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A COMPUTATIONAL SYSTEM FOR AERODYNAMIC DESIGN AND ANALYSIS OF SUPERSONIC AIRCRAFT

Part 2 - User's Manual

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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, covered in three separate documents:

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.

These three documents supersede NASA contractor reports CR-2520, CR-2521, and CR-2522, which described an earlier version of the system.

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CONTENTS

A COMPUTATIONAL SYSTEM FOR AERODYNAMIC DESIGN AND ANALYSIS OF SUPERSONIC AIRCRAFT

PART 2 - USER'S MANUAL

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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.

These three documents supersede NASA contractor reports CR-2520, CR-2521, and CR-2522 which described an earlier version of the system.

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 or KRONOS operating system and library file. It is designed for the CDC 6000 series of computers and is overlay-structured. The system requires approximately 115000_8 central memory words and uses eight peripheral disc files in addition to the input and output files.

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.

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.

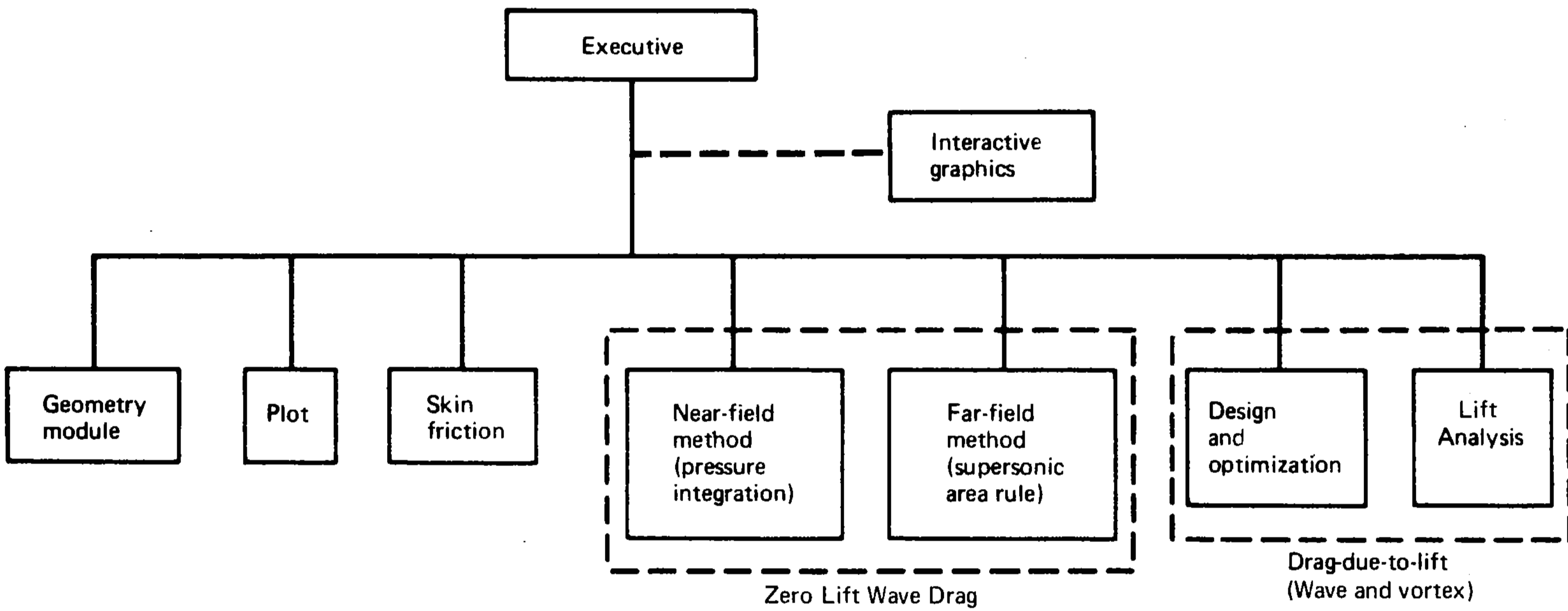


FIGURE 3.0-1.—INTEGRATED SUPERSONIC DESIGN AND ANALYSIS SYSTEM

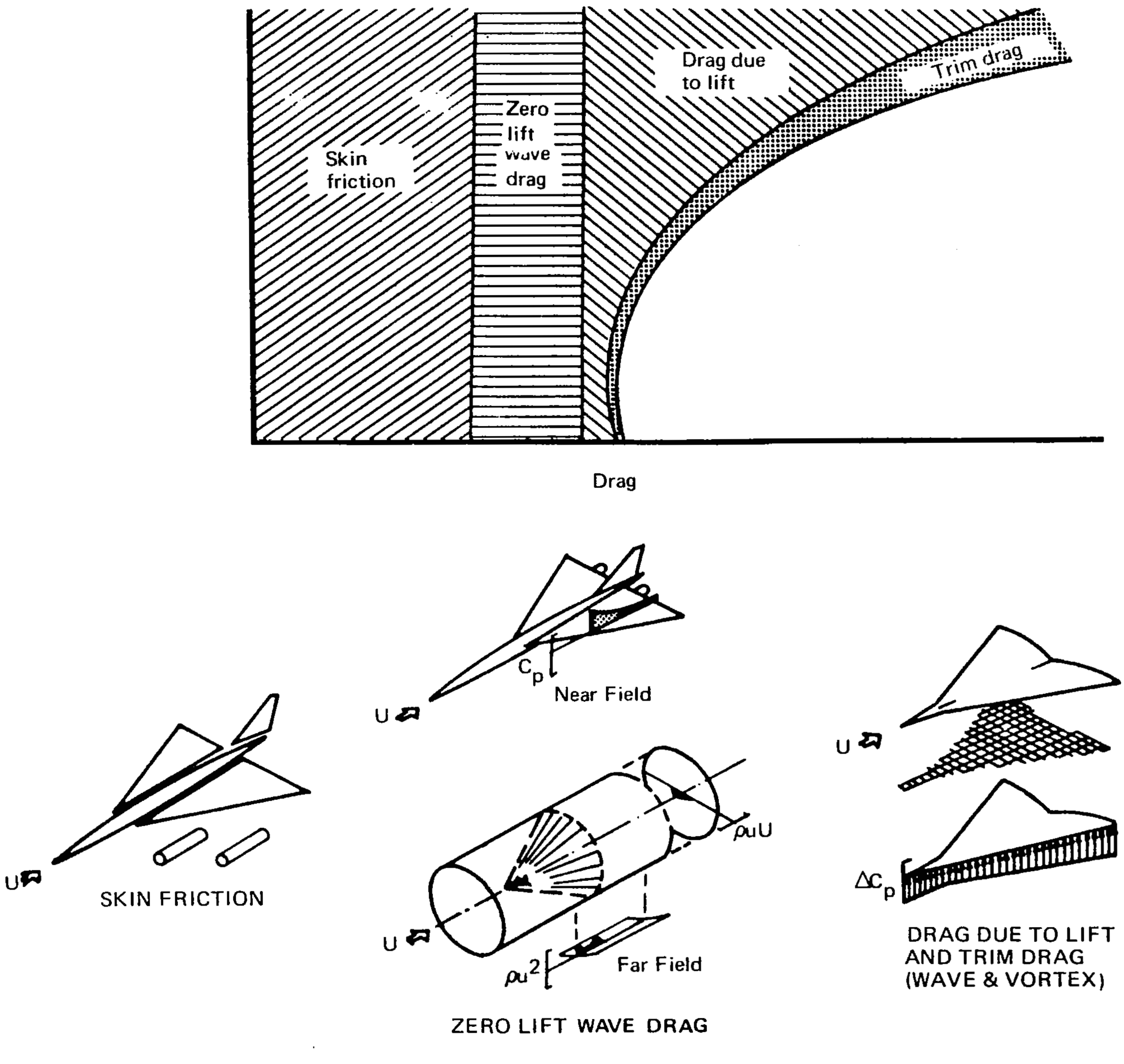


FIGURE 3.0-2.—DRAG BUILDUP

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 wing design module is the largest single program in the system. Therefore, some common blocks used in the wing design 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). There are some restrictions (relative to the reference 2 input) in the allowable number of input defining stations, however. The geometry format and limits are given in section 4. Some optional geometry has also been added. This includes provisions for fuselage perimeters to be input (if needed by the skin friction program), and provisions for

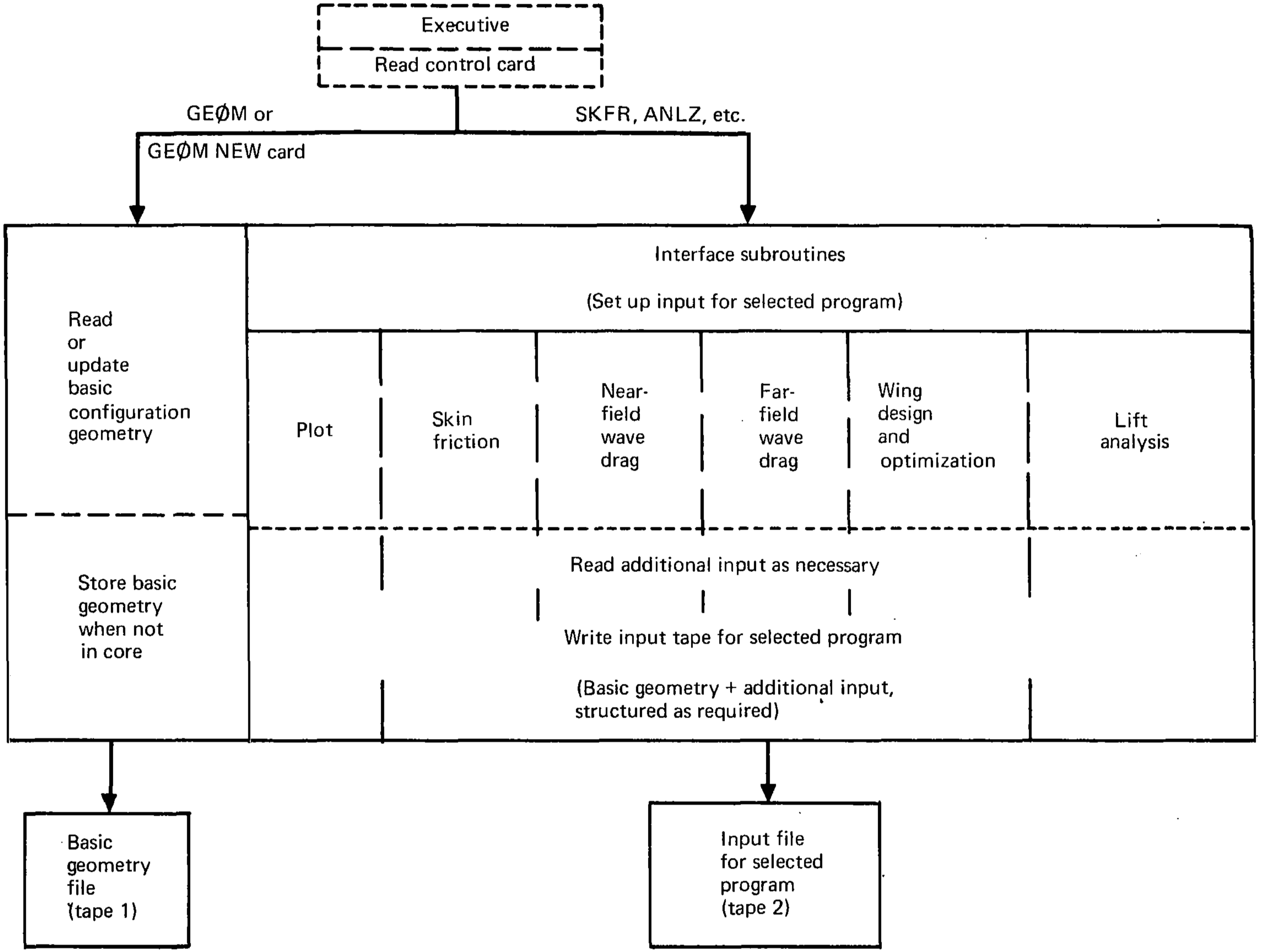


FIGURE 3.2-1.—SCHEMATIC OF GEOMETRY MODULE

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 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.

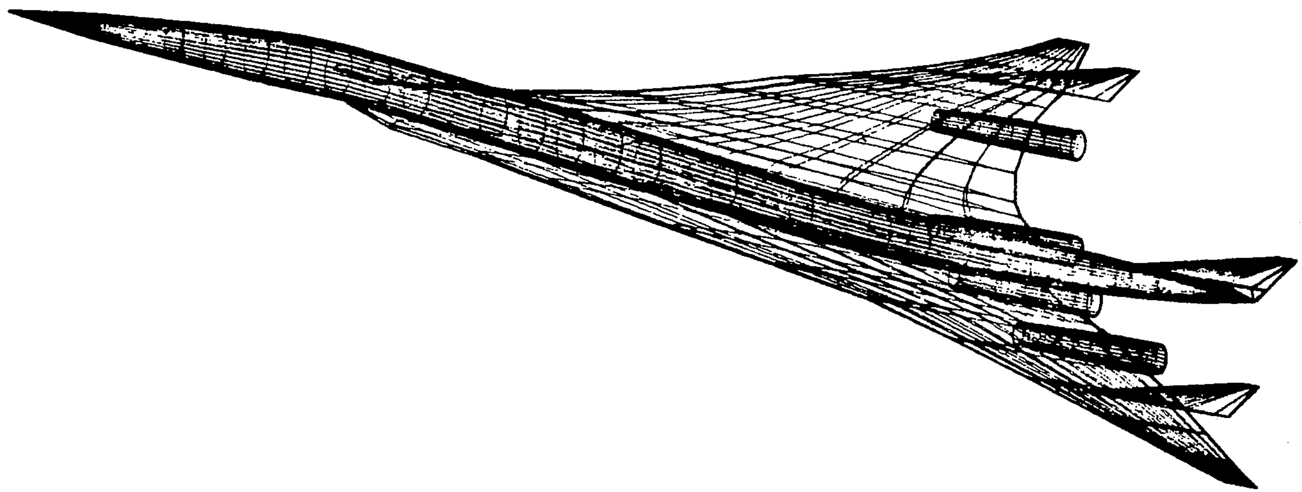


FIGURE 3.3-1.—TYPICAL PLOT PROGRAM DRAWING

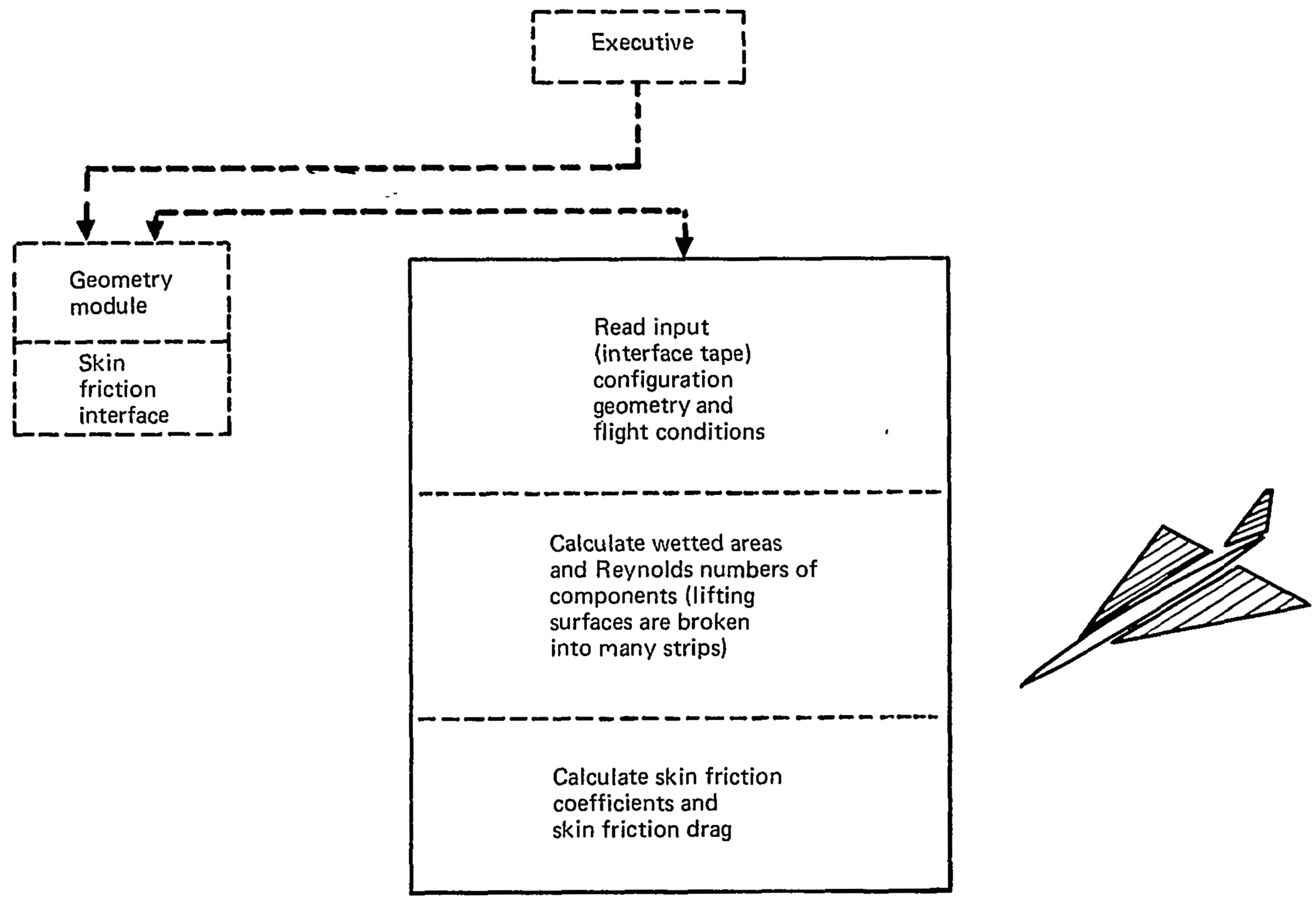


FIGURE 3.4-1.—SCHEMATIC OF SKIN FRICTION PROGRAM

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

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

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

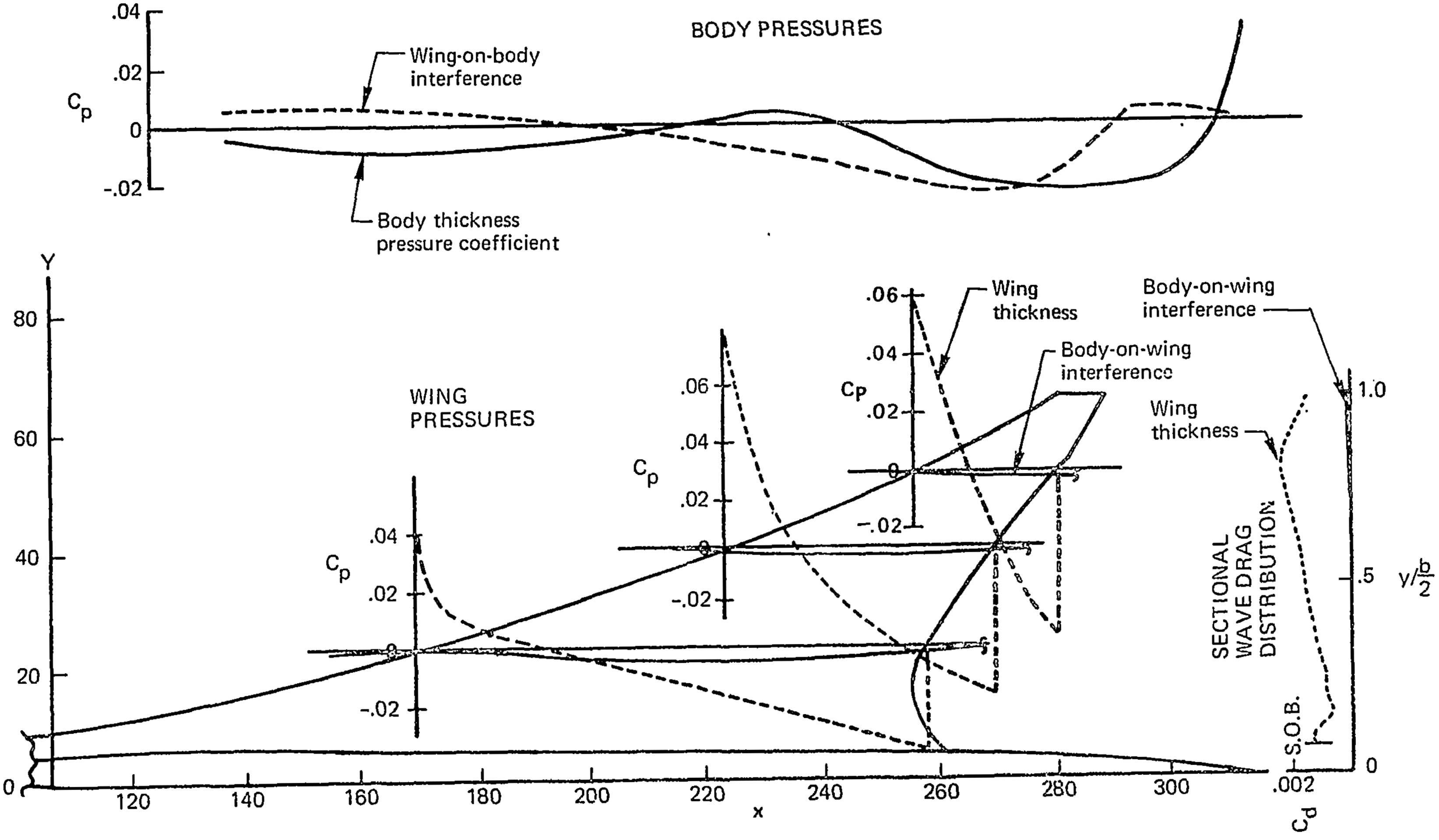
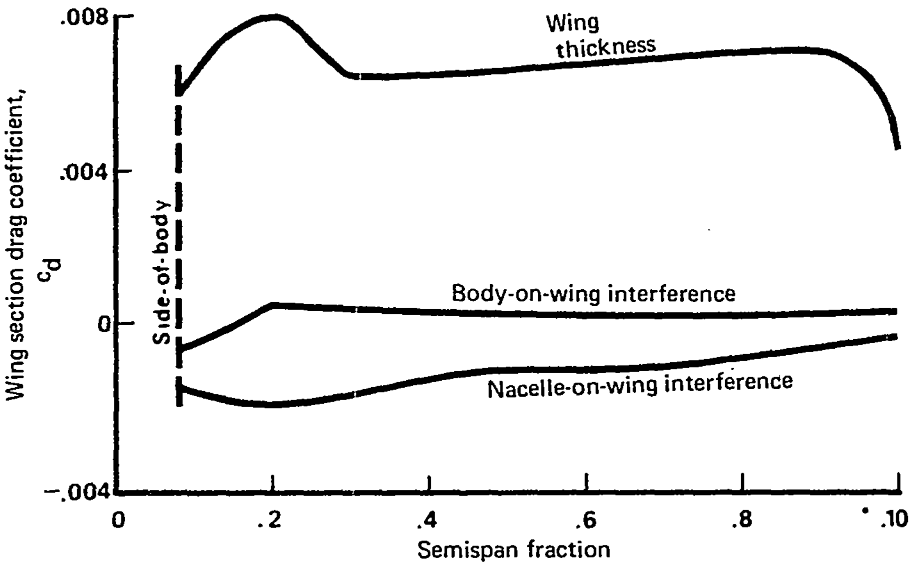


FIGURE 3.6-1.— WING-BODY SOLUTION, M = 2.6



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).

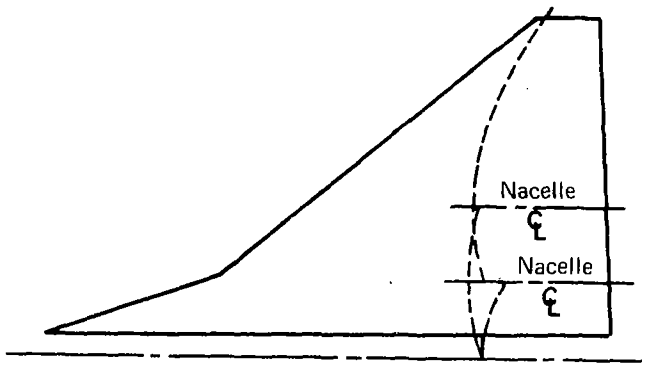
3.7 Wing Design and Lift Analysis

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.

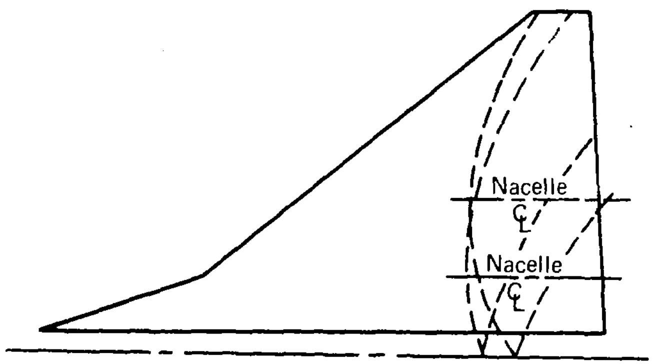
- Lift analysis - to define the lifting pressure distribution acting on a given wing camber surface shape, and calculate the associated force coefficients.

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 at zero lift

- Upper surface pressure coefficient level and/or streamwise gradient

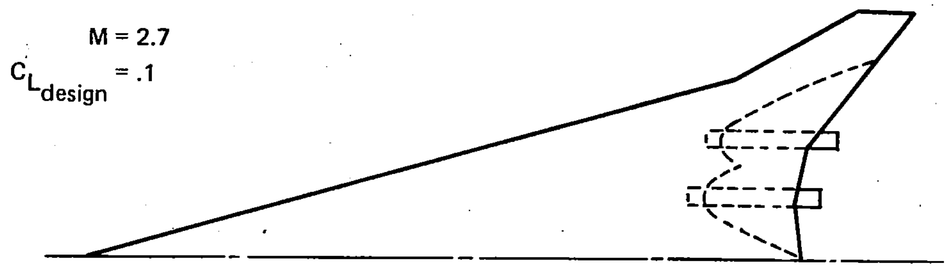
- Ordinate at defined planform locations

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 level and pressure gradient 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).



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

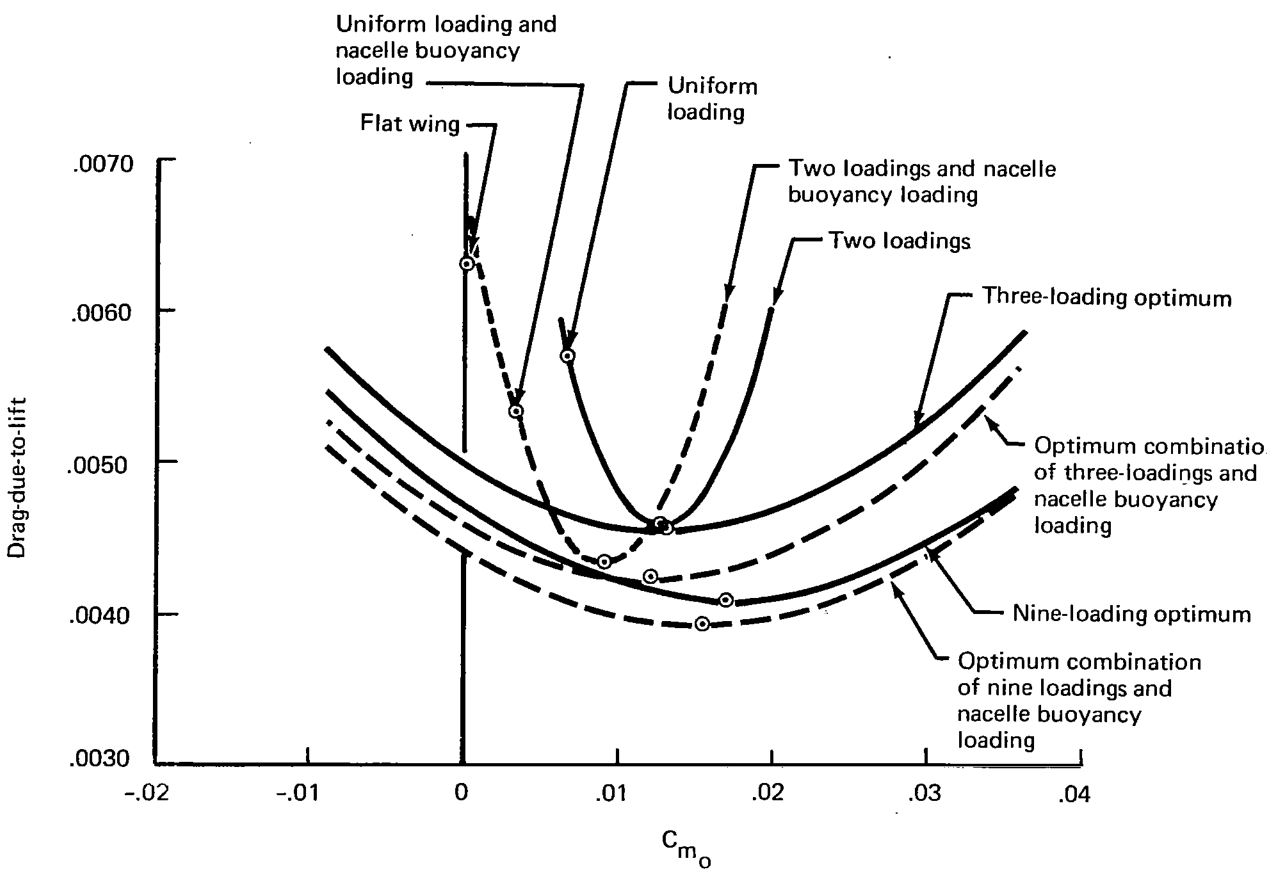


FIGURE 3.7-1.—EFFECT OF NUMBER OF LOADINGS ON WING DESIGN

- 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.

- Optional constraints on camber surface ordinate at specified planform locations.

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 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. (The bucket plot is not used with ordinate constraints, however, and only the solution corresponding to the design point values of C_L and C_{mo} is printed.)

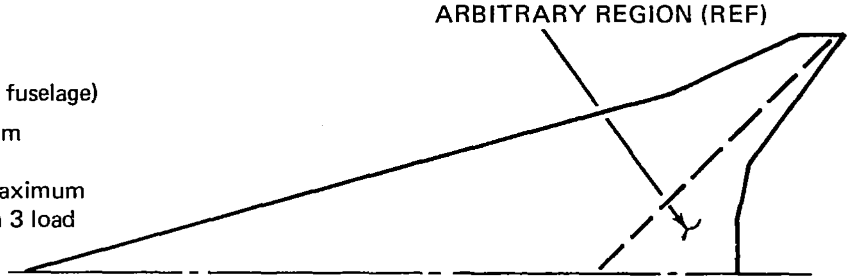
Pressure constraints. - The use of a large number of basic wing loadings permits great flexibility in identifying a theoretically optimum lifting pressure distribution. Such an optimum may be physically unrealistic, however. Linear theory contains no limitations on allowable surface pressures, e.g.,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

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 for a basic wing planform (no fuselage) is illustrated in figure 3.7-2. For a given planform and set of loadings, the program cycles to find an optimum pressure distribution (least drag) subject to input constraint conditions. First an optimum loading combination is found, then the corresponding peak pressure level and gradient are located. If either violate the input limits, a new optimum loading is found

DESIGN POINT
$M = 2.7,\ C_L = .10$

1. Basic wing solution (no fuselage)
2. $C_{P_{level}}$ limit = .7 vacuum
3. Gradient limit set by maximum obtained with optimum 3 load solution



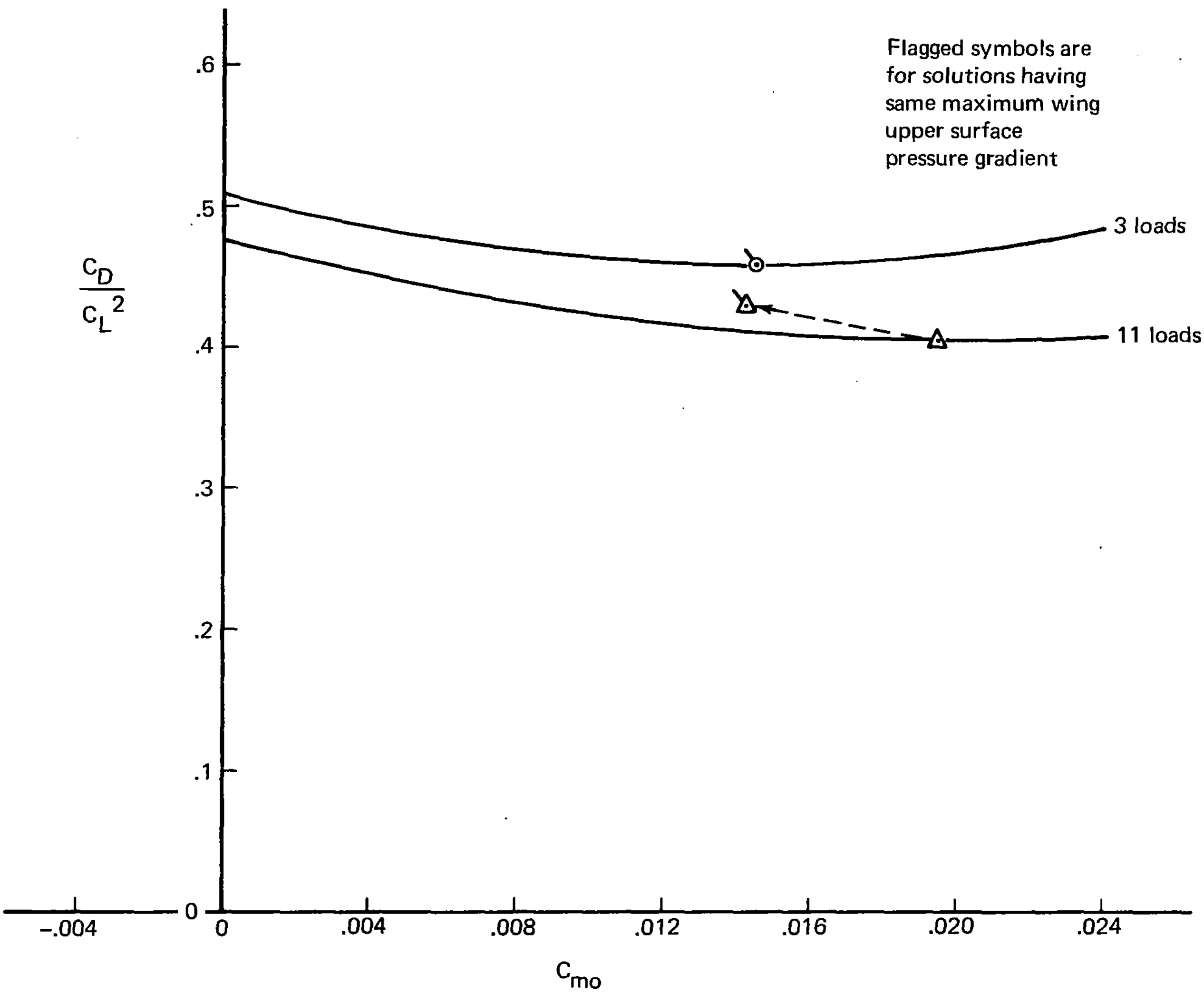


FIGURE 3.7-2.—EFFECT OF PRESSURE CONSTRAINTS

with a pressure constraint applied at the location of the maximum pressure violation. The new optimum is then examined, etc. Gradient is everywhere satisfied before level is constrained, as described in the theory document, volume 1.

The cyclic operation continues until the wing pressure distribution everywhere satisfies the pressure constraints. In the example case shown, the effect of adding pressure constraints shifts the drag minimum from the bucket plot level to the level indicated by the flagged symbol.

It can occur that the input pressure gradient constraint cannot be satisfied within the other constraint bounds of Cmo, wing thickness pressures and/or Z constraints. In this case, the program automatically increases the input acceptable gradient level by 20 percent and tries again. This process will continue until the gradient level is satisfied. No similar option is applied to the pressure level constraint, however; if pressure level cannot be satisfied, the program halts.

A further discussion of pressure constraint application is given on page 29.

Loading definitions. - A tabulation of the pressure loadings available within the design program is given in Table I on page 24. The configuration dependent loadings may be used both as a superimposed, 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).

- As a loading definition (12-14), a configuration-dependent loading produced by wing camber may be introduced in addition to its independent effect. The optimization then could cancel the lift of the independent effect with this camber-generated 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.

Fuselage in wing design solution. - The fuselage may be included in the wing design solution by input of fuselage geometry and specifying a side-of-fuselage semi-span station. The resulting solution is then split into two parts: the wing part (outboard of side-of-fuselage station) with loading definitions as described previously, and the "fuselage" part (inboard of side-of-fuselage station). Loadings inboard of the side-of-fuselage station are of

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 $(1.05\ c\text{-}x)^{0.5}$
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

the "carry-over" type, and are dependent functions of the loadings outboard of the side-of-fuselage.

Drag of the "fuselage" part is calculated by applying the carry-over loadings to the fuselage camberline. The outboard, or wing, part is handled as for the wing alone case, with integrations beginning at the side-of-fuselage. The wing-fuselage solution thus reflects the interdependence of wing and fuselage contributions to the wing design optimization.

There are several considerations of importance in the wing/fuselage solution:

- Wing paneling (internal definition of wing geometry) may require a slight shift in the input side-of-fuselage station. This is accomplished automatically in the program and an explanatory note is printed.

- The fuselage attitude and wing camberline at the side-of-fuselage must approximately align for the drag integrations to be valid. Experience has shown that it is necessary for Z constraints to be applied to the wing camberline at the side-of-fuselage for this to occur. (Z constraints are discussed in more detail on page 33).

- For convenience, the fuselage attitude (relative to the basic geometry definition) in both the lift analysis program (to generate upwash loading) and the wing design program can be changed without revising the basic geometry. (In the lift analysis module, this is done by a special application of the pressure limiting option. Set FLIMIT=1.0, an appropriate value of vacuum fraction VACFR, and increment the fuselage angle of attack in TLALP).

Inclusion of the fuselage in the wing design solution is illustrated in figure 3.7-3. In the example case shown, the wing camberline at the side-of-fuselage was constrained at the four locations indicated. A bucket plot is not produced when Z constraints are used; however, the effect of adding pressure constraints to the solution is illustrated by the symbols on the drag-due-to-lift versus C_{mo} plot.

Use of configuration-dependent loadings. - An example of the inclusion of a configuration-dependent loading is illustrated in figure 3.7-4, showing "reflexing" of the wing due to nacelle influences. The wing trailing edge is bent upward locally, or reflexed, to take advantage of positive pressure coefficients from the nacelle pressure field.

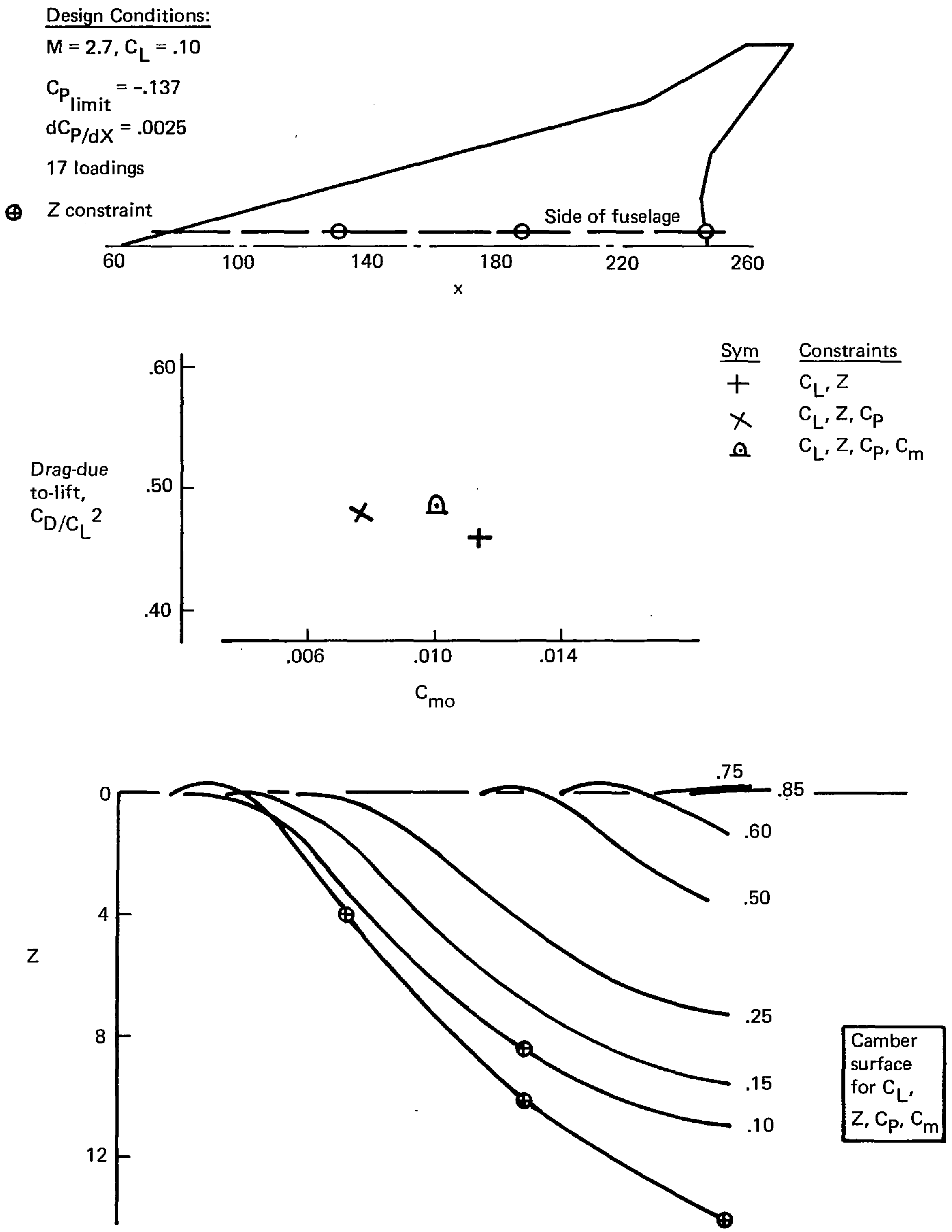


FIGURE 3.7-3.—WING DESIGN OPTIMIZATION WITH FUSELAGE AND Z CONSTRAINTS

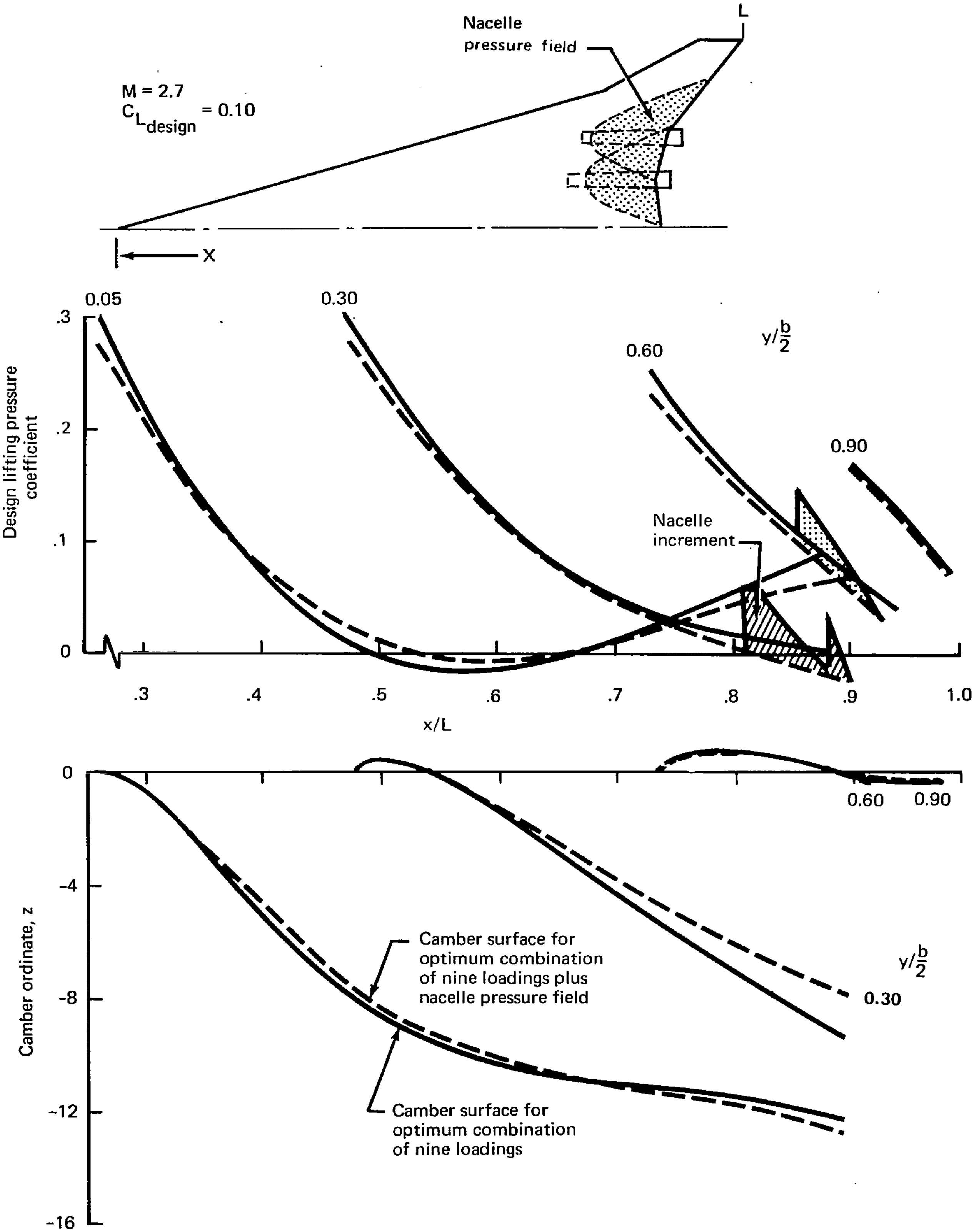


FIGURE 3.7-4.—EFFECT OF ADDING NACELLES TO WING DESIGN SOLUTION

The loadings due to the fuselage include both lift caused by the fuselage upwash field and also lift due to asymmetric distribution of fuselage volume above and below the wing (non mid-wing arrangement).

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 a mid-wing arrangement. However, if the fuselage buoyancy lift is zero, 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 upwash field and corresponding loading is calculated by the analysis program, and the design solution is performed. Because the resulting wing shape probably differs from the shape used in the initial upwash solution, the upwash loading is incorrect. It may be desirable to then rerun the wing design solution and/or alter the fuselage angle of attack. A representative program executive card sequence would be:

Event	Executive Card
Define fuselage	GEØM
Calculate upwash loading	ANLZ (WHUP=1.0)
Wing design solution	WDEZ
Recalculate upwash loading with new camber surface	ANLZ (WHUP=1.0, TIFZC=3.0)
Wing design	WDEZ

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

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 3.7-5. 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

If ordinate constraints are specified, they are included with each of these four types of solutions.

The four types of solutions are not completely independent. If the C_L and constrained pressure solution is requested, then the program must first generate the C_L only solution. Similarly, if the C_L, C_{mo}, and constrained 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.

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).

Pressure constraint application. - The constraints applied on the pressure distribution of the wing upper surface are of two types. Both pressure level and the longitudinal gradient of pressure can be constrained. This has been done because linear aerodyanmic theory can produce pressure distributions (in terms of pressure

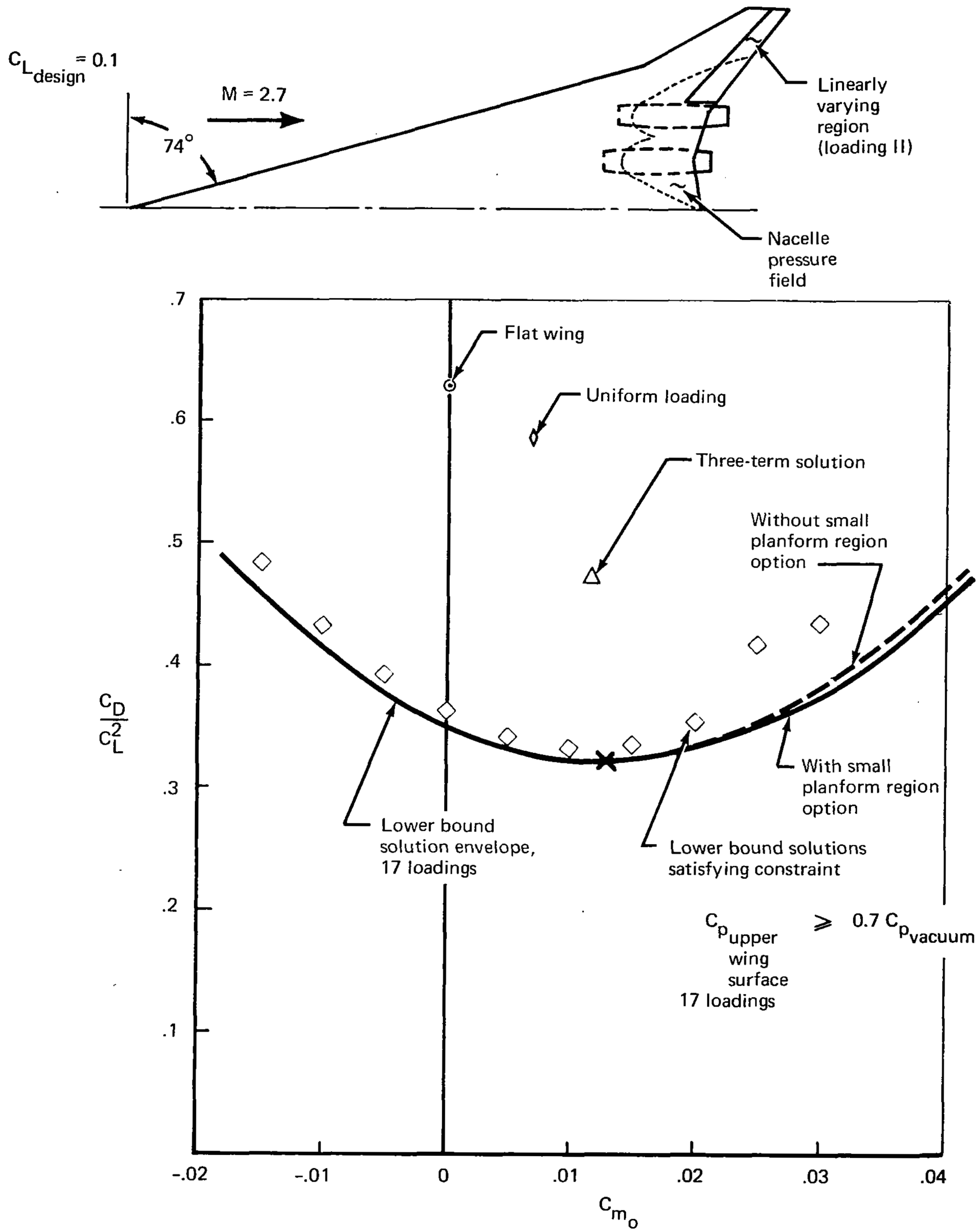


FIGURE 3.7-5. —PLANFORM REGION OPTION

coefficient) requiring pressure lower than vacuum on the wing upper surface; the first type of constraint can be used to restrict pressure levels to more realistic values. High, positive values of pressure gradient tend to produce boundary layer separation, and the second type of constraint offers a means of control. Constraints on pressure gradient are also quite valuable in controlling wide variations of pressure level.

Pressure constraints of both types are specified by the user with tables of acceptable values. The tabular format has been chosen so that the values can be varied depending on planform location. After a design solution has been obtained in terms of the component loading factors A_i, the pressure level and gradient on the wing upper surface are compared with the acceptable values. If either or both types of conditions are violated, the program retains the planform locations of the violations in order to apply constraints.

When pressure level violations are detected, the program computes the level of lifting pressure coefficient at the critical planform location which in combination with thickness pressure will produce a wing upper surface pressure equal to the acceptable value. A constrained lifting pressure coefficient set at 95% of the allowable value is then imposed on subsequent solutions for the optimum combination of loadings.

When pressure gradient violations are detected, an acceptable value of pressure gradient due to lifting pressure coefficients is calculated at the critical planform location. Account is taken of the gradient due to wing thickness pressures and any configuration dependent loadings. A constrained gradient set at 75% of the allowable value is then imposed on subsequent optimizations.

The two factors (95% and 75%) have been applied to diminish a tendency of the solution to vary slightly above the limiting value in the immediate vicinity of the constraint. This type of behavior is most noticable for cases involving the configuration-dependent loadings, which are specified by tables as a function of planform location and are not necessarily smoothly varying functions.

The application of constraints is cyclic. If the solution without pressure constraints does not satisfy the pressure criteria, a pressure constraint is imposed and a second solution is obtained and tested. If the second solution does not satisfy the criteria, a second pressure constraint is imposed, and a third solution generated and tested, and so forth.

Constraints imposed early in this cyclic fashion can become unnecessary as later constraints are imposed. The solution process includes logic to search for unnecessary constraints after the second and subsequent cycles (the first constraint would not be applied if it were not necessary). At each solution cycle, any unnecessary constraints are removed, and the next cycle has fewer pressure constraints. The constraint identified on the current cycle as most critical is also deleted from the next solution cycle; with fewer pressure constraints, it might no longer be the most critical pressure location.

In summary, the results of each solution cycle are tested in three ways. First, the Lagrange multipliers of the solution corresponding to the pressure constraints are tested. If any constraints are unnecessary, they are removed, and another solution cycle is begun. If all of the pressure constraints are necessary, then the most critical pressure gradient is tested. If it exceeds the allowable value, another gradient constraint is imposed, and another solution cycle is begun. If the solution pressure gradient is satisfactory, then the most critical pressure level is tested. If it exceeds the allowable value, another level constraint is imposed, and another solution cycle is begun. If the most critical pressure level is acceptable, the solution as a whole is acceptable. Solution cycling continues until either an acceptable solution is obtained (cycling stops), or until a limit on the number of constraints is reached.

There are two types of limits on the maximum number of pressure constraints that can be imposed. One limit is imposed by the number of loadings used. The total number of constraints, including those on lift, pitching moment, configuration-dependent loadings, ordinates, and pressure, can at most be equal to the number of loadings. This situation is undesirable for it leaves no degrees of freedom for drag minimization; consequently, a program limit has been set so that two degrees of freedom remain free for drag minimization (for small numbers of loadings, this is reduced to one).

The second limit imposed on the number of pressure constraints is dictated by the number of loadings that are free to influence longitudinal pressure gradient. A maximum of ten loadings does so - loadings 2, 5-9, and 11-14. It has been thought desirable to leave one degree of freedom for drag minimization for gradient constraints. The number of permissible gradient constraints is reduced by one more whenever a constraint is imposed on C_{mo} since C_{mo} constraints are satisfied primarily by the same x-dependent loadings used to satisfy gradient criteria.

If the program reaches a limit on the number of pressure constraints, it checks to see if gradient constraints have been imposed. If one or more gradient constraints have been imposed, the program arbitrarily increases the gradient criterion table by 20 percent, and begins anew with no pressure constraints. This process is also cyclic and can be repeated up to 50 times before halting with the solution produced by the last cycle.

If no pressure gradient constraints have been used, the program stops cycling upon reaching either one of the two constraint limits, and retains the last solution.

Ordinate constraints.- Linear theory produces wrinkles, or kinks, in the computed camber surface aft of wing leading edge breaks. This is especially noticeable near the wing apex, or at the side-of-fuselage station if there is a fuselage. As a result, 8 constraint provisions are provided in the wing design input.

As many as five ordinate constraints can be applied at arbitrary locations on the wing planform, provided that each of the constraints be on a camber-calculation line. The program tests the span stations of the ordinate constraints and shifts them to the nearest calculated camber line if they do not lie on one. (This has been done to avoid difficulties with two-dimensional interpolation near the side-of-fuselage).

The program next checks constraint planform locations chordwise. If any constraints are downstream of the trailing edge, they are moved to the trailing edge. If any are ahead of the leading edge, the case is stopped.

When ordinate constraints are used, several cases should probably be run. For the first case, both the maximum number of component loadings being considered and the maximum number of ordinate constraints being considered should be used, and a RESTART deck should be generated. Given RESTART capability (described on page 35), a reduced number of ordinate constraints may be imposed. Any number of ordinate constraints can be deleted, starting with the last one. (The ordinate constraint order cannot be changed, however). Cases without ordinate constraints can also be run from a RESTART deck which includes the component loading ordinate data.

Given this logic, some thought should be devoted to the order in which ordinate constraints are applied. Difficulties possibly requiring ordinate constraints sometimes arise at the centerline of a wing alone, at the side-of-body wing station, and at wing stations having substantial change of sweep angle. One would be tempted, for example, to impose ordinate constraints at, say, 65 percent, 35 percent, and 85 percent chord of the side-of-body wing station, leaving two ordinate constraints free for use further outboard. It would then be feasible, using RESTART, to run a case with one ordinate constraint at 65 percent chord on the side-of-body station, another case with two ordinate constraints at 65 and 35 percent, and so on.

