
0.0       0.55      0.715     0.876     1.126     1.174     1.235     1.25      1.229     1.087           8-4-1
0.94      0.474     0.0                                                                                   8-4-2
0.0       0.57      0.727     0.902     1.098     1.22      1.289     1.315     1.262     1.105           8-5-1
0.942     0.473     0.0                                                                                   8-5-2
0.0       0.58      0.729     0.911     1.134     1.268     1.343     1.375     1.32      1.155           8-6-1
0.88      0.495     0.0                                                                                   8-6-2
0.0       0.134     0.261     0.435     0.88      1.155     1.32      1.375     1.32      1.155           8-7-1
0.88      0.495     0.0                                                                                   8-7-2
0.0       0.134     0.261     0.491     0.88      1.155     1.285     1.375     1.32      1.155           8-8-1
0.88      0.495     0.0                                                                                   8-8-2
0.0       16.67     33.33     50.0      66.67     83.33     100.0     116.67    133.33    150.0           10-1
166.66    183.33    200.0     216.67    233.33    250.0     266.67    283.3     295.0                     10-2
10.0      8.55      7.10      5.64      4.17      2.73      1.28      -.14      -1.6      -3.04           11-1
-4.5      -5.9      -7.4      -8.85     -10.25    -11.7     -13.2     -14.6     -15.7                     11-2
0.0       23.5      57.5      89.0      117.0     125.0     119.8     108.0     105.0     107.0           12-1
107.0     105.0     102.0     94.0      73.0      59.0      33.0      8.0       0.0                       12-2
213.42    16.33               -5.8                                                                        16-1
0.0       2.008     15.47     21.525    28.017    32.067    35.04                                         17
2.865     2.983     3.633     3.77      3.654     3.42      3.42                                          18
218.67    31.25               -4.9                                                                        16-2
0.0       2.008     15.47     21.525    28.017    32.067    35.04                                         17
2.865     2.983     3.633     3.77      3.654     3.42      3.42                                          18
261.      2.0       -14.      25.       277.      11.       -14.      9.                                  22
0.        50.       100.                                                                                  23
0.        1.5       0.                                                                                    24
NEWD                                                                                                      1
WING THICKNESS PRESSURE GENERATION                                                                        2
0.        0.                                                                                              3
0.0       2.7       0.0                                                                                   4
4.76                -1.                                                                                   5
END                                                                                                       8
ANLZ                                                                                                      1
FUSELAGE UPWASH AND NACELLE PRESSURE FIELD LOADINGS.                                                      2
1.        1.                                                                                              3
12.       12.       1.                                                                                    4
                              -1.                 -1.                                                     5
4.76                                    1.0                                                               6
```

```
2.7                                                                         7
  0.000   5.000 10.000 20.000 30.000 40.000 50.000 60.000 70.000 80.000    9-1
 90.000100.000                                                              9-2
  0.000   5.000 10.000 20.000 30.000 40.000 50.000 60.000 70.000 80.000    10-1
 90.000100.000                                                              10-2
  0.000   -.573 -1.667 -4.135 -6.618 -8.925-10.956-12.677-14.019-14.957    11-1-1
-15.465-15.521                                                              11-1-2
  0.000   -.093  -.453 -1.478 -2.557 -3.896 -5.034 -6.213 -7.218 -8.082    11-2-1
 -8.781  -9.297                                                             11-2-2
  0.000    .066  -.144  -.857 -1.747 -2.715 -3.700 -4.662 -5.572 -6.406    11-3-1
 -7.143  -7.758                                                             11-3-2
  0.000    .086  -.006  -.425 -1.040 -1.754 -2.528 -3.327 -4.130 -4.918    11-4-1
 -5.676  -6.330                                                             11-4-2
  0.000    .232   .265   .020  -.410  -.958 -1.594 -2.258 -2.364 -3.685    11-5-1
 -4.410  -5.127                                                             11-5-2
  0.000    .146   .268   .180  -.106  -.519 -1.017 -1.576 -2.184 -2.825    11-6-1
 -3.489  -4.159                                                             11-6-2
  0.000    .281   .493   .561   .410   .149  -.211  -.647 -1.137 -1.669    11-7-1
 -2.239  -2.842                                                             11-7-2
  0.000    .074   .436   .688   .717   .594   .397   .082  -.265  -.669    11-8-1
 -1.114  -1.598                                                             11-8-2
  0.000    .280   .547  1.073  1.362  1.520  1.633  1.704  1.722  1.730    11-9-1
  1.704   1.647                                                             11-9-2
  0.000   -.360  -.638  -.850  -.994 -1.141 -1.349 -1.556 -1.750 -1.973    11-10-1
 -2.211  -2.456                                                             11-10-2
  0.000   -.336  -.655 -1.241 -1.750 -2.211 -2.557 -2.900 -3.264 -3.622    11-11-1
 -3.970  -4.308                                                             11-11-2
  0.000   -.339  -.653 -1.209 -1.666 -2.015 -2.238 -2.412 -2.542 -2.627    11-12-1
 -2.646  -2.597                                                             11-12-2
END                                                                         26
WDEZ                                                                        1
WING DESIGN    7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS              2
1.        40.       12.       0.                                            3
1.                  1.                                                      4
2.7       .015      .1        -7.       -12.                                5
2.        2.        1.                                                      6
1.        1.        1.        3.                                            7
0.0       2.0       4.0       8.0       12.0      16.0      20.0      24.0      28.0      32.0   8-1
36.0      40.0                                                              8-2
1.0       2.0       3.0       16.       17.       9.        7.              10
0.0       100.0                                                             14
0.0       100.0                                                             15
-0.137    -0.137                                                            16-1
-0.137    -0.137                                                            16-2
END                                                                         20
WGUP
ANLZ                                                                        1
ANALYSIS OF DRAG-DUE-TO-LIFT     M=2.7                                      2
1.        1.                  1.0                                           3
                    3.0                                                     4
0.                            -1.                 -1.                       5
4.76      5.0                                                               6
2.7                                                                         7
-2.       -1.       0.        1.        2.                                  22
END                                                                         26
SKFR
SKIN FRICTION CALCULATIONS  M=2.7, H=60000 FT                               2

