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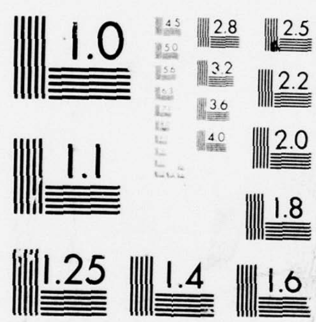
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SCHOOL OF ENGINEERING

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CALCULATION OF AERODYNAMIC PRESSURE
DISTRIBUTIONS ON ARBITRARY AIRCRAFT GEOMETRIES
USING THE WOODWARD AERODYNAMIC ANALYSIS
PROGRAM

THESIS

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PREFACE

My objective in this study was to assemble a computer capability which would calculate aerodynamic force distributions on an arbitrary aircraft geometry and tailor that capability to the preliminary design environment. The Woodward USSAERO program was selected for the analysis and no attempt was made to modify that program, although through use, certain problems with the code surfaced. These problems were corrected by Mr. Woodward and the author. What was required to use USSAERO in preliminary design was to automate the input preparation process. This was accomplished by interfacing the program with existing data bases and by writing a geometry program which allowed rapid vehicle definition in the new format. An additional aid to input preparation for USSAERO is the complete input listing of Appendix C which includes corrections and additions since it was originally published.

I wish to thank my thesis advisor, Major Stephen Koob, for his advice and assistance. I owe thanks to Captain Mike Freeman for his help with the TACT aircraft analysis. Finally I wish to thank Ed Brown, whose skill and knowledge of computer programming is reflected in the code of the Geometry and Interface programs.

Glynn E. Sisson

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LIST OF SYMBOLS

Symbol	
C_p	Pressure Coefficient
F	Function Defining Solid Surface of Aircraft
M_∞	Free Stream Mach Number
q	Perturbation Velocity
r	Distance from Point (x,y,z) to Surface of Aircraft
S	Surface of Aircraft
U_∞	Free Stream Velocity
X	Cartesian Coordinate in Free Stream Direction when Angle of Attack is Zero
Y	Cartesian Coordinate out Right Wing of Aircraft
Z	Cartesian Coordinate in Vertical Direction
ϕ	Perturbation Velocity Potential
ϕ_X	Partial Derivative of Potential in X direction
∇	Gradient Operator
$\frac{D}{Dt}$	Material Derivative
γ	Ratio of Specific Heats

ABSTRACT

There is a frequent need for accurate aerodynamic force data in preliminary design. Data must be available on many configurations, some of which deviate significantly from existing aircraft. A method which fills the gap between statistically based predictions and wind tunnel testing is a computer solution to the linearized potential flow equations of motion. The Woodward USSAERO program was selected to calculate potential flow force distributions on arbitrary aircraft geometry. Five major aspects of the program must be understood by the user to insure that the program is capable of supplying the required data. They are the differential equation, boundary conditions, singularity types, matrix operations, and force and moment calculations. In order to operate this program in the preliminary design environment, it was interfaced with existing geometry data bases with a separate Interface program. A third program, the Geometry program, was written to speed the definition of a complete aircraft configuration in a format compatible with several existing analysis programs. It defines arbitrary fuselage geometry as a series of cross sections using a Tektronix Interactive Terminal and Digitizer. It defines lifting surface geometry as a series of streamwise airfoil sections with several different airfoil shapes being available. To perform an aerodynamic analysis using the system of programs is a five step process: aircraft components (wing, body, fin, canard) must be identified; the geometry must be defined; the data must be converted into the USSAERO format; the singularity paneling must be defined; and finally, additional nongeometric data must be defined. When data from an existing geometric data base are to be used and they are compatible with USSAERO,

only the last three steps are required. The system of programs was applied to the F-111A aircraft as an example case. The results of that analysis show excellent agreement with wind tunnel data for pressure distributions on the wing at moderately high subsonic Mach numbers.

I. INTRODUCTION

Background

There is a frequent need for accurate aerodynamic force data in aircraft preliminary design. The data must be available quickly and cheaply. The source of the data must be flexible enough to handle the entire spectrum of configurations considered in preliminary design. Additionally, the data are often the starting basis for expensive detailed analyses which may require accurate initial data for numerical stability. These requirements cannot be met by either the wind tunnel or a statistically based program.

Several approximate theoretical methods which provide results by numerical means are available. The potential flow methods most closely satisfy the requirements of preliminary design and computer programs have been developed for specific applications. A general potential flow program can be made a valuable supplement to existing preliminary design data bases by careful integration with them.

Problem

The objective of this effort was to assemble a practical computational tool for potential flow aerodynamic analysis of arbitrary aircraft configurations at Wright-Patterson Air Force Base, Ohio and to apply this capability to the analysis of the F-111. The resulting computer programs provide a means of determining aerodynamic pressure distributions quickly, economically, and accurately. They bridge the gap between the wind tunnel and existing statistical prediction programs.

Approach

This section describes the steps taken to solve this problem while outlining the scope of the effort and summarizing the contents of the thesis. The format is more functional in nature than chronological, describing first the analysis program, how it was interfaced with existing data, how new data were generated, the different operating procedures, and the analysis of the F-111.

Aerodynamic Analysis Program. The potential flow computer program selected for this effort was the Unified Subsonic Supersonic Aerodynamic Analysis Program (hereafter referred to as USSAERO) written by Frank A. Woodward of Analytical Methods, Inc., 9320 S. E. Shoveland Drive., Bellevue, Washington (1-206-454-6119). The program was made available, installed and checked out on the Wright-Patterson Air Force Base computer system by Mr. Woodward, the author, and ASD/ENFTA personnel. This program was selected because of its availability, its complete configuration analysis capability, and its computer requirements which allow daytime operation. In addition, the program enjoys widespread use throughout the aerospace industry and is well documented. The program numerically solves the Prandtl-Glauert equation for potential flow of an inviscid, nonheat conducting, irrotational, perfect gas at subsonic and supersonic Mach numbers. The program solves a system of linear equations relating unknown singularity strengths to known perturbation velocities resulting in no flow through solid configuration boundaries. The system of equations is solved by iterative techniques with the final results being aerodynamic pressure distributions on the configuration. Chapter II describes the program in more detail. It is not intended to document the program as that already exists in the literature (Ref 6). Appendix C describes the inputs to USSAERO as they have changed due to additions and corrections since publication of reference six.

Geometry Preparation. The USSAERO program requires a point-by-point description of the configuration which can take from one to several days to define by hand. Without substantial reduction in this input time, the program would be ill-suited to preliminary design applications. Within the Deputy for Development Planning, Aeronautical Systems Division, there exists a preliminary design program known as ICAD (Interactive Computer Aided Design). ICAD has a point-wise description of the configuration that it is analyzing. An Interface program has been written which makes that geometry available to USSAERO in a matter of seconds. A Geometry program has been written to define a complete aircraft configuration in a format that can be used by ICAD and the Interface program to speed up the geometry definition process and to aid in the stand-alone operation of USSAERO. The Geometry program makes use of a Tektronix Interactive Terminal and Digitizing Tablet to provide computer graphic feedback on the accuracy of the geometry as it is being defined. Chapter III discusses the theory and capabilities of the Geometry and Interface programs. Appendices A and B describe the inputs to these programs.

Operating Procedures. Chapter IV describes the operating procedures for the stand-alone operation of USSAERO and for the interfaced operation of USSAERO and ICAD. Both are five step processes requiring component identification, geometry definition, conversion to the USSAERO input format, paneling definition, and nongeometric data definition. All five steps are required for any configuration analysis though the time and effort involved in each step are problem dependent. The first three steps are performed only once for each configuration whereas the last two are required for each run of the USSAERO program. The input formats of the Geometry program and USSAERO are discussed along with their affect on the operating procedures. The ultimate consideration of how the USSAERO program analyzes the configuration components and the influence it has on the procedure are discussed.

F-111 Aircraft Analysis. Whereas Chapter IV provides the information necessary to conduct an analysis on any aircraft in general, Chapter V provides the results of applying that information to a particular configuration. The aircraft analyzed was a modified F-111A which had been refitted with a supercritical wing. This highly complex configuration taxed the capabilities of all the programs and served to illustrate the procedure to follow when the actual geometry must be modified to satisfy input requirements. With the Geometry program the configuration definition process takes less than four hours. Without the Geometry program, 10 man-days would have been required.

II. USSAERO PROGRAM

The "real world" aerodynamic problem for the design engineer is the determination of surface force distributions resulting from real fluid flow about arbitrary aircraft geometries at subsonic and supersonic speeds. The efficient solution to this general problem has yet to be found. The most reasonable approach is to use an approximate solution method to solve a linearized problem. The USSAERO program is an efficient method for finding surface pressure distributions resulting from "small" perturbations to a uniform flow caused by an arbitrary aircraft configuration. This chapter discusses the governing differential equation, singularity types, boundary conditions, matrix operations, and force and moment calculations.

Differential Equation

The governing partial differential equation is the Prandtl-Glauert equation for small perturbations in potential flow:

$$(1 - M_\infty^2)\, \phi_{xx} + \phi_{yy} + \phi_{zz} = 0 \tag{1}$$

The equation is easily derived by a coordinate transformation from the equation(s) governing acoustics (Ref 4:64-65). Although this derivation gives a good account of the physics of small perturbations, it masks the effects of the simplifying assumptions. A good derivation of the full potential equation which is subsequently simplified to the above result is in Liepmann and Roshko (Ref 3:180-205). A third derivation, which does not assume small perturbations but expands the velocity potential in a small parameter power series, shows the Prandtl-Glauert equation to be the first equation of an infinite series which may or may not converge (Ref 5:121). This first term only has meaning when the series converges uniformly; the fact that the series does not converge uniformly in the transonic and hypersonic Mach number ranges shows the equation is not valid there. The equation is valid for subsonic and supersonic flow when the following conditions are observed: no large temperature gradients, no large velocity gradients outside the boundary layer, no strong shocks, and small perturbations.

Singularity Types

Due to the linearity of the governing differential equation, the solution for the specified boundary conditions is determined by superposition of elementary solutions. These solutions have a singular behavior at their point of application, thus the name singularities.

The governing differential equation can be transformed by the Prandtl-Glauert transformation to Laplace's equation whose solution can be written in the form (Ref 4:126-130):

$$\phi(x,y,z) = -\frac{1}{4\pi}\iint_S \left[\frac{1}{r}\frac{\partial\phi}{\partial n} - \phi\,\frac{\partial}{\partial n}\left(\frac{1}{r}\right)\right] dS \qquad (2)$$

The first term on the right hand side is a surface source sheet of strength $\frac{\partial\phi}{\partial n}$ and the second term is a surface doublet sheet of strength $-\phi$. The doublet is simply the derivative of a source in the direction of the normal to the surface of integration. The USSAERO program makes use of both types of singularities in solving for the unknown velocity potential.

Due to the addition of fluid to the flow from the source (or deletion in the case of a negative source), the streamlines of the flow are displaced. Thus, the source is useful in stopping and deflecting the flow as occurs near the stagnation point on a body. The program uses source sheets to simulate bodies and wing thickness. The integration suface of Equation (2) is defined as quadrilateral panels on each of which the singularity strengths are constant (Ref 6:7).

When a doublet is integrated to infinity in a direction perpendicular to its axis and then for a finite distance perpendicular to both its axis and the first integration, a horseshoe vortex is formed. These vorticity type singularities are sometimes used instead of the doublet type and have an equivalent effect. Horseshoe vorticities have been used before to represent a wing as in the Prandtl Lifting Line Theory (Ref 2:97-122). Vorticity can be integrated over a panel giving a constant pressure panel (Ref 6:19). If the vorticity is of linearily increasing strength, a linearily increasing pressure panel is formed. The USSAERO program uses constant pressure and linear pressure panels to represent the unsymmetrical effects of a lifting surface geometry.

Boundary Conditions

This section describes the origin of boundary conditions and how USSAERO uses them. This is done by showing the linking effect that the differential equation has between the known (surface condition) and the unknown (aerodynamic forces). The condition at a solid surface fluid interface is known to be (Ref 1:190-192).

$$\frac{DF}{Dt} = \frac{\partial F}{\partial t} + U_\infty \frac{\partial F}{\partial x} + \nabla\phi \cdot F = 0 \qquad (3)$$

Dividing through by $|\nabla F|$ and solving for the last term

$$\frac{\partial\phi}{\partial n} = \nabla\phi \cdot \frac{\nabla F}{|\nabla F|} = - \frac{1}{|\nabla F|} \left[\frac{\partial F}{\partial t} + U_\infty \frac{\partial F}{\partial x}\right] \tag{4}$$

Thus, if the equation defining the solid surface is known for all time, the normal derivative of the velocity potential is known as well. The pressure coefficient at a point to second order is (Ref 3:206).

$$Cp = -\left[\frac{2\phi_x}{U_\infty} + (1 - M_\infty^2) \frac{\phi_x^{\,2}}{U_\infty^{\,2}} + \frac{\phi_y^{\,2}}{U_\infty^{\,2}} + \frac{\phi_z^{\,2}}{U_\infty^{\,2}}\right] \tag{5}$$

For planar flows only the first term is significant. The unknown aerodynamic force is a function of the x-component of the perturbation velocity. The link between the surface condition and the force is the governing differential equation. It provides the condition that the velocity potential (and thus its derivatives) must satisfy at every point in the flow.

The USSAERO program follows the above procedure for finding the pressure coefficients when the nonplanar option is specified. When the planar option is specified the boundary condition is split into two parts as in thin airfoil theory, a symmetrical portion due to thickness and an unsymmetrical portion due to camber and angle of attack (Ref 1: 494-499). The former is a direct problem requiring only a simple integration; the latter is an indirect problem requiring the solution of an integral equation. By using the planar option a smaller number of equations can be used to represent the same geometry resulting in less computer time being used. The nonplanar option uses the linear pressure panel which is a better approximation to the real vorticity in the boundary layer.

Matrix Operations

This section describes the matrix operations in the USSAERO program and how they affect the results. The velocity potential at a point is a linear sum of the contributions from all singularities in the flow in terms of their unknown strengths. By writing an equation for the velocity induced at a point on the surface of the vehicle where the magnitude of that induced velocity is known, and repeating this process once for each singularity in the flow, a system of linear equations is formed. This system of equations can be solved for the magnitudes of the unknown strengths. This solution process can be any valid numerical process such as direct inversion, relaxation, or iteration. Direct inversion is the most accurate of the three, but it can require large amounts of computer time when the number of coefficients exceeds the core storage available. The USSAERO program uses direct inversion when the number of equations does not exceed sixty. For more than 60 equations four techniques are available, Blocked Jacobi, Blocked Gauss-Seidel, Successive Over Relaxation, and Controlled Over Relaxation. The last three methods are additions to the program since it was documented. No effort has been made to determine the capabilities of these methods in terms of accuracy and speed. When more than 60 equations are being solved, the program writes the coefficients on to TAPE7. The data are read from this tape and the tape rewound once for each pass through the solution procedure. This is not a very efficient mass storage technqiue. By saving this tape after it is created, but before the solution begins, a restart capability could be added to the program. As the calculation of the aerodynamic influence coefficients requires the largest portion of time for a given run, this modification has great potential.

Force and Moment Calculations

The pressure coefficient is calculated using the full isentropic relationship for Mach numbers other than zero.

$$Cp = \frac{2}{\gamma M^2} \{[1 + \frac{\gamma - 1}{2} M_\infty^2 (1 - q^2)]^{3.5} - 1\}$$

For the Mach number of zero a linearized expression is used (Ref 6:57).

$$Cp = 1 - q^2$$

This coefficient is assumed to act on the entire panel so that multiplying by the panel area and the proper angular relationships, gives lift, drag, and moment coefficients. By summing the contributions from each panel, total force and moment coefficients for body-alone, wing-along, and total configuration result. Due to this summation process, the force and moment coefficients for a particular part of the configuration cannot be isolated. Such data may be the reason for the analysis. Another problem with this process is that, by assuming flat panels, different paneling arrangements are required to get accurate lift and drag values. To get good drag data, denser paneling is required in those portions of the configuration that are most severely inclined to the flow. Since these same regions violate the assumptions made in deriving the governing differential equation, a practical limit exists on the accuracy of the calculated drag.

III. GEOMETRY PREPARATION

In order to make use of the analytical aerodynamic analysis capability of the USSAERO program early in the aircraft design cycle, it was recognized that configuration geometry in the proper input format was required quickly, and with a minimum of human effort. This chapter describes two computer programs, the Interface program and the Geometry program, which provide an interface between the ICAD design synthesis program and USSAERO by allowing both to use a common geometric input. The Interface program converts data from a very general format into the USSAERO format automatically. The Geometry program allows the user to define an entire aircraft configuration in that general format which is compatible with both ICAD and USSAERO. The remainder of this chapter describes these two programs.