The ordinate constraint capability has one other important feature with respect to RESTART capability. Although planform location of ordinate constraints cannot be changed in a RESTART deck, the constrained values of ordinate can be changed from case to case.

In general, ordinate constraints should be used sparingly, since they compromise the number of component loadings available for drag minimization.

Solution over-constraint. - There are six types of constraints that can be imposed on the wing design optimization. Constraints can be imposed on:

(1) Lift coefficient

(2) Pitching moment coefficient at zero lift

(3) Body buoyancy, body upwash, and nacelle buoyancy loadings

(4) Pressure gradient on the wing upper surface

(5) Pressure level on the wing upper sucface

(6) Camber (Z) ordinates

It is certainly possible to specify an over-constrained solution -- 5 camber ordinate constraints for a case combining three component loadings, for example. Consequently, a test has been placed in the first part of the optimization program OPTIMUM to detect and correct this situation. The test first sums the number of constraints types (1), (2), (3), and (6) above. If the sum is either greater than or equal to the number of component loadings, the number of ordinate constraints is reduced so that the sum is one less than the number of loadings. If the altered number of ordinate constraints is negative, the program halts; otherwise it proceeds normally.

Loading selection. - The loading definitions used in the program are tabulated on page 24, consisting of both analytically defined and configuration - dependent type loadings. These may be input in any order, subject to the condition that the camber-generating version (12-14) of the configuration dependent loadings may not be used without also using the corresponding configuration dependent loading (15-17).

Experience with the wing design program has shown that combinations of the higher order X term loadings (i.e., the X term loadings other than loading 2) tend to produce excessive twist or waviness in the calculated camber surface unless constraints are imposed on wing upper surface pressure level and gradient.

It is always good practice to run the wing alone with three basic loadings (uniform + linear spanwise + linear chordwise) in addition to any more sophisticated wing design case. Although the three term case has little capability for handling multiple constraints, it serves as a check on the average pressure gradient the wing can be designed to, in addition to providing a quick approximation to the optimum 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}, ordinate, 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. 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 (using RESTART=1.) 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 are written 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 much the same as RESTART= -1., but RESTART= -1. is intended for use 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 all lesser combinations of loadings are available. Successive cases can then be run with different loading combinations to check the design sensitivity to certain loadings, without repeating the basic loadings calculations. Any combination of loadings involving the set used when the RESTART data were generated may be employed. The number and the order of the loadings may be altered as desired.

If the RESTART data deck is used, it is not necessary to recalculate any configuration-dependent data (since these are preserved along with the basic wing loading data). With respect to the wing design solution that was possible at the time the RESTART data were generated, the RESTART deck may be used for any wing design having:

1) The same or fewer loadings (order may vary)

2) The same or fewer ordinate constraints (order cannot vary). The value of Z at these locations can be changed, however.

3) Same fuselage geometry, angle of attack, and side of fuselage station.

4) Any C_L, C_{mo}, or pressure constraint condition.

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 a standard set of 23 semi-span stations 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, for a basic wing case, 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

<u>Fuselage solutions</u>. - 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.

Downwash "shift" options. - The basic theoretical solution employed from canard or wing propagates directly aft. Since the downwash in the real flow case must follow the fuselage contour, a shift feature in the program translates the downwash field laterally to account for fuselage radii change between a generating (canard or wing) and affected (wing or tail lifting surface. The downwasy shift can have an appreciable effect on the calculated characteristics, as shown in figure 3.7-6.

The shift feature is controlled by input codes. If the control codes are left blank, the downwash will be shifted according to the side-of-fuselage Y values of canard, wing, or tail. Alternatively, the downwash can be either unshifted, or shifted a specified amount, as described in the data input section (4.0).

Experimental comparisons. - Theoretical calculations for a typical supersonic transport configuration are compared with corresponding wind tunnel data in figures 3.7-7 and 3.7-8 (wing-fuselage-nacelles) and figures 3.7-9 and 3.7-10 (incremental effects of

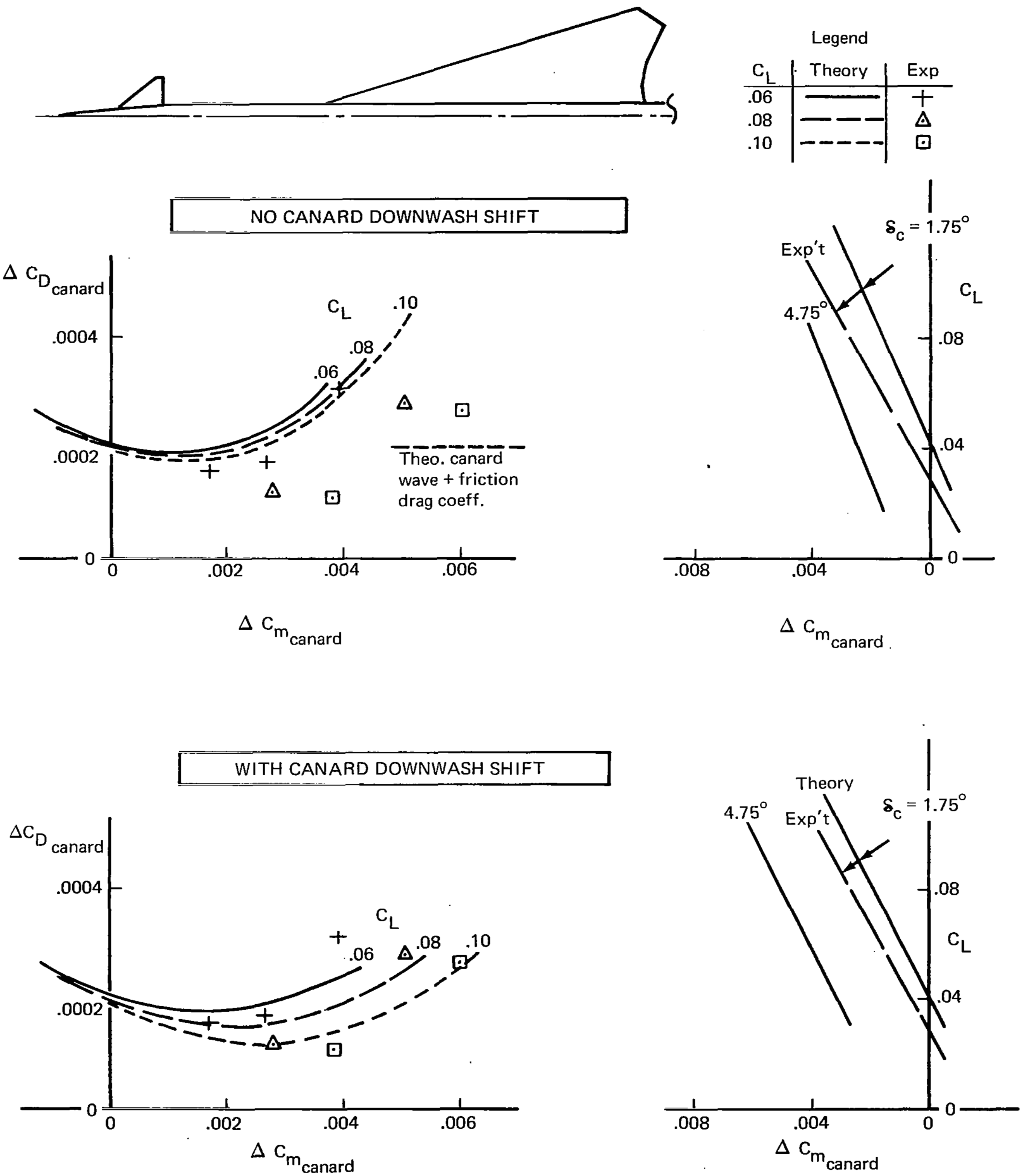


FIGURE 3.7-6.—EFFECT OF CANARD DOWNWASH SHIFT ≈ M = 2.7

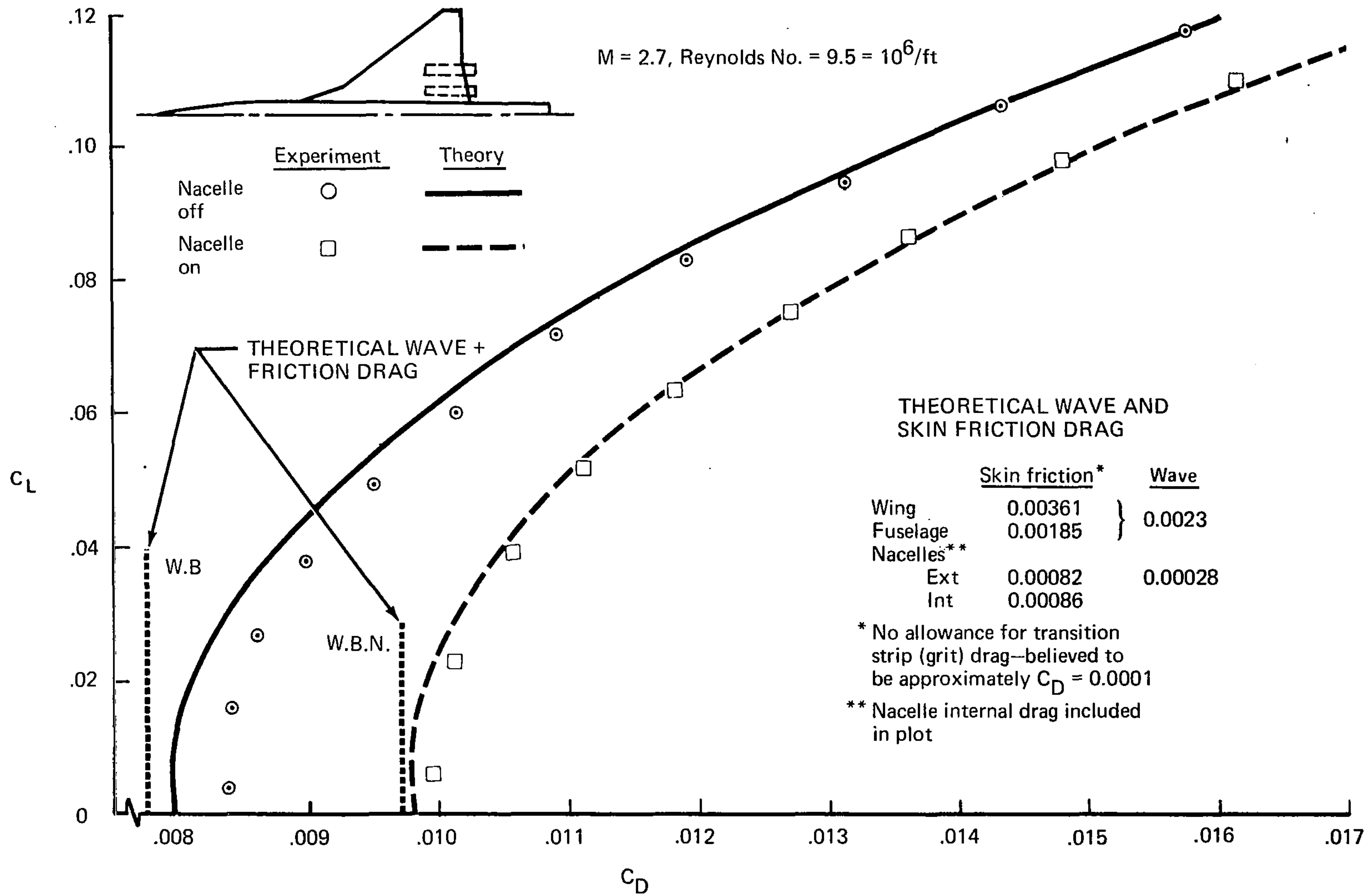


FIGURE 3.7-7. —DRAG POLAR COMPARISON

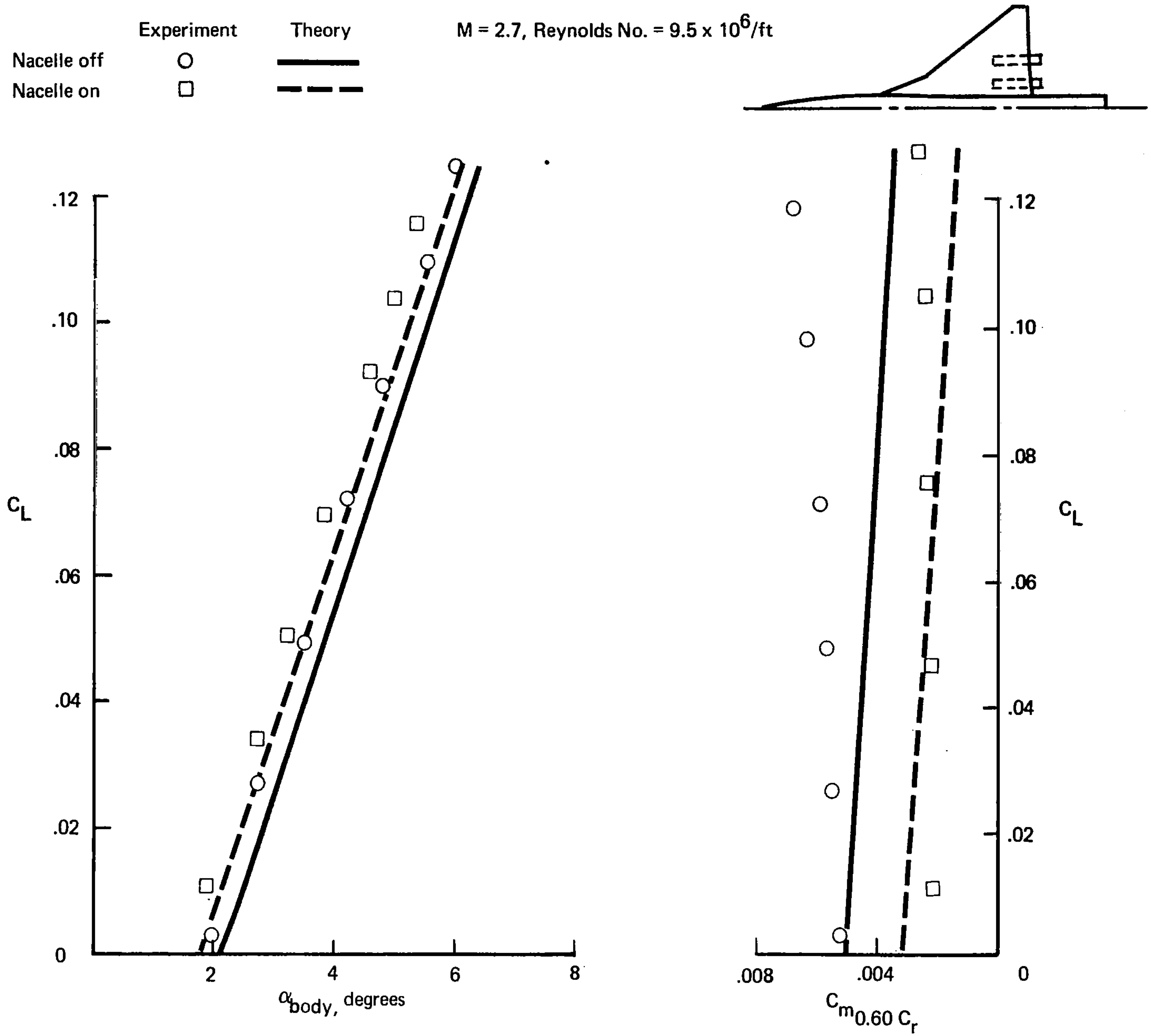
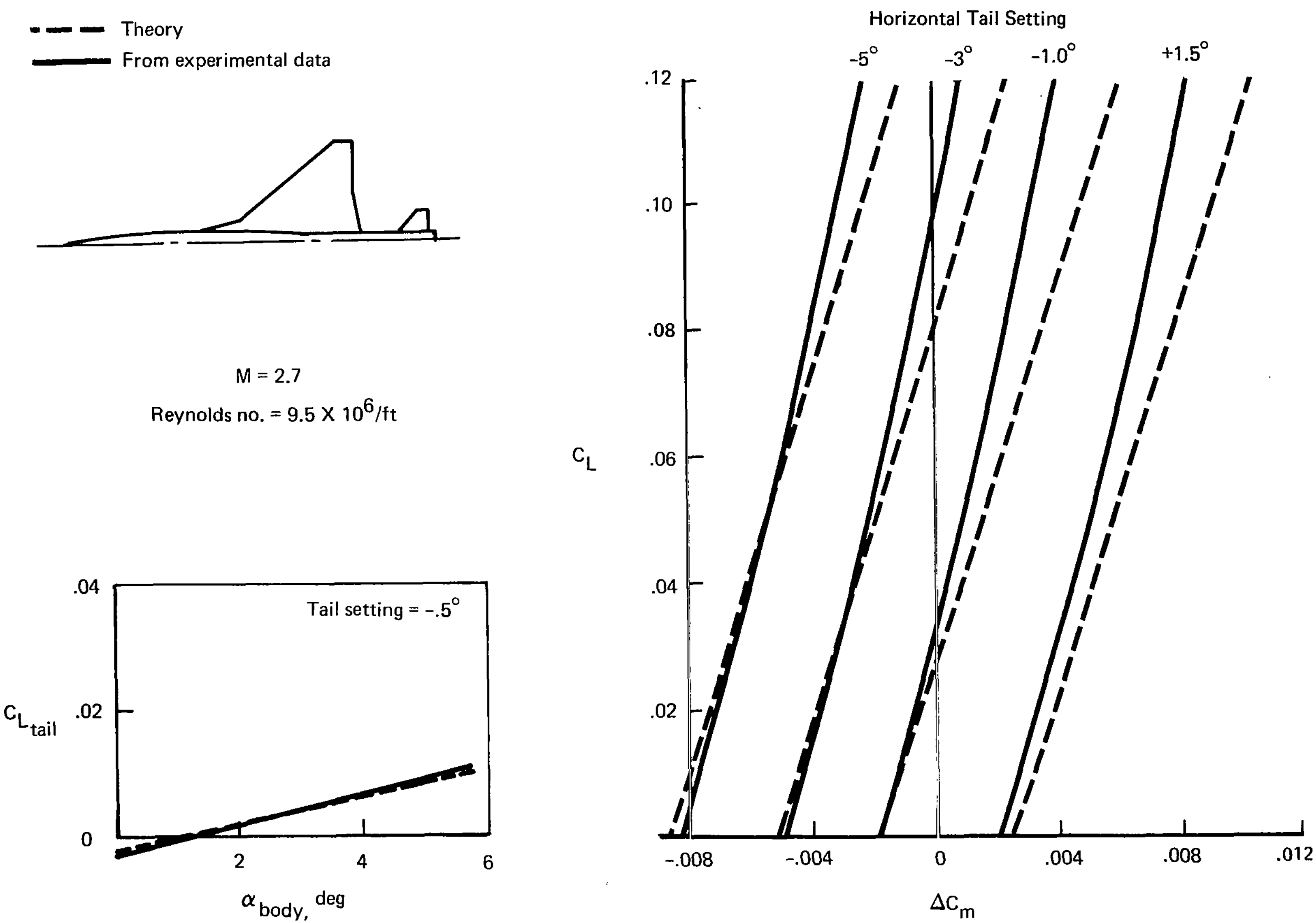


FIGURE 3.7-8.—FORCE COEFFICIENT COMPARISON



Note: Horizontal tail setting referred to wl. Angle relative to wing z = 0 plane is 1.25° larger.

FIGURE 3.7-9. —HORIZONTAL TAIL EFFECTS

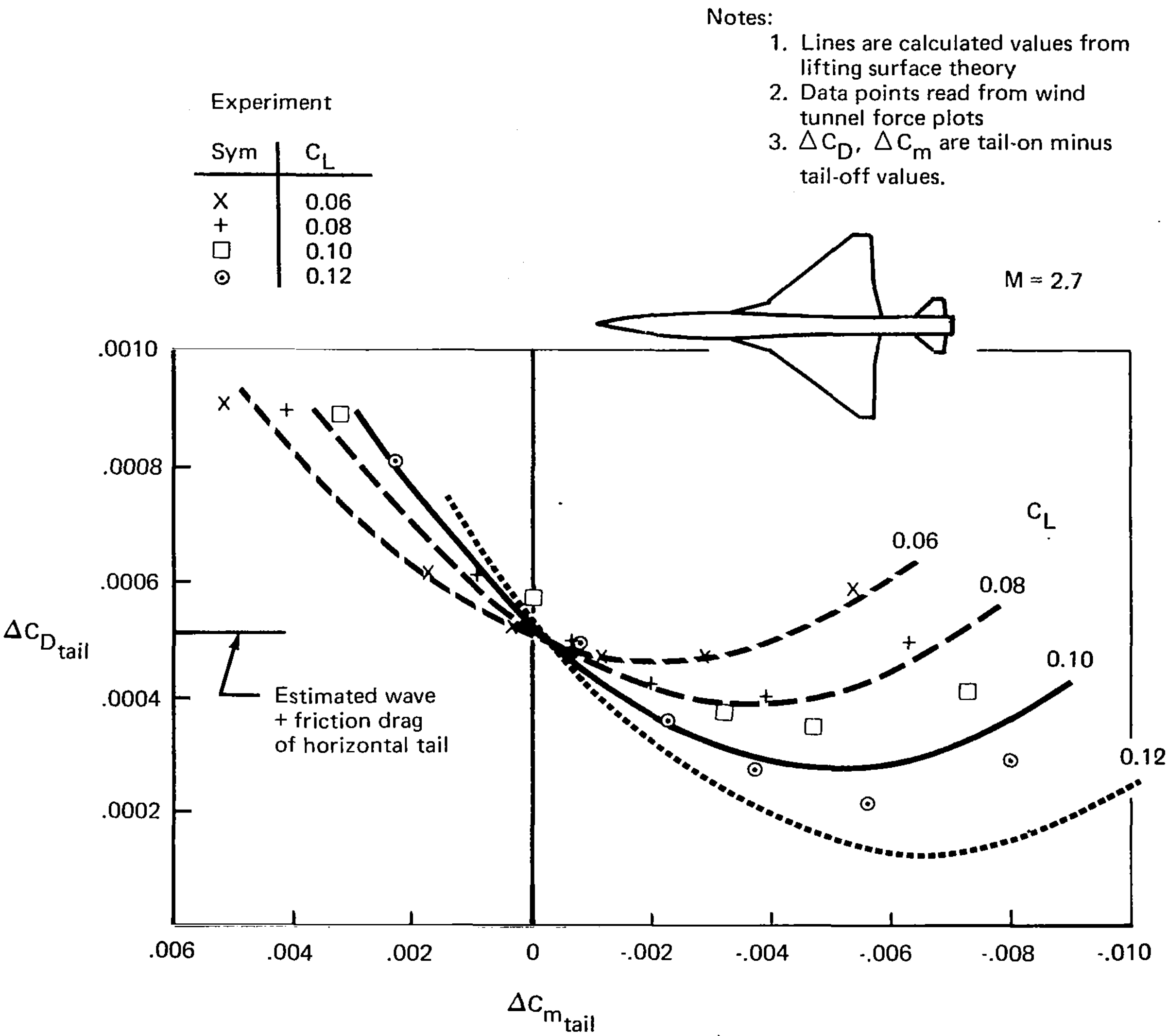


FIGURE 3.7-10. —HORIZONTAL TAIL EFFECTS

horizontal tail). The theoretical buildup of the zero-lift drag coefficient is given in figure 3.7-7.

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 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 3.7-11 and 3.7-12. 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 ≠ 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

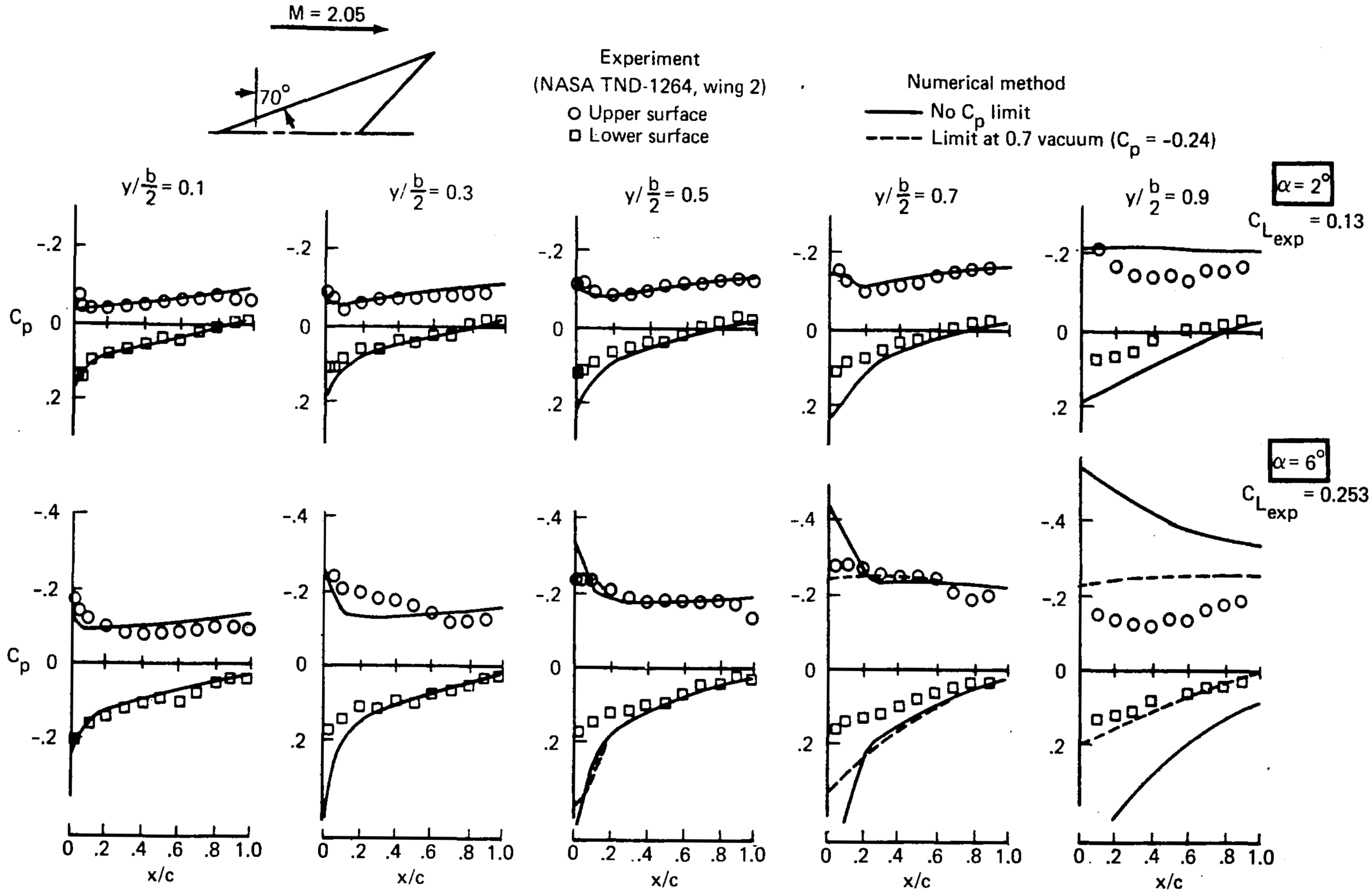


FIGURE 3.7-11.—PRESSURE COEFFICIENT COMPARISON—WING 2 (TWISTED AND CAMBERED WING, C_L = 0.08)

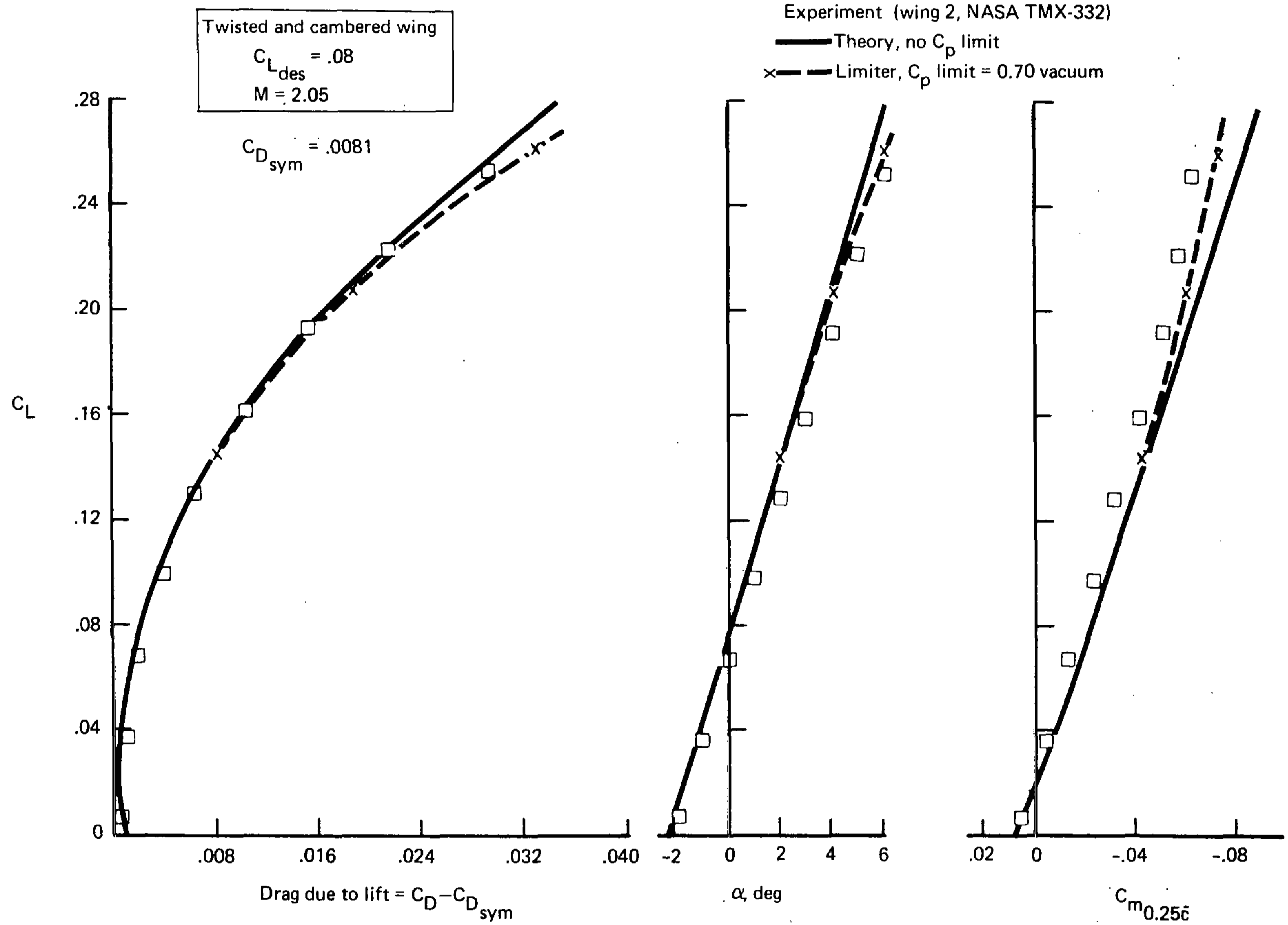


FIGURE 3.7-12. —TEST—THEORY COMPARISON, WING 2
($C_{L_{des}} = 0.08$)

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 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 ZFUS in the basic geometry).

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 allowable input number sizes are:

- No more than 5 digits to left of decimal
- No more than 4 digits to right of decimal

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 tc 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	With camber surface will be updated to the definition contained in /CAMBER/. The /CAMBER/ definition is created each time the wing design program executes and 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 used for the geometry input, 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 ≤ 20.
3	25-27	NO	NWAFOR	Number of ordinates defining each airfoil section. $3 \leq$ NWAFOR ≤ 20.
3	28-30	NO	NFUS	Number of fuselage segments. $0 \leq$ NFUS ≤ 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) ≤ 30.
3	34-36	NO	NFORX(1)	Number of stations for first fuselage segment. $4 \leq$ NFORX(1) ≤ 20.
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 JO (Card 3) is 0 or 2.

Wing Description

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 ≤20 defining stations. First segment must begin at x=0.

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. Make the first x value of the succeeding segment slightly larger than the last x value of the preceding segment.

Card Number	Card Column	Decimal Required	Variable Name	Description
10	1-70	YES	XFUS	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
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. Three different types of card 4 inputs are possible, for orthographic, perspective, or stacked three-view options.

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.
		For Orthographic Projections		
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
				For Perspective Views (See fig. 4.0-1)
4	8-12	YES	PHI	X of eye point in data coordinate system.
4	13-17	YES	THETA	Y of eye point in data coordinate system.
4	18-22	YES	PSI	Z of eye point in data coordinate system.
4	23-27	YES	XF*	X of focal point in data coordinate system.
4	28-32	YES	YF*	Y of focal point in data coordinate system.
4	33-37	YES	ZF*	Z of focal point in data coordinate system.
4	38-42	YES	DIST	Distance from eye point to view plane in inches.
4	43-47	YES	FMAG	View plane magnification factor. Controls size of projected image.
4	48-52	YES	PLOTSZ	Diameter of view plane in inches. DIST and PLOTSZ determine a cone which is the field of vision.
4	53-55		TYPE	The letters PER

* Inside configuration

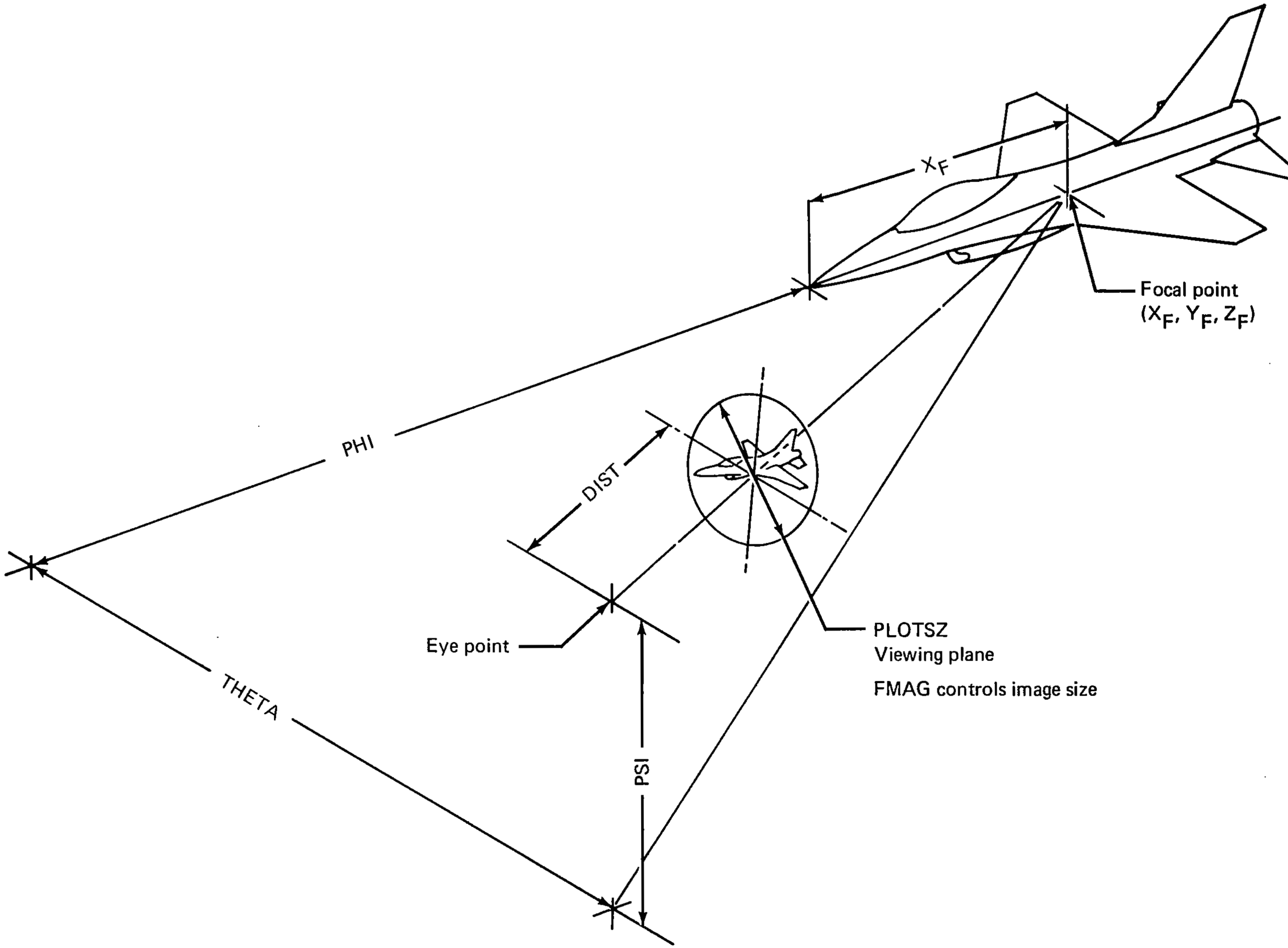


FIGURE 4.0-1.—PERSPECTIVE VIEW INPUTS

For Plan, Front and Side Views (Stacked)

Card Number	Card Column	Decimal Required	Variable Name	Description
4	8-12	YES	PHI	Ordinate of plan view centerline on paper (in inches).
4	13-17	YES	THETA	Ordinate of side view centerline on paper (in inches).
4	18-22	YES	PSI	Ordinate of front view centerline on paper (in Inches).
4	48-52	YES	PLOTSZ	Length (in inches) of maximum plot dimension.
4	53-55		TYPE	The letters VU3

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 card 3 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 Columns	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	1-7	YES	AKI	Mach number-altitude combination code. -AKI = Combination same as preceding case. 0 = Use Mach number-Reynolds combinations. AKI = Number of Mach-altitude combinations. AKI $\leqq$ 20.
4	8-14	YES	AK4	Mach number-Reynolds combination code. -AK4 = Combinations same as preceding case. 0 = Use Mach number-altitude combinations. AK4 = Number of Mach-Reynolds combinations. AK4 $\leqq$ 20.
4	15-21	YES	AXTPT	Miscellaneous components code. -AXTPT = Same components as preceding case. 0 = No miscellaneous components defined. AXTPT = Number of miscellaneous components. NXTPT $\leqq$ 10.
4	22-28	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 AKI of card(s) 5. Omit card set 5 if AKI 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 AK4 of card(s) 6. Omit card set 6 if AK4 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 card (s) 9. 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. Do not input restraint stations at nose or aft end of fuselage (those are automatically assumed). 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 $\leq$ 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. NTHETA $\leq$ 36., and a multiple of 4. If the area distribution at -90° only is desired, input NTHETA = 1.
4	29-35	YES	NREST	Number of X stations for fuselage restraint points ($\leq$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 $\neq$ 0.

6	1-3			END

4.6 Near-Field Wave Drag Program

Codes on card 3 control inclusion of basic geometry as requested.

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.
3	15-21	YES	AJ4	Fin input code 0. = Ignore fin definitions. 1. = Include fin definitions.
3	22-28	YES	AJ5	Canard input code 0. = Ignore canard definitions. 1. = Include canard definitions.

Card Number	Card Column	Decimal Required	Variable Name	Description
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.
4	29-35	YES	TNON	Number of semi-span element rows in wing calculations. TNON $\leq$ 40. If blank, TNON set to 40.
4	36-42	YES	TJBYMX	Number of spanwise stations at which wing thickness pressures are calculated. TJBYMX $\leq$ 24. Leave blank if TNON not specified.
4	43-49	YES	TNCUT	Number of body stations at which pressure coefficients are calculated ($\leq$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.

Card Number	Card Column	Decimal Required	Variable Name	Description
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 $\leq$ 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.

Card Number	Card Column	Decimal Required	Variable Name	Description
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.

Card Number	Card Column	Decimal Required	Variable Name	Description
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 73. (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 blank, 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 $(1.05\,c\text{-}x)^{0.5}$
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 the specified incidence. If the upwash fields corresponding to a series of fuselage angles of attack are desired, it will be necessary to rerun the lift analysis program to produce each upwash pressure loading. Notice that the fuselage angle of attack in both the lift analysis and wing design programs can be different from the incidence in the basic geometry. In the wing design program, angle of attack is input on card 3. In the lift analysis program, angle of attack is input on card 24 as a special case of pressure limiting option (requires FLIMIT=1.0 on card 4, appropriate value of VACFR on card 23).

The number of elements in the wing grid system is controlled by input TNØN (normally set at 40.). Depending upon the Mach number and planform, some program dimension limits may be exceeded with TNØN = 40. If this occurs, the program solves for the allowable TNØN, prints it, and stops the design case. It is then necessary for the user to reduce TNØN and the associated camber line variables TJBYMX and TJBYS.

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.

If the fuselage is included in the solution, it will be necessary to also use Z constraints at the side of fuselage station to obtain a directly usable camber surface shape and a valid drag integration.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			WDEZ
2	1-70			Any desired TITLE information
3	1-7	YES	TNON	Numbers of semispan elements in wing grid system. 2. ≤ TNON ≤ 50. If blank, TNON set to 40.
3	8-14	YES	TJBYMX	Number of semispan stations at which camber surface is calculated. 2. ≤ TJBYMX ≤ 25.
3	15-21	YES	TIFAF	Flat plate calculation code -1. = Use data from previous case. 0. = Flat plate calculation will be made. 1. = Flat plate data will be input on card 9.
3	22-28	YES	AJ1	Fuselage input code 0. = Ignore fuselage definition 1. = Include fuselage
3	29-35	YES	YSOB	Y value of wing-fuselage intersection
3	36-42	YES	ALPB	Fuselage angle of attack, deg. (relative to attitude in basic geometry).
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.

Card Number	Card Column	Decimal Required	Variable Name	Description
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 18 and 19. 3. = Data are read from tape 3 (written by previous case)

Note: See page 35 for RESTART discussion.

Card Number	Card Column	Decimal Required	Variable Name	Description
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. ≤ TLOADS ≤ 17. TLOADS < 0. = Loading numbers will be input on card(s) 10. Loading numbers will be taken from table on page 73. TLOADS > 0. = Loadings will be in the order tabulated on page 73. E.g., if TLOADS = 3.0, first 3 loadings from page 73 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) ≤ 20. 0. = Default locations of 0., 5., 10., 20., 30., ... 90., 100. are 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 ≤ 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 ≤ 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 ≤ 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.
7	29-35	YES	ANZ	Number of Z constraint locations (≤ 5).

Card Number	Card Column	Decimal Required	Variable Name	Description
7A	1-35	YES	XZCØN	X location of Z constraint
7B	1-35	YES	YZCØN	Y location of Z constraint
7C	1-35	YES	ZCØN	Z values at XZCØN and YZCØN.

Note: There will be ANZ values on cards 7A, 7B, and 7C. Z values are with respect to local wing leading edge, consistent with units of basic geometry. Omit cards 7A, 7B, and 7C if ANZ = 0.

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 23 values are used: 0., 1., 2., 4., 5., 6., 8., 10., 12., 14., 16., ..., 38., 40. Omit cards(s) 8 if TJBYMX = 0.

Card Number	Card Column	Decimal Required	Variable Name	Description
9	1-7	YES	XF	Wing aerodynamic center fraction.
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	KF	Flat wing lift-dependent drag factor.
9	22-28	YES	AREA9	Planform area in program units.
9	29-35	YES	FACTOR	Gross wing area/reference area.

Note: Omit this card if TIFAF (card 4) $\leq$ 0. The data on card 9 would normally be calculated by a previous run of the same planform at the same Mach number.

Card Number	Card Column	Decimal Required	Variable Name	Description
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.

Card Number	Card Column	Decimal Required	Variable Name	Description
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 = 0.
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.		
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 ≠ 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.		
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.		
17	1-70	YES	CPGRAD	Array of limiting upper surface pressure gradients (dCp/dx). Input at same X and Y stations as CPLIMIT, in same order.
*18	1-80		TITLE	Title card of RESTART data.
*19	1-80		RESTRT	Array of force and moment coefficients for component and interference loading, as punched from a previous run, for restarting program execution.
	Note:	Omit cards 18-19 if RESTART (card 4) is not equal to 2.0.		
20	1-3			END

*The restart card sets 18-19 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 gecmetry 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. Both are assumed to be mounted on the fuselage.

The fuselage definition (in basic geometry) cannot have discontinuities in z (meanline shape) between segments, when used in the lift analysis program.

The downwash shift options (for canard and wing downwash) are controlled by inputs on card 6. If the shift codes are left blank, the downwash will be shifted according to the respective side-of-fuselage stations of canard, wing and tail.

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.
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 $\leq$ 25.
4	8-14	YES	TNOPCT	Number of percent chords defining each spanwise station. TNOPCT $\leq$ 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.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	22-28	YES	TNOM	Number of Mach numbers in addition to basic Mach number XM. TNOM $\leq$ 5.
4	29-35	YES	FNON	Number of semi-span rows in wing grid system. FNON $\leq$ 40. If left blank, will be set to 40.
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 $\leq$ 5.
5	8-14	YES	TNTWST	Number of values (Y in percent, and angle) to define wing twist. Relative to input wing shape. TNTWST $\leq$ 40.
5	15-21	YES	TNALP	Number of canard angles of attack ($\leq$ 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 $\leq$ 40.

Card Number	Card Column	Decimal Required	Variable Name	Description
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.
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).
6	36-42	YES	FWSH	Wing downwash shift code 0. = Shift according to DYWH 1. = No shift
6	43-49	YES	DYWH	Shift of wing downwash at tail 0. = Use increment between side-of-fuselage stations of wing and tail DYWH = Y shift increment (+ = outboard)

Card Number	Card Column	Decimal Required	Variable Name	Description
6	50-56	YES	FCSH	Canard downwash shift code 0. = Shift according to DYCW and DYCH 1. = No shift
6	57-63	YES	DYCW	Shift of canard downwash at wing 0. = Use increment between side-of-fuselage stations of canard and wing DYCW = Y shift increment (+ = outboard)
6	64-70	YES	DYCH	Shift of canard downwash at horizontal tail 0. = Use increment between inboard Y stations of canard and tail DYCH = Y shift increment (+ = outboard)
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) $\leq$ 5.)
7	22-28	YES	CLIN(2)	Same as CLIN(1) for second Mach number (TMACH(1)).
7	29-35	YES	CLIN(3)	Same as CLIN(1) but for third Mach number (TMACH(2)).
7	36-42	YES	CLIN(4)	Same as CLIN(1) but for fourth Mach number (TMACH(3)).
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.

Card Number	Card Column	Decimal Required	Variable Name	Description

Wing Camber Surface Definition

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

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.

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.

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.

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.

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.

Card Number	Card Column	Decimal Required	Variable Name	Description

Wing-Fuselage Intersection

Omit cards 14-16 if ANYBOD ≠ -10.

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 ≠ 2.0

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.

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.

Card Number	Card Column	Decimal Required	Variable Name	Description

Wing Flap Definition

Omit cards 20 if TNFLAP = 0.

20	1-7	YES	XI	Inboard X value of flap leading edge.
20	8-14	YES	YI	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.

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.

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.

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.

Card Number	Card Column	Decimal Required	Variable Name	Description
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} = .010, in presence of fuselage and nacelles with pressure constraints.

- 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 99.

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 109). Only the wing geometry is required for this calculation; output for the complete configuration from the near-field program is illustrated later (page 234).

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 TIFZC = 1.0 option. The ANLZ interface program inserts this definition into the basic geometry and prints it (page 110). The lift analysis program then computes the wing upwash field (page 115), the nacelle pressure field (page 117), the asymmetric fuselage buoyancy pressure field (page 116), and the loading on the wing due to the fuselage upwash field (page 121). 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 +2.0, to illustrate output format, and then edited. The design case shown uses all seventeen loadings; first to generate a RESTART deck, then to obtain a wing design for a specific design point using the RESTART option. The fuselage is included in the solution.

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.

Five Z ordinate constraint locations are specified in the input. These are checked to see if the Y and X values of the constraints are on a computed Y station and on the planform, respectively. (The Y values were shifted slightly and a note printed).

The program next computes and prints the flat wing solution (page 133). This includes lift and drag coefficients, the lengthwise center of pressure position (as a fraction of overall wing length), and the drag-due-to-lift factor. Since the fuselage is used in the solution, the aerodynamic center location of the wing-fuselage combination (computed in the lift analysis program at the time of the fuselage upwash calculation) is substituted for the aerodynamic center of the wing planform only (page 133).

The program then calculates the carry-over lift distribution of all the camber-type loadings (page 134), and the associated force coefficients.

The program next 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 then printed, together with the lifting pressure distribution and upper and lower wing surface pressure distributions. The camber surface inboard of the side-of-fuselage station is set to zero, since it is replaced by the fuselage shape. The individual camber surface data are not shown, but have the same format as the final solution (page 182).

The program next summarizes the force and interference drag coefficients of all the component loadings (page 156). The order of the data are:

1) Lift coefficients for all loadings and their respective C_{mo} constributions (for the exposed wing part).
2) Interference drag coefficients for all wing loadings (page 156).
3) Drag coefficients of wing-on-nacelle (page 158).
4) Fuselage input to wing design point (lift, drag, and pitching moment), transferred from lift analysis program (page 159).
5) Lift, drag, and C_{mo} contributions of the carry-over lift distributions (page 159).

All of these data, plus the configuration dependent pressure distributions, are then punched into a RESTART deck, and the deck image printed (page 159). Only a portion of the RESTART listing is shown since it is quite long. (The size of the RESTART deck is a function of the number of loadings, whether fuselage is used, number of constraints, etc).

With all component loading data defined, the program then solves for the wing designs requested on card 7. (If the design case is run from a RESTART deck, the program solution commences at this point.) The solution conditions are summarized (i.e., C_L, C_{mo}, Z constraints, etc), followed by the optimized values of C_{mo}, K_E (drag-due-to-lift factor, C_D/C_L2), and the associated loading combination factors A_iC_{Li} The respective contributions of exposed wing, fuselage, carry-over lift, and nacelles to the configuration is then printed.

The solution pressure distribution is next printed and scanned for pressure constraint violations. If any occur (either in level or gradient), they are noted in the right hand margin. At the conclusion of the pressure distribution print-out, the locations and magnitude of the largest solution pressure level and gradient are noted (page 171). If violations of input pressure limits occur, the solution repeats with a constraint added at the location of worst violation.

For the test case shown, a wing design was obtained in a subsequent run using RESTART. Since all of the basic solution data were preserved in the RESTART deck, it was not necessary to recalculate the configuration-dependent data. The RESTART deck is valid for any case having:

1) The same or fewer loadings (order can be changed)
2) Same fuselage geometry, angle of attack, and side of fuselage station
3) The same or fewer Z constraint locations (in same order). The value of Z at these locations can be changed, however.
4) Any C_L, C_{mo}, or pressure constraint.

In the particular test case shown, the value of C_{mo} was changed from the case which generated the RESTART data, and only four (of the five available) Z ordinate locations were used. Solution pressure distributions were requested for all four camber surface options (C_L only, C_L plus pressure constraint, C_L plus C_{mo}, C_L plus C_{mo} + pressure constraint). The resulting camber surface for $C_L + C_{mo} + C_p$ was requested to be output and also punched into cards.

The solution commences for the C_L (and Z) case. It then continues by applying the pressure and C_{mo} constraints.

In order to illustrate program output, the solution involving all constraints has been edited and is shown (beginning on page 160). The initial solution has a number of pressure violations, the worst of which is identified (page 171), and a constraint applied there. The solution then recycles, and identifies a second constraint to be applied.

Subsequent solution cycles build up to four gradient constraints, one of which is found redundant (i.e., made unnecessary by a later constraint), as shown on page 177. That constraint is removed, together with the last constraint applied (since it involved a redundant constraint), as shown on page 178. The solution continues until the gradient constraint is everywhere satisfied, and then checks pressure level. In this case, level was already satisfied, so the final solution summary is printed, including a summary of the twelve largest pressure gradients on the wing upper surface for the final solution (page 181).

After the final solution is obtained, the program calculates any requested camber surfaces. The spanwise drag summary, force coefficient summary, Z values are printed as was done for each of the component loadings earlier. The camber surface is then interpolated at the requested percent chord values, printed and punched into cards (page 194).

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 196.

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 camber surface definition punched by the wing design program was input into the lift analysis program. The wing camberline definition at .075 semi-span (side-of-fuselage station in wing design program) was substituted for the zeros punched by the wing design program in the fuselage region, in order to allow calculation of the wing-fuselage intersection.

The lift analysis program output consists of the input, the wing-fuselage intersection definition, fuselage upwash definition (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 209).

The force coefficient summary, tail-off, is shown on page 215. 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 wing lifting pressure distribution at a specified C_L is next printed (in this case, at a $C_L = .10$). These data (page 222) are for the pressures generated by the lifting surface, but do not include pressures due to the nacelles or asymmetric fuselage buoyancy field.

The spanwise lift distribution is printed last (page 224). 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/9), and drag coefficient, for each input flight condition (page 227).

Far-Field Wave Drag

The far-field wave drag program prints an enriched area distribution for the fuselage (page 228), 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 233).

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.) Thickness pressure distributions for the empennage surfaces are then printed (page 235).

