MISSILE DATA COMPENDIUM (DATCOM)
User Manual – 2014 Revision

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U.S. Army Aviation and Missile Research, Development, and Engineering Center

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**Aerospace Vehicles Technical Assessment and Simulation Branch
Power and Control Division**

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Final Report

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UNITED STATES AIR FORCE**

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PREFACE

This report was prepared by the System Simulation and Development Directorate of the US Army Aviation and Missile Research, Development and Engineering Center (AMRDEC) in conjunction with the Aerospace Systems Directorate of the Air Force Research Laboratory (AFRL/RQ), Wright-Patterson Air Force Base, Ohio. It documents the October 2014, FORTRAN 90 version of Missile Datcom. The development of the original FORTRAN 77 version of Missile Datcom was performed by the McDonnell Douglas Corporation, St. Louis, Missouri. This report supersedes AFRL-RB-WP-TR-2011-3071 that documents the 2011 Datcom revision and AFRL-RB-WP-TR-2009-3015 that documents the 2008 revision of Missile Datcom.

A list of individuals who made significant contributions to the development of Missile Datcom from it's inception is provided below.

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The System Simulation and Development Directorate of the US Army AMRDEC and the Air Vehicles Directorate of the AFRL are planning to continue 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:

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SUMMARY OF MISSILE DATCOM RELEASES

Contract	Investigator (Govt engineer)	Documentation	Release	Rev.	Capability added
F33615-80-C-3605 (McDonnell Douglas)	S.R. Vukelich (J.E. Jenkins)	AFWAL TR-81-3130			Feasibility study only Recommended methods, code structure
F33615-81-C-3617 (McDonnell Douglas)	S.R. Vukelich (J.E. Jenkins)		1	12/84	Axisymmetric bodies Two fin sets with up to four fins each Automatic configuration trim
Same	S.R. Vukelich (J.E. Jenkins)		2	11/85	Elliptical bodies, Inlets at supersonic speeds Dynamic derivatives, exp data substitution Four fin sets with up to eight fins each
Same	S.L. Stoy (J.E. Jenkins)	AFWAL-TR-86-3091 (ADA 211086, 210128)	3	12/88	Expanded data substitution Configuration incrementing
None	(W.B. Blake)		4	7/89	Expanded body dynamic derivatives
F33615-86-C-3626 (McDonnell Douglas) F33615-87-C-3604 (NEAR Inc)	A.A. Jenn (J.E. Jenkins) M.F.E. Dillenius (W.B. Blake)	WL-TR-91-3039 (ADA 237817)	5	4/91	Inlets at sub/transonic speeds, additive drag Plume effects on body Six types of body protuberances Modified fin lateral center of pressure
F33657-89-D-2198 (McDonnell Douglas)	K.A. Burns (J.W. Herrmann)	WL-TR-93-3043 (ADA 267447)	6	6/93	UNIX workstation, PC compatibility Trailing edge flaps, Folding fins Semi-submerged inlets
None	(W.B. Blake)	AFRL-TR-1998-3009 (ADA-344707)	7	5/97	Fortran 90 compatibility Expanded dynamic derivatives Revised body-fin upwash, Modified base drag Modified fin longitudinal center of pressure
None	(W.B. Blake)	AIAA-1999-4258	8	09/02	High AOA, airfoil, friction drag corrections
None	M. Underwood (L. Auman, Army) (W.B. Blake, AF)	AIAA-2003-3668 AIAA-2005-4833 AIAA-2005-4971	9	01/06	Code clean-up and restructurng Cambered body capability Revised body drag (bluntness, high AOA)
DAAH01-02-C-R170 W31P4Q-07-D-0001	M. Underwood C.C. Rosema (L. Auman, Army)	AIAA-2007-3936 AIAA-2007-3937	10	07/07	Rolling moment for elliptical bodies Improvements to TE Flap control increments Improvements for low aspect ratio fins
W31P4Q-07-D-0001	M. Underwood J.B. Doyle (C.C. Rosema) (L. Auman, Army)	AIAA-2009-0907 AFRL- TR-2009-3015	11	08/08	Nine fin sets with up to 8 fins each Revised body drag Moment contribution from protuberances
None	(C.C. Rosema) J.B. Doyle	AIAA 2009-3853 AIAA 2011-1240 AIAA 2011-1241	12	03/11	Improved fin-shed vortex modeling Up to 20 vortices shed per fin Output file option for vortex visualization
None	(C.C. Rosema) (J.B. Doyle)		13	2014	Improved supersonic (M>1.2) nose drag Improved body-shed vortex model

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.

2.0 PROGRAM CAPABILITIES/INSTALLATION

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:

- An axisymmetric or elliptically-shaped body.

- One to nine 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. Each fin may be deflected independently, as an all moving panel or as a fixed panel with a plain trailing edge flap.

- 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 overridden by the user in order to more accurately model the configuration of interest.

2.1 TYPES OF DATA COMPUTED

2.1.1 Aerodynamics

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

C_N	Normal Force Coefficient (body axis)
C_L	Lift Coefficient (wind axis)
C_M	Pitching Moment Coefficient (body axis)
X_{cp}	Center of Pressure in calibers from the moment reference center
C_A	Axial Force Coefficient (body axis)
C_D	Drag Coefficient (wind axis)
C_Y	Side Force Coefficient (body axis)
C_n	Yawing Moment Coefficient (body axis)
C_l	Rolling Moment Coefficient (body axis)
$C_{N\alpha}$	Normal force coefficient derivative with angle of attack
$C_{m\alpha}$	Pitching moment coefficient derivative with angle of attack
$C_{Y\beta}$	Side force coefficient derivative with sideslip angle
$C_{n\beta}$	Yawing moment coefficient derivative with sideslip angle (body axis)
$C_{l\beta}$	Rolling moment coefficient derivative with sideslip angle (body axis)
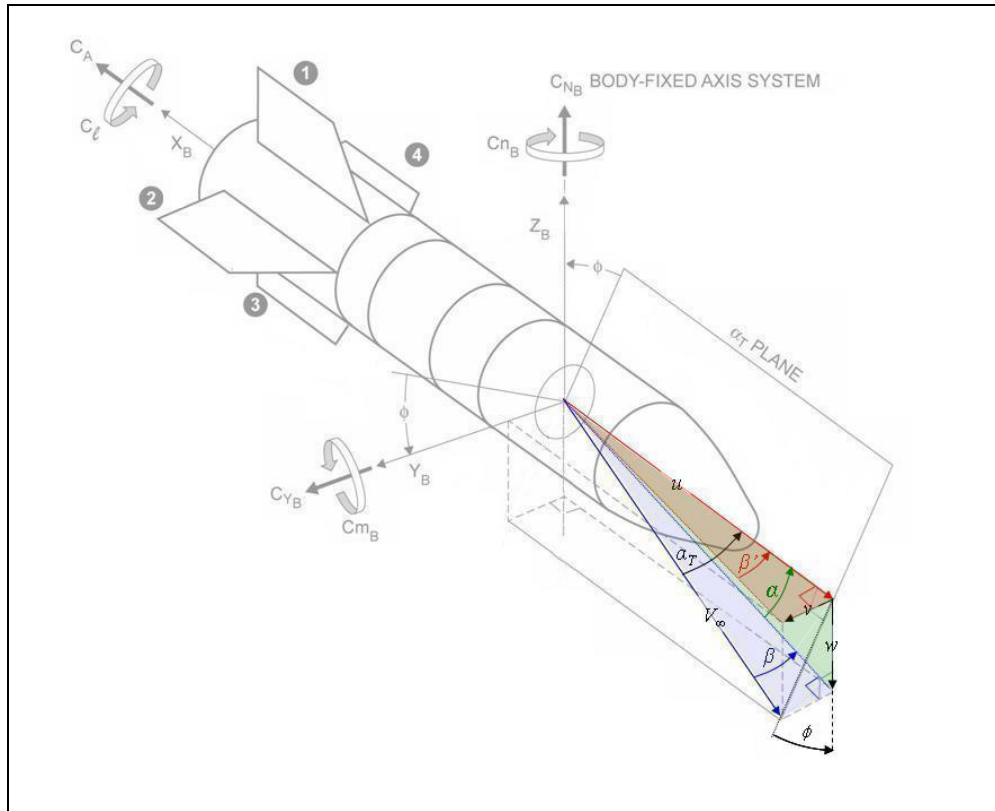
Figure 1. Coefficient Axis System

For body alone or body plus fin combinations, the following parameters are also computed, all in the body axis system:

C_{Mq}	Pitching moment coefficient derivative with pitch rate
C_{Nq}	Normal force coefficient derivative with pitch rate
C_{Aq}	Axial force coefficient derivative with pitch rate
$C_{M\dot{\alpha}}$	Pitching moment derivative with rate of change of angle of attack
$C_{N\dot{\alpha}}$	Normal force derivative with rate of change of angle of attack
C_{lp}	Rolling moment coefficient derivative with roll rate
C_{np}	Yawing moment coefficient derivative with roll rate
C_{Yp}	Side force coefficient derivative with roll rate
C_{lr}	Rolling moment coefficient derivative with yaw rate
C_{nr}	Yawing moment coefficient derivative with yaw rate
C_{Yr}	Side force coefficient derivative with yaw rate

The output units of the aerodynamic derivatives can be in degrees or radians and is controlled by the DERIV control card. This control card has no effect on input angles, input angles are always specified in degrees. Partial output results, which detail the components used in the calculations, are also optionally available using the PART and/or BUILD control cards.

The drag force (and drag coefficient) is different between the wind and stability axes systems if the missile body is at a sideslip angle (β) to the wind. However, wind axis drag and stability axis drag are the same at zero sideslip. In Missile Datcom, drag force methods are assumed to be in the stability axes system and axial force methods are assumed to be in the body axes system unless otherwise noted.

The program has the capability to perform a static trim of the configuration, using any fin set 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.

Fin data is also calculated and made available in tabulated form in the plot043.csv file as follows:

CNx_IPB	Finset x normal force coefficient in presence of the body
CMx_IPB	Finset x pitching moment coefficient in presence of the body
CAx_IPB	Finset x axial force coefficient in presence of the body
CYx_IPB	Finset x side force coefficient in presence of the body
CLNx_IPB	Finset x yawing moment coefficient in presence of the body
CLLx_IPB	Finset x rolling moment coefficient in presence of the body
PBMx_Py	Finset x, panel y root bending moment coefficient [$RBM/(qS_{ref}L_{ref})$]
PHMx_Py	Finset x, panel y hinge moment coefficient [$HM/(qS_{ref}L_{ref})$]
PCNx_Py	Finset x, panel y normal force coefficient [$CN/(qS_{ref})$] normal to the panel surface

2.1.2 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. The reference area and reference length are user defined.

2.2 INSTALLATION ON COMPUTER SYSTEMS

This section details the steps necessary to make the computer code functional on the user's computer system. Although conversion of the program can be easily accomplished by someone with a good understanding of the FORTRAN language, it is highly recommended that someone familiar with the computer operating system be consulted.

2.2.1 Requirements

In order for the Missile Datcom code to be successfully compiled on the user's computer system, there are three requirements which must be met, as follows:

- Language - The code is written in FORTRAN 77 but is also compatible with FORTRAN 90/95.

- Namelist - The code has been designed with an internal FORTRAN NAMELIST emulator to allow the input and output (I/O) to be handled by namelist variables. This is an exception to Standard FORTRAN but with the emulator as part of the code the program will run under Standard FORTRAN. The code is not easily converted to fixed field, rather than namelist input.

- I/O Scratch Files - The code uses the logical file units listed in **Table 1**. All file units are accessed using formatted reads and writes. File units 2 and 8 are used internally; file units 3, 5 ,6, 9, 10, 11, 12, 20, 21, 22, 42 and 43 transfer data between the user and the code.

2.2.2 Input/Output

Fourteen file units are used by the program. They are used as follows:

Table 1. Input/Output Logical Units

Unit	Name	Usage	Subroutine
2	for002.dat	Namelists for the input "case" are read from unit 8 and written to unit 2.	READIN
3	for003.dat	Plot file of aerodynamic data, written at user request (using PLOT card) to unit 3	PLOT3 PLTTRM PLTUT9
5	for005.dat	User input file read from unit 5	CONERR
6	for006.dat	Program output file written to unit 6	PRINTS PRIIOM
8	for008.dat	User input cards read from unit 5 are written to unit 8 after they have been checked for errors.	CONERR
9	for009.dat	Body geometry data, written at user request to unit 9	SBODY
10	for010.dat	Body pressure coefficient data at angle of attack, written at user request to unit 10 when using PRESSURES card.	SOSE HYPERS
11	for011.dat	Fin pressure coefficient data, written at user request to unit 11 when using PRESSURES card	FCAWPF
12	for012.dat	Body pressure coefficient and local Mach number at zero angle of attack, written at user request to unit 12 when using PRESSURES card	SOSE
20	for020.dat	Total configuration force and moment coefficient data, damping derivatives and flight conditions, written at user request (using PLOT card). File is formatted for use with software developed with Adaptive Modeling Language (AML).	PLOT20
21	for021.dat	Total configuration force and moment coefficient data,	PLOT21

		damping derivatives, flight conditions and control deflections, written at user request (using PLOT card). File is formatted for use with the Aviator Visual Design Simulator (AVDS).	
22	for022.dat	Configuration geometry file compatible with the commercial software package Tecplot. Only geometry for the body and fins are printed. No geometry is provided for inlets or probuterances.	TECGRD
42	for042.csv	All standard data written in rows and columns with headers	PLOT42
43	for043.csv	Fin data written in rows and columns with headers	PLOT43
	vpath*.dat	Fin-shed vortex path data, including cartesian coordinate position and strength. Written at user request. One vpath*.dat file will be written for each alpha-Mach flight condition.	VPRINT

As the number of output columns change for the for042.csv file depending on what input flags the user has set, it is recommended that any user-developed software that reads this file should key off the column header names and not specific column locations.

The program is run in a "batch" mode. The user prepares an input file in accordance with the rules given in *Section 3* of this report. This file must be renamed "for005.dat" prior to program execution. The program is then exectued by double clicking the executable file. The program then exectues and creates the output files requested by the case inputs. The primary output files, "for006.dat" and "for042.csv" are always written. A complete discussion of what is contained in these output files is given in *Section 4* of this report. The optional plot, geometry, pressure distribution and vortex path output files are written only if requested.

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, for example, 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. 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.

The program includes an error checking routine which scans all inputs and identifies errors. This process is a single-pass error checking routine; errors are identified in a single "run". In addition, the program checks for necessary valid inputs, such as a non-zero Reynolds number. In general the code will skip over cases with errors and provide an error message to the user.

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

- The unit system can be English or Metric and is controlled by the DIM card. The four options are feet, inches, meters, and centimeters. DIM FT and DIM IN toggle the English unit system while DIM M and DIM CM toggle the Metric unit system. Missile geometry inputs are specified by the chosen length following DIM while other input flight conditions and reference quantities such as altitude, velocity, pressure, and temperature are associated with the unit system chosen rather than actual length. The default is DIM FT. Refer to **Table 23** for a detailed listing of corresponding units.

- Output derivatives can be expressed in degrees or radians, and are specified via the DERIV control card. 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.

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3.1 NAMELIST INPUTS

The required program inputs use FORTRAN namelists. Missile Datcom is similar to other codes which use the namelist input technique, but differs 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, however this is NOT a recommended practice and should be avoided.

     The three namelist inputs

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

     are equivalent to

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

- If the SAVE card is used, the total number of namelists read, including repeat occurrences of the same namelist name from all previous SAVEs, must not exceed 300.

- The last occurrence of a namelist variable in a case is the value 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

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

- Certain variables may be input as arrays instead of single values, such as ALPHA. If the array list is too long (80 columns) to fit on one line, it must be continued on the following line. This can be done by repeating the variable name with the array index of the first continued value, or continuing to the next line with the next value. For example:

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```
$FLTCON
    NALPHA=20., ALPHA=0.,2.,4.,6.,8.,10.,12.,14.,16.,18.,20.,
    ALPHA(12)=22.,24.,28.,32.,36.,40.,44.,48.,52.,
    NMACH=5., MACH=0.2,0.8,1.5,2.0,3.0, ALT=0.,10000.,
    ALT(3)=20000.,30000.,40000.,$
```

or

```
$FLTCON
    NALPHA=20., ALPHA=0.,2.,4.,6.,8.,10.,12.,14.,16.,18.,20.,
    22.,24.,28.,32.,36.,40.,44.,48.,52.,
    NMACH=5., MACH=0.2,0.8,1.5,2.0,3.0, ALT=0.,10000.,
    20000.,30000.,40000.,$
```

• The namelists can be input in any order.

• Only those namelists required to execute the case need be entered.

• Certain alphanumeric constants are permitted. They are summarized in *Table 2*.

• Note the constant UNUSED can be used in the input deck, but it cannot be guaranteed to reset the value truly to the value used within the code (i.e., the read in value may not be truly .EQ. to UNUSED within the code due to round-off error). It is therefore recommended that UNUSED be avoided. The recommended way of resetting a variable within a namelist input is to use the DELETE command on the namelist, then set the values as desired.

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. **ALL namelist and variable names must be input in CAPITAL LETTERS**. This also applies to numerical values input in "E" format, i.e. REN=6.0E06 is acceptable, while REN=6.0e06 is not.

The namelist names have been selected to be mnemonically related to their physical meaning. Each component of the configuration requires a separate namelist input.

$FLTCON	to define the flight conditions
$AXIBOD or $ELLBOD	to define the body
$FINSET1	to define the first fin set
$FINSET2	to define the second fin set
$FINSET3	to define the thirdfin set
$FINSET4	to define the fourth fin set
$FINSET5	to define the fifth fin set
$FINSET6	to define the sixth fin set
$FINSET7	to define the seventh fin set
$FINSET8	to define the eigth fin set
$FINSET9	to define the ninth fin set
$PROTUB	to define protuberance option inputs
$INLET	to define inlet geometry

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
$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.

Table 2. Namelist Alphanumeric Constants

NAMELIST	PERMITTED ALPHANUMERIC CONSTANTS	CONVERTED VALUE
(ALL)	UNUSED	1.E-30 (not recommended)
REFQ	TURB	0.
	NATURAL	1.
AXIBOD or ELLBOD	CONICAL	0.
	CONE	0.
	OGIVE	1.
	POWER	2.
	HAACK	3.
	KARMAN	4.
PROTUB	VCYL	1.
	HCYL	2.
	LUG	3.
	SHOE	4.
	BLOCK	5.
	FAIRING	6.
FINSETn	HEX	0.
	NACA	1.
	ARC	2.
	USER	3.
INLET	2DTOP	3.
	2DSIDE	1.
	AXI	2.
EXPR	BODY	1.
	F1	2.
	F2	3.
	F3	4.
	F4	5.
	BF1	6.
	BF12	7.
	BF123	8.
	BF1234	9.

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 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 "0" and the alphabetic letter "O", it should be noted that none of the namelist or namelist variable names contain the number zero in them. In general, the number zero and the letter "O" are not interchangeable 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. In the 7/07 and subsequent releases, over-specified geometry should always cause an error and STOP the code.

3.1.1 Namelist FLTCON - Flight Conditions

This namelist defines the flight conditions to be run for the case. The program is limited to no more than 100 angles of attack and 20 Mach number/Altitude combinations per case at a fixed sideslip angle, aerodynamic roll angle, and panel deflection angle. (Releases prior to 7/07 limit the angles of attack to 20.) Therefore, a "case" is defined as a fixed geometry with variable Mach number/Altitude and angles of attack.

The inputs are given in **Table 3**. 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 (β) are defined.

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NOTE: The 7/07 and all prior releases defined BETA as $\beta = \tan^{-1}(v/u)$, which is shown as β' in *Figure 1*. The 8/08 and subsequent releases define BETA as $\beta = \sin^{-1}(v/V_o)$ which is shown as β in *Figure 1*. Equations for converting between the two definitions are given in Table 3.

- 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).

    NOTE: Input ALPHA, BETA and PHI is NOT allowed. The 7/07 and subsequent releases will generate an error message and STOP execution

    NOTE: Missile Datcom will not run if two or more identical ALPHA values are defined in the input file.

As a minimum the following variables must be defined:

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

    NOTE: The ALT, REN, TINF and PINF data must correspond to the MACH or VINF inputs. The ALPHA and MACH (or VINF) dependent data can be input in any order; the code will sort the data into ascending order. If MACH or VINF is sorted by Missile Datcom, the corresponding REN, ALT, TINF, and PINF will also be reordered to maintain the correct pairing with MACH or VINF.

Missile Datcom requires Mach number and Reynolds number to operate. The program can calculate these from several combinations of free-stream input conditions.

1. MACH and REN   (no computations required – primary input)
2. MACH and ALT   (Reynolds number is computed using the 1962 Standard Atmosphere)
3. MACH, TINF and PINF   (Reynolds number is computed)
4. VINF and ALT   (Reynolds and Mach number are computed using the 1962 Standard Atmosphere)
5. VINF, TINF and PINF   (Reynolds and Mach number are computed)
6. VINF, TINF and REN   (Mach number is computed)

User supplied data will take precedence over program calculations. Hence, the user can override any default or Standard Atmosphere calculation.

NOTE: Previous versions of Missile Datcom used ALT=0 (sea-level) as a default condition. This default has been elimintated in the 7/07 version as it was found to be a source of user error. The user must specify the flight conditions, however the user may mix and match the data if he chooses. For example:

