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MISSILE DATCOM

VOLUME II – USER'S MANUAL

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This technical report has been reviewed and is approved for publication.

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This user's guide summarizes the Missile Datcom Phase IV code which is version 12/88. The Missile Datcom computer code calculates aerodynamics and stability and control characteristics with predictive accuracy suitable for preliminary design.

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PREFACE

This report was prepared for the Flight Dynamics Laboratory of the Air Force Wright Aeronautical Laboratories, Wright-Patterson AFB, Ohio, under Contract F33615-81-C-3617. The work was performed by McDonnell Douglas Astronautics Company, St Louis, Missouri (MDAC-STL), a division of McDonnell Douglas Corporation. The initial development of Missile Datcom was initiated in September 1981 and completed in December 1985. Mr. Jerry E. Jenkins (AFWAL/FIGC) was the Air Force project engineer. Mr. Steven R. Vukelich was the principal investigator for the period September 1981 to February 1985. Mr. Stanley L. Stoy assumed the responsibilities of Principal Investigator in February 1985.

The authors gratefully acknowledge the contributions of the AFWAL/FIGC staff, whose guidance was instrumental for the successful completion of the study. The encouragement and support from numerous professionals from the aerospace community were significant to the effort.

AFWAL is committed to the continuing development of Missile Datcom. This development is dependent to a large extent on user feedback. Questions about the program or suggestions for future improvements to the program should be directed to the current Air Force project engineer, Mr. William Blake, at (513) 255-6764.

Volume I of this report summarizes the method selection philosophy and method selection criteria for the Missile Datcom development program. Volume II is the program User's Manual.

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Table of Contents

Table of Contents (Continued)

List of Illustrations

List of Illustrations (Continued)

List of Tables

1.0 INTRODUCTION

In missile preliminary design it is necessary to quickly and economically estimate the aerodynamics of a wide variety of missile configuration designs. Since the ultimate shape and aerodynamic performance are so dependent upon the subsystems utilized, such as payload size, propulsion system selection and launch mechanism, the designer must be capable of predicting a wide variety of configurations accurately. The fundamental purpose of Missile Datcom is to provide an aerodynamic design tool which has the predictive accuracy suitable for preliminary design, and the capability for the user to easily substitute methods to fit specific applications.

The Missile Datcom computer code calculates the static stability and control characteristics using the methods discussed in the Missile Datcom Handbook. Although the companion Handbook is not required to use the computer code, it serves as documentation for the incorporated methods.

This manual consists of two volumes. This Volume is the program User's Guide. Instructions for incorporating the program on the user's computer system are detailed in Volume 2. The program has been written to closely conform to the American National Standards Institute (ANSI) Fortran IV language, and is written for the Control Data Corporation (CDC) computers. The code is compatible with the Fortran V language and is available in a VAX version. It can be easily converted to other computer systems.

Questions regarding the code should be directed to the United States Air Force Wright Aeronautical Laboratories, AFWAL/FIGC, Wright-Patterson AFB, Ohio 45433, Telephone (513) 255-8485.

2.0 PROGRAM CAPABILITIES

The computer code is capable of addressing a wide variety of conventional missile designs. For the purposes of this document, a conventional missile is one which is comprised of the following:

- o An axisymmetric or elliptically-shaped body
- o One to four fin sets located along the body between the nose and base. Each fin set can be comprised of one to eight identical panels attached around the body at a common longitudinal position
- o An airbreathing propulsion system.

To minimize the quantity of input data required, commonly used values for many inputs are assumed as defaults. However, all program defaults can be over-ridden by the user in order to more accurately model the configuration of interest.

The following paragraphs detail the configurations that can be analyzed. Later paragraphs describe the range of aerodynamic coefficients that can be predicted. Finally, the program constraints are discussed.

2.1 Addressable Configurations

The following configurations can be analyzed:

a) Circular or elliptically-shaped cross section bodies, with or without airbreathing inlets

b) Fin alone (1 to 8 panels attached at the root)

c) Body and up to 4 finsets (1 to 8 panels in each finset)

d) The body and finset configurations with deflected fins

Certain restrictions exist due to method limitations and are summarized in the following paragraphs.

2.1.1 Axisymmetric or Elliptical Bodies - Methodology is incorporated that permits analysis of the configuration components summarized in Table 1. Due to the types of methods selected restrictions also apply to the manner in which these components are joined to form a complete configuration:

Subsonic/transonic speeds - The aerodynamic methods assume that the body is, as a minimum, composed of a nose-cylinder combination. The afterbody (boattail or flare) is optional, but if used, it must be attached to a cylindrical center body whose length is at least four body diameters; this restriction minimizes nose flow field coupling over the afterbody. If an afterbody is specified it must not be cylindrical, e.g., the base diameter must be different than the centerbody diameter. Table 2 summarizes the other restrictions on the configurations.

Supersonic speeds - The aerodynamic methods used are not restricted to nose-cylinder combinations. Any arbitrary radii distribution can be defined since theoretical techniques are employed at Mach numbers above 1.4. Care should be taken to avoid introducing unexpected corners into the contour. If the contour has any concaved regions the marching may fail due to shock impingement on the body as it starts to curve out.

2.1.2 Panels - The program will accept inputs to describe most airfoil sections or planforms. Certain assumptions and limitations are made and summarized in the following paragraphs.

2.1.2.1 Airfoil Section - The program will accept virtually any symmetrical airfoil section or NASA subsonic cambered section. The airfoil section can be defined using a NACA designation or by supplying the coordinates of the section. Circular arc, hexagonal, or diamond shaped sections can also be specified. A symmetric hexagonal cross-section is the default; its shape is computed using the planform inputs. Hence, explicit definition of the airfoil section is optional. Although cambered airfoil sections can be input, their use in the code is currently limited to subsonic applications.

2.1.2.2 Planform - Each set of fins may be comprised of up to eight separate panels. It is assumed that each panel is geometrically identical. Although planforms may be described by up to 10 separate pieces or sections, an equivalent straight-tapered panel is computed and used at all speeds. There is no capability to specify a panel with outboard dihedral.

2.1.3 Airbreathing Inlets - Both axisymmetric and two-dimensional airbreathing inlets can be defined on the geometry. They can be in combinations of 1, 2, 3 or 4 identical inlets and can be positioned around the body at arbitrary angles. Due to strong component interference effects, inletted vehicles can only be analyzed at supersonic speeds.

2.2 Types of Data Computed

2.2.1 Aerodynamics - The program computes the following aerodynamic parameters as a function of angle of attack for each configuration:

C_N	Normal Force Coefficient
C_L	Lift Coefficient
C_m	Pitching Moment Coefficient
X_{cp}	Center of Pressure in calibers from the moment reference center
C_A	Axial Force Coefficient
C_D	Drag Coefficient
C_Y	Side Force Coefficient
C_n	Yawing Moment Coefficient
C_ℓ	Rolling Moment Coefficient
$C_{N\alpha}$	Normal force coefficient derivative with angle of attack
$C_{m\alpha}$	Pitching moment coefficient derivative with angle of attack
C_{Y_β}	Side force coefficient derivative with sideslip angle
C_{n_β}	Yawing moment coefficient derivative with sideslip angle
C_{ℓ_β}	Rolling moment coefficient derivative with sideslip angle

The derivative output can be in degrees or radians. Partial output results, which detail the components used in the calculations, are also optionally available.

The program has the capability to perform a static trim of the configuration, using any finset for control with fixed incidence on the other sets. The two types of aerodynamic output available from the trim option are as follows:

- Untrimmed data - Each of the aerodynamic force and moment coefficients are printed in a matrix, which is a function of angle of attack and panel deflection angle. This output is optional.

- Trimmed data - The trimmed aerodynamic coefficients, and trim deflection angle, are output as a function of angle of attack.

2.2.2 Geometry - All components of the configuration have their physical properties calculated and output for reference if requested. All data is supplied in the user selected system of units.

2.2.3 Other - The reference area and reference length are user defined. The user may optionally select to print the calculated body or fin pressure coefficient distributions at supersonic speeds. Outputs of the partial aerodynamic results and a summary of method extrapolations are also optionally available.

2.3 Operational Considerations

The code has been written to conform to the coding standards for the American National Standards Institute (ANSI) FORTRAN IV or FORTRAN 66. The code has also been designed to be compatible with ANSI Standard FORTRAN X3.9-1978, often referred to as FORTRAN V or FORTRAN 77.

There are only four exceptions to the ANSI standards used in the computer code:

1. Transfer on end-of-file. The FORTRAN IV statement IF(EOF(UNIT).NE.0) is used in the standard CDC compatible code. The FORTRAN V statement END=label in the FORTRAN READ statements is incorporated in the code, but is inactive in the CDC code.

2. NAMELIST. The use of namelist for input and output (I/O) is used. Although this appears to be a violation of FORTRAN IV, it is really not since a namelist emulator has been written for the Missile Datcom code using FORTRAN IV.

3. Mathematical functions. A few mathematical functions are not considered "standard," such as the trigonometric tangent. Standard FORTRAN equivalents for these functions are available on request.

4. PROGRAM card. The code was developed on the Control Data Corporation CYBER computers. This system requires that the first card of the main routine be a PROGRAM statement. An IBM or VAX compatible version of the code is also maintained which has a different format for the program card.

Procedures for making the code operational on other than Control Data Corporation computer equipment are given in Volume 2 to this manual. Only minor coding changes are required and can be accomplished by anyone having an understanding of the FORTRAN language. The code is currently operational on CYBER, VAX, PRIME, IBM, UNIVAC and HARRIS computer systems.

3.0 INPUT DEFINITION

Inputs to the program are grouped by "case." A "case" consists of a set of input cards which define the flight conditions and geometry to be run. Provisions are made to allow multiple cases to be run. The successive cases can either incorporate the data of the previous case (using the input card SAVE) or be a completely new configuration design. The SAVE feature permits the user to define a body and wing (or canard) configuration in the first case and vary the tail design for subsequent cases.

The scheme used to input data to the computer program is a mixture of namelist and control cards. (See Section 4.5 on alternative input schemes.) This combination permits the following:

- Inputs are column independent and can be input in any order.

- All numeric inputs are related to mnemonic (variable) names.

- Program input "flags" are greatly reduced. Required "flags" are identified by a unique alphabetic name which corresponds to the option selected.

The program includes an error checking routine which scans all inputs and identifies all errors. This process is a single-pass error checking routine; all errors are identified in a single "run." In addition, the program checks for necessary valid inputs, such as a non-zero Reynolds number. In some cases, the code will take corrective action. The type of corrective action taken is summarized later in this section.

Flexibility has been maintained for all user inputs and outputs. The following summarize the program generality available:

- The units system can be feet, inches, meters or centimeters. The default is feet.

- Derivatives can be expressed in degree or radian measure. Degree measure is the default.

- The body geometry can be defined either by shape type or by surface coordinates.

- The airfoil can be user defined, NACA or supersonic shaped sections. The NACA sections are defined using the NACA designation. A hexagonally shaped supersonic section is the default.

- The configuration can be run at a fixed sideslip angle and varying body angle of attack, or a fixed aerodynamic roll angle and varying total angle of attack.

- The flight conditions can be user defined, or set using a Standard Atmosphere model. The capability to define wind tunnel test conditions as the flight conditions is also available. The default flight condition is zero altitude.

3.1 Namelist Inputs

The required program inputs use FORTRAN namelist. Missile Datcom is similar to other codes which use the namelist input technique, but differ as follows:

- Namelist inputs are column independent, and can begin in any column including the first. If a namelist is continued to a second card, the continued card must leave column 1 blank. Also, the card before the continued card must end with a comma. The last usable column is number 79 if column 1 is used, and column 80 if column 1 is blank.

- The same namelist can be input multiple times for the same input case. The total number of namelists read, including repeat occurrences of the same namelist name, must not exceed 300.

The three namelist inputs

```
$REFQ  SREF=1.,$
$REFQ  LREF=2.,$
$REFQ  ROUGH=0.001,$
```

are equivalent to

```
$REFQ  SREF=1.,LREF=2.,ROUGH=0.001,$
```

The last occurence of a namelist variable in a case is the value of that variable used for the calculations.

The three namelist inputs

```
$REFQ  SREF=1.,$
$FLTCON NMACH=2.,MACH=1.0, 2.0,$
$REFQ SREF=2.,$
```

are equivalent to

```
$REFQ SREF=2.,$
$FLTCON NMACH=2., MACH=1.0, 2.0,$
```

- o The namelists can be input in any order.
- o Only those namelists required to execute the case need be entered.
- o Certain hollerith constants are permitted. They are summarized in Table 3. Note that any variable can be initialized by using the constant UNUSED; for example, LREF=UNUSED sets the reference length to its initialized value.

All Missile Datcom namelist inputs are either real numbers or logical constants. Integer constants will produce a nonfatal error message from the error checking routine and should be avoided.

The namelist names have been selected to be mnemonically related to their physical meaning. The eight namelists available are as follows:

Namelist	Inputs
$FLTCØN	Flight Conditions (Angles of attack, Mach numbers, etc.)
$REFQ	Reference quantities (Reference area, length, etc.)
$AXIBØD	Axisymmetric body definition
$ELLBØD	Elliptical body definition
$FINSETn	Fin descriptions by fin set (n is the fin set number; 1, 2, 3 or 4)
$DEFLCT	Panel incidence (deflection) values
$TRIM	Trimming information
$INLET	Inlet geometry
$EXPR	Experimental data
$ARBBØD	Arbitrary body geometry

Each component of the configuration requires a separate namelist input. Hence, an input case of a body-wing-tail configuration requires at least one of each of the following namelist inputs, since not all variables have default values assigned:

$FLTCØN	to define the flight conditions
$AXIBØD, ARBBØD or $ELLBØD	to define the body
$FINSET1	to define the most forward fin set
$FINSET2	to define the first following fin set
$FINSET3	to define the second following fin set
$FINSET4	to define the third following fin set

The following namelists are optional since defaults exist for all inputs:

$REFQ	to define the reference quantities
$DEFLCT	to define the panel incidence (deflection angles)
$TRIM	to define a trim case
$INLET	to define inlet geometry
$EXPR	to define experimental input data

Defaults for all namelists should be checked to verify the configuration being modeled does not include an unexpected characteristic introduced by a default.

The following sections describe each of the namelist inputs. Each section is accompanied by a figure which summarizes the input variables, their definitions, and units. Since the system of units can be optionally selected, the column "Units" specifies the generic system of units as follows:

ℓ	Units of length; feet, inches, centimeters or meters
F	Units of force; pounds or Newtons
deg	Units of degrees; if angular, in angular degrees; if temperature, either degrees Rankine or degrees Kelvin
sec	Units of time in seconds

Exponents are added to modify the above. For example, ℓ^2 means units of length squared, or area. Combinations of the above are also used to specify other units. For example, F/ℓ^2 means force divided by area, which is a pressure.

Since it is difficult to discern the difference between the number zero and the alphabetic letter "O," this manual will always express the alphabetic letter as "Ø" In general, the number zero and the letter "Ø" are not interchangable unless so stated.

The program ascertains the configuration being modeled by the presence of each component namelist, even if no data is entered. The following rules for namelist input apply:

- Do not include a namelist unless it is required. Once read, the presence of a namelist (and, hence, a configuration component) can only be removed using the DELETE control card in a subsequent case. Simply setting all variables to their initialized values will not remove the configuration component.

- Do not include a variable within a namelist unless it is required. Program actions are often determined from the number and types of input provided.

- Do not over-specify the geometry. User inputs will take precedence over program calculations. Inputs that define a shape that is physically impossible will be used as specified. The program does not "fix-up" inconsistent or contradictory inputs.

3.1.1 NAMELIST FLTCØN - Flight Conditions

This namelist defines the flight conditions to be run for the case. The program is limited to no more than 20 angles of attack and 20 Mach numbers per case at a fixed sideslip angle, aerodynamic roll angle, altitude, and panel deflection angle. Therefore, a "case" is defined as a fixed geometry with variable Mach number and angles of attack.

The inputs are given in Figure 1. There are two ways in which the aerodynamic pitch and yaw angles can be defined:

- Input ALPHA and BETA. If BETA is input and PHI is not, it is assumed that the body axis angles of attack (α) and sideslip angles ( ) are defined.

- Input ALPHA and PHI. If PHI is input and non-zero, it is assumed that ALPHA is the total angle of attack (α) and PHI is the aerodynamic roll angle (ϕ).

- Input ALPHA, BETA and PHI. The value for BETA is ignored if PHI is non-zero.

As a minimum the following variables must be defined:

NALPHA	number of angles of attack to run (NALPHA$\geq$2)
ALPHA	angle of attack schedule (matching NALPHA)
NMACH	number of Mach numbers or speeds (NMACH$\geq$1)
MACH or VINF	Mach number or speed schedule (matching NMACH)

The REN, TINF and PINF data must correspond to the MACH or VINF inputs. The ALPHA and MACH dependent data can be input in any order; the code will sort the data into ascending order.

Reynolds number is always required. Three types of inputs are permitted to satisfy the Reynolds number requirement:

(1) Specify Reynolds number per unit length using REN

(2) Specify the altitude using ALT, and the speed using MACH or VINF (Reynolds number is computed using the Standard Atmosphere model)

(3) Specify pressure and temperature using PINF and TINF, and the speed using MACH or VINF (typical of data available from a wind tunnel test)

User supplied data will take precedence over program calculations. Hence, the user can override any default or Standard Atmosphere calculation. The default condition is sea-level altitude (ALT=0.) if the wrong combination of inputs are provided for the Reynolds number cannot be calculated.

3.1.2 NAMELIST REFQ - Reference Quantities

Inputs for this namelist are optional and are defined in Figure 2. A vehicle scale factor (SCALE) permits the user to input a geometry that is scaled to the size desired. This scale factor is used as a multiplier to the user defined geometry inputs; it is not applied to the user input reference quantities (SREF, LREF, LATREF). If no reference quantities are input, they are computed based upon the scaled geometry. XCG is input relative to the origin (X=0) and is scaled using SCALE.

In lieu of specifying the surface roughness height RØUGH, the surface Roughness Height Rating (RHR) can be specified. The RHR represents the arithmetic average roughness height variation in millionths of an inch. Typical values of RØOUGH and RHR are given in Table 4.

3.1.3 Namelist AXIBØD - Axisymmetric Body Geometry

An axisymmetric body is defined using this namelist. The body can be specified in one of two ways:

OPTION 1: The geometry is divided into nose, centerbody, and aft body sections. The shape, overall length, and base diameter for each section are specified. Note that not all three body sections need to exist on a configuration; for example, a nose-cylinder configuration does not require definition of an aft body.

OPTION 2: The longitudinal stations and corresponding body radii are defined, from nose to tail.

It is highly recommended that Option 1 be used when possible. The program automatically calculates the body contour based upon the segment shapes using geometry generators. Hence, more accurate calculations are possible. Even when Option 2 is used, appropriate Option 1 inputs should be included. This identifies where the code should insert break points in the contour.

The namelist input variables are given in Figure 3 and a sketch of the geometric inputs are given in Figure 4. The program uses the input value for NX to determine which option is being used. If NX is not input then Option 1 inputs are assumed. If both shapes and body coordinates (Options 1 and 2) are used, the body coordinate information will take precedence. NX can be set to its initialized value (to simulate the variable as not input) by specifying "NX=UNUSED".

Most of the subsonic and transonic methods are dependent upon nose and afterbody size and shape. Hence, Option 2 inputs should also specify the variables LNØSE, DNØSE, LCENTR, DCENTR, LAFT, and DAFT to further define the geometry. If these parameters are not input, they are selected as follows:

LNØSE	-	Length of the body segment to where the radius first reaches a maximum
DNØSE	-	The diameter at the first radius maximum
LCENTR	-	Length of the body segment where the radius is constant
DCENTR	-	Diameter of the constant radius segment
LAFT	-	The remaining body length
DAFT	-	Diameter at the base
DEXIT	-	Not defined (implies that base drag is not to be included in the axial force calculations)

If DEXIT is not input, or set to UNUSED, the base drag computed for the body geometry will not be included in the final computed axial force calculations. To include a "full" base drag increment, a zero exit diameter must be specified (DEXIT=0.).

If body coordinates are input using the variables NX, X, R, and DISCØN, and the nose is spherically blunted, the geometry must be additionally defined using the following:

- o BNØSE must be specified (even if zero)

- TRUNC must be set to .FALSE.

- The first five (5) points in the X and R arrays must lie on the spherical nose cap [i.e., X(1), X(2), X(3), X(4), X(5), R(1), R(2), R(3), R(4), and R(5) are spherical cap coordinates]

The following summarizes the input generality available:

- X(1) does not have to be 0.0; an arbitrary origin can be selected.

- Five shapes can be specified by name:

  CØNICAL (CONE) - cone or cone frustrum (default for boattails and flares)
  ØGIVE - tangent ogive (default for noses) POWER - power law*
  PØWER - power law*
  HAACK - L-V Haack (length-volume constrained)*
  KARMAN - von Karman (L-D Haack; length-diameter constrained)*

- If the nose is truncated (TRUNC=.TRUE.) it is assumed that the nose is open (i.e., the forward flat face produced will not be a drag producing element)

- If DAFT<DCENTR the afterbody is a boattail.**

- If DAFT>DCENTR the afterbody is a flare.**

- If LAFT is not input, aft body (boattail or flare) does not exist.

  * applies to noses only
  ** DAFT must not be equal to DCENTR

3.1.4 NAMELIST ELLBØD - Define Elliptically-Shaped Bodies

Elliptically-shaped cross section bodies are defined using this namelist. The inputs are similar to those for the axisymmetric body geometry (AXIBØD), and are shown in Figure 5a-5b. The types of shapes available, and the limitations, are the same as those given for axisymmetric bodies. Please read Section 3.1.3 for limitations.

Note that the body cross section ellipticity can vary along the body longitudinal axis. Sections which are taller-than-wide and wider-than-tall can be mixed to produce "shaped" designs. The shape of the sections is controlled by the variables ENØSE, ECENTR, and EAFT or ELLIP, H and W.

3.1.5 NAMELIST FINSETn - Define Fin Set n

Figure 6a describes the variables needed to be input for fin set planform geometry descriptions. Optional fin cross-section inputs are described in Figure 6b. Special user specified fin cross-sections can be input using the variables in Figure 6c. The user may specify up to four non-overlapping fin sets. The variable "n" in the namelist specifies the fin set number. Fin sets must be numbered sequentially from the front to the back of the mission beginning with fin set one. An input error will occur if "n" is zero or omitted. The code allows for between 1 and 8 geometrically identical panels to be input per fin set. The panels may be arbitrarily rolled about the body and can be given dihedral.

Four types of airfoil sections are permitted--hexagonal (HEX), circular arc (ARC), NACA airfoils (NACA) and user defined (USER). Only one type of airfoil section can be specified per fin set, and this type is used for all chord wise cross sections from root to tip. Diamond-shaped sections are considered a special case of the HEX type; hence, hexagonal and diamond sections can coexist on the same panel. The airfoil proportions can be varied from span station to span station.

The user selects "break points" on the panel (Figure 7). A "break point" specifies a change in leading or trailing edge sweep angle. Also a break point may specify a change in airfoil section, but the section must be of the same type (i.e., a change in section type cannot go from a NACA to an ARC) only the proportions can change. The location of each "break point" is defined by specifying its semi-span station (SSPAN) from the vehicle centerline and position of the chord leading edge (XLE) from the first body station. The "break point" chord leading edge array (XLE) can be defined by simply specifying the root chord leading edge [XLE(1)] and the sweep angles of each successive panel segment if the semi-span stations are input. Note that only those variables that uniquely define the fin need to be entered. Redundant inputs can lead to numerical inconsistencies and subsequent computational errors.

The panel sweep angle (SWEEP) can be specified at any span station for each segment of the panels. If STA=0., the sweep angle input is measured at the segment leading edge; if STA=1., the sweep angle input is measured at the segment trailing edge. Note that some aerodynamic methods are very sensitive to panel sweep angle. For small span fins, small errors in the planform inputs can create large sweep angle calculation errors. It is recommended that exact sweep angles be specified wherever possible; for example, if the panel trailing edge is unswept, specifying SWEEP=0. and STA=1. will minimize calculation error. Then the leading edge sweep will be computed by the code internally using the SSPAN and CHORD inputs.

Since all panels are assumed to be planar (i.e., no tip dihedral), all inputs must be "true view." Once the planform of a single panel is defined, all fins of the set are assumed to be identical. The number of panels present is defined using the variable NPANEL. Each panel may be rolled to an arbitrary position around the body using the variable PHIF. PHIF is measured clockwise from top vertical center (looking forward from behind the missile) as shown in Figure 8. Each panel may also contain a constant dihedral. A panel has zero dihedral when it is aligned along a radial ray from the centerline (see Figure 10). The variable used to specify dihedral is GAM. GAM is positive if the panel tip chord is rotated clockwise (see Figure 10).

Different aerodynamics will be computed depending upon whether the FLTCØN namelist variable PHI, or the FINSETn namelist variable PHIF, is used to roll the geometry. Figure 8 depicts the usage of the roll options. The variable "PHI" means that the body axes system is to be rolled with the missile body, whereas PHIF keeps the aerodynamics in a non-rolled body axis, but rather locates the fin positions around the body. PHIF must be input for each panel, while PHI rolls the whole configuration.

It is the user's responsibility to assure that the fins are (1) on the body surface, and (2) do not lie internal to the body mold line. The program does not check for these peculiarities. If SSPAN(1)=0 is input, the program will assume that the panel semi-span data relative to its root chord are supplied. The code will automatically interpolate the body geometry to place the panel on the body surface with the root chord parallel to the body centerline. See Section 3.4 for modeling fins on body segments of varying radii.

When defining more than one fin set, the fin sets must never have their planforms overlap one another. There must be sufficient space between the forward fin trailing edge and aft fin leading edge to avoid violating the assumptions made by the aerodynamic computations. It is assumed by the aerodynamic model that the vortices are fully rolled up when they pass the control points of the next downstream set of fins. In reality the vortex sheet does not fully roll up until it is at least four semispans downstream. If two fin sets are closer than this the results may be in error since the use of a vortex filament model may introduce too much vorticity. The closer the spacing the larger the error may be. No algorithm error will result from too close a fin set spacing.

Panels with cut-out portions can be modeled by using one of the ten available fin segments as a transition segment. This is accomplished by giving the segment a small span, such as 0.0001, and specifying the segment root and tip chords to transition into the cut-out portion of the fin.

3.1.6 NAMELIST DEFLCT - Panel Deflection Angles

This namelist permits the user to fix the incidence angle for each panel in each fin set. The variables are given in Figure 9. Note that the panel numbering scheme is assumed to be that shown in Figure 10. The array element of each deflection array corresponds to the panel number.

The scheme for specifying deflection angles is unique, yet concise. The scheme used is based upon the body axis rolling moment:

> "In Missile Datcom a positive panel deflection is one which will produce a negative (counterclockwise when viewed from the rear) roll moment increment at zero angle of attack and sideslip."

3.1.7 NAMELIST TRIM - Trim Aerodynamics

This namelist instructs the program to statically trim the vehicle longitudinally (C_m=0). The inputs are given in Figure 11. Note that only one fin set can be used for trimming. The user only specifies the range of deflection angles desired using DELMIN and DELMAX; the code will try to trim the vehicle for each angle of attack specified using the allowable fin deflections. This option will not trim the vehicle at a specific angle of attack if the deflection required is outside the range set by the values of DELMIN and DELMAX.

The deflection sign convention used is that described in Section 3.1.6; hence, DELMIN and DELMAX are input as if deflecting the panel to the maximum will produce a negative rolling moment from the panels resulting normal force increment. The magnitude of DELMIN must always be less than the magnitude of DELMAX.

A logical variable, ASYM, has been included to permit reverse panel deflections. For example, deflecting all panels in one sense results in a rolling moment and no pitching moment. The ASYM flag will permit analysis of an elevator (or pitch deflection) effect, by deflecting panels on one side of the vehicle only, with opposite panels mirroring those deflections. Since a maximum of eight panels are allowed in a finset, only four panels of the finset can be deflected in the reverse direction using the ASYM flag. Both trimmed and untrimmed results are available for output.

3.1.8 NAMELIST INLET - Axisymmetric and 2-Dimensional Inlet Geometry

The namelist inputs shown in Figure 12 will model both 2- dimensional (2-D) and axisymmetric inlet installations of the type depicted in Figure 13. The variables INTYPE and POSD are used to identify the type of inlet and the attachment scheme. If the diverter attaches to the top of a 2-D inlet, the inlet is a top attached 2-D inlet. If the diverter attaches to the side of the 2-D inlet, the inlet is a side attached 2-D inlet. For an axisymmetric inlet, this distinction is unnecessary because of the symmetry.

Figure 14 illustrates the procedure Missile Datcom uses to obtain the proper inlet orientation based upon the input data. To provide capability for more general inlet geometries to be incorporated in the future, the inlet installation is considered to consist of two major components: an inlet and a diverter which attaches the inlet to the missile body. To setup the namelist inputs, the inlet and diverter are assumed to be mounted in the horizontal position on the missile body as shown at the top of Figure 14. Missile Datcom will combine the diverter and inlet inputs into a fully integrated inlet installation at this position, and then rotate it to the specified angular orientation, PHI.

The variable NIN defines the number of inlets being modeled. The program can handle 1, 2, 3, and 4 inlets in combination. Figure 15 illustrates the type PHI values for single, twin, triple and quadruple inlet configurations. For four inlet combinations, a positive angle should be input for PHI(1) and a negative angle for PHI(2) to obtain cruciform inlets. For inlet geometries with inlets in the vertical plane, an angle of 90^{o} should be used; the program will then position one inlet on the bottom vertical and other on the top vertical.

The namelist inputs of inlet coordinates and dimensions require the definition of an inlet coordinate system (X_I, Y_I, Z_I) and its location relative to the missile coordinate system (X_B, Y_B, Z_B). Figure 16 shows the coordinate systems for the three inlet installations assuming they are initially in the horizontal position. For all inlets, the inlet coordinate system axial location is located relative to X=0 with input X_{BO}. (Although the origin is shown at the inlet leading edge, it can be at any arbitrary X-station). For the side attached 2-D inlet the origin location relative to the missile horizontal centerline is located by coordinate Z_{BO}. The coordinates Y_I and Z_I relative to the missile are computed from the input geometry.

<u>2-D Inlet</u> - The nomenclature for the 2-D inlet is provided in Figure 17. The inlet is considered to be composed of four major sections: forebody, midbody, boattail and base. The user identifies these sections through inputs of selected coordinates and dimensions at locations 1 through 10. The forebody is the most complicated section and includes the top and bottom lip leading edge which form the top and bottom of the inlet opening. The external surface on the top and outside edge can be chamfered. (The side external transfer can end at or before the inlet maximum depth location). The maximum depth of the inlet occurs on the forebody. Internal to the forebody at the top of the inlet opening is the internal compression ramp.

The midbody is a rectangular tube connecting the forebody and boattail. The boattail begins with a rectangular cross section and transitions into a semicircular base section.

