    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 ELEMENTDESIGNATION - (N)
    D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED
    E - ASSIGNED VALUES EXCEED ARRAY DIMENSION
    F - SYNTAX ERROR

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

```
   1 DIM M
   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  $AXIBOD LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,DEXIT=2.,$
   7  $AXIBOD BASE=.TRUE.,BETAN=10.,JMACH=2.5,PRAT=4.,TRAT=4.,$
   8  $FINSET1 XLE=15.42,NPANEL=2.,PHIF=90.,270.,SWEEP=0.,STA=1.,
   9           CHORD=6.96,0.,SSPAN=1.875,5.355,
  10           ZUPPER=2*0.02238,LMAXU=0.238,0.238,
  11           LFLATU=0.524,0.524,LER=2*0.015,$
  12  $FINSET2 XLE=31.915,NPANEL=4.,PHIF=0.,90.,180.,270.,LER=2*0.015,
  13           SWEEP=0.,STA=1.,SSPAN=1.875,6.26,CHORD=5.585,2.792,
  14           ZUPPER=2*0.02238,LMAXU=2*0.288,LFLATU=2*0.428,$
  15 PART
  16 DAMP DB14
  17 PLOT
  18 PRESSURES
  19 SAVE
  20 NEXT CASE
  21  $TRIM SET=2.,$
  22 PRINT AERO TRIM PLOT
  23 NEXT CASE
```

```
 * REV 3/99 *****     CASE   1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   1
    CASE INPUTS
    FOLLOWING ARE THE CARDS INPUT FOR THIS CASE

 DIM M
  $FLTCON NALPHA=8.,NMACH=1.,MACH=2.36,REN=3000000.,
          ALPHA=0.,4.,8.,12.,
          ALPHA(5)=16.,20.,24.,28.,$
  $REFQ XCG=18.75,$
  $AXIBOD LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,DEXIT=2.,$
  $AXIBOD BASE=.TRUE.,BETAN=10.,JMACH=2.5,PRAT=4.,TRAT=4.,$
  $FINSET1 XLE=15.42,NPANEL=2.,PHIF=90.,270.,SWEEP=0.,STA=1.,
           CHORD=6.96,0.,SSPAN=1.875,5.355,
           ZUPPER=2*0.02238,LMAXU=0.238,0.238,
           LFLATU=0.524,0.524,LER=2*0.015,$
  $FINSET2 XLE=31.915,NPANEL=4.,PHIF=0.,90.,180.,270.,LER=2*0.015,
           SWEEP=0.,STA=1.,SSPAN=1.875,6.26,CHORD=5.585,2.792,
           ZUPPER=2*0.02238,LMAXU=2*0.288,LFLATU=2*0.428,$
 PART
 DAMP DB14
 PLOT
 PRESSURES
 SAVE
 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 METERS, THE SCALE FACTOR IS   1.0000
1         *****
```

AXISYMMETRIC BODY DEFINITION

	NOSE	CENTERBODY	AFT BODY	TOTAL	
SHAPE	OGIVE	CYLINDER	--------		
LENGTH	11.250	26.250	0.000	37.500	M
FINENESS RATIO	3.000	7.000	0.000	10.000	
PLANFORM AREA	28.280	98.437	0.000	126.717	M**2
AREA CENTROID	7.016	24.375	0.000	20.501	M
WETTED AREA	89.818	309.250	0.000	399.069	M**2
VOLUME	66.789	289.923	0.000	356.711	M**3
VOL. CENTROID	7.714	24.375	0.000	21.255	M

MOLD LINE CONTOUR

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

NOTE - * INDICATES SLOPE DISCONTINUOUS POINTS

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
0.32000	0.32000	0.02250	0.32000	-0.02250	0.00000	0.04500
0.34000	0.34000	0.02250	0.34000	-0.02250	0.00000	0.04500
0.36000	0.36000	0.02250	0.36000	-0.02250	0.00000	0.04500
0.38000	0.38000	0.02250	0.38000	-0.02250	0.00000	0.04500
0.40000	0.40000	0.02250	0.40000	-0.02250	0.00000	0.04500
0.42000	0.42000	0.02250	0.42000	-0.02250	0.00000	0.04500
0.45000	0.45000	0.02250	0.45000	-0.02250	0.00000	0.04500
0.50000	0.50000	0.02250	0.50000	-0.02250	0.00000	0.04500
0.55000	0.55000	0.02250	0.55000	-0.02250	0.00000	0.04500
0.60000	0.60000	0.02250	0.60000	-0.02250	0.00000	0.04500
0.65000	0.65000	0.02250	0.65000	-0.02250	0.00000	0.04500
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