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 244), 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 249) consists of wing section data, tabulated fuselage and nacelle drag coefficients, empennage drag, 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.

```
GEOM NEW                                                                   1
969-500A  SYSTEM CHECK CASE   12/75                                        2
  1 -1 -1  1  1  1  0  8 13  1  9 19  0  0  0  0  0  0  2  7  2  4  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
77.328 4.9688 0.       166.07                                              6-1
83.104 6.625  0.       160.133                                             6-2
93.165 9.51   0.0      149.79                                              6-3
116.96 16.333 0.0      125.35                                              6-4
168.98 31.25  0.0      77.295                                              6-5
225.81 47.544 0.0      32.681                                              6-6
225.81 47.545 0.0      32.681                                              6-7
258.21 66.25  0.0      14.445                                              6-8
0.0     0.57    0.714   0.872   1.05    1.145   1.2     1.23    1.249   1.17     8-1-1
0.997   0.546   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   0.872   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
0.84    0.476   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
0.842   0.479   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   0.495   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   126.0   119.8   108.0   105.0   107.0    12-1
107.0   106.0   102.0   94.0    79.0    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
225.8   47.55   0.      38.75   262.5   47.55   10.     5.                 19-1
0.      32.5    67.5    100.                                               20-1
0.      1.5     1.5     0.                                                 21-1
270.    0.      -13.    24.2    282.5   0.      -.9     9.2                19-2
0.      32.5    67.5    100.                                               20-2
0.      1.5     1.5     0.                                                 21-2
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
4.969           -1.                                                              5
END                                                                              8
ANLZ                                                                             1
CONFIGURATION-DEPENDENT LOADING GENERATION                                       2
1.      1.                                                                       3
12.     12.     1.                                                               4
                        1.              -1.                                      5
-1.             1.              1.                                               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  -.093   -.453 -1.478 -2.667 -3.896 -5.094 -6.213 -7.218 -8.082          11-2-1
 -8.781 -9.297                                                                   11-2-2
 0.000  -.093   -.453 -1.478 -2.667 -3.896 -5.094 -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 -7.768                                                                   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 -6.390                                                                   11-4-2
 0.000   .232    .265   .020  -.410  -.958 -1.584 -2.258 -2.964 -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 -4.169                                                                   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   .387   .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  -.984 -1.141 -1.348 -1.556 -1.750 -1.973          11-10-
 -2.211 -2.456                                                                   11-10-
 0.000  -.336   -.655 -1.241 -1.760 -2.211 -2.557 -2.900 -3.264 -3.622          11-11-
 -3.970 -4.308                                                                   11-11-
 0.000  -.339   -.653 -1.208 -1.666 -2.015 -2.238 -2.412 -2.542 -2.627          11-12-
 -2.646 -2.597                                                                   11-12-
END                                                                              26
WDEZ                                                                             1
969-500  17 LOAD CHECK CASE   22 SPAN STA.  WITH FUSELAGE AND Z TERMS            2
40.     22.     0.      1.      4.9688 0.                                        3
2.      1.      1.                                                               4
2.7     .006    .1      -17.    0.      2.                                       5
2.      2.      1.                                                               6
1.      0.      0.      0.      5.                                               7
130.85 189.     243.39 189.     189.                                             7A
4.9688 4.9688 4.9688 6.625     8.2813                                            7B
-4.07   -10.16 -14.11 -8.32    -7.0                                              7C
0.      1.      2.      3.      4.      5.      6.      7.      8.      10.      8-1
12.     14.     16.     19.     22.     25.     28.     30.     32.     36.      8-2
38.     40.                                                                      8-3
1.      2.      3.      4.      5.      6.      7.      8.      9.      10.      10-1
11.     16.     17.     14.     13.     15.     12.                              10-2
```

```
0.      66.25                                                                                12
207.    269.8                                                                                13
0.      100.                                                                                 14
0.      100.                                                                                 15
-.137   -.137                                                                                16-1
-.137   -.137                                                                                16-2
.0025   .0025                                                                                17-1
.0025   .0025                                                                                17-2
END                                                                                          20
WDEZ                                                                                         1
969-500A   CHECK CASE   17 LOADS, 4 Z CONSTRAINTS   RESTARTED                                2
40.     22.     -1.     1.      4.9688  0.                                                   3
2.      1.      2.                                                                           4
2.7     .01     .1      -17.    0.      2.                                                   5
2.      2.      1.                                                                           6
1.      1.      1.      3.      4.                                                           7
130.85  189.    243.39  189.    189.                                                         7A
4.9688  4.9688  4.9688  6.625   8.2813                                                       7B
-4.07   -10.16  -14.11  -8.32   -7.0                                                         7C
0.      1.      2.      3.      4.      5.      6.      7.      8.      10.                  8-1
12.     14.     16.     19.     22.     25.     28.     30.     32.     36.                  8-2
38.     40.                                                                                  8-3
1.      2.      3.      4.      5.      6.      7.      8.      9.      10.                  10-1
11.     16.     17.     14.     13.     15.     12.                                          10-2
0.      66.25                                                                                12
207.    269.8                                                                                13
0.0     100.0                                                                                14
0.0     100.0                                                                                15
-0.137  -0.137                                                                               16-1
-0.137  -0.137                                                                               16-2
.0025   .0025                                                                                17-1
.0025   .0025                                                                                17-2
969-500   17 LOAD CHECK CASE   22 SPAN STA.   WITH FUSELAGE AND Z TERMS                       RESTART

                    *** RESTART DECK INPUT *** (NOT LISTED) ***

END                                                                                          20
WGUP
ANLZ                                                                                         1
969-500   17 LOAD CHECK CASE   22 SPAN STA.   WITH FUSELAGE AND Z TERMS                      2
1.      1.              1.                                                                   3
22.     12.     1.                                                                           4
                        1.              -1.                                                  5
-1.     2.      1.                                                                           6
2.7             1.                                                                           7
  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   2.500   5.000   7.500 10.000 12.500 15.000 17.500 20.000 25.000
 30.000 35.000 40.000 47.500 55.000 62.500 70.000 75.000 80.000 90.000
 95.000100.000
  0.000    .166    .182   -.624 -2.108 -3.584 -4.783 -5.668 -6.255 -6.625
 -7.132 -8.500
  0.000    .166    .182   -.624 -2.108 -3.584 -4.783 -5.668 -6.255 -6.625
 -7.132 -8.500
```

```
  0.000    .166    .182   -.624 -2.108 -3.584 -4.783 -5.668 -6.255 -6.625
 -7.132 -8.500
  0.000    .166    .182   -.624 -2.108 -3.584 -4.783 -5.668 -6.255 -6.625
 -7.132 -8.500
  0.000    .004   -.102   -.799 -1.887 -3.009 -4.001 -4.808 -5.397 -5.777
 -6.161 -6.858
  0.000   -.005   -.098   -.707 -1.644 -2.632 -3.523 -4.251 -4.779 -5.085
 -5.343 -5.742
  0.000   -.039   -.185   -.879 -1.876 -2.936 -3.920 -4.754 -5.393 -5.807
 -6.132 -6.500
  0.000   -.032   -.175   -.838 -1.799 -2.833 -3.806 -4.640 -5.284 -5.702
 -5.974 -6.282
  0.000   -.017   -.134   -.749 -1.657 -2.649 -3.592 -4.407 -5.036 -5.434
 -5.688 -5.938
  0.000    .020   -.071   -.625 -1.486 -2.447 -3.386 -4.212 -4.864 -5.280
 -5.520 -5.770
  0.000    .058   -.025   -.562 -1.402 -2.375 -3.354 -4.262 -5.034 -5.624
 -6.028 -6.391
  0.000    .085    .040   -.432 -1.214 -2.141 -3.103 -4.016 -4.826 -5.486
 -5.972 -6.409
  0.000    .137    .176   -.105  -.714 -1.479 -2.304 -3.109 -3.835 -4.424
 -4.940 -5.431
  0.000    .193    .316    .172  -.315  -.992 -1.758 -2.570 -3.301 -3.923
 -4.530 -5.190
  0.000    .328    .532    .596   .336  -.114  -.695 -1.316 -1.925 -2.525
 -3.152 -3.842
  0.000    .403    .719   1.087  1.156  1.020   .753    .410  -.024  -.378
  -.817 -1.322
  0.000    .456    .862   1.533  1.972  2.249  2.394  2.474  2.584  2.600
  2.523  2.322
  0.000   -.047   -.066    .012   .156   .242   .320    .387   .448   .533
   .582   .492
  0.000   -.017   -.032   -.064  -.075  -.055   .022    .138   .287   .465
   .613   .706
  0.000   -.131   -.242   -.401  -.467  -.465  -.404   -.266  -.013   .263
   .549   .848
  0.000    .103    .156    .026  -.320  -.611  -.855 -1.032 -1.118 -1.157
 -1.173 -1.167
  0.000   -.339   -.660 -1.245 -1.756 -2.187 -2.526 -2.821 -3.071 -3.277
 -3.434 -3.542
0.       2.                                                                  22
.1                                                                           25
END
SKFR                                                                         1
-500A                                                                        2
  1  1  1 -1  1                                                              3
2.                                                                           4
2.7     60.      0.       1.                                                 5-1
1.1     35.      0.       1.                                                 5-2
END                                                                          10
FFWD                                                                         1
FAR-FIELD WAVE DRAG  OPTIMIZATION BASED ON MAX. AREA                         2
1.       1.       1.       1.       1.                                       3
   1    2.7      50.      36.                                                4
```

```
END                                                    6
NFWD                                                   1
NEAR-FIELD WAVE DRAG                                   2
1.      1.      1.      1.                             3
0.      2.7                                            4
4.97    -1.     -1.                                    5
END                                                    8
PLOT                                                   1
969-500 CONFIGURATION                                  2
1.      1.      1.      1.      1.                     3
X Z                                  10.  ORT          4-1
X Y                                  10.  ORT          4-2
Y Z                                  10.  ORT          4-3
END                                                    5
```

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

REFA = 9898.0000 CBAR = 106.4100 XBARIN = 187.0000

	XO = 77.3280			XO = 83.1040			XO = 93.1650		
	YO = 4.9688			YO = 6.6250			YO = 9.5100		
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000		
	CHORD = 166.0700			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	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 = 168.9800			XO = 225.8100		
	YO = 16.3330			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	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	= 32.6810	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	0.0000	.1340	.1340	0.0000	.1340	.1340
5.0	0.0000	.2610	.2610	0.0000	.2610	.2610
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
16.6700	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
66.6700	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
116.6700	-.1400	5.8632	108.0000	36.8398
133.3300	-1.6000	5.7812	105.0000	36.3245
150.0000	-3.0400	5.8360	107.0000	36.6688
166.6600	-4.5000	5.8360	107.0000	36.6688
183.3300	-5.9000	5.8087	106.0000	36.4971
200.0000	-7.4000	5.6980	102.0000	35.8018
216.6700	-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	-14.6000	1.5958	8.0000	10.0265
295.0000	-15.7000	0.0000	0.0000	0.0000

3

**** **** **** **** **** 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

**** FIN ***** **** ****

XL =	225.8000	XL =	270.0000
YL =	47.5500	YL =	0.0000
ZL =	0.0000	ZL =	-13.0000
CL =	38.7500	CL =	24.2000
XU =	262.5000	XU =	282.5000
YU =	47.5500	YU =	0.0000
ZU =	10.0000	ZU =	-.9000
CU =	5.0000	CU =	9.2000

PERCENT CHORD	HALF THICK	PERCENT CHORD	HALF THICK
0.00	0.00	0.00	0.00
32.50	1.50	32.50	1.50
67.50	1.50	67.50	1.50
100.00	0.00	100.00	0.00

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

XI =	261.0000
YI =	2.0000
ZI =	-14.0000
CI =	25.0000
XO =	277.0000
YO =	11.0000
ZO =	-14.0000
CO =	9.0000

PERCENT CHORD	UPPER ORD	LOWER ORD
0.000	0.000	0.000
50.000	1.500	1.500
100.000	0.000	0.000

4

WING THICKNESS PRESSURE GENERATION

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

PLANFORM BREAKPOINTS

	X	Y	CHORD
1	77.3280	0.0000	166.0700
2	77.3280	4.9688	166.0700
3	83.1040	6.6250	160.1330
4	93.1650	9.5100	149.7900
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	32.6810
9	258.2100	66.2500	14.4450

	XLE	XTE	Y
0	77.3280	243.3980	0.0000
1	77.3280	243.3980	1.6563
2	77.3280	243.3980	3.3125
3	77.3280	243.3980	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	11.5938
8	106.2081	242.6014	13.2500
9	111.9843	242.4449	14.9063
10	117.7603	242.3710	16.5625
11	123.5362	242.8112	18.2188
12	129.3120	243.2515	19.8750
13	135.0878	243.6917	21.5313
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	28.1563
18	163.9670	245.8929	29.8125
19	169.7430	246.4390	31.4688
20	175.5196	247.6807	33.1250
21	181.2962	248.9225	34.7813
22	187.0729	250.1642	36.4375
23	192.8495	251.4059	38.0938
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	44.7188
28	221.7328	257.6146	46.3750
29	226.6523	258.8592	48.0313
30	229.5211	260.1134	49.6875
31	232.3900	261.3675	51.3438
32	235.2589	262.6217	53.0000
33	238.1278	263.8759	54.6563
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	61.2813
38	252.4722	270.1467	62.9375
39	255.3411	271.4008	64.5938
40	258.2100	272.6550	66.2500

INBOARD WING END DEFINITION

CHORD	X	Y	Z	T
0.00	77.328698	4.969000	0.000000	0.000000
2.50	81.480430	4.969000	0.000000	1.893190
5.00	85.632162	4.969000	0.000000	2.371469
10.00	93.935626	4.969000	0.000000	2.896248
20.00	110.542554	4.969000	0.000000	3.487455
30.00	127.149482	4.969000	0.000000	3.802987
40.00	143.756411	4.969000	0.000000	3.985663

50.00	160.363339	4.969000	0.000000	4.085304
60.00	176.970267	4.969000	0.000000	4.148411
70.00	193.577196	4.969000	0.000000	3.886021
80.00	210.184124	4.969000	0.000000	3.112138
90.00	226.791052	4.969000	0.000000	1.813477
100.00	243.397981	4.969000	0.000000	0.000000

TABLE OF INPUT Z/C ORDINATES

XPCT / Y/B/2	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
0.0000	0.000000	.570000	.714000	.872000	1.050000	1.145000	1.200000	1.230000	1.249000	1.170000	.937000	.546000	0.000000
.0750	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	1.156000	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	.902000	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

Y/B/2 / 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
0.000	0.000000	.007181	.015607	.020424	.012817	.007863	.005973	.003528	.002811	.005289	.003303	.000610	.000151	-.001318	-.003688	-.004128	-.007772	-.013488	-.017615	-.021555	-.026499
.025	.003049	.007607	.013422	.014378	.009965	.007961	.008164	.005860	.003421	.002533	.001086	.000594	.001558	.000311	-.002880	-.006007	-.010054	-.014312	-.017090	-.020651	-.025437
.050	.010939	.011780	.015375	.013403	.012492	.008389	.004969	.003865	.004275	.002871	.002566	.001463	-.000675	-.002347	-.003482	-.006735	-.010090	-.014023	-.018806	-.023430	-.026351
.075	.035284	.010912	.005615	.005186	.009271	.008633	.004337	.004304	.003884	.001148	.001523	.001518	-.001747	-.003713	-.005689	-.010283	-.013900	-.016427	-.021427	-.025760	-.027668
.100	.063472	.007600	-.005832	.004328	.007403	.004838	.002677	.003110	.002020	.001265	.001828	-.000367	-.003840	-.006206	-.009535	-.013783	-.017184	-.020046	-.024428	-.028060	-.029891
.125	.093863	.006988	-.006320	.002556	.004160	.003531	.001621	.001257	.001579	.000937	.000031	-.001399	-.003455	-.007376	-.012355	-.015598	-.018511	-.021984	-.025640	-.029708	-.032117
.150	.133990	.005068	-.010583	-.000360	.003827	.002310	-.000515	.001040	.001054	-.000971	-.001206	-.000325	-.004685	-.010334	-.013589	-.014761	-.019423	-.023994	-.026222	-.029065	-.032547
.200	.050564	-.005361	-.009168	.000630	-.001391	-.000912	.000929	-.000706	-.002469	-.001061	-.001079	-.004266	-.007965	-.011250	-.015679	-.019393	-.021710	-.025177	-.028616	-.031299	-.033278
.250	.040388	-.005435	-.012106	-.004102	-.005151	-.004461	-.002725	-.000706	-.001555	-.003227	-.003446	-.004986	-.008411	-.013225	-.017520	-.020189	-.023190	-.026821	-.030448	-.033071	-.034131
.300	.027466	-.006828	-.011185	-.009507	-.006045	-.005855	-.004519	-.002755	-.003994	-.001841	-.004837	-.007420	-.011106	-.014519	-.017333	-.022887	-.026213	-.029833	-.031124	-.034304	-.037308
.350	.049029	.003490	-.008104	-.014617	-.010100	-.008771	-.003495	-.003808	-.004304	-.005878	-.006262	-.008352	-.011633	-.016311	-.021072	-.023907	-.027272	-.030746	-.034898	-.036204	-.038300
.400	.040513	.001386	-.010265	-.012448	-.011351	-.010549	-.007385	-.004560	-.005188	-.005760	-.008216	-.011057	-.014561	-.017424	-.021129	-.025144	-.029719	-.033393	-.036282	-.038077	-.038769
.450	.032322	-.002308	-.013220	-.015858	-.013421	-.007985	-.008625	-.006773	-.007687	-.009129	-.008486	-.012453	-.014931	-.020065	-.023851	-.026846	-.030958	-.033339	-.037421	-.040173	-.042339
.500	.018075	-.002695	-.013526	-.017029	-.017991	-.011116	-.007332	-.007855	-.007451	-.010320	-.012964	-.014798	-.018562	-.021491	-.024274	-.028923	-.032109	-.036115	-.039599	-.041182	-.042446
.600	.020833	-.001276	-.010419	-.015099	-.014387	-.014226	-.013563	-.011754	-.012966	-.015005	-.017232	-.019086	-.021376	-.026207	-.029886	-.033134	-.037964	-.041284	-.043432	-.044431	-.045429
.700	.001309	-.004798	-.010906	-.014763	-.015342	-.015964	-.017024	-.018084	-.018067	-.017740	-.019248	-.022904	-.026569	-.030291	-.034013	-.038357	-.042826	-.046341	-.048955	-.051593	-.054323
.800	.041524	.034978	.028432	.021860	.015246	.008632	.002186	-.003937	-.010060	-.015977	-.021424	-.026872	-.031841	-.035512	-.039184	-.042754	-.045989	-.049224	-.052262	-.054495	-.056729
.900	.045388	.041996	.038604	.035212	.031819	.026961	.021219	.015477	.009735	.003178	-.003818	-.010815	-.017812	-.024015	-.029836	-.035658	-.041480	-.045631	-.049070	-.052509	-.055948

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.950	.046229	.043122	.040014	.036906	.033160	.029018	.024876	.020734	.016592	.010999	.005275	-.000450
	-.006175	-.011970	-.018024	-.024077	-.030131	-.036184	-.042748	-.049790	-.056833			
1.000	.034932	.032537	.030143	.027748	.025354	.022959	.020565	.018170	.015466	.012046	.008627	.005208
	.001788	-.001631	-.005038	-.008434	-.011831	-.015227	-.018624	-.022020	-.025417			

UPDATED WING DEFINITION

WING CAMBER SURFACE READ INTO BASIC GEOMETRY

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

REFA = 9898.0000 CBAR = 106.4100 XBARIN = 187.0000

	XO = 77.3280			XO = 83.1040			XO = 93.1650	
	YO = 4.9688			YO = 6.6250			YO = 9.5100	
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000	
	CHORD = 166.0700			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	-.0122	.5700	.5700	.0528	.5700	.5700	.0554	.5500	.5500
5.0	-.0244	.7140	.7140	.1057	.7140	.7140	.1107	.7120	.7120
10.0	-.4914	.8720	.8720	-.2306	.8720	.8720	-.1337	.8720	.8720
20.0	-1.9134	1.0500	1.0500	-1.3723	1.0500	1.0500	-1.0272	1.0540	1.0540
30.0	-3.6133	1.1450	1.1450	-2.7975	1.1450	1.1450	-2.1970	1.1560	1.1560
40.0	-5.4088	1.2000	1.2000	-4.3476	1.2000	1.2000	-3.4961	1.2130	1.2130
50.0	-7.1922	1.2300	1.2300	-5.9249	1.2300	1.2300	-4.8463	1.2350	1.2350
60.0	-8.8916	1.2490	1.2490	-7.4654	1.2490	1.2490	-6.1905	1.2370	1.2370
70.0	-10.4547	1.1700	1.1700	-8.9226	1.1700	1.1700	-7.4901	1.1270	1.1270
80.0	-11.8399	.9370	.9370	-10.2581	.9370	.9370	-8.7121	.8830	.8830
90.0	-13.0104	.5460	.5460	-11.4383	.5460	.5460	-9.8285	.5070	.5070
100.0	-13.9393	0.0000	0.0000	-12.4391	0.0000	0.0000	-10.8176	0.0000	0.0000

	XO = 116.9600			XO = 168.9800			XO = 225.8100	
	YO = 16.3330			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	.3681	0.0000	0.0000

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

XO = 225.8100		XO = 258.2100	
YO = 47.5450		YO = 66.2500	
ZO = 0.0000		ZO = 0.0000	
CHORD = 32.6810		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	.1308	.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	.4283	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	.3966	.4950	.4950	-.3822	.4950	.4950
100.0	.3679	0.0000	0.0000	-.3751	0.0000	0.0000

PROGRAM CONTROL CARD
ANLZ

CONFIGURATION DEPENDENT LOADING GENERATION

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

TIFZC= 1.00 TNOM= 0.00 SYMM= 1.00 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	77.3280	0.0000	0.0000	166.0700	166.0700
2	77.3280	4.9688	0.0000	166.0700	166.0700
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	77.3280	243.3980	243.3980
1	77.3280	243.3980	243.3980
2	77.3280	243.3980	243.3980
3	77.3280	243.3980	243.3980
4	83.1040	243.2370	243.2370
5	88.8799	243.0751	243.0751
6	94.6559	242.9146	242.9146
7	100.4320	242.7580	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	247.6807	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

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	6.10264	117.00000	4.17000
83.33000	6.33301	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	5.80869	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
283.30000	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	16.33000	-5.80000	0.00000	2.86500	25.78696
			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	25.78696
			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

FUSELAGE AREAS ABOVE AND BELOW WING

PER CENT CHORD	X	AREA ABOVE	AREA BELOW
0.00	79.01	100.08	25.03
5.00	88.63	89.74	34.93
10.00	96.87	83.91	37.50
20.00	112.35	77.52	32.66
30.00	128.02	78.63	26.78
40.00	144.01	84.62	21.89
50.00	160.36	91.27	15.85
60.00	176.97	103.09	3.65
70.00	193.58	102.02	2.02
80.00	210.18	96.69	1.33
90.00	226.79	84.77	.84
100.00	243.40	67.21	.79

TABLE OF INPUT Z/C ORDINATES

Y/B/2 \ 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
0.0000	0.00000	-.09300	-.45300	-1.47800	-2.66700	-3.89600	-5.09400	-6.21300	-7.21800	-8.08200	-8.78100	-9.29700
.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	.06600	-.14400	-.85700	-1.74700	-2.71500	-3.70000	-4.66200	-5.57200	-6.40600	-7.14300	-7.76800
.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	-1.58400	-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	-4.16900
.5000	0.00000	.28100	.49300	.56100	.41000	.14900	-.21100	-.64700	-1.13700	-1.66900	-2.23900	-2.84200
.6000	0.00000	.07400	.43600	.68800	.71700	.59400	.38700	.08200	-.26500	-.66900	-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	-1.97300	-2.21100	-2.45600
.9000	0.00000	-.33600	-.65500	-1.24100	-1.76000	-2.21100	-2.55700	-2.90000	-3.26400	-3.62200	-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	79.0096	5.4510	0.0000
5.00	88.6320	5.8758	.0469
10.00	96.8711	5.9071	-.3437
20.00	112.3501	5.6214	-1.7002
30.00	128.0228	5.3311	-3.4348
40.00	144.0122	5.0936	-5.3289
50.00	160.3630	4.7128	-7.3881
60.00	176.9700	3.0533	-10.3179
70.00	193.5770	2.5045	-11.9869
80.00	210.1840	2.1222	-13.4218
90.00	226.7910	1.7648	-14.5826
100.00	243.3980	1.6051	-15.4395

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	-2.262	-.634	-.298	-1.447	-2.058	-1.944	-1.834	-1.596	-.931	.070	.915
.025	-2.262	-.634	-.298	-1.447	-2.058	-1.944	-1.834	-1.596	-.931	.070	.915
.050	3.278	4.120	3.761	2.752	2.180	2.164	1.964	2.113	2.186	2.346	2.394
.075	3.355	4.123	3.980	3.525	3.356	3.564	3.543	3.882	3.839	3.680	3.433
.100	3.169	3.464	3.290	3.087	3.066	3.230	3.207	3.473	3.322	3.035	2.718
.125	2.526	2.577	2.410	2.286	2.272	2.364	2.296	2.467	2.317	2.066	1.829
.150	1.923	1.894	1.754	1.678	1.666	1.718	1.637	1.749	1.639	1.451	1.237
.175	1.463	1.408	1.300	1.254	1.244	1.276	1.194	1.264	1.194	1.057	.869
.200	1.130	1.066	.986	.960	.953	.974	.912	.939	.897	.796	.645
.250	.710	.645	.610	.603	.602	.614	.581	.560	.568	.515	.434
.300	.473	.418	.406	.408	.413	.415	.399	.373	.378	.354	.316
.350	.327	.289	.290	.294	.300	.299	.294	.277	.266	.267	.241
.400	.229	.210	.216	.220	.226	.227	.226	.214	.201	.199	.195
.450	.160	.159	.167	.171	.176	.179	.177	.172	.164	.155	.153
.500	.122	.125	.132	.135	.141	.145	.143	.141	.135	.130	.123
.550	.097	.101	.108	.110	.116	.120	.119	.117	.115	.110	.106
.600	.079	.084	.089	.091	.096	.099	.101	.099	.098	.096	.092
.700	.058	.062	.064	.065	.067	.070	.072	.074	.074	.073	.072
.800	.039	.041	.043	.046	.048	.049	.050	.051	.053	.055	.056
.900	.024	.026	.027	.028	.029	.031	.032	.034	.035	.037	.037
1.000	.025	.023	.020	.018	.016	.017	.017	.018	.018	.019	.020

LIFTING PRESSURE COEFFICIENTS DUE TO ASYMMETRIC BODY VOLUME

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	.0164	.0353	-.0127	-.0253	-.0243	-.0307	-.0141	.0177	.0223	.0281	.0158
.025	.0164	.0353	-.0127	-.0253	-.0243	-.0307	-.0141	.0177	.0223	.0281	.0158
.050	.0164	.0353	-.0127	-.0253	-.0243	-.0307	-.0141	.0177	.0223	.0281	.0158
.075	.0164	.0353	-.0127	-.0253	-.0243	-.0307	-.0141	.0177	.0223	.0281	.0158
.100	.0404	.0122	-.0176	-.0262	-.0235	-.0333	-.0075	.0178	.0220	.0274	.0175
.125	.0379	.0084	-.0162	-.0235	-.0211	-.0278	-.0147	.0146	.0179	.0241	.0267
.150	.0362	.0054	-.0153	-.0216	-.0193	-.0237	-.0211	.0124	.0163	.0216	.0261
.175	.0350	.0029	-.0146	-.0200	-.0179	-.0205	-.0231	.0112	.0148	.0195	.0217
.200	.0340	.0006	-.0140	-.0188	-.0168	-.0179	-.0249	.0101	.0134	.0158	.0192
.250	.0277	-.0030	-.0133	-.0169	-.0153	-.0147	-.0233	-.0035	.0101	.0126	.0165
.300	.0209	-.0042	-.0131	-.0156	-.0141	-.0132	-.0202	-.0143	.0085	.0109	.0115
.350	.0141	-.0049	-.0131	-.0146	-.0133	-.0124	-.0162	-.0185	.0008	.0084	.0105
.400	.0050	-.0056	-.0129	-.0138	-.0126	-.0117	-.0128	-.0180	-.0115	.0072	.0086
.450	-.0009	-.0069	-.0122	-.0131	-.0122	-.0111	-.0110	-.0157	-.0165	-.0046	.0069
.500	-.0035	-.0083	-.0116	-.0126	-.0118	-.0106	-.0103	-.0123	-.0158	-.0130	-.0011
.550	-.0047	-.0095	-.0110	-.0121	-.0114	-.0102	-.0099	-.0104	-.0140	-.0154	-.0111
.600	-.0069	-.0107	-.0105	-.0116	-.0110	-.0099	-.0096	-.0094	-.0107	-.0138	-.0146
.700	-.0100	-.0099	-.0106	-.0108	-.0104	-.0095	-.0091	-.0089	-.0088	-.0088	-.0096
.800	-.0081	-.0094	-.0094	-.0095	-.0102	-.0102	-.0099	-.0094	-.0088	-.0085	-.0083
.900	-.0038	-.0055	-.0070	-.0083	-.0089	-.0089	-.0089	-.0094	-.0099	-.0097	-.0095
1.000	-.0008	-.0022	-.0027	-.0032	-.0040	-.0051	-.0062	-.0071	-.0080	-.0085	-.0085

NACELLES BELOW WING WITH ORIGINS AT

X= 213.42000 Y= 16.33000 Z= -5.80000
X= 219.67000 Y= 31.25000 Z= -4.90000

FOR NACELLE(S) AT X= 213.42000 Y= 16.33000 Z= -5.80000

Y	D	AREA	CP	Y	S(Y)
213.420000	2.865000	25.766502	.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	3.023380	28.716525	.044722	208.465308	.089467

NACELLE PRESSURE FIELD

Y/B/2

X, PER CENT CHORD AND PRESSURE COEFFICIENT
GLANCE SOLUTION

NACELLES BELOW WING

0.000	77.328	243.398										
	0.000	100.000										
	0.00000	0.00000										
.050	77.328	238.690	238.700	238.995	239.290	239.584	239.879	240.174	240.469	240.764	241.058	241.353
	241.648	241.943	242.238	242.533	242.827	243.122	243.417	243.712				
	0.000	97.165	97.171	97.349	97.526	97.704	97.881	98.059	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	.03721	.03663	.03606	.03550	.03493	.03436	.03379
	.03322	.03266	.03209	.03153	.03097	.03041	.02985	.02929				
.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	92.790	92.796	93.247	93.698	94.149	94.600	95.051	95.502	95.953	96.404	96.855
	97.306	97.757	98.208	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	229.786	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	83.866	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	.06088	.05645	.05202	.04762	.04329	.03909	.03500	.03100	.02709	.02357
	.02379	.02120	.01450	.00784	.00131	-.00484	-.01032	-.01497				
.246	116.926	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	85.956	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	.02382
	.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
	233.516	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				
	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	218.836	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				
	0.00000	0.00000	.06527	.06017	.05504	.04996	.04497	.04018	.03550	.03092	.02645	.02389
	.02360	.01674	.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	239.509	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	96.707	96.716	98.104	99.492	100.248	100.248				
	0.00000	0.00000	.05970	.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	231.346	232.529	232.539	233.819	235.100	236.380
	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	96.214	97.454	98.694	99.934	100.512	100.512				
	0.00000	0.00000	.05092	.04754	.04417	.04083	.03753	.03455	.08402	.07760	.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	.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	239.716	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
	.01426	.01146	.04967	.03699	.02510	.01428	.00288	.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	.07180	.06511	.05841	.05180	.04541	.03926	.03328	.02766	.02812	.02118
	.01081	.00072	-.00837	-.00860	.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	246.379	246.388				
	0.000	68.583	68.596	70.841	73.086	75.331	77.576	79.820	82.065	84.310	86.555	88.800
	91.045	93.290	95.535	95.625	95.638	97.883	100.128	100.139				
	0.00000	0.00000	.07180	.06511	.05840	.05180	.04541	.03925	.03327	.02765	.02813	.02116
	.01078	.00069	-.00840	-.00874	-.00879	-.01686	-.02661	-.02665				
.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	65.526	67.991	70.455	72.919	75.384	77.848	80.312	82.776	85.241	87.705
	90.169	92.634	95.098	97.562	99.196	99.210	100.196	100.196				
	0.00000	0.00000	.06910	.06252	.05594	.04946	.04320	.03718	.03131	.02612	.02621	.01837
	.00827	-.00154	-.01012	-.01821	-.02460	-.02465	-.02852	-.02852				
.550	187.073	227.153	227.163	228.602	230.041	231.480	232.918	234.357	235.796	237.235	238.673	240.112
	241.551	242.989	244.428	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	84.067
	86.348	88.628	90.908	93.189	95.469	97.750	100.030	100.652				
	0.00000	0.00000	.05808	.05368	.04929	.04494	.04066	.03654	.03251	.02856	.02469	.02167
	.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	247.856	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	.04839	.04539	.04241	.03944	.03650	.03363	.03082	.02807	.02536	.02269
	.02007	.01823	.01912	.01619	.01175	.00738	.00306	-.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	91.876	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	.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	255.196	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	.03650	.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	257.820	258.109
	258.398	258.687	258.976	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

         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              -.000006          -.000217
     CD       .000001          -.000000               .000000          -.000004
     CM       .003958           .000795               .003882           .000889
```

TABLE OF CAMBER CP AT BASIC ALPHA

Y/B/2 \ 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
0.000	.00105	.00327	.00860	.02294	.02623	.02142	.01980	.02729	.02850	.02160	.01697	.01243
.025	.00121	.00421	.00945	.02295	.02614	.02163	.02055	.02811	.02634	.02157	.01790	.01300
.050	.00395	.00744	.01244	.02319	.02593	.02247	.02428	.03304	.03058	.02208	.01599	.01113
.075	.01128	.01288	.01662	.02464	.02682	.02528	.02827	.02896	.02825	.02093	.01320	.00905
.100	.04489	.03327	.02565	.02196	.02210	.02209	.02309	.02454	.02595	.01971	.01148	.00787
.125	.05756	.03935	.02802	.01968	.02050	.02061	.02085	.02198	.02394	.01926	.01116	.00745
.150	.06408	.04193	.02852	.01821	.01939	.01966	.01925	.02019	.02211	.01917	.01159	.00746
.175	.06010	.04133	.02738	.01753	.01852	.01880	.01822	.01874	.02061	.01913	.01247	.00772
.200	.06110	.04094	.02779	.01669	.01806	.01796	.01762	.01762	.01922	.01882	.01358	.00846
.225	.06322	.04038	.02743	.01607	.01738	.01748	.01710	.01683	.01787	.01844	.01474	.00942
.250	.05586	.03909	.02653	.01563	.01673	.01725	.01665	.01616	.01674	.01799	.01569	.01068
.275	.05674	.03907	.02683	.01471	.01655	.01687	.01637	.01553	.01596	.01725	.01626	.01196
.300	.05056	.03694	.02539	.01468	.01639	.01655	.01604	.01499	.01539	.01651	.01645	.01331
.325	.05054	.03666	.02582	.01565	.01606	.01639	.01561	.01478	.01485	.01581	.01629	.01438
.350	.05134	.03653	.02600	.01616	.01598	.01601	.01537	.01470	.01440	.01516	.01587	.01516
.375	.04557	.03486	.02530	.01662	.01558	.01567	.01534	.01456	.01413	.01452	.01526	.01543
.400	.04622	.03559	.02658	.01732	.01498	.01564	.01523	.01444	.01392	.01391	.01468	.01529

.425	.04198 .01413	.03378 .01493	.02600	.01744	.01482	.01559	.01510	.01447	.01366	.01348
.450	.04226 .01359	.03406 .01431	.02594	.01847	.01479	.01540	.01516	.01436	.01346	.01322
.475	.04320 .01312	.03486 .01368	.02729	.01935	.01484	.01535	.01505	.01418	.01343	.01312
.500	.03933 .01275	.03316 .01317	.02724	.01975	.01534	.01509	.01477	.01409	.01353	.01293
.525	.04038 .01264	.03435 .01287	.02817	.02092	.01600	.01461	.01462	.01426	.01346	.01275
.550	.03772 .01273	.03297 .01312	.02824	.02081	.01636	.01443	.01477	.01416	.01336	.01284
.575	.03861 .01302	.03372 .01317	.02887	.02166	.01740	.01475	.01448	.01403	.01354	.01313
.600	.04002 .01341	.03506 .01330	.03004	.02275	.01831	.01506	.01411	.01421	.01394	.01361
.625	.03724 .01381	.03343 .01347	.02963	.02261	.01874	.01568	.01437	.01461	.01443	.01418
.650	.03819 .01419	.03454 .01365	.03084	.02411	.02033	.01733	.01535	.01491	.01491	.01470
.675	.04017 .01441	.03598 .01389	.03218	.02591	.02193	.01914	.01681	.01534	.01526	.01492
.700	.03867 .01435	.03592 .01350	.03317	.02783	.02369	.02080	.01840	.01606	.01509	.01476
.725	.03987 .01405	.03749 .01384	.03511	.03013	.02573	.02219	.01959	.01726	.01550	.01442
.750	.03559 .01423	.03415 .01387	.03270	.02955	.02583	.02276	.02022	.01808	.01611	.01493
.775	.03422 .01494	.03262 .01425	.03117	.02874	.02612	.02353	.02110	.01915	.01733	.01564
.800	.02999 .01515	.02957 .01390	.02915	.02777	.02607	.02398	.02195	.01999	.01830	.01669
.825	.02694 .01619	.02682 .01489	.02669	.02635	.02528	.02391	.02224	.02055	.01894	.01752
.850	.02451 .01735	.02444 .01646	.02437	.02423	.02382	.02308	.02200	.02079	.01945	.01824
.875	.02183 .01787	.02201 .01616	.02218	.02244	.02264	.02239	.02195	.02110	.02018	.01907
.900	.01952 .01861	.01976 .01763	.02000	.02049	.02089	.02127	.02113	.02088	.02026	.01956

.925	.01774	.01794	.01814	.01855	.01895	.01940	.01985	.01986	.01977	.01944
	.01899	.01853									
.950	.01462	.01510	.01559	.01645	.01707	.01769	.01813	.01856	.01866	.01857
	.01820	.01706									
.975	.01300	.01331	.01362	.01424	.01482	.01532	.01582	.01611	.01633	.01646
	.01608	.01570									
1.000	.01037	.01049	.01061	.01084	.01108	.01130	.01143	.01157	.01165	.01158
	.01151	.01145									

```
PROGRAM CONTROL CARD
WREZ
ENTER INPTS---TAPE INPUTS
EXIT INPTS
ENTER GEOMETRY INTERFACE WITH TEA253A
J1=  1
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* * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * *
                         TEA253, 17 LOADING VERSION OF OCTOBER 30, 1975.

                          OPTIMUM COMBINATION OF 17 WING LOADINGS

                969-500  17 LOAD CHECK CASE  22 SPAN STA.  WITH FUSELAGE AND Z TERMS
* * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * *

           NUMBER OF PLANFORM BREAKPOINTS =    9.0                     FLAT PLATE CONTROL FLAG =   0.0
              NUMBER OF SEMISPAN ELEMENTS =   40.0                              PRINT FLAG =   2.0
 NUMBER OF SPAN STATIONS FOR CAMBER SURFACE =   22.0                          SMOOTHING FLAG =   1.0
          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 =    -17.0000
                PITCHING MOMENT CENTER AT    187.0000        NUMBER OF CAMBER ORDINATES =     12.0000
                         REFERENCE AREA =   9898.0000  NUMBER OF POINTS DEFINING ARBITRARY REGION =  2.0000
                        C-M-O CONSTRAINT =      .0060                    FUSELAGE ALPHA =      0.0000
            SPAN STATION FOR SIDE-OF-BODY =     4.9688      NUMBER OF BODY CAMBER ORDINATES =     19.0000

                   NUMBER OF CHORDWISE AND SPANWISE LOCATIONS FOR
                                                     BODY BUOYANCY TABLES =  -11.0  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   1.0
```

5 CONSTRAINTS ARE APPLIED ON ORDINATE

CONSTRAINT LOCATIONS

I	X(I)	Y(I)	Z(I)
1	130.850000	4.968800	-4.070000
2	189.000000	4.968800	-10.160000
3	243.390000	4.968800	-14.110000
4	189.000000	6.625000	-8.320000
5	189.000000	8.281300	-7.000000

PLANFORM DEFINITION

	X (LEADING EDGE)	Y	CHORD	X (TRAILING EDGE)
1	59.999300	0.000000	183.881700	243.881000
2	77.328000	4.968800	166.070000	243.398000
3	83.104000	6.625000	160.133000	243.237000
4	93.165000	9.510000	149.790000	242.955000
5	116.960000	16.333000	125.350000	242.310000
6	168.980000	31.250000	77.295000	246.275000
7	225.810000	47.544000	32.681000	258.491000
8	225.810000	47.545000	32.681000	258.491000
9	258.210000	66.250000	14.445000	272.655000

ORDINATES FOR BODY CAMBER LINE

I	X	Z	I	X	Z	I	X	Z	I	X	Z
1	0.00000	10.00000	2	16.67000	8.55000	3	33.33000	7.10000	4	50.00000	5.64000
5	66.67000	4.17000	6	83.33000	2.73000	7	100.00000	1.28000	8	116.67000	-.14000
9	133.33000	-1.60000	10	150.00000	-3.04000	11	166.66000	-4.50000	12	183.33000	-5.90000
13	200.00000	-7.40000	14	216.67000	-8.85000	15	233.33000	-10.25000	16	250.00000	-11.70000
17	266.67000	-13.20000	18	283.30000	-14.60000	19	295.00000	-15.70000			

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

```
  0.0000     1.0000     2.0000     3.0000     4.0000     5.0000     6.0000     7.0000     8.0000    10.0000
 12.0000    14.0000    16.0000    19.0000    22.0000    25.0000    28.0000    30.0000    32.0000    36.0000
 38.0000    40.0000
```

WING GRID SYSTEM PUTS SIDE-OF-FUSELAGE AT Y=   4.14063 AT EDGE OF ELEMENT ROW= 3

SPAN STATION OF ORDINATE CONSTRAINT 1 IS CHANGED FROM   4.96880 TO   4.96875

SPAN STATION OF ORDINATE CONSTRAINT 2 IS CHANGED FROM   4.96880 TO   4.96875

SPAN STATION OF ORDINATE CONSTRAINT 3 IS CHANGED FROM   4.96880 TO   4.96875

SPAN STATION OF ORDINATE CONSTRAINT 5 IS CHANGED FROM   8.28130 TO   8.28125

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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  QUADRATIC SPANWISE  (LOADING  4 IN THE LOADING DEFINITIONS)
LOADING  5 FOR THIS CASE IS QUADRATIC CHORDWISE  (LOADING  5 IN THE LOADING DEFINITIONS)
LOADING  6 FOR THIS CASE IS PARABOLIC CHORDWISE  (LOADING  6 IN THE LOADING DEFINITIONS)
LOADING  7 FOR THIS CASE IS   CUBIC CHORDWISE    (LOADING  7 IN THE LOADING DEFINITIONS)
LOADING  8 FOR THIS CASE IS SIMILAR TO FLAT WING (LOADING  8 IN THE LOADING DEFINITIONS)
LOADING  9 FOR THIS CASE IS SQ. ROOT FROM T. E.  (LOADING  9 IN THE LOADING DEFINITIONS)
LOADING 10 FOR THIS CASE IS  ELLIPTICAL C-SUB-P  (LOADING 10 IN THE LOADING DEFINITIONS)
LOADING 11 FOR THIS CASE IS LINEAR IN ARB.REGION (LOADING 11 IN THE LOADING DEFINITIONS)
LOADING 12 FOR THIS CASE IS BODY UPWASH LOADING  (LOADING 16 IN THE LOADING DEFINITIONS)
LOADING 13 FOR THIS CASE IS  NACELLE BUOYANCY    (LOADING 17 IN THE LOADING DEFINITIONS)
LOADING 14 FOR THIS CASE IS NACELLE BUOY(CAMBER) (LOADING 14 IN THE LOADING DEFINITIONS)
LOADING 15 FOR THIS CASE IS BODY UPWASH (CAMBER) (LOADING 13 IN THE LOADING DEFINITIONS)
LOADING 16 FOR THIS CASE IS  BODY BUOYANCY TERM  (LOADING 15 IN THE LOADING DEFINITIONS)
LOADING 17 FOR THIS CASE IS BODY BUOY. (CAMBER)  (LOADING 12 IN THE LOADING DEFINITIONS)
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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
```

DEFINITION OF ARBITRARY REGION FOR LOADING 11.

Y

```
       0.00000     66.25000
```

X

```
     207.00000    269.80000
```

ARBITRARY REGION DEFINITION (LOADING 11)

FRACTION OF SEMISPAN

```
       0.00000      1.00000
```

FRACTION OF LOCAL CHORD

```
        .79943       .80235
```

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	2.008000	15.470000	21.525000	28.017000	32.067000	35.040000

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	2.983000	3.633000	3.770000	3.654000	3.420000	3.420000

BODY BUOYANCY PRESSURES AT THE FOLLOWING X/C (PERCENT)

0.00000	10.00000	20.00000	30.00000	40.00000	50.00000	60.00000	70.00000	80.00000	90.00000	100.00000

AND AT THE FOLLOWING SPANWISE LOCATIONS (PERCENT SEMISPAN

0.00000	2.50000	5.00000	7.50000	10.00000	12.50000	15.00000	17.50000	20.00000	25.00000
30.00000	35.00000	40.00000	45.00000	50.00000	55.00000	60.00000	70.00000	80.00000	90.00000
100.00000									

BODY PRESSURES ON THE WING UPPER SURFACE

-.024245	-.054833	-.019740	.001568	.007146	.011193	-.001136	-.016040	-.021326	-.027706	-.034637
-.024245	-.054833	-.019740	.001568	.007146	.011193	-.001136	-.016040	-.021326	-.027706	-.034637
-.024245	-.054833	-.019740	.001568	.007146	.011193	-.001136	-.016040	-.021326	-.027706	-.034637
-.024245	-.054833	-.019740	.001568	.007146	.011193	-.001136	-.016040	-.021326	-.027706	-.034637
-.039550	-.039251	-.012045	.005122	.006814	.012885	-.003674	-.016136	-.021097	-.027050	-.032863
-.039027	-.033581	-.010241	.004473	.006134	.010506	-.000148	-.013167	-.016876	-.023801	-.027867
-.038867	-.029376	-.008856	.003979	.005636	.008635	.002994	-.011134	-.015103	-.021234	-.025682
-.038903	-.025919	-.007717	.003584	.005252	.007200	.005572	-.010105	-.013580	-.018985	-.021290
-.039051	-.022972	-.006746	.003254	.004944	.006159	.007757	-.009246	-.012211	-.015181	-.018958
-.036235	-.018070	-.005118	.002745	.004478	.004476	.008656	-.003031	-.009079	-.011662	-.016107
-.032104	-.014913	-.003527	.002573	.004163	.003724	.008427	.001389	-.007670	-.009904	-.010955
-.027202	-.012715	-.002147	.002442	.003931	.003508	.006388	.005286	-.004006	-.007507	-.009595
-.019293	-.010836	-.001230	.002340	.003750	.003337	.004609	.006803	.000449	-.006560	-.007717
-.014481	-.008147	-.000948	.002259	.003604	.003204	.003382	.006652	.004703	-.001743	-.006266
-.011306	-.005504	-.000662	.002344	.003459	.003085	.002895	.004727	.006129	.002387	-.002727
-.009317	-.003140	-.000379	.002422	.003319	.002975	.002811	.003530	.005965	.005534	.001253
-.005979	-.001043	-.000067	.002431	.003197	.002880	.002739	.002618	.004053	.005885	.005082
-.000799	-.000227	.001339	.002458	.002996	.002790	.002639	.002553	.002476	.002630	.003561

-.002570	-.000805	-.000395	.000316	.001596	.002287	.002753	.002704	.002557	.002466	.002404
-.007132	-.005187	-.003308	-.001570	-.000695	-.000397	-.000058	.000881	.001821	.002170	.002508
-.009527	-.008352	-.007788	-.007223	-.006449	-.005123	-.003796	-.002603	-.001424	-.000731	-.000535

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THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE .01288515 -.05483304
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BODY PRESSURES ON THE WING LOWER SURFACE

-.007867	-.019534	-.032427	-.023714	-.017120	-.019483	-.015240	.001680	.000964	.000383
-.018793									
-.007867	-.019534	-.032427	-.023714	-.017120	-.019483	-.015240	.001680	.000964	.000383
-.018793									
-.007867	-.019534	-.032427	-.023714	-.017120	-.019483	-.015240	.001680	.000964	.000383
-.018793									
-.007867	-.019534	-.032427	-.023714	-.017120	-.019483	-.015240	.001680	.000964	.000383
-.018793									
.000830	-.027034	-.029671	-.021035	-.016722	-.020463	-.011212	.001668	.000927	.000361
-.015359									
-.001092	-.025207	-.026487	-.019036	-.014962	-.017311	-.014807	.001427	.001000	.000340
-.001158									
-.002632	-.023954	-.024121	-.017571	-.013663	-.015054	-.018062	.001231	.001170	.000385
.000453									
-.003922	-.023053	-.022272	-.016440	-.012654	-.013259	-.017554	.001049	.001258	.000512
.000379									
-.005076	-.022382	-.020771	-.015535	-.011880	-.011778	-.017189	.000896	.001200	.000615
.000241									
-.008527	-.021027	-.018452	-.014157	-.010789	-.010188	-.014634	-.006483	.001008	.000902
.000379									
-.011253	-.019127	-.016663	-.013021	-.009958	-.009482	-.011820	-.012904	.000792	.000980
.000568									
-.013142	-.017644	-.015251	-.012138	-.009319	-.008929	-.009794	-.013205	-.003240	.000848
.000885									
-.014315	-.016444	-.014095	-.011426	-.008810	-.008360	-.008188	-.011177	-.011095	.000655
.000836									
-.015361	-.015032	-.013123	-.010837	-.008564	-.007889	-.007652	-.009042	-.011840	-.006381
.000659									
-.014811	-.013757	-.012252	-.010251	-.008306	-.007526	-.007431	-.007610	-.009648	-.010659
-.003779									
-.014037	-.012630	-.011389	-.009677	-.008035	-.007270	-.007127	-.006852	-.007998	-.009864
-.009895									
-.012861	-.011723	-.010611	-.009169	-.007802	-.007050	-.006830	-.006743	-.006690	-.007941
-.009529									
-.010751	-.010088	-.009266	-.008300	-.007425	-.006755	-.006485	-.006326	-.006317	-.006207
-.006018									
-.010636	-.010180	-.009754	-.009194	-.008603	-.007898	-.007179	-.006698	-.006229	-.006080
-.005931									
-.010980	-.010722	-.010306	-.009897	-.009588	-.009278	-.008912	-.008482	-.008052	-.007529
-.007005									
-.010305	-.010505	-.010474	-.010444	-.010413	-.010257	-.009982	-.009707	-.009432	-.009222
-.009018									

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THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE .00168000 -.03242652

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BODY BUOYANCY LOADING

.016378	.035299	-.012687	-.025282	-.024267	-.030676	-.014104	.017720	.022290	.028090
.015844									
.016378	.035299	-.012687	-.025282	-.024267	-.030676	-.014104	.017720	.022290	.028090
.015844									
.016378	.035299	-.012687	-.025282	-.024267	-.030676	-.014104	.017720	.022290	.028090
.015844									
.016378	.035299	-.012687	-.025282	-.024267	-.030676	-.014104	.017720	.022290	.028090
.015844									
.040379	.012217	-.017626	-.026157	-.023536	-.033348	-.007539	.017804	.022024	.027412
.017503									
.037935	.008374	-.016246	-.023510	-.021096	-.027816	-.014658	.014594	.017875	.024141
.026708									
.036234	.005422	-.015265	-.021550	-.019299	-.023689	-.021056	.012365	.016273	.021619
.026135									
.034981	.002866	-.014555	-.020024	-.017906	-.020459	-.023126	.011154	.014838	.019497
.021669									
.033975	.000589	-.014025	-.018790	-.016824	-.017937	-.024947	.010141	.013411	.015796
.019199									
.027708	-.002957	-.013334	-.016902	-.015268	-.014663	-.023291	-.003452	.010086	.012563
.016486									
.020851	-.004215	-.013136	-.015593	-.014121	-.013206	-.020247	-.014293	.008462	.010884
.011523									
.014061	-.004930	-.013105	-.014580	-.013251	-.012437	-.016183	-.018491	.000767	.008356
.010480									
.004978	-.005608	-.012866	-.013767	-.012560	-.011697	-.012797	-.017981	-.011544	.007215
.008552									
-.000879	-.006886	-.012175	-.013096	-.012168	-.011093	-.011035	-.015695	-.016544	-.004639
.006925									
-.003504	-.008253	-.011589	-.012595	-.011765	-.010610	-.010326	-.012337	-.015777	-.013046
-.001052									
-.004721	-.009490	-.011010	-.012099	-.011353	-.010244	-.009938	-.010382	-.013962	-.015398
-.011148									
-.006882	-.010679	-.010544	-.011599	-.010999	-.009931	-.009569	-.009361	-.010743	-.013826
-.014611									
-.009952	-.009861	-.010606	-.010758	-.010422	-.009545	-.009125	-.008878	-.008793	-.008836
-.009579									
-.008066	-.009375	-.009359	-.009511	-.010199	-.010185	-.009933	-.009402	-.008787	-.008547
-.008336									
-.003848	-.005536	-.006998	-.008327	-.008893	-.008881	-.008853	-.009363	-.009873	-.009699
-.009513									
-.000777	-.002153	-.002687	-.003221	-.003964	-.005134	-.006185	-.007104	-.008008	-.008491
-.008483									

****
THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE .04037947 -.03334836
****

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

.001046	.003268	.008605	.022937	.026235	.021424	.019804	.027286	.028499	.021600
.016973	.012435								
.001210	.004207	.009450	.022951	.026135	.021630	.020551	.028112	.028343	.021572
.017904	.012998								
.003948	.007438	.012438	.023190	.025926	.022471	.024283	.033044	.030582	.022076
.015995	.011131								
.011283	.012877	.016625	.024645	.026816	.025275	.028270	.028964	.028247	.020933
.013199	.009054								
.044891	.033265	.025648	.021962	.022104	.022091	.023088	.024544	.025948	.019707
.011485	.007869								
.057560	.039350	.028015	.019679	.020503	.020611	.020846	.021983	.023942	.019261
.011161	.007446								
.064083	.041929	.028525	.018210	.019390	.019663	.019254	.020186	.022105	.019174
.011588	.007461								
.060097	.041327	.027377	.017532	.018519	.018796	.018220	.018739	.020612	.019127
.012466	.007724								
.061100	.040936	.027786	.016688	.018059	.017962	.017621	.017622	.019225	.018820
.013584	.008458								
.063218	.040383	.027430	.016071	.017378	.017479	.017102	.016827	.017870	.018444
.014742	.009421								
.055863	.039094	.026533	.015627	.016730	.017247	.016653	.016164	.016736	.017987
.015691	.010682								
.056742	.039069	.026826	.014715	.016546	.016873	.016372	.015525	.015960	.017246
.016255	.011958								
.050563	.036938	.025393	.014684	.016385	.016550	.016037	.014992	.015388	.016511
.016454	.013308								
.050544	.036662	.025820	.015651	.016061	.016391	.015614	.014781	.014853	.015814
.016289	.014378								
.051339	.036534	.026003	.016162	.015982	.016013	.015365	.014700	.014403	.015164
.015865	.015159								
.045565	.034861	.025304	.016620	.015576	.015669	.015338	.014558	.014135	.014519
.015260	.015428								
.046219	.035585	.026576	.017320	.014980	.015641	.015234	.014439	.013924	.013915
.014679	.015292								
.041978	.033780	.025999	.017439	.014819	.015593	.015098	.014471	.013663	.013476
.014129	.014927								
.042263	.034063	.025941	.018467	.014794	.015399	.015161	.014359	.013458	.013217
.013592	.014309								
.043198	.034859	.027294	.019351	.014837	.015349	.015047	.014178	.013428	.013122
.013117	.013680								
.039325	.033159	.027245	.019746	.015340	.015086	.014767	.014094	.013531	.012933
.012754	.013173								
.040382	.034349	.028171	.020924	.015997	.014606	.014623	.014257	.013457	.012753
.012639	.012868								