Interface Program

This section shows how a single format was used to describe any vehicle component and how that format was converted into the USSAERO format without human intervention. It begins with a discussion of the theory of operation of the Interface program and how the program fits into an overall system of computer programs. The capabilities and restrictions of the general input format and the program structure are discussed.

The overall system of computer programs is illustrated in the block diagram of Figure 1. The Interface program exists as a separate program from the analysis programs and operates from a common geometrical format which exists in the ICAD data base. This format, hereafter referred to as the

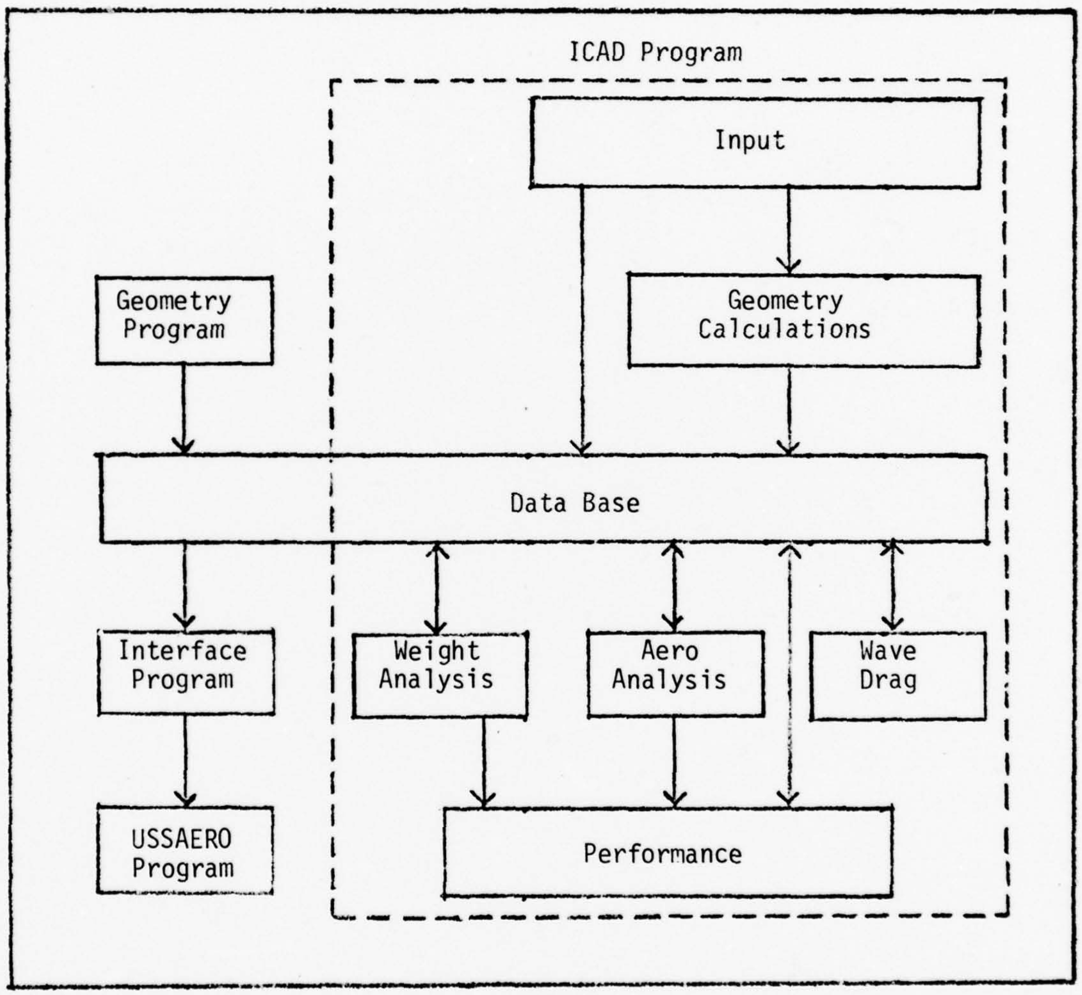

Figure 1. Schematic of Overall Computer System

3-D Item Format, is different from the input formats in either of the analysis programs. This third format was selected for several reasons. The 3-D Item Format contained sufficient information to supply geometric inputs to both analysis programs thus limiting the number of ways that a given configuration need be defined. It had the advantage of not requiring additional code to modify USSAERO to accept the multiple forms of ICAD inputs which would have increased the size and execution times of USSAERO. The most significant reason for the third format was that it maintained program integrity for the two analysis programs.

The 3-D Item Format can represent almost any continuous surface. It has no preferred direction so that the same format can be used on both bodies and lifting surfaces. Since the data are a series of points begin with, there are no constraints on the shape of a given cross-section. Due to the lack of flags or other means of establishing special connectivity between points, the format cannot represent a discontinuous surface as required for a fan-in-wing. Also for connectivity requirements, each cross-section must have the same number of points describing it. The only means of representing a discontinuity, as occurs at an engine inlet or exhaust, is to butt two or more 3-D Items together at the same plane as shown in Figure 2.

The Interface program is not overlayed and is written in FORTRAN IV. It converts one input component into one or more output components, each with separate subroutines. The program provides hands off processing of the data from the 3-D Item Format into the USSAERO format and checks the validity of the data for use in USSAERO. Diagnostic messages are printed when an incompatibility is discovered. There are no provisions to define

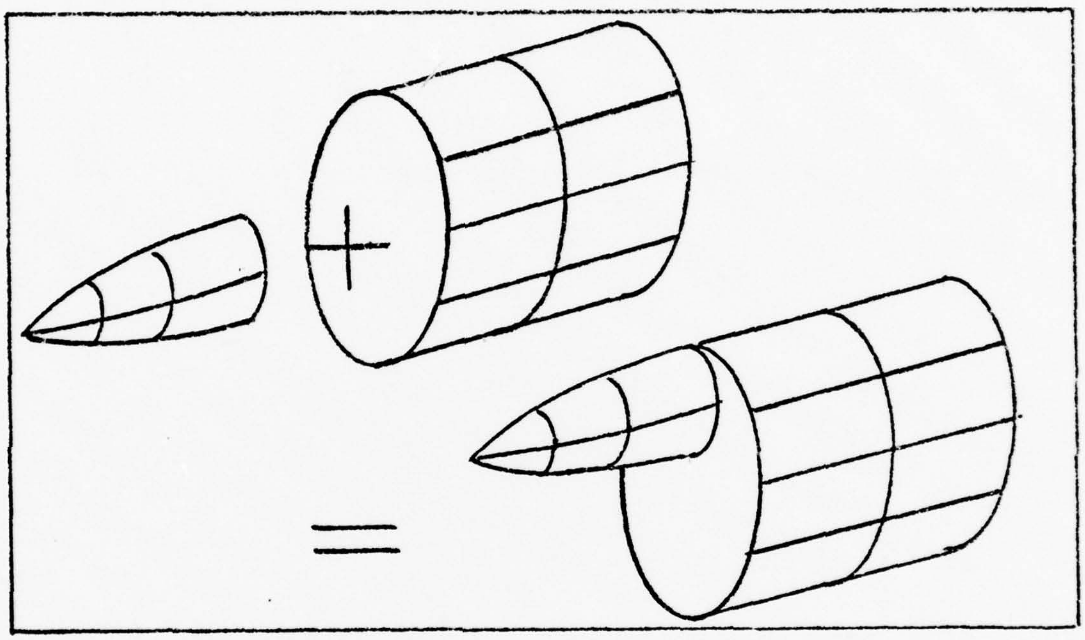

Figure 2. Engine Inlet Representation

any input but the geometry for USSAERO. A further restriction is that new geometry cannot be added to old geometry. This must be done external to the program.

Geometry Program

The Geometry Program in conjunction with a digitizer, provides a means of quickly generating aircraft components in the 3-D Item format by defining the geometry as a series of two-dimensional cross-sections.

Components of aircraft geometry usually have principal axes. Such axes are in the direction of least rate of change of the geometry. For bodies, the direction is towards increasing fuselage station numbers, and for wings, it is towards increasing span. Aircraft geometry is normally defined in this format: for bodies, as a series of two-dimensional cross-sections, and for wings, as a series of streamwise airfoils. The Geometry program was written to rapidly convert this graphical presentation of the aircraft into a mathematical model in the computer. The input geometry can be from a three-view drawing, an inboard profile, or from lofting data. The program can define the cross sectional data only perpendicular to the principle axis. Canted engine inlets, as occur on many supersonic aircraft, canted cross-sections, and nonstreamwise airfoils can only be approximated. The program can define bodies that have symmetry about the aircraft centerplane, symmetry about some plane other than the aircraft centerplane, and bodies without symmetry.

The Geometry program is overlayed and written in FORTRAN IV. It uses standard Tektronix software and can be run from any Tektronix terminal. The program consists of two primary overlays, the first being devoted to body definition, and the second to lifting surface definition.

Body definition is done using the Tektronix Digitizing Tablet to input the cross sectional data. Surface definition is done by entering dimensional data and NACA type airfoil designators using the teletype keyboard. These two processes are further described in the next two sections.

Body Definition. Figure 3 illustrates three frequently occurring special shapes that can be defined with a minimum of information by the Geometry program and digitizer. The first two digitized points of each cross-section define the horizontal axis of the cross-section relative to the digitizer tablet. Thus each cross-section can be on a different piece of paper. If only one additional point is defined (3 total), the only possible shape is a single point. All the output points for this cross-section are set equal to that point. If two additional points are defined (4 total), a circle with center at the first point and radius equal to the distance between the two points results. The output points are evenly distributed around the perimeter of the circle. When three additional points are defined (5 total), two quadrants of ellipses are drawn, one between the first two points with its major and minor axes aligned in the horizontal and vertical directions, and a second between the second two points.

When more than five total points are defined, a curve is fitted to the data with a rotating cubic as shown in Figure 4. The curve fit for the data in each interval is in a different coordinate system which appears to rotate with the curve. This feature avoids infinite slopes and the resulting error condition in the computer. The only way that this method can cause an error is when three adjacent points span 180 degrees as illustrated in Figure 5.

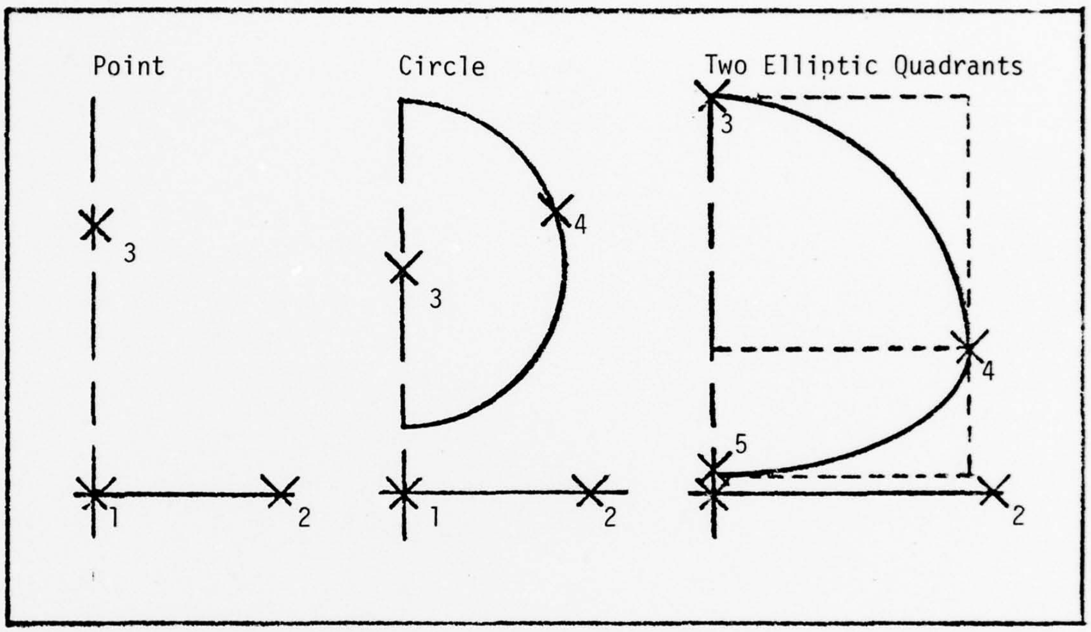

Figure 3. Required Inputs for Three Special Shapes

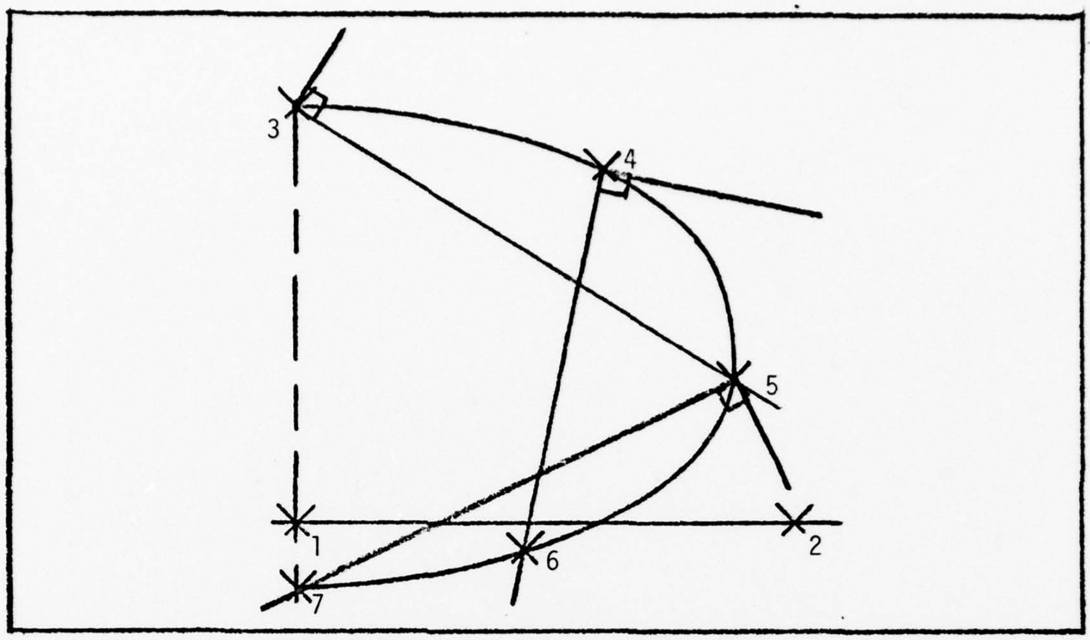

Figure 4. Axis System for Rotating Cubic Curve Fit

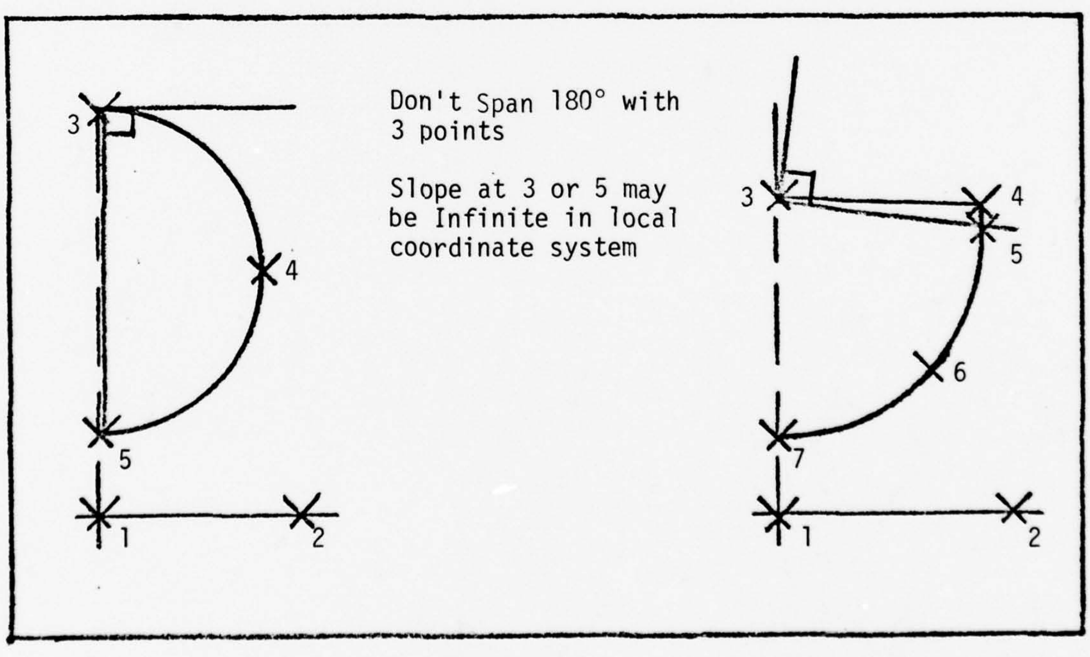

Figure 5. Examples of Improper Data for Rotating Cubic Curve Fit

The cubic curve which results is continuous with a continuous first derivative. The first derivative can be made discontinuous by breaking the curve into more than one segment by defining the same point twice. The curve fit searches for this condition and the program advises the user of the number of segments found in a given cross-section. The user then inputs the point number of the last point in each segment. This allows the user to control the number of output points per segment. The output points are evenly spaced on perimeter within segments with the specified points at the ends. When an input point is repeated three times, the previous segment is terminated and a zero length segment is created at that point. This allows several output points to be co-located on a given cross-section as may be required at the beginning or end of a protuberance like the canopy.