    1    1    1         1                                                   3
    1                                                                       4
2.7       60.       0.        1.                                            5
END                                                                         10
FFWD                                                                        1
FAR-FIELD WAVE DRAG    OPTIMIZATION BASED ON MAX. AREA                      2
1.        1.        1.0                 1.0                                 3
    1     2.7       50.       36.                                           4
END                                                                         6
NFWD                                                                        1
NEAR FIELD WAVE DRAG                                                        2
1.        1.                                                                3
0.        2.7                                                               4
4.76      -1.       -1.                                                     5
END                                                                         8
PLOT                                                                        1
969-500 WING CAMBER DESIGN                                                  2
1.0       1.0       1.0       0.0       1.0                                 3
X Z                                                     10.   ORT           4-1
X Y                                                     10.   ORT           4-2
Y Z                                                     10.   ORT           4-3
END                                                                         5
```

**** **** **** **** **** WING **** **** **** **** ****

REFA = 9890.0000 CBAR = 106.4100 XBARIN = 107.0000

	XO = 76.5900			XO = 83.1040			XO = 93.1650		
	YO = 4.7570			YO = 6.6250			YO = 9.5100		
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000		
	CHORD = 166.8300			CHORD = 160.1330			CHORD = 149.7300		
PERCENT	CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS	
CHORD	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	0.0000	.5700	.5700	0.0000	.5700	.5700	0.0000	.5500	.5500
5.0	0.0000	.7140	.7140	0.0000	.7140	.7140	0.0000	.7120	.7120
10.0	0.0000	.8720	.8720	0.0000	.8720	.8720	0.0000	.8720	.8720
20.0	0.0000	1.0500	1.0500	0.0000	1.0500	1.0500	0.0000	1.0540	1.0540
30.0	0.0000	1.1450	1.1450	0.0000	1.1450	1.1450	0.0000	1.1560	1.1560
40.0	0.0000	1.2000	1.2000	0.0000	1.2000	1.2000	0.0000	1.2130	1.2130
50.0	0.0000	1.2300	1.2300	0.0000	1.2300	1.2300	0.0000	1.2350	1.2350
60.0	0.0000	1.2490	1.2490	0.0000	1.2490	1.2490	0.0000	1.2370	1.2370
70.0	0.0000	1.1700	1.1700	0.0000	1.1700	1.1700	0.0000	1.1270	1.1270
80.0	0.0000	.9370	.9370	0.0000	.9370	.9370	0.0000	.8830	.8830
90.0	0.0000	.5460	.5460	0.0000	.5460	.5460	0.0000	.5070	.5070
100.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000

	XO = 116.9600			XO = 168.9900			XO = 225.8100		
	YO = 15.3330			YO = 31.2300			YO = 47.5440		
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000		
	CHORD = 125.3590			CHORD = 77.2350			CHORD = 32.6510		
PERCENT	CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS	
CHORD	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	0.0000	.5500	.5500	0.0000	.5700	.5700	0.0000	.5800	.5800
5.0	0.0000	.7150	.7150	0.0000	.7270	.7270	0.0000	.7290	.7290
10.0	0.0000	.8760	.8760	0.0000	.9020	.9020	0.0000	.9110	.9110
20.0	0.0000	1.1260	1.1260	0.0000	1.0980	1.0980	0.0000	1.1340	1.1340
30.0	0.0000	1.1740	1.1740	0.0000	1.2200	1.2200	0.0000	1.2680	1.2680
40.0	0.0000	1.2350	1.2350	0.0000	1.2890	1.2890	0.0000	1.3430	1.3430
50.0	0.0000	1.2500	1.2500	0.0000	1.3150	1.3150	0.0000	1.3750	1.3750
60.0	0.0000	1.2290	1.2290	0.0000	1.2620	1.2620	0.0000	1.3200	1.3200
70.0	0.0000	1.0870	1.0870	0.0000	1.1050	1.1050	0.0000	1.1550	1.1550
80.0	0.0000	.8400	.8400	0.0000	.8420	.8420	0.0000	.8800	.8800
90.0	0.0000	.4740	.4740	0.0000	.4730	.4730	0.0000	.4950	.4950
100.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000

****** **** **** **** **** WING **** **** **** **** ******

XO	=	225.8100	XO	=	258.2100
YO	=	47.5450	YO	=	66.2500
ZO	=	0.0000	ZO	=	0.0000
CHORD	=	32.5810	CHORD	=	14.4450

PERCENT CHORD	CAMBER (Z)	HALF-THICKNESS UPPER	HALF-THICKNESS LOWER	CAMBER (Z)	HALF-THICKNESS UPPER	HALF-THICKNESS LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	0.0000	.1340	.1340	0.0000	.1340	.1340
5.0	0.0000	.2610	.2610	0.0000	.2610	.2510
10.0	0.0000	.4950	.4950	0.0000	.4910	.4910
20.0	0.0000	.8800	.8800	0.0000	.8800	.8800
30.0	0.0000	1.1550	1.1550	0.0000	1.1550	1.1550
40.0	0.0000	1.3200	1.3200	0.0000	1.2850	1.2850
50.0	0.0000	1.3750	1.3750	0.0000	1.3750	1.3750
60.0	0.0000	1.3200	1.3200	0.0000	1.3200	1.3200
70.0	0.0000	1.1550	1.1550	0.0000	1.1550	1.1550
80.0	0.0000	.8800	.8800	0.0000	.8800	.8800
90.0	0.0000	.4950	.4950	0.0000	.4950	.4950
100.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000

****** **** **** **** **** FUSELAGE **** **** **** **** ******

X CENTERLINE	Z CENTERLINE	RADIUS	AREA	PERIMETER
0.0000	10.0000	0.0000	0.0000	0.0000
16.5700	8.5500	2.7350	23.5000	17.1846
33.3300	7.1000	4.2782	57.5000	26.8806
50.0000	5.6400	5.3226	89.0000	33.4426
66.5700	4.1700	6.1026	117.0000	38.3440
83.3300	2.7300	6.3330	126.0000	39.7915
100.0000	1.2800	6.1752	119.8000	38.8001
116.5700	-.1400	5.8632	108.0000	36.8398
133.3300	-1.5000	5.7812	105.0000	36.3245
150.0000	-3.0400	5.8360	107.0000	36.6688
166.5500	-4.5000	5.8360	107.0000	36.5688
183.3300	-5.9000	5.8087	106.0000	36.4971
200.0000	-7.4000	5.6980	102.0000	35.8018
216.5700	-8.8500	5.4700	94.0000	34.3692
233.3300	-10.2500	5.0146	79.0000	31.5078
250.0000	-11.7000	4.3336	59.0000	27.2290
266.6700	-13.2000	3.2410	33.0000	20.3639
283.3000	-14.5000	1.5958	8.0000	10.0265
295.0000	-15.7000	0.0000	0.0000	0.0000

**** **** **** **** **** NACELLE **** **** **** **** ****

XO =	213.4200		XO =	218.6700
YO =	16.3300		YO =	31.2500
ZO =	-5.8000		ZO =	-4.9000
DO =	-5.8000		DO =	-4.9000

X	RADIUS	X	RADIUS
0.0000	2.8650	0.0000	2.8650
2.0080	2.9830	2.0080	2.9830
15.4700	3.6330	15.4700	3.6330
21.5250	3.7700	21.5250	3.7700
28.0170	3.6540	28.0170	3.6540
32.0670	3.4200	32.0670	3.4200
35.0400	3.4200	35.0400	3.4200

**** **** **** **** **** CANARD **** **** **** **** ****

XI = 261.0000
YI = 2.0000
ZI = -14.0000
CI = 25.0000
XO = 277.0000
YO = 11.0000
ZO = -14.0000
CO = 3.0000

PERCENT CHORD	UPPER ORD	LOWER ORD
0.00	0.00	0.00
50.00	1.50	1.50
100.00	0.00	0.00

WING THICKNESS PRESSURE GENERATION

MACH NO.= 2.70000 NON= 40 NOPCT= 13 JBYMAX= 20 RATIO= 4.15385

PLANFORM BREAKPOINTS

	X	Y	CHORD
1	76.5900	0.0000	166.8300
2	76.5900	4.7570	166.9300
3	83.1040	6.6250	160.1330
4	93.1650	9.5100	149.7900
5	116.9500	16.3330	125.3300
6	165.9800	31.2500	77.2350
7	225.8100	47.5440	32.6810
8	225.8100	47.5450	32.6810
9	258.2100	66.2500	14.4450

	XLE	XTE	Y
0	76.5900	243.4200	0.0000
1	76.5900	243.4200	1.6563
2	76.5900	243.4200	3.3125
3	77.3284	243.3993	4.9688
4	83.1040	243.2370	6.6250
5	88.8799	243.0751	8.2813
6	94.6559	242.9146	9.9375
7	100.4320	242.7580	11.5937
8	106.2081	242.6014	13.2500
9	111.9843	242.4449	14.9062
10	117.7603	242.3710	16.5525
11	123.5362	242.8112	18.2187
12	129.3120	243.2515	19.8750
13	135.0878	243.6917	21.5312
14	140.8637	244.1320	23.1875
15	146.6395	244.5722	24.8438
16	152.4153	245.0124	26.5000
17	158.1912	245.4527	28.1562
18	163.9670	245.8929	29.8125
19	169.7430	246.4390	31.4687
20	175.5196	247.6807	33.1250
21	181.2962	248.9225	34.7812
22	187.0729	250.1642	36.4375
23	192.8495	251.4059	38.0937
24	198.6262	252.6477	39.7500
25	204.4028	253.8894	41.4063
26	210.1795	255.1311	43.0625
27	215.9561	256.3728	44.7187
28	221.7328	257.6146	46.3750
29	226.6523	258.8592	48.0312
30	229.5211	260.1134	49.6875
31	232.3900	261.3675	51.3437
32	235.2589	262.6217	53.0000
33	238.1278	263.8759	54.6562
34	240.9967	265.1300	56.3125
35	243.8656	265.3842	57.9688
36	246.7345	267.6383	59.6250
37	249.6033	268.8925	61.2812
38	252.4722	270.1467	62.9375
39	255.3411	271.4008	64.5937
40	258.2100	272.6550	66.2500

INBOARD WING END DEFINITION

CHORD	X	Y	Z	T
0.00	76.600461	4.760000	0.000000	0.000000
2.50	80.770943	4.760000	0.000000	1.901739
5.00	84.941424	4.760000	0.000000	2.392179
10.00	93.282386	4.760000	0.000000	2.933328
20.00	109.964310	4.760000	0.000000	3.503204

30.00	126.646235	4.760000	0.000000	3.820161
40.00	143.328159	4.760000	0.000000	4.003662
50.00	160.010084	4.760000	0.000000	4.103753
60.00	176.592008	4.760000	0.000000	4.167145
70.00	193.373933	4.760000	0.000000	3.933570
80.00	210.055857	4.760000	0.000000	3.125193
90.00	226.737782	4.760000	0.000000	1.821666
100.00	243.419706	4.760000	0.000000	0.000000

TABLE OF INPUT Z/C ORDINATES

XPCT	0.000000	2.500000	5.000000	10.000000	20.000000	30.000000	40.000000	50.000000
	60.000000	70.000000	80.000000	90.000000	100.000000			
Y/B/2								
0.0000	0.000000	.570000	.714000	.872000	1.050000	1.145000	1.200000	1.230000
	1.249000	1.170000	.937000	.546000	0.000000			
.0718	0.000000	.570000	.714000	.872000	1.050000	1.145000	1.200000	1.230000
	1.249000	1.170000	.937000	.546000	0.000000			
.1000	0.000000	.570000	.714000	.872000	1.050000	1.145000	1.200000	1.230000
	1.249000	1.170000	.937000	.546000	0.000000			
.1435	0.000000	.550000	.712000	.872000	1.054000	1.155000	1.213000	1.235000
	1.237000	1.127000	.883000	.507000	0.000000			
.2465	0.000000	.550000	.715000	.876000	1.126000	1.174000	1.235000	1.250000
	1.229000	1.087000	.840000	.474000	0.000000			
.4717	0.000000	.570000	.727000	.302000	1.098000	1.220000	1.289000	1.315000
	1.262000	1.105000	.842000	.473000	0.000000			
.7176	0.000000	.580000	.729000	.911000	1.134000	1.268000	1.343000	1.375000
	1.320000	1.155000	.880000	.495000	0.000000			
.7177	0.000000	.134000	.261000	.495000	.880000	1.155000	1.320000	1.375000
	1.320000	1.155000	.880000	.495000	0.000000			
1.0000	0.000000	.134000	.261000	.491000	.880000	1.155000	1.285000	1.375000
	1.320000	1.155000	.880000	.495000	0.000000			

TABLE OF THICKNESS PRESSURE COEFFICIENT

XPCT	0.00	5.00	10.00	15.00	20.00	25.00	30.00	35.00	40.00	45.00	50.00	55.00
	60.00	65.00	70.00	75.00	80.00	85.00	90.00	95.00	100.00			
Y/B/2 /												
0.000	0.000000	.008208	.017194	.020513	.012988	.007425	.005438	.003264	.002663	.005482	.003488	.000479
	-.000100	-.001515	-.003906	-.004349	-.008065	-.013937	-.018165	-.022130	-.027107			
.025	.003384	.008515	.013914	.013905	.009557	.007861	.008343	.005986	.003147	.002164	.000875	.000549
	.001641	.000329	-.003097	-.006390	-.010492	-.014792	-.017603	-.021228	-.026055			
.050	.012141	.012961	.015735	.014073	.012224	.008119	.004613	.003518	.004264	.002316	.002561	.001351
	-.000865	-.002505	-.003729	-.007037	-.010502	-.014541	-.019393	-.024072	-.026998			
.075	.044193	.010107	.004035	.005410	.008955	.008058	.003985	.004022	.003647	.000982	.001429	.001289
	-.002097	-.004195	-.006313	-.011001	-.014695	-.017282	-.022324	-.026679	-.028459			
.100	.064049	.007115	-.006118	.004790	.007254	.004428	.002431	.002909	.001848	.001128	.001735	-.000430
	-.004020	-.006435	-.009853	-.014174	-.017639	-.020563	-.024990	-.028653	-.030494			
.125	.093723	.006411	-.005317	.003027	.004030	.003180	.001413	.001083	.001427	.000812	-.000060	-.001436
	-.003597	-.007547	-.012586	-.015899	-.018869	-.022399	-.026103	-.030207	-.032645			
.150	.133021	.004494	-.010407	.000092	.003698	.001997	-.000703	.000883	.000915	-.001088	-.001295	-.006437
	-.004801	-.010476	-.013770	-.015004	-.019718	-.024340	-.026617	-.029497	-.033011			
.200	.049035	-.005775	-.008818	.001025	-.001539	-.001172	.000763	-.000841	-.002589	-.001167	-.001164	-.004334
	-.008043	-.011358	-.015803	-.019553	-.021919	-.025428	-.028903	-.031633	-.033646			
.250	.040300	-.005285	-.011646	-.003849	-.005326	-.004686	-.002863	-.000826	-.001664	-.003324	-.003529	-.005053
	-.008474	-.013308	-.017513	-.020301	-.023333	-.027004	-.030665	-.033324	-.034423			
.300	.027563	-.006494	-.010754	-.009395	-.006239	-.006047	-.004647	-.002865	-.004094	-.001931	-.004918	-.007486
	-.011162	-.014583	-.017413	-.022979	-.026318	-.029964	-.031283	-.034498	-.037529			
.350	.049360	.003395	-.007744	-.014614	-.010288	-.008938	-.003613	-.003909	-.004396	-.005963	-.006340	-.008419
	-.011683	-.016364	-.021135	-.023984	-.027357	-.030841	-.035015	-.036348	-.038466			
.400	.041224	.001324	-.010040	-.012519	-.011536	-.010695	-.007434	-.004655	-.005275	-.005841	-.008291	-.011125

**** **** **** **** **** WING **** **** **** **** ****

REFA = 9898.0000 CBAR = 106.4100 XBARIN = 187.0000

	XO = 76.5900			XO = 83.1040			XO = 93.1650		
	YO = 4.7570			YO = 6.6250			YO = 9.5100		
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000		
	CHORD = 166.8300			CHORD = 160.1330			CHORD = 149.7300		
PERCENT	CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS	
CHORD	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	-.0207	.5700	.5700	.0528	.5700	.5700	.0554	.5500	.5500
5.0	-.0414	.7140	.7140	.1057	.7140	.7140	.1107	.7120	.7120
10.0	-.5267	.8720	.8720	-.2306	.8720	.8720	-.1337	.8720	.8720
20.0	-1.9889	1.0500	1.0500	-1.3723	1.0500	1.0500	-1.0272	1.0540	1.0540
30.0	-3.7290	1.1450	1.1450	-2.7975	1.1450	1.1450	-2.1970	1.1560	1.1560
40.0	-5.5612	1.2000	1.2000	-4.3476	1.2000	1.2000	-3.4361	1.2130	1.2130
50.0	-7.3761	1.2300	1.2300	-5.9249	1.2300	1.2300	-4.8463	1.2350	1.2350
60.0	-9.1005	1.2490	1.2490	-7.4654	1.2490	1.2490	-6.1905	1.2370	1.2370
70.0	-10.6815	1.1700	1.1700	-8.9226	1.1700	1.1700	-7.4901	1.1270	1.1270
80.0	-12.0769	.9370	.9370	-10.2581	.9370	.9370	-8.7121	.8830	.8830
90.0	-13.2491	.5460	.5460	-11.4383	.5460	.5460	-9.8285	.5070	.5070
100.0	-14.1710	0.0000	0.0000	-12.4391	0.0000	0.0000	-10.8175	0.0000	0.0000

	XO = 116.9600			XO = 168.9800			XO = 225.8100		
	YO = 15.3330			YO = 31.2500			YO = 47.5440		
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000		
	CHORD = 125.3500			CHORD = 77.2950			CHORD = 32.6810		
PERCENT	CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS	
CHORD	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	.0929	.5500	.5500	.0918	.5700	.5700	.0327	.5800	.5800
5.0	.1857	.7150	.7150	.1836	.7270	.7270	.0654	.7290	.7290
10.0	.1361	.8760	.8760	.3253	.9020	.9020	.1308	.9110	.9110
20.0	-.2993	1.1260	1.1260	.3374	1.0980	1.0980	.2760	1.1340	1.1340
30.0	-.9758	1.1740	1.1740	.1843	1.2200	1.2200	.3550	1.2680	1.2680
40.0	-1.7870	1.2350	1.2350	-.0589	1.2890	1.2890	.3941	1.3430	1.3430
50.0	-2.6833	1.2500	1.2500	-.3757	1.3150	1.3150	.4175	1.3750	1.3750
60.0	-3.6234	1.2290	1.2290	-.7478	1.2620	1.2620	.4284	1.3200	1.3200
70.0	-4.5833	1.0870	1.0870	-1.1606	1.1050	1.1050	.4244	1.1550	1.1550
80.0	-5.5402	.8400	.8400	-1.6038	.8420	.8420	.4160	.8800	.8800
90.0	-6.4773	.4740	.4740	-2.0728	.4730	.4730	.3968	.4950	.4950
100.0	-7.3782	0.0000	0.0000	-2.5630	0.0000	0.0000	.3581	0.0000	0.0000

**** **** **** **** **** WINGS **** **** **** **** ****

XO	= 225.8100		XO	= 258.2100	
YO	= 47.5450		YO	= 66.2500	
ZO	= 0.0000		ZO	= 0.0000	
CHORD	= 32.6810		CHORD	= 14.4450	

PERCENT CHORD	CAMBER (Z)	HALF-THICKNESS UPPER	HALF-THICKNESS LOWER	CAMBER (Z)	HALF-THICKNESS UPPER	HALF-THICKNESS LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	.0327	.1340	.1340	-.0245	.1340	.1340
5.0	.0653	.2610	.2610	-.0490	.2610	.2610
10.0	.1309	.4950	.4950	-.0943	.4910	.4910
20.0	.2759	.8800	.8800	-.1745	.8800	.8800
30.0	.3549	1.1550	1.1550	-.2407	1.1550	1.1550
40.0	.3939	1.3200	1.3200	-.2911	1.2850	1.2850
50.0	.4173	1.3750	1.3750	-.3233	1.3750	1.3750
60.0	.4263	1.3200	1.3200	-.3484	1.3200	1.3200
70.0	.4242	1.1550	1.1550	-.3672	1.1550	1.1550
80.0	.4158	.8800	.8800	-.3795	.8800	.8800
90.0	.3965	.4950	.4950	-.3822	.4950	.4950
100.0	.3679	0.0000	0.0000	-.3751	0.0000	0.0000

FUSELAGE UPWASH AND NACELLE PRESSURE FIELD LOADINGS.

MACH NO.= 2.70000   XMAX= 272.65500   NON= 40   CBAR= 105.41000   XBAR= 187.00000

TIFZC= 1.00   TNOM= 0.00   SYMM= -0.00   SMOGO= -0.00

NOPCT= 12   JBYMAX= 12   RATIO= 4.153856

	XPCT		YB2
1	0.000	1	0.000
2	5.000	2	5.000
3	10.000	3	10.000
4	20.000	4	20.000
5	30.000	5	30.000
6	40.000	6	40.000
7	50.000	7	50.000
8	60.000	8	60.000
9	70.000	9	70.000
10	80.000	10	80.000
11	90.000	11	90.000
12	100.000	12	100.000

PLANFORM BREAKPOINTS

	X	Y	Z	CHORD	AUX. CHORD
1	76.5900	0.0000	0.0000	166.8300	166.8300
2	76.5900	4.7570	0.0000	166.8300	166.8300
3	93.1040	6.6250	0.0000	160.1330	160.1330
4	93.1650	9.5100	0.0000	149.7900	149.7900
5	116.9600	16.3330	0.0000	125.3500	125.3500
6	169.9900	31.2500	0.0000	77.2950	77.2950
7	225.8100	47.5440	0.0000	32.6810	32.6810
8	225.8100	47.5450	0.0000	32.6810	32.6810
9	259.2100	66.2500	0.0000	14.4450	14.4450

	XLE	XTE	AUX XTE
0	76.5900	243.4200	243.4200
1	76.5900	243.4200	243.4200
2	76.5900	243.4200	243.4200
3	77.3284	243.3993	243.3993
4	93.1040	243.2370	243.2370
5	88.8799	243.0751	243.0751
6	94.6559	242.9146	242.9146
7	100.4320	242.7580	242.7580
8	106.2081	242.6014	242.6014
9	111.9843	242.4449	242.4449
10	117.7603	242.3710	242.3710
11	123.5362	242.8112	242.8112
12	129.3120	243.2515	243.2515
13	135.0878	243.6917	243.6917
14	140.8637	244.1320	244.1320
15	146.6395	244.5722	244.5722
16	152.4153	245.0124	245.0124
17	158.1912	245.4527	245.4527
18	163.9670	245.8929	245.8929
19	169.7430	246.4390	246.4390
20	175.5196	247.6907	247.6907
21	181.2962	248.9225	248.9225
22	187.0729	250.1642	250.1642
23	192.8495	251.4059	251.4059
24	198.6262	252.6477	252.6477
25	204.4028	253.8894	253.8894
26	210.1795	255.1311	255.1311
27	215.9561	256.3728	256.3728
28	221.7328	257.6146	257.6146
29	226.6523	258.8592	258.8592
30	229.5211	260.1134	260.1134
31	232.3900	261.3675	261.3675
32	235.2589	262.6217	262.6217
33	239.1278	263.8759	263.8759
34	240.9957	265.1300	265.1300
35	243.8656	266.3842	266.3842
36	246.7345	267.6383	267.6383
37	249.6033	268.8925	268.8925
38	252.4722	270.1467	270.1467
39	255.3411	271.4008	271.4008
40	258.2100	272.6550	272.6550

FUSELAGE DEFINITION

X	RAD	AREA	Z
0.00000	0.00000	0.00000	10.00000
16.67000	2.73501	23.50000	8.55000
33.33000	4.27919	57.50000	7.10000
50.00000	5.32255	89.00000	5.64000
66.67000	6.10264	117.00000	4.17000
83.33000	6.33301	126.00000	2.73000
100.00000	6.17523	119.90000	1.28000
116.67000	5.86323	108.00000	-.14000

133.33000	5.78122	105.00000	-1.60000
150.00000	5.83602	107.00000	-3.04000
166.66000	5.83602	107.00000	-4.50000
183.33000	5.80869	106.00000	-5.90000
200.00000	5.59804	102.00000	-7.40000
216.67000	5.47002	94.30000	-8.85000
233.33000	5.01463	79.00000	-10.25000
250.00000	4.33362	59.00000	-11.70000
266.67000	3.24102	33.00000	-13.20000
283.30000	1.59577	8.00000	-14.60000
295.00000	0.00000	0.00000	-15.70000

NACELLE GEOMETRY

ORIGIN (X,Y,Z)			X	RADIUS	AREA
213.42000	15.33000	-5.80000	0.00000	2.86500	25.78596
			2.00800	2.98300	27.95486
			15.47000	3.63300	41.46500
			21.52500	3.77000	44.65125
			28.01700	3.65400	41.94575
			32.06700	3.42000	36.74541
			35.04000	3.42000	36.74541

ORIGIN (X,Y,Z)			X	RADIUS	AREA
218.67000	31.25000	-4.90000	0.00000	2.86500	25.78596
			2.00800	2.98300	27.95486
			15.47000	3.63300	41.46500
			21.52500	3.77000	44.65125
			28.01700	3.65400	41.94575
			32.06700	3.42000	36.74541
			35.04000	3.42000	36.74541

WING SLOPES SET TO ZERO FOR UPWASH PRESSURE FIELD SOLUTION

TABLE OF INPUT Z/C ORDINATES

XPCT / Y/B/2	0.00	5.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.0000	0.00000	-.57300	-1.66700	-4.13500	-6.61800	-8.32500	-10.96600	-12.67700	-14.01900	-14.95700	-15.46500	-15.52100
.0500	0.00000	-.09300	-.45300	-1.47800	-2.66700	-3.89600	-5.09400	-6.21300	-7.21800	-8.08200	-8.78100	-9.29700
.1000	0.00000	.05600	-.14400	-.85700	-1.74700	-2.71500	-3.70000	-4.66200	-5.57200	-6.40600	-7.14300	-7.75800
.2000	0.00000	.08600	-.00600	-.42500	-1.04000	-1.75400	-2.52800	-3.32700	-4.13000	-4.91800	-5.67600	-6.39000
.3000	0.00000	.23200	.26500	.02000	-.41000	-.95800	-1.59400	-2.25800	-2.96400	-3.68500	-4.41000	-5.12700
.4000	0.00000	.14600	.26800	.18000	-.10600	-.51900	-1.01700	-1.57600	-2.18400	-2.82500	-3.48900	-4.15900
.5000	0.00000	.28100	.49300	.56100	.41000	.14900	-.21100	-.64700	-1.13700	-1.66300	-2.23900	-2.84200
.6000	0.00000	.07400	.43600	.68800	.71700	.59400	.38700	.08200	-.26500	-.56900	-1.11400	-1.59800
.7000	0.00000	.28000	.54700	1.07300	1.36200	1.52000	1.63300	1.70400	1.72200	1.73000	1.70400	1.64700
.8000	0.00000	-.36000	-.63800	-.85000	-.98400	-1.14100	-1.34800	-1.55600	-1.75000	-1.37300	-2.21100	-2.45600
.9000	0.00000	-.33600	-.65500	-1.24100	-1.75000	-2.21100	-2.55700	-2.30000	-3.26400	-3.52200	-3.97000	-4.30800
1.0000	0.00000	-.33900	-.65300	-1.20800	-1.66600	-2.01500	-2.23800	-2.41200	-2.54200	-2.62700	-2.64600	-2.59700

WING-FUSELAGE INTERSECTION

CHORD	X	Y	Z
0.00	75.6005	4.7500	0.0000
5.00	84.9414	4.7600	-.0412
10.00	93.2824	4.7500	-.5263
20.00	109.9643	4.7600	-1.9879
30.00	126.6462	4.7600	-3.7275
40.00	143.3292	4.7600	-5.5593
50.00	160.0101	4.7500	-7.3738
60.00	175.5920	4.7500	-9.0980
70.00	193.3739	4.7600	-10.6788
80.00	210.0559	4.7600	-12.0739
90.00	226.7378	4.7600	-13.2462
100.00	243.4197	4.7500	-14.1682

FUSELAGE UPWASH ACTING ON WING AT ALPHA= 0.00 DEG.

XPCT	0.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
Y/B/2											
0.000	-3.685	-2.542	-1.594	-1.429	-1.933	-2.048	-1.760	-1.466	-1.260	-.879	-.271
.025	-3.685	-2.542	-1.594	-1.429	-1.933	-2.048	-1.760	-1.466	-1.260	-.879	-.271
.050	1.723	3.227	3.702	3.145	2.223	1.919	1.783	1.785	1.902	1.985	2.155
.075	2.142	3.542	4.159	3.944	3.502	3.438	3.533	3.664	3.927	3.820	3.631
.100	2.404	3.296	3.534	3.341	3.136	3.173	3.272	3.328	3.528	3.331	3.002
.125	2.065	2.574	2.612	2.432	2.310	2.318	2.380	2.335	2.475	2.301	2.019
.150	1.643	1.343	1.305	1.760	1.688	1.681	1.724	1.643	1.746	1.619	1.393
.175	1.303	1.473	1.405	1.298	1.256	1.245	1.275	1.188	1.253	1.178	1.020
.200	1.047	1.134	1.062	.981	.957	.949	.971	.908	.922	.883	.771
.250	.706	.712	.645	.601	.598	.597	.604	.578	.549	.559	.504
.300	.501	.476	.417	.396	.402	.403	.406	.398	.371	.366	.349
.350	.365	.334	.284	.277	.287	.291	.295	.294	.277	.261	.262
.400	.267	.243	.202	.203	.213	.218	.225	.222	.216	.204	.194
.450	.201	.181	.150	.153	.163	.168	.175	.176	.173	.166	.157
.500	.154	.135	.114	.117	.127	.131	.135	.143	.140	.138	.131
.550	.119	.093	.083	.093	.101	.106	.112	.117	.117	.115	.112
.600	.091	.072	.071	.075	.082	.087	.090	.095	.100	.098	.095
.700	.043	.046	.048	.051	.054	.059	.061	.062	.066	.069	.072
.800	.040	.033	.032	.033	.035	.038	.040	.043	.046	.047	.049
.900	.038	.036	.032	.028	.023	.023	.024	.025	.027	.028	.030
1.000	.034	.033	.032	.030	.029	.028	.025	.023	.021	.018	.015

NACELLE PRESSURE FIELD

Y/B/2 — X, PER CENT CHORD AND PRESSURE COEFFICIENT

WRAP SOLUTION

NACELLES BELOW WING

Y/B/2												
0.000	76.590	243.420										
	0.000	100.000										
	0.00000	0.00000										
.050	76.590	238.590	238.700	238.996	239.292	239.589	239.885	240.181	240.477	240.773	241.069	241.366
	241.662	241.958	242.254	242.550	242.847	243.143	243.439	243.735				
	0.000	97.165	97.171	97.348	97.526	97.703	97.881	98.058	98.236	98.414	98.591	98.769
	98.946	99.124	99.301	99.479	99.656	99.834	100.011	100.189				
	0.00000	0.00000	.03894	.03836	.03778	.03720	.03662	.03605	.03548	.03491	.03434	.03377
	.03320	.03263	.03206	.03149	.03093	.03037	.02981	.02925				
.100	83.104	231.592	231.702	232.424	233.146	233.868	234.590	235.312	236.035	236.757	237.479	238.201
	238.923	239.645	240.367	241.090	241.812	242.534	243.256	243.875				
	0.000	92.790	92.796	93.247	93.698	94.149	94.600	95.051	95.502	95.953	96.404	96.855
	97.305	97.757	98.209	98.659	99.110	99.561	100.012	100.398				
	0.00000	0.00000	.04458	.04294	.04129	.03967	.03804	.03642	.03481	.03322	.03165	.03010
	.02858	.02706	.02555	.02408	.02261	.02115	.01971	.01848				
.150	94.655	225.394	225.404	226.499	227.595	228.690	229.785	230.882	231.977	233.073	234.169	235.264
	236.360	237.455	238.551	239.647	240.742	241.838	242.934	244.029				
	0.000	88.182	88.189	88.928	89.667	90.406	91.145	91.884	92.623	93.362	94.101	94.840
	95.579	96.318	97.057	97.796	98.535	99.274	100.013	100.752				
	0.00000	0.00000	.05213	.04913	.04616	.04322	.04030	.03741	.03461	.03186	.02915	.02548
	.02385	.02176	.01949	.02071	.01821	.01377	.00937	.00503				
.200	106.208	220.585	220.595	221.972	223.348	224.725	226.101	227.478	228.855	230.231	231.608	232.984
	234.361	235.738	237.114	238.491	239.867	241.244	242.620	243.846				
	0.000	83.858	83.866	84.875	85.884	86.893	87.903	88.912	89.921	90.930	91.940	92.949
	93.958	94.968	95.977	96.986	97.995	99.005	100.014	100.913				
	0.00000	0.00000	.06039	.05645	.05202	.04762	.04329	.03909	.03500	.03100	.02709	.02357
	.02379	.02120	.01450	.00784	.00131	-.00484	-.01032	-.01497				
.240	116.925	218.815	218.825	220.294	221.763	223.232	224.701	226.170	227.639	229.108	230.577	232.047
	233.516	234.985	236.454	237.923	239.392	240.861	242.330	243.541				
	0.000	81.261	81.269	82.441	83.612	84.784	85.955	87.127	88.299	89.470	90.642	91.814
	92.985	94.157	95.329	96.500	97.672	98.844	100.015	100.981				
	0.00000	0.00000	.06530	.06020	.05508	.05001	.04503	.04025	.03557	.03100	.02653	.02352
	.02370	.01593	.00914	.00153	-.00557	-.01187	-.01837	-.02450				

.247	116.973	218.815	218.825	220.294	221.763	223.232	224.701	226.170	227.639	229.108	230.578	232.047
	233.515	234.985	236.454	237.923	239.392	240.851	242.333	243.541				
	0.000	81.254	81.262	82.434	83.606	84.778	85.950	87.122	88.294	89.467	90.639	91.811
	92.983	94.155	95.327	96.499	97.671	98.843	100.015	100.981				
	0.00000	0.00000	.06530	.06020	.05508	.05001	.04503	.04025	.03557	.03100	.02653	.02382
	.02372	.01593	.00914	.00153	-.00557	-.01187	-.01837	-.02450				
.250	117.760	218.926	218.935	220.308	221.780	223.252	224.724	226.196	227.669	229.141	230.613	232.085
	233.557	235.029	236.501	237.974	239.446	240.918	242.390	243.553				
	0.000	81.105	81.113	82.294	83.476	84.657	85.839	87.020	88.201	89.383	90.564	91.746
	92.927	94.108	95.293	96.471	97.652	98.834	100.015	100.949				
	0.00000	0.00000	.06527	.06017	.05504	.04996	.04497	.04018	.03550	.03092	.02645	.02339
	.02360	.01574	.00895	.00133	-.00575	-.01205	-.01863	-.02449				
.300	129.312	221.119	221.129	222.710	224.292	225.873	227.455	229.037	230.618	232.200	233.781	235.363
	236.944	238.526	239.493	239.509	241.091	242.673	243.534	243.534				
	0.000	80.575	80.584	81.972	83.360	84.748	85.136	87.524	88.912	90.300	91.688	93.076
	94.464	95.852	96.707	96.716	98.104	99.492	100.248	100.248				
	0.00000	0.00000	.05973	.05472	.04975	.04483	.04004	.03541	.03088	.02648	.02271	.02319
	.01775	.01020	.00553	.04786	.03721	.02746	.02239	.02239				
.350	140.864	226.214	226.224	227.505	228.785	230.065	231.346	232.529	232.539	233.819	235.100	235.380
	237.661	238.941	240.222	241.502	242.783	244.063	244.661	244.661				
	0.000	82.649	82.659	83.899	85.139	86.379	87.619	88.764	88.774	90.014	91.254	92.494
	93.734	94.974	95.214	97.454	98.694	99.934	100.512	100.512				
	0.00000	0.00000	.05392	.04754	.04417	.04083	.03753	.03455	.08402	.07750	.07125	.06498
	.05881	.05320	.05106	.04549	.03756	.02970	.02606	.02606				
.400	152.415	226.431	226.441	227.769	229.097	230.425	231.753	232.642	232.652	233.980	235.307	236.635
	237.963	239.291	240.619	241.947	243.275	244.603	245.931	246.037				
	0.000	79.933	79.944	81.378	82.812	84.246	85.680	86.640	86.651	88.085	89.519	90.953
	92.387	93.821	95.255	96.689	98.123	99.557	100.992	101.107				
	0.00000	0.00000	.05957	.05540	.05122	.04709	.04300	.04033	.08399	.07713	.07048	.06375
	.05722	.05156	.04981	.04305	.03404	.02523	.01722	.01559				
.450	163.967	222.474	222.484	224.157	225.831	227.504	229.178	230.851	232.524	234.198	235.871	237.545
	239.218	239.706	239.715	241.389	243.063	244.736	246.371	246.371				
	0.000	71.414	71.427	73.469	75.512	77.554	79.597	81.640	83.682	85.725	87.767	89.810
	91.853	92.448	92.460	94.503	96.546	98.588	100.583	100.583				
	0.00000	0.00000	.07015	.06386	.05756	.05133	.04533	.03948	.03383	.02831	.02602	.02401
	.01426	.01146	.04967	.03699	.02510	.01428	.00285	.00288				
.472	168.957	222.002	222.012	223.746	225.481	227.215	228.949	230.683	232.418	234.152	235.886	237.621
	239.355	241.089	242.824	242.870	242.880	244.614	246.349	246.388				

	0.000	68.608	68.621	70.864	73.107	75.350	77.593	79.836	82.080	84.323	86.566	88.809
	91.052	93.295	95.538	95.598	95.611	97.854	100.097	100.148				
	0.00000	0.00000	.07183	.06511	.05841	.05180	.04541	.03926	.03328	.02766	.02812	.02118
	.01081	.00072	-.00937	-.00860	.02772	.01652	.00373	.00344				
.472	163.003	222.002	222.012	223.747	225.481	227.216	228.951	230.686	232.421	234.155	235.890	237.625
	233.360	241.035	242.823	242.839	242.909	244.644	245.373	246.388				
	0.000	68.583	68.595	70.841	73.086	75.331	77.575	79.820	82.065	84.310	86.559	88.800
	91.045	93.290	95.535	95.525	95.638	97.883	100.128	100.139				
	0.00000	0.00000	.07183	.06511	.05840	.05180	.04541	.03925	.03327	.02765	.02813	.02116
	.01079	.00359	-.00843	-.00874	.02755	.01637	.00355	.00348				
.500	175.520	222.794	222.804	224.582	226.361	228.139	229.917	231.695	233.474	235.252	237.030	238.808
	240.587	242.365	244.143	245.921	247.101	247.111	247.822	247.822				
	0.000	65.513	65.525	67.991	70.455	72.919	75.384	77.848	80.312	82.776	85.241	87.705
	90.169	92.634	95.099	97.562	99.196	99.210	100.195	100.196				
	0.00000	0.00000	.06913	.06252	.05594	.04946	.04320	.03718	.03131	.02612	.02621	.01837
	.00827	-.00154	-.01012	-.01821	-.02460	.00352	.00445	.00446				
.550	187.073	227.153	227.153	228.602	230.041	231.480	232.918	234.357	235.796	237.235	238.673	240.112
	241.551	242.989	244.429	245.867	247.306	248.744	250.183	250.576				
	0.000	63.528	63.544	65.824	68.104	70.385	72.665	74.946	77.225	79.506	81.787	84.057
	86.348	88.528	90.909	93.189	95.469	97.750	100.030	100.652				
	0.00000	0.00000	.05809	.05368	.04929	.04494	.04065	.03654	.03251	.02856	.02469	.02157
	.02229	.01739	.01075	.00423	-.00214	-.00772	-.01295	-.01436				
.600	198.626	233.415	233.425	234.627	235.830	237.032	238.235	239.438	240.640	241.843	243.046	244.248
	245.451	246.654	247.855	249.059	250.261	251.464	252.667	253.869				
	0.000	64.397	64.416	66.642	68.868	71.095	73.321	75.547	77.773	79.999	82.226	84.452
	86.678	88.904	91.130	93.357	95.583	97.809	100.035	102.261				
	0.00000	0.00000	.04833	.04539	.04241	.03944	.03650	.03363	.03082	.02807	.02536	.02269
	.02007	.01823	.01912	.01519	.01175	.00738	.00305	-.00119				
.650	210.179	240.457	240.467	241.385	242.302	243.220	244.138	245.056	245.973	246.891	247.809	248.726
	249.644	250.562	251.479	252.397	253.315	254.232	255.150	255.928				
	0.000	67.356	67.378	69.420	71.461	73.503	75.544	77.586	79.627	81.669	83.710	85.752
	87.793	89.835	91.875	93.918	95.959	98.001	100.042	101.772				
	0.00000	0.00000	.04149	.03956	.03765	.03575	.03385	.03198	.03012	.02831	.02652	.02475
	.02301	.02128	.01957	.01788	.01657	.01652	.01647	.01513				
.700	221.733	247.875	247.885	248.494	249.103	249.712	250.322	250.931	251.540	252.150	252.759	253.368
	253.978	254.587	255.195	255.806	256.415	257.024	257.634	258.243				
	0.000	72.855	72.883	74.581	76.279	77.977	79.675	81.374	83.072	84.770	86.468	88.156
	89.864	91.562	93.260	94.959	96.657	98.355	100.053	101.751				

	0.00000	0.00000	.03653	.03540	.03430	.03321	.03212	.03103	.02994	.02887	.02780	.02674
	.02570	.02467	.02365	.02263	.02162	.02062	.01962	.01863				
.750	229.521	255.497	255.507	255.796	256.085	256.374	256.663	256.952	257.241	257.530	257.828	258.109
	258.399	258.587	258.975	259.265	259.554	259.843	260.132	260.421				
	0.000	84.909	84.942	85.887	86.832	87.777	88.722	89.667	90.612	91.557	92.502	93.447
	94.392	95.337	96.282	97.227	98.172	99.117	100.062	101.007				
	0.00000	0.00000	.03276	.03229	.03183	.03137	.03092	.03046	.03000	.02955	.02909	.02864
	.02818	.02773	.02727	.02682	.02637	.02592	.02547	.02503				
.800	235.259	262.622										
	0.000	100.000										
	0.00000	0.00000										
.850	240.997	265.130										
	0.000	100.000										
	0.00000	0.00000										
.900	246.734	267.638										
	0.000	100.000										
	0.00000	0.00000										
.950	252.472	270.147										
	0.000	100.000										
	0.00000	0.00000										
1.000	258.210	272.655										
	0.000	100.000										
	0.00000	0.00000										

DEBUG PARAMETER =10

DEBUG PARAMETER =11

DEBUG PARAMETER =12

DEBUG PARAMETER =13

DEBUG PARAMETER =14

DEBUG PARAMETER =15

DEBUG PARAMETER =16

TABLE OF CAMBER CP AT BASIC ALPHA

XPCT	0.00	5.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00
	90.00	100.00								
Y/B/2										
0.000	.00216	.00615	.01402	.03004	.03054	.02223	.01544	.01273	.01168	.01132
	.01038	.00667								
.025	.00339	.00770	.01504	.02995	.03029	.02213	.01542	.01280	.01173	.01129
	.01025	.00649								
.050	.00825	.01392	.01898	.02986	.02943	.02130	.01361	.01305	.01202	.01142
	.00983	.00604								
.075	.02706	.02627	.02743	.03020	.02729	.02058	.01580	.01354	.01269	.01172
	.00873	.005[illegible]2								
.100	.04991	.04028	.03400	.02978	.02472	.01347	.01591	.01356	.01267	.01153
	.00729	.00386								
.125	.06008	.04526	.03607	.02753	.02315	.01855	.01351	.01322	.01197	.01124
	.00694	.00355								
.150	.06631	.04765	.03624	.02554	.02189	.01814	.01503	.01301	.01139	.01091
	.00723	.00384								
.175	.06375	.04712	.03477	.02425	.02087	.01758	.01480	.01281	.01105	.01052
	.00783	.00444								
.200	.06522	.04699	.03490	.02285	.02024	.01697	.01479	.01272	.01082	.01036
	.00840	.00528								
.225	.06808	.04667	.03432	.02174	.01935	.01658	.01472	.01276	.01056	.00990
	.00884	.00615								
.250	.06185	.04565	.03329	.02076	.01852	.01655	.01464	.01279	.01058	.00967
	.00910	.00700								
.275	.06351	.04596	.03347	.01932	.01818	.01641	.01467	.01263	.01071	.00953
	.00915	.00765								
.300	.05808	.04393	.03168	.01882	.01789	.01622	.01457	.01251	.01095	.00951
	.00913	.00815								
.325	.05856	.04387	.03208	.01950	.01746	.01617	.01433	.01265	.01112	.00958
	.00899	.00841								
.350	.06008	.04392	.03217	.01974	.01732	.01584	.01426	.01287	.01127	.00954
	.00882	.00853								
.375	.05421	.04227	.03128	.02000	.01677	.01554	.01444	.01238	.01152	.01001
	.00874	.00853								

.400	.05546	.04324	.03263	.02054	.01603	.01561	.01449	.01309	.01175	.01014
	.00889	.00856								
.425	.05290	.04119	.03177	.02042	.01580	.01554	.01449	.01335	.01182	.01035
	.00922	.00940								
.450	.05152	.04161	.03161	.02146	.01574	.01550	.01472	.01340	.01189	.01058
	.00956	.00843								
.475	.05294	.04252	.03305	.02236	.01576	.01552	.01469	.01333	.01210	.01189
	.00986	.00879								
.500	.04911	.04036	.03278	.02266	.01631	.01524	.01443	.01337	.01244	.01123
	.01005	.00926								
.525	.04349	.04175	.03374	.02395	.01705	.01468	.01439	.01371	.01254	.01134
	.01046	.00991								
.550	.04588	.03372	.03358	.02361	.01744	.01451	.01463	.01373	.01260	.01171
	.01131	.01037								
.575	.04584	.04047	.03411	.02452	.01870	.01437	.01438	.01370	.01237	.01227
	.01173	.01127								
.600	.04835	.04187	.03534	.02576	.01985	.01545	.01403	.01404	.01360	.01334
	.01254	.01203								
.625	.04436	.03948	.03459	.02550	.02038	.01531	.01445	.01462	.01432	.01399
	.01325	.01234								
.650	.04528	.04059	.03584	.02722	.02230	.01839	.01375	.01508	.01500	.01456
	.01390	.01391								
.675	.04737	.04207	.03726	.02932	.02424	.02065	.01763	.01558	.01552	.01534
	.01433	.01359								
.700	.04506	.04152	.03819	.03152	.02632	.02268	.01962	.01656	.01541	.01435
	.01438	.01319								
.725	.04540	.04342	.04045	.03427	.02880	.02440	.02114	.01822	.01600	.01453
	.01413	.01373								
.750	.04146	.03963	.03779	.03382	.02915	.02528	.02208	.01937	.01686	.01535
	.01445	.01395								
.775	.04122	.03814	.03627	.03311	.02974	.02643	.02333	.02084	.01850	.01632
	.01543	.01454								
.800	.03530	.03469	.03407	.03219	.02991	.02719	.02456	.02203	.01984	.01774
	.01575	.01414								
.825	.03180	.03156	.03133	.03074	.02922	.02736	.02514	.02292	.02080	.01892
	.01716	.01548								
.850	.02902	.02887	.02871	.02840	.02773	.02661	.02510	.02344	.02164	.02031
	.01890	.01759								

.875	.02520	.02631	.02541	.02652	.02659	.02603	.02327	.02405	.02274	.02121
	.01955	.01724								
.900	.02352	.02374	.02395	.02438	.02467	.02493	.02455	.02403	.02308	.02234
	.02070	.01931								
.925	.02147	.02165	.02183	.02219	.02256	.02232	.02327	.02308	.02277	.02220
	.02150	.02090								
.950	.01900	.01851	.01902	.01991	.02051	.02110	.02145	.02178	.02172	.02141
	.02072	.01898								
.975	.01607	.01640	.01672	.01736	.01794	.01940	.01986	.01904	.01913	.01910
	.01942	.01774								
1.000	.01279	.01290	.01300	.01321	.01342	.01361	.01365	.01369	.01357	.01347
	.01328	.01309								

TEA253, N LOADING VERSION OF JANUARY 28, 1974

OPTIMUM COMBINATION OF 7 WING LOADINGS

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

NUMBER OF PLANFORM BREAKPOINTS = 9.0 FLAT PLATE CONTROL FLAG = 0.0
NUMBER OF SEMISPAN ELEMENTS = 40.0 PRINT FLAG = 1.0
NUMBER OF SPAN STATIONS FOR CAMBER SURFACE = 12.0 SMOOTHING FLAG = -0.0
SPAN STATION FOR PARABOLIC APEX = -0.0 RESTART FLAG = 1.0

BASIC MACH NUMBER = 2.7000 DESIGN C-L = .1000
CBAR = 106.4100 NUMBER OF LOADINGS = -7.0000
PITCHING MOMENT CENTER AT 187.0000 NUMBER OF CAMBER ORDINATES = 12.0000
REFERENCE AREA = 9898.0000 NUMBER OF POINTS DEFINING ARBITRARY REGION = -0.0000
C-M-O CONSTRAINT = .0150 NUMBER OF NACELLES = 2.0000

NUMBER OF CHORDWISE AND SPANWISE LOCATIONS FOR
BODY BUOYANCY TABLES = 0.0 21.0
BODY UPWASH LOADING TABLE = -12.0 41.0
NACELLE BUOYANCY LOADING TABLES = -20.0 25.0
WING UPPER SURFACE LIMITING PRESSURES = 2.0 2.0
WING THICKNESS PRESSURES = -21.0 20.0

CAMBER SURFACE OPTION FLAGS = 1.0 1.0 1.0 3.0

PLANFORM DEFINITION

	X (LEADING EDGE)	Y	CHORD
1	60.001600	0.000000	183.884400
2	76.590000	4.757000	166.830000
3	83.104000	6.625000	160.133000
4	93.165000	9.510000	149.790000
5	116.960000	16.333000	125.350000
6	159.980000	31.250000	77.295000
7	225.810000	47.544000	32.681000
8	225.810000	47.545000	32.681000
9	258.210000	66.250000	14.445000

VALUES OF SEMISPAN LOCATION AT WHICH WING CAMBER SURFACE WILL BE CALCULATED

0.0000	2.0000	4.0000	8.0000	12.0000	16.0000	20.0000	24.0000	28.0000	32.0000
36.0000	40.0000								

LOADING 1 FOR THIS CASE IS UNIFORM OR CONSTANT (LOADING 1 IN THE LOADING DEFINITIONS)
LOADING 2 FOR THIS CASE IS LINEAR CHORDWISE (LOADING 2 IN THE LOADING DEFINITIONS)
LOADING 3 FOR THIS CASE IS LINEAR SPANWISE (LOADING 3 IN THE LOADING DEFINITIONS)
LOADING 4 FOR THIS CASE IS BODY UPWASH LOADING (LOADING 16 IN THE LOADING DEFINITIONS)
LOADING 5 FOR THIS CASE IS NACELLE BUOYANCY (LOADING 17 IN THE LOADING DEFINITIONS)
LOADING 6 FOR THIS CASE IS SIMILAR TO FLAT WING (LOADING 8 IN THE LOADING DEFINITIONS)
LOADING 7 FOR THIS CASE IS CUBIC CHORDWISE (LOADING 7 IN THE LOADING DEFINITIONS)

X/C(PERCENT) FOR INTERPOLATED CAMBER SURFACE ORDINATES

0.000000	5.000000	10.000000	20.000000	30.000000	40.000000	50.000000	60.000000	70.000000	80.000000
90.000000	100.000000								

NACELLE NUMBER 1, ORIGIN AT X = 213.42000000
Y = 16.33000000
Z = -5.80000000

NACELLE LONGITUDINAL COORDINATES (X HAS BEEN MULTIPLIED BY 1.00000000)

0.000000	2.028000	15.470000	21.525000	28.017000	32.067000	35.340000

NACELLE RADII (R HAS BEEN MULTIPLIED BY 1.00000000)

2.865000	2.983000	3.633000	3.770000	3.654000	3.420000	3.420000

NACELLE X AND RADIUS TABLES EXPANDED TO 40 ENTRIES BY LINEAR INTERPOLATION, AND X HAS BEEN TRANSLATED BY THE ORIGIN X.

NACELLE LONGITUDINAL COORDINATES (X HAS BEEN MULTIPLIED BY 1.00000000)

213.420000	214.318462	215.216923	216.115385	217.013846	217.912308	218.810769	219.709231	220.607692	221.506154
222.404615	223.303077	224.201538	225.100000	225.998462	226.896923	227.795385	228.693846	229.592308	230.490769
231.389231	232.287692	233.186154	234.084615	234.983077	235.881538	236.780000	237.678462	238.576923	239.475385
240.373846	241.272308	242.170769	243.069231	243.967692	244.866154	245.764615	246.663077	247.561538	248.460000

NACELLE RADII (R HAS BEEN MULTIPLIED BY 1.00000000)

2.865000	2.917798	2.970596	3.016190	3.059571	3.102952	3.146334	3.189715	3.233097	3.276478
3.319859	3.363241	3.406622	3.450003	3.493385	3.536766	3.580148	3.623529	3.648890	3.669219
3.689547	3.709876	3.730204	3.750533	3.769320	3.753266	3.737212	3.721158	3.705104	3.689050
3.672997	3.656943	3.611604	3.559693	3.507782	3.455871	3.420000	3.420000	3.420000	3.420000

NACELLE NUMBER 2, ORIGIN AT X = 218.67000000
Y = 31.25000000
Z = -4.90000000

NACELLE LONGITUDINAL COORDINATES (X HAS BEEN MULTIPLIED BY 1.00000000)

0.000000	2.008000	15.470000	21.525000	28.017000	32.067000	35.040000

NACELLE RADII (R HAS BEEN MULTIPLIED BY 1.00000000)

2.865000	2.953000	3.633000	3.770000	3.654000	3.420000	3.420000

NACELLE X AND RADIUS TABLES EXPANDED TO 40 ENTRIES BY LINEAR INTERPOLATION, AND X HAS BEEN TRANSLATED BY THE ORIGIN X.

NACELLE LONGITUDINAL COORDINATES (X HAS BEEN MULTIPLIED BY 1.00000000)

218.670000	219.568462	220.466923	221.365385	222.263846	223.162308	224.060769	224.959231	225.857692	226.756154
227.654615	228.553077	229.451538	230.350000	231.248462	232.146923	233.045385	233.943846	234.842308	235.740769
236.639231	237.537692	238.436154	239.334615	240.233077	241.131538	242.030000	242.928462	243.826923	244.725385
245.623846	246.522308	247.420769	248.319231	249.217692	250.116154	251.014615	251.913077	252.811538	253.710000

NACELLE RADII (R HAS BEEN MULTIPLIED BY 1.00000000)

2.865000	2.917798	2.970595	3.016190	3.059571	3.102952	3.146334	3.189715	3.233097	3.276478
3.319859	3.363241	3.406622	3.450003	3.493385	3.536766	3.580148	3.623529	3.648890	3.669219
3.689547	3.709875	3.730204	3.750533	3.769320	3.753266	3.737212	3.721158	3.705104	3.689050
3.672997	3.656943	3.611604	3.559693	3.507782	3.455871	3.420000	3.420000	3.420000	3.420000

X/C (PERCENT) FOR BODY UPWASH LOADING

0.00000	5.00000	10.00000	20.00000	30.00000	40.00000	50.00000	60.00000	70.00000	80.00000
90.00000	100.00000								

AND SPANWISE LOCATIONS (PERCENT SEMISPAN)

0.00000	2.50000	5.00000	7.50000	10.00000	12.50000	15.00000	17.50000	20.00000	22.50000
25.00000	27.50000	30.00000	32.50000	35.00000	37.50000	40.00000	42.50000	45.00000	47.50000
50.00000	52.50000	55.00000	57.50000	60.00000	62.50000	65.00000	67.50000	70.00000	72.50000
75.00000	77.50000	80.00000	82.50000	85.00000	87.50000	90.00000	92.50000	95.00000	97.50000
100.00000									

BODY UPWASH LOADING

.002164	.006149	.014021	.030043	.030536	.022227	.015442	.012728	.011576	.011318
.010381	.006670								
.003093	.007701	.015038	.029951	.030290	.022127	.015417	.012802	.011732	.011293
.010250	.006490								
.008257	.013017	.018983	.029865	.029430	.021896	.015606	.013047	.012022	.011423
.009834	.006039								
.027051	.026273	.027434	.030203	.027290	.020679	.015802	.013544	.012688	.011721
.009732	.005020								
.049307	.040284	.033935	.029780	.024725	.019471	.015912	.013557	.012669	.011532
.007291	.003862								
.060084	.045259	.036058	.027526	.023152	.018648	.015509	.013224	.011973	.011239
.006942	.003551								
.066313	.047663	.036241	.025539	.021889	.018143	.015028	.013013	.011395	.010914
.007233	.003843								
.063745	.047115	.034769	.024253	.020866	.017578	.014799	.012806	.011055	.010522
.007334	.004440								
.065224	.046990	.034904	.022853	.020245	.016969	.014791	.012717	.010818	.010064
.008399	.005283								
.068384	.046673	.034318	.021740	.019348	.016677	.014723	.012762	.010658	.009796
.008941	.006153								
.061954	.045663	.033294	.020758	.018520	.016649	.014645	.012794	.010576	.009667
.009100	.007002								
.063512	.045958	.033471	.019316	.018179	.016407	.014674	.012626	.010708	.009535
.009199	.007646								
.058578	.043926	.031684	.018820	.017894	.016216	.014573	.012514	.010955	.009513
.009128	.008146								
.058560	.043869	.032078	.019497	.017460	.016173	.014327	.012648	.011117	.009600
.009989	.008406								
.060077	.043921	.032167	.019739	.017317	.015845	.014263	.012870	.011273	.009842
.008823	.008531								
.054212	.042273	.031282	.020001	.016774	.015544	.014436	.012982	.011515	.010008
.008737	.008531								
.055458	.043239	.032633	.020537	.016030	.015610	.014491	.013087	.011749	.010140
.008994	.008560								
.050902	.041190	.031770	.020422	.015803	.015636	.014493	.013349	.011820	.010349
.009225	.008402								
.051520	.041608	.031605	.021463	.015736	.015497	.014718	.013395	.011889	.010683
.009564	.008435								
.052937	.042516	.033049	.022356	.015760	.015522	.014693	.013330	.012099	.011089
.009862	.008705								
.048110	.040360	.032784	.022657	.016309	.015237	.014434	.013366	.012441	.011231
.010051	.009260								

.049494	.041747	.033739	.023948	.017048	.014678	.014347	.013714	.012542	.011340
.010461	.009913								
.045890	.039717	.033577	.023613	.017439	.014509	.014635	.013735	.012598	.011709
.011014	.010366								
.046941	.040468	.034111	.024516	.018703	.014973	.014378	.013699	.012973	.012274
.011734	.011265								
.048350	.041868	.035340	.025761	.019851	.015449	.014030	.014043	.013598	.013038
.012541	.012033								
.044361	.039476	.034530	.025496	.020384	.016307	.014446	.014623	.014319	.013892
.013245	.012342								
.045276	-.040589	.035843	.027216	.022304	.018391	.015748	.015080	.014397	.014656
.013997	.013010								
.047368	.042066	.037260	.029323	.024236	.020645	.017627	.015684	.015524	.015040
.014333	.013587								
.045054	.041624	.038190	.031516	.026322	.022675	.019621	.016658	.015409	.014949
.014391	.013194								
.046396	.043424	.040453	.034273	.028802	.024401	.021142	.018221	.016001	.014627
.014131	.013793								
.041462	.039628	.037794	.033816	.029149	.025280	.022081	.019372	.016857	.015353
.014447	.013963								
.040215	.038138	.036265	.033106	.029741	.026428	.023335	.020842	.018504	.016318
.015427	.014535								
.035299	.034686	.034074	.032186	.029908	.027193	.024561	.022026	.019837	.017743
.015751	.014141								
.031801	.031564	.031327	.030741	.029223	.027359	.025142	.022925	.020798	.018917
.017164	.015478								
.029024	.028868	.028712	.028400	.027727	.026609	.025097	.023436	.021637	.020006
.018795	.017585								
.026200	.025306	.026413	.026524	.026549	.026029	.025271	.024047	.022739	.021210
.019554	.017241								
.023524	.023738	.023951	.024378	.024668	.024926	.024546	.024031	.023080	.022044
.020704	.019311								
.021455	.021647	.021829	.022193	.022557	.022915	.023274	.023085	.022767	.022197
.021496	.020795								
.018300	.018511	.019023	.019914	.020506	.021099	.021446	.021783	.021723	.021406
.020722	.018980								
.016275	.016397	.016719	.017364	.017945	.018401	.018858	.019042	.019135	.019101
.018421	.017742								
.012790	.012896	.013001	.013212	.013424	.013609	.013650	.013690	.013666	.013473
.013279	.013086								

THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE .06808402 .00216401

SEMISPAN LOCATION (PERCENT) FOR NACELLE BUOYANCY LOADING

0.00000	5.00000	10.00000	15.00000	20.00000	24.63906	24.65906	25.00000	30.00000	35.00000
40.00000	45.00000	47.15981	47.17981	50.00000	55.00000	60.00000	65.00000	70.00000	75.00000
80.00000	85.00000	90.00000	95.00000	100.00000					

X/C PERCENT FOR EACH SPAN STATION

0.000000	100.000000	100.000060	100.000120	100.000180	100.000240	100.000300	100.000360	100.000420	100.000480
100.000539	100.000599	100.000659	100.000719	100.000779	100.000839	100.000899	100.000959	100.001019	100.001079
0.000000	97.164738	97.170732	97.348273	97.525814	97.703355	97.880896	98.058437	98.235978	98.413519
98.591061	98.768602	98.946143	99.123684	99.301225	99.478766	99.656307	99.833848	100.011389	100.188930
0.000000	92.790119	92.795354	93.247333	93.698301	94.149270	94.600239	95.051208	95.502177	95.953146
96.404114	96.855083	97.306052	97.757021	98.207990	98.658959	99.109927	99.560896	100.011865	100.398212
0.000000	88.182111	88.188856	88.927853	89.666851	90.405848	91.144846	91.883843	92.622841	93.361838
94.100836	94.839833	95.578831	96.317828	97.056826	97.795823	98.534820	99.273818	100.012815	100.751813
0.000000	83.858219	83.865551	84.874825	85.884098	86.893372	87.902646	88.911920	89.921193	90.930467
91.939741	92.949014	93.958288	94.967562	95.976835	96.986109	97.995383	99.004657	100.013930	100.912714
0.000000	81.260981	81.268355	82.440393	83.612231	84.783868	85.955505	87.127143	88.298780	89.470417
90.642055	91.813692	92.985329	94.156967	95.328604	96.500241	97.671879	98.843516	100.015153	100.980929
0.000000	81.254037	81.262016	82.434087	83.606159	84.778230	85.950302	87.122373	88.294444	89.466516
90.638597	91.810659	92.982730	94.154802	95.326873	96.498945	97.671016	98.843088	100.015159	100.981247
0.000000	81.104986	81.113011	82.294401	83.475791	84.657180	85.838570	87.019960	88.201350	89.382739
90.564129	91.745519	92.926909	94.108299	95.289688	96.471078	97.652468	98.833858	100.015247	100.948746
0.000000	80.575273	80.583950	81.971909	83.359968	84.748027	86.136086	87.524145	88.912204	90.300263
91.688322	93.076381	94.464440	95.852499	96.706975	96.715751	98.103810	99.491869	100.248347	100.248347
0.000000	82.649125	82.658828	83.898779	85.138750	86.378720	87.618691	88.764151	88.773834	90.013805
91.253776	92.493747	93.733717	94.973688	96.213659	97.453630	98.693600	99.933571	100.512454	100.512454
0.000000	79.933015	79.943814	81.377864	82.811915	84.245965	85.680015	85.640212	86.651011	88.085062
89.519112	90.953163	92.387213	93.821263	95.255314	96.689364	98.123414	99.557465	100.991515	101.106727
0.000000	71.414306	71.426513	73.469133	75.511752	77.554372	79.596992	81.639612	83.682232	85.724852
87.767472	89.810092	91.852712	92.448200	92.460406	94.503026	96.545646	98.588266	100.583312	100.583312
0.000000	69.607764	69.620698	70.863832	73.106966	75.350100	77.593234	79.836368	82.079502	84.322636
86.565770	88.808904	91.052038	93.295172	95.538306	95.598170	95.611104	97.854238	100.097372	100.147868
0.000000	68.583019	68.595959	70.840861	73.085763	75.330665	77.575567	79.820469	82.065371	84.310273
86.555175	88.800077	91.044979	93.289881	95.534783	95.625433	95.638373	97.883275	100.128177	100.139237
0.000000	65.512610	65.526467	67.990743	70.455019	72.919295	75.383571	77.847847	80.312123	82.776399
85.240675	87.704950	90.169226	92.633502	95.097778	97.562054	99.195971	99.209828	100.195331	100.195331
0.000000	63.527670	63.543520	65.823933	68.104345	70.384757	72.665169	74.945581	77.225993	79.506406
81.786818	84.067230	86.347642	88.628054	90.908466	93.188879	95.469291	97.749703	100.030115	100.652380
0.000000	64.397380	64.415891	66.542096	68.868301	71.094506	73.320711	75.546916	77.773121	79.999326

82.225531	84.[illegible]1736	86.677941	88.904146	91.130351	93.356556	95.582761	97.808966	100.035171	102.261376
0.000000	67.3[illegible]037	67.378293	69.419782	71.461281	73.502780	75.544279	77.585778	79.627277	81.668776
83.710275	85.751774	87.793274	89.834773	91.876272	93.917771	95.959270	98.000769	100.042268	101.771747
0.000000	72.555263	72.883132	74.581246	76.279360	77.977473	79.675587	81.373701	83.071814	84.769928
86.468042	88.166156	89.864269	91.562383	93.260497	94.958610	96.656724	98.354838	100.052952	101.751065
0.000000	84.969301	84.941989	85.886996	86.832004	87.777011	88.722019	89.667026	90.612033	91.557041
92.502048	93.447055	94.392063	95.337070	96.282078	97.227085	98.172092	99.117100	100.062107	101.007115
0.000000	100.000000	100.000365	100.000731	100.001096	100.001462	100.001827	100.002193	100.002558	100.002924
100.003289	100.003655	100.004020	100.004386	100.004751	100.005116	100.005482	100.005847	100.006213	100.006578
0.000000	100.000000	100.000414	100.000829	100.001243	100.001657	100.002072	100.002486	100.002901	100.003315
100.003729	100.004144	100.004558	100.004972	100.005387	100.005801	100.006215	100.006630	100.007044	100.007459
0.000000	100.000000	100.000478	100.000957	100.001435	100.001914	100.002392	100.002870	100.003349	100.003827
100.004305	100.004784	100.005262	100.005741	100.006219	100.006697	100.007176	100.007654	100.008132	100.008611
0.000000	100.000000	100.000565	100.001132	100.001697	100.002263	100.002829	100.003395	100.003961	100.004526
100.005092	100.005658	100.006224	100.006789	100.007355	100.007921	100.008487	100.009053	100.009618	100.010184
0.000000	100.000000	100.000692	100.001385	100.002077	100.002769	100.003461	100.004154	100.004846	100.005538
100.006231	100.006923	100.007615	100.008307	100.009000	100.009692	100.010384	100.011076	100.011769	100.012461

NACELLE BUOYANCY LOADING

0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	.038936	.038356	.037777	.037198	.036622	.036050	.035479	.034908
.034337	.033768	.033199	.032630	.032061	.031492	.030931	.030370	.029810	.029250
0.000000	0.000000	.044585	.042938	.041291	.039666	.038041	.036422	.034811	.033218
.031646	.030102	.028575	.027061	.025565	.024076	.022610	.021147	.019710	.018479
0.000000	0.000000	.052097	.049132	.046164	.043224	.040295	.037414	.034610	.031858
.029149	.026479	.023849	.021260	.019492	.020714	.018210	.013770	.009372	.005027
0.000000	0.000000	.060878	.056453	.052016	.047623	.043291	.039094	.035005	.031004
.027086	.023573	.023787	.021204	.014502	.007839	.001306	-.004838	-.010320	-.014967
0.000000	0.000000	.065304	.060203	.055082	.050014	.045032	.040246	.035573	.031001
.026531	.023824	.023698	.016929	.009143	.001529	-.005569	-.011870	-.018369	-.024499
0.000000	0.000000	.065304	.060203	.055082	.050014	.045032	.040246	.035573	.031001
.026531	.023824	.023698	.016928	.009143	.001529	-.005570	-.011871	-.018370	-.024499
0.000000	0.000000	.065275	.060165	.055037	.049960	.044973	.040180	.035500	.030923
.026446	.023895	.023602	.016742	.008948	.001328	-.005752	-.012050	-.018604	-.024489
0.000000	0.000000	.059700	.054717	.049746	.044827	.040038	.035498	.030884	.025477
.022711	.023190	.017749	.010203	.005628	.047861	.037207	.027461	.022391	.022391
0.000000	0.000000	.050921	.047539	.044171	.040828	.037525	.034549	.084023	.077599
.071254	.064984	.058818	.053205	.051062	.045493	.037561	.029699	.026059	.026059
0.000000	0.000000	.059567	.055400	.051222	.047085	.042999	.040326	.083987	.077134
.070401	.063753	.057223	.051559	.049810	.043046	.034044	.025227	.017219	.015591
0.000000	0.000000	.070154	.063862	.057557	.051333	.045301	.039478	.033825	.028311
.026021	.024013	.014260	.011463	.049672	.036989	.025098	.014280	.002881	.002881
0.000000	0.000000	.071797	.065107	.058406	.051801	.045413	.039263	.033279	.027661
.028119	.021193	.010818	.003719	-.008372	-.008599	.027716	.016523	.003734	.003443
0.000000	0.000000	.071797	.065105	.058402	.051795	.045406	.039254	.033269	.027654
.028130	.021157	.010780	.000689	-.008400	-.008743	.027555	.016369	.003546	.003482
0.000000	0.000000	.069097	.062516	.055938	.049458	.043202	.037182	.031312	.025120
.026205	.018370	.008273	-.001542	-.010117	-.018209	-.024600	.009523	.004462	.004462
0.000000	0.000000	.058077	.053693	.049293	.044940	.040658	.036536	.032506	.028558
.024693	.021671	.022290	.017395	.010750	.004229	-.002142	-.007722	-.012950	-.014359
0.000000	0.000000	.048395	.045392	.042406	.039441	.036505	.033628	.030824	.028069
.025357	.022689	.020069	.018231	.019122	.016194	.011753	.007381	.003056	-.001192
0.000000	0.000000	.041494	.039563	.037651	.035749	.033853	.031977	.030119	.028309
.026522	.024752	.023308	.021280	.019570	.017883	.016573	.016616	.016466	.015131
0.000000	0.000000	.036502	.035396	.034298	.033207	.032117	.031030	.029943	.028870

.027799	.0[illegible]744	.025705	.024669	.023648	.022628	.021621	.020617	.019622	.018635
0.000000	0.0[illegible]000	.032756	.032293	.031829	.031373	.030916	.030459	.030002	.029546
.029090	.028635	.028190	.027725	.027270	.026820	.026372	.025923	.025475	.025026
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000

THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE .08402283 -.02460018

UPPER WING SURFACE LIMITING CP TABLES

X STATIONS

0.00000 100.00000

Y STATIONS

0.00000 100.00000

LIMIT C-P

-.137000	-.137000
-.137000	-.137000

THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE -.13700000 -.13700000

X/C(PERCENT) FOR WING THICKNESS PRESSURE COEFFICIENT

0.00000	5.00000	10.00000	15.00000	20.00000	25.00000	30.00000	35.00000	40.00000	45.00000
50.00000	55.00000	60.00000	65.00000	70.00000	75.00000	80.00000	85.00000	90.00000	95.00000
100.00000									

SPANWISE LOCATION(PERCENT SEMISPAN)

0.00000	2.50000	5.00000	7.50000	10.00000	12.50000	15.00000	20.00000	25.00000	30.00000
35.00000	40.00000	45.00000	50.00000	60.00000	70.00000	80.00000	90.00000	95.00000	100.00000

WING THICKNESS PRESSURE COEFFICIENT

0.000000	.008208	.017194	.020513	.012988	.007425	.005438	.003264	.002663	.005482
.003488	.000479	-.000100	-.001515	-.003906	-.004349	-.008065	-.013937	-.018165	-.022130
-.027107									
.003384	.008515	.013914	.013905	.009557	.007861	.008349	.005986	.003147	.002164
.000875	.000549	.001641	.000329	-.003097	-.006390	-.010492	-.014792	-.017609	-.021228
-.026055									
.012141	.012961	.015735	.014073	.012224	.008119	.004619	.003518	.004264	.002916
.002561	.001391	-.000866	-.002606	-.003728	-.007037	-.010502	-.014541	-.019393	-.024072
-.026938									
.044193	.010107	.004035	.005410	.008955	.008058	.003985	.004022	.003647	.000982
.001429	.001289	-.002097	-.004195	-.006319	-.011001	-.014695	-.017282	-.022324	-.026679
-.028459									
.064049	.007115	-.006118	.004790	.007254	.004428	.002431	.002909	.001848	.001128
.001735	-.000490	-.004020	-.006435	-.009850	-.014174	-.017639	-.020563	-.024930	-.028658
-.030494									
.093720	.006411	-.006317	.003027	.004030	.003180	.001410	.001083	.001427	.000812
-.000060	-.001496	-.003597	-.007547	-.012586	-.015899	-.018869	-.022399	-.026103	-.030207
-.032645									
.133021	.004434	-.010407	.000092	.003698	.001997	-.000703	.000883	.000915	-.001188
-.001295	-.000407	-.004801	-.010476	-.013770	-.015004	-.019718	-.024340	-.026617	-.029497
-.033311									
.049035	-.005775	-.008818	.001025	-.001539	-.001172	.000768	-.000841	-.002589	-.001167
-.001164	-.004334	-.008049	-.011358	-.015803	-.019553	-.021919	-.025428	-.028908	-.031633
-.033646									
.040300	-.005286	-.011646	-.003849	-.005326	-.004686	-.002869	-.000826	-.001664	-.003324
-.003529	-.005053	-.008474	-.013308	-.017619	-.020301	-.023333	-.027004	-.030666	-.033324
-.034423									
.027568	-.006494	-.010754	-.009395	-.006239	-.006047	-.004647	-.002865	-.004094	-.001931
-.004918	-.007486	-.011162	-.014583	-.017413	-.022979	-.026318	-.029964	-.031289	-.034498
-.037529									
.049360	.003985	-.007744	-.014614	-.010288	-.008938	-.003613	-.003909	-.004336	-.005963
-.006340	-.008419	-.011689	-.016364	-.021135	-.023984	-.027357	-.030841	-.035015	-.036348
-.038456									
.041224	.001924	-.010040	-.012519	-.011536	-.010695	-.007494	-.004655	-.005275	-.005841
-.008291	-.011125	-.014620	-.017474	-.021179	-.025203	-.029789	-.033471	-.036367	-.038179
-.038831									
.033210	-.001862	-.013157	-.015995	-.013605	-.008113	-.008726	-.006863	-.007759	-.009207
-.008559	-.012921	-.014992	-.020118	-.023898	-.026893	-.031013	-.033403	-.037491	-.040248
-.042427									
.018110	-.002790	-.013687	-.017204	-.018154	-.011229	-.007426	-.007940	-.007530	-.010394
-.013034	-.014864	-.018624	-.021547	-.024323	-.028967	-.032153	-.036164	-.039656	-.041246
-.042514									
.019845	-.001756	-.010743	-.015305	-.014502	-.014319	-.013645	-.011831	-.013038	-.015074
-.017298	-.019150	-.021437	-.026264	-.029940	-.033184	-.038010	-.041326	-.043471	-.044470

-.045470									
.001231	-.004876	-.010984	-.014841	-.015417	-.016037	-.017096	-.018154	-.018134	-.017805
-.019311	-.022965	-.026629	-.030350	-.034070	-.038411	-.042879	-.046391	-.049004	-.051538
-.054363									
.041532	.034965	.028397	.021810	.015191	.008571	.002120	-.004005	-.010130	-.016048
-.021493	-.026937	-.031904	-.035574	-.039245	-.042813	-.046047	-.049281	-.052317	-.054549
-.056781									
.044927	.041580	.038232	.034884	.031536	.026710	.020995	.015279	.009564	.003329
-.003948	-.010925	-.017902	-.024095	-.029913	-.035731	-.041549	-.045697	-.049134	-.052571
-.056008									
.049965	.042864	.039763	.036662	.032927	.028799	.024670	.020541	.016412	.010831
.005117	-.000597	-.006310	-.012095	-.018139	-.024183	-.030226	-.036270	-.042826	-.049362
-.056898									
.034998	.032584	.030170	.027756	.025342	.022928	.020514	.018100	.015391	.011956
.008532	.005108	.001693	-.001741	-.005151	-.008549	-.011948	-.015346	-.018744	-.022143
-.025541									

THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE .13302127 -.05689805

DELTAT = .879 SEC., T = 147.340 SEC.

FLAT WING FORCE COEFFICIENTS

$C_L = .027674$ $C_D = .0004830$ $\frac{XF}{L} = .720745$ $\frac{C_M}{C_L} = -.089415$ $\frac{C_D}{C_L^2} = .530681$

WING DATA FOR UNIFORM OR CONSTANT LOADING

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$\frac{Y}{B/2}$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8844000	2.5798467	1.0000000	-.8263583
.0500000	172.0087010	1.2893861	1.0000000	-.9161196
.1000000	160.1330000	.9483733	1.0000000	-1.0192347
.2000000	136.3933123	.6226274	1.0000000	-1.2787009
.3000000	113.9334744	.4166064	1.0000000	-1.6352923
.4000000	92.5970892	.2760153	1.0000000	-2.1461007
.5000000	72.1611317	.1388743	1.0000000	-2.9330184
.6000000	54.0214637	.0284964	1.0000000	-4.1768068
.7000000	35.9817958	-.2072074	1.0000000	-6.6821074
.8000000	27.3627760	.1246594	1.0000000	-9.0987203
.9000000	20.3038890	.2635956	1.0000000	-12.3067405
1.0000000	14.4450000	.1572312	1.0000000	-18.3990688

C_L = 1.087735 C_D = .696259 $\frac{X_{CP}}{L}$ = .693907 K_E = .588470

$\frac{S_{REF}}{S_{PROG}}$ = .919341 $\frac{C_M}{C_L}$ = -.020647 C_{M_0} = .074801

C_D WING-ON-NACELLE(S) = .002214

INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 1 (UNIFORM OR CONSTANT) IS 1.19789334E+00
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 1 (UNIFORM OR CONSTANT) IS 2.72828925E-01
INTERFERENCE DRAG OF LOADING 4 (BODY UPWASH LOADING) ON LOADING 1 (UNIFORM OR CONSTANT) IS 9.81740777E-03
INTERFERENCE DRAG OF LOADING 5 (NACELLE BUOYANCY) ON LOADING 1 (UNIFORM OR CONSTANT) IS 3.06543352E-03
INTERFERENCE DRAG OF LOADING 6 (SIMILAR TO FLAT WING) ON LOADING 1 (UNIFORM OR CONSTANT) IS 5.77825109E-01
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 1 (UNIFORM OR CONSTANT) IS 1.99721554E+00

MINIMUM OF (C_{P} UPPER SURFACE − C_{P} LIMIT) = −.5568 AT 80.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

DELTAT = 11.582 SEC., T = 196.690 SEC.

WING DATA FOR LINEAR CHORDWISE LOADING

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/(B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.9844000	9.5852340	1.7003484	-1.6884083
.0500000	172.0097010	3.8380694	1.5907850	-1.7220469
.1000000	160.1330000	2.3922455	1.4813403	-1.7557711
.2000000	136.3933123	1.0417997	1.2610877	-1.8227090
.3000000	113.9394744	.4392285	1.0542456	-1.8987452
.4000000	92.5970892	.1310471	.8562936	-1.9607872
.5000000	72.1611317	-.0107430	.6681052	-2.0700081
.6000000	54.0214637	-.0590931	.4994753	-2.1702495
.7000000	35.9917958	-.0705393	.3334622	-2.2833441
.8000000	27.3627760	-.0029394	.2534695	-2.3491900
.9000000	20.9034840	.0083485	.1946099	-2.4256667
1.0000000	14.4450000	.0047213	.1398801	-2.5920430

C_L = 1.155897 C_D = 1.681279 X_{CP}/L = .731415 K_E = 1.258349

S_{REF}/S_{PROG} = .919341 C_M/C_L = -.116755 C_{M_0} = -.031602

C_D = .002290

WING-ON-NACELLE(S)

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING	2 (LINEAR CHORDWISE)	IS	7.49967972E-01
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING	2 (LINEAR CHORDWISE)	IS	1.46353882E-01
INTERFERENCE DRAG OF LOADING	4 (BODY UPWASH LOADING)	ON LOADING	2 (LINEAR CHORDWISE)	IS	9.01875490E-03
INTERFERENCE DRAG OF LOADING	5 (NACELLE BUOYANCY)	ON LOADING	2 (LINEAR CHORDWISE)	IS	3.31050846E-03
INTERFERENCE DRAG OF LOADING	6 (SIMILAR TO FLAT WING)	ON LOADING	2 (LINEAR CHORDWISE)	IS	5.55868631E-01
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	2 (LINEAR CHORDWISE)	IS	3.18287578E+00

MINIMUM OF ($C_{P_{UPPER\ SURFACE}}$ - $C_{P_{LIMIT}}$) = -1.7273 AT 0.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

DELTAT = 11.648 SEC., T = 208.338 SEC.

WING DATA FOR LINEAR SPANWISE LOADING

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$\frac{Y}{B/2}$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8844000	0.0000000	0.0000000	0.0000000
.0500000	172.0087010	-.0089937	.1500000	-.1374173
.1000000	160.1330000	.1125175	.3000000	-.3057704
.2000000	136.3933123	.4660583	.6000000	-.7672205
.3000000	113.9394744	.8422840	.9000000	-1.4717630
.4000000	92.5970892	1.1772889	1.2000000	-2.5753209
.5000000	72.1611317	1.3608273	1.5000000	-4.3995275
.6000000	54.0214637	1.3869116	1.8000000	-7.5182522
.7000000	35.8817959	.5546854	2.1000000	-14.0324255
.8000000	27.3627760	2.0575935	2.4000000	-21.8369285
.9000000	20.9038880	3.0910470	2.7000000	-33.2281995
1.0000000	14.4450000	2.3485096	3.0000000	-55.1972064

C_L = 1.029478 C_D = .819515 $\frac{X_{CP}}{L}$ = .776981 K_E = .773256

$\frac{S_{REF}}{S_{PROG}}$ = .919341 $\frac{C_M}{C_L}$ = -.233510 C_{M_0} = -.148343

C_{D} WING-ON-NACELLE(S) = .002385

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 3 (LINEAR SPANWISE) IS 5.77550395E-01
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 3 (LINEAR SPANWISE) IS 4.44059566E-01
INTERFERENCE DRAG OF LOADING 4 (BODY UPWASH LOADING) ON LOADING 3 (LINEAR SPANWISE) IS 8.98668033E-03
INTERFERENCE DRAG OF LOADING 5 (NACELLE BUOYANCY) ON LOADING 3 (LINEAR SPANWISE) IS 5.87012595E-03
INTERFERENCE DRAG OF LOADING 6 (SIMILAR TO FLAT WING) ON LOADING 3 (LINEAR SPANWISE) IS 5.09554775E-01
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 3 (LINEAR SPANWISE) IS -2.68536648E-02

MINIMUM OF ($C_{P\ \mathrm{UPPER\ SURFACE}}$ - $C_{P\ \mathrm{LIMIT}}$) = -1.5255 AT 100.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

DELTAT = 11.668 SEC., T = [illegible] SEC.

WING DATA FOR BODY UPWASH LOADING LOADING

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y / (B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8844000	0.0000000	.0161836	-.0123403
.0500000	172.0087010	0.0000000	.0169017	-.0140941
.1000000	160.1330000	0.0000000	.0194431	-.0169000
.2000000	136.3933123	0.0000000	.0185409	-.0207081
.3000000	113.9394744	0.0000000	.0172857	-.0258329
.4000000	92.5970892	0.0000000	.0174857	-.0350691
.5000000	72.1611317	0.0000000	.0176114	-.0494701
.6000000	54.0214637	0.0000000	.0194278	-.0788286
.7000000	35.9817358	0.0000000	.0227637	-.1497045
.8000000	27.3627760	0.0000000	.0247868	-.2235879
.9000000	20.3038590	0.0000000	.0233790	-.2873805
1.0000000	14.4450000	0.0000000	.0134356	-.2472103

C_L = .020149 C_D = 0.000000 $\dfrac{X_{CP}}{L}$ = .647725 K_E = 0.000000

$\dfrac{S_{REF}}{S_{PROG}}$ = .919341 $\dfrac{C_M}{C_L}$ = .097683 C_{M_O} = .003770

C_D = .000029
WING-ON-NACELLE(S)

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 4 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 4 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 4 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 5 (NACELLE BUOYANCY) ON LOADING 4 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 6 (SIMILAR TO FLAT WING) ON LOADING 4 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 4 (BODY UPWASH LOADING) IS 0.

MINIMUM OF ($C_{P\ \mathrm{UPPER\ SURFACE}}$ - $C_{P\ \mathrm{LIMIT}}$) = -.0657 AT 90.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

DELTAT = .550 SEC., T = 220.556 SEC.

WING DATA FOR NACELLE BUOYANCY LOADING

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y / (B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8844000	0.0000000	0.0000000	0.0000000
.0500000	172.0087010	0.0000000	.0009363	-.0013137
.1000000	160.1330000	0.0000000	.0026260	-.0038721
.2000000	136.3933123	0.0000000	.0037379	-.0062593
.3000000	113.9394744	0.0000000	.0062126	-.0126341
.4000000	92.5970882	0.0000000	.0096895	-.0246218
.5000000	72.1611317	0.0000000	.0094939	-.0300187
.6000000	54.0214637	0.0000000	.0082822	-.0370251
.7000000	35.8817958	0.0000000	.0082231	-.0577493
.8000000	27.3627750	0.0000000	.0000000	-.0000000
.9000000	20.9038880	0.0000000	0.0000000	0.0000000
1.0000000	14.4450000	0.0000000	0.0000000	0.0000000

C_L = .005272 C_D = 0.000000 X_{CP}/L = .860673 K_E = 0.000000

S_{REF}/S_{PROG} = .919341 C_M/C_L = -.447954 C_{M0} = -.001890

C_D WING-ON-NACELLE(S) = 0.000000

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 5 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 5 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 5 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 4 (BODY UPWASH LOADING) ON LOADING 5 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 6 (SIMILAR TO FLAT WING) ON LOADING 5 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 5 (NACELLE BUOYANCY) IS 0.

MINIMUM OF (C_P UPPER SURFACE - C_P LIMIT) = -.0568 AT 80.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

DELTAT = .543 SEC., T = 221.899 SEC.

NG DATA FOR SIMILAR TO FLAT WING LOADING

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$Y/(B/2)$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8844000	3.1492958	.9993676	-.7261872
.0500000	172.0087010	1.3843790	.9980726	-.8154605
.1000000	160.1330000	.9099207	.9961829	-.9175504
.2000000	136.3933123	.5799414	.9997471	-1.1785240
.3000000	113.9394744	.2921199	.9948314	-1.5298263
.4000000	92.5970892	.1234851	1.0008060	-2.0472526
.5000000	72.1611317	.0692555	.9920661	-2.8142721
.6000000	54.0214637	-.2011954	1.0049852	-4.0944839
.7000000	35.8817959	-.3384355	.9839211	-6.4831123
.8000000	27.3627760	.0802501	.9978718	-8.9798354
.9000000	20.9038880	.2934192	.9677070	-11.8278473
1.0000000	14.4450000	-.1068973	.9098076	-16.6859025

C_L = 1.083094 C_D = .645434 X_{CP}/L = .651715 K_E = .550199

S_{REF}/S_{PROG} = .919341 C_M/C_L = .087461 C_{M_0} = .191574

C_D WING-ON-NACELLE(S) = .001814

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 6 (SIMILAR TO FLAT WING) IS 7.29786232E-01
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 6 (SIMILAR TO FLAT WING) IS 1.19313180E+00
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 6 (SIMILAR TO FLAT WING) IS 2.11723944E-01
INTERFERENCE DRAG OF LOADING 4 (BODY UPWASH LOADING) ON LOADING 6 (SIMILAR TO FLAT WING) IS 1.09232242E-02
INTERFERENCE DRAG OF LOADING 5 (NACELLE BUOYANCY) ON LOADING 6 (SIMILAR TO FLAT WING) IS 2.27038495E-03
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 6 (SIMILAR TO FLAT WING) IS 2.00960098E+00

MINIMUM OF ($C_{P_{UPPER\ SURFACE}}$ - $C_{P_{LIMIT}}$) = -1.2500 AT 0.0000 PERCENT SEMISPAN AND 0.0000 PERCENT CHORD

DELTAT = 14.365 SEC., T = 235.464 SEC.

WING DATA FOR CUBIC CHORDWISE LOADING

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$Y/(B/2)$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8844000	61.9447617	3.6402351	-3.7361262
.0500000	172.0087010	14.5306294	2.9795580	-3.3251632
.1000000	160.1330000	5.0768548	2.4041342	-2.9304335
.2000000	136.3933123	.0303302	1.4855070	-2.1966141
.3000000	113.3394744	-.4594161	.8660677	-1.5890832
.4000000	92.5970492	-.2787932	.4648139	-1.0906171
.5000000	72.1511317	-.1050435	.2200213	-.6891720
.6000000	54.0214637	-.0311981	.0922852	-.4039907
.7000000	35.8917358	-.0057752	.0270614	-.1862215
.8000000	27.3627760	-.0010861	.0119949	-.1115644
.9000000	20.3038880	-.0002099	.0053627	-.0670416
1.0000000	14.4450000	-.0000292	.0018107	-.0336120

C_L = 1.444349 C_D = 5.189257 X_{CP}/L = .724380 K_E = 2.966839

S_{REF}/S_{PROG} = .919341 C_M/C_L = -.098730 C_{M_0} = -.013455

$C_{D\ \text{WING-ON-NACELLE(S)}}$ = .002105

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 7 (CUBIC CHORDWISE) IS 1.04931558E+00
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 7 (CUBIC CHORDWISE) IS 2.78913038E+00
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 7 (CUBIC CHORDWISE) IS -9.77387010E-02
INTERFERENCE DRAG OF LOADING 4 (BODY UPWASH LOADING) ON LOADING 7 (CUBIC CHORDWISE) IS 1.23605007E-02
INTERFERENCE DRAG OF LOADING 5 (NACELLE BUOYANCY) ON LOADING 7 (CUBIC CHORDWISE) IS -1.15331425E-04
INTERFERENCE DRAG OF LOADING 6 (SIMILAR TO FLAT WING) ON LOADING 7 (CUBIC CHORDWISE) IS 7.59418183E-01

MINIMUM OF ($C_{P\ \text{UPPER SURFACE}}$ - $C_{P\ \text{LIMIT}}$) = -3.6673 AT 0.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

DELTAT = 12.817 SEC., T = 247.451 SEC.

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FORCE COEFFICIENTS OF COMPONENT AND INTERFERENCE LOADINGS

WING DESIGN    7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

   GROSS WING AREA =    10765.401856        SREF/SPROG =          .919341

  CL 1 =        1.087735 FOR UNIFORM OR CONSTANT  LOADING
  CL 2 =        1.155897 FOR   LINEAR CHORDWISE   LOADING
  CL 3 =        1.029478 FOR   LINEAR SPANWISE    LOADING
  CL 4 =         .020149 FOR BODY UPWASH LOADING  LOADING
  CL 5 =         .005272 FOR  NACELLE BUOYANCY    LOADING
  CL 6 =        1.083094 FOR SIMILAR TO FLAT WING LOADING
  CL 7 =        1.444349 FOR   CUBIC CHORDWISE    LOADING

  C-M-0 1 =          .074801
  C-M-0 2 =         -.031602
  C-M-0 3 =         -.149343
  C-M-0 4 =          .003770
  C-M-0 5 =         -.001890
  C-M-0 6 =          .191574
  C-M-0 7 =         -.013455

  CD 1 1/(CL 1)(CL 1) =         .589470
  CD 2 2/(CL 2)(CL 2) =        1.258349
  CD 3 3/(CL 3)(CL 3) =         .773256
  CD 4 4/(CL 4)(CL 4) =        0.000000
  CD 5 5/(CL 5)(CL 5) =        0.000000
  CD 6 6/(CL 6)(CL 6) =         .550199
  CD 7 7/(CL 7)(CL 7) =        2.965839

  (CD 1 2+CD 2 1)/(CL 1)(CL 2) =        1.549229
  (CD 1 3+CD 3 1)/(CL 1)(CL 3) =         .759404
  (CD 1 4+CD 4 1)/(CL 1)(CL 4) =         .447950
  (CD 1 5+CD 5 1)/(CL 1)(CL 5) =         .534544
  (CD 1 6+CD 6 1)/(CL 1)(CL 6) =        1.109915
  (CD 1 7+CD 7 1)/(CL 1)(CL 7) =        1.939145
  (CD 2 3+CD 3 2)/(CL 2)(CL 3) =         .496158
  (CD 2 4+CD 4 2)/(CL 2)(CL 4) =         .397242
  (CD 2 5+CD 5 2)/(CL 2)(CL 5) =         .543238
  (CD 2 6+CD 6 2)/(CL 2)(CL 6) =        1.397025
  (CD 2 7+CD 7 2)/(CL 2)(CL 7) =        3.577116
  (CD 3 4+CD 4 3)/(CL 3)(CL 4) =         .433249
  (CD 3 5+CD 5 3)/(CL 3)(CL 5) =        1.091547
  (CD 3 6+CD 6 3)/(CL 3)(CL 6) =         .646874
  (CD 3 7+CD 7 3)/(CL 3)(CL 7) =        -.083792
  (CD 4 5+CD 5 4)/(CL 4)(CL 5) =        0.000000
  (CD 4 6+CD 6 4)/(CL 4)(CL 6) =         .500815
  (CD 4 7+CD 7 4)/(CL 4)(CL 7) =         .424736
  (CD 5 6+CD 6 5)/(CL 5)(CL 6) =         .397549
  (CD 5 7+CD 7 5)/(CL 5)(CL 7) =        -.015154
  (CD 6 7+CD 7 6)/(CL 6)(CL 7) =        1.770058

  CD WING-LIFT-ON-NACELLES  1 =         .002214
  CD WING-LIFT-ON-NACELLES  2 =         .002290
  CD WING-LIFT-ON-NACELLES  3 =         .002385
  CD WING-LIFT-ON-NACELLES  4 =         .000029
  CD WING-LIFT-ON-NACELLES  5 =        0.000000
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     CD WING-LIFT-ON-NACELLES  6 =        .001814
     CD WING-LIFT-ON-NACELLES  7 =        .002105
                                                                           DELTAT =      .088 SEC., T =    247.559 SEC.

RESTART DATA PUNCHED, DECK IMAGE FOLLOWS.

WING DESIGN   7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS           RESTART
 1.2401396111736E+00 1.7907497324428E+00 7.8178908862871E-01 9.0255503597905E-03
 2.8181802436644E-03 1.2021422868333E+00 2.8008024785434E+00 0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  1.7907497324423E+00 3.0913392404302E+00 5.4279158286634E-01
 8.2913156334630E-03 3.0434878032637E-03 1.6079286727437E+00 5.4903682398974E+00
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  7.8178908862871E-01 5.4279158286634E-01
 1.5068289621450E+00 8.2618281457997E-03 5.3966503797511E-03 6.6310145733337E-01
-1.1454287690183E-01 0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  9.0255503597905E-03
 8.2913156334630E-03 8.2618281457997E-03 0.                  0.
 1.0047689300188E-02 1.1363521243807E-02 0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 2.8181802436644E-03 3.0434978032637E-03 5.3966503797511E-03 0.
 0.                  2.0869832958645E-03-1.0608412520359E-04 0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  1.2021422868333E+00 1.6079286727437E+00 6.6310145733337E-01
 1.0047689300188E-02 2.0869932958645E-03 1.1867492555130E+00 2.5456742207045E+00
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  2.8008024785434E+00 5.4903682398974E+00
-1.1454287690183E-01 1.1363521243807E-02-1.0608412520359E-04 2.5456742207045E+00
 1.1380081722139E+01 0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
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 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
 0.                  0.                  0.                  0.
```