```
VINF(1) = 300.,  ALT(1) = 30000.,
MACH(2) = 2.,    ALT(2) = 0.,
```
or
```
VINF(1) = 300., TINF(1) = 288., REN(1) = 1.E+6,
MACH(2) = 2.,   REN(2) = 2000000.,
```

are valid entries.

Table 3. NAMELIST FLTCON Variables

NAMELIST FLTCON				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS****	**DEFAULT**
NALPHA	-	Number of angles of attack (must be > 1)	-	-
ALPHA	100	Angle of attack or total angle of attack, $\alpha = \tan^{-1}(w/u)$ or $\alpha_T = \tan^{-1}(\sqrt{(v^2 + w^2}/u)$ (see Figure 1)	Deg	-
BETA***	-	Sideslip angle, $\beta = \sin^{-1}(v/V_o)$, (see Fig. 1)	Deg	0.
PHI***	-	Aerodynamic roll angle $\phi = \tan^{-1}(v/w)$, (see Figure 1)	Deg	0.
NMACH	-	Number of Mach numbers or velocities	-	-
MACH*	20	Mach numbers	-	-
ALT*	20	Altitudes	L	0.
REN*	20	Reynolds numbers per unit length	1/L	-
VINF*	20	Freestream velocities	L/sec	-
TINF*	20	Freestream static temperatures	Deg	-
PINF*	20	Freestream static pressures	F/L^2	-

Note:
  *   Any of the following combinations satisfy the minimum requirements for calculating atmospheric conditions (Mach and Reynolds number):
        1. MACH and REN
        2. MACH and ALT
        3. MACH and TINF and PINF
        4. VINF and ALT
        5. VINF and TINF and PINF
        6. VINF and TINF and REN
  **   Lengths are in feet for English units and meters for metric units.
  *** PHI and BETA can NOT be used in the same case.

(α_T, ϕ) to $(\alpha, \beta$ or $\beta')$

$\tan \alpha = \tan \alpha_T \cos \phi$

$\sin \beta = \sin \alpha_T \sin \phi$

$\tan \beta' = \tan \alpha_T \sin \phi$

$(\alpha, \beta$ or $\alpha, \beta')$ to (α_T, ϕ)

$\cos \alpha_T = \cos \alpha \cos \beta$

$\tan \phi = \tan \beta / \sin \alpha$

$\tan \alpha_T = \sqrt{\tan^2 \alpha + \tan^2 \beta'}$

$\tan \phi = \tan \beta' / \tan \alpha$

(α, β') to (α, β)

$\tan \beta = \tan \beta' \cos \alpha$

3.1.2 Namelist REFQ - Reference Quantities

Inputs for this namelist are optional and are defined in **Table 4**. 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 and to the user input reference quantities (SREF, LREF, LATREF, XCG). If no reference quantities are input, they are computed based upon the scaled geometry. XCG is input relative to the origin of the global coordinate system (X=0, **Figure 2**) and is scaled using SCALE.

Table 4. NAMELIST REFQ Variables

NAMELIST REFQ				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
SREF	-	Reference area	L^2	*
LREF	-	Longitudinal reference length	L	**
LATREF	-	Lateral reference length	L	LREF
XCG	-	Longitudinal rosition of C.G. (+aft)	L	0.
ZCG	-	Vertical position of C.G. (+up)	L	0.
BLAYER	-	Boundary layer type: TURB for fully turbulent NATURAL for natural transition	-	TURB
ROUGH ***	-	Surface roughness height (see **Table 5** for range of values)	L	0.
RHR ***	-	Roughness Height Rating (see **Table 6** for range of values)	-	0.
SCALE	-	Vehicle scale factor	-	1.
Note: * Default is maximum body cross-sectional area. If no body is input, default is maximum fin panel area. ** Default is maximum body diameter. If no body is input, default is fin panel mean geometric chord. *** Either ROUGH or RHR can be used. If ROUGH is used, the units must be inches (for English) or centimeters (for Metric).				

In lieu of specifying the surface roughness height ROUGH, 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 ROUGH and RHR are given in *Table 6*.

Table 5. Equivalent Sand Roughness

TYPE OF SURFACE	EQUIVALENT SAND ROUGHNESS k (INCHES)	RHR
Aerodynamically Smooth	0.0	0.0
Polished Metal or Wood	0.00002 to 0.00008	6 to 26
Natural Sheet Metal	0.00016	53
Smooth Matte Paint, Carefully Applied	0.00025	83
Standard Camouflage Paint, Average Application	0.00040	133
Camouflage Paint, Mass Production Spray	0.0012	400
Dip Galvanized Metal Surface	0.006	2000
Natural Surface of Cast Iron	0.01	3333

Table 6. Preferred RHR Values

APPLICATION	RHR
Steel Structural Parts	250
Aluminum and Titanium Structural Parts	125
Close Tolerance Surfaces	63
Seals	32

3.1.3 Namelist AXIBOD - Axisymmetric Body Geometry

An axisymmetric body is defined using this namelist. The namelist input variables are given in **Table 7** and **Table 8** and a sketch of the geometric inputs are given in **Figure 2** and **Figure 3**. 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.

NOTE: Versions prior to the 1/06 release could NOT handle CONIC centerbodies. The 1/06 and subsequent releases include changes that now allow for conic centerbodies.

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

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.

If Option 2 is selected, the program generates a body contour based on the user specified values of X, R, and DISCON. Many additional points in between the user specified input coordinates will be generated. The resulting contour can contain up to 300 points. If the PRINT GEOM BODY control card is used, this contour will be written to tape unit 9 ("for009.dat").

An option to specify a cambered body was added in the 1/06 and subsequent releases. This new functionality added the Z namelist input as an optional input for the Option 2 body input. Cambered should only be run with the aerodynamic roll angle (PHI) set to zero.

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, and more accurate aeroprediction methods are used for certain flight conditions for Option 1. 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. If these parameters are not input, they are selected as defined in **Table 9**.

If DEXIT is not input, 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.). Regardless of the DEXIT input, the plot042.csv file includes zero base drag and full base drag axial force data.

If body coordinates are input using the variables NX, X, R, and DISCON, the nose is spherically blunted, and results using the Second Order Shock Expansion method are desired (only if M>1.2), the geometry must be additionally defined using the following:

• BNOSE must be specified

• TRUNC must be set to .FALSE.

- The first five (5) points in the X and R arrays must lie on the spherical nose cap with the fifth point being the transition from the spherical nose cap to the rest of the nose or body [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 with X(5) and R(5) being the transition point].

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

- 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.

For the 2014 release, a new body-shed vortex model has been implemented where vortices are shed incrementally down the axial direction of the body, thus forming a 'cloud' of vortices. The separation lines and incremental vortex strengths have been determined through a large scale computational fluid dynamics (CFD) study. Unlike the previous model which simply used a single symmetric vortex pair with empirically derived strengths and positions, this new model tracks the vortices similar to the updated fin-shed vortex model from the 3/11 release, thus allowing for mutual interaction of all of the vortices as they move downstream. For symmetric configurations and flight conditions, little to no changes will be evident in the output. However, for cases with asymmetric canard vortex shedding, the new model shows a robust improvement in calculated forces and moments.

Because the new model requires more memory and processing, it can slow down execution time (O(milliseconds) to O(seconds)). As a result, use of the new model is optional and at the user's discretion, with the input options shown in *Table 10*. The default will use the legacy model. While the model has been tuned to best match available computational and empirical data, the user has the ability to change the strength of the shed vortices if desired with the BVGAM parameter. However, it is recommended to use the default BVGAM value of 1.

If the new body-shed vortex model is used and the PRINT VORTEX control card is set, the vortex paths and strengths will be written in the vpath*.dat output file along with the fin-shed vortices. Giving the user the ability to visualize the entire vortex cloud in Tecplot.

The inputs for base-jet plume interaction effects are defined using Option 1 and are shown in *Table 11*. Incremental forces and moments due to jet induced boattail separation and separation locations on aft fins are calculated if these inputs are used.

- This option should only be run for supersonic cases (i.e. $M_\infty > 1.2$)

- The calculations will be done for three types of aft bodies conical boattail, ogival boattail, or cylindrical (i.e. no boattail). Error messages will be printed to the output file and the calculations skipped if any other aft body is defined.

- If BASE=.FALSE. or is not input the calculations will be skipped.

- DEXIT must not equal zero if this option is used.

- The jet Mach number (JMACH), jet to freestream static pressure ratio (PRAT), and jet to freestream stagnation temperature ratio (TRAT) must be specified for each freestream Mach number or velocity input in the namelist FLTCON. For subsonic or transonic freestream Mach numbers or velocities, dummy values must be input for JMACH, PRAT, and TRAT. The user must be careful to match these inputs with the proper freestream conditions.

- If a portion of the fins in a fin set are located on the boattail or base, the boattail separation locations will be calculated and output at each fin roll angle. However, if the fins do not extend to the boattail the separation locations will be skipped.

- Results may be inaccurate if excessive extrapolation is required.   If extrapolation occurs, a warning message will be printed to the output file. To avoid extrapolation and minimize inaccuracy, the input parameters should be kept within the ranges shown in *Table 12*.

Table 7. NAMELIST AXIBOD Variables (Option 1 Inputs)

VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
NAMELIST AXIBOD - (Option 1 Inputs)				
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
TNOSE	-	Nose shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive) * POWER <or 2> (power law) HAACK <or 3> (L-V constrained) KARMAN <or 4> (L-D constrained)	-	OGIVE
POWER	-	Exponent, n, for power law shape: $(r/R)=(x/L)^n$	-	0.
LNOSE	-	Nose length	L	-
DNOSE	-	Nose diameter at base	L	1.
BNOSE	-	Nose bluntness radius or radius of truncation	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
LCENTR	-	Centerbody length	L	0.
DCENTR	-	Centerbody diameter at base	L	DNOSE
TAFT	-	Afterbody shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive)	-	CONICAL
LAFT	-	Afterbody length	L	0.
DAFT	-	Afterbody diameter at base (must be > 0 and not equal to DCENTR)	L	-
DEXIT	-	Nozzle diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. Gives "full" base drag DEXIT= exit gives base drag of annulus around exit only	L	-

Table 8. NAMELIST AXIBOD Variables (Option 2 Inputs)

NAMELIST AXIBOD – (Option 2 Inputs)				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
NX	-	Number of input stations (2 < NX < 50)	-	-
X	50	Longitudinal coordinates X(NX) must be the end of the body	L	-
R	50	Radius at each X station	L	-
Z	50	Distance of bent missile centerline from the un-bent centerline.	L	0
DISCON	20	Indices of X stations where the surface slope is discontinuous. Example: X(1)=0.,4.,8.,12.,16.,20., DISCON=3., defines a discontinuity at X=8. (third value)	-	-
BNOSE	-	Nose bluntness radius or radius of truncation	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
DEXIT	-	Nozzle diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. gives "full" base drag DEXIT= exit gives base drag of annulus around exit only	L	-
Note: LNOSE, DNOSE, LCENTR, DCENTR, LAFT, DAFT, and DEXIT should also be defined when using the AXIBOD option 2 input scheme. If they are not specified, Missile Datcom will use the default values defined in *Table 9*.				

Table 9. Default values assigned when using AXIBOD Option 2 input scheme

LNOSE	Length of the body segment to where the radius first reaches a maximum
DNOSE	The diameter at the first radius maximum
LCENTR	Length of the body segment
DCENTR	Diameter at the end of the body segment
LAFT	The second body segment
DAFT	Diameter at the base of the second body segment
DEXIT	Not defined (implies that base drag is not to be included in the axial force calculations)

Table 10. NAMELIST AXIBOD Body Vortex Model Variables

NAMELIST AXIBOD (Body Vortex Cloud Model Inputs)				
VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
BVCLOUD	-	Flag for body vortex cloud model: .TRUE. for enacting the vortex cloud model .FALSE. for enacting the legacy model	-	.FALSE.
BVGAM	-	Body vortex strength factor	-	1.0

Table 11. NAMELIST AXIBOD Base-Jet Plume Interaction Variables

NAMELIST AXIBOD (Base-Jet Plume Interaction Inputs)				
VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
BASE *	-	Flag for base plume interaction: .TRUE. for plume calculations .FALSE. for no plume calculations	-	.FALSE.
BETAN *	-	Nozzle exit angle	deg	-
JMACH **	20**	Jet Mach number at nozzle exit	-	-
PRAT **	20**	Jet/freestream static pressure ratio	-	-
TRAT **	20**	Jet/freestream stagnation temperature ratio	-	-
Note:				
* Only required if base plume interaction calculations are desired.				
** JMACH, PRAT and TRAT must be specified for each freestream Mach number or velocity input in Namelist $FLTCON.				

Table 12. Base-Jet Plume Interaction Parameter Limitations

Input Parameter	Symbol	Min Value	Max Value
Boattail shape	--	Cylinder, Cone, Ogive	
Boattail fineness ratio	L/D	0	2
Boattail terminal angle	β_E	0	12
Jet pressure ratio	P_J/P_{inf}	0	10
Freestream Mach number	M	2	5
Angle of Attack	α	0	8
Jet Mach number	M_J	M_{inf} -1	M_{inf} +1
Nozzle terminal angle	β_N	5	25
Jet diameter ratio	D_J / D_B	0.80	0.95
Jet temperature ratio	T_J / T_{inf}	4	10

Note: If input parameter is not between minimum and maximum values the code will extrapolate.
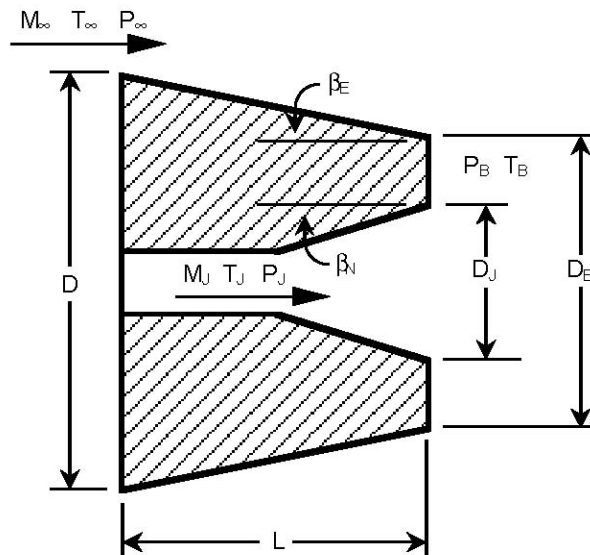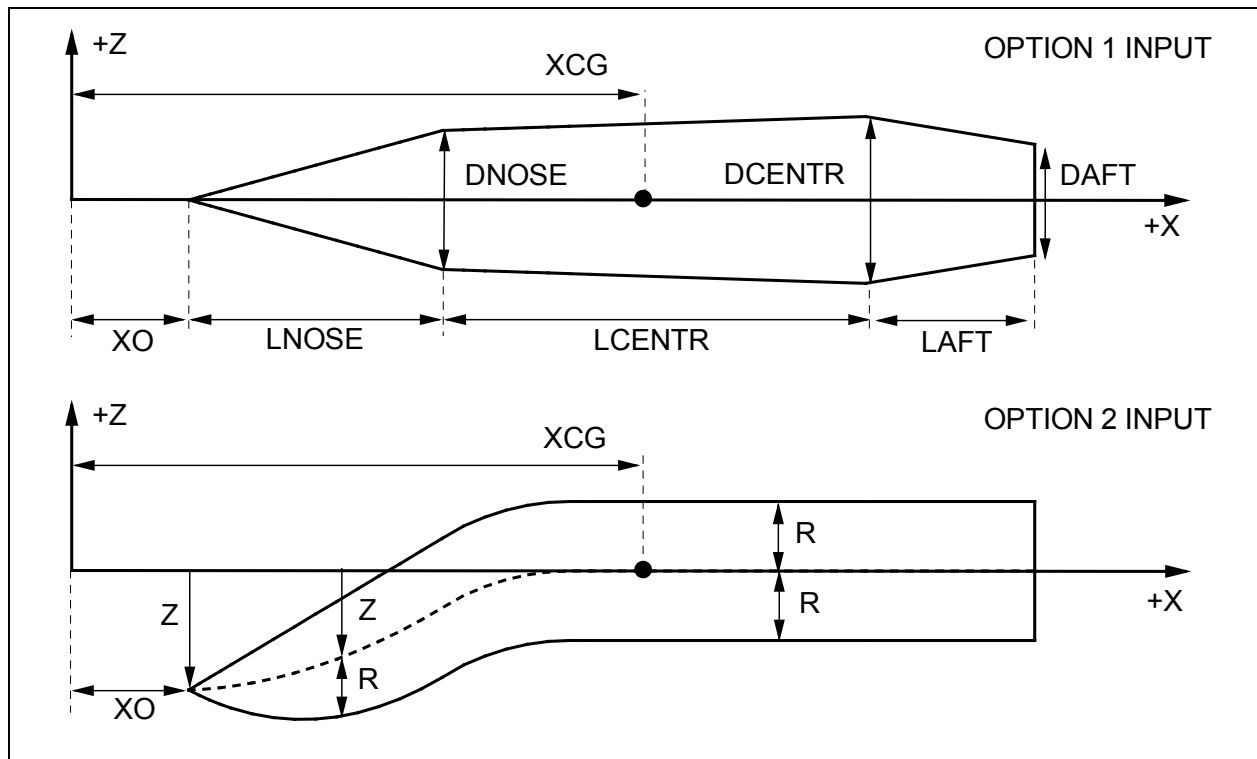
Figure 2. Axisymmetric Body Geometry Variables
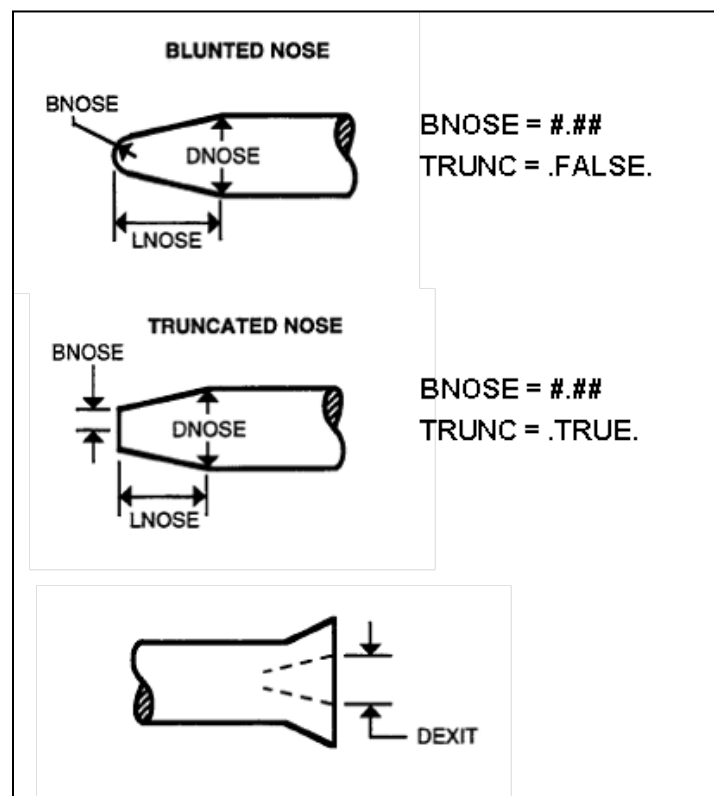
Figure 3. Blunted and Truncated Nose Variables

3.1.4 Namelist ELLBOD - Elliptical Body Geometry

Elliptically-shaped cross section bodies are defined using this namelist. The inputs are similar to those for the axisymmetric body geometry (AXIBOD), and are shown in **Table 13** and **Table 14**. The types of shapes available, and the limitations, are the same as those given for axisymmetric bodies. **However, the base-jet plume interaction input options in namelist AXIBOD are not available in namelist ELLBOD**. 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 ENOSE, ECENTR, and EAFT or ELLIP, H and W.

The new body vortex model is also available for the ELLBOD namelist. However,the vortices will be shed from the equivalent axisymmetric body that corresponds to the same cross-sectional area.

Table 13. NAMELIST ELLBOD Variables (Option 1 Inputs)

NAMELIST ELLBOD – (OPTION 1 INPUTS)				
VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
TNOSE	-	Nose shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive)* POWER <or 2> (power law) HAACK <or 3> (L-V constrained) KARMAN <or 4> (L-D constrained)	-	OGIVE
POWER	-	Exponent, n, for power law shape: $(r/R)=(x/L)^n$	-	0.
LNOSE	-	Nose length	L	-
WNOSE	-	Nose width at base	L	1.
ENOSE	-	Ellipticity at nose base (height/width)	-	1.0
BNOSE	-	Nose bluntness radius or radius of truncated nose. **BNOSE is relative to width (WNOSE). Height is calculated by BNOSE * ENOSE**	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
LCENTR	-	Centerbody length	L	0.
WCENTR	-	Centerbody width at base	L	WNOSE
ECENTR	-	Ellipticity at centerbody base (height/width)	-	1.0
TAFT	-	Afterbody shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive)	-	CONICAL
LAFT	-	Afterbody length	L	0.
WAFT	-	Afterbody diameter at base (must be > 0 and not equal to WCENTR)	L	-
EAFT	-	Ellipticity at aft body base (height/width)	-	1.0
DEXIT	-	Nozzle equivalent diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. gives "full" base drag DEXIT= exit gives base drag of annulus around exit only	L	-

Table 14. NAMELIST ELLBOD Variables (Option 2 Inputs)

NAMELIST ELLBOD – (Option 2 Inputs)				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
NX	-	Number of input stations (2 < NX < 50)	-	-
X	50	Longitudinal coordinates X(NX) must be the end of the body	L	-
H *	50	Body half-height at each X station		
W *	50	Body half-width at each X station	L	-
ELLIP *	50	Body height to width ratio at each X station	-	1.0
DISCON	20	Indices of X stations where the surface slope is discontinuous. Example: X(1) = 0., 4., 8., 12., 16., 20., DISCON=3., defines a discontinuity at X=8. (third value)	-	-
BNOSE	-	Nose bluntness radius or radius of truncated nose. **BNOSE is relative to width (WNOSE). Height is calculated by BNOSE * ENOSE**	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
DEXIT	-	Nozzle equivalent diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. gives "full" base drag DEXIT= exit gives base drag of annulus around exit only	L	-

Note:
 *   One of the following combinations is required:
        1. W and H
        2. W and ELLIP
        3. H and ELLIP				

OPTION 1

WNOSE

OPTION 2

H

ELLIP=H/W

W
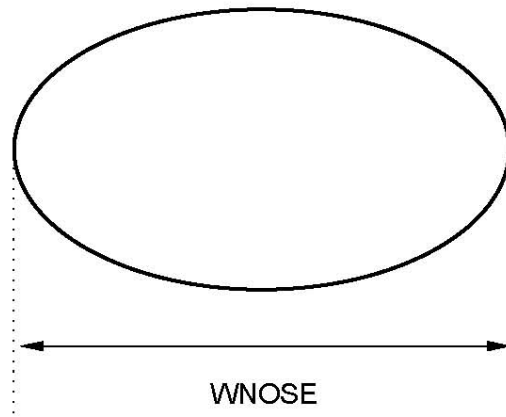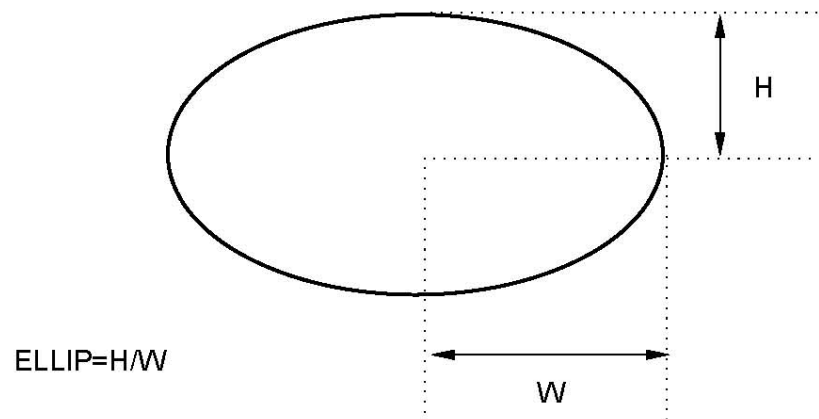
NOTE: Option 1 input WNOSE is TOTAL Width, Option 2 input W is HALF width

Figure 4. Elliptical Body Variables

3.1.5 Namelist PROTUB - Protuberance Geometry

Missile protuberances can be input using this namelist. Axial force coefficient is calculated for the protuberances and added to the body axial force coefficient. **Table 15** shows the inputs required. **Figure 5** shows the different protuberance shapes available. The following defines the inputs required for protuberance calculations:

- NPROT is the number of protuberance sets. A protuberance set is made up of protuberances at the same axial location with the same size and shape. Therefore, it is only necessary to describe the geometry of one individual protuberance per set. The maximum number of protuberance sets is 20.

- NLOC is the number of protuberances in each protuberance set. NLOC accounts for the number of identical protuberances located around the missile body at a given axial location.

- The following equation helps to clarify the relationship between NLOC and NPROT:
  NLOC(1)+NLOC(2)+NLOC(3)+ . . . +NLOC(NPROT) = (Total number of protuberances on the missile)

- The axial location of a protuberance (XPROT) should be input at the protuberance geometric centroid with respect to the coordinate system origin. An approximation of the centroid will be adequate for the analysis. The location is used to calculate the average boundary layer thickness over the protuberance length.

- For 8/08 and subsequent releases, the angular orientation of protuberances can be defined with PHIPRO. This variable is only used to calculate a moment from the axial force of the protuberance. If PHIPRO is not defined, no moment will be computed.

- VCYL, HCYL, BLOCK, and FAIRING type protuberances have 1 member. LUG types have 4 members and SHOE types have 3 members (**Figure 5**).

- All inputs for LPROT, WPROT, HPROT, and OPROT are in sequential order based upon the members specified with the protuberance type (PTYPE) input.

- The FAIRING type protuberance should always have a zero offset. The code will assume a zero offset even if a non-zero offset is input.

More complex protuberance shapes can be analyzed by a component build-up method. Each member is treated as a separate protuberance. Combinations of vertical cylinders, horizontal cylinders, and flat plates or blocks can be input at specified offsets from the missile body. If a FAIRING type protuberance is used in a component build-up, the offset should be zero. The user must manually add axial force of the individual members of the component build-up if the total protuberance axial force is desired.

Figure 6 shows an example input file for a missile with several protuberances.

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Table 15. NAMELIST PROTUB Variables

VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
NPROT	-	Number of protuberance sets (20 maximum)	-	0.
PTYPE	20	Protuberance set type: VCYL <or 1.> (cyl. perp. to flow) HCYL <or 2.> (cyl. aligned with flow) LUG <or 3.> (launch lug) ** SHOE <or 4.> (launch shoe)** BLOCK <or 5.> FAIRING <or 6.> (streamline half body)	-	-
XPROT	20	Longitudinal distance from coordinate system origin to the geometric centroid of the protuberance set	L	-
NLOC *	20	Number of protuberances in set	-	0.
PHIPRO***	400	Protuberance angular orientation measured clockwise from top vertical center looking forward	deg	-
LPROT	100	Length of protuberance	L	-
WPROT	100	Width of protuberance	L	-
HPROT	100	Height of protuberance	L	-
OPROT	100	Vertical offset of protuberance	L	0.

Note:

*   NLOC defines for identical protuberances (same size and shape) located around the body at the same axial station.

**   LUG type has 4 members. SHOE type has three members. LPROT, WPROT, HPROT, and OPROT must be specified for each member.

***   Angles for each protuberance are entered in the same order as the sets. An angle must be defined for each protuberance if this option is used.

 Example: A case has 2 protuberance sets with 3 protuberances in the first set (0, 120, and 240 degrees) and 2 in the second set (90, 270). Enter the data as NPROT = 2, NLOC = 3, 2, PHIPRO = 0, 120, 240, 90, 270,

NOTE: For the 2014 release, XPROT is now measured from the coordinate system origin.

Vertical Circular Cylinder Oriented Perpendicular To Flow (VCYL):

Horizontal Circular Cylinder Oriented In Flow Direction (HCYL):

Vertical Elliptical Cylinder Oriented Perpendicular To Flow With L> W Or W>L (VCYL):

Block (BLOCK):

Streamline Half Bodies (FAIRING):

Launch Shoe (SHOE):
1) Base Member
2) Vertical Member
3) Top Member

Launch Lug (LUG):
1) Base Member
2) & 3) Vertical Members
4) Top Member

TOP VIEW   3-D VIEW

L-Length   W-Width   H-Height

Note; Length, width, height, and offset must be input for each member of launch lug and launch shoe types

Note; Offset is the perpendicular distance from the missile mold line to the bottom of the protuberance or protuberance member

Figure 5. Available Protuberances Shapes