.037722	.032968	.028238	.020806	.016363	.014430	.014770	.014159	.013358	.012844
.012727	.013117								
.038605	.033723	.028866	.021656	.017401	.014746	.014475	.014027	.013537	.013130
.013015	.013172								
.040015	.035058	.030037	.022750	.018315	.015057	.014109	.014213	.013939	.013607
.013409	.013299								
.037241	.033434	.029628	.022613	.018743	.015685	.014367	.014606	.014435	.014179
.013814	.013475								
.038194	.034540	.030836	.024108	.020328	.017334	.015352	.014909	.014906	.014699
.014189	.013646								
.040175	.035981	.032178	.025911	.021925	.019136	.016809	.015337	.015264	.014922
.014411	.013888								
.038668	.035922	.033175	.027831	.023690	.020804	.018396	.016061	.015093	.014763
.014348	.013505								
.039868	.037489	.035110	.030134	.025728	.022193	.019588	.017263	.015498	.014417
.014054	.013837								
.035593	.034145	.032698	.029548	.025832	.022755	.020221	.018084	.016109	.014929
.014229	.013871								
.034223	.032617	.031171	.028744	.026123	.023529	.021100	.019155	.017335	.015638
.014942	.014246								
.029987	.029567	.029147	.027768	.026066	.023979	.021946	.019985	.018301	.016686
.015147	.013899								
.026942	.026816	.026690	.026349	.025279	.023912	.022242	.020555	.018938	.017521
.016186	.014895								
.024511	.024440	.024368	.024225	.023815	.023077	.021997	.020789	.019452	.018242
.017350	.016458								
.021830	.022005	.022180	.022445	.022637	.022389	.021947	.021101	.020184	.019073
.017866	.016164								
.019520	.019762	.020004	.020488	.020888	.021267	.021129	.020881	.020256	.019559
.018614	.017626								
.017739	.017941	.018143	.018546	.018949	.019399	.019852	.019861	.019765	.019436
.018985	.018534								
.014616	.015102	.015588	.016452	.017071	.017691	.018127	.018555	.018665	.018569
.018203	.017060								
.013004	.013313	.013622	.014241	.014820	.015322	.015824	.016114	.016335	.016456
.016076	.015696								
.010373	.010490	.010608	.010842	.011077	.011296	.011432	.011568	.011648	.011580
.011513	.011445								

```
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
****************************************************************************************************

C-P LONGITUDINAL GRADIENT LIMIT
              .002500     .002500
              .002500     .002500

****************************************************************************************************
           THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE          .00250000      .00250000
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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	.007181	.015607	.020424	.012817	.007863	.005973	.003528	.002811	.005289
.003303	.000610	.000151	-.001318	-.003688	-.004128	-.007772	-.013488	-.017615	-.021555
-.026499									
.003049	.007607	.013422	.014378	.009965	.007961	.008164	.005860	.003421	.002533
.001086	.000594	.001558	.000311	-.002880	-.006007	-.010054	-.014312	-.017090	-.020651
-.025437									
.010939	.011780	.015375	.013403	.012492	.008389	.004969	.003865	.004275	.002871
.002566	.001463	-.000675	-.002347	-.003482	-.006735	-.010090	-.014023	-.018806	-.023430
-.026351									
.035284	.010912	.005615	.005186	.009271	.008633	.004337	.004304	.003884	.001148
.001523	.001518	-.001747	-.003713	-.005689	-.010283	-.013900	-.016427	-.021427	-.025760
-.027668									
.063472	.007600	-.005832	.004328	.007403	.004838	.002677	.003110	.002020	.001265
.001828	-.000367	-.003840	-.006206	-.009535	-.013783	-.017184	-.020046	-.024428	-.028060
-.029891									
.093863	.006988	-.006320	.002556	.004160	.003531	.001621	.001257	.001579	.000937
.000031	-.001399	-.003455	-.007376	-.012355	-.015598	-.018511	-.021984	-.025640	-.029708
-.032117									
.133990	.005068	-.010583	-.000360	.003827	.002310	-.000515	.001040	.001054	-.000971
-.001206	-.000325	-.004685	-.010334	-.013589	-.014761	-.019423	-.023994	-.026222	-.029065
-.032547									
.050564	-.005361	-.009168	.000630	-.001391	-.000912	.000929	-.000706	-.002469	-.001061
-.001079	-.004266	-.007965	-.011250	-.015679	-.019393	-.021710	-.025177	-.028616	-.031299
-.033278									
.040388	-.005435	-.012106	-.004102	-.005151	-.004461	-.002725	-.000706	-.001555	-.003227
-.003446	-.004986	-.008411	-.013225	-.017520	-.020189	-.023190	-.026821	-.030448	-.033071
-.034131									
.027466	-.006828	-.011185	-.009507	-.006045	-.005855	-.004519	-.002755	-.003994	-.001841
-.004837	-.007420	-.011106	-.014519	-.017333	-.022887	-.026213	-.029833	-.031124	-.034304
-.037308									
.049029	.003490	-.008104	-.014617	-.010100	-.008771	-.003495	-.003808	-.004304	-.005878
-.006262	-.008352	-.011633	-.016311	-.021072	-.023907	-.027272	-.030746	-.034898	-.036204
-.038300									
.040513	.001386	-.010265	-.012448	-.011351	-.010549	-.007385	-.004560	-.005188	-.005760
-.008216	-.011057	-.014561	-.017424	-.021129	-.025144	-.029719	-.033393	-.036282	-.038077
-.038769									
.032322	-.002308	-.013220	-.015858	-.013421	-.007985	-.008625	-.006773	-.007687	-.009129
-.008486	-.012453	-.014931	-.020065	-.023851	-.026846	-.030958	-.033339	-.037421	-.040173
-.042339									
.018075	-.002695	-.013526	-.017029	-.017991	-.011116	-.007332	-.007855	-.007451	-.010320
-.012964	-.014798	-.018562	-.021491	-.024274	-.028923	-.032109	-.036115	-.039599	-.041182
-.042446									
.020833	-.001276	-.010419	-.015099	-.014387	-.014226	-.013563	-.011754	-.012966	-.015005
-.017232	-.019086	-.021376	-.026207	-.029886	-.033134	-.037964	-.041284	-.043432	-.044431
-.045429									

.001309	-.004798	-.010906	-.014763	-.015342	-.015964	-.017024	-.018084	-.018067	-.017740
-.019248	-.022904	-.026569	-.030291	-.034013	-.038357	-.042826	-.046341	-.048955	-.051593
-.054323									
.041524	.034978	.028432	.021860	.015246	.008632	.002186	-.003937	-.010060	-.015977
-.021424	-.026872	-.031841	-.035512	-.039184	-.042754	-.045989	-.049224	-.052262	-.054495
-.056729									
.045388	.041996	.038604	.035212	.031819	.026961	.021219	.015477	.009735	.003178
-.003818	-.010815	-.017812	-.024015	-.029836	-.035658	-.041480	-.045631	-.049070	-.052509
-.055948									
.046229	.043122	.040014	.036906	.033160	.029018	.024876	.020734	.016592	.010999
.005275	-.000450	-.006175	-.011970	-.018024	-.024077	-.030131	-.036184	-.042748	-.049790
-.056833									
.034932	.032537	.030143	.027748	.025354	.022959	.020565	.018170	.015466	.012046
.008627	.005208	.001788	-.001631	-.005038	-.008434	-.011831	-.015227	-.018624	-.022020
-.025417									

****
THE MAXIMUM AND MINIMUM OF THE PRECEDING ARRAY ARE .13398973 -.05683268
****

OVERLAY 1, DEPART

OVERLAY 2, ENTER

FLAT WING FORCE COEFFICIENTS

C_L = .027674 C_D = .0004830 $\frac{XF}{L}$ = .720744 $\frac{C_M}{C_L}$ = -.089414 $\frac{C_D}{C_L^2}$ = .630682

CARD 9 PARAMETERS.

XF = .72074445 SCL9 = .02767366 KF = .63068237 AREA9 = 781.13547925 FACTOR = 1.08589045

NOTE XF HAS BEEN CHANGED TO THE WING-BODY VALUE OF .72238572

DELTAT = 61.633 SEC., T = 374.177 SEC.

OVERLAY 2, DEPART

OVERLAY 3, ENTER

DELTAT = .022 SEC., T = 374.199 SEC.

FUSELAGE CONTRIBUTION AND CARRY-OVER LIFT

FUSELAGE CL= -.00000 CD= .000001 CM= .00396 XAC/XMAX= .72239 CMO= .00396

CARRY-OVER CP FOR LOADING 1 OF THIS CASE (UNIFORM OR CONSTANT)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.09609	.62278	.97071	.86523	.87947	.91622	.92619	.92993	.93711	.94577
.025	0.00000	0.00000	0.00000	.06494	.42603	.78994	.95960	.86806	.89054	.91949	.92996	.93619	.94577	.94861
.050	0.00000	0.00000	.12231	.51132	.68960	.92131	.95010	.89851	.91921	.94254	.94951	.95346	.95621	.96248

CARRY-OVER CP FOR LOADING 2 OF THIS CASE (LINEAR CHORDWISE)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.00142	.06658	.36442	.67677	.96826	1.27295	1.58644	1.90209	2.22009	2.52535
.025	0.00000	0.00000	0.00000	.00096	.04115	.14744	.45471	.75185	1.03801	1.33679	1.64239	1.94942	2.25864	2.55742
.050	0.00000	0.00000	.00180	.06077	.15679	.28976	.59490	.88633	1.17484	1.47197	1.77305	2.07520	2.37899	2.67447

CARRY-OVER CP FOR LOADING 3 OF THIS CASE (LINEAR SPANWISE

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.02162	.14013	.25787	.26940	.29815	.32205	.34178	.35670	.37042	.38695
.025	0.00000	0.00000	0.00000	.01461	.09586	.18720	.26451	.27227	.29696	.31657	.33648	.35249	.36906	.37757
.050	0.00000	0.00000	.02752	.11505	.16473	.22602	.26219	.26853	.28577	.30381	.31805	.32886	.33591	.34763

CARRY-OVER CP FOR LOADING 4 OF THIS CASE (QUADRATIC SPANWISE)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.00324	.02102	.04770	.06160	.07942	.09385	.10975	.12440	.13857	.15651
.025	0.00000	0.00000	0.00000	.00219	.01438	.02997	.05147	.06361	.07849	.09052	.10630	.12132	.13789	.14856
.050	0.00000	0.00000	.00413	.01726	.02662	.03812	.05191	.06051	.07071	.08193	.09358	.10402	.11211	.12475

CARRY-OVER CP FOR LOADING 5 OF THIS CASE (QUADRATIC CHORDWISE)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.00001	.00553	.09921	.33698	.70923	1.21733	1.86474	2.65610	3.59019	4.61368
.025	0.00000	0.00000	0.00000	.00001	.00337	.02257	.15315	.41932	.81145	1.33146	1.98557	2.77348	3.69595	4.70929
.050	0.00000	0.00000	.00002	.00531	.02780	.07416	.26222	.57725	1.01419	1.57517	2.26305	3.07872	4.02271	5.05940

CARRY-OVER CP FOR LOADING 6 OF THIS CASE (PARABOLIC CHORDWISE)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.00578	.26154	1.28034	2.04478	2.43691	2.59806	2.50492	2.12203	1.45458	.55339
.025	0.00000	0.00000	0.00000	.00390	.16177	.55681	1.53498	2.17883	2.51116	2.62806	2.49415	2.08979	1.41760	.51820
.050	0.00000	0.00000	.00735	.23820	.58449	1.03169	1.88383	2.41143	2.67341	2.71473	2.51129	2.05063	1.33448	.39642

CARRY-OVER CP FOR LOADING 7 OF THIS CASE (CUBIC CHORDWISE)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.00004	.01373	.22895	.71164	1.35238	2.07129	2.80113	3.46304	3.97682	4.25469
.025	0.00000	0.00000	0.00000	.00002	.00836	.05447	.34405	.86302	1.51117	2.22080	2.92903	3.56362	4.05131	4.31218
.050	0.00000	0.00000	.00005	.01319	.06676	.17208	.56544	1.14239	1.82128	2.54044	3.23875	3.84929	4.30423	4.53138

CARRY-OVER CP FOR LOADING 8 OF THIS CASE (SIMILAR TO FLAT WING)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.23201	1.27193	1.45810	.93603	.85827	.82254	.74669	.68248	.63953	.61009
.025	0.00000	0.00000	0.00000	.15679	.88787	1.45715	1.32356	.90573	.85082	.80357	.73625	.68382	.65017	.61093
.050	0.00000	0.00000	.29531	1.02276	1.21363	1.49740	1.18172	.90264	.84564	.80039	.73832	.68818	.64407	.61603

CARRY-OVER CP FOR LOADING 9 OF THIS CASE (SQ. ROOT FROM T. E.)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.18862	1.20478	1.77554	1.44398	1.35015	1.29083	1.16160	1.00145	.81854	.59611
.025	0.00000	0.00000	0.00000	.12747	.82552	1.50380	1.71926	1.42141	1.34689	1.27419	1.14707	.99337	.81614	.58554
.050	0.00000	0.00000	.24008	.98726	1.30328	1.71225	1.65294	1.43792	1.36006	1.27407	1.14068	.98235	.79139	.55233

CARRY-OVER CP FOR LOADING 10 OF THIS CASE (ELLIPTICAL C-SUB-P)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	.10540	.68313	1.06339	.94608	.96008	.99915	1.00862	1.01133	1.01786	1.02563
.025	0.00000	0.00000	0.00000	.07123	.46732	.86618	1.05083	.94901	.97235	1.00307	1.01309	1.01851	1.02743	1.02951
.050	0.00000	0.00000	.13416	.56087	.75611	1.00994	1.04034	.98279	1.00460	1.02922	1.03578	1.03912	1.04136	1.04702

CARRY-OVER CP FOR LOADING 11 OF THIS CASE (LINEAR IN ARB.REGION)

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	.02361	.32718
.025	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	.04937	.34620
.050	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	.12444	.41148

CARRY-OVER CP FOR LOADING 14 OF THIS CASE (NACELLE BUOY(CAMBER))

Y/B/2 \ XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00	90.00	100.00
0.000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	.00544
.025	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	.00827
.050	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	.01860

CARRY-OVER CP FOR LOADING 15 OF THIS CASE (BODY UPWASH (CAMBER))

XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00
	70.00	80.00	90.00	100.00						
Y/B/2										
0.000	0.00000	0.00000	0.00000	0.00000	.00110	.00772	.02543	.02471	.02049	.02167
	.02353	.02272	.01651	.00890						
.025	0.00000	0.00000	0.00000	.00074	.00523	.01423	.02704	.02396	.02034	.02212
	.02361	.02243	.01631	.00896						
.050	0.00000	0.00000	.00140	.00640	.01343	.01935	.02636	.02333	.02187	.02377
	.02461	.02182	.01523	.00919						

CARRY-OVER CP FOR LOADING 17 OF THIS CASE (BODY BUOY. (CAMBER))

XPCT	0.00	2.50	5.00	10.00	15.00	20.00	30.00	40.00	50.00	60.00
	70.00	80.00	90.00	100.00						
Y/B/2										
0.000	0.00000	0.00000	0.00000	0.00000	.00166	.01430	.01566	-.02464	-.02607	-.02291
	-.01435	.01581	.02350	.02257						
.025	0.00000	0.00000	0.00000	.00112	.00951	.02316	.00324	-.02757	-.02486	-.02301
	-.01038	.01681	.02371	.02181						
.050	0.00000	0.00000	.00211	.01212	.02198	.01677	-.00961	-.02720	-.02542	-.02068
	-.00235	.01807	.02452	.01991						

DELTAT = 253.616 SEC., T = 627.815 SEC.

OVERLAY 3, DEPART

OVERLAY 4, ENTER

WING DATA FOR UNIFORM OR CONSTANT LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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	183.8817000	0.0000000	.7409501	-.6845282
.0250000	177.9445264	0.0000000	.7758395	-.7372558
.0500000	172.0073528	0.0000000	.8387086	-.8141165
.0750000	166.0701792	3.3007611	1.0000000	-.9656284
.1000000	160.1330000	1.6568419	1.0000000	-1.0192347
.1250000	154.1951859	1.2350506	1.0000000	-1.0764594
.1500000	148.2586941	1.0146702	1.0000000	-1.1383784
.1750000	142.3260032	.8688242	1.0000000	-1.2058524
.2000000	136.3933123	.7616182	1.0000000	-1.2787008
.2500000	124.6106675	.6020349	1.0000000	-1.4451936
.3000000	113.9394744	.4860737	1.0000000	-1.6352923
.3500000	103.2682813	.3920641	1.0000000	-1.8640200
.4000000	92.5970882	.3116704	1.0000000	-2.1461007
.4750000	76.6960487	.1999098	1.0000000	-2.7130599
.5500000	63.0912977	.1024884	1.0000000	-3.4670189
.6250000	49.4865467	.0054721	1.0000000	-4.6326307
.7000000	35.8817958	-.1934528	1.0000000	-6.6821074
.7500000	30.5922200	-.0261647	1.0000000	-8.0063824
.8000000	27.3627760	.1156497	1.0000000	-9.0987203
.9000000	20.9038880	.2639022	1.0000000	-12.3067406
.9500000	17.6744440	.2982005	1.0000000	-14.7858920
1.0000000	14.4450000	.1423389	1.0000000	-18.3990688

C_L = .938399 C_D = .647121 $\frac{X_{CP}}{L}$ = .714229 K_E = .734870

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = -.072721 C_{M_0} = .019612

INTERFERENCE DRAG OF LOADING	2 (LINEAR CHORDWISE)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.10231462E+01
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.35038758E+00
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.20308055E+00
INTERFERENCE DRAG OF LOADING	5 (QUADRATIC CHORDWISE)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.12990764E+01
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.10857848E+01
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.14564008E+01
INTERFERENCE DRAG OF LOADING	8 (SIMILAR TO FLAT WING)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.54114581E+00
INTERFERENCE DRAG OF LOADING	9 (SQ. ROOT FROM T. E.)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.72564265E+00
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.69049248E+00
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.47908493E-01
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.12048131E-01
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.38898304E-02
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.38898304E-02
INTERFERENCE DRAG OF LOADING	15 (BODY UPWASH (CAMBER))	ON LOADING	1 (UNIFORM OR CONSTANT) IS	.12048131E-01
INTERFERENCE DRAG OF LOADING	16 (BODY BUOYANCY TERM)	ON LOADING	1 (UNIFORM OR CONSTANT) IS	-.12190312E-02
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING	1 (UNIFORM OR CONSTANT) IS	-.12190312E-02

WING DATA FOR LINEAR CHORDWISE LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/B/2	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	1.0303936	-1.0852044
.0250000	177.9445264	0.0000000	1.0841903	-1.1771041
.0500000	172.0073528	0.0000000	1.2019202	-1.3400052
.0750000	166.0701792	9.2051767	1.5353591	-1.7384780
.1000000	160.1330000	4.2757079	1.4812924	-1.7556970
.1250000	154.1951859	2.9089002	1.4257731	-1.7721179
.1500000	148.2586941	2.1906517	1.3705690	-1.7887482
.1750000	142.3260032	1.6984455	1.3164540	-1.8061801
.2000000	136.3933123	1.3400430	1.2610898	-1.8227120
.2500000	124.6106675	.8456530	1.1525092	-1.8573530
.3000000	113.9394744	.5429794	1.0541730	-1.8985884
.3500000	103.2682813	.3337833	.9545366	-1.9387352
.4000000	92.5970882	.1806995	.8562689	-1.9807214
.4750000	76.6960487	.0447445	.7089300	-2.0419749
.5500000	63.0912977	-.0230816	.5855507	-2.1258335
.6250000	49.4865467	-.0532899	.4595165	-2.2041253
.7000000	35.8817958	-.0637878	.3334628	-2.2833479
.7500000	30.5922200	-.0215506	.2845145	-2.3231477
.8000000	27.3627760	.0018451	.2534699	-2.3491939
.9000000	20.9038880	.0104310	.1946102	-2.4256708
.9500000	17.6744440	.0121128	.1638388	-2.4521495
1.0000000	14.4450000	.0058128	.1387718	-2.5710074

C_L = .908181 C_D = 1.318625 $\frac{X_{CP}}{L}$ = .746701 K_E = 1.598735

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = -.155923 C_{M_0} = -.056583

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING	2 (LINEAR CHORDWISE) IS	.64364539E+00
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING	2 (LINEAR CHORDWISE) IS	.24261463E+00
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING	2 (LINEAR CHORDWISE) IS	.63018484E-01
INTERFERENCE DRAG OF LOADING	5 (QUADRATIC CHORDWISE)	ON LOADING	2 (LINEAR CHORDWISE) IS	.19323669E+01
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING	2 (LINEAR CHORDWISE) IS	.11180578E+01
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	2 (LINEAR CHORDWISE) IS	.21009686E+01
INTERFERENCE DRAG OF LOADING	8 (SIMILAR TO FLAT WING)	ON LOADING	2 (LINEAR CHORDWISE) IS	.47843654E+00
INTERFERENCE DRAG OF LOADING	9 (SQ. ROOT FROM T. E.)	ON LOADING	2 (LINEAR CHORDWISE) IS	.64562905E+00
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING	2 (LINEAR CHORDWISE) IS	.70224286E+00
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING	2 (LINEAR CHORDWISE) IS	.82971350E-01
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING	2 (LINEAR CHORDWISE) IS	.11909504E-01
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING	2 (LINEAR CHORDWISE) IS	.47298498E-02
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING	2 (LINEAR CHORDWISE) IS	.47298498E-02
INTERFERENCE DRAG OF LOADING	15 (BODY UPWASH (CAMBER))	ON LOADING	2 (LINEAR CHORDWISE) IS	.11909504E-01
INTERFERENCE DRAG OF LOADING	16 (BODY BUOYANCY TERM)	ON LOADING	2 (LINEAR CHORDWISE) IS	.35710802E-02
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING	2 (LINEAR CHORDWISE) IS	.35710802E-02

WING DATA FOR LINEAR SPANWISE LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$Y/(B/2)$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.2524992	-.2420649
.0250000	177.9445264	0.0000000	.2592031	-.2555230
.0500000	172.0073528	0.0000000	.2596443	-.2602768
.0750000	166.0701792	-.0996978	.2250000	-.2172664
.1000000	160.1330000	.0476099	.3000000	-.3057704
.1250000	154.1951859	.1467936	.3750000	-.4036723
.1500000	148.2586941	.2434399	.4500000	-.5122703
.1750000	142.3260032	.3415410	.5250000	-.6330725
.2000000	136.3933123	.4407091	.6000000	-.7672205
.2500000	124.6106675	.6381523	.7500000	-1.0838952
.3000000	113.9394744	.8306856	.9000000	-1.4717630
.3500000	103.2682813	1.0074875	1.0500000	-1.9572210
.4000000	92.5970882	1.1627786	1.2000000	-2.5753208
.4750000	76.6960487	1.3200773	1.4250000	-3.8661103
.5500000	63.0912977	1.3938920	1.6500000	-5.7205812
.6250000	49.4865467	1.3085455	1.8750000	-8.6861825
.7000000	35.8817958	.5532453	2.1000000	-14.0324256
.7500000	30.5922200	1.2209408	2.2500000	-18.0143604
.8000000	27.3627760	1.9458385	2.4000000	-21.8369286
.9000000	20.9038880	3.0278847	2.7000000	-33.2281995
.9500000	17.6744440	3.3840285	2.8500000	-42.1397923
1.0000000	14.4450000	2.1109021	3.0000000	-55.1972064

C_L = 1.020416 C_D = .803209 X_{CP}/L = .778783 K_E = .771391

S_{REF}/S_{PROG} = .920569 C_M/C_L = -.238127 C_{M_0} = -.147457

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING	3 (LINEAR SPANWISE)	IS	.63674079E+00
INTERFERENCE DRAG OF LOADING	2 (LINEAR CHORDWISE)	ON LOADING	3 (LINEAR SPANWISE)	IS	.60668757E+00
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING	3 (LINEAR SPANWISE)	IS	.84049145E+00
INTERFERENCE DRAG OF LOADING	5 (QUADRATIC CHORDWISE)	ON LOADING	3 (LINEAR SPANWISE)	IS	.48663795E+00
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING	3 (LINEAR SPANWISE)	IS	.53436029E+00
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	3 (LINEAR SPANWISE)	IS	.40799350E+00
INTERFERENCE DRAG OF LOADING	8 (SIMILAR TO FLAT WING)	ON LOADING	3 (LINEAR SPANWISE)	IS	.55157197E+00
INTERFERENCE DRAG OF LOADING	9 (SQ. ROOT FROM T. E.)	ON LOADING	3 (LINEAR SPANWISE)	IS	.58883880E+00
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING	3 (LINEAR SPANWISE)	IS	.60785379E+00
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING	3 (LINEAR SPANWISE)	IS	.22843435E-01
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING	3 (LINEAR SPANWISE)	IS	.10442415E-01
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING	3 (LINEAR SPANWISE)	IS	.60074767E-02
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING	3 (LINEAR SPANWISE)	IS	.60074767E-02
INTERFERENCE DRAG OF LOADING	15 (BODY UPWASH (CAMBER))	ON LOADING	3 (LINEAR SPANWISE)	IS	.10442415E-01
INTERFERENCE DRAG OF LOADING	16 (BODY BUOYANCY TERM)	ON LOADING	3 (LINEAR SPANWISE)	IS	-.56983528E-02
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING	3 (LINEAR SPANWISE)	IS	-.56983528E-02

WING DATA FOR QUADRATIC SPANWISE LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/B/2	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.0750794	-.0760302
.0250000	177.9445264	0.0000000	.0754519	-.0786752
.0500000	172.0073528	0.0000000	.0689181	-.0730018
.0750000	166.0701792	-.0148206	.0337500	-.0325900
.1000000	160.1330000	-.0106846	.0600000	-.0611541
.1250000	154.1951859	-.0059499	.0937500	-.1009181
.1500000	148.2586941	.0047263	.1350000	-.1536811
.1750000	142.3260032	.0241547	.1837500	-.2215754
.2000000	136.3933123	.0547094	.2400000	-.3068882
.2500000	124.6106675	.1584247	.3750000	-.5419476
.3000000	113.9394744	.3321832	.5400000	-.8830578
.3500000	103.2682813	.5872992	.7350000	-1.3700547
.4000000	92.5970882	.9311917	.9600000	-2.0602566
.4750000	76.6960487	1.5881544	1.3537500	-3.6728048
.5500000	63.0912977	2.3892802	1.8150000	-6.2926394
.6250000	49.4865467	3.1386757	2.3437500	-10.8577281
.7000000	35.8817958	2.5837882	2.9400000	-19.6453959
.7500000	30.5922200	4.3301086	3.3750000	-27.0215406
.8000000	27.3627760	6.6331417	3.8400000	-34.9390858
.9000000	20.9038880	11.5393497	4.8600000	-59.8107591
.9500000	17.6744440	13.8329800	5.4150000	-80.0656054
1.0000000	14.4450000	9.9778845	6.0000000	-110.3944127

C_L = 1.033028 C_D = 1.386804 $\frac{X_{CP}}{L}$ = .829257 K_E = 1.299544

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = -.367456 C_{M_0} = -.282881

INTERFERENCE DRAG OF LOADING	1	(UNIFORM OR CONSTANT)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.62736474E+00
INTERFERENCE DRAG OF LOADING	2	(LINEAR CHORDWISE)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.39317316E+00
INTERFERENCE DRAG OF LOADING	3	(LINEAR SPANWISE)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.10849319E+01
INTERFERENCE DRAG OF LOADING	5	(QUADRATIC CHORDWISE)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.18173140E+00
INTERFERENCE DRAG OF LOADING	6	(PARABOLIC CHORDWISE)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.26964457E+00
INTERFERENCE DRAG OF LOADING	7	(CUBIC CHORDWISE)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.76256456E-01
INTERFERENCE DRAG OF LOADING	8	(SIMILAR TO FLAT WING)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.55597137E+00
INTERFERENCE DRAG OF LOADING	9	(SQ. ROOT FROM T. E.)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.48665544E+00
INTERFERENCE DRAG OF LOADING	10	(ELLIPTICAL C-SUB-P)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.53074380E+00
INTERFERENCE DRAG OF LOADING	11	(LINEAR IN ARB.REGION)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.12634418E-01
INTERFERENCE DRAG OF LOADING	12	(BODY UPWASH LOADING)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.10336168E-01
INTERFERENCE DRAG OF LOADING	13	(NACELLE BUOYANCY)	ON LOADING	4 (QUADRATIC SPANWISE) IS	.56882088E-02
INTERFERENCE DRAG OF LOADING	14	(NACELLE BUOY(CAMBER))	ON LOADING	4 (QUADRATIC SPANWISE) IS	.56882088E-02
INTERFERENCE DRAG OF LOADING	15	(BODY UPWASH (CAMBER))	ON LOADING	4 (QUADRATIC SPANWISE) IS	.10336168E-01
INTERFERENCE DRAG OF LOADING	16	(BODY BUOYANCY TERM)	ON LOADING	4 (QUADRATIC SPANWISE) IS	-.69055808E-02
INTERFERENCE DRAG OF LOADING	17	(BODY BUOY. (CAMBER))	ON LOADING	4 (QUADRATIC SPANWISE) IS	-.69055808E-02

WING DATA FOR QUADRATIC CHORDWISE LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/(B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	1.2783719	-1.4312949
.0250000	177.9445264	0.0000000	1.3542289	-1.5638501
.0500000	172.0073528	0.0000000	1.5393599	-1.8291181
.0750000	166.0701792	20.5465001	2.0953535	-2.5471187
.1000000	160.1330000	8.7261141	1.9482178	-2.4720552
.1250000	154.1951859	5.4330349	1.8063386	-2.3957348
.1500000	148.2586941	3.6945433	1.6699737	-2.3184911
.1750000	142.3260032	2.5937813	1.5391427	-2.2403421
.2000000	136.3933123	1.8400386	1.4135907	-2.1609479
.2500000	124.6106675	.9155049	1.1800841	-2.0003908
.3000000	113.9394744	.4417276	.9861979	-1.8587434
.3500000	103.2682813	.1832124	.8103071	-1.7132585
.4000000	92.5970882	.0494787	.6521124	-1.5629463
.4750000	76.6960487	-.0238774	.4471943	-1.3252851
.5500000	63.0912977	-.0336361	.3030265	-1.1260259
.6250000	49.4865467	-.0235780	.1867411	-.9117735
.7000000	35.8817958	-.0118632	.0985107	-.6831024
.7500000	30.5922200	-.0037483	.0709289	-.5852814
.8000000	27.3627760	-.0006476	.0573165	-.5360979
.9000000	20.9038880	.0002949	.0331481	-.4160066
.9500000	17.6744440	.0003210	.0243550	-.3666617
1.0000000	14.4450000	.0000840	.0159019	-.2960677

C_L = .917149 C_D = 2.443731 $\frac{X_{CP}}{L}$ = .771585 K_E = 2.905185

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = -.219684 C_{M_0} = -.115619

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.67869558E+00
INTERFERENCE DRAG OF LOADING	2 (LINEAR CHORDWISE)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.15462416E+01
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.22331264E+00
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.40590513E-01
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.10871694E+01
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.25862970E+01
INTERFERENCE DRAG OF LOADING	8 (SIMILAR TO FLAT WING)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.48266304E+00
INTERFERENCE DRAG OF LOADING	9 (SQ. ROOT FROM T. E.)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.62254553E+00
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.74315703E+00
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.11814812E+00
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.12057017E-01
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.55920414E-02
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.55920414E-02
INTERFERENCE DRAG OF LOADING	15 (BODY UPWASH (CAMBER))	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.12057017E-01
INTERFERENCE DRAG OF LOADING	16 (BODY BUOYANCY TERM)	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.78468824E-02
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING	5 (QUADRATIC CHORDWISE)	IS	.78468824E-02

WING DATA FOR PARABOLIC CHORDWISE LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$Y/(B/2)$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	1.4917912	-1.3918589
.0250000	177.9445264	0.0000000	1.5615864	-1.4985080
.0500000	172.0073528	0.0000000	1.6968809	-1.6570225
.0750000	166.0701792	20.1820783	2.0958671	-2.0240129
.1000000	160.1330000	7.8361132	1.9519239	-1.9876386
.1250000	154.1951859	4.3180221	1.8077514	-1.9456747
.1500000	148.2586941	2.5981171	1.6700038	-1.9015728
.1750000	142.3260032	1.4855479	1.5412931	-1.8573958
.2000000	136.3933123	.7664999	1.4136547	-1.8075831
.2500000	124.6106675	-.0557157	1.1807185	-1.7056169
.3000000	113.9394744	-.2671609	.9890139	-1.6160763
.3500000	103.2682813	-.3572653	.8097533	-1.5097411
.4000000	92.5970882	-.3646834	.6506543	-1.3961343
.4750000	76.6960487	-.2814541	.4461790	-1.2107498
.5500000	63.0912977	-.1735138	.3047960	-1.0547725
.6250000	49.4865467	-.0971234	.1871792	-.8657318
.7000000	35.8817958	-.0440600	.0979868	-.6538292
.7500000	30.5922200	-.0224898	.0727414	-.5817040
.8000000	27.3627760	-.0119420	.0563681	-.5126662
.9000000	20.9038880	-.0035512	.0340047	-.4181634
.9500000	17.6744440	-.0015462	.0226862	-.3352519
1.0000000	14.4450000	-.0008254	.0176194	-.3235225

C_L = .917731 C_D = 1.898925 X_{CP}/L = .649283 K_E = 2.254638

S_{REF}/S_{PROG} = .920569 C_M/C_L = .093692 C_{M_D} = .171901

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INTERFERENCE DRAG OF LOADING  1 (UNIFORM OR CONSTANT ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .69566615E+00
INTERFERENCE DRAG OF LOADING  2 (  LINEAR CHORDWISE  ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .15511156E+01
INTERFERENCE DRAG OF LOADING  3 (  LINEAR SPANWISE   ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS  -.83655690E-01
INTERFERENCE DRAG OF LOADING  4 ( QUADRATIC SPANWISE ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS  -.32125585E+00
INTERFERENCE DRAG OF LOADING  5 (QUADRATIC CHORDWISE ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .23368009E+01
INTERFERENCE DRAG OF LOADING  7 (  CUBIC CHORDWISE   ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .29348159E+01
INTERFERENCE DRAG OF LOADING  8 (SIMILAR TO FLAT WING) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .55414950E+00
INTERFERENCE DRAG OF LOADING  9 (SQ. ROOT FROM T. E. ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .90853487E+00
INTERFERENCE DRAG OF LOADING 10 ( ELLIPTICAL C-SUB-P ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .79890021E+00
INTERFERENCE DRAG OF LOADING 11 (LINEAR IN ARB.REGION) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .62786249E-01
INTERFERENCE DRAG OF LOADING 12 (BODY UPWASH LOADING ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .15910892E-01
INTERFERENCE DRAG OF LOADING 13 ( NACELLE BUOYANCY   ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS  -.71940765E-03
INTERFERENCE DRAG OF LOADING 14 (NACELLE BUOY(CAMBER)) ON LOADING  6 (PARABOLIC CHORDWISE ) IS  -.71940765E-03
INTERFERENCE DRAG OF LOADING 15 (BODY UPWASH (CAMBER)) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .15910892E-01
INTERFERENCE DRAG OF LOADING 16 ( BODY BUOYANCY TERM ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .24557139E-02
INTERFERENCE DRAG OF LOADING 17 (BODY BUOY. (CAMBER) ) ON LOADING  6 (PARABOLIC CHORDWISE ) IS   .24557139E-02
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WING DATA FOR CUBIC CHORDWISE LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/B/2	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	1.6742763	-1.8104941
.0250000	177.9445264	0.0000000	1.7683932	-1.9714472
.0500000	172.0073528	0.0000000	1.9896429	-2.2781662
.0750000	166.0701792	35.3653518	2.6814571	-3.1256052
.1000000	160.1330000	12.7152031	2.4041490	-2.9304524
.1250000	154.1951859	6.5658502	2.1464063	-2.7395879
.1500000	148.2586941	3.5714341	1.9079038	-2.5535465
.1750000	142.3260032	1.8842747	1.6879747	-2.3726189
.2000000	136.3933123	.8868117	1.4855144	-2.1966251
.2500000	124.6106675	-.0663247	1.1328514	-1.8636165
.3000000	113.9394744	-.2547880	.8660750	-1.5890976
.3500000	103.2682813	-.2772580	.6447524	-1.3309260
.4000000	92.5970882	-.2212113	.4648162	-1.0906226
.4750000	76.6960487	-.1185498	.2641315	-.7695172
.5500000	63.0912977	-.0514031	.1471426	-.5393039
.6250000	49.4865467	-.0185476	.0710040	-.3430296
.7000000	35.8817958	-.0049046	.0270615	-.1862225
.7500000	30.5922200	-.0018927	.0167955	-.1377510
.8000000	27.3627760	-.0008447	.0119949	-.1115649
.9000000	20.9038880	-.0001467	.0053628	-.0670419
.9500000	17.6744440	-.0000544	.0032269	-.0483892
1.0000000	14.4450000	-.0000181	.0018156	-.0337047

C_L = .941350 C_D = 3.238649 $\frac{X_{CP}}{L}$ = .738845 K_E = 3.654782

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = -.135793 C_{M_0} = -.039700

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 7 (CUBIC CHORDWISE) IS .73422736E+00
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 7 (CUBIC CHORDWISE) IS .17607187E+01
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 7 (CUBIC CHORDWISE) IS .93863772E-01
INTERFERENCE DRAG OF LOADING 4 (QUADRATIC SPANWISE) ON LOADING 7 (CUBIC CHORDWISE) IS -.97737164E-01
INTERFERENCE DRAG OF LOADING 5 (QUADRATIC CHORDWISE) ON LOADING 7 (CUBIC CHORDWISE) IS .28631180E+01
INTERFERENCE DRAG OF LOADING 6 (PARABOLIC CHORDWISE) ON LOADING 7 (CUBIC CHORDWISE) IS .14767418E+01
INTERFERENCE DRAG OF LOADING 8 (SIMILAR TO FLAT WING) ON LOADING 7 (CUBIC CHORDWISE) IS .53411507E+00
INTERFERENCE DRAG OF LOADING 9 (SQ. ROOT FROM T. E.) ON LOADING 7 (CUBIC CHORDWISE) IS .75248098E+00
INTERFERENCE DRAG OF LOADING 10 (ELLIPTICAL C-SUB-P) ON LOADING 7 (CUBIC CHORDWISE) IS .81793543E+00
INTERFERENCE DRAG OF LOADING 11 (LINEAR IN ARB.REGION) ON LOADING 7 (CUBIC CHORDWISE) IS .11746090E+00
INTERFERENCE DRAG OF LOADING 12 (BODY UPWASH LOADING) ON LOADING 7 (CUBIC CHORDWISE) IS .14489258E-01
INTERFERENCE DRAG OF LOADING 13 (NACELLE BUOYANCY) ON LOADING 7 (CUBIC CHORDWISE) IS .28542288E-02
INTERFERENCE DRAG OF LOADING 14 (NACELLE BUOY(CAMBER)) ON LOADING 7 (CUBIC CHORDWISE) IS .28542288E-02
INTERFERENCE DRAG OF LOADING 15 (BODY UPWASH (CAMBER)) ON LOADING 7 (CUBIC CHORDWISE) IS .14489258E-01
INTERFERENCE DRAG OF LOADING 16 (BODY BUOYANCY TERM) ON LOADING 7 (CUBIC CHORDWISE) IS .88370895E-02
INTERFERENCE DRAG OF LOADING 17 (BODY BUOY. (CAMBER)) ON LOADING 7 (CUBIC CHORDWISE) IS .88370895E-02

WING DATA FOR SIMILAR TO FLAT WING LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/(B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.7511525	-.6347600
.0250000	177.9445264	0.0000000	.7866845	-.6827532
.0500000	172.0073528	0.0000000	.8458607	-.7473634
.0750000	166.0701792	3.7361336	1.0000451	-.8656535
.1000000	160.1330000	1.7742890	.9961879	-.9175580
.1250000	154.1951859	1.3331175	.9988378	-.9758993
.1500000	148.2586941	1.0168803	1.0007716	-1.0388276
.1750000	142.3260032	.8616729	.9968933	-1.1039064
.2000000	136.3933123	.7536686	.9997471	-1.1785240
.2500000	124.6106675	.5591254	.9978125	-1.3432789
.3000000	113.9394744	.3860239	.9948416	-1.5298483
.3500000	103.2682813	.2351510	1.0021850	-1.7667948
.4000000	92.5970882	.1818981	1.0008105	-2.0472644
.4750000	76.6960487	-.0151047	1.0033199	-2.6200887
.5500000	63.0912977	-.1060708	.9835553	-3.3191919
.6250000	49.4865467	-.1596947	.9835181	-4.4653032
.7000000	35.8817958	-.3177683	.9839211	-6.4831123
.7500000	30.5922200	-.1770755	.9710085	-7.6907138
.8000000	27.3627760	.0609439	.9978718	-8.9798354
.9000000	20.9038880	.2549810	.9677070	-11.8278473
.9500000	17.6744440	.2206933	1.0165444	-14.9184371
1.0000000	14.4450000	-.1545978	.9109414	-16.7074209

C_L = .933996 C_D = .600950 X_{CP}/L = .675634 K_E = .688888

S_{REF}/S_{PROG} = .920569 C_M/C_L = .026172 C_{M0} = .111885

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.66895198E+00
INTERFERENCE DRAG OF LOADING	2 (LINEAR CHORDWISE)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.10010980E+01
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.29189613E+00
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.10375287E+00
INTERFERENCE DRAG OF LOADING	5 (QUADRATIC CHORDWISE)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.12407691E+01
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.12016898E+01
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.14425819E+01
INTERFERENCE DRAG OF LOADING	9 (SQ. ROOT FROM T. E.)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.82715982E+00
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.72627049E+00
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.41371082E-01
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.12856143E-01
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.29457836E-02
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.29457836E-02
INTERFERENCE DRAG OF LOADING	15 (BODY UPWASH (CAMBER))	ON LOADING	8 (SIMILAR TO FLAT WING) IS	.12856143E-01
INTERFERENCE DRAG OF LOADING	16 (BODY BUOYANCY TERM)	ON LOADING	8 (SIMILAR TO FLAT WING) IS	-.16301299E-02
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING	8 (SIMILAR TO FLAT WING) IS	-.16301299E-02

WING DATA FOR SQ. ROOT FROM T. E. LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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	183.8817000	0.0000000	1.0131671	-.8713977
.0250000	177.9445264	0.0000000	1.0608846	-.9370673
.0500000	172.0073528	0.0000000	1.1433267	-1.0265615
.0750000	166.0701792	7.2748445	1.3630046	-1.1985779
.1000000	160.1330000	3.3366139	1.3381735	-1.2487639
.1250000	154.1951859	2.2714838	1.3133294	-1.3005662
.1500000	148.2586941	1.6893780	1.2878869	-1.3549958
.1750000	142.3260032	1.3009306	1.2615795	-1.4126423
.2000000	136.3933123	1.0139926	1.2351348	-1.4727876
.2500000	124.6106675	.5768761	1.1803914	-1.6040857
.3000000	113.9394744	.3325216	1.1287772	-1.7488981
.3500000	103.2682813	.1328174	1.0748411	-1.9105994
.4000000	92.5970882	-.0174514	1.0172594	-2.0948971
.4750000	76.6960487	-.2040746	.9261461	-2.4324628
.5500000	63.0912977	-.2685511	.8385092	-2.8352564
.6250000	49.4865467	-.3159346	.7420212	-3.3734019
.7000000	35.8817958	-.3682630	.6307752	-4.1594252
.7500000	30.5922200	-.2153391	.5844066	-4.6303483
.8000000	27.3627760	-.1155193	.5517096	-4.9707846
.9000000	20.9038880	-.0216555	.4831362	-5.9061697
.9500000	17.6744440	-.0062481	.4402294	-6.4658459
1.0000000	14.4450000	-.0315881	.3973887	-7.2827496

$C_L = .986059$ $C_D = .929121$ $\frac{X_{CP}}{L} = .652844$ $K_E = .955578$

$\frac{S_{REF}}{S_{PROG}} = .920569$ $\frac{C_M}{C_L} = .084568$ $C_{M_0} = .175704$

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.71557952E+00
INTERFERENCE DRAG OF LOADING	2 (LINEAR CHORDWISE)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.12438662E+01
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.13596999E+00
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	-.12756533E+00
INTERFERENCE DRAG OF LOADING	5 (QUADRATIC CHORDWISE)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.16418523E+01
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.15489058E+01
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.19773895E+01
INTERFERENCE DRAG OF LOADING	8 (SIMILAR TO FLAT WING)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.61885110E+00
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.80243527E+00
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.49048708E-01
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.14383407E-01
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.20855132E-02
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.20855132E-02
INTERFERENCE DRAG OF LOADING	15 (BODY UPWASH (CAMBER))	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	.14383407E-01
INTERFERENCE DRAG OF LOADING	16 (BODY BUOYANCY TERM)	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	-.84264670E-03
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING	9 (SQ. ROOT FROM T. E.) IS	-.84264670E-03

WING DATA FOR ELLIPTICAL C-SUB-P LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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	183.8817000	0.0000000	.8080698	-.7459070
.0250000	177.9445264	0.0000000	.8464148	-.8036624
.0500000	172.0073528	0.0000000	.9161870	-.8887469
.0750000	166.0701792	4.0274333	1.0969019	-1.0591997
.1000000	160.1330000	2.0130837	1.0944862	-1.1155383
.1250000	154.1951859	1.4895591	1.0913724	-1.1748181
.1500000	148.2586941	1.2109220	1.0875546	-1.2380487
.1750000	142.3260032	1.0225564	1.0830253	-1.3059686
.2000000	136.3933123	.8809192	1.0777755	-1.3781524
.2500000	124.6106675	.6642529	1.0650704	-1.5392330
.3000000	113.9394744	.5013291	1.0493331	-1.7159663
.3500000	103.2682813	.3671192	1.0304247	-1.9207322
.4000000	92.5970882	.2523484	1.0081667	-2.1636271
.4750000	76.6960487	.1012056	.9679844	-2.6261996
.5500000	63.0912977	-.0200900	.9186811	-3.1850848
.6250000	49.4865467	-.1188605	.8586872	-3.9779808
.7000000	35.8817958	-.2495885	.7855571	-5.2491771
.7500000	30.5922200	-.1327435	.7275816	-5.8252966
.8000000	27.3627760	-.0559525	.6600000	-6.0051554
.9000000	20.9038880	-.0171421	.4794789	-5.9008222
.9500000	17.6744440	-.0202062	.3434749	-5.0785826
1.0000000	14.4450000	0.0000000	0.0000000	0.0000000

$C_L = .919594$ $C_D = .704516$ $\frac{X_{CP}}{L} = .698575$ $K_E = .833103$

$\frac{S_{REF}}{S_{PROG}} = .920569$ $\frac{C_M}{C_L} = -.032610$ $C_{M_0} = .056104$

INTERFERENCE DRAG OF LOADING	1	(UNIFORM OR CONSTANT)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.64299531E+00
INTERFERENCE DRAG OF LOADING	2	(LINEAR CHORDWISE)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.10858583E+01
INTERFERENCE DRAG OF LOADING	3	(LINEAR SPANWISE)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.25876723E+00
INTERFERENCE DRAG OF LOADING	4	(QUADRATIC SPANWISE)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.50978324E-01
INTERFERENCE DRAG OF LOADING	5	(QUADRATIC CHORDWISE)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.14114779E+01
INTERFERENCE DRAG OF LOADING	6	(PARABOLIC CHORDWISE)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.11683273E+01
INTERFERENCE DRAG OF LOADING	7	(CUBIC CHORDWISE)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.15949953E+01
INTERFERENCE DRAG OF LOADING	8	(SIMILAR TO FLAT WING)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.53394344E+00
INTERFERENCE DRAG OF LOADING	9	(SQ. ROOT FROM T. E.)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.74733741E+00
INTERFERENCE DRAG OF LOADING	11	(LINEAR IN ARB.REGION)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.51419306E-01
INTERFERENCE DRAG OF LOADING	12	(BODY UPWASH LOADING)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.12107682E-01
INTERFERENCE DRAG OF LOADING	13	(NACELLE BUOYANCY)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.37141478E-02
INTERFERENCE DRAG OF LOADING	14	(NACELLE BUOY(CAMBER))	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.37141478E-02
INTERFERENCE DRAG OF LOADING	15	(BODY UPWASH (CAMBER))	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	.12107682E-01
INTERFERENCE DRAG OF LOADING	16	(BODY BUOYANCY TERM)	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	-.60321758E-03
INTERFERENCE DRAG OF LOADING	17	(BODY BUOY. (CAMBER))	ON LOADING	10	(ELLIPTICAL C-SUB-P) IS	-.60321758E-03

WING DATA FOR LINEAR IN ARB.REGION LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/B/2	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.0187135	-.0240441
.0250000	177.9445264	0.0000000	.0222419	-.0293964
.0500000	172.0073528	0.0000000	.0330258	-.0448299
.0750000	166.0701792	.0297939	.0616348	-.0862040
.1000000	160.1330000	.0177053	.0592769	-.0860747
.1250000	154.1951859	.0144896	.0569657	-.0859933
.1500000	148.2586941	.0125877	.0548954	-.0862554
.1750000	142.3260032	.0112873	.0527201	-.0863849
.2000000	136.3933123	.0106112	.0504032	-.0862948
.2500000	124.6106675	.0091111	.0461476	-.0866876
.3000000	113.9394744	.0071755	.0417704	-.0863995
.3500000	103.2682813	.0057944	.0379834	-.0873057
.4000000	92.5970882	.0044758	.0344417	-.0888993
.4750000	76.6960487	.0033489	.0283823	-.0893197
.5500000	63.0912977	.0022243	.0233725	-.0911995
.6250000	49.4865467	.0014684	.0186231	-.0944147
.7000000	35.8817958	.0008995	.0140130	-.0998602
.7500000	30.5922200	.0004789	.0105987	-.0894721
.8000000	27.3627760	.0004990	.0100896	-.0963497
.9000000	20.9038880	.0002184	.0080284	-.1021017
.9500000	17.6744440	.0003026	.0077910	-.1190176
1.0000000	14.4450000	.0000724	.0031771	-.0599916

C_L = .036338 C_D = .007826 X_{CP}/L = .867120 K_E = 5.926973

S_{REF}/S_{PROG} = .920569 C_M/C_L = -.464472 C_{M_0} = -.013476

INTERFERENCE DRAG OF LOADING	1	(UNIFORM OR CONSTANT)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.23637590E-01
INTERFERENCE DRAG OF LOADING	2	(LINEAR CHORDWISE)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.52556546E-01
INTERFERENCE DRAG OF LOADING	3	(LINEAR SPANWISE)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.17462539E-01
INTERFERENCE DRAG OF LOADING	4	(QUADRATIC SPANWISE)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.12610489E-01
INTERFERENCE DRAG OF LOADING	5	(QUADRATIC CHORDWISE)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.83848568E-01
INTERFERENCE DRAG OF LOADING	6	(PARABOLIC CHORDWISE)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.10234734E-01
INTERFERENCE DRAG OF LOADING	7	(CUBIC CHORDWISE)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.68675837E-01
INTERFERENCE DRAG OF LOADING	8	(SIMILAR TO FLAT WING)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.15157762E-01
INTERFERENCE DRAG OF LOADING	9	(SQ. ROOT FROM T. E.)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.12692167E-01
INTERFERENCE DRAG OF LOADING	10	(ELLIPTICAL C-SUB-P)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.24741670E-01
INTERFERENCE DRAG OF LOADING	12	(BODY UPWASH LOADING)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.29265031E-03
INTERFERENCE DRAG OF LOADING	13	(NACELLE BUOYANCY)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.52214341E-03
INTERFERENCE DRAG OF LOADING	14	(NACELLE BUOY(CAMBER))	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.52214341E-03
INTERFERENCE DRAG OF LOADING	15	(BODY UPWASH (CAMBER))	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.29265031E-03
INTERFERENCE DRAG OF LOADING	16	(BODY BUOYANCY TERM)	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.33628352E-03
INTERFERENCE DRAG OF LOADING	17	(BODY BUOY. (CAMBER))	ON LOADING	11	(LINEAR IN ARB.REGION) IS	.33628352E-03

WING DATA FOR BODY UPWASH LOADING LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/(B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.0199352	-.0171715
.0250000	177.9445264	0.0000000	.0203094	-.0183712
.0500000	172.0073528	0.0000000	.0217258	-.0203907
.0750000	166.0701792	0.0000000	.0222595	-.0212843
.1000000	160.1330000	0.0000000	.0221333	-.0212475
.1250000	154.1951859	0.0000000	.0216492	-.0216256
.1500000	148.2586941	0.0000000	.0211931	-.0222990
.1750000	142.3260032	0.0000000	.0203227	-.0228787
.2000000	136.3933123	0.0000000	.0200290	-.0239821
.2500000	124.6106675	0.0000000	.0191149	-.0262688
.3000000	113.9394744	0.0000000	.0182813	-.0287738
.3500000	103.2682813	0.0000000	.0182212	-.0326650
.4000000	92.5970882	0.0000000	.0177127	-.0366937
.4750000	76.6960487	0.0000000	.0174484	-.0458375
.5500000	63.0912977	0.0000000	.0171522	-.0580714
.6250000	49.4865467	0.0000000	.0182157	-.0829274
.7000000	35.8817958	0.0000000	.0209287	-.1379502
.7500000	30.5922200	0.0000000	.0218173	-.1728406
.8000000	27.3627760	0.0000000	.0220862	-.1994720
.9000000	20.9038880	0.0000000	.0201745	-.2481318
.9500000	17.6744440	0.0000000	.0173637	-.2569930
1.0000000	14.4450000	0.0000000	.0112932	-.2078505

C_L = .018252 C_D = 0.000000 X_{CP}/L = .684021 K_E = 0.000000

S_{REF}/S_{PROG} = .920569 C_M/C_L = .004682 C_{M0} = .001794

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 4 (QUADRATIC SPANWISE) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 5 (QUADRATIC CHORDWISE) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 6 (PARABOLIC CHORDWISE) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 8 (SIMILAR TO FLAT WING) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 9 (SQ. ROOT FROM T. E.) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 10 (ELLIPTICAL C-SUB-P) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 11 (LINEAR IN ARB.REGION) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 13 (NACELLE BUOYANCY) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 14 (NACELLE BUOY(CAMBER)) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 15 (BODY UPWASH (CAMBER)) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 16 (BODY BUOYANCY TERM) ON LOADING 12 (BODY UPWASH LOADING) IS 0.
INTERFERENCE DRAG OF LOADING 17 (BODY BUOY. (CAMBER)) ON LOADING 12 (BODY UPWASH LOADING) IS 0.

WING DATA FOR NACELLE BUOYANCY LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/B/2	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.0001173	-.0001540
.0250000	177.9445264	0.0000000	.0004694	-.0006370
.0500000	172.0073528	0.0000000	.0009759	-.0013649
.0750000	166.0701792	0.0000000	.0017408	-.0024903
.1000000	160.1330000	0.0000000	.0020751	-.0030665
.1250000	154.1951859	0.0000000	.0029497	-.0044669
.1500000	148.2586941	0.0000000	.0032910	-.0051617
.1750000	142.3260032	0.0000000	.0038770	-.0062537
.2000000	136.3933123	0.0000000	.0045904	-.0076394
.2500000	124.6106675	0.0000000	.0054555	-.0098394
.3000000	113.9394744	0.0000000	.0072213	-.0145887
.3500000	103.2682813	0.0000000	.0088303	-.0200926
.4000000	92.5970882	0.0000000	.0105369	-.0267138
.4750000	76.6960487	0.0000000	.0093157	-.0277860
.5500000	63.0912977	0.0000000	.0096136	-.0355341
.6250000	49.4865467	0.0000000	.0092359	-.0454217
.7000000	35.8817958	0.0000000	.0081753	-.0574132
.7500000	30.5922200	0.0000000	.0043955	-.0370185
.8000000	27.3627760	0.0000000	.0003417	-.0032440
.9000000	20.9038880	0.0000000	0.0000000	0.0000000
.9500000	17.6744440	0.0000000	0.0000000	0.0000000
1.0000000	14.4450000	0.0000000	0.0000000	0.0000000

C_L = .005585 C_D = 0.000000 $\frac{X_{CP}}{L}$ = .857603 K_E = 0.000000

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = -.440087 C_{M_0} = -.001935

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 4 (QUADRATIC SPANWISE) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 5 (QUADRATIC CHORDWISE) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 6 (PARABOLIC CHORDWISE) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 8 (SIMILAR TO FLAT WING) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 9 (SQ. ROOT FROM T. E.) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 10 (ELLIPTICAL C-SUB-P) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 11 (LINEAR IN ARB.REGION) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 12 (BODY UPWASH LOADING) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 14 (NACELLE BUOY(CAMBER)) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 15 (BODY UPWASH (CAMBER)) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 16 (BODY BUOYANCY TERM) ON LOADING 13 (NACELLE BUOYANCY) IS 0.
INTERFERENCE DRAG OF LOADING 17 (BODY BUOY. (CAMBER)) ON LOADING 13 (NACELLE BUOYANCY) IS 0.

WING DATA FOR NACELLE BUOY(CAMBER) LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/(B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.0002720	-.0003518
.0250000	177.9445264	0.0000000	.0004138	-.0005529
.0500000	172.0073528	0.0000000	.0009311	-.0012872
.0750000	166.0701792	.0000244	.0017408	-.0024903
.1000000	160.1330000	.0000020	.0020751	-.0030665
.1250000	154.1951859	.0000473	.0029497	-.0044669
.1500000	148.2586941	.0000281	.0032910	-.0051617
.1750000	142.3260032	.0000616	.0038770	-.0062537
.2000000	136.3933123	.0001236	.0045904	-.0076394
.2500000	124.6106675	.0002436	.0054555	-.0098394
.3000000	113.9394744	.0002085	.0072213	-.0145887
.3500000	103.2682813	.0001189	.0088303	-.0200926
.4000000	92.5970882	.0002700	.0105369	-.0267138
.4750000	76.6960487	.0003084	.0093157	-.0277860
.5500000	63.0912977	.0002672	.0096136	-.0355341
.6250000	49.4865467	.0000543	.0092359	-.0454217
.7000000	35.8817958	.0000691	.0081753	-.0574132
.7500000	30.5922200	.0000369	.0043955	-.0370185
.8000000	27.3627760	.0000009	.0003417	-.0032440
.9000000	20.9038880	0.0000000	0.0000000	0.0000000
.9500000	17.6744440	0.0000000	0.0000000	0.0000000
1.0000000	14.4450000	0.0000000	0.0000000	0.0000000

C_L = .005585 C_D = .000128 X_{CP}/L = .857603 K_E = 4.112798

S_{REF}/S_{PROG} = .920569 C_M/C_L = -.440087 C_{M0} = -.001935

INTERFERENCE DRAG OF LOADING	1	(UNIFORM OR CONSTANT)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.37395446E-02
INTERFERENCE DRAG OF LOADING	2	(LINEAR CHORDWISE)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.59821625E-02
INTERFERENCE DRAG OF LOADING	3	(LINEAR SPANWISE)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.42389670E-02
INTERFERENCE DRAG OF LOADING	4	(QUADRATIC SPANWISE)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.37495071E-02
INTERFERENCE DRAG OF LOADING	5	(QUADRATIC CHORDWISE)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.72949422E-02
INTERFERENCE DRAG OF LOADING	6	(PARABOLIC CHORDWISE)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.17314144E-02
INTERFERENCE DRAG OF LOADING	7	(CUBIC CHORDWISE)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.53332648E-02
INTERFERENCE DRAG OF LOADING	8	(SIMILAR TO FLAT WING)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.25098526E-02
INTERFERENCE DRAG OF LOADING	9	(SQ. ROOT FROM T. E.)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.22607846E-02
INTERFERENCE DRAG OF LOADING	10	(ELLIPTICAL C-SUB-P)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.37418132E-02
INTERFERENCE DRAG OF LOADING	11	(LINEAR IN ARB.REGION)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.57341500E-03
INTERFERENCE DRAG OF LOADING	12	(BODY UPWASH LOADING)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.52252367E-04
INTERFERENCE DRAG OF LOADING	13	(NACELLE BUOYANCY)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.12827839E-03
INTERFERENCE DRAG OF LOADING	15	(BODY UPWASH (CAMBER))	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.52252367E-04
INTERFERENCE DRAG OF LOADING	16	(BODY BUOYANCY TERM)	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.94075333E-05
INTERFERENCE DRAG OF LOADING	17	(BODY BUOY. (CAMBER))	ON LOADING	14	(NACELLE BUOY(CAMBER)) IS	.94075333E-05

WING DATA FOR BODY UPWASH (CAMBER) LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS.

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	183.8817000	0.0000000	.0165823	-.0150016
.0250000	177.9445264	0.0000000	.0173935	-.0161667
.0500000	172.0073528	0.0000000	.0186505	-.0177343
.0750000	166.0701792	.0012257	.0222595	-.0212843
.1000000	160.1330000	.0012041	.0221333	-.0212475
.1250000	154.1951859	.0008419	.0216492	-.0216256
.1500000	148.2586941	.0006261	.0211931	-.0222990
.1750000	142.3260032	.0003266	.0203227	-.0228787
.2000000	136.3933123	.0003031	.0200290	-.0239821
.2500000	124.6106675	.0001082	.0191149	-.0262688
.3000000	113.9394744	.0000016	.0182813	-.0287738
.3500000	103.2682813	.0000806	.0182212	-.0326650
.4000000	92.5970882	.0000677	.0177127	-.0366937
.4750000	76.6960487	.0000285	.0174484	-.0458375
.5500000	63.0912977	-.0000430	.0171522	-.0580714
.6250000	49.4865467	-.0000590	.0182157	-.0829274
.7000000	35.8817958	-.0000608	.0209287	-.1379502
.7500000	30.5922200	-.0000107	.0218173	-.1728406
.8000000	27.3627760	.0000405	.0220862	-.1994720
.9000000	20.9038880	.0000421	.0201745	-.2481318
.9500000	17.6744440	.0000266	.0173637	-.2569930
1.0000000	14.4450000	-.0000333	.0112932	-.2078505

C_L = .018252 C_D = .000260 $\frac{X_{CP}}{L}$ = .684021 K_E = .780685

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = .004682 C_{M_D} = .001794

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.13207657E-01
INTERFERENCE DRAG OF LOADING	2 (LINEAR CHORDWISE)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.20614524E-01
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.57260391E-02
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.22282178E-02
INTERFERENCE DRAG OF LOADING	5 (QUADRATIC CHORDWISE)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.26842756E-01
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.23686154E-01
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.31360523E-01
INTERFERENCE DRAG OF LOADING	8 (SIMILAR TO FLAT WING)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.11730779E-01
INTERFERENCE DRAG OF LOADING	9 (SQ. ROOT FROM T. E.)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.16014873E-01
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.14316308E-01
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.86849194E-03
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.26006707E-03
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.49983912E-04
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	.49983912E-04
INTERFERENCE DRAG OF LOADING	16 (BODY BUOYANCY TERM)	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	-.13606499E-04
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING 15 (BODY UPWASH (CAMBER)) IS	-.13606499E-04

WING DATA FOR BODY BUOYANCY TERM LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

$Y/(B/2)$	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	.0012633	-.0027446
.0250000	177.9445264	0.0000000	.0012696	-.0028101
.0500000	172.0073528	0.0000000	.0012584	-.0028545
.0750000	166.0701792	0.0000000	.0012441	-.0029140
.1000000	160.1330000	0.0000000	-.0000400	-.0023109
.1250000	154.1951859	0.0000000	-.0006373	-.0015437
.1500000	148.2586941	0.0000000	-.0013829	-.0004346
.1750000	142.3260032	0.0000000	-.0019972	.0005587
.2000000	136.3933123	0.0000000	-.0026088	.0017498
.2500000	124.6106675	0.0000000	-.0045431	.0052926
.3000000	113.9394744	0.0000000	-.0059995	.0087494
.3500000	103.2682813	0.0000000	-.0071245	.0125372
.4000000	92.5970882	0.0000000	-.0084617	.0176057
.4750000	76.6960487	0.0000000	-.0103943	.0282688
.5500000	63.0912977	0.0000000	-.0112420	.0393547
.6250000	49.4865467	0.0000000	-.0105717	.0491104
.7000000	35.8817958	0.0000000	-.0096770	.0645111
.7500000	30.5922200	0.0000000	-.0095236	.0761343
.8000000	27.3627760	0.0000000	-.0093318	.0848577
.9000000	20.9038880	0.0000000	-.0083788	.1034734
.9500000	17.6744440	0.0000000	-.0066879	.0994734
1.0000000	14.4450000	0.0000000	-.0053462	.0989333

C_L = -.005353 C_D = 0.000000 X_{CP}/L = .687736 K_E = 0.000000

S_{REF}/S_{PROG} = .920569 C_M/C_L = -.004838 C_{M0} = -.000475

INTERFERENCE DRAG OF LOADING	1 (UNIFORM OR CONSTANT)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	2 (LINEAR CHORDWISE)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	3 (LINEAR SPANWISE)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	4 (QUADRATIC SPANWISE)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	5 (QUADRATIC CHORDWISE)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	6 (PARABOLIC CHORDWISE)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	7 (CUBIC CHORDWISE)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	8 (SIMILAR TO FLAT WING)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	9 (SQ. ROOT FROM T. E.)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	10 (ELLIPTICAL C-SUB-P)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	11 (LINEAR IN ARB.REGION)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	12 (BODY UPWASH LOADING)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	13 (NACELLE BUOYANCY)	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	14 (NACELLE BUOY(CAMBER))	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	15 (BODY UPWASH (CAMBER))	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.
INTERFERENCE DRAG OF LOADING	17 (BODY BUOY. (CAMBER))	ON LOADING	16 (BODY BUOYANCY TERM)	IS	0.

WING DATA FOR BODY BUOY. (CAMBER) LOADING

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y/(B/2)	CHORD	SECTION C_D	SECTION C_L	SECTION C_M
0.0000000	183.8817000	0.0000000	-.0010249	-.0001839
.0250000	177.9445264	0.0000000	-.0008688	-.0003048
.0500000	172.0073528	0.0000000	-.0002469	-.0009537
.0750000	166.0701792	.0005866	.0012441	-.0029140
.1000000	160.1330000	.0005146	-.0000400	-.0023109
.1250000	154.1951859	.0002164	-.0006373	-.0015437
.1500000	148.2586941	.0001188	-.0013829	-.0004346
.1750000	142.3260032	.0001143	-.0019972	.0005587
.2000000	136.3933123	.0001008	-.0026088	.0017498
.2500000	124.6106675	.0000502	-.0045431	.0052926
.3000000	113.9394744	.0000442	-.0059995	.0087494
.3500000	103.2682813	.0000425	-.0071245	.0125372
.4000000	92.5970882	.0000489	-.0084617	.0176057
.4750000	76.6960487	.0000450	-.0103943	.0282688
.5500000	63.0912977	.0000269	-.0112420	.0393547
.6250000	49.4865467	.0000108	-.0105717	.0491104
.7000000	35.8817958	-.0000128	-.0096770	.0645111
.7500000	30.5922200	.0000004	-.0095236	.0761343
.8000000	27.3627760	.0000162	-.0093318	.0848577
.9000000	20.9038880	.0000176	-.0083788	.1034734
.9500000	17.6744440	.0000120	-.0066879	.0994734
1.0000000	14.4450000	.0000064	-.0053462	.0989333

C_L = -.005353 C_D = .000110 X_{CP}/L = .687736 K_E = 3.845320

S_{REF}/S_{PROG} = .920569 C_M/C_L = -.004838 C_{M0} = -.000475

INTERFERENCE DRAG OF LOADING 1 (UNIFORM OR CONSTANT) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.26418061E-02
INTERFERENCE DRAG OF LOADING 2 (LINEAR CHORDWISE) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.16409844E-02
INTERFERENCE DRAG OF LOADING 3 (LINEAR SPANWISE) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.28823449E-02
INTERFERENCE DRAG OF LOADING 4 (QUADRATIC SPANWISE) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.23857417E-02
INTERFERENCE DRAG OF LOADING 5 (QUADRATIC CHORDWISE) ON LOADING 17 (BODY BUOY. (CAMBER)) IS .12895546E-02
INTERFERENCE DRAG OF LOADING 6 (PARABOLIC CHORDWISE) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.60267578E-02
INTERFERENCE DRAG OF LOADING 7 (CUBIC CHORDWISE) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.37045677E-03
INTERFERENCE DRAG OF LOADING 8 (SIMILAR TO FLAT WING) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.23457428E-02
INTERFERENCE DRAG OF LOADING 9 (SQ. ROOT FROM T. E.) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.34717632E-02
INTERFERENCE DRAG OF LOADING 10 (ELLIPTICAL C-SUB-P) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.26583267E-02
INTERFERENCE DRAG OF LOADING 11 (LINEAR IN ARB.REGION) ON LOADING 17 (BODY BUOY. (CAMBER)) IS .31926719E-03
INTERFERENCE DRAG OF LOADING 12 (BODY UPWASH LOADING) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.57886323E-04
INTERFERENCE DRAG OF LOADING 13 (NACELLE BUOYANCY) ON LOADING 17 (BODY BUOY. (CAMBER)) IS .16865586E-05
INTERFERENCE DRAG OF LOADING 14 (NACELLE BUOY(CAMBER)) ON LOADING 17 (BODY BUOY. (CAMBER)) IS .16865586E-05
INTERFERENCE DRAG OF LOADING 15 (BODY UPWASH (CAMBER)) ON LOADING 17 (BODY BUOY. (CAMBER)) IS -.57886323E-04
INTERFERENCE DRAG OF LOADING 16 (BODY BUOYANCY TERM) ON LOADING 17 (BODY BUOY. (CAMBER)) IS .11019121E-03

FORCE COEFFICIENTS OF COMPONENT AND INTERFERENCE LOADINGS

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

GROSS WING AREA = 10752.043141 SREF/SPROG = .920569

CL 1 =	.938399	FOR UNIFORM OR CONSTANT	LOADING
CL 2 =	.908181	FOR LINEAR CHORDWISE	LOADING
CL 3 =	1.020416	FOR LINEAR SPANWISE	LOADING
CL 4 =	1.033028	FOR QUADRATIC SPANWISE	LOADING
CL 5 =	.917149	FOR QUADRATIC CHORDWISE	LOADING
CL 6 =	.917731	FOR PARABOLIC CHORDWISE	LOADING
CL 7 =	.941350	FOR CUBIC CHORDWISE	LOADING
CL 8 =	.933996	FOR SIMILAR TO FLAT WING	LOADING
CL 9 =	.986059	FOR SQ. ROOT FROM T. E.	LOADING
CL10 =	.919594	FOR ELLIPTICAL C-SUB-P	LOADING
CL11 =	.036338	FOR LINEAR IN ARB.REGION	LOADING
CL12 =	.018252	FOR BODY UPWASH LOADING	LOADING
CL13 =	.005585	FOR NACELLE BUOYANCY	LOADING
CL14 =	.005585	FOR NACELLE BUOY(CAMBER)	LOADING
CL15 =	.018252	FOR BODY UPWASH (CAMBER)	LOADING
CL16 =	-.005353	FOR BODY BUOYANCY TERM	LOADING
CL17 =	-.005353	FOR BODY BUOY. (CAMBER)	LOADING

C-M-O 1 =	.019612
C-M-O 2 =	-.056583
C-M-O 3 =	-.147457
C-M-O 4 =	-.282881
C-M-O 5 =	-.115619
C-M-O 6 =	.171901
C-M-O 7 =	-.039700
C-M-O 8 =	.111885
C-M-O 9 =	.175704
C-M-O10 =	.056104
C-M-O11 =	-.013476
C-M-O12 =	.001794
C-M-O13 =	-.001935
C-M-O14 =	-.001935
C-M-O15 =	.001794
C-M-O16 =	-.000475
C-M-O17 =	-.000475

CD 1 1/(CL 1)(CL 1) =	.734870
CD 2 2/(CL 2)(CL 2) =	1.598735
CD 3 3/(CL 3)(CL 3) =	.771391
CD 4 4/(CL 4)(CL 4) =	1.299544
CD 5 5/(CL 5)(CL 5) =	2.905185
CD 6 6/(CL 6)(CL 6) =	2.254638
CD 7 7/(CL 7)(CL 7) =	3.654782

CD 8 8/(CL 8)(CL 8) =	.688888
CD 9 9/(CL 9)(CL 9) =	.955578
CD1010/(CL10)(CL10) =	.833103
CD1111/(CL11)(CL11) =	5.926973
CD1212/(CL12)(CL12) =	0.000000
CD1313/(CL13)(CL13) =	0.000000
CD1414/(CL14)(CL14) =	4.112798
CD1515/(CL15)(CL15) =	.780685
CD1616/(CL16)(CL16) =	0.000000
CD1717/(CL17)(CL17) =	3.845320