The purpose of controlling the distribution of the output points on each cross-section is to establish the connectivity of the data. The only effort the user must make to generate three-dimensional data from the two-dimensional cross-sections is to assure the same numbered point lies on the same geometric feature of each section. For example, if the fifteenth point on each cross section is to lie on the maximum width point, it is required to break each cross-section into segments above and below the maximum width point and to specify the fifteenth point to be the last point on the upper segments.

No provisions were made in the Geometry program to edit the digitized data. It was found that the entire cross-section could be redefined in less time than would be required to edit the data point by point.

Lifting Surface Definition. The purpose of the lifting surface definition portion of the Geometry program was to provide a means of quickly defining lifting surface geometry in terms of airfoil data when the geometry can be so defined. This is done by using standard NACA type descriptors for the airfoil shapes and dimensional data from a three-view of the configuration to locate the airfoils in the body coordinate system. Each chord of the surface is located by an airfoil reference point, usually the leading edge point, a chord length, a thickness-to-chord ratio, a local incidence angle in degrees, a parameter defining the fraction of chord length of the reference point, and the airfoil designator. The program can define four digit, four digit modified, five digit, five digit modified, 60-series, hexagonal, bicircular arc, and NASA supercritical airfoils. If the surface can be described as a series of streamwise airfoils of these types, then this portion of the Geometry program will very quickly define the surface.

One restriction that the program places on the output data is that all the airfoil sections must be at the same angle relative to the vertical axis. For gull wings where the separate panels have different dihedrals, this can be a problem. A second restriction on the output data is caused by the spacing of the output points. The user has the option to enter the fractions of chord at which he wishes the surface to be defined. The upper surface of the airfoil is defined first going front to back followed by the lower surface going back to front. The result is a closed loop beginning at the airfoil leading edge, going around the airfoil, and ending with the leading edge point again. With only one output point at the trailing edge of the airfoil, shapes with finite thickness at the trailing edge cannot be properly defined. For the NASA supercritical airfoils, the trailing edge point is located on the upper surface.

IV. OPERATING PROCEDURES

The purpose of this chapter is to describe the application of the Geometry, Interface, ICAD, and USSAERO programs to the analysis of a configuration. Two similar procedures are described: (1) stand-alone operation of USSAERO, and (2) interfaced operation of USSAERO with ICAD. Both procedures require five specific steps with the time and effort devoted to each step being strongly problem dependent.

Stand-Alone Operation

The stand-alone operation of USSAERO is the most general and each of the required steps will be discussed in detail: component identification, geometry definition, conversion to USSAERO format, paneling definition, and nongeometric data definition.

Component Identification. Each configuration to be analyzed must be divided into components. Some configurations consist of single components as in wing-alone or body-alone problems. For the majority of multi-component configurations, it is not difficult to decide which part of the configuration is to be analyzed as a wing, a fin, a canard, or a body. However, on blended wing body configurations and variable sweep aircraft with large glove areas the best choice is not always obvious.

The body can consist of a maximum of four segments. Each segment describes a three-dimensional shape by defining cross-sections. The cross-sections must be perpendicular to the longitudinal (X) axis. Each point on a cross-section must be single valued when defined in a cylindrical coordinate system whose origin is located half the distance between the upper and lower extremes of the cross-section as in Figure 6.

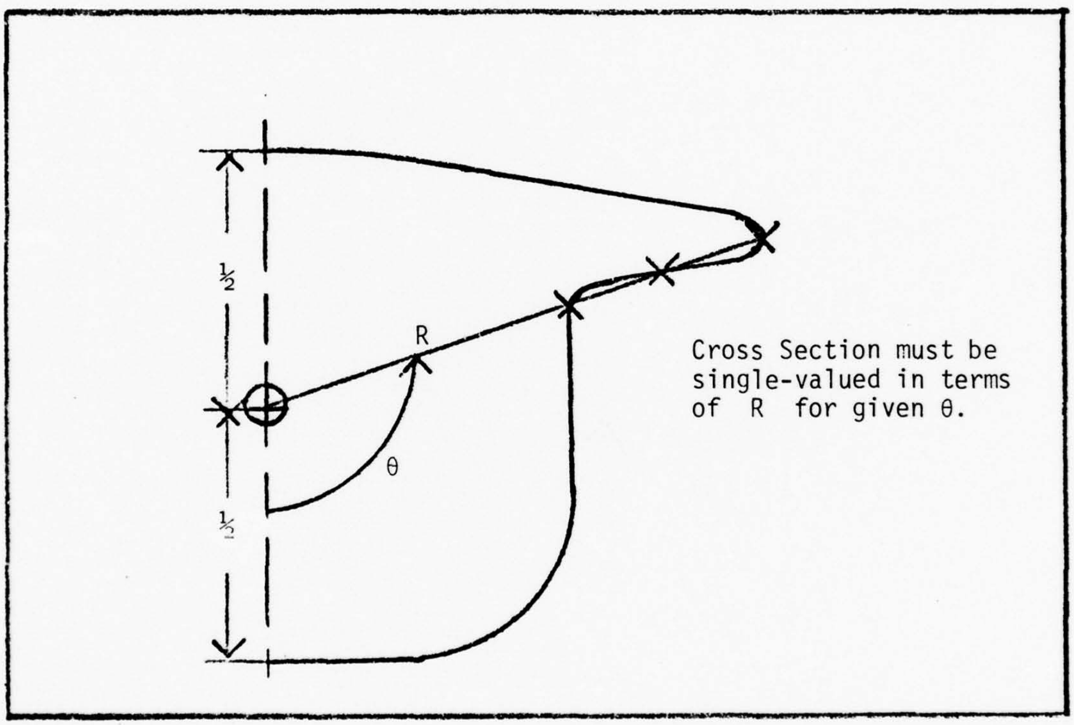

Figure 6. Example of Improper Data for USSAERO Program

Each body segement must be symmetrical about the configuration centerplane. The break between adjacent segments can be used to represent discontinuities in the geometry such as occur at engine inlets and exhausts. When the available geometric data for the body does not meet these requirements, it must be modified until it does. Chapter V discusses such modifications for the F-111 aircraft geometry.

The wing is described by a number of streamwise airfoil sections each of which satisfies the Kutta condition at its trailing edge and may be cambered. Only one wing is allowed, and it cannot have a panel with a 90 degree sweep.

The fins are the same as the wing with two additional requirements. They may not be cambered and may consist of only two chords. Three chord surfaces can be described by defining two fins, the outboard fin's root chord being colocated with the inboard fin's tip chord.

The canards are defined by two chord surfaces with camber permitted. Although USSAERO does not analyze the effects of pods, they are included in the input for compatibility with other aerodynamic programs in use at NASA Langley.

By making these considerations before defining any geometric inputs, it is possible to divide complex configurations into the proper components for analysis with the USSAERO program. In Chapter V these considerations are applied to the F-111 as an example.

Geometry Definition. The previcus section discussed the requirements that body cross-sections be single valued in terms of a cylindrical coordinate system, and that lifting surface sections must be streamwise. Modifications to the actual geometry may be required to satisfy these requirements. Such modifications can change a simple circular body into a more complicated shape when part of the wing fillet is included in the body. Likewise, part of the actual body cross-sections may be removed so that the wing body junction is streamwise. This modification to one cross-section may impact all the sections through the requirement to maintain connectivity of sections and unusual spacing of points may require even circular cross-sections to be defined using the rotating cubic in order to control the output point spacing. Only after a satisfactory scheme for connectivity can be visualized should the first body section be defined and then proceed from front to back with each body segment being treated as a completely new body in the digitizing portion of the Geometry program. The centerline body option should be used as USSAERO requires the body to be symmetrical about the centerplane and the number of output points per cross section should be kept less than or equal to 30 because of USSAERO requirements.

Modification to the wing body junction can influence the way the wing is defined. A wing describable with standard NACA airfoils may require modification to the root section for the above reason. When the entire wing is composed of nonstandard airfoil sections it is best to record the airfoil descriptions on coding forms directly in the USSAERO format. Then data can be added to the body data at a later time by changing only the first input card of the geometry. When defining tail surface geometry, the number of stations where the airfoil sections will be defined should be kept to less than eleven.

Conversion to USSAERO Format. The Interface program requires very little user involvement. While the program is converting the geometry into the USSAERO format it is checking the validity of the data and diagnostics will be printed out when a discrepancy occurs. Such an event will normally require the geometry for that component to be redefined. When a component is keypunched in the USSAERO format originally, it must be added to the Interface program output and the first Geometry card changed to reflect the new data. To convert a body defined in the Geometry program into a pod, it is required to change the treatment code on the second card of the 3-D Item format from BOD to POD.

If the geometry is not defined in the order wing, body, fins, and canards, the output from the Interface program must be sorted into this order to be ready for use in USSAERO.

Paneling Definition. Single component analyses should be performed first to verify the adequacy of the geometry. When multi-component runs are made, paneling density should decrease as distance from the point of concern increases. If wing pressures are being sought, forebody and afterbody paneling can be made less dense to cut down the computer resources spent. Also, tail surfaces can have no effect on wing pressures at supersonic speeds. Panel density should increase in regions of highest anticipated pressure gradient. Density should be increased in steps from run to run until the pressures no longer change with increased density. This minimizes the expense of geometry input errors for the initial runs and provides confidence in the converged solution of the last run.

Nongeometric Data Definition. The primary requirement for additional data is the MACH, ALPHA card for each case. Since new aerodynamic influence coefficients must be calculated for each new MACH, the cards should be grouped together for the same MACH number to save computer time. Other information on the same card are indicators for specifying whether nonzero normal velocities will be specified at the panel control points. There are also indicators for specifying whether off body points will be read in. The program determines the velocity components of the flow induced at these points and these are used to determine flow inclinations and magnitudes near inlets and other protuberances. A negative Mach number serves as a flag to terminate calculations on that geometry, and new geometry may follow.

Two other requirements for additional data are the plot control cards and the job card. The plot control cards are located before and after the paneling information and result in geometry, the pressure distributions are plotted for each strip of panels for each Mach number and angle of attack. Care should be taken not to overdo these as they require large amounts of Calcomp paper and time. The last data required are central processor and input-output time for the job card. Figures 7 and 8 provide a means of estimating these for the Wright-Patterson AFB system of CDC 6000 and 7000 series computers. ASD/ENFTA experience indicates much less time is required.

Interfaced Operation

The procedures for operating the programs in the interfaced mode are very similar to the stand-alone mode. In this case, the program execution sequence varies based on the results of the component identification step. If it is desired to perform an aerodynamic analysis on a configuration that already exists in ICAD, the ICAD geometry must be examined for compatibility with USSAERO.

ICAD Components Satisfactory. When it is determined that the component division of ICAD is the same as for USSAERO, the 3-D Item nonzero normal velocities will be specified at the panel control points. There are also indicators for specifying whether off body points will be read in. The program determines the velocity components of the flow induced at these points and this data is used to determine flow inclinations and magnitudes near inlets and other proturberances. A negative Mach number serves as a flag to terminate calculations on that geometry, and new geometry may follow.

Two other requirements for additional data are the plot control cards and the job card. The plot control cards are located before and after the paneling information and result in geometry plots before and after paneling. When plots are made of the geometry, the pressure distributions are plotted for each strip of panels for each Mach number and angle of attack. Care should be taken not to over do these as they require large amounts of Calcomp paper and time. The last data required is central processor and input-output time for the job card. Figures 7 and 8 provide a means of estimating these requirements based on the number of singularities used.

Interfaced Operation

The procedure for operating the programs in the interfaced mode are very similar to the stand-alone mode. In this case, the program execution sequence varies based on the results of the component identification step. If it is desired to perform an aerodynamic analysis on configuration that already exists in ICAD, the ICAD geometry must be examined for compatibility with USSAERO.

ICAD Components Satisfactory. When it is determined that the component division of ICAD is the same as for USSAERO, the 3-D Item

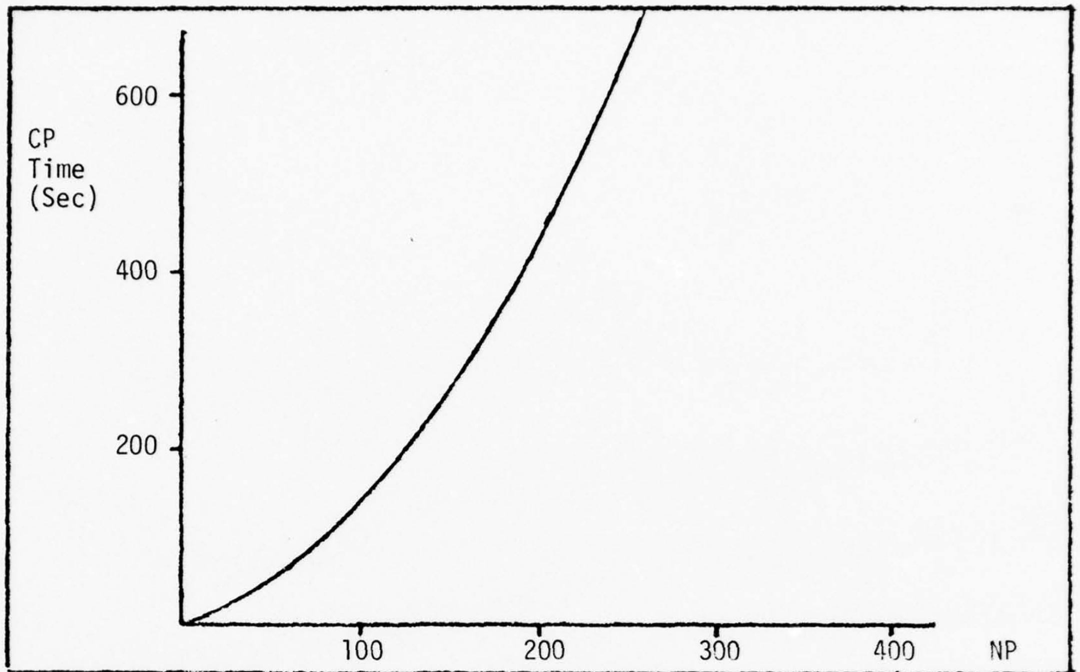

Figure 7. Central Processor Time Required vs. Number of Panels

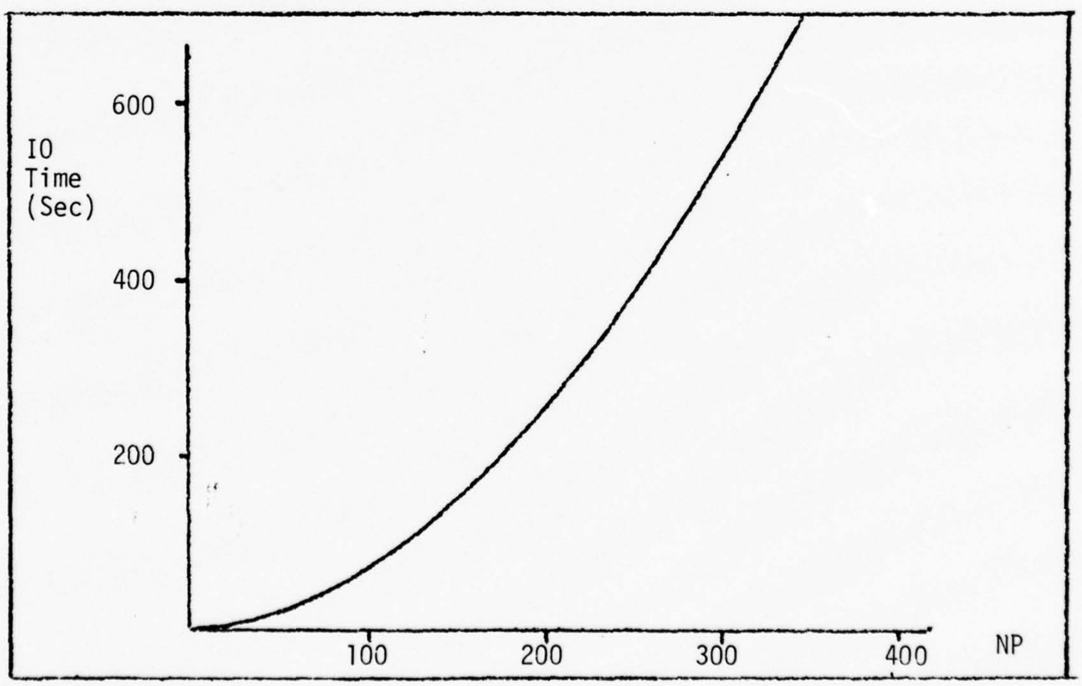

Figure 8. Input/Output Time Required vs. Number of Panels

geometry from ICAD is used for input into the Interface program. Thus, regardless of the original form of input to ICAD, the same geometry can be made available for analysis in the USSAERO program in a matter of a few seconds. The steps for paneling definition and nongeometric data definition are performed next, as for the stand-alone operation.