```
CASEID Protuberance Example Case
DIM IN
$FLTCON
   NALPHA =  4.00,
   ALPHA  =  0.00,   2.00,   4.00,   6.00,
   NMACH  =  3.00,
   MACH   =  0.40,   0.80,   2.00,
   REN    =  3.0000E+06,  3.0000E+06,  3.0000E+06,
$END
$REFQ
   SREF = 113.0973,
   LREF = 12.,
   XCG = 39.,
   BLAYER = TURB,
$END
$AXIBOD
   TNOSE =    OGIVE,
   LNOSE =  12.0000,
   DNOSE =  12.0000,
   TRUNC =  .FALSE.,
   LCENTR =  54.0000,
   DCENTR =  12.0000,
   TAFT =   CONICAL,
   LAFT =  12.0000,
   DAFT =   6.0000,
$END
$PROTUB
   NPROT = 4.,
   PTYPE = FAIRING, VCYL, SHOE, BLOCK,
   XPROT = 14., 22., 39., 56.,
   NLOC = 4., 4., 1., 1.,
   LPROT = 5.00, 1.00, 10.00, 10.00, 10.00, 0.50,
   WPROT = 2.00, 1.00, 4.00, 0.25, 1.00, 2.00,
   HPROT = 1.00, 0.50, 0.10, 0.75, 0.25, 1.50,
   OPROT = 0.00, 0.00, 0.00, 0., 0., 0.00,
   PHIPRO = 0., 90., 180., 270., 45., 135., 225., 315., 0., 0.,
$END
$FINSET1
   SECTYP =       HEX,
   SSPAN  =   0.0000,   9.0000,
   CHORD  =  14.0000,   8.0000,
   XLE    =  64.0000,
   SWEEP  =   0.0000,
   STA    =   1.0000,
   NPANEL =   4.0000,
   PHIF   =  45.0000, 135.0000, 225.0000, 315.0000,
$END
SAVE
NEXT CASE
```

Figure 6. Sample Missile with Protuberances

3.1.6 Namelist FINSETn - Define Fin Set n

Table 16 describes the variables needed to be input for fin set planform geometry descriptions. Optional fin cross-section inputs are described in **Figure 8**. Special user specified fin cross-sections can be input using the variables in **Figure 9**. **The user may specify up to nine sets of fins.** The variable "n" in the namelist specifies the fin set number. Fin sets are normally numbered sequentially from the front to the back of the missile 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. **Table 17** describes the input variables for HEX and ARC airfoil sections. **Table 18** described the input variables for USER airfoil sections. It is not possible in the current version to specify a cambered airfoil using the HEX or ARC airfoil definitions. Cambered airfoils can only be defined with a USER or NACA airfoil definition.

Note that the HEX and ARC airfoils can vary across the span, and must be defined at each span station for each fin set where they are used. If they are only defined once (at the root chord), the default values will be used at all other chord stations. NACA and USER airfoils cannot vary across the span and are only defined once for each fin set.

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 distance from the coordinate system origin to the chord leading edge (XLE). 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.

NOTE: For the 2014 release, XLE is now measured from the coordinate system origin, and thus is consistent with XHINGE (see *Figure 7*).

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.

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SSPAN(1):

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, so that the center of the root chord is on the surface mold line (see *Figure 7*).

CFOC:

Plain trailing edge devices may be modelled in Missile Datcom. This is accomplished via the CFOC array which is the flap chord to fin chord ratio, c_f/c. Trailing edge devices can be either full span or partial span subject to certain limits specified below. The trailing edge devices can not have a taper ratio greater than 1.0, and the hinge line must be straight regardless of the number of segments comprising the trailing edge device. A partial span trailing edge device is specified by setting CFOC=0 for those chord/span stations that are not part of the trailing edge device. Examples of acceptable and unacceptable geometries are shown in *Figure 8* as well as the corresponding input values for the variable arrays CFOC, CHORD and SSPAN. **A special case where the trailing edge device extends to the tip of a fin with a taper ratio of zero** is also shown in *Figure 8*. While any value of CFOC will result in the correct flap chord at the tip (since the tip chord is zero), the user is recommended to specify the same CFOC value from the adjacent inboard station. The user should also be aware of the following:

- All trailing edge deflection angles are measured with respect to the freestream and not relative to the hinge line. This becomes an important distinction as the hinge line sweep angle is increased. The following equation converts the hinge line sweep to the freestream sweep: $\delta_{flap} = \tan^{-1}\left(\tan\delta_{Hinge\,Line}\cos\Lambda_{Hinge\,Line}\right)$
- The variable SKEW does not apply to trailing edge devices.
- The hinge moments for trailing edge devices are not calculated.
- The increase in profile drag due to trailing edge deflection is not calculated.

Internally, Missile Datcom calculates the flap chord ratio between non-zero values of CFOC. For example,

CFOC = 0.0, 0.30, 0.25, 0.0,

This fin panel has three sections. The inner-most section has no flap as CFOC(1) is set to zero, The center panel has a flap that varies from 30% of the chord length at the spanwise beginning of the panel and varies linearly to 25% of the chord length at the end of the center panel. The outer-most panel has no flap as CFOC(4) is set to zero.

NPANEL, PHIF and GAM:

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 9*. 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 9*). For a circular body, this makes the fin perpendicular to the body mold line. For an elliptical body, the fin may not be perpendicular to the body mold line if there is no dihedral. The variable used to specify dihedral is GAM. GAM is positive if the panel tip chord is rotated clockwise when viewed from behind.

Different aerodynamics will be computed depending upon whether the FLTCON namelist variable PHI, or the FINSETn namelist variable PHIF, is used to roll the geometry. *Figure 10.* 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 entire configuration.

> **NOTE: When defining more than one fin set, the sets should be input in order as they are mounted on the body from nose to tail. This will ensure correct tracking of the fin-shed vortices. The code only tracks vortices from a given finset to a finset with a higher index. For example, vortices from FINSET2 will be tracked to FINSET3 and FINSET4 only, not to FINSET1.**

> **NOTE: The code will not track vortices between finsets if there is not a positive longitudinal gap between the trailing edge of the forward fin and leading edge of the aft fin. This allows multiple fin planforms to be defined at the same longutidinal station, for example, a horizontal and vertical tail of differing planform.**

NVOR:

For the 3/11 and subsequent releases, an input is available that allows the user to specify up to twenty vortices to be shed from each fin. A single value for the NVOR variable can be included within each fin set namelist, with each value denoting the number of vortices that will be shed from each fin within the specified fin set. Within the code, fin-shed vortices only affect fins that are downstream. Therefore NVOR does not need to be specified for the furthest downstream finset.

8/08 and prior releases were limited to modeling the fin-shed vortex wake with a single vortex; and to maintain backward compatibility with previous versions, if NVOR is not specified in the input file the 3/11 release will default to a single vortex per panel. If the default single vortex is used, there should be sufficient space between the forward fin trailing edge and aft fin leading edge to avoid violating the assumptions made by the aerodynamic computations. A single vortex model assumes that the vortex sheet is fully rolled up before it passes the control points of the next downstream set of fins. In reality, the roll up process takes several semispans of downstream movement to complete. If two fin sets are closer than this the results may be in error since the use of a single vortex filament model may introduce too much vorticity in certain regions. The closer the spacing, the larger the error may be. No algorithm error will result from too close a fin set spacing. For such cases where fin sets are close to each other, it is recommended that the user specify more than one shed vortex from the upstream fin set in order to better approximate the vortex sheet.

While increasing the number of fin-shed vortices in the calculations never makes the solution worse, it will cause the code run time to slow down. For a nominal configuration, changing NVOR from the default value of 1 to the maximum value of 20 will likely cause the run time to increase from the order of tenths of a second to the order of seconds.

Additional information regarding the improved fin-shed vortex models can be found in the AIAA papers corresponding to the 3/11 release listed on page vii.

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Table 16. NAMELIST FINSETn Variables – (Nominal Inputs)

NAMELIST FINSETn – (NOMINAL INPUTS)				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
SECTYP	-	Airfoil section type and numerical equivalent: HEX <or 0.> (Hexagonal section) NACA <or 1.> (Requires NACA control card) ARC <or 2.> (Circular arc section) USER <or 3.> (User defined airfoil section)	-	HEX
SSPAN	10	Semi-span locations. To automatically place fin on body moldline, use SSPAN(1)=0.0 with other values relative to fin root chord.	L	-
CHORD	10	Panel chord at each semi-span location	L	-
XLE	10	Distance from coordinate system origin to chord leading edge at each span location. Specify only XLE(1) if using SWEEP to define planform.	L	0.0
SWEEP	9	Sweepback angle at each span station.	deg	0.0
STA	9	Chord station used in measuring sweep: STA=0.0 is leading edge STA=1.0 is trailing edge	-	1.0
LER	10	Leading edge radius at each span station. Not required if SECTYP=NACA	L	0.0
NPANEL	8	Number of panels in fin set (1-8)	-	4.0
PHIF *	8	Roll angle of each fin measured clockwise from top vertical center looking forward	deg	*
GAM	8	Dihedral of each fin, positive when PHIF is Increased, see *Figure 9*.	deg	0.0
CFOC	10	Flap chord to fin chord ratio at each span station	-	1.0
NVOR	-	Number of vortices shed from each panel, up to a maximum of 20	-	1.0
Note: * If PHIF not used, panels will be evenly spaced around the body with the first fin placed at PHIF = 0.				

Multi-Segment Fin Placement

NOTE 1: XLE and XHINGE are both measured from the origin for the 2014 release

NOTE 2: Define either XLE(1) with various values of SWEEP
or multiple values of XLE with no SWEEP

Varying Body Radius Placement
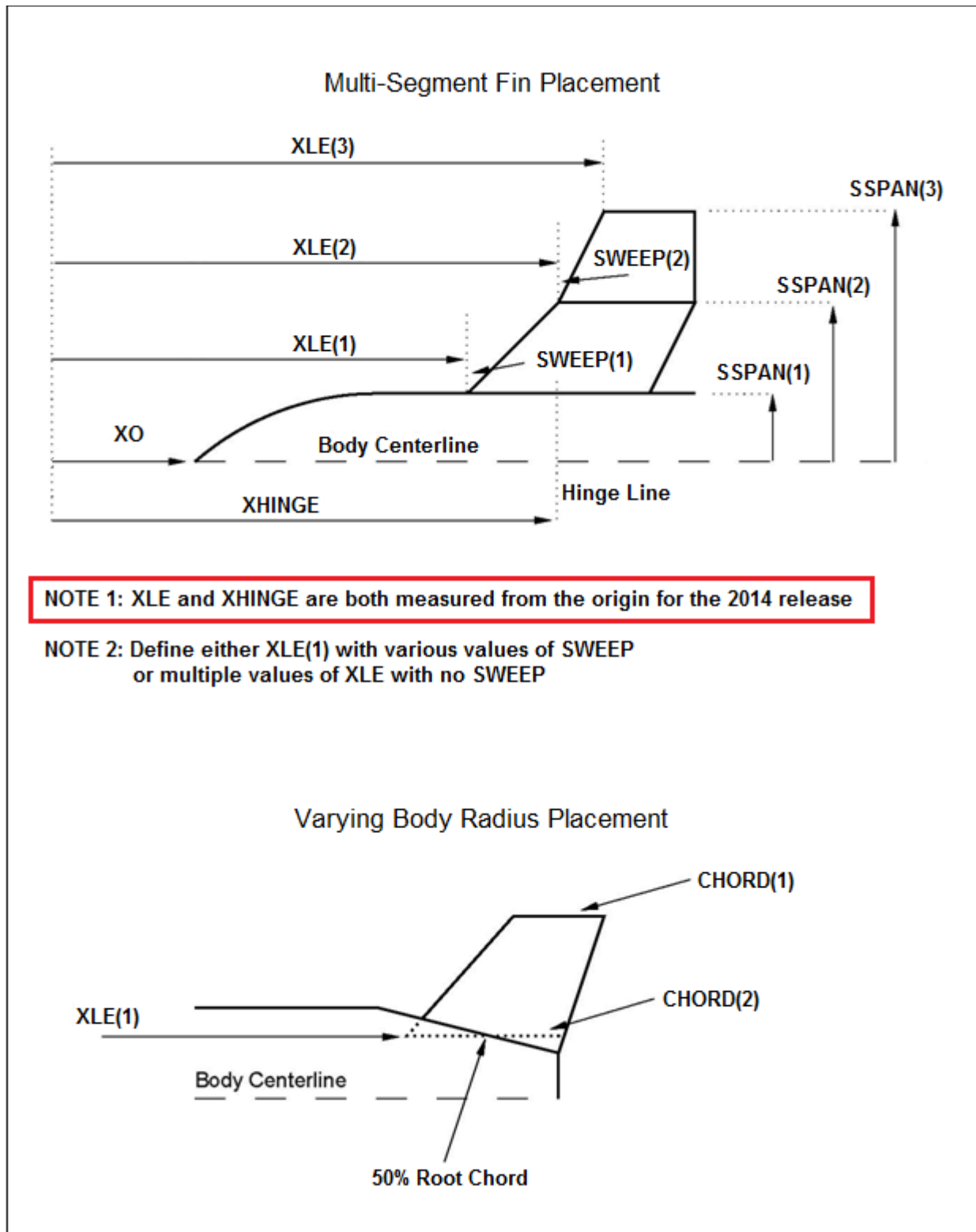
Figure 7. Fin Placement on Body

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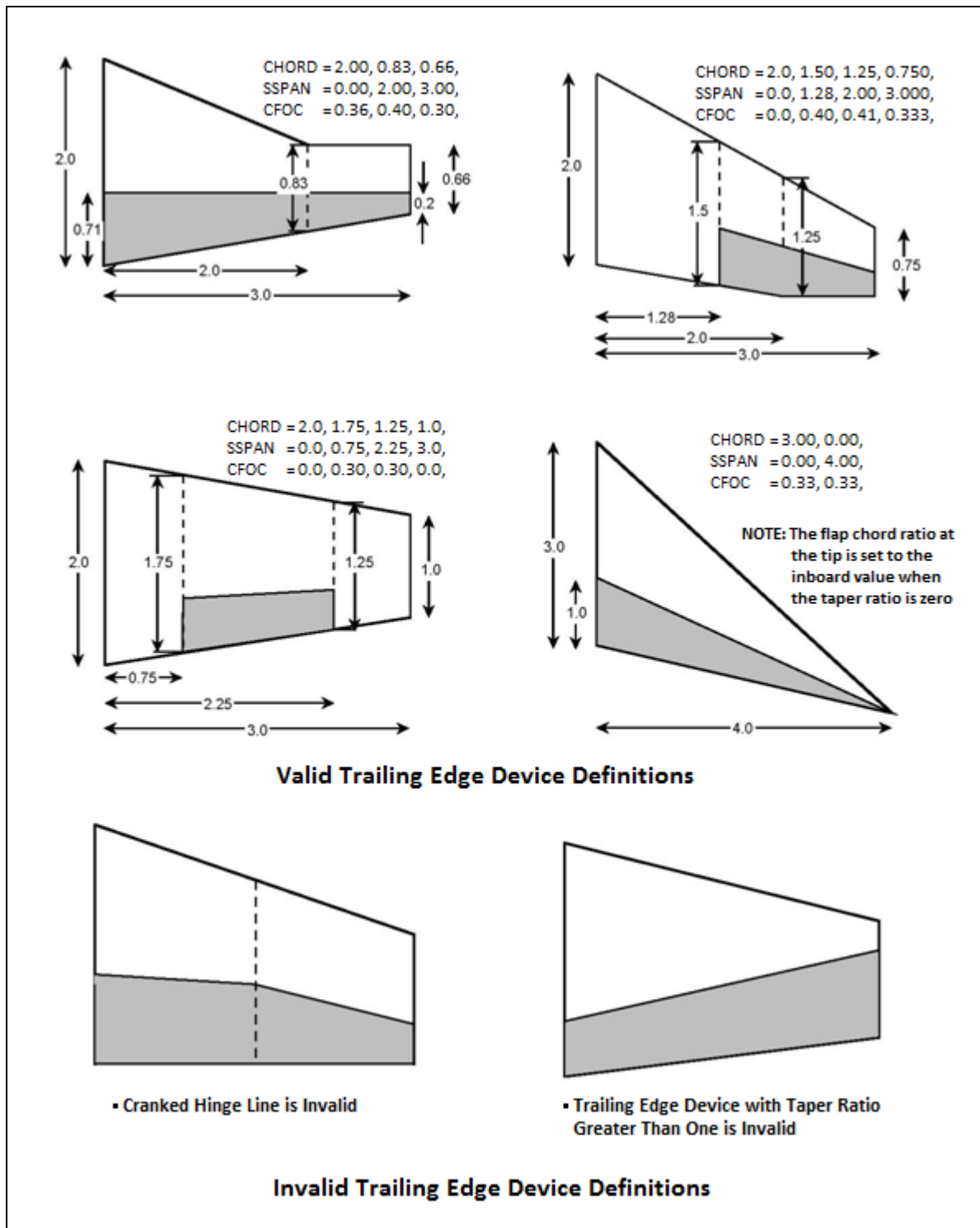
Figure 8. Definition of Trailing Edge Flaps
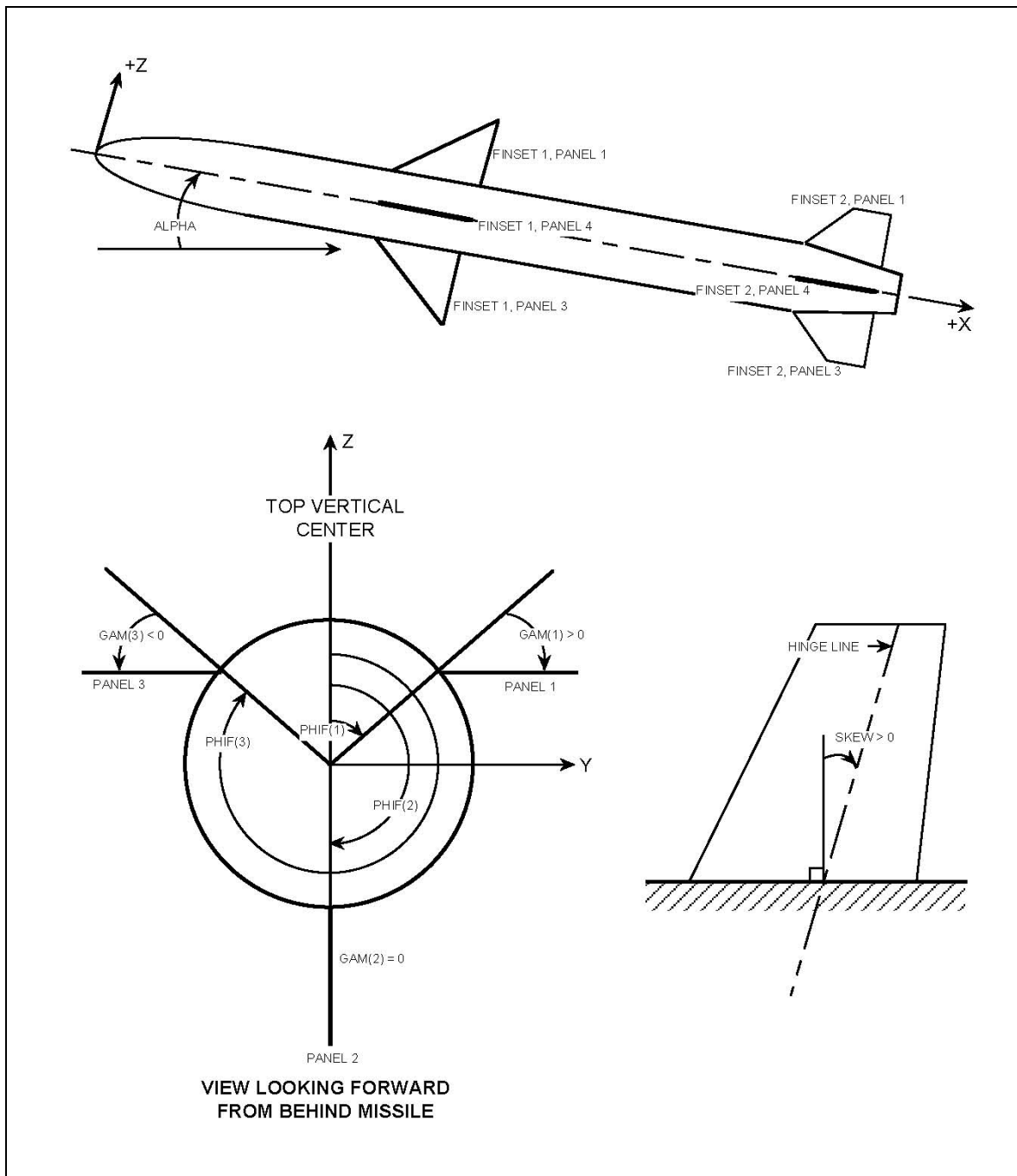
Figure 9. Fin Numbering and Orientation

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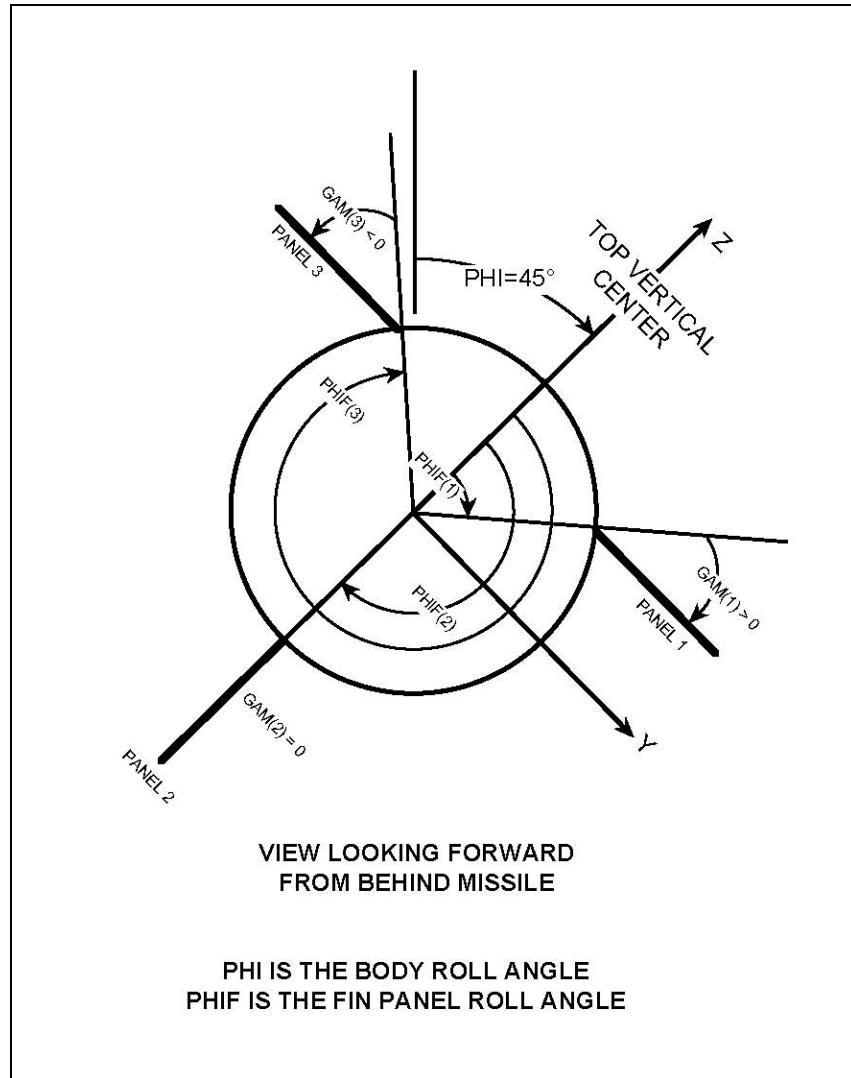
Figure 10. Roll Attitude vs Fin Orientation

Table 17. NAMELIST FINSETn Variables for SECTYP= HEX, ARC inputs

NAMELIST FINSETn - (SECTYP= HEX, ARC INPUTS)				
VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
ZUPPER	10	Thickness-to-chord ratio of upper surface. Input separate value for each span station.	-	0.025
ZLOWER	10	Thickness-to-chord ratio of lower surface. Input separate value for each span station.	-	ZUPPER
LMAXU	10	Fraction of chord from leading edge to maximum thickness of upper surface. Input separate value for each span station.	-	0.5
LMAXL	10	Fraction of chord from leading edge to maximum thickness of lower surface. Input separate value for each span station.	-	LMAXU
LFLATU*	10	Fraction of chord of constant thickness section of upper surface. Input separate value for each span station.	-	0.0
LFLATL*	10	Fraction of chord of constant thickness section of lower surface. Input separate value for each span station.	-	LFLATU
NOTE: All values must be defined at each span station				
* The variables LMAXL or LFLATL are not used in any aerodynamic calculation. They remain in the code for consistency and possible future use.				
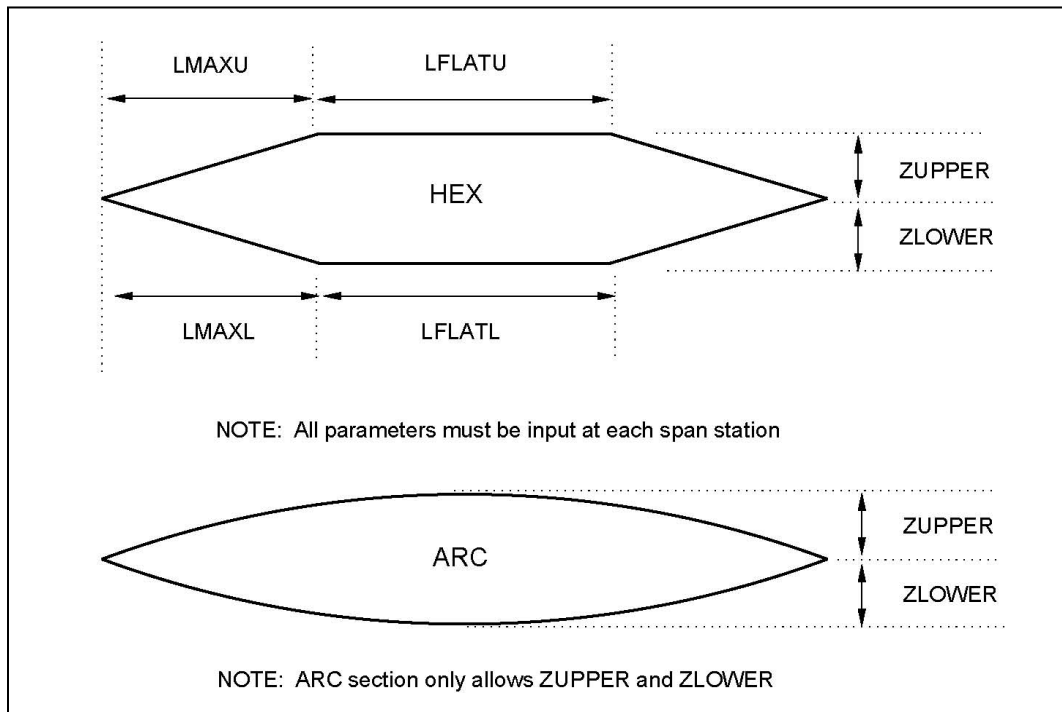
Figure 11. HEX and ARC Airfoil Variables

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Table 18. NAMELIST FINSETn - (SECTYP= USER Inputs)

| NAMELIST FINSETn - (SECTYP= USER INPUTS) |||||
VARIABLE NAME*	ARRAY SIZE **	DEFINITION	UNITS	DEFAULT
XCORD	50	Chord station, fraction of chord from leading edge. Values must range from 0.0 (leading edge) to 1.0 (trailing edge).	-	-
MEAN	50	Distance between the mean line and chord at each XCORD station in fraction of chord	-	-
THICK	50	Thickness-to-chord ratio at each XCORD station	-	-
YUPPER ***	50	Upper surface coordinates at each XCORD station in fraction of chord	-	-
YLOWER ***	50	Lower surface coordinates at each XCORD station in fraction of chord	-	-

Note:
 	
* One of the following combinations is required:
    1. XCORD, MEAN and THICK
    2. XCORD, YUPPER and YLOWER
 	
** Values apply to the entire fin set
 	
*** The airfoil must be closed at the leading and trailing edges, i.e. YUPPER=YLOWER or THICK=0.				
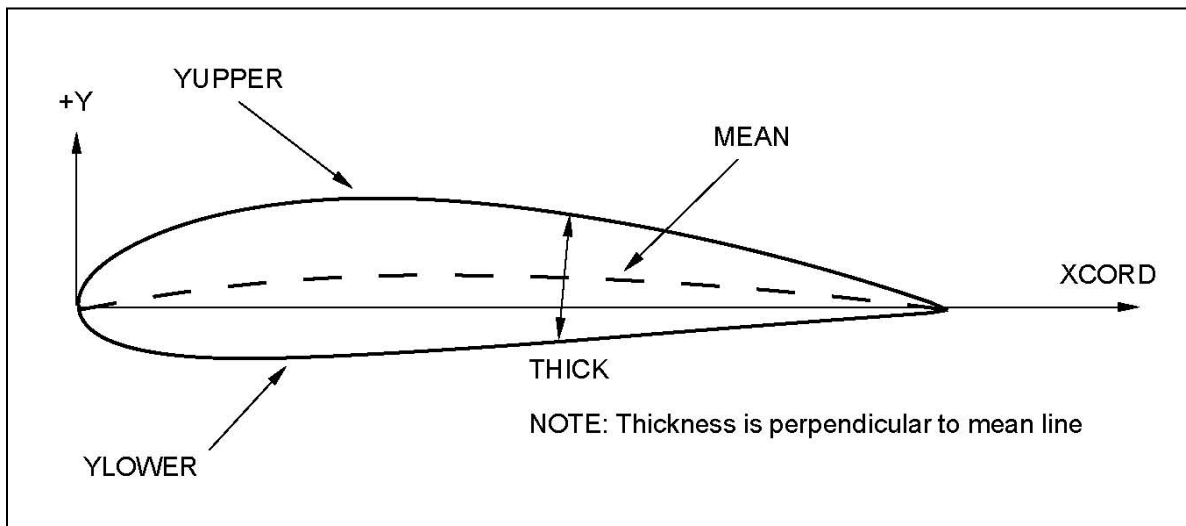
Figure 12. USER Airfoil Variables

Table 19. Airfoil Designation Using the NACA Control Card

NACA Series	Digit	Series Designation Rules	Samples (fin set and series follow NACA)
4 digit Series	1 2 3, 4	Maximum value of mean line ordinate (% chord) Distance to maximum thickness (tenths of chord) Maximum Thickness (% chord)	NACA-1-4-0008 NACA-2-4-2412
Modified 4 digit Series	1 – 4 5 6 7	Same as 4 digit series 'dash' (i.e. '-') Leading Edge Radius: 0 = sharp, 6 = normal radius Position of maximum thickness (tenths of chord), must be 2, 3, 4, 5, or 6	NACA-1-4-0012-62 NACA-2-4-4410-35 NACA-3-4-2004-04
5 digit Series	1 2, 3 4, 5	2/3 of design lift coefficient in tenths (2 indicates design Cl of 0.3) Twice distance to maximum camber point (% chord, i.e. 20 yields maximum camber point at 10% chord) Maximum thickness (% chord)	NACA-1-5-23012 NACA-2-5-42008
Modified 5 digit Series	1 – 5 6 7 8	Same as 5 digit series 'dash' (i.e. '-') Leading Edge Radius: 0 = sharp, 6 = normal radius Position of maximum thickness (tenths of chord), must be 2, 3, 4, 5, or 6	NACA-1-5-23012-32 NACA-2-5-42008-05 NACA-3-5-22406-63
1 Series	1 2 3 4 5, 6	Series designation Distance to minimum pressure point (tenths of chord, must be 6, 8, or 9) 'dash' (i.e. '-') Design lift coefficient in tenths Maximum thickness (% chord)	NACA-1-1-16-212 NACA-2-1-18-006 NACA-3-1-19-110
6 series	1 2 3 4 5, 6 7	Series designation Distance to minimum pressure point (tenths of chord) 'dash' (i.e. '-'): conventional section 'A': section straight from 80% chord to TE Design lift coefficient in tenths Maximum thickness (% chord) Optional mean line parameter (A=xx), must be decimal between 0.1 and 1.0 (default = 1.0)	NACA-1-6-64-005 NACA-2-6-61-205 A=0.6 NACA-3-6-65A010
Supersonic – Diamond or Circular Arc	1 2 3, 4 5 6	Type: 1 for diamond, 2 for circular arc 'dash' (i.e. '-') Distance to Maximum thickness (% chord) 'dash' (i.e. '-') Maximum thickness (% chord)	NACA-1-S-1-45-6 NACA-1-S-2-50-7
Supersonic – Hexagonal	1 2 3,4 4 5 6 7	Type: 3 'dash' (i.e. '-') Distance to Maximum thickness (% chord) 'dash' (i.e. '-') Maximum thickness (% chord) 'dash' (i.e. '-') Length of constant thickness section (% chord)	NACA-1-S-3-25-5-50
Note: Thickness can be expressed to nearest 0.01% chord for 1,4,5 and 6 series sections and nearest 0.1% chord for supersonic series. Examples: NACA-1-4-4411.75, NACA-1-S-1-50-7.5			

NOTE: A cambered section is oriented to provide positive normal force (body axes) at all angular orientations. For top and bottom fins ($\phi=0°$ or $180°$), the camber is oriented to provide positive side force.

3.1.7 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 **Table 20**. Note that the panel numbering scheme is assumed to be that shown in **Figure 9**. The array element of each deflection array corresponds to the panel number. Fin deflection angle sign convention is illustrated in **Figure 13**.

Table 20. NAMELIST DEFLCT Variables

NAMELIST DEFLCT				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
DELTA1	8	Deflection angles for each panel in fin set 1	deg	0.0
DELTA2	8	Deflection angles for each panel in fin set 2	deg	0.0
DELTA3	8	Deflection angles for each panel in fin set 3	deg	0.0
DELTA4	8	Deflection angles for each panel in fin set 4	deg	0.0
DELTA5	8	Deflection angles for each panel in fin set 5	deg	0.0
DELTA6	8	Deflection angles for each panel in fin set 6	deg	0.0
DELTA7	8	Deflection angles for each panel in fin set 7	deg	0.0
DELTA8	8	Deflection angles for each panel in fin set 8	deg	0.0
DELTA9	8	Deflection angles for each panel in fin set 9	deg	0.0
XHINGE *	4	Position of the panel hinge line for each fin set, measured from the coordinate system origin.	L	XO+XLE+ CR/2*
SKEW	4	Hinge line sweepback for each fin set	deg	0.0
Note:				
* Default is at one-half of the exposed root chord, as measured from the coordinate system origin.				
A POSITIVE DEFLECTION ANGLE PRODUCES A NEGATIVE BODY-AXIS ROLLING MOMENT AT ZERO ANGLE OF ATTACK.				
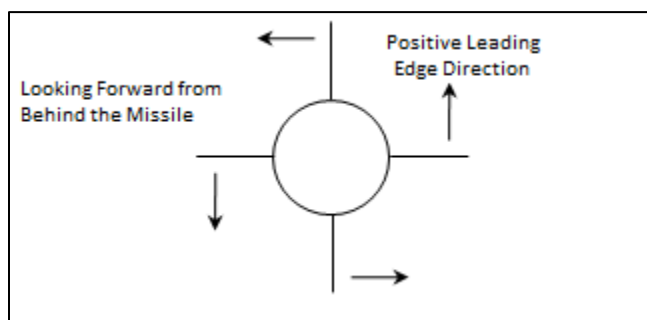
Figure 13. Fin Deflection Sign Convention

3.1.8 Namelist TRIM - Trim Aerodynamics

This namelist instructs the program to statically trim the vehicle longitudinally ($C_M=0$). The inputs are given in **Table 21**. Note that only one fin set can be used for trimming. The user specifies the range of deflection angles desired using DELMIN and DELMAX as well as the number of increments between the bounds using NINCR; 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 3.1.7**; 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. DELMIN must always be less than DELMAX.

Regardless of the values input for DELMIN, DELMAX and NINCR, as long as DELMIN and DELMAX span zero, a value of zero deflection will be run to avoid interpolation errors in that region. While the NINCR parameter allows the user to increase the resolution of the trim method to find a more exact trim value, such actions will increase the run time of the code. Therefore the user must decide on a value that satisfies their desired trim deflection angle accuracy as well as their run time preferences. If NINCR is not specified in the input file, the default value of 10 is used, which corresponds to the hard-coded value from previous releases.

NOTE: The logical variable ASYM has been deleted from the 8/08 and subsequent releases of the code.

Table 21. NAMELIST TRIM Variables

NAMELIST TRIM				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
SET	-	Fin set to be used for trim	-	1.
PANL1	-	.TRUE. if panel to be used	-	.FALSE.
PANL2	-	.TRUE. if panel to be used	-	.FALSE.
PANL3	-	.TRUE. if panel to be used	-	.FALSE.
PANL4	-	.TRUE. if panel to be used	-	.FALSE.
PANL5	-	.TRUE. if panel to be used	-	.FALSE.
PANL6	-	.TRUE. if panel to be used	-	.FALSE.
PANL7	-	.TRUE. if panel to be used	-	.FALSE.
PANL8	-	.TRUE. if panel to be used	-	.FALSE.
DELMIN *	-	Minimum negative deflection	Deg	-25.
DELMAX *	-	Maximum positive deflection	Deg	+20.
NINCR	-	Number of deflection cases		10.
* If only one of DELMIN or DELMAX is specified, the other will default to 0.				

3.1.9 Namelist INLET - Axisymmetric and 2-Dimensional Inlet Geometry

This namelist is used to model the inlet and diverter geometry. Axisymmetric, two-dimensional side mounted, and two-dimensional top mounted inlets can be described. The inlets may be covered or uncovered and oriented in any position about the missile body. Inlet normal force, pitching moment, side force, yawing moment, and axial force are calculated. The methods are valid for subsonic, transonic, and supersonic speeds. **Table 22** shows the INLET namelist inputs, and **Figures 14 - 17** show the inlet/diverter geometry for each type of inlet. The inlets may have a boundary layer diverter, be conformal (diverter height HDIV=0), or be semi-submerged (diverter height HDIV<0). The methods used for the inlets are the same regardless of whether the inlet has a diverter or is semi-submerged, but they are **not** applicable to chin inlets. The variable HDIV is used to determine whether a diverter exists. **Figure 17.** shows examples of two-dimensional and axisymmetric inlets that are conformal or semi-submerged.

- Inlet roll orientation uses the same convention as the fin panel roll orientation.

- Inlet height and width or inlet diameter is input at five axial locations described in **Figures 14 - 16**:

   1) leading edge or tip
   2) cowl lip leading edge
   3) midbody start
   4) boattail start
   5) boattail end

- If the inlet is covered (COVER=.TRUE.), no flow is allowed into the inlet. The inlet is plugged between stations 1 and 2, flush with the inlet face.

Inlet additive drag or spillage drag can be calculated for external compression inlets operating at off-design conditions ($M_\infty < M_{design}$) for Mach numbers greater than 1. Whenever flow spillage occurs, the mass flow ratio is less than one, and additive forces are generated on the deflected streamtube captured by the inlet. If the inlet operates on-design, the ramp shock lies on the inlet face and on the cowl lip. In these cases, the maximum mass flow ratio is one (zero spillage) and the minimum additive forces are zero.

- If the inlet is covered (COVER=.TRUE.), the additive drag calculations will be skipped.

- If ADD=.FALSE., or is not input the additive drag calculations will be skipped.

- Mass flow ratio (MFR) must be specified for each freestream Mach number or velocity given in namelist FLTCON. For Mach numbers less than 1, dummy values must be input for MFR. The user must be careful to match these inputs with the proper freestream conditions.

- The additive drag is calculated at zero angle of attack and assumed to remain constant for all angles of attack.

NOTE: **For the 2014 release, XINLT is now measured from the coordinate system origin (see *Figures 14-16*).**

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Table 22. NAMELIST INLET Variables

NAMELIST INLET				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
NIN	-	Number of inlets (maximum 20)	-	-
INTYPE	-	Type of inlet and numerical equivalent: 2DSIDE <or 1.> (side mounted 2-D) AXI <or 2.> (axisymmetric) 2DTOP <or 3.> (top mounted 2-D)	-	-
XINLT	-	Longitudinal distance from coordinate system origin to inlet leading edge	L	-
XDIV	-	Longitudinal distance from inlet leading edge to diverter leading edge	L	-
HDIV	-	Diverter height at leading edge. HDIV=0 defines conformal inlet. HDIV<0 defines semi-submerged inlet	L	-
LDIV	-	Length of diverter	L	-
PHI	20	Inlet roll orientations measured clockwise from top vertical center looking forward	deg	-
X *	5	Inlet longitudinal positions relative to inlet leading edge	L	-
H *	5	Inlet heights at the longitudinal positions. Not required if INTYPE=AXI	L	-
W *	5	Inlet widths at the longitudinal positions if INTYPE=2DTOP or 2DSIDE. Inlet diameters if INTYPE=AXI	L	-
COVER	-	Flag for covered inlet: .TRUE. (inlet covered) .FALSE. (inlet open)	-	.FALSE.
RAMP	-	External compression inlet ramp angle	deg	-
ADD	-	Flag for inlet additive drag: .TRUE. (calculate additive drag) .FALSE. (do not calculate)	-	.FALSE.
MFR	20	Mass flow ratio for each Mach number in namelist $FLTCON. 0.0 <MFR <1.0. Only required if ADD=.TRUE.	-	-
Note:* Specify X, H and W at five inlet locations as shown in *Figure 14 – 16*. (1) leading edge, (2) cowl lip, (3) midbody start (4) boattail start, and (5) boattail end. The inlet must be boattailed, meaning H(5)*W(5) < H(4)*W(4) for 2D inlets, or W(5) < W(4) for axisymmetric inlets.				

SIDE VIEW

XO

XDIV

LDIV

HDIV

RAMP

XINLT

(1)   (2) (3)   (4)   (5)
X(1) = 0
H(1) = 0

BOTTOM VIEW

(1)   (2) (3)   (4)   (5)

TOP VERTICAL
CENTER

0°

90° ——   —— 270°

PHI(1)=180°

180°

FRONT VIEW

INLET FACE

INPUT INLET LONGITUDINAL COORDINATE (X), HEIGHT (H),
AND WIDTH (W) AT EACH STATION

STATION   INLET POSITION
(1)   LEADING EDGE (X(1) & H(1) = 0)
(2)   COWL LIP LEADING EDGE
(3)   MIDBODY START
(4)   BOATTAIL START
(5)   BOATTAIL END

INLET ALONE

X

RAMP

(1)   (2) (3)   (4)   (5)

H

NOTES;
• INLET ROLL ORIENTATION IS SAME CONVENTION AS FIN ROLL ORIENTATION.
• RAMP IS THE EXTERNAL COMPRESSION RAMP ANGLE (SHOWN SHADED IN THE SIDE VIEW)
• HEIGHT OF THE DIVERTER IS SPECIFIED AT THE DIVERTER LEADING EDGE
• THE DIVERTER WIDTH IS EQUAL TO THE INLET WIDTH AT LDIV
• IF INLET IS COVERED (COVER=.TRUE.) A PLUG IS PLACED BETWEEN STATIONS 1 AND 2
  FLUSH WITH THE INLET FACE

Figure 14. Top-Mounted 2-D Inlet/Diverter Geometry

TOP VIEW

XO

XDIV   LDIV

HDIV

XINLT

(1)   (2) (3)   (4)   (5)

X(1) = 0
H(1) = 0

SIDE VIEW

RAMP

(1)   (2) (3)   (4)   (5)

X(1) = 0
H(1) = 0

TOP VERTICAL CENTER

$0°$

$PHI(1)=90°$

$90°$ — — $270°$

$PHI(2)=270°$

$180°$

FRONT VIEW

INLET FACE

INPUT INLET LONGITUDINAL COORDINATE (X), HEIGHT (H), AND WIDTH (W) AT EACH STATION

STATION	INLET POSITION
(1)	LEADING EDGE (X(1) & H(1) = 0)
(2)	COWL LIP LEADING EDGE
(3)	MIDBODY START
(4)	BOATTAIL START
(5)	BOATTAIL END

INLET ALONE

RAMP   X

(1)   (2) (3)   (4)   (5)

H

NOTES:
- INLET ROLL ORIENTATION IS SAME CONVENTION AS FIN ROLL ORIENTATION.
- RAMP IS THE EXTERNAL COMPRESSION RAMP ANGLE (SHOWN SHADED IN THE SIDE VIEW)
- HEIGHT OF THE DIVERTER IS SPECIFIED AT THE DIVERTER LEADING EDGE
- THE DIVERTER WIDTH IS EQUAL TO THE INLET WIDTH AT LDIV
- IF INLET IS COVERED (COVER=.TRUE.) A PLUG IS PLACED BETWEEN STATIONS 1 AND 2
  FLUSH WITH THE INLET FACE

Figure 15. Side-Mounted 2-D Inlet/Diverter Geometry

SIDE VIEW

XO

XDIV

LDIV

HDIV

RAMP

XINLT

(1) (2) (3)   (4)   (5)

X(1) = 0
W(1) = 0

BOTTOM VIEW

RAMP

(1)   (2) (3)   (4)   (5)

TOP VERTICAL
CENTER

0°

PHI(1)=120°

90°   270°

PHI(2)=250°

180°

(2 INLETS ROTATED AT 120° AND 250°)

FRONT VIEW

INLET FACE

W

INPUT INLET LONGITUDINAL COORDINATE (X), AND
DIAMETER (W) AT EACH STATION

STATION   INLET POSITION
(1)   TIP (X(1) & W(1) = 0)
(2)   COWL LIP LEADING EDGE
(3)   MIDBODY START
(4)   BOATTAIL START
(5)   BOATTAIL END

INLET ALONE

RAMP

X

(1)   (2) (3)   (4)   (5)

NOTES:
• INLET ROLL ORIENTATION IS SAME CONVENTION AS FIN ROLL ORIENTATION.
• RAMP IS THE EXTERNAL COMPRESSION CONE HALF-ANGLE (SHOWN SHADED
  IN THE SIDE VIEW)
• HEIGHT OF THE DIVERTER IS SPECIFIED AT THE DIVERTER LEADING EDGE
• THE DIVERTER WIDTH IS EQUAL TO THE INLET DIAMETER AT LDIV
• IF INLET IS COVERED (COVER=.TRUE.) A PLUG IS PLACED BETWEEN STATIONS 1 AND 2
  FLUSH WITH THE INLET FACE

Figure 16. Axisymmetric Inlet/Diverter Geometry

Figure 17. Geometry Definition for Conformal and Semi-Submerged Inlets

3.1.10 Namelist EXPR - Experimental Data Substitution

This namelist is used to substitute experimental data for the theoretical data generated by the program. The variables to be input are shown in **Table 23**. Use of namelist EXPR 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. 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) is 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.

- It is assumed that the coefficients in EXPR are for the same sideslip and/or aerodynamic roll as the case being run.

- Separate namelist EXPR must be specified for each Mach number.

- 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)

- 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.

Table 23. NAMELIST EXPR Variables

NAMELIST EXPR				
VARIABLE NAME	**ARRAY SIZE**	**DEFINITION**	**UNITS**	**DEFAULT**
MACH	-	Mach number	-	-
NALPHA	-	Number of angles of attack	-	-
ALPHA	100	Angle of attack for data	deg	-
SREF	-	Reference area for data	L^2	*
LREF	-	Longitudinal reference length for data	L	**
LATREF	-	Lateral reference length for data	L	LREF
XCG	-	Longitudinal C.G. for data	L	0.
ZCG	-	Vertical C.G. for data	L	0.
CONF	-	Configuration for which data is to be supplied and numerical equivalent: BODY <or 1.> (body) F1 <or 2.> (fin set 1) F2 <or 3.> (fin set 2) F3 <or 4.> (fin set 3) F4 <or 5.> (fin set 4) BF1 <or 6.> (body +1 fin set) BF12 <or 7.> (body +2 fin sets) BF123 <or 8.> (body +3 fin sets) BF1234 <or 9.> (body +4 fin sets)	-	-
CN	100	C_N data vs alpha	-	-
CM	100	C_M data vs alpha	-	-
CA	100	C_A data vs alpha	-	-
CY	100	C_Y data vs alpha	-	-
CSN	100	C_n data vs alpha	-	-
CSL	100	C_l data vs alpha	-	-
Notes: * Default is maximum body cross-sectional area. If no body is input, default is maximum fin panel area. ** Default is maximum body diameter. If no body is input, default is fin panel mean geometric chord.				

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:

- Outputting intermediate calculations (PART, BUILD, PRESSURES, PRINT AERO, PRINT GEOM, PLOT, PRINT VORTEX, NAMELIST)

- Selecting the system of units to be used (DIM, DERIV)

- Defining multiple cases, permitting the reuse of previously input namelist data or deleting namelists of a prior case (SAVE, DELETE, NEXT CASE)

- Adding case titles or comments to the input file and output pages (*, CASEID)

- Limits the calculations to longitudinal aerodynamics (NO LAT) or to fin-shed vortex effects (NO VORTEX)