Figure 18 shows a side attached 2-D inlet, the 19 inputs required to define its geometry, and the set of assumptions used to construct the inlet geometry. For example, the forebody and midbody sides are assumed parallel. Therefore, the width of the midbody top at the start of the boattail is B_2. Because the midbody top and bottom are assumed parallel, the coordinates at location 3 and 8 are $Z_3=Z_2$, $X_8=X_3$ and $Z_8=Z_7$. The leading edge of the internal compression ramp begins at location 1, the top lip leading edge, and is assumed parallel to the X_I axis to location 9, i.e., $Z_9=Z_1$. The compression ramp is assumed to terminate at the axial location of the bottom lip and, therefore $X_{10}=X_5$.

In the program, the diverter is integrated with the inlet and missile body. Figure 19 defines the side attached diverter nomenclature, inputs and assumptions. The diverter leading edge width, $TIND_1$, sets the spacing between the body and inlet side and $XIND_1$ sets the diverter leading edge axial location in the inlet coordinate system. The vertical location is assumed to be at the half height of the inlet midbody. The only other input required is the axial position, $XIND_2$, at which the diverter height becomes equal to the inlet midbody height. The diverter height is assumed constant from $XIND_2$ to the start of the inlet boattail at X_3. Because of this assumption, $XIND_2$ must not exceed X_3. The diverter then fairs into the boattail and has a height at the inlet base equal to the base diameter. Other diverter parameter such as width are computed based upon these assumptions.

Top Attached Inlet - Figure 20 shows the 18 inputs required to define the geometry of the top attached 2-D inlet. This is an inlet with the diverter attached to its top surface. The base semicircle is oriented with the flat side on top to provide a flat top surface to integrate with the diverter.

The diverter construction is somewhat different than for the side-mounted inlet. Figure 21 defines the nomenclature, inputs and assumptions. The three diverter inputs serve the same function as for the side mounted 2-D inlet. The other dimensions are computing assuming their inlet midbody is parallel with the missile centerline. The diverter maximum height is set equal to the midbody height, B_2 and its base height is set to $2*B_4$. The input value of $XIND_1$ must not be less than the inlet coordinate X_2 and $XIND_2$ must not be greater than X_3.

Axisymmetric Inlet - Figure 22 describes the nomenclature for the axisymmetric inlet. Five major inlet sections are defined: centerbody, lip, midbody, boattail, and base. The centerbody and lip are modeled by inputs at 3 axial stations to provide an approximate description of their actual contours.

Figure 23 shows the 15 inputs required to define the geometry and the assumptions made internal to the program to construct the remaining geometry. B_2 through B_6 and B_8 represent radii at the location indicated. B_9 is the offset of the base from the midbody X_1 through X_8 represent the identified axial stations. Both the centerbody and cowl lip are modeled as conical frustums. The midbody of the inlet is assumed cylindrical. Therefore, the radius as station 7 is equal to B_6.

The diverter is integrated with the inlet and missile body as shown in Figure 24. The diverter inputs include the leading edge width, $XIND_1$ which sets the spacing between the body and inlet side, and $XIND_1$, which sets the diverter leading edge axial location in the inlet coordinate system ($XIND_1$ must be between X_6 and X_7 so that the diverter starts on the midbody). The only other input required is the axial position, $XIND_2$, at which the diverter height becomes equal to the inlet midbody diameter which is $2B_6$. the geometry for the remainder of the diverter is constructed assuming a constant diverter height from station $XIND_2$ until the start of the inlet boattail. At the inlet base the diverter height is assumed to equal $2B_8$. Other diverter parameters and widths are computed based upon these assumptions.

3.1.9 NAMELIST ARBBØD - Arbitrary Body Geometry

Namelist ARBBØD allows the user more flexibility in creating nonconventional missile body shapes. Body shapes input should not be too flat and should have fineness ratios greater than five and cross-sectional aspect ratios less than five.

Geometric reference values (e.g., S_{REF}, L_{REF}, X_{CG}, etc) and aerodynamic conditions (α, M_∞, Re, etc) are input using the REFQ and FLTCON NAMELISTs. These NAMELISTs have already been described. The user is directed to their description for details of their input variables.

The geometry of a configuration is developed using the variables described in Figure 25. Although it is not possible to describe every arbitrary shape with the present method, it does expand the class of shapes that can be analyzed by NAMELISTs AXIBØD and ELLBØD.

The planform and forebody base cross-section are the most important shape details needed for this method. The planform must be trapezoidal or triangular, to simplify the input. Consequently, the nose must converge either to a point or to a flat spatular tip. No nose bluntness is allowed for either the pointed or spatular nose geometries.

The base cross-section of the nose is defined by connecting one or more linear segments and/or elliptic arc segments. The start and end points of the segment define the extent of that segment. Subsequent segments require only the end point, since the segment begins at the point where the previous segment ends.

Mirror symmetry about the vertical plane reduces the chance of erroneous input data. Therefore, only that portion of the base with zero or positive y values may be input. That portion which falls to the left of the vertical plane will be generated by the program. At the present time, only eight segments may be used to describe the half cross-section. The first segment must start at the x-z plane, proceed counterclockwise, and end at the x-z plane.

The centerbody must have the same cross-section as the base of the forebody. The only input required to describe the centerbody therefore is the length. No afterbody is permitted at the present time.

To illustrate the inputs required to describe the arbitrary cross section bodies, four examples are presented next. The first example is very simple. The second illustrates a pointed, arbitrary cross section nose. The third details the inputs for a spatular nose and the fourth describes another form of a pointed, arbitrary cross section nose.

The forebody in Figure 26 has a pointed tip, and a circular base cross section. The forebody length is 1.111 ft and the centerbody length is 2.222 ft. The tip width is zero and is placed at the origin. The nose is a right circular cone, so the center of the base lies on the x-axis; the radius is 0.1667 ft.

The segment may be described in two ways. First, the angle of the radius at the starting point is -90 degrees, relative to the positive x-axis; the end point is +90 degrees from the x-axis. Likewise, the y-z coordinate pair for the starting point is (0.0000, - .1667), and (0.0000, + .1667) for the end point.

Only one segment is used to describe the base, and the base type for that segment is 3. The tip is a point, so the tip type must be zero. The calculation of $C_{m\alpha}$ requires the integration of $C_{N\alpha}$ along the longitudinal axis segment by segment. The code divides the planform into equal length segments for which $C_{N\alpha}$ is calculated.

The forebody in Figure 27 also has a pointed tip, but the cross section is a rounded equilateral triangular. Except for the base radius, which is not applicable here, the reference lengths are the same as in Figure 1.

Four segments, two linear and two elliptic arcs, are used to describe the basic half cross-section. The first segment is linear, beginning at (0.0000, - .1700) and extending to (0.1200, - .1700). The second segment is a circular arc; the origin is located at (0.1200, -.1033). The arc extends from -90 degrees to +30 degrees. The third segment is a line, extending from the end of the second segment to (0.0578, 0.1367). The fourth segment is a circular arc with a radius of 0.0667. The end point may be specified either as +90 degrees, or as (0.0000, 0.1700). The base types are 2, 3, 2 and 3 respectively; the tip is a point so all four tip types must be zero.

The third example, Figure 28, is a wedge with a trapezoid planform and base cross-section. The forebody length is 0.6460, and because there is no centerbody, that length is zero. The nose tip is spatular; the total tip width is 0.1667 ft. The center of the tip has been placed at the origin, so the distance to the tip is zero.

The base half cross-section may be described using three line segments. The first starts at (0.0000, 0.0000) and extends to (0.2088, 0.0000). The second segments continues from the end of the first segment to (0.0833, 0.0565). The third segment ends on the z-axis at (0.0000, 0.0565).

Since all three segments are linear, all three base types are 2. The first segment lies in the x-y plane, and narrows to a finite width at the tip, so the tip type is 1. The second segment slopes up, and to the left, and narrows to a point, so the 2nd tip type is 0. the final segment narrows to a finite width at the tip, so the third tip type is 1.

Figure 29 shows a forebody with a pointed tip and a half ellipsoid base. The forebody is 1.111 ft and the centerbody is 2.222 ft long.

Two segments describe the base half cross-section; an elliptsoid and a linear segment. Both extend to a point at the nose the tip types are both zero. The linear segment starts at (0.000, 0.000) and goes to (.2886, 0.000). The ellipsoid segment automatically begins at the end of the last segment, therefore no DSTART value must be specified. The ellipsoid segment is centered around (0.000, 0.000) and has major and minor axes of lengths .2886 and .1924. The ellipsoid is traced from 0 through 90 degrees, therefore DFINAL=90.0.

3.1.10 <u>NAMELIST EXPR</u> - Experimental Data Substitution

This namelist is used to substitute experimental data for the theoretical data generated by the program. The variables to be input ae shown in Figure 30. Use of namelist EXP does not stop the program from calculating theoretical data, but rather the experimental data is used in configuration synthesis, and it is the experimental data that is used for the component aerodynamics for which it is input.

Experimental data may be substituted for any configuration component or partial configuration. Experimental data is input at a specific Mach number. when using namelist EXPR the case must be run at the Mach number for which you are substituting experimental data. However, the experimental data being input may have different reference quantities and a different center of gravity location than the case being run.

Experimental data input for a fin alone is input as panel data - not as total fin set data. The user should note that experimental data for fin alone $C_{m\alpha}$ is not used in the configuration synthesis process. Instead fin alone $C_{N\alpha}$ (the experimental value if input) i used to determine the fin contribution to $C_{m\alpha}$ during configuration synthesis. If body alone experimental data and body-fin experimental data are input for the same case the body data is ignored in configuration synthesis. If experimental $C_{m\alpha}$ data is input for a body + 1 fin set for a multi-fin set configuration, the calculated contributions to $C_{m\alpha}$ of the other fin sets are added to the experimental data.

Since the experimental namelist forms the basis for configuration incrementing, the lateral directional coefficients are included to allow for sideslip cases. These coefficients are input the same as the longitudinal coefficients. However, if the lateral directional coefficients are input, the lateral directional beta derivatives will not be computed our output.

The following rules apply to the use of namelist EXPR.

1. It is assumed that the coefficients in EXPR are for the same sideslip and/or aerodynamic roll as the case being run.
2. Separate namelist EXPR must be specified for each Mach number.
3. Each namelist EXPR must end with a $END card.
4. Separate namelist EXPR must be specified for each partial configuration for which experimental data is to be input, (i.e., body, body + 1 fin set, etc) since
5. Separate namelist EXPR must be specified for each reference quantity change.

   Example:

   The user has experimental data available for a body + 2 fin set configurations and is interested in the effects of adding a booster containing a third fin set. he would then use namelist EXPR to input the experimental data. When the configuration is synthesized, it would use the experimental data for body + 2 fin sets and theoretical data for fin set three.

3.2 Control Card Inputs

Control cards are one line commands which select program options. Although they are not required inputs, they permit user control over program execution and the types of output desired. Control cards enable the following:

- o Printing internal data array results for diagnostic purposes (DUMP)
- o Outputting intermediate calculations (PART, BUILD, PRESSURES, PRINT AERØ, PRINT EXTRAP, PRINT GEØM, PLØT, NAMELIST, WRITE, FØRMAT)
- o Selecting the system of units to be used (DIM, DERIV)
- o Defining multiple cases, permitting the reuse of previously input namelist data or deleting namelists of a prior case (SAVE, DELETE, NEXT CASE)
- o Adding case titles or comments to the input file and output pages (*, CASEID)
- o Limits the calculations to longitudinal aerodynamics (NO LAT)

3.2.1 Control Card - General Remarks

A total of 42 different control cards are available. There is no limit to the number of control cards that can be present in a case. If two or more control cards contradict each other, the last control card input will take precedence. All control cards must be input as shown, including any blanks. Control cards can start in any column but they cannot be continued to a second card. Misspelled cards are ignored. Control cards can be located anywhere within a case.

Once input, the following control cards remain in effect for all subsequent cases:

DIM FT	DIM IN	DIM CM	DIM M	FORMAT
HYPER	INCREMENT	NOGO	NO LAT	PLOT
SOSE	WRITE			

The following control cards are effective only for the case in which they appear:

BUILD	CASEID	DAMP	DELETE	DUMP CASE
DUMP NAME	NAMELIST	PART	PRESSURES	PRINT AERO
PRINT EXTRAP	PRINT GEOM	SAVE	SPIN	TRIM

The control cards can be changed from case to case:

DERIV DEG	DERIV RAD	NACA

The only control card that can be optionally saved, from case-to-case, is the NACA card.

Available control cards are summarized as follows:

BUILD

This control card instructs the program to print the results of a configuration build-up. All configurations which can be built from the components defined will be synthesized and output, including isolated data (e.g., body alone, fin alone, etc.). Component build-up data is not provided if the TRIM option is selected.

CASEID

A user supplied title to be printed on each output page is specified. Up to 72 characters can be specified (card columns 8 to 80).

DAMP

When DAMP control card is input longitudinal dynamic derivatives are computed and the results output for the configuration. Dynamic derivatives are configuration components or partial configurations may be output using the PART or BUILD control cards respectively.

DELETE name1,name2

This control card instructs the program to ignore a previous case namelist input that was retained using the SAVE control card. All previously saved namelists with the names specified will be purged from the input file. Any new inputs of the same namelist will be retained. At least one name (name1) must be specified.

DIM IN, DIM FT, DIM CM or DIM M

This control card sets the system of units for the user inputs and program outputs. The four options are inches (DIM IN), feet (DIM FT), centimeters (DIM CM), and meters (DIM M). The default system of units is feet. Once the system of units has been set, it remains set for all subsequent cases of the "run".

DERIV DEG or DERIV RAD

All output derivatives are set to either degree (DERIV DEG) or radian (DERIV RAD) measure*. The derivative units can be changed more than once during the run by inputting multiple DERIV cards.

DUMP CASE

Internal data blocks, used in the computation of the case, are written on Tape 6. This control card automatically selects partial output (PART).

* Default is degree.

DUMP name1,name2

This permits the user to write selected internal data blocks on Tape 6. At least one name (name1) must be specified. Dump names are identified in Volume 2 - Program Implementation Guide. The arrays will be dumped in units of feet, pounds, degrees or degrees Rankine.

FØRMAT (format)

This control card is used in conjunction with the WRITE control card. It specifies the format of the data to be printed to tape unit 3. The format is input starting with a left parenthesis, the format and a right parenthesis. This is exactly the same as a FORTRAN FORMAT statement. Because of the code structure alphanumeric data must not be printed. For example

```
FØRMAT ((8(2X,F10.4))     is legal
FØRMAT (2HX=,F10.4)       is illegal
```

The default format is 8F10.4, and will be used if the FØRMAT control card is not present. Multiple formats can be used. The last FØRMAT read will be used for all successive WRITE statements until another FØRMAT is encountered. Hence, the FØRMAT must precede the applicable WRITE.

HYPER

This control card causes the program to select the Newtonian flow method for bodies at any Mach number above 1.4. HYPER should normally be selected at Mach numbers greater than 6.

INCRMT

This card is used to set the configuration incrementing flag. Configuration incrementing uses the first case of a run to determine correction factors for the longitudinal and lateral aerodynamic coefficients. These correction factors are computed by comparing theoretical and experimental values for each coefficient for which data is input. The experimental values are input using elist EXPR. During subsequent cases of the run, the correction factors are applied to coefficients for which experimental data was input in the first case. This provides the user with a method to evaluate changes in a configuration.

The INCRMT card must be input in the first case of a run. The first case must be run at the same Mach number as the experimental data which is input. Once the increment flag is set it can not be deleted during that run.

The following restrictions apply:

1. All cases of a run must have the same number of fin sets.
2. All cases of a run must have the same sideslip or aerodynamic roll angle as the first case (BETA or PHI as specified in namelist FLTCON).
3. The first case must be run at exactly the same angles of attack as the experimental data being input.
4. All cases must be run within the same Mach regime (subsonic, transonic, or supersonic) as the experimental data.
5. Experimental data can only be input in the first case and only for the complete configuration. No additional data can be substituted.
6. To increment C_{Y_β} and $C_{N\alpha}$ experimental data must be input for C_Y and C_N.

Use of configuration incrementing may or may not increase the accuracy of the results. The following guidelines will produce better results when using configuration incrementing:

1. The user may run different angles of attack in each case. However, no angle of attack should exceed the upper or lower limit of the angles of attack for which experimental data was input in the first case.

2. Experimental data should be input at as many angles of attack as possible.

3. The user should remember that the effect of a change in Mach number from case to case is not corrected by inputting experimental data at one Mach number as is required.

NACA

This card defines the NACA airfoil section designation (or supersonic airfoil definition). Note that if airfoil coordinates and the NACA card are specified for the same aerodynamic surface, the airfoil coordinate specification will be used. Therefore, if coordinates have been specified in a previous case and the SAVE option is in effect, the saved namelist must be deleted or the namelist variable SECTYP must be changed for the NACA card to be recognized for that aerodynamic surface. The airfoil designated with this card will be used for all segments and panels of the fin set.

The form of this control card and the required parameters are as follows:

Card Column(s)	Input(s)	Purpose
1 thru 4	NACA	The unique letters NACA designate that an airfoil is to be defined
5	Any delimeter	
6	1 or 2	Fin set number for which the airfoil designation applies
7	Any delimeter	
8	1,4,5,6,S	Type of NACA airfoil section; 1-series (1), 4-digit (4), 5-digit (5), 6-series (6), or supersonic (S)
9	Any delimeter	
10 thru 80	Designation	Input designation (see Table 5); columns are free-field (blanks are ignored)

Only fifteen (15) characters are accepted in the airfoil designation. The vocabulary consists of the following characters:

0 1 2 3 4 5 6 7 8 9 A , = . -

Any characters input that are not in the vocabulary list will be interpreted as the number zero (0). Table 5 details the restrictions on the NACA designation.

NAMELIST

This control card instructs the program to print all namelist data. This is useful when multiple inputs of the same variable or namelist are used.

NEXT CASE

This card indicates termination of the case input data and instructs the program to begin case execution. It is required for multiple case "runs." This card must be the last card input for the case.

NØ GØ

This control card permits the program to cycle through all of the input cases without computing configuration aerodynamics. It can be present anywhere in the input stream and only needs to appear once. This option is useful for performing error checking to insure all cases have been correctly set up.

NØ LAT

This control card inhibits the calculation of the lateral-directional derivatives due to sideslip angle. Savings in computation time can be realized by using this option. This option is automatically selected when using TRIM.

PART

This control card permits printing of partial aerodynamic output, such as a summary of the normal force and axial force contributors. Partial output of the configuration synthesis methods is only provided if the TRIM option is not selected. Use of this card is equivalent to inputing all PRINT AERO and PRINT GEOM control cards (except *, -, +).

PLØT

A data file for use with a post-processing plotting program is provided. The BUILD control card must also be present in order to generate a data file with component buildup data. If the TRIM option is selected, only the trimmed and untrimmed total configuration results are provided.

PRESSURES

This control card instructs the program to print the body and fin alone pressure coefficient distributions at supersonic speeds. Only pressure data to 15 degrees angle of attack for bodies and at zero angle of attack for fins are printed.

PRINT AERØ name

This control card instructs the program to print the incremental aerodynamics for "name," which can be one of the following:

BØDY	for body aerodynamics
FIN1	for FINSET1 aerodynamics
FIN2	for FINSET2 aerodynamics
FIN3	for FINSET3 aerodynamics
FIN4	for FINSET4 aerodynamics
SYNTHS	for configuration synthesis aerodynamics
TRIM	for trim/untrimmed aerodynamics
BEND	for panel bending moments
HINGE	for panel hinge moments
INLET	for inlet aerodynamics

All options are automatically selected when the control card PART is used. Details of the output obtained with these options are presented in Section 4.2.

PRINT EXTRAP

This control card enables the printing of method extrapolation messages produced during execution of the case.Extrapolation messages are not normally provided.

PRINT GEØM name

This control card instructs the program to print the geometric characteristics of the configuration component "name," which can be one of the following:

BØDY	for body geometry
FIN1	for FINSET1 geometry
FIN2	for FINSET2 geometry
FIN3	for FINSET3 geometry
FIN4	for FINSET4 geometry
INLET	for inlet geometry

All options are automatically selected when the control card PART is used.

SAVE

The SAVE card saves namelist inputs from one case to the following case but not for the entire run. This permits the user to build-up or change a complex configuration, case-to-case, by adding new namelist cards without having to re-input namelist cards of the previous case. When changing a namelist that has been saved, the namelist must first be deleted using the delete control card.

The only control card that can be optionally saved, case-to-case, is the NACA card.

SØSE

The presence of this control card selects the Second-Order Shock Expansion Method for axisymmetric bodies at supersonic speeds. SØSE should be selected if any Mach number is higher than 2.0.

SPIN

When the SPIN control card is input, spin and magnus derivatives are computed for body alone. If the configuration being run is a body + fin sets, the spin derivatives are still computed for body alone. A PART or BUILD card must be input for body alone derivatives to be printed out.

TRIM

This control card causes the program to perform a trim calculation. Component buildup data cannot be dumped if TRIM is selected. The use of this control card is the same as if the namelist TRIM was included except that the defaults for namelist TRIM are used.

WRITE name, start, end

This control card causes the common block "name" to be printed to tape unit 3 using the most recent FØRMAT control card. Locations from "start" to "end" are dumped (see Volume II for a list of block names). Multiple WRITE statements may be input, and there is no limit to the number which may be present. The presence of a WRITE will cause the block "name" to be printed for all cases of the run. The output will be in units of feet, pounds, degrees, or degrees Rankine. If the PLØT option is also selected, this output will be "mixed" with the PLØT file output on tape unit 3.

*

Any card with an asterisk (*) in Column 1 will be interpreted as a comment card. This permits detailed documentation of case inputs.

3.3 Typical Case Set-Up

Figure 31 schematically shows how Missile Datcom inputs are structured. This example illustrates a multiple case job in which case 2 uses part of the case 1 inputs. This is accomplished through use of the SAVE control card. Case 1 is a body-wing-tail configuration; partial output, component buildup data, and a plot file are created. Case 2 uses the body and tail data of case 1 (the wing is deleted using DELETE), specifies panel deflection angles and sets the data required to trim.

There is no limit to the number of cases that can be "stacked" in a single run, provided that no more than 300 namelist inputs are "saved" between cases. If a SAVE control card is not present in a case, all previous case inputs are deleted.

3.3.1 Configuration Incrementing Case Setup

A "configuration incrementing" case setup is shown in Figure 32. This figure shows the inputs for a three case setup fin parametric analysis. The first case is the calibration case with the remaining cases being used for the parametric analysis. Therefore, the first case must contain both the INCRMT control card and EXPR namelist. These should only appear in the first case.

3.4 Special Usage of Input Parameters

It is possible to manipulate the input geometry, such that unique configurations can be modeled. This section defines those capabilities.

3.4.1 Locating Panels on Varying Body Radii Segments

The fin panels can be located anywhere on the geometry. If they are to be positioned on a varying radii segment, select the root chord span station [SSPAN(1)] such that the center of the exposed root chord is on the surface mold line. Physically this places part of the panel within the body and part offset from the body.

If SSPAN(1) is input precisely as zero, the code will assume that panel semi-span stations relative to the root chord are defined. It will then interpolate the body geometry at the root chord center and add the body radius at this point to the user defined values in the SSPAN array.

Table 1 Body Addressable Configurations

M11-14065

	Subsonic $M \leq 0.6$	Transonic $0.6 < M \leq 1.2$	Supersonic $M > 1.2$
1. Nose Shape			
Conical			
Sharp	x	x	x
Blunted	x	x	x
Truncated			x
Tangent Ogive			
Sharp	x	x	x
Blunted	x	x	x
Truncated			x
Other			x
2. Centerbody Shape			
Cylinder	x	x	x
Elliptically Variable	x	x	x
Arbitrary Cross Section	x	x	x
3. Afterbody Shape			
Boattails			
Conical	x	x	x
Tangent Ogive	x	x	x
Other			x
Flares			
Conical	x	x	x
Ogive			x
Other			x

Table 2 Subsonic/Transonic Method Limitations

M11-14066

	Range Permitted	Action Taken If Exceeded
Nose Shape	CONE or OGIVE	Uses OGIVE
Conical Nose Bluntness (C_A)	Sharp Only	Uses Sharp Method
Nose Bluntness (C_N, C_m)	Sharp Only	Uses Sharp Method
Conical Nose Slope	0 to 25 Degrees	Uses 25 Degrees
Boattail Shape	CONE or OGIVE	Uses CONE
Conical Boattail Slope	0 to 16 Degrees	Extrapolates
Ogive Boatttail Slope	0 to 28 Degrees	Extrapolates
Flare Shape	CONE	Uses CONE
Flare Slope	0 to 10 Degrees	Extrapolates
Airfoil t/c	0 to 12%	Continues, if possible

Table 3 Namelist Alphanumeric Constants

M11-14067

NAME LIST	PERMITTED ALPHANUMERIC CONSTANTS	CONVERTED VALUE
(ALL)	UNUSED	1.E-30 (Initialized Value)
REFQ	TURB	0.
	NATURAL	1.
AXIBØD or	CØNICAL	0.
ELLBØD	CØNE	0.
	ØGIVE	1.
	PØWER	2.
	HAACK	3.
	KARMAN	4.
FINSETn	HEX	0.
	NACA	1.
	ARC	2.
	USER	3.
INLET	AXI	0.
	2D	1.

Table 4 Equivalent Sand Roughness

M11-14068

TYPE OF SURFACE	EQUIVALENT SAND ROUGHNESS k (inches)	RHR
Aerodynamically smooth	0	0
Polished metal or wood	$0.02\text{-}0.08 \times 10^{-3}$	6-26
Natural sheet metal	0.16×10^{-3}	53
Smooth matte paint, carefully applied	0.25×10^{-3}	83
Standard camouflage paint, average application	0.40×10^{-3}	133
Camouflage paint, mass-production spray	1.20×10^{-3}	400
Dip-galvanized metal surface	6×10^{-3}	2000
Natural surface of cast iron	10×10^{-3}	3333

Preferred RHR Values

Application	RHR
Steel structural parts	250
Aluminum an titanium structural parts	125
Close tolerance surfaces	63
Seals	32

NAMELIST FLTCON

M11-14069

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
NALPHA	–	NUMBER OF ANGLES OF ATTACK (AT LEAST 2)	–	
ALPHA	20	ANGLE OF ATTACK OR TOTAL ANGLE OF ATTACK	DEG	
BETA	–	SIDESLIP ANGLE	DEG	0.
PHI	–	AERODYNAMIC ROLL ANGLE	DEG	0.
NMACH	–	NUMBER OF MACH NUMBERS (AT LEAST 1)	–	
MACH ①	20	MACH NUMBERS	–	
REN ①	20	REYNOLDS NUMBER PER FOOT (METER)	$1/\lambda$ ②	
ALT ①	–	GEOMETRIC ALTITUDE	λ ③	0.
VINF ①	20	FREESTREAM SPEED	λ /SEC ④	
TINF ①	20	FREESTREAM STATIC TEMPERATURE	DEG	
PINF ①	20	FREESTREAM STATIC PRESSURE	F/λ^2 ⑤	

① FOLLOWING COMBINATIONS SATISFY THE REYNOLDS NUMBER AND MACH NUMBER INPUT REQUIREMENTS.

USER INPUT	PROGRAM COMPUTES
1. MACH, REN	(NONE)
2. MACH, ALT	PINF, TINF, REN
3. VINF, ALT	PINF, TINF, MACH, REN
4. VINF, TINF, PINF	MACH, REN
5. MACH, TINF, PINF	VINF, REN

② INPUT AS 1/FT FOR ENGLISH UNITS AND 1/M FOR METRIC UNITS
③ INPUT AS FT FOR ENGLISH UNITS AND M FOR METRIC UNITS
④ INPUT AS FT/SEC FOR ENGLISH UNITS AND M/SEC FOR METRIC UNITS
⑤ INPUT AS LB/FT^2 FOR ENGLISH UNITS AND N/M^2 FOR METRIC UNITS

Figure 1 Flight Condition Inputs

NAMELIST REFQ

M11-14070

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
SREF	–	REFERENCE AREA	λ^2	②
LREF	–	REFERENCE LENGTH (LONGITUDINAL)	λ	③
LATREF	–	REFERENCE LENGTH (LATERAL-DIRECTIONAL)	λ	= LREF
RØUGH ①	–	SURFACE ROUGHNESS HEIGHT	λ ④	0.
RHR ①	–	ROUGHNESS HEIGHT RATING	–	0.
XCG	–	LONGITUDINAL POSITION OF C.G. (+ AFT)	λ	0.
ZCG	–	VERTICAL POSITION OF C.G. (+ UP)	λ	0.
SCALE	–	VEHICLE SCALE FACTOR (MULTIPLIER TO GEOMETRY)	–	1.
BLAYER	–	BOUNDARY LAYER TYPE: TURB FOR FULLY TURBULENT, NATURAL FOR NATURAL TRANSITION	–	TURB

① EITHER CAN BE USED.

② DEFAULT IS BODY MAXIMUM CROSS-SECTIONAL AREA. IF NO BODY IS INPUT, MAXIMUM FIN PANEL AREA IS USED.

③ DEFALUT IS BODY MAXIMUM DIAMETER. IF NO BODY IS INPUT, MAXIMUM FIN PANEL MEAN GEOMETRIC CHORD IS USED.

④ INPUT AS INCHES FOR ENGLISH UNITS AND CENTIMETERS FOR METRIC UNITS.