ICAD Components Unsatisfactory. When one or more components require redefinition before running the USSAERO program, cross-sections defining these components are determined and plotted by the appropriate routines in ICAD. The new geometry is sketched on these cross-sections and digitized with the Geometry program as in the stand-alone operation.

V. TACT AIRCRAFT ANALYSIS

Transonic Aircraft Technology (TACT) aircraft is an F-111A that has been refitted with a supercritical wing which improves performance in the high subsonic speed regime. This chapter describes the entire analysis procedure for this complex configuration as an example of how to use the various programs. The input preparation procedure will follow the same five steps covered in Chapter IV for stand-alone operation of USSAERO. The type and amount of data available and the effect they had on the procedure are discussed. How the configuration was divided into components, the connectibility of cross-sections, the wing input procedure, the paneling definition and how planar surfaces can affect that paneling, and the amount of time and number of runs required to generate the data are discussed. Limited comparison with wind tunnel and flight test results were made.

Component Identification

The available aerodynamic data were for a wind tunnel model which deviated slightly from the full scale aircraft. Fuselage cross-sections were not available beyond station number 600 whereas the body extended another 300 inches. The data describing the wing were airfoil coordinates given on the drawings. Coordinates were available for all major breaks in planform and thickness geometry. The data were for the wing positioned at 16 degrees leading edge sweep. Tail surface geometry was described as bicircular airfoils with dimensional data from the drawings.

The first decision in the component identification process was to include that part of the glove area beyond span station 72 in the wing as shown in Figure 9. This aligned the rear fuselage and the wing root chord and required modification to the body cross-sections to make a flat section at the spanwise ordinate of 72 inches.

The body was broken into three segments: from the nose to the start of the glove, the glove to the engine inlet, and from the inlet to the engine nozzle. The first segment had smooth cross-sections and was not a problem. The second segment had cross-sections that required special paneling to properly define the glove protuberance. Due to the lack of data immediately behind the engine inlet and near the rear fuselage, cross-sections were translated to make the third body segment the proper length. The section at fuselage station 500 was moved forward to define the body just behind the inlet. Data from station 600 was moved aft to the engine nozzle plane to approximate the real geometry and to deiine something to hand the tails on.

Geometry Definition

Since the wing airfoil coordinates were available, and they were not standard NACA type airfoils, the wing data were coded and keypunched directly into the USSAERO format. The wing data so defined were for a wing sweep of 16 degrees. The aerodynamic data that was to be used for comparison was for a sweep of 26 degrees. A wuick computer program was written to rotate the 16 degree data and punch out 26 degree data define at new span stations by linear interpolation.

The root chord required modification to include the glove area that had been removed from the body. The upper and lower surfaces of the glove at span station 72 for each of the body cross-sections modified were plotted and an

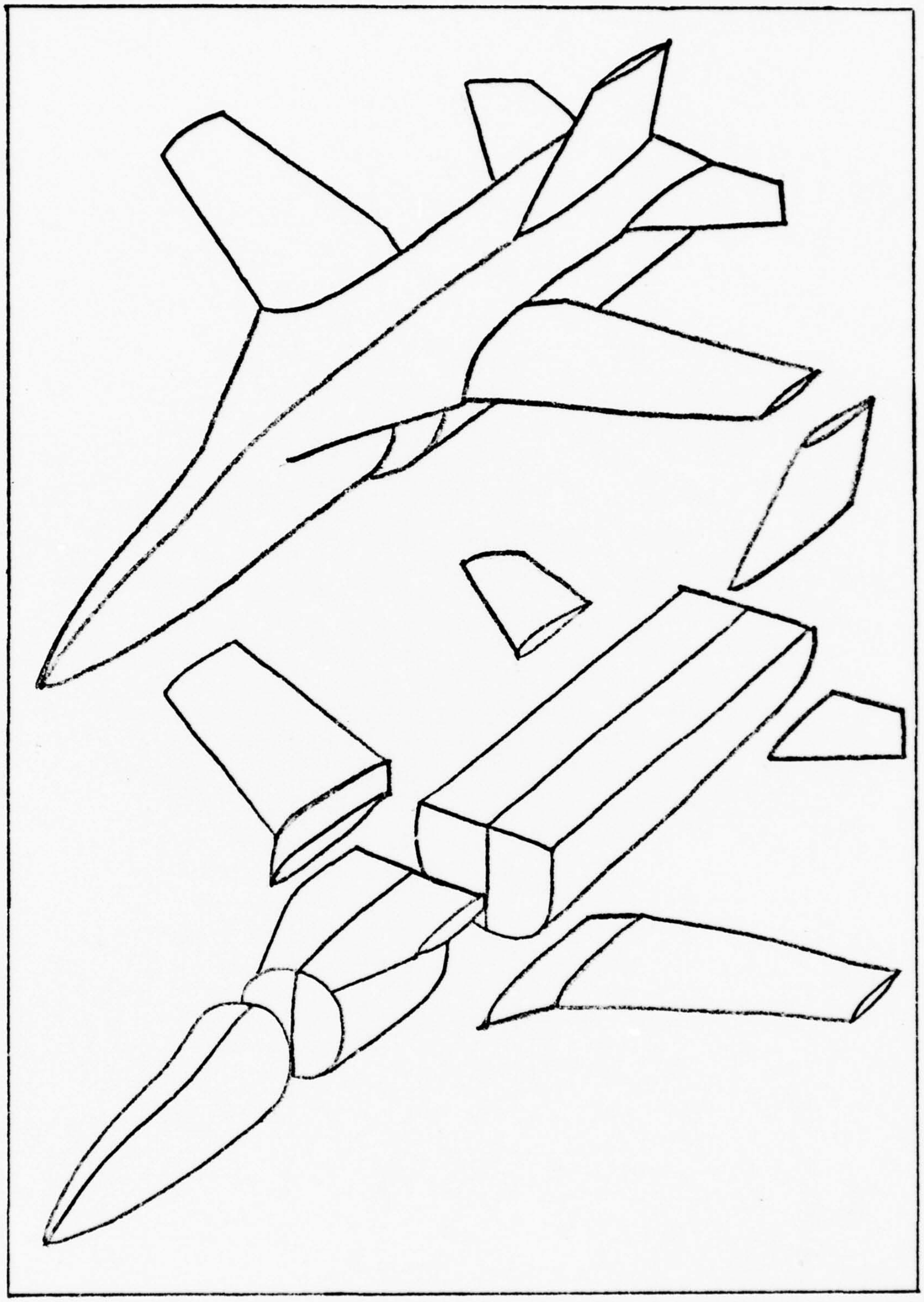

Figure 9. Component Identification for the F-111A Aircraft

airfoil shape was faired through this data and the rear portion of the root chord. This shape was used for the root section of the wing, as shown in Figure 10.

The first body segment had simple smooth sections, and was no problem to input. The second body segment had slope discontinuities where the glove had been removed. In addition, some of these sections had to be modified to satisfy the requirement that $R(\theta)$ be single-valued. The lower surface of the glove protuberance was modified to be a straight line from its old value at span station 72 to where a line from slightly below center intersected the fuselage side as in Figure 11. The first section of the second body segment had points grouped together which would later describe the glove on a zero length segment. This points out why the connectivity of the entire body must be established before defining the first section.

The tail surfaces were describable in terms of standard bicircular airfoil sections. They were input using the surface portion of the Geometry program.

Conversion to USSAERO Program

The conversion process was completed in a matter of seconds. Only the body and tail surfaces were defined in the Geometry program, and thus, in the 3-D Item Format required by the Interface program. The wing was already in the proper format, and was added by changing the first geometry card and inserting the data cards into the deck created by the Interface program.

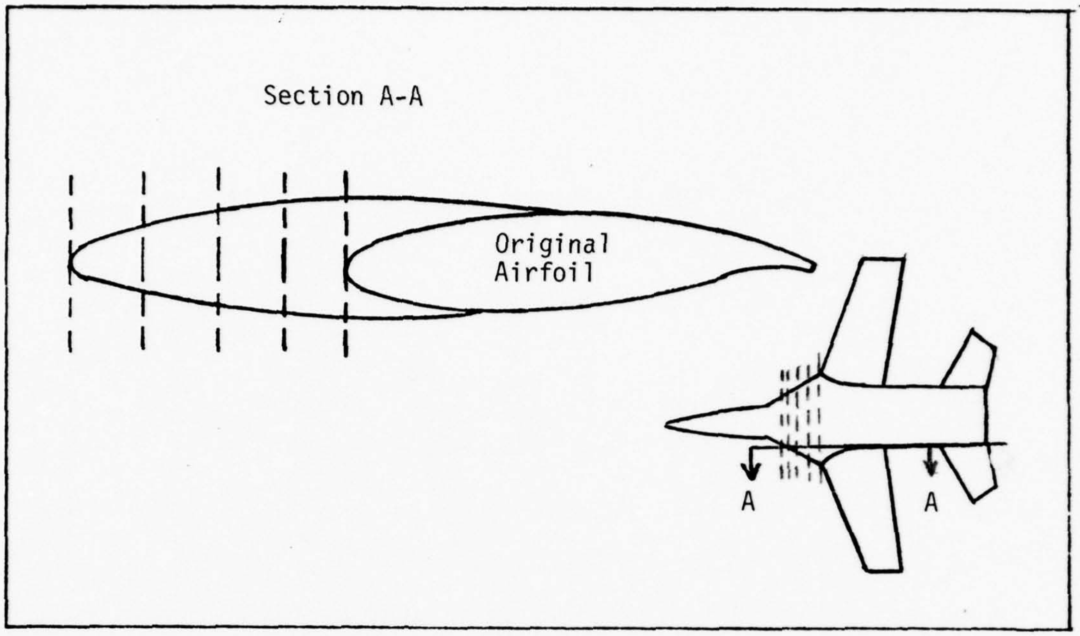

Figure 10. Root Chord Modification for F-111A Aircraft

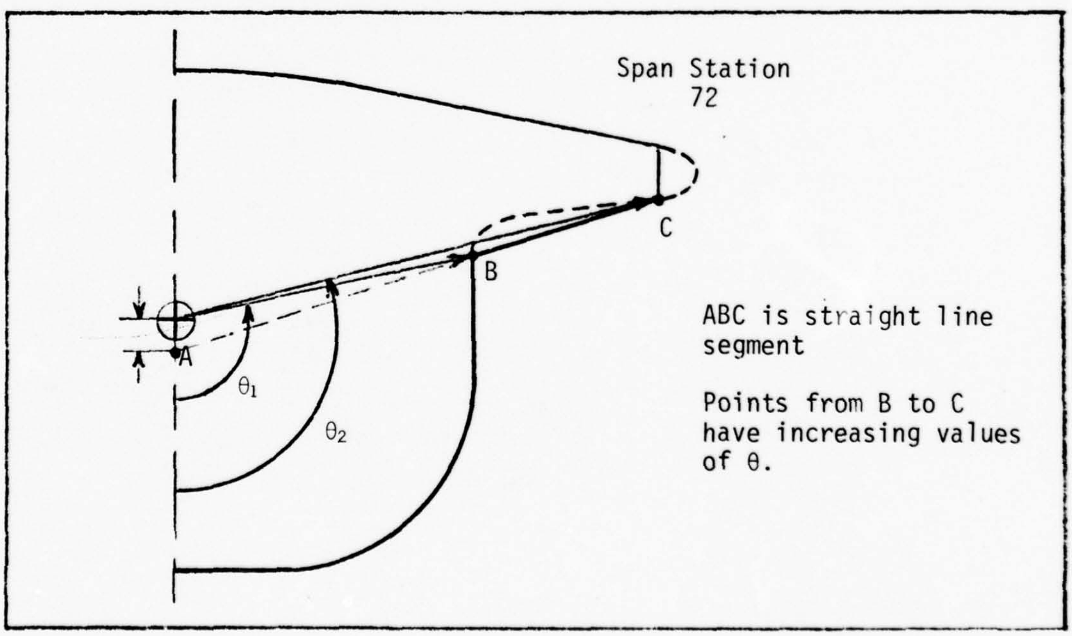

Figure 11. Glove Modification for Fuselage Cross Sections

Paneling Definition

Particular theta angles were specified for the body which place panel edges on the breaks in the geometry. This was required because the breaks did not occur at even increments of theta. Each cross-section of a given body segment was paneled at the same thetas. Thus, even if the input geometry is the exact paneling desired, USSAERO will interpolate the paneling at specified increments. This is a severe restriction in the program and it cannot be eliminated without a major recoding of the program. The thetas selected must be a compromise over all the sections in a given segment. When the data in a segment differs too widely, that segment must be divided into two.

It was a simple process to specify the paneling for the lifting surface geometry. The panels were approximately the same size, and the width of two adjacent panel strips did not differ by more than a ratio of .5. The paneling density was increased near the leading edge of the wing and near the break in sweep. Extra care was required in paneling for the runs which included the tails because the horizontal surface was nearly coplanar with the wing. Concentrated vorticity was shed from wing panel edges and when it approached downstream control points it induced infinite velocities. This was eliminated by placing the wing and horizontal tail panel edges at the same span stations.

Nongeometric Data Definition

This section describes the remaining information required to perform the analysis. The first run made was for body alone geometry for the purpose of checking the geometry definition and paneling information.

The second run included wing-body geometry before paneling, but performed wing alone analysis only for the same reasons as before. Plots of the complete geometry before paneling revealed any errors in the relative location of the wing. A complete wing-body analysis was performed next to reveal any interference problems between body and wing. Tail geometry was added next with the analysis being wing-tail to point out problems with paneling coplanar lifting surfaces. The final runs included wing-body-tail analysis. This piecewise development of the geometry is recommended to hold down the cost of the initial runs and to provide confidence in the geometric modeling. The large core storage requirements limit the number of runs possible per day. The job can be run during the day shift, but cannot be run under INTERCOM without an extended password.

Discussion of Results

Pressure coefficient data on the wing at 26 degrees of sweep were available from both flight test and wind tunnel testing. The wing pressure ports were aligned with the freestream at this sweep only. Flight test data on the full sized aircraft were available at many subsonic Mach numbers and at five spanwise stations. Wind tunnel data were available on a one-twelth scale model with two different wings. Test results for a flexible wing made from composite material to simulate the aerodynamic twist of the full scale wing were also available. The data were for Mach numbers between 0.6 and 0.85, and for the same five spanwise stations as the full scale wing. Test results for a "rigid" steel wing were available at Mach numbers of 0.6 and 0.85, but at only one spanwise station.

The geometry analyzed in the USSAERO program was for the tail-off configuration since wind tunnel tests showed little effect of the horizontal tail on wing pressure distributions.

The comparison with the rigid wing wind tunnel data at a Mach number of 0.6 is shown in Figure 12. Aerodynamic twist is less of a problem with this wing since maximum values up to 0.4 degrees have been observed. Such twist explains some of the differences between the USSAERO data and experiment. The discrepancy at the leading edge is due to the square root singularity in the potential flow solution which results from the sharp leading edge assumed for the lifting portion of the solution. The discrepancy near the trailing edge is due to the thickening of the boundary layer which is ignored in potential flow.

Figure 13 is a comparison of USSAERO data with flight test data at several spanwise stations. The observed twist distribution on the full scale aircraft was input into the program using the separate camber line option. The amount of twist at varying dynamic pressure ratios for the flexible wing wind tunnel model are presently being determined. Without such information, the proper geometry for analysis is not available.

Wind tunnel results in Figure 14 at a Mach number of 0.8 clearly show a shock existing on the upper surface which cannot be predicted by potential flow and is thus beyond the capability of the USSAERO program.