3.2.1 Control Card - General Remarks

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, but may be subsequently changed in later cases:

DIM FT	DIM IN	DIM CM	DIM M
DERIV DEG	DERIV RAD	HYPER	NOGO
NO LAT	PLOT	SOSE	

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

BUILD	CASEID	DAMP	DELETE
NAMELIST	PART	PRESSURES	PRINT AERO
PRINT GEOM	SAVE	SPIN	TRIM
PRINT VORTEX	NO VORTEX		

The following control card is the ONLY control card that may be saved:

NACA

3.2.2 Control Card Definitions

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. This control card is effective only for the case in which it appears.

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). This control card is effective only for the case in which it appears.

DAMP

When DAMP control card is input dynamic derivatives are computed and the results output for the configuration. The longitudinal (pitch rate) derivatives are non-dimensionalized by the quantity (q*LREF/2*VINF). The lateral-directional (roll rate, yaw rate) derivatives are non-dimensionalized by the quantities (p*LATREF/2*VINF) and (r*LATREF/2*VINF) respectively. Dynamic derivatives for configuration components or partial configurations may be output using the PART or BUILD control cards respectively. This control card is effective only for the case in which it appears. Damping derivatives are NOT computed if PHI or BETA are non-zero.

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. The DELETE control cards are effective only for the case in which they appear.

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". All units for geometric length are associated with DIM while others listed below in **Table 24** are associated with whether DIM is in the English or Metric system.

Table 24. Units Required for Flight Condition Variables and Surface Roughness

Namelist	VAR	DIM IN	DIM FT	DIM CM	DIM M
FLTCON	REN	1/feet	1/feet	1/meter	1/meter
FLTCON	ALT	feet	Feet	meter	meter
FLTCON	VINF	feet/sec	feet/sec	meter/sec	meter/sec
FLTCON	PINF	pounds/feet^2	pounds/feet^2	Newtons/meter^2	Newtons/meter^2
FLTCON	TINF	Rankine	Rankine	Kelvin	Kelvin
REFQ	ROUGH	inches	Inches	centimeters	centimeters

DERIV DEG or DERIV RAD

All output derivatives are set to either degree (DERIV DEG) or radian (DERIV RAD) measure. The default setting is degree. The derivative units can be changed between cases by inputting additional DERIV cards.

HYPER

This control card causes the program to select the Newtonian flow method for bodies at any Mach number above 1.2. HYPER should normally be selected at Mach numbers greater than 6. HYPER causes carryover effects to be ignored for all supersonic Mach numbers.

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 delimiter	
6	1 to 9	Fin set number for which the airfoil designation applies
7	Any delimiter	
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 delimiter	
10 thru 80	Designation	Input designation (see Table 6); 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 19* 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. This control card is effective only for the case in which it appears.

<u>NEXT CASE</u>

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.

<u>NOGO</u>

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.

<u>NO LAT</u>

This control card inhibits the calculation of the lateral-directional derivatives due to sideslip angle, and the roll rate and yaw rate derivatives if the control card DAMP is selected. Large savings in computation time can be realized by using this option. This option is automatically selected when using TRIM.

<u>NO VORTEX</u>

This control card inhibits the calculation of body-shed and fin-shed vortex effects on downstream fin surfaces. This will rarely, if ever, improve the aerodynamic prediction; however, it may be useful to quantify the vortex effects by running cases with and without this control card and differencing the output.

<u>PART</u>

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. This control card is effective only for the case in which it appears.

<u>PLOT</u>

A data file for use with a post-processing plotting program is provided when this control card is used. A formatted file is written to tape unit 3 ("for003.dat").

<u>PRESSURES</u>

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. The body pressure output at positive angle of attack is written to tape unit 10 ("for010.dat"). The fin pressure output is written to tape unit 11 ("for011.dat"). The body pressure output and local Mach number at zero angle of attack is written to tape unit 12 ("for012.dat"). This control card is effective only for the case in which it appears.

PRINT AERO name

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

BODY   for body aerodynamics
FIN   for all fin set 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**. The PRINT AERO control cards are effective only for the case in which they appear.

PRINT GEOM name

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

BODY   for body geometry
FIN   for all fin set geometries
INLET   for inlet geometry

If PRINT GEOM BODY is selected the body contour coordinates (X,R,Z) used by the program for Mach numbers greater than 1.2 are written to tape unit 9 ("for009.dat"). This contour will contain many additional points in between the user specified input coordinates, and is useful for verifying that the DISCON values have been properly entered.

All options are automatically selected when the control card PART is used. The PRINT GEOM control cards are effective only for the case in which they appear.

PRINT VORTEX

A data file for each flight condition (NALPHA*NMACH files) is provided for use with a post-processing plotting program when this control card is used. A formatted file is written to "vpath*.dat" (ie. "vpath_a4.00_m1.60.dat" for α = 4°, Mach = 1.6 condition). Combined with the for022.dat file, vortex tracks in relation to the missile body can be visualized. This control card is effective only for the case in which it appears.

SAVE

The SAVE card saves namelist inputs from one case to the following case but not for the entire run (only effective for the case in which it appears). 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. To reinitialize a namelist that has been saved, the namelist must first be deleted using the DELETE control card. If the SAVE card is used, the total number of namelists read, including repeat occurrences of the same namelist name from all previous SAVEs, must not exceed 300. A warning message is output when this limit would be exceeded. For more information see the comments on SAVE listed under "typical set-up" below.

SOSE

The presence of this control card selects the Second-Order Shock Expansion method for bodies at supersonic speeds (Mach > 1.2). The Van Dyke hybrid method has been removed in the 10/14 release, and thus SOSE is the default method used.

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. As the SPIN card affects only damping derivatives, the DAMP card must also be specified to see any results in the output file. A PART or BUILD card must be input for body alone derivatives to be printed out. This control card is effective only for the case in which it appears. The following data will be computed and output:

Table 25. Magnus derivatives calculated with SPIN Control Card

CLLP	Rolling moment due to roll rate (roll damping)
CYPA	Side force derivative with respect to alpha due to roll rate
CNP	Is a polynomial fit of CNP1, CNP3, and CNP5 given by: CNP=CNP1*SIN(ALPHA)+CNP3*(SIN(ALPHA)**3)+CNP5*(SIN(ALPHA)**5)
CNP1	Zero yaw Magnus coefficient derivative per sin alpha
CNP3	Cubic Magnus moment coefficient per (sin alpha)**3
CNP5	Quintic Magnus moment coefficient per (sin alpha)**5
CNPY5	5 degree secant slope of Magnus moment coefficient derivative (at five degree yaw) per sin alpha

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. This control card is effective only for the case in which it appears.

COMMENT CARDS

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 18 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.
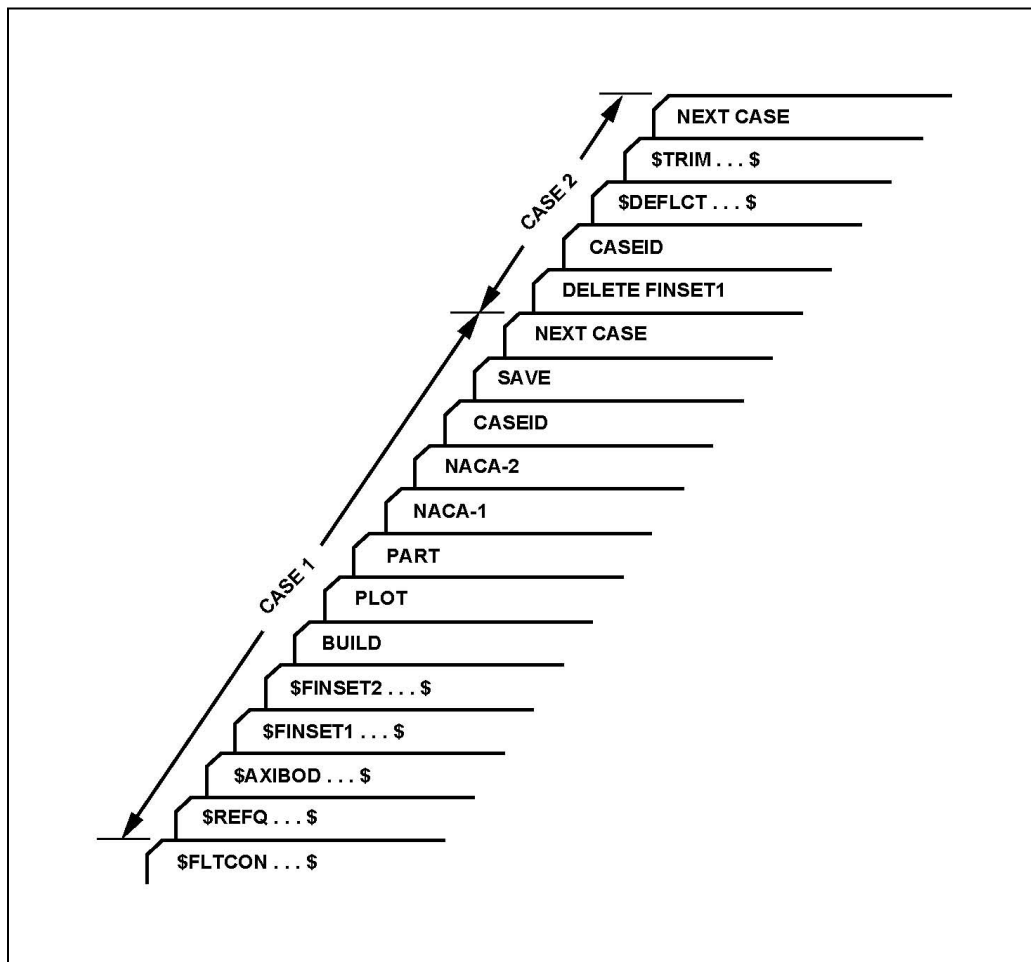
Figure 18. Typical "Stacked" Case Set-up.

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4.0 OUTPUT DESCRIPTION

This section describes the types of output available from the code. In many 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 three 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 be 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.

The remainder of the section describes each of these output data. Examples of each output page are also included and were created from the example problems, described in Appendix A, 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. A complete listing of the output files is given in **Table 1**.

4.1 NOMINAL OUTPUT

Without the use of any program options the code will provide three types of output (written to for006.dat). 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:

- No terminating comma on a namelist input card

- No terminating "$" or "$END" on a namelist input ("&" on IBM systems)

- 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. Missile Datcom inputs are either REAL or LOGICAL regardless of the variable name assigned.

An example output from CONERR is shown in **Figure 19**. This figure illustrates the array of input errors checked by CONERR. Several additional features of the output are as follows:

- All user defined input cards are assigned a sequential "line number". This serves to identify user inputs from the code generated inputs (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.

- 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.

- In many cases alphanumeric constants are available (see **Table 2**). 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.

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 20 shows the first page of outputs for a case without CONERR detected errors. Then **Figure 21** 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

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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 set-up 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 correction taken by the code. In some cases a correction is not possible, and an appropriate error message and a suggestion for correcting the error is provided. If a correction is not possible the case will not run.

4.1.3 Case Total Configuration Aerodynamic Output Summary

As shown in *Figure 22,* 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, and are given in the body axis system. 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 position from the moment reference point divided by reference length, positive values for XCP correspond to being forward of the moment reference point.

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 may not be numerically identical for those conditions which should produce identical results if they were both calculated by the same method.

A 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 23***. 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 format 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.

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 24*** 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 25***, the description of the panel airfoil section is provided. Following that, shown in ***Figure 26***, 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.

An input file including a 2-D inlet is provided in ***Figure 27***. If an airbreathing inlet is specified the output is similar to that in ***Figure 28***. 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.

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4.2.2 Aerodynamic Partial Output

The output on the configuration aerodynamics is most extensive when PRINT AERO or PART is specified. 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 flow analysis of the airfoil section using conformal mapping. If a configuration has inlets additional partial output is included to summarize the inlet external aerodynamics.

If base-jet plume interaction calculations are specified (BASE=.TRUE. in namelist AXIBOD), then there will be one or two separate pages of output. **Figure 29** shows an example of the first page of output. This page will always be printed if BASE=.TRUE. The base pressure coefficient, axial force coefficient, and freestream pressure and temperature ratios are shown versus angle of attack. Also, the incremental forces and moments due to separation are shown versus angle of attack. If extrapolation of the base pressures and separation conditions database occurs, a warning message is printed explaining what input variable required extrapolation. A second page of output containing the boattail separation parameters will be printed if there are any fins on the missile boattail. The separation location aft of the nose and the Mach cone angle are shown versus angle of attack for each panel on the fin set. This output is provided if the PRINT AERO BODY or PART control card is input.

The protuberance partial output is printed if PRINT AERO BODY or PART is used. This output will only be shown if the namelist PROTUB is present in the input file. **Figure 30** is an example of the protuberance output. Protuberance type, location, number, and axial force coefficient are listed for each protuberance set. The total axial force coefficient or zero lift drag coefficient is printed at the bottom of the page.

As shown in **Figure 31**, 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 32 details the fin normal force and pitching moment 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 33** illustrates the fin axial force contributors.

The analysis by the Airfoil Section Module is provided in a format similar to **Figure 34**. 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.

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Figure 35 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 can include additive drag results if the user inputs the additive drag calculation flag. The maximum mass flow ratio is printed at the bottom of the page if the additive drag is calculated. If additive drag cannot be calculated, a warning message is printed.

After the aerodynamic details for each component of the configuration are output, the aerodynamic calculations for the synthesis of the complete configuration follows. For the example case, fin set 1 results would be followed by fin set 2 results for each of the following outputs:

- "FIN SET IN PRESENCE OF THE BODY" - This summarizes the aerodynamic incrementals of each set of fins with the influence of the body. *Figure 36 and 37* presents the example of this output. The data represented in *Figure 36* include fin set forces and moments from the effect of body-on-fin component interference. The data represented in *Figure 37* include the equivalent angle of attack of each panel within the fin set along with the panel normal force coefficient (in panel reference frame) and the longitudinal and lateral center of pressure values which are non-dimensionalized by panel root chord and semi-span, respectively. The sign convention corresponds to the fin deflection angles as shown in *Figure 13*, and is as follows: a positive panel normal force, hence, equivalent angle of attack, produces a negative roll moment. Therefore, 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.

- "CARRYOVER INTERFERENCE FACTORS" - This page of partial output summarizes the carryover factors. The results shown in *Figure 38* 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-W(B) (wing in presence of the body carryover due to panel deflection), KK-B(W) (body in presence of the wing carryover due to panel deflection), and XCP-B(W) (body in presence of the wing carryover center of pressure). This output is repeated for the body plus each additional aft fin set in the configuration. If additional fin sets are present on the configuration additional pages are output with each one successively included.

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 39*. 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 40 illustrates the panel hinge moment coefficients computed when the control card PRINT AERO HINGE or PART is 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.

4.3 DYNAMIC DERIVATIVES

As shown in **Figure 41**, 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. All derivatives are in the body axis system, with assumed rates of rotation also in that system. The coefficients provided are as follows:

CMQ
Pitching moment coefficient derivative with pitch rate:
$$C_{mq} = \partial C_m / \partial(q * lref / 2 / V)$$

CNQ
Normal force coefficient derivative with pitch rate:
$$C_{Nq} = \partial C_N / \partial(q * lref / 2 / V)$$

CAQ
Axial force coefficient derivative with pitch rate:
$$C_{Aq} = \partial C_A / \partial(q * lref / 2 / V)$$

CMAD
Pitching moment derivative with rate of change of angle of attack:
$$C_{m\dot{\alpha}} = \partial C_m / \partial(\dot{\alpha} * lref / 2 / V)$$

CNAD
Normal force derivative with rate of change of angle of attack:
$$C_{N\dot{\alpha}} = \partial C_N / \partial(\dot{\alpha} * lref / 2 / V)$$

CLLP
Rolling moment coefficient derivative with roll rate:
$$C_{lp} = \partial C_l / \partial(p * latref / 2 / V)$$

CLNP
Yawing moment coefficient derivative with roll rate:
$$C_{np} == \partial C_n / \partial(p * latref / 2 / V)$$

CYP
Side force coefficient derivative with roll rate:
$$C_{Yp} == \partial C_Y / \partial(p * latref / 2 / V)$$

CLLR
Rolling moment coefficient derivative with yaw rate:
$$C_{lr} == \partial C_l / \partial(r * latref / 2 / V)$$

CLNR
Yawing moment coefficient derivative with yaw rate:
$$C_{nr} == \partial C_n / \partial(r * latref / 2 / V)$$

CYR
Side force coefficient derivative with yaw rate:
$$C_{Yr} == \partial C_Y / \partial(r * latref / 2 / V)$$

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. Also, the complete set of six force and moment components due to each of the three body axis rotations is written to the for042.dat file

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. Both fixed and user defined format data files can be created with the addition of control cards.

4.4.1 for003.dat

Fixed format aerodynamic data is output as an external data file with the addition of the PLOT control card. The PLOT data are written to file "for003.dat". 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. If TRIM is specified, the user can selectively print the six untrimmed static aerodynamic tables used in the trim process to file for003.dat. An example is shown in **Figure 42**. The code computes the six-component aerodynamics at the calculated 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 or more deflection angles, up to 100 angles of attack, and up to 20 Mach numbers per input case.

4.4.2 for009.dat

Body contour coordinates (X,R,Z) will be printed to tape unit 9 when the user specifies PRINT GEOM BODY or PART. These body coordinates are used by the program for Mach numbers greater than 1.2 for the Hybrid, SOSE or HYPER routines. This contour will contain many additional points in between the user specified input coordinates, and is useful for verifying that the DISCON values have been properly entered. **Figure 43** shows an example.

4.4.3 for010.dat, for011.dat & for012.dat

If the Mach number is supersonic (M > 1.2), the user has the option to write out the surface pressure distributions over the body and fins. This option is selected only through the addition of the control card PRESSURES. Since two body alone supersonic methods are available (Second-Order Shock Expansion (SOSE) and Newtonian flow (HYPER)) the capability exists to output the pressure distribution data from either 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 SOSE method is used. Because of the nature of the calculations, body alone pressures are printed for angles of attack less than or equal to 15 degrees when using the SOSE technique.

The primary body pressure distribution output is written to file "for010.dat". Local Mach number data is computed using the SOSE method only, and is written to file "for012.dat" if the PRESSURES option is used. 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.

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 output. The fin pressure data is written to file "for011.dat".

Examples of these files are shown in *Figures 44 – 46*.

4.4.4 for020.dat & for021.dat

The output files "for020.dat" and "for021.dat" are generated if the control card PLOT is used. The file "for020.dat" generates data used by software developed with Adaptive Modeling Language (AML). The file "for021.dat" generates data that can be used with the Aviator Visual Design Simulator (AVDS). *Tables 26 and 27* list the contents of these files.

4.4.5 for022.dat

The "for022.dat" output file is always generated and is a configuration geometry file compatible with the commercial software package Tecplot. Only geometry for the body and fins are printed. No geometry is provided for inlets or probuterances. Fin planforms are correctly defined; however, the cross sections are always defined as double wedges. *Figure 47* shows an example.

4.4.6 for042.dat & for043.dat

The output file "for042.csv" is always generated and contains the majority of the data supplied in the "for006.dat" file in a row/column file format with a header indicating the contents of the columns. *Table 28* lists the dictionary for the contents of the "for042.csv" file.

The output file "for043.csv" is generated if the control card PRINT AERO SYNTHS, PRINT AERO BEND, PRINT AERO HINGE, or PART is used. This file generates fin related variables in a comma delimited format. *Table 29* lists the contents of this file.

4.4.7 vpath*.dat

If the PRINT VORTEX control card is specified, a "vpath*.dat" Tecplot-compatible output file will be written for each flight condition (alpha-Mach combination). This formatted file (ie. "vpath_a4.00_m1.60.dat" for α = 4, Mach = 1.6 condition) includes vortex X,Y,Z position as well as strength for every fin-shed vortex used within the code. Combined with the for022.dat file, vortex tracks in relation to the missile body can be visualized. An example is provided in *Figure 48*.

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Table 26. Dictionary listing for the FOR020.DAT file

MACH	Mach Number
ALT	Altitude
VINF	Freestream Velocity
Q	Dynamic Pressure
ALPHA	Angle of Attack
BETA	Sideslip Angle
CN	Normal Force Coefficient
CM	Pitching Moment Coefficient
CA	Axial Force Coefficient
CY	Side-Force Coefficient
CLN	Yawing Moment Coefficient
CLL	Rolling Moment Coefficient
CL	Lift Force Coefficient
CD	Drag Force Coefficient
CL/CD	Lift-to-Drag Ratio
CNA	Normal Force Coefficient change wrt Angle of Attack
CMA	Pitching Moment Coefficient change wrt Angle of Attack
CYB	Side-force Coefficient change wrt Sideslip Angle
CLNB	Yawing Moment Coefficient change wrt Sideslip Angle
CLLB	Rolling Moment Coefficient change wrt Sideslip Angle
CMQ	Pitching Moment Coefficient change wrt Pitch Rate
CLNR	Yawing Moment Coefficient change wrt Yaw Rate
CLLP	Rolling Moment Coefficient change wrt Roll Rate

Table 27. Dictionary listing for the FOR021.DAT file

Row 1	
MACH	Mach Number
ALT	Altitude
BETA	Sideslip Angle
DELTA	Fin deflections
Row 2	
NALPHA	Number of angles of attack
NCOL	Number of columns of data in row 3
NCDD	Number of columns of dynamic derivatives in row 3
Row 3	
ALPHA	Angle of Attack
CN	Normal Force Coefficient
CM	Pitching Moment Coefficient
CA	Axial Force Coefficient
CY	Side-Force Coefficient
CLN	Yawing Moment Coefficient
CLL	Rolling Moment Coefficient
CNAD	Normal Force Coefficient change wrt Angle of Attack Change Rate
CMAD	Pitching Moment Coefficient change wrt Angle of Attack Change Rate
CNQ	Normal Force Coefficient change wrt Pitch Rate
CMQ	Pitching Moment Coefficient change wrt Pitch Rate
CAQ	Axial Force Coefficient change wrt Pitch Rate
CYQ	Side-force Coefficient change wrt Pitch Rate
CLNQ	Yawing Moment Coefficient change wrt Pitch Rate
CLLQ	Rolling Moment Coefficient change wrt Pitch Rate
CNP	Normal Force Coefficient change wrt Roll Rate
CMP	Pitching Moment Coefficient change wrt Roll Rate
CAP	Axial Force Coefficient change wrt Roll Rate
CYP	Side-force Coefficient change wrt Roll Rate
CLNP	Yawing Moment Coefficient change wrt Roll Rate
CLLP	Rolling Moment Coefficient change wrt Roll Rate
CNR	Normal Force Coefficient change wrt Yaw Rate
CMR	Pitching Moment Coefficient change wrt Yaw Rate
CAR	Axial Force Coefficient change wrt Yaw Rate
CYR	Side-force Coefficient change wrt Yaw Rate
CLNR	Yawing Moment Coefficient change wrt Yaw Rate
CLLR	Rolling Moment Coefficient change wrt Yaw Rate
Row 3 will be printed for each angle of attack in the case. Rows 1-3 are then repeated for each succeeding Mach number in the case. Damping derivatives will only be printed if the altitude is zero AND the sideslip angle if zero.	

Table 28. Dictionary listing for the FOR042.CSV file

CASE	Case Number currently being run
RAD?	Defines units as radians or degrees (0 = degrees, 1 = radians)
TRIM?	Defines case as trim case or not (0 = not trim, 1 = trim)
MACH	Mach Number
RE	Reynolds Number
ALT	Altitude
Q	Dynamic Pressure
BETA	Sideslip Angle
PHI	Roll Angle
SREF	Reference Area
XCG	Distance from coordinate system origin to Center of Gravity
XMRP	Distance from coordinate system origin to Center of Gravity (in calibers)
LREF	Longitudinal Reference Length
LATREF	Latitudinal Reference Length
ALPHA	Angle of Attack
CN	Normal Force Coefficient
CM	Pitching Moment Coefficient
CA	Axial Force Coefficient
CA_0B	Axial Force Coefficient with no base drag included
CA_FB	Axial Force Coefficient with full base drag included
CY	Side-Force Coefficient
CLN	Yawing Moment Coefficient
CLL	Rolling Moment Coefficient
CL	Lift Force Coefficient
CD	Drag Force Coefficient
CL/CD	Lift-to-Drag Ratio
X-C.P.	Distance from coordinate system origin to Center of Pressure
CNA	Normal Force Coefficient change wrt Angle of Attack
CMA	Pitching Moment Coefficient change wrt Angle of Attack
CYB	Side-force Coefficient change wrt Sideslip Angle
CLNB	Yawing Moment Coefficient change wrt Sideslip Angle
CLLB	Rolling Moment Coefficient change wrt Sideslip Angle
CNQ	Normal Force Coefficient change wrt Pitch Rate
CMQ	Pitching Moment Coefficient change wrt Pitch Rate
CAQ	Axial Force Coefficient change wrt Pitch Rate
CNAD	Normal Force Coefficient change wrt Angle of Attack change Rate
CMAD	Pitching Moment Coefficient change wrt Angle of Attack change Rate
CYQ	Side-force Coefficient change wrt Pitch Rate
CLNQ	Yawing Moment Coefficient change wrt Pitch Rate
CLLQ	Rolling Moment Coefficient change wrt Pitch Rate
CYR	Side-force Coefficient change wrt Yaw Rate
CLNR	Yawing Moment Coefficient change wrt Yaw Rate
CLLR	Rolling Moment Coefficient change wrt Yaw Rate
CYP	Side-force Coefficient change wrt Roll Rate
CLNP	Yawing Moment Coefficient change wrt Roll Rate

Table 29. Dictionary listing for the FOR043.CSV file

CASE	Case Number Currently Being Run
MACH	Mach Number
ALPHA	Angle of Attack
CNx_IPB	Finset x Normal Force Coefficient in Presence of Body
CMx_IPB	Finset x Pitching Moment Coefficient in Presence of Body
CAx_IPB	Finset x Axial Force Coefficient in Presence of Body
CYx_IPB	Finset x Side Force Coefficient in Presence of Body
CLNx_IPB	Finset x Yawing moment Coefficient in Presence of Body
CLLx_IPB	Finset x Rolling Moment Coefficient in Presence of body
PBMx_Py	Fin y of Finset x Bending Moment Coefficient
PHMx_Py	Fin y of Finset x Hinge Moment Coefficient
PCNx_Py	Fin y of Finset x Normal Force Coefficient in Panel Coordinate System (ie. CN is normal to the panel)

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****
              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 ************************

       CASEID Missile Datcom ERROR flags              comments

          NALPH  =  4.0000,                   << Mis-spelled
          ** ERROR **  1*A  0*B  0*C  0*D  0*E  0*F
          * FATAL ERROR *

        $REFQ SREF 100.,                      << missing "="
          ** ERROR **  0*A  1*B  0*C  0*D  0*E  0*F
          * FATAL ERROR *

          DNOSE(2) = 2.5,                      << not an array
          ** ERROR **  0*A  0*B  1*C  0*D  0*E  0*F
          * FATAL ERROR *

          NPANEL = 1., 2., 3.,                 << not an array
          ** ERROR **  0*A  0*B  0*C  1*D  0*E  0*F
          * FATAL ERROR *

        $FINSET1 PHIF(10) = 33.0, $            << exceeds array
          ** ERROR **  0*A  0*B  0*C  0*D  1*E  0*F
          * FATAL ERROR *

         $INLET NIN=1, $                       << should be 1.
          ** ERROR **  0*A  0*B  0*C  0*D  0*E  1*F

      BUILT                                    << Mis-spelled
          ** ERROR ** UNKNOWN CONTROL CARD - IGNORED
      NEXT CASE
          FATAL ERROR ENCOUNTERED IN CONERR.
          PROGRAM STOPPED
```