****
SOLUTION FOR DESIGN C_L = .100000
****

RIGHT-SIDE SOLUTION MATRIX

-.00203567	-.00210551	-.00219299	-.00002706	-0.00000000	-.00166741	-.00193357	.09193415	1.00000000	1.00000000

LEFT-SIDE SOLUTION MATRIX

1.280200	1.790750	.781789	.003026	.002818	1.202142	2.800302	1.000000	0.000000	0.000000
1.790750	3.091339	.542792	.008291	.003043	1.607929	5.490368	1.062664	0.000000	0.000000
.781789	.542792	1.536829	.008262	.005397	.663101	-.114543	.946442	0.000000	0.000000
.003026	.008231	.008262	0.000000	0.000000	.010048	.011364	.018523	1.000000	0.000000
.002818	.003043	.005397	0.000000	0.000000	.002087	-.000106	.004947	0.000000	1.000000
1.202142	1.607929	.653101	.010048	.002087	1.186749	2.545574	.995733	0.000000	0.000000
2.900802	5.490354	-.114543	.011364	-.000106	2.545674	11.380382	1.327850	0.000000	0.000000
1.000000	1.062664	.946442	.018523	.004847	.995733	1.327850	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	1.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
0.000000	0.000000	0.000000	0.000000	1.000000	0.000000	0.000000	0.000000	0.000000	0.000000

DETERMINANT OF LHS = -.003080484 = -3.080484E-03

TEST OF SOLUTION ACCURACY BY MULTIPLICATION OF SQUARE MATRIX BY SOLUTION COLUMN MATRIX
THE FOLLOWING TABLE IS THE PRODUCT.

-2.0357E-03	-2.1055E-03	-2.1930E-03	-2.7061E-05	-8.6736E-19	-1.6674E-03	-1.9356E-03	9.1934E-02	1.0000E+00	1.0000E+00

A_I $C\text{-}L_I$

C_M	K_S	I = 1	2	3	4	5	6	7
.012647	.366875	-.073210	-.003980	.032005	.020149	.005272	.115044	.004721

CL AND CD FROM ALGEBRAIC COMBINATION OF LOADINGS ARE .100000000 .003668746

MINIMUM OF (C_P UPPER SURFACE $- C_P$ LIMIT) = -.0161 AT 70.0000 PERCENT SEMISPAN AND 0.0000 PERCENT CHORD

DELTAT = 2.011 SEC., T = 249.538 SEC.

LIFT-DEPENDENT DRAG FACTOR K_E AS A FUNCTION OF C_{M_0} FOR $C_{L_{DESIGN}}$ = .1000

C_{M_0}	K_E	A_I C-L_I, I = 1	2	3	4	5	6	7
-.018000	.575535	.059585	.001594	.071107	.020149	.005272	-.076022	.019215
-.015000	.536693	.045774	.001048	.067279	.020149	.005272	-.057318	.017797
-.012000	.501830	.032963	.000502	.063451	.020149	.005272	-.039615	.016378
-.009000	.470976	.019352	-.000043	.059624	.020149	.005272	-.019911	.014959
-.006000	.444120	.007040	-.000589	.055796	.020149	.005272	-.001208	.013540
-.003000	.421264	-.005871	-.001135	.051968	.020149	.005272	.017495	.012121
.000000	.402407	-.018782	-.001680	.048141	.020149	.005272	.036199	.010702
.003000	.387549	-.031693	-.002226	.044313	.020149	.005272	.054902	.009283
.006000	.376689	-.044604	-.002771	.040485	.020149	.005272	.073605	.007864
.009000	.369829	-.057516	-.003317	.036658	.020149	.005272	.092309	.006445
.012000	.366967	-.070427	-.003863	.032830	.020149	.005272	.111012	.005027
.015000	.368105	-.083338	-.004408	.029002	.020149	.005272	.129716	.003508
.018000	.373241	-.096249	-.004954	.025175	.020149	.005272	.148419	.002189
.021000	.382377	-.109160	-.005500	.021347	.020149	.005272	.167122	.000770
.024000	.395511	-.122072	-.006045	.017519	.020149	.005272	.185826	-.000649
.027000	.412644	-.134983	-.006591	.013692	.020149	.005272	.204529	-.002068
.030000	.433776	-.147894	-.007137	.009864	.020149	.005272	.223233	-.003487
.033000	.458907	-.160805	-.007682	.006036	.020149	.005272	.241936	-.004905
.036000	.488038	-.173716	-.008228	.002209	.020149	.005272	.260639	-.006324
.039000	.521166	-.186628	-.008773	-.001619	.020149	.005272	.279343	-.007743
.042000	.558294	-.199539	-.009319	-.005447	.020149	.005272	.298046	-.009162

DELTAT = 1.019 SEC., T = 250.539 SEC.