Figure 2 Reference Quantity Inputs

NAMELIST AXIBOD

OPTION 1 INPUTS

M11-14071

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
X0 OR XØ	–	LONGITUDINAL COORDINATE AT NOSE TIP	–	0.
TNØSE	–	NOSE SHAPE NAME: CONICAL, CONE, OGIVE, POWER, HAACK, KARMAN		OGIVE
PØWER	–	EXPONENT, n, FOR POWER SERIES SHAPES, $(r_i/R) = (X_i/\lambda)^n$	–	0.
LNØSE	–	NOSE LENGTH	λ	
DNØSE	–	NOSE SECTION BASE DIAMETER	λ	1.0
BNØSE	–	NOSE BLUNTNESS RADIUS OR TRUNCATED NOSE RADIUS	λ	0.
TRUNC	–	.TRUE. IF NOSE IS TRUNCATED	–	.FALSE
LCENTR	–	CENTERBODY LENGTH	λ	0.
DCENTR	–	CENTERBODY BASE DIAMETER	λ	= DNOSE
TAFT (1)	–	AFTRBODY SHAPE NAME: CONICAL, OGIVE	–	CONICAL
LAFT (1)	–	AFTERBODY LENGTH	λ	0.
DAFT (1)	–	AFTERBODY BASE DIAMETER	λ	
DEXIT (1)		DIAMETER OF NOZZLE EXIT	λ	

(1) AFT BODY MUST NOT BE CYLINDRICAL (i.e. DAFT ≠ DCENTR)

NAMELIST AXIBOD

OPTION 2 INPUTS

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
X0 OR XØ	–	LONGITUDINAL COORDINATE AT NOSE TIP	λ	0.
BNØSE	– –	NOSE BLUNTNESS RADIUS OR TRUNCATED NOSE RADIUS	λ	0.
TRUNC	–	.TRUE. IF NOSE IS TRUNCATED	–	.FALSE
DEXIT		DIAMETER OF NOZZLE EXIT	λ	
NX	–	NUMBER OF INPUT STATIONS	–	
X (1)	50	LONGITUDINAL COORDINATES	λ	
R	50	RADIUS AT EACH X STATION	λ	
DISCØN	20	INDICES OF X STATIONS WHERE THE SURFACE SLOPE IS DISCONTINUOUS	–	
LNOSE (2)		NOSE LENGTH	λ	
LCENTR (2)		CENTERBODY LENGTH	λ	

(1) X (NX) MUST BE END OF BODY

(2) THESE INPUTS CAN BE USED TO IMPROVE MODELLING ACCURACY

Figure 3 Axisymmetric Body Geometry Inputs

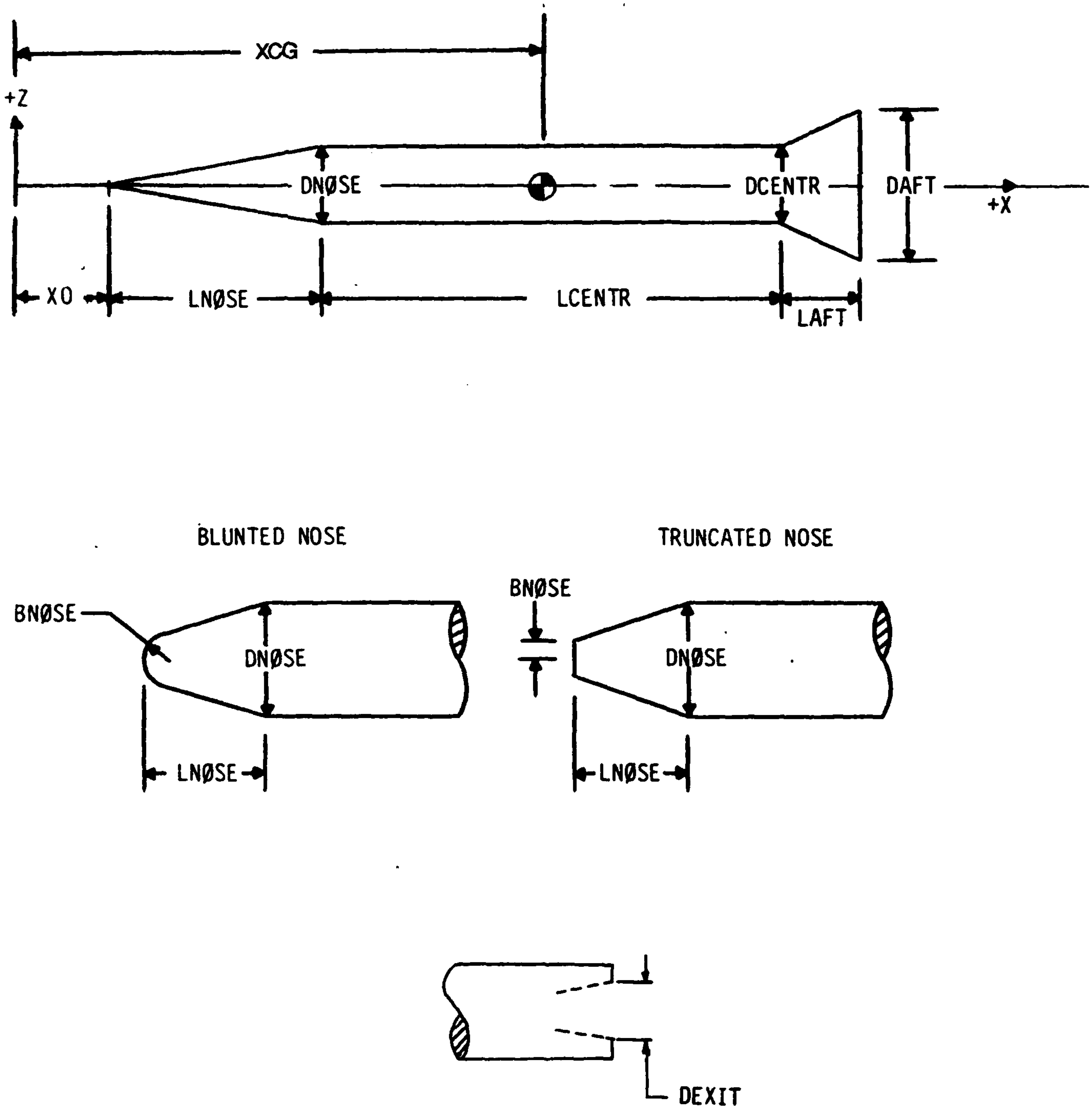


Figure 4 Body Geometry Inputs

NAMELIST ELLBOD
OPTION 1 INPUTS

M11-14072

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
X0 OR XØ	–	LONGITUDINAL COORDINATE NOSE TIP	λ	0.
TNØSE	–	NOSE SHAPE NAME: CONICAL CONE OGIVE POWER HAACK KARMAN	–	ØGIVE
PØWER	–	EXPONENT, n, FOR POWER SERIES SHAPES, $(r_i/R) = (X_i/\lambda)^n$	–	0.
LNØSE	–	NOSE LENGTH	λ	
WNØSE	–	NOSE SECTION BASE WIDTH	λ	1.
BNØSE	–	NOSE BLUNTNESS RADIUS OR TRUNCATED NOSE RADIUS	λ	0.
TRUNC	–	.TRUE. IF NOSE IS TRUNCATED	–	.FALSE
ENØSE	–	ELLIPTICITY AT NOSE BASE, H/W	–	1.
LCENTR	–	CENTERBODY LENGTH	λ	0.
WCENTR		CENTERBODY BASE WIDTH	λ	= WNØSE
ECENTR	–	ELLIPTICITY AT CENTERBODY BASE, H/W	–	1.
TAFT (1)	–	AFTERBODY SHAPE NAME:	–	CØNICAL
	–	CONICAL CONE OGIVE		
LAFT (1)		AFTERBODY LENGTH	λ	0.
WAFT (1)		AFTERBODY BASE WIDTH	λ	
EAFT (1)		ELLIPTICITY AT AFT BODY BASE, H/W	–	1.
DEXIT		DIAMETER OF NOZZLE EXIT	λ	

(1) AFT BODY MUST NOT BE CYLINDRICAL (i.e. WAFT ≠ DCENTR)

Figure 5a Elliptical Body Geometry Inputs-Option 1

NAMELIST ELLBOD

OPTION 2 INPUTS

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
XO OR XØ	–	LONGITUDINAL COORDINATE AT NOSE TIP	λ	0.
BNØSE	–	NOSE BLUNTNESS RADIUS OR TRUNCATED NOSE RADIUS	λ	0.
TRUNC	–	. TRUE, IF NOSE IS TRUNCATED	–	.FALSE.
NX	–	NUMBER OF INPUT STATIONS	–	
X	50	LONGITUDINAL COORDINATES	λ	
W (1)	50	BODY HALF-WIDTH AT EACH X STATION	λ	
DISCØN	20	INDICES OF X STATIONS WHERE THE SURFACE SLOPE IS DISCONTINUOUS	–	
H (1)	50	BODY HALF-HEIGHT AT EACH X STATION	λ	
ELLIP (1)	50	BODY HEIGHT TO WIDTH RATIO AT EACH X STATION	–	1.

(1) ONE OF THE FOLLOWING COMBINATIONS IS REQUIRED:
W AND H, W AND ELLIP, OR H AND ELLIP

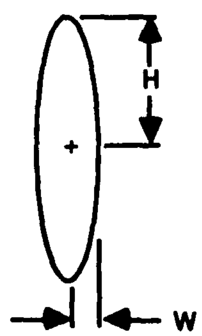


Figure 5b Elliptical Body Geometry Inputs-Option 2

NAMELIST FINSETn

NOMINAL INPUTS

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
SECTYP	--	TYPE OF SECTION TO BE DEFINED: HEX, NACA, ARC, USER	--	HEX
SSPAN (1)	10	SEMI-SPAN	λ	
CHORD	10	PANEL CHORD LENGTH AT EACH SSPAN	λ	
XLE	10	AXIAL STATION (X) OF CHORD LEADINGL EDGE AT EACH SSPAN	λ	0.0
SWEEP (2)	10	SWEEPBACK ANGLE AT EACH SSPAN	DEG	0.0
STA	10	CHORD STATION USED IN MEASURING SWEEP AT EACH SSPAN (0. = LEADING EDGE, 1. = TRAILING EDGE)	--	0.0
LER (3)	10	PANEL LEADING EDGE RADIUS AT EACH SPAN STATION	λ	
NPANEL	--	NUMBER OF PANELS IN SET (1 - 8)	--	4
PHIF (4)	8	ROLL ANGLE OF EACH FIN MEASURED CLOCKWISE FROM TOP VERTICAL CENTER	DEG	--
GAM	8	DIHEDRAL OF EACH FIN (POSITIVE WHEN PHIF IS INCREASED, SEE FIG. 10)	DEG	0.0
SKEW	--	ANGLE BETWEEN THE Y AXIS AND THE FIN HINGE LINE (POSITIVE SWEPT BACK)	DEG	0.0

(1) IF SSPAN (1) = 0 INPUTS ARE RELATIVE TO ROOT CHORD NOT BODY CENTERLINE.

(2) IF USING SWEEP, SPECIFY ONLY XLE (1); IF USING XLE DO NOT SPECIFY SWEEP.

(3) NOT REQUIRED FOR NACA AIRFOILS, REQUIRED FOR USER AIRFOILS

(4) IF NOT INPUT THE NUMBER OF PANELS SPECIFIED ARE EVENLY SPACED ABOUT THE BODY.

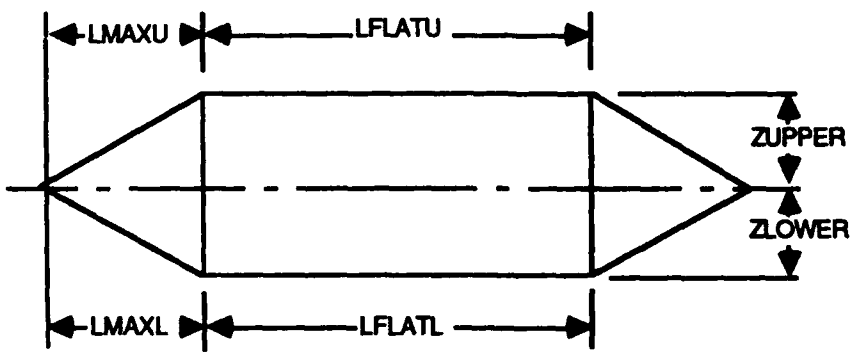
Figure 6a Fin Geometry Inputs-Nominal

NAMELIST FINSETn

M11-14075

OPTIONAL INPUTS

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
ZUPPER	10	MAXIMUM THICKNESS OF UPPER SECTION RATIOED TO LOCAL CHORD LENGTH	—	0.025
ZLØWER	10	MAXIMUM THICKNESS OF LOWER SECTION RATIOED TO LOCAL CHORD LENGTH	—	= ZUPPER
LMAXU	10	FRACTION CHORD FROM SECTION LEADING EDGE TO MAXIMUM THICKNESS FOR UPPER SURFACE	—	0.5
LMAXL	10	FRACTION CHORD FROM SECTION LEADING EDGE TO MAXIMUM THICKNESS FOR LOWER SURFACE	—	= LMAXU
LFLATU	10	FRACTION CHORD LENGTH OF CONSTANT THICKNESS FOR UPPER SURFACE	—	0.0
LFLATL	10	FRACTION CHORD LENGTH OF CONSTANT THICKNESS FOR LOWER SURFACE	—	= LFLATU



NOTE: THESE PARAMATERS MUST BE INPUT AT EACH SPAN STATION.

Figure 6b Fin Geometry Inputs-Optional

NAMELIST FINSETn
INPUTS FOR "USER" SECTIONS

M11-14076

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
XCØRD	50	PANEL CHORD STATION	–	
MEAN ①	50	DISTANCE BETWEEN THE MEAN LINE AND CHORD LINE AT EACH XCORD FOR AVERAGE AIRFOIL SECTION	–	
THICK ①	50	THICKNESS TO LOCAL CHORD FRACTION AT EACH XCORD FOR AIRFOIL SECTION	–	
YUPPER ①	50	AIRFOIL SECTION UPPER SURFACE COORDINATES, FRACTION CHORD, AT EACH XCORD	–	
YLOWER ①	50	AVERAGE AIRFOIL SECTION LOWER SURFACE COORDINATES, FRACTION CHORD, AT EACH XCORD	–	

NOTE: ALL VARIABLES ARE EXPRESSED AS UNITS OF CHORD

① EITHER MEAN AND THICK OR YUPPER AND YLOWER ARE REQUIRED

② LEADING EDGE RADIUS (VARIABLE LER) MUST BE DEFINED

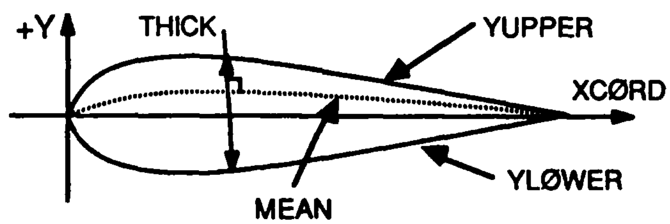


Figure 6c Fin Geometry Input User Airfoils

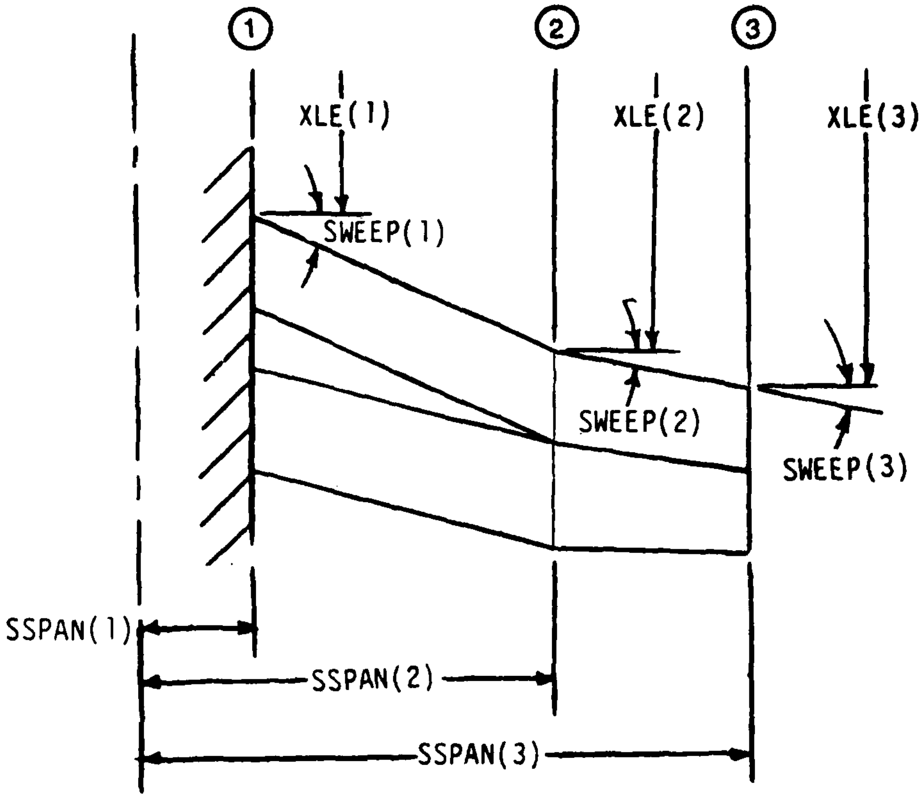


Figure 7 Selecting Panel Break Points

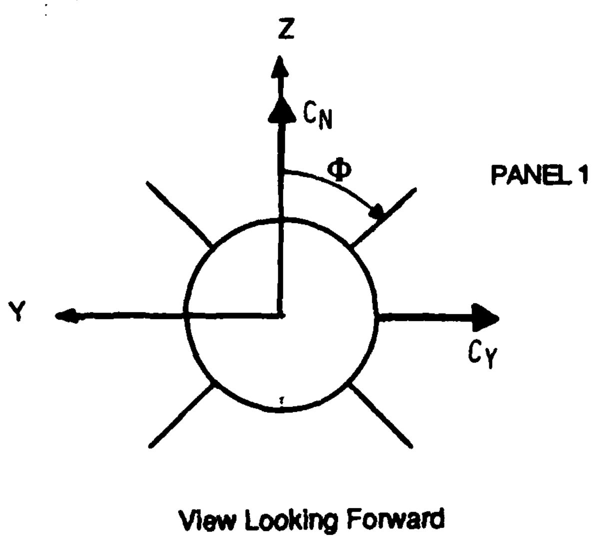


Figure 8 Roll Attitude, PHIF

NAMELIST DEFLCT

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
DELTA1 (1)	8	DEFLECTION ANGLES FOR EACH PANEL IN FIN SET 1 (SUBSCRIPT IS FIN NUMBER)	DEG	0.
DELTA2 (1)	8	DEFLECTION ANGLES FOR EACH PANEL IN FIN SET 2 (SUBSCRIPT IS FIN NUMBER)	DEG	0.
DELTA3 (1)	8	DEFLECTION ANGLES FOR EACH PANEL IN FIN SET 3 (SUBSCRIPT IS FIN NUMBER)	DEG	0.
DELTA4 (1)	8	DEFLECTION ANGLES FOR EACH PANEL IN FIN SET 4 (SUBSCRIPT IS FIN NUMBER)	DEG	0.
XHINGE	4	AXIAL STATION (FROM X = 0) OF PANEL LINE	λ	CR/2 (2)
SKEW	4	SWEEPBACK OF HINGE LINE	DEG	0.

(1) PANEL #1 IS TOP (WHEN $\phi = 0$)

(2) DEFAULT IS AT ONE-HALF OF THE EXPOSED ROOT CHORD

NOTE: A POSITIVE DEFLECTION ANGLE PRODUCES A NEGATIVE BODY AXIS ROLLING MOMENT AT ZERO ANGLE OF ATTACK.

Figure 9 Panel Deflection Inputs

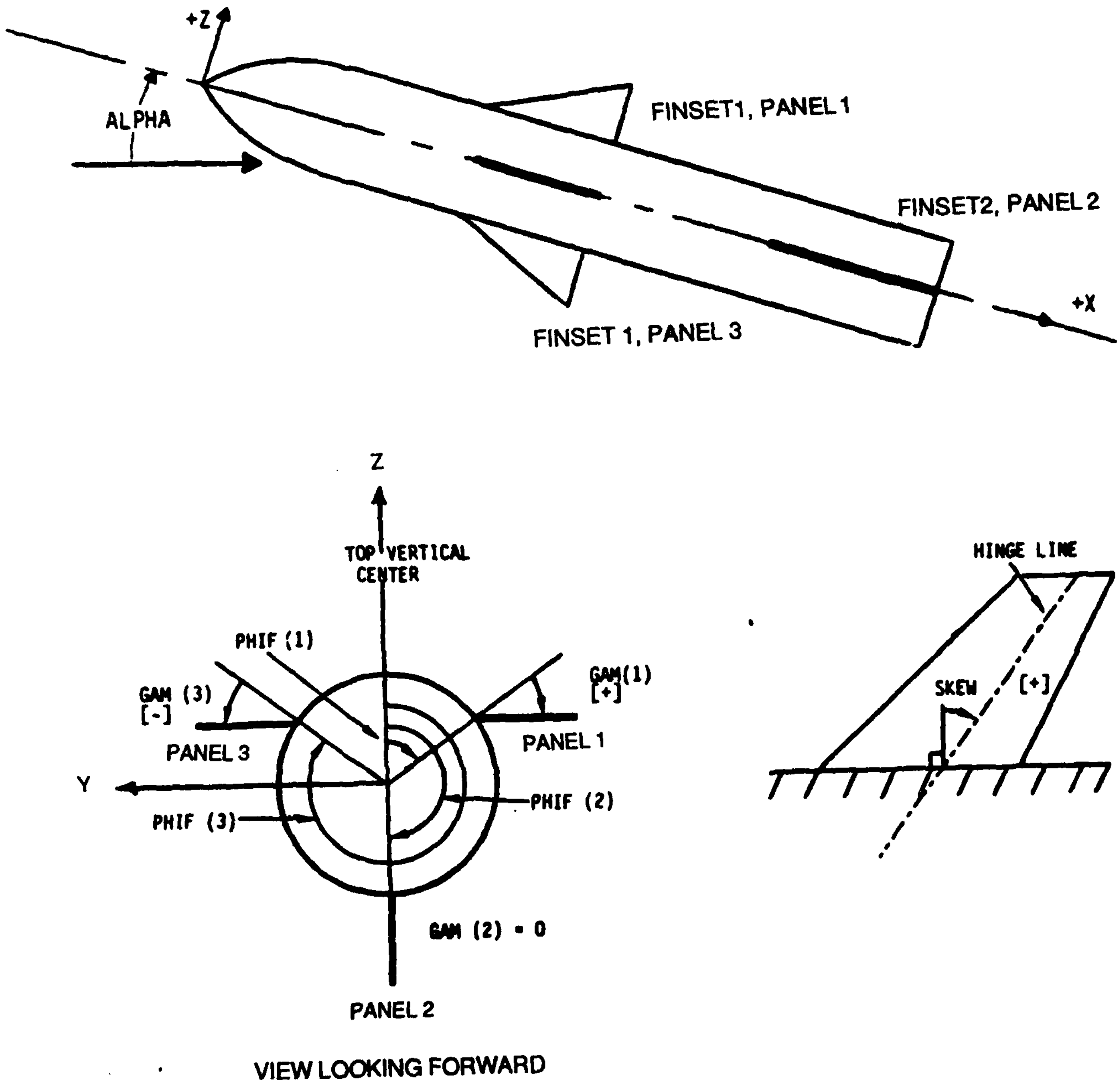


Figure 10 Fin Numbering and Orientation

M11-14081

NAMELIST TRIM

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
SET	--	FIN SET TO BE USED FOR TRIM	--	1.
PANL1 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
PANL2 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
PANL3 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
PANL4 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
PANL5 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
PANL6 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
PANL7 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
PANL8 (1)	--	.TRUE. IF PANEL TO BE USED	--	.FALSE.
DELMIN (2)	--	MINIMUM NEGATIVE DEFLECTION	DEG	-25.
DELMAX (2)	--	MAXIMUM POSITIVE DEFLECTION	DEG	+20.
ASYM	4	.TRUE. IF PANEL WITH SUBSCRIPT IS TO BE DEFLECTED OPPOSITE TO NORMAL SIGN CONVENTION (ASYMMETRIC DEFLECTIONS)	--	.FALSE.

(1) DEFAULTS APPLY ONLY IF ALL PANLX DATA ARE NOT INPUT OR .FALSE.

(2) BOTH DELMIN AND DELMAX MUST BE SPECIFIED

Figure 11 Trim Inputs

M11-14078

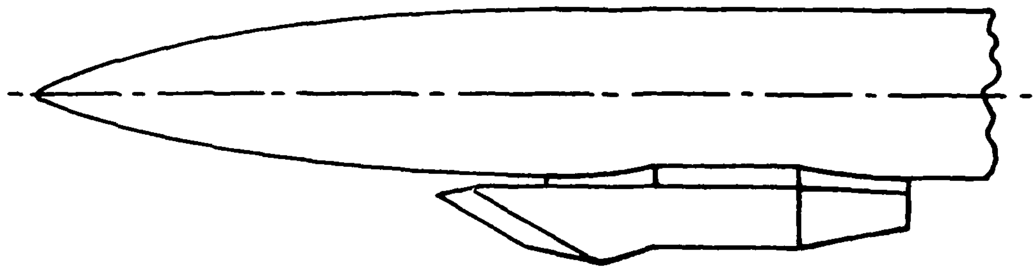
NAMELIST INLET

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
INTYPE	--	INLET TYPE: 2D - TWO DIMENSIONAL = 1 AXI - AXISYMMETRIC = 0	--	0
PØSD	--	FLAG TO IDENTIFY THE TYPE OF DIVERTER 0 = SIDE MOUNTED DIVERTER 1 = TOP MOUNTED DIVERTER	--	0.
PHI (1) (4)	--	ANGULAR ORIENTATION OF INLETS IN SET 1	DEG.	0.
PHI (2) (4)	--	ANGULAR ORIENTATION OF NLETS IN SET 2	DEG.	0.
NIN	--	NUMBER OF INLETS 1, 2, 3, OR 4	--	2.
XBØ OR XBO	--	LONGITUDINAL DIST. FROM X = 0 TO INLET COORDINATE SYSTEM ORIGIN	λ	--
ZBØ OR ZBO	--	DISTANCE FROM VEHICLE CENTERLINE TO INLET COORDINATE SYSTEM ORIGIN	λ	--
X (1)	10	INLET LONGITUDINAL POSITIONS RELATIVE TO XBO	λ	--
B (2)	10	INLET RADIUS OR HALF WIDTH	λ	--
Z (3) (5)	10	INLET HEIGHT RELATIVE TO ZBO	λ	--
XIND (1)	--	DIVERTER LEADING EDGE LOCATION RELATIVE TO INLET COORDINATE SYSTEM ORIGIN	λ	--
XIND (2)	--	DIVERTER MAXIMUM HEIGHT LOCATION RELATIVE TO INLET COORDINATE SYSTEM ORIGIN	λ	--
TIND	--	DIVERTER LEADING EDGE WIDTH	λ	--

(1) THE VARIABLES X1, X2, X3... X10 CAN BE USED INSTEAD OF ARRAY ELEMENT DESIGNATIONS.
(2) THE VARIABLES B1, B2, B3... B10 CAN BE USED INSTEAD OF ARRAY ELEMENT DESIGNATIONS.
(3) THE VARIABLES Z1, Z2, Z3... Z10 CAN BE USED INSTEAD OF ARRAY ELEMENT DESIGNATIONS.
(4) MEASURED POSITIVE DOWNWARD FROM HORIZONTAL POSITION.
(5) Z(1) MUST EQUAL ZERO.

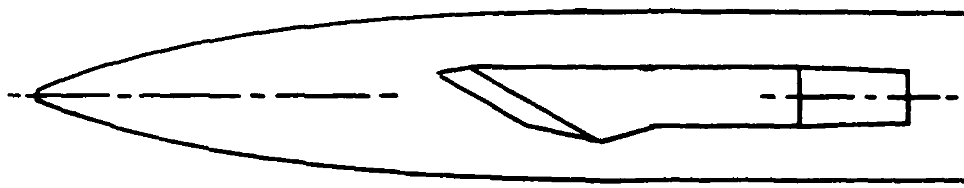
Figure 12 Inlet Geometry Inputs

TOP ATTACHED 2D INLET



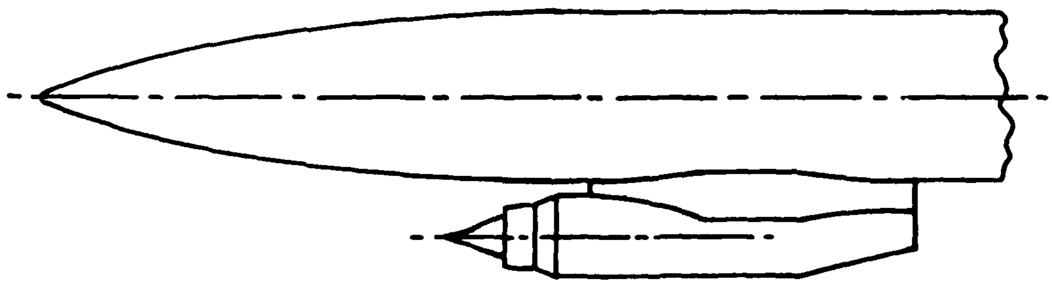
INPUTS: INTYPE = 2D POSD = 1

SIDE ATTACHED 2D INLET



INPUTS: INTYPE = 2D POSD = 0

AXISYMMETRIC INLET



INPUTS: INTYPE = AXI POSD = UNUSED

Figure 13 Typical Missile Inlet Installations

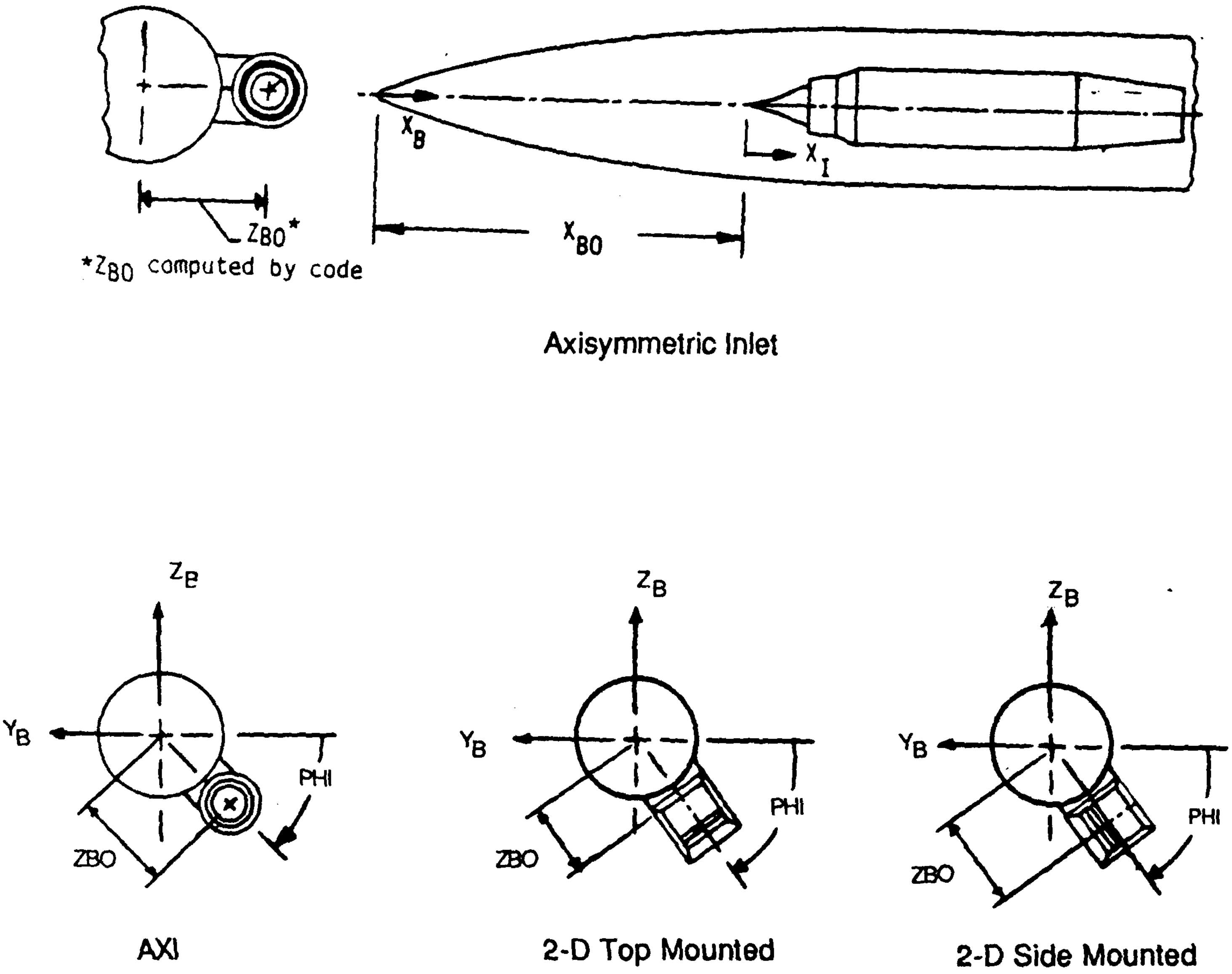


Figure 14 Obtaining Inlet Orientation

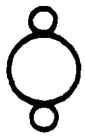
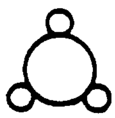
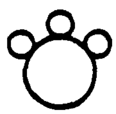
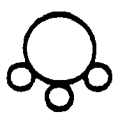
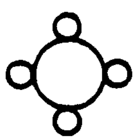
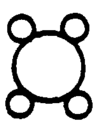
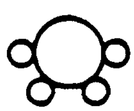
CONFIGURATION	NIN	PHI(1)	PHI(2)
	1	90	–
	2	0	–
	2	90	–
	2	45	–
	3	45	-90
	3	-45	-90
	3	45	90
	4	0	90
	4	60	-60
	4	0	60