Figure 15 is included to show the effects of changing panel number and solution method. Increasing the number of panels in each chordwise strip from eight to 18 for 10 spanwise strips improves the comparison with the wind tunnel data. Changing to the nonplanar solution method more closely approximates the physical surface vorticity distribution at the expense of doubling the number of panels and quadrupline solution times. The

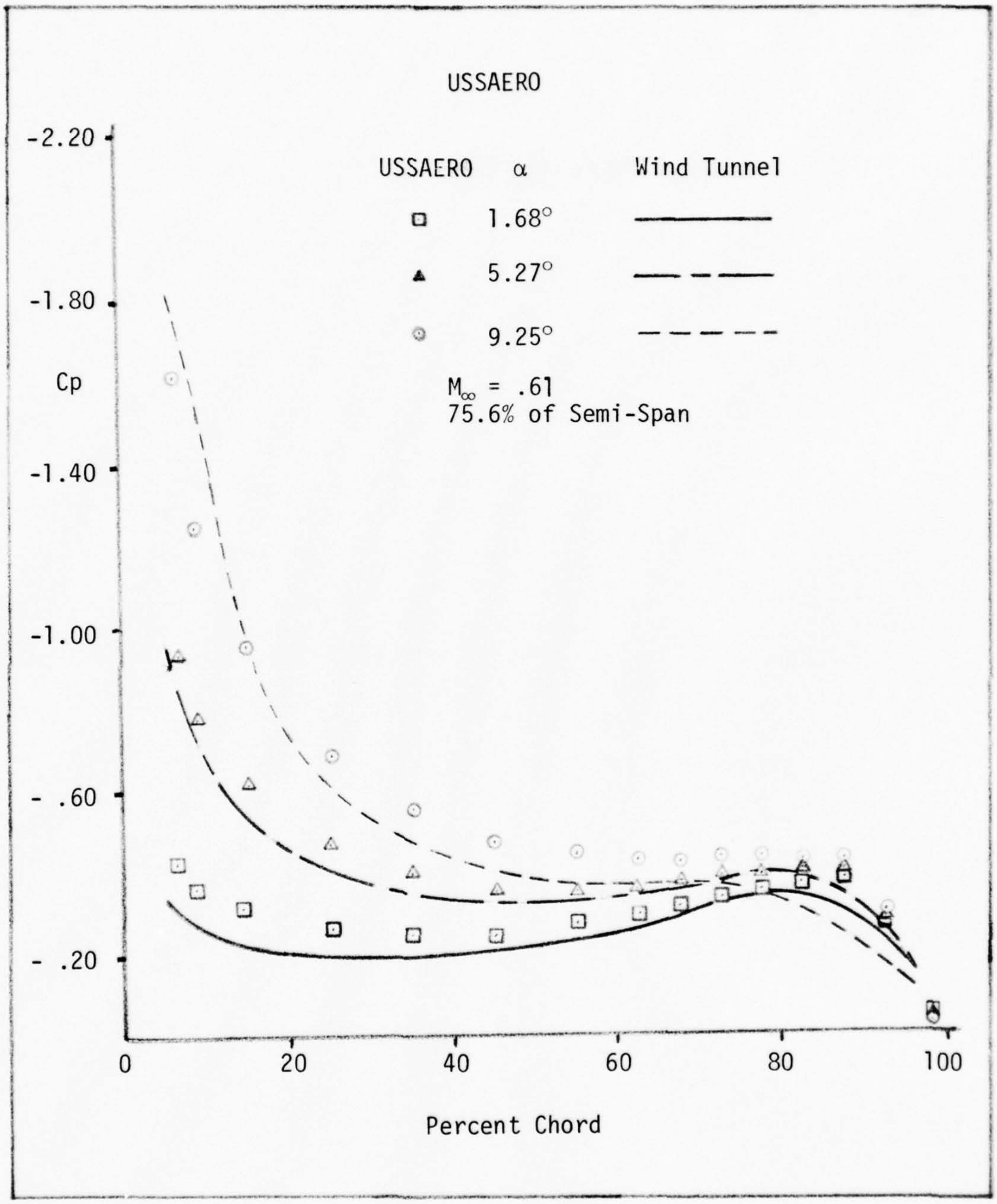

Figure 12. Comparison of USSAERO vs. Wind Tunnel Data for Airfoil Upper Surface Pressure Distribution

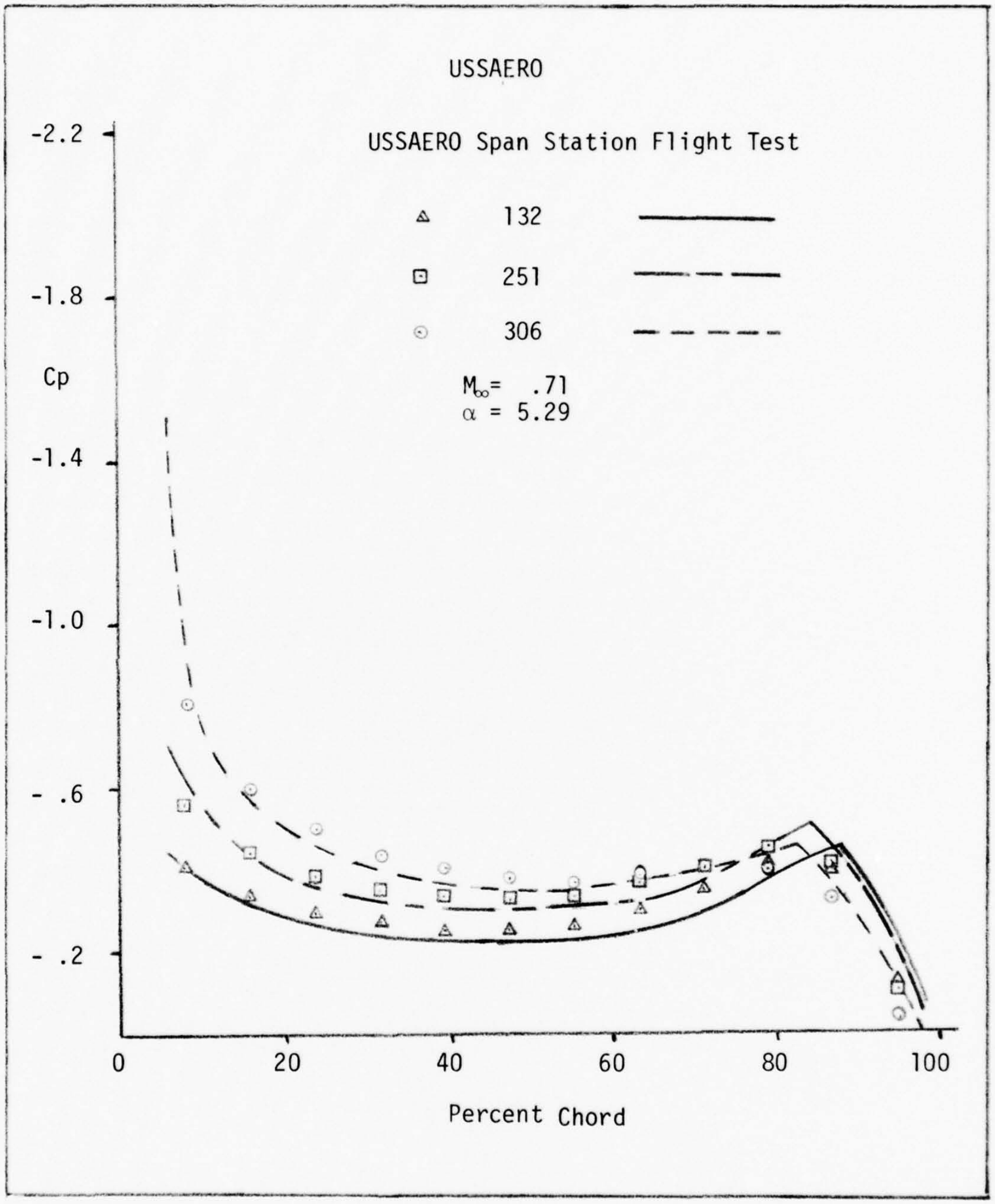

Figure 13. Comparison of USSAERO vs. Flight Test Data for Wing Upper Surface Pressure Distribution

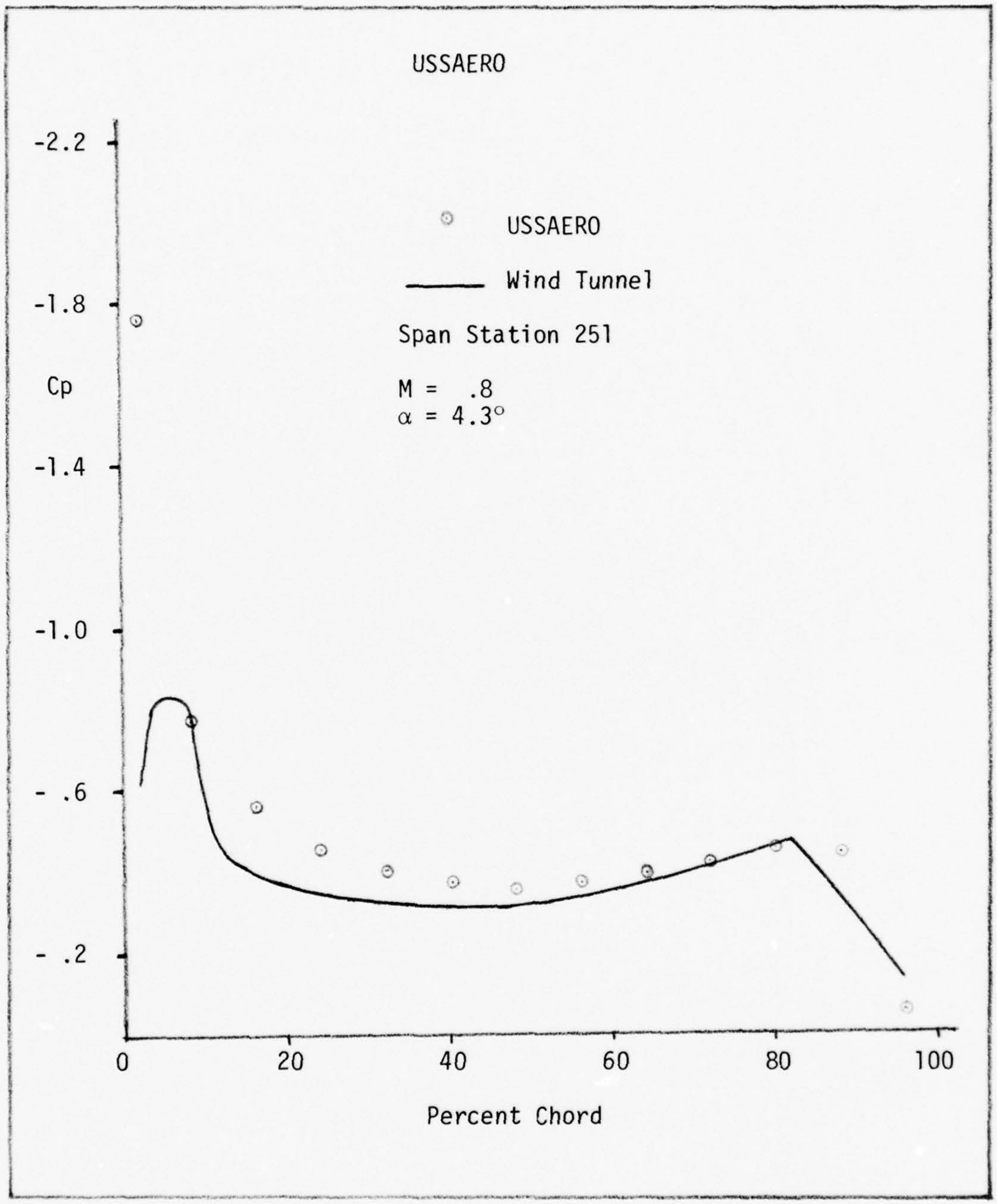

Figure 14. Comparison of USSAERO vs. Wind Tunnel Data for Supercritical Conditions on Airfoil Upper Surface

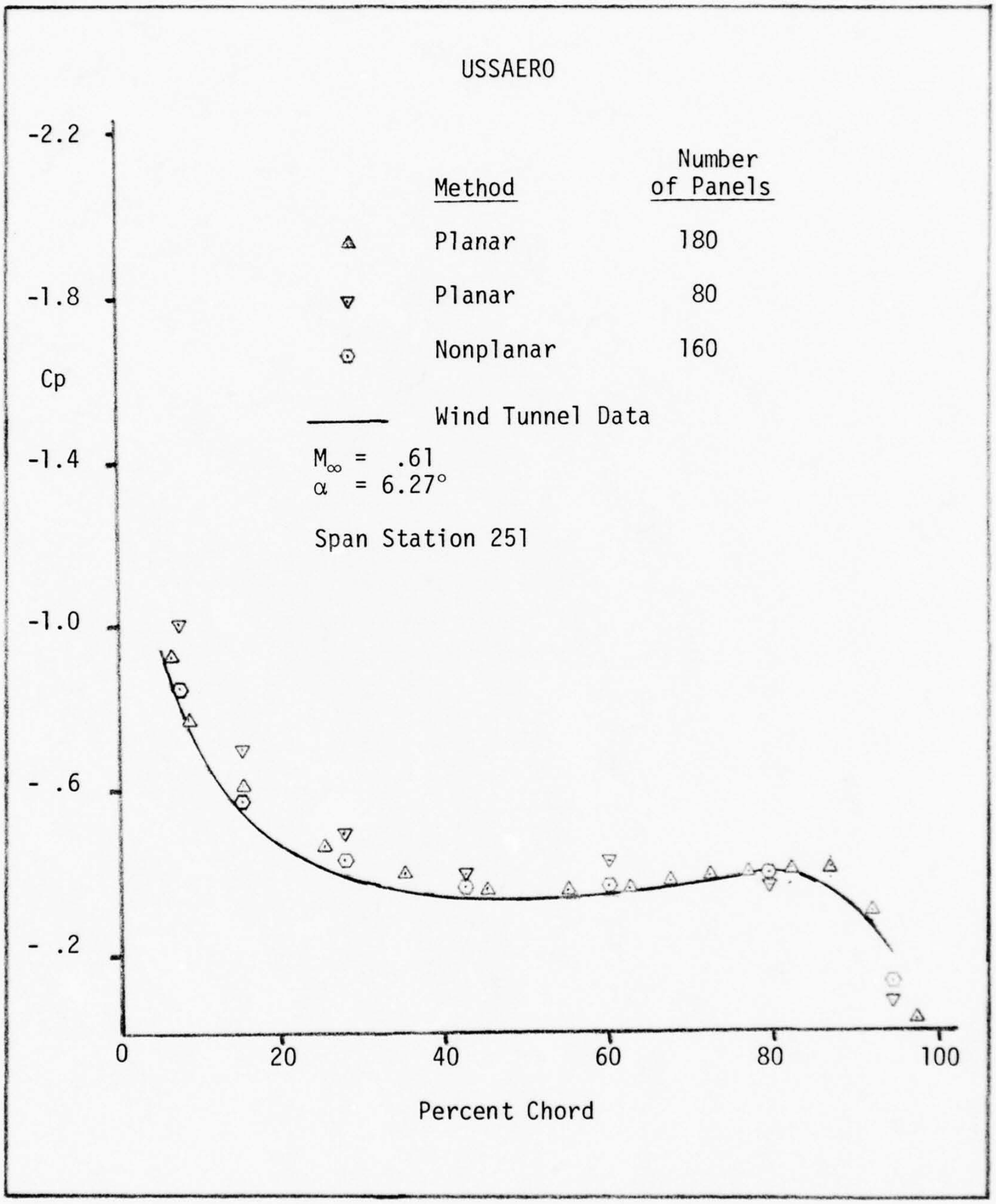

Figure 15. Effects of Variations in Number of Wing Panels and Solution Methods on Airfoil Upper Surfaces

nonplanar solution exhibits some improvement over the 80 panel planar case but shows no significant difference when compared to the 180 panel planar case.

VI. CONCLUSIONS AND RECOMMENDATIONS

Conclusions

Detailed, point-design aerodynamic analysis programs such as the USSAERO program can be used in the preliminary design environment. They significantly supplement the analytical capability of statistically based programs. They can be operated from common data bases with design synthesis programs. A single input procedure can be used for both types of programs. The input procedure can be significantly automated to the point that it no longer paces the design process. Results can be achieved in a timely manner to make them an active part of the design process.

Recommendations

USSAERO consists of over 10,500 source statements. This large size makes debugging and checkout difficult. The program consists of five different parts: paneling definition, aerodynamic influence coefficient calculations, boundary condition calculations, matrix operations, and force and moment calculations. These five parts should be separated into five different programs. Such a change could be made nearly transparent to the user by using cataloged procedures. Not only would debugging and checkout be made simplier, but the resulting program structure would be ideal for checking out new technologies such as higher order panels, new matrix techniques, etc. Each separate part could be tailored to a particular situation so that different programs could be used for a given operation. For example, direct matrix inversion could be used when the number of panels was less than 200 resulting in significant savings in computer resources. A program restart capability would exist at any step by simply cataloging the results of prior operations and beginning at the desired step instead of at the beginning with the geometry calculations. New geometric arrangements could be analyzed, such as multiple wings, multiple fuselages, wind tunnel walls, etc., by changing just the required portions while leaving the balance of the code unchanged and unaffected. This trend towards multiple programs is drawing attention in industry which makes the potential greater since provisions would exist to just "plug in" new technologies as they become available.