Figure 19. Example of Missile Datcom Error Messages
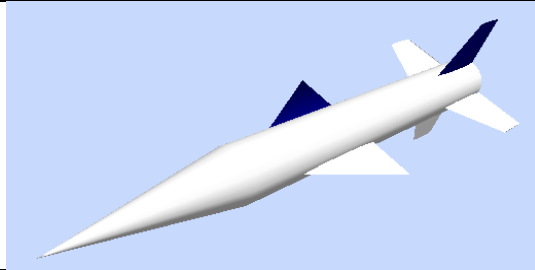
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```
1           ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS


    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  $FLTCON NALPHA=8.,NMACH=1.,MACH=2.36,REN=3000000.,
  3           ALPHA=0.,4.,8.,12.,
  4           ALPHA(5)=16.,20.,24.,28.,$
  5  $REFQ XCG=18.75,$
  6  $AXIBOD LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,DEXIT=0.,$
  7  $FINSET1 XLE=15.42,NPANEL=2.,PHIF=90.,270.,SWEEP=0.,STA=1.,
  8           CHORD=6.96,0.,SSPAN=1.875,5.355,
  9           ZUPPER=2*0.02238,LMAXU=0.238,0.238,
 10           LFLATU=0.524,0.524,LER=2*0.015,$
 11  $FINSET2 XLE=31.915,NPANEL=4.,PHIF=0.,90.,180.,270.,LER=2*0.015,
 12           SWEEP=0.,STA=1.,SSPAN=1.875,6.26,CHORD=5.585,2.792,
 13           ZUPPER=2*0.02238,LMAXU=2*0.288,LFLATU=2*0.428,$
 14 PART
 15 PLOT
 16 DAMP
 17 SOSE
 18 SAVE
 19 DIM IN
 20 NEXT CASE
 21 CASEID TRIM OF CASE NUMBER 1
 22  $TRIM SET=2.,$
 23 PRINT AERO TRIM
 24 NEXT CASE
```

Figure 20. Case Input Listing

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****     CASE   1
               AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   1
      CASE INPUTS
      FOLLOWING ARE THE CARDS INPUT FOR THIS CASE

  CASEID  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
   $FLTCON NALPHA=8.,NMACH=1.,MACH=2.36,REN=3000000.,
           ALPHA=0.,4.,8.,12.,
           ALPHA(5)=16.,20.,24.,28.,$
   $REFQ XCG=18.75,$
   $AXIBOD LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,DEXIT=0.,$
   $FINSET1 XLE=15.42,NPANEL=2.,PHIF=90.,270.,SWEEP=0.,STA=1.,
           CHORD=6.96,0.,SSPAN=1.875,5.355,
           ZUPPER=2*0.02238,LMAXU=0.238,0.238,
           LFLATU=0.524,0.524,LER=2*0.015,$
   $FINSET2 XLE=31.915,NPANEL=4.,PHIF=0.,90.,180.,270.,LER=2*0.015,
           SWEEP=0.,STA=1.,SSPAN=1.875,6.26,CHORD=5.585,2.792,
           ZUPPER=2*0.02238,LMAXU=2*0.288,LFLATU=2*0.428,$
  PART
  PLOT
  DAMP
  SOSE
  SAVE
  DIM IN
  NEXT CASE
      * WARNING * THE REFERENCE AREA IS UNSPECIFIED, DEFAULT VALUE ASSUMED
      * WARNING * THE REFERENCE LENGTH IS UNSPECIFIED, DEFAULT VALUE ASSUMED
      * WARNING * CENTER SECTION DEFINED BUT BASE DIAMETER NOT INPUT
                  CYLINDRICAL SECTION ASSUMED
      THE BOUNDARY LAYER IS ASSUMED TO BE TURBULENT
      THE INPUT UNITS ARE IN INCHES, THE SCALE FACTOR IS   1.0000
```