DETERMINANT OF SOLUTION MATRIX FOR PRECEDING SOLUTIONS IS .00009780 = 9.708E-05

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                    ***********************************
                    SOLUTION FOR DESIGN C   =      .100000
                                         L

                    WITH  1 CONSTRAINT(S) ON PRESSURE
                    ***********************************

       RIGHT-SIDE SOLUTION MATRIX
 -.00203567  -.00210551  -.00219299  -.00002706  -0.00000000  -.00166741  -.00193557  .09193415  .26263958  1.00000000
  1.00000000

       LEFT-SIDE SOLUTION MATRIX
   1.290200   1.790750    .781789    .003026    .002818   1.202142   2.800302   1.000000   1.000000   0.000000
   0.000000
   1.790750   3.091339    .542792    .008291    .003043   1.607929   5.490368   1.062664   0.000000   0.000000
   0.000000
    .781789    .542792   1.506929    .005262    .005397    .663101   -.114543    .946442   2.100000   0.000000
   0.000000
    .003026    .008291    .005262   0.000000   0.000000    .010048    .011364    .018523    .045059   1.000000
   0.000000
    .002818    .003043    .005397   0.000000   0.000000    .002087   -.000105    .004847   0.000000   0.000000
   1.000000
   1.202142   1.607929    .663101    .010048    .002087   1.186749   2.545674    .995733   2.500000   0.000000
   0.000000
   2.800302   5.490358   -.114543    .011364   -.000106   2.545674  11.380382   1.327850   0.000000   0.000000
   0.000000
   1.000000   1.062664    .946442    .018523    .004847    .995733   1.327850   0.000000   0.000000   0.000000
   0.000000
   1.000000   0.000000   2.100000    .045059   0.000000   2.500000   0.000000   0.000000   0.000000   0.000000
   0.000000
   0.000000   0.000000   0.000000   1.000000   0.000000   0.000000   0.000000   0.000000   0.000000   0.000000
   0.000000
   0.000000   0.000000   0.000000   0.000000   1.000000   0.000000   0.000000   0.000000   0.000000   0.000000
   0.000000

   DETERMINANT OF LHS =         .573031772 =        5.730318E-01

   TEST OF SOLUTION ACCURACY BY MULTIPLICATION OF SQUARE MATRIX BY SOLUTION COLUMN MATRIX
   THE FOLLOWING TABLE IS THE PRODUCT.

 -2.0357E-03 -2.1055E-03 -2.1930E-03 -2.7061E-05  8.6736E-19 -1.6674E-03 -1.9356E-03  9.1934E-02  2.6264E-01  1.0000E+00
  1.0000E+00

                A C-L
                 I   I
   C
    M          K
    O          E     I = 1        2          3          4          5          6          7

  .012173   .367492   -.013674   -.019256    .022905    .020149    .005272    .079468    .005136

   CL AND CD FROM ALGEBRAIC COMBINATION OF LOADINGS ARE          .100000000         .003674918

 MINIMUM OF (C                 - C      ) =      .0059 AT   70.0000 PERCENT SEMISPAN AND    0.0000 PERCENT CHORD
               P                 P
             UPPER SURFACE      LIMIT
                                                                DELTAT =     .429 SEC., T =    251.028 SEC.
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                                   ************************************
                                   SOLUTION FOR DESIGN C   =      .100000
                                                         L
                                   WITH C   CONSTRAINED TO       .015000
                                         M
                                          O
                                   ************************************

        RIGHT-SIDE SOLUTION MATRIX
 -.00203567  -.00210551  -.00219299  -.00002706 -0.00000000  -.00166741  -.00193557   .09193415  1.00000000  1.00000000
  .00538192

        LEFT-SIDE SOLUTION MATRIX
  1.280200    1.790750     .781789     .009026     .002818    1.202142    2.800802    1.000000    0.000000    0.000000
   .026938
  1.790750    3.691339     .542792     .008291     .003043    1.607929    5.490368    1.062664    0.000000    0.000000
  -.011339
   .781789     .542792    1.536829     .008262     .005397     .663101    -.114343     .946442    0.000000    0.000000
  -.053225
   .009026     .008291     .008262    0.000000    0.000000     .010048     .011364     .018523    1.000000    0.000000
   .001353
   .002818     .003043     .005397    0.000000    0.000000     .002097    -.000106     .004947    0.000000    1.000000
  -.000678
  1.202142    1.607929     .653101     .010048     .002087    1.186749    2.545674     .995733    0.000000    0.000000
   .068736
  2.800802    5.490368    -.114343     .011364    -.000106    2.545674   11.380982    1.327850    0.000000    0.000000
  -.004927
  1.000000    1.062664     .946442     .018523     .004847     .395733    1.327850    0.000000    0.000000    0.000000
  0.000000
  0.000000    0.000000    0.000000    1.000000    0.000000    0.000000    0.000000    0.000000    0.000000    0.000000
  0.000000
  0.000000    0.000000    0.000000    0.000000    1.000000    0.000000    0.000000    0.000000    0.000000    0.000000
  0.000000
   .026938    -.011339    -.053225     .001353    -.000678     .068736    -.004927    0.000000    0.000000    0.000000
  0.000000

  DETERMINANT OF LHS =          .000097080 =          9.708037E-05

  TEST OF SOLUTION ACCURACY BY MULTIPLICATION OF SQUARE MATRIX BY SOLUTION COLUMN MATRIX
  THE FOLLOWING TABLE IS THE PRODUCT.

-2.0357E-03  -2.1055E-03  -2.1930E-03  -2.7061E-05  -3.4694E-18  -1.6674E-03  -1.9356E-03   9.1934E-02   1.0000E+00   1.0000E+00
 5.3819E-03

                              A C-L
                               I   I
  C
   M          K
    O          E      I = 1        2          3          4          5          6          7

  .015000    .368105   -.083335   -.004408    .029002    .020149    .005272    .129716    .003608

  CL AND CD FROM ALGEBRAIC COMBINATION OF LOADINGS ARE            .100000000          .003681050

MINIMUM OF (C                  - C        ) =      -.0253 AT    70.0000 PERCENT SEMISPAN AND     0.0000 PERCENT CHORD
             P                    P
              UPPER SURFACE        LIMIT
```

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                    ************************************
                    SOLUTION FOR DESIGN C   =      .100000
                                         L
                    WITH  1 CONSTRAINT(S) ON PRESSURE
                    WITH C    CONSTRAINED TO       .015000
                          M
                          0
                    ************************************

      RIGHT-SIDE SOLUTION MATRIX
 -.00203567  -.00210551  -.00219299  -.00002706 -0.00000000  -.00166741  -.00193557   .09193415   .26263958  1.00000000
 1.00000000   .00538192

      LEFT-SIDE SOLUTION MATRIX
   1.280200    1.790750     .731789     .009026     .002818    1.202142    2.800802    1.000000    1.000000    0.000000
   0.000000     .026838
   1.790750    3.091339     .542792     .008231     .003043    1.607929    5.490368    1.062664    0.000000    0.000000
   0.000000    -.011339
    .731789     .542792    1.505929     .008262     .005397     .663101    -.114543     .946442    2.100000    0.000000
   0.000000    -.053225
    .009026     .008231     .008262    0.000000    0.000000     .010049     .011364     .018523     .045059    1.000000
   0.000000     .001353
    .002818     .003043     .005397    0.000000    0.000000     .002087    -.000106     .004847    0.000000    0.000000
   1.000000    -.000678
   1.202142    1.607929     .663101     .010048     .002087    1.186749    2.545574     .995733    2.500000    0.000000
   0.000000     .068736
   2.800802    5.490358    -.114543     .011364    -.000106    2.545674   11.380382    1.327850    0.000000    0.000000
   0.000000    -.004827
   1.000000    1.062664     .946442     .018523     .004847     .995733    1.327850    0.000000    0.000000    0.000000
   0.000000    0.000000
   1.000000    0.000000    2.100000     .045059    0.000000    2.500000    0.000000    0.000000    0.000000    0.000000
   0.000000    0.000000
   0.000000    0.000000    0.000000    1.000000    0.000000    0.000000    0.000000    0.000000    0.000000    0.000000
   0.000000    0.000000
   0.000000    0.000000    0.000000    0.000000    1.000000    0.000000    0.000000    0.000000    0.000000    0.000000
   0.000000    0.000000
    .026338    -.011339    -.053225     .001353    -.000679     .068735    -.004827    0.000000    0.000000    0.000000
   0.000000    0.000000

   DETERMINANT OF LHS =          -.016601415 =         -1.660141E-02

   TEST OF SOLUTION ACCURACY BY MULTIPLICATION OF SQUARE MATRIX BY SOLUTION COLUMN MATRIX
   THE FOLLOWING TABLE IS THE PRODUCT.

 -2.0357E-03 -2.1055E-03 -2.1930E-03 -2.7061E-05 -3.2526E-19 -1.6674E-03 -1.9356E-03  9.1934E-02  2.6264E-01  1.0000E+00
 1.0000E+00  5.3819E-03

                  A C-L
                  I  I
   C
    M        K
    0        E     I = 1       2          3         4         5         6         7

  .015000  .369423  .004296  -.027821   .014213   .020149   .005272   .079992   .003899

   CL AND CD FROM ALGEBRAIC COMBINATION OF LOADINGS ARE          .100000000         .003694229
```

MINIMUM OF $(C_{P_{\text{UPPER SURFACE}}} - C_{P_{\text{LIMIT}}}) =$ .0059 AT 70.0000 PERCENT SEMISPAN AND 0.0000 PERCENT CHORD

DELTAT = .445 SEC., T = 251.896 SEC.

CAMBER SURFACES WILL BE CALCULATED FOR OPTION FLAGS GREATER THAN 1.

THE OPTION FLAGS ARE 1 1 1 3

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

CAMBER SURFACE CORRESPONDING TO OPTION 4

WING DESIGN 7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$\frac{Y}{B/2}$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8844000	.0120815	.0628434	-.0386852
.0500000	172.0087010	.0058188	.0673265	-.0486681
.1000000	160.1330000	.0042313	.0745697	-.0624364
.2000000	136.3933123	.0034472	.0820059	-.0917101
.3000000	113.9394744	.0023880	.0903103	-.1368193
.4000000	92.5970882	.0019675	.1022516	-.2101913
.5000000	72.1611317	.0013754	.1095467	-.3116996
.6000000	54.0214637	-.0003815	.1189513	-.4874020
.7000000	35.8917958	-.0024269	.1286438	-.8519349
.8000000	27.3627760	.0021708	.1295006	-1.1679725
.9000000	20.9036880	.0049164	.1314056	-1.6100835
1.0000000	14.4450000	.0001665	.1226355	-2.2519781

$C_L =$ .100000 $C_D =$ .003694 $\frac{X_{CP}}{L} =$ .662204 $K_E =$ .369423

$\frac{S_{REF}}{S_{PROG}} =$ .919341 $\frac{C_M}{C_L} =$ .060585 $C_{M_0} =$ .015000

$C_{D_{\text{WING-ON-NACELLE(S)}}} =$ .000156

TABLE OF INTERPOLATED ORDINATES FROM DESIGN PROGRAM (Z/C, PER CENT)

XPCT	0.00 90.00	5.00 100.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00
Y/B/2										
0.0000	0.00000 -14.70198	-.74415 -15.29110	-2.02164	-4.49123	-6.69671	-8.52011	-10.28533	-11.71095	-12.91591	-13.31893
.0500	0.00000 -7.66569	-.12104 -8.14319	-.55773	-1.62639	-2.71841	-3.75894	-4.72442	-5.60392	-6.39106	-7.07978
.1000	0.00000 -5.58244	.08951 -6.10764	-.16497	-.90010	-1.69577	-2.48856	-3.24499	-3.95045	-4.59753	-5.17684
.2000	0.00000 -4.50173	.08960 -4.94170	-.05979	-.53553	-1.11977	-1.72420	-2.33148	-2.92150	-3.48475	-4.01423
.3000	0.00000 -3.25116	.28145 -3.66767	.27214	-.03541	-.44740	-.90790	-1.38870	-1.87000	-2.34726	-2.80849
.4000	0.00000 -2.23968	.14636 -2.66108	.25281	.11749	-.14006	-.46438	-.81474	-1.18030	-1.55544	-1.92336
.5000	0.00000 -1.74222	.32522 -2.11346	.50851	.45809	.23498	-.02783	-.33224	-.67014	-1.02147	-1.37736
.6000	0.00000 -.03042	.01648 -.21119	.43040	.64228	.70362	.55600	.57824	.43855	.30112	.14182
.7000	0.00000 1.42270	.31783 1.43728	.58652	.97647	1.14175	1.21063	1.27495	1.33560	1.35786	1.39782
.8000	0.00000 -1.10801	-.45395 -1.14500	-.78663	-.95695	-.96250	-.94417	-.99348	-1.03268	-1.04004	-1.07078
.9000	0.00000 -3.26027	-.30131 -3.48038	-.58965	-1.12748	-1.64316	-2.05742	-2.29615	-2.52455	-2.77939	-3.02681
1.0000	0.00000 -.15076	-.19035 .13814	-.35730	-.62100	-.79111	-.85722	-.80039	-.70049	-.56135	-.38412

PUNCHED ORDINATES HAVE BEEN REQUESTED. CHORDWISE AND SPANWISE LOCATIONS OF ORDINATES ARE PUNCHED FIRST.
AN IMAGE OF THE PUNCHED DECK FOLLOWS.

```
WING DESIGN   7 LOADINGS INCLUDING FUSELAGE AND NACELLE LOADS          OPTION 4
  0.000  5.000 10.000 20.000 30.000 40.000 50.000 60.000 70.000 80.000
 90.000100.000
  0.000  5.000 10.000 20.000 30.000 40.000 50.000 60.000 70.000 80.000
 90.000100.000
  0.000  -.744 -2.022 -4.491 -6.637 -8.620-10.285-11.711-12.916-13.911
-14.702-15.291
  0.000  -.121  -.558 -1.626 -2.718 -3.759 -4.724 -5.604 -6.391 -7.080
 -7.666 -8.143
  0.000   .090  -.165  -.900 -1.636 -2.489 -3.245 -3.950 -4.598 -5.177
 -5.682 -6.108
```

```
  0.000   .090  -.060  -.536 -1.120 -1.724 -2.331 -2.922 -3.485 -4.014
-4.502 -4.942
  0.000   .281   .272  -.039  -.447  -.908 -1.399 -1.870 -2.347 -2.808
-3.251 -3.668
  0.000   .146   .253   .117  -.140  -.464  -.815 -1.190 -1.555 -1.929
-2.300 -2.661
  0.000   .325   .509   .459   .235  -.028  -.332  -.670 -1.021 -1.378
-1.742 -2.113
  0.000   .016   .430   .642   .704   .656   .579   .439   .301   .142
 -.030  -.211
  0.000   .318   .587   .976  1.142  1.211  1.275  1.336  1.358  1.398
 1.423  1.437
  0.000  -.454  -.787  -.957  -.962  -.944  -.933 -1.033 -1.040 -1.071
-1.108 -1.145
  0.000  -.301  -.590 -1.127 -1.643 -2.057 -2.295 -2.525 -2.780 -3.027
-3.260 -3.480
  0.000  -.190  -.357  -.621  -.791  -.857  -.800  -.700  -.562  -.384
 -.151   .178
                                                                          DELTAT =     .100 SEC., T =   284.806 SEC.

END OF DATA *** STOP

                              ----TOTAL ELAPSED TIME, CP= 138.066 ----

PROGRAM CONTROL CARD
WGUP
ENTER INPTS---TAPE INPUTS
EXIT INPTS
ENTER WRGEOM---WRITE GEOMETRY ON TAPE
EXIT WRGEOM

                                  UPDATED WING DEFINITION

                        WING CAMBER SURFACE READ INTO BASIC GEOMETRY
```

**** **** **** **** **** WING **** **** **** **** ****

REFA = 9898.0000 CBAR = 105.4100 XBARIN = 187.0000

	XO = 76.5900			XO = 83.1040			XO = 93.1650		
	YO = 4.7570			YO = 6.6250			YO = 9.5100		
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000		
	CHORD = 165.9300			CHORD = 160.1330			CHORD = 149.7900		
PERCENT	CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS	
CHORD	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	-.0257	.5700	.5700	.0717	.5700	.5700	.0671	.5500	.5500
5.0	-.0514	.7140	.7140	.1433	.7140	.7140	.1341	.7120	.7120
10.0	-.6399	.8720	.8720	-.2642	.8720	.8720	-.1846	.8720	.8720
20.0	-2.1585	1.0500	1.0500	-1.4414	1.0500	1.0500	-1.1318	1.0540	1.0540
30.0	-3.7415	1.1450	1.1450	-2.7155	1.1450	1.1450	-2.1981	1.1560	1.1560
40.0	-5.2742	1.2000	1.2000	-3.9850	1.2000	1.2000	-3.2737	1.2130	1.2130
50.0	-6.7107	1.2300	1.2300	-5.1963	1.2300	1.2300	-4.3183	1.2350	1.2350
60.0	-8.0307	1.2490	1.2490	-6.3260	1.2490	1.2490	-5.3064	1.2370	1.2370
70.0	-9.2232	1.1700	1.1700	-7.3623	1.1700	1.1700	-6.2260	1.1270	1.1270
80.0	-10.2756	.9370	.9370	-8.2898	.9370	.9370	-7.0641	.8830	.8830
90.0	-11.1799	.5460	.5460	-9.0935	.5460	.5460	-7.8107	.5070	.5070
100.0	-11.9260	0.0000	0.0000	-9.7803	0.0000	0.0000	-8.4564	0.0000	0.0000

	XO = 116.3600			XO = 168.9800			XO = 225.8100		
	YO = 16.3330			YO = 31.2300			YO = 47.5440		
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000		
	CHORD = 125.3500			CHORD = 77.2350			CHORD = 32.6810		
PERCENT	CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS	
CHORD	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	.1073	.5500	.5500	.1033	.5700	.5700	.0360	.5800	.5800
5.0	.2145	.7150	.7150	.2066	.7270	.7270	.0720	.7290	.7290
10.0	.1007	.8760	.8760	.3293	.9020	.9020	.1353	.9110	.9110
20.0	-.4093	1.1260	1.1260	.2678	1.0980	1.0980	.2423	1.1340	1.1340
30.0	-1.0538	1.1740	1.1740	.0849	1.2200	1.2200	.2909	1.2680	1.2680
40.0	-1.7387	1.2350	1.2350	-.1361	1.2890	1.2890	.3122	1.3430	1.3430
50.0	-2.4365	1.2500	1.2500	-.3854	1.3150	1.3150	.3288	1.3750	1.3750
60.0	-3.1229	1.2290	1.2290	-.6560	1.2620	1.2620	.3448	1.3200	1.3200
70.0	-3.7857	1.0870	1.0870	-.9361	1.1050	1.1050	.3510	1.1550	1.1550
80.0	-4.4164	.8400	.8400	-1.2186	.8420	.8420	.3614	.8800	.8800
90.0	-5.0065	.4740	.4740	-1.5041	.4730	.4730	.3669	.4950	.4950
100.0	-5.5483	0.0000	0.0000	-1.7908	0.0000	0.0000	.3694	0.0000	0.0000

**** **** **** **** **** WING **** **** **** **** ****

	XO = 225.9100			XO = 258.2100		
	YO = 47.5450			YO = 66.2500		
	ZO = 0.0000			ZO = 0.0000		
	CHORD = 32.6810			CHORD = 14.4450		
PERCENT	CAMBER	HALF-THICKNESS		CAMBER	HALF-THICKNESS	
CHORD	(Z)	UPPER	LOWER	(Z)	UPPER	LOWER
0.0	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2.5	.0360	.1340	.1340	-.0137	.1340	.1340
5.0	.0720	.2610	.2610	-.0275	.2610	.2610
10.0	.1393	.4950	.4950	-.0516	.4910	.4910
20.0	.2423	.8800	.8800	-.0897	.8800	.8800
30.0	.2909	1.1550	1.1550	-.1143	1.1550	1.1550
40.0	.3121	1.3200	1.3200	-.1238	1.2850	1.2850
50.0	.3287	1.3750	1.3750	-.1156	1.3750	1.3750
60.0	.3447	1.3200	1.3200	-.1012	1.3200	1.3200
70.0	.3509	1.1550	1.1550	-.0812	1.1550	1.1550
80.0	.3612	.8800	.8800	-.0555	.8800	.8800
90.0	.3668	.4950	.4950	-.0218	.4950	.4950
100.0	.3693	0.0000	0.0000	.0200	0.0000	0.0000

ANALYSIS OF DRAG-DUE-TO-LIFT M=2.7

MACH NO.= 2.70000 XMAX= 272.65500 NON= 40 CBAR= 106.41000 XBAR= 187.00000

TIFZC= 1.00 INOM= 0.00 SYMM= -0.00 SMOGO= -0.00

NOPCT= 12 JBYMAX= 12 RATIO= 4.153854

	XPCT		YB2
1	0.000	1	0.000
2	5.000	2	5.000
3	10.000	3	10.000
4	20.000	4	20.000
5	30.000	5	30.000
6	40.000	6	40.000
7	50.000	7	50.000
8	60.000	8	60.000
9	70.000	9	70.000
10	80.000	10	80.000
11	90.000	11	90.000
12	100.000	12	100.000

PLANFORM BREAKPOINTS

	X	Y	Z	CHORD	AUX. CHORD
1	76.5900	0.0000	0.0000	165.8300	166.8300
2	76.5900	4.7570	0.0000	165.8300	166.8300
3	83.1040	6.6250	0.0000	160.1330	160.1330
4	93.1650	9.5100	0.0000	149.7900	149.7900
5	116.9600	16.3330	0.0000	125.3500	125.3500
6	168.9800	31.2500	0.0000	77.2950	77.2950
7	225.8100	47.5440	0.0000	32.6810	32.6810
8	225.8100	47.5450	0.0000	32.6810	32.6810
9	258.2100	66.2500	0.0000	14.4450	14.4450

	XLE	XTE	AUX XTE
0	76.5900	243.4200	243.4200
1	76.5900	243.4200	243.4200
2	76.5900	243.4200	243.4200
3	77.3284	243.3993	243.3993
4	83.1040	243.2370	243.2370
5	88.8799	243.0751	243.0751
6	94.6559	242.9146	242.9146
7	100.4320	242.7550	242.7580
8	105.2081	242.6014	242.5014
9	111.9843	242.4449	242.4449
10	117.7603	242.3710	242.3710
11	123.5362	242.8112	242.8112
12	129.3120	243.2515	243.2515
13	135.0878	243.6917	243.6917
14	140.8637	244.1320	244.1320
15	146.6395	244.5722	244.5722
16	152.4153	245.0124	245.0124
17	158.1912	245.4527	245.4527
18	163.9670	245.8929	245.8929
19	169.7430	246.4390	246.4390
20	175.5196	247.6807	247.6807
21	181.2962	248.9225	248.9225
22	187.0729	250.1642	250.1642
23	192.8495	251.4059	251.4059
24	198.6262	252.6477	252.6477
25	204.4028	253.8894	253.8894
26	210.1795	255.1311	255.1311
27	215.9561	256.3728	256.3728
28	221.7328	257.6146	257.6146
29	226.6523	258.8592	258.8592
30	229.5211	260.1134	260.1134
31	232.3900	261.3675	261.3675
32	235.2589	262.6217	262.6217
33	238.1278	263.8759	263.8759
34	240.9967	265.1300	265.1300
35	243.8656	266.3842	266.3842
36	246.7345	267.6383	267.6383
37	249.6033	268.8925	268.8925
38	252.4722	270.1467	270.1467
39	255.3411	271.4008	271.4008
40	258.2100	272.6550	272.6550

HORIZONTAL TAIL PLANFORM

	X	Y	Z	CHORD
1	261.0000	2.0000	-14.0000	25.0000
2	277.0000	11.0000	-14.0000	9.0000

BY	HXLE	HXTE
1	260.3889	285.0000
2	263.3333	286.0000
3	266.2778	286.0000
4	269.2222	286.0000
5	272.1667	286.0000
6	275.1111	286.0000
7	278.0556	286.0000

FUSELAGE DEFINITION

X	RAD	AREA	Z
0.00000	0.00000	0.00000	10.00000
16.67000	2.73501	23.50000	8.55000
33.33000	4.27818	57.50000	7.10000
50.00000	5.32255	89.00000	5.64000
66.67000	6.10254	117.00000	4.17000
83.33000	6.33331	126.00000	2.73000
100.00000	6.17523	119.80000	1.28000
116.67000	5.86323	108.00000	-.14000
133.33000	5.78122	105.00000	-1.60000
150.00000	5.83602	107.00000	-3.04000
166.66000	5.83602	107.00000	-4.50000
183.33000	5.80859	106.00000	-5.90000
200.00000	5.69804	102.00000	-7.40000
216.67000	5.47002	94.00000	-8.85000
233.33000	5.01463	79.00000	-10.25000
250.00000	4.33362	59.00000	-11.70000
266.67000	3.24102	33.00000	-13.20000
283.33000	1.59577	8.00000	-14.60000
295.00000	0.00000	0.00000	-15.70000

NACELLE GEOMETRY

ORIGIN (X,Y,Z)			X	RADIUS	AREA
213.42000	15.33000	-5.80000	0.00000	2.86500	25.78696
			2.00800	2.93300	27.95485
			15.47000	3.63300	41.46500
			21.52500	3.77000	44.65125
			28.01700	3.65400	41.94575
			32.06700	3.42000	36.74541
			35.04000	3.42000	36.74541

ORIGIN (X,Y,Z)			X	RADIUS	AREA
218.67000	31.25000	-4.90000	0.00000	2.86500	25.78696
			2.00800	2.93300	27.95486
			15.47000	3.63300	41.46500
			21.52500	3.77000	44.65125
			28.01700	3.65400	41.94575
			32.06700	3.42000	36.74541
			35.04000	3.42000	36.74541

TABLE OF INPUT Z/C ORDINATES

XPCT	0.00 90.00	5.00 100.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00
Y/B/2										
0.0000	0.00000 -14.70198	-.74415 -15.29110	-2.02164	-4.49123	-6.69671	-8.52011	-10.28533	-11.71095	-12.31531	-13.31093
.0500	0.00000 -7.66569	-.12104 -8.14319	-.55773	-1.62639	-2.71841	-3.75834	-4.72442	-5.60392	-6.39106	-7.07978
.1000	0.00000 -5.68244	.08951 -6.10764	-.16437	-.90010	-1.69577	-2.48856	-3.24499	-3.95045	-4.59753	-5.17684
.2000	0.00000 -4.50173	.03950 -4.94170	-.05973	-.53553	-1.11977	-1.72420	-2.33148	-2.92150	-3.48475	-4.01423
.3000	0.00000 -3.25116	.28145 -3.66767	.27214	-.03541	-.44740	-.90790	-1.38870	-1.87000	-2.34726	-2.80849
.4000	0.00000 -2.29968	.14636 -2.66108	.25281	.11749	-.14006	-.46438	-.81474	-1.18030	-1.55544	-1.92936
.5000	0.00000 -1.74222	.32522 -2.11346	.50851	.45809	.23498	-.02783	-.33224	-.57014	-1.02147	-1.37796
.6000	0.00000 -.03042	.01648 -.21119	.43040	.64228	.70362	.65600	.57824	.43855	.30112	.14182
.7000	0.00000 1.42270	.31783 1.43728	.58652	.97647	1.14175	1.21063	1.27495	1.33560	1.35786	1.39782
.8000	0.00000 -1.10801	-.45395 -1.14500	-.78663	-.95695	-.96250	-.94417	-.99348	-1.03268	-1.04004	-1.07078
.9000	0.00000 -3.26027	-.30131 -3.48038	-.58965	-1.12748	-1.64316	-2.05742	-2.29615	-2.52455	-2.77999	-3.12681
1.0000	0.00000 -.15076	-.19035 .13814	-.35730	-.62100	-.79111	-.85722	-.80039	-.70049	-.56135	-.38412

WING-FUSELAGE INTERSECTION

CHORD	X	Y	Z
0.00	75.6305	4.7600	0.0000
5.00	84.9414	4.7600	-.0511
10.00	93.2824	4.7600	-.6393
20.00	109.9643	4.7600	-2.1575
30.00	125.6462	4.7600	-3.7400
40.00	143.3232	4.7600	-5.2721
50.00	160.0101	4.7600	-6.7083
60.00	176.6920	4.7600	-8.0280
70.00	193.3739	4.7600	-9.2202
80.00	210.0559	4.7600	-10.2724
90.00	226.7378	4.7600	-11.1766
100.00	243.4197	4.7600	-11.9226

FUSELAGE UPWASH ACTING ON WING AT ALPHA= 0.00 DEG.

XPCT Y/B/2	0.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	-3.685	-2.401	-1.451	-1.471	-2.078	-2.210	-1.935	-1.653	-1.411	-.950	-.268
.025	-3.685	-2.401	-1.451	-1.471	-2.078	-2.210	-1.935	-1.653	-1.411	-.950	-.268
.050	1.723	3.950	4.947	4.594	4.093	4.442	4.617	5.302	5.792	5.842	5.719
.075	2.142	3.336	4.855	4.973	4.784	5.151	5.485	5.960	6.282	5.885	3.173
.100	2.404	3.503	3.839	3.701	3.623	3.753	3.812	3.780	3.756	3.251	2.507
.125	2.065	2.650	2.694	2.517	2.437	2.454	2.464	2.336	2.345	2.031	1.545
.150	1.643	1.955	1.915	1.763	1.705	1.695	1.699	1.565	1.589	1.394	1.073
.175	1.303	1.478	1.399	1.279	1.242	1.229	1.230	1.118	1.128	1.010	.805
.200	1.047	1.133	1.047	.959	.937	.928	.933	.853	.830	.767	.631
.250	.705	.709	.633	.584	.581	.581	.579	.545	.505	.499	.432
.300	.501	.473	.409	.384	.391	.393	.392	.379	.349	.333	.312
.350	.365	.332	.273	.269	.280	.284	.287	.282	.263	.244	.239
.400	.267	.242	.197	.197	.208	.213	.221	.215	.208	.195	.182
.450	.201	.180	.147	.150	.160	.164	.172	.173	.169	.161	.152
.500	.154	.135	.113	.116	.125	.130	.137	.142	.139	.136	.129
.550	.119	.099	.089	.092	.100	.105	.110	.116	.115	.113	.110
.600	.091	.072	.070	.074	.080	.085	.089	.094	.099	.097	.095
.700	.045	.046	.049	.051	.055	.059	.061	.063	.066	.069	.072
.800	.040	.033	.031	.033	.035	.037	.040	.043	.045	.047	.048
.900	.038	.036	.032	.028	.023	.022	.024	.025	.025	.028	.030
1.000	.034	.033	.032	.030	.029	.028	.025	.023	.020	.018	.015

INCREMENTAL FUSELAGE UPWASH ON WING PER DEGREE ALPHA

XPCT	0.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
Y/B/2											
0.000	-.287	-.472	-.510	-.471	-.438	-.416	-.391	-.360	-.325	-.285	-.237
.025	-.287	-.472	-.510	-.471	-.438	-.416	-.390	-.360	-.326	-.285	-.237
.050	.683	.807	.824	.798	.827	.902	.948	1.012	1.055	1.016	.898
.075	.667	.941	.967	.910	.951	1.043	1.117	1.184	1.209	1.163	1.020
.100	.599	.715	.749	.717	.723	.756	.772	.770	.739	.676	.570
.125	.472	.531	.528	.496	.486	.494	.495	.485	.462	.424	.367
.150	.359	.391	.381	.353	.341	.340	.339	.329	.313	.290	.258
.175	.276	.294	.283	.260	.249	.246	.244	.236	.225	.211	.189
.200	.217	.226	.217	.198	.189	.185	.183	.177	.170	.159	.145
.250	.142	.143	.135	.125	.118	.115	.114	.111	.107	.102	.095
.300	.099	.098	.093	.085	.081	.078	.077	.075	.073	.070	.067
.350	.073	.070	.067	.062	.059	.057	.056	.055	.054	.052	.050
.400	.055	.053	.050	.047	.045	.043	.042	.042	.041	.040	.039
.450	.043	.041	.039	.037	.035	.034	.033	.033	.033	.032	.032
.500	.034	.033	.031	.030	.028	.028	.027	.027	.027	.026	.026
.550	.028	.027	.026	.024	.023	.023	.022	.022	.022	.022	.022
.600	.023	.022	.021	.020	.020	.019	.019	.018	.018	.018	.018
.700	.016	.016	.015	.015	.014	.014	.014	.014	.014	.013	.013
.800	.013	.012	.012	.012	.012	.011	.011	.011	.011	.011	.011
.900	.010	.010	.010	.010	.010	.010	.009	.009	.009	.009	.009
1.000	.008	.008	.008	.008	.008	.008	.008	.008	.008	.008	.008

FUSELAGE UPWASH ACTING ON TAIL AT ALPHA= 0.00 DEG.