Figure 15 Representative Inlet Orientation

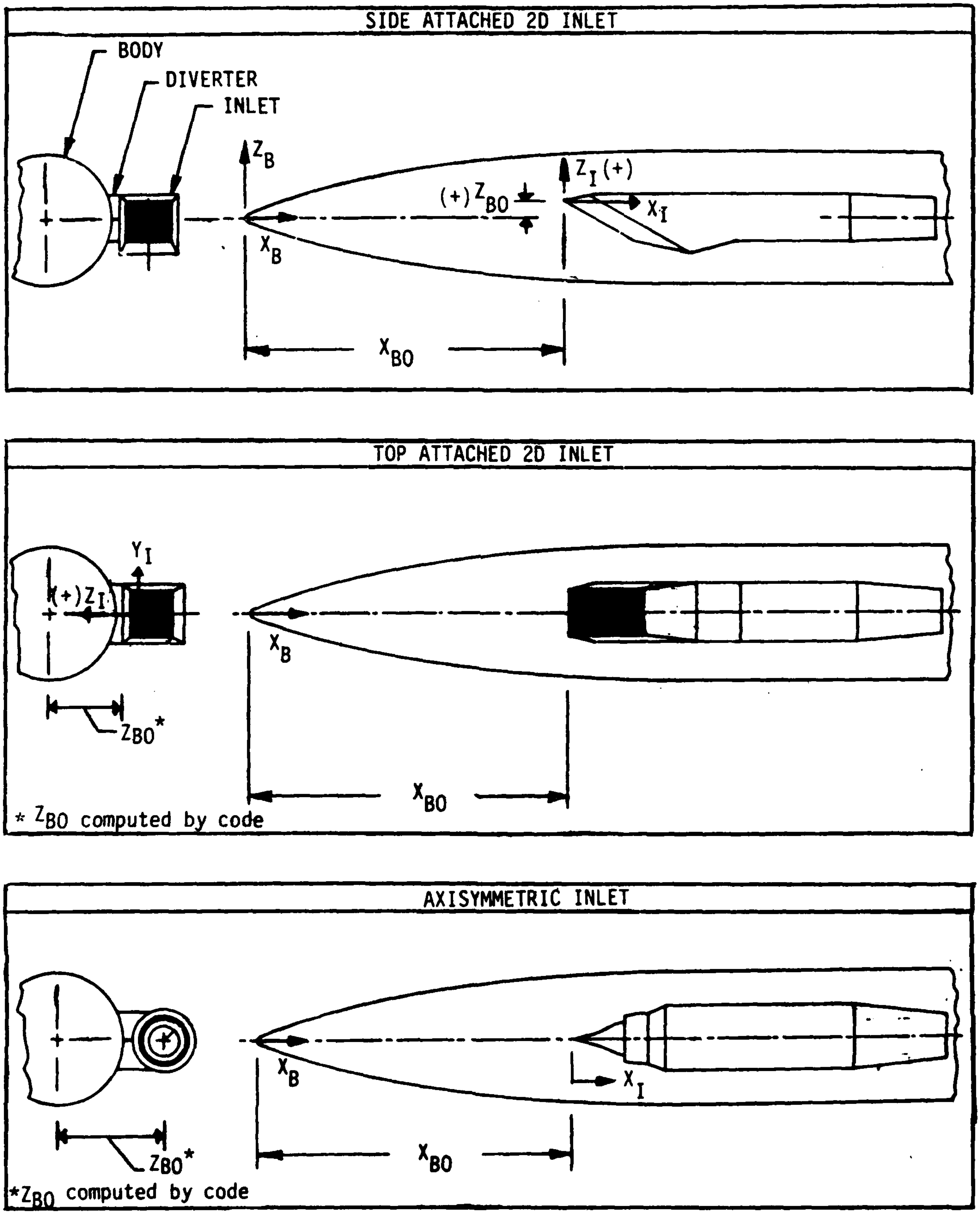
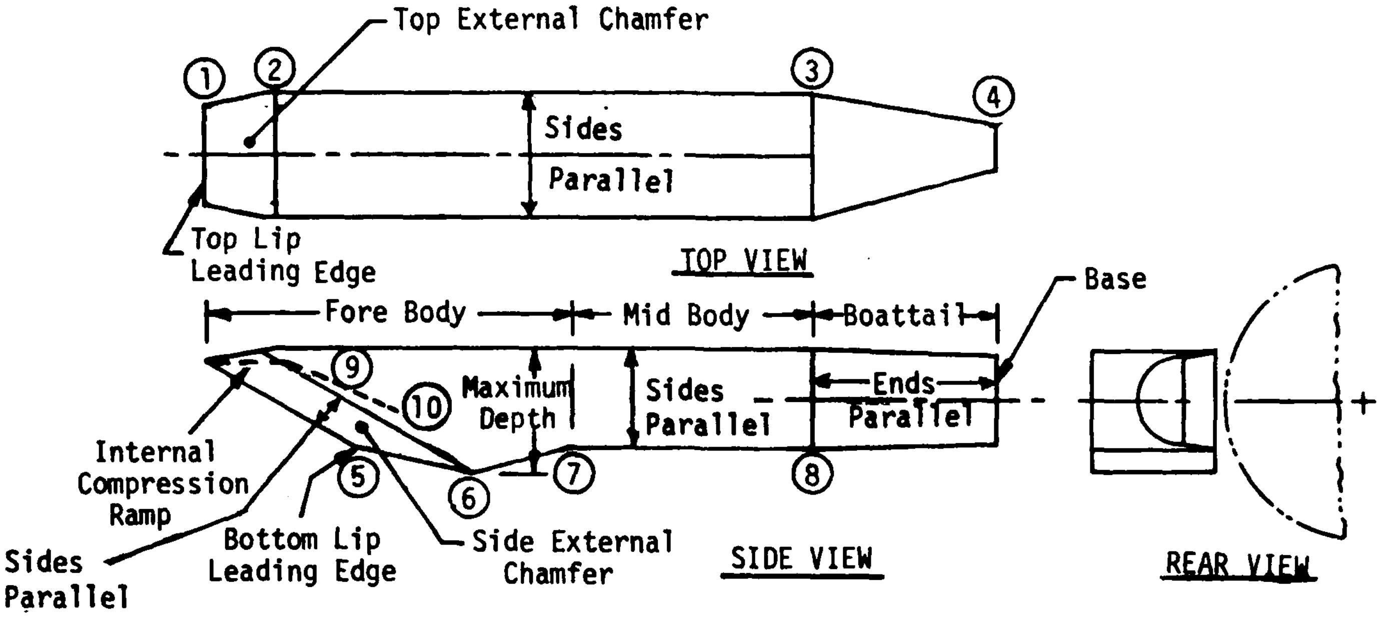


Figure 16 Inlet Coordinate System Defined



Definitions:

Location Number	Definition
1	Top lip leading edge
2	Top lip chamfer end
3	Midbody end and boattail start
4	Boattail end or base
5	Bottom lip leading edge
6	Forebody maximum height
7	Forebody end and Midbody start
8	Midbody end and boattail start
9	Internal compression ramp start
10	Internal compression ramp end

Figure 17 2-D Inlet Nomenclature and Assumptions

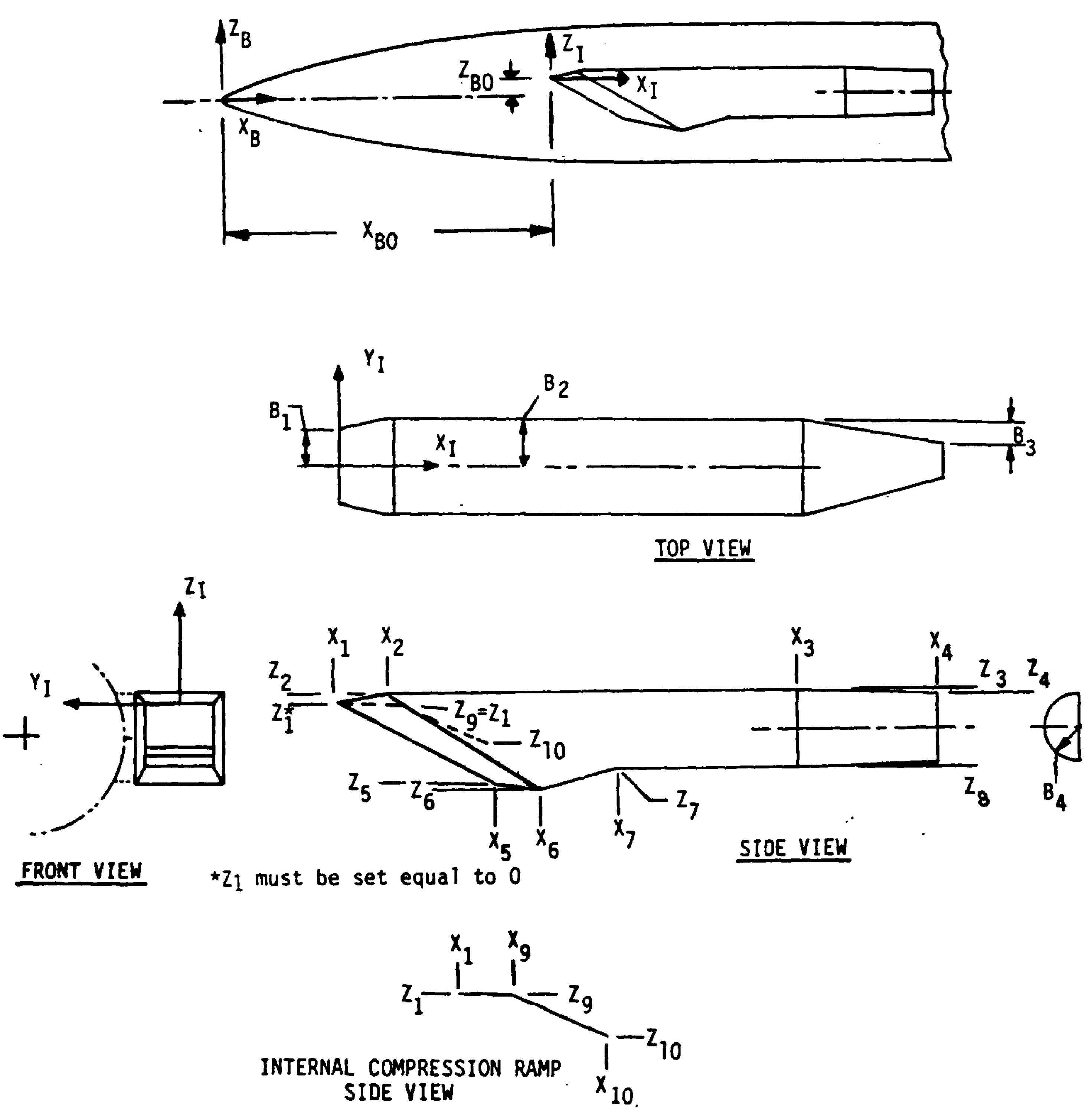


Figure 18 Side Attached 2-D Inlet Nomenclature

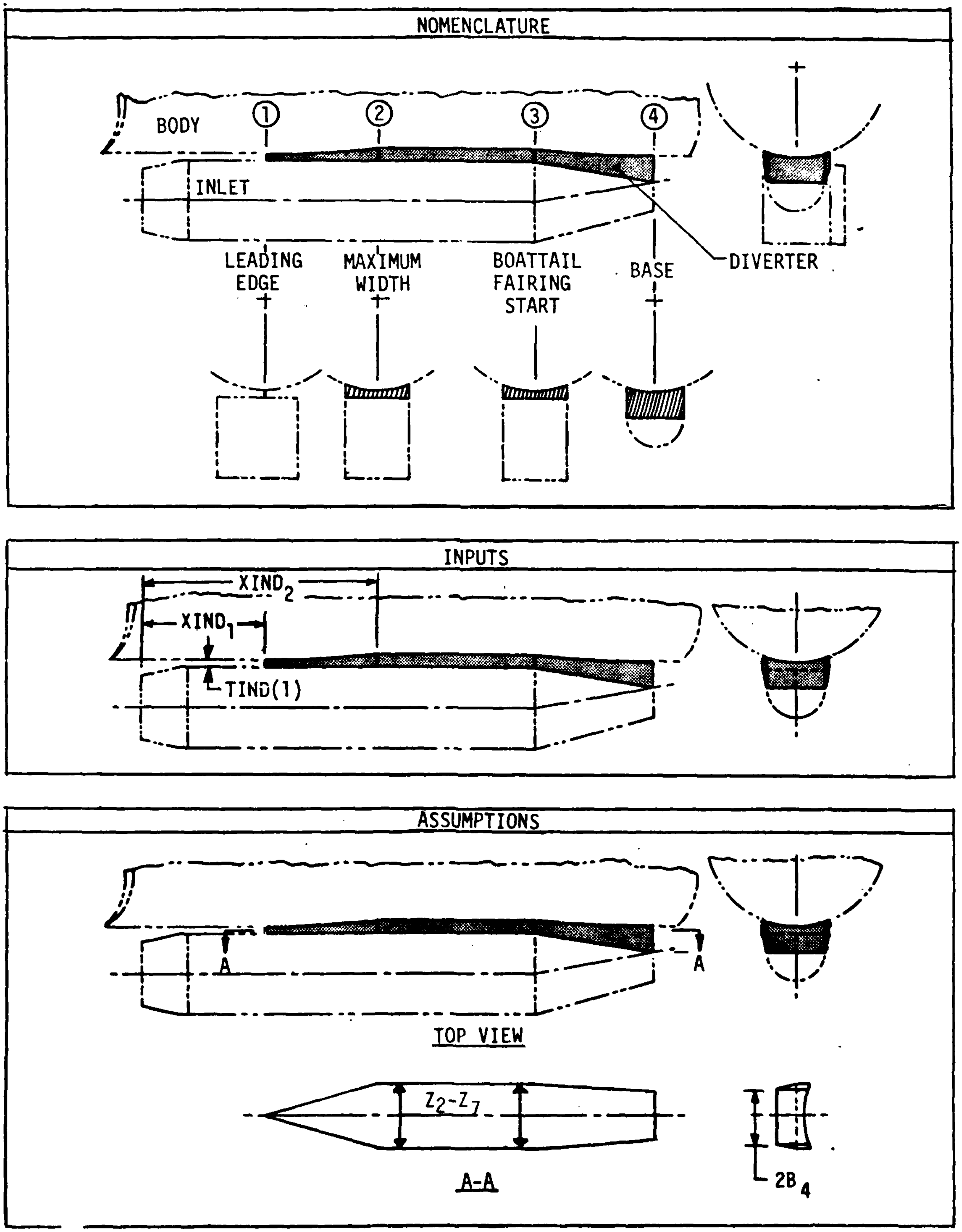


Figure 19 Side Attached Inlet Diverter Nomenclature

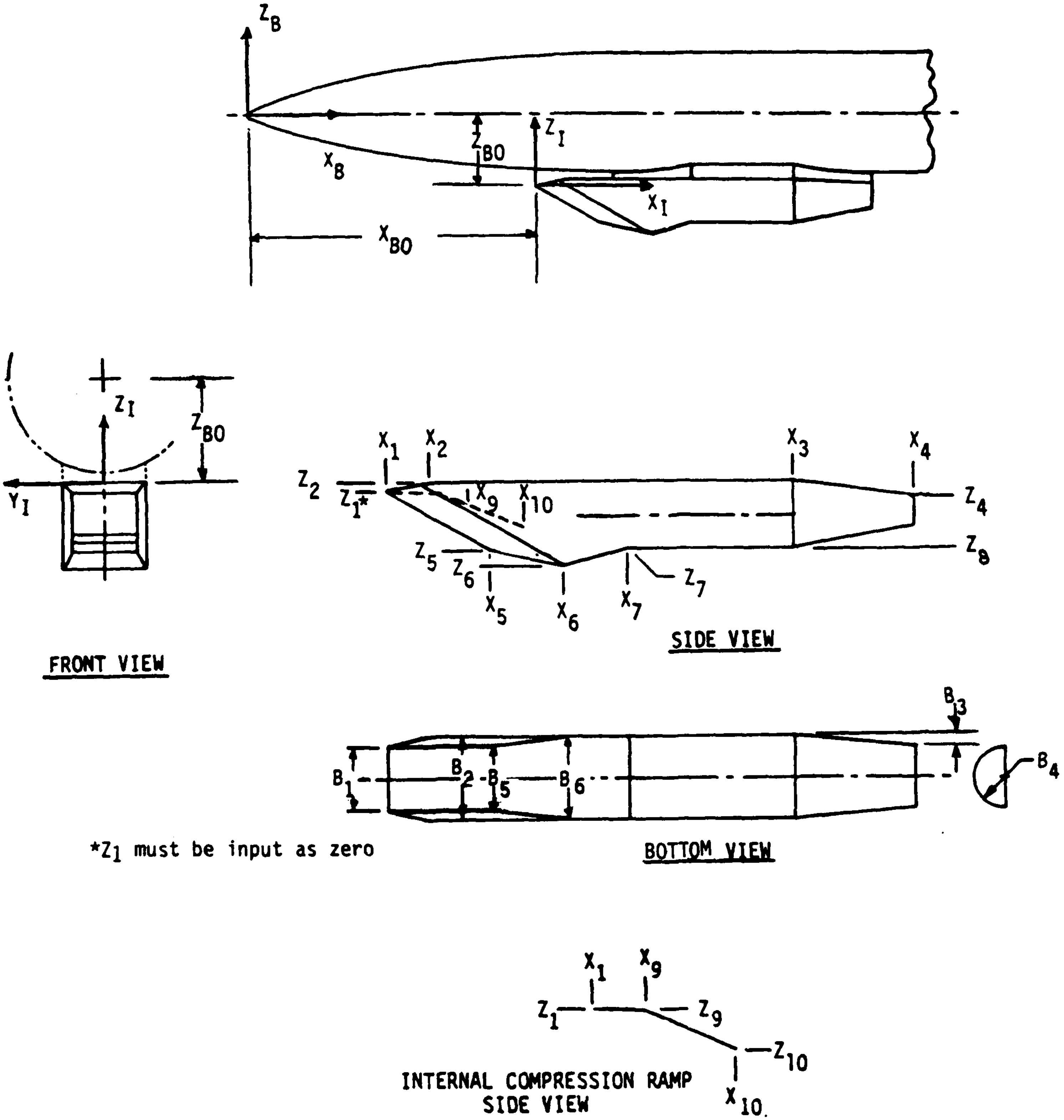


Figure 20 Top Attached 2-D Inlet Nomenclature

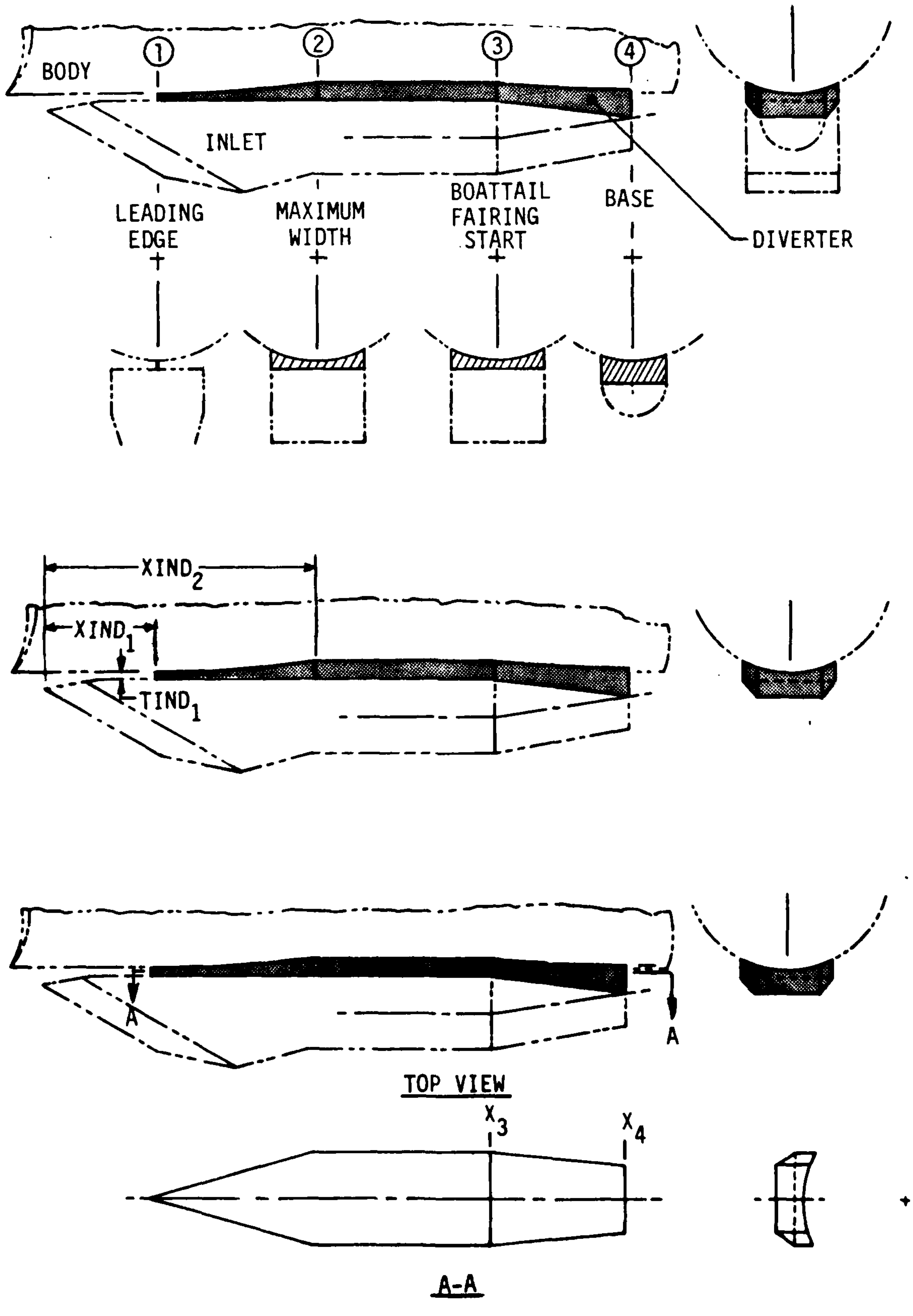
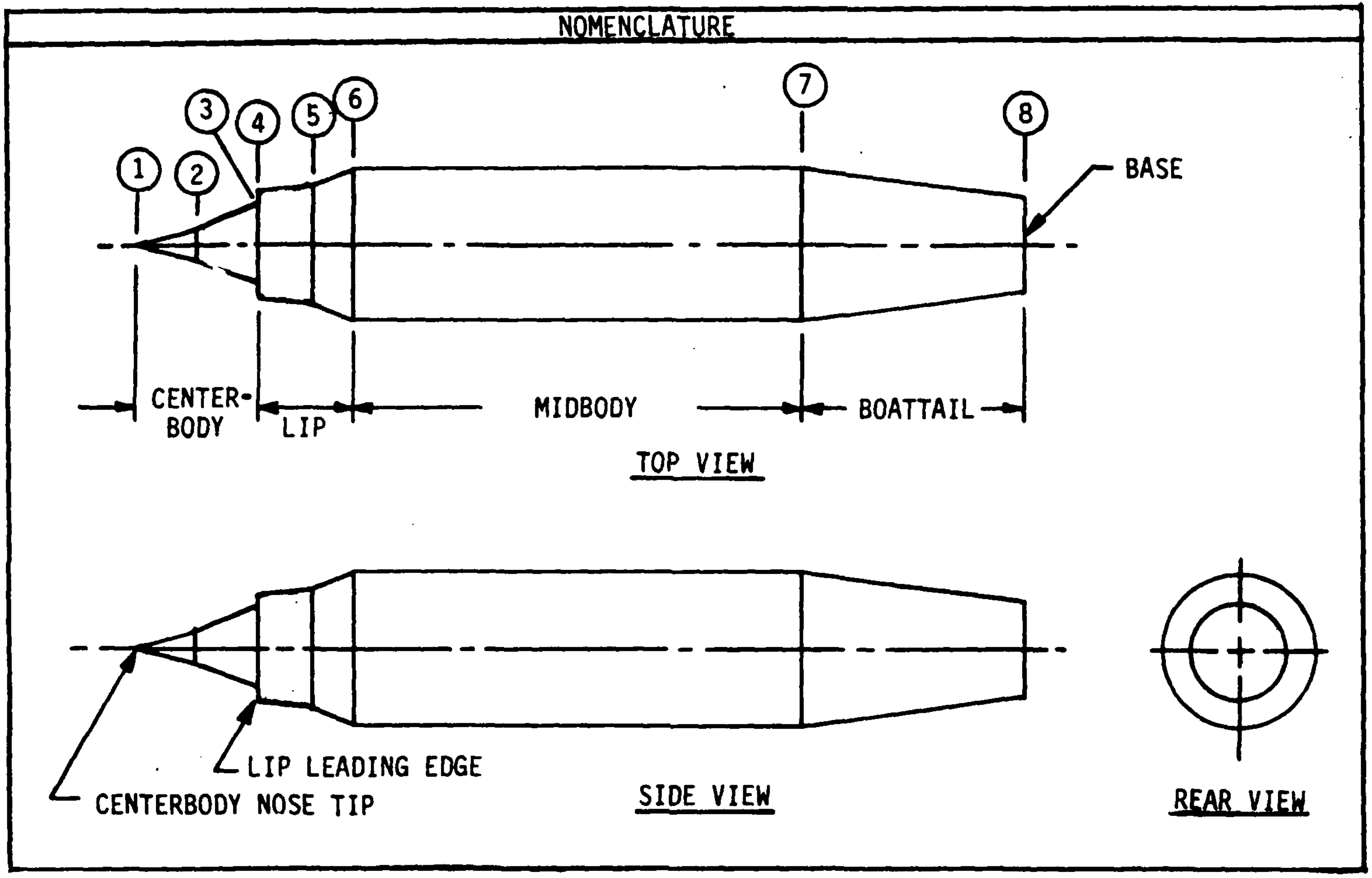


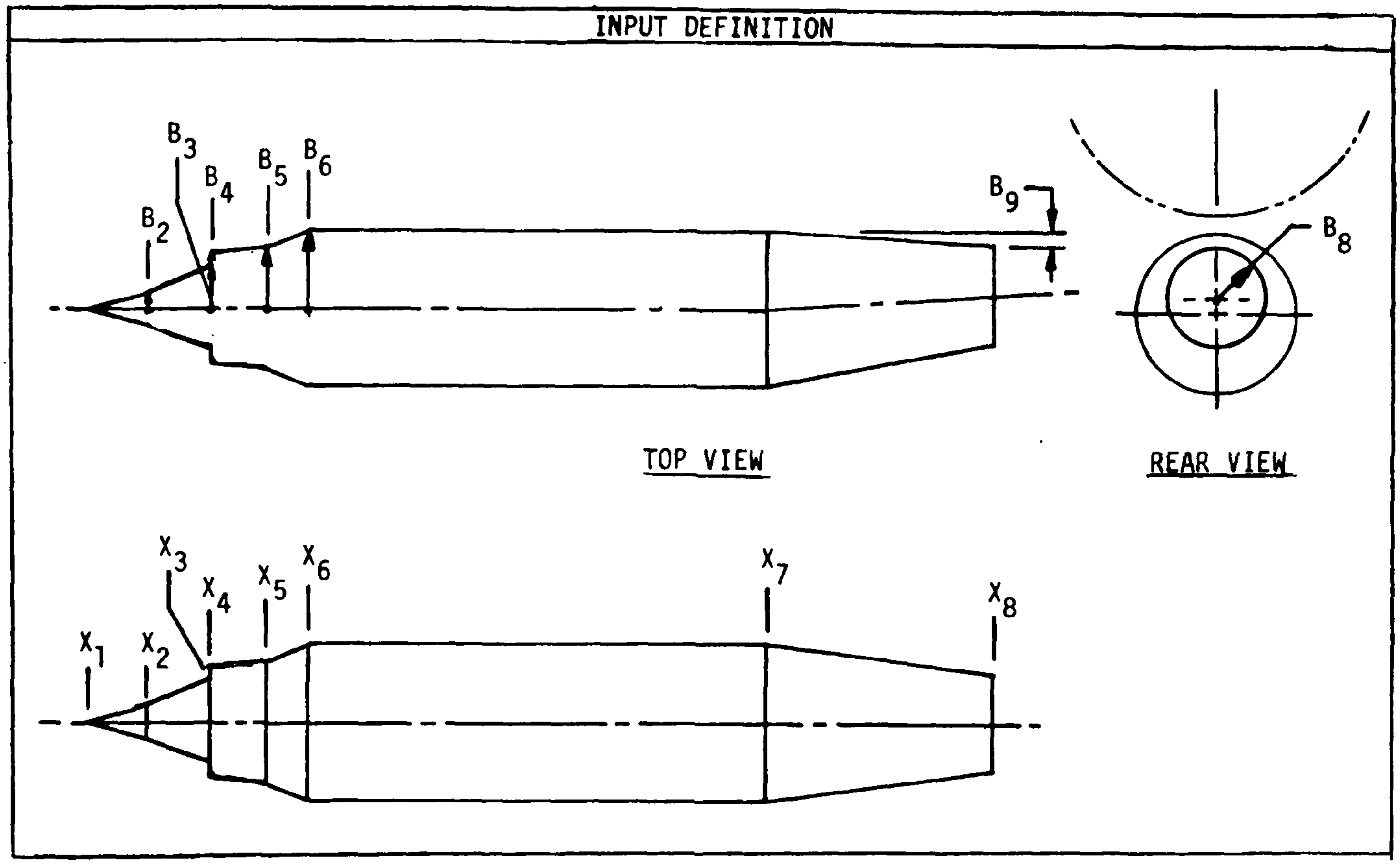
Figure 21 Top Attached Inlet Diverter Nomenclature



DEFINITIONS:

LOCATION NUMBER	DEFINITION
1	CENTERBODY NOSE TIP
2	CENTERBODY INTERMEDIATE LOCATION
3	CENTERBODY END
4	LIP LEADING EDGE
5	LIP INTERMEDIATE LOCATION
6	LIP END AND MIDBODY START
7	MIDBODY END AND BOATTAIL START
8	BOATTAIL END OR BASE