1 *****

FIN SET NUMBER 2 AIRFOIL SECTION

NACA S-3-28.8-04.5-42.8

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	0.00008	0.00100	-0.00008	0.00000	0.00016
0.00200	0.00200	0.00016	0.00200	-0.00016	0.00000	0.00031
0.00300	0.00300	0.00023	0.00300	-0.00023	0.00000	0.00047
0.00400	0.00400	0.00031	0.00400	-0.00031	0.00000	0.00063
0.00500	0.00500	0.00039	0.00500	-0.00039	0.00000	0.00078
0.00600	0.00600	0.00047	0.00600	-0.00047	0.00000	0.00094
0.00800	0.00800	0.00063	0.00800	-0.00063	0.00000	0.00125
0.01000	0.01000	0.00078	0.01000	-0.00078	0.00000	0.00156
0.02000	0.02000	0.00156	0.02000	-0.00156	0.00000	0.00313
0.03000	0.03000	0.00234	0.03000	-0.00234	0.00000	0.00469
0.04000	0.04000	0.00313	0.04000	-0.00313	0.00000	0.00625
0.05000	0.05000	0.00391	0.05000	-0.00391	0.00000	0.00781
0.06000	0.06000	0.00469	0.06000	-0.00469	0.00000	0.00937
0.08000	0.08000	0.00625	0.08000	-0.00625	0.00000	0.01250
0.10000	0.10000	0.00781	0.10000	-0.00781	0.00000	0.01562
0.12000	0.12000	0.00937	0.12000	-0.00937	0.00000	0.01875
0.14000	0.14000	0.01094	0.14000	-0.01094	0.00000	0.02187
0.16000	0.16000	0.01250	0.16000	-0.01250	0.00000	0.02500
0.18000	0.18000	0.01406	0.18000	-0.01406	0.00000	0.02812
0.20000	0.20000	0.01562	0.20000	-0.01562	0.00000	0.03125
0.22000	0.22000	0.01719	0.22000	-0.01719	0.00000	0.03437
0.24000	0.24000	0.01875	0.24000	-0.01875	0.00000	0.03750
0.26000	0.26000	0.02031	0.26000	-0.02031	0.00000	0.04062
0.28000	0.28000	0.02187	0.28000	-0.02187	0.00000	0.04375
0.30000	0.30000	0.02250	0.30000	-0.02250	0.00000	0.04500
0.32000	0.32000	0.02250	0.32000	-0.02250	0.00000	0.04500
0.34000	0.34000	0.02250	0.34000	-0.02250	0.00000	0.04500
0.36000	0.36000	0.02250	0.36000	-0.02250	0.00000	0.04500
0.38000	0.38000	0.02250	0.38000	-0.02250	0.00000	0.04500
0.40000	0.40000	0.02250	0.40000	-0.02250	0.00000	0.04500
0.42000	0.42000	0.02250	0.42000	-0.02250	0.00000	0.04500
0.45000	0.45000	0.02250	0.45000	-0.02250	0.00000	0.04500
0.50000	0.50000	0.02250	0.50000	-0.02250	0.00000	0.04500
0.55000	0.55000	0.02250	0.55000	-0.02250	0.00000	0.04500
0.60000	0.60000	0.02250	0.60000	-0.02250	0.00000	0.04500
0.65000	0.65000	0.02250	0.65000	-0.02250	0.00000	0.04500
0.70000	0.70000	0.02250	0.70000	-0.02250	0.00000	0.04500
0.75000	0.75000	0.01981	0.75000	-0.01981	0.00000	0.03961
0.80000	0.80000	0.01585	0.80000	-0.01585	0.00000	0.03169
0.82000	0.82000	0.01426	0.82000	-0.01426	0.00000	0.02852
0.84000	0.84000	0.01268	0.84000	-0.01268	0.00000	0.02535
0.86000	0.86000	0.01109	0.86000	-0.01109	0.00000	0.02218
0.88000	0.88000	0.00951	0.88000	-0.00951	0.00000	0.01901
0.90000	0.90000	0.00792	0.90000	-0.00792	0.00000	0.01585
0.92000	0.92000	0.00634	0.92000	-0.00634	0.00000	0.01268
0.94000	0.94000	0.00475	0.94000	-0.00475	0.00000	0.00951
0.96000	0.96000	0.00317	0.96000	-0.00317	0.00000	0.00634
0.98000	0.98000	0.00158	0.98000	-0.00158	0.00000	0.00317
1.00000	1.00000	0.00000	1.00000	0.00000	0.00000	0.00000

1 *****

                      GEOMETRIC RESULTS FOR FIN SETS

FIN SET NUMBER 1
(DATA FOR ONE PANEL ONLY)