XPCT	0.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
Y/B/2											
0.000	1.653	7.319	14.273	22.134	28.658	30.349	25.534	17.932	8.322	1.008	-3.094
.100	1.653	7.319	14.273	22.134	28.658	30.349	25.534	17.932	8.322	1.008	-3.094
.200	9.126	10.482	11.403	11.617	11.156	9.863	7.905	5.900	3.921	2.106	.511
.300	6.853	6.819	6.594	6.155	5.567	4.886	3.975	3.042	2.224	1.469	.770
.400	4.554	4.332	4.055	3.723	3.339	2.930	2.521	2.014	1.533	1.113	.723
.500	3.101	2.921	2.714	2.493	2.250	1.996	1.743	1.498	1.207	.924	.666
.600	2.206	2.087	1.942	1.792	1.639	1.475	1.310	1.149	.995	.821	.640
.700	1.635	1.558	1.462	1.359	1.255	1.152	1.043	.933	.827	.721	.620
.800	1.256	1.201	1.145	1.075	1.003	.932	.861	.789	.715	.638	.567
.900	.992	.954	.915	.875	.829	.779	.730	.682	.633	.582	.530
1.000	.797	.776	.753	.723	.696	.668	.634	.601	.567	.533	.508

INCREMENTAL FUSELAGE UPWASH ON TAIL PER DEGREE ALPHA

XPCT	0.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
Y/B/2											
0.000	-.897	.078	1.525	3.321	5.025	5.899	5.544	4.143	2.469	1.163	.358
.100	-.897	.078	1.526	3.321	5.025	5.899	5.544	4.143	2.469	1.163	.358
.200	1.228	1.939	1.803	1.970	2.034	1.927	1.678	1.369	1.046	.746	.486
.300	1.074	1.106	1.110	1.086	1.039	.972	.851	.719	.589	.468	.356
.400	.753	.733	.705	.672	.633	.589	.541	.470	.401	.336	.275
.500	.530	.508	.485	.459	.432	.403	.374	.345	.304	.264	.226
.600	.389	.372	.354	.336	.318	.299	.280	.261	.242	.219	.194
.700	.297	.285	.272	.259	.247	.234	.221	.208	.195	.184	.171
.800	.235	.226	.217	.208	.199	.190	.181	.173	.164	.156	.147
.900	.190	.184	.178	.172	.166	.159	.153	.147	.142	.136	.130
1.000	.157	.154	.149	.145	.141	.137	.133	.129	.124	.120	.116

FUSELAGE PRESSURE ACTING ON WING MID-WING

XPCT Y/B/2	0.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	-.0077	-.0194	-.0195	-.0049	.0007	-.0019	-.0028	-.0053	-.0084	-.0135	-.0162
.025	-.0077	-.0194	-.0196	-.0049	.0007	-.0019	-.0028	-.0053	-.0084	-.0135	-.0162
.050	-.0077	-.0194	-.0195	-.0049	.0007	-.0019	-.0028	-.0053	-.0084	-.0135	-.0162
.075	-.0073	-.0193	-.0193	-.0048	.0007	-.0019	-.0029	-.0054	-.0084	-.0136	-.0162
.100	-.0111	-.0183	-.0167	-.0039	.0002	-.0020	-.0027	-.0052	-.0083	-.0132	-.0159
.125	-.0109	-.0165	-.0149	-.0036	.0003	-.0017	-.0024	-.0047	-.0070	-.0112	-.0128
.150	-.0108	-.0151	-.0135	-.0033	.0004	-.0015	-.0022	-.0039	-.0059	-.0083	-.0110
.175	-.0109	-.0140	-.0125	-.0031	.0004	-.0013	-.0020	-.0033	-.0048	-.0064	-.0103
.200	-.0110	-.0132	-.0115	-.0029	.0005	-.0012	-.0018	-.0027	-.0035	-.0059	-.0091
.250	-.0113	-.0119	-.0102	-.0027	.0004	-.0009	-.0014	-.0018	-.0033	-.0045	-.0054
.300	-.0110	-.0109	-.0085	-.0025	.0002	-.0005	-.0012	-.0016	-.0024	-.0031	-.0046
.350	-.0099	-.0102	-.0074	-.0023	.0001	-.0002	-.0010	-.0013	-.0017	-.0027	-.0034
.400	-.0092	-.0095	-.0062	-.0021	-.0000	-.0000	-.0003	-.0011	-.0013	-.0019	-.0025
.450	-.0088	-.0090	-.0050	-.0020	-.0001	.0001	-.0006	-.0009	-.0012	-.0013	-.0020
.500	-.0085	-.0082	-.0039	-.0019	-.0002	.0002	-.0003	-.0008	-.0010	-.0012	-.0013
.550	-.0081	-.0072	-.0032	-.0018	-.0003	.0003	-.0001	-.0007	-.0008	-.0010	-.0011
.600	-.0078	-.0060	-.0026	-.0016	-.0004	.0003	.0000	-.0004	-.0007	-.0008	-.0009
.700	-.0050	-.0027	-.0018	-.0014	-.0006	.0000	.0003	.0001	-.0002	-.0005	-.0006
.800	-.0062	-.0054	-.0030	-.0021	-.0016	-.0014	-.0008	-.0001	.0001	.0003	.0001
.900	-.0063	-.0064	-.0052	-.0057	-.0050	-.0031	-.0023	-.0016	-.0015	-.0013	-.0018
1.000	-.0059	-.0059	-.0060	-.0050	-.0060	-.0060	-.0061	-.0058	-.0055	-.0052	-.0043

NACELLES BELOW WING WITH ORIGINS AT

X= 213.42000 Y= 16.33000 Z= -5.60000
X= 219.67000 Y= 31.25000 Z= -4.90000

FOR NACELLE(S) AT X= 213.42000 Y= 16.33000 Z= -5.60000

X	R	AREA	CP	Y	F(Y)
213.420000	2.865000	25.786902	.044364	206.234617	0.000000
214.296000	2.917145	26.734131	.044364	206.979940	.071776

215.172000	2.968253	27.679093	.041510	207.727055	.088660
216.048000	3.023389	28.716925	.044722	208.465398	.089467
216.924000	3.078753	29.778283	.041297	209.202729	.087381
217.800000	3.132102	30.819232	.038072	209.944909	.082733
218.676000	3.183437	31.837746	.035034	210.692015	.077956
219.552000	3.232747	32.831689	.031741	211.444536	.073240
220.428000	3.290033	33.799196	.028018	212.201305	.068517
221.304000	3.325305	34.738638	.025436	212.954233	.063831
222.180000	3.368561	35.648295	.023974	213.731900	.059237
223.056000	3.409802	36.526524	.021627	214.504415	.054766
223.932000	3.449029	37.371759	.019384	215.281337	.050371
224.808000	3.486240	38.182510	.017069	216.064793	.046047
225.684000	3.521422	38.957048	.014794	216.852498	.041782
226.560000	3.554586	39.694273	.012784	217.645235	.037574
227.436000	3.585734	40.392995	.010884	218.443238	.033948
228.312000	3.614858	41.052037	.009062	219.246088	.032089
229.188000	3.645250	41.745016	.015208	220.045335	.034675
230.064000	3.678046	42.499407	.010698	220.840090	.031368
230.940000	3.705878	43.145178	.005120	221.646124	.023316
231.816000	3.728732	43.678952	.001725	222.464699	.013680
232.692000	3.746538	44.099478	-.002307	223.296016	.004201
233.568000	3.759598	44.405086	-.006083	224.139233	-.004928
234.444000	3.767511	44.594568	-.009717	224.995135	-.012406
235.320000	3.771353	44.697421	-.010834	225.850635	-.019351
236.196000	3.772455	44.709314	-.015373	226.735129	-.028106
237.072000	3.767133	44.583267	-.019720	227.624547	-.037803
237.948000	3.755979	44.319647	-.023882	228.528697	-.047459
238.824000	3.739023	43.920381	-.027744	229.446807	-.056711
239.700000	3.716263	43.387321	-.031271	230.380025	-.065428
240.576000	3.687701	42.722959	-.034632	231.327916	-.081908
241.452000	3.651176	41.880845	-.052835	232.298647	-.104360
242.328000	3.579379	40.249953	-.060489	233.349797	-.115289
243.204000	3.518753	38.898023	-.046477	234.377576	-.100671
244.080000	3.470759	37.844383	-.032996	235.375019	-.072147
244.956000	3.435395	37.076899	-.020238	236.339130	-.045849
245.832000	3.412632	36.587164	-.008520	237.271855	-.021072
246.708000	3.402478	36.369766	.002926	238.174245	.002586
247.584000	3.404934	36.422293	.013989	239.043270	.018817
248.460000	3.420030	36.745328	.023086	239.889816	.021834

FOR NACELLE(S) AT X= 218.67000 Y= 31.25000 Z= -4.30000

NACELLE PRESSURE FIELD

X, PER CENT CHORD AND PRESSURE COEFFICIENT
WRAP SOLUTION

NACELLES BELOW WING

Y/B/2												
0.000	76.590	243.420										
	0.000	100.000										
	0.00000	0.00000										
.050	76.590	238.690	238.700	238.996	239.292	239.589	239.885	240.181	240.477	240.773	241.069	241.366
	241.662	241.958	242.254	242.550	242.847	243.143	243.439	243.735				
	0.000	97.155	97.171	97.348	97.526	97.703	97.881	98.058	98.236	98.414	98.591	98.759
	98.945	99.124	99.301	99.479	99.656	99.834	100.011	100.189				
	0.00000	0.00000	.03894	.03836	.03778	.03720	.03662	.03605	.03548	.03491	.03434	.03377
	.03320	.03263	.03205	.03149	.03093	.03037	.02981	.02925				
.100	83.104	231.692	231.702	232.424	233.146	233.868	234.590	235.312	236.035	236.757	237.479	238.201
	238.923	239.645	240.367	241.090	241.812	242.534	243.256	243.875				
	0.000	92.730	92.795	93.247	93.698	94.149	94.600	95.051	95.502	95.953	96.404	96.855
	97.305	97.757	98.209	98.659	99.110	99.561	100.012	100.398				
	0.00000	0.00000	.04458	.04294	.04129	.03967	.03804	.03642	.03481	.03322	.03165	.03010
	.02858	.02706	.02556	.02408	.02261	.02115	.01971	.01848				
.150	94.656	225.394	225.404	226.499	227.595	228.690	229.785	230.882	231.977	233.073	234.169	235.264
	236.360	237.455	238.551	239.647	240.742	241.838	242.934	244.029				
	0.000	88.182	88.189	88.928	89.667	90.406	91.145	91.884	92.623	93.362	94.101	94.840
	95.579	96.318	97.057	97.796	98.535	99.274	100.013	100.752				
	0.00000	0.00000	.05210	.04913	.04616	.04322	.04030	.03741	.03461	.03186	.02915	.02648
	.02385	.02126	.01949	.02071	.01821	.01377	.00937	.00503				
.200	106.208	220.585	220.595	221.972	223.348	224.725	226.101	227.478	228.855	230.231	231.608	232.984
	234.361	235.738	237.114	238.491	239.867	241.244	242.620	243.846				
	0.000	83.858	83.865	84.875	85.884	86.893	87.903	88.912	89.921	90.930	91.940	92.949
	93.958	94.968	95.977	96.966	97.995	99.005	100.014	100.913				
	0.00000	0.00000	.06389	.05545	.05202	.04752	.04323	.03909	.03500	.03100	.02709	.02337
	.02379	.02120	.01450	.00784	.00131	-.00484	-.01032	-.01497				
.246	116.925	218.815	218.825	220.234	221.763	223.232	224.701	226.170	227.639	229.108	230.577	232.047
	233.516	234.985	236.454	237.923	239.392	240.861	242.330	243.541				
	0.000	81.261	81.269	82.441	83.612	84.784	85.955	87.127	88.299	89.470	90.642	91.814
	92.985	94.157	95.329	96.500	97.672	98.844	100.015	100.981				
	0.00000	0.00000	.06530	.06020	.05508	.05001	.04503	.04025	.03557	.03100	.02653	.02392
	.02370	.01693	.00914	.00153	-.00557	-.01187	-.01837	-.02450				

.247	116.973	218.815	218.825	220.294	221.763	223.232	224.701	226.170	227.639	229.108	230.578	232.047
	233.515	234.985	236.454	237.923	239.392	240.861	242.330	243.541				
	0.000	81.254	81.262	82.434	83.606	84.778	85.950	87.122	88.294	89.467	90.639	91.811
	92.983	94.155	95.327	96.499	97.671	98.843	100.015	100.981				
[illegible]	0.00000	0.00000	.06530	.06020	.05508	.05001	.04503	.04025	.03557	.03100	.02653	.02382
	.02370	.01693	.00914	.00153	-.00557	-.01187	-.01837	-.02450				
.250	117.760	218.826	218.835	220.308	221.780	223.252	224.724	226.196	227.669	229.141	230.613	232.085
	233.557	235.029	236.501	237.974	239.446	240.918	242.390	243.553				
	0.000	81.105	81.113	82.294	83.476	84.657	85.839	87.020	88.201	89.383	90.564	91.746
	92.927	94.108	95.290	96.471	97.652	98.834	100.015	100.949				
[illegible]	0.00000	0.00000	.06527	.06017	.05504	.04996	.04497	.04018	.03550	.03092	.02645	.02399
	.02363	.01574	.00895	.00133	-.00575	-.01205	-.01860	-.02449				
.300	129.312	221.119	221.129	222.710	224.292	225.873	227.455	229.037	230.618	232.200	233.781	235.363
	236.944	238.526	239.499	239.539	241.091	242.673	243.534	243.534				
	0.000	80.575	80.584	81.972	83.360	84.748	86.136	87.524	88.912	90.300	91.688	93.076
	94.464	95.852	96.767	96.716	98.104	99.492	100.248	100.248				
[illegible]	0.00000	0.00000	.05973	.05472	.04975	.04483	.04004	.03541	.03088	.02648	.02271	.02319
	.01775	.01020	.00563	.04786	.03721	.02746	.02239	.02239				
.350	140.864	226.214	226.224	227.505	228.785	230.065	231.345	232.529	232.539	233.819	235.100	235.380
	237.661	238.941	240.222	241.502	242.783	244.063	244.661	244.661				
	0.000	82.649	82.653	83.899	85.139	86.379	87.613	88.764	88.774	90.014	91.254	92.434
	93.734	94.974	95.214	97.454	98.694	99.934	100.512	100.512				
[illegible]	0.00000	0.00000	.05092	.04754	.04417	.04083	.03753	.03455	.08402	.07760	.07125	.06438
	.05881	.05320	.05105	.04549	.03756	.02970	.02605	.02506				
.400	152.415	226.431	226.441	227.769	229.097	230.425	231.753	232.642	232.652	233.980	235.307	236.635
	237.963	239.291	240.513	241.347	243.275	244.603	245.931	246.037				
	0.000	79.333	79.344	81.378	82.812	84.246	85.680	86.640	86.651	88.085	89.519	90.953
	92.387	93.821	95.255	96.689	98.123	99.557	100.992	101.107				
[illegible]	0.00000	0.00000	.05957	.05540	.05122	.04709	.04300	.04033	.08399	.07713	.07040	.06375
	.05722	.05156	.04981	.04305	.03404	.02523	.01722	.01659				
.450	163.967	222.474	222.484	224.157	225.831	227.504	229.178	230.851	232.524	234.198	235.871	237.545
	239.218	239.706	239.715	241.389	243.063	244.736	246.371	246.371				
	0.000	71.414	71.427	73.469	75.512	77.554	79.597	81.640	83.682	85.725	87.767	89.810
	91.853	92.448	92.460	94.503	96.546	98.588	100.583	100.583				
	0.00000	0.00000	.07015	.06386	.05756	.05133	.04530	.03948	.03383	.02831	.02602	.02401
	.01425	.01146	.04967	.03699	.02510	.01428	.00289	.00288				
.472	168.957	222.002	222.012	223.746	225.481	227.215	228.949	230.683	232.418	234.152	235.886	237.621
	239.355	241.089	242.924	242.870	242.880	244.614	245.349	246.388				

	0.000	68.608	68.521	70.854	73.107	75.350	77.593	79.836	82.080	84.323	86.566	88.809
	91.052	93.295	95.538	95.598	95.611	97.854	100.097	100.148				
	0.00000	0.00000	.07180	.06511	.05841	.05180	.04541	.03926	.03328	.02766	.02812	.02118
	.01081	.00072	-.00937	-.00850	.02772	.01652	.00373	.00344				
.472	169.003	222.002	222.012	223.747	225.481	227.216	228.951	230.686	232.421	234.155	235.890	237.625
	239.360	241.095	242.829	242.899	242.909	244.644	246.373	246.388				
	0.000	68.583	68.595	70.841	73.086	75.331	77.575	79.820	82.065	84.310	86.555	88.800
	91.045	93.290	95.535	95.625	95.638	97.883	100.125	100.139				
	0.00000	0.00000	.07180	.06511	.05840	.05180	.04541	.03925	.03327	.02765	.02813	.02116
	.01078	.00059	-.00940	-.00874	.02755	.01637	.00355	.00348				
.500	175.520	222.794	222.904	224.582	226.361	228.139	229.917	231.595	233.474	235.252	237.030	238.808
	240.587	242.365	244.143	245.921	247.101	247.111	247.822	247.822				
	0.000	65.513	65.525	67.991	70.455	72.919	75.384	77.848	80.312	82.776	85.241	87.735
	90.169	92.534	95.099	97.562	99.196	99.210	100.196	100.196				
	0.00000	0.00000	.06313	.06252	.05594	.04946	.04320	.03718	.03131	.02612	.02621	.01937
	.00827	-.00154	-.01012	-.01821	-.02460	.00952	.00446	.00446				
.550	187.073	227.153	227.163	228.602	230.041	231.480	232.919	234.357	235.795	237.235	238.673	240.112
	241.551	242.989	244.429	245.867	247.306	248.744	250.183	250.576				
	0.000	63.528	63.544	65.824	68.104	70.385	72.665	74.946	77.226	79.506	81.787	84.057
	86.348	88.528	90.909	93.189	95.469	97.750	100.033	100.652				
	0.30000	0.00000	.05833	.05368	.04929	.04494	.04065	.03654	.03251	.02856	.02469	.02167
	.02229	.01739	.01075	.00423	-.00214	-.00772	-.01295	-.01436				
.600	199.625	233.415	233.425	234.627	235.830	237.032	238.235	239.438	240.640	241.843	243.046	244.248
	245.451	246.654	247.855	249.059	250.261	251.464	252.667	253.869				
	0.000	64.397	64.416	66.642	68.868	71.095	73.321	75.547	77.773	79.999	82.225	84.452
	86.678	88.304	91.130	93.357	95.583	97.809	100.035	102.261				
	0.00000	0.00000	.04833	.04539	.04241	.03944	.03653	.03363	.03082	.02807	.02536	.02259
	.02007	.01523	.01912	.01619	.01175	.00738	.00305	-.00119				
.650	210.179	240.457	240.467	241.385	242.302	243.220	244.139	245.056	245.973	246.891	247.809	248.726
	249.644	250.562	251.479	252.397	253.315	254.232	255.150	255.928				
	0.000	67.356	67.379	69.420	71.461	73.503	75.544	77.586	79.627	81.669	83.710	85.752
	87.793	89.835	91.875	93.918	95.959	98.001	100.042	101.772				
	0.00000	0.00000	.04149	.03956	.03765	.03575	.03385	.03198	.03012	.02831	.02652	.02475
	.02301	.02128	.01357	.01788	.01657	.01662	.01647	.01513				
.700	221.733	247.875	247.885	248.494	249.103	249.712	250.322	250.931	251.540	252.150	252.759	253.368
	253.978	254.587	255.195	255.806	256.415	257.024	257.634	258.243				
	0.000	72.855	72.883	74.581	76.279	77.977	79.676	81.374	83.072	84.770	86.468	88.166
	89.864	91.562	93.260	94.959	96.657	98.355	100.053	101.751				

	0.00000	0.00000	.03653	.03540	.03430	.03321	.03212	.03103	.02994	.02887	.02780	.02574
	.02573	.02467	.02365	.02263	.02162	.02062	.01962	.01863				
.750	229.521	255.497	255.507	255.796	256.085	256.374	256.663	256.952	257.241	257.530	257.820	258.109
	258.399	258.687	258.975	259.265	259.554	259.843	260.132	260.421				
	0.000	84.909	84.942	85.887	86.832	87.777	88.722	89.667	90.612	91.557	92.502	93.447
	94.392	95.337	96.282	97.227	98.172	99.117	100.062	101.007				
	0.00000	0.00000	.03275	.03229	.03183	.03137	.03092	.03046	.03000	.02955	.02909	.02854
	.02819	.02773	.02727	.02582	.02637	.02592	.02547	.02503				
.800	235.259	262.522										
	0.000	100.000										
	0.00000	0.00000										
.850	240.997	265.130										
	0.000	100.000										
	0.00000	0.00000										
.900	246.734	267.638										
	0.000	100.000										
	0.00000	0.00000										
.950	252.472	270.147										
	0.000	100.000										
	0.00000	0.00000										
1.000	258.210	272.655										
	0.000	100.000										
	0.00000	0.00000										

DEBUG PARAMETER =10

TABLE OF CAMBER CP AT BASIC ALPHA

XPCT	0.00 90.00	5.00 100.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00
Y/B/2										
0.000	.00208 .03328	.00945 .01989	.02230	.06280	.08005	.07322	.06284	.05607	.04998	.04343
.025	.00401 .03272	.01130 .01855	.02515	.06360	.07978	.07311	.06292	.05604	.04986	.04308
.050	.01173 .03365	.02117 .01725	.03454	.06610	.07889	.07285	.06313	.05585	.04952	.04138
.075	.04449 .02522	.05191 .01414	.05917	.07302	.07739	.07179	.06317	.05530	.04848	.03938
.100	.07978 .02105	.08320 .01214	.08396	.07991	.07399	.06381	.06151	.05288	.04515	.03528
.125	.09517 .02137	.09924 .01325	.09621	.08199	.07505	.06377	.06060	.05194	.04376	.03458
.150	.10531 .02315	.10912 .01553	.10252	.08315	.07510	.06980	.06066	.05203	.04333	.03457
.175	.11484 .02555	.11234 .01825	.10440	.08466	.07586	.06336	.06147	.05263	.04356	.03508
.200	.11844 .02818	.11529 .02108	.10634	.08636	.07719	.07043	.06272	.05355	.04408	.03607
.225	.12211 .03364	.11721 .02392	.10802	.08858	.07845	.07172	.06387	.05446	.04488	.03766
.250	.11986 .03309	.11749 .02643	.10945	.09092	.08003	.07323	.06495	.05539	.04616	.03924
.275	.11877 .03513	.11837 .02873	.11174	.09349	.08184	.07437	.06383	.05522	.04783	.04119
.300	.11763 .03709	.11731 .03066	.11113	.09616	.08359	.07537	.06552	.05751	.05004	.04329
.325	.11647 .03860	.11708 .03266	.11333	.09876	.08476	.07513	.06730	.05952	.05240	.04536
.350	.11885 .03997	.11789 .03458	.11398	.09969	.08568	.07653	.06379	.06136	.05479	.04751
.375	.11989 .04153	.11805 .03678	.11311	.09988	.08620	.07773	.07112	.06432	.05714	.04982

.400	.12184	.11860	.11335	.09990	.08744	.08000	.07367	.06676	.05919	.05274
	.04335	.03910								
.425	.12213	.11695	.11116	.09990	.08991	.08269	.07508	.06893	.06057	.05238
	.04558	.03933								
.450	.12033	.11775	.11326	.10377	.09332	.08530	.07929	.07015	.06181	.05429
	.04900	.04178								
.475	.11943	.11823	.11579	.10733	.09704	.08770	.07929	.07098	.06320	.05632
	.05027	.04434								
.500	.11950	.11862	.11693	.10966	.09994	.08942	.07933	.07157	.06480	.05927
	.05228	.04740								
.525	.11946	.12043	.11886	.11139	.10020	.08939	.07946	.07288	.06624	.05988
	.05421	.05049								
.550	.12164	.11904	.11631	.10750	.09801	.08945	.08985	.07400	.06763	.06164
	.05643	.05150								
.575	.11932	.11703	.11410	.10668	.09951	.08976	.08183	.07519	.06921	.06383
	.05935	.05575								
.600	.11760	.11418	.11144	.10563	.09841	.09039	.08303	.07665	.07133	.06677
	.06286	.06024								
.625	.11470	.11228	.10987	.10325	.09731	.09046	.08418	.07886	.07402	.07031
	.06581	.06236								
.650	.11248	.11103	.10913	.10366	.09848	.09271	.08597	.08199	.07788	.07439
	.07081	.06678								
.675	.11230	.10954	.10785	.10429	.10018	.09575	.09123	.08666	.08236	.07822
	.07442	.07106								
.700	.11227	.11170	.11113	.10860	.10504	.10037	.09554	.09157	.08611	.08117
	.07863	.06992								
.725	.11825	.11763	.11700	.11426	.11024	.10532	.10026	.09485	.08988	.08339
	.07817	.07304								
.750	.11252	.11199	.11146	.11009	.10801	.10422	.09998	.09538	.09040	.08538
	.08070	.07658								
.775	.11264	.11112	.10974	.10740	.10506	.10233	.09982	.09471	.09049	.08526
	.08329	.08032								
.800	.11281	.11111	.10940	.10621	.10316	.10041	.09752	.09441	.09043	.08661
	.08279	.07805								
.825	.11382	.11205	.11029	.10683	.10401	.10093	.09757	.09428	.09089	.08715
	.08386	.08081								
.850	.11615	.11407	.11199	.10783	.10390	.10024	.09709	.09413	.09093	.08778
	.08476	.08174								

.875	.11958	.11808	.11658	.11271	.10809	.10398	.10009	.09650	.09296	.08884
	.08421	.07695								
.900	.11964	.11818	.11672	.11380	.10944	.10473	.10009	.09546	.09117	.08694
	.08203	.07693								
.925	.11159	.11008	.10847	.10525	.10203	.09769	.09331	.08836	.08462	.08090
	.07727	.07373								
.950	.10152	.10004	.09855	.09534	.09162	.08730	.08385	.07979	.07565	.07147
	.06524	.05801								
.975	.08095	.07974	.07863	.07641	.07389	.07078	.06767	.06406	.06029	.05641
	.05201	.04761								
1.000	.05038	.04945	.04852	.04667	.04482	.04288	.04047	.03806	.03577	.03331
	.03185	.02990								

TABLE OF FLAT PLATE CP AT 1 DEG ANGLE OF ATTACK

XPCT	0.00	5.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00
	90.00	100.00								
Y/B/2										
0.000	.00161	.00413	.00869	.01647	.01773	.01690	.01674	.01656	.01613	.01601
	.01545	.01420								
.025	.00220	.00503	.00911	.01643	.01774	.01535	.01572	.01669	.01617	.01534
	.01543	.01418								
.050	.00551	.00805	.01097	.01641	.01773	.01725	.01584	.01674	.01637	.01588
	.01530	.01417								
.075	.01716	.01487	.01446	.01652	.01774	.01752	.01716	.01689	.01660	.01577
	.01484	.01393								
.100	.02996	.02287	.01831	.01737	.01764	.01750	.01724	.01657	.01644	.01553
	.01454	.01331								
.125	.03905	.02737	.02115	.01756	.01786	.01758	.01710	.01670	.01640	.01578
	.01467	.01417								
.150	.04533	.03136	.02335	.01793	.01806	.01777	.01599	.01686	.01647	.01588
	.01496	.01451								
.175	.04573	.03396	.02472	.01865	.01936	.01793	.01715	.01697	.01669	.01538
	.01529	.01484								
.200	.05186	.03694	.02734	.01911	.01891	.01795	.01750	.01713	.01683	.01613
	.01562	.01521								
.225	.05840	.03966	.02928	.01973	.01909	.01920	.01778	.01743	.01685	.01541
	.01596	.01550								
.250	.05791	.04210	.03100	.02034	.01932	.01875	.01909	.01758	.01593	.01675
	.01630	.01585								
.275	.06279	.04563	.03365	.02039	.02002	.01914	.01857	.01771	.01724	.01730
	.01557	.01603								
.300	.06151	.04687	.03437	.02134	.02068	.01961	.01985	.01783	.01767	.01727
	.01708	.01622								
.325	.06610	.05067	.03727	.02341	.02121	.02024	.01994	.01829	.01799	.01751
	.01725	.01648								
.350	.07226	.05342	.03975	.02506	.02206	.02045	.01933	.01885	.01933	.01818
	.01734	.01685								
.375	.06929	.05457	.04114	.02684	.02229	.02076	.02305	.01928	.01881	.01829
	.01750	.01731								

.400	.07512	.05917	.04526	.02884	.02229	.02161	.02063	.01979	.01928	.01840
	.01786	.01778								
.425	.07295	.05953	.04640	.03011	.02301	.02239	.02120	.02055	.01948	.01958
	.01841	.01784								
.450	.07759	.06324	.04858	.03311	.02387	.02333	.02216	.02030	.01970	.01922
	.01887	.01814								
.475	.08392	.06772	.05303	.03591	.02493	.02398	.02267	.02113	.02024	.01930
	.01921	.01856								
.500	.07992	.06738	.05499	.03799	.02686	.02431	.02286	.02166	.02112	.02025
	.01953	.01944								
.525	.08504	.07279	.05897	.04168	.02899	.02420	.02342	.02277	.02159	.02060
	.02034	.02032								
.550	.08320	.07204	.06093	.04254	.03071	.02480	.02468	.02333	.02203	.02151
	.02145	.02183								
.575	.08924	.07624	.06424	.04584	.03423	.02655	.02496	.02387	.02320	.02286
	.02305	.02346								
.600	.09459	.08177	.06893	.04981	.03763	.02940	.02518	.02526	.02495	.02478
	.02497	.02485								
.625	.08959	.07958	.06976	.05098	.04006	.03123	.02700	.02728	.02706	.02736
	.02690	.02638								
.650	.09461	.08470	.07465	.05623	.04545	.03571	.03379	.02928	.02938	.02935
	.02859	.02775								
.675	.10239	.09063	.08005	.06253	.05114	.04235	.03509	.03177	.03156	.03830
	.02998	.02897								
.700	.10029	.09250	.08471	.06953	.05763	.04315	.04201	.03517	.03236	.03153
	.03059	.02895								
.725	.10679	.09951	.09284	.07835	.06546	.05338	.04712	.04013	.03488	.03172
	.03375	.03047								
.750	.09382	.09433	.08984	.08012	.06876	.05322	.05124	.04444	.03816	.03447
	.03233	.03129								
.775	.09377	.09439	.08353	.08128	.07265	.06418	.05524	.04972	.04361	.03734
	.03570	.03346								
.800	.09105	.08920	.08735	.08200	.07569	.06839	.06134	.05435	.04855	.04230
	.03758	.03343								
.825	.08536	.08442	.08348	.08130	.07671	.07128	.06499	.05878	.05280	.04742
	.04254	.03793								
.850	.08113	.08037	.07960	.07806	.07554	.07179	.06714	.06215	.05688	.05238
	.04846	.04484								

ANALYSIS OF .87[illegible]	.07747	.07728	.07710	.07638	.07536	.07302	.07008	.06609	.06191	.05718
	.05207	.04495								
.900	.07230	.07306	.07322	.07354	.07329	.07230	.07086	.06846	.06506	.06146
	.05701	.05239								
.925	.06967	.06977	.06986	.07005	.07025	.07011	.06995	.06835	.06640	.06332
	.06118	.05843								
.950	.06382	.06467	.06551	.06682	.06726	.06759	.06731	.06659	.06539	.06318
	.05365	.05233								
.975	.05953	.05930	.06026	.06099	.06143	.06130	.06118	.06015	.05884	.05713
	.05346	.04978								
1.000	.04773	.04754	.04735	.04697	.04659	.04608	.04482	.04356	.04220	.04058
	.03996	.03735								

FUSELAGE FORCE COEFFICIENTS BASED ON WING REF. GEOMETRY

	IGNORING WING DOWNWASH		INCLUDING WING DOWNWASH	
	AT ALPHA= 0.000	PER DEG.	AT ALPHA= 0.000	PER DEG.
CL	-.000000	-.000000	-.000403	-.000173
CD	.000001	-.000000	-.000034	-.000003
CM	.003958	.000795	.004009	.000855

ANALYSIS OF DRAG-DUE-TO-LIFT M=2.7 MACH NUMBER = 2.7000

HORIZONTAL TAIL CONTRIBUTION EXCLUDED

FORCE COEFFICIENTS

	CAMBER	FP AT 1 DEG	NAC ON WING	WING ON NAC
CD	2.71745138E-03	4.91443934E-04	1.95422614E-04	1.50747428E-04
CL	7.48765710E-02	2.81576546E-02	5.50084455E-03	
CMXBAR	3.33018195E-04	-2.93415831E-03	-2.43051283E-03	

INTERFERENCE DRAG COEFFICIENTS

FLAT WING PRESSURES ON CAMBERED SURFACE	CAMBERED WING PRESSURES ON FLAT SURFACE
CD = 7.90719816E-04	CD = 1.30684326E-03
NACELLE PRESSURES ON FLAT SURFACE	**FLAT WING PRESSURES ON NACELLE**
CD = 9.60078903E-05	CD = 4.41380312E-05

INCLUDE FUSELAGE TERMS

FORCE COEFFICIENTS

	CAMBER	FP AT 1 DEG	NAC ON WING	WING ON NAC
CD	2.68340862E-03	4.88422519E-04	1.95422614E-04	1.50747428E-04
CL	7.44732831E-02	2.73845370E-02	5.50084455E-03	
CMXBAR	4.34157796E-03	-2.07774440E-03	-2.43051283E-03	

INTERFERENCE DRAG COEFFICIENTS

FLAT WING PRESSURES ON CAMBERED SURFACE	CAMBERED WING PRESSURES ON FLAT SURFACE
CD = 7.75868329E-04	CD = 1.29980450E-03
NACELLE PRESSURES ON FLAT SURFACE	**FLAT WING PRESSURES ON NACELLE**
CD = 9.60078903E-05	CD = 4.41380312E-05

POLAR W/O NAC CD = .002683 + .074172(CL - .074473) + .623677(CL - .074473)**2

POLAR WITH NAC CD = .003030 + .079180(CL - .079974) + .623677(CL - .079974)**2

	CAMBERED WING				FLAT WING	
	W/O NACELLES		WITH NACELLES		W/O NAC	WITH NAC
CL	CD	CM	CD	CM	CD	CD
0.00	.000619	.00957	.000685	.00785	0.000000	-.000003
.01	.000494	.00913	.000543	.00711	.000062	.000035
.02	.000494	.00839	.000524	.00636	.000249	.000204
.03	.000618	.00764	.000630	.00562	.000561	.000497
.04	.000868	.00690	.000861	.00488	.000998	.000915
.05	.001242	.00616	.001217	.00414	.001559	.001458
.06	.001741	.00542	.001697	.00339	.002245	.002125
.07	.002364	.00457	.002302	.00265	.003055	.002918
.08	.003112	.00393	.003032	.00191	.003992	.003935
.09	.003985	.00319	.003885	.00117	.005052	.004975
.10	.004983	.00245	.004865	.00042	.006237	.006143
.11	.006106	.00170	.005969	-.00032	.007545	.007334
.12	.007353	.00095	.007199	-.00106	.008981	.008750
.13	.008725	.00022	.008551	-.00180	.010540	.010291
.14	.010222	-.00052	.010030	-.00255	.012224	.011956
.15	.011843	-.00127	.011633	-.00329	.014033	.013746
.16	.013589	-.00201	.013360	-.00403	.015965	.015561
.17	.015460	-.00275	.015213	-.00477	.018024	.017760
.18	.017456	-.00349	.017190	-.00552	.020207	.019365
.19	.019576	-.00424	.019291	-.00626	.022515	.022154
.20	.021821	-.00498	.021518	-.00700	.024347	.024368