Figure 22 Axisymmetric Inlet Nomenclature



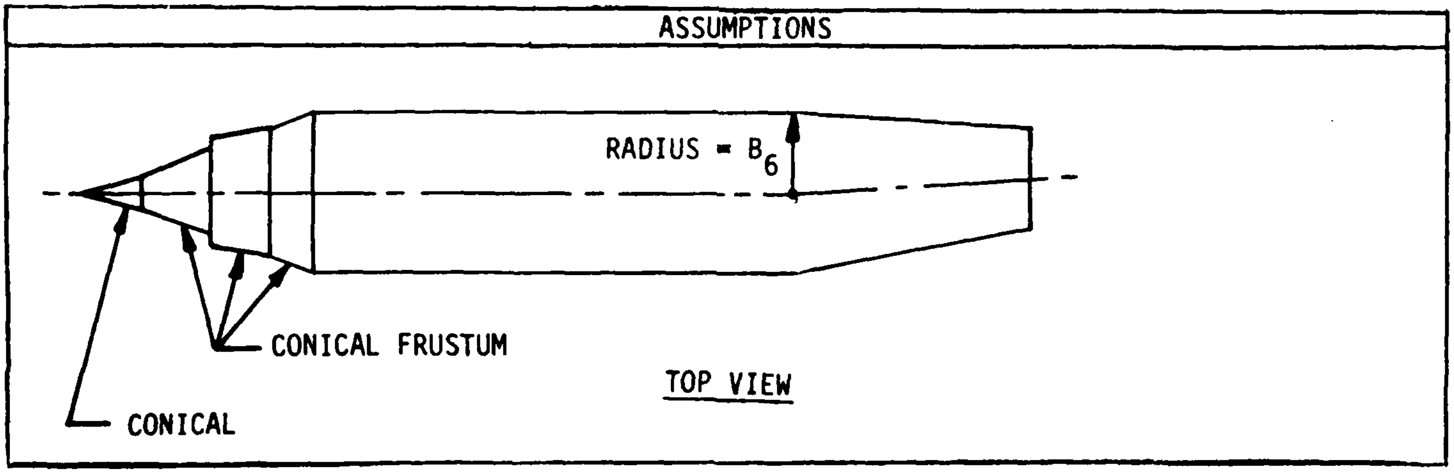


Figure 23 Axisymmetric Inlet Inputs

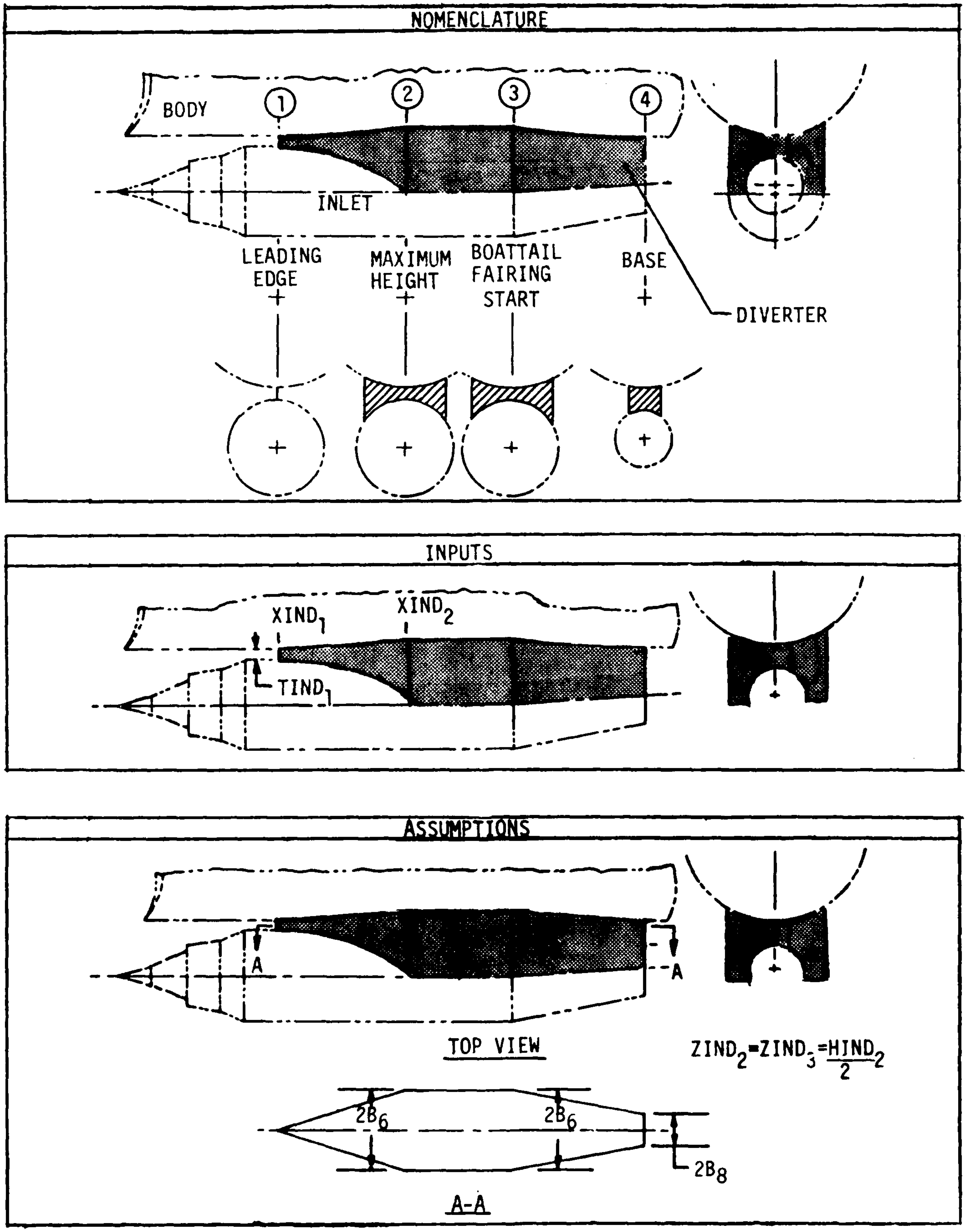


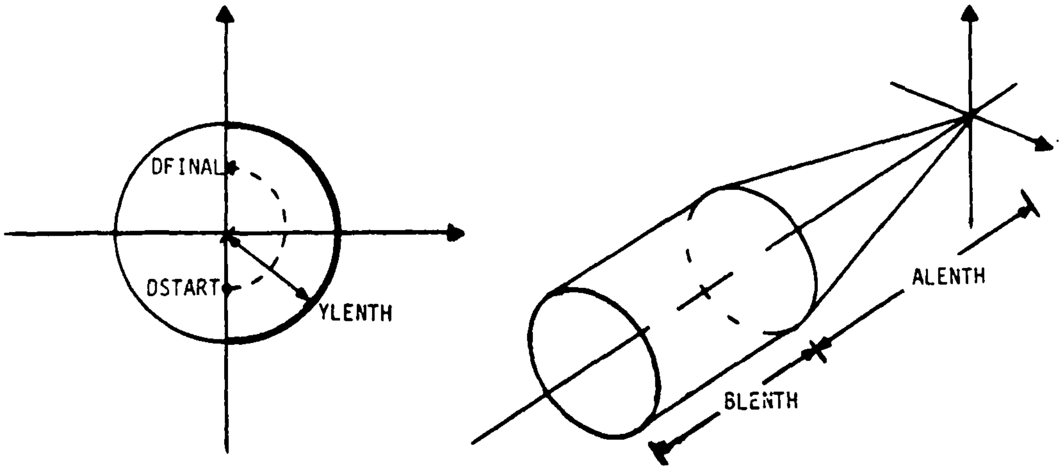
Figure 24 Axisymmetric Diverter Nomenclature

NAMELIST ARBBOD

M11-14079

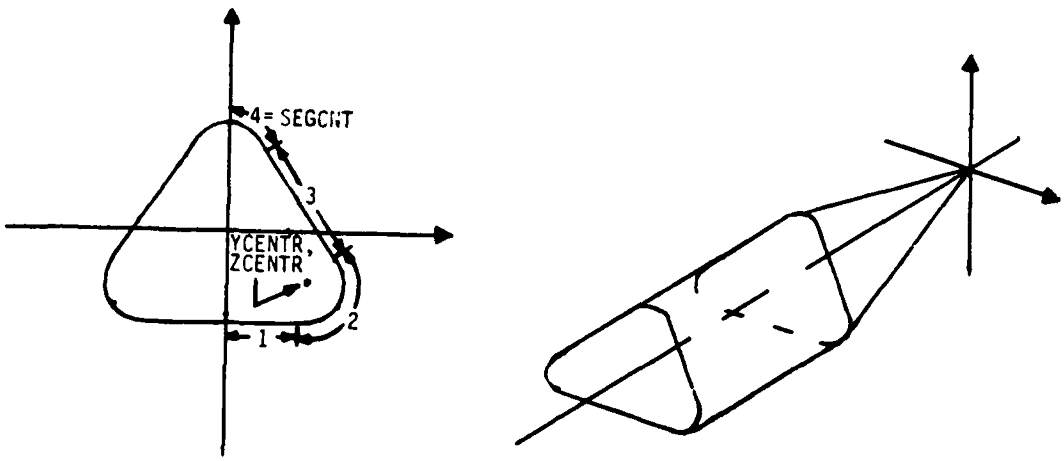
VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
XTIP	–	The distance along the x-axis from the origin to the tip of the forebody	1	
DTIP	–	The width of the forebody at the tip	1	
ALENTH	–	The length of the forebody, measured from the tip to the base	1	
BLENTH	–	The length of the forebody, measured from the base of the nose segment to the base of the centerbody	1	
STACNT	–	The number of stations the nose segment will be divided into when calculating C_{m_α} up to a maximum of nine	–	
SEGCNT	–	The number of linear and elliptic arc segments used to descibe the right half of the forebody base cross section	–	
TTYPE	9	0 indicates the corresponding surface segment will converge to a point at the tip. 1 indicates the corresponding surfacebase segment will converge to a line segment at the tip.	–	
BTYPE	9	1 - point (unused) 2 - line segment 3 - elliptic arc segment	–	
YSTART, ZSTART	9	The coordinates of the starting point of a segment in the base cross section	λ	
YFINAL, ZFINAL	9	The coordinates of the end point of a segment in the base cross section	λ	
YCENTR, ZCENTR	9	The coordinates of the center of an elliptic arc segment	λ	
YLENTH, ZLENTH	9	The semimajor and semiminor axis lengths. Note: YLENGTH does not have to be the semimajor axis. For a circle, only YLENTH needs to be specified.	λ	
DSTART	9	Angle of the starting point measured from the center of the starting ellipse, relative to the positive y-axis.	deg.	
DFINAL	9	Angle of the end point measured from the center of the ellipse to the end point relative to the positive y-axis.	deg.	

Figure 25 Arbitrary Body Inputs



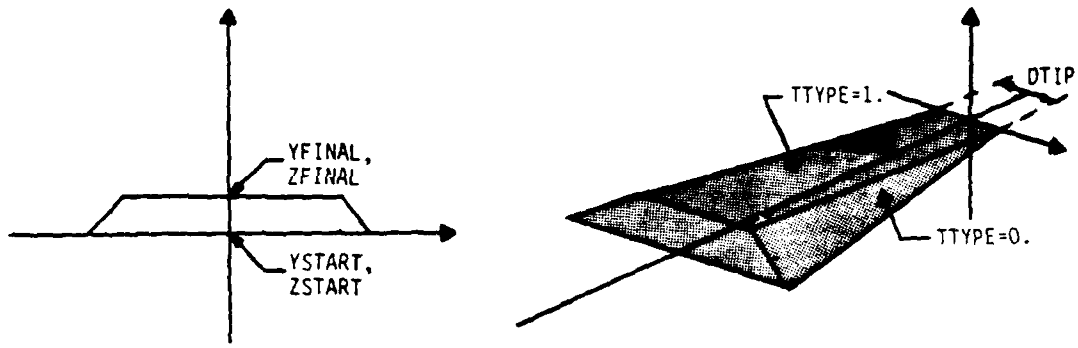

```
CASEID ARBITRARY BODY
NAMELIST
NO LAT
$FLTCON
  NALPHA= 2.,
  ALPHA = 0.,        1.,
  NMACH = 4.,
  MACH  = 2.50,      3.00,      3.50,      4.00,
  REN   = 2.5E06,    2.5E06,    2.5E06,    2.5E06, $
 $REFQ XCG=0., LREF = 3.333, $
 $ARBBOD
  STACNT= 9.000,
  SEGCNT= 1.000,
  ALENTH= 1.111,
  BLENTH= 2.222,
  DTIP  = 0.000,
  BTYPE = 3.,
  "TYPE = 0.,
  DSTART=-90.0000,
  DFINAL= 90.0000,
  YCENTR= 0.0000,
  ZCENTR= 0.0000,
  YLENTH= 0.1667,
NEXT CASE
```

Figure 26 Cone/Cylinder Input Case



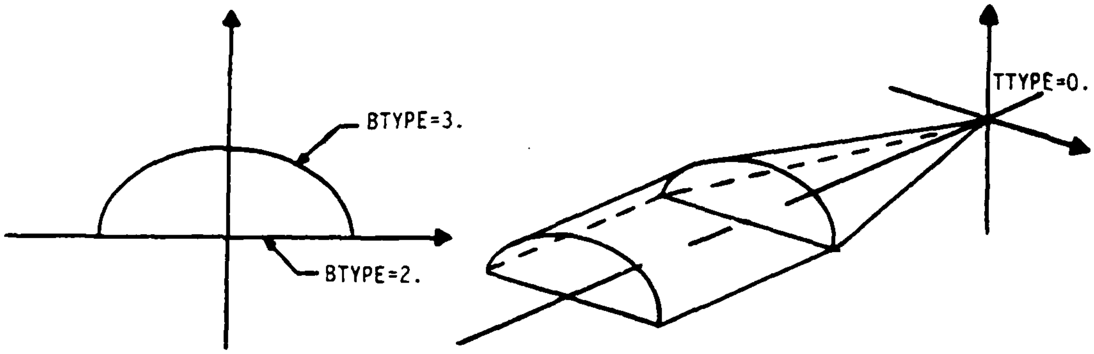

```
CASEID ARBITRARY BODY
NAMELIST
NO LAT
$FLTCON
  NALPHA= 2.,
  ALPHA = 0.,         1.,
  NMACH = 4.,
  MACH  = 2.50,       3.00,       3.50,       4.00,
  REN   = 2.5E06,     2.5E06,     2.5E06,     2.5E06, $
 $REFQ XCG=0., LREF = 3.333, $
 $ARBBOD
  STACNT= 9.000,
  SEGCNT= 4.000,
  ALENTH= 1.111,
  BLENTH= 2.222,
  DTIP  = 0.000,
  BTYPE = 2.,         3.,         2.,         3.,
  TTYPE = 0.,         0.,         0.,         0.,
  YSTART= 0.0000,
  ZSTART=-0.1700,
  YFINAL= 0.1200,     1.0E-30,    0.0578,     1.0E-30,
  ZFINAL=-0.1700,     1.0E-30,    0.1367,     1.0E-30,
  YCENTR= 1.0E-30,  0.1200,     1.0E-30,  0.000,
  ZCENTR= 1.0E-30, -0.1033,     1.0E-30,  0.1033,
  YLENTH= 1.0E-30,  0.0667,     1.0E-30,  0.0667,
  DFINAL= 1.0E-30, 30.0000,     1.0E-30, 90.0000, $
NEXT CASE
```

Figure 27 Rounded Triangular Cone/Cylinder Input Case




```
CASEID ARBITRARY BODY
NAMELIST
NO LAT
$FLTCON
  NALPHA= 2.,
  ALPHA = 0.,        1.,
  NMACH = 4.,
  MACH  = 2.50,      3.00,      3.50,      4.00,
  REN   = 2.5E06,    2.5E06,    2.5E06,    2.5E06, $
 $REFQ XCG=0., LREF = 3.333, $
 $ARBBOD
  STACNT= 9.000,
  SEGCNT= 3.000,
  ALENTH= .646,
  DTIP  = 0.1667,
  BTYPE = 2.,        2.,        2.,
  TTYPE = 1.,        0.,        1.,
  YSTART= 0.0000,
  ZSTART=-0.0000,
  YFINAL= 0.2088,    0.0833,    0.0000,
  ZFINAL= 0.0000,    0.0565,    0.0565,
NEXT CASE
```

Figure 28 Rhombus/Spatular Nose Input Case




```
CASEID MISSILE DATCOM ARBITRARY BODY EXAMPLE
DIM FT
 $FLTCON NMACH=2.0, MACH=2.5,4.0, REN=2.5E6,2.5E6,
         NALPHA=5., ALPHA=0.,5.,10.,15.,20.,$
 $REFQ XCG=1.667,$
 $ARBBOD XTIP=0.0, DTIP=0.0,
         ALENTH=1.111, BLENTH=2.222,
         STACNT=9.0, SEGCNT=2.0,
         TTYPE=0.0,       0.0,
         BTYPE=2.0,       3.0,
         YCENTR=1.E-30,   0.0,
         ZCENTR=1.E-30,   0.0,
         YLENTH=1.E-30,   .2886,
         ZLENTH=1.E-30,   .1924,
         YSTART=0.0,      1.E-30,
         ZSTART=0.0,      1.E-30,
         YFINAL=.2886,    1.E-30,
         ZFINAL=0.0,      1.E-30,
         DFINAL=90.0,$
SOSE
NEXT CASE
```

Figure 29 Flat Bottom Ellipse/Centerbody Input Case

NAMELIST EXPR

M11-14080

VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	DEFAULT
MACH	1	MACH NUMBER	--	--
NALPHA	1	NO. ANGLES OF ATTACK (2-20)	--	--
ALPHA	20	ANGLES OF ATTACK FOR DATA	--	--
SREF (1)	1	REFERENCE AREA FOR DATA	--	--
LREF (1)	1	LONGITUDINAL REF. LENGTH FOR DATA	--	--
LATREF (1)	1	LATERAL REF. LENGTH FOR DATA	--	--
XCG (1)	1	LONGITUDINAL C.G. FOR DATA	--	--
ZCG (1)	1	VERTICAL C.G. FOR DATA	--	--
CONF	1	CONFIGURATION FOR DATA	--	--
		SELECT ONE OF THE FOLLOWING		
		BODY - BODY		
		F1 - WING		
		F2 - TAIL		
		F3 - 3RD FIN SET		
		F4 - 4TH FIN SET		
		BF1 - BODY-WING		
		BF12 - BODY-3 FIN SETS		
		BF1234 - BODY-4 FIN SETS		
CN	20	C_N DATA VS ALPHA	--	--
CM	20	C_M DATA VS ALPHA	--	--
CA	20	C_A DATA VS ALPHA	--	--
CY	20	C_Y DATA VS ALPHA	--	--
CSN	20	C_n DATA VS ALPHA	--	--
CSL	20	C_l DATA VS ALPHA	--	--

(1) FOR DEFAULT VALUES, SEE NAMELIST REFQ.

Figure 30 Experimental Data Inputs

Table 5 Airfoil Designation Using the NACA Control Card

M11-14099

INPUT NACA DESIGNATION	NACA SERIES AIRFOIL	RESTRICTIONS
0012.25	4-Digit	None. Fractional thickness may be specified.
23118.50	5-digit	None. Fractional thickness may be specified.
2406-32	4-Digit modified	Sixth digit specifies position of maximum thickness, (%chord/10), and must be a 2, 3, 4, 5, or 6.
43006-65	5-Digit modified	Seventh digit specifies postion of maximum, (%chord/10), and must be a 2, 3, 4, 5, or 6.
16-212.25	1-Series	Second digit specifies location of minimum pressure, (%chord/10), and must be a 6, 8, or 9. Fractional thickness may be specified.
64-005 64-205 A = 0.6 63A005 652A215 A = 0.8 65,2A215 A = 0.8	6-Series	Second digit specifies location of minimum pressure, (%chord/10), and must be a 3, 4, 5 or 6. The mean line parameter (A = xx) must be a decimal between 0.1 and 1.0 (Defalut is 1.0). See Note 1.
3-30.02.5-40.1 A B C D	Supersonic	See Note 2. A - Section type: 1 = Double Wedge 2 = Circular Arc 3 = Hexagonal B - Distance from leading edge to position of maximum thickness, % of chord. C - Maximum thickness, % of chord. D - For hexagonal sections, length of surface of constant thickness, % of chord.

Note 1. The program does not distinguish between a 64, 2-220 and a 64-220 specification. The difference in coordinates between the two is negligible.

Note 2. All parameters can be expressed to 0.1%. The delimeter "–" must be used.

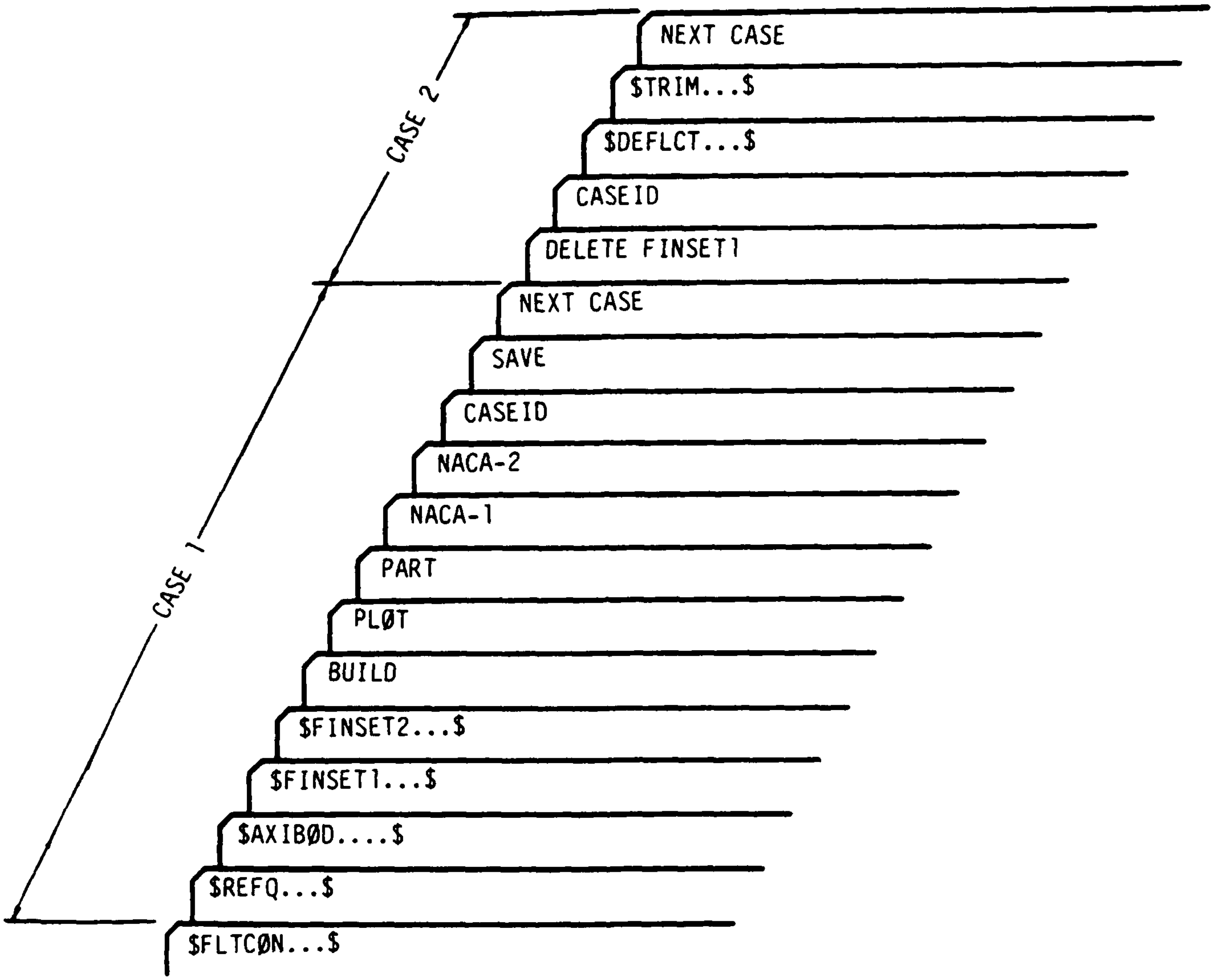


Figure 31 Typical "Stacked" Case Setup

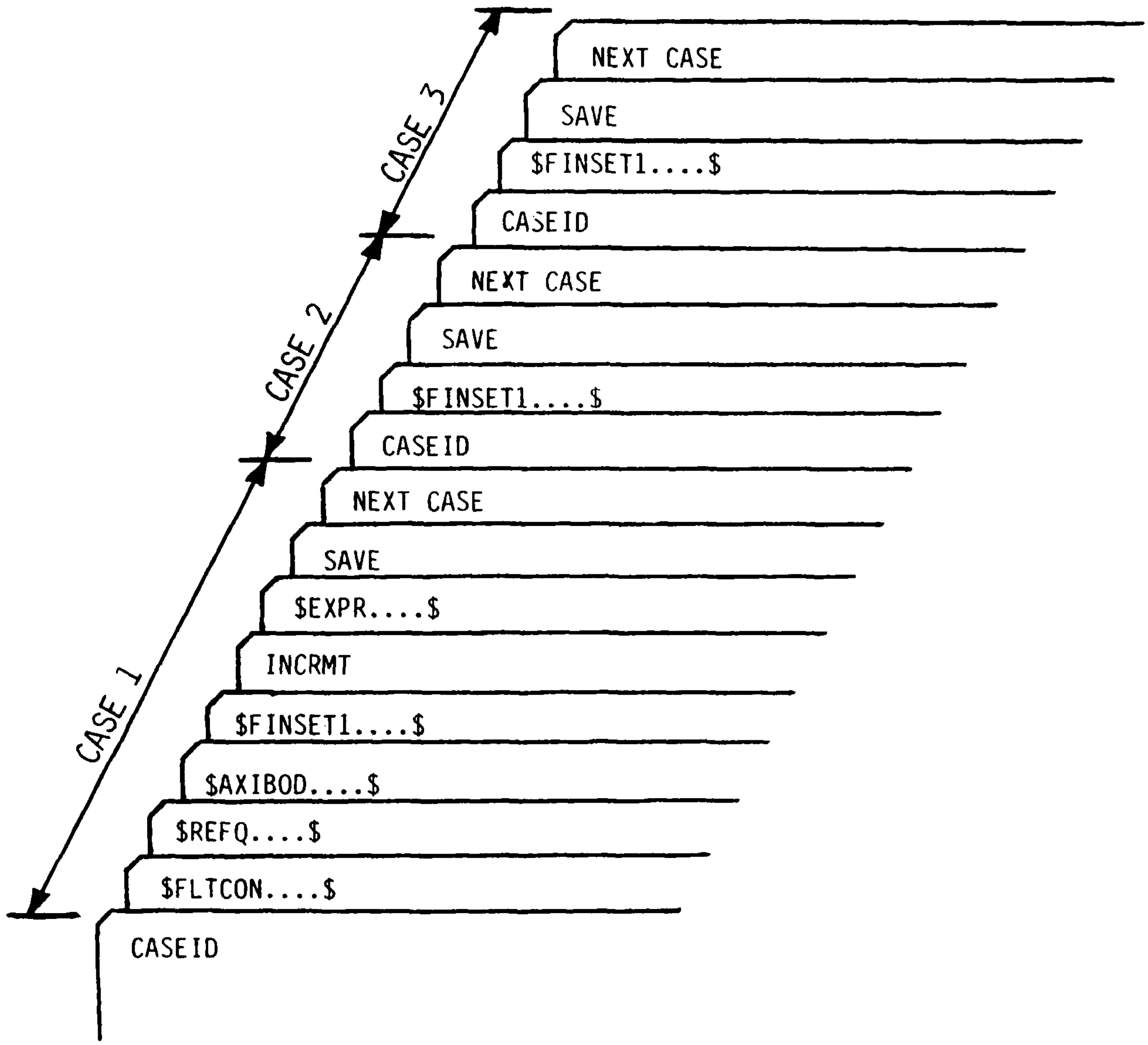


Figure 32 "Configuration Incrementing" Case Setup

4.0 OUTPUT DESCRIPTION

This section describes the types of output available from the code. In most cases the available output is user selectable, that is, it is not normally provided and must be specifically requested using a specialized control card. This feature permits the user to tailor the code output to fit his particular application without extensive reprogramming. This allows him to find the output that he is interested in without having to wade through output that does not interest him.

The following four types of output are available from the code:

- Nominal output - This output is always provided by the code and consists of output from the input error checking module (CONERR), a listing of the inputs for each case, and the final aerodynamic results for the configuration.

- Partial output - This output details the configuration geometry and the intermediate aerodynamic calculations. Special control cards are available so that the user can select the quantity and types of output desired.

- External data files - This output permits the user to create external data files which can the used in post-processing programs, such as plotting or trajectory programs. Both fixed and user defined format data files can be created with the addition of simple control cards.

- Array dumps and extrapolation messages - This output permits the user to print internal data arrays (DUMP) or to investigate design chart extrapolations during the course of the case execution (PRINT EXTRAP).

A special feature of the code is the ability to utilize it as a subroutine package to a larger missile design code. This feature is described in Section 4.5.

The remainder of the section describes each of these output data. Examples of each output page are also included and were treated from the example problems, described in the Appendix, which can be used as a model for setting-up another, similar configuration or be used as a means to check the proper operation of the code.

4.1 Nominal Output

Without the use of any program options the code will provide three types of output. First, an analysis by the input error checking routine is provided. It lists all input cards provided by the user and identifies any input errors detected. Second, a listing of all input cards, grouped by case, are provided; included in this output is an error analysis from the major input error routine MAJERR. Finally, the total configuration aerodynamics are provided in summary form; one page of aerodynamic output is supplied for each Mach number specified. The MAJERR results and the total configuration aerodynamics results are listed in succession for each case.

4.1.1 Input Error Checking

The purpose of the input error checking module is to provide single pass error checking of all inputs. If an error is detected, it is identified and an appropriate error message provided. The error messages are designed to be self-explanatory. In some cases, errors are automatically corrected by the routine, although the routine was not designed to be a comprehensive error correction utility.

The following errors are automatically corrected by the code:

- o No terminating comma on a namelist input card
- o No terminating "$" or "$END" on a namelist input ("&" on IBM systems)
- o No terminating NEXT CASE for the case inputs for single case or last case inputs.

Errors detected by the error checking routine are considered either "FATAL" or "NON-FATAL", a "FATAL" error is one which will cause the code to terminate execution abnormally; examples of "FATAL" errors include incorrect spelling of any namelist name, incorrect spelling of any variable name, and any drastic input error in a namelist input, such as leaving out an equals sign in a constant definition. All "FATAL" errors are clearly identified on the output. A "NON-FATAL" error is one which will not cause the program to terminate execution; an example of a "NON-FATAL" error is leaving off the decimal point on numeric constants all Missile Datcom inputs are either REAL or LOGICAL regardless of the variable name assigned. "NON-FATAL" errors will not cause the code to stop execution, whereas, "FATAL" errors will cause the code to stop execution after input error checking has been completed.