SEGMENT NUMBER	PLAN AREA	ASPECT RATIO	TAPER RATIO	L.E. SWEEP	T.E. SWEEP	M.A.C. CHORD	T/C RATIO
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 NUMBER	PLAN AREA	ASPECT RATIO	TAPER RATIO	L.E. SWEEP	T.E. SWEEP	M.A.C. CHORD	T/C RATIO
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

1   *****

                                BASE-JET PLUME INTERACTION FLOW PARAMETERS

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

     WARNING EXTRAPOLATION WILL BE REQUIRED FOR THE FOLLOWING CONDITIONS:

     * ANGLE OF ATTACK GREATER THAN 8.0
     * BOATTAIL TERMINAL ANGLE GREATER THAN 12.0
     * NOZZLE EXIT TO BASE DIAMETER RATIO LESS THAN 0.8

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

 0.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
 4.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
 8.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
12.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
16.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
20.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
24.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
28.00   -0.1605   0.1149   3.9010   0.3742   0.0000   0.0000   0.0000
1   *****

                                BASE-JET PLUME INTERACTION FLOW PARAMETERS

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

          --------- BOATTAIL SEPARATION PARAMETERS ---------

            PANEL 1 PHI= 0.0   PANEL 2 PHI= 90.0   PANEL 3 PHI=180.0   PANEL 4 PHI
             X-SEP MACH CONE   X-SEP MACH CONE   X-SEP MACH CONE   X-SEP MAC
 ALPHA   (FT)   ANGLE   (FT)   ANGLE   (FT)   ANGLE   (FT)   A

  0.00   123.000   25.070   123.000   25.070   123.000   25.070   123.000   25
  4.00   123.000   29.070   123.000   25.070   123.000   21.070   123.000   25
  8.00   123.000   33.070   123.000   25.070   123.000   17.070   123.000   25
 12.00   123.000   37.070   123.000   25.070   123.000   13.070   123.000   25
 16.00   123.000   41.070   123.000   25.070   123.000   9.070   123.000   25
 20.00   123.000   45.070   123.000   25.070   123.000   5.070   123.000   25
 24.00   123.000   49.070   123.000   25.070   123.000   1.070   123.000   25
 28.00   123.000   53.070   123.000   25.070   123.000   -2.930   123.000   25
1   *****

                          BODY ALONE PARTIAL OUTPUT

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

ALPHA	CA-FRIC	CA-PRES/WAVE	CA-BASE	CA-PROT	CA-SEP	CA-ALP
0.00	0.0560	0.0969	0.0897		0.0000	0.0000
4.00	0.0558	0.0967	0.0895		0.0000	0.0000
8.00	0.0549	0.0964	0.0889		0.0000	0.0000
12.00	0.0536	0.0958	0.0878		0.0000	0.0000
16.00	0.0518	0.0950	0.0863		0.0000	0.0000
20.00	0.0495	0.0939	0.0843		0.0000	0.0000
24.00	0.0468	0.0926	0.0820		0.0000	0.0000
28.00	0.0437	0.0910	0.0792		0.0000	0.0000

CROSS FLOW DRAG PROPORTIONALITY FACTOR = 1.00000

ALPHA	CN-POTEN	CN-VISC	CN-SEP	CM-POTEN	CM-VISC	CM-SEP	CDC
0.00	0.000	0.000	0.000	0.000	-0.000	0.000	0.740
4.00	0.196	0.047	0.000	0.697	-0.022	0.000	0.841
8.00	0.388	0.232	0.000	1.377	-0.108	0.000	1.044
12.00	0.571	0.665	0.000	2.026	-0.310	0.000	1.340
16.00	0.741	1.308	0.000	2.628	-0.610	0.000	1.500
20.00	0.894	2.005	0.000	3.170	-0.936	0.000	1.494
24.00	1.027	2.608	0.000	3.640	-1.218	0.000	1.374
28.00	1.136	3.330	0.000	4.028	-1.555	0.000	1.317

1   *****

                         FIN SET 1 CA PARTIAL OUTPUT

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

    SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS

    SKIN FRICTION   0.0047
    SUBSONIC PRESSURE   0.0000
    TRANSONIC WAVE   0.0000
    SUPERSONIC WAVE   0.0065
    LEADING EDGE   0.0011
    TRAILING EDGE   0.0000
    TOTAL CAO   0.0123

    FIN AXIAL FORCE DUE TO ANGLE OF ATTACK

    ALPHA   CA DUE TO LIFT (SINGLE PANEL)   CA-TOTAL (2 FINS)

       0.00   0.0000   0.0246
       4.00   0.0000   0.0245
       8.00   0.0000   0.0243
      12.00   0.0000   0.0240
      16.00   0.0000   0.0236
      20.00   0.0000   0.0231
      24.00   0.0000   0.0224
      28.00   0.0000   0.0217
1   *****

                      FIN SET 1 CN, CM PARTIAL OUTPUT

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

    NORMAL FORCE SLOPE AT ALPHA ZERO, CNA =   0.03132/DEG (1 PANEL)
         CENTER OF PRESSURE FOR LINEAR CN =   -0.35552 (CALIBERS FROM C.G.)
     CENTER OF PRESSURE FOR NON-LINEAR CN =   -0.34933 (CALIBERS FROM C.G.)