CMXBAR W/O NAC = .004342 -(.074473 -CL)(-.074246) FOR CL = 0. , CMXBAR = .009871

CMXBAR WITH NAC = .001911 -(.079974 -CL)(-.074246) FOR CL = 0. , CMXBAR = .007849

PROGRAM WING AREA= 10665.9372
REFERENCE AREA = 9895.0000

CONFIGURATION STREAMWISE LIFT DISTRIBUTION

		BASIC LIFT DISTRIBUTION				INCREMENT PER DEGREE ALPHA		
X	X/L	W-B-C	NAC	TAIL	SUM	W-B-C	TAIL	SUM
4.154	.01408	.00081	0.00000	0.00000	.00081	.00047	0.00000	.00047
8.308	.02816	.00197	0.00000	0.00000	.00197	.00113	0.00000	.00113
12.462	.04224	.00342	0.00000	0.00000	.00342	.00196	0.00000	.00196
16.615	.05632	.00516	0.00000	0.00000	.00516	.00296	0.00000	.00296
20.769	.07040	.00696	0.00000	0.00000	.00696	.00399	0.00000	.00399
24.923	.08449	.00881	0.00000	0.00000	.00881	.00504	0.00000	.00504
29.077	.09857	.01070	0.00000	0.00000	.01070	.00612	0.00000	.00612
33.231	.11265	.01256	0.00000	0.00000	.01256	.00717	0.00000	.00717
37.385	.12673	.01438	0.00000	0.00000	.01438	.00820	0.00000	.00820
41.539	.14081	.01618	0.00000	0.00000	.01618	.00921	0.00000	.00921
45.692	.15489	.01800	0.00000	0.00000	.01800	.01021	0.00000	.01021
49.846	.16897	.01983	0.00000	0.00000	.01983	.01121	0.00000	.01121
54.000	.18305	.02160	0.00000	0.00000	.02160	.01219	0.00000	.01219
58.154	.19713	.02314	0.00000	0.00000	.02314	.01311	0.00000	.01311
62.308	.21121	.02438	0.00000	0.00000	.02438	.01389	0.00000	.01389
66.462	.22529	.02539	0.00000	0.00000	.02539	.01455	0.00000	.01455
70.616	.23937	.02617	0.00000	0.00000	.02617	.01508	0.00000	.01508
74.769	.25346	.02677	0.00000	0.00000	.02677	.01541	0.00000	.01541
78.923	.26754	.02768	0.00000	0.00000	.02768	.01620	0.00000	.01620
83.077	.28162	.02921	0.00000	0.00000	.02921	.01768	0.00000	.01768
87.231	.29570	.03201	0.00000	0.00000	.03201	.02057	0.00000	.02057
91.385	.30978	.03593	0.00000	0.00000	.03593	.02445	0.00000	.02445
95.539	.32386	.04115	0.00000	0.00000	.04115	.02922	0.00000	.02922
99.692	.33794	.04809	0.00000	0.00000	.04809	.03521	0.00000	.03521
103.846	.35202	.05708	0.00000	0.00000	.05708	.04260	0.00000	.04260
108.000	.36610	.06774	0.00000	0.00000	.06774	.05081	0.00000	.05081
112.154	.38018	.07999	0.00000	0.00000	.07999	.05976	0.00000	.05976
116.308	.39426	.09445	0.00000	0.00000	.09445	.07057	0.00000	.07057
120.462	.40834	.11010	0.00000	0.00000	.11010	.08229	0.00000	.08229
124.616	.42243	.12668	0.00000	0.00000	.12668	.09463	0.00000	.09463
128.769	.43651	.14439	0.00000	0.00000	.14439	.10802	0.00000	.10802
132.923	.45059	.16345	0.00000	0.00000	.16345	.12300	0.00000	.12300
137.077	.46467	.18328	0.00000	0.00000	.18328	.13849	0.00000	.13849
141.231	.47875	.20388	0.00000	0.00000	.20388	.15442	0.00000	.15442
145.385	.49283	.22572	0.00000	0.00000	.22572	.17186	0.00000	.17186
149.539	.50691	.24825	0.00000	0.00000	.24825	.19014	0.00000	.19014
153.693	.52099	.27119	0.00000	0.00000	.27119	.20877	0.00000	.20877
157.846	.53507	.29437	0.00000	0.00000	.29437	.22811	0.00000	.22811
162.000	.54915	.31844	0.00000	0.00000	.31844	.24921	0.00000	.24921
166.154	.56323	.34275	0.00000	0.00000	.34275	.27069	0.00000	.27069
170.308	.57732	.36741	0.00000	0.00000	.36741	.29246	0.00000	.29246
174.462	.59140	.39324	0.00000	0.00000	.39324	.31563	0.00000	.31563
178.616	.60548	.41971	0.00000	0.00000	.41971	.33981	0.00000	.33981
182.770	.61956	.44648	0.00000	0.00000	.44648	.36425	0.00000	.36425
186.923	.63364	.47322	0.00000	0.00000	.47322	.38911	0.00000	.38911
191.077	.64772	.50064	0.00000	0.00000	.50064	.41604	0.00000	.41604
195.231	.66180	.52788	0.00000	0.00000	.52788	.44323	0.00000	.44323
199.385	.67588	.55491	0.00000	0.00000	.55491	.47056	0.00000	.47056
203.539	.68996	.58211	0.00000	0.00000	.58211	.49906	0.00000	.49906
207.693	.70404	.60936	0.00000	0.00000	.60936	.52875	0.00000	.52875
211.847	.71812	.63611	0.00000	0.00000	.63611	.55847	0.00000	.55847
216.000	.73220	.66230	0.00000	0.00000	.66230	.58821	0.00000	.58821

220.154	.74 29	.68876	.00056	0.00000	.68932	.62005	0.00000	.62005
224.308	.76037	.71481	.00702	0.00000	.72183	.65230	0.00000	.65230
228.462	.77445	.74076	.01935	0.00000	.76011	.68588	0.00000	.68588
232.616	.78853	.76765	.03187	0.00000	.79952	.72345	0.00000	.72345
236.770	.80261	.79576	.04562	0.00000	.84138	.76507	0.00000	.76507
240.924	.81369	.82458	.05617	0.00000	.88074	.80906	0.00000	.80906
245.077	.83077	.85215	.06299	0.00000	.91514	.85118	0.00000	.85118
249.231	.84485	.87471	.06481	0.00000	.93952	.88542	0.00000	.88542
253.385	.85893	.89572	.06666	0.00000	.96238	.91853	0.00000	.91853
257.539	.87301	.91367	.06821	0.00000	.98188	.94930	0.00000	.94930
261.693	.88709	.92698	.06878	0.00000	.99576	.97531	0.00000	.97531
265.847	.90118	.93471	.06878	0.00000	1.00349	.99381	0.00000	.99381
270.001	.91526	.93723	.06878	0.00000	1.00601	1.00320	0.00000	1.00320
274.154	.92934	.93621	.06878	0.00000	1.00499	1.00362	0.00000	1.00362
278.308	.94342	.93483	.06878	0.00000	1.00361	1.00258	0.00000	1.00258
282.462	.95750	.93358	.06878	0.00000	1.00236	1.00165	0.00000	1.00165
286.616	.97158	.93259	.06878	0.00000	1.00138	1.00095	0.00000	1.00095
290.770	.98566	.93181	.06878	0.00000	1.00059	1.00040	0.00000	1.00040
294.924	.99974	.93123	.06878	0.00000	1.00001	1.00001	0.00000	1.00001
295.000	1.00000	.93122	.06878	0.00000	1.00000	1.00000	0.00000	1.00000

HORIZONTAL TAIL ALPHA= 2.000

HORIZONTAL TAIL COEFFICIENTS BASED ON WING GEOMETRY

	AT GIVEN ALPHA	PER DEGREE
CL	.002300	.001001
CD	.000118	.000017
CM	-.001815	-.000819

FORCE COEFFICIENTS

	CAMBER	FP AT 1 DEG	NAC ON WING	WING ON NAC
CD	2.80108688E-03	5.05886939E-04	1.95422614E-04	1.50747428E-04
CL	7.67732716E-02	2.89851758E-02	5.50084455E-03	
CMXBAR	2.52634517E-03	-2.89681130E-03	-2.43051283E-03	

INTERFERENCE DRAG COEFFICIENTS

FLAT WING PRESSURES ON CAMBERED SURFACE	CAMBERED WING PRESSURES ON FLAT SURFACE
CD = 8.23941730E-04	CD = 1.33994693E-03
NACELLE PRESSURES ON FLAT SURFACE	FLAT WING PRESSURES ON NACELLE
CD = 9.60075903E-05	CD = 4.41380312E-05

POLAR W/O NAC CD = .002801 + .074655(CL - .076773) + .602146(CL - .076773)**2

POLAR WITH NAC CD = .003147 + .079490(CL - .082274) + .602146(CL - .082274)**2

	CAMBERED WING				FLAT WING	
	W/O NACELLES		WITH NACELLES		W/O NAC	WITH NAC
CL	CD	CM	CD	CM	CD	CD
0.00	.000619	.01020	.000683	.00832	0.000000	-.000108
.01	.000501	.00920	.000548	.00732	.000060	.000134
.02	.000504	.00820	.000532	.00632	.000241	.000197
.03	.000627	.00720	.000637	.00532	.000542	.000480
.04	.000870	.00620	.000863	.00432	.000963	.000983
.05	.001234	.00520	.001209	.00332	.001505	.001408
.06	.001718	.00420	.001675	.00232	.002169	.002052
.07	.002323	.00320	.002262	.00132	.002951	.002817
.08	.003048	.00220	.002970	.00032	.003854	.003702
.09	.003894	.00120	.003797	-.00068	.004877	.004708
.10	.004860	.00021	.004745	-.00168	.005021	.005934
.11	.005946	-.00079	.005814	-.00268	.007285	.007081
.12	.007153	-.00179	.007003	-.00367	.008671	.008448
.13	.008481	-.00279	.008313	-.00467	.010176	.009935
.14	.009928	-.00379	.009742	-.00567	.011802	.011543
.15	.011497	-.00479	.011293	-.00667	.013548	.013271
.16	.013185	-.00579	.012963	-.00767	.015415	.015120
.17	.014994	-.00679	.014755	-.00867	.017402	.017389
.18	.016924	-.00779	.016665	-.00967	.019510	.019179
.19	.018974	-.00879	.018698	-.01067	.021737	.021389
.20	.021144	-.00979	.020851	-.01167	.024085	.023720

CMXBAR W/O NAC = .002526 -(.076773 -CL)(-.099941) FOR CL = 0. , CMXBAR = .010199

CMXBAR WITH NAC = .000396 -(.082274 -CL)(-.099941) FOR CL = 0. , CMXBAR = .008318

PROGRAM WING AREA= 10665.9372
REFERENCE AREA = 9898.0000

CONFIGURATION STREAMWISE LIFT DISTRIBUTION

		BASIC LIFT DISTRIBUTION				INCREMENT PER DEGREE ALPHA		
X	X/L	W-B-C	NAC	TAIL	SUM	W-B-C	TAIL	SUM
4.154	.01408	.00079	0.00000	0.00000	.00079	.00045	0.00000	.00045
8.308	.02816	.00191	0.00000	0.00000	.00191	.00109	0.00000	.00109
12.462	.04224	.00333	0.00000	0.00000	.00333	.00190	0.00000	.00190
16.615	.05632	.00502	0.00000	0.00000	.00502	.00286	0.00000	.00286
20.769	.07040	.00677	0.00000	0.00000	.00677	.00385	0.00000	.00385
24.923	.08449	.00856	0.00000	0.00000	.00856	.00487	0.00000	.00487
29.077	.09857	.01040	0.00000	0.00000	.01040	.00591	0.00000	.00591
33.231	.11265	.01220	0.00000	0.00000	.01220	.00692	0.00000	.00692
37.385	.12673	.01398	0.00000	0.00000	.01398	.00792	0.00000	.00792
41.539	.14081	.01573	0.00000	0.00000	.01573	.00889	0.00000	.00889
45.692	.15489	.01750	0.00000	0.00000	.01750	.00986	0.00000	.00986
49.846	.16897	.01927	0.00000	0.00000	.01927	.01082	0.00000	.01082
54.000	.18305	.02099	0.00000	0.00000	.02099	.01177	0.00000	.01177
58.154	.19713	.02250	0.00000	0.00000	.02250	.01266	0.00000	.01266
62.308	.21121	.02370	0.00000	0.00000	.02370	.01341	0.00000	.01341
66.462	.22529	.02468	0.00000	0.00000	.02468	.01405	0.00000	.01405
70.616	.23937	.02544	0.00000	0.00000	.02544	.01456	0.00000	.01456
74.769	.25346	.02603	0.00000	0.00000	.02603	.01488	0.00000	.01488
78.923	.26754	.02690	0.00000	0.00000	.02690	.01564	0.00000	.01564
83.077	.28162	.02839	0.00000	0.00000	.02839	.01707	0.00000	.01707
87.231	.29570	.03112	0.00000	0.00000	.03112	.01986	0.00000	.01986
91.385	.30978	.03492	0.00000	0.00000	.03492	.02361	0.00000	.02361
95.539	.32386	.04000	0.00000	0.00000	.04000	.02821	0.00000	.02821
99.692	.33794	.04675	0.00000	0.00000	.04675	.03400	0.00000	.03400
103.846	.35202	.05548	0.00000	0.00000	.05548	.04113	0.00000	.04113
108.000	.36610	.06585	0.00000	0.00000	.06585	.04906	0.00000	.04906
112.154	.38018	.07775	0.00000	0.00000	.07775	.05770	0.00000	.05770
116.308	.39426	.09181	0.00000	0.00000	.09181	.06813	0.00000	.06813
120.462	.40834	.10702	0.00000	0.00000	.10702	.07944	0.00000	.07944
124.616	.42243	.12314	0.00000	0.00000	.12314	.09136	0.00000	.09136
128.769	.43651	.14035	0.00000	0.00000	.14035	.10429	0.00000	.10429
132.923	.45059	.15688	0.00000	0.00000	.15688	.11875	0.00000	.11875
137.077	.46467	.17816	0.00000	0.00000	.17816	.13371	0.00000	.13371
141.231	.47875	.19618	0.00000	0.00000	.19618	.14909	0.00000	.14909
145.385	.49283	.21941	0.00000	0.00000	.21941	.16593	0.00000	.16593
149.539	.50691	.24131	0.00000	0.00000	.24131	.18357	0.00000	.18357
153.693	.52099	.26361	0.00000	0.00000	.26361	.20157	0.00000	.20157
157.846	.53507	.28614	0.00000	0.00000	.28614	.22024	0.00000	.22024
162.000	.54915	.30954	0.00000	0.00000	.30954	.24061	0.00000	.24061
166.154	.56323	.33317	0.00000	0.00000	.33317	.26134	0.00000	.26134
170.308	.57732	.35714	0.00000	0.00000	.35714	.28236	0.00000	.28236
174.462	.59140	.38224	0.00000	0.00000	.38224	.30473	0.00000	.30473
178.616	.60548	.40797	0.00000	0.00000	.40797	.32808	0.00000	.32808
182.770	.61956	.43400	0.00000	0.00000	.43400	.35167	0.00000	.35167
186.923	.63364	.46000	0.00000	0.00000	.46000	.37568	0.00000	.37568
191.077	.64772	.48665	0.00000	0.00000	.48665	.40167	0.00000	.40167
195.231	.66180	.51312	0.00000	0.00000	.51312	.42793	0.00000	.42793
199.385	.67588	.53939	0.00000	0.00000	.53939	.45431	0.00000	.45431
203.539	.68996	.56584	0.00000	0.00000	.56584	.48184	0.00000	.48184
207.693	.70404	.59233	0.00000	0.00000	.59233	.51050	0.00000	.51050
211.847	.71812	.61833	0.00000	0.00000	.61833	.53919	0.00000	.53919
216.000	.73220	.64378	0.00000	0.00000	.64378	.56790	0.00000	.56790

220.154	.74629	.66951	.00054	0.00000	.67005	.59864	0.00000	.59864
224.308	.76037	.69483	.00682	0.00000	.70165	.62979	0.00000	.62979
228.462	.77445	.72005	.01880	0.00000	.73886	.66220	0.00000	.66220
232.616	.78353	.74619	.03098	0.00000	.77717	.69847	0.00000	.69847
236.770	.80261	.77352	.04434	0.00000	.81786	.73866	0.00000	.73866
240.924	.81663	.80152	.05460	0.00000	.85612	.78113	0.00000	.78113
245.077	.83077	.82632	.06123	0.00000	.88955	.82180	0.00000	.82180
249.231	.84465	.85026	.06300	0.00000	.91326	.85485	0.00000	.85485
253.385	.85893	.87066	.06460	0.00000	.93548	.88662	0.00000	.88662
257.539	.87301	.88613	.06630	0.00000	.95443	.91653	0.00000	.91653
261.693	.88709	.90107	.06686	.00047	.96840	.94164	.00037	.94201
265.847	.90118	.90858	.06686	.00517	.96060	.95950	.00386	.96336
270.001	.91526	.91103	.06686	.01294	.99084	.96857	.01026	.97883
274.154	.92934	.91004	.06686	.01995	.99685	.96897	.01732	.98629
278.308	.94342	.90870	.06686	.02500	1.00056	.96797	.02442	.99239
282.462	.95750	.90748	.06686	.02769	1.00203	.96707	.03052	.99758
286.616	.97158	.90652	.06686	.02796	1.00134	.96639	.03452	1.00092
290.770	.98566	.90576	.06686	.02796	1.00058	.96587	.03452	1.00039
294.924	.99974	.90520	.06686	.02796	1.00001	.96548	.03452	1.00001
295.000	1.00000	.90518	.06686	.02796	1.00000	.96548	.03452	1.00000

1WING SPANWISE LIFT DISTRIBUTION

	CAMBERED WINGS	FLAT WING	NACELLE INC.
Y/B/2	LIFT FRACTION AT Y/B/2	LIFT FRACTION AT Y/B/2	LIFT FRACTION AT Y/B/2
0.000000	.018495	.014771	0.000000
.025000	.037224	.029747	.002293
.050000	.038036	.030437	.009991
.075000	.040328	.032119	.016798
.100000	.040937	.032764	.022372
.125000	.040653	.032740	.027658
.150000	.040139	.032604	.032319
.175000	.039397	.032183	.035336
.200000	.038731	.031902	.037479
.225000	.037942	.031539	.037975
.250000	.037129	.031016	.037905
.275000	.036391	.030763	.044369
.300000	.035419	.030193	.041821
.325000	.034543	.029905	.048567
.350000	.033487	.029401	.056278
.375000	.032329	.028725	.058179
.400000	.031226	.028302	.059955
.425000	.029972	.027510	.055388
.450000	.028911	.026895	.049762
.475000	.027776	.026351	.042304
.500000	.026584	.025593	.037770
.525000	.025289	.025170	.035650
.550000	.023635	.024397	.035063
.575000	.022141	.023925	.033203
.600000	.020625	.023468	.030351
.625000	.019127	.022728	.027426
.650000	.017794	.022375	.024129
.675000	.016461	.021939	.020133
.700000	.015254	.021242	.016539
.725000	.014239	.020773	.011796
.750000	.013400	.020560	.008178
.775000	.012590	.020420	.002012
.800000	.011927	.020202	-.000000
.825000	.011209	.019631	0.000000
.850000	.010570	.018765	0.000000
.875000	.010111	.018030	0.000000
.900000	.009344	.016877	0.000000
.925000	.008044	.015413	0.000000
.950000	.006594	.013751	0.000000
.975000	.004785	.011289	0.000000
1.000000	.001308	.003772	0.000000

END OF DATA ***STOP

----TOTAL ELAPSED TIME, CP= 122.043 ----

PROGRAM CONTROL CARD
SKFR
ENTER INPTS---TAPE INPUTS
EXIT INPTS
ENTER GEOM158---GEOMETRY INTERFACE WITH PROGRAM TEA15AA
SKIN FRICTION CALCULATIONS M=2.7, H=60000 FT

SKIN FRICTION CALCULATIONS M=2.7, H=60000 FT

NUMBER OF MACH-ALTITUDE COMBINATIONS = 1 NUMBER OF MACH-REYNOLDS COMBINATIONS = -0

NWAF= 8 NWAFOR= 13 NFUSOR= 19 NPOD= 2 NPODOR= 7 NFIN= 0 NFINOR= 0 NCANOR= 3

J1= 1 J2= 1 J3= 1 J4= -0 J5= 1

NCAN= 1 NO. OF EXTRA PARTS= -0 TOTAL NACELLE OVERLAP AREA= -0.00000 REFERENCE AREA= 9898.00000

	MACH NO.	ALTITUDE/1000	TEMPERATURE DEVIATION	SCALE FACTOR
1	2.70	60.000	0.00000	1.00000

	XFUS	PFUS
1	0.00000	0.00000
2	16.67000	17.18460
3	33.33000	26.89060
4	50.00000	33.44260
5	66.67000	38.34400
6	83.33000	39.73150
7	100.00000	38.80010
8	116.67000	36.83980
9	133.33000	35.32450
10	150.00000	36.65980
11	166.66000	36.65380
12	183.33000	36.49710
13	200.00000	35.80180
14	216.67000	34.35920
15	233.33000	31.50790
16	250.00000	27.22900
17	266.67000	20.35390
18	283.30000	10.02650
19	295.00000	0.00000

WING PLANFORM

	X	Y	Z	CHORD LENGTH
1	76.59000	4.75700	0.00000	166.83000
2	83.10400	6.62500	0.00000	160.13300
3	93.16500	9.51000	0.00000	149.79000
4	116.96000	16.33300	0.00000	125.35000
5	168.98000	31.25000	0.00000	77.29500
6	225.81000	47.54400	0.00000	32.68100
7	225.81000	47.54500	0.00000	32.68100
8	258.21000	66.25000	0.00000	14.44500

WING AIRFOIL AT SIDE OF FUSELAGE

	X/C	Z/C
1	0.00	0.0000
2	2.50	.5700
3	5.00	.7140
4	10.00	.8720
5	20.00	1.0500
6	30.00	1.1450

7	40.00	1.2000
8	50.00	1.2300
9	60.00	1.2490
10	70.00	1.1700
11	80.00	.9370
12	90.00	.5460
13	100.00	0.0000

THE NO. OF WING PARTITIONS IS 53

NACELLE GEOMETRY 1

	X	RADIUS	PERIMETER
1	0.0000	2.8650	18.0013
2	2.0080	2.9830	18.7427
3	15.4700	3.6330	22.8268
4	21.5250	3.7700	23.6876
5	28.0170	3.6540	22.9588
6	32.0570	3.4200	21.4885
7	35.0400	3.4200	21.4885

NACELLE GEOMETRY 2

	X	RADIUS	PERIMETER
1	0.0000	2.8650	18.0013
2	2.0080	2.9830	18.7427
3	15.4700	3.6330	22.8268
4	21.5250	3.7700	23.6876
5	28.0170	3.6540	22.9588
6	32.0570	3.4200	21.4885
7	35.0400	3.4200	21.4885

INPUT DATA FOR CANARD 1

ROOT AIRFOIL

261.00000	2.00000	-14.00000	25.00000

TIP AIRFOIL

277.00000	11.00000	-14.00000	9.00000

Z/C COORDINATES FOR CANARD 1

	PERCENT CHORD	Z
1	0.00000	0.00000
2	50.00000	1.50000
3	100.00000	0.00000

NO EXTRA PARTS

DRAG COEFFICIENT CALCULATIONS

MACH NO.= 2.70000 ALTITUDE= 60000.00000

TEMPERATURE VARIATION= 0.00000 INPUT SCALE= 1.00000

	SWET	D/Q	CDF
FUSELAGE	7833.735014	8.037180	.000812
WING	18157.073120	22.091565	.002232
NACELLES	3051.292786	4.254190	.000430
FIN	0.000000	0.000000	0.000000
CANARD	612.000000	.951778	.000096
TOTAL	29654.100920	35.334713	.003570

FUSELAGE 1 AREA DISTRIBUTION (D/Q = 5.34631)

N	X	Z	R	S	N	X	Z	R	S
0	0.0000	10.0000	0.0000	0.0000	50	147.5000	-2.8240	5.8284	106.7199
1	2.9500	9.7434	.7805	1.9136	51	150.4500	-3.0794	5.8370	107.0351
2	5.9000	9.4858	1.3012	5.3195	52	153.4000	-3.3380	5.8408	107.1755
3	8.8500	9.2302	1.7477	9.5952	53	156.3500	-3.5965	5.8420	107.2198
4	11.8000	8.9736	2.1478	14.4929	54	159.3000	-3.8550	5.8415	107.2007
5	14.7500	8.7170	2.5132	19.8427	55	162.2500	-4.1135	5.8399	107.1370
6	17.7000	8.4604	2.8490	25.4990	56	165.2000	-4.3721	5.8373	107.0461
7	20.6500	8.2036	3.1592	31.3552	57	168.1500	-4.6251	5.8350	106.9627
8	23.5000	7.9468	3.4490	37.3701	58	171.1000	-4.8729	5.8323	106.8844
9	26.5500	7.6901	3.7204	43.4837	59	174.0500	-5.1206	5.8298	106.7736
10	29.5000	7.4333	3.9745	49.5273	60	177.0000	-5.3684	5.8254	106.6106
11	32.4500	7.1766	4.2113	55.7162	61	179.9500	-5.6161	5.8190	106.3783
12	35.4000	6.9187	4.4277	61.5902	62	182.9000	-5.8639	5.8102	106.0565
13	38.3500	6.6603	4.5278	67.2818	63	185.8500	-6.1268	5.7980	105.6119
14	41.3000	6.4020	4.8161	72.8675	64	188.8000	-6.3922	5.7828	105.0556
15	44.2500	6.1436	4.9947	78.3745	65	191.7500	-6.6576	5.7646	104.3359
16	47.2000	5.8852	5.1656	83.8271	66	194.7000	-6.9231	5.7435	103.6333
17	50.1500	5.6269	5.3309	89.2804	67	197.6500	-7.1885	5.7194	102.7662
18	53.1000	5.3666	5.4941	94.8280	68	200.6000	-7.4522	5.6923	101.7943
19	56.0500	5.1065	5.6497	100.2791	69	203.5500	-7.7088	5.6520	100.7150
20	59.0000	4.8464	5.7949	105.4964	70	206.5000	-7.9654	5.6279	99.5060
21	61.9500	4.5862	5.9270	110.3623	71	209.4500	-8.2220	5.5894	98.1469
22	64.9000	4.3261	6.0431	114.7272	72	212.4000	-8.4786	5.5455	96.6135
23	67.8500	4.0690	6.1362	118.2310	73	215.3500	-8.7352	5.4952	94.8537
24	70.8000	3.8133	6.2048	120.3510	74	218.3000	-8.9870	5.4357	92.8242
25	73.7500	3.5580	6.2567	122.9836	75	221.2500	-9.2349	5.3559	90.4992
26	76.7000	3.3031	6.2947	124.4737	76	224.2000	-9.4828	5.2908	87.9427
27	79.6500	3.0491	6.3198	125.4758	77	227.1500	-9.7307	5.2082	85.2159
28	82.6000	2.7931	6.3322	125.9661	78	230.1000	-9.9786	5.1192	82.3284
29	85.5500	2.5369	6.3295	125.8539	79	233.0500	-10.2265	5.0240	79.2949
30	88.5000	2.2803	6.3144	125.2591	80	236.0000	-10.4822	4.9227	76.1302
31	91.4500	2.0237	6.2897	124.2821	81	238.9500	-10.7388	4.8149	72.8285
32	94.4000	1.7671	6.2569	122.9909	82	241.9000	-10.9954	4.6994	69.3811
33	97.3500	1.5105	6.2170	121.4265	83	244.8500	-11.2520	4.5757	65.7747
34	100.3000	1.2544	6.1701	119.6001	84	247.8000	-11.5086	4.4419	61.9839
35	103.2500	1.0032	6.1158	117.5043	85	250.7500	-11.7675	4.2943	57.9349
36	106.2000	.7519	6.0577	115.2818	86	253.7000	-12.0329	4.1296	53.5767
37	109.1500	.5006	5.9984	113.0362	87	256.6500	-12.2984	3.9510	49.0426
38	112.1000	.2493	5.9406	110.8700	88	259.6000	-12.5638	3.7592	44.3360
39	115.0500	-.0020	5.8879	108.9101	89	262.5500	-12.8293	3.5539	39.6764
40	118.0000	-.2566	5.8473	107.4130	90	265.5000	-13.0947	3.3334	34.9073
41	120.9500	-.5151	5.8206	106.4343	91	268.4500	-13.3498	3.0937	30.0687
42	123.9000	-.7735	5.8020	105.7567	92	271.4000	-13.5982	2.8340	25.2311
43	126.8500	-1.0321	5.7897	105.3073	93	274.3500	-13.8465	2.5554	20.5141
44	129.8000	-1.2906	5.7827	105.0528	94	277.3000	-14.0949	2.2575	16.0117
45	132.7500	-1.5492	5.7809	104.9872	95	280.2500	-14.3432	1.9463	11.8279
46	135.7000	-1.8047	5.7854	105.1525	96	283.2000	-14.5916	1.6071	8.1140
47	138.6500	-2.0596	5.7943	105.4742	97	286.1500	-14.8679	1.2762	5.1167
48	141.6000	-2.3144	5.8053	105.8767	98	289.1000	-15.1453	.9320	2.7287

49	144.5500	-2.5692	5.8171	106.3087	99	292.0500	-15.4226	.5497	.9493
50	147.5000	-2.8240	5.8284	106.7139	100	295.0000	-15.7000	0.0000	0.0000

EXIT START
ENTER FUSFIT
EXIT FUSFIT
ENTER SLOPE
EXIT SLOPE
ENTER XMAT
EXIT XMAT
ENTER ADIST
EXIT ADIST
ENTER OUT

FAR-FIELD WAVE DRAG   OPTIMIZATION BASED ON MAX. AREA   2

CASE NO. 1
MACH = 2.700   NX = 50   NTHETA = 36

S(X) COMPONENT BUILDUP AT THETA = -90.000

S(B),CAPTURE = .0000   S(P),CAPTURE = 103.1476

X	S(B)	S(BW)	S(BWP)	S(BWPF)	S(BWPF2)
25.0739	0.0000	0.0000	0.0000	0.0000	0.0000
30.0324	11.0592	11.0592	11.0592	11.0592	11.0592
34.9849	25.9873	25.9873	25.9873	25.9873	25.9873
39.9374	42.3793	42.3793	42.3793	42.3793	42.3793
44.8899	58.9684	58.9684	58.9684	58.9684	58.9684
49.8424	75.3376	75.3376	75.3376	75.3376	75.3376
54.7949	91.6512	91.6512	91.6512	91.6512	91.6512
59.7474	107.4040	107.4040	107.4040	107.4040	107.4040
64.6999	121.3580	121.3580	121.3580	121.3580	121.3580
69.6524	132.8915	132.8915	132.8915	132.8915	132.8915
74.6049	142.2923	142.2923	142.2923	142.2923	142.2923
79.5574	149.6250	151.1674	151.1674	151.1674	151.1674
84.5099	154.8615	161.8952	161.8952	161.8952	161.8952
89.4624	157.7770	172.6803	172.6803	172.6803	172.6803
94.4150	158.2309	182.3309	182.3309	182.3309	182.3309
99.3675	156.7219	191.0438	191.0438	191.0438	191.0438
104.3200	154.0842	198.8616	198.8616	198.8616	198.8616
109.2725	150.9659	206.5285	206.5285	206.5285	206.5285
114.2250	147.7354	214.2626	214.2626	214.2626	214.2626
119.1775	144.8576	222.4065	222.4065	222.4065	222.4065
124.1300	142.4728	230.7178	230.7178	230.7178	230.7178
129.0825	140.8530	240.1248	240.1248	240.1248	240.1248
134.0350	140.1071	250.1360	250.1360	250.1360	250.1360
138.9875	139.9615	260.5026	260.5026	260.5026	260.5026
143.9400	140.0853	270.4176	270.4176	270.4176	270.4176
148.8925	140.2215	279.9327	279.9327	279.9327	279.9327
153.8450	140.0634	288.6939	288.6939	288.6939	288.6939
158.7975	139.7173	296.7680	296.7680	296.7680	296.7680
163.7500	139.1160	303.5730	303.5730	303.5730	303.5730
168.7025	138.2982	309.3469	309.3469	309.3469	309.3469
173.6550	137.1030	313.0520	313.0520	313.0520	313.0520
178.6076	135.1738	314.6210	314.6210	314.6210	314.6210
183.5601	132.0711	312.8232	312.8232	312.8232	312.8232

188.5126	127.5666	307.1600	307.1600	307.1600	307.1600
193.4651	122.1921	298.2177	298.8802	298.8802	298.8802
198.4176	115.2164	285.2214	289.8101	289.8101	289.8101
203.3701	109.5624	268.7550	281.8752	281.8752	281.8752
208.3226	101.8677	247.0494	273.7096	273.7096	273.7096
213.2751	93.2729	220.5419	252.0171	262.0171	262.0171
218.2276	83.6164	189.9056	243.9417	243.9417	243.9417
223.1801	72.8880	158.3252	217.8203	217.8203	217.8203
228.1326	61.0979	127.9535	198.0241	188.0241	188.1951
233.0851	49.1279	99.5808	155.2949	155.2949	157.0530
238.0376	34.5013	74.6059	124.7390	124.7390	129.7272
242.9901	21.5237	53.8072	100.0566	100.0566	107.0232
247.9426	10.4035	36.6984	80.5321	80.5321	83.7314
252.8951	2.2911	23.3737	67.2074	67.2074	67.2074
257.8476	-.0000	15.8392	59.6729	59.6729	59.6729
262.8001	-.0000	9.0729	52.9057	52.9057	52.9057
267.7527	-.0000	2.7703	46.6040	46.6040	46.6040
272.7052	-.0000	-.0000	43.8337	43.8337	43.8337

FAR-FIELD WAVE DRAG   OPTIMIZATION BASED ON MAX. AREA   2

CASE NO.   1

MACH = 2.700   NX = 50   NTHETA = 36

S(X) COMPONENT BUILDUP AT THETA = 0.000

S(B),CAPTURE = .0000   S(P),CAPTURE = 103.1476

X	S(B)	S(BW)	S(BWP)	S(BWPF)	S(BWPFC)
.0000	0.0000	0.0000	0.0000	0.0000	0.0000
8.7752	11.3791	11.3791	11.3791	11.3791	11.3791
17.5524	28.0948	28.0948	28.0948	28.0948	28.0948
26.3285	45.1601	46.1601	46.1601	46.1601	46.1601
35.1047	63.9203	63.9203	63.9203	63.9203	63.9203
43.8809	81.1552	81.1552	81.1552	81.1552	81.1552
52.6571	97.0451	97.0451	97.0451	97.0451	97.0451
61.4333	109.9154	109.9154	109.9154	109.9154	109.9154
70.2095	119.2583	126.0384	126.0384	126.0384	126.0384
78.9856	124.4676	151.1320	151.1320	151.1320	151.1320
87.7618	125.3983	173.2611	173.2611	173.2511	173.2611
96.5380	122.9679	195.0744	195.0744	195.0744	195.0744
105.3142	119.5517	217.8486	217.8486	217.8486	217.8486
114.0904	114.0389	229.2987	229.2987	229.2987	229.2987
122.8666	110.4307	232.2009	232.2009	232.2009	232.2009
131.6427	108.6744	231.7380	231.7380	231.7380	231.7380
140.4189	109.5425	228.6657	231.2640	231.2640	231.2640
149.1951	109.9440	223.8450	233.0828	233.0828	233.0828
157.9713	109.3794	217.7540	232.5753	232.5753	232.5753
166.7475	109.3504	210.5487	225.7596	225.7596	225.7596
175.5237	109.7787	202.9756	219.9928	219.9928	219.9928

184.2998	107.7584	194.7342	217.3460	217.3460	217.3460
193.0760	106.0109	186.4917	212.9446	212.9446	212.9446
201.8522	103.1446	177.9707	203.3819	203.3819	203.3819
210.6284	98.8614	169.1414	191.6257	191.6257	191.6257
219.4046	93.0833	159.5283	181.4452	181.4452	181.4452
229.1809	85.7537	149.7645	171.6813	171.6313	171.6813
236.9569	76.7882	138.9194	160.8362	160.8362	160.8352
245.7331	66.2492	124.5774	146.4942	146.4342	146.4942
254.5093	54.1822	106.0635	130.5843	130.5943	131.6335
263.2855	40.6162	85.0542	116.2182	116.2182	118.9636
272.0617	26.1776	64.1655	100.9078	100.9378	102.1159
280.8379	12.5480	45.4470	81.9173	81.9173	83.3611
289.6140	2.7715	30.4645	65.0100	65.0100	66.7325
298.3902	-.0000	23.3220	59.4772	59.4772	60.4875
307.1664	-.0000	19.4181	62.5498	62.5498	62.9778
315.9426	-.0000	16.0406	64.1671	64.1671	64.1671
324.7188	-.0000	13.2528	60.9072	60.9372	60.9072
333.4950	-.0000	10.8885	55.9833	55.9933	55.9833
342.2711	-.0000	8.7568	52.5905	52.5305	52.5935
351.0473	-.0000	6.7430	50.5767	50.5767	50.5737
359.8235	-.0000	5.4913	49.3250	49.3250	49.3250
368.5997	-.0000	4.6909	48.5246	48.5246	48.5246
377.3759	-.0000	4.0172	47.8509	47.8509	47.8539
386.1521	-.0000	3.3997	47.2334	47.2334	47.2334
394.9282	-.0000	2.8337	46.6674	46.6574	46.6674
403.7044	-.0000	2.3192	46.1529	46.1529	46.1529
412.4806	-.0000	1.8563	45.6900	45.6300	45.6910
421.2568	-.0000	1.4448	45.2785	45.2785	45.2735
430.0330	-.0000	.7842	44.6179	44.6179	44.6179
438.8092	-.0000	-.0000	43.8337	43.8337	43.8337