An example output from CONERR is shown in Figure 33. This figure illustrates the array of input errors checked by CONERR. Several additional features of the output are as follows:

o All user defined input cards are assigned a sequential "line number". This serves as means of inputs from the code generated ones (all code-created input cards are not identified with a "line number"). This scheme also permits the user to quickly identify those input cards in error so that efficient correction of input errors can be performed.

o All input cards are listed as input by the user. To the right of each input card is a listing of any errors encountered in processing that card. If no such error message appears then the input was interpreted as being correct.

o In many cases alphanumeric constants are available (see Table 3). Hence the user does not need to memorize a numeric scheme of "flags". Since some computers do not recognize alphanumeric constants as namelist constants, they are automatically converted by the code to their numeric equivalent. A message is printed to identify the substitutions performed. The example input in Figure 34 shows replacements for CONE and OGIVE.

In order to permit column independent inputs the code will automatically adjust some of the input cards to begin in columns 1 or 2. All control cards will be automatically shifted to start in column 1; all namelists which begin in column 1 will be shifted to column 2. If any input card cannot be shifted to conform to this scheme, an error message will be produced. As a general rule, column 80 of namelist inputs should be left blank so that the code can shift the card image, if necessary.

4.1.2 Listing of Case Input Data

Figure 34 shows the first page of outputs for a case without CONERR detected errors. Then Figure 35 shows the next page of output which lists all input cards for the case (down to the NEXT CASE control card). If the input for a case is from a previous case (through use of the SAVE control card) only the new case inputs are listed. All saved inputs are not repeated in subsequent case input summaries.

After the case data have been read, the data setup for the case is analyzed by the case major error checking module (MAJERR). The purpose of this second error checking is to insure that the data input, although syntax error free, properly defines a case to be run. Examples of errors detected in MAJERR include valid flight condition inputs, valid reference condition inputs, and that geometry has been defined. In most cases errors detected by MAJERR are corrected with assumed defaults. If any MAJERR error message is produced, the user should verify the "fix-up" taken by the code. In some cases a "fix-up" is not possible; an appropriate error message and a suggestion for correcting the error is provided. If a "fix-up" is not possible the case will not run.

4.1.3 Case Total Configuration Aerodynamic Output Summary

As shown in Figure 36, the total configuration aerodynamics are provided in compact form for easy review. The aerodynamics are summarized as a function of angle of attack (ALPHA) in the user specified system of units. the nomenclature is as follows:

CN	- Normal force coefficient
CM	- Pitching moment coefficient
CA	- Axial force coefficient
CY	- Side force coefficient
CLN	- Yawing moment coefficient
CLL	- Rolling moment coefficient
CNA	- Normal force coefficient derivative with ALPHA
CMA	- Pitching moment coefficient derivative with ALPHA
CYB	- Side force coefficient derivative with BETA
CLNB	- Yawing moment coefficient derivative with BETA
CLLB	- Rolling moment coefficient derivative with BETA
CL	- Lift coefficient
CD	- Drag coefficient
CL/CD	- Lift to drag ratio
XCP	- Center of pressure from the moment reference center divided by reference length
DX/L	- (XCP - XCG)/LREF

All coefficients are based upon the reference areas and lengths specified at the top of the output page. The derivatives CNA and CMA are computed by numeric differentiation of the CN and CM curves, respectively; precise derivatives are only obtained when the angle of attack range specified is narrow. The derivatives CYB, CLNB and CLLB are determined by perturbing the sideslip angle by one degree, recalculating the configuration forces and moments, and then differencing with the user specified orientation. Hence, the longitudinal and lateral derivatives will probably not be numerically identical for those conditions which should produce identical results if they were both calculated by the same method.

A significant decrease in computational time is realized when the calculation of lateral-directional derivatives are suppressed using the control card NO LAT. For these cases, the CYB, CLNB, and CLLB data fields are filled with blanks.

When selecting TRIM, the output is provided in a form similar to Figure 37. When running a trim case the derivatives due to ALPHA and BETA are not available. The panels which were deflected to trim the configuration are indicated by the "VARIED" citation next to them.

The format for the values of the numbers in the printed output has been assumed based on typical magnitudes for missile aerodynamic coefficients. In some cases, a user specified reference area and/or length will cause the results to underflow or overflow the formal selected. For these cases the user should adjust his reference quantities by powers of ten to get the data to fit the format specified.

4.2 Partial Output

Partial output consists of geometry calculation details, intermediate aerodynamic results, or auxiliary data, such as pressure distributions. Each of these output types are printed through the addition of control cards input for each case. In all cases, partial output requested for one case is not automatically selected for subsequent cases, and the control cards must be re-input. This permits the user to be selective on the amount and types of output desired.

A special control card PART permits the user to request all geometric and aerodynamic partial output. Due to the amount of output produced, this option should be used sparingly or when details of the calculations are desired.

There is one geometry for which no partial output is obtained. This is when bodies with arbitrary cross sections are input.

The following paragraphs describe the output received when partial output is requested.

4.2.1 Geometric Partial Output

Details of the geometry are provided when the PART or PRINT GEOM control cards are included in the case inputs. Figure 38 shows the output created when the PRINT GEOM BODY control card is used. Detailed are the results of the geometric calculations for the body. Included are such items as planform area, surface (wetted) area, and the mold line contour.

If fins are present on the configuration, two types of fin geometry data are produced when PRINT GEOM FIN1 or PART is requested. As shown in Figure 39, the description of the panel airfoil section is provided. Following that, shown in Figure 40, is a summary of the major geometric characteristics of such planform; note that fin planform geometry data is given for one panel of each fin set, since it is assumed that each fin of a fin set is identical. If a panel is made up of multiple segments, the geometric data is provided by panel segment (each segment is assigned a number starting at the root). Total panel set of characteristics is also provided. This total panel data represents an equivalent straight-tapered panel, which is used for most of the aerodynamic calculations. The thickness-to-chord ratio shown for each segment is that value at the segment root; for the total panel, it is an "effective" value.

If an airbreathing inlet is specified (presently available only at supersonic speeds) the output is similar to that in Figure 41. This output reflects the user input definition for the inlet design specified. It is provided if the PRINT GEOM INLET or PART control cards are included in the input case.

4.2.2 Aerodynamic Partial Output

The output on the configuration aerodynamics is most extensive when PRINT AERO or PART is specified. As shown in Figures 42, 43, 44 and 45, output is created for the body and each fin set on the configuration. In addition, for any subsonic/transonic Mach number (less than 1.4) an analysis by the Airfoil Section Module is made, which involves a potential low analysis of the airfoil section using conformal mapping is made. Figure 46 shows the output produced from this analysis. If an inletted configuration additional partial output is included to summarize the inlet external aerodynamics. Figure 48 presents an example of this output.

As shown in Figure 42, the body alone partial aerodynamic output for normal force lists the axial force contributors, potential normal force (CN-POTENTIAL), viscous normal forces (CN-VISCOUS), potential pitching moment (CM-POTENTIAL), viscous pitching moment (CM-VISCOUS), and the crossflow drag coefficient (CDC). The cross-flow drag proportionality factor at subsonic and transonic speeds is also given for reference. These data are similar to that obtained for elliptical bodies.

Figure 43 details the fin normal force calculations by fin set. Each panel's contribution to the configuration normal force is described. The column titled CN-POTENTIAL is the potential contribution and the column titled CN-VISCOUS is the viscous contribution. Their sum is given in the column titled CN-TOTAL. CNAA is the nonlinear variation of normal force due to angle of attack and ALPHA EQUIV is the panel angle of attack due to its roll position on the body. Figure 45 illustrates the fin axial force contributors and Figure 46 presents an example of the fin pitching moment contributors.

The analysis by the Airfoil Section Module is provided in a format similar to Figure 46. If any mach number specified produces supersonic flow on the airfoil surface, the message "CREST CRITICAL MACH NUMBER EXCEEDED" will be printed; approximation of the airfoil section data is then assumed. These fin aerodynamic increments are repeated for each fin set on the configuration. Note that the Airfoil Section Module assumes that the panels have sharp trailing edges. Any panel input with a non-sharp trailing edge will have its aerodynamic characteristics set as though the airfoil was "ideal". This assumption is approximate for preliminary design.

Figure 47 shows the aerodynamic output available when inlets are specified on the configuration. It is provided when PRINT AERO INLET or PART is specified in the case inputs. The aerodynamics summarized for inlets assume that the inlets are operating at their critical Mach number. No internal aerodynamics are calculated or included in the output.

After the aerodynamic details for each component of the configuration are output, the aerodynamic calculations for the synthesis of the complete configuration follows. As shown in Figure 48, the aerodynamics for the body are first presented. This is followed by the following output for each fin set. In the case of the example case fin set 1 results would be followed by fin set 2 results for each of the following outputs:

- "FIN SET PRESENCE OF THE BODY" - This summarizes the aerodynamic incrementals of the most forward set of fins with the influence of the body. Figure 49 presents the example of this output. The left-most six columns include the effect of body-on-fin component interference. The right-most columns represent the contribution to each panel to configuration aerodynamics, and include the effect of body-on-fin interference, these values are, in effect, individual panel loads. The panel characteristic values included are "AEQn" (the panel equivalent (local) angle of attack) and "CNn" (the panel normal force coefficient). The sign convention is as follows: a positive panel normal force, hence, equivalent angle of attack, produces a negative roll moment. Hence, panels on the right side of the configuration will produce loads and angles of attack opposite in sign to those on the left side of the configuration even though they produce the same physical force loading.

- "BODY-FIN SET" - Aerodynamics for the body plus most forward set of fins configuration. It is produced through addition of the body alone and wing in presence of the body incrementals, described above. The results shown in Figure 50, include the component carryover factors K-W(B) (wing in presence of the body carryover due to angle of attack), K-B(W) (body in presence of the wing carryover due to angle of attack), KK-B(W) (body in presence of the wing carryover due to panel deflection), JCP-W(B) (wing in presence of the body carryover center of pressure), and XCP-B(W) (body in presence of the wing carryover center of pressure). This output is repeated for the body plus the first and second most forward lifting surfaces, if one exists on the configuration. This example includes two fin sets so the next page of partial output would look like Figure 51. If additional fin sets are present on the configuration additional pages are output with each one successively included.

- "CARRYOVER INTERFERENCE FACTORS" - This page of partial output summarizes the carryover factors listed in the paragraph above. These were included in the body plus fin set calculations. An example of this output is presented in Figure 52.

- "COMPLETE CONFIGURATION" - Complete configuration aerodynamics. This output is illustrated in Figure 36. The values are obtained by summing the body-wing and tail in the presence of the wing flow field data.

If the PRINT AERO BEND or PART control card is used, the code will compute and print panel bending moment coefficients for each fin set on a separate page. One page is shown in Figure 53. The sign convention is that assumed for the individual panel loads and equivalent angles of attack, noted above. The bending moment coefficients are based upon the reference area and longitudinal length given at the top of the page. The moments are referenced about the fin-body structure specified by the root chord span station.

Figure 54 illustrates the panel hinge moments coefficients computed when the control cards PRINT AERO HINGE or PART are used. The reference area and longitudinal reference length given at the top of the page are used. All moments are computed about the hinge line, which is defined using namelist DEFLCT.

If TRIM is specified, the user can selectively print the six untrimmed static aerodynamic tables used in the trim process. An example is shown in Figure 55. The code computes the six-component aerodynamics at ten deflection angles for each specified angle of attack, then interpolates for C_m=0. Note that this trim process can be used to create control authority data, effectively giving the user 10 deflection angles, 20 angles of attack, and 20 Mach numbers per input case.

4.2.3 Pressure Distribution Data

If the Mach number is supersonic ($M \geq 1.2$), the user has the option to print the surface pressure distributions over the body and fins. This option is selected only through the addition of the control card PReSSURES. Since three body alone supersonic methods are available (Van Dyke Hybrid, Second-Order Shock Expansion (SOSE), and Newtonian flow) the capability exists to output the pressure distribution data from any one of these methods. The method to be used in the calculation of the pressure data is controlled with the control cards SOSE and HYPER; if neither control card is input, the Van Dyke Hybrid method is selected. Because of the nature of the calculations, body alone pressure are printed for angles of attack less than or equal to 15 degrees when using the Hybrid or SOSE techniques.

The capability also exists for the user to output the pressure distribution data over fins at any Mach number greater than 1.05. This option is also controlled by the PRESSURES control card. Due to the nature of the method, only pressure distribution data at zero angle of attack is presently output.

Figures 56, 57 and 58 illustrate typical output produced when PRESSURES is specified. The format of Figure 56 is only available when SOSE is specified; all other body alone pressure methods produce output similar to Figure 57 for bodies. Figure 58 is representative of fin pressure distribution output. Note that calculation of pressures is a time-consuming process; much higher computational times will be required.

All body pressure distribution data is based on a configuration that has body diameter of unity; that is, the configuration is expressed in calibers (or body diameters). The longitudinal stations at which pressure coefficient data is desired cannot be user specified; however, sufficient data is provided to permit accurate interpolation for most applications.

4.3 Dynamic Derivatives

As shown in Figure 59, the total configuration dynamic derivatives are provided in compact form for easy interpretation. The dynamic derivatives are summarized as a function of angle of attack in the user specified units. The coefficients provided are as follows:

CNQ - Normal force coefficient due to pitch rate

CNAD - Normal force coefficient due to acceleration in angle of attack (α)

CMQ - Pitching moment coefficient due to pitch rate

CMAD - Pitching moment coefficient due to acceleration in angle of attack (α)

Note: For body alone and body + fin set data CMQ and CMAD are presented as the sum CMQ+CMAD.

The dynamic derivatives are printed after all static coefficients and partial static aerodynamics are printed. If a build or part card is input, additional dynamic derivatives for partial configurations and/or configuration components are printed.

4.4 External Data Files

The code has the capability to be used in conjunction with other missile design tools, such as post-processing plotting programs or trajectory programs. Fixed format aerodynamic data is output as an external data file with the addition of the PLOT control card. Included in this data file are the six component forces and moments based upon the user specified reference quantities. In order to print component buildup data to the plot file the BUILD and PLOT control cards must be present in the case. Refer to Volume 2 for a complete description of the plot file output format.

An option to create a user specified format data file is also available. The control cards WRITE and FORMAT have been designed for easy access to this capability.

4.5 Extrapolation Messages and Array Dumps

As shown in Figure 60, the extrapolation messages are summarized for all design charts which have been extrapolated during the execution of the case. Since many of the aerodynamic methods do not include design charts, but are either closed-form equations or complete theoretical methods, this option is most useful in the subsonic and transonic Mach regimes. Extrapolation messages are only provided if the control card PRINTE EXTRAP appears in the case inputs. The data titled "ROUTINE TRACE-BACK" lists the decks called when the look-up was performed; "X" represents the independent variable and "Y" represents the dependent variable in the extrapolation.

When it is necessary to examine the values stored in internal data arrays the DUMP control card can be used. This control card causes the contents of the named data arrays to be printed in a form similar to Figure 61. Array dumps are provided for each Mach number of the input case, and represent the data block contents at aerodynamic calculation completion. Volume 2 of this User's Guide details the arrays and their contents.

Note that all data arrays are initialized to a constant named "UNUSED", which is preset to a value of 1×10^{-30}. Hence, any array element which contains this constant was not changed during execution of the case (since it is highly unlikely that this constant will result from any calculation). This scheme permits rapid "tracking" of program calculation sequences while in "debug" mode.

4.6 Special Program Capabilities

The code is currently being used both as a stand-alone aerodynamic prediction code and as a subroutine package to a larger missile design code. Since all of the printed output is controlled by print flags, some of which are not directly user selectable, the code can be used as part of a larger system relatively easily. Users desiring to utilize this capability should contact the procuring agency for instructions.

On computer systems which do not utilize the namelist input scheme, a fixed-format input mode can be easily treated. The procuring agency can provide advice on this option when necessary.

```
                              THE USAF AUTOMATED MISSILE DATCOM * REV 8/85 *
                              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
                                     CONERR - INPUT ERROR CHECKING

ERROR CODES - N* DENOTES THE NUMBER OF OCCURENCES OF EACH ERROR

A - UNKNOWN VARIABLE NAME

B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME

C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N).

D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED

E - ASSIGNED VALUES EXCEED ARRAY DIMENSION

F - SYNTAX ERROR

   ****************************  INPUT DATA CARDS  ****************************

 1 *
 2 * INPUT ERROR CHECKING TEST CASES
 3 *
 4 CASEID CONERR ERROR CHECKING TEST CASE
 5                                                    ** BLANK CARD - IGNORED
 6  $FLTCON NMCH=1.,$                                 ** ERROR **  1*A  0*B  0*C  0*D  0*E  0*F
                                                      ** FATAL ERROR **
 7  $REFQ SREF 1.,$                                   ** ERROR **  0*A  1*B  0*C  C*D  0*E  0*F
                                                      ** FATAL ERROR **
 8  $REFQ ROUGH(2)=0.,$                               ** ERROR **  0*A  0*B  1*C  0*D  0*E  0*F
                                                      ** FATAL ERROR **
 9  $REFQ LATREF=1.,1.,$                              ** ERROR **  0*A  0*B  0*C  1*D  0*E  0*F
                                                      ** FATAL ERROR **
10  $FLTCON MACH(21)=0.6,                             ** ERROR **  0*A  0*B  0*C  0*D  1*E  0*F
                                                      ** FATAL ERROR **
11  $END
12  $FLTCON NMACH=1,$                                 ** ERROR **  0*A  0*B  0*C  0*D  0*E  1*F
13 BUILT                                              ** ERROR ** UNKNOWN CONTROL CARD - IGNORED
14  $AXIBOD TNOSE=CONE,$                              ** SUBSTITUTING NUMERIC FOR NAME CONE
15  $AXIBD LNOSE=1.,$                                 ** ERROR ** UNKNOWN NAMELIST NAME
16  $AXIBOD TNOSE=CONE,TAFT=OGIVE,$                   ** SUBSTITUTING NUMERIC FOR NAME CONE
                                                      ** SUBSTITUTING NUMERIC FOR NAME OGIVE
17 DUMP SSS                                           ** ERROR **   1 INCORRECT ARRAY NAMES
18  $                                                 ** ERROR ** UNKNOWN NAMELIST NAME
19 NEXT CASE

FATAL ERROR ENCOUNTERED IN CONERR.  EXECUTION TERMINATED.
```

Figure 33 Output from Input Error Checking

```
                                       THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
                                       AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
                                             CONERR - INPUT ERROR CHECKING

ERROR CODES - N* DENOTES THE NUMBER OF OCCURENCES OF EACH ERROR

A - UNKNOWN VARIABLE NAME

B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME

C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)

D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED

E - ASSIGNED VALUES EXCEED ARRAY DIMENSION

F - SYNTAX ERROR

   ****************************  INPUT DATA CARDS  ****************************

 1 CASEID EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
 2 SOSE
 3 DIM IN
 4 NO LAT
 5  $FLTCON NMACH=1.,MACH=2.36,REN=3.E6,
 6          NALPHA=8.,ALPHA=0.,4.,8.,12.,16.,20.,24.,28.,$
 7  $REFQ XCG=18.75,$
 8  $AXIBOD LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,$
 9  $FINSET1 CHORD=6.96,0.,SSPAN=1.875,5.355,XLE=15.42,
10           SWEEP=0.,STA=1.,ZUPPER=2*0.02238,NPANEL=4.,
11           LMAXU=0.288,LER=2*.015,LFLATU=0.428,FINPHI=0.,$
12  $FINSET2 CHORD=5.585,2.792,SSPAN=1.875,6.260,XLE=31.915,
13           SWEEP=0.,STA=1.,ZUPPER=2*0.02238,NPANEL=4.,
14           LMAXU=0.288,LER=2*.015,LFLATU=0.428,FINPHI=0.,$
15 PART
16 BUILD
17 SAVE
18 NEXT CASE
```

Figure 34 Case Input Listing

```
                    THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *                      CASE  1
                    AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS                       PAGE  1
                                     CASE INPUTS
FOLLOWING ARE THE CARDS INPUT FOR THIS CASE

CASEID EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
SOSE
DIM IN
NO LAT
 $FLTCON NMACH=1.,MACH=2.36,REN=3.E6,
         NALPHA=8.,ALPHA=0.,4.,8.,12.,16.,20.,24.,28.,$
 $REFQ XCG=18.75,$
 $AXIBOD LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,$
 $FINSET1 CHORD=6.96,0.,SSPAN=1.875,5.355,XLE=15.42,
         SWEEP=0.,STA=1.,ZUPPER=2*0.02238,NPANEL=4.,
         LMAXU=0.288,LER=2*.015,LFLATU=0.428,FINPHI=0.,$
 $FINSET2 CHORD=5.585,2.792,SSPAN=1.875,6.260,XLE=31.915,
         SWEEP=0.,STA=1.,ZUPPER=2*0.02238,NPANEL=4.,
         LMAXU=0.288,LER=2*.015,LFLATU=0.428,FINPHI=0.,$
PART
BUILD
SAVE
NEXT CASE

* WARNING * THE REFERENCE AREA IS UNSPECIFIED, DEFAULT VALUE ASSUMED

* WARNING * THE REFERENCE LENGTH IS UNSPECIFIED, DEFAULT VALUE ASSUMED

* WARNING * A CENTER SECTION IS DEFINED BUT THE BASE DIAMETER IS NOT INPUT. CYLINDRICAL SECTION ASSUMED.

THE BOUNDARY LAYER IS ASSUMED TO BE TURBULENT OVER ALL COMPONENTS OF THE CONFIGURATION

THE INPUT UNITS ARE IN INCHES, THE SCALE FACTOR IS   1.0000
```

Figure 35 Example of Default Substitutions for Incomplete Case Inputs

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

CASE 1
PAGE 20

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/IN**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA IN**2	REF. LENGTH LONG. IN	REF. LENGTH LAT. IN	MOMENT REF. CENTER LONG. IN	MOMENT REF. CENTER VERTICAL IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

	LONGITUDINAL			LATERAL DIRECTIONAL			DERIVATIVES (PER DEGREE) LONGITUDINAL		DERIVATIVES (PER DEGREE) LATERAL DIRECTIONAL		
ALPHA	CN	CM	CA	CY	CLN	CLL	CNA	CMA	CYB	CLNB	CLLB
0.00	0.000	0.000	0.367	0.000	0.000	0.000	2.588E-01	-3.424E-01			
4.00	1.124	-1.472	0.368	0.000	0.000	0.000	3.034E-01	-3.934E-01			
8.00	2.428	-3.149	0.368	0.000	0.000	0.000	3.717E-01	-4.804E-01			
12.00	4.104	-5.327	0.368	0.000	0.000	0.000	4.317E-01	-5.783E-01			
16.00	5.882	-7.780	0.369	0.000	0.000	0.000	4.414E-01	-6.214E-01			
20.00	7.635	-10.298	0.370	0.000	0.000	0.000	4.194E-01	-6.175E-01			
24.00	9.239	-12.720	0.371	0.000	0.000	0.000	4.035E-01	-5.909E-01			
28.00	10.863	-15.026	0.372	0.000	0.000	0.000	4.085E-01	-5.624E-01			

ALPHA	CL	CD	CL/CD	X-C.P.
0.00	0.000	0.367	0.000	-1.323
4.00	1.096	0.445	2.462	-1.309
8.00	2.353	0.702	3.352	-1.297
12.00	3.937	1.213	3.245	-1.298
16.00	5.553	1.976	2.810	-1.323
20.00	7.048	2.959	2.382	-1.349
24.00	8.289	4.097	2.023	-1.377
28.00	9.416	5.429	1.735	-1.383

PANEL DEFLECTION ANGLES (DEGREES)

FIN SET	FIN 1	FIN 2	FIN 3	FIN 4
1	0.00	0.00	0.00	0.00
2	0.00	0.00	0.00	0.00

LINEAR DATA FOR BODY ALONE WAS GENERATED USING THE SECOND-ORDER SHOCK EXPANSION METHOD

Figure 36 Total Configuration Aerodynamic Output Summary

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
TRIMMED STATIC AERODYNAMIC COEFFICIENTS

CASE 1
PAGE 8

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
0.00					1.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

ALPHA	DELTA	CN	CA	CY	CLN	CLL
0.00	0.0000	0.0000	0.2690	0.0000	0.0000	0.0000
8.00	15.9550	4.0599	0.8585	0.0000	0.0000	0.0000
16.00	*NT*	*NT*	*NT*	*NT*	*NT*	*NT*

PANELS FROM FIN SET 1 WERE DEFLECTED OVER THE RANGE -25.0000 TO 20.0000 DEG.
PANEL 1 WAS FIXED
PANEL 2 WAS VARIED
PANEL 3 WAS FIXED
PANEL 4 WAS VARIED

NOTE - *NT* PRINTED WHEN NO TRIM POINT COULD BE FOUND

Figure 37 Trimmed Output Summary

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
AXISYMMETRIC BODY DEFINITION

CASE 1
PAGE 2

	NOSE	CENTERBODY	AFT BODY	TOTAL	
SHAPE	OGIVE	CYLINDER	———		
LENGTH	11.2500	26.2500	0.0000	37.5000	IN
FINENESS RATIO	3.0000	7.0000	0.0000	10.0000	
PLANFORM AREA	28.2799	98.4376	0.0000	126.7175	IN**2
AREA CENTROID	7.0157	24.3750	0.0000	20.5008	IN FROM NOSE TIP
WETTED AREA	89.8180	309.2500	0.0000	399.0687	IN**2
VOLUME	66.7887	289.9221	0.0000	356.7109	IN**3
VOLUME CENTROID	7.7135	24.3750	0.0000	21.2554	IN FROM NOSE TIP

MOLD LINE CONTOUR

LONGITUDINAL STATIONS	0.0000	1.1250	2.2500	3.3750	4.5000	5.6250	6.7500	7.8750	9.0000	10.1250
	11.2500	13.8750	16.5000	19.1250	21.7500	24.3750	27.0000	29.6250	32.2500	34.8750
	37.5000*									
BODY RADII	0.0000	0.3644	0.6871	0.9693	1.2119	1.4159	1.5819	1.7104	1.8020	1.8568
	1.8750	1.8750	1.8750	1.8750	1.8750	1.8750	1.8750	1.8750	1.8750	1.8750
	1.8750*									

NOTE - * INDICATES SLOPE DISCONTINUOUS POINTS

Figure 38 Body Geometry Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
FIN SET NUMBER 1 AIRFOIL SECTION

CASE 1
PAGE 3

NACA S-3-35.9-04.5-28.5

UPPER ABSCISSA	UPPER ORDINATE	LOWER ABSCISSA	LOWER ORDINATE	X-FRACTION CHORD	MEAN LINE	THICKNESS
0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
0.00100	0.00006	0.00100	-0.00006	0.00100	0.00000	0.00013
0.00200	0.00013	0.00200	-0.00013	0.00200	0.00000	0.00025
0.00300	0.00019	0.00300	-0.00019	0.00300	0.00000	0.00038
0.00400	0.00025	0.00400	-0.00025	0.00400	0.00000	0.00050
0.00500	0.00031	0.00500	-0.00031	0.00500	0.00000	0.00063
0.00600	0.00038	0.00600	-0.00038	0.00600	0.00000	0.00075
0.00800	0.00050	0.00800	-0.00050	0.00800	0.00000	0.00100
0.01000	0.00063	0.01000	-0.00063	0.01000	0.00000	0.00125
0.02000	0.00125	0.02000	-0.00125	0.02000	0.00000	0.00251
0.03000	0.00188	0.03000	-0.00188	0.03000	0.00000	0.00376
0.04000	0.00251	0.04000	-0.00251	0.04000	0.00000	0.00501
0.05000	0.00313	0.05000	-0.00313	0.05000	0.00000	0.00627
0.06000	0.00376	0.06000	-0.00376	0.06000	0.00000	0.00752
0.08000	0.00501	0.08000	-0.00501	0.08000	0.00000	0.01003
0.10000	0.00627	0.10000	-0.00627	0.10000	0.00000	0.01253
0.12000	0.00752	0.12000	-0.00752	0.12000	0.00000	0.01504
0.14000	0.00877	0.14000	-0.00877	0.14000	0.00000	0.01755
0.16000	0.01003	0.16000	-0.01003	0.16000	0.00000	0.02006
0.18000	0.01128	0.18000	-0.01128	0.18000	0.00000	0.02256
0.20000	0.01253	0.20000	-0.01253	0.20000	0.00000	0.02507
0.22000	0.01379	0.22000	-0.01379	0.22000	0.00000	0.02758
0.24000	0.01504	0.24000	-0.01504	0.24000	0.00000	0.03008
0.26000	0.01630	0.26000	-0.01630	0.26000	0.00000	0.03259
0.28000	0.01755	0.28000	-0.01755	0.28000	0.00000	0.03510
0.30000	0.01880	0.30000	-0.01880	0.30000	0.00000	0.03760
0.32000	0.02006	0.32000	-0.02006	0.32000	0.00000	0.04011
0.34000	0.02131	0.34000	-0.02131	0.34000	0.00000	0.04262
0.36000	0.02250	0.36000	-0.02250	0.36000	0.00000	0.04500
0.38000	0.02250	0.38000	-0.02250	0.38000	0.00000	0.04500
0.40000	0.02250	0.40000	-0.02250	0.40000	0.00000	0.04500
0.42000	0.02250	0.42000	-0.02250	0.42000	0.00000	0.04500
0.45000	0.02250	0.45000	-0.02250	0.45000	0.00000	0.04500
0.50000	0.02250	0.50000	-0.02250	0.50000	0.00000	0.04500
0.55000	0.02250	0.55000	-0.02250	0.55000	0.00000	0.04500
0.60000	0.02250	0.60000	-0.02250	0.60000	0.00000	0.04500
0.65000	0.02212	0.65000	-0.02212	0.65000	0.00000	0.04424
0.70000	0.01896	0.70000	-0.01896	0.70000	0.00000	0.03792
0.75000	0.01580	0.75000	-0.01580	0.75000	0.00000	0.03160
0.80000	0.01264	0.80000	-0.01264	0.80000	0.00000	0.02528
0.82000	0.01138	0.82000	-0.01138	0.82000	0.00000	0.02275
0.84000	0.01011	0.84000	-0.01011	0.84000	0.00000	0.02022
0.86000	0.00885	0.86000	-0.00885	0.86000	0.00000	0.01770
0.88000	0.00758	0.88000	-0.00758	0.88000	0.00000	0.01517
0.90000	0.00632	0.90000	-0.00632	0.90000	0.00000	0.01264
0.92000	0.00506	0.92000	-0.00506	0.92000	0.00000	0.01011
0.94000	0.00379	0.94000	-0.00379	0.94000	0.00000	0.00758
0.96000	0.00253	0.96000	-0.00253	0.96000	0.00000	0.00506
0.98000	0.00126	0.98000	-0.00126	0.98000	0.00000	0.00253
1.00000	0.00000	1.00000	0.00000	1.00000	0.00000	0.00000

Figure 39 Airfoil Geometry Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
GEOMETRIC RESULTS FOR FIN SETS

CASE 1
PAGE 5

FIN SET NUMBER 1
(DATA FOR ONE PANEL ONLY)

SEGMENT NUMBER	PLANFORM AREA IN**2	TAPER RATIO	ASPECT RATIO	LEADING EDGE SWEEP DEG	TRAILING EDGE SWEEP DEG	MEAN AERODYNAMIC CHORD IN	LEADING M.A.C. POSITION IN	LATERAL M.A.C. POSITION IN	THICKNESS TO CHORD RATIO
1	12.11040	0.0000	1.00000	63.43495	0.00000	4.6400	2.3200	3.0350	0.04500
TOTAL	12.11040	0.0000	1.00000	63.43495	0.00000	4.6400	2.3200	3.0350	0.04500