```
(CD 1 2+CD 2 1)/(CL 1)(CL 2) =     1.955786
(CD 1 3+CD 3 1)/(CL 1)(CL 3) =     1.030882
(CD 1 4+CD 4 1)/(CL 1)(CL 4) =      .856666
(CD 1 5+CD 5 1)/(CL 1)(CL 5) =     2.297994
(CD 1 6+CD 6 1)/(CL 1)(CL 6) =     2.068573
(CD 1 7+CD 7 1)/(CL 1)(CL 7) =     2.479876
(CD 1 8+CD 8 1)/(CL 1)(CL 8) =     1.380664
(CD 1 9+CD 9 1)/(CL 1)(CL 9) =     1.557544
(CD 110+CD10 1)/(CL 1)(CL10) =     1.545274
(CD 111+CD11 1)/(CL 1)(CL11) =     2.098149
(CD 112+CD12 1)/(CL 1)(CL12) =      .703440
(CD 113+CD13 1)/(CL 1)(CL13) =      .742224
(CD 114+CD14 1)/(CL 1)(CL14) =     1.455772
(CD 115+CD15 1)/(CL 1)(CL15) =     1.474581
(CD 116+CD16 1)/(CL 1)(CL16) =      .242672
(CD 117+CD17 1)/(CL 1)(CL17) =      .768576
(CD 2 3+CD 3 2)/(CL 2)(CL 3) =      .916458
(CD 2 4+CD 4 2)/(CL 2)(CL 4) =      .486254
(CD 2 5+CD 5 2)/(CL 2)(CL 5) =     4.176316
(CD 2 6+CD 6 2)/(CL 2)(CL 6) =     3.202498
(CD 2 7+CD 7 2)/(CL 2)(CL 7) =     4.517035
(CD 2 8+CD 8 2)/(CL 2)(CL 8) =     1.744246
(CD 2 9+CD 9 2)/(CL 2)(CL 9) =     2.109941
(CD 210+CD10 2)/(CL 2)(CL10) =     2.141035
(CD 211+CD11 2)/(CL 2)(CL11) =     4.106713
(CD 212+CD12 2)/(CL 2)(CL12) =      .718483
(CD 213+CD13 2)/(CL 2)(CL13) =      .932539
(CD 214+CD14 2)/(CL 2)(CL14) =     2.111984
(CD 215+CD15 2)/(CL 2)(CL15) =     1.962127
(CD 216+CD16 2)/(CL 2)(CL16) =     -.734548
(CD 217+CD17 2)/(CL 2)(CL17) =     -.397008
(CD 3 4+CD 4 3)/(CL 3)(CL 4) =     1.826574
(CD 3 5+CD 5 3)/(CL 3)(CL 5) =      .758597
(CD 3 6+CD 6 3)/(CL 3)(CL 6) =      .481282
(CD 3 7+CD 7 3)/(CL 3)(CL 7) =      .522459
(CD 3 8+CD 8 3)/(CL 3)(CL 8) =      .885007
(CD 3 9+CD 9 3)/(CL 3)(CL 9) =      .720350
(CD 310+CD10 3)/(CL 3)(CL10) =      .923541

(CD 311+CD11 3)/(CL 3)(CL11) =     1.087002
(CD 312+CD12 3)/(CL 3)(CL12) =      .560685
(CD 313+CD13 3)/(CL 3)(CL13) =     1.054161
(CD 314+CD14 3)/(CL 3)(CL14) =     1.797993
(CD 315+CD15 3)/(CL 3)(CL15) =      .868134
(CD 316+CD16 3)/(CL 3)(CL16) =     1.043194
(CD 317+CD17 3)/(CL 3)(CL17) =     1.570863
(CD 4 5+CD 5 4)/(CL 4)(CL 5) =      .234655
(CD 4 6+CD 6 4)/(CL 4)(CL 6) =     -.054440
(CD 4 7+CD 7 4)/(CL 4)(CL 7) =     -.022089
(CD 4 8+CD 8 4)/(CL 4)(CL 8) =      .683763
(CD 4 9+CD 9 4)/(CL 4)(CL 9) =      .352524
(CD 410+CD10 4)/(CL 4)(CL10) =      .612361
(CD 411+CD11 4)/(CL 4)(CL11) =      .672512
(CD 412+CD12 4)/(CL 4)(CL12) =      .548205
(CD 413+CD13 4)/(CL 4)(CL13) =      .985951
(CD 414+CD14 4)/(CL 4)(CL14) =     1.635863
(CD 415+CD15 4)/(CL 4)(CL15) =      .666384
(CD 416+CD16 4)/(CL 4)(CL16) =     1.248766
(CD 417+CD17 4)/(CL 4)(CL17) =     1.680190
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(CD 5 6+CD 6 5)/(CL 5)(CL 6) =     4.067942
(CD 5 7+CD 7 5)/(CL 5)(CL 7) =     6.311882
(CD 5 8+CD 8 5)/(CL 5)(CL 8) =     2.011914
(CD 5 9+CD 9 5)/(CL 5)(CL 9) =     2.503859
(CD 510+CD10 5)/(CL 5)(CL10) =     2.554688
(CD 511+CD11 5)/(CL 5)(CL11) =     6.060975
(CD 512+CD12 5)/(CL 5)(CL12) =      .720270
(CD 513+CD13 5)/(CL 5)(CL13) =     1.091748
(CD 514+CD14 5)/(CL 5)(CL14) =     2.515958
(CD 515+CD15 5)/(CL 5)(CL15) =     2.323820
(CD 516+CD16 5)/(CL 5)(CL16) =    -1.598270
(CD 517+CD17 5)/(CL 5)(CL17) =    -1.860929
(CD 6 7+CD 7 6)/(CL 6)(CL 7) =     5.106523
(CD 6 8+CD 8 6)/(CL 6)(CL 8) =     2.048445
(CD 6 9+CD 9 6)/(CL 6)(CL 9) =     2.715592
(CD 610+CD10 6)/(CL 6)(CL10) =     2.331005
(CD 611+CD11 6)/(CL 6)(CL11) =     2.189628
(CD 612+CD12 6)/(CL 6)(CL12) =      .949892
(CD 613+CD13 6)/(CL 6)(CL13) =     -.140363
(CD 614+CD14 6)/(CL 6)(CL14) =      .197451
(CD 615+CD15 6)/(CL 6)(CL15) =     2.363973
(CD 616+CD16 6)/(CL 6)(CL16) =     -.499868
(CD 617+CD17 6)/(CL 6)(CL17) =      .727303
(CD 7 8+CD 8 7)/(CL 7)(CL 8) =     2.248246
(CD 7 9+CD 9 7)/(CL 7)(CL 9) =     2.940951
(CD 710+CD10 7)/(CL 7)(CL10) =     2.787390
(CD 711+CD11 7)/(CL 7)(CL11) =     5.441504
(CD 712+CD12 7)/(CL 7)(CL12) =      .843316
(CD 713+CD13 7)/(CL 7)(CL13) =      .542912

(CD 714+CD14 7)/(CL 7)(CL14) =     1.557370
(CD 715+CD15 7)/(CL 7)(CL15) =     2.668586
(CD 716+CD16 7)/(CL 7)(CL16) =    -1.753682
(CD 717+CD17 7)/(CL 7)(CL17) =    -1.680166
(CD 8 9+CD 9 8)/(CL 8)(CL 9) =     1.570087
(CD 810+CD10 8)/(CL 8)(CL10) =     1.467247
(CD 811+CD11 8)/(CL 8)(CL11) =     1.665571
(CD 812+CD12 8)/(CL 8)(CL12) =      .754155
(CD 813+CD13 8)/(CL 8)(CL13) =      .564739
(CD 814+CD14 8)/(CL 8)(CL14) =     1.045905
(CD 815+CD15 8)/(CL 8)(CL15) =     1.442296
(CD 816+CD16 8)/(CL 8)(CL16) =      .326039
(CD 817+CD17 8)/(CL 8)(CL17) =      .795207
(CD 910+CD10 9)/(CL 9)(CL10) =     1.709106
(CD 911+CD11 9)/(CL 9)(CL11) =     1.723090
(CD 912+CD12 9)/(CL 9)(CL12) =      .799197
(CD 913+CD13 9)/(CL 9)(CL13) =      .378706
(CD 914+CD14 9)/(CL 9)(CL14) =      .789239
(CD 915+CD15 9)/(CL 9)(CL15) =     1.689045
(CD 916+CD16 9)/(CL 9)(CL16) =      .159638
(CD 917+CD17 9)/(CL 9)(CL17) =      .817356
(CD1011+CD1110)/(CL10)(CL11) =     2.279159
(CD1012+CD1210)/(CL10)(CL12) =      .721374
(CD1013+CD1310)/(CL10)(CL13) =      .723195
(CD1014+CD1410)/(CL10)(CL14) =     1.451776
(CD1015+CD1510)/(CL10)(CL15) =     1.574337
(CD1016+CD1610)/(CL10)(CL16) =      .122538
(CD1017+CD1710)/(CL10)(CL17) =      .662552
(CD1112+CD1211)/(CL11)(CL12) =      .441247
(CD1113+CD1311)/(CL11)(CL13) =     2.572882
```

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(CD1114+CD1411)/(CL11)(CL14) =      5.398406
(CD1115+CD1511)/(CL11)(CL15) =      1.750728
(CD1116+CD1611)/(CL11)(CL16) =     -1.728767
(CD1117+CD1711)/(CL11)(CL17) =     -3.370056
(CD1213+CD1312)/(CL12)(CL13) =      0.000000
(CD1214+CD1412)/(CL12)(CL14) =       .512617
(CD1215+CD1512)/(CL12)(CL15) =       .780685
(CD1216+CD1612)/(CL12)(CL16) =      0.000000
(CD1217+CD1712)/(CL12)(CL17) =       .592467
(CD1314+CD1413)/(CL13)(CL14) =      4.112798
(CD1315+CD1513)/(CL13)(CL15) =       .490363
(CD1316+CD1613)/(CL13)(CL16) =      0.000000
(CD1317+CD1713)/(CL13)(CL17) =      -.056414
(CD1415+CD1514)/(CL14)(CL15) =      1.002980
(CD1416+CD1614)/(CL14)(CL16) =      -.314674
(CD1417+CD1714)/(CL14)(CL17) =      -.371088
(CD1516+CD1615)/(CL15)(CL16) =       .139263
(CD1517+CD1715)/(CL15)(CL17) =       .731729
(CD1617+CD1716)/(CL16)(CL17) =      3.845320


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 =        .001882
CD WING-LIFT-ON-NACELLES  5 =        .001639
CD WING-LIFT-ON-NACELLES  6 =        .002983
CD WING-LIFT-ON-NACELLES  7 =        .002105
CD WING-LIFT-ON-NACELLES  8 =        .001814
CD WING-LIFT-ON-NACELLES  9 =        .002427
CD WING-LIFT-ON-NACELLES 10 =        .002254
CD WING-LIFT-ON-NACELLES 11 =       -.000199
CD WING-LIFT-ON-NACELLES 12 =        .000037
CD WING-LIFT-ON-NACELLES 13 =       0.000000
CD WING-LIFT-ON-NACELLES 14 =       -.000027
CD WING-LIFT-ON-NACELLES 15 =        .000037
CD WING-LIFT-ON-NACELLES 16 =       0.000000
CD WING-LIFT-ON-NACELLES 17 =       -.000022
```

BODY TERMS

C-L = -.00000 C-D = .000001 C-M = .003958 XF = .722386 ALPHA = 0.0000 C-M-0 = .0040

LIFT, DRAG, AND MOMENT INCREMENTS
DUE TO BODY CARRY-OVER OF WING LIFT

I	TYPE OF WING LIFT LOADING	C L	C D	C M 0	I
1	UNIFORM OR CONSTANT	.118088	.010218	.032205	1
2	LINEAR CHORDWISE	.165429	.014306	.006463	2
3	LINEAR SPANWISE	.038359	.003319	.008244	3
4	QUADRATIC SPANWISE	.010797	.000934	.001281	4
5	QUADRATIC CHORDWISE	.207608	.017928	-.016465	5
6	PARABOLIC CHORDWISE	.238510	.020680	.060842	6
7	CUBIC CHORDWISE	.271410	.023470	-.003052	7
8	SIMILAR TO FLAT WING	.120023	.010382	.049734	8
9	SQ. ROOT FROM T. E.	.161934	.014018	.062718	9
10	ELLIPTICAL C-SUB-P	.128894	.011153	.035309	10
11	LINEAR IN ARB.REGION	.003426	.000294	-.001259	11
12	BODY UPWASH LOADING	.003074	.000266	.001050	12
13	NACELLE BUOYANCY	.000086	.000007	-.000035	13
14	NACELLE BUOY(CAMBER)	.000053	.000005	-.000022	14
15	BODY UPWASH (CAMBER)	.002659	.000230	.000820	15
16	BODY BUOYANCY TERM	.000182	.000016	-.000350	16
17	BODY BUOY. (CAMBER)	-.000069	-.000007	-.000295	17

DELTAT = .422 SEC., T = 2082.401 SEC.

RESTART DATA PUNCHED, DECK IMAGE FOLLOWS.

```
969-500  17 LOAD CHECK CASE  22 SPAN STA.  WITH FUSELAGE AND Z TERMS     RESTART
  21  20  12  41  20  25  11  21  19  17  22  46  45  43  42  40  39  37  35  34
  31  29  27  24  20  16  13  10   9   8   6   6   5   5   1   2   3   4   5   6
   7   8   9  10  11  16  17  14  13  15  12
  .1191438946281E+01  .1534397023106E+01  .9087199921286E+00  .7644823748055E+00
  .1820676006052E+01  .1639948897481E+01  .2016624882681E+01  .1113978786573E+01
  .1326744771816E+01  .1227567816936E+01  .6586312236193E-01  .1109113885135E-01
  .3580858154972E-02  .7023367855027E-02  .2324970154520E-01 -.1122202583347E-02
 -.3554167929359E-02  .1534397023106E+01  .2427771011671E+01  .7818414627068E+00
  .4199559911423E+00  .3202299855232E+01  .2457158862649E+01  .3554950491973E+01
  .1362013949550E+01  .1739411186432E+01  .1646070902093E+01  .1247628099813E+00
  .1096352327210E-01  .4354154169237E-02  .9861148813370E-02  .2994061940195E-01
  .3287426509946E-02  .1776786738630E-02  .9087199921286E+00  .7818414627068E+00
  .1478819747834E+01  .1772485513297E+01  .6535586609634E+00  .4149047863636E+00
  .4619943611736E+00  .7764707785822E+00  .6672366652443E+00  .7977846344839E+00
  .3710443961398E-01  .9612965507468E-02  .5530298196860E-02  .9432560756286E-02
  .1488418085144E-01 -.5245728207967E-02 -.7899126215912E-02  .7644823748055E+00

  .4199559911423E+00  .1772485513297E+01  .2553297933167E+01  .2046627129120E+00
 -.4751175313797E-01 -.1977447936120E-01  .6073218385443E+00  .3305673019399E+00
  .5355154876602E+00  .2323968421968E-01  .9515158406377E-02  .5236389929867E-02
  .8688070732743E-02  .1156638715958E-01 -.6357065106674E-02 -.8553305482497E-02
  .1820676006052E+01  .3202299855232E+01  .6535586609634E+00  .2046627129120E+00
  .4499247395492E+01  .3152001624528E+01  .5016563758293E+01  .1586538597773E+01
  .2084534927924E+01  .1983490600667E+01  .1859519365264E+00  .1109931845477E-01
  .5147861220275E-02  .1186336045607E-01  .3580993308925E-01  .7223598429944E-02
  .8410722727820E-02  .1639948897481E+01  .2457158862649E+01  .4149047863636E+00
 -.4751175313797E-01  .3152001624528E+01  .3496183138605E+01  .4061144276037E+01
  .1616371580898E+01  .2262244205410E+01  .1810969126480E+01  .6722086987816E-01
```

* *

TEA253, 17 LOADING VERSION OF OCTOBER 30, 1975.

OPTIMUM COMBINATION OF 17 WING LOADINGS

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

* *

NUMBER OF PLANFORM BREAKPOINTS =	9.0	FLAT PLATE CONTROL FLAG =	-1.0
NUMBER OF SEMISPAN ELEMENTS =	40.0	PRINT FLAG =	2.0
NUMBER OF SPAN STATIONS FOR CAMBER SURFACE =	22.0	SMOOTHING FLAG =	1.0
SPAN STATION FOR PARABOLIC APEX =	-0.0	RESTART FLAG =	2.0
BASIC MACH NUMBER =	2.7000	DESIGN C-L =	.1000
CBAR =	100.4100	NUMBER OF LOADINGS =	-17.0000
PITCHING MOMENT CENTER AT	187.0000	NUMBER OF CAMBER ORDINATES =	12.0000
REFERENCE AREA =	9898.0000	NUMBER OF POINTS DEFINING ARBITRARY REGION =	2.0000
C-M-O CONSTRAINT =	.0100	FUSELAGE ALPHA =	0.0000
SPAN STATION FOR SIDE-OF-BODY =	4.9688	NUMBER OF BODY CAMBER ORDINATES =	19.0000

NUMBER OF CHORDWISE AND SPANWISE LOCATIONS FOR

BODY BUOYANCY TABLES =	0.0	21.0
BODY UPWASH LOADING TABLE =	0.0	0.0
NACELLE BUOYANCY LOADING TABLES =	0.0	0.0
WING UPPER SURFACE LIMITING PRESSURES =	2.0	2.0
WING THICKNESS PRESSURES =	-21.0	0.0

CAMBER SURFACE OPTION FLAGS = 1.0 1.0 1.0 3.0

4 CONSTRAINTS ARE APPLIED ON ORDINATE

CONSTRAINT LOCATIONS

I	X(I)	Y(I)	Z(I)
1	130.850000	4.968800	-4.070000
2	189.000000	4.968800	-10.160000
3	243.390000	4.968800	-14.110000
4	189.000000	6.625000	-8.320000

PLANFORM DEFINITION

	X (LEADING EDGE)	Y	CHORD	X (TRAILING EDGE)
1	59.999300	0.000000	183.881700	243.881000
2	77.328000	4.968800	166.070000	243.398000
3	83.104000	6.625000	160.133000	243.237000
4	93.165000	9.510000	149.790000	242.955000
5	116.960000	16.333000	125.350000	242.310000
6	168.980000	31.250000	77.295000	246.275000
7	225.810000	47.544000	32.681000	258.491000
8	225.810000	47.545000	32.681000	258.491000
9	258.210000	66.250000	14.445000	272.655000

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

0.0000	1.0000	2.0000	3.0000	4.0000	5.0000	6.0000	7.0000	8.0000	10.0000
12.0000	14.0000	16.0000	19.0000	22.0000	25.0000	28.0000	30.0000	32.0000	36.0000
38.0000	40.0000								

WING GRID SYSTEM PUTS SIDE-OF-FUSELAGE AT Y= 4.14063 AT EDGE OF ELEMENT ROW= 3

SPAN STATION OF ORDINATE CONSTRAINT 1 IS CHANGED FROM 4.96880 TO 4.96875

SPAN STATION OF ORDINATE CONSTRAINT 2 IS CHANGED FROM 4.96880 TO 4.96875

SPAN STATION OF ORDINATE CONSTRAINT 3 IS CHANGED FROM 4.96880 TO 4.96875

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 QUADRATIC SPANWISE (LOADING 4 IN THE LOADING DEFINITIONS)
LOADING 5 FOR THIS CASE IS QUADRATIC CHORDWISE (LOADING 5 IN THE LOADING DEFINITIONS)
LOADING 6 FOR THIS CASE IS PARABOLIC CHORDWISE (LOADING 6 IN THE LOADING DEFINITIONS)
LOADING 7 FOR THIS CASE IS CUBIC CHORDWISE (LOADING 7 IN THE LOADING DEFINITIONS)
LOADING 8 FOR THIS CASE IS SIMILAR TO FLAT WING (LOADING 8 IN THE LOADING DEFINITIONS)
LOADING 9 FOR THIS CASE IS SQ. ROOT FROM T. E. (LOADING 9 IN THE LOADING DEFINITIONS)
LOADING 10 FOR THIS CASE IS ELLIPTICAL C-SUB-P (LOADING 10 IN THE LOADING DEFINITIONS)
LOADING 11 FOR THIS CASE IS LINEAR IN ARB.REGION (LOADING 11 IN THE LOADING DEFINITIONS)
LOADING 12 FOR THIS CASE IS BODY UPWASH LOADING (LOADING 16 IN THE LOADING DEFINITIONS)
LOADING 13 FOR THIS CASE IS NACELLE BUOYANCY (LOADING 17 IN THE LOADING DEFINITIONS)
LOADING 14 FOR THIS CASE IS NACELLE BUOY(CAMBER) (LOADING 14 IN THE LOADING DEFINITIONS)
LOADING 15 FOR THIS CASE IS BODY UPWASH (CAMBER) (LOADING 13 IN THE LOADING DEFINITIONS)
LOADING 16 FOR THIS CASE IS BODY BUOYANCY TERM (LOADING 15 IN THE LOADING DEFINITIONS)
LOADING 17 FOR THIS CASE IS BODY BUOY. (CAMBER) (LOADING 12 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								

DEFINITION OF ARBITRARY REGION FOR LOADING 11.

Y

0.00000 66.25000

X

207.00000 269.80000

ARBITRARY REGION DEFINITION (LOADING 11)

FRACTION OF SEMISPAN

0.00000 1.00000

FRACTION OF LOCAL CHORD

.79943 .80235

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

****

****

C-P LONGITUDINAL GRADIENT LIMIT

.002500 .002500
.002500 .002500

****

```
*************************************
SOLUTION FOR DESIGN C   =      .100000
                     L
WITH C   CONSTRAINED TO       .010000
      M
       O
*************************************

AT Y =   4.969 AND X =  130.850, Z IS CONSTRAINED TO   -4.070
AT Y =   4.969 AND X =  189.000, Z IS CONSTRAINED TO  -10.160
AT Y =   4.969 AND X =  243.390, Z IS CONSTRAINED TO  -14.110
AT Y =   6.625 AND X =  189.000, Z IS CONSTRAINED TO   -8.320

                        A C-L
                         I   I
 C
  M            K
   O            E     I = 1       2        3        4        5        6        7        8        9       10
                         11      12       13       14       15       16       17

 .010000    .461402   .257066  .257426  .187933 -.059010 -.111751  .009099  .035220 -.005353  .005585  .018252
                     -.000444 -.012777 -.040320 -.072880  .146896 -.133854 -.390144

LAGRANGE MULTIPLIERS   -.091783  -.000080   .000704   .000257  -.000203   .000121   .000001   .000012   .029473

             CONFIGURATION FORCE AND MOMENT BREAKDOWN

                                                     C
                               C          C           M
                                L          D           O

                       WING   .09095    .003674    .002835
                   FUSELAGE   .00000    .000001    .003958
    WING INDUCED ON FUSELAGE  .00905    .000785    .003207
    WING INDUCED ON NACELLES  0.00000   .000154   0.000000
                              -------   --------   --------
                     TOTALS   .10000    .004614    .010000
```

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

****

SOLUTION PRESSURE DISTRIBUTION

****

Y / B/2	X / L	X-PRIME / CHORD	LIFTING C P	THICKNESS C P	LOWER SURFACE C P	UPPER SURFACE C P	UPPER SURFACE DC / DX P	
.075000	.081486	0.000000	-.051162	.035300	-.006281	.044881	-.013234	
.075000	.098755	.022113	.007950	.024509	.007807	-.000143	-.011471	
.075000	.118288	.047126	.063143	.012303	.017907	-.045236	-.010333	
.075000	.137822	.072138	.113434	.008553	.034013	-.079421	-.007656	
.075000	.157355	.097151	.156756	.005902	.048732	-.110023	-.007102	
.075000	.176888	.122164	.150150	.005423	.045808	-.104342	.002641	
								CP GRADIENT LIMIT = .00250
.075000	.196421	.147176	.131771	.005223	.039195	-.092577	.003014	
								CP GRADIENT LIMIT = .00250
.075000	.215955	.172189	.110507	.007019	.033136	-.077371	.003881	
								CP GRADIENT LIMIT = .00250
.075000	.235488	.197202	.086696	.009071	.026058	-.060638	.004171	
								CP GRADIENT LIMIT = .00250
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.075000	.294088	.272240	.076608	.006687	.029777	-.046830	.001023	
.075000	.313621	.297252	.070436	.004536	.028292	-.042144	.001231	
.075000	.333154	.322265	.070443	.004300	.029808	-.040636	.000339	
.075000	.352687	.347277	.069867	.004300	.031020	-.038847	.000520	
.075000	.372220	.372290	.067842	.004122	.031330	-.036512	.000640	
.075000	.391754	.397303	.064463	.003942	.030941	-.033522	.000798	
.075000	.411287	.422315	.057171	.002650	.026387	-.030785	.000706	
.075000	.430820	.447328	.048384	.001250	.020817	-.027566	.000842	
.075000	.450353	.472341	.038504	.001279	.016132	-.022372	.001354	
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.075000	.489420	.522366	.031712	.001500	.012311	-.019401	-.000908	
.075000	.508953	.547379	.036842	.001500	.013876	-.022966	-.000810	
.075000	.528486	.572391	.041201	.000067	.013622	-.027579	-.001108	
.075000	.548020	.597434	.044867	-.001534	.012854	-.032014	-.001029	
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.075000	.645686	.722467	.088262	-.007767	.028540	-.059722	.001041		
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.075000	.684752	.772492	.058466	-.011919	.007989	-.050477	.001189		
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.075000	.743352	.847530	.052077	-.016277	-.002052	-.054129	-.001160		
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.100000	.313621	.272202	.061293	.003868	.022956	-.038337	.001053		
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.150000	.430820	.384192	.070280	.001068	.031724	-.038557	.000405	
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.150000	.489420	.468245	.042812	-.001073	.016829	-.025983	.001488	
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.200000	.294088	.119731	.125839	-.005333	.036662	-.089176	.003435	
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.200000	.665219	.698376	.068376	-.015554	.014476	-.053900	-.003528	
.200000	.684752	.728831	.059711	-.017833	.007483	-.052228	.000640	
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.200000	.743352	.820196	.036485	-.023114	-.010735	-.047220	-.001027	
.200000	.762885	.850651	.090976	-.025244	.041658	-.049318	-.001167	
.200000	.782418	.881106	.082365	-.027315	.028125	-.054239	-.001205	
.200000	.801952	.911561	.075523	-.029224	.016058	-.059465	-.001328	
.200000	.821485	.942015	.075117	-.030869	.010094	-.065023	-.001412	
.200000	.841018	.972470	.066710	-.032199	-.004606	-.071317	-.001621	
.200000	.858675	1.000000	.060163	-.033300	-.017730	-.077894	-.001918	
.250000	.271618	0.000000	.237221	.040400	.136661	-.100561	-.005823	
.250000	.294088	.038346	.207094	.005275	.087526	-.119568	-.002677	
.250000	.313621	.071681	.169926	-.008305	.056279	-.113647	.003821	
								CP GRADIENT LIMIT = .00250
.250000	.333154	.105016	.133371	-.011298	.036182	-.097189	.002876	
								CP GRADIENT LIMIT = .00250
.250000	.352687	.138350	.123114	-.005964	.039004	-.084111	.003396	
								CP GRADIENT LIMIT = .00250
.250000	.372220	.171685	.109077	-.004577	.035988	-.073088	.002336	
.250000	.391754	.205020	.094943	-.005130	.030893	-.064050	.000482	
.250000	.411287	.236354	.094688	-.004663	.033232	-.061456	.000760	
.250000	.430820	.271689	.092279	-.003719	.034971	-.057307	.001179	
.250000	.450353	.305024	.088768	-.002499	.036266	-.052503	.000518	

.250000	.469887	.338358	.088051	-.001166	.038107	-.049944	.000711	
.250000	.489420	.371693	.085804	-.001090	.037926	-.047878	.000421	
.250000	.508953	.405027	.081853	-.001761	.036031	-.045822	.000536	
.250000	.528486	.438362	.074853	-.002828	.031564	-.043289	.000681	
.250000	.548020	.471697	.066707	-.003287	.027132	-.039575	.001037	
.250000	.567553	.505031	.055955	-.003561	.021561	-.034393	.002149	
.250000	.587066	.538366	.035447	-.004628	.010207	-.025239	.002256	
.250000	.606619	.571701	.014100	-.006476	-.002347	-.016448	.002062	
.250000	.626153	.605035	-.002996	-.008883	-.013422	-.010426	-.002192	
.250000	.645686	.638370	.007311	-.012084	-.012069	-.019380	-.002121	
.250000	.665219	.671705	.017072	-.015067	-.010771	-.027843	-.001981	
.250000	.684752	.705039	.025448	-.017772	-.009763	-.035211	-.000727	
.250000	.704285	.738374	.028206	-.019572	-.009951	-.038157	-.000693	
.250000	.723819	.771709	.030739	-.021503	-.010381	-.041120	-.000722	
.250000	.743352	.805043	.033562	-.023563	-.010900	-.044462	-.001362	
.250000	.762885	.838378	.068450	-.025963	.040428	-.048022	-.001513	
.250000	.782418	.871713	.081572	-.028363	.027204	-.054369	-.001544	
.250000	.801952	.905047	.075813	-.030673	.015023	-.060790	-.001427	
.250000	.821485	.938382	.074193	-.032473	.007422	-.066771	-.001612	
.250000	.841018	.971717	.062081	-.033534	-.011107	-.073188	-.001554	
.250000	.857591	1.000000	.055852	-.034100	-.023283	-.079135	-.001875	
.300000	.325939	0.000000	.224271	.027500	.117985	-.106285	-.005351	
.300000	.352587	.049923	.194242	-.006747	.071045	-.123197	-.001432	
.300000	.372220	.086380	.159095	-.010001	.051913	-.107182	.004373	
								CP GRADIENT LIMIT = .00250
.300000	.391754	.122837	.137441	-.010424	.042884	-.094557	.002346	
.300000	.411287	.159293	.123374	-.008849	.039958	-.083415	.003149	
								CP GRADIENT LIMIT = .00250
.300000	.430820	.195750	.105773	-.006297	.036244	-.069529	.003523	
								CP GRADIENT LIMIT = .00250
.300000	.450353	.232207	.104052	-.005936	.037602	-.066449	.000502	
.300000	.469887	.268663	.102467	-.005377	.039136	-.063331	.000988	
.300000	.489420	.305120	.099338	-.004326	.040264	-.059075	.000657	
.300000	.508953	.341577	.097235	-.003086	.041308	-.055927	.000855	
.300000	.528486	.378033	.093577	-.003473	.039950	-.053628	.000523	
.300000	.548020	.414490	.087979	-.003362	.037770	-.050209	.001234	
.300000	.567553	.450947	.080299	-.001857	.035417	-.044882	.000464	
.300000	.587086	.487403	.071495	-.004044	.028809	-.042685	.000592	
.300000	.606619	.523860	.054950	-.006041	.018821	-.036130	.002124	
.300000	.626153	.560317	.033956	-.008163	.006638	-.027318	.002033	
.300000	.645686	.596773	.012173	-.010861	-.006514	-.018686	.002121	
.300000	.665219	.633230	.005588	-.013360	-.013611	-.019200	-.000308	
.300000	.684752	.669687	-.000030	-.015602	-.020140	-.020110	-.000140	
.300000	.704285	.706143	-.002470	-.017988	-.024832	-.022362	-.002627	
.300000	.723819	.742600	.0[illegible]2714	-.022071	-.020484	-.033199	-.002593	
.300000	.743352	.779057	.027680	-.024818	-.014909	-.042589	-.002171	
.300000	.762885	.815513	.091127	-.027317	.042749	-.048378	-.001782	
.300000	.782418	.851970	.087056	-.029851	.031331	-.055725	-.001398	
.300000	.801952	.888427	.084092	-.030799	.022475	-.061617	-.001447	
.300000	.821485	.924883	.081594	-.032693	.014950	-.066645	-.001061	
.300000	.841018	.961340	.072796	-.034980	.000818	-.071978	-.001414	
.300000	.860551	.997797	.094949	-.037168	.017854	-.077095	-.001850	
.300000	.861732	1.000000	.094101	-.037300	.016537	-.077564	-.001887	
.350000	.380260	0.000000	.225556	.049000	.141628	-.083928	-.008299	
.350000	.391754	.023669	.218000	.027461	.117494	-.100505	-.005592	
.350000	.411287	.063893	.168867	.000277	.077755	-.111112	.001847	
.350000	.430820	.104117	.157037	-.008635	.055001	-.102036	.000656	

.350000	.450353	.144341	.143522	-.013864	.045629	-.097893	.001296		
.350000	.469887	.184565	.125554	-.011489	.041635	-.083919	.003879		
								CP GRADIENT LIMIT =	.00250
.350000	.489420	.224789	.116898	-.009455	.041273	-.075625	.000880		
.350000	.508953	.265013	.113489	-.007209	.043324	-.070165	.001937		
.350000	.528486	.305237	.106456	-.003531	.045909	-.062547	.000448		
.350000	.548020	.345461	.105112	-.003773	.044860	-.060251	.000653		
.350000	.567553	.385684	.100171	-.004157	.042871	-.057300	.000794		
.350000	.587086	.425908	.093020	-.005129	.038644	-.054376	.000687		
.350000	.606619	.466132	.084181	-.006029	.033344	-.050836	.001060		
.350000	.626153	.506356	.073211	-.006567	.027402	-.045809	.001732		
.350000	.645686	.546580	.055782	-.008256	.017400	-.038382	.001841		
.350000	.665219	.586804	.037506	-.010755	.006166	-.031340	.001723		
.350000	.684752	.627028	.020193	-.014141	-.006352	-.026545	.000896		
.350000	.704285	.667252	.003097	-.017956	-.019621	-.022718	.000942		
.350000	.723819	.707476	-.006771	-.021519	-.029828	-.021057	-.002067		
.350000	.743352	.747700	.004024	-.023771	-.025542	-.029566	-.002033		
.350000	.762865	.787924	.016619	-.026479	-.021812	-.038431	-.002135		
.350000	.782418	.828147	.077660	-.029214	.031337	-.046323	-.002637		
.350000	.801952	.868371	.083580	-.032243	.025942	-.057638	-.002829		
.350000	.821485	.908595	.129129	-.035123	.062703	-.066426	-.001477		
.350000	.841018	.948819	.119329	-.036169	.046491	-.072839	-.001621		
.350000	.860551	.989043	.113689	-.037840	.032900	-.080789	-.002197		
.350000	.865872	1.000000	.110719	-.038300	.027354	-.083365	-.002360		
.400000	.434581	0.000000	.198791	.040500	.123095	-.075695	-.010542		
.400000	.450353	.036223	.199737	.012173	.096401	-.103336	-.006616		
.400000	.469887	.081083	.181358	-.005873	.070600	-.110758	.000695		
.400000	.489420	.125942	.162652	-.011390	.057680	-.104772	.002037		
.400000	.508953	.170802	.144679	-.011984	.050968	-.093711	.003298		
								CP GRADIENT LIMIT =	.00250
.400000	.528486	.215661	.129202	-.011118	.046311	-.082891	.001103		
.400000	.548020	.260521	.123013	-.009846	.045855	-.077159	.001898		
.400000	.567553	.305380	.115279	-.007099	.046048	-.069231	.001135		
.400000	.587086	.350239	.111744	-.004603	.047674	-.064070	.000617		
.400000	.606619	.395099	.106548	-.005141	.045435	-.061113	.000802		
.400000	.626153	.439958	.098348	-.005680	.040914	-.057433	.000964		
.400000	.645686	.484818	.086716	-.007471	.034329	-.054387	.000714		
.400000	.665219	.529677	.075654	-.009921	.025578	-.050076	.001231		
.400000	.684752	.574537	.060461	-.012818	.015422	-.045039	.001204		
.400000	.704285	.619396	.041962	-.015686	.003417	-.038545	.002019		
.400000	.723819	.664256	.019508	-.018455	-.010758	-.030266	.001894		
.400000	.743352	.709115	.000299	-.021829	-.024167	-.024466	-.000636		
.400000	.762885	.753974	-.004565	-.025466	-.031648	-.027083	-.000734		
.400000	.782418	.798834	-.009592	-.029593	-.039702	-.030110	-.000727		
.400000	.801952	.843693	.066030	-.032933	.019132	-.046898	-.004758		
.400000	.821485	.888553	.124880	-.035636	.060286	-.064595	-.004731		
.400000	.841018	.933412	.121740	-.037503	.046970	-.074770	-.001706		
.400000	.860551	.978272	.115628	-.038496	.034034	-.081795	-.001883		
.400000	.870013	1.000000	.110013	-.038800	.024172	-.085842	-.002160		
.475000	.516063	0.000000	.188089	.025200	.105245	-.082845	-.011252		
.475000	.528486	.034447	.197722	.006116	.092148	-.105573	-.006716		
.475000	.548020	.088607	.186268	-.010878	.071269	-.114999	-.000009		
.475000	.567553	.142767	.172107	-.016002	.061099	-.111009	.001552		
.475000	.587086	.196927	.153805	-.015746	.054288	-.099517	.003196		
								CP GRADIENT LIMIT =	.00250
.475000	.606619	.251087	.141911	-.009515	.056031	-.085880	.002315		
.475000	.626153	.305247	.128733	-.007887	.052442	-.076291	.000872		

.475000	.645686	.359407	.123546	-.007397	.051246	-.072300	.000895	
.475000	.665219	.413567	.116352	-.008170	.047580	-.068772	.000502	
.475000	.684752	.467727	.107126	-.010072	.041159	-.065967	.000939	
.475000	.704285	.521887	.095641	-.012019	.033548	-.062093	.000886	
.475000	.723819	.576047	.081627	-.015265	.023350	-.058278	.000945	
.475000	.743352	.630207	.063162	-.019197	.010474	-.052689	.001669	
.475000	.762885	.684367	.047593	-.023068	.002770	-.044823	.001913	
.475000	.782418	.738527	.071663	-.026990	.034573	-.037090	.001045	
.475000	.801952	.792687	.038153	-.031009	.005495	-.032658	.001076	
.475000	.821485	.846847	.039073	-.034501	-.004491	-.043564	-.003141	
.475000	.841018	.901007	.045352	-.038544	-.012695	-.058047	-.004010	
.475000	.860551	.955167	.052647	-.040870	-.022387	-.075034	-.004090	
.475000	.876721	1.000000	.062685	-.042350	-.027177	-.089862	-.004651	
.550000	.597556	0.000000	.176762	.019450	.096181	-.080581	-.013170	
.550000	.626153	.096389	.189315	-.011231	.075439	-.113876	-.000704	
.550000	.645686	.162228	.178650	-.016087	.066602	-.112048	.002053	
.550000	.665219	.228067	.163456	-.014207	.062252	-.101204	.003167	
								CP GRADIENT LIMIT = .00250
.550000	.684752	.293905	.147518	-.010718	.059254	-.088264	.003177	
								CP GRADIENT LIMIT = .00250
.550000	.704285	.359744	.136390	-.009928	.055364	-.081026	.001459	
.550000	.723819	.425583	.126105	-.011478	.049251	-.076854	.000211	
.550000	.743352	.491422	.117673	-.014680	.042031	-.075642	.000369	
.550000	.762885	.557260	.104542	-.017393	.032750	-.071793	.000731	
.550000	.782418	.623099	.089172	-.021779	.020761	-.068411	.000917	
.550000	.801952	.688938	.115693	-.026381	.053604	-.062089	.001051	
.550000	.821485	.754777	.081185	-.031387	.025673	-.055512	.001659	
.550000	.841018	.820616	.054029	-.036555	.001382	-.052648	-.001536	
.550000	.860551	.886454	.048860	-.040741	-.009674	-.056535	-.001481	
.550000	.880085	.952293	.045291	-.042850	-.024196	-.069487	-.003016	
.550000	.894238	1.000000	.049397	-.043900	-.029867	-.079264	-.003610	
.625000	.679049	0.000000	.171818	.015925	.093296	-.078521	-.016798	
.625000	.704285	.108449	.195115	-.011285	.080310	-.114805	-.001260	
.625000	.723819	.192388	.184300	-.014686	.072373	-.111928	.002374	
.625000	.743352	.276327	.168901	-.014545	.066220	-.102682	.002570	
								CP GRADIENT LIMIT = .00250
.625000	.762885	.360267	.153665	-.013560	.060600	-.093065	.001662	
.625000	.782418	.444206	.140201	-.015513	.052424	-.087778	.000919	
.625000	.801952	.528145	.127978	-.019023	.042931	-.085047	.000474	
.625000	.821485	.612084	.115328	-.023794	.031811	-.083517	-.000105	
.625000	.841018	.696023	.137940	-.030631	.056511	-.081429	.000203	
.625000	.860551	.779962	.111349	-.037286	.032387	-.078962	.000286	
.625000	.880085	.863901	.096604	-.043176	.015018	-.081586	-.000672	
.625000	.899618	.947840	.091762	-.046140	.005865	-.085898	-.001477	
.625000	.911756	1.000000	.091609	-.047625	.001147	-.090462	-.002208	
.700000	.760541	0.000000	.169120	.001300	.080060	-.089060	-.019392	
.700000	.782418	.129655	.210460	-.013213	.087208	-.123252	-.001271	
.700000	.801952	.245420	.197779	-.015936	.079453	-.118326	.001832	
.700000	.821485	.361185	.181432	-.018100	.070144	-.111288	.002156	
.700000	.841018	.476950	.165691	-.018508	.062329	-.103362	.001165	
.700000	.860551	.592715	.147346	-.026061	.045662	-.101684	.000280	
.700000	.880085	.708480	.131016	-.034746	.028908	-.102109	-.000722	
.700000	.899618	.824245	.148218	-.044497	.042896	-.105322	-.002526	
.700000	.919151	.940010	.146140	-.051081	.032012	-.114129	-.001866	
.700000	.929273	1.000000	.146750	-.054300	.027691	-.119060	-.002879	

.750000	.797166	0.000000	.155218	.021400	.092784	-.062434	-.023857
.750000	.821485	.169049	.206373	.002178	.100575	-.105798	-.001764
.750000	.841018	.304831	.193839	-.007748	.085537	-.108302	-.000122
.750000	.860551	.440612	.175541	-.016334	.068780	-.106761	.000497
.750000	.880085	.576394	.157126	-.027170	.049266	-.107860	-.000431
.750000	.899618	.712175	.138308	-.037574	.029664	-.108644	-.000652
.750000	.919151	.847956	.129573	-.047613	.015359	-.114214	-.002834
.750000	.938684	.983738	.162116	-.054703	.037978	-.124138	-.003276
.750000	.941024	1.000000	.163303	-.055500	.037440	-.125863	-.003692
.800000	.824147	0.000000	.127774	.041500	.098737	-.029037	-.027647
.800000	.841018	.131115	.185463	.024355	.111711	-.073752	-.006896
.800000	.860551	.282922	.187646	.004386	.093648	-.093998	-.003406
.800000	.880085	.434729	.171052	-.014198	.066071	-.102981	-.001931
.800000	.899618	.586536	.152971	-.030480	.043767	-.109203	-.001135
.800000	.919151	.738342	.136096	-.041961	.024164	-.111932	-.000620
.800000	.938684	.890149	.131219	-.051689	.012165	-.119054	-.002773
.800000	.952819	1.000000	.138215	-.056700	.010658	-.127558	-.003739
.900000	.878110	0.000000	.074407	.045400	.073604	-.000804	-.034481
.900000	.899618	.218797	.145431	.029996	.096109	-.049323	-.004934
.900000	.919151	.417509	.144435	.007424	.074544	-.069891	-.005422
.900000	.938684	.616221	.135577	-.019811	.043549	-.092028	-.003647
.900000	.958218	.814933	.120490	-.042724	.014445	-.106045	-.004421
.900000	.976410	1.000000	.130533	-.055900	.007116	-.123416	-.005756
.950000	.905092	0.000000	.043389	.046200	.058445	.015055	-.041484
.950000	.919151	.169159	.117863	.035482	.066208	-.031656	-.007426
.950000	.938684	.404179	.126733	.016132	.072745	-.053988	-.005768
.950000	.958218	.639199	.120069	-.010747	.043857	-.076213	-.004679
.950000	.977751	.874220	.113889	-.039349	.013674	-.100216	-.008362
.950000	.988205	1.000000	.121169	-.056800	.000284	-.120884	-.010597
1.000000	.932073	0.000000	.020245	.034900	.035123	.014877	-.049889
1.000000	.958218	.384889	.106719	.016316	.061215	-.045504	-.003359
1.000000	.977751	.672452	.096514	-.003127	.038773	-.057740	-.002603
1.000000	.997284	.960016	.098822	-.022681	.021900	-.076922	-.007513
1.000000	1.000000	1.000000	.102811	-.025400	.021256	-.081556	-.008717

MINIMUM OF ($C_{P\ \text{UPPER SURFACE}}$ - $C_{P\ \text{LIMIT}}$) = .0094 AT 80.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

MAXIMUM OF ($C_{P\ \text{GRADIENT}}$ - LIMITING $C_{P\ \text{GRADIENT}}$) .00187 AT 30.00 PERCENT SEMISPAN AND 8.64 PERCENT CHORD.

DELTAT = 2.707 SEC., T = 23.278 SEC.