Figure 21. Example of Default Substitutions for Incomplete Case Inputs

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****     CASE   1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE  21
               PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
               STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

     ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
     MACH NO  =      2.36               REYNOLDS NO = 3.000E+06 /FT
     SIDESLIP =      0.00 DEG                  ROLL =     0.00 DEG
     REF AREA =     11.045 IN**2     MOMENT CENTER =    18.750 IN
     REF LENGTH =    3.75 IN         LAT REF LENGTH =     3.75 IN

                    ----- LONGITUDINAL -----    -- LATERAL DIRECTIONAL --
          ALPHA      CN        CM        CA        CY        CLN       CLL
          0.00     0.000     0.000     0.465     0.000     0.000     0.000
          4.00     1.159    -1.429     0.464     0.000     0.000     0.000
          8.00     2.476    -3.082     0.462     0.000     0.000     0.000
         12.00     3.977    -4.833     0.458     0.000     0.000     0.000
         16.00     5.554    -6.657     0.453     0.000     0.000     0.000
         20.00     7.123    -8.576     0.447     0.000     0.000     0.000
         24.00     8.573   -10.443     0.439     0.000     0.000     0.000
         28.00    10.067   -12.345     0.430     0.000     0.000     0.000


          ALPHA      CL        CD      CL/CD      X-C.P.
          0.00     0.000     0.465     0.000    -1.220
          4.00     1.124     0.544     2.068    -1.233
          8.00     2.388     0.802     2.978    -1.245
         12.00     3.795     1.275     2.977    -1.215
         16.00     5.214     1.966     2.652    -1.198
         20.00     6.541     2.856     2.290    -1.204
         24.00     7.653     3.888     1.968    -1.218
         28.00     8.686     5.106     1.701    -1.226

   X-C.P. MEAS. FROM MOMENT CENTER IN REF. LENGTHS, NEG. AFT OF MOMENT CENTER
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 03/11 *****     CASE   1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE  22
               PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
               STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

     ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
     MACH NO  =      2.36               REYNOLDS NO = 3.000E+06 /FT
     SIDESLIP =      0.00 DEG                  ROLL =     0.00 DEG
     REF AREA =     11.045 IN**2     MOMENT CENTER =    18.750 IN
     REF LENGTH =    3.75 IN         LAT REF LENGTH =     3.75 IN


                    ---------- DERIVATIVES (PER DEGREE) ----------
          ALPHA      CNA        CMA        CYB        CLNB       CLLB
          0.00     0.2702    -0.3297    -0.1994     0.4782     0.0000
          4.00     0.3094    -0.3850    -0.2084     0.4692    -0.0156
          8.00     0.3520    -0.4253    -0.2154     0.4406    -0.0232
         12.00     0.3848    -0.4468    -0.2159     0.3541    -0.0162
         16.00     0.3933    -0.4679    -0.2046     0.2233     0.0038
         20.00     0.3773    -0.4733    -0.1899     0.1140     0.0297
         24.00     0.3679    -0.4711    -0.1727     0.0369     0.0486
         28.00     0.3790    -0.4798    -0.1613    -0.0167     0.0632


   PANEL DEFLECTION ANGLES (DEGREES)
   SET    FIN 1   FIN 2   FIN 3   FIN 4   FIN 5   FIN 6   FIN 7   FIN 8
    1     0.00    0.00
    2     0.00    0.00    0.00    0.00
   BODY ALONE LINEAR DATA GENERATED FROM SECOND ORDER SHOCK EXPANSION METHOD
```

Figure 22. Total Configuration Aerodynamic Output Summary

```
1           ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****    CASE   2
                AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   2
                           TRIM OF CASE NUMBER 1
                 STATIC AERODYNAMIC COEFFICIENTS TRIMMED IN PITCH


        ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
     MACH NO  =       2.36            REYNOLDS NO = 3.000E+06 /FT
     SIDESLIP =       0.00 DEG               ROLL =     0.00 DEG
     REF AREA =      11.045 IN**2   MOMENT CENTER =    18.750 IN
     REF LENGTH =     3.75 IN       LAT REF LENGTH =     3.75 IN


         ALPHA      DELTA      CL        CD        CN        CA
          0.00       0.00     0.000     0.465     0.000     0.465
          4.00      -3.48     0.794     0.524     0.829     0.467
          8.00      -7.33     1.685     0.696     1.765     0.455
         12.00     -11.13     2.712     1.021     2.864     0.435
         16.00     -14.74     3.755     1.506     4.025     0.413
         20.00     -18.60     4.707     2.141     5.155     0.402
         24.00     -22.58     5.487     2.867     6.179     0.387
         28.00      *NT*      *NT*      *NT*      *NT*      *NT*
     PANELS FROM FIN SET 2 WERE DEFLECTED OVER THE RANGE -25.00 TO  20.00 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
     *** END OF JOB ***
```

Figure 23. Trimmed Output Summary

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014  *****    CASE    1
                AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE    2
                 PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                         AXISYMMETRIC BODY DEFINITION

                        NOSE      CENTERBODY    AFT BODY      TOTAL
        SHAPE           OGIVE      CYLINDER     --------
        LENGTH          11.250      26.250       0.000       37.500   IN
        FINENESS RATIO   3.000       7.000       0.000       10.000
        PLANFORM AREA   28.280      98.437       0.000      126.717   IN**2
        AREA CENTROID    7.016      24.375       0.000       20.501   IN
        WETTED AREA     89.818     309.251       0.000      399.069   IN**2
        VOLUME          66.789     289.922       0.000      356.711   IN**3
        VOL. CENTROID    7.714      24.375       0.000       21.255   IN


                            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

                Z-OFFSET    0.0000      0.0000      0.0000      0.0000      0.0000
        0.0000      0.0000      0.0000      0.0000      0.0000      0.0000      0.0000
        0.0000      0.0000      0.0000      0.0000      0.0000      0.0000      0.0000
        0.0000      0.0000

                BODY RADII  0.0000      0.3644      0.6871      0.9693      1.2119
        1.4159      1.5819      1.7104      1.8019      1.8568      1.8750      1.8750
        1.8750      1.8750      1.8750      1.8750      1.8750      1.8750      1.8750
        1.8750      1.8750
```

Figure 24. Body Geometry Output

```
1           ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014  *****    CASE   1
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS             PAGE   3
                   PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                         FIN SET NUMBER 1 AIRFOIL SECTION


                            NACA S-3-23.8-04.5-52.4


      X/C       X-UPPER     Y-UPPER     X-LOWER     Y-LOWER    MEAN LINE    THICKNESS


    0.00000     0.00000     0.00000     0.00000     0.00000     0.00000      0.00000
    0.00100     0.00100     0.00009     0.00100    -0.00009     0.00000      0.00019
    0.00200     0.00200     0.00019     0.00200    -0.00019     0.00000      0.00038
    0.00300     0.00300     0.00028     0.00300    -0.00028     0.00000      0.00057
    0.00400     0.00400     0.00038     0.00400    -0.00038     0.00000      0.00076
    0.00500     0.00500     0.00047     0.00500    -0.00047     0.00000      0.00095
    0.00600     0.00600     0.00057     0.00600    -0.00057     0.00000      0.00113
    0.00800     0.00800     0.00076     0.00800    -0.00076     0.00000      0.00151
    0.01000     0.01000     0.00095     0.01000    -0.00095     0.00000      0.00189
    0.02000     0.02000     0.00189     0.02000    -0.00189     0.00000      0.00378
    0.03000     0.03000     0.00284     0.03000    -0.00284     0.00000      0.00567
    0.04000     0.04000     0.00378     0.04000    -0.00378     0.00000      0.00756
    0.05000     0.05000     0.00473     0.05000    -0.00473     0.00000      0.00945
    0.06000     0.06000     0.00567     0.06000    -0.00567     0.00000      0.01134
    0.08000     0.08000     0.00756     0.08000    -0.00756     0.00000      0.01513
    0.10000     0.10000     0.00945     0.10000    -0.00945     0.00000      0.01891
    0.12000     0.12000     0.01134     0.12000    -0.01134     0.00000      0.02269
    0.14000     0.14000     0.01324     0.14000    -0.01324     0.00000      0.02647
    0.16000     0.16000     0.01513     0.16000    -0.01513     0.00000      0.03025
    0.18000     0.18000     0.01702     0.18000    -0.01702     0.00000      0.03403
    0.20000     0.20000     0.01891     0.20000    -0.01891     0.00000      0.03782
    0.22000     0.22000     0.02080     0.22000    -0.02080     0.00000      0.04160
    0.24000     0.24000     0.02250     0.24000    -0.02250     0.00000      0.04500
    0.26000     0.26000     0.02250     0.26000    -0.02250     0.00000      0.04500
    0.28000     0.28000     0.02250     0.28000    -0.02250     0.00000      0.04500
    0.30000     0.30000     0.02250     0.30000    -0.02250     0.00000      0.04500
       :           :           :           :           :           :            :
       :           : (points deleted for editing purposes) :         :
       :           :           :           :           :           :            :
    0.70000     0.70000     0.02250     0.70000    -0.02250     0.00000      0.04500
    0.75000     0.75000     0.02250     0.75000    -0.02250     0.00000      0.04500
    0.80000     0.80000     0.01891     0.80000    -0.01891     0.00000      0.03782
    0.82000     0.82000     0.01702     0.82000    -0.01702     0.00000      0.03403
    0.84000     0.84000     0.01513     0.84000    -0.01513     0.00000      0.03025
    0.86000     0.86000     0.01324     0.86000    -0.01324     0.00000      0.02647
    0.88000     0.88000     0.01134     0.88000    -0.01134     0.00000      0.02269
    0.90000     0.90000     0.00945     0.90000    -0.00945     0.00000      0.01891
    0.92000     0.92000     0.00756     0.92000    -0.00756     0.00000      0.01513
    0.94000     0.94000     0.00567     0.94000    -0.00567     0.00000      0.01134
    0.96000     0.96000     0.00378     0.96000    -0.00378     0.00000      0.00756
    0.98000     0.98000     0.00189     0.98000    -0.00189     0.00000      0.00378
    1.00000     1.00000     0.00000     1.00000     0.00000     0.00000      0.00000
```

Figure 25. Airfoil Geometry Output

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014  *****    CASE   1
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   5
                  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                      GEOMETRIC RESULTS FOR FIN SETS


                             FIN SET NUMBER  1
                          (DATA FOR ONE PANEL ONLY)


     SEGMENT    PLAN       ASPECT    TAPER     L.E.      T.E.     M.A.C.    T/C
     NUMBER     AREA       RATIO     RATIO    SWEEP     SWEEP     CHORD    RATIO
                IN**2                          DEG       DEG       IN
       1      12.1104     1.000     0.000    63.435     0.000     4.640    0.045
     TOTAL    12.1104     1.000     0.000    63.435     0.000     4.640    0.045


                             FIN SET NUMBER  2
                          (DATA FOR ONE PANEL ONLY)


     SEGMENT    PLAN       ASPECT    TAPER     L.E.      T.E.     M.A.C.    T/C
     NUMBER     AREA       RATIO     RATIO    SWEEP     SWEEP     CHORD    RATIO
                IN**2                          DEG       DEG       IN
       1      18.3666     1.047     0.500    32.495     0.000     4.344    0.045
     TOTAL    18.3666     1.047     0.500    32.495     0.000     4.344    0.045
```

Figure 26. Fin Geometry Output

```
CASEID NASA-TM-84559 2-D Inlet
BUILD
DIM M
PART
SOSE
$FLTCON   NALPHA =  11.0000,
   ALPHA =   0.0000,   2.0000,   4.0000,   6.0000,   8.0000,  10.0000,
            12.0000,  14.0000,  16.0000,  18.0000,  20.0000,
   NMACH  =   3.0000,
   MACH   =   0.6000,   0.8000,   0.9500,
   ALT    =   0.0000,   0.0000,   0.0000,
$END
$REFQ     SREF = 0.00456,     LREF = 0.0762,     XCG = 0.5334,
   BLAYER = TURB,
$END
$AXIBOD
   X0 =    0.0000,      TNOSE =    OGIVE,      LNOSE =   0.2286,
   DNOSE =   0.0762,   TRUNC =  .FALSE.,
   LCENTR =   0.8382,  DCENTR =   0.0762,   DEXIT =   0.0762,
$END
$INLET
   NIN = 1.,
   INTYPE = 2DTOP,
   XINLT = 0.29845,
   XDIV = 0.01905,
   HDIV = 0.00432,
   LDIV = 0.16566,
   PHI = 180.,
   X = 0., 0.11153, 0.16741, 0.4572, 0.7684,
   H = 0., 0.0477, 0.0584, 0.0584, 0.005,
   W = 0.05842, 0.05842, 0.05842, 0.05842, 0.01,
   COVER = .TRUE.,
   RAMP = 12.,
   ADD = .FALSE.,
$END
SAVE
NEXT CASE
```

Figure 27. Inlet Geometry Input

```
1          *****  THE USAF AUTOMATED MISSILE DATCOM * REV 2014   *****    CASE   1
                  AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   3
                             NASA-TM-84559 2-D Inlet
                               INLET GEOMETRY


          INLET IS A TOP MOUNTED TWO-DIMENSIONAL TYPE

          THE INLETS ARE COVERED

          NUMBER OF INLETS = 1

          INLET ANGULAR ROLL POSITIONS FROM TOP VERTICAL CENTER (DEG)
               (SAME CONVENTION AS FIN ROLL POSITIONS)
             180.0

          LONGITUDINAL DISTANCE FROM MISSILE NOSE TIP TO
          INLET LEADING EDGE =   0.30

               INLET POSITIONS RELATIVE TO THE LEADING EDGE
                POSITION              LONGITUDINAL      WIDTH         HEIGHT
           TOP LIP LEADING EDGE          0.000          0.058         0.000
           COWL LIP LEADING EDGE         0.112          0.058         0.048
              MID BODY START             0.167          0.058         0.058
              BOATTAIL START             0.457          0.058         0.058
               BOATTAIL END              0.768          0.010         0.005

          LONGITUDINAL DISTANCE FROM INLET LEADING EDGE TO
          DIVERTER LEADING EDGE =   0.02

          DIVERTER LENGTH =   0.17

          HEIGHT OF DIVERTER LEADING EDGE =   0.00
```

Figure 28. Inlet Geometry Output

```
1         ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
             AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   6
             BASE-JET PLUME INTERACTION FLOW PARAMETERS


     ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
     MACH NO  =        2.36              REYNOLDS NO = 3.000E+06 /FT
     SIDESLIP =        0.00 DEG                  ROLL =       0.00 DEG
     REF AREA =       11.045 FT**2      MOMENT CENTER =     18.750 FT
     REF LENGTH =       3.75 FT         LAT REF LENGTH =       3.75 FT



         WARNING EXTRAPOLATION WILL BE REQUIRED FOR THE FOLLOWING
CONDITIONS:

         * ANGLE OF ATTACK GREATER THAN 8.0



             ----- BASE FLOW PARAMETERS -----    ---- INCREMENTAL  DATA ---
-
     ALPHA     CP-BASE  CA-BASE   TB/TINF   PB/PINF    DEL CN    DEL CM    DEL CA

      0.00     0.0828   -0.0162   5.9018    1.3226    0.0000    0.0000   -0.0010
      4.00     0.0828   -0.0162   5.9018    1.3226    0.0004   -0.0028   -0.0010
      8.00     0.0828   -0.0162   5.9018    1.3226    0.0009   -0.0057   -0.0010
     12.00     0.0828   -0.0162   5.9018    1.3226    0.0013   -0.0085   -0.0010
     16.00     0.0828   -0.0162   5.9018    1.3226    0.0017   -0.0113   -0.0010
     20.00     0.0828   -0.0162   5.9018    1.3226    0.0022   -0.0142   -0.0010
     24.00     0.0828   -0.0162   5.9018    1.3226    0.0026   -0.0170   -0.0010
     28.00     0.0828   -0.0162   5.9018    1.3226    0.0030   -0.0198   -0.0010
```

Figure 29. Base-Jet Plume Interaction Output

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****     CASE   1
                AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   6
                         Protuberance Example Case
                           PROTUBERANCE OUTPUT


      ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
     MACH NO  =       0.40             REYNOLDS NO = 3.000E+06 /FT
     SIDESLIP =       0.00 DEG               ROLL =      0.00 DEG
     REF AREA =    113.097 IN**2    MOMENT CENTER =    39.000 IN
     REF LENGTH =   12.00 IN      LAT REF LENGTH =     12.00 IN

     PROTUBERANCE AXIAL FORCE COEFFICIENT IS CALCULATED AT ZERO ANGLE OF
     ATTACK AND ASSUMED CONSTANT FOR ALL ANGLES OF ATTACK.  PROTUBERANCES
     ARE CONSIDERED PART OF THE BODY WHEN CALCULATING TOTAL AXIAL FORCE.
     PROTUBERANCE AXIAL FORCE IS INCLUDED IN THE TOTAL CONFIGURATION RESULTS.

     ---------- PROTUBERANCE  CALCULATIONS (AT ZERO ANGLE OF ATTACK) ----------


                                  LONG.      NUMBER  INDIVIDUAL     TOTAL
     NUMBER        TYPE         LOCATION (IN)            CA          CA
       1         FAIRING          14.000       4       0.0009      0.0037

           **** INDUCED MOMENT BREAKDOWN ****
            PROTUBERANCE PHI =   0.000   CM =   0.0005   CLN =   0.0000
            PROTUBERANCE PHI =  90.000   CM =   0.0000   CLN =   0.0005
            PROTUBERANCE PHI = 180.000   CM =  -0.0005   CLN =   0.0000
            PROTUBERANCE PHI = 270.000   CM =   0.0000   CLN =  -0.0005

       2     VERTICAL CYLINDER    22.000       4       0.0018      0.0073

           **** INDUCED MOMENT BREAKDOWN ****
            PROTUBERANCE PHI =  45.000   CM =   0.0007   CLN =   0.0007
            PROTUBERANCE PHI = 135.000   CM =  -0.0007   CLN =   0.0007
            PROTUBERANCE PHI = 225.000   CM =  -0.0007   CLN =  -0.0007
            PROTUBERANCE PHI = 315.000   CM =   0.0007   CLN =  -0.0007

       3        LAUNCH SHOE       39.000       1       0.0028      0.0028
           **** INDUCED MOMENT BREAKDOWN ****
            PROTUBERANCE PHI =   0.000   CM =   0.0014   CLN =   0.0000

       4     FLAT PLATE OR BLOCK    56.000      1       0.0298      0.0298
           **** INDUCED MOMENT BREAKDOWN ****
            PROTUBERANCE PHI =   0.000   CM =   0.0167   CLN =   0.0000


     TOTAL CA DUE TO PROTUBERANCES =  0.0435

     TOTAL CM DUE TO PROTUBERANCES =  0.0182

     TOTAL CLN DUE TO PROTUBERANCES =  0.0000
```

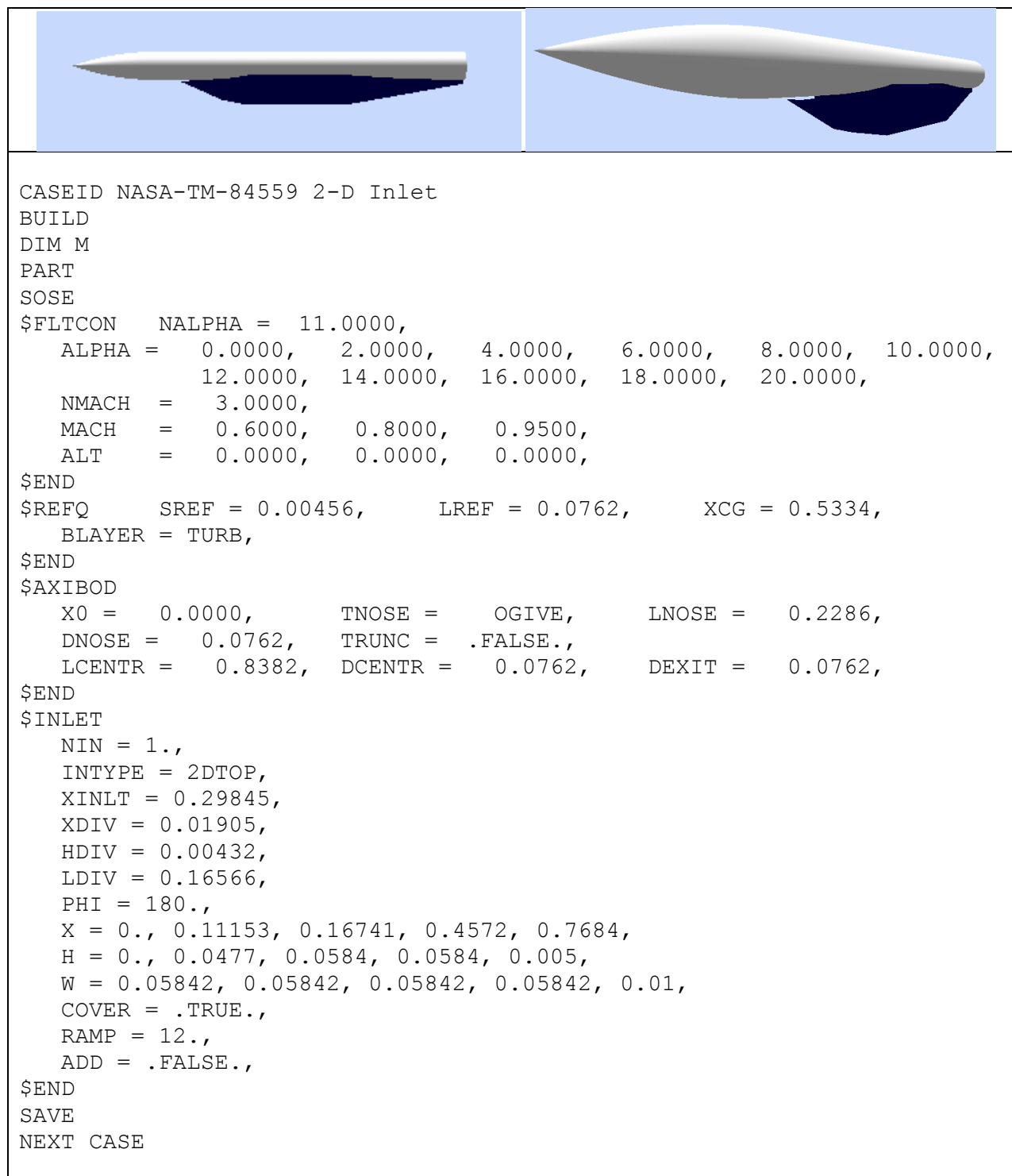Figure 30. Protuberance Output

```
1           ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   6
               PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                         BODY ALONE PARTIAL OUTPUT


         ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
         MACH NO  =       2.36              REYNOLDS NO = 3.000E+06 /FT
         SIDESLIP =       0.00 DEG                 ROLL =      0.00 DEG
         REF AREA =     11.045 IN**2   MOMENT CENTER =     18.750 IN
         REF LENGTH =     3.75 IN      LAT REF LENGTH =      3.75 IN


          ALPHA    CA-FRIC  CA-PRES/WAVE CA-BASE    CA-PROT      CA-SEP      CA-ALP
          0.00     0.0845     0.1029     0.1254                              0.0000
          4.00     0.0841     0.1029     0.1251                              0.0000
          8.00     0.0828     0.1029     0.1242                              0.0000
         12.00     0.0808     0.1028     0.1227                              0.0000
         16.00     0.0781     0.1026     0.1205                              0.0000
         20.00     0.0746     0.1025     0.1178                              0.0000
         24.00     0.0705     0.1023     0.1146                              0.0000
         28.00     0.0659     0.1020     0.1107                              0.0000


                      CROSS FLOW DRAG PROPORTIONALITY FACTOR = 1.00000

ALPHA     CN-POTEN  CN-VISC   CN-SEP      CM-POTEN   CM-VISC   CM-SEP   CM-PROT   CDC
0.00      0.000     0.000                 0.000      0.000                        0.740
4.00      0.222     0.047                 0.582     -0.022                        0.841
8.00      0.438     0.232                 1.151     -0.108                        1.044
12.00     0.645     0.665                 1.693     -0.310                        1.340
16.00     0.836     1.308                 2.196     -0.610                        1.500
20.00     1.009     2.005                 2.649     -0.936                        1.494
24.00     1.159     2.608                 3.042     -1.218                        1.374
28.00     1.282     3.330                 3.367     -1.555                        1.317
```