FIN SET NUMBER 2
(DATA FOR ONE PANEL ONLY)

SEGMENT NUMBER	PLANFORM AREA IN**2	TAPER RATIO	ASPECT RATIO	LEADING EDGE SWEEP DEG	TRAILING EDGE SWEEP DEG	MEAN AERODYNAMIC CHORD IN	LEADING M.A.C. POSITION IN	LATERAL M.A.C. POSITION IN	THICKNESS TO CHORD RATIO
1	18.36658	0.4999	1.04001	32.40486	0.00000	4.3437	1.2413	3.8238	0.04500
TOTAL	18.36658	0.4999	1.04001	32.40486	0.00000	4.3437	1.2413	3.8238	0.04500

Figure 40 Fin Geometry Output

THE USAF AUTOMATED MISSILE DATCOM * REV 8/86 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
TOP ATTACHED TWO-DIMENSIONAL INLET ON NASA-TESTED BODY
INLET GEOMETRY

INLET TYPE IS TWO-DIMENSIONAL

INLET BOATTAIL TYPE IS AXISYMMETRIC

NUMBER OF INLETS = 1

INLET ANGULAR POSITION 90.000 0.000 DEG. FROM HORIZONTAL

LONGITUDINAL POSITION OF INLET COORDINATE SYSTEM = 0.000E+00

LATERAL POSITION OF INLET COORDINATE SYSTEM = 1.000E-30

STATION	AXIAL STATION	HEIGHT	DIA. OR SPAN
1	2.98450E+01	0.00000E+00	5.84200E+00
2	3.17500E+01	3.55000E-01	5.84200E+00
3	7.55650E+01	0.00000E+00	0.00000E+00
4	1.06680E+02	0.00000E+00	1.27000E+00
5	4.09900E+01	-4.64500E+00	0.00000E+00
6	4.70000E+01	-5.14500E+00	0.00000E+00
7	5.80000E+01	-4.84500E+00	0.00000E+00
8	0.00000E+00	0.00000E+00	0.00000E+00
9	3.17500E+01	0.00000E+00	0.00000E+00
10	4.40000E+01	0.00000E+00	0.00000E+00

*** DIVERTER GEOMETRY ***

DIVERTER IS TOP MOUNTED

THE ZBOD POSITION OF DIVERTER = 1.0000E-30

DIVERTER DEFINITION

X	HEIGHT	THICKNESS
3.17500E+01	1.00000E-30	4.32000E-01
4.83160E+01	1.00000E-30	1.00000E-30
1.00000E-30	1.00000E-30	1.00000E-30
1.00000E-30	1.00000E-30	1.00000E-30

Figure 41 Inlet Geometry Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
BODY ALONE PARTIAL OUTPUT

CASE 1
PAGE 6

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
2.30					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

ALPHA	CA-FRICTION	CA-PRESSURE/WAVE	CA-BASE	CA-ALPHA
0.000	0.08447	0.10279	0.12540	0.00000
4.000	0.08459	0.10277	0.12540	0.00000
8.000	0.08493	0.10271	0.12540	0.00000
12.000	0.08551	0.10262	0.12540	0.00000
16.000	0.08633	0.10250	0.12540	0.00000
20.000	0.08740	0.10234	0.12540	0.00000
24.000	0.08871	0.10215	0.12540	0.00000
28.000	0.09028	0.10194	0.12540	0.00000

NOTE - THE BASE DRAG INCREMENT IS NOT INCLUDED IN THE AXIAL FORCE CALCULATIONS

CROSS FLOW DRAG PROPORTIONALITY FACTOR = 1.00000

ALPHA	CN-POTENTIAL	CN-VISCOUS	CM-POTENTIAL	CM-VISCOUS	CDC
0.000	0.0000	0.0000	0.0000	0.0000	0.2800
4.000	0.2220	0.0248	0.5821	-0.0116	0.4446
8.000	0.4390	0.1809	1.1508	-0.0845	0.8140
12.000	0.6458	0.6578	1.6930	-0.3071	1.3264
16.000	0.8377	1.3075	2.1963	-0.6105	1.5000
20.000	1.0106	2.0045	2.6494	-0.9359	1.4935
24.000	1.1604	2.6081	3.0423	-1.2177	1.3741
28.000	1.2842	3.3290	3.3667	-1.5547	1.3168

Figure 42 Body Alone Aerodynamic Partial Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
FIN SET 1 CN PARTIAL OUTPUT

CASE 1
PAGE 7

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

POTENTIAL NORMAL FORCE SLOPE AT ALPHA ZERO (1 PANEL), CNA = 0.03186/DEG

PANEL NO.	ALPHA TOTAL DEG	FIN PHI DEG	ALPHA EQUIV DEG	CNAA	CN POTENTIAL	CN VISCOUS	CN TOTAL
1	0.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	0.000	90.000	0.000	1.75439	0.00000	0.00000	0.00000
3	0.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	0.000	270.000	0.000	1.75439	0.00000	0.00000	0.00000
SET					0.00000	0.00000	0.00000
1	4.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	4.000	90.000	4.000	1.56610	0.12703	0.00762	0.13465
3	4.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	4.000	270.000	4.000	1.56610	0.12703	0.00762	0.13465
SET					0.25406	0.01524	0.26930
1	8.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	8.000	90.000	8.000	1.38431	0.25159	0.02681	0.27840
3	8.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	8.000	270.000	8.000	1.38431	0.25159	0.02681	0.27840
SET					0.50318	0.05363	0.55680
1	12.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	12.000	90.000	12.000	1.21073	0.37125	0.05234	0.42359
3	12.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	12.000	270.000	12.000	1.21073	0.37125	0.05234	0.42359
SET					0.74250	0.10467	0.84717
1	16.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	16.000	90.000	16.000	1.06690	0.48368	0.08106	0.56474
3	16.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	16.000	270.000	16.000	1.06690	0.48368	0.08106	0.56474
SET					0.96737	0.16212	1.12949
1	20.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	20.000	90.000	20.000	0.93611	0.58670	0.10950	0.69621
3	20.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	20.000	270.000	20.000	0.93611	0.58670	0.10950	0.69621
SET					1.17341	0.21901	1.39242
1	24.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	24.000	90.000	24.000	0.82120	0.67831	0.13586	0.81416
3	24.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	24.000	270.000	24.000	0.82120	0.67831	0.13586	0.81416
SET					1.35661	0.27171	1.62832
1	28.000	0.000	0.000	1.75439	0.00000	0.00000	0.00000
2	28.000	90.000	28.000	0.73626	0.75670	0.16227	0.91898
3	28.000	180.000	0.000	1.75439	0.00000	0.00000	0.00000
4	28.000	270.000	28.000	0.73626	0.75670	0.16227	0.91898
SET					1.51341	0.32456	1.83796

Figure 43 Fin Normal Force Partial Output

CASE 1
PAGE 8

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
FIN SET 1 CA PARTIAL OUTPUT

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS

SKIN FRICTION	0.00755
SUBSONIC PRESSURE	0.00000
TRANSONIC WAVE	0.00000
SUPERSONIC WAVE	0.00598
LEADING EDGE	0.00108
TRAILING EDGE	0.00000
TOTAL CAO	0.01461

FIN AXIAL FORCE DUE TO ANGLE OF ATTACK

ALPHA DEG	CA DUE TO ALPHA	CA-TOTAL (4 FINS)
0.000	0.00000	0.05845
4.000	0.00000	0.05845
8.000	0.00000	0.05845
12.000	0.00000	0.05845
16.000	0.00000	0.05845
20.000	0.00000	0.05845
24.000	0.00000	0.05845
28.000	0.00000	0.05845

Figure 44 Fin Axial Force Partial Output

CASE 1
PAGE 9

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
FIN SET 1 CM PARTIAL OUTPUT

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/IN**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA IN**2	REF. LENGTH LONG. IN	REF. LENGTH LAT. IN	MOMENT REF. CENTER LONG. IN	MOMENT REF. CENTER VERTICAL IN
2.30					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

CENTER OF PRESSURE FOR LINEAR CN = -0.35552 (CALIBERS FROM C.G.)
CENTER OF PRESSURE FOR NON-LINEAR CN = -0.34933 (CALIBERS FROM C.G.)

ALPHA DEG.	CM LINEAR	CM NON-LINEAR	CM TOTAL
0.00000	0.00000	0.00000	0.00000
4.00000	-0.09032	-0.00532	-0.09565
8.00000	-0.17889	-0.01873	-0.19762
12.00000	-0.26397	-0.03657	-0.30054
16.00000	-0.34392	-0.05663	-0.40055
20.00000	-0.41717	-0.07651	-0.49368
24.00000	-0.48230	-0.09492	-0.57722
28.00000	-0.53806	-0.11338	-0.65142

Figure 45 Fin Pitching Moment Partial Output

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                 THE USAF AUTOMATED MISSILE DATCOM * REV 8/85 *
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
               EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
                        FIN SET 1 SECTION AERODYNAMICS

               IDEAL ANGLE OF ATTACK =   0.00000 DEG.

           ZERO LIFT ANGLE OF ATTACK =   0.00000 DEG.

             IDEAL LIFT COEFFICIENT =   0.00000

 ZERO LIFT PITCHING MOMENT COEFFICIENT =   0.00000

          MACH ZERO LIFT-CURVE-SLOPE =   0.09275 /DEG.

                 LEADING EDGE RADIUS =   0.00323 FRACTION CHORD

          MAXIMUM AIRFOIL THICKNESS =   0.04500 FRACTION CHORD

                            DELTA-Y =   0.36684 PERCENT CHORD

     MACH= 0.0000 LIFT-CURVE-SLOPE =   0.11536 /DEG.      XAC =   0.28382
```

Figure 46 Airfoil Section Aerodynamics Partial Output

CASE 1
PAGE 5

THE USAF AUTOMATED MISSILE DATCOM * REV 8/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
TOP ATTACHED TWO-DIMENSIONAL INLET ON NASA-TESTED BODY
INLET AERODYNAMIC INCREMENTALS

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/FT**2	DEG R	1/FT	DEG	DEG	FT**2	FT	FT	FT	FT
2.50					6.560E+06	0.00	0.00	45.604	7.620	7.620	53.340	0.000

ALPHA	CN-INLET	CM-INLET	CA-INLET
0.00	-0.2049	-1.6747	0.1243
4.00	-0.0776	-1.3963	0.1099
8.00	0.0577	-1.1681	0.0966
12.00	0.1806	-0.6951	0.0844
16.00	0.3224	-0.1149	0.0703
20.00	0.4567	0.4308	0.0526

Figure 47 Inlet Aerodynamic Partial Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
BODY ALONE STATIC AERODYNAMIC CHARACTERISTICS

CASE 1
PAGE 22

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/IN**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA IN**2	REF. LENGTH LONG. IN	REF. LENGTH LAT. IN	MOMENT REF. CENTER LONG. IN	MOMENT REF. CENTER VERTICAL IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

	LONGITUDINAL			LATERAL DIRECTIONAL			DERIVATIVES (PER DEGREE) LONGITUDINAL		LATERAL DIRECTIONAL		
ALPHA	CN	CM	CA	CY	CLN	CLL	CNA	CMA	CYB	CLNB	CLLB
0.00	0.000	0.000	0.187	0.000	0.000	0.000	4.597E-02	1.520E-01	-4.991E-02	-1.496E-01	0.000E+00
4.00	0.247	0.571	0.187	0.000	0.000	0.000	7.746E-02	1.333E-01	-6.208E-02	-1.421E-01	0.000E+00
8.00	0.620	1.066	0.188	0.000	0.000	0.000	1.319E-01	1.019E-01	-7.739E-02	-1.322E-01	0.000E+00
12.00	1.304	1.386	0.188	0.000	0.000	0.000	1.906E-01	6.492E-02	-1.073E-01	-1.136E-01	0.000E+00
16.00	2.145	1.586	0.189	0.000	0.000	0.000	2.139E-01	4.095E-02	-1.307E-01	-9.642E-02	0.000E+00
20.00	3.015	1.713	0.190	0.000	0.000	0.000	2.029E-01	2.985E-02	-1.446E-01	-8.211E-02	0.000E+00
24.00	3.769	1.825	0.191	0.000	0.000	0.000	1.998E-01	1.231E-02	-1.478E-01	-7.149E-02	0.000E+00
28.00	4.614	1.812	0.192	0.000	0.000	0.000	2.229E-01	-1.862E-02	-1.515E-01	-5.944E-02	0.000E+00

ALPHA	CL	CD	CL/CD	X-C.P.
0.00	0.000	0.187	0.000	3.306
4.00	0.233	0.204	1.142	2.311
8.00	0.588	0.272	2.160	1.720
12.00	1.236	0.455	2.716	1.063
16.00	2.010	0.773	2.601	0.739
20.00	2.768	1.210	2.289	0.568
24.00	3.365	1.707	1.971	0.484
28.00	3.984	2.336	1.705	0.393

LINEAR DATA FOR BODY ALONE WAS GENERATED USING THE SECOND-ORDER SHOCK EXPANSION METHOD

Figure 48 Body Alone Synthesis Partial Output

CASE 1
PAGE 13

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
AERODYNAMIC FORCE AND MOMENT SYNTHESIS

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/IN**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA IN**2	REF. LENGTH LONG. IN	REF. LENGTH LAT. IN	MOMENT REF. CENTER LONG. IN	MOMENT REF. CENTER VERTICAL IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

FIN SET 1 IN PRESENCE OF THE BODY

ANGLE OF ATTACK, DEG.	CN	CM	CA	CY	CLN	CLL
0.0000	0.0000	0.0000	0.0584	0.0000	0.0000	0.0000
4.0000	0.3541	-0.1250	0.0584	0.0000	0.0000	0.0000
8.0000	0.7285	-0.2500	0.0584	0.0000	0.0000	0.0000
12.0000	1.0019	-0.3882	0.0584	0.0000	0.0000	0.0000
16.0000	1.3782	-0.4900	0.0584	0.0000	0.0000	0.0000
20.0000	1.6192	-0.5756	0.0584	0.0000	0.0000	0.0000
24.0000	1.8547	-0.6594	0.0584	0.0000	0.0000	0.0000
28.0000	2.0750	-0.7377	0.0584	0.0000	0.0000	0.0000

PANEL CHARACTERISTICS

PANEL NUMBER	ALPHA EQUIV, DEG.	CN
1	0.0000	0.0000
2	0.0000	0.0000
3	0.0000	0.0000
4	0.0000	0.0000
1	0.0000	0.0000
2	5.1963	0.1770
3	0.0000	0.0000
4	-5.1963	-0.1770
1	0.0000	0.0000
2	10.3582	0.3642
3	0.0000	0.0000
4	-10.3582	-0.3642
1	0.0000	0.0000
2	15.4529	0.5459
3	0.0000	0.0000
4	-15.4529	-0.5459
1	0.0000	0.0000
2	19.7740	0.6891
3	0.0000	0.0000
4	-19.7740	-0.6891
1	0.0000	0.0000
2	23.8356	0.8096
3	0.0000	0.0000
4	-23.8356	-0.8096
1	0.0000	0.0000
2	28.3397	0.9274
3	0.0000	0.0000
4	-28.3397	-0.9274
1	0.0000	0.0000
2	32.7188	1.0375
3	0.0000	0.0000
4	-32.7188	-1.0375

Figure 49 Fin Set in Presence of the Body Partial Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/86 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
AERODYNAMIC FORCE AND MOMENT SYNTHESIS

CASE 1
PAGE 15

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

SYNTHESIS AERODYNAMICS FOR BODY-FIN SET 1

ANGLE OF ATTACK, DEG.	CN	CM	CA	CY	CLN	CLL
0.0000	0.0000	0.0000	0.2457	0.0000	0.0000	0.0000
4.0000	0.7177	0.3174	0.2458	0.0000	0.0000	0.0000
8.0000	1.5885	0.5455	0.2461	0.0000	0.0000	0.0000
12.0000	2.7555	0.6052	0.2466	0.0000	0.0000	0.0000
16.0000	3.9779	0.6004	0.2473	0.0000	0.0000	0.0000
20.0000	5.1681	0.5558	0.2482	0.0000	0.0000	0.0000
24.0000	6.2347	0.4986	0.2493	0.0000	0.0000	0.0000
28.0000	7.3731	0.3285	0.2507	0.0000	0.0000	0.0000

Figure 50 Body Plus Fin Set Partial Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
AERODYNAMIC FORCE AND MOMENT SYNTHESIS

CASE 1
PAGE 16

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

SYNTHESIS AERODYNAMICS FOR BODY-FIN SET 1 AND 2

ANGLE OF ATTACK, DEG.	CN	CM	CA	CY	CLN	CLL
0.0000	0.0000	0.0000	0.3674	0.0000	0.0000	0.0000
4.0000	1.1243	-1.4717	0.3675	0.0000	0.0000	0.0000
8.0000	2.4282	-3.1491	0.3678	0.0000	0.0000	0.0000
12.0000	4.1036	-5.3268	0.3683	0.0000	0.0000	0.0000
16.0000	5.8824	-7.7796	0.3690	0.0000	0.0000	0.0000
20.0000	7.6349	-10.2964	0.3699	0.0000	0.0000	0.0000
24.0000	9.2386	-12.7199	0.3710	0.0000	0.0000	0.0000
28.0000	10.8628	-15.0265	0.3724	0.0000	0.0000	0.0000

Figure 51 Body Plus Two Fin Sets Partial Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
AERODYNAMIC FORCE AND MOMENT SYNTHESIS

CASE 1
PAGE 17

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
2.30					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

CARRYOVER INTERFERENCE FACTORS

FIN SET	K-W(B)	K-B(W)	KK-W(B)	KK-B(W)	XCP-W(B)	XCP-B(W)
1	1.300543E+00	4.287623E-01	9.347357E-01	3.658075E-01	3.555199E-01	1.090307E+00
2	1.252313E+00	1.126450E-01	9.358542E-01	3.164502E-01	4.300780E+00	4.503571E+00

PANEL CENTERS OF PRESSURE

FIN SET	X-CP	Y-CP/(B/2)
1	3.5552E-01	4.1740E-01
2	4.3008E+00	4.1801E-01

Figure 52 Carryover Interference Factors Partial Output

CASE 1
PAGE 18

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
FIN SET 1 PANEL BENDING MOMENTS (ABOUT EXPOSED ROOT CHORD)

	FLIGHT CONDITIONS							REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/IN**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA IN**2	REF. LENGTH LONG. IN	REF. LENGTH LAT. IN	MOMENT REF. CENTER LONG. IN	MOMENT REF. CENTER VERTICAL IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

ANGLE OF ATTACK, DEG.	PANEL 1	PANEL 2	PANEL 3	PANEL 4
0.0000	0.00000E+00	0.00000E+00	0.00000E+00	0.00000E+00
4.0000	0.00000E+00	6.85767E-02	1.04372E-06	-6.85767E-02
8.0000	0.00000E+00	1.41008E-01	2.20559E-06	-1.41008E-01
12.0000	0.00000E+00	2.11471E-01	3.49735E-06	-2.11471E-01
16.0000	-1.93078E-06	2.66924E-01	3.00203E-06	-2.66924E-01
20.0000	-3.65773E-06	3.13589E-01	6.52849E-06	-3.13589E-01
24.0000	-4.35311E-06	3.50210E-01	1.26578E-07	-3.50210E-01
28.0000	-7.75470E-06	4.01004E-01	1.54505E-07	-4.01004E-01

Figure 53 Panel Bending Moment Partial Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
FIN SET 1 PANEL HINGE MOMENTS (ABOUT HINGE LINE)

CASE 1
PAGE 20

	FLIGHT CONDITIONS							REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/IN**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA IN**2	REF. LENGTH LONG. IN	REF. LENGTH LAT. IN	MOMENT REF. CENTER LONG. IN	MOMENT REF. CENTER VERTICAL IN
2.30					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

ANGLE OF ATTACK, DEG.	PANEL 1	PANEL 2	PANEL 3	PANEL 4
0.0000	0.00000E+00	0.00000E+00	0.00000E+00	0.00000E+00
4.0000	0.00000E+00	-5.58607E-02	-8.50186E-09	5.58607E-02
8.0000	0.00000E+00	-1.14926E-01	-1.79661E-08	1.14926E-01
12.0000	0.00000E+00	-1.72258E-01	-2.84885E-08	1.72258E-01
16.0000	1.57276E-08	-2.17429E-01	-2.44537E-08	2.17429E-01
20.0000	2.97949E-08	-2.55441E-01	-5.31793E-08	2.55441E-01
24.0000	3.54592E-08	-2.92603E-01	-1.03107E-07	2.92603E-01
28.0000	6.31677E-08	-3.27347E-01	-1.25904E-07	3.27347E-01

Figure 54 Panel Hinge Moment Partial Ouptut

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
UNTRIMMED STATIC AERODYNAMIC COEFFICIENTS

CASE 1
PAGE 2

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE FT	VELOCITY FT/SEC	PRESSURE LB/IN**2	TEMPERATURE DEG R	REYNOLDS NUMBER 1/FT	SIDESLIP ANGLE DEG	ROLL ANGLE DEG	REF. AREA IN**2	REF. LENGTH LONG. IN	REF. LENGTH LAT. IN	MOMENT REF. CENTER LONG. IN	MOMENT REF. CENTER VERTICAL IN
0.60					1.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

TABLE OF UNTRIMMED NORMAL FORCE COEFFICIENTS

PANEL DEFLECTION ANGLE, DEG.

ALPHA	-25.0000	-20.0000	-15.0000	-10.0000	-5.0000	0.0000	5.0000	10.0000	15.0000	20.0000
0.00	-1.0653	-0.9540	-0.7642	-0.5250	-0.2621	0.0000	0.2621	0.5250	0.7642	0.9540
8.00	1.7768	2.2004	2.5928	2.9030	3.1780	3.4112	3.6712	3.8942	4.0495	4.1039
16.00	4.4598	5.0204	5.6504	6.3256	6.9756	7.5348	7.9343	8.1266	8.0797	7.6763

NOMINAL DEFLECTION ANGLES (DEGREES)

FIN SET	FIN 1	FIN 2	FIN 3	FIN 4
1	0.00	0.00	0.00	0.00
2	0.00	0.00	0.00	0.00

PANELS IN FIN SET 1 HAVE BEEN DEFLECTED

Figure 55 Untrimmed Partial Output

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
BODY ALONE PRESSURE OUTPUT

CASE 1
PAGE 3

MACH = 2.36 ANGLE OF ATTACK = 0.00 DEG.

PRESSURE COEFFICIENTS

X/DMAX	CP	M-LOCAL
0.000000	0.284442	1.867502
0.250000	0.245955	1.915320
0.500000	0.207539	1.966646
0.750000	0.172672	2.016886
1.000000	0.140477	2.066912
1.250000	0.110632	2.116958
1.500000	0.082994	2.167009
1.750000	0.057462	2.216970
2.000000	0.033921	2.266767
2.250000	0.012215	2.316413
2.500000	-0.007059	2.364015
2.750000	-0.027219	2.417976
3.000000	-0.035327	2.441060
3.250000	-0.036312	2.443921
3.500000	-0.033083	2.434587
3.750000	-0.027454	2.418635
4.000000	-0.022511	2.404955
4.250000	-0.019338	2.396327
4.500000	-0.016612	2.389009
4.750000	-0.014270	2.382789
5.000000	-0.012258	2.377495
5.250000	-0.010530	2.372982
5.500000	-0.009046	2.369132
5.750000	-0.007771	2.365842
6.000000	-0.006675	2.363030
6.250000	-0.005734	2.360625
6.500000	-0.004926	2.358566
6.750000	-0.004232	2.356802
7.000000	-0.003635	2.355291
7.250000	-0.003123	2.353996
7.500000	-0.002682	2.352885
7.750000	-0.002304	2.351933
8.000000	-0.001979	2.351116
8.250000	-0.001700	2.350415
8.500000	-0.001461	2.349814
8.750000	-0.001255	2.349298
9.000000	-0.001078	2.348854
9.250000	-0.000926	2.348474
9.500000	-0.000795	2.348147
9.750000	-0.000683	2.347867
10.000000	-0.000587	2.347626

Figure 56 Body Pressure Distribution from SOSE, $\alpha = 0^{0}$

CASE 1
PAGE 4

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
BODY ALONE PRESSURE OUTPUT

MACH = 2.36 ANGLE OF ATTACK = 4.00 DEG.

PRESSURE COEFFICIENTS

X/DMAX	PHI=0	PHI=30	PHI=60	PHI=90	PHI=120	PHI=150	PHI=180
0.000000	0.205017	0.213251	0.238086	0.277548	0.323402	0.361022	0.375649
0.250000	0.172947	0.180311	0.202796	0.239111	0.281890	0.317305	0.331133
0.500000	0.140746	0.147313	0.167598	0.200856	0.240519	0.273616	0.286588
0.750000	0.112104	0.117857	0.135912	0.166105	0.202684	0.233509	0.245647
1.000000	0.086064	0.091014	0.106863	0.134023	0.167542	0.196111	0.207421
1.250000	0.062291	0.066450	0.080129	0.104290	0.134774	0.161099	0.171582
1.500000	0.040606	0.043995	0.055551	0.076768	0.104257	0.128357	0.138018
1.750000	0.020869	0.023514	0.033012	0.051350	0.075910	0.097816	0.106665
2.000000	0.002918	0.004852	0.012375	0.027948	0.049639	0.069396	0.077448
2.250000	-0.013446	-0.012184	-0.006536	0.006384	0.025312	0.042978	0.050249
2.500000	-0.027749	-0.027105	-0.023191	-0.012746	0.003588	0.019278	0.025809
2.750000	-0.043149	-0.043069	-0.040768	-0.032005	-0.018930	-0.005244	0.000528
3.000000	-0.047487	-0.047905	-0.046969	-0.040776	-0.028908	-0.016623	-0.011367
3.250000	-0.045538	-0.046380	-0.046571	-0.041840	-0.031346	-0.020010	-0.015088
3.500000	-0.039890	-0.041110	-0.042280	-0.038755	-0.029313	-0.018651	-0.013956
3.750000	-0.032259	-0.033816	-0.035836	-0.033306	-0.024669	-0.014475	-0.009926
4.000000	-0.025604	-0.027451	-0.030200	-0.028522	-0.020568	-0.010768	-0.006340
4.250000	-0.020960	-0.023048	-0.026411	-0.025464	-0.018120	-0.008688	-0.004379
4.500000	-0.016949	-0.019246	-0.023144	-0.022837	-0.016028	-0.006921	-0.002717
4.750000	-0.013486	-0.015967	-0.020330	-0.020580	-0.014239	-0.005417	-0.001305
5.000000	-0.010536	-0.013170	-0.017924	-0.018642	-0.012690	-0.004104	-0.000068
5.250000	-0.008004	-0.010769	-0.015859	-0.016977	-0.011358	-0.002974	0.000997
5.500000	-0.005830	-0.008706	-0.014085	-0.015546	-0.010214	-0.002003	0.001912
5.750000	-0.003956	-0.006932	-0.012558	-0.014317	-0.009234	-0.001174	0.002693
6.000000	-0.002346	-0.005406	-0.011247	-0.013262	-0.008392	-0.000462	0.003363
6.250000	-0.000964	-0.004096	-0.010120	-0.012355	-0.007668	0.000150	0.003939
6.500000	0.000223	-0.002971	-0.009153	-0.011578	-0.007047	0.000677	0.004435
6.750000	0.001242	-0.002005	-0.008322	-0.010907	-0.006512	0.001129	0.004861
7.000000	0.002118	-0.001175	-0.007608	-0.010332	-0.006054	0.001518	0.005227
7.250000	0.002871	-0.000462	-0.006995	-0.009838	-0.005659	0.001851	0.005541
7.500000	0.003517	0.000151	-0.006468	-0.009414	-0.005321	0.002138	0.005811
7.750000	0.004072	0.000677	-0.006015	-0.009049	-0.005030	0.002384	0.006043
8.000000	0.004549	0.001129	-0.005627	-0.008736	-0.004780	0.002596	0.006242
8.250000	0.004959	0.001518	-0.005293	-0.008468	-0.004566	0.002777	0.006413
8.500000	0.005311	0.001851	-0.005006	-0.008237	-0.004381	0.002933	0.006560
8.750000	0.005613	0.002138	-0.004759	-0.008038	-0.004223	0.003067	0.006686
9.000000	0.005873	0.002384	-0.004548	-0.007868	-0.004087	0.003183	0.006795
9.250000	0.006096	0.002596	-0.004366	-0.007721	-0.003970	0.003282	0.006888
9.500000	0.006288	0.002777	-0.004210	-0.007595	-0.003870	0.003367	0.006968
9.750000	0.006452	0.002933	-0.004076	-0.007487	-0.003783	0.003440	0.007037
10.000000	0.006594	0.003068	-0.003960	-0.007305	-0.003709	0.003502	0.007096

NOTE - PHI= 0 IS TOP VERTICAL CENTER (LEEWARD)
PHI=180 IS BOTTOM VERTICAL CENTER (WINDWARD)

Figure 57 Body Pressure Distribution at Angle of Attack

CASE 1
PAGE 7

THE USAF AUTOMATED MISSILE DATCOM * REV 11/85 *
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
PRESSURE COEFFICIENTS ON FIN SET 1 AT MACH = 2.360

FOR ALPHA = 0.0

LOCAL CHORD = 0.5798 FT

Y/(B/2)	X/C	CP
0.0003	0.0000	0.3413
0.0003	0.0000	0.3310
0.0003	0.0001	0.3013
0.0003	0.0003	0.2559
0.0003	0.0005	0.2003
0.0003	0.0008	0.1410
0.0003	0.0011	0.0853
0.0003	0.0014	0.0399
0.0003	0.0023	0.0229
0.0003	0.0025	0.0200
0.0003	0.0026	0.0173
0.0003	0.0028	0.0148
0.0003	0.0029	0.0124
0.0003	0.0031	0.0103
0.0003	0.0032	0.0052
0.0003	0.0034	-0.0052
0.0003	0.0035	0.0302
0.0003	0.0037	0.0197
0.0003	0.0039	-0.0325
0.0003	0.0041	0.0018
0.0003	0.0041	0.0018
0.0003	0.0041	0.0018
0.0003	0.0041	0.0011
0.0003	0.0041	0.0025
0.0003	0.0325	0.0430
0.0003	0.0608	0.0430
0.0003	0.0892	0.0430
0.0003	0.1176	0.0430
0.0003	0.1460	0.0430
0.0003	0.1744	0.0430
0.0003	0.2028	0.0430
0.0003	0.2312	0.0430
0.0003	0.2596	0.0430
0.0003	0.2880	-0.0464
0.0003	0.2880	-0.0464
0.0003	0.3186	-0.0049
0.0003	0.3491	-0.0049
0.0003	0.3797	-0.0049
0.0003	0.4103	-0.0049
0.0003	0.4409	-0.0049
0.0003	0.4714	-0.0049
0.0003	0.5020	-0.0049
0.0003	0.5326	-0.0049
0.0003	0.5631	-0.0049
0.0003	0.5937	-0.0049
0.0003	0.6243	-0.0049
0.0003	0.6549	-0.0049
0.0003	0.6854	-0.0049
0.0003	0.7160	-0.0814
0.0003	0.7160	-0.0814
0.0003	0.7444	-0.0682
0.0003	0.7728	-0.0682
0.0003	0.8012	-0.0682
0.0003	0.8296	-0.0682
0.0003	0.8580	-0.0682
0.0003	0.8864	-0.0682
0.0003	0.9148	-0.0682
0.0003	0.9432	-0.0682
0.0003	0.9716	-0.0682
0.0003	1.0000	0.0055

Figure 58 Fin Pressure Distribution Output

THE USAF AUTOMATED MISSILE DATCOM • REV 11/85 •
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
BODY + 2 FIN SETS DYNAMIC DERIVATIVES

CASE 1
PAGE 3

FLIGHT CONDITIONS								REFERENCE DIMENSIONS				
MACH NUMBER	ALTITUDE	VELOCITY	PRESSURE	TEMPERATURE	REYNOLDS NUMBER	SIDESLIP ANGLE	ROLL ANGLE	REF. AREA	REF. LENGTH LONG.	REF. LENGTH LAT.	MOMENT REF. CENTER LONG.	MOMENT REF. CENTER VERTICAL
	FT	FT/SEC	LB/IN**2	DEG R	1/FT	DEG	DEG	IN**2	IN	IN	IN	IN
2.36					3.000E+06	0.00	0.00	11.045	3.750	3.750	18.750	0.000

DYNAMIC DERIVATIVES (PER DEGREE)			
ALPHA	CNQ	CNAD	CMQ+CMAD
0.0	5.584E-01	3.808E-01	-4.64119E+00
4.0	6.547E-01	4.465E-01	-5.33195E+00
8.0	8.022E-01	5.470E-01	-6.51044E+00
12.0	9.317E-01	6.353E-01	-7.83775E+00
16.0	9.877E-01	6.735E-01	-8.78341E+00

Figure 59 Dynamic Derivative Output

THE USAF AUTOMATED MISSILE DATCOM • REV 8/85 •
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
EXAMPLE PROBLEM - PLANAR WING PLUS TAIL CONFIGURATION
EXTRAPOLATION SUMMARY FOR INPUT MACH 2

CASE 1
PAGE 153

ROUTINE TRACE-BACK								LOWER BOUNDS	UPPER BOUNDS	FINAL RESULT
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 1.6000000E-01	X= 2.0000000E+01 Y= 1.2320000E+00	X= 1.1180340E+00 Y= 8.0292538E-02
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 1.8799999E-01	X= 2.0000000E+01 Y= 1.6670001E+00	X= 1.1180340E+00 Y= 9.8790073E-02
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.1100000E-01	X= 2.0000000E+01 Y= 2.2620001E+00	X= 1.1180340E+00 Y= 1.1387300E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.3100001E-01	X= 2.0000000E+01 Y= 3.0039999E+00	X= 1.1180340E+00 Y= 1.2806654E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.4699999E-01	X= 2.0000000E+01 Y= 4.0650001E+00	X= 1.1180340E+00 Y= 1.3878787E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.5799999E-01	X= 2.0000000E+01 Y= 5.8140001E+00	X= 1.1180340E+00 Y= 1.4306137E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.6600000E-01	X= 2.0000000E+01 Y= 9.2589998E+00	X= 1.1180340E+00 Y= 1.4670274E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.6899999E-01	X= 2.0000000E+01 Y= 2.0830017E+01	X= 1.1180340E+00 Y= 1.4442408E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.7000001E-01	X= 2.0000000E+01 Y= 1.0000000E+03	X= 1.1180340E+00 Y= 1.4067334E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 4.6999998E-02	X= 2.0000000E+01 Y= 3.0599999E-01	X= 1.1180340E+00 Y= 1.6011753E-02
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 8.9000002E-02	X= 2.0000000E+01 Y= 6.2900001E-01	X= 1.1180340E+00 Y= 3.4630015E-02
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 1.2700000E-01	X= 2.0000000E+01 Y= 9.2500001E-01	X= 1.1180340E+00 Y= 5.7321966E-02
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 1.6000000E-01	X= 2.0000000E+01 Y= 1.2320000E+00	X= 1.1180340E+00 Y= 8.0292538E-02
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 1.8799999E-01	X= 2.0000000E+01 Y= 1.6670001E+00	X= 1.1180340E+00 Y= 9.8790073E-02
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.1100000E-01	X= 2.0000000E+01 Y= 2.2620001E+00	X= 1.1180340E+00 Y= 1.1387300E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.3100001E-01	X= 2.0000000E+01 Y= 3.0039999E+00	X= 1.1180340E+00 Y= 1.2806654E-01
MISDAT	AERO	FINS	FINCN	FCNAAH	FIG60B	MVLOOK	LNTRP	X= 1.2500000E+00 Y= 2.4699999E-01	X= 2.0000000E+01 Y= 4.0650001E+00	X= 1.1180340E+00 Y= 1.3878787E-01

Figure 60 Extrapolation Message Output