SUMMARY OF PRESSURE LEVEL AND PRESSURE GRADIENT CONSTRAINT CYCLES

CYCLE NUMBER	C_{M_0}	K_E	MOST CRITICAL DELTA C_P (POSITIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE	MOST CRITICAL C_P GRADIENT INCREMENT (NEGATIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE
1	.01000	.46140	.009442	80.0000	100.0000	.001873	30.0000	8.6380

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                    ***************************************
                    SOLUTION FOR DESIGN C  =      .100000
                                         L
                    WITH  1 CONSTRAINTS ON PRESSURE GRADIENT
                    WITH C   CONSTRAINED TO      .010000
                          M
                          O
                    ***************************************

   PLANFORM LOCATION OF SOLUTION PRESSURE CONSTRAINTS (  1 GRADIENT AND  0 LEVEL)

                                          SPANWISE  CHORDWISE
                                          (PERCENT) (PERCENT)

                   GRADIENT CONSTRAINT AT    30.0000    8.6380

            AT Y =    4.969 AND X =  130.850, Z IS CONSTRAINED TO   -4.070
            AT Y =    4.969 AND X =  189.000, Z IS CONSTRAINED TO  -10.160
            AT Y =    4.969 AND X =  243.390, Z IS CONSTRAINED TO  -14.110
            AT Y =    6.625 AND X =  189.000, Z IS CONSTRAINED TO   -8.320
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C_MO	K_E	A_I / C-L_I, I = 1 / 11	2 / 12	3 / 13	4 / 14	5 / 15	6 / 16	7 / 17	8	9	10
.010000	.468216	.338353	-.454879	.251809	-.134100	.206683	.123566	-.042776	-.172716	-.040738	-.024299
		.008370	.018252	.005585	-.000389	.022227	-.005353	-.008901			
LAGRANGE MULTIPLIERS		-.093930	.046614	-.000128	.000794	.000233	-.000231	.000058	.000040	-.000005	-.018791

CONFIGURATION FORCE AND MOMENT BREAKDOWN

	C_L	C_D	C_{M_O}
WING	.09069	.003698	.003017
FUSELAGE	.00000	.000001	.003958
WING INDUCED ON FUSELAGE	.00931	.000806	.003025
WING INDUCED ON NACELLES	0.00000	.000177	0.000000
TOTALS	.10000	.004682	.010000

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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SOLUTION PRESSURE DISTRIBUTION
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Y / B/2	X / L	X-PRIME / CHORD	LIFTING C_P	THICKNESS C_P	LOWER SURFACE C_P	UPPER SURFACE C_P	UPPER SURFACE DC_P / DX
.075000	.081486	0.000000	-.092133	.035300	-.026767	.065367	-.016963
.075000	.098755	.022113	-.013946	.024509	-.003141	.010805	-.013186
.075000	.118288	.047126	.050834	.012303	.011503	-.038832	-.010932

.750000	.919151	.847956	.133726	-.047613	.017436	-.116290	-.001870
.750000	.938684	.983738	.155842	-.054703	.034841	-.121001	-.001446
.750000	.941024	1.000000	.155074	-.055500	.033325	-.121749	-.001574
.800000	.824147	0.000000	-.013436	.041500	.028132	.041568	-.063228
.800000	.841018	.131115	.148856	.024355	.093407	-.055448	-.012943
.800000	.860551	.282922	.180579	.004386	.090114	-.090464	-.005422
.800000	.880085	.434729	.175466	-.014198	.070278	-.105188	-.002713
.800000	.899618	.586536	.160786	-.030480	.047675	-.113111	-.001237
.800000	.919151	.738342	.142513	-.041961	.027373	-.115141	-.000222
.800000	.938684	.890149	.131746	-.051689	.012429	-.119317	-.001729
.800000	.952819	1.000000	.129894	-.056700	.006497	-.123397	-.001589
.900000	.878110	0.000000	-.074724	.045400	-.000962	.073762	-.081259
.900000	.899618	.218797	.123800	.029996	.085293	-.038507	-.009156
.900000	.919151	.417509	.142244	.007424	.073448	-.068796	-.006536
.900000	.938684	.616221	.137765	-.019811	.044643	-.093122	-.003842
.900000	.958218	.814933	.121250	-.042724	.014825	-.106425	-.003658
.900000	.976410	1.000000	.121135	-.055900	.002417	-.118717	-.003391
.950000	.905092	0.000000	-.110712	.046200	-.018606	.092106	-.096914
.950000	.919151	.169159	.081089	.035482	.067821	-.013269	-.014582
.950000	.938684	.404179	.120127	.016132	.069442	-.050685	-.007386
.950000	.958218	.639199	.119832	-.010747	.043738	-.076094	-.004879
.950000	.977751	.874220	.110815	-.039349	.012136	-.098678	-.007327
.950000	.988205	1.000000	.111397	-.056800	-.004602	-.115998	-.008170
1.000000	.932073	0.000000	-.143336	.034900	-.046668	.096668	-.118133
1.000000	.958218	.384889	.093031	.016316	.054371	-.038660	-.005931
1.000000	.977751	.672452	.093851	-.003127	.037442	-.056409	-.002977
1.000000	.997284	.960016	.091394	-.022681	.018186	-.073208	-.005630
1.000000	1.000000	1.000000	.092779	-.025400	.016239	-.076539	-.005976

MINIMUM OF ($C_{P\,\text{UPPER SURFACE}} - C_{P\,\text{LIMIT}}$) = .0136 AT 80.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

MAXIMUM OF ($C_{P\,\text{GRADIENT}}$ - LIMITING $C_{P\,\text{GRADIENT}}$) = .00108 AT 7.50 PERCENT SEMISPAN AND 19.72 PERCENT CHORD.

DELTAT = 2.688 SEC., T = 25.966 SEC.

SUMMARY OF PRESSURE LEVEL AND PRESSURE GRADIENT CONSTRAINT CYCLES

CYCLE NUMBER	C_{M0}	K_E	MOST CRITICAL DELTA C_P (POSITIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE	MOST CRITICAL C_P GRADIENT INCREMENT (NEGATIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE
1	.01000	.46140	.009442	80.0000	100.0000	.001873	30.0000	8.6380
2	.01000	.46822	.013603	80.0000	100.0000	.001080	7.5000	19.7202

****
SOLUTION FOR DESIGN C_L = .100000

WITH 2 CONSTRAINTS ON PRESSURE GRADIENT
WITH C_{M_0} CONSTRAINED TO .010000

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PLANFORM LOCATION OF SOLUTION PRESSURE CONSTRAINTS (2 GRADIENT AND 0 LEVEL)

	SPANWISE (PERCENT)	CHORDWISE (PERCENT)
GRADIENT CONSTRAINT AT	30.0000	8.6380
GRADIENT CONSTRAINT AT	7.5000	19.7202

AT Y = 4.969 AND X = 130.850, Z IS CONSTRAINED TO -4.070
AT Y = 4.969 AND X = 189.000, Z IS CONSTRAINED TO -10.160
AT Y = 4.969 AND X = 243.390, Z IS CONSTRAINED TO -14.110
AT Y = 6.625 AND X = 189.000, Z IS CONSTRAINED TO -8.320

A_I $C\text{-}L_I$

C_{M_0}	K_E	I = 1 / 11	2 / 12	3 / 13	4 / 14	5 / 15	6 / 16	7 / 17	8	9	10
.010000	.472958	.487378	-.425316	.189910	-.140481	.190797	.115615	-.030996	-.038986	-.169180	-.112950
		.004064	.018252	.095585	-.000354	.003075	-.005353	-.000426			
LAGRANGE MULTIPLIERS		-.095586	.034491	.055210	-.000240	.000852	.000234	-.000235	.000054	.000049	-.000066
		-.039364									

CONFIGURATION FORCE AND MOMENT BREAKDOWN

	C_L	C_D	C_{M_0}
WING	.09063	.003752	.003178
FUSELAGE	.00000	.000001	.003958
WING INDUCED ON FUSELAGE	.00937	.000812	.002864
WING INDUCED ON NACELLES	0.00000	.000165	0.000000
TOTALS	.10000	.004730	.010000

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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SOLUTION PRESSURE DISTRIBUTION
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Y / B/2	X / L	X-PRIME / CHORD	LIFTING C_P	THICKNESS C_P	LOWER SURFACE C_P	UPPER SURFACE C_P	UPPER SURFACE DC_P / DX
.075000	.081486	0.000000	.010929	.035300	.024764	.013836	-.008219
.075000	.098755	.022113	.036603	.024509	.022133	-.014469	-.007295

.750000	.919151	.847956	.121752	-.047613	.011449	-.110303	-.002189
.750000	.938684	.983738	.152031	-.054703	.032935	-.119096	-.003421
.750000	.941024	1.000000	.153476	-.055500	.032526	-.120950	-.004081
.800000	.824147	0.000000	.147710	.041500	.108765	-.039005	-.015853
.800000	.841018	.131115	.171793	.024355	.104876	-.066917	-.004810
.800000	.860551	.282922	.165059	.004386	.082354	-.082705	-.003004
.800000	.880085	.434729	.149961	-.014198	.057525	-.092435	-.002175
.800000	.899618	.586536	.134502	-.030480	.034533	-.099969	-.001532
.800000	.919151	.738342	.120911	-.041961	.016571	-.104339	-.001101
.800000	.938684	.890149	.116033	-.051689	.004572	-.111461	-.002521
.800000	.952819	1.000000	.123040	-.056700	.003070	-.119970	-.004294
.900000	.878110	0.000000	.096638	.045400	.084719	-.011919	-.018834
.900000	.899618	.218797	.126936	.029996	.086861	-.040075	-.003728
.900000	.919151	.417509	.119639	.007424	.062146	-.057493	-.004904
.900000	.938684	.616221	.109115	-.019811	.030318	-.078797	-.004699
.900000	.958218	.814933	.099039	-.042724	.003724	-.095319	-.003854
.900000	.976410	1.000000	.108057	-.055900	-.004122	-.112178	-.006472
.950000	.905092	0.000000	.071318	.046200	.072409	.001091	-.022756
.950000	.919151	.169159	.105961	.035482	.080256	-.025705	-.004845
.950000	.938684	.404179	.104100	.016132	.061429	-.042671	-.005173
.950000	.958218	.639199	.095804	-.010747	.031724	-.064060	-.005029
.950000	.977751	.874220	.091902	-.039349	.002680	-.089222	-.008005
.950000	.988205	1.000000	.100134	-.056800	-.010233	-.110367	-.011597
1.000000	.932073	0.000000	.069148	.034900	.059574	-.009574	-.027420
1.000000	.958218	.384889	.107535	.016316	.061623	-.045912	-.002865
1.000000	.977751	.672452	.099795	-.003127	.040414	-.059381	-.003305
1.000000	.997284	.960016	.104305	-.022681	.024641	-.079664	-.007561
1.000000	1.000000	1.000000	.108688	-.025400	.024194	-.084494	-.009446

MINIMUM OF (C_{p} UPPER SURFACE $-$ C_{p} LIMIT) = .0161 AT 75.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

MAXIMUM OF (C_{p} GRADIENT $-$ LIMITING C_{p} GRADIENT) = .00067 AT 55.00 PERCENT SEMISPAN AND 22.81 PERCENT CHORD.

DELTAT = 2.698 SEC., T = 28.604 SEC.

SUMMARY OF PRESSURE LEVEL AND PRESSURE GRADIENT CONSTRAINT CYCLES

CYCLE NUMBER	C_{M_0}	K_E	MOST CRITICAL DELTA C_p (POSITIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE	MOST CRITICAL C_p GRADIENT INCREMENT (NEGATIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE
1	.01000	.46140	.009442	80.0000	100.0000	.001873	30.0000	8.6380
2	.01000	.46822	.013603	80.0000	100.0000	.001080	7.5000	19.7202
3	.01000	.47296	.016050	75.0000	100.0000	.000672	55.0000	22.8067

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*************************************
SOLUTION FOR DESIGN C  =      .100000
                     L
WITH  3 CONSTRAINTS ON PRESSURE GRADIENT
WITH C    CONSTRAINED TO      .010000
      M
       O
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PLANFORM LOCATION OF SOLUTION PRESSURE CONSTRAINTS (3 GRADIENT AND 0 LEVEL)

	SPANWISE (PERCENT)	CHORDWISE (PERCENT)
GRADIENT CONSTRAINT AT	30.0000	8.6380
GRADIENT CONSTRAINT AT	7.5000	19.7202
GRADIENT CONSTRAINT AT	55.0000	22.8067

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AT Y =    4.969 AND X =  130.850, Z IS CONSTRAINED TO   -4.070
AT Y =    4.969 AND X =  189.000, Z IS CONSTRAINED TO  -10.160
AT Y =    4.969 AND X =  243.390, Z IS CONSTRAINED TO  -14.110
AT Y =    6.625 AND X =  189.030, Z IS CONSTRAINED TO   -8.320
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A_I $C\text{-}L_I$

C_{M_0}	K_E	I = 1 / 11	2 / 12	3 / 13	4 / 14	5 / 15	6 / 16	7 / 17	8	9	10
.010000	.494800	-.145461	.054701	.213185	-.073990	-.155339	-.041060	.128217	-.027758	.134221	-.028291
		.016580	.018252	.005585	-.004342	.000490	-.005353	.000810			
LAGRANGE MULTIPLIERS		-.104929	-.147483	.281509	.395908	-.000694	.001247	.000233	-.000172	.000075	.000019
		-.000317	-.141623								

CONFIGURATION FORCE AND MOMENT BREAKDOWN

	C_L	C_D	C_{M_0}
WING	.09051	.003912	.003217
FUSELAGE	.00000	.000001	.003958
WING INDUCED ON FUSELAGE	.00949	.000824	.002825
WING INDUCED ON NACELLES	0.00000	.000212	0.000000
TOTALS	.10000	.004948	.010000

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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SOLUTION PRESSURE DISTRIBUTION
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$\frac{Y}{B/2}$	$\frac{X}{L}$	$\frac{\text{X-PRIME}}{\text{CHORD}}$	LIFTING C_P	THICKNESS C_P	LOWER SURFACE C_P	UPPER SURFACE C_P	UPPER SURFACE DC_P / DX
.075000	.081486	0.000000	.074924	.035300	.056762	-.018162	-.004951
.075000	.098755	.022113	.078737	.024509	.043200	-.035537	-.004579
.075000	.118288	.047126	.081185	.012303	.026928	-.054257	-.004467

.750000	.938684	.983738	.123596	-.054703	.018718	-.104878	-.002591
.750000	.941024	1.000000	.123579	-.055500	.017578	-.106002	-.001899
.800000	.824147	0.000000	.109456	.041500	.089578	-.019878	-.013494
.800000	.841018	.131115	.133076	.024355	.085518	-.047559	-.005596
.800000	.860551	.282922	.134484	.004386	.067067	-.067417	-.004057
.800000	.880085	.434729	.127827	-.014198	.046459	-.081368	-.003070
.800000	.899618	.586536	.118248	-.030480	.026406	-.091842	-.002015
.800000	.919151	.738342	.105846	-.041961	.009039	-.096807	-.000837
.800000	.938684	.890149	.108641	-.051689	.000876	-.107765	-.003845
.800000	.952819	1.000000	.114095	-.056700	-.001403	-.115497	-.001236
.900000	.878110	0.000000	.083804	.045400	.078302	-.005502	-.015143
.900000	.899618	.218797	.112844	.029996	.079815	-.033029	-.004509
.900000	.919151	.417509	.111909	.007424	.058281	-.053628	-.005549
.900000	.938684	.616221	.104697	-.019811	.028109	-.076588	-.004206
.900000	.958218	.814933	.092433	-.042724	.000417	-.092016	-.005752
.900000	.976410	1.000000	.101239	-.055900	-.007531	-.108769	-.002287
.950000	.905092	0.000000	.068169	.046200	.070835	.002665	-.017980
.950000	.919151	.169159	.097539	.035482	.076045	-.021493	-.005212
.950000	.938684	.404179	.100089	.016132	.059424	-.040666	-.005646
.950000	.958218	.639199	.093180	-.010747	.030412	-.062768	-.004781
.950000	.977751	.874220	.088008	-.039349	.000733	-.087275	-.008685
.950000	.988205	1.000000	.090474	-.056800	-.015063	-.105537	-.006582
1.000000	.932073	0.000000	.056750	.034900	.053375	-.003375	-.021102
1.000000	.958218	.384889	.088811	.016316	.052261	-.036550	-.003088
1.000000	.977751	.672452	.079809	-.003127	.030421	-.049388	-.002530
1.000000	.997284	.960016	.078340	-.022681	.011659	-.066681	-.004785
1.000000	1.000000	1.000000	.077821	-.025400	.008761	-.069061	-.003234

MINIMUM OF ($C_{P\ \mathrm{UPPER\ SURFACE}}$ − $C_{P\ \mathrm{LIMIT}}$) = .0215 AT 80.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

MAXIMUM OF ($C_{P\ \mathrm{GRADIENT}}$ − LIMITING $C_{P\ \mathrm{GRADIENT}}$) = .00073 AT 15.00 PERCENT SEMISPAN AND 10.40 PERCENT CHORD.

DELTAT = 2.727 SEC., T = 31.391 SEC.

SUMMARY OF PRESSURE LEVEL AND PRESSURE GRADIENT CONSTRAINT CYCLES

CYCLE NUMBER	C_{M_O}	K_E	MOST CRITICAL DELTA C_P (POSITIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	CHORDWISE	MOST CRITICAL C_P GRADIENT INCREMENT (NEGATIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	CHORDWISE
1	.01000	.46140	.009442	80.0000	100.0000	.001873	30.0000	8.6380
2	.01000	.46822	.013603	80.0000	100.0000	.001080	7.5000	19.7202
3	.01000	.47296	.016050	75.0000	100.0000	.000672	55.0000	22.8067
4	.01000	.49480	.021503	80.0000	100.0000	.000735	15.0000	10.4016

PRESSURE GRADIENT CONSTRAINT REMOVED AT 30.0000 8.6380

****
SOLUTION FOR DESIGN C_L = .100000

WITH 2 CONSTRAINTS ON PRESSURE GRADIENT
WITH C_{M_0} CONSTRAINED TO .010000

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PLANFORM LOCATION OF SOLUTION PRESSURE CONSTRAINTS (2 GRADIENT AND 0 LEVEL)

	SPANWISE (PERCENT)	CHORDWISE (PERCENT)
GRADIENT CONSTRAINT AT	7.5000	19.7202
GRADIENT CONSTRAINT AT	55.0000	22.8067

AT Y = 4.969 AND X = 130.850, Z IS CONSTRAINED TO -4.070
AT Y = 4.969 AND X = 189.000, Z IS CONSTRAINED TO -10.160
AT Y = 4.969 AND X = 243.390, Z IS CONSTRAINED TO -14.110
AT Y = 6.625 AND X = 189.000, Z IS CONSTRAINED TO -8.320

C_{M_0}	K_E	A_{C-L} $T\ I$ I = 1 / 11	2 / 12	3 / 13	4 / 14	5 / 15	6 / 16	7 / 17	8	9	10
.010000	.483143	.285119	-.302363	.207329	-.116617	.085936	.051099	.029776	-.116159	-.001211	-.057462
		.008814	.018252	.005585	-.001694	-.000964	-.005353	.000379			
LAGRANGE MULTIPLIERS		-.099604	.121338	.135208	-.000398	.001024	.000221	-.000228	.000027	.000058	-.000148
		-.096228									

CONFIGURATION FORCE AND MOMENT BREAKDOWN

	C_L	C_D	C_{M_0}
WING	.09047	.003809	.003257
FUSELAGE	.00000	.000001	.003958
WING INDUCED ON FUSELAGE	.00953	.000827	.002785
WING INDUCED ON NACELLES	0.00000	.000195	0.000000
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TOTALS	.10000	.004831	.010000

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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SOLUTION PRESSURE DISTRIBUTION
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Y / B/2	X / L	X-PRIME / CHORD	LIFTING C_P	THICKNESS C_P	LOWER SURFACE C_P	UPPER SURFACE C_P	UPPER SURFACE DC_P / DX
.075000	.081486	0.000000	-.010196	.035300	.014202	.024398	-.010886
.075000	.098755	.022113	.028937	.024509	.018301	-.010637	-.008485

.800000	.952819	1.000000	.116870	-.056700	-.000015	-.116885	-.001541
.900000	.878110	0.000000	-.023012	.045400	.024894	.047906	-.055482
.900000	.899618	.218797	.109793	.029996	.078289	-.031503	-.007375
.900000	.919151	.417509	.120382	.007424	.062518	-.057865	-.006056
.900000	.938684	.616221	.114578	-.019811	.033049	-.081529	-.004189
.900000	.958218	.814933	.101682	-.042724	.005041	-.096641	-.004050
.900000	.976410	1.000000	.103289	-.055900	-.006505	-.109795	-.003141
.950000	.905092	0.000000	-.049868	.046200	.011816	.061684	-.066083
.950000	.919151	.169159	.078174	.035482	.066363	-.011811	-.010807
.950000	.938684	.404179	.102396	.016132	.060577	-.041819	-.006656
.950000	.958218	.639199	.099634	-.010747	.033639	-.065995	-.005022
.950000	.977751	.874220	.092731	-.039349	.003094	-.089636	-.007540
.950000	.988205	1.000000	.093826	-.056800	-.013387	-.107213	-.007880
1.000000	.932073	0.000000	-.066477	.034900	-.008238	.058238	-.080495
1.000000	.958218	.384889	.090808	.016316	.053259	-.037548	-.004952
1.000000	.977751	.672452	.090328	-.003127	.035681	-.054647	-.003176
1.000000	.997284	.960016	.088812	-.022681	.016895	-.071917	-.005205
1.000000	1.000000	1.000000	.089507	-.025400	.014604	-.074904	-.005141

MINIMUM OF ($C_{P\,UPPER\ SURFACE}$ - $C_{P\,LIMIT}$) = .0201 AT 80.0000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

MAXIMUM OF ($C_{P\,GRADIENT}$ - LIMITING $C_{P\,GRADIENT}$) = .00000 AT 15.00 PERCENT SEMISPAN AND 13.20 PERCENT CHORD.

DELTAT = 2.722 SEC., T = 34.113 SEC.

SUMMARY OF PRESSURE LEVEL AND PRESSURE GRADIENT CONSTRAINT CYCLES

CYCLE NUMBER	$C_{M\,O}$	K_E	MOST CRITICAL DELTA C_P (POSITIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE	MOST CRITICAL C_P GRADIENT INCREMENT (NEGATIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	PLANFORM LOCATION (IN PERCENT) CHORDWISE
1	.01000	.46140	.013603	80.0000	100.0000	.001080	7.5000	19.7202
2				75.0000	100.0000		55.0000	22.8067
3	.01000	.48314	.020115	80.0000	100.0000	.000001	15.0000	13.2033

****
SOLUTION FOR DESIGN C_L = .100000

WITH 3 CONSTRAINTS ON PRESSURE GRADIENT
WITH C_{M_0} CONSTRAINED TO .010000
****

PLANFORM LOCATION OF SOLUTION PRESSURE CONSTRAINTS (3 GRADIENT AND 0 LEVEL)

	SPANWISE (PERCENT)	CHORDWISE (PERCENT)
GRADIENT CONSTRAINT AT	7.5000	19.7202
GRADIENT CONSTRAINT AT	55.0000	22.8067
GRADIENT CONSTRAINT AT	15.0000	13.2033

AT Y = 4.969 AND X = 130.850, Z IS CONSTRAINED TO -4.070
AT Y = 4.969 AND X = 189.000, Z IS CONSTRAINED TO -10.160
AT Y = 4.969 AND X = 243.390, Z IS CONSTRAINED TO -14.119
AT Y = 6.625 AND X = 189.000, Z IS CONSTRAINED TO -8.320

C_{M_0}	K_E	A_I $C\text{-}L_I$ I = 1 / 11	2 / 12	3 / 13	4 / 14	5 / 15	6 / 16	7 / 17	8	9	10
.010000	.485022	.500824	-.456754	.197412	-.136682	.190669	.091543	-.014601	-.149810	-.074707	-.078543
		.004940	.018252	.005585	-.001068	-.001501	-.005353	.000185			
LAGRANGE MULTIPLIERS		-.097487	.071945	.089635	.122549	-.000425	.000942	.000213	-.000117	-.000083	.000098
		-.000079	-.085738								

CONFIGURATION FORCE AND MOMENT BREAKDOWN

	C_L	C_D	C_{M_0}
WING	.09039	.003824	.003292
FUSELAGE	.00000	.000001	.003958
WING INDUCED ON FUSELAGE	.00961	.000833	.002750
WING INDUCED ON NACELLES	0.00000	.000193	0.000000
	-------	--------	--------
TOTALS	.10000	.004850	.010000

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

****
SOLUTION PRESSURE DISTRIBUTION
****

Y / B/2	X / L	X-PRIME / CHORD	LIFTING C_P	THICKNESS C_P	LOWER SURFACE C_P	UPPER SURFACE C_P	UPPER SURFACE DC_P / DX
.075000	.081486	0.000000	-.044770	.035300	-.003085	.041685	-.013245

SUMMARY OF PRESSURE LEVEL AND PRESSURE GRADIENT CONSTRAINT CYCLES

CYCLE NUMBER	C_{M_0}	K_F	MOST CRITICAL DELTA C_P (POSITIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	CHORDWISE	MOST CRITICAL C_P GRADIENT INCREMENT (NEGATIVE IS SATISFACTORY)	PLANFORM LOCATION (IN PERCENT) SPANWISE	CHORDWISE
1	.01000	.46140	.013603	80.0000	100.0000	.001080	7.5000	19.7202
2				75.0000	100.0000		55.0000	22.8067
3	.01000	.48314	.020115	80.0000	100.0000	.000001	15.0000	13.2033
4	.01000	.48502	.018510	7.5000	100.0000	-.000116	35.0000	18.4565

LARGEST VALUES OF WING UPPER SURFACE
LONGITUDINAL PRESSURE GRADIENT
DUE TO BODY BUOYANCY AND UPWASH LOADINGS,
AND TO WING THICKNESS PRESSURES

	LARGEST GRADIENTS	PLANFORM LOCATION (IN PERCENT) SPANWISE	CHORDWISE
1	.003106	15.00	10.40
2	.003106	15.00	13.20
3	.003079	10.00	11.66
4	.003079	10.00	14.25
5	.003039	20.00	11.97
6	.003032	17.50	12.61
7	.002941	12.50	11.05
8	.002941	12.50	13.75
9	.002765	25.00	10.50
10	.002765	25.00	13.84
11	.002372	35.00	18.46
12	.002366	7.50	17.22

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

CAMBER SURFACE CORRESPONDING TO OPTION 4

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

SPANWISE DISTRIBUTION OF SECTION DRAG, LIFT, AND PITCHING MOMENT

Y / --- / B/2	CHORD	SECTION C D	SECTION C L	SECTION C M
0.0000000	183.8817000	0.0000000	.0609469	-.0551104
.0250000	177.9445264	0.0000000	.0635613	-.0597203
.0500000	172.0073528	0.0000000	.0688513	-.0668628
.0750000	166.0701792	.0073901	.0777107	-.0776302
.1000000	160.1330000	.0053355	.0765027	-.0774657
.1250000	154.1951859	.0044110	.0757925	-.0788723
.1500000	148.2586941	.0049669	.0750135	-.0805416
.1750000	142.3260032	.0048712	.0753591	-.0840463
.2000000	136.3933123	.0047023	.0762417	-.0892302
.2500000	124.6106675	.0047351	.0784822	-.1025623
.3000000	113.9394744	.0053322	.0844334	-.1262182
.3500000	103.2682813	.0055635	.0905118	-.1566280
.4000000	92.5970882	.0049410	.0991832	-.2009582
.4750000	76.6960487	.0050838	.1107863	-.2881715
.5500000	63.0912977	.0038330	.1254547	-.4233732
.6250000	49.4355457	.0011358	.1.073651	-.6443739
.6250000	49.4865467	.0011858	.1407865	-.6448738
.7000000	35.8817958	-.0038208	.1587721	-1.0585959
.7500000	30.5922200	-.0008203	.1547333	-1.2372215
.8000000	27.3627760	-.0009678	.1345567	-1.2247928
.9000000	20.9038880	-.0010630	.1040856	-1.2831989
.9500000	17.6744440	.0012518	.0782925	-1.1654958
1.0000000	14.4450000	.0028522	.0848762	-1.5637930

SPANWISE INTEGRATION BY TRAPAZOID RULE = 311.572109 FOR KOPT, LOADNO(KOPT), KVAK = 18 0 2

Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)
0.00000	44.26773	.39873	42.83842	.79745	41.40910	1.19618	39.97978	1.59490	38.55047
1.99363	37.12099	2.39236	35.69184	2.79108	34.26360	3.18981	32.83537	3.98726	29.99881
4.78471	27.42982	5.58217	24.86084	6.37962	22.29185	7.57580	18.46383	8.77197	15.18862
9.96815	11.91341	11.16433	8.63819	11.96178	7.36478	12.75924	6.58732	14.35414	5.03241
15.15159	4.25495	15.94904	3.47749						

SPANWISE INTEGRATION BY TRAPAZOID RULE = 8584.760860 FOR KOPT, LOADNO(KOPT), KVAR = 18 0 3

Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)
0.00000	1959.63231	.39873	1835.13002	.79745	1714.71362	1.19618	1598.38311	1.59490	1486.13839
1.99363	1377.96822	2.39236	1273.90755	2.79108	1173.99455	3.18981	1078.16126	3.98726	899.92852
4.78471	752.39516	5.58217	618.06119	6.37962	496.92659	7.57580	340.91299	8.77197	230.69410
9.96815	141.92924	11.16433	74.61840	11.96178	54.23998	12.75924	43.39282	14.35414	25.32513
15.15159	18.10461	15.94904	12.09296						

SPANWISE INTEGRATION BY TRAPAZOID RULE = 25.932097 FUR KOPT, LOADNO(KOPT), KVAR = 18 0 4

Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)
0.00000	2.69798	.39873	2.72287	.79745	2.85137	1.19618	3.10686	1.59490	2.94922
1.99363	2.81349	2.39236	2.67737	2.79108	2.59207	3.18981	2.50342	3.98726	2.35437
4.78471	2.31599	5.58217	2.25020	6.37962	2.21098	7.57580	2.04554	8.77197	1.90548
9.96815	1.67725	11.16433	1.37150	11.96178	1.13958	12.75924	.88637	14.35414	.52380
15.15159	.33313	15.94904	.29516						

SPANWISE INTEGRATION BY TRAPAZOID RULE = 1.099969 FOR KOPT, LOADNO(KOPT), KVAR = 18 0 5

Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)
0.00000	0.00000	.39873	0.00000	.79745	0.00000	1.19618	.29546	1.59490	.20569
1.99363	.16374	2.39236	.17728	2.79108	.16690	3.18981	.15440	3.98726	.14205
4.78471	.14626	5.58217	.13831	6.37962	.11014	7.57580	.09387	8.77197	.05822
9.96815	.01413	11.16433	-.03301	11.96178	-.00604	12.75924	-.00637	14.35414	-.00535
15.15159	.00533	15.94904	.00992						

SPANWISE INTEGRATION BY TRAPAZOID RULE = -1205.475049 FOR KOPT, LOADNO(KOPT), KVAR = 18 0 6

Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)	Y	F(Y)
0.00000	-107.99616	.39873	-109.59460	.79745	-114.65063	1.19618	-124.08284	1.59490	-115.12479
1.99363	-108.68349	2.39236	-102.60252	2.79108	-98.67226	3.18981	-96.20450	3.98726	-92.29871
4.78471	-94.96596	5.58217	-96.80569	6.37962	-99.86148	7.57580	-98.24141	8.77197	-97.66969
9.96815	-91.52644	11.16433	-78.99073	11.96178	-67.10688	12.75924	-53.14722	14.35414	-32.49718
15.15159	-21.10084	15.94904	-18.91089						

C_L = .100020 C_D = .004862 $\frac{X_{CP}}{L}$ = .708204 K_E = .485974

$\frac{S_{REF}}{S_{PROG}}$ = .920569 $\frac{C_M}{C_L}$ = -.057282 C_{M_0} = .009993

CONFIGURATION FORCE AND MOMENT BREAKDOWN

	C_L	C_D	C_{M_0}
WING	.09041	.003835	.003285
FUSELAGE	.00000	.000001	.003958
WING INDUCED ON FUSELAGE	.00961	.000833	.002750
WING INDUCED ON NACELLES	0.00000	.000193	0.000000
	--------	---------	--------
TOTALS	.10002	.004862	.009993

Y / B/2	X / LENGTH	X-PRIME / CHORD	Z / CHORD	Z	LIFTING C_P	THICKNESS C_P	LOWER SURFACE C_P	UPPER SURFACE C_P
0.00000	0.00000	0.00000	0.00000	0.00000	.01740	0.00000	-.00730	-.02470
0.00000	.02062	.02385	0.00000	0.00000	.02300	.00343	-.00611	-.02911
0.00000	.04016	.04644	0.00000	0.00000	.02831	.00669	-.00498	-.03329
0.00000	.05969	.06903	0.00000	0.00000	.03476	.01040	-.00282	-.03758
0.00000	.07922	.09162	0.00000	0.00000	.04143	.01419	-.00047	-.04190
0.00000	.09876	.11421	0.00000	0.00000	.03748	.01696	.00013	-.03735
0.00000	.11829	.13680	0.00000	0.00000	.02728	.01913	-.00029	-.02757
0.00000	.13782	.15939	0.00000	0.00000	.01839	.01897	-.00239	-.02078
0.00000	.15735	.18198	0.00000	0.00000	.01133	.01554	-.00685	-.01818
0.00000	.17689	.20457	0.00000	0.00000	.00828	.01235	-.00887	-.01715
0.00000	.19642	.22716	0.00000	0.00000	.02104	.01014	-.00132	-.02236
0.00000	.21595	.24975	0.00000	0.00000	.03380	.00792	.00623	-.02756
0.00000	.23549	.27234	0.00000	0.00000	.04655	.00705	.01513	-.03142

0.00000	.25502	.29493	0.00000	0.00000	.05931	.00619	.02404	-.03527
0.00000	.27455	.31752	0.00000	0.00000	.06599	.00512	.02812	-.03787
0.00000	.29409	.34011	0.00000	0.00000	.07090	.00399	.03080	-.04010
0.00000	.31362	.36270	0.00000	0.00000	.07581	.00332	.03394	-.04187
0.00000	.33315	.38529	0.00000	0.00000	.08072	.00301	.03744	-.04329
0.00000	.35269	.40788	0.00000	0.00000	.08247	.00319	.03945	-.04302
0.00000	.37222	.43047	0.00000	0.00000	.07831	.00432	.03870	-.03961
0.00000	.39175	.45306	0.00000	0.00000	.07415	.00518	.03768	-.03647
0.00000	.41129	.47565	0.00000	0.00000	.06999	.00427	.03490	-.03509
0.00000	.43082	.49824	0.00000	0.00000	.06582	.00337	.03212	-.03371
0.00000	.45035	.52083	0.00000	0.00000	.06767	.00218	.03103	-.03664
0.00000	.46989	.54341	0.00000	0.00000	.07002	.00096	.03008	-.03994
0.00000	.48942	.56600	0.00000	0.00000	.07237	.00047	.02987	-.04250
0.00000	.50895	.58859	0.00000	0.00000	.07472	.00029	.02996	-.04476
0.00000	.52849	.61118	0.00000	0.00000	.07782	-.00014	.03074	-.04708
0.00000	.54802	.63377	0.00000	0.00000	.08168	-.00081	.03221	-.04947
0.00000	.56755	.65636	0.00000	0.00000	.08554	-.00161	.03358	-.05196
0.00000	.58709	.67895	0.00000	0.00000	.08940	-.00269	.03465	-.05475
0.00000	.60662	.70154	0.00000	0.00000	.09267	-.00371	.03543	-.05725
0.00000	.62615	.72413	0.00000	0.00000	.08796	-.00389	.03221	-.05575
0.00000	.64569	.74672	0.00000	0.00000	.08324	-.00407	.02899	-.05424
0.00000	.66522	.76931	0.00000	0.00000	.07852	-.00553	.02450	-.05402
0.00000	.68475	.79190	0.00000	0.00000	.07380	-.00720	.01979	-.05401
0.00000	.70429	.81449	0.00000	0.00000	.07054	-.00945	.01516	-.05538
0.00000	.72382	.83708	0.00000	0.00000	.06810	-.01203	.01057	-.05753
0.00000	.74335	.85967	0.00000	0.00000	.06566	-.01429	.00630	-.05936
0.00000	.76289	.88226	0.00000	0.00000	.06322	-.01615	.00243	-.06078
0.00000	.78242	.90485	0.00000	0.00000	.06204	-.01799	-.00125	-.06329
0.00000	.80195	.92744	0.00000	0.00000	.06553	-.01980	-.00426	-.06979
0.00000	.82148	.95003	0.00000	0.00000	.06901	-.02160	-.00728	-.07629
0.00000	.84102	.97262	0.00000	0.00000	.07249	-.02382	-.01070	-.08319
0.00000	.86055	.99521	0.00000	0.00000	.07597	-.02603	-.01412	-.09009
0.00000	.86469	1.00000	0.00000	0.00000	.07671	-.02650	-.01484	-.09156
.02500	.02716	0.00000	0.00000	0.00000	.01760	.00300	-.00420	-.02180
.02500	.04016	.01553	0.00000	0.00000	.02147	.00443	-.00412	-.02559
.02500	.05969	.03887	0.00000	0.00000	.02728	.00658	-.00401	-.03128
.02500	.07922	.06222	0.00000	0.00000	.03267	.00902	-.00381	-.03648
.02500	.09876	.08556	0.00000	0.00000	.03767	.01172	-.00353	-.04121
.02500	.11829	.10890	0.00000	0.00000	.03741	.01358	-.00388	-.04129
.02500	.13782	.13225	0.00000	0.00000	.02859	.01404	-.00523	-.03382
.02500	.15735	.15559	0.00000	0.00000	.02232	.01391	-.00591	-.02823
.02500	.17689	.17893	0.00000	0.00000	.02414	.01185	-.00447	-.02860
.02500	.19642	.20228	0.00000	0.00000	.02687	.00991	-.00236	-.02923
.02500	.21595	.22562	0.00000	0.00000	.03803	.00898	.00579	-.03225
.02500	.23549	.24896	0.00000	0.00000	.04920	.00804	.01394	-.03526
.02500	.25502	.27231	0.00000	0.00000	.06036	.00809	.02307	-.03730
.02500	.27455	.29565	0.00000	0.00000	.07153	.00818	.03224	-.03928
.02500	.29409	.31899	0.00000	0.00000	.07535	.00733	.03509	-.04026
.02500	.31362	.34234	0.00000	0.00000	.07750	.00625	.03649	-.04101
.02500	.33315	.36568	0.00000	0.00000	.07965	.00512	.03783	-.04182
.02500	.35269	.38902	0.00000	0.00000	.08180	.00395	.03914	-.04266
.02500	.37222	.41237	0.00000	0.00000	.08043	.00318	.03845	-.04198
.02500	.39175	.43571	0.00000	0.00000	.07594	.00276	.03600	-.03994
.02500	.41129	.45906	0.00000	0.00000	.07145	.00225	.03345	-.03800
.02500	.43082	.48240	0.00000	0.00000	.06696	.00159	.03077	-.03620
.02500	.45035	.50574	0.00000	0.00000	.06418	.00104	.02875	-.03543
.02500	.46989	.52909	0.00000	0.00000	.06663	.00081	.02881	-.03782
.02500	.48942	.55243	0.00000	0.00000	.06908	.00065	.02894	-.04014

.02500	.50895	.57577	0.00000	0.00000	.07153	.00112	.02970	-.04183
.02500	.52849	.59912	0.00000	0.00000	.07398	.00158	.03046	-.04352
.02500	.54802	.62246	0.00000	0.00000	.07774	.00102	.03196	-.04578
.02500	.56755	.64580	0.00000	0.00000	.08155	.00041	.03349	-.04806
.02500	.58709	.66915	0.00000	0.00000	.08536	-.00093	.03430	-.05107
.02500	.60662	.69249	0.00000	0.00000	.08917	-.00242	.03494	-.05423
.02500	.62615	.71583	0.00000	0.00000	.08728	-.00388	.03213	-.05514
.02500	.64569	.73918	0.00000	0.00000	.08267	-.00533	.02768	-.05499
.02500	.66522	.76252	0.00000	0.00000	.07806	-.00703	.02298	-.05508
.02500	.68475	.78586	0.00000	0.00000	.07346	-.00894	.01806	-.05539
.02500	.70429	.80921	0.00000	0.00000	.07020	-.01087	.01375	-.05645
.02500	.72382	.83255	0.00000	0.00000	.06901	-.01283	.01038	-.05863
.02500	.74335	.85590	0.00000	0.00000	.06783	-.01463	.00718	-.06065
.02500	.76289	.87924	0.00000	0.00000	.06664	-.01594	.00446	-.06218
.02500	.78242	.90258	0.00000	0.00000	.06595	-.01729	.00170	-.06425
.02500	.80195	.92593	0.00000	0.00000	.06927	-.01897	-.00137	-.07063
.02500	.82148	.94927	0.00000	0.00000	.07258	-.02065	-.00444	-.07702
.02500	.84102	.97261	0.00000	0.00000	.09522	-.02283	.01132	-.08390
.02500	.86055	.99596	0.00000	0.00000	.09480	-.02502	.00400	-.09080
.02500	.86393	1.00000	0.00000	0.00000	.09475	-.02540	.00276	-.09199
.05000	.05432	0.00000	0.00000	0.00000	.02030	.01090	.00505	-.01525
.05000	.07922	.03078	0.00000	0.00000	.02658	.01145	.00223	-.02435
.05000	.09876	.05493	0.00000	0.00000	.02800	.01216	-.00146	-.02946
.05000	.11829	.07908	0.00000	0.00000	.03815	.01389	.00024	-.03791
.05000	.13782	.10323	0.00000	0.00000	.04712	.01527	.00204	-.04508
.05000	.15735	.12738	0.00000	0.00000	.04852	.01430	.00445	-.04407
.05000	.17689	.15153	0.00000	0.00000	.04991	.01337	.00690	-.04301
.05000	.19642	.17568	0.00000	0.00000	.05118	.01294	.00978	-.04140
.05000	.21595	.19983	0.00000	0.00000	.05246	.01250	.01266	-.03980
.05000	.23549	.22398	0.00000	0.00000	.05920	.01053	.01768	-.04152
.05000	.25502	.24813	0.00000	0.00000	.06597	.00855	.02271	-.04326
.05000	.27455	.27227	0.00000	0.00000	.07275	.00689	.02805	-.04470
.05000	.29409	.29642	0.00000	0.00000	.07952	.00524	.03342	-.04610
.05000	.31362	.32057	0.00000	0.00000	.08044	.00455	.03495	-.04549
.05000	.33315	.34472	0.00000	0.00000	.08034	.00402	.03582	-.04452
.05000	.35269	.36887	0.00000	0.00000	.08023	.00405	.03725	-.04298
.05000	.37222	.39302	0.00000	0.00000	.08013	.00424	.03884	-.04129
.05000	.39175	.41717	0.00000	0.00000	.07718	.00382	.03751	-.03967
.05000	.41129	.44132	0.00000	0.00000	.07307	.00314	.03500	-.03807
.05000	.43082	.46547	0.00000	0.00000	.06896	.00281	.03283	-.03614
.05000	.45035	.48962	0.00000	0.00000	.06486	.00266	.03085	-.03401
.05000	.46989	.51377	0.00000	0.00000	.06464	.00230	.02992	-.03472
.05000	.48942	.53792	0.00000	0.00000	.06735	.00177	.02978	-.03758
.05000	.50895	.56207	0.00000	0.00000	.07007	.00097	.02937	-.04070
.05000	.52849	.58621	0.00000	0.00000	.07279	-.00009	.02870	-.04409
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.05000	.58709	.65866	0.00000	0.00000	.08231	-.00251	.03108	-.05123
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.05000	.62615	.70696	0.00000	0.00000	.08657	-.00395	.03198	-.05459
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.05000	.66522	.75526	0.00000	0.00000	.07713	-.00706	.02270	-.05443
.05000	.68475	.77941	0.00000	0.00000	.07240	-.00870	.01797	-.05443
.05000	.70429	.80356	0.00000	0.00000	.06855	-.01038	.01363	-.05493
.05000	.72382	.82771	0.00000	0.00000	.06976	-.01226	.01150	-.05826
.05000	.74335	.85186	0.00000	0.00000	.07096	-.01418	.00934	-.06162
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.07500	.15735	.09715	.00189	.31305	.09998	.00590	.01934	-.08063
.07500	.17689	.12216	.00096	.15928	.10328	.00542	.02237	-.08091
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.07500	.21595	.17219	-.00300	-.49834	.09714	.00702	.02645	-.07069
.07500	.23549	.19720	-.00588	-.97664	.09098	.00907	.02820	-.06278
.07500	.25502	.22221	-.00921	-1.52977	.08977	.00899	.03116	-.05861
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.07500	.29409	.27224	-.01675	-2.78179	.06569	.00669	.03432	-.05137
.07500	.31362	.29725	-.02066	-3.43028	.08240	.00454	.03427	-.04813
.07500	.33315	.32226	-.02451	-4.07077	.08065	.00430	.03491	-.04574
.07500	.35269	.34728	-.02830	-4.69912	.07863	.00430	.03540	-.04323
.07500	.37222	.37229	-.03197	-5.30896	.07619	.00412	.03550	-.04069
.07500	.39175	.39730	-.03547	-5.89064	.07339	.00392	.03540	-.03798
.07500	.41129	.42232	-.03878	-6.44029	.06963	.00265	.03262	-.03701
.07500	.43082	.44733	-.04190	-6.95752	.06556	.00125	.02941	-.03616
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.07500	.46989	.49735	-.04755	-7.89650	.05695	.00148	.02578	-.03117
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.07500	.74335	.84753	-.06806	-11.30213	.07931	-.01628	.01156	-.06774
.07500	.76289	.87254	-.06937	-11.52109	.08852	-.01865	.01292	-.07560
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.10000	.21595	.14250	-.00326	-.52163	.10035	.00279	.02503	-.07532
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.10000	.25502	.19438	-.00745	-1.19371	.09224	.00705	.03168	-.06057
.10000	.27455	.22032	-.01002	-1.60393	.09040	.00634	.03335	-.05706
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.10000	.33315	.29814	-.01866	-2.98787	.08228	.00278	.03568	-.04660
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.10000	.37222	.35002	-.02457	-3.93396	.07862	.00310	.03593	-.04268
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.10000	.46989	.47972	-.03814	-6.10674	.05444	.00160	.02483	-.02962
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.10000	[illegible]	[illegible]	[illegible]	-7.[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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.10000	.74335	.84288	-.05920	-9.47955	.06537	-.01960	.00157	-.06380
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.12500	.17689	.05665	-.00009	-.01442	.10079	.00523	.03028	-.07051
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.12500	.35269	.29910	-.01635	-2.52172	.08841	.00163	.03849	-.04992
.12500	.37222	.32604	-.01904	-2.93659	.08611	.00144	.03798	-.04813
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.12500	.54802	.56849	-.04042	-6.23242	.04920	-.00216	.01625	-.03295
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.12500	.60662	.64931	-.04538	-6.99789	.05273	-.00735	.01233	-.04040
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.12500	.64569	.70319	-.04792	-7.38888	.05807	-.01260	.01046	-.04760
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.12500	.68475	.75707	-.04984	-7.68533	.04845	-.01601	.00115	-.04731
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.12500	.74335	.83788	-.05174	-7.97880	.05091	-.02115	-.00509	-.05600
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.12500	.82148	.94564	-.05486	-8.45944	.10972	-.02934	.02889	-.08083
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.15000	.19642	.04798	-.00036	-.05284	.10811	.01031	.04073	-.06738
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.15000	.29409	.18807	-.00774	-1.14724	.10739	.00280	.03873	-.06866
.15000	.31362	.21609	-.01026	-1.52102	.10384	.00332	.04026	-.06359
.15000	.33315	.24410	-.01297	-1.92224	.10172	.00248	.04109	-.06063
.15000	.35269	.27212	-.01584	-2.34789	.09841	.00106	.04075	-.05766
.15000	.37222	.30014	-.01877	-2.78345	.09417	-.00050	.03980	-.05438
.15000	.39175	.32816	-.02176	-3.22592	.09127	.00034	.03995	-.05132
.15000	.41129	.35617	-.02475	-3.66876	.08775	.00101	.03963	-.04812
.15000	.43082	.38419	-.02771	-4.10792	.08374	.00107	.03846	-.04529
.15000	.45035	.41221	-.03062	-4.54017	.07847	.00059	.03587	-.04260
.15000	.46989	.44023	-.03345	-4.95979	.07176	-.00059	.03156	-.04020
.15000	.48942	.46824	-.03621	-5.36818	.06480	-.00107	.02782	-.03698
.15000	.50895	.49626	-.03885	-5.75993	.05766	-.00119	.02436	-.03329
.15000	.52849	.52428	-.04138	-6.13492	.05232	-.00076	.02110	-.03122
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.15000	.64569	.69239	-.05352	-7.93498	.05043	-.01310	.00701	-.04341
.15000	.66522	.72040	-.05497	-8.14997	.04765	-.01409	.00442	-.04323
.15000	.68475	.74842	-.05625	-8.33965	.04258	-.01476	.00063	-.04194
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.15000	.72382	.80446	-.05821	-8.63012	.03486	-.01981	-.00949	-.04434
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.15000	.82148	.94454	-.06279	-9.30932	.09134	-.02878	.01941	-.07193
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.17500	.19013	0.00000	0.00000	0.00000	.06842	.09230	.10511	.03669
.17500	.21595	.03858	-.00020	-.02855	.11186	.02096	.05432	-.05755
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.17500	.29409	.15532	-.00489	-.69563	.11993	.00022	.04096	-.07897
.17500	.31362	.18451	-.00707	-1.00648	.11495	.00086	.04187	-.07308

.17500	.33315	.21370	-.00958	-1.36292	.11076	.00106	.04262	-.06814
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[illegible]	[illegible]	[illegible]	[illegible]	-2.15744	[illegible]	[illegible]	[illegible]	[illegible]
.17500	.37222	.27207	-.01516	-2.15744	.10452	.00048	.04394	-.06058
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.17500	.41129	.33044	-.02113	-3.00754	.09618	.00017	.04269	-.05348
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.17500	.48942	.44718	-.03305	-4.70417	.07374	-.00103	.03247	-.04127
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.62500	.89962	.94784	-.01049	-.51917	.08690	-.04614	.00343	-.08347
.62500	.91176	1.00000	-.01322	-.65403	.08080	-.04763	-.00426	-.08506
.70000	.76054	0.00000	0.00000	0.00000	.03946	.00130	.01523	-.02423
.70000	.78242	.12966	.01080	.38765	.16891	-.01321	.06643	-.10248
.70000	.80195	.24542	.01767	.63406	.18387	-.01594	.07250	-.11137
.70000	.82148	.36119	.02158	.77428	.18010	-.01810	.06948	-.11062

.70000	.84102	.47695	.02382	.85466	.16923	-.01851	.06410	-.10513
.70000	.86055	.59271	.02462	.88335	.15401	-.02606	.04900	-.10502
.70000	.88008	.70848	.02589	.92906	.13765	-.03475	.03222	-.10543
.70000	.89962	.82424	.02589	.92914	.14773	-.04450	.04265	-.10508
.70000	.91915	.94001	.02458	.88184	.13622	-.05108	.02705	-.10917
.70000	.92927	1.00000	.02322	.83307	.13190	-.05430	.02026	-.11163
.75000	.79717	0.00000	0.00000	0.00000	.01926	.02140	.02481	.00554
.75000	.82148	.16905	-.00044	-.01332	.16599	.00218	.08038	-.08560
.75000	.84102	.30483	.00162	.04957	.17657	-.00775	.07690	-.09967
.75000	.86055	.44061	.00274	.08390	.16991	-.01633	.06596	-.10394
.75000	.88008	.57639	.00374	.11429	.15685	-.02717	.04913	-.10772
.75000	.89962	.71217	.00456	.13952	.14072	-.03757	.03087	-.10985
.75000	.91915	.84796	.00552	.16900	.12812	-.04761	.01463	-.11348
.75000	.93868	.98374	.00520	.15898	.14589	-.05470	.02986	-.11602
.75000	.94102	1.00000	.00492	.15051	.14543	-.05550	.02850	-.11693
.80000	.82415	0.00000	0.00000	0.00000	-.00822	.04150	.03074	.03896
.80000	.84102	.13112	-.00040	-.01098	.13252	.02436	.08524	-.04728
.80000	.86055	.28292	-.00073	-.02001	.15948	.00439	.07957	-.07992
.80000	.88008	.43473	-.00036	-.00989	.15683	-.01420	.06096	-.09587
.80000	.89962	.58654	.00119	.03253	.14522	-.03048	.03989	-.10533
.80000	.91915	.73834	.00347	.09498	.13031	-.04196	.02127	-.10904
.80000	.93868	.89015	.00601	.16437	.12052	-.05169	.00682	-.11370
.80000	.95282	1.00000	.00706	.19330	.11913	-.05670	.00111	-.11801
.90000	.87811	0.00000	0.00000	0.00000	-.06759	.04540	.00261	.07019
.90000	.89962	.21880	-.00421	-.08805	.10495	.03000	.07587	-.02908
.90000	.91915	.41751	-.00457	-.09562	.12036	.00742	.06251	-.05785
.90000	.93868	.61622	-.00236	-.04923	.11586	-.01981	.03369	-.08217
.90000	.95822	.81493	.00305	.06368	.10394	-.04272	.00617	-.09777
.90000	.97641	1.00000	.00848	.17718	.10379	-.05590	-.00625	-.11005
.95000	.90509	0.00000	0.00000	0.00000	-.09918	.04620	-.01284	.08634
.95000	.91915	.16916	.00145	.02555	.06663	.03548	.06059	-.00604
.95000	.93868	.40418	-.00622	-.10999	.09962	.01613	.05919	-.04043
.95000	.95822	.63920	-.01083	-.19138	.09935	-.01075	.03350	-.06586
.95000	.97775	.87422	-.01171	-.20704	.09296	-.03935	.00321	-.08975
.95000	.98820	1.00000	-.01167	-.20626	.09453	-.05680	-.01304	-.10756
1.00000	.93207	0.00000	0.00000	0.00000	-.11596	.03490	-.03298	.08298
1.00000	.95822	.38489	-.02131	-.30787	.08974	.01632	.05273	-.03701
1.00000	.97775	.67245	-.03006	-.43416	.09364	-.00313	.03734	-.05630
1.00000	.99728	.96002	-.03505	-.50625	.09364	-.02268	.01931	-.07433
1.00000	1.00000	1.00000	-.03542	-.51161	.09531	-.02540	.01750	-.07780

MINIMUM OF $(C_{P\ \mathrm{UPPER\ SURFACE}} - C_{P\ \mathrm{LIMIT}})$ = .0185 AT 7.5000 PERCENT SEMISPAN AND 100.0000 PERCENT CHORD

DELTAT = 178.863 SEC., T = 215.681 SEC.