Figure 31. Body Alone Aerodynamic Partial Output

```
1           ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
               AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   10
                 PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                       FIN SET 2 CN, CM PARTIAL OUTPUT


       ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
      MACH NO  =        2.36                  REYNOLDS NO = 3.000E+06 /FT
      SIDESLIP =        0.00 DEG                   ROLL =        0.00 DEG
      REF AREA =      11.045 IN**2       MOMENT CENTER =     18.750 IN
      REF LENGTH =      3.75 IN          LAT REF LENGTH =       3.75 IN


     NORMAL FORCE SLOPE AT ALPHA ZERO, CNA =    0.04956/DEG (1 PANEL)
                    FIN AERODYNAMIC CENTER =    0.53282 (FRACTION OF ROOT CHORD)
                     FIN PLANFORM CENTROID =    0.61113 (FRACTION OF ROOT CHORD)


        ALPHA     CN        CN         CN        CM         CM         CM
                LINEAR   NON-LINEAR   TOTAL    LINEAR   NON-LINEAR   TOTAL


         0.00    0.0000     0.0000    0.0000    0.0000     0.0000     0.0000
         4.00    0.3952     0.0033    0.3985   -1.7043    -0.0143    -1.7186
         8.00    0.7826     0.0264    0.8091   -3.3820    -0.1143    -3.4962
        12.00    1.1549     0.0900    1.2449   -4.9996    -0.3897    -5.3893
        16.00    1.5047     0.2185    1.7232   -6.5251    -0.9477    -7.4727
        20.00    1.8251     0.3375    2.1626   -7.9279    -1.4658    -9.3937
        24.00    2.1101     0.4759    2.5860   -9.1799    -2.0706   -11.2505
        28.00    2.3540     0.6323    2.9863  -10.2561    -2.7551   -13.0112
```

Figure 32. Fin Normal Force and Pitching Moment Partial Output

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****     CASE   1
                AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   9
                  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                         FIN SET 2 CA PARTIAL OUTPUT


        ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
      MACH NO  =       2.36              REYNOLDS NO = 3.000E+06 /FT
      SIDESLIP =       0.00 DEG                  ROLL =     0.00 DEG
      REF AREA =      11.045 IN**2    MOMENT CENTER =    18.750 IN
      REF LENGTH =      3.75 IN       LAT REF LENGTH =     3.75 IN


      SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS


      SKIN FRICTION          0.0116
      SUBSONIC PRESSURE      0.0000
      TRANSONIC WAVE         0.0000
      SUPERSONIC WAVE        0.0092
      LEADING EDGE           0.0096
      TRAILING EDGE          0.0000
      TOTAL CAO              0.0304


      FIN AXIAL FORCE DUE TO ANGLE OF ATTACK


      ALPHA      CA DUE TO LIFT (SINGLE PANEL)    CA-TOTAL (4 FINS)


       0.00              0.0000                        0.1216
       4.00              0.0000                        0.1214
       8.00              0.0000                        0.1205
      12.00              0.0000                        0.1190
      16.00              0.0000                        0.1169
      20.00              0.0000                        0.1143
      24.00              0.0000                        0.1111
      28.00              0.0000                        0.1074
```

Figure 33. Fin Axial Force Partial Output

```
1           ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   5
                         NACA 4312 AIRFOIL SECTION CHECK
                           FIN SET 1 SECTION AERODYNAMICS


                       IDEAL ANGLE OF ATTACK =     1.1260 DEG.
                  ZERO LIFT ANGLE OF ATTACK =    -3.3053 DEG.
                      IDEAL LIFT COEFFICIENT =     0.5441
     ZERO LIFT PITCHING MOMENT COEFFICIENT =    -0.0895
                  MACH ZERO LIFT-CURVE-SLOPE =     0.1020 /DEG.
                         LEADING EDGE RADIUS =     0.0159 FRACTION CHORD
                MAXIMUM AIRFOIL THICKNESS =     0.1200 FRACTION CHORD
                                    DELTA-Y =     3.1690 PERCENT CHORD



   MACH = 0.300    CL-ALPHA = 0.1084 /DEG.   XAC = 0.2484   CL MAX = 1.5493

   MACH = 0.500    CL-ALPHA = 0.1158 /DEG.   XAC = 0.2620   CL MAX = 1.5874

   MACH = 0.700    CL-ALPHA = 0.1282 /DEG.   XAC = 0.2879   CL MAX = 1.6254


                     *** CREST CRITICAL MACH NUMBER EXCEEDED ***
```

Figure 34. Airfoil Section Aerodynamic Partial Output

```
1           ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   5
                           NASA-TM-84559 2-D Inlet
                         INLET AERODYNAMIC INCREMENTALS


      ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
     MACH NO  =        0.60              REYNOLDS NO = 1.391E+07 /M
     ALTITUDE =        0.0 M      DYNAMIC PRESSURE =  25533.94 N/M**2
     SIDESLIP =        0.00 DEG                ROLL =      0.00 DEG
     REF AREA =       0.005 M**2      MOMENT CENTER =     0.533 M
     REF LENGTH =     0.08 M       LAT REF LENGTH =      0.08 M


    ALPHA   CN-INLT  CM-INLT  CA-INLT   CA-ADD  CY-INLT  CLN-INLT CLL-INLT
     0.00   0.0000  -0.0806   0.0914           0.0000   0.0000   0.0000
     2.00   0.0676   0.0611   0.0914           0.0000   0.0000   0.0000
     4.00   0.1458   0.1999   0.0914           0.0000   0.0000   0.0000
     6.00   0.2342   0.3350   0.0914           0.0000   0.0000   0.0000
     8.00   0.3324   0.4659   0.0914           0.0000   0.0000   0.0000
    10.00   0.4398   0.5918   0.0914           0.0000   0.0000   0.0000
    12.00   0.5560   0.7121   0.0914           0.0000   0.0000   0.0000
    14.00   0.6804   0.8264   0.0914           0.0000   0.0000   0.0000
    16.00   0.8124   0.9339   0.0914           0.0000   0.0000   0.0000
    18.00   0.9514   1.0343   0.0914           0.0000   0.0000   0.0000
    20.00   1.0971   1.1268   0.0914           0.0000   0.0000   0.0000
```

Figure 35. Inlet Aerodynamic Partial Output

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   12
                PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                 AERODYNAMIC FORCE AND MOMENT SYNTHESIS


       ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
       MACH NO  =       2.36              REYNOLDS NO = 3.000E+06 /FT
       SIDESLIP =       0.00 DEG               ROLL =      0.00 DEG
       REF AREA =      11.045 IN**2    MOMENT CENTER =    18.750 IN
       REF LENGTH =     3.75 IN        LAT REF LENGTH =     3.75 IN


           ---------------FIN SET 2 IN PRESENCE OF THE BODY---------------


        ALPHA       CN          CM          CA          CY         CLN         CLL

         0.00      0.0000      0.0000      0.1216      0.0000      0.0000      0.0000
         4.00      0.3746     -1.6154      0.1216      0.0000      0.0000      0.0000
         8.00      0.7747     -3.3470      0.1216      0.0000      0.0000      0.0000
        12.00      1.1636     -5.0356      0.1216      0.0000      0.0000      0.0000
        16.00      1.5457     -6.6987      0.1216      0.0000      0.0000      0.0000
        20.00      1.9335     -8.3913      0.1216      0.0000      0.0000      0.0000
        24.00      2.3027    -10.0073      0.1216      0.0000      0.0000      0.0000
        28.00      2.6555    -11.5556      0.1216      0.0000      0.0000      0.0000
```

Figure 36. Fin Set in Presence of the Body Partial Output

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
                AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS              PAGE  14
                  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                   AERODYNAMIC FORCE AND MOMENT SYNTHESIS

          ****** FLIGHT CONDITIONS AND REFERENCE QUANTITIES ******
        MACH NO  =       2.36              REYNOLDS NO = 3.000E+06 /FT
        SIDESLIP =       0.00 DEG                   ROLL =      0.00 DEG
        REF AREA =      11.045 IN**2      MOMENT CENTER =    18.750 IN
        REF LENGTH =     3.75 IN        LAT REF LENGTH =      3.75 IN

            ---------------FIN SET 2 PANEL CHARACTERISTICS-----------------

         ALPHA     PANEL   AEQ(PANEL AXIS SYS.)   PANEL_CN   PANEL_XCP/CR   PANEL_YCP/(B/2)

         0.00      1          0.0000             0.0000       0.5328          0.4219
         0.00      2          0.0000             0.0000       0.5328          0.4219
         0.00      3          0.0000             0.0000       0.5328          0.4219
         0.00      4          0.0000             0.0000       0.5328          0.4219

         4.00      1          0.0000             0.0000       0.5328          0.4219
         4.00      2          3.7624             0.1873       0.5383          0.4232
         4.00      3          0.0000             0.0000       0.5328          0.4219
         4.00      4         -3.7624            -0.1873       0.5383          0.4232

         8.00      1          0.0000             0.0000       0.5328          0.4219
         8.00      2          7.6721             0.3873       0.5438          0.4245
         8.00      3          0.0000             0.0000       0.5328          0.4219
         8.00      4         -7.6721            -0.3873       0.5438          0.4245

        12.00      1          0.0000             0.0000       0.5328          0.4219
        12.00      2         11.2753             0.5818       0.5486          0.4259
        12.00      3          0.0000             0.0000       0.5328          0.4219
        12.00      4        -11.2753            -0.5818       0.5486          0.4259

        16.00      1          0.0000             0.0000       0.5328          0.4219
        16.00      2         14.5505             0.7728       0.5528          0.4272
        16.00      3          0.0000             0.0000       0.5328          0.4219
        16.00      4        -14.5505            -0.7728       0.5528          0.4272

        20.00      1          0.0000             0.0000       0.5328          0.4219
        20.00      2         17.8953             0.9667       0.5569          0.4285
        20.00      3          0.0000             0.0000       0.5328          0.4219
        20.00      4        -17.8953            -0.9667       0.5569          0.4285

        24.00      1          0.0000             0.0000       0.5328          0.4219
        24.00      2         21.3032             1.1513       0.5609          0.4297
        24.00      3          0.0000             0.0000       0.5328          0.4219
        24.00      4        -21.3032            -1.1513       0.5609          0.4297

        28.00      1          0.0000             0.0000       0.5328          0.4219
        28.00      2         24.6754             1.3277       0.5646          0.4309
        28.00      3          0.0000             0.0000       0.5328          0.4219
        28.00      4        -24.6754            -1.3277       0.5646          0.4309
```

Figure 37. Fin Set in Presence of the Body Partial Output (continued)

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****     CASE    1
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   15
                  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                  AERODYNAMIC FORCE AND MOMENT SYNTHESIS


          ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
        MACH NO  =       2.36               REYNOLDS NO = 3.000E+06 /FT
        SIDESLIP =       0.00 DEG               ROLL =      0.00 DEG
        REF AREA =      11.045 IN**2      MOMENT CENTER =     18.750 IN
        REF LENGTH =     3.75 IN          LAT REF LENGTH =     3.75 IN


               CARRYOVER INTERFERENCE FACTORS - FIN SET 1


         ALPHA      K-W(B)      K-B(W)      KK-W(B)     KK-B(W)     XCP-B(W)


         0.00       1.4031      0.4360      0.9347      0.3658      1.0903
         4.00       1.3650      0.4360      0.9347      0.3658      1.0903
         8.00       1.3042      0.4360      0.9347      0.3658      1.0903
        12.00       1.2404      0.4360      0.9347      0.3658      1.0903
        16.00       1.1819      0.4360      0.9347      0.3658      1.0903
        20.00       1.1325      0.4360      0.9347      0.3658      1.0903
        24.00       1.0934      0.4360      0.9347      0.3658      1.0903
        28.00       1.0638      0.4360      0.9347      0.3658      1.0903
```

Figure 38. Carryover Interference Factors Partial Output

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****     CASE    1
                 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   18
                  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
           FIN SET 2 PANEL BENDING MOMENTS (ABOUT EXPOSED ROOT CHORD)


          ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
        MACH NO  =       2.36               REYNOLDS NO = 3.000E+06 /FT
        SIDESLIP =       0.00 DEG               ROLL =      0.00 DEG
        REF AREA =      11.045 IN**2      MOMENT CENTER =     18.750 IN
        REF LENGTH =     3.75 IN          LAT REF LENGTH =     3.75 IN


     ALPHA   PANL 1    PANL 2    PANL 3    PANL 4    PANL 5    PANL 6    PANL 7    PANL 8

     0.0  0.00E+00  0.00E+00  0.00E+00  0.00E+00
     4.0  2.21E-08  1.11E-01 -9.30E-09 -1.11E-01
     8.0  4.21E-08  2.23E-01 -2.33E-08 -2.23E-01
    12.0  8.01E-08  3.27E-01 -5.81E-08 -3.27E-01
    16.0  8.10E-08  4.26E-01 -3.83E-08 -4.26E-01
    20.0  7.79E-08  5.16E-01 -1.29E-07 -5.16E-01
    24.0 -1.16E-08  6.07E-01 -6.45E-08 -6.07E-01
    28.0  2.50E-08  6.92E-01 -7.92E-08 -6.92E-01
```

Figure 39. Panel Bending Moment Partial Output

```
1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   20
               PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
             FIN SET 2 PANEL HINGE MOMENTS (ABOUT HINGE LINE)


        ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
     MACH NO  =       2.36                 REYNOLDS NO = 3.000E+06 /FT
     SIDESLIP =       0.00 DEG                    ROLL =     0.00 DEG
     REF AREA =      11.045 IN**2       MOMENT CENTER =    18.750 IN
     REF LENGTH =     3.75 IN           LAT REF LENGTH =     3.75 IN


 ALPHA   PANL 1    PANL 2    PANL 3    PANL 4    PANL 5    PANL 6    PANL 7    PANL 8


  0.0  0.00E+00  0.00E+00  0.00E+00  0.00E+00
  4.0 -2.19E-09 -1.32E-02  9.21E-10  1.32E-02
  8.0 -4.17E-09 -3.03E-02  2.31E-09  3.03E-02
 12.0 -7.94E-09 -4.91E-02  5.75E-09  4.91E-02
 16.0 -8.03E-09 -6.88E-02  3.80E-09  6.88E-02
 20.0 -7.72E-09 -8.93E-02  1.27E-08  8.93E-02
 24.0  1.15E-09 -1.11E-01  6.39E-09  1.11E-01
 28.0 -2.47E-09 -1.34E-01  7.85E-09  1.34E-01
```

Figure 40. Panel Hinge Moment Partial Output

```
1        ***** THE USAF AUTOMATED MISSILE DATCOM * REV 2014 *****      CASE   1
               AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   25
                  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                       BODY + 2 FIN SETS DYNAMIC DERIVATIVES


         ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
       MACH NO  =        2.36              REYNOLDS NO = 3.000E+06 /FT
       SIDESLIP =        0.00 DEG                    ROLL =      0.00 DEG
       REF AREA =      11.045 IN**2     MOMENT CENTER =    18.750 IN
       REF LENGTH =     3.75 IN         LAT REF LENGTH =      3.75 IN


                     ------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------
            ALPHA      CNQ        CMQ        CAQ        CNAD       CMAD
            0.00     1.854     -7.598      0.000      1.238     -1.923
            4.00     1.877     -7.697      0.000      1.238     -1.923
            8.00     1.818     -7.466      0.000      1.238     -1.923
           12.00     1.723     -7.089      0.000      1.238     -1.923
           16.00     1.600     -6.588      0.000      1.238     -1.923
           20.00     1.505     -6.202      0.000      1.238     -1.923
           24.00     1.410     -5.817      0.000      1.238     -1.923
           28.00     1.318     -5.440      0.000      1.238     -1.923

      PITCH RATE DERIVATIVES NON-DIMENSIONALIZED BY Q*LREF/2*V
1        ***** THE USAF AUTOMATED MISSILE DATCOM * REV 03/11 *****    CASE   1
               AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS           PAGE   26
                  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
                       BODY + 2 FIN SETS DYNAMIC DERIVATIVES


         ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
       MACH NO  =        2.36              REYNOLDS NO = 3.000E+06 /FT
       SIDESLIP =        0.00 DEG                    ROLL =      0.00 DEG
       REF AREA =      11.045 IN**2     MOMENT CENTER =    18.750 IN
       REF LENGTH =     3.75 IN         LAT REF LENGTH =      3.75 IN


                     ------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------
            ALPHA      CYR        CLNR       CLLR       CYP        CLNP        CLLP
            0.00     1.843     -7.804      0.000      0.000      0.000      -0.466
            4.00     1.811     -7.663      0.000      0.002     -0.009      -0.469
            8.00     1.765     -7.464      0.000      0.004     -0.019      -0.475
           12.00     1.724     -7.284      0.000      0.006     -0.028      -0.483
           16.00     1.694     -7.156      0.000      0.008     -0.034      -0.476
           20.00     1.679     -7.092      0.000      0.009     -0.038      -0.448
           24.00     1.680     -7.096      0.000      0.009     -0.038      -0.429
           28.00     1.697     -7.169      0.000      0.008     -0.035      -0.408

      YAW AND ROLL RATE DERIVATIVES NON-DIMENSIONALIZED BY R*LATREF/2*V
```

Figure 41. Dynamic Derivative Output

```
VARIABLES=ALPHA,CN,CM,CA,CY,CLN,CLL,DELTA,CL,CD
ZONE T="UNTRIMMED MACH=  0.30"
  -3.00   -6.8754   18.8254   0.4952   0.0000   0.0000   0.0000  -25.0000
  -2.00   -5.6970   15.5945   0.5283   0.0000   0.0000   0.0000  -25.0000
  -1.00   -4.5390   12.5304   0.5438   0.0000   0.0000   0.0000  -25.0000
   0.00   -3.4086    9.4947   0.5420   0.0000   0.0014   0.0140  -25.0000
   1.00   -2.2997    6.4886   0.5226   0.0000   0.0033   0.0253  -25.0000
   2.00   -1.2134    3.5370   0.4864   0.0000   0.0084   0.0411  -25.0000
   3.00   -0.1484    0.6385   0.4333   0.0000   0.0365   0.0532  -25.0000
ZONE T="UNTRIMMED MACH=  0.30"
  -3.00   -6.2225   16.7872   0.4487   0.0000   0.0000   0.0000  -20.0000
  -2.00   -5.0514   13.5915   0.4841   0.0000   0.0000   0.0000  -20.0000
  -1.00   -3.9059   10.5543   0.5021   0.0000   0.0000   0.0000  -20.0000
   0.00   -2.7851    7.5369   0.5023   0.0000   0.0013   0.0132  -20.0000
   1.00   -1.6890    4.5598   0.4850   0.0000   0.0035   0.0230  -20.0000
   2.00   -0.6164    1.6410   0.4507   0.0000   0.0118   0.0374  -20.0000
   3.00    0.4450   -1.2566   0.3982   0.0000  -0.0270   0.0484  -20.0000
ZONE T="UNTRIMMED MACH=  0.30"
  -3.00   -5.1144   13.3298   0.3892   0.0000   0.0000   0.0000  -15.0000
  -2.00   -3.9649   10.2180   0.4286   0.0000   0.0000   0.0000  -15.0000
  -1.00   -2.8370    7.2158   0.4499   0.0000   0.0000   0.0000  -15.0000
   0.00   -1.7371    4.2442   0.4532   0.0000   0.0015   0.0115  -15.0000
   1.00   -0.6660    1.3268   0.4388   0.0000   0.0058   0.0196  -15.0000
   2.00    0.3961   -1.5770   0.4058   0.0000  -0.0172   0.0309  -15.0000
   3.00    1.4661   -4.5196   0.3526   0.0000  -0.0034   0.0399  -15.0000
```

Figure 42. Untrimmed Aerodynamic Output ("for003.dat")

```
VARIABLES=X,R,Z
ZONE T="BODY GEOMETRY IN FEET"
     0.00000    0.00000    0.00000
     0.00781    0.00664    0.00000
     0.01561    0.01110    0.00000
     0.02342    0.01497    0.00000
     0.03122    0.01848    0.00000
     0.03903    0.02172    0.00000
     0.04683    0.02476    0.00000
     0.05464    0.02764    0.00000
     0.06244    0.03036    0.00000
     0.07025    0.03296    0.00000
     0.07806    0.03545    0.00000
     0.08586    0.03783    0.00000
     0.09367    0.04011    0.00000
     0.10147    0.04229    0.00000
     0.10928    0.04439    0.00000
     0.11708    0.04640    0.00000
     0.12489    0.04833    0.00000
     0.13269    0.05018    0.00000
     0.14050    0.05194    0.00000
     0.14831    0.05363    0.00000
     0.15611    0.05523    0.00000
     0.16392    0.05675    0.00000
     0.17172    0.05818    0.00000
     0.17953    0.05952    0.00000
     0.18733    0.06077    0.00000
     0.19514    0.06193    0.00000
     0.20294    0.06297    0.00000
     0.21075    0.06389    0.00000
     0.21856    0.06468    0.00000
     0.22636    0.06529    0.00000
     0.23417    0.06563    0.00000
     0.23417    0.06563    0.00000
     0.24167    0.06563    0.00000
     0.24916    0.06563    0.00000
     0.25666    0.06563    0.00000
     0.26416    0.06563    0.00000
     0.27166    0.06563    0.00000
     0.27916    0.06563    0.00000
     0.28666    0.06563    0.00000
     0.29416    0.06563    0.00000
     0.30165    0.06563    0.00000
         :          :          :
 (points deleted for editing purposes)
         :          :          :
```

Figure 43. Body Coordinate Data Output ("for009.dat")

```
VARIABLES=X/D,CP(0),CP(30),CP(60),CP(90),CP(120),CP(150),CP(180)
ZONE T="BODY CP AT MACH=  1.60 ALPHA=  0.00"
     0.000000   1.27042   1.27042   1.27042   1.27042   1.27042   1.27042   1.27042
     0.059471   0.78154   0.78154   0.78154   0.78154   0.78154   0.78154   0.78154
     0.118942   0.62338   0.62338   0.62338   0.62338   0.62338   0.62338   0.62338
     0.178413   0.53666   0.53666   0.53666   0.53666   0.53666   0.53666   0.53666
     0.237884   0.47825   0.47825   0.47825   0.47825   0.47825   0.47825   0.47825
     0.297354   0.43443   0.43443   0.43443   0.43443   0.43443   0.43443   0.43443
     0.356825   0.37405   0.37405   0.37405   0.37405   0.37405   0.37405   0.37405
     0.416296   0.33557   0.33557   0.33557   0.33557   0.33557   0.33557   0.33557
     0.475767   0.30667   0.30667   0.30667   0.30667   0.30667   0.30667   0.30667
     0.535238   0.28311   0.28311   0.28311   0.28311   0.28311   0.28311   0.28311
     0.594709   0.26271   0.26271   0.26271   0.26271   0.26271   0.26271   0.26271
     0.654180   0.24461   0.24461   0.24461   0.24461   0.24461   0.24461   0.24461
     0.713651   0.22811   0.22811   0.22811   0.22811   0.22811   0.22811   0.22811
     0.773122   0.21274   0.21274   0.21274   0.21274   0.21274   0.21274   0.21274
     0.832593   0.19820   0.19820   0.19820   0.19820   0.19820   0.19820   0.19820
     0.892063   0.18423   0.18423   0.18423   0.18423   0.18423   0.18423   0.18423
     0.951534   0.17067   0.17067   0.17067   0.17067   0.17067   0.17067   0.17067
     1.011005   0.15735   0.15735   0.15735   0.15735   0.15735   0.15735   0.15735
     1.070476   0.14415   0.14415   0.14415   0.14415   0.14415   0.14415   0.14415
     1.129947   0.13094   0.13094   0.13094   0.13094   0.13094   0.13094   0.13094
     1.189418   0.11761   0.11761   0.11761   0.11761   0.11761   0.11761   0.11761
     1.248889   0.10401   0.10401   0.10401   0.10401   0.10401   0.10401   0.10401
     1.308360   0.09002   0.09002   0.09002   0.09002   0.09002   0.09002   0.09002
     1.367831   0.07544   0.07544   0.07544   0.07544   0.07544   0.07544   0.07544
     1.427302   0.06006   0.06006   0.06006   0.06006   0.06006   0.06006   0.06006
     1.486772   0.04355   0.04355   0.04355   0.04355   0.04355   0.04355   0.04355
     1.546243   0.02529   0.02529   0.02529   0.02529   0.02529   0.02529   0.02529
     1.605714   0.00462   0.00462   0.00462   0.00462   0.00462   0.00462   0.00462
     1.665185  -0.01997  -0.01997  -0.01997  -0.01997  -0.01997  -0.01997  -0.01997
     1.724656  -0.05237  -0.05237  -0.05237  -0.05237  -0.05237  -0.05237  -0.05237
     1.784127  -0.13282  -0.13282  -0.13282  -0.13282  -0.13282  -0.13282  -0.13282
     1.784127  -0.13282  -0.13282  -0.13282  -0.13282  -0.13282  -0.13282  -0.13282
     1.841260  -0.12618  -0.12618  -0.12618  -0.12618  -0.12618  -0.12618  -0.12618
     1.898393  -0.11988  -0.11988  -0.11988  -0.11988  -0.11988  -0.11988  -0.11988
     1.955526  -0.11390  -0.11390  -0.11390  -0.11390  -0.11390  -0.11390  -0.11390
     2.012659  -0.10821  -0.10821  -0.10821  -0.10821  -0.10821  -0.10821  -0.10821
     2.069792  -0.10280  -0.10280  -0.10280  -0.10280  -0.10280  -0.10280  -0.10280
     2.126925  -0.09767  -0.09767  -0.09767  -0.09767  -0.09767  -0.09767  -0.09767
        :         :         :         :         :         :         :         :
        :         :      (points deleted for editing purposes)   :         :
        :         :         :         :         :         :         :         :
```

Figure 44. Body Pressure Data Output ("for010.dat")