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APPENDIX A

Geometry Program

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Geometry Program

This appendix is a listing of the inputs required to define aircraft components using the Geometry program. It is an INTERCOM program and the user is assumed to be familiar with INTERCOM operation. Most of the inputs are self-explanatory. The information appearing in capital letters is printed on the screen. The origin is the routine that writes this information. When invalid data are input, the program will normally repeat the question. The program runs within the INTERCOM core limit of 60,000 octal locations. The time required depends on the amount of geometry defined. An experienced user will average three minutes per fuselage cross section and one minute per airfoil section of wall clock time. The central processor time required is minimal. The documentation of the airfoil indicator is a repeat of information from unpublished, inhouse documents from the Deputy for Development Planning, Aeronautical Systems Division, Wright-Patterson Air Force Base, Ohio.

ENTER NUMBER OF CHARACTERS PER SECOND:

Origin: SUBROUTINE INITT

Valid Inputs: 11 30 120

Remarks: This data is the BAUD rate divided by 10. 4800 BAUD rates are not possible with this program.

ENTER TERMINAL TYPE:

Origin: SUBROUTINE INITT

Valid Inputs: 1 2 3

Remarks: Enter 1 if using a Tektronix 4008, 2 if using 4010, or 3 if using 4014.

ENTER ONE OF THE FOLLOWING:

1) CREATE A 3-D BODY

2) CREATE A 3-D SURFACE

Origin: PROGRAM INIT

Valid Inputs: 1 2

ENTER BODY NAME (MAX. 20):

Origin: PROGRAM CROSS

Valid Inputs: Any 20 character alpha-numeric name.

Remarks: Used for identification purposes only.

IS THIS A CENTER-LINE BODY? (Y/N):

Origin: PROGRAM CROSS

Valid Inputs: Y N

Remarks: If YES, define only right hand side of cross section. If NO, define entire cross section. No negative Y-coordinates are permitted since the total configuration must be symmetrical about x - z plane

NO. OF POINTS PER HOOP DESIRED (MAX. 50):

Origin: PROGRAM CROSS

Valid Inputs: 3-50

Remarks: This is the number of output points which will be defined for each cross section. There is no relationship between the number of input points and the number of output points.

TABLET CALIBRATION

ENTER XMIN,XMAX,YMIN,YMAX,UNITS,SCALE AS FOLLOWS:

1) FOR GRAPHING PURPOSES: ENTER XMIN,XMAX,YMIN,YMAX.

2) FOR GENERAL USE: UNITS DESIGNATED AS 1=FEET,2=INCHES,3=METERS,4=OTHERS

3) IF UNITS=4, ENTER SCALE = (FEET)/(UNITS).

4) FOR EQUAL SCALING: ENTER XMIN,XMAX,,,UNITS - OR -
,,YMIN,YMAX,UNITS

5) FOR EQUAL SCALING, BUT DIFFERENT OFFSET VALUES
ENTER XMIN,XMAX,YMIN,,UNITS - OR -
XMIN,,YMIN,YMAX,UNITS

6) FOR UNEQUAL SCALING ENTER ALL.

Origin: SUBROUTINE TABCAL

Valid Inputs: As described.

Remarks: The purpose of the scale is to convert the input coordinates into feet by multiplication. This is required by ICAD. The option most frequently used when defining body geometry is to enter XMIN,XMAX,YMIN,,UNITS.

MARK XMIN,XMAX,YMAX: THEN ENTER ANY KEYBOARD CHARACTER

Origin: SUBROUTINE TABCAL

Valid Inputs: Using digitizer mouse, mark the two points that were specified above.

Remarks: This data provides means of properly scaling and translating cross sectional data into the body axis system. The first point defined for each cross section is assumed to be the point (XMIN,YMIN).

ENTER XSTA IN SAME UNITS AS SECTION:

ALSO ENTER A (1) IF TABLET NEEDS TO BE RECALIBRATED

ENTER "END," IF FINISHED WITH THIS BODY:

Origin: PROGRAM CROSS

Valid Inputs: Any fuselage station number in the same units as the Reference point (XMIN,YMIN).

Remarks: The inputs 250,1 will define this cross section as station number 250 and reexecute SUBROUTINE TABCAL allowing redefinition of the scale factor and reference point.

FIRST TWO POINTS DEFINE THE REFERENCE LINE.

NUMBER OF POINTS (IGNORING FIRST TWO) DEFINE CURVE TYPE:

NUM=1 ; POINT

=2 ; CIRCLE

=3 ; ELLIPSE

≥4 ; ROTATING CUBIC

BEGIN DIGITIZING CROSS SECTION.

Origin: SUBROUTINE SECTION

Valid Inputs: Using the digitizer mouse, mark the reference point (XMIN,YMIN) and one other point on the same water line. Digitize the cross section beginning with the top and proceeding in a clockwise direction. Use a double point (same point defined twice) to mark the ends of segments. Segments are used to control the distribution of output points around the section and to allow for breaks in slope continuity.

NUMBER OF SEGMENTS = 3

ENTER VALUE OF END POINT OF EACH SEGMENT.

LAST SEGMENT MUST BE EQUAL TO 30

ENTER BLANK TO USE PREVIOUSLY DEFINED VALUES:

IF NO SEGMENTS ARE THE SAME.

Origin: SUBROUTINE FDSEG

Valid Inputs: Three integer values specifying the value of the end points of each segment, such as 10,25,30.

Remarks: The output points 1-10 will be evenly distributed on the first segment. Points 10-25 will be distributed on the second segment, and points 25-30 will be distributed on the third segment.

Figure 16 is the next output of the program. The input points are marked with the symbols, and the output points form the solid line of the cross section.

TYPE (Y/N) TO ACCEPT THIS SECTION:

TYPE (M) TO MODIFY SLOPE CONTROLS:

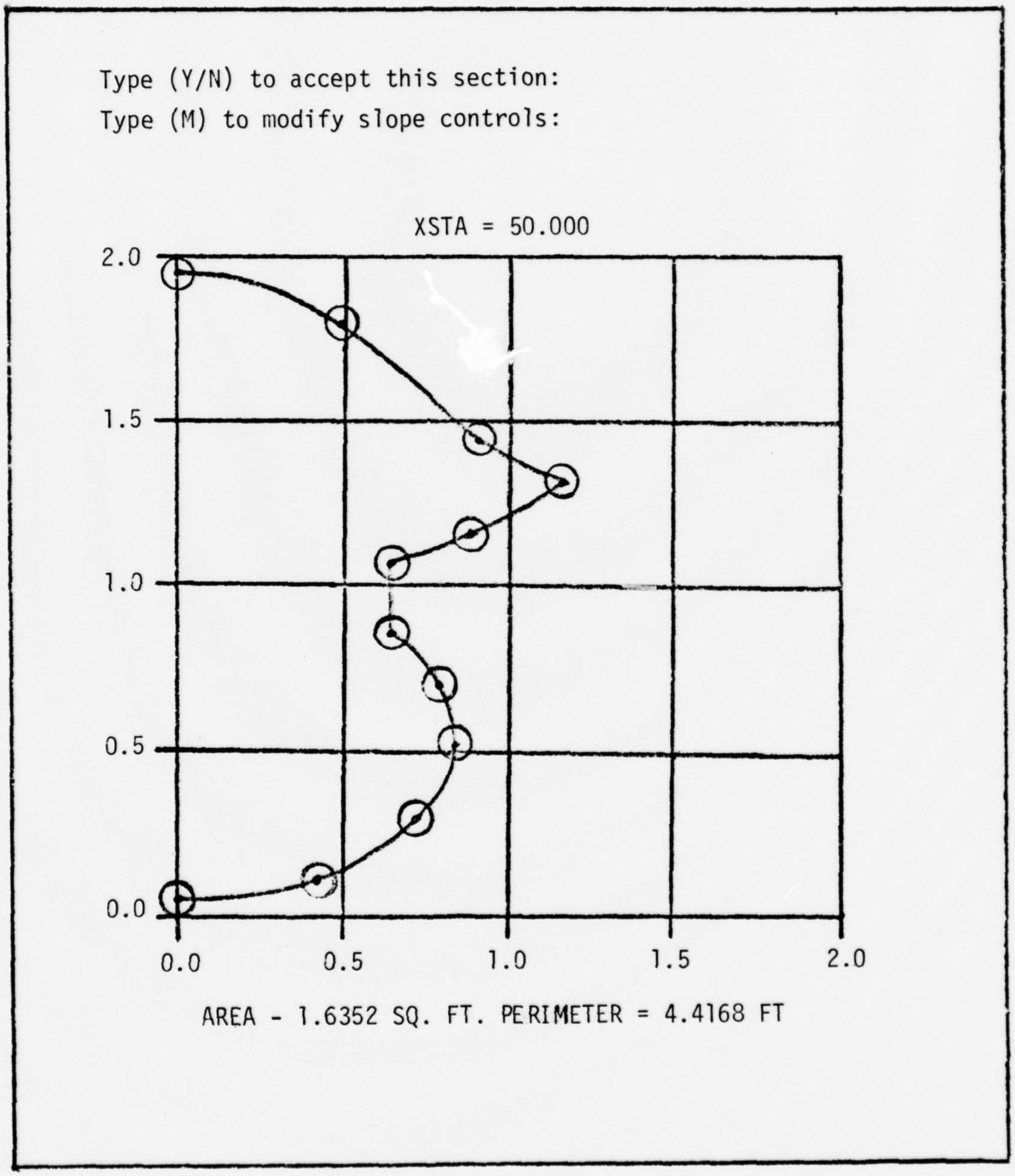

Figure 16. Sample Output From Geometry Program

Origin: SUBROUTINE DRHOOP

Valid Inputs: Y N M

Remarks: When a section is in error, it will usually require redefinition. An N should be entered and the same station number repeated. The modify option allows the slope at the ends of the segments only to be specified as horizontal or vertical.

SLOPE OPTIONS

1=EXTRA WILL DETERMINE SLOPE

2=SLOPE WILL EQUAL ZERO

3=SLOPE WILL EQUAL INFINITY

NSEG	IS	IF
1	1	1
2	1	1
3	1	1

INPUT SEG NO., IS,IF,ETC:

Origin: SUBROUTINE MDSLP

Valid Inputs: As described.

Remarks: NSEG=segment number, IS=inclination at start of segment, IF=inclination at finish of segment. Inputs such as 1,2,1,3,1,2 will force horizontal slopes at the beginning of first and the end of the third segments.

DO YOU WISH TO USE DEFAULT MERGE AND DRAG INDICATORS (Y/N)?

Origin: PROGRAM CROSS

Valid Inputs: Y N

Remarks: These inputs are required only when the geometry will be used in ICAD. Input Y when geometry is for the USSAERO program.

DO YOU WANT TO DIGITIZE ANOTHER BODY? (Y/N):

Origin: PROGRAM CROSS

Valid Inputs: Y N

Remarks: Input Y to define another body segment.

DO YOU WISH TO CREATE A 3-D SURFACE? (Y/N):

Origin: PROGRAM CROSS

Valid Inputs: Y N

Remarks: Input Y to define a wing, fin, or canard.

ENTER SURFACE NAME (MAX. 20):

Origin: PROGRAM SURF

Valid Inputs: Any 20 character alpha-numeric name.

Remarks: Used for identification purposes only.

ENTER ONE OF THE FOLLOWING TYPES:

1) WING 2) FIN 3) CANARD

SCALE = MULTIPLIER TO CONVERT TO FEET.

YOO,ZOO,ARE SAME AS YO,ZO OF FIRST CHORD, IF NOT ENTERED.

ENTER TYPE NO.,DIHEDRAL,SCALE,BODREF,YOO,ZOO:

Origin: PROGRAM SURF

Valid Inputs: As described.

Remarks: The dihedral is in degrees. Default for scale is 1. YOO,ZOO are only required when the inputs will be used in ICAD.

(MIN. OF 2 CHORDS)/(MAX. OF 5 CHORDS: FOR ICAD)

ENTER (-999) IF DONE WITH THIS SURFACE

ENTER XO,YO,ZO,CH,T/C,ALPHA,X/C,AIRFOIL:

INPUT:

Origin: PROGRAM SURF

Valid Inputs: As described.

Remarks: The first three inputs locate the airfoil reference point in the body coordinate system. This reference point can be any point on the airfoil from the leading edge to the trailing edge. The fraction of chord is specified by the X/C input. CH is the chord length. T/C is the thickness ratio in fraction of chord (leave blank). ALPHA is the local incidence in degrees. AIRFOIL is a 10 digit parameter defining the type of airfoil. The inputs are detailed below. These same inputs are repeated for each airfoil section for the given surface.

DO YOU WISH TO CREATE ANOTHER 3-D SURFACE? (Y/N):

Origin: PROGRAM SURF

Valid Inputs: Y N

Remarks: Input Y to define another wing, fin, or canard.

DO YOU WISH TO CREATE A 3-D BODY? (Y/N):

Origin: PROGRAM SURF

Valid Inputs; Y N

Remarks: Input Y to define a body segment.

IAF - Airfoil Section Input:

The airfoil section description is input through the IAF-array as a ten-digit array for specifying the pertinent parameters of the particular section designation desired. The IAF-array must have zeroes in those columns not required by the shorter designators.

A description of the available airfoil designations follows, with each parameter being defined and an appropriate example given. A summary is given at the end for quick reference once the basic scheme of input and parameter definitions is understood. The examples given for each option in the quick reference chart are shown as they would be input. An important condition to note is that the locations used to input the thickness ratio, t/c, must be left blank, or zeroed, as it is more accurately defined by other inputs to ICAD.

Option	Series	
1	4 digit	eg; 2415

1st digit - max value of mean line ordinate, $y_{c_{max}}$ in % chord

2nd digit - distance from leading edge to $y_{c_{max}}$ in tenths chord

last two - section thickness in % chord

above example; 2 percent camber at .4 chord from leading edge and is 15 percent thick

2	4 digit MOD	eg: 0012-64

1st four digits - same as above

following dash

1st digit - indicates relative magnitude of leading edge radius

6 = normal leading edge radius

0 = sharp leading edge

varies as square of integer except > 8, then variation becomes arbitrary. Use $0 \le$ 1st digit ≤ 8

2nd digit - position of maximum thickness in tenths of chord

(The suffix -63 indicates sections very nearly but not exactly the same as the basic sections)

above example: Symmetric airfoil that is 12 percent thick with a leading edge radius very nearly the same and the maximum thickness 10 percent chord aft of the normal, nonsuffixed, 4-digit section.

Option	Series	
3	5 digit	eg: 23012

1st digit - indicates amount of camber in terms of the relative magnitude of the design lift coefficient; the design C_l in tenths is 3/2 of the first digit.

2d & 3d digits - measure of the distance from the leading edge to the maximum camber location, the actual distance being 1/2 the number represented.

last two digits - section thickness in % chord

above example -- design C_l of .3, maximum camber at 15 percent chord and t/c of 12 percent chord

4 5-digit MOD eg: 23012-64

1st five digits - same as above

following dash

2 digits - same as 4-digit MOD

5 6-series eg: 65_3A218 a = .5

1st digit - the 6 series designation

2nd digit - chordwise position of minimum pressure in tenths-chord

3rd digit - (subscript) - indicator of low-drag range (width of drag bucket) measured as C_l in tenths

Capital Letter - indicates a modified thickness distribution and mean line (A-sections are substantially straight on both surfaces aft of 0.8c)

4th digit - design C_l in tenths

Last 2 digits - indicate thickness of section in % chord

"a = _" - indicates type of mean line used, where the value of 'a' is the fraction of the chord from the leading edge over which loading is uniform at the ideal angle of attack

above example: 6-series section with minimum pressure position at .5c, low drag range of .3, A-section, design C_l of .2, t/c of 18 percent c, and mean line of .5

Option Series

6 Hexagonal eg: 20251206

1st two digits - section design C_{l_i} in hundredths*

2nd two digits - position along chord of start of plateau in % chord

3rd two digits - thickness of section in % chord

4th two digits - indicator for type of meanline ("a" designation) $\leq$ 10 (=1.0) tenths

above example - 12 percent thick section with design C_l of .20, wedge section being .25 of chord for leading and trailing edge, meanline designator of a = .6

7 Circular Arc eg: 301108

1st two digits - section design C_{l_i} in hundredths

2nd two digits = thickness of section in % chord

3rd two digits - meanline indicator ('a' designation) $\leq$ 10 (= 1.0); tenths

above example - 11 percent thick section with design C_l of .30 with meanline designator of a = .8

8 Supercritical eg: 1244535

1st two digits - section thickness in % chord

3rd digit - section design C_{l_i} in tenths

4th & 5th digits - location, in % chord, of maximum thickness of upper surface (default = .4)

6th & 7th digits - location, in % chord, of maximum thickness of lower surface (default = .33)

above example - 12 percent thick supercritical section with design C_l of .4, upper surface maximum thickness location at .45 chord and lower surface maximum thickness location at .35 chord

*C_{l_i} = design lift coefficient

Quick reference guide to airfoil section options available:

Series	Opt. No.	Section Parameters								
4-Digit	1	2	4	1	5					
4-Digit Mod	2	0	0	1	2		6	4		
5-Digit	3	2	3	0	1	2				
5-Digit Mod	4	2	3	0	1	2		6	4	
6-Series	5	6	5	3	1*	2	1	8	0	5
Hexagonal	6	2	0	2	5	1	2	0	6	
Circular Arc	7	3	0	1	1	0	8			
Supercritical	8	1	2	4	4	5	3	5		

IAF-array

*Numerical value corresponding to letter in designation for a modified series (=A).