```
DUMP OF INTERNAL DATA ARRAYS IN FOOT-POUND-RANKINE UNITS

FLT(  1)= 5.00000E+00  FLT(  2)= 0.00000E+00  FLT(  3)= 4.00000E+00  FLT(  4)= 8.00000E+00  FLT(  5)= 1.20000E+01
FLT(  6)= 1.00000E+01  FLT(  7)= 1.00000E-30  FLT(  8)= 1.00000E-30  FLT(  9)= 1.00000E-30  FLT( 10)= 1.00000E-30
FLT( 11)= 1.00000E-30  FLT( 12)= 1.00000E-30  FLT( 13)= 1.00000E-30  FLT( 14)= 1.00000E-30  FLT( 15)= 1.00000E-30
FLT( 16)= 1.00000E-30  FLT( 17)= 1.00000E-30  FLT( 18)= 1.00000E-30  FLT( 19)= 1.00000E-30  FLT( 20)= 1.00000E-30
FLT( 21)= 1.00000E-30  FLT( 22)= 0.00000E+00  FLT( 23)= 0.00000E+00  FLT( 24)= 1.00000E+00  FLT( 25)= 2.36000E+00
FLT( 26)= 1.00000E-30  FLT( 27)= 1.00000E-30  FLT( 28)= 1.00000E-30  FLT( 29)= 1.00000E-30  FLT( 30)= 1.00000E-30
FLT( 31)= 1.00000E-30  FLT( 32)= 1.00000E-30  FLT( 33)= 1.00000E-30  FLT( 34)= 1.00000E-30  FLT( 35)= 1.00000E-30
FLT( 36)= 1.00000E-30  FLT( 37)= 1.00000E-30  FLT( 38)= 1.00000E-30  FLT( 39)= 1.00000E-30  FLT( 40)= 1.00000E-30
FLT( 41)= 1.00000E-30  FLT( 42)= 1.00000E-30  FLT( 43)= 1.00000E-30  FLT( 44)= 1.00000E-30  FLT( 45)= 1.00000E-30
FLT( 46)= 3.00000E+06  FLT( 47)= 1.00000E-30  FLT( 48)= 1.00000E-30  FLT( 49)= 1.00000E-30  FLT( 50)= 1.00000E-30
FLT( 51)= 1.00000E-30  FLT( 52)= 1.00000E-30  FLT( 53)= 1.00000E-30  FLT( 54)= 1.00000E-30  FLT( 55)= 1.00000E-30
FLT( 56)= 1.00000E-30  FLT( 57)= 1.00000E-30  FLT( 58)= 1.00000E-30  FLT( 59)= 1.00000E-30  FLT( 60)= 1.00000E-30
FLT( 61)= 1.00000E-30  FLT( 62)= 1.00000E-30  FLT( 63)= 1.00000E-30  FLT( 64)= 1.00000E-30  FLT( 65)= 1.00000E-30
FLT( 66)= 1.00000E-30  FLT( 67)= 1.00000E-30  FLT( 68)= 1.00000E-30  FLT( 69)= 1.00000E-30  FLT( 70)= 1.00000E-30
FLT( 71)= 1.00000E-30  FLT( 72)= 1.00000E-30  FLT( 73)= 1.00000E-30  FLT( 74)= 1.00000E-30  FLT( 75)= 1.00000E-30
FLT( 76)= 1.00000E-30  FLT( 77)= 1.00000E-30  FLT( 78)= 1.00000E-30  FLT( 79)= 1.00000E-30  FLT( 80)= 1.00000E-30
FLT( 81)= 1.00000E-30  FLT( 82)= 1.00000E-30  FLT( 83)= 1.00000E-30  FLT( 84)= 1.00000E-30  FLT( 85)= 1.00000E-30
FLT( 86)= 1.00000E-30  FLT( 87)= 1.00000E-30  FLT( 88)= 1.00000E-30  FLT( 89)= 1.00000E-30  FLT( 90)= 1.00000E-30
FLT( 91)= 1.00000E-30  FLT( 92)= 1.00000E-30  FLT( 93)= 1.00000E-30  FLT( 94)= 1.00000E-30  FLT( 95)= 1.00000E-30
FLT( 96)= 1.00000E-30  FLT( 97)= 1.00000E-30  FLT( 98)= 1.00000E-30  FLT( 99)= 1.00000E-30  FLT(100)= 1.00000E-30
FLT(101)= 1.00000E-30  FLT(102)= 1.00000E-30  FLT(103)= 1.00000E-30  FLT(104)= 1.00000E-30  FLT(105)= 1.00000E-30
FLT(106)= 1.00000E-30  FLT(107)= 1.00000E-30  FLT(108)= 1.00000E-30  FLT(109)= 1.00000E-30  FLT(110)= 1.00000E-30
FLT(111)= 1.00000E-30  FLT(112)= 1.00000E-30  FLT(113)= 1.00000E-30  FLT(114)= 1.00000E-30  FLT(115)= 1.00000E-30
FLT(116)= 1.00000E-30  FLT(117)= 1.00000E-30  FLT(118)= 1.00000E-30  FLT(119)= 1.00000E-30  FLT(120)= 1.00000E-30
FLT(121)= 1.00000E-30  FLT(122)= 1.00000E-30  FLT(123)= 1.00000E-30  FLT(124)= 1.00000E-30  FLT(125)= 1.00000E-30

FLCT(  1)= 0.00000E+00  FLCT(  2)= 4.00000E+00  FLCT(  3)= 8.00000E+00  FLCT(  4)= 1.20000E+01  FLCT(  5)= 1.60000E+01
FLCT(  6)= 1.00000E-30  FLCT(  7)= 1.00000E-30  FLCT(  8)= 1.00000E-30  FLCT(  9)= 1.00000E-30  FLCT( 10)= 1.00000E-30
FLCT( 11)= 1.00000E-30  FLCT( 12)= 1.00000E-30  FLCT( 13)= 1.00000E-30  FLCT( 14)= 1.00000E-30  FLCT( 15)= 1.00000E-30
FLCT( 16)= 1.00000E-30  FLCT( 17)= 1.00000E-30  FLCT( 18)= 1.00000E-30  FLCT( 19)= 1.00000E-30  FLCT( 20)= 1.00000E-30
FLCT( 21)= 0.00000E+00  FLCT( 22)= 0.00000E+00  FLCT( 23)= 0.00000E+00  FLCT( 24)= 0.00000E+00  FLCT( 25)= 0.00000E+00
FLCT( 26)= 1.00000E-30  FLCT( 27)= 1.00000E-30  FLCT( 28)= 1.00000E-30  FLCT( 29)= 1.00000E-30  FLCT( 30)= 1.00000E-30
FLCT( 31)= 1.00000E-30  FLCT( 32)= 1.00000E-30  FLCT( 33)= 1.00000E-30  FLCT( 34)= 1.00000E-30  FLCT( 35)= 1.00000E-30
FLCT( 36)= 1.00000E-30  FLCT( 37)= 1.00000E-30  FLCT( 38)= 1.00000E-30  FLCT( 39)= 1.00000E-30  FLCT( 40)= 1.00000E-30
FLCT( 41)= 0.00000E+00  FLCT( 42)= 0.00000E+00  FLCT( 43)= 0.00000E+00  FLCT( 44)= 0.00000E+00  FLCT( 45)= 0.00000E+00
FLCT( 46)= 1.00000E-30  FLCT( 47)= 1.00000E-30  FLCT( 48)= 1.00000E-30  FLCT( 49)= 1.00000E-30  FLCT( 50)= 1.00000E-30
FLCT( 51)= 1.00000E-30  FLCT( 52)= 1.00000E-30  FLCT( 53)= 1.00000E-30  FLCT( 54)= 1.00000E-30  FLCT( 55)= 1.00000E-30
FLCT( 56)= 1.00000E-30  FLCT( 57)= 1.00000E-30  FLCT( 58)= 1.00000E-30  FLCT( 59)= 1.00000E-30  FLCT( 60)= 1.00000E-30
FLCT( 61)= 0.00000E+00  FLCT( 62)= 4.00000E+00  FLCT( 63)= 8.00000E+00  FLCT( 64)= 1.20000E+01  FLCT( 65)= 1.60000E+01
FLCT( 66)= 1.00000E-30  FLCT( 67)= 1.00000E-30  FLCT( 68)= 1.00000E-30  FLCT( 69)= 1.00000E-30  FLCT( 70)= 1.00000E-30
FLCT( 71)= 1.00000E-30  FLCT( 72)= 1.00000E-30  FLCT( 73)= 1.00000E-30  FLCT( 74)= 1.00000E-30  FLCT( 75)= 1.00000E-30
FLCT( 76)= 1.00000E-30  FLCT( 77)= 1.00000E-30  FLCT( 78)= 1.00000E-30  FLCT( 79)= 1.00000E-30  FLCT( 80)= 1.00000E-30

REFQ(  1)= 7.80000E-02  REFQ(  2)= 3.12500E-01  REFQ(  3)= 3.12500E-01  REFQ(  4)= 0.00000E+00  REFQ(  5)= 1.56250E+00
REFQ(  6)= 0.00000E+00  REFQ(  7)= 1.00000E+00  REFQ(  8)= 0.00000E+00  REFQ(  9)= 1.00000E-30
```

Figure 61 Internal Array Dump Output

A. EXAMPLE PROBLEMS

This appendix presents two example problems for use in verifying the proper operation of the computer code, or for use as a model for the setup of inputs of similar configurations. The first example is a simple tangent ogive nose-cylinder circular body. It has a planar wing (two panels) and a cruciform set of tails orientated in the "plus" configuration. There are two input cases for this problem. The first demonstrates the output generated when the PART control card is included. The second demonstrates the output created when trim has been requested; note that trim partial output has been requested with the PRINT AERO TRIM control card.

The second problem is a body-tail-inlet configuration. Three Mach numbers are requested, one subsonic, one transonic, and one supersonic. Although all three Mach numbers could be run in a single case, they have been divided into three separate cases to illustrate the "SAVE" feature as well as selecting particular output for illustration. The inlet computations are done only for the supersonic Mach number case, since the methods are only valid supersonically.

A.1 Example Problem 1

The first example problem is shown in Figure A-1. It is comprised of a 3-caliber tangent ogive nose attached to a cylindrical body; a triangular monoplane set of wings; and a cruciform set of tails orientated in the "plus" position. The first case is a simple angle of attack sweep; component build-up data and partial output are requested. The second case is a trim of the configuration using the two horizontal tail surfaces.

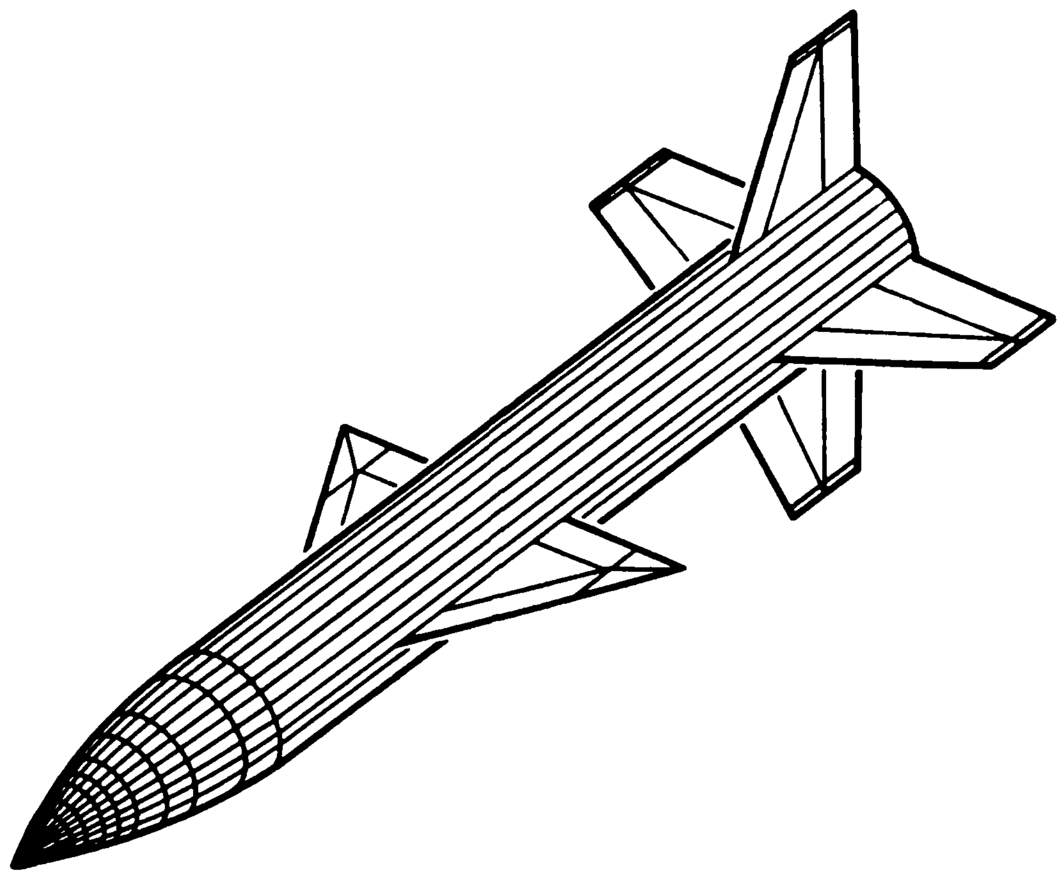


Figure A-1 Example Problem 1 Configuration

```
                    THE USAF AUTOMATED MISSILE DATCOM * REV 8/85 *
                    AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
                            CONERR - INPUT ERROR CHECKING

ERROR CODES - N* DENOTES THE NUMBER OF OCCURENCES OF EACH ERROR

A - UNKNOWN VARIABLE NAME

B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME

C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)

D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED

E - ASSIGNED VALUES EXCEED ARRAY DIMENSION

F - SYNTAX ERROR

   ******************************  INPUT DATA CARDS  ******************************

 1 CASEID PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
 2 SOSE
 3 DIM IN
 4 NO LAT
 5  $FLTCON NMACH=1.,MACH=2.36,REN=3.E6,
 6          NALPHA=8.,ALPHA=0.,4.,8.,12.,16.,20.,24.,28.,$
 7  $REFQ XCG=18.75,$
 8  $AXIBOD LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,$
 9  $FINSET1 CHORD=6.96,0.,SSPAN=1.875,5.355,XLE=15.42,
10           SWEEP=0.,STA=1.,ZUPPER=2*0.02238,NPANEL=2.,
11           LMAXU=0.238,LER=2*.015,LFLATU=0.524,$
12  $FINSET2 CHORD=5.585,2.792,SSPAN=1.875,6.260,XLE=31.915,
13           SWEEP=0.,STA=1.,ZUPPER=2*0.02238,NPANEL=4.,
14           LMAXU=0.288,LER=2*.015,LFLATU=0.428,FINPHI=0.,$
15 BUILD
16 SAVE
17 PART
18 NEXT CASE
19 CASEID TRIM OF CASE NUMBER 1
20  $TRIM SET=2.,$
21 PRINT AERO TRIM
22 NEXT CASE
```

A.2 Example Problem 2

The configuration for this example is sketched in Figure A-2. The figure is a modified copy of the wind tunnel model drawings from NASA Technical Memorandum 84557. The model definition in Figure A-2 is representative of the detail normally found on design drawings.

This example has been divided into subsonic, transonic, and supersonic cases. Each case is run for one Mach number. Although all three Mach numbers could have been run in one case they were run separately to demonstrate the SAVE capability.

This example provides a check case for the inlet option. It can be used by the user to make sure that he understands the inputs.

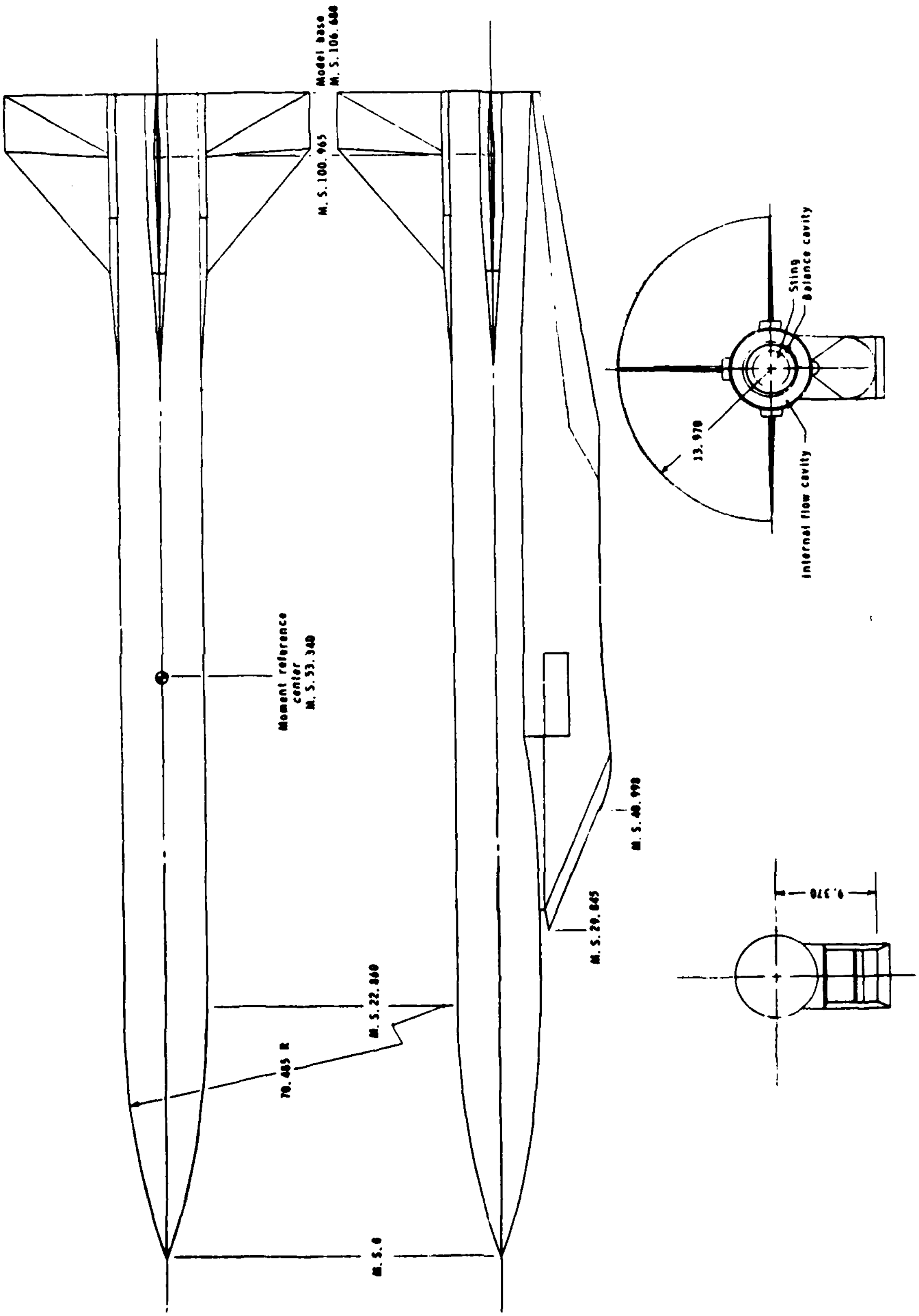
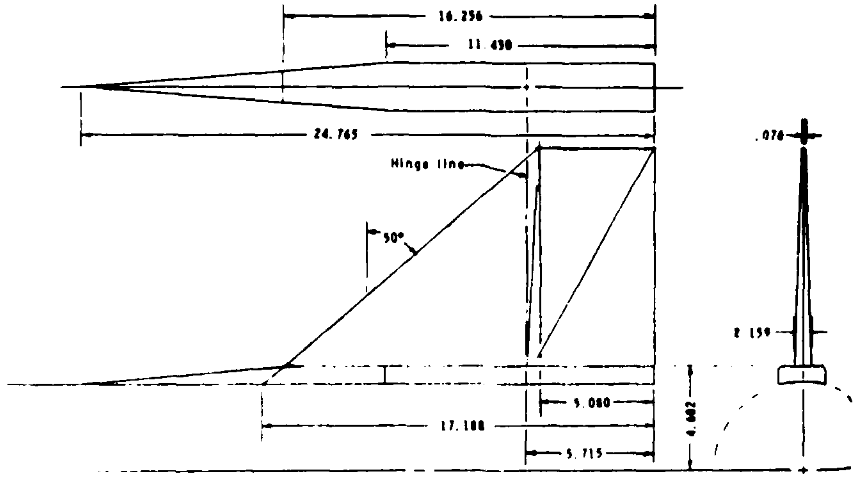
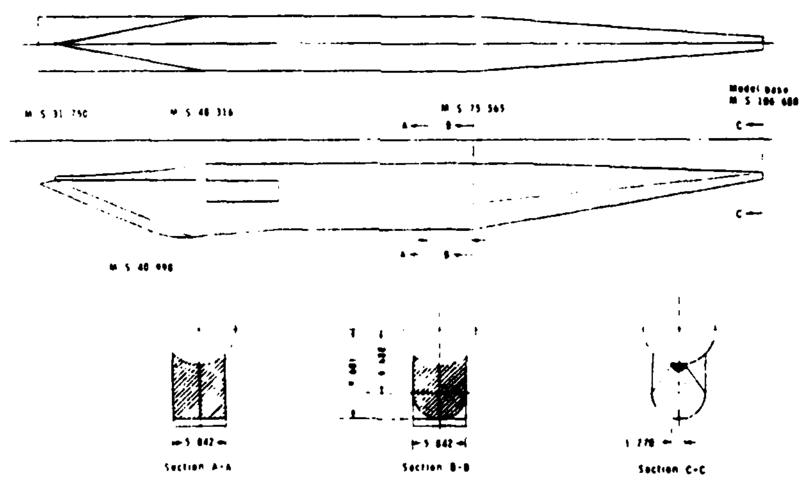


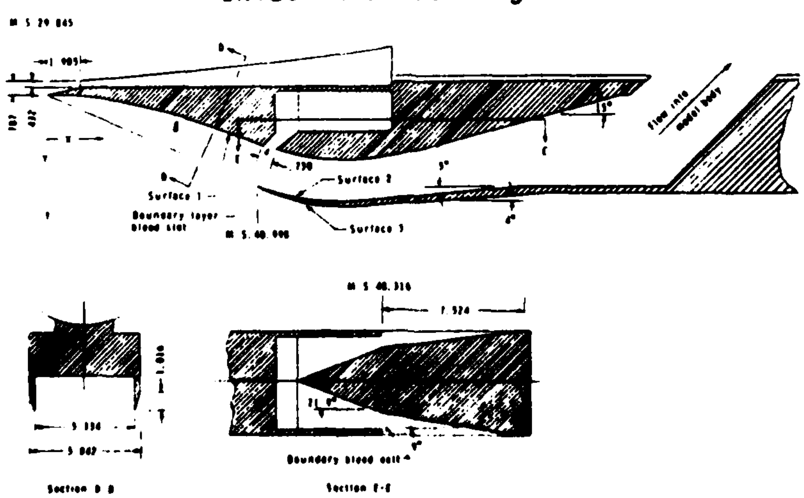
Figure A-2 Example 2 Configuration-Body/Tail/Inlet



Tail-surface details



Inlet-inlet fairing details



Inlet details - 2x size

Figure A-2 (cont.)

```
                    THE USAF AUTOMATED MISSILE DATCOM * REV 8/85 *
                    AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
                             CONERR - INPUT ERROR CHECKING

ERROR CODES - N* DENOTES THE NUMBER OF OCCURENCES OF EACH ERROR

A - UNKNOWN VARIABLE NAME

B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME

C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)

D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED

E - ASSIGNED VALUES EXCEED ARRAY DIMENSION

F - SYNTAX ERROR

  ******************************  INPUT DATA CARDS  ******************************

 1 *
 2 * USE DEFAULT REFERENCE AREA AND LENGTH
 3 *
 4  $REFQ XCG=53.34,$
 5 *
 6 * DEFINE FLIGHT CONDITIONS
 7 *
 8  $FLTCON NMACH=1., MACH=.6, REN=6.56E6,
 9          NALPHA=5., ALPHA=0.,5.,10.,20.,25.,$
10 *
11 * DEFINE BODY
12 * NO BASE DRAG INCLUDED (DEXIT NOT INPUT)
13 *
14  $AXIBOD TNOSE=OGIVE, LNOSE=22.86, LCENTR=83.821,                  ** SUBSTITUTING NUMERIC FOR NAME OGIVE
15          DNOSE=7.62, DCENTR=7.62,$
16 *
17 * DEFINE FINS
18 *
19  $FINSET1 SECTYP=HEX, SSPAN=9.368,13.97, CHORD=17.188,5.715,         ** SUBSTITUTING NUMERIC FOR NAME HEX
20           XLE=89.492, SWEEP=0., STA=1., NPANEL=3., PHIF=0.,90.,270.,
21           ZUPPER=.1256,0., LFLATU=0.,1., LMAXU=.665,0.,$
22 *
23 * DEFINE INLET GEOMETRY
24 *
25  $INLET  INTYPE=2D, POSD=1., NIN=1., PHI(1)=90., XBO=0.,            ** SUBSTITUTING NUMERIC FOR NAME 2D
26          X=29.845,31.750,75.565,106.68,40.998,47.,58.,0.,31.750,44.,
27          B=5.842,5.842,0.,1.27,0.,0.,0.,0.,0.,0.,
28          Z=0.,.355,0.,0.,-4.645,-5.145,-4.845,0.,0.,0.,
29          XIND=31.750,48.316, TIND=.432,$
30 *
31 * OPTIONS
32 *
33 CASEID MISSILE DATCOM EXAMPLE PROBLEM, SUBSONIC
34 SOSE
35 DIM CM
36 PART
37 SAVE
38 NEXT CASE
39 *
40 * CASE TWO, TRANSONIC
41 *
```

```
 $FLTCON NMACH=1., MACH=1.0,$
*
* OPTIONS
*
CASEID MISSILE DATCOM EXAMPLE PROBLEM, TRANSONIC
NO LAT
PRINT AERO HINGE
PRINT AERO BEND
SAVE
NEXT CASE
*
* CASE THREE, SUPERSONIC
*
 $FLTCON NMACH=1., MACH=3.95,$
*
* OPTIONS
*
CASEID MISSILE DATCOM EXAMPLE PROBLEM, SUPERSONIC
PRESSURES
NEXT CASE
```

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