```
VARIABLES=Y/(B/2),X/C,CP
ZONE T="FIN SET  1 CP, MACH=  1.60"
   0.00121   0.00000   0.14193
   0.00121   0.03110   0.06516
   0.00121   0.06220   0.06515
   0.00121   0.09330   0.06514
   0.00121   0.12440   0.06514
   0.00121   0.15550   0.06514
   0.00121   0.18660   0.06514
   0.00121   0.21770   0.06514
   0.00121   0.24880   0.06514
   0.00121   0.27990   0.06514
   0.00121   0.31100  -0.21573
   0.00121   0.31100  -0.21573
   0.00121   0.34251  -0.00804
   0.00121   0.37402  -0.00803
   0.00121   0.40553  -0.00803
   0.00121   0.43703  -0.00803
   0.00121   0.46854  -0.00803
   0.00121   0.50005  -0.00803
   0.00121   0.53156  -0.00803
   0.00121   0.56307  -0.00803
   0.00121   0.59458  -0.00803
   0.00121   0.62608  -0.00803
   0.00121   0.65759  -0.00803
   0.00121   0.68910  -0.13506
   0.00121   0.68910  -0.13506
   0.00121   0.72019  -0.09503
   0.00121   0.75128  -0.09502
   0.00121   0.78237  -0.09502
   0.00121   0.81346  -0.09502
   0.00121   0.84455  -0.09502
   0.00121   0.87564  -0.09502
   0.00121   0.90673  -0.09502
   0.00121   0.93782  -0.09502
   0.00121   0.96891  -0.09502
   0.00121   1.00000   0.01746
ZONE T="FIN SET  1 CP, MACH=  1.60"
   0.12500   0.00000   0.34589
   0.12500   0.03110   0.09402
   0.12500   0.06220   0.08073
   0.12500   0.09330   0.07546
   0.12500   0.12440   0.07270
      :         :         :
 (points deleted for editing purposes)
      :         :         :
```

Figure 45. Fin Pressure Data Output ("for011.dat")

```
VARIABLES=X/D,CP,MACH
ZONE T="BODY CP, MLOCAL AT MACH=  1.60 ALPHA= 0.0"
          0.000000      1.270417       0.467128
          0.059471      0.781539       0.838630
          0.118942      0.623381       0.954855
          0.178413      0.536661       1.020750
          0.237884      0.478252       1.066457
          0.297354      0.434425       1.101623
          0.356825      0.374052       1.151520
          0.416296      0.335567       1.184344
          0.475767      0.306672       1.209579
          0.535238      0.283108       1.230571
          0.594709      0.262706       1.249066
          0.654180      0.244608       1.265738
          0.713651      0.228108       1.281168
          0.773122      0.212745       1.295742
          0.832593      0.198199       1.309733
          0.892063      0.184235       1.323351
          0.951534      0.170669       1.336762
          1.011005      0.157351       1.350109
          1.070476      0.144149       1.363525
          1.129947      0.130942       1.377140
          1.189418      0.117607       1.391091
          1.248889      0.104015       1.405531
          1.308360      0.090020       1.420644
          1.367831      0.075444       1.436660
          1.427302      0.060061       1.453886
          1.486772      0.043555       1.472756
          1.546243      0.025292       1.494133
          1.605714      0.004622       1.519003
          1.665185     -0.019972       1.549608
          1.724656     -0.052369       1.591783
          1.784127     -0.132816       1.707668
          1.784127     -0.132816       1.707668
          1.841260     -0.126183       1.697410
          1.898393     -0.119882       1.687794
          1.955526     -0.113895       1.678772
          2.012659     -0.108208       1.670300
          2.069792     -0.102804       1.662339
          2.126925     -0.097670       1.654853
          2.184057     -0.092793       1.647808
              :             :              :
          (points deleted for editing purposes)
              :             :              :
```

Figure 46. Body Pressure & Local Mach Number Data Output ("for012.dat")

```
VARIABLES=X,Y,Z
 ZONE T="FUSELAGE" I=31 J=49 K=1 F=BLOCK
  0.0000000E+00  2.3416668E-02  4.6833336E-02  7.0250005E-02  9.3666673E-02
  0.1170833      0.1405000      0.1639167      0.1873333      0.2107500
  0.2341667      0.4261333      0.6181000      0.8100667      1.002033
   1.194000       1.385967       1.577933       1.769900       1.961867
   2.153833       2.161575       2.169317       2.177059       2.184800
   2.192542       2.200284       2.208025       2.215767       2.223508
   2.231250
  0.0000000E+00  2.3416668E-02  4.6833336E-02  7.0250005E-02  9.3666673E-02
  0.1170833      0.1405000      0.1639167      0.1873333      0.2107500
  0.2341667      0.4261333      0.6181000      0.8100667      1.002033
   1.194000       1.385967       1.577933       1.769900       1.961867
   2.153833       2.161575       2.169317       2.177059       2.184800
   2.192542       2.200284       2.208025       2.215767       2.223508
   2.231250
  0.0000000E+00  2.3416668E-02  4.6833336E-02  7.0250005E-02  9.3666673E-02
  0.1170833      0.1405000      0.1639167      0.1873333      0.2107500
  0.2341667      0.4261333      0.6181000      0.8100667      1.002033
   1.194000       1.385967       1.577933       1.769900       1.961867
   2.153833       2.161575       2.169317       2.177059       2.184800
   2.192542       2.200284       2.208025       2.215767       2.223508
   2.231250
  0.0000000E+00  2.3416668E-02  4.6833336E-02  7.0250005E-02  9.3666673E-02
  0.1170833      0.1405000      0.1639167      0.1873333      0.2107500
  0.2341667      0.4261333      0.6181000      0.8100667      1.002033
   1.194000       1.385967       1.577933       1.769900       1.961867
   2.153833       2.161575       2.169317       2.177059       2.184800
   2.192542       2.200284       2.208025       2.215767       2.223508
   2.231250
      :              :              :              :              :
                (points deleted for editing purposes)
      :              :              :              :              :
ZONE T="FINSET  1"  I=21 J=2 K=1 F=BLOCK
  0.3106094      0.3286276      0.3466460      0.3646642      0.3826825
  0.4007008      0.4185621      0.4364234      0.4542848      0.4721461
  0.4900074      0.4719892      0.4539708      0.4359526      0.4179343
  0.3999160      0.3820547      0.3641934      0.3463320      0.3284707
  0.3106094
  0.4425332      0.4484531      0.4543730      0.4602930      0.4662129
  0.4721328      0.4780012      0.4838696      0.4897379      0.4956063
  0.5014747      0.4955547      0.4896348      0.4837149      0.4777949
  0.4718750      0.4660066      0.4601383      0.4542699      0.4484015
  0.4425332
      :              :              :              :              :
                (points deleted for editing purposes)
      :              :              :              :              :
```

Figure 47. Tecplot-Compatible Body Coordinate Data Output ("for022.dat")

```
TITLE = "Fin-shed Vortex Tracking"
 VARIABLES = "X" "Y" "Z" "Gamma"
zone t="Finset 1, Panel 1, Vortex  1 of 10", I=  50, J=  1, K=  1
      0.49025        0.00556        0.14824        0.00126
      0.52578        0.00486        0.14823        0.00126
      0.56131        0.00417        0.14818        0.00126
      0.59684        0.00349        0.14808        0.00126
      0.63237        0.00281        0.14793        0.00126
      0.66790        0.00215        0.14773        0.00126
      0.70343        0.00151        0.14749        0.00126
      0.73896        0.00088        0.14719        0.00126
      0.77450        0.00028        0.14686        0.00126
      0.81003       -0.00030        0.14648        0.00126
      0.84556       -0.00084        0.14606        0.00126
      0.88109       -0.00135        0.14561        0.00126
      0.91662       -0.00183        0.14512        0.00126
      0.95215       -0.00227        0.14459        0.00126
      0.98768       -0.00267        0.14404        0.00126
      1.02321       -0.00303        0.14346        0.00126
      1.05874       -0.00334        0.14286        0.00126
      1.09427       -0.00360        0.14224        0.00126
      1.12980       -0.00382        0.14160        0.00126
      1.16533       -0.00399        0.14094        0.00126
      1.20086       -0.00411        0.14028        0.00126
         :              :              :              :
          (points deleted for editing purposes)
         :              :              :              :
      2.01807        0.00450        0.13008        0.00126
      2.05360        0.00507        0.13009        0.00126
      2.08913        0.00562        0.13013        0.00126
      2.12466        0.00615        0.13021        0.00126
      2.16019        0.00661        0.12929        0.00126
      2.19572        0.00680        0.12365        0.00126
      2.23125        0.00698        0.11807        0.00126
zone t="Finset 1, Panel 1, Vortex  2 of 10", I=  50, J=  1, K=  1
      0.49025        0.00556        0.14740        0.00126
      0.52578        0.00493        0.14740        0.00126
      0.56131        0.00430        0.14735        0.00126
      0.59684        0.00368        0.14726        0.00126
      0.63237        0.00307        0.14712        0.00126
      0.66790        0.00246        0.14694        0.00126
      0.70343        0.00188        0.14672        0.00126
      0.73896        0.00131        0.14646        0.00126
      0.77450        0.00076        0.14616        0.00126
         :              :              :              :
          (points deleted for editing purposes)
         :              :              :              :
```

Figure 48. Tecplot-Compatible Vortex Coordinate & Strength Data Output ("vpath*.dat")

5.0 AERODYNAMIC METHODOLOGY

The methods incorporated in the program are summarized at the end of this section. Most methods are coded in individual subroutines so that revision or replacement is easily accomplished. The program subroutines corresponding to the individual methods are given in tables immediately following the tables of methods. The tables are listed in the following order:

Table 30. Body Alone Aerodynamic Methodology

Parameter	Subsonic/Transonic (M<1.2)	Supersonic (M>1.2)
CN-potential	Option 1: Nose-cylinder: MBB charts, TN-WE-2-9769 Boattail: NSWC charts, NSWC-TR-81-156 Flare: Army charts AMCP 706-280 Option 2: Slender Body Theory Camber: AIAA 2003-3668 (Option 2 only)	Option 1 and Option 2: Second Order Shock Expansion, NSWC-TR-81-156 Van Dyke Hybrid theory, NSWC-TR-81-156 Modified Newtonian theory, NASA-TND-176 Camber: AIAA-2003-3668 (Option 2 only)
CN-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124
Cm-potential	Option 1: Nose-cylinder: MBB charts, TN-WE-2-9769 Boattail: NSWC charts, NSWC-TR-81-156 Flare: Army charts AMCP 706-280, or Option 2: Slender Body Theory Camber: AIAA-2005-4971 (Option 2 only)	Option 1 and Option 2: Second Order Shock Expansion, NSWC-TR-81-156 Van Dyke Hybrid theory, NSWC-TR-81-156 Modified Newtonian theory, NASA-TND-176 Camber: AIAA-2005-4971 (Option 2 only)
Cm-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124
CLL (elliptic body)	AIAA 2007-3936	AIAA 2007-3936
CA-skin friction	Turb: Van Driest II, MDAC West Handbook Lam: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom section 4.1.5.1	Turb: Van Driest II, MDAC West Handbook Lam: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom section 4.1.5.1
CA-pressure/ wave	M<Mcrit: USAF Datcom section 4.2.3.1 M>Mcrit: Cubic Fairing between to M = 1.2	Second Order Shock Expansion, NSWC-TR-81-156 Van Dyke Hybrid theory, NSWC-TR-81-156 Modified Newtonian theory, NASA-TND-176 AIAA-2009-0907
CA-base	Cylinder: NSWC charts, NSWC-TR-92/509 Boattail: NASA method, NASA-TR-R-100 Flare: NSWC charts, NSWC-TR-81-358	Cylinder: NSWC charts, NSWC-TR-92/509 Boattail: NASA method, NASA-TR-R-100 Flare: NSWC charts, NSWC-TR-81-156
CA-protuberance	M<0.6: Hoerner Fluid Dynamic Drag M>0.6: cubic fairing, AIAA-94-0027	M<5.0 Modified Newtonian theory, AIAA-94-0027 M>5.0: Modified Newtonian theory
CM, CLN-protuberance	AIAA-2009-0907	AIAA-2009-0907
CA-α	Allen and Perkins Crossflow, NASA TR-1048 AIAA 2005-4833	Second Order Shock Expansion, NSWC-TR-81-156 Assumed zero for Van Dyke Hybrid and Modified Newtonian theory AIAA 2005-4833
Dynamic derivatives	LMSC code, LMSC-D646354 andD646354A Slender Body Theory, AIAA 97-2280	LMSC code, LMSC-D646354 andD646354A Slender Body Theory, AIAA 97-2280
Magnus derivatives	SPIN 73 code, FRL-TR-4588	SPIN 73 code, FRL-TR-4588
Plume effects	not calculated	Chapman Korst model, AIAA 90-0618

Table 31. Body Alone Subroutines

Parameter	Subsonic/Transonic (M<1.2)	Supersonic (M>1.2)
CN-potential	Option 1: Nose-cylinder: BDCNAN, BDCNP Boattail: BDCNAB, BDCNP Flare: BDCNAF, BDCNP Option 2: SUBPTS, BDCNP Canmer: CAMBOD	Option 1 and Option 2: Second Order Shock Expansion: SOSE, BDCNP Modified Newtonian theory: HYPERS, BDCNP Camber: CAMBOD
CN-viscous	CDCS, GETETA, BDCNV	CDCS, GETETA, BDCNV
Cm-potential	Option 1: Nose-cylinder: BDXCPN, BDCMP Boattail: BDXCPB, BDCMP Flare: BDXCPF, BDCMP Option 2: SUBPTS, BDCMP Camber: CAMBOD	Option 1 and Option 2: Second Order Shock Expansion: SOSE, BDCMP Modified Newtonian theory: HYPERS, BDCMP Camber: CAMBOD
Cm-viscous	CDCS, GETETA, BDCMV	CDCS, GETETA, BDCMV
CLL (elliptic body)	ELROLL	ELROLL
CA-skin friction	SKINF, CAFRIC, BODYCA	SKINF, CAFRIC, SUPBOD
CA-pressure/ wave	M<Mcrit: BDCAPR, BODYCA M>Mcrit: CDPRES, BODYCA	Second Order Shock Expansion: SOSE, SUPBOD Modified Newtonian theory: HYPERS, SUPBOD
CA-base	BDCAB, BODYCA	BDCAB, SUPBOD
CA-protuberance	CAPROT, BODYCA	CAPROT, BODYCA
CM-protuberance	PROTCM	PROTCM
CLN-protuberance	PROTCM	PROTCM
CA-angle of attack	BDCALP, BODYCA	SOSE, SUPBOD
Dynamic derivatives	BDAMP, DAMP2	BDAMP, DAMP2
Magnus derivatives	SPIN83, DAMP2	SPIN83, DAMP2
Plume effects		BOTCNM, BOTCA, BASPRS

Table 32. Fin Alone Aerodynamic Methodology

Parameter	Subsonic (M<0.8)	Transonic (0.8<M<1.4)	Supersonic (M>1.4)
Airfoil Section	ADDFL-TR-71-87	M < Mcrit: AFFDL-TR-71-87 M > Mcrit: not calculated	not calculated
CN-potential	USAF Datcom 4.1.3.2 AIAA-2007-3937	RAS Data Sheets AIAA-2007-3937	Λ>0: USAF Datcom 4.1.3.2 Λ<0: AFWAL-TR-84-3084 AIAA 84-0575 AIAA-2007-3937
CN-viscous	Λ>0: USAF Datcom 4.1.3.3 Λ<0: AFWAL-TR-84-3084	Λ>0: USAF Datcom 4.1.3.3 Λ<0: AFWAL-TR-84-3084	Λ>0: USAF Datcom 4.1.3.3 Λ<0: AFWAL-TR-84-3084
XCP at α=0	Λ>0: USAF Datcom 4.1.4.2 Λ<0: AFWAL-TR-84-3084	Λ>0: USAF Datcom 4.1.4.2 Λ<0: AFWAL-TR-84-3084	Λ>0: USAF Datcom 4.1.4.2 Λ<0: AFWAL-TR-84-3084 AIAA-2007-3937
XCP vs α	AIAA-2007-3937	AIAA-2007-3937	AIAA-2007-3937
YCP	M > 0.4: AIAA-91-0708 M < 0.4: M=0.4 value used Flaps: AIAA-2007-3937	Tri-service data base: AIAA-91-0708 Flaps: AIAA-2007-3937	Tri-service data base: AIAA-91-0708 Flaps: AIAA-2007-3937
Flap (α/δ)	USAF Datcom section 6.1.4.1	cubic fairing	NACA-TR-1041
CA-skin friction	MDAC West Handbook Hoerner Fluid Dynamic Drag USAF Datcom section 4.1.5.1	MDAC West Handbook Hoerner Fluid Dynamic Drag USAF Datcom section 4.1.5.1	MDAC West Handbook Hoerner Fluid Dynamic Drag USAF Datcom section 4.1.5.1
CA-pressure	Hoerner Fluid Dynamic Drag	Hoerner Fluid Dynamic Drag	not applicable
CA-wave	not applicable	M<1.05: 0 1.05 M< <1.4: linear fairing	Potential Flow Theory, NWL-TR-3018
CA-bluntness	USAF Datcom section 4.1.5.1	USAF Datcom section 4.1.5.1	Potential Flow Theory, NWL-TR-3018
CA-base	Empirical, NWL-TR-2796	Empirical, NWL-TR-2796	Empirical, NWL-TR-2796
CA-induced	USAF Datcom section 4.1.5.2	USAF Datcom section 4.1.5.2	0

Table 33. Fin Alone Subroutines

Parameter	Subsonic (M<0.8)	Transonic (0.8<M<1.4)	Supersonic (M>1.4)
Airfoil Section	THEORY, CLMAX	THEORY, CLMAX	
CN-potential	FCNASB, FCNA	FCNATR, FCNA	FCNASP, FCNA
CN-viscous	FCNAAS, FCNAA	FCNAAT, FCNAA	FCNAAH, FCNAA
XCP at α=0	Λ>0: FALCP, FINXAC Λ<0: FWDXAC, FINXAC	Λ>0: FALCP, FINXAC Λ<0: FWDXAC, FINXAC	Λ>0: FALCP, FINXAC Λ<0: FWDXAC, FINXAC
XCP vs α	ALPXCP	ALPXCP	ALPXCP
YCP	YCP, YCP0, PANLCN	YCP, YCP0, PANLCN	YCP, YCP0, PANLCN
Flap (α/δ)	FLAPS	FLAPS	FLAPS
CA-skin friction	SKINF, CAFRIC, FINXCA	SKINF, CAFRIC, FINXCA	SKINF, CAFRIC, FINXCA
CA-pressure	FINCAP, FINXCA	FCAWT, FINXCA	
CA-wave		FINXCA	FCAWPF, FINXCA
CA-bluntness	FCALE, FINXCA	FCALE, FINXCA	FCAWPF, FINXCA
CA-base	FINCAB, FINXCA	FINCAB, FINXCA	FINCAB, FINXCA
CA-induced	FCALP, FINXCA	FCALP, FINXCA	

Table 34. Inlet Aerodynamic Methodology

Parameter	Subsonic (M<1.0)	Supersonic (M>1.0)
CN-potential	Engineering method, AIAA 90-3091	Engineering method, AIAA 90-3091
CN-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124
Cm-potential	Engineering method, AIAA 90-3091	Engineering method, AIAA 90-3091
Cm-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124
YCP		
CA-skin friction	Turb.: Van Driest II, MDAC West Handbook Lam.: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom 4.1.5.1	Turb.: Van Driest II, MDAC West Handbook Lam.: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom 4.1.5.1
CA-pressure/ wave	M<Mcrit: USAF Datcom 4.2.3.1 M>Mcrit: Transonic area rule, AIAA-90-0280	Supersonic Area rule, AIAA-90-0280
CA-additive	not applicable	Engineering method, AIAA 91-0712
CA-α	0	0

Table 35. Inlet Subroutines

Parameter	Subsonic (M<1.0)	Supersonic (M>1.0)
CN-viscous	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO
Cm-potential	ILTCDC, ILTCFD, ILTVIS, ILTARO	ILTCDC, ILTCFD, ILTVIS, ILTARO
Cm-viscous	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO
YCP	ILTCDC, ILTCFD, ILTVIS, ILTARO	ILTCDC, ILTCFD, ILTVIS, ILTARO
CA-skin friction	SKINF, CAFRIC, ILTARO	SKINF, CAFRIC, ILTARO
CA-pressure/ wave	M<Mcrit: BDCAPR, ILTARO M>Mcrit: CDPRES, ILTARO	CDPRES, ILTARO
CA-additive	IAD2D, or IADAXI, ILTARO	IAD2D or IADAXI, ILTARO
CA-α	not applicable	not applicable

Table 36. Body-Fin Synthesis Aerodynamic Methodology

Parameter	Subsonic (M<1.0)	Supersonic (M>1.0)
Body-Fin Upwash, K_W	Empirical correlation, AIAA 96-3395 Folding fin: AIAA 94-0027	Empirical correlation, AIAA 96-3395 Folding fin: AIAA 94-0027
Fin-Body Carryover, K_B	Slender body theory, NACA-TR-1307	Slender body theory, NACA-TR-1307 AIAA Journal, May 1981 AIAA-2007-3937
Body-Fin Upwash Center of Pressure, xcp_{WB}	$\Lambda>0$: USAF Datcom section 4.1.4.2 $\Lambda<0$: AFWAL-TR-84-3084	$\Lambda>0$: USAF Datcom section 4.1.4.2 $\Lambda<0$: AFWAL-TR-84-3084
Body-Fin-Body Carryover Center of Pressure, xcp_{bW}	Lifting line theory, NACA-TR-1307 AIAA 94-0027	Slender body theory, NACA-TR-1307 AIAA Journal, August 1982
Fin Deflection, Λ_{IJ}	Slender body theory, AGARD-R-711	Slender body theory, AGARD-R-711
Equivalent angle of attack	AIAA J. S&R, July-Aug 1983	AIAA J. S&R, July-Aug 1983
Body Vortex Strength	Empirical, NWC-TP-5761	Empirical, NWC-TP-5761
Body Vortex Track	Empirical, NWC-TP-5761	Empirical, NWC-TP-5761
Fin Vortex Strength	Line vortex theory, NACA-TR-1307	Line vortex theory, NACA-TR-1307 AIAA 2011-1240
Fin Vortex Track	Crossflow, 2-D potential flow marched downstream	AIAA 2009-3853
Dynamic derivatives	Equiv. angle of attack, AIAA 97-2280	Equiv. angle of attack, AIAA 97-2280

Table 37. Body-Fin Synthesis Subroutines

Parameter	Subsonic (M<1.0)	Supersonic (M>1.0)
Body-Fin Upwash, K_W	KWBNEW, SYNTHS Folding fin: PANLCN	KWBNEW, SYNTHS Folding fin: PANLCN
Fin-Body Carryover, K_B	CARRYO, SYNTHS	CARRYO, SYNTHS
Body-Fin Upwash Center of Pressure, xcp_{WB}	$\Lambda >0$: FALCP, CARRYO, SYNTHS $\Lambda <0$: FWDXAC, CARRYO, SYNTHS	$\Lambda >0$: FALCP, CARRYO, SYNTHS $\Lambda <0$: FWDXAC, CARRYO, SYNTHS
Body-Fin-Body Carryover Center of Pressure, xcp_{BW}	CARRYO, SYNTHS	CARRYO, SYNTHS
Fin Deflection, Λ_{IJ}	FINFIN, PANLCN	FINFIN, PANLCN
Equivalent angle of attack	ALPEQ, PANLCN	ALPEQ, PANLCN
Body Vortex Strength	CLVR, ALPEQ, PANLCN	CLVR, ALPEQ, PANLCN
Body Vortex Track	CLVR, ALPEQ, PANLCN	CLVR, ALPEQ, PANLCN
Fin Vortex Strength	VRINTS, SYNTHS	VRINTS, SYNTHS
Fin Vortex Track	SYNTHS, VORDIS, VTRACK	SYNTHS, VORDIS, VTRACK
Dynamic derivatives	ALPEQ2, FDAMP, DAMP2	ALPEQ2, FDAMP, DAMP2