APPENDIX B

Interface Program

APPENDIX B

Interface Program

This section is a listing of the input required by the Interface program. The purpose of this program is to convert the geometry data from the 3-D Item Format into the input format of the USSAERO program. In addition, the validity of the data is checked and diagnostic messages printed out declaring why the geometry data cannot be executed in the USSAERO program. The input data required by the Interface program is exactly the output data from the Geometry program, and the output data from the Interface program is exactly the input geometry required by the USSAERO program. It is an INTERCOM program which has small core and processor time requirements. Each geometry component has the following format:

Card 1 - Locator Card.

THE hollerith variable THREED must appear in columns six through 11. This card tells the program that a geometry component follows. This allows geometry data to be mixed with other data forms on the same file as is the normal operating procedure for ICAD.

Card 2 - Control card.

Columns	Variable	Value	Description
1-5	NCARD	1-20	Number of comments cards that follow
7-9	TYP	BOD	Body geometry follows
		POD	Pod geometry follows
		INE	Wing geometry follows

Columns	Variable	Value	Description
		FIN	Fin geometry follows
		CAN	Canard geometry follows
11-30	NAME		For identification purposes only.

Card 3 - Comment Card.

Any alphanumerical data for identification purposes only.

Card 4 - Control Integers

Columns	Variable	Value	Description
1-5	NH	1-30	Number of cross sections or airfoil sections for body or lifting surface geometry respectively.
6-10	NPH	3-50	Number of points per cross section or airfoil section.

Card 5 - Point Card.

There are NH* NPH input cards of this type. Each card has the X, Y, Z coordinate data for a single point in a 3F10.0 format. For bodies, the data begins with the top of the first cross section and proceeds in a clockwise manner for that section and continuing on to the next. For lifting surface geometry, the data begins at the leading edge of the root chord and continues over the airfoil upper surface to the trailing edge and then forward over the lower surface, ending with the leading edge point before going to the next airfoil section.

APPENDIX C

USSAERO Program

APPENDIX C

USSAERO Program

This appendix is a listing of the inputs required by the USSAERO program. Basically it is the Program Input Data section from NASA CR 2228, Part 1, with corrections and additions recommended by Mr. Woodward. The program required 120,000 octal core locations. The time required to run can be estimated using Figures 7 and 8.

Description of configuration geometry input cards. The configuration is defined to be symmetrical about the xz plane, therefore only one side of the configuration need be described. The convention used in this program is to present that half of the configuration located on the positive y side of the xz plane. The number of input cards depends on the number of components used to describe the configuration, and the amount of detail used to describe each component.

Card 1 - Identification. Card 1 contains any desired identifying information in columns 1 - 80.

Card 2 - Control Integers. Card 2 contains 24 integers, each punched right justified in a three column field. Card 2 contains the following:

Columns	Variable	Value	Description
1-3	J0	0	No reference area
		1	Reference area to be read
4-6	J1	0	No wing data
		1	Cambered wing data to be read
		-1	Uncambered wing data to be read
7-9	J2	0	No fuselage data
		1	Data for arbitrarily shaped fuselage to be read
		-1	Data for circular fuselage to be read (with J6 = 0, fuselage will be cambered. With J6 = -1, fuselage will be symmetrical with xy-plane. With J6 = 1, entire configuration will be symmetrical with xy-plane)
10-12	J3	0	No pod (nacelle) data
		1	Pod (nacelle) data to be read

Columns	Variable	Value	Description
13-15	J4	0	No fin (vertical tail) data
		1	Fin (vertical tail) data to be read
16-18	J5	0	No canard (horizontal tail) data
		1	Canard (horizontal Tail) data to be read
19-21	J6	0	A cambered circular or arbitrary fuselage if J2 is nonzero
		1	Complete configuration is symmetrical with respect to xy-plane, which implies an uncambered circular fuselage if there is a fuselage
		-1	Uncambered circular fuselage with J2 nonzero
22-24	NWAF	2-20	Number of airfoil sections used to describe the wing
25-27	NWAFOR	3-30	Number of ordinates used to define each wing airfoil section. If the value of NWAFOR is input with a negative sign, the program will expect to read lower surface ordinates also
28-30	NFUS	1-4	Number of fuselage segments
31-33	NRADX(1)	3-30	Number of points used to represent half-section of first fuselage segment. If fuselage is circular, the program computes the indicated number of y and z ordinates
34-36	NFORX(1)	2-30	Number of stations for first fuselage segment
37-39	NRADX(2)	3-30	Same as NRADX(1), but for second fuselage segment
40-42	NFORX(2)	2-30	Same as NFORX(1), but for second fuselage segment
43-45	NRADX(3)	3-30	Same as NRADX(1), but for third fuselage segment
46-48	NFORX(3)	2-30	Same as NFORX(1), but for third fuselage segment

Columns	Variable	Value	Description
49-51	NRADX(4)	3-30	Same as NRADX(1), but for fourth fuselage segment
52-54	NFORX(4)	2-30	Same as NFORX(1), but for fourth fuselage segment
55-57	NP	0-9	Number of pods described
58-60	NPODOR	4-30	Number of stations at which pod radii are to be specified
61-63	NF	0-6	Number of fins (vertical tails) to be described
64-66	NFINOR	3-10	Number of ordinates used to describe each fin (vertical tail) airfoil section
67-69	NCAN	0-2	Number of canards (horizontal tails) to be described
70-72	NCANOR	3-10	Number of ordinates used to define each canard (horizontal tail) airfoil section. If the value of NCANOR is input with a negative sign, the program will expect to read lower surface ordinates also, otherwise the airfoil is assumed to be symmetrical
73-75	PLOT	0	No plot output
		1	Plot singularity paneling and CP distributions
		-1	Plot input geometry, singularity paneling, and CP distributions

Cards 3, 4, ... - remaining input data cards - The remaining input data cards contain a detailed description of each component of the configuration. Each card contains up to 10 values, each value punched in a seven column field with a decimal point and may be identified in columns 73-80. The cards are arranged in the following order: reference area, wing data cards, fuselage data cards, pod data cards, fin (vertical tail) data cards, and canard (horizontal tail) data cards.

Reference area card: The reference area value is punched in columns 1-7 and may be identified as REFA in columns 73-80.

Wing data cards: The first wing data card (or cards) contains the locations in percent chord at which the ordinates of all the wing airfoils are to be specified. There will be exactly NWAFOR locations in percent chord given. Each card may be identified in columns 73-80 by the symbol XAFJ where J denotes the last location in percent chord given on that card.

The next wing data cards (there will be NWAF cards) each contain four numbers which give the origin and chord length of each of the wing airfoils that is to be specified. The card representing the most inboard airfoil is given first, followed by the cards for successive airfoils. These cards contain the following:

Columns	Contents
1-7	x ordinate of airfoil leading edge
8-14	y ordinate of airfoil leading edge
15-21	z ordinate of airfoil leading edge
22-28	airfoil streamwise chord length
73-80	card identification, WAFORGJ where J denotes the particular airfoil, thus WAFORG1 denotes the most inboard airfoil

If a cambered wing has been specified, the next set of wing data cards is the mean camber line cards. There will be NWAFOR values of delta z referenced in the z ordinate of the airfoil leading edge, each value corresponding to a specified percent chord location on the airfoil. These cards are arranged in the order which begins with the most inboard airfoil and proceeds outboard. Each card may be identified in columns 73-80. as TZORDJ where J denotes the particular airfoil. Note that the z ordinates are dimensional.

Next are the wing ordinate cards. If NWAFOR > 0, there will be NWAFOR values of half thickness specified for each airfoil expressed as percent chord. If NWAFOR ≤ 0, |NWAFOR| values of upper ordinates are followed by |NWAFOR| values of lower ordinates. The program expects both upper and lower ordinates to be punched as positive values in percent chords.

Fuselage data cards: The first card (or cards) specifies the x values of the fuselage stations of the first segment. There will be NFORX(1) values and the cards may be identified in columns 73-80 by the symbol XFUSJ where J denotes the number of the last fuselage station given on that card.

If the fuselage is circular, the next card (or cards) gives the fuselage cross sectional areas, and may be identified in columns 73-80 by the symbol FUSARDJ where J denotes the number of the last fuselage station given on that card. If the fuselage is of arbitrary shape, NRADX(1) values of the y-ordinates for a half section are given and identified in columns 73-80 as YJ where J is the station number. Following the y-ordinates are the NRADX(1) values of the corresponding z-ordinates for the half section identified in columns 73-80 as ZJ where J is the station number. Each station will have a set of y and z, and the convention of ordering the ordinates from bottom to top is observed.

For each fuselage segment a new set of cards as described must be provided. The segment descriptions should be given in order of increasing values of x.

Pod data cards: The first pod (nacelle) data card specifies the location of the origin of the first pod. The card contains the following:

Columns	Contents
1-7	x ordinate of origin of first pod
8-14	y ordinate of origin of first pod
15-21	z ordinate of origin of first pod
73-80	card identification, PODORGJ where J denotes the pod number.

The next pod input data card (or cards) contains the x-ordinates, referenced to the pod origin, at which NPODOR values of the pod radii are to be specified. The first x value must be zero and the last x value is the length of the pod. These cards may be identified in columns 73-80 by the symbol XPODJ where J denotes the pod number.

For each additional pod, new PODORG, XPOD, and PODR cards must be provided. Only single pods are described but the program assumes that if the y-ordinate is not zero an exact duplicate is located symmetrically with respect to the xz-plane, a y-ordinate of zero implies a single pod.

Fin data cards: Exactly three data input cards are used to describe a fin (vertical tail). The first fin data card contains the following:

Columns	Contents
1-7	x-ordinate on inboard airfoil leading edge
8-14	y-ordinate of inboard airfoil leading edge
15-21	z-ordinate of inboard airfoil leading edge
22-28	chord length of inboard airfoil
29-35	x-ordinate of outboard airfoil leading edge
36-42	y-ordinate of outboard airfoil leading edge
43-49	z-ordinate of outboard airfoil leading edge
50-56	chord length of outboard airfoil
73-80	card identification, FINORGJ where J denotes the fin number

The second fin input data card contains NFINOR values of x expressed in percent chord at which the fin airfoil ordinates are to be specified. The card may be identified in columns 73-80 as XFINJ where J denotes the fin number.

The third fin input data card contains NFINOR values of the fin airfoil half-thickness expressed in percent chord. Since the fin airfoil must be symmetrical, only the ordinates on the positive y side of the fin chord plane are specified. The card identification FINORDJ may be given in columns 73-80 where J denotes the fin number.

For each fin, new FINORG, XFIN, and FINORD cards must be provided. Only single fins are described but the program assumes that if the y-ordinate is not zero an exact duplicate is located symmetrically with respect to the xz-plane, a y-ordinate of zero implies a single fin.

Canard data cards: If the canard (or horizontal tail) airfoil is symmetrical, exactly three cards are used to describe a canard, and the input is given in the same manner as for a fin. If, however, the canard airfoil is not symmetrical (indicated by a negative value of NCANOR), a fourth canard input data card will be required to give the lower ordinates. The information presented on the first canard input data card is as follows:

Columns	Contents
1-7	x-ordinate of inboard airfoil leading edge
8-14	y-ordinate of inboard airfoil leading edge
15-21	z-ordinate of inboard airfoil leading edge
22-28	chord length of inboard airfoil
29-35	x-ordinate of outboard airfoil leading edge
36-42	y-ordinate of outboard airfoil leading edge
43-49	z-ordinate of outboard airfoil leading edge
50-56	chord length of outboard airfoil
73-80	card identification, CANORGH where J denotes the canard number

The second canard input data card contains NCANOR values of x expressed in percent chord at which the canard airfoil ordinates are to be specified. The card may be identified in columns 73-80 as XCANJ where J denotes the canard number.

The third canard input data card contains NCANOR values of the canard airfoil half-thickness expressed in percent chord. This card may be identified in columns 73-80 as CANORDJ where J denotes the canard number. If the canard airfoil is not symmetrical, the lower ordinates are presented on a second CANORD card. The program expects both upper and lower ordinates to be punched as positive values in percent chord.

For another canard, new CANORG, XCAN, and CANORD cards must be provided.

Plot Control Cards. If the geometry plot option was specified, one or more plot control cards follow. The plot control card contains all the needed information for a single plot. Several plot options are available, each with its own input. For all options, the plot card columns 1-7 and 53-55 contain alphanumeric information and card columns 8-52 contain real numbers with a decimal point required. The geometry paneling plot cards may be identified in columns 73-80 as GPLOT. The orthographic projection option plot card contains the following information.

Columns	Variable	Value	Description
1	HORZ	X,Y,Z	Horizontal axis
3	VERT	X,Y,Z	Vertical Axis
5-7	TEST1	OUT	Deletes hidden lines, TEST1 should be blank in order to plot the hidden lines.
8-12	PHI	0.-360	Roll angle, degrees
13-17	THETA	0.-360	Pitch Angle, degrees
18-22	PSI	0.-360	Yaw angle, degrees
48-52	PLOTSZ	Real	PLOTSZ determines the size of the plot. The scale factor is computed using PLOTSZ and the maximum dimension of the configuration.
53-55	TYPE	ORT	Indicates orthographic projection.
72	KODE	0 1	Continue reading plot cards. After processing this plot, read new configuration description.

The stacked three-view (plan, front, and side views) plot card contains the following information:

Columns	Variable	Value	Description
8-12	PHI	Real	y-origin on paper of plan view, inches
13-17	THETA	Real	y-origin on paper of side view, inches
18-22	PSI	Real	y-origin on paper of front view, inches
48-52	PLOTSZ	Real	PLOTSZ determines the size of the plot. The scale factor is computed using PLOTSZ and the maximum dimension of the configuration.
53-55	TYPE	VU3	Indicates three-view plot.
72	KODE	0 1	Continue reading plot cards. After processing this plot, read new configuration description.

The perspective view plot card contains the following information:

Columns	Variable	Value	Description
8-12	PHI	Real	x of view point (location of viewer) in data coordinate system
13-17	THETA	Real	y of view point in data coordinate system
18-22	PSI	Real	z of view point in data coordinate system
23-27	XF	Real	x of focal point (determines direction and focus) in data coordinate system.
28-32	YF	Real	y of focal point in data coordinate system.
33-37	ZF	Real	z of focal point in data coordinate system.
38-42	DIST	Real	Distance from eye to viewing plane, inches.
43-47	FMAG	Real	Viewing plane magnification factor. FMAG controls the size of the projected image.
48-52	PLOTSZ	Real	Diameter of viewing plane, inches. DIST and PLOTSZ together determine a cone which is the field of vision.
53-55	TYPE	PER	Indicates perspective plot.
72	KODE	0	Continue reading plot cards.
		1	After processing this plot, read new configuration description.

The stereo plot card contains the following information:

Columns	Variable	Value	Description
8-12	PHI	Real	x of view point (location of viewer) in data coordinate system.
13-17	THETA	Real	y of view point in data coordinate system.
18-22	PSI	Real	z of view point in data coordinate system.

Columns	Variable	Value	Description
23-27	XF	Real	x of focal point (determines direction and focus) in data coordinate system.
28-32	XF	Real	y of focal point in data coordinate system.
33-37	ZF	Real	z of focal point in data coordinate system.
38-42	DIST	Real	Distance from eye to viewing plane in inches.
43-47	FMAG	Real	Viewing plane magnification factor. FMAG controls the size of the projected image.
48-52	PLOTSZ	Real	Diameter of viewing plane inches. DIST and PLOTSZ together determine a cone which is the field of vision. The value of PLOTSZ is also relative to the type of viewer which is to be used.
53-55	TYPE	STE	Indicates stereo plot.
	KODE	0	Continue reading plot cards
		1	After processing this plot, read new configuration description

Description of Auxiliary Input Cards

Card 1.1 - Identification. Card 1.1 contains any desired identifying information in columns 1-80.