INTERNAL RESTRAINT POINTS (XI*)

SN= 0.0000 SB= 43.8337 ELL= 438.8092

XF	SF
175.5237	252.4718
298.3902	58.8910
307.1664	57.2945
315.9426	55.7961
324.7188	53.3598
333.4950	50.8050
342.2711	48.6796
351.0473	47.7230
359.8235	46.8593
368.5997	46.2921
377.3759	45.7988
386.1521	45.3910
394.9282	45.0197
403.7044	44.7157
412.4806	44.4611
421.2568	44.2408
430.0330	44.0327

FAR-FIELD WAVE DRAG   OPTIMIZATION BASED ON MAX. AREA   2

CASE NO. 1

MACH = 2.700   NX = 50   NTHETA = 36

SBAR(X*) AVERAGE EQUIVALENT BODY

X*	SBAR(B)	SBAR(BW)	SBAR(BWP)	SBAR(BWPF)	SBAR(BWPFC)	SBAR(RESTRAINED)	DELTA SBAR
-.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
8.7762	12.0011	12.0010	12.0009	12.0009	12.0009	5.4076	-6.5933
17.5524	29.2052	29.2049	29.2048	29.2048	29.2048	15.0095	-14.1963
26.3285	47.6364	47.6957	47.6957	47.6957	47.6957	27.0350	-20.5607
35.1047	65.8924	65.8913	65.8913	65.8913	65.8913	40.7780	-25.1133
43.8809	83.5420	83.5387	83.5387	83.5387	83.5387	55.7814	-27.7573
52.6571	99.5192	99.5194	99.5194	99.5194	99.5194	71.7017	-27.8177
61.4333	112.4259	112.4563	112.4562	112.4562	112.4562	88.2559	-24.2003
70.2095	121.7994	125.6941	125.6940	125.6940	125.6940	105.1969	-20.4971
78.9856	127.0706	140.9071	140.9070	140.9070	140.9070	122.2983	-18.6087
87.7618	128.1202	156.2821	156.2820	156.2820	156.2820	139.3457	-16.9362
96.5380	125.7341	171.7180	171.7178	171.7178	171.7178	156.1295	-15.5883
105.3142	121.5899	187.1709	187.1706	187.1706	187.1706	172.4383	-14.7322
114.0904	117.1367	201.0514	201.0503	201.0503	201.0503	188.0538	-12.9965
122.8666	113.5483	213.1301	213.1307	213.1307	213.1307	202.7432	-10.3875
131.6427	111.7088	223.9191	223.9016	223.9016	223.9016	216.2508	-7.6508
140.4189	111.4154	233.2742	233.7312	233.7312	233.7312	228.2854	-5.4458
149.1951	111.7403	240.2943	242.5294	242.5294	242.5294	238.4990	-4.0304
157.9713	112.0871	244.5413	249.4263	249.4263	249.4263	246.4484	-2.9773
166.7475	112.0594	245.4188	252.4396	252.4396	252.4396	251.5028	-.9368
175.5237	111.5325	243.3122	252.4719	252.4719	252.4718	252.4718	-.0000
184.2998	110.4496	239.0465	250.8213	250.8213	250.8213	247.2430	-3.5784
193.0760	108.6586	223.3690	246.5213	246.5213	246.5212	238.0024	-8.5189
201.8522	105.7592	217.5885	238.0028	238.0028	238.0028	226.0156	-11.9872
210.6284	101.2715	202.4647	224.8619	224.8619	224.8618	211.9916	-12.8701
219.4046	95.2913	184.5675	208.6989	208.6989	208.6988	196.4576	-12.2413
228.1808	87.7394	164.4996	190.7393	190.7393	190.7387	179.8525	-10.8862
236.9569	78.6728	143.9183	172.4070	172.4070	172.4085	162.5702	-9.8383
245.7331	67.9938	122.5146	153.4990	153.4990	153.4767	144.9852	-8.4915
254.5093	55.7131	100.1112	134.2557	134.2557	134.6494	127.4725	-7.1769
263.2855	41.9052	78.1147	115.7791	115.7791	117.7712	110.4238	-7.3414
272.0617	27.0913	55.8421	97.1638	97.1638	100.0146	94.3071	-5.7075
280.8379	13.0905	37.5324	78.7675	78.7675	81.1680	79.6628	-1.5052
289.6140	2.9552	23.0039	64.4551	64.4551	65.8802	67.2932	1.4130
298.3902	-.0000	15.1347	58.3262	58.3262	58.8910	58.8910	-.0000
307.1664	-.0000	13.0030	57.1655	57.1655	57.2945	57.2945	-.0000
315.9426	-.0000	10.3324	55.8025	55.8025	55.7961	55.7961	-.0000
324.7188	-.0000	9.1526	53.3592	53.3592	53.3598	53.3598	-.0000
333.4950	-.0000	5.4335	50.8053	50.8053	50.8050	50.8050	-.0000
342.2711	-.0000	5.0497	48.8796	48.8796	48.8796	48.8736	-.0000
351.0473	-.0000	3.8881	47.7230	47.7230	47.7230	47.7230	-.0000
359.8235	-.0000	3.0360	46.8693	46.8693	46.8693	46.8693	-.0000
368.5997	-.0000	2.4484	46.2821	46.2821	46.2821	46.2821	-.0000
377.3759	-.0000	1.9551	45.7888	45.7888	45.7888	45.7888	-.0000
386.1521	-.0000	1.5473	45.3810	45.3810	45.3810	45.3810	-.0000
394.9282	-.0000	1.1860	45.0198	45.0198	45.0197	45.0197	.0000
403.7044	-.0000	.8820	44.7157	44.7157	44.7157	44.7157	-.0000
412.4806	-.0000	.5274	44.4611	44.4611	44.4611	44.4611	-.0000
421.2568	-.0000	.4071	44.2408	44.2408	44.2408	44.2408	-.0000

430.0330	-.0000	.1990	44.0327	44.0327	44.0327	44.0327	-.0000
438.8092	-.0000	-.0000	43.8337	43.8337	43.8337	43.8337	0.0000

FAR-FIELD WAVE DRAG OPTIMIZATION BASED ON MAX. AREA 2

CASE NO. 1

MACH = 2.700 NX = 50 NTHETA = 36

OPTIMUM FUSELAGE AREA DISTRIBUTION WITH RESTRAINTS AT

X= 175.5237

N	X	Z	R	S	N	X	Z	R	S
0	-.0000	10.0000	0.0000	0.0000	25	219.4046	-9.0798	5.0379	79.7363
1	8.7762	9.2366	.9587	2.8876	26	228.1808	-9.8173	4.8316	73.3382
2	17.5524	8.4732	1.8725	11.0147	27	236.9569	-10.5655	4.5569	65.2363
3	26.3285	7.7094	2.6680	22.3621	28	245.7331	-11.3289	4.2284	56.1695
4	35.1047	6.9445	3.3804	35.8997	29	254.5093	-12.1058	3.7918	45.1701
5	43.8809	6.1759	3.9867	49.9318	30	263.2855	-12.8955	3.1489	31.1502
6	52.6571	5.4057	4.5837	66.1796	31	272.0617	-13.6539	2.4235	18.4517
7	51.4333	4.6318	5.2120	85.3410	32	280.8379	-14.3927	1.7425	9.5385
8	70.2095	3.8641	5.6412	99.9761	33	289.6140	-15.1936	1.0986	3.7917
9	78.9856	3.1055	5.8274	106.6859	34	298.3902	-15.7000	-.0000	-.0000
10	87.7618	2.3445	5.8771	108.5116	35	307.1664	-15.7000	-.0000	-.0000
11	96.5380	1.5811	5.8157	106.2344	36	315.9425	-15.7000	-.0000	-.0000
12	105.3142	.8273	5.6763	101.2240	37	324.7188	-15.7000	-.0000	-.0000
13	114.0904	.0797	5.5428	96.5182	38	333.4950	-15.7000	-.0000	-.0000
14	122.8666	-.6830	5.5157	95.5780	39	342.2711	-15.7000	-.0000	-.0000
15	131.6427	-1.4521	5.5663	97.3376	40	351.0473	-15.7000	-.0000	-.0000
16	140.4189	-2.2124	5.6493	100.2635	41	359.8235	-15.7000	-.0000	-.0000
17	149.1951	-2.9779	5.7229	102.8912	42	368.5997	-15.7000	-.0000	-.0000
18	157.9713	-3.7386	5.7602	104.2379	43	377.3759	-15.7000	-.0000	-.0000
19	166.7475	-4.5048	5.8104	106.0607	44	386.1521	-15.7000	-.0000	-.0000
20	175.5237	-5.2444	5.8278	106.6997	45	394.9282	-15.7000	.0000	.0000
21	184.2998	-5.9896	5.7059	102.2827	46	403.7044	-15.7000	-.0000	-.0000
22	193.0760	-6.7770	5.5149	95.5471	47	412.4806	-15.7000	-.0000	-.0000
23	201.8522	-7.5611	5.3334	89.3635	48	421.2568	-15.7000	-.0000	-.0000
24	210.6284	-8.3245	5.1920	84.6868	49	430.0330	-15.7000	-.0000	-.0000
25	219.4046	-9.0799	5.0379	79.7363	50	438.8092	-15.7000	0.0000	0.0000

FAR-FIELD WAVE DRAG OPTIMIZATION BASED ON MAX. AREA 2

CASE NO. 1

MACH = 2.700 NX = 50 NTHETA = 36

D/Q ASSOCIATED WITH VARIOUS VALUES OF THETA

N	THETA	D/Q
0	-90.000	32.42244
1	-85.000	32.43852
2	-80.000	29.06960
3	-75.000	32.21580
4	-70.000	39.14779
5	-65.000	40.81831
6	-60.000	31.09613
7	-55.000	24.76395
8	-50.000	21.32500
9	-45.000	18.24720
10	-40.000	16.59317
11	-35.000	14.70569
12	-30.000	13.65178
13	-25.000	12.47812
14	-20.000	12.00596
15	-15.000	11.66950
16	-10.000	11.46378
17	-5.000	11.16181
18	0.000	10.75983
19	5.000	10.84299
20	10.000	10.43451
21	15.000	9.25607
22	20.000	8.65404
23	25.000	8.42336
24	30.000	7.86690
25	35.000	7.40971
26	40.000	7.68032
27	45.000	8.20368
28	50.000	9.06418
29	55.000	9.74315
30	60.000	9.23257
31	65.000	9.07063
32	70.000	10.07143
33	75.000	10.01783
34	80.000	12.48336
35	85.000	15.68759
36	90.000	25.03472

WING VOLUME CHECK		ENTIRE AIRCRAFT		DRAG OF TRANSFERRED AREA DISTRIBUTIONS	
EXACT VOLUME =	17856.35813	D/Q =	16.00341	OPTIMUM EQ. BODY CDW*=	6.87029430E-04
EQUIVALENT BODY VOLUME =	17854.82340	CDW =	1.61703448E-03	AVERAGE EQ. BODY CDW*=	8.29027269E-04
		OPT. CDW*=	1.47503664E-03	POTENTIAL CDW* CHANGE=	-1.41997839E-04

EXIT OUT

NEAR FIELD WAVE DRAG

MACH NC.= 2.70000 NON= 40 NOPCT= 13 JBYMAX= 20 RATIO= 4.15385

PLANFORM BREAKPOINTS

	X	Y	CHORD
1	76.5900	0.0000	166.8300
2	76.5900	4.7570	166.8300
3	83.1040	6.6250	160.1330
4	93.1650	9.5100	149.7300
5	116.9600	16.3330	125.3500
6	168.9800	31.2500	77.2950
7	225.8100	47.5440	32.6810
8	225.8100	47.5450	32.6910
9	258.2100	66.2500	14.4450

	XLE	XTE	Y
0	76.5900	243.4200	0.0000
1	76.5900	243.4200	1.6563
2	76.5900	243.4200	3.3125
3	77.3284	243.3993	4.9588
4	83.1040	243.2370	6.6250
5	88.8799	243.0751	8.2913
6	94.6559	242.9146	9.9375
7	100.4320	242.7580	11.5937
8	106.2081	242.6014	13.2500
9	111.9843	242.4449	14.9062
10	117.7603	242.3710	16.5625
11	123.5362	242.8112	18.2187
12	129.3120	243.2515	19.8750
13	135.0878	243.5917	21.5312
14	140.8637	244.1320	23.1875
15	146.6395	244.5722	24.8438
16	152.4153	245.0124	26.5000
17	158.1912	245.4527	28.1562
18	163.9670	245.8929	29.8125
19	159.7430	246.4390	31.4687
20	175.5196	247.6807	33.1250
21	181.2962	248.9225	34.7812
22	187.0729	250.1542	36.4375
23	192.8495	251.4059	38.0937
24	198.6262	252.6477	39.7500
25	204.4028	253.8894	41.4063
26	210.1795	255.1311	43.0625
27	215.9561	256.3728	44.7187
28	221.7328	257.6146	46.3750
29	226.6523	258.8592	48.0312
30	229.5211	260.1134	49.6875
31	232.3900	261.3675	51.3437
32	235.2589	262.6217	53.0000
33	238.1278	263.8759	54.6562
34	240.9967	265.1300	56.3125
35	243.8656	266.3842	57.9688
36	246.7345	267.6383	59.6250
37	249.6033	268.8925	61.2812
38	252.4722	270.1467	62.9375
39	255.3411	271.4608	64.5937
40	258.2100	272.6550	66.2500

WING-BODY INTERSECTION

CHORD	X	Y	Z	T
0.00	76.600461	4.760000	-3.336402	0.000000
2.50	80.770943	4.760000	-2.986316	1,911739
5.00	84.941424	4.760000	-2.748377	2.382179
10.00	93.282386	4.760000	-2.628692	2.909328
20.00	109.964310	4.760000	-2.606061	3.503204

30.00	126.646235	4.760000	-2.715941	3.820161
40.00	143.328159	4.760000	-2.788028	4.003662
50.00	160.010084	4.760000	-2.765999	4.103753
60.00	176.692008	4.760000	-2.657383	4.167145
70.00	193.373933	4.760000	-2.386124	3.903570
80.00	210.055857	4.760000	-1.965603	3.126193
90.00	226.737782	4.760000	-1.446089	1.821666
100.00	243.419706	4.760000	-.620226	0.000000

NACELLE GEOMETRY

ORIGIN (X,Y,Z)			X	RADIUS	AREA
213.42000	15.33000	-5.80000	0.00000	2.85500	25.78596
			2.00800	2.99300	27.95486
			15.47000	3.63300	41.46500
			21.52500	3.77000	44.65125
			28.01700	3.65400	41.94575
			32.06700	3.42000	36.74541
			35.04000	3.42000	36.74541

ORIGIN (X,Y,Z)			X	RADIUS	AREA
215.67000	31.25000	-4.90000	0.00000	2.85500	25.78595
			2.00800	2.98300	27.95485
			15.47000	3.63300	41.46500
			21.52500	3.77000	44.65125
			28.01700	3.65400	41.94575
			32.06700	3.42000	36.74541
			35.04000	3.42000	36.74541

BUOYANCY FIELD OF BODY ON NACELLES

NACELLE(S) AT Y= 16.33000

NEAR-FIELD PRESSURE SIGNATURE
1 SHOCK WAVES
X= 39.588462 CP1= 0.000000 CP2= .030315

X	CP1	CP2
39.588462	0.000000	.030315
43.341083		.027189
50.127321		.022016
56.200167		.018743
62.741337		.014860
68.803991		.012385
74.243239		.011448
79.757524		.010410
86.528565		.006857
94.143385		.001842
101.796107		-.002798
108.871673		-.005862
115.169625		-.009143
123.136136		-.011303
129.365965		-.011720
136.029681		-.012959
141.793655		-.011923
146.382478		-.008484
150.954921		-.005185
155.980955		-.003057
161.161171		-.001359
165.256541		.000366
172.073526		.000422
178.278771		-.000331
184.440366		-.000939
190.467519		-.001223
196.371297		-.001219
202.316363		-.001316
208.485705		-.001861
214.644554		-.002322
220.855524		-.002862
227.139305		-.003485
233.296427		-.003759
239.743019		-.004635
246.499121		-.006031
253.228764		-.007212

NACELLE(S) AT Y= 31.25000

NEAR-FIELD PRESSURE SIGNATURE
1 SHOCK WAVES
X= 72.238336 CP1= 0.000000 CP2= .021457

X	CP1	CP2
72.238336	0.000000	.021457
74.078021		.020434
81.649347		.016546
88.217356		.014087

95.348762	.011168
101.786243	.009308
107.367500	.008604
113.033337	.007424
120.349235	.005154
128.724913	.001384
137.081253	-.002103
144.621606	-.004406
152.415174	-.006871
159.711239	-.008495
166.004442	-.008808
172.856067	-.009740
179.462943	-.008961
182.530101	-.005376
186.602353	-.003897
191.305488	-.002298
196.228155	-.001022
201.061973	.000275
206.870373	.000317
213.189814	-.000249
219.443635	-.000706
225.513304	-.000919
231.416913	-.000916
237.376583	-.000989
243.629754	-.001399
249.857543	-.001745
256.150483	-.002151

BUOYANCY FIELD OF NACELLES ON BODY

FUSELAGE AREAS IN WING REGION

X	ABOVE WING	BELOW WING
78.26865	93.74599	25.63999
81.60504	97.00015	28.99523
84.94142	95.93133	29.31994
88.27781	94.74395	29.50536
91.61419	93.40572	29.60164
94.95058	91.92340	29.60858
98.28696	90.29913	29.52674
101.62335	88.58375	28.13317
104.95973	87.11385	26.87659
108.29512	85.86749	25.76785
111.63250	84.84582	24.79966
114.96889	84.04878	23.96591
118.30527	83.60100	23.40222
121.64166	83.28522	22.91489
124.97804	83.09932	22.50036
128.31443	83.04322	22.15702
131.65081	83.11712	21.88355
134.98720	83.59150	21.96565
138.32358	84.01550	22.02163
141.65997	84.38783	22.05027
144.99635	84.70778	22.05149
148.33274	84.97525	22.02532
151.66912	84.88047	22.20010
155.00551	84.75309	22.36152

158.34189	84.61101	22.50945
161.67828	84.43634	22.64380
165.01466	84.23527	22.76578
168.35105	83.99115	23.05301
171.68743	83.66534	23.29852
175.02382	83.26151	23.50180
178.36020	82.77745	23.66242
181.69659	82.21082	23.78555
185.03297	81.15109	24.37850
188.36936	79.98160	24.91169
191.70574	78.70551	25.38320
195.04213	77.32618	25.79175
198.37851	75.84787	26.14098
201.71489	74.27253	26.73292
205.05128	72.50359	27.20008
208.38766	70.54376	27.54006
211.72405	68.42049	27.75127
215.06043	66.12213	27.83666
218.39682	63.45889	27.99709
221.73320	60.65942	28.02585
225.06959	57.73733	27.92337
228.40597	54.70698	27.69082
231.74235	51.57098	27.35679
235.07874	47.78901	27.90609
238.41513	43.83102	28.20016
241.75151	39.74122	28.23515

NACELLE(S) AT Y= 16.33000

NEAR-FIELD PRESSURE SIGNATURE
2 SHOCK WAVES

X= 247.773591 CP1= 0.000000 CP2= .016929
X= 285.922955 CP1= -.019327 CP2= .001573

X	CP1	CP2
247.773591	0.000000	.016929
248.453943		.016363
249.909538		.015169
251.374028		.013971
252.837103		.012792
254.287727		.011656
255.735252		.010543
257.181773		.009448
258.627060		.008372
260.071353		.007314
261.330493		.006675
261.933280		.007331
263.393652		.006369
265.503337		.003902
267.607455		.001485
269.703175		-.000874
271.603904		-.002768
273.449613		-.004515
275.626045		-.006945
277.848329		-.009429
280.030443		-.011779
282.168818		-.013988

285.364090		-.018454
285.922355	-.019327	.001573
286.490111		.001202
287.625209		.001028
288.755188		.000866
289.875102		.000724
290.989775		.000598
292.033558		.000493
293.185066		.000414
294.274588		.000341
295.361162		.000273

NACELLE(S) AT Y= 31.25000

NEAR-FIELD PRESSURE SIGNATURE
2 SHOCK WAVES

X= 286.638518 CP1= 0.000000 CP2= .011993
X= 327.354033 CP1= -.013072 CP2= .000798

X	CP1	CP2
286.638518	0.000000	.011993
287.719571		.011400
289.365751		.010500
291.007652		.009614
292.630564		.008760
294.246918		.007924
295.859509		.007101

BUOYANCY FIELD OF NACELLES ON NACELLE

NACELLE AT Y= 16.33000 Z= -5.80000
NACELLE AFT END AT X= 248.45000

PRESSURE SIGNATURE FROM NACELLE AT Y= -16.33000 Z= -5.80000

X	CP
283.133	0.00000

PRESSURE SIGNATURE FROM NACELLE AT Y= 31.25000 Z= -4.90000

X	CP
245.608	0.00000
245.609	.01893
246.614	.01798
248.020	.01667
249.435	.01535

PRESSURE SIGNATURE FROM NACELLE AT Y= -31.25000 Z= -4.90000

X	CP
324.817	0.00000

COMPOSITE SIGNATURE

X	CP
0.000	0.00000
245.608	0.00000
245.609	.01893
246.614	.01798
248.020	.01667
249.435	.01535

NACELLE AT Y= 31.25000 Z= -4.90000
NACELLE AFT END AT X= 253.71000

PRESSURE SIGNATURE FROM NACELLE AT Y= 16.33000 Z= -5.80000

X	CP
240.358	0.00000
240.359	.01893
241.364	.01798
242.770	.01667
244.185	.01535
245.600	.01405
247.004	.01281
248.405	.01158
249.807	.01038
251.208	.00920
252.608	.00804
253.841	.00733

PRESSURE SIGNATURE FROM NACELLE AT Y= -16.33000 Z= -5.80000

X	CP
313.567	0.00000

PRESSURE SIGNATURE FROM NACELLE AT Y= -31.25000 Z= -4.90000

X	CP
351.416	0.00000

COMPOSITE SIGNATURE

X	CP
0.000	0.00000
240.359	0.00000
240.359	.01893
241.364	.01798
242.770	.01667
244.185	.01535
245.600	.01405
247.004	.01281
248.405	.01158
249.807	.01038
251.218	.00920
252.609	.00804
253.841	.00733

BOUYANCY FIELD OF NACELLE ON ITSELF(IMAGE EFFECT)

NACELLE AT Y= 16.33000 Z= -5.80000

X	CP
0.000	0.00000
232.389	0.00000
232.390	.02197
232.438	.02192
233.783	.02041
235.130	.01892
236.486	.01742
237.841	.01595
239.189	.01454
240.535	.01315
241.882	.01178
243.229	.01044
244.578	.00912
245.779	.00833
246.531	.00914
247.814	.00794
249.639	.00487

BUOYANCY FIELD OF OTHER IMAGE NACELLES

PRESSURE SIGNATURE FROM NACELLE AT Y= -16.33000 Z= -5.80000

X	CP
288.000	0.00000

PRESSURE SIGNATURE FROM NACELLE AT Y= 31.25000 Z= -4.90000

X	CP
253.795	0.00000

PRESSURE SIGNATURE FROM NACELLE AT Y= -31.25000 Z= -4.90000

X	CP
327.704	0.00000

NO EFFECT

BOUYANCY FIELD OF NACELLE ON ITSELF(IMAGE EFFECT)

NACELLE AT Y= 31.25000 Z= -4.90000

X	CP
0.000	0.00000

233.394	0.00000
233.395	.02422
233.696	.02384
235.006	.02220
236.317	.02058
237.637	.01896
238.959	.01736
240.271	.01582
241.584	.01431
242.899	.01282
244.214	.01136
245.531	.00992
246.713	.00926
247.455	.00995
248.739	.00864
250.551	.00523
252.363	.00202
254.174	-.00119

BUOYANCY FIELD OF OTHER IMAGE NACELLES

PRESSURE SIGNATURE FROM NACELLE AT Y= 16.33000 Z= -5.80000

X	CP
248.535	0.00000
248.536	.01675
249.184	.01622
250.644	.01504
252.113	.01385
253.581	.01268
255.036	.01155

PRESSURE SIGNATURE FROM NACELLE AT Y= -16.33000 Z= -5.80000

X	CP
322.454	0.00000

PRESSURE SIGNATURE FROM NACELLE AT Y= -31.25000 Z= -4.90000

X	CP
353.293	0.00000

COMPOSITE SIGNATURE

X	CP
0.000	0.00000
248.535	0.00000
248.536	.01675
249.184	.01622
250.644	.01504
252.113	.01385
253.591	.01268
255.036	.01155

FUSELAGE DATA AND PRESSURE FIELD ACTING ON WINS

X	R	AREA	CP	Y	F(Y)
0.000000	0.000000	0.000000	.091103	0.000000	0.000000
5.900000	1.104211	3.830417	.091103	3.130679	.148984
11.800000	2.059176	13.320996	.070564	6.635614	.154546
17.700000	2.844897	25.426294	.041354	10.565034	.129324
23.600000	3.437564	37.123729	.031026	14.978632	.104722
29.500000	3.967615	49.454854	.024325	19.549273	.089156
35.400000	4.422238	61.437582	.014505	24.309084	.070684
41.300000	4.810474	72.698526	.012070	29.235393	.058911
47.200000	5.165634	83.828574	.009575	34.244732	.054456
53.100000	5.509234	95.352574	.010591	39.282910	.049518
59.000000	5.812011	106.121003	.003634	44.423575	.032618
64.900000	5.045934	114.934503	-.002900	49.736950	.008751
70.800000	5.195919	120.503895	-.010545	55.260715	-.013309
76.700000	6.287810	124.207758	-.013702	60.930253	-.027884
82.600000	6.331048	125.921846	-.017987	66.721813	-.043489
88.500000	5.300591	124.712819	-.021066	72.698224	-.053763
94.400000	5.245443	122.539577	-.020010	78.736509	-.055748
100.300000	6.167530	119.503576	-.023954	84.831763	-.061641
106.200000	6.032329	114.319412	-.019952	91.070395	-.056715
112.100000	5.925892	110.320412	-.013555	97.237364	-.040355
118.000000	5.851554	107.574327	-.005632	103.324102	-.024663
123.900000	5.810850	106.078961	-.003905	109.326462	-.014543
129.800000	5.787195	105.216692	-.001663	115.285814	-.006466
135.700000	5.792359	105.404912	.003245	121.172838	.001739
141.600000	5.815263	106.240143	.001478	127.015395	.002005
147.500000	5.831331	106.826936	.000127	132.875173	-.001576
153.400000	5.838243	107.081451	-.001267	138.757761	-.004468
159.300000	5.839392	107.123621	-.001642	144.654878	-.005817
165.200000	5.837117	107.040141	-.001933	150.560586	-.005796
171.100000	5.836835	107.031627	-.001684	156.461168	-.006257
177.000000	5.828881	106.738322	-.002814	162.381240	-.008852
182.900000	5.810442	106.064052	-.003482	168.327486	-.011045
188.800000	5.785318	105.148821	-.004336	174.290496	-.013616
194.700000	5.745943	103.722411	-.005541	180.289247	-.016577
200.600000	5.693782	101.847774	-.004977	186.320068	-.017880
206.500000	5.636215	99.798749	-.007118	192.364443	-.022049
212.400000	5.553124	96.773228	-.009539	198.480361	-.028655
218.300000	5.435404	92.814012	-.011534	204.668077	-.034303
224.200000	5.292076	87.983666	-.013044	210.927541	-.038625
230.100000	5.120513	82.371466	-.014194	217.257820	-.040103
236.000000	4.933236	76.456389	-.012314	223.627506	-.040381
241.900000	4.715935	69.863160	-.016290	230.072495	-.045988
247.800000	4.447074	62.129622	-.013013	236.646794	-.050171
253.700000	4.139234	53.825726	-.013064	243.318854	-.051802
259.600000	3.772458	44.709618	-.022089	250.138699	-.054137
265.500000	3.335833	34.950219	-.022318	257.133623	-.050431
271.400000	2.847433	25.472723	-.019924	264.258523	-.042301
277.300000	2.273537	15.241080	-.017316	271.597598	-.029909
283.200000	1.607849	8.121575	-.006312	279.167536	-.005395
289.100000	.849956	2.269533	.023198	286.968337	.034512
295.000000	.000000	.000000	.164528	295.000000	.062392

BODY PRESSURE FIELD ACTING ON WING

XPCT								
	0.000000	5.000000	10.000000	15.000000	20.000000	25.000000	30.000000	35.000000
	40.000000	45.000000	50.000000	55.000000	60.000000	65.000000	70.000000	75.000000
	80.000000	85.000000	90.000000	95.000000	100.000000			
Y/B/2								
0.0000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000	0.000000
	0.000000	0.000000	0.000000	0.000000	0.000000			
.0250	-.035223	-.031952	-.041376	-.029911	-.014951	-.005791	.001250	-.000221
	-.003145	-.004030	-.004299	-.006535	-.008811	-.011518	-.013521	-.018620
	-.023910	-.027153	-.027952	-.031621	-.035007			
.0500	-.021056	-.026650	-.029143	-.027871	-.015917	-.007092	-.001157	.008978
	-.001264	-.002683	-.002878	-.003771	-.005347	-.007124	-.008563	-.010884
	-.014969	-.019149	-.019621	-.020420	-.023519			
.0750	-.013569	-.020113	-.022323	-.024093	-.017640	-.008542	-.003130	.000732
	-.000237	-.001846	-.002326	-.002489	-.003790	-.005088	-.006539	-.007749
	-.010582	-.013611	-.015608	-.016124	-.017935			
.1000	-.012847	-.017923	-.019502	-.020791	-.015179	-.007417	-.002923	.000527
	-.000050	-.001472	-.002016	-.002110	-.003022	-.004066	-.005306	-.006146
	-.007983	-.010683	-.012807	-.013834	-.014192			
.1250	-.012462	-.015496	-.017624	-.018513	-.013445	-.006640	-.002625	.000554
	.000093	-.001157	-.001805	-.001846	-.002439	-.003348	-.004354	-.005302
	-.006363	-.008531	-.010639	-.012068	-.012477			
.1500	-.012254	-.015284	-.016261	-.016815	-.012121	-.006059	-.002486	.000500
	.000209	-.000912	-.001571	-.001649	-.001987	-.002822	-.003650	-.004622
	-.005203	-.006762	-.008761	-.010361	-.011221			
.2000	-.011651	-.013398	-.014395	-.014395	-.010165	-.005227	-.002307	.000305
	.000394	-.000543	-.001209	-.001425	-.001494	-.002018	-.002641	-.003326
	-.004089	-.004565	-.005786	-.007389	-.008715			
.2500	-.011327	-.012132	-.013166	-.012706	-.008724	-.004636	-.002194	-.000031
	.000432	-.000254	-.000914	-.001255	-.001272	-.001390	-.001962	-.002454
	-.003025	-.003652	-.003980	-.004881	-.006230			
.3000	-.010854	-.011305	-.012310	-.011405	-.007434	-.004073	-.002017	-.000189
	.000385	-.000051	-.000635	-.001026	-.001163	-.001198	-.001432	-.001890
	-.002309	-.002788	-.003322	-.003564	-.004263			
.3500	-.010214	-.010831	-.011197	-.009875	-.006344	-.003613	-.001879	-.000327
	.000349	.000119	-.000404	-.000839	-.001053	-.001076	-.001130	-.001394
	-.001771	-.002121	-.002515	-.002962	-.003226			
4000	-.009775	-.010474	-.010214	-.008410	-.005389	-.003224	-.001767	-.000451
	.000318	.000265	-.000194	-.000582	-.000861	-.001009	-.001006	-.001051
	-.001291	-.001612	-.001896	-.002222	-.002578			
.4500	-.009619	-.003964	-.009378	-.007089	-.004523	-.002884	-.001674	-.000566

	.000292	.000323	.000013	-.000348	-.000658	-.000845	-.000951	-.000948
	-.000992	-.001132	-.001410	-.001652	-.001902			
.5000	-.009505	-.009132	-.008060	-.005806	-.003717	-.002506	-.001525	-.000589
	.000272	.000298	.000156	-.000161	-.000435	-.000593	-.000818	-.000902
	-.000900	-.000929	-.000978	-.001210	-.001432			
.6000	-.008065	-.006553	-.004952	-.003436	-.002586	-.001814	-.001171	-.000514
	.000152	.000251	.000279	.000131	-.000084	-.000285	-.000461	-.000635
	-.000719	-.000804	-.000823	-.000822	-.000838			
.7000	-.004198	-.003210	-.002673	-.002137	-.001676	-.001273	-.000870	-.000452
	-.000034	.000232	.000243	.000254	.000245	.000113	-.000018	-.000149
	-.000267	-.000375	-.000482	-.000588	-.000640			
.8000	-.005983	-.005070	-.004310	-.003555	-.002909	-.002516	-.002124	-.001745
	-.001452	-.001159	-.000866	-.000564	-.000260	.000044	.000217	.000224
	.000232	.000240	.000227	.000134	.000041			
.9000	-.006984	-.006853	-.006742	-.005632	-.006353	-.005753	-.005153	-.004560
	-.003991	-.003422	-.002853	-.002563	-.002274	-.001985	-.001695	-.001477
	-.001262	-.001047	-.000832	-.000612	-.000389			
.9500	-.006817	-.006899	-.005891	-.006800	-.006708	-.006617	-.006525	-.006433
	-.006152	-.005646	-.005141	-.004635	-.004151	-.003672	-.003193	-.002745
	-.002504	-.002263	-.002022	-.001782	-.001566			
1.0000	-.006389	-.006453	-.006518	-.006583	-.006648	-.006713	-.006746	-.006673
	-.006600	-.006527	-.006453	-.006380	-.006307	-.006233	-.005872	-.005460
	-.005048	-.004635	-.004234	-.003843	-.003453			