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

Y/B/2 \ 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
0.0000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
.0250	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
.0500	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
.0750	0.00000	.16630	.18163	-.62367	-2.10829	-3.58371	-4.78263	-5.66752	-6.25520	-6.62485	-7.13243	-8.49974
.1000	0.00000	.00403	-.10230	-.79912	-1.88718	-3.00902	-4.00117	-4.80766	-5.39719	-5.77695	-6.16086	-6.85766
.1250	0.00000	-.00491	-.09838	-.70727	-1.64437	-2.63207	-3.52281	-4.25127	-4.77861	-5.08516	-5.34348	-5.74196
.1500	0.00000	-.03862	-.18518	-.87891	-1.87596	-2.93633	-3.91957	-4.75383	-5.39322	-5.80655	-6.13194	-6.49993
.1750	0.00000	-.03240	-.17466	-.83759	-1.79916	-2.83283	-3.80642	-4.64048	-5.28416	-5.70208	-5.97417	-6.28197
.2000	0.00000	-.01684	-.13390	-.74872	-1.65728	-2.64865	-3.59249	-4.40722	-5.03630	-5.43432	-5.68780	-5.93845
.2500	0.00000	.01974	-.07134	-.62511	-1.48563	-2.44750	-3.38556	-4.21159	-4.86372	-5.28004	-5.52025	-5.77012
.3000	0.00000	.05803	-.02459	-.56201	-1.40243	-2.37491	-3.35447	-4.26204	-5.03421	-5.62440	-6.02812	-6.39148
.3500	0.00000	.08513	.03953	-.43159	-1.21389	-2.14059	-3.10266	-4.01632	-4.82574	-5.48616	-5.97235	-6.40887
.4000	0.00000	.13662	.17613	-.10503	-.71362	-1.47939	-2.30367	-3.10867	-3.83531	-4.42376	-4.93999	-5.43073
.4750	0.00000	.19323	.31608	.17228	-.31549	-.99202	-1.75841	-2.56968	-3.30083	-3.92337	-4.53006	-5.18962
.5500	0.00000	.32818	.53171	.59647	.33567	-.11437	-.69498	-1.31599	-1.92497	-2.52456	-3.15228	-3.84239
.6250	0.00000	.40330	.71902	1.08715	1.15626	1.01990	.75331	.40958	.02436	-.37779	-.81664	-1.32164
.7000	0.00000	.45558	.86225	1.53291	1.97211	2.24904	2.39403	2.47361	2.58420	2.60034	2.52261	2.32170
.7500	0.00000	-.04748	-.06590	.01153	.15634	.24203	.31978	.38693	.44809	.53314		

	.58172	.49200								
.8000	0.00000	-.01658	-.03159	-.06378	-.07487	-.05503	.02153	.13809	.28706	.46515
	.61285	.70642								
.9000	0.00000	-.13149	-.24211	-.40072	-.46719	-.46459	-.40392	-.26554	-.01318	.26251
	.54944	.84760								
.9500	0.00000	.10343	.15592	.02647	-.32016	-.61133	-.85495	-1.03183	-1.11780	-1.15702
	-1.17343	-1.16701								
1.0000	0.00000	-.33905	-.65954	-1.24481	-1.75582	-2.18661	-2.52635	-2.82071	-3.07107	-3.27721
	-3.43413	-3.54181								

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

```
969-500  17 LOAD CHECK CASE  22 SPAN STA.  WITH FUSELAGE AND 7 TERMS    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  2.500  5.000  7.500 10.000 12.500 15.000 17.500 20.000 25.000
 30.000 35.000 40.000 47.500 55.000 62.500 70.000 75.000 80.000 90.000
 95.000100.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000
  0.000   .166   .182  -.624 -2.108 -3.584 -4.783 -5.668 -6.255 -6.625
 -7.132 -8.500
  0.000   .004  -.102  -.799 -1.887 -3.009 -4.001 -4.808 -5.397 -5.777
 -6.161 -6.858
  0.000  -.005  -.098  -.707 -1.644 -2.632 -3.523 -4.251 -4.779 -5.085
 -5.343 -5.742
  0.000  -.039  -.185  -.879 -1.876 -2.936 -3.920 -4.754 -5.393 -5.807
 -6.132 -6.500
  0.000  -.032  -.175  -.838 -1.799 -2.833 -3.806 -4.640 -5.284 -5.702
 -5.974 -6.282
  0.000  -.017  -.134  -.749 -1.657 -2.649 -3.592 -4.407 -5.036 -5.434
 -5.688 -5.938
  0.000   .020  -.071  -.625 -1.486 -2.447 -3.386 -4.212 -4.864 -5.280
 -5.520 -5.770
  0.000   .058  -.025  -.562 -1.402 -2.375 -3.354 -4.262 -5.034 -5.624
 -6.028 -6.391
  0.000   .085   .040  -.432 -1.214 -2.141 -3.103 -4.016 -4.826 -5.486
 -5.972 -6.409
  0.000   .137   .176  -.105  -.714 -1.479 -2.304 -3.109 -3.835 -4.424
 -4.940 -5.431
  0.000   .193   .316   .172  -.315  -.992 -1.758 -2.570 -3.361 -3.923
 -4.530 -5.190
  0.000   .328   .532   .596   .336  -.114  -.695 -1.316 -1.925 -2.525
 -3.152 -3.842
  0.000   .403   .719  1.087  1.156  1.020   .753   .410   .024  -.376
  -.817 -1.322
  0.000   .456   .862  1.533  1.972  2.249  2.394  2.474  2.584  2.600
  2.523  2.322
  0.000  -.047  -.066   .012   .156   .242   .320   .387   .448   .533
   .582   .492
```

```
  0.000  -.017  -.032  -.064  -.075  -.055   .022   .138   .287   .465
   .613   .706
  0.000  -.131  -.242  -.401  -.467  -.465  -.404  -.266  -.013   .263
   .549   .848
  0.000   .103   .156   .026  -.320  -.611  -.855 -1.032 -1.118 -1.157
 -1.173 -1.167
  0.000  -.339  -.660 -1.245 -1.756 -2.187 -2.526 -2.821 -3.071 -3.277
 -3.434 -3.542

*****************************************************OVERLAY   4, DEPART*****************************************************

*****************************************************OVERLAY   1,  ENTER*****************************************************

   ENTER WRGEOM---WRITE GEOMETRY ON TAPE                                       DELTAT =     .274 SEC., T =   215.955 SEC.
   EXIT WRGEOM
```

UPDATED WING DEFINITION

WING CAMBER SURFACE READ INTO BASIC GEOMETRY

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

REFA = 9898.0000 CBAR = 106.4100 XBARIN = 187.0000

	XO = 77.3280			XO = 83.1040			XO = 93.1650	
	YO = 4.9688			YO = 6.6250			YO = 9.5100	
	ZO = 0.0000			ZO = 0.0000			ZO = 0.0000	
	CHORD = 166.0700			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	.1378	.5700	.5700	.0032	.5700	.5700	-.0224	.5500	.5500
5.0	.2757	.7140	.7140	.0064	.7140	.7140	-.0449	.7120	.7120
10.0	.3022	.8720	.8720	-.1633	.8720	.8720	-.2475	.8720	.8720
20.0	-1.0363	1.0500	1.0500	-1.2795	1.0500	1.0500	-1.2482	1.0540	1.0540
30.0	-3.5007	1.1450	1.1450	-3.0217	1.1450	1.1450	-2.7177	1.1560	1.1560
40.0	-5.9519	1.2000	1.2000	-4.8184	1.2000	1.2000	-4.2765	1.2130	1.2130
50.0	-7.9431	1.2300	1.2300	-6.4069	1.2300	1.2300	-5.7138	1.2350	1.2350
60.0	-9.4128	1.2490	1.2490	-7.6992	1.2490	1.2490	-6.9209	1.2370	1.2370
70.0	-10.3876	1.1700	1.1700	-8.6424	1.1700	1.1700	-7.8339	1.1270	1.1270
80.0	-11.0021	.9370	.9370	-9.2509	.9370	.9370	-8.4110	.8830	.8830
90.0	-11.8441	.5460	.5460	-9.8658	.5460	.5460	-8.8712	.5070	.5070
100.0	-14.1159	0.0000	0.0000	-10.9819	0.0000	0.0000	-9.4347	0.0000	0.0000

XO = 116.9600	XO = 168.9800	XO = 225.8100
YO = 16.3330	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	.0108	.5500	.5500	.0735	.5700	.5700	.0504	.5800	.5800
5.0	.0216	.7150	.7150	.1471	.7270	.7270	.1008	.7290	.7290
10.0	-.0950	.8760	.8760	.2389	.9020	.9020	.1930	.9110	.9110
20.0	-.7956	1.1260	1.1260	.1218	1.0980	1.0980	.3572	1.1340	1.1340
30.0	-1.8800	1.1740	1.1740	-.2601	1.2200	1.2200	.4747	1.2680	1.2680
40.0	-3.0883	1.2350	1.2350	-.7876	1.2890	1.2890	.5483	1.3430	1.3430
50.0	-4.2664	1.2500	1.2500	-1.3829	1.3150	1.3150	.5904	1.3750	1.3750
60.0	-5.3014	1.2290	1.2290	-2.0111	1.2620	1.2620	.6182	1.3200	1.3200
70.0	-6.1170	1.0870	1.0870	-2.5766	1.1050	1.1050	.6483	1.1550	1.1550
80.0	-6.6371	.8400	.8400	-3.0567	.8420	.8420	.6612	.8800	.8800
90.0	-6.9394	.4740	.4740	-3.5228	.4730	.4730	.6486	.4950	.4950
100.0	-7.2530	0.0000	0.0000	-4.0267	0.0000	0.0000	.5923	0.0000	0.0000

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

XO = 225.8100	XO = 258.2100
YO = 47.5450	YO = 66.2500
ZO = 0.0000	ZO = 0.0000
CHORD = 32.6810	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	.0504	.1340	.1340	-.0245	.1340	.1340
5.0	.1008	.2610	.2610	-.0490	.2610	.2610
10.0	.1929	.4950	.4950	-.0953	.4910	.4910
20.0	.3571	.8800	.8800	-.1798	.8800	.8800
30.0	.4745	1.1550	1.1550	-.2537	1.1550	1.1550
40.0	.5481	1.3200	1.3200	-.3159	1.2850	1.2850
50.0	.5902	1.3750	1.3750	-.3649	1.3750	1.3750
60.0	.6160	1.3200	1.3200	-.4075	1.3200	1.3200
70.0	.6481	1.1550	1.1550	-.4436	1.1550	1.1550
80.0	.6610	.8800	.8800	-.4734	.8800	.8800
90.0	.6484	.4950	.4950	-.4960	.4950	.4950
100.0	.5921	0.0000	0.0000	-.5116	0.0000	0.0000

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS

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

TIFZC= 1.00 TNOM= 0.00 SYMF= 1.00 SMCGC= -0.00

NOPCT= 12 JBYMAX= 22 RATIO= 4.153854

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

	YB2
1	0.000
2	2.500
3	5.000
4	7.500
5	10.000
6	12.500
7	15.000
8	17.500
9	20.000
10	25.000
11	30.000
12	35.000
13	40.000
14	47.500
15	55.000
16	62.500
17	70.000
18	75.000
19	80.000
20	90.000
21	95.000
22	100.000

PLANFORM BREAKPOINTS

	X	Y	Z	CHORD	AUX. CHORD
1	77.3280	0.0000	0.0000	166.0700	166.0700
2	77.3280	4.9688	0.0000	166.0700	166.0700
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	77.3280	243.3980	243.3980
1	77.3280	243.3980	243.3980
2	77.3280	243.3980	243.3980
3	77.3280	243.3980	243.3980
4	83.1040	243.2370	243.2370
5	88.8799	243.0751	243.0751
6	94.6559	242.9146	242.9146
7	100.4320	242.7580	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

FUSFLAGE 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	6.10264	117.00000	4.17000
83.33000	6.33301	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	5.80869	106.00000	-5.90000
200.00000	5.69804	102.00000	-7.46000
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
283.30000	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	16.33000	-5.80000	0.00000	2.86500	25.78656
			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	21.25000	-4.90000	0.00000	2.86500	25.78656
			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

HORIZONTAL TAIL PLANFORM

	X	Y	Z	CHORD	BY	HXLE	HXTE
1	261.0000	2.0000	-14.0000	25.0000	1	260.3889	286.0000
2	277.0000	11.0000	-14.0000	9.0000	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

WING DOWNWASH AT TAIL SHIFTED PER W-B INTSCTN

TABLE OF INPUT Z/C ORDINATES

XPCT Y/B/2	0.00 90.00	5.00 100.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00
0.0000	0.00000 -7.13200	.16600 -8.50000	.18200	-.62400	-2.10800	-3.58400	-4.78300	-5.66800	-6.25500	-6.62500
.0250	0.00000 -7.13200	.16600 -8.50000	.18200	-.62400	-2.10800	-3.58400	-4.78300	-5.66800	-6.25500	-6.62500
.0500	0.00000 -7.13200	.16600 -8.50000	.18200	-.62400	-2.10800	-3.58400	-4.78300	-5.66800	-6.25500	-6.62500
.0750	0.00000 -7.13200	.16600 -8.50000	.18200	-.62400	-2.10800	-3.58400	-4.78300	-5.66800	-6.25500	-6.62500
.1000	0.00000 -6.16100	.00400 -6.85800	-.10200	-.79900	-1.88700	-3.00900	-4.00100	-4.80800	-5.39700	-5.77700
.1250	0.00000 -5.34300	-.00500 -5.74200	-.09800	-.70700	-1.64400	-2.63200	-3.52300	-4.25100	-4.77900	-5.08500
.1500	0.00000 -6.13200	-.03900 -6.50000	-.18500	-.87900	-1.87600	-2.93600	-3.92000	-4.75400	-5.39300	-5.80700
.1750	0.00000 -5.97400	-.03200 -6.28200	-.17500	-.83800	-1.79900	-2.83300	-3.80600	-4.64000	-5.28400	-5.70200
.2000	0.00000 -5.68800	-.01700 -5.93800	-.13400	-.74900	-1.65700	-2.64900	-3.59200	-4.40700	-5.03600	-5.43400
.2500	0.00000 -5.52000	.02000 -5.77000	-.07100	-.62500	-1.48600	-2.44700	-3.38600	-4.21200	-4.86400	-5.28000
.3000	0.00000 -6.02800	.05800 -6.39100	-.02500	-.56200	-1.40200	-2.37500	-3.35400	-4.26200	-5.03400	-5.62400
.3500	0.00000 -5.97200	.08500 -6.40900	.04000	-.43200	-1.21400	-2.14100	-3.10300	-4.01600	-4.82600	-5.48600
.4000	0.00000 -4.94000	.13700 -5.43100	.17600	-.10500	-.71400	-1.47900	-2.30400	-3.10900	-3.83500	-4.42400
.4750	0.00000 -4.53000	.19300 -5.19000	.31600	.17200	-.31500	-.99200	-1.75800	-2.57000	-3.30100	-3.92300
.5500	0.00000 -3.15200	.32800 -3.84200	.53200	.59600	.33600	-.11400	-.69500	-1.31600	-1.92500	-2.52500
.6250	0.00000 -.81700	.40300 -1.32200	.71900	1.08700	1.15600	1.02000	.75300	.41000	.02400	-.37800
.7000	0.00000 2.52300	.45600 2.32200	.86200	1.53300	1.97200	2.24900	2.39400	2.47400	2.58400	2.60000
.7500	0.00000	-.04700	-.06600	.01200	.15600	.24200	.32000	.38700	.44800	.53300

	.58200	.49200								
.8000	0.00000	-.01700	-.03200	-.06400	-.07500	-.05500	.02200	.13800	.28700	.46500
	.61300	.70600								
.9000	0.00000	-.13100	-.24200	-.40100	-.46700	-.46500	-.40400	-.26600	-.01300	.26300
	.54900	.84800								
.9500	0.00000	.10300	.15600	.02600	-.32000	-.61100	-.85500	-1.03200	-1.11800	-1.15700
	-1.17300	-1.16700								
1.0000	0.00000	-.33900	-.66000	-1.24500	-1.75600	-2.18700	-2.52600	-2.82100	-3.07100	-3.27700
	-3.43400	-3.54200								

WING-FUSELAGE INTERSECTION

CHORD	X	Y	Z
0.00	79.0096	5.4510	0.0000
5.00	88.7350	5.9069	.1232
10.00	97.2051	6.0139	.0085
20.00	112.7924	5.7811	-1.1555
30.00	128.0753	5.3528	-3.3897
40.00	143.7560	4.6800	-5.9519
50.00	160.3630	4.2520	-7.9431
60.00	176.9700	4.1805	-9.4128
70.00	193.5770	4.4984	-10.3877
80.00	210.1840	4.8711	-11.0021
90.00	226.7910	4.7302	-11.8441
100.00	243.3980	3.4996	-14.1160

FUSELAGE AREAS ABOVE AND BELOW WING

PER CENT CHORD	X	AREA ABOVE	AREA BELOW
0.00	79.01	100.08	25.03
5.00	88.73	88.71	35.93
10.00	97.21	79.26	41.99
20.00	112.79	70.75	39.18
30.00	128.08	78.10	27.31
40.00	143.76	90.93	15.55
50.00	160.36	96.26	10.86
60.00	176.97	96.42	10.32
70.00	193.58	90.25	13.78
80.00	210.18	78.11	19.91
90.00	226.79	64.54	21.07
100.00	243.40	59.77	8.23

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	-2.262	-.290	.307	-.684	-1.145	-.802	-.589	-.057	.781	1.813	2.665
.025	-2.262	-.290	.307	-.684	-1.145	-.802	-.589	-.057	.781	1.813	2.665
.050	3.278	4.521	4.466	3.753	3.476	3.777	3.776	4.271	4.415	4.430	4.312
.075	3.355	4.097	3.933	3.456	3.258	3.442	3.399	3.714	3.667	3.525	3.299
.100	3.169	3.553	3.438	3.321	3.388	3.580	3.583	3.816	3.552	3.076	2.584
.125	2.526	2.631	2.488	2.412	2.430	2.481	2.364	2.417	2.124	1.677	1.281
.150	1.923	1.898	1.761	1.693	1.687	1.730	1.640	1.725	1.593	1.366	1.112
.175	1.463	1.407	1.298	1.255	1.247	1.273	1.165	1.242	1.159	1.005	.797
.200	1.130	1.064	.983	.958	.951	.968	.903	.920	.871	.762	.599
.250	.710	.644	.609	.603	.602	.613	.580	.559	.566	.512	.431
.300	.473	.422	.412	.414	.417	.423	.410	.387	.400	.380	.346
.350	.327	.292	.294	.299	.304	.306	.303	.287	.282	.287	.266
.400	.229	.211	.218	.223	.229	.230	.230	.219	.208	.208	.206
.450	.160	.160	.169	.173	.179	.182	.181	.177	.169	.162	.162
.500	.122	.126	.134	.137	.143	.147	.145	.145	.139	.134	.129
.550	.097	.101	.109	.111	.117	.122	.120	.119	.117	.112	.109
.600	.079	.084	.090	.092	.096	.100	.101	.100	.098	.096	.093
.700	.058	.062	.064	.065	.067	.069	.071	.074	.074	.073	.071
.800	.039	.041	.043	.045	.047	.048	.049	.050	.052	.054	.055
.900	.024	.025	.026	.027	.028	.029	.031	.032	.034	.035	.036
1.000	.025	.023	.021	.018	.016	.017	.017	.018	.019	.019	.020

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	-.269	-.252	-.225	-.198	-.183	-.164	-.160	-.130	-.092	-.079	-.056
.025	-.269	-.252	-.225	-.198	-.183	-.164	-.160	-.130	-.092	-.079	-.056
.050	.776	.791	.735	.711	.730	.756	.755	.761	.742	.649	.516
.075	.747	.752	.696	.665	.674	.689	.667	.681	.653	.578	.474
.100	.654	.675	.652	.656	.688	.719	.735	.735	.704	.635	.528
.125	.506	.509	.485	.480	.490	.497	.492	.472	.437	.387	.318
.150	.380	.371	.346	.337	.341	.345	.343	.336	.321	.296	.255
.175	.289	.280	.258	.250	.251	.253	.251	.245	.234	.217	.192
.200	.224	.215	.198	.192	.191	.191	.189	.185	.176	.165	.149
.250	.143	.136	.125	.121	.120	.120	.119	.117	.113	.108	.101
.300	.098	.092	.086	.084	.083	.083	.083	.082	.081	.078	.075
.350	.070	.066	.062	.061	.060	.061	.060	.060	.059	.058	.056
.400	.053	.049	.047	.046	.045	.045	.045	.045	.044	.044	.043
.450	.040	.038	.037	.036	.035	.035	.035	.035	.035	.035	.034
.500	.032	.030	.029	.029	.028	.028	.028	.028	.028	.028	.028
.550	.026	.025	.024	.024	.023	.023	.023	.023	.023	.023	.023
.600	.021	.020	.020	.020	.019	.019	.019	.019	.019	.019	.019
.700	.015	.015	.014	.014	.014	.014	.014	.014	.014	.014	.013
.800	.012	.011	.011	.011	.011	.011	.011	.011	.011	.011	.011
.900	.010	.010	.009	.009	.009	.009	.009	.009	.009	.009	.009
.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	7.019	14.279	22.134	28.658	30.349	26.534	17.932	8.322	1.008	-3.094
.100	1.653	7.019	14.279	22.134	28.658	30.349	26.534	17.932	8.322	1.008	-3.094
.200	9.126	10.482	11.409	11.617	11.156	9.863	7.906	5.900	3.921	2.106	.511
.300	6.859	6.819	6.594	6.155	5.567	4.886	3.976	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	1.636	1.558	1.462	1.359	1.256	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	.828	.779	.730	.682	.633	.582	.530
1.000	.797	.776	.750	.723	.696	.668	.634	.601	.567	.533	.500

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	1.526	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	1.539	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	.750	.733	.706	.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	.260	.261	.242	.219	.194
.700	.297	.285	.272	.259	.247	.234	.221	.208	.196	.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

LIFTING PRESSURE COEFFICIENTS DUE TO ASYMMETRIC BODY VOLUME

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	.0179	.0534	-.0138	-.0581	-.0332	.0001	.0181	.0309	.0282	.0033	-.0252
.025	.0179	.0534	-.0138	-.0581	-.0332	.0001	.0181	.0309	.0282	.0033	-.0252
.050	.0179	.0534	-.0138	-.0581	-.0332	.0001	.0181	.0309	.0282	.0033	-.0252
.075	.0179	.0534	-.0138	-.0581	-.0332	.0001	.0181	.0309	.0282	.0033	-.0252
.100	.0448	.0319	-.0307	-.0566	-.0222	.0034	.0197	.0313	.0272	.0044	-.0251
.125	.0433	.0256	-.0291	-.0510	-.0228	.0010	.0152	.0252	.0283	.0137	-.0248
.150	.0423	.0205	-.0283	-.0469	-.0235	-.0007	.0115	.0218	.0284	.0171	-.0195
.175	.0416	.0159	-.0279	-.0437	-.0239	-.0020	.0086	.0201	.0267	.0177	-.0048
.200	.0400	.0116	-.0279	-.0411	-.0242	-.0031	.0075	.0182	.0241	.0183	.0070
.250	.0351	.0055	-.0286	-.0372	-.0248	-.0048	.0062	.0129	.0175	.0218	.0144
.300	.0291	.0004	-.0298	-.0342	-.0247	-.0058	.0029	.0084	.0154	.0194	.0168
.350	.0240	-.0019	-.0281	-.0318	-.0243	-.0066	-.0001	.0056	.0120	.0148	.0185
.400	.0161	-.0040	-.0269	-.0299	-.0237	-.0093	-.0021	.0047	.0078	.0129	.0150
.450	.0075	-.0085	-.0258	-.0284	-.0232	-.0129	-.0038	.0012	.0049	.0091	.0126
.500	-.0000	-.0152	-.0252	-.0270	-.0226	-.0147	-.0049	-.0011	.0038	.0052	.0096
.550	-.0031	-.0213	-.0250	-.0258	-.0218	-.0159	-.0067	-.0027	.0008	.0043	.0058
.600	-.0108	-.0215	-.0249	-.0247	-.0212	-.0169	-.0098	-.0041	-.0015	.0017	.0041
.700	-.0202	-.0225	-.0234	-.0228	-.0203	-.0178	-.0148	-.0104	-.0056	-.0034	-.0018
.800	-.0176	-.0186	-.0202	-.0218	-.0220	-.0215	-.0197	-.0179	-.0160	-.0140	-.0109
.900	-.0025	-.0087	-.0162	-.0169	-.0177	-.0189	-.0201	-.0209	-.0208	-.0203	-.0191
1.000	.0075	.0033	-.0002	-.0014	-.0027	-.0077	-.0150	-.0155	-.0160	-.0165	-.0172

NACELLES BELOW WING WITH ORIGINS AT

X= 213.42000 Y= 16.33000 Z= -5.80000
X= 218.67000 Y= 31.25000 Z= -4.90000

FOR NACELLE(S) AT X= 213.42000 Y= 16.33000 Z= -5.80000

X	P	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

NACELLE PRESSURE FIELD

X,PER CENT CHORD AND PRESSURE COEFFICIENT
GLANCE SOLUTION

NACELLES BELOW WING

Y/B/2												
0.000	77.328	243.398										
	0.000	100.000										
	0.00000	0.00000										
.050	77.328	238.690	238.700	238.995	239.290	239.584	239.879	240.174	240.469	240.764	241.058	241.353
	241.648	241.943	242.238	242.533	242.827	243.122	243.417	243.712				
	0.000	97.165	97.171	97.349	97.526	97.704	97.881	98.059	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	.03721	.03663	.03606	.03550	.03493	.03436	.03379
	.03322	.03266	.03209	.03153	.03097	.03041	.02985	.02929				
.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	239.648	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
	97.306	97.757	98.208	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	229.786	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	83.866	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	.06088	.05645	.05202	.04762	.04329	.03909	.03500	.03100	.02709	.02357
	.02379	.02120	.01450	.00784	.00131	-.00484	-.01032	-.01497				
.246	116.926	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	85.956	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	.02382
	.02270	.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

	233.516	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				
	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	218.836	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				
	0.00000	0.00000	.06527	.06017	.05504	.04996	.04497	.04018	.03550	.03092	.02645	.02389
	.02360	.01674	.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	239.509	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	96.707	96.716	98.104	99.492	100.248	100.248				
	0.00000	0.00000	.05970	.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	231.346	232.529	232.539	233.819	235.100	236.380
	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	96.214	97.454	98.694	99.934	100.512	100.512				
	0.00000	0.00000	.05092	.04754	.04417	.04083	.03753	.03455	.08402	.07760	.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	.07040	.06375
	.05722	.05156	.04981	.04305	.03404	.02523	.01722	.01655				
.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	239.716	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
	.01426	.01146	.04967	.03699	.02510	.01428	.00288	.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	.07180	.06511	.05841	.05180	.04541	.03926	.03328	.02766	.02812	.02118
	.01081	.00072	-.00837	-.00860	.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	246.379	246.388				
	0.000	68.583	68.596	70.841	73.086	75.331	77.576	79.820	82.065	84.310	86.555	88.800
	91.045	93.290	95.535	95.625	95.638	97.883	100.128	100.139				
	0.00000	0.00000	.07180	.06511	.05840	.05180	.04541	.03925	.03327	.02765	.02813	.02116
	.01078	.00069	-.00840	-.00874	-.00879	-.01686	-.02661	-.02665				
.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	65.526	67.991	70.455	72.919	75.384	77.848	80.312	82.776	85.241	87.705
	90.169	92.634	95.098	97.562	99.196	99.210	100.196	100.196				
	0.00000	0.00000	.06910	.06252	.05594	.04946	.04320	.03718	.03131	.02612	.02621	.01837
	.00827	-.00154	-.01012	-.01821	-.02460	-.02465	-.02852	-.02852				
.550	187.073	227.153	227.163	228.602	230.041	231.480	232.918	234.357	235.796	237.235	238.673	240.112
	241.551	242.989	244.428	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	84.067
	86.348	88.628	90.908	93.189	95.469	97.750	100.030	100.652				
	0.00000	0.00000	.05808	.05368	.04929	.04494	.04066	.03654	.03251	.02856	.02469	.02167
	.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	247.856	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	.04839	.04539	.04241	.03944	.03650	.03363	.03082	.02807	.02536	.02269
	.02007	.01823	.01912	.01619	.01175	.00738	.00306	-.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	255.528				
	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	91.876	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	.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	255.196	255.806	256.415	257.024	257.634	258.243				
	0.000	72.855	72.893	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	.03650	.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	257.820	258.109
	258.398	258.687	258.976	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

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	-.000674	-.000211
CD	.000001	-.000000	-.000058	-.000004
CM	.003958	.000795	.004322	.000858

TABLE OF CAMBER CP AT BASIC ALPHA

XPCT Y/B/2	0.00 90.00	5.00 100.00	10.00	20.00	30.00	40.00	50.00	60.00	70.00	80.00
0.000	.00040 .02892	.00236 .07038	.00838	.03600	.07643	.10746	.11036	.09127	.06299	.03731
.025	.00048 .03229	.00339 .07497	.00990	.03792	.07823	.10806	.10591	.09034	.06200	.03694
.050	.00274 .04715	.00714 .09544	.01492	.04407	.08408	.11030	.10848	.08674	.05830	.03589
.075	.00903 .06774	.01487 .08898	.02389	.05884	.10286	.12044	.10589	.07559	.04626	.03514
.100	.05517 .04603	.05251 .06686	.05857	.08138	.09596	.10505	.09644	.07223	.04497	.03041
.125	.07577 .03262	.07094 .05274	.07510	.08957	.09791	.10159	.09221	.06897	.04123	.02470
.150	.09367 .02345	.08615 .04137	.08734	.09650	.10144	.10142	.09034	.06718	.03902	.02015
.175	.10319 .01791	.09504 .03303	.09508	.10343	.10590	.10242	.08569	.06602	.03683	.01610
.200	.10920 .01360	.10272 .02629	.10336	.11002	.11115	.10418	.09005	.06530	.03537	.01428
.225	.11637 .01136	.10936 .02141	.11027	.11698	.11606	.10688	.09106	.06546	.03494	.01355
.250	.11962 .00994	.11601 .01718	.11741	.12380	.12110	.11016	.09253	.06638	.03550	.01369
.275	.12620 .00806	.12477 .01354	.12591	.12981	.12656	.11343	.09463	.06798	.03758	.01446
.300	.13375 .00685	.13057 .01053	.13095	.13564	.13183	.11700	.09718	.07092	.04092	.01595
.325	.13832 .00699	.13854 .00921	.13906	.14192	.13622	.12065	.10001	.07453	.04484	.01825
.350	.14472 .00856	.14492 .00921	.14508	.14642	.14059	.12414	.10371	.07889	.04918	.02234
.375	.14861 .01266	.14816 .01161	.14824	.15047	.14413	.12823	.10837	.08336	.05431	.02777
.400	.15039 .01900	.15274 .01599	.15336	.15401	.14783	.13349	.11337	.08807	.05965	.03392

.425	.15741 .02686	.15511 .02191	.15442	.15627	.15255	.13909	.11847	.09289	.06460	.04093
.450	.15825 .03560	.15935 .02860	.15984	.16246	.15763	.14430	.12362	.09718	.06994	.04860
.475	.16098 .04450	.16397 .03646	.16524	.16715	.16235	.14934	.12786	.10166	.07627	.05696
.500	.16544 .05261	.16674 .04534	.16790	.16991	.16598	.15210	.13085	.10598	.08252	.06438
.525	.16677 .06045	.17125 .05434	.17284	.17353	.16774	.15404	.13444	.11173	.08949	.07195
.550	.17277 .06865	.17227 .05973	.17180	.17124	.16705	.15622	.13914	.11728	.09657	.08011
.575	.16962 .07771	.17192 .07041	.17327	.17336	.16945	.15857	.14199	.12217	.10324	.08812
.600	.16730 .08715	.17141 .08136	.17296	.17359	.16915	.15928	.14451	.12723	.11073	.09715
.625	.16736 .09763	.16824 .08664	.16912	.16892	.16626	.15884	.14736	.13356	.11912	.10724
.650	.16120 .10823	.16428 .09787	.16650	.16701	.16485	.15948	.15095	.14019	.12894	.11833
.675	.15613 .11859	.15868 .10986	.16063	.16325	.16316	.16052	.15564	.14847	.13905	.12859
.700	.15752 .12462	.15980 .10767	.16208	.16510	.16597	.16455	.16072	.15425	.14540	.13519
.725	.16531 .12606	.16648 .11445	.16765	.16828	.16691	.16359	.15873	.15203	.14424	.13555
.750	.15340 .12707	.15336 .12006	.15332	.15324	.15317	.15141	.14872	.14498	.13973	.13367
.775	.13624 .12501	.13764 .12089	.13871	.13994	.14077	.14085	.13967	.13706	.13353	.12913
.800	.12112 .11754	.12309 .11010	.12506	.12759	.12924	.12971	.12924	.12808	.12573	.12216
.825	.10902 .11268	.11041 .10938	.11180	.11452	.11668	.11797	.11837	.11778	.11684	.11535
.850	.09838 .10581	.09898 .10551	.09958	.10078	.10220	.10390	.10503	.10594	.10610	.10612
.875	.08311 .09988	.08440 .10084	.08568	.08807	.09029	.09224	.09408	.09576	.09742	.09867
.900	.07325	.07415	.07504	.07684	.07929	.08189	.08442	.08692	.08956	.09221

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
											.09436	.09643
.925	.07218	.07286	.07353	.07487	.07621	.07874	.08131	.08364	.08591	.08758	.08893	.09028
.950	.07256	.07399	.07542	.07778	.07904	.08029	.08110	.08189	.08250	.08297	.08316	.08253
.975	.07574	.07554	.07533	.07492	.07440	.07364	.07289	.07182	.07063	.06941	.06798	.06655
1.000	.06307	.06185	.06062	.05818	.05573	.05328	.05079	.04830	.04620	.04511	.04402	.04293

TABLE OF FLAT PLATE CP AT 1 DEG ANGLE OF ATTACK

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
0.000	.00053	.00165	.00436	.01198	.01640	.01885	.02047	.01967	.01693	.01459	.01414	.01742
.025	.00060	.00212	.00480	.01209	.01656	.01905	.02047	.01964	.01687	.01451	.01442	.01778
.050	.00199	.00376	.00633	.01247	.01711	.01987	.02069	.01946	.01663	.01435	.01560	.01958
.075	.00567	.00651	.00837	.01355	.01935	.02243	.02174	.01890	.01530	.01373	.01634	.01784
.100	.02252	.01682	.01379	.01480	.01759	.02020	.02050	.01899	.01671	.01452	.01432	.01595
.125	.03221	.02264	.01725	.01484	.01744	.01949	.01984	.01897	.01713	.01508	.01400	.01520
.150	.03995	.02706	.01975	.01531	.01746	.01928	.01944	.01909	.01755	.01560	.01420	.01482
.175	.04188	.02992	.02142	.01612	.01766	.01908	.01934	.01914	.01798	.01603	.01461	.01467
.200	.04714	.03304	.02411	.01670	.01813	.01889	.01946	.01922	.01828	.01649	.01512	.01477
.225	.05362	.03596	.02614	.01743	.01829	.01907	.01955	.01941	.01841	.01704	.01569	.01505
.250	.05256	.03846	.02794	.01813	.01850	.01945	.01969	.01957	.01857	.01759	.01627	.01531

.275	.05824 .01686	.04195 .01562	.03057	.01831	.01919	.01971	.02002	.01951	.01889	.01802
.300	.05718 .01748	.04328 .01600	.03141	.01935	.01983	.02007	.02017	.01954	.01931	.01842
.325	.06168 .01788	.04642 .01650	.03426	.02142	.02035	.02060	.02015	.01990	.01961	.01887
.350	.06768 .01815	.04971 .01710	.03671	.02310	.02119	.02074	.02043	.02034	.01991	.01941
.375	.06491 .01849	.05100 .01781	.03813	.02489	.02144	.02098	.02102	.02068	.02034	.01968
.400	.07056 .01898	.05535 .01850	.04215	.02690	.02147	.02173	.02148	.02108	.02075	.01983
.425	.06856 .01961	.05576 .01855	.04335	.02821	.02218	.02243	.02195	.02173	.02089	.02011
.450	.07314 .02014	.05940 .01905	.04552	.03116	.02304	.02300	.02279	.02198	.02105	.02062
.475	.07914 .02052	.06377 .01964	.04987	.03393	.02408	.02388	.02320	.02212	.02150	.02124
.500	.07544 .02084	.06357 .02061	.05186	.03604	.02598	.02419	.02333	.02257	.02228	.02155
.525	.08135 .02161	.06881 .02153	.05577	.03967	.02807	.02408	.02383	.02357	.02267	.02185
.550	.07881 .02266	.06827 .02295	.05778	.04061	.02978	.02466	.02499	.02404	.02304	.02268
.575	.08373 .02416	.07239 .02455	.06108	.04387	.03320	.02632	.02522	.02450	.02410	.02391
.600	.08990 .02596	.07780 .02588	.06569	.04776	.03649	.02809	.02539	.02578	.02570	.02569
.625	.08538 .02767	.07602 .02734	.06665	.04901	.03890	.03080	.02710	.02765	.02766	.02782
.650	.09032 .02934	.08096 .02863	.07147	.05414	.04411	.03605	.03069	.02951	.02982	.02994
.675	.09791 .03061	.08681 .02972	.07681	.06028	.04962	.04203	.03572	.03184	.03184	.03137
.700	.09621 .03122	.08883 .02971	.08145	.06712	.05592	.04801	.04139	.03505	.03256	.03191

.725	.10259 .03123	.09598 .03108	.08937	.07566	.06347	.05361	.04626	.03975	.03489	.03202
.750	.09504 .03263	.09078 .03175	.08653	.07733	.06659	.05761	.05013	.04377	.03793	.03455
.775	.09593 .03572	.09084 .03367	.08624	.07843	.07028	.06230	.05483	.04873	.04304	.03777
.800	.08766 .03745	.08592 .03365	.08419	.07914	.07318	.06629	.05966	.05328	.04767	.04240
.825	.08229 .04208	.08141 .03778	.08053	.07848	.07415	.06903	.06309	.05726	.05165	.04662
.850	.07832 .04759	.07759 .04421	.07687	.07541	.07303	.06549	.06511	.06042	.05547	.05097
.875	.07484 .05099	.07467 .04433	.07450	.07384	.07289	.07068	.06791	.06415	.06022	.05578
.900	.07053 .05562	.07069 .05128	.07084	.07116	.07093	.07058	.06865	.06639	.06319	.05981
.925	.06751 .05949	.06759 .05690	.06768	.06786	.06804	.06792	.06779	.06627	.06443	.06209
.950	.06197 .05794	.06278 .05097	.06358	.06482	.06522	.06563	.06525	.06485	.06341	.06130
.975	.05788 .05185	.05822 .04837	.05855	.05923	.05962	.05947	.05932	.05832	.05704	.05538
1.000	.04641 .03775	.04621 .03618	.04602	.04562	.04523	.04471	.04346	.04222	.04089	.03932

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS MACH NUMBER = 2.7000

HORIZONTAL TAIL CONTRIBUTION EXCLUDED

FORCE COEFFICIENTS

	CAMBER	FP AT 1 DEG	NAC ON WING	WING ON NAC
CD	.45556369E-02	.48442753E-03	.25709114E-03	.20665611E-03
CL	.91417717E-01	.27755641E-01	.54860437E-02	
CMXBAR	-.17180262E-02	-.33006941E-02	-.24223439E-02	

(CAMBER CL INCLUDES -.00407 DUE TO ASYMMETRIC FUSELAGE VOLUME)

INTERFERENCE DRAG COEFFICIENTS

FLAT WING PRESSURES ON CAMBERED SURFACE

CD = .10477146E-02

CAMBERED WING PRESSURES ON FLAT SURFACE

CD = .15955408E-02

NACELLE PRESSURES ON FLAT SURFACE

CD = .95749567E-04

FLAT WING PRESSURES ON NACELLE

CD = .47480742E-04

INCLUDE FUSELAGE TERMS

FORCE COEFFICIENTS

	CAMBER	FP AT 1 DEG	NAC ON WING	WING ON NAC
CD	.44974731E-02	.48074130E-03	.25709114E-03	.20665611E-03
CL	.90743590E-01	.27544436E-01	.54860437E-02	
CMXBAR	.26142037E-02	-.24026498E-02	-.24223439E-02	

(CAMBER CL INCLUDES -.00407 DUE TO ASYMMETRIC FUSELAGE VOLUME)

INTERFERENCE DRAG COEFFICIENTS

FLAT WING PRESSURES ON CAMBERED SURFACE

CD = .10294943E-02

CAMBERED WING PRESSURES ON FLAT SURFACE

CD = .15837749E-02

NACELLE PRESSURES ON FLAT SURFACE

CD = .95749567E-04

FLAT WING PRESSURES ON NACELLE

CD = .47480742E-04

POLAR W/O NAC CD = .004497 + .094875(CL - .090744) + .633642(CL - .090744)**2

POLAR WITH NAC CD = .004961 + .100075(CL - .096230) + .633642(CL - .096230)**2

	CAMBERED WING				FLAT WING	
	W/O NACELLES		WITH NACELLES		W/O NAC	WITH NAC
CL	CD	CM	CD	CM	CD	CD
0.00	.001106	.01053	.001199	.00859	0.000000	-.000009
.01	.000968	.00966	.001043	.00771	.000063	.000036
.02	.000957	.00879	.001015	.00684	.000253	.000209
.03	.001072	.00791	.001113	.00597	.000570	.000508
.04	.001315	.00704	.001337	.00510	.001014	.000934
.05	.001684	.00617	.001689	.00422	.001584	.001487
.06	.002180	.00530	.002167	.00335	.002281	.002167
.07	.002802	.00442	.002772	.00248	.003105	.002973
.08	.003551	.00355	.003504	.00161	.004055	.003906
.09	.004427	.00268	.004362	.00074	.005132	.004965
.10	.005430	.00181	.005348	-.00014	.006336	.006152
.11	.006559	.00093	.006459	-.00101	.007667	.007465
.12	.007816	.00006	.007698	-.00188	.009124	.008905
.13	.009198	-.00081	.009063	-.00275	.010709	.010471
.14	.010708	-.00168	.010555	-.00363	.012419	.012165
.15	.012344	-.00255	.012174	-.00450	.014257	.013985
.16	.014107	-.00343	.013920	-.00537	.016221	.015931
.17	.015997	-.00430	.015792	-.00624	.018312	.018005
.18	.018014	-.00517	.017791	-.00712	.020530	.020205
.19	.020157	-.00604	.019917	-.00799	.022874	.022532
.20	.022427	-.00692	.022169	-.00886	.025346	.024986

CMXBAR W/O NAC = .002614 -(.090744 -CL)(-.087228) FOR CL = 0. , CMXBAR = .010530

CMXBAR WITH NAC = .000192 -(.096230 -CL)(-.087228) FOR CL = 0. , CMXBAR = .008586

PROGRAM WING AREA= 10659.6406
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	.00067	0.00000	0.00000	.00067	.00047	0.00000	.00047
8.308	.02816	.00163	0.00000	0.00000	.00163	.00115	0.00000	.00115
12.462	.04224	.00285	0.00000	0.00000	.00285	.00200	0.00000	.00200
16.615	.05632	.00429	0.00000	0.00000	.00429	.00301	0.00000	.00301
20.769	.07040	.00579	0.00000	0.00000	.00579	.00406	0.00000	.00406
24.923	.08449	.00732	0.00000	0.00000	.00732	.00512	0.00000	.00512
29.077	.09857	.00889	0.00000	0.00000	.00889	.00621	0.00000	.00621
33.231	.11265	.01043	0.00000	0.00000	.01043	.00728	0.00000	.00728
37.385	.12673	.01195	0.00000	0.00000	.01195	.00833	0.00000	.00833
41.539	.14081	.01344	0.00000	0.00000	.01344	.00935	0.00000	.00935
45.692	.15489	.01496	0.00000	0.00000	.01496	.01037	0.00000	.01037
49.846	.16897	.01648	0.00000	0.00000	.01646	.01139	0.00000	.01139
54.000	.18305	.01795	0.00000	0.00000	.01795	.01239	0.00000	.01239
58.154	.19713	.01923	0.00000	0.00000	.01923	.01332	0.00000	.01332
62.308	.21121	.02026	0.00000	0.00000	.02026	.01412	0.00000	.01412
66.462	.22529	.02110	0.00000	0.00000	.02110	.01479	0.00000	.01479
70.616	.23937	.02175	0.00000	0.00000	.02175	.01532	0.00000	.01532
74.769	.25346	.02225	0.00000	0.00000	.02225	.01566	0.00000	.01566
78.923	.26754	.02307	0.00000	0.00000	.02307	.01605	0.00000	.01605
83.077	.28162	.02484	0.00000	0.00000	.02484	.01668	0.00000	.01668
87.231	.29570	.02828	0.00000	0.00000	.02828	.01837	0.00000	.01837
91.385	.30978	.03305	0.00000	0.00000	.03305	.02094	0.00000	.02094
95.539	.32386	.03927	0.00000	0.00000	.03927	.02429	0.00000	.02479
99.692	.33794	.04625	0.00000	0.00000	.04625	.02880	0.00000	.02880
103.846	.35202	.05410	0.00000	0.00000	.05410	.03466	0.00000	.03466
108.000	.36610	.06238	0.00000	0.00000	.06238	.04138	0.00000	.04138
112.154	.38018	.07104	0.00000	0.00000	.07104	.04891	0.00000	.04891
116.308	.39426	.08109	0.00000	0.00000	.08109	.05831	0.00000	.05831
120.462	.40834	.09171	0.00000	0.00000	.09171	.06876	0.00000	.06876
124.616	.42243	.10280	0.00000	0.00000	.10280	.08001	0.00000	.08001
128.769	.43651	.11468	0.00000	0.00000	.11468	.09250	0.00000	.09250
132.923	.45059	.12851	0.00000	0.00000	.12851	.10675	0.00000	.10675
137.077	.46467	.14388	0.00000	0.00000	.14388	.12178	0.00000	.12178
141.231	.47875	.16095	0.00000	0.00000	.16095	.13752	0.00000	.13752
145.385	.49283	.18043	0.00000	0.00000	.18043	.15499	0.00000	.15499
149.539	.50691	.20177	0.00000	0.00000	.20177	.17353	0.00000	.17353
153.693	.52099	.22466	0.00000	0.00000	.22466	.19265	0.00000	.19265
157.846	.53507	.24898	0.00000	0.00000	.24898	.21266	0.00000	.21266
162.000	.54915	.27538	0.00000	0.00000	.27538	.23454	0.00000	.23454
166.154	.56323	.30281	0.00000	0.00000	.30281	.25694	0.00000	.25694
170.308	.57732	.33127	0.00000	0.00000	.33127	.27973	0.00000	.27973
174.462	.59140	.36145	0.00000	0.00000	.36145	.30394	0.00000	.30394
178.616	.60548	.39269	0.00000	0.00000	.39269	.32916	0.00000	.32916
182.770	.61956	.42440	0.00000	0.00000	.42440	.35464	0.00000	.35464
186.923	.63364	.45621	0.00000	0.00000	.45621	.38052	0.00000	.38052
191.077	.64772	.48880	0.00000	0.00000	.48880	.40832	0.00000	.40832
195.231	.66180	.52107	0.00000	0.00000	.52107	.43634	0.00000	.43634
199.385	.67588	.55275	0.00000	0.00000	.55275	.46440	0.00000	.46440
203.539	.68996	.58415	0.00000	0.00000	.58415	.49355	0.00000	.49355
207.693	.70404	.61499	0.00000	0.00000	.61499	.52378	0.00000	.52378
211.847	.71812	.64477	0.00000	0.00000	.64477	.55401	0.00000	.55401
216.000	.73220	.67336	0.00000	0.00000	.67336	.58425	0.00000	.58425
220.154	.74629	.70156	.00046	0.00000	.70202	.61659	0.00000	.61659
224.308	.76037	.72896	.00583	0.00000	.73479	.64944	0.00000	.64944

228.462	.77445	.75638	.01608	0.00000	.77246	.68373	0.00000	.68373
232.616	.78953	.78512	.02649	0.00000	.81161	.72214	0.00000	.72214
236.770	.80261	.81523	.03791	0.00000	.85314	.76476	0.00000	.76476
240.924	.81669	.84611	.04668	0.00000	.89279	.80997	0.00000	.80997
245.077	.83077	.87449	.05235	0.00000	.92684	.85305	0.00000	.85305
249.231	.84485	.89646	.05371	0.00000	.95018	.88744	0.00000	.88744
253.385	.85893	.91586	.05525	0.00000	.97111	.92047	0.00000	.92047
257.539	.87301	.93156	.05653	0.00000	.98809	.95101	0.00000	.95101
261.693	.88709	.94231	.05701	0.00000	.99932	.97670	0.00000	.97670
265.847	.90118	.94785	.05701	0.00000	1.00486	.99483	0.00000	.99483
270.001	.91526	.94941	.05701	0.00000	1.00642	1.00392	0.00000	1.00392
274.154	.92934	.94821	.05701	0.00000	1.00522	1.00416	0.00000	1.00416
278.308	.94342	.94675	.05701	0.00000	1.00376	1.00296	0.00000	1.00296
282.462	.95750	.94543	.05701	0.00000	1.00244	1.00189	0.00000	1.00189
286.616	.97158	.94441	.05701	0.00000	1.00142	1.00109	0.00000	1.00109
290.770	.98566	.94360	.05701	0.00000	1.00061	1.00046	0.00000	1.00046
294.924	.99974	.94300	.05701	0.00000	1.00001	1.00001	0.00000	1.00001
295.000	1.00000	.94299	.05701	0.00000	1.00000	1.00000	0.00000	1.00000

969-500 17 LOAD CHECK CASE 22 SPAN STA. WITH FUSELAGE AND Z TERMS MACH NUMBER = 2.7000

HORIZONTAL TAIL ALPHA= 2.000

HORIZONTAL TAIL COEFFICIENTS BASED ON WING GEOMETRY

	AT GIVEN ALPHA	PER DEGREE
CL	-.000515	.000595
CD	-.000040	.000010
CM	.000439	-.000492

FORCE COEFFICIENTS

	CAMBER	FP AT 1 DEG	NAC ON WING	WING ON NAC
CD	.44579131E-02	.49111890E-03	.25709114E-03	.20665611E-03
CL	.90228307E-01	.28139028E-01	.54860437E-02	
CMXBAR	.30528447E-02	-.28949778E-02	-.24223439E-02	

(CAMBER CL INCLUDES -.00407 DUE TO ASYMMETRIC FUSELAGE VOLUME)

INTERFERENCE DRAG COEFFICIENTS

FLAT WING PRESSURES ON CAMBERED SURFACE — CD = .10556330E-02

CAMBERED WING PRESSURES ON FLAT SURFACE — CD = .15747817E-02

NACELLE PRESSURES ON FLAT SURFACE — CD = .95749567E-04

FLAT WING PRESSURES ON NACELLE — CD = .47480742E-04

POLAR W/O NAC CD = .004458 + .093479(CL - .090228) + .620252(CL - .090228)**2

POLAR WITH NAC CD = .004922 + .098569(CL - .095714) + .620252(CL - .095714)**2

	CAMBERED WING				FLAT WING	
	W/O NACELLES		WITH NACELLES		W/O NAC	WITH NAC
CL	CD	CM	CD	CM	CD	CD
0.00	.001072	.01234	.001169	.01048	0.000000	-.000009
.01	.000951	.01131	.001030	.00945	.000062	.000036
.02	.000952	.01028	.001014	.00842	.000248	.000205
.03	.001078	.00925	.001123	.00739	.000558	.000498
.04	.001327	.00822	.001355	.00636	.000992	.000915
.05	.001701	.00719	.001712	.00533	.001551	.001456
.06	.002199	.00616	.002192	.00430	.002233	.002121
.07	.002821	.00513	.002797	.00328	.003039	.002910
.08	.003567	.00411	.003526	.00225	.003970	.003823
.09	.004437	.00308	.004379	.00122	.005024	.004860
.10	.005431	.00205	.005355	.00019	.006203	.006022
.11	.006549	.00102	.006456	-.00084	.007505	.007307
.12	.007791	-.00001	.007681	-.00187	.008932	.008717
.13	.009157	-.00104	.009030	-.00290	.010482	.010250
.14	.010647	-.00207	.010503	-.00393	.012157	.011908
.15	.012261	-.00310	.012100	-.00495	.013956	.013689
.16	.014000	-.00413	.013822	-.00598	.015878	.015595
.17	.015862	-.00515	.015667	-.00701	.017925	.017624
.18	.017848	-.00618	.017636	-.00804	.020096	.019778
.19	.019959	-.00721	.019729	-.00907	.022391	.022056
.20	.022193	-.00824	.021947	-.01010	.024810	.024458

CMXBAR W/O NAC = .003053 -(.090228 -CL)(-.102881) FOR CL = 0. , CMXBAR = .012336

CMXBAR WITH NAC = .000631 -(.095714 -CL)(-.102881) FOR CL = 0. , CMXBAR = .010478

PROGRAM WING AREA= 10659.6406
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	.00068	0.00000	0.00000	.00068	.00046	0.00000	.00046
8.308	.02816	.00164	0.00000	0.00000	.00164	.00112	0.00000	.00112
12.462	.04224	.00286	0.00000	0.00000	.00286	.00195	0.00000	.00195
16.615	.05632	.00431	0.00000	0.00000	.00431	.00295	0.00000	.00295
20.769	.07040	.00582	0.00000	0.00000	.00582	.00397	0.00000	.00397
24.923	.08449	.00736	0.00000	0.00000	.00736	.00502	0.00000	.00502
29.077	.09857	.00894	0.00000	0.00000	.00894	.00608	0.00000	.00608
33.231	.11265	.01049	0.00000	0.00000	.01049	.00713	0.00000	.00713
37.385	.12673	.01202	0.00000	0.00000	.01202	.00815	0.00000	.00815
41.539	.14081	.01352	0.00000	0.00000	.01352	.00916	0.00000	.00916
45.692	.15489	.01504	0.00000	0.00000	.01504	.01015	0.00000	.01015
49.846	.16897	.01657	0.00000	0.00000	.01657	.01115	0.00000	.01115
54.000	.18305	.01804	0.00000	0.00000	.01804	.01213	0.00000	.01213
58.154	.19713	.01934	0.00000	0.00000	.01934	.01304	0.00000	.01304
62.308	.21121	.02037	0.00000	0.00000	.02037	.01382	0.00000	.01382
66.462	.22529	.02122	0.00000	0.00000	.02122	.01447	0.00000	.01447
70.616	.23937	.02187	0.00000	0.00000	.02187	.01499	0.00000	.01499
74.769	.25346	.02237	0.00000	0.00000	.02237	.01533	0.00000	.01533
78.923	.26754	.02319	0.00000	0.00000	.02319	.01571	0.00000	.01571
83.077	.28162	.02497	0.00000	0.00000	.02497	.01632	0.00000	.01632
87.231	.29570	.02843	0.00000	0.00000	.02843	.01799	0.00000	.01799
91.385	.30978	.03323	0.00000	0.00000	.03323	.02049	0.00000	.02049
95.539	.32386	.03948	0.00000	0.00000	.03948	.02378	0.00000	.02378
99.692	.33794	.04649	0.00000	0.00000	.04649	.02819	0.00000	.02819
103.846	.35202	.05439	0.00000	0.00000	.05439	.03393	0.00000	.03393
108.000	.36610	.06271	0.00000	0.00000	.06271	.04051	0.00000	.04051
112.154	.38018	.07142	0.00000	0.00000	.07142	.04788	0.00000	.04788
116.308	.39426	.08153	0.00000	0.00000	.08153	.05708	0.00000	.05708
120.462	.40834	.09220	0.00000	0.00000	.09220	.06731	0.00000	.06731
124.616	.42243	.10335	0.00000	0.00000	.10335	.07832	0.00000	.07832
128.769	.43651	.11529	0.00000	0.00000	.11529	.09054	0.00000	.09054
132.923	.45059	.12920	0.00000	0.00000	.12920	.10449	0.00000	.10449
137.077	.46467	.14466	0.00000	0.00000	.14466	.11920	0.00000	.11920
141.231	.47875	.16181	0.00000	0.00000	.16181	.13461	0.00000	.13461
145.385	.49283	.18140	0.00000	0.00000	.18140	.15172	0.00000	.15172
149.539	.50691	.20286	0.00000	0.00000	.20286	.16986	0.00000	.16986
153.693	.52099	.22587	0.00000	0.00000	.22587	.18858	0.00000	.18858
157.846	.53507	.25032	0.00000	0.00000	.25032	.20817	0.00000	.20817
162.000	.54915	.27686	0.00000	0.00000	.27686	.22959	0.00000	.22959
166.154	.56323	.30444	0.00000	0.00000	.30444	.25151	0.00000	.25151
170.308	.57732	.33306	0.00000	0.00000	.33306	.27382	0.00000	.27382
174.462	.59140	.36340	0.00000	0.00000	.36340	.29751	0.00000	.29751
178.616	.60548	.39480	0.00000	0.00000	.39480	.32221	0.00000	.32221
182.770	.61956	.42668	0.00000	0.00000	.42668	.34715	0.00000	.34715
186.923	.63364	.45866	0.00000	0.00000	.45866	.37247	0.00000	.37247
191.077	.64772	.49143	0.00000	0.00000	.49143	.39970	0.00000	.39970
195.231	.66180	.52387	0.00000	0.00000	.52387	.42712	0.00000	.42712
199.385	.67588	.55573	0.00000	0.00000	.55573	.45459	0.00000	.45459
203.539	.68996	.58729	0.00000	0.00000	.58729	.48312	0.00000	.48312
207.693	.70404	.61830	0.00000	0.00000	.61830	.51271	0.00000	.51271
211.847	.71812	.64824	0.00000	0.00000	.64824	.54230	0.00000	.54230
216.000	.73220	.67698	0.00000	0.00000	.67698	.57190	0.00000	.57190
220.154	.74629	.70534	.00046	0.00000	.70580	.60356	0.00000	.60356
224.308	.76037	.73288	.00586	0.00000	.73875	.63572	0.00000	.63572

228.462	.77445	.76045	.01616	0.00000	.77662	.66928	0.00000	.66928
232.616	.78853	.78935	.02663	0.00000	.81598	.70688	0.00000	.70688
236.770	.80261	.81961	.03812	0.00000	.85773	.74860	0.00000	.74860
240.924	.81669	.85066	.04693	0.00000	.89759	.79285	0.00000	.79285
245.077	.83077	.87920	.05263	0.00000	.93183	.83503	0.00000	.83503
249.231	.84485	.90129	.05400	0.00000	.95529	.86868	0.00000	.86868
253.385	.85893	.92079	.05555	0.00000	.97634	.90102	0.00000	.90102
257.539	.87301	.93657	.05684	0.00000	.99341	.93092	0.00000	.93092
261.693	.88709	.94739	.05732	-.00083	1.00387	.95606	-.00012	.95594
265.847	.90118	.95296	.05732	-.00231	1.00797	.97381	.00098	.97479
270.001	.91526	.95452	.05732	-.00207	1.00977	.98271	.00450	.98721
274.154	.92934	.95332	.05732	-.00103	1.00960	.98294	.00930	.99224
278.308	.94342	.95184	.05732	-.00056	1.00860	.98176	.01456	.99632
282.462	.95750	.95051	.05732	-.00221	1.00562	.98072	.01875	.99947
286.616	.97158	.94949	.05732	-.00538	1.00143	.97994	.02113	1.00107
290.770	.98566	.94868	.05732	-.00538	1.00061	.97932	.02113	1.00045
294.924	.99974	.94808	.05732	-.00538	1.00001	.97888	.02113	1.00001
295.000	1.00000	.94807	.05732	-.00538	1.00000	.97887	.02113	1.00000