Card 1.2 - Boundary condition and control point definition. Nonplanar boundary conditions are always applied on a body, however, card 1.2 permits the selection of boundary conditions to apply on a wing, fin (vertical tail), or canard (horizontal tail). This card also selects the output print options. This card contains the following:

Columns	Varialbe	Value	Description
1-3	LINBC	0	Control points on surface of wing, fin (vertical tail), and canard (horizontal tail). This is referred to as the non-planar boundary condition option.
		1	Control points in plane of wing, fin (vertical tail), and canard (horizontal tail). This is referred to as the planar boundary condition option.
4-6	THICK	0	Do not calculate wing thickness matrix
		1	Calculate wing thickness matrix if LINBC = 1
7-9	PRINT	0	Print out the pressures and the forces and moments
		1	Print out option 0 and the spanwise loads on the wing, fins, and canards.
		2	Print out option 1 and the velocity components and source and vortex strengths
		3	Print out option 2 and the steps in the iterative solution
		4	Print out option 3 and the axial and normal velocity matrices
18	ITMATH	0	Gauss - Siedel
		1	Jacobi
		2	Gauss - Siedel
		3	Controlled successive overrelaxation
		4	Successive overrelaxation

A negative value of print adds the panel geometry print out to the output indicated for options 1-4.

LINBC, THICK, and PRINT are punched as right justified integers. THICK is not used if LINBC = 0.

Card 2.1 - Revised configuration paneling description control integers. The contents of card 2.1 are punched as right justified integers as follows:

Columns	Variable	Value	Description
1-3	K0	0	No reference lengths
		1	Reference length data to be read.
4-6	K1	0	No wing data
		1	Wing data to be read, wing has a sharp leading edge
		3	Wing data to be read, wing has a round leading edge
7-9	K2	0	No body data
		1	Body data follows
10-12	K3		Not used
13-15	K4	0	No fin (vertical tail) data
		1	Fin (vertical tail) data to be read, fin has a sharp leading edge
		3	Fin (vertical tail) data to be read, fin has a round leading edge
16-18	K5	0	No canard (horizontal tail) data
		1	Canard (horizontal tail) data to be read, canard has a sharp leading edge.
		3	Canard (horizontal tail) data to be read, canard has a round leading edge.
19-21	K6		Not used
22-24	KWAF	0, 2-20	Number of wing sections used to define the inboard and outboard panel edges. If KWAF = 0, the panel edges are defined by NWAF in the geometry input
25-27	KWAFOR	0, 3-30	Number of ordinates used to define the leading and trailing edges of the wing panels. If KWAFOR = 0, the panel edges are defined by NWAFOR in the geometry input.
28-30	KFUS		The number of fuselage segments. The program sets KFUS = NFUS

Columns	Variable	Value	Description
31-33	KRADX(1)	0, 3-20	Number of meridian lines used to define panel edges on first body segment. There are three options for defining the panel edges. If KRADX(1) = 0, the meridian lines are defined by NRADX(1) in the geometry input. If KRADX(1) is positive, the meridian lines are calculated at KRADX(1) equally spaced PHIKs. If KRADX(1) is negative, the meridian lines are calculated at specified values of PHIK
34-36	KFORX(1)	0, 2-30	Number of axial stations used to define leading and trailing edges of panels on first body segment. If KFORX(1) = 0, the panel edges are defined by NFORX(1) in the geometry input
37-39	KRADX(2)	0, 3-20	Same as KRADX(1), but for second body segment
40-42	KFORX(2)	0, 2-30	Same as KFORX(1), but for second body segment
43-45	KRADX(3)	0, 3-20	Same as KRADX(1), but for third body segment
46-48	KFORX(3)	0, 2-30	Same as KFORX(1), but for third body segment
49-51	KRADX(4)	0, 3-20	Same as KRADX(1), but for fourth body segment
52-54	KFORX(4)	0, 2-30	Same as KFORX(1), but for fourth body segment

The program is restricted to 600 body singularity panels. For this program there is an additional restriction that the total number of singularity panels in the axial direction on the body (fuselage) cannot exceed 30. The arbitrary body (fuselage) capability of this program is limited to those shapes for which the radius is a single-valued function of PHIK for each cross section of the body.

Card 2.2 - Additional revised configuration paneling description control integers. The contents of card 2.2 are punched as right justified integers as follows:

Columns	Variable	Value	Description
1-3	KF(1)	0, 2-20	Number of fin sections used to define the inboard and outboard panel edges on the first fin. If KF(1) = 0, the root and tip chords define the panel edges
4-6	KFINOR(1)	0, 3-30	Number of ordinates used to define the leading and trailing edges of the fin panels on the first fin. If KFINOR(1) = 0, the panel edges are defined by NFINOR
7-9	KF(2)	0, 2-20	Same as for KF(1), but for second fin
10-12	KFINOR(2)	0, 3-30	Same as for KFINOR(1), but for second fin
13-15	KF(3)	0, 2-20	Same as for KF(1), but for third fin
16-18	KFINOR(3)	0, 3-30	Same as for KFINOR(1), but for third fin
19-21	FK(4)	0, 2-20	Same as for KF(1), but for fourth fin
22-24	KFINOR(4)	0, 3-30	Same as for KFINOR(1), but for fourth fin
25-27	KF(5)	0, 2-20	Same as for KF(1), but for fifth fin
28-30	KFINOR(5)	0, 3-30	Same as for KFINOR(1), but for fifth fin
31-33	KF(6)	0, 2-20	Same as for KF(1), but for sixth fin
34-36	KFINOR(6)	0, 3-30	Same as for KFINOR(1), but for sixth fin
37-39	KCAN(1)	0, 2-20	Number of canard sections used to define the inboard and outboard panel edges on the first canard. If KCAN(1) = 0, the root tip chords define the panel edges. If KCAN(N) negative, no vortex sheets carry through the body and concentrated vortices are shed from the inboard edge of the canard or tail surface.

Columns	Variable	Value	Description
40-42	KCANOR(1)	0, 3-30	Number of ordinates used to define the leading and trailing edges of the first canard. If KCANOR(1) = 0, the panel edges are defined by NCANOR
43-45	KCAN(2)	0, 2-20	Same as for KCAN(1), but for second canard
46-48	KCANOR(2)	0, 3-30	Same as for KCANOR(1), but for second canard
49-51	KCAN(3)	0, 2-20	Same as for KCAN(1), but for third canard
52-54	KCANOR(3)	0, 3-30	Same as for KCANOR(1), but for third canard
55-57	KCAN(4)	0, 2-20	Same as for KCAN(1), but for fourth canard
58-60	KCANOR(4)	0, 3-30	Same as for KCANOR(1), but for fourth canard
61-63	KCAN(5)	0, 2-20	Same as for KCAN(1), but for fifth canard
64-66	KCANOR(5)	0, 3-30	Same as for KCANOR(1), but for fifth canard
67-69	KCAN(6)	0, 2-20	Same as for KCAN(1), but for ixth canard
70-72	KCANOR(6)	0, 3-30	Same as for KCANOR(1), but for ixth canard

The program is restricted to a total of 600 singularity panels on the wing-fin-canard combination.

For this program there is an additional restriction that the total number of singularity panels in the spanwise direction on the wing-fin-canard combination cannot exceed 20.

Cards 3, 4, ... - remaining input data cards. The remaining input data cards contain a detailed description of the singularity paneling of each component of the configuration. Each card contains up to 10 values, each value punched in a seven-column field with a decimal point and may be identified in columns 73-80. The cards are arranged in the following order: reference lengths, wing data cards, fuselage (body) data cards, fin (vertical tail) data cards, canard (horizontal tail) data cards, singularity paneling plot cards, and finally Mach number and angle of attack case cards. Note that the present program will not handle a pod and therefore there are no pod panel inputs. However, if the geometry input contains a pod description it will be read and ignored.

Reference length card: This card may be identified as REFL in columns 73-80 and contains the following:

Columns	Variable	
1-7	REFA	Wing reference area. If REFA = 0, the reference area is defined by the value of REFA in the geometry input
8-14	REFB	Wing semispan. If REFB = 0, a value of 1.0 is used for the reference semispan
15-21	REFC	Wing reference chord. If REFC = 0, a value of 1.0 is used for the reference chord
22-28	REFD	Body (fuselage) reference diameter. If REFD = 0, a value of 1.0 is used for the reference diameter
29-35	REFL	Body (fuselage) reference length. If REFL = 0, a value of 1.0 is used for the reference length
36-42	REFX	x coordinate of moment center
43-49	REFZ	z coordinate of moment center

Wing data cards: The first wing data card is the wing leading edge radius card and is required only when K1 = 3. This card contains NWAF values of leading edge radius expressed in percent chord. It may be identified in columns 73-80 as RHOJ where J denotes the number of the last radius given on that card.

Next is the wing panel leading edge card. This card contains KWAFOR values of wing panel leading edge locations expressed in percent chord. This card may be identified in columns 73-80 as XAFKJ where J denotes the last location in percent chord given on that card. Omit if KWAFOR = 0.

The last wing data card gives the wing panel side edge data. This card contains KWAF values of the y ordinate of the panel inboard edges. This card may be identified in columns 73-80 as YKJ where J denotes the last y ordinate on that card. These values are arranged in the order which begins with the most inboard panel edge and proceeds outboard. Omit if KWAF = 0.

Fuselage (body) data cards: The first body card is the body meridian angle card. This card contains KRADX(1) values of body meridian angle expressed in degrees and may be identified in columns 73-80 as PHIKJ where J denotes the body segment number. The convention is observed that PHIK = 0, at the bottom of the body and PHIK = 180 at the top of the body. Omit unless KRADX(1) is negative. Repeat this card for each fuselage segment.

The second body card is the body axial station card. This card contains KFORX(1) values of the x ordinate of the body axial stations and may be identified in columns 73-80 as XFUSKJ where J denotes the body segment number. Omit if KFORX(1) = 0. Repeat this card for each fuselage segment.

Fin (vertical tail) data cards: The first fin data card is the fin leading edge radius card and is required only when K4 = 3. This card contains NF values of leading edge radius expressed in percent chord, one value for each fin. It may be identified in columns 73-80 as RHOFIN.

Next is the fin panel leading edge card for the first fin. This card contains KFINOR(1) values of fin panel leading edge locations expressed in percent chord. This card may be identified in columns 73-80 as XFINKJ where J denotes the fin number. Repeat this card for each fin.

The last fin data card gives the fin panel side edge data for the first fin. This card contains KF(1) values of the z-ordinate of the panel inboard edges. This card may be identified in columns 73-80 as the order that begins with the most inboard panel edge and proceeds outboard. Repeat this card for each fin.

Canard (horizontal tail) data cards: The first canard data card is the canard leading edge radius card and is required only when K5 = 3. This card contains NCAN values of leading edge radius expressed in percent chord, one value for each canard. It may be identified in columns 73-80 as RHOCAN.

Next is the canard panel leading edge card for the first canard. This card contains KCANOR(1) values of canard panel leading edge locations expressed in percent chord. This card may be identified in columns 73-80 as XCANKJ where J denotes the canard number. Repeat this card for each canard.

The last canard data cards gives the canard panel side edge data for the first canard. This card contains KCAN(1) values of the y-ordinate of the panel inboard edges. This card may be identified in columns 73-80 as YCANKJ where J denotes the canard number. These values are arranged in the order that begins with the most inboard panel edge and proceeds outboard. Repeat this card for each canard.

Plot Control Cards for Singularity Paneling: Input identification is the same as for the configuration plots.

Card 3.0 - Aerodynamics Input Cards

Columns	Variable	Value	Description
1-7	MACH	REAL	The free stream subsonic Mach number (including MACH = 0.) or supersonic Mach number at which aerodynamic output is desired
		-1	Indicates the termination of the aerodynamic calculation for the given configuration. Geometry cards for a new configuration can follow such a terminal card
8-14	ALPHA	REAL	The angle-of-attack in degrees at which aerodynamic output is desired
15-21	NORVEL	0	The usual boundary condition of zero normal velocity is applied at body panel control points
		1.0	Modified boundary condition applied at body panel control points (non-zero normal velocities are read on card set 3.1)
22-28	LMACH	0	Perform one pass through the program to obtain the solution corresponding to the free stream Mach number.
29-35	FLDPTS	REAL	Velocities and pressures calculated at field points read in on card set 3.2. The number of field points equals FLDPTS
		0	No field point calculations.

Card 3.1 - Normal Velocity Input Cards. These cards contain the values of the normal velocities specified at the control point of each body panel. The data is input in each 10F7.0 format. One value of the normal velocity is input for each body panel, in order of the body panel numbers assigned by the program.

Card Set 3.2 - Field Point Input Cards. One card is required for each field point containing the following:

Columns	Variable	
1-7	XPT	x coordinate of the field point
8-14	YPT	y coordinate of the field point
15-21	XPT	z coordinate of the field point

A maximum of 600 field points may be read.

A series of Mach number and angle-of-attack values for the same configuration geometry may be calculated by repeating card set 3.0 with the desired values.

Program Output Data

All output is processed by a standard 132 characters-per-line printer. The output from each run is always preceded by a complete list of the input data cards. The amount and type of the remaining output depend on the PRINT option selected, the number of panels used, and whether the configuration being analyzed is an isolated wing, an isolated body, or a complete wing-body-tail combination. The program output options are described below:

PRINT = 0 The program prints the case description, Mach number and angle of attack, followed by a table listing the panel number, control point coordinates (both dimensional and nondimensional), pressure coefficient, normal force, axial force, and pitching moment. Separate tables are printed for the body and wing panels, noting that any tail, fin or canard panels are included with the wing output. If the planar boundary condition option has been selected, the results for the wing upper surface are given in one table, followed by a separate table giving the results for the wing lower surface. Additional tables giving the total coefficients on the body, the wing and the complete configuration follow the pressure coefficient tables. These include the reference area, reference span and reference chord, the normal force, axial force, pitching moment, lift, and drag coefficients, and the center of pressure of the component.

PRINT = 1 In addition to the output described for PRINT = 0, the program prints out additional tables giving the normal force, axial force, pitching moment, lift and drag coefficients, and the center of pressure of each column of panels on the wing and tail surfaces. In addition, the indices of the first and last panel in the column are listed, together with the span, chord and origin of the column.

PRINT = 2 In addition to the output described for PRINT = 1, the program prints out tables listing the panel number, the source or vortex strength of that panel, and the axial velocity u, lateral velocity v, and vertical veloxity w at the panel control point. The normal velocity is also calculated for body panels. Separate tables are printed for the body and wing panels, noting again that any tail, fin, or canard panels are included with the wing output. If the planar boundary condition option has been selected, separate tables are given for the wing upper and lower surfaces.

PRINT = 3 In addition to the output described for PRINT = 2, the program prints out the iteration number, and the source and vortex strength arrays obtained at each step of the iterative solution procedure.

PRINT = 4 In addition to the output described for PRINT = 3, the program prints out tables of the axial and normal velocity components which make up the elements of the aerodynamic matrices. The program prints out the matrix row number, and gives the number of elements in that row. A maximum of four matrix partitions will be printed if this option is selected, each of which is identified by number and its influence description prior to printing the velocity component tables.

If a negative value of PRINT is selected, the program prints all the information described above for the positive values, together with the complete panel geometry description of the configuration following the list of input cards. This consists of tables giving the wing panel corner points, control points, inclination angles, areas, and chords. If the configuration has a horizontal tail, fin or canard, additional tables are printed giving the same information as listed above for the wing. Finally, if the configuration includes a body, the body panel corner points, control points, areas, and inclination angles are listed.

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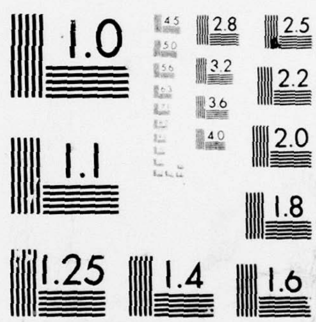

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VITA

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There is a frequent need for accurate aerodynamic force data in preliminary design. Data must be available on many configurations, some of which deviate significantly from existing aircraft. A method which fills the gap between statistically based predictions and wind tunnel testing is a computer solution to the linearized potential flow equations of motion. The Woodward USSAERO program was selected to calculate potential flow force distributions on arbitrary aircraft geometry. Five major aspects of the program must be understood by the user to insure that the program is capable of supplying the required

data. They are the differential equation, boundary conditions, singularity types, matrix operations, and force and moment calculations. In order to operate this program in the preliminary design environment, it was interfaced with existing geometry data bases with a separate Interface program. A third program, the Geometry program, was written to speed the definition of a complete aircraft configuration in a format compatible with several existing analysis programs. It defines arbitrary fuselage geometry as a series of cross-sections using an Tektronix Interactive Terminal and Digitizer. It defines lifting surface geometry as a series of streamwise airfoil sections with several different airfoil shapes being available. To perform an aerodynamic analysis using the system of programs is a five step process: aircraft components (wing, body, fin, canard) must be identified; the geometry must be defined; the data must be converted into the USSAERO format; the singularity paneling must be defined; and finally, additional nongeometric data must be defined. When data from an existing geometric data base are to be used and they are compatible with USSAERO, only the last three steps are required. The system of programs were applied to the F-111A aircraft as an example case. The results of that analysis show excellent agreement with wind tunnel data for pressure distributions on the wing at moderately high subsonic Mach numbers.