TABLE OF THICKNESS PRESSURE COEFFICIENT

XPCT	0.00	5.00	10.00	15.00	20.00	25.00	30.00	35.00	40.00	45.00	50.00	55.00
	60.00	65.00	70.00	75.00	80.00	85.00	90.00	95.00	100.00			
Y/B/2												
0.000	0.000000	.008208	.017194	.020613	.012988	.007425	.005438	.003264	.002663	.005482	.003488	.000479
	-.000100	-.001515	-.003906	-.004349	-.008065	-.013937	-.018165	-.022130	-.027107			
.025	.003384	.008515	.013914	.013905	.009557	.007861	.008349	.005986	.003147	.002164	.000873	.000549
	.001641	.000329	-.003097	-.006390	-.010492	-.014792	-.017609	-.021228	-.026055			
.050	.012141	.012361	.015735	.014073	.012224	.008119	.004619	.003518	.004264	.002916	.002561	.001381
	-.000866	-.002606	-.003728	-.007037	-.010502	-.014541	-.019393	-.024072	-.026998			
.075	.044193	.010107	.004035	.005410	.008955	.008058	.003985	.004022	.003647	.000982	.001429	.001239
	-.002097	-.004195	-.006319	-.011001	-.014695	-.017282	-.022324	-.026579	-.028459			
.100	.064049	.007115	-.006118	.004790	.007254	.004428	.002431	.002909	.001848	.001128	.001735	-.000430
	-.004020	-.006435	-.009853	-.014174	-.017639	-.020563	-.024990	-.028658	-.030494			
.125	.093720	.006411	-.006317	.003027	.004030	.003180	.001410	.001083	.001427	.000812	-.000068	-.001436
	-.003597	-.007547	-.012585	-.015999	-.018869	-.022399	-.026103	-.030207	-.032645			
.150	.133021	.004494	-.010407	.000092	.003698	.001997	-.000703	.000883	.000915	-.001088	-.001293	-.000437
	-.004801	-.010476	-.013770	-.015004	-.019718	-.024340	-.026617	-.029497	-.033011			
.200	.049035	-.005775	-.008915	.001025	-.001539	-.001172	.000765	-.000841	-.002589	-.001167	-.001164	-.004334
	-.008049	-.011358	-.015903	-.019553	-.021919	-.025428	-.028908	-.031633	-.033646			
.250	.040300	-.005286	-.011646	-.003849	-.005326	-.004686	-.002863	-.000826	-.001664	-.003324	-.003529	-.005053
	-.009474	-.013308	-.017619	-.020301	-.023333	-.027004	-.030666	-.033324	-.034423			
.300	.027568	-.006494	-.010754	-.009395	-.006239	-.006047	-.004647	-.002865	-.004094	-.001931	-.004918	-.007486
	-.011162	-.014583	-.017413	-.022979	-.026318	-.029364	-.031283	-.034498	-.037529			
.350	.049360	.003985	-.007744	-.014514	-.010288	-.008938	-.003613	-.003909	-.004396	-.005963	-.006348	-.008419
	-.011589	-.016364	-.021135	-.023984	-.027357	-.030841	-.035015	-.036348	-.038465			
.400	.041224	.001924	-.010040	-.012519	-.011536	-.010695	-.007494	-.004655	-.005275	-.005841	-.008291	-.011125
	-.014620	-.017474	-.021173	-.025203	-.029789	-.033471	-.036367	-.038175	-.038881			
.450	.033210	-.001862	-.013157	-.015995	-.013605	-.008113	-.008725	-.006863	-.007769	-.009207	-.008559	-.012521
	-.014992	-.020118	-.023898	-.026893	-.031013	-.033403	-.037491	-.040248	-.042427			
.500	.018110	-.002790	-.013687	-.017204	-.018154	-.011229	-.007425	-.007940	-.007530	-.010394	-.013034	-.014864
	-.018624	-.021547	-.024323	-.028967	-.032153	-.036164	-.039656	-.041246	-.042514			
.600	.019845	-.001756	-.010743	-.015305	-.014502	-.014319	-.013645	-.011831	-.013038	-.015074	-.017298	-.019150
	-.021437	-.026264	-.029940	-.033184	-.038010	-.041326	-.043471	-.044470	-.045470			
.700	.001231	-.004976	-.010984	-.014841	-.015417	-.016037	-.017095	-.018154	-.018134	-.017805	-.019311	-.022355
	-.026629	-.030350	-.034070	-.038411	-.042679	-.046391	-.049004	-.051638	-.054363			
.800	.041532	.034965	.028397	.021810	.015191	.008571	.002120	-.004005	-.010130	-.016048	-.021493	-.026337
	-.031904	-.035574	-.039245	-.042813	-.046047	-.049281	-.052317	-.054549	-.056781			

.900	.044927	.041580	.038232	.034884	.031536	.026710	.020995	.015279	.009564	.003029	-.003948	-.010325
	-.017902	-.024095	-.029913	-.035731	-.041549	-.045697	-.049134	-.052571	-.056008			
.950	.045965	.042864	.039763	.036662	.032927	.028799	.024670	.020541	.016412	.010831	.005117	-.000537
	-.006310	-.012095	-.018139	-.024183	-.030226	-.036270	-.042826	-.049862	-.056898			
1.000	.034998	.032584	.030170	.027756	.025342	.022928	.020514	.018100	.015381	.011956	.008532	.005108
	.001683	-.001741	-.005151	-.008549	-.011948	-.015346	-.018744	-.022143	-.025541			

BODY PRESSURE DATA

X	CPB	CPW/B	BODY DRAG	WING INTF.
2.950000	.091103	0.000000	.058115	0.000000
8.850000	.080834	0.000000	.212837	0.000000
14.750000	.055959	0.000000	.377722	0.000000
20.650000	.036190	0.000000	.461300	0.000000
26.550000	.027675	0.000000	.536467	0.000000
32.450000	.019415	0.000000	.590210	0.000000
38.350000	.013288	0.000000	.620923	0.000000
44.250000	.010823	0.000000	.647952	0.000000
50.150000	.010083	0.000000	.671402	0.000000
56.050000	.007112	0.000000	.686339	0.000000
61.950000	.000367	0.000000	.687200	0.000000
67.850000	-.006723	0.000000	.679353	0.000000
73.750000	-.012123	0.000000	.671013	0.000000
79.650000	-.015845	0.000000	.660735	0.000000
85.550000	-.019527	.010173	.666457	-.010810
91.450000	-.020538	.014542	.674838	-.032328
97.350000	-.021982	.006981	.684010	-.042390
103.250000	-.021953	.007173	.700763	-.062740
109.150000	-.016753	.009987	.713101	-.089409
115.050000	-.010093	.011184	.719138	-.113549
120.950000	-.005268	.010190	.720073	-.120359
126.850000	-.002787	.008125	.720595	-.125714
132.750000	.000788	.005889	.720539	-.126981
138.650000	.002361	.005025	.720822	-.124796
144.550000	.000802	.004384	.720913	-.122889
150.450000	-.000570	.003669	.720830	-.122212
156.350000	-.001455	.002665	.720830	-.122212
162.250000	-.001787	.002179	.720890	-.122212
168.150000	-.001808	.001999	.720970	-.122537
174.050000	-.002249	.001279	.721104	-.122812
179.950000	-.003148	-.000082	.721231	-.122794
185.850000	-.003909	-.001435	.722179	-.121611
191.750000	-.004939	-.002525	.723365	-.119412
197.650000	-.005259	-.003926	.724774	-.115599
203.550000	-.006047	-.005956	.727802	-.104783
209.450000	-.008328	-.008428	.731979	-.089455
215.350000	-.010536	-.010641	.738810	-.064437
221.250000	-.012289	-.012990	.751239	-.016562
227.150000	-.013619	-.015990	.765110	.042236
233.050000	-.013254	-.019211	.781714	.129509
238.950000	-.014302	-.022008	.803443	.250760
244.850000	-.017652	-.024370	.829755	.382543
250.750000	-.019038	-.025156	.866690	.559417
256.650000	-.020576	-.025273	.908917	.747497
262.550000	-.022204	-.021929	.949954	.894500
268.450000	-.021121	-.011440	.997958	.988783
274.350000	-.018620	-.004260	1.034676	1.019241
280.250000	-.011814	.000605	1.052492	1.015934
286.150000	.008443	.002570	1.041378	1.003668
292.050000	.093863	.002295	1.000000	1.000000

SECTION DRAG COEFFICIENTS

Y/B/2	CDW/C	CDBOW/C	CDNOW/C	SUM CD/C	DRAG FR.	CHORD
0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	166.83000
.02500	0.00000	0.00000	0.00000	0.00000	0.00000	166.83000
.05000	0.00000	0.00000	0.00000	0.00000	0.00000	166.83000
.07500	.00053	-.00004	-.00006	.00054	.04050	166.17085
.10000	.00134	-.00007	-.00013	.00115	.08377	160.13300
.12500	.00148	-.00010	-.00015	.00123	.08634	154.19519
.15000	.00151	-.00013	-.00017	.00122	.08219	148.25869
.20000	.00095	-.00015	-.00019	.00061	.03774	136.39331
.25000	.00098	-.00018	-.00019	.00061	.03486	124.61067
.30000	.00094	-.00018	-.00023	.00052	.02693	113.93947
.35000	.00102	-.00018	-.00037	.00047	.02192	103.25828
.40000	.00105	-.00018	-.00041	.00046	.01933	92.59709
.45000	.00110	-.00018	-.00033	.00059	.02198	81.92590
.50000	.00098	-.00017	-.00025	.00056	.01847	72.16113
.60000	.00101	-.00013	-.00026	.00063	.01541	54.02146
.70000	.00104	-.00008	-.00029	.00068	.01112	35.88180
.80000	.00184	-.00010	0.00000	.00174	.02165	27.36278
.90000	.00210	-.00014	0.00000	.00196	.01866	20.90389
.95000	.00182	-.00011	0.00000	.00171	.01376	17.67444
1.00000	.00128	-.00005	0.00000	.00123	.00809	14.44500

DRAG TERMS

CDW= .001066   CDB= .000466   CDB/W= -.000131   CDW/B= .003129   CD WING-BODY= .001529

NACELLE DRAG COEFFICIENTS

NACELLE(S) AT Y=	16.33000	31.25000
Z=	-5.80000	-4.90000
WETTED AREA	1535.75183	1535.75183
ISOL. CDWAVE	.00016	.00016
BODY-ON-NACELLE CD	-.00000	-.00000
NACELLE-ON-BODY CD	-.00002	-.00001
OTHER NACELLES EFFECT CD		
DIRECT EFFECT	0.00000	-.00002
NAC-ON-ITSELF(IMAGE)	-.00001	.00000
OTHER NAC IMAGES	0.00000	-.00001
WING-ON-NACELLE CD	-.00003	-.00001
NACELLE-ON-WING CD	-.00820	

SUM NACELLE CD= .00002

TOTAL CD= .001545   REF. AREA= 9898.0000
BODY SWET= 7861.56   WING SWET= 18155.07

----TOTAL ELAPSED TIME, CP= 143.197 ----

PROGRAM CONTROL CARD
PLOT
ENTER INPTS---TAPE INPUTS
EXIT INPTS
ENTER GEOMPLT---GEOMETRY INTERFACE WITH PROGRAM GPLOT
969-500 WING CAMBER DESIGN 2

1 1 -1 1 0 1 0 0 13 1 9 19 0 0 0 0 0 0 2 7 0 0 1 3

9898.0000									
0.0000	2.5000	5.0000	10.0000	20.0000	30.0000	40.0000	50.0000	60.0000	70.0000
80.0000	90.0000	100.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
76.5900	4.7570	0.0000	166.8300						
83.1040	6.6250	0.0000	160.1330						
93.1650	9.5100	0.0000	149.7900						
116.9600	16.3330	0.0000	125.3500						
168.9300	31.2500	0.0000	77.2950						
225.8100	47.5440	0.0000	32.6810						
225.8100	47.5450	0.0000	32.6810						
258.2100	66.2500	0.0000	14.4450						
0.0000	-.0257	-.0514	-.5399	-2.1596	-3.7416	-5.2742	-6.7107	-8.0317	-9.2232
-10.2756	-11.1799	-11.9260	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	.0717	.1433	-.2642	-1.4414	-2.7155	-3.9850	-5.1363	-6.3250	-7.3623
-8.2899	-9.0945	-9.7303	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	.0571	.1341	-.1846	-1.1318	-2.1981	-3.2737	-4.3183	-5.3054	-6.2260
-7.0641	-7.8107	-8.4554	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	.1073	.2146	.1007	-.4093	-1.0538	-1.7387	-2.4365	-3.1219	-3.7857
-4.4164	-5.0366	-5.5493	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	.1033	.2066	.3293	.2678	.0849	-.1361	-.3854	-.6550	-.9361
-1.2186	-1.5041	-1.7909	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	.0360	.0720	.1353	.2423	.2909	.3122	.3288	.3448	.3510
.3614	.3669	.3694	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	.0360	.0720	.1353	.2423	.2908	.3121	.3287	.3447	.3509
.3612	.3668	.3693	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	-.0137	-.0275	-.0516	-.0897	-.1143	-.1238	-.1156	-.1012	-.0812
-.0555	-.0218	.0200	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
0.0000	.5700	.7140	.8720	1.0500	1.1450	1.2000	1.2300	1.2430	1.1700
.9370	.5460	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
0.0000	.5700	.7140	.8720	1.0500	1.1450	1.2000	1.2300	1.2430	1.1700
.9370	.5460	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
0.0000	.5500	.7120	.8720	1.0540	1.1560	1.2130	1.2350	1.2370	1.1270
.8830	.5070	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
0.0000	.5500	.7150	.8760	1.1250	1.1740	1.2350	1.2500	1.2230	1.0870
.8400	.4740	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
0.0000	.5700	.7270	.9020	1.0940	1.2200	1.2890	1.3150	1.2620	1.1050
.8420	.4730	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
0.0000	.5900	.7230	.9110	1.1340	1.2640	1.3430	1.3750	1.3200	1.1550
.8800	.4950	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
0.0000	.1340	.2610	.4950	.8800	1.1550	1.3200	1.3750	1.3200	1.1550
.8800	.4950	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000
0.0000	.1340	.2610	.4910	.8800	1.1550	1.2850	1.3750	1.3200	1.1550
.8800	.4950	0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000	-0.0000

0.0000	16.6700	33.3300	50.0000	66.6700	83.3300	100.0000	116.6700	133.3300	150.0000
166.6600	183.3300	200.0000	216.6700	233.3300	250.0000	266.6700	283.3000	295.0030	-0.0000
10.0000	8.5500	7.1000	5.6400	4.1700	2.7300	1.2800	-.1400	-1.6030	-3.0400
-4.5000	-5.9000	-7.4000	-8.8500	-10.2500	-11.7000	-13.2000	-14.6000	-15.7000	-0.0000
0.0000	23.5000	57.5000	89.0000	117.0000	126.0000	119.8000	108.0000	105.0000	107.0000
107.0000	106.0000	102.0000	94.0000	79.0000	59.0000	33.0000	8.0000	0.0030	-0.0000
213.4200	16.3390	-5.0000							
0.0000	2.0090	15.4700	21.5250	28.0170	32.0670	35.0400	-0.0000	-0.0030	-0.0000
2.8650	2.9830	3.6330	3.7700	3.6540	3.4200	3.4200	-0.0000	-0.0000	-0.0000
218.6703	31.2500	-4.9000							
0.0000	2.0080	15.4700	21.5250	28.0170	32.0670	35.0400	-0.0000	-0.0030	-0.0000
2.8650	2.9830	3.6330	3.7700	3.6540	3.4200	3.4200	-0.0000	-0.0000	-0.0000
261.0000	2.0000	-14.0000	25.0000	277.0000	11.0000	-14.0000	9.3300		
0.0000	50.0000	100.0000							
0.0000	1.5000	0.0000							

PLOT DATA

X Z	-0	-0	-0	-0	-0	-0	-0	-0	100RT
X Y	-0	-0	-0	-0	-0	-0	-0	-0	100RT
Y Z	-0	-0	-0	-0	-0	-0	-0	-0	100RT

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6.0 REFERENCES

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APPENDIX A

INTERACTIVE GRAPHICS

The cathode ray tube (CRT) display and program coding for the design and analysis system are based on the NASA-LRC CRT and associated software. However, all display portions of the system coding are subroutined or overlaid from the basic programs, so that the system could be readily converted to other CRT arrangements.

The basic input parameter required to activate the graphics routines is the executive card CRT (punched in columns 1-3), which may be placed at the beginning of the data deck, or anywhere else in the input that an executive card may be read. If the CRT card does not appear in the data deck, no graphic displays will be generated.

The CRT card is actually an on-off device. Successive readings of the CRT card either turn on the graphics, or turn them off and place an end-of-file mark on the hard-copy file, depending upon the previous status of the graphics routines. However, the usual mode of graphics operation is to place a CRT card at the beginning of the data deck, if graphics are desired.

BASIC CRT OPERATION

Several types of video displays are generated by the design and analysis system, using the NASA-LRC software. These include:

- Menus
  A list of display choices with corresponding function keys

- Edit tables
  A list of numbers with variable names

- Plots
  Displays of x-y plots

When a display is complete, one of two system messages will appear at the top of the video screen. If the display is a menu, the message AWAITING OPERATOR ACTION will appear. To continue processing, the user must press a defined function key, selected from the menu. The second system message is PLOT FRAME COMPLETE. When this message appears, the graphics software is programmed to allow (a) editing of the display variables, (b) resumption of program execution, or (c) hard copy plot generation. If (b) is selected, the user presses function key 3 (NEXT FRAME). Editing and hard copy options are discussed on pages 190 and 191.

Menus

Menus consist of a set of display choices, together with defined function keys. Some menu lines display sets of function keys. For instance, a menu line may say FN KEY 6 DISPLAYS WING THICKNESS. Pressing key 6 will bring up a second menu, with the message FN KEYS 1 THRU 20 IDENTIFY AIRFOIL NUMBER, which would require the user to select one of the input airfoils. It should be noted that the upper key number (20) is the maximum number of airfoils allowed in the input. For a particular case, however, the user may have input only 7 airfoils. If the user now presses an undefined function key (8 thru 54) the message AWAITING OPERATOR ACTION will appear and another function button must be chosen.

Edit Tables

If the display contains an edit table, the user may now use the console keyboard to type in a new value for any variable in the table. The variable name used on the display is first typed, followed by an equal sign and the new value, followed by the console keys RETURN and EØM. (e.g., CONSTR(3) = -1.0 will change the third value of the array CØNSTR to -1.0). The new value will be displayed at the top left of the video screen.

The LRC software allows the definition of only one edit format per display in the using program. It can happen that there are both fixed and floating point numbers on the screen to be edited. If this happens, the edit format can be changed by the typed-in message FØRMAT = XXX RETURN EØM (where XXX is the desired format). This format remains in effect so long as the display is up, i.e., until key 3 is pressed. In case of doubt, the display will identify the current format if the message FØRMAT = RETURN EØM is typed.

Special Usage of Key 55

Function key 55 is used in two ways. If the statement "RESUMES EXECUTION" appears on the menu line and key 55 is selected, the current graphic program will be terminated and execution will continue at the next executable statement encountered. If the statement "DISPLAYS PROGRAM ØPTION MENU" appears on the menu line, and key 55 is selected, the current menu will be erased and the previous menu redisplayed.

Hard-Copy Plots

Each time the system message "PLOT FRAME COMPLETE" appears on the display screen, the user has the option of generating Varian hard-copy plots of the current display, assuming the run terminates normally and the job control cards specify the correct post-processor. Selecting key 6 (RECORD PLOT) or key 8 (RECORD PICTURE) followed by key 3 (NEXT FRAME) will save the display information and continue processing.

GRAPHICS USAGE

The principal uses of the graphics routines in the design and analysis system are to display the configuration, edit input geometry, and to display and/or alter the basic program calculations.

There is no provision in the system to alter the input data stream on-line, so the intended usage of the graphics and the input data card set up must be carefully coordinated. Limited capability to redirect the system calculation sequence is available and these options are displayed on the CRT screen when encountered.

Geometry

Configuration geometry may be displayed either from the PLOT module, or, in simplified form, from the geometry module. The PLOT display draws a picture of the configuration on the screen (as instructed by the input view cards), but has no edit capability. All editing of geometry must be performed in the geometry module.

When the geometry module is entered from the executive to read or change configuration geometry (executive cards GEØM, GEØM NEW, FSUP or WGUP), the CRT program DISGEØM is used to display and/or edit the configuration geometry. The first menu generated gives the user the option of executing or bypassing the video displays:

```
FN KEY  1  DISPLAY AND EDITS GEOMETRY
FN KEY 55  RESUME EXECUTION
```

When key 1 is selected, the program option menu appears:

```
FN KEY  1  DISPLAYS CONFIGURATION PLANFORM
```

FN KEY 2 DISPLAYS FUSELAGE AREA VS X
FN KEY 3 DISPLAYS WING CAMBER (Z vs X)
FN KEY 4 DISPLAYS WING CAMBER (Z vs Y)
FN KEY 5 DISPLAYS WING CAMBER (Z/C vs Y)
FN KEY 6 DISPLAYS WING THICKNESS (Z/C vs X/C)
FN KEY 7 DISPLAYS FUSELAGE SECTIONS (NON-CIRCULAR)
FN KEY 8 EDITS CONFIGURATION CODES
FN KEY 9 EDITS PERCENT CHORD ARRAY
FN KEY 10 EDITS X,Y,Z AND CHORD (AIRFØILS 1-10)
FN KEY 11 EDITS X,Y,Z AND CHORD (AIRFØILS 11-20)
FN KEY 12 EDIT/DISPLAY WING T.E. (TZØRD)
FN KEY 13 EDIT/DISPLAY WING T.E. (TZØRD + ZLE)
FN KEY 14 EDIT/DISPLAY WING THICKNESS (Z/C vs X/C)
FN KEY 15 EDITS FUSELAGE X ARRAY
FN KEY 16 EDITS FUSELAGE Z ARRAY
FN KEY 17 EDITS FUSELAGE AREA ARRAY
FN KEY 18 EDITS X,Y,Z AND D OF NACELLES
FN KEY 19 EDITS NACELLE X ARRAY
FN KEY 20 EDITS NACELLE R ARRAY
FN KEY 21 EDITS X,Y,Z AND CHORD OF FIN AIRFOILS
FN KEY 22 EDITS X,Y,Z AND CHORD OF CANARD AIRFOILS
FN KEY 23 EDITS CAMBER Z ARRAY
FN KEY 55 RESUMES EXECUTION

The table below describes the function of each key.

KEY	FUNCTION
1	A plan view of the configuration geometry is displayed.
2.	A plot of fuselage area versus station is displayed.
3.	Given an airfoil number 1 through 20 (1 being most inboard) a side view plot of camber (camber value + Z of leading edge) versus station at the Y of the specified airfoil is displayed.
4.	Given a percent chord number 1 through 21 (1 at leading edge), a rear view plot of camber (camber value + Z of leading edge) versus Y at the percent chord specified is displayed.
5.	Same as key 4 but camber value versus Y
6.	Given an airfoil number 1 through 20 (1 being most inboard), a side view plot of airfoil half thickness (upper and lower) versus percent chord at the specified airfoil, is displayed. The array of thicknesses (THK) is displayed below the plot and may be edited by the

user. THK (1) represents the half thickness at the leading edge.

7. Given the fuselage segment number 1 through 4, and the section number 1 through 30 within the segment, the Y and Z coordinates defining the fuselage half-section are displayed. The horizontal X axis is positioned vertically at the fuselage centerline Z value (ZFUS).

8. The basic geometry input parameters JO through XBARIN are displayed on the screen and may be edited by the user. The program defined format is I4. If it is necessary to modify variables REFA, CBAR or XBARIN the user must first change the format to floating point, such as F8.4.

9. The percent chord array (XAF) is displayed on the screen and may be edited by the user.

10/11 Four arrays, XLED, YLED, ZLED and CLED representing the X, Y and Z coordinates of the input airfoil locations of the wing leading edge and the airfoil chord lengths are displayed on the screen and may be edited by the user. Key 10 displays coordinates of first 10 airfoils and key 11 the last 10 airfoils.

12/13 Keys 12 and 13 provide a special capability to remove "spikes" or irregularities in the wing camber surface. A plot of camberline Z values (from array WZØRD) or $Z + Z_{LE}$ versus Y along the wing trailing edge is displayed. The corresponding table of Z or $Z + Z_{LE}$ values is displayed in a table under the plot, which may be edited. When the NEXT FRAME key is depressed, the following menu appears:

```
FN KEYS  1  THRU 21 DISPLAY PERCENT CHORD LINES
            TRAILING EDGE MAY BE EDITED
FN KEY 33  TWISTS WING TO MATCH EDITED T.E.
FN KEY 34  RESTARTS WITH ORIGINAL CAMBER DEFINITION
FN KEY 44  SAVES NEW CAMBER DEFINITION
FN KEY 55  DISPLAYS PRØGRAM ØPTION MENU
```

KEY	FUNCTION
1-21	A plot of Z or $Z + Z_{LE}$ versus Y at the percent chord selected is displayed.
21	The Z or $Z + Z_{LE}$ array is displayed below the plot and may be edited.
33	If the trailing edge has been edited, the remainder of the camber surface definition is altered, by

	linear twist, to agree with the trailing edge change. The trailing edge is redisplayed.
34	Restart option. If the change to the trailing edge was made incorrectly, the original camber surface may be recalled and the editing redone. (The restart option is available until key 44 is depressed)
44	The wing camber surface, WZORD, which was altered in a scratch array until now, is permanently changed to match the surface displayed under key 33.
55	Return to redisplay complete option menu.

14	Given an airfoil number 1 through 20, a side view plot of airfoil thickness versus percent chord is displayed. The thickness array of the specified airfoil is also displayed below the plot and may be edited.
15	Given a fuselage segment number 1 through 4, the array of fuselage X values for the segment are displayed and may be edited.
16	Given a fuselage segment number 1 through 4, the array of fuselage Z values for the segment are displayed and may be edited.
17	Given a fuselage segment number 1 through 4, the array (A) of fuselage area values for the segment are displayed and may be edited.
18	Four arrays, X, Y, Z and D, representing the coordinates of the nacelle origins are displayed and may be edited.
19	Given a nacelle number 1 through 9, the array of nacelle X coordinates are displayed and may be edited.
20	Given a nacelle number 1 through 9, the array (R) of nacelle radii values are displayed and may be edited.
21	Given a fin number 1 through 6, the variables XL, YL, Z L, CL, XU, YU, ZU and CU, representing the X,Y,Z and chord lengths of the lower and upper fin airfoils are displayed and may be edited.
22	Given a canard number 1 or 2, the variables XI, YI, ZI, CI, XO, YO, ZO and CO, representing the X,Y,Z and chord lengths of the inboard and outboard canard airfoils are displayed and may be edited.
23	Given an airfoil number 1 through 20, the array (C) of camber values for the airfoil are displayed and may be edited.

Skin Friction Module/Near-Field Wave Drag Module

When the skin friction program executes, the force coefficient summary from the program may be seen, or bypassed, according to the menu below:

FN KEY 1 DISPLAYS SKIN FRICTION RESULTS

FN KEY 55 RESUMES EXECUTION

Similarily, function keys 1 and 55 display or bypass the summary results from the near field program when it executes.

Far-Field Wave Drag Module

When the far-field wave drag program executes, the menu choice of display or bypass first comes up. If display (FN key 1) is selected, the display program (DIS080) will give the user the option of generating displays as follows:

```
FN KEY  1  ERASES SCREEN
FN KEY  2  DISPLAYS GRID
FN KEY  3  DISPLAYS BODY AREA VS X
FN KEY  4  DISPLAYS ØPTIMUM BODY AREA VS X
FN KEY  5  DISPLAYS CONFIG AREA VS X
FN KEY  6  DISPLAYS RESTRAINED CONFIG AREA VS X
FN KEY  7  INTERRUPT PROGRAM TO ALLOW HARD COPY PLOT GENERATION
FN KEY  8  DISPLAYS FAR FIELD WAVE DRAG SUMMATION
FN KEY 55  RESUMES EXECUTION
```

The user's options at this point are different from the other displays. Here the user constructs the plot to include as many curves as desired, with or without a grid, and may or may not generate a hard copy plot. To view the configuration area plot, the user need only select key 5 (followed by key 1 to remove the plot). If the user wants a hard copy plot of all curves with a grid, he selects keys 2,3,4,5,6 and 7 followed by keys 6 or 8 and 3 (NEXT FRAME). He may then resume execution, display the drag summation or build a new display after erasing the current display with key 1.

If the user selects key 8, the menu is erased and the wave drag program drag summary is printed (illustrated by typical values):

```
          70 CHARACTER TITLE ARRAY FOR CURRENT CASE

     CASE= 14     MACH= 2.700        NX= 50     NTHETA= 36
     WING VOLUME CHECK
          EXACT VØLUME =                          11432.023
          EQUIVALENT BØDY VOLUME =                11429.954

     ENTIRE AIRCRAFT
          D/Q =                                    20.27199
          CDW =                                      .00263
          ØPT. CDW* =                                .002481

DRAG OF TRANSFERRED AREA DISTRIBUTIONS
```

ØPTIMUM EQ. BØDY CDW* = .00089225
AVERAGE EQ. BØDY CDW* = .00104445
POTENTIAL CDW* CHANGE = -.00015220

At this point, the system message PLOT FRAME CØMPLETE will appear. To get a hard copy plot of the display, press key 6 or 8.

To continue, the user selects key 3 which erases the screen and re-displays the function key menu.

NOTE: There is one instance when the wave drag display subroutine will not be called. That is when the restraint points exceed allowable storage of 33, which causes the optimization calculations to be omitted.

Wing Design Module

The graphics capability of the wing design program consists of:

- display of "bucket" plot, drag-due-to-lift factor (K_E) versus Cmo.
- K_E versus Cmo for camber surface constraint solutions, if requested
- Editing of the design solution variables (C_{mo}, C_L) and constraint or restart codes .
- Continuation to next input case or return to executive

The design camber surface, which is automatically stored in common block CAMBER, can be viewed in the geometry module, but not in the wing design module.

The initial display to appear in the wing design module is the bypass or display menu:

FN KEY 1 DISPLAYS BUCKET PLOT
FN KEY 55 RESUMES EXECUTION

When key 1 is selected, the optimum drag-due-to-lift versus Cmo "bucket" plot is displayed. Additional symbols are also plotted, giving the flat wing (+), uniform load (x), and three term (Δ) solutions. (The uniform load and three term solutions will be plotted only if those solutions have been calculated).

Symbols (θ) are then plotted, corresponding to solutions from the constraint options 1,2,3, and 4, if requested. And, finally, up to 10 symbols (θ) are plotted giving the option 4 solutions from previous design cases (if the current case is one of a series of wing design cases).

After the bucket plot is generated, the NEXT FRAME key beings up the set of current design inputs:

```
70 CHARACTER TITLE OF CURRENT CASE

          CMO =  .0200
       CLDZIN =  .1000
      RESTART = 2.0000
    CØNSTR(1) = 1.0000
    CØNSTR(2) = 1.0000
    CØNSTR(3) = 1.0000
    CØNSTR(4) = 1.0000
```

The user may edit any of the variables on the display. If editing is performed, the wing design case may then be re-executed when the NEXT FRAME key is again depressed, which generates the menu:

```
FN KEY  1 EXECUTES NEXT CASE
FN KEY 55 CALCULATES EDITED DESIGN POINT
```

If key 55 is selected, the program returns to the wing optimization overlay, and recalculates the wing design for the edited design inputs. If key 1 is selected, the program continues to the next statement in the normal execution process.

When the wing design case is completed, and key 1 is selected, a final option menu is displayed:

```
FN KEY  1 TERMINATES WING DESIGN PROGRAM EXECUTION
FN KEY 55 READS NEXT DATA CASE
```

The purpose of this choice is to permit the user to abort a series of wing design input cases once the desired wing design has been obtained.

Lift Analysis Module

Graphics options provided in the analysis module consist of:

- Display and editing of wing twist array
- Editing of configuration angle of attack, and canard and horizontal tail setting (if used)
- Editing of Mach number, and inputs SYMM, WHUP, and ANYBØD
- Display of wing pressure coefficients and fuselage upwash
- Display of force coefficient summary

The initial menu seen is:

```
FN KEY  1  DISPLAYS WING TWIST (DEG) VERSUS SPAN
FN KEY  2  EDITS WING TWIST ARRAY
FN KEY  3  EDITS CANARD ANGLES OF ATTACK
FN KEY  4  EDITS SYMM, WHUP and ANYBØD
FN KEY 55  RESUMES EXECUTION
```

The user selects the function key associated with the task desired, noting the following conditions:

1. If function key 1 is selected and no twist array was input, no plot will be generated, and the user will be required to select another function Key.
2. If function key 2 is selected, the variable TWISTN (the current number of twist angles in the array ATWIST) and the ATWIST array are displayed. If entries are added or deleted in ATWIST, a corresponding change must be made in TWISTN.
3. If function key 3 is selected, the variable ALPN (the current number of canard angles of attack in array TCA) and the TCA array are displayed. If entries are added or deleted in TCA, a corresponding change must be made to ALPN.

When key 55 is selected, the analysis module continues execution, halting with the menu,

```
FN KEY  1  DISPLAYS UPWASH VERSUS PERCENT CHORD
FN KEY  2  DISPLAYS UPWASH VERSUS PERCENT SEMI-SPAN
FN KEY  3  DISPLAYS WING PRESSURE VERSUS PERCENT CHORD
FN KEY  4  DISPLAYS WING PRESSURE VERSUS PERCENT SEMI-SPAN
FN KEY 55  RESUMES EXECUTION
```

which provides the display options indicated.

Selection of keys 1 through 4 brings up one of the following secondary menus:

```
FN KEYS  1  THRU 21 IDENTIFY SEMI-SPAN PERCENT
FN KEY  55  DISPLAYS PROGRAM OPTION MENU
```

```
FN KEYS  1  THRU 11 IDENTIFY PERCENT CHØRD
FN KEY  55  DISPLAYS PROGRAM OPTION MENU

FN KEYS  1  THRU 41 IDENTIFY SEMI-SPAN PERCENT
FN KEY  55  DISPLAYS PROGRAM OPTION MENU

FN KEYS  1  THRU 20 IDENTIFY PERCENT CHORD
FN KEY  55  DISPLAYS PROGRAM OPTION MENU
```

If no fuselage was input, function keys 1 or 2 will produce no response. Key 55 returns to the primary menu.

Upon resumption of the analysis calculations, program FINISH is entered which halts with the menu:

```
FN KEY  1  DISPLAY DRAG DUE TO LIFT PROGRAM RESULTS AND
           EDIT NEXT HORIZONTAL TAIL ANGLE
FN KEY 55  RESUMES EXECUTION
```

If key 1 is selected, the drag summary table is printed (illustrated with typical values):

```
               70 CHARACTER TITLE FOR CURRENT CASE

               MACH NUMBER =               2.70
               CONFIGURATION ALPHA =       0.00
               CANARD ALPHA =              0.00
               HORIZONTAL TAIL ALPHA =     0.00
```

CL	CD(OFF)	CM(OFF)	CD(ON)	CM(ON)
.00	.000551	.00801	.000645	.00598
.01	.000451	.00677	.000525	.00474
.02	.000480	.00554	.000533	.00351
.	.	.	.	.
.	.	.	.	.
.	.	.	.	.
.18	.018392	-.01424	.018121	.01627
.19	.020603	-.01548	.020311	.01751
.20	.022942	-.01671	.022639	.01874

NEXT HORIZONTAL TAIL ALPHA (THALP) = 1.50

In the table, the (off) and (on) refer to nacelles. Canard alpha and horizontal tail alphas are not printed if no canard or horizontal tail is present.

It is possible to trim the configuration by the proper selection of horizontal tail angle. If there will be another horizontal tail angle, its value is indicated as shown. The value may be edited by typing THALP = XXX RETURN EØM. Key 3 (NEXT FRAME) will then resume execution.

A broader editing capability for altering the calculation sequence is enabled by the next menus to appear. The primary menu sets up the choice:

```
FN KEY  1  ALLOWS USER TO VIEW AND EDIT MACH NUMBER,
           CONFIGURATION ALPHA AND CANARD ALPHA, FOR
           CURRENT AND NEXT EXECUTION CYCLE
FN KEY 55  RESUMES EXECUTION
```

Selection of key 1 displays the current Mach number, configuration alpha and canard alpha. In addition, it displays the next parameter in the cycle to change, which may be edited by the user (typical values are shown):

```
CURRENT MACH NUMBER =                          2.14
CURRENT CONFIGURATION ALPHA =                  1.70
CURRENT CANARD ALPHA =                          .50

NEXT CANARD ALPHA (CAN) =                       .95
                - or -
NEXT CONFIGURATION ALPHA (CØN) =               1.87
                - or -
NEXT MACH NUMBER (XMCH) =                      2.30
```

The program execution sequence is canard alpha loop, configuration alpha loop and Mach number loop, in that order. When the individual loops are complete, the words CURRENT and NEXT are replaced with LAST.

The user has the option of editing the variables CAN, CØN and XMCH when they appear on the screen.

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