TABLE OF COMBINED CAMBER AND FLAT PLATE CP FOR CL = .1000 ALPHA = .1369

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
0.000	.000468	.002585	.008981	.037639	.078672	.110037	.113156	.093962	.065304	.039304	.030859	.072764
.025	.000565	.003680	.010553	.039580	.080493	.110664	.112708	.093024	.064309	.038925	.034261	.077306
.050	.003012	.007654	.015789	.045777	.086420	.113025	.111314	.089401	.060581	.037859	.049285	.098123
.075	.009810	.015760	.025035	.060695	.105510	.123511	.108865	.078174	.048350	.037020	.069982	.091320
.100	.058257	.054817	.060457	.083404	.098370	.107817	.099245	.074826	.047258	.032394	.047995	.069044
.125	.080174	.074044	.077464	.091604	.100297	.104262	.094930	.071571	.043578	.026768	.034532	.054818
.150	.099141	.089853	.090041	.098596	.103835	.104061	.093004	.069790	.041424	.022290	.025390	.043401
.175	.108924	.099140	.098012	.105638	.108318	.105034	.092339	.068635	.039295	.018293	.019908	.035041
.200	.115695	.107239	.106657	.112308	.113628	.106771	.092717	.067932	.037869	.016538	.015668	.028314
.225	.123708	.114282	.113848	.119369	.118564	.109489	.093732	.068120	.037465	.015887	.013505	.023465
.250	.126814	.121278	.121232	.126278	.123632	.112818	.095229	.069063	.038046	.016102	.012164	.019275
.275	.134174	.130511	.130098	.132311	.129190	.116124	.097375	.070654	.040168	.016923	.010368	.015680
.300	.141576	.138499	.135246	.138289	.134546	.119746	.099943	.073593	.043569	.018476	.009245	.012716
.325	.146759	.144896	.143754	.144848	.139006	.123470	.102773	.077254	.047523	.020831	.009435	.011467
.350	.153981	.151728	.150101	.149581	.143487	.126978	.106510	.081670	.051902	.024996	.011049	.011546
.375	.157498	.155140	.153456	.153876	.147066	.131105	.111242	.086189	.057092	.030465	.015188	.014050
.400	.160050	.160315	.159133	.157690	.150766	.136465	.116314	.090956	.062492	.036637	.021593	.018525
.425	.166799	.162743	.160353	.160129	.155586	.142157	.121470	.095867	.067460	.043683	.029543	.024445

.450	.168260	.167482	.166069	.166724	.160784	.147449	.126737	.100185	.072819	.051418	.038355	.031207
.475	.171811	.172700	.172063	.171798	.165651	.152613	.130979	.104692	.079212	.059863	.047305	.039148
.500	.175771	.175437	.174997	.174845	.169533	.155415	.134042	.109066	.085579	.067330	.055458	.048163
.525	.177902	.180667	.180473	.178964	.171579	.157336	.137702	.114955	.092594	.074936	.063403	.057282
.550	.183555	.181615	.179707	.176803	.171130	.159597	.142556	.120570	.099724	.083215	.071749	.062871
.575	.181078	.181833	.181631	.179365	.173997	.162173	.145440	.125527	.106535	.091353	.081022	.073773
.600	.179608	.182061	.181951	.180127	.174148	.163129	.147990	.130761	.114247	.100666	.090704	.084903
.625	.179043	.178643	.178244	.175627	.171584	.163061	.151066	.137343	.122908	.111046	.101418	.090380
.650	.173560	.175361	.176284	.174417	.170890	.164411	.155151	.144234	.133019	.122424	.112241	.101791
.675	.169537	.170558	.171145	.171500	.169953	.166269	.160532	.152828	.143405	.132882	.122783	.113931
.700	.170686	.171959	.173232	.174284	.173630	.171126	.166384	.159048	.149859	.139557	.128896	.111733
.725	.179354	.179621	.179888	.178637	.175598	.170929	.165065	.157475	.149016	.139930	.130334	.118703
.750	.166411	.165789	.165167	.163829	.162281	.159299	.155583	.150971	.144926	.138401	.131533	.124410
.775	.149376	.150072	.150516	.150677	.150393	.149376	.147180	.143734	.139416	.134297	.129898	.125498
.800	.133123	.134854	.136586	.138421	.139253	.138783	.137407	.135377	.132252	.127968	.122671	.114710
.825	.120282	.121553	.122824	.125262	.126832	.127423	.127011	.125613	.123911	.121737	.118440	.114550
.850	.109097	.109598	.110100	.111102	.112199	.113413	.113944	.114215	.113689	.113094	.112327	.111560
.875	.093353	.094618	.095882	.098176	.100269	.101919	.103275	.104545	.105662	.106309	.106860	.106908
.900	.082906	.083824	.084742	.086579	.088999	.091555	.093814	.096011	.098205	.100398	.101977	.103448
.925	.081425	.082109	.082793	.084160	.085527	.088038	.090593	.092712	.094729	.096077	.097074	.098071

.950	.081039	.082580	.084121	.086656	.087965	.089275	.090035	.090771	.091175	.091364	.091095	.089506
.975	.083663	.083505	.083346	.083028	.082557	.081786	.081014	.079799	.078438	.076986	.075076	.073166
1.000	.069423	.068172	.066922	.064421	.061921	.059396	.056739	.054082	.051797	.050491	.049185	.047879

WING SPANWISE LIFT DISTRIBUTION

	CAMBERED WING	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	.018740	.014641	0.000000
.025000	.038011	.029567	.002284
.050000	.040252	.030664	.009877
.075000	.043411	.032389	.016848
.100000	.041478	.032719	.022433
.125000	.039462	.032626	.027733
.150000	.038056	.032535	.032406
.175000	.036845	.032191	.035431
.200000	.035852	.031959	.037580
.225000	.034927	.031654	.038077
.250000	.034188	.031193	.038007
.275000	.033649	.030977	.045091
.300000	.033071	.030448	.041934
.325000	.032752	.030077	.048698
.350000	.032386	.029675	.056430
.375000	.031923	.029002	.058336
.400000	.031574	.028558	.060016
.425000	.030934	.027751	.055537
.450000	.030452	.027102	.049896
.475000	.029878	.026520	.039771
.500000	.029075	.025750	.037871
.525000	.028288	.025287	.035746
.550000	.027067	.024492	.035158
.575000	.025910	.023886	.033293
.600000	.024542	.023487	.031034
.625000	.022910	.022719	.027500
.650000	.021342	.022323	.024194
.675000	.019626	.021848	.020187
.700000	.017917	.021123	.016584
.725000	.016093	.020626	.011828
.750000	.014248	.020368	.008200
.775000	.012458	.020190	.002018
.800000	.010685	.019942	0.000000
.825000	.009124	.019354	0.000000
.850000	.007573	.018483	0.000000
.875000	.006255	.017746	0.000000
.900000	.005136	.016607	0.000000
.925000	.004688	.015165	0.000000
.950000	.004391	.013532	0.000000
.975000	.003658	.011111	0.000000
1.000000	.001173	.003717	0.000000

-500A

NUMBER OF MACH-ALTITUDE COMBINATIONS = 2 NUMBER OF MACH-REYNOLDS COMBINATIONS = 0

NWAF= 8 NWAFOR= 13 NFUSOR= 19 NPOD= 2 NPODOR= 7 NFIN= 2 NFINOR= 4 NCANOR= 3

J1= 1 J2= 1 J3= 1 J4= -1 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
2	1.10	35.000	0.00000	1.00000

	XFUS	PFUS
1	0.00000	0.00000
2	16.67000	17.18460
3	33.33000	26.88060
4	50.00000	33.44260
5	66.67000	38.34400
6	83.33000	39.79150
7	100.00000	38.80010
8	116.67000	36.83980
9	133.33000	36.32450
10	150.00000	36.66880
11	166.66000	36.66880
12	183.33000	36.49710
13	200.00000	35.80180
14	216.67000	34.36920
15	233.33000	31.50790
16	250.00000	27.22900
17	266.67000	20.36390
18	283.30000	10.02650
19	295.00000	0.00000

WING PLANFORM

	X	Y	Z	CHORD LENGTH
1	77.32800	4.96880	0.00000	166.07000
2	83.10400	6.62500	0.00000	160.13300
3	93.34500	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	258.72000	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 52

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	32.0670	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	32.0670	3.4200	21.4885
7	35.0400	3.4200	21.4885

INPUT DATA FOR FIN 1

ROOT AIRFOIL

225.80000	47.55000	0.00000	38.75000

TIP AIRFOIL

262.50000	47.55000	10.00000	5.00000

INPUT DATA FOR FIN 2

ROOT AIRFOIL

270.00000	0.00000	-13.00000	24.20000

TIP AIRFOIL

282.50000	0.00000	-.90000	9.20000

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	7843.441513	8.047139	.000813
WING	18016.054169	21.935069	.002216
NACELLES	3051.292786	4.254190	.000430
FIN1	875.000000	1.296285	.000131
FIN2	404.140000	.630591	.000064
CANARD	612.000000	.951778	.000096
TOTAL	30801.928468	37.115051	.003750

DRAG COEFFICIENT CALCULATIONS

MACH NO.= 1.10000 ALTITUDE= 35000.00000

TEMPERATURE VARIATION= 0.00000 INPUT SCALE= 1.00000

	SWET	D/Q	CDF
FUSELAGE	7843.441513	11.200269	.001132
WING	18016.054169	30.185945	.003050
NACELLES	3051.292786	5.802858	.000586
FIN1	875.000000	1.760317	.000178
FIN2	404.140000	.853285	.000086
CANARD	612.000000	1.288193	.000130
TOTAL	30801.928468	51.090866	.005162

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PROGRAM CONTROL CARD
FFWD
ENTER INPTS---TAPE INPUTS
EXIT INPTS
ENTER GEOMBO---GEOMETRY INTERFACE WITH PROGRAM TA80
FAR-FIELD WAVE DRAG OPTIMIZATION BASED ON MAX. AREA
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FAR-FIELD WAVE DRAG OPTIMIZATION BASED ON MAX. AREA

FUSELAGE 1 AREA DISTRIBUTION (D/Q = 5.64631)

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	9.4868	1.3012	5.3195	52	153.4000	-3.3380	5.8408	107.1755
3	8.8500	9.2302	1.7477	9.5962	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	5.8398	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	23.6000	7.9468	3.4490	37.3701	58	171.1000	-4.8729	5.8329	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	49.6279	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	6.9197	4.4277	61.5902	62	182.9000	-5.8639	5.8102	106.0565
13	38.3500	6.6603	4.6278	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	104.3959
16	47.2000	5.8852	5.1656	83.8271	66	194.7000	-6.9231	5.7435	103.6333
17	50.1500	5.6268	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	100.2781	69	203.5500	-7.7088	5.6620	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	4.0680	6.1362	118.2910	73	215.3500	-8.7352	5.4952	94.8687
24	70.8000	3.8130	6.2048	120.9510	74	218.3000	-8.9870	5.4357	92.8242
25	73.7500	3.5580	6.2567	122.9836	75	221.2500	-9.2349	5.3665	90.4892
26	76.7000	3.3031	6.2947	124.4797	76	224.2000	-9.4828	5.2908	87.9427
27	79.6500	3.0481	6.3198	125.4756	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	125.8599	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	4.8148	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	117.5049	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	44.3960
39	115.0500	-.0020	5.8879	108.9101	89	262.5500	-12.8293	3.5538	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	-.7736	5.8020	105.7567	92	271.4000	-13.5982	2.8340	25.2313
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	2.2576	16.0117
45	132.7500	-1.5492	5.7809	104.9872	95	280.2500	-14.3432	1.9403	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	106.7199	100	295.0000	-15.7000	0.0000	0.0000

FAR-FIELD WAVE DRAG OPTIMIZATION BASED ON MAX. AREA

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(BWPFC)
25.0799	0.0000	0.0000	0.0000	0.0000	0.0000
30.4299	12.1826	12.1826	12.1826	12.1826	12.1826
35.7799	28.5733	28.5733	28.5733	28.5733	28.5733
41.1299	46.3976	46.3976	46.3976	46.3976	46.3976
46.4799	64.2596	64.2596	64.2596	64.2596	64.2596
51.8299	81.8788	81.8788	81.8788	81.8788	81.8788
57.1799	99.3691	99.3691	99.3691	99.3691	99.3691
62.5299	115.5583	115.5583	115.5583	115.5583	115.5583
67.8799	129.0135	129.0135	129.0135	129.0135	129.0135
73.2299	139.8853	139.8853	139.8853	139.8853	139.8853
78.5799	148.3366	148.5787	148.5787	148.5787	148.5787
83.9299	154.3777	159.3326	159.3326	159.3326	159.3326
89.2799	157.7145	170.1389	170.1389	170.1389	170.1389
94.6299	158.2070	179.9189	179.9189	179.9189	179.9189
99.9799	156.4359	189.4803	189.4803	189.4803	189.4803
105.3299	153.4846	198.3019	198.3019	198.3019	198.3019
110.6799	150.0538	208.6754	208.6754	208.6754	208.6754
116.0299	146.6765	219.6455	219.6455	219.6455	219.6455
121.3799	143.7128	230.6082	230.6082	230.6082	230.6082
126.7299	141.5199	242.1708	242.1708	242.1708	242.1708
132.0799	140.3103	254.0663	254.0663	254.0663	254.0663
137.4299	139.9647	266.4037	266.4037	266.4037	266.4037
142.7799	140.0481	277.8753	277.8753	277.8753	277.8753
148.1299	140.2153	288.6652	288.6652	288.6652	288.6652
153.4799	140.0873	298.0467	298.0467	298.0467	298.0467
158.8299	139.7141	306.1559	306.1559	306.1559	306.1559
164.1799	139.0588	312.4712	312.4712	312.4712	312.4712
169.5299	138.1186	316.7141	316.7141	316.7141	316.7141
174.8799	136.6939	317.9505	317.9505	317.9505	317.9505
180.2299	134.3240	315.7996	315.7996	315.7996	315.7996
185.5799	130.3446	309.5113	309.5113	309.5113	309.5113
190.9299	125.0448	299.1318	299.1318	299.1318	299.1318
196.2799	118.8412	284.3132	286.9141	286.9141	286.9141
201.6299	111.9526	265.9303	275.3266	275.3266	275.3266
206.9799	104.0157	240.2157	262.9756	262.9756	262.9756
212.3299	95.0056	209.6759	248.4415	248.4415	248.4415
217.6799	84.7320	175.6946	228.5789	228.5789	228.5789
223.0299	73.2233	142.6157	202.0174	202.0174	202.0174
228.3799	60.4757	112.8608	171.9810	172.0805	172.2912
233.7299	46.3546	84.9236	140.0782	141.0178	143.1050
239.0799	31.6746	63.0047	111.9576	114.6210	120.3595
244.4299	18.0724	44.8162	89.8770	95.0457	101.5074
249.7799	6.9649	30.2157	74.0494	81.9846	83.1883
255.1299	.1348	19.6609	63.4946	73.6541	73.6541
260.4799	-.0000	13.2643	57.0980	67.2481	67.2481
265.8299	-.0000	4.5454	48.3791	56.3208	56.3208
271.1799	-.0000	.0064	43.8401	49.4238	49.4238
276.5299	-.0000	-.0000	43.8337	47.4801	47.4801
281.8799	-.0000	-.0000	43.8337	45.9287	45.9287

287.2299	-.0000	-.0000	43.8337	44.4770	44.4770
292.5799	-.0000	-.0000	43.8337	43.8337	43.8337

INTERNAL RESTRAINT POINTS (XT*)

SN= 0.0000 SB= 43.8337 ELL= 438.8092

XF	SF
166.7475	252.0980
298.3902	60.1186
307.1664	58.1049
315.9426	56.3420
324.7188	53.8978
333.4950	51.4124
342.2711	49.6452
351.0473	48.6435
359.8235	48.1189
368.5997	47.7728
377.3759	47.1759
386.1521	46.1431
394.9282	45.1971
403.7044	44.7138
412.4806	44.4607
421.2568	44.2401
430.0330	44.0305

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
0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
8.7762	12.0020	12.0018	12.0018	12.0018	12.0018	5.6513	-6.3505
17.5524	29.2039	29.2035	29.2035	29.2035	29.2035	15.6762	-13.5273
26.3285	47.6961	47.6954	47.6954	47.6954	47.6954	28.2209	-19.4744
35.1047	65.8925	65.8915	65.8915	65.8915	65.8915	42.5391	-23.3524
43.8809	83.5417	83.5378	83.5378	83.5378	83.5378	58.1489	-25.3890
52.6571	99.5205	99.5263	99.5262	99.5262	99.5262	74.6859	-24.8403
61.4333	112.4254	112.3871	112.3870	112.3870	112.3870	91.8489	-20.5381
70.2095	121.7892	125.3886	125.3885	125.3885	125.3885	109.3729	-16.0156
78.9856	127.0695	140.1881	140.1880	140.1880	140.1880	127.0141	-13.1739
87.7618	128.1213	155.2423	155.2422	155.2420	155.2420	144.5398	-10.7023
96.5380	125.7939	170.5643	170.5640	170.5642	170.5642	161.7200	-8.8442
105.3142	121.5897	186.3067	186.3064	186.3021	186.3021	178.3211	-7.9811
114.0904	117.1366	200.4752	200.4741	200.5683	200.5683	194.0975	-6.4709
122.8666	113.5481	212.7445	212.7452	213.3427	213.3427	208.7831	-4.5596
131.6427	111.7098	223.5815	223.5636	224.9645	224.9645	222.0786	-2.8860
140.4189	111.4153	232.8360	233.2932	235.1206	235.1206	233.6307	-1.4904
149.1951	111.7403	239.6077	241.8449	243.0872	243.0872	242.9924	-.0948
157.9713	112.0879	243.6745	248.5590	249.5331	249.5330	249.5381	.0050
166.7475	112.0586	244.2037	251.2216	252.0980	252.0980	252.0980	-.0000
175.5237	111.5324	241.9135	251.0730	251.8693	251.8693	248.6092	-3.2600
184.2998	110.4487	236.5157	249.2868	250.0186	250.0186	241.1718	-8.8468
193.0760	108.6595	227.9217	245.0690	245.7738	245.7737	230.9928	-14.7810
201.8522	105.7554	216.3262	236.7438	237.3898	237.3898	218.7388	-18.6510
210.6284	101.2719	201.3533	223.7461	224.3957	224.3957	204.8977	-19.4979
219.4046	95.2899	183.5978	207.7443	208.3487	208.3487	189.8693	-18.4793
228.1808	87.8000	163.4533	189.6853	190.3173	190.3167	174.0050	-16.3117
236.9569	78.6724	142.9456	171.4427	172.0156	172.0173	157.6309	-14.3864
245.7331	67.9893	121.5347	152.5224	153.1343	153.1107	141.0641	-12.0466
254.5093	58.7121	99.0758	133.2076	133.7938	134.1782	124.6276	-9.5505
263.2855	41.9103	77.0621	114.7430	115.7128	117.6973	108.6680	-9.0293
272.0617	27.0924	55.8629	96.1733	98.3426	101.2075	93.5807	-7.6268
280.8379	13.0816	36.6908	77.9325	80.8510	83.2523	79.8585	-3.3938
289.6140	2.9601	22.2674	63.7056	66.1195	67.5446	68.2116	.6670
298.3902	-.0000	15.5400	57.7381	59.5568	60.1186	60.1186	-.0000
307.1664	-.0000	12.5568	56.7200	57.9724	58.1049	58.1049	-.0000
315.9426	-.0000	9.9877	55.4650	56.3494	56.3420	56.3420	-.0000
324.7188	-.0000	7.9120	53.1268	53.8971	53.8978	53.8978	-.0000
333.4950	-.0000	6.2798	50.6488	51.4127	51.4124	51.4124	-.0000
342.2711	-.0000	4.9514	48.7890	49.6452	49.6452	49.6452	-.0000
351.0473	-.0000	3.8282	47.6623	48.6435	48.6435	48.6435	-.0000
359.8235	-.0000	3.0144	46.8479	48.1189	48.1189	48.1189	-.0000
368.5997	-.0000	2.4435	46.2772	47.7728	47.7728	47.7728	-.0000
377.3759	-.0000	1.9626	45.7963	47.1759	47.1759	47.1759	-.0000
386.1521	-.0000	1.5555	45.3892	46.1431	46.1431	46.1431	-.0000
394.9282	-.0000	1.1896	45.0233	45.1971	45.1971	45.1971	-.0000
403.7044	-.0000	.8846	44.7183	44.7138	44.7138	44.7138	-.0000
412.4806	-.0000	.6269	44.4606	44.4607	44.4607	44.4607	-.0000
421.2568	-.0000	.4067	44.2404	44.2401	44.2401	44.2401	-.0000
430.0330	-.0000	.1968	44.0305	44.0305	44.0305	44.0305	-.0000
438.8092	-.0000	-.0000	43.8337	43.8337	43.8337	43.8337	0.0000

FAR-FIELD WAVE DRAG OPTIMIZATION BASED ON MAX. AREA

CASE NO. 1

MACH = 2.700 NX = 50 NTHETA = 36

OPTIMUM FUSELAGE AREA DISTRIBUTION WITH RESTRAINTS AT

X= 166.7475

N	X	Z	R	S	N	X	Z	R	S
0	0.0000	10.0000	0.0000	0.0000	25	219.4046	-9.0798	4.8369	73.4983
1	8.7762	9.2366	.9982	3.1304	26	228.1808	-9.8173	4.6494	67.9128
2	17.5524	8.4732	1.9285	11.6837	27	236.9569	-10.5655	4.3552	60.6882
3	26.3285	7.7094	2.7378	23.5484	28	245.7331	-11.3289	4.0924	52.6144
4	35.1047	6.9445	3.4623	37.6606	29	254.5093	-12.1058	3.6909	42.7964
5	43.8809	6.1759	4.0802	52.3001	30	263.2855	-12.8955	3.0624	29.4623
6	52.6571	5.4057	4.6918	69.1570	31	272.0617	-13.6539	2.2940	16.5324
7	61.4333	4.6318	5.3226	89.0032	32	280.8379	-14.3927	1.5605	7.6499
8	70.2095	3.8641	5.7663	104.4577	33	289.6140	-15.1936	.9837	3.0397
9	78.9856	3.1055	5.9740	112.1207	34	298.3902	-15.7000	-.0000	-.0000
10	87.7618	2.3445	6.0436	114.7455	35	307.1664	-15.7000	-.0000	-.0000
11	96.5380	1.5811	5.9984	113.0386	36	315.9426	-15.7000	-.0000	-.0000
12	105.3142	.8273	5.8626	107.9751	37	324.7188	-15.7000	-.0000	-.0000
13	114.0904	.0797	5.7271	103.0439	38	333.4950	-15.7000	-.0000	-.0000
14	122.8666	-.6830	5.6814	101.4060	39	342.2711	-15.7000	-.0000	-.0000
15	131.6427	-1.4521	5.7009	102.1024	40	351.0473	-15.7000	-.0000	-.0000
16	140.4189	-2.2124	5.7597	104.2188	41	359.8235	-15.7000	-.0000	-.0000
17	149.1951	-2.9708	5.8313	106.8268	42	368.5997	-15.7000	-.0000	-.0000
18	157.9713	-3.7386	5.8420	107.2209	43	377.3759	-15.7000	-.0000	-.0000
19	166.7475	-4.5048	5.8360	106.9976	44	386.1521	-15.7000	-.0000	-.0000
20	175.5237	-5.2444	5.7381	103.4397	45	394.9282	-15.7000	-.0000	-.0000
21	184.2998	-5.9886	5.5570	97.0143	46	403.7044	-15.7000	-.0000	-.0000
22	193.0760	-6.7770	5.3311	89.2850	47	412.4806	-15.7000	-.0000	-.0000
23	201.8522	-7.5611	5.1307	82.6997	48	421.2568	-15.7000	-.0000	-.0000
24	210.6284	-8.3245	4.9847	78.0590	49	430.0330	-15.7000	-.0000	-.0000
25	219.4046	-9.0798	4.8369	73.4983	50	438.8092	-15.7000	0.0000	0.0000

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	28.16357
1	-85.000	28.94670
2	-80.000	28.01920
3	-75.000	29.27614
4	-70.000	32.21735
5	-65.000	27.58607
6	-60.000	29.20454
7	-55.000	24.[illegible]9772
8	-50.000	21.28687
9	-45.000	18.95825
10	-40.000	17.46070
11	-35.000	15.25333
12	-30.000	14.35137
13	-25.000	12.65905
14	-20.000	1[illegible].70322
15	-15.000	11.82619
16	-10.000	11.78099
17	-5.000	11.44315
18	0.000	10.92529
19	5.000	11.66043
20	10.000	11.51566
21	15.000	9.43732
22	20.000	9.27867
23	25.000	[illegible].67515
24	30.000	7.66778
25	35.000	7.69786
26	40.000	8.61762
27	45.000	9.63512
28	50.000	9.91157
29	55.000	10.65450
30	60.000	1[illegible].20551
31	65.000	11.69679
32	70.000	12.03396
33	75.000	12.07994
34	80.000	16.49410
35	85.000	23.13706
36	90.000	47.71676

WING VOLUME CHECK

EXACT VOLUME = 17631.22221

EQUIVALENT BODY VOLUME = 17630.6[illegible]452

ENTIRE AIRCRAFT

D/Q = 16.54588

CDW = .16720426E-02

OPT. CDW* = .15231231E-02

DRAG OF TRANSFERRED AREA DISTRIBUTIONS

OPTIMUM EQ. BODY CDW* = .66673369E-03

AVERAGE EQ. BODY CDW* = .81765325E-03

POTENTIAL CDW* CHANGE = -.14891956E-03

EXIT OUT

SUCCESS STOP REACHED

NEAR-FIELD WAVE DRAG

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

PLANFORM BREAKPOINTS

	X	Y	CHORD
1	77.3280	0.0000	166.0700
2	77.3280	4.9688	166.0700
3	83.1040	6.6250	160.1320
4	93.1650	9.5100	149.7500
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	32.6810
9	258.2100	66.2500	14.4450

	XLE	XTE	Y
0	77.3280	243.3980	0.0000
1	77.3280	243.3980	1.6563
2	77.3280	243.3980	3.3125
3	77.3280	243.3980	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	11.5938
8	106.2081	242.6014	13.2500
9	111.9843	242.4449	14.9063
10	117.7603	242.3710	16.5625
11	123.5362	242.8112	18.2188
12	129.3120	243.2515	19.8750
13	135.0878	243.6917	21.5313
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	28.1563
18	163.9670	245.8929	29.8125
19	169.7430	246.4390	31.4688
20	175.5196	247.6807	33.1250
21	181.2962	248.9225	34.7813
22	187.0729	250.1642	36.4375
23	192.8495	251.4059	38.0938
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	44.7188
28	221.7328	257.6146	46.3750
29	226.6523	258.8592	48.0313
30	229.5211	260.1134	49.6875
31	232.3900	261.3675	51.3438
32	235.2589	262.6217	53.0000
33	238.1278	263.8759	54.6563
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	61.2813
38	252.4722	270.1467	62.9375
39	255.3411	271.4008	64.5938
40	258.2100	272.6550	66.2500

EMPENNAGE INPUT

CANARD 1

X	Y	Z	CHORD
261.00000	2.00000	-14.00000	25.00000
277.00000	11.00000	-14.00000	9.00000

X/C	0.000	50.000	100.000
Z/C	0.000	1.500	0.000

FIN 1

X	Y	Z	CHORD
225.80000	47.55000	0.00000	38.75000
262.50000	47.55000	10.00000	5.00000

X/C	0.000	32.500	67.500	100.000
Z/C	0.000	1.500	1.500	0.000

FIN 2

X	Y	Z	CHORD
270.00000	0.00000	-13.00000	24.20000
282.50000	0.00000	-.90000	9.20000

X/C	0.000	32.500	67.500	100.000
Z/C	0.000	1.500	1.500	0.000

WING-BODY INTERSECTION

CHORD	X	Y	Z	T
0.00	77.332785	4.970000	-3.120741	0.000000
2.50	81.483827	4.970000	-2.712181	1.693149
5.00	85.635473	4.970000	-2.382362	2.371418
10.00	93.939755	4.970000	-1.853436	2.856155
20.00	110.545325	4.970000	-1.680796	3.467350
30.00	127.151895	4.970000	-2.438581	3.802934
40.00	143.758464	4.970000	-3.341331	3.955577
50.00	160.365034	4.970000	-3.866869	4.065216
60.00	176.971604	4.970000	-3.922773	4.148321
70.00	193.578174	4.970000	-3.477534	3.685937
80.00	210.184744	4.970000	-2.769022	3.112071
90.00	226.791314	4.970000	-2.481563	1.813437
100.00	243.397883	4.970000	-2.439228	0.000000

TABLE OF THICKNESS CP FOR CANARD 1

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	.000608	.001963	.003944	.006145	.009561	.009272	.007887	.003950	-.001610	-.006792	-.011178
.100	.002078	.003379	.004974	.006672	.007940	.007768	.005402	.001553	-.003004	-.007631	-.010948
.200	.012434	.010004	.009880	.006935	.002991	-.002345	-.005908	-.007914	-.008972	-.009650	-.011083
.300	.020705	.018177	.014376	.008444	.000367	-.008693	-.015714	-.017524	-.016365	-.014236	-.012393
.400	.024442	.022545	.018247	.010955	.000759	-.010159	-.019762	-.023467	-.022544	-.019935	-.016458
.600	.029045	.026850	.021915	.014974	.004598	-.005628	-.016547	-.024522	-.030156	-.030149	-.029631
.800	.033224	.027052	.021303	.015766	.008291	.000263	-.008221	-.016994	-.023510	-.027753	-.031997
1.000	.020068	.017704	.014391	.011078	.006693	.001726	-.003242	-.006919	-.009673	-.012427	-.015180

TABLE OF THICKNESS CP FOR FIN 1

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	.023163	.016795	.009929	.001280	-.004790	-.004794	-.008411	-.016436	-.019829	-.017861	-.016626
.100	.036235	.024278	.012591	-.006819	-.009144	-.008495	-.012001	-.023460	-.026663	-.022615	-.019596
.200	.055239	.030200	.013628	-.002456	-.011758	-.012541	-.014711	-.026307	-.031862	-.028586	-.024020
.300	.042099	.031200	.014288	-.002657	-.013740	-.016217	-.019163	-.029730	-.036192	-.034809	-.030089
.400	.056024	.033882	.014953	-.003047	-.015775	-.019600	-.023724	-.030827	-.038312	-.039674	-.037908
.600	.053975	.029135	.012938	-.003203	-.016866	-.026046	-.032186	-.037063	-.041223	-.045329	-.044572
.800	.032107	.018311	.004589	-.009989	-.021011	-.028141	-.035208	-.041747	-.048285	-.048865	-.049274
1.000	-.015019	-.020436	-.025859	-.031281	-.036704	-.039279	-.041207	-.043135	-.045063	-.046990	-.048918

TABLE OF THICKNESS CP FOR FIN 2

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	.029787	.021047	.013422	.006298	-.001728	-.008557	-.014315	-.019754	-.022346	-.022974	-.023150
.100	.032664	.026993	.019037	.009503	-.001844	-.011560	-.019654	-.025878	-.029180	-.030011	-.030313
.200	.036999	.031046	.023026	.012700	.000638	-.010923	-.020824	-.028831	-.033595	-.035303	-.036406
.300	.034564	.031370	.025540	.015963	.006043	-.007390	-.019019	-.028561	-.036340	-.038957	-.040717
.400	.035787	.031670	.025973	.018889	.008231	-.003046	-.015542	-.026388	-.035770	-.039827	-.043034
.600	.031785	.028319	.024410	.017175	.009531	.000970	-.007701	-.017190	-.026773	-.033473	-.039857
.800	.026563	.024363	.019137	.013910	.007422	.000844	-.006172	-.013732	-.021292	-.027519	-.033688
1.000	.012716	.012716	.009624	.004873	.000122	-.004184	-.007670	-.011156	-.014630	-.017894	-.021159

NACELLE GEOMETRY

ORIGIN (X,Y,Z)			X	RADIUS	AREA
213.42000	16.33000	-5.80000	0.00000	2.86500	25.78696
			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	25.78696
			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

BUOYANCY FIELD OF BODY ON NACELLES

NACELLE(S) AT Y= 16.33000

NEAR-FIELD PRESSURE SIGNATURE
1 SHOCK WAVES
Y= 40.355189 CP1= 0.000000 CP2= .029985

Y	CP1	CP2
40.355189	0.000000	.029985
44.358056		.026941
50.967514		.021816
56.953502		.018574
63.511580		.014725
69.584027		.012273
75.027210		.011344
81.545952		.010316
87.331814		.006795
94.967496		.001825
102.639001		-.002773
109.728334		-.005809
117.039064		-.009060
124.015645		-.011200
130.347226		-.011614
136.916145		-.012841
142.675770		-.011815
147.250150		-.008407
151.808639		-.005138
156.925829		-.003030
161.998925		-.001347
167.987150		.000362
172.903801		.000418
179.112298		-.000328
185.276356		-.000931
191.304799		-.001212
197.208458		-.001208
203.153932		-.001304
209.326565		-.001844
215.486350		-.002301
221.699589		-.002837
227.986584		-.003453
234.144256		-.003725
240.594528		-.004593
247.356491		-.005976
254.091093		-.007146

NACELLE(S) AT Y= 31.25000

NEAR-FIELD PRESSURE SIGNATURE
1 SHOCK WAVES
Y= 72.633184 CP1= 0.000000 CP2= .021388

Y	CP1	CP2
72.633184	0.000000	.021388
74.452278		.020381
82.031820		.016503
88.605628		.014050
95.742604		.011139
102.194016		.009284

107.766961	.008582
113.440307	.007804
120.755790	.005140
129.139332	.001381
137.503162	-.002097
145.048367	-.004394
152.848147	-.006854
160.146702	-.008473
166.440510	-.008786
173.294103	-.009714
178.999332	-.008938
182.961077	-.006360
187.327736	-.003987
191.727793	-.002292
196.647764	-.001019
201.478840	.000274
207.297151	.000316
213.607780	-.000248
219.862576	-.000704
225.933295	-.000917
231.936297	-.000913
237.796222	-.000986
244.050158	-.001395
250.278686	-.001741
256.572470	-.002146

BUOYANCY FIELD OF NACELLES ON BODY

FUSELAGE AREAS IN WING REGION

Y	ABOVE WING	BELOW WING
78.99284	96.92532	28.64354
82.31416	93.61770	32.25399
85.63547	90.63376	34.43103
88.95679	87.48811	36.53056
92.27810	84.19379	38.54261
95.59941	80.76391	40.45781
98.92073	77.65352	41.57514
102.24204	76.69513	39.49853
105.56335	75.93396	37.60845
108.88467	75.36764	35.89529
112.20598	74.99460	34.35017
115.52730	75.07633	32.75428
118.84861	76.70057	30.15993
122.16992	78.39843	27.69466
125.49124	80.17202	25.25423
128.81255	82.02373	23.13465
132.13387	83.97360	21.10919
135.45518	86.02440	19.60369
138.77649	87.99889	18.09644
142.09781	89.89184	16.59217
145.41912	91.69775	15.09590
148.74044	93.29095	13.72116
152.06175	93.94674	13.13850
155.38306	94.57092	12.55165
158.70438	95.15670	11.96139
162.02569	95.70524	11.36853
165.34701	96.15138	10.85925
168.66832	95.97972	11.06222

171.98963	95.71769	11.23429
175.31095	95.36598	11.37491
178.63226	94.92536	11.48366
181.95357	94.32483	11.64254
185.27489	92.83765	12.65265
188.59620	91.21038	13.63516
191.91752	89.44971	14.58433
195.23883	87.56226	15.49569
198.56014	85.56074	16.38474
201.88146	83.46683	17.48375
205.20277	81.15091	18.48952
208.52409	78.62678	19.39454
211.84540	75.90870	20.19236
215.16671	73.06056	20.80812
218.48803	71.46499	19.92472
221.80934	69.66986	18.95574
225.13066	67.70161	17.90838
228.45197	65.56820	16.79066
231.77328	63.28486	15.62181
235.09460	61.01197	14.67352
238.41591	58.44786	13.59315
241.73723	55.61189	12.39218

NACELLE(S) AT Y= 16.33000

NEAR-FIELD PRESSURE SIGNATURE
2 SHOCK WAVES
Y= 248.570653 CP1= 0.000000 CP2= .016746
Y= 286.795522 CP1= -.019090 CP2= .001529

Y	CP1	CP2
248.570653	0.000000	.016746
249.217278		.016215
250.677085		.015031
252.147403		.013844
253.615428		.012676
255.070825		.011556
256.523024		.010447
257.974143		.009362
259.423949		.008296
260.872682		.007247
262.134503		.006615
262.794540		.007264
264.198049		.006312
266.318695		.003867
268.433262		.001472
270.538898		-.000666
272.447472		-.002743
274.300618		-.004474
276.487255		-.006882
278.719069		-.009343
280.911059		-.011672
283.059603		-.013861
286.273630		-.016287
286.795522	-.019090	.001525
287.317110		.001191
288.452036		.001019
289.583594		.000858
290.705117		.000718

291.919312		.000592
292.923533		.000488
294.015370		.000411
295.105203		.000338

NACELLE(S) AT Y= 31.25000

NEAR-FIELD PRESSURE SIGNATURE
2 SHOCK WAVES

X=	287.043461	CP1=	0.000000	CP2=	.011955
X=	327.782663	CP1=	-.013024	CP2=	.000795

Y	CP1	CP2
287.043461	0.000000	.011955
288.113022		.011371
289.761005		.010473
291.404779		.009589
293.029495		.008737
294.647418		.007903
296.261348		.007082

BUOYANCY FIELD OF NACELLES ON NACELLE

NACELLE AT Y= 16.33000 Z= -5.80000
NACELLE AFT END AT X= 248.46000

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
252.609	.00804
253.841	.00733

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

X	CP
319.567	0.00000

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

X	CP
361.416	0.00000

COMPOSITE SIGNATURE

X	CP
0.000	0.00000

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

BOUYANCY FIELD OF NACELLE ON ITSELF(IMAGE EFFECT)

NACELLE AT Y= 16.33000 Z= -5.80000

Y	CP
0.000	0.00000
232.389	0.00000
232.390	.02197
232.438	.02192
233.793	.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
246.501	.00914
247.814	.00794
249.609	.00487

BUOYANCY FIELD OF OTHER IMAGE NACELLES

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

Y	CP
288.000	0.00000

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

Y	CP
253.785	0.00000

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

Y	CP
227.704	0.00000

NO EFFECT

BOUYANCY FIELD OF NACELLE ON ITSELF(IMAGE EFFECT)

NACELLE AT Y= 31.25000 Z= -4.90000

Y	CP
0.000	0.00000
233.394	0.00000
233.395	.02422
233.696	.02384
235.006	.02220

236.317	.02058
237.637	.01896
238.958	.01736
240.271	.01582
241.584	.01431
242.899	.01282
244.214	.01136
245.531	.00992
246.713	.00906
247.455	.00995
248.729	.00864
250.551	.00529
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
332.454	0.00000

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

X	CP
363.293	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
253.581	.01268
255.036	.01155

FUSELAGE DATA AND PRESSURE FIELD ACTING ON WING

Y	R	AREA	CP	Y	F(Y)
0.000000	0.000000	0.000000	.091103	0.000000	0.000000
5.900000	1.104201	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	4.422228	61.437582	.014505	24.309084	.070684
41.300000	4.810474	72.698526	.012070	29.235393	.058931
47.200000	5.165604	83.828574	.009575	34.244732	.054454
53.100000	5.509224	95.352574	.010591	39.282910	.049518
59.000000	5.812001	106.121003	.003634	44.423575	.032618
64.900000	6.045904	114.834503	-.002900	49.736950	.008761
70.800000	6.195919	120.603895	-.010545	55.260715	-.013209
76.700000	6.287810	124.207758	-.013702	60.930253	-.027884
82.600000	6.331048	125.921846	-.017987	66.721813	-.043489
88.500000	6.300581	124.712819	-.021066	72.698224	-.053763
94.400000	6.245443	122.539577	-.020010	78.736509	-.055748
100.300000	6.167590	119.503576	-.023954	84.831763	-.061641
106.200000	6.032229	114.319412	-.019952	91.070995	-.056715
112.100000	5.925882	110.320412	-.013555	97.237964	-.040355
118.000000	5.851664	107.574327	-.006632	103.324102	-.024663
123.900000	5.810850	106.078961	-.003905	109.326462	-.014543
129.800000	5.787186	105.216692	-.001669	115.285914	-.006466
135.700000	5.792359	105.404912	.003245	121.172838	.001739
141.600000	5.815263	106.240143	.001478	127.015395	.002005
147.500000	5.831301	106.926936	.000127	132.875173	-.001576
153.400000	5.838243	107.081451	-.001267	138.757761	-.004468
159.300000	5.839292	107.123621	-.001642	144.654878	-.005817
165.200000	5.837117	107.040141	-.001933	150.560586	-.005796
171.100000	5.836885	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	5.785218	105.148821	-.004336	174.290496	-.013616
194.700000	5.745943	103.722411	-.005541	180.289247	-.016577
200.600000	5.693782	101.847774	-.004977	186.220068	-.017880
206.500000	5.636215	99.798749	-.007118	192.364443	-.022049
212.400000	5.550124	96.773228	-.009539	198.480361	-.028685
218.300000	5.435404	92.814012	-.011534	204.668077	-.034309
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	69.869160	-.016290	230.072495	-.045988
247.800000	4.447074	62.129622	-.019013	236.646794	-.050171
253.700000	4.139234	53.825726	-.019064	243.318854	-.051802
259.600000	3.772468	44.709618	-.022089	250.138699	-.054137
265.500000	3.335892	34.960219	-.022318	257.133623	-.050431
271.400000	2.847493	25.472723	-.019924	264.258523	-.042301
277.300000	2.273697	16.241080	-.017316	271.597598	-.029909
283.200000	1.607849	8.121575	-.006312	279.167536	-.005395
289.100000	.849950	2.269533	.023196	286.968337	.034512
295.000000	.000010	.000000	.164521	294.999975	.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	-.036061	-.039409	-.041007	-.028666	-.014230	-.005254	.001266	-.000412
	-.003213	-.004029	-.004318	-.006616	-.008895	-.011554	-.013617	-.018743
	-.023971	-.027168	-.027954	-.031630	-.035004			
.0500	-.021732	-.026761	-.029445	-.027074	-.015086	-.006706	-.000787	.000927
	-.001366	-.002727	-.002852	-.003839	-.005398	-.007182	-.008585	-.010970
	-.015024	-.018179	-.019627	-.020427	-.023512			
.0750	-.013569	-.020113	-.022323	-.024093	-.017641	-.008543	-.003130	.000732
	-.000237	-.001846	-.002326	-.002489	-.003790	-.005088	-.006628	-.007749
	-.010581	-.013611	-.015607	-.016134	-.017935			
.1000	-.012847	-.017923	-.019502	-.020791	-.015179	-.007417	-.002823	.000627
	-.000050	-.001472	-.002016	-.002110	-.003022	-.004066	-.005306	-.006146
	-.007993	-.010683	-.012807	-.013834	-.014192			
.1250	-.012462	-.016496	-.017624	-.018513	-.013445	-.006640	-.002625	.000554
	.000093	-.001157	-.001805	-.001846	-.002439	-.003348	-.004354	-.005302
	-.006362	-.008531	-.010629	-.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	-.013396	-.014395	-.014295	-.010165	-.005227	-.002307	.000305
	.000394	-.000543	-.001269	-.001425	-.001494	-.002018	-.002641	-.003326
	-.004089	-.004565	-.005786	-.007369	-.008715			
.2500	-.011327	-.012132	-.013166	-.012706	-.008724	-.004636	-.002194	-.000031
	.000432	-.000254	-.000914	-.001255	-.001272	-.001390	-.001967	-.002454
	-.003025	-.003652	-.003980	-.004881	-.006230			
.3000	-.010854	-.011305	-.012316	-.011405	-.007434	-.004073	-.002017	-.000189
	.000385	-.000051	-.000635	-.001026	-.001163	-.001198	-.001432	-.001890
	-.002209	-.002766	-.003322	-.003564	-.004263			
.3500	-.010214	-.010831	-.011197	-.009875	-.006344	-.003613	-.001879	-.000327
	.000348	.000119	-.000404	-.000839	-.001053	-.001076	-.001130	-.001394
	-.001771	-.002121	-.002515	-.002962	-.003226			
.4000	-.009776	-.010474	-.010214	-.008410	-.005389	-.003224	-.001767	-.000453
	.000218	.000265	-.000194	-.000582	-.000861	-.001009	-.001006	-.001063
	-.001291	-.001612	-.001896	-.002222	-.002578			
.4500	-.009619	-.009964	-.009378	-.007089	-.004529	-.002884	-.001674	-.000566
	.000292	.000322	.000013	-.000348	-.000656	-.000845	-.000951	-.000949
	-.000992	-.001132	-.001410	-.001652	-.001902			

.5000	-.009595	-.009132	-.008060	-.005806	-.003717	-.002506	-.001525	-.000589
	.000272	.000298	.000156	-.000161	-.000435	-.000693	-.000818	-.000902
	-.000930	-.000929	-.000978	-.001210	-.001432			
.6000	-.008046	-.006553	-.004952	-.003436	-.002586	-.001814	-.001171	-.000514
	.000152	.000261	.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	-.005863	-.005070	-.004310	-.003555	-.002909	-.002516	-.002124	-.001745
	-.001452	-.001159	-.000866	-.000564	-.000260	.000044	.000217	.000224
	.000232	.000240	.000227	.000134	.000041			
.9000	-.006964	-.006853	-.006742	-.006632	-.006353	-.005753	-.005153	-.004560
	-.003991	-.003422	-.002853	-.002563	-.002274	-.001985	-.001695	-.001477
	-.001262	-.001047	-.000832	-.000612	-.000389			
.9500	-.006817	-.006899	-.006891	-.006800	-.006708	-.006617	-.006525	-.006433
	-.006152	-.005646	-.005141	-.004635	-.004151	-.003672	-.003193	-.002745
	-.002504	-.002263	-.002022	-.001782	-.001566			
1.0000	-.006388	-.006453	-.006518	-.006583	-.006648	-.006713	-.006746	-.006673
	-.006600	-.006527	-.006453	-.006380	-.006307	-.006233	-.005872	-.005460
	-.005048	-.004635	-.004234	-.003843	-.003453			

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			
.0750	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	1.156000	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	.902000	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	.007172	.011599	.020427	.012819	.007865	.005974	.003529	.002812	.005289	.003304	.000610
	.000152	-.001317	-.003687	-.004127	-.007770	-.013485	-.017613	-.021552	-.026497			
.025	.002045	.007599	.013417	.014380	.009967	.007963	.008165	.005861	.003422	.002533	.001087	.000594
	.001559	.000311	-.002979	-.006005	-.010052	-.014310	-.017087	-.020648	-.025434			
.050	.010926	.011774	.015375	.013402	.012494	.008291	.004970	.003867	.004276	.002871	.002567	.001463
	-.000674	-.002246	-.003480	-.006723	-.010088	-.014021	-.018803	-.023427	-.026348			
.075	.035241	.010936	.005622	.005185	.009273	.008635	.004339	.004305	.003885	.001149	.001523	.001519
	-.001746	-.003711	-.005686	-.010280	-.013856	-.016423	-.021423	-.025755	-.027664			
.100	.063470	.007602	-.005830	.004326	.007404	.004840	.002678	.003111	.002021	.001266	.001829	-.000366
	-.003839	-.006205	-.009534	-.013781	-.017182	-.020044	-.024426	-.028057	-.029888			
.125	.093862	.006991	-.006320	.002554	.004160	.003533	.001622	.001258	.001580	.000938	.000032	-.001399
	-.003455	-.007375	-.012354	-.015596	-.018509	-.021982	-.025638	-.029706	-.032115			
.150	.133994	.005071	-.010584	-.000362	.003828	.002311	-.000514	.001041	.001054	-.000970	-.001206	-.000325
	-.004684	-.010333	-.013588	-.014760	-.019421	-.023992	-.026220	-.029063	-.032545			
.200	.050571	-.005359	-.009170	.000628	-.001291	-.000910	.000930	-.000705	-.002468	-.001060	-.001078	-.004266
	-.007965	-.011249	-.015678	-.019392	-.021709	-.025176	-.028614	-.031297	-.033276			
.250	.040388	-.005436	-.012108	-.004103	-.005151	-.004460	-.002724	-.000705	-.001555	-.003226	-.003445	-.004986
	-.008411	-.013225	-.017519	-.020188	-.023189	-.026820	-.030447	-.033070	-.034130			
.300	.027668	-.006829	-.011187	-.009568	-.006044	-.005854	-.004518	-.002754	-.003994	-.001840	-.004837	-.007419
	-.011106	-.014519	-.017332	-.022887	-.026212	-.029832	-.031124	-.034303	-.037307			
.350	.049020	.003489	-.008106	-.014617	-.010099	-.008770	-.003495	-.003807	-.004303	-.005877	-.006262	-.008351
	-.011632	-.016311	-.021072	-.023907	-.027272	-.030745	-.034898	-.036203	-.038299			
.400	.040510	.001382	-.010267	-.012448	-.011350	-.010549	-.007384	-.004559	-.005187	-.005759	-.008216	-.011057
	-.014561	-.017424	-.021129	-.025144	-.029718	-.033393	-.036282	-.039076	-.038768			
.450	.022318	-.002310	-.013220	-.015817	-.013420	-.007985	-.008674	-.006773	-.007686	-.009128	-.008486	-.012453
	-.014931	-.020065	-.023951	-.026846	-.030958	-.033339	-.037421	-.040173	-.042339			
.500	.018075	-.002695	-.013525	-.017028	-.017990	-.011116	-.007332	-.007855	-.007451	-.010319	-.012963	-.014798
	-.018562	-.021491	-.024274	-.028922	-.032109	-.036115	-.039555	-.041192	-.042446			
.600	.020839	-.001274	-.010417	-.015098	-.014386	-.014226	-.013563	-.011754	-.012966	-.015004	-.017231	-.019086
	-.021376	-.026206	-.029886	-.033133	-.037564	-.041284	-.043432	-.044421	-.045429			
.700	.001309	-.004798	-.010905	-.014763	-.015341	-.015964	-.017024	-.018084	-.018067	-.017739	-.019248	-.022903
	-.026568	-.030291	-.034013	-.038356	-.042826	-.046340	-.048955	-.051593	-.054323			
.800	.041524	.034979	.028432	.021860	.015246	.008632	.002186	-.003937	-.010666	-.015977	-.021424	-.026871
	-.031840	-.035512	-.039184	-.042754	-.045989	-.049224	-.052261	-.054495	-.056729			
.900	.045390	.041998	.038605	.035213	.031821	.026962	.021220	.015478	.009736	.003179	-.003818	-.010814

	-.017811	-.024014	-.029836	-.035658	-.041460	-.045631	-.049070	-.052509	-.055948			
.950	.044230	.042123	.040015	.036907	.033162	.029019	.024877	.020735	.016592	.011000	.005275	-.000449
	-.006174	-.011970	-.018023	-.024077	-.030130	-.036184	-.042747	-.049790	-.056832			
1.000	.034932	.032537	.030143	.027748	.025354	.022959	.020565	.018170	.015466	.012047	.008628	.005208
	.001789	-.001630	-.005037	-.008434	-.011830	-.015227	-.018623	-.022020	-.025416			

BODY PRESSURE DATA

X	CPP	CPW/B	BODY DRAG	WING INTF.
2.950000	.091103	0.000000	.058116	0.000000
8.850000	.080834	0.000000	.212810	0.000000
14.750000	.055959	0.000000	.377728	0.000000
20.650000	.036190	0.000000	.461307	0.000000
26.550000	.027675	0.000000	.536476	0.000000
32.450000	.019415	0.000000	.590219	0.000000
38.350000	.013289	0.000000	.620923	0.000000
44.250000	.010823	0.000000	.647962	0.000000
50.150000	.010083	0.000000	.671412	0.000000
56.050000	.007112	0.000000	.686400	0.000000
61.950000	.000367	0.000000	.687211	0.000000
67.850000	-.006723	0.000000	.679373	0.000000
73.750000	-.012123	0.000000	.671023	0.000000
79.650000	-.015845	0.000000	.660559	0.000000
85.550000	-.019527	.009765	.666300	-.010476
91.450000	-.020538	.013305	.674708	-.030352
97.350000	-.021982	.006848	.683901	-.040802
103.250000	-.021953	.007067	.700692	-.060526
109.150000	-.016753	.009676	.713059	-.086591
115.050000	-.010093	.011142	.719678	-.110835
120.950000	-.005268	.010623	.720051	-.117991
126.850000	-.002787	.008589	.720558	-.123697
132.750000	.000788	.006276	.720511	-.125058
138.650000	.002361	.005306	.720795	-.122733
144.550000	.000802	.004646	.720891	-.120697
150.450000	-.000570	.003863	.720862	-.119979
156.350000	-.001455	.002850	.720862	-.119979
162.250000	-.001787	.002310	.720862	-.119979
168.150000	-.001908	.002122	.720943	-.120326
174.050000	-.002249	.001426	.721077	-.120635
179.950000	-.003148	.000117	.721264	-.120660
185.850000	-.003909	-.001201	.722153	-.119663
191.750000	-.004939	-.002227	.723340	-.117710
197.650000	-.005259	-.003569	.724750	-.114219
203.550000	-.006047	-.005518	.727780	-.104129
209.450000	-.008328	-.007923	.731960	-.089621
215.350000	-.010536	-.010078	.738793	-.065773
221.250000	-.012289	-.012377	.751285	-.019862
227.150000	-.013619	-.015329	.765098	.036868
233.050000	-.013254	-.018522	.781705	.121551
238.950000	-.014302	-.021318	.803439	.239757
244.850000	-.017652	-.023706	.829761	.368746
250.750000	-.019039	-.024824	.866676	.544379
256.650000	-.020576	-.024965	.908904	.731325
262.550000	-.022204	-.022138	.949951	.880658
268.450000	-.021121	-.012199	.997956	.981828
274.350000	-.018620	-.004827	1.034675	1.016561
280.250000	-.011814	.000313	1.052491	1.014840
286.150000	.008443	.002365	1.041377	1.003479
292.050000	.093860	.002163	1.000000	1.000000

NEAR-FIELD WAVE DRAG MACH NO.= 2.7000

SECTION DRAG COEFFICIENTS

Y/B/2	CDW/C	CDBGW/C	CDNCW/C	SUM CD/C	DRAG FR.	CHORD
0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	166.07000
.02500	0.00000	0.00000	0.00000	0.00000	0.00000	166.07000
.05000	0.00000	0.00000	0.00000	0.00000	0.00000	166.07000
.07500	.00046	-.00003	-.00005	.00039	.02962	166.07000
.10000	.00133	-.00007	-.00013	.00113	.08378	160.13300
.12500	.00148	-.00010	-.00015	.00122	.08729	154.19519
.15000	.00152	-.00013	-.00017	.00122	.08365	148.25869
.20000	.00096	-.00016	-.00019	.00062	.03885	136.39331
.25000	.00097	-.00018	-.00019	.00061	.03502	124.61067
.30000	.00093	-.00018	-.00023	.00051	.02694	113.93947
.35000	.00101	-.00018	-.00037	.00046	.02174	103.26828
.40000	.00103	-.00018	-.00041	.00045	.01908	92.59709
.45000	.00109	-.00018	-.00033	.00058	.02180	81.92590
.50000	.00098	-.00017	-.00025	.00056	.01874	72.16113
.60000	.00103	-.00013	-.00026	.00064	.01594	54.02146
.70000	.00104	-.00005	-.00029	.00068	.01129	35.88180
.80000	.00184	-.00010	0.00000	.00174	.02156	27.36278
.90000	.00211	-.00014	0.00000	.00197	.01901	20.90389
.95000	.00183	-.00011	0.00000	.00171	.01400	17.67444
1.00000	.00129	-.00005	0.00000	.00123	.00818	14.44500

DRAG TERMS

CDW= .001054 CDB= .003466 CDB/W= -.000131 CDW/B= .000128 CD WING-BODY= .001517

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	.00000	-.00002
NAC-ON-ITSELF(IMAGE)	-.00001	.00000
OTHER NAC IMAGES	0.00000	-.00001
WING-ON-NACELLE CD	-.00003	-.00001
NACELLE-ON-WING CD	-.00020	

SUM NACELLE CD= .00002

EMPENNAGE DRAG COEFFICIENTS

CANARD 1 = .00003
FIN 1 = .00007
FIN 2 = .00003

TOTAL CD= .001663 REF. AREA= 9898.0000
BODY SWET= 7871.99 WING SWET= 18015.26

----TOTAL ELAPSED TIME, CP= 90.154 ----

6.0 REFERENCES

1. Sommer, Simon C. and Short, Barbara J.: Free Flight Measurements of Turbulent Boundary Layer Skin Friction in the Presence of Severe Aerodynamic Heating at Mach Numbers from 2.8 to 7.9. NACA TN 3391, 1955.

2. Craidon, Charlotte: Description of a Digital Computer Program for Airplane Configuration Plots. NASA TM X-2074, 1970.

3. Harris, Roy V., Jr.: An Analysis and Correlation of Aircraft Wave Drag. NASA TM X-947, 1964.

4. Carlson, Harry W. and Middleton, Wilbur D.: A Numerical Method for the Design of Camber Surfaces of Supersonic Wings with Arbitrary Planforms. NASA TN D-2341, 1964.

5. Sorrells, Russell and Miller, David S.: Numerical Method for Design of Minimum Drag Supersonic Wing Camber with Constraints on Pitching Moment and Surface Deformation. NASA TN D-7097, 1972.

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 254 and 255.

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 IRC 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 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
```

<u>KEY</u>	<u>FUNCTION</u>
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, WƵØRD, 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 Ƶ 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, Ƶ 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, Ƶ L, CL, XU, YU, ƵU and CU, representing the X,Y,Ƶ 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, ƵI, CI, XO, YO, ƵO and CO, representing the X,Y,Ƶ 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 (Cmo, 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 (△) 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
       CLDEIN =  .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:

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