ALPHA	CN LINEAR	CN NON-LINEAR	CN TOTAL	CM LINEAR	CM NON-LINEAR	CM TOTAL
0.00	0.0000	0.0000	0.0000	-0.0000	-0.0000	0.0000
4.00	0.2498	0.0153	0.2651	-0.0888	-0.0053	-0.0941
8.00	0.4947	0.0541	0.5488	-0.1759	-0.0189	-0.1948
12.00	0.7300	0.1058	0.8358	-0.2595	-0.0370	-0.2965
16.00	0.9511	0.1649	1.1160	-0.3381	-0.0576	-0.3957
20.00	1.1537	0.2235	1.3771	-0.4102	-0.0781	-0.4882
24.00	1.3338	0.2774	1.6112	-0.4742	-0.0969	-0.5711
28.00	1.4879	0.3277	1.8156	-0.5290	-0.1145	-0.6435

1   *****

                         FIN SET 2 CA PARTIAL OUTPUT

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

    SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS

    SKIN FRICTION   0.0072
    SUBSONIC PRESSURE   0.0000
    TRANSONIC WAVE   0.0000
    SUPERSONIC WAVE   0.0092
    LEADING EDGE   0.0096
    TRAILING EDGE   0.0000
    TOTAL CAO   0.0260

    FIN AXIAL FORCE DUE TO ANGLE OF ATTACK

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

       0.00   0.0000   0.1041
       4.00   0.0000   0.1039
       8.00   0.0000   0.1031
      12.00   0.0000   0.1018
      16.00   0.0000   0.1001
      20.00   0.0000   0.0978
      24.00   0.0000   0.0951
      28.00   0.0000   0.0919
1   *****

                      FIN SET 2 CN, CM PARTIAL OUTPUT

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

    NORMAL FORCE SLOPE AT ALPHA ZERO, CNA =   0.04956/DEG (1 PANEL)
         CENTER OF PRESSURE FOR LINEAR CN =   -4.39078 (CALIBERS FROM C.G.)
     CENTER OF PRESSURE FOR NON-LINEAR CN =   -4.42084 (CALIBERS FROM C.G.)

      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   -1.7351   -0.0146   -1.7497
       8.00   0.7826   0.0264   0.8091   -3.4364   -0.1169   -3.5533
      12.00   1.1549   0.0900   1.2449   -5.0709   -0.3980   -5.4689
      16.00   1.5047   0.2185   1.7232   -6.6066   -0.9661   -7.5727
      20.00   1.8251   0.3375   2.1626   -8.0138   -1.4919   -9.5056
      24.00   2.1101   0.4759   2.5860   -9.2649   -2.1041 -11.3690
      28.00   2.3540   0.6323   2.9863 -10.3358   -2.7955 -13.1313
1   *****

  ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******

MACH NO =	2.36	REYNOLDS NO =	3.000E+06 /M
SIDESLIP =	0.00 DEG	ROLL =	0.00 DEG
REF AREA =	11.045 M**2	MOMENT CENTER =	18.750 M
REF LENGTH =	3.75 M	LAT REF LENGTH =	3.75 M

----------FIN SET 1 IN PRESENCE OF THE BODY----------

ALPHA	CN	CM	CA	CY	CLN	CLL
0.00	0.0000	0.0000	0.0246	0.0000	-0.0000	0.0000
4.00	0.3667	-0.1304	0.0246	-0.0000	0.0000	0.0000
8.00	0.7206	-0.2562	0.0246	-0.0000	0.0000	0.0000
12.00	1.0316	-0.3667	0.0246	-0.0000	0.0000	0.0000
16.00	1.2509	-0.4447	0.0246	-0.0000	0.0000	0.0000
20.00	1.4289	-0.5080	0.0246	-0.0000	0.0000	0.0000
24.00	1.6160	-0.5745	0.0246	-0.0000	0.0000	0.0000
28.00	1.7821	-0.6336	0.0246	-0.0000	0.0000	0.0000

1 *****

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

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

ALPHA	CN	CM	CA	CY	CLN	CLL
0.00	0.0000	0.0000	0.1041	0.0000	-0.0000	0.0000
4.00	0.3934	-1.7272	0.1041	0.0000	-0.0000	-0.0000
8.00	0.7753	-3.4042	0.1041	0.0000	-0.0000	-0.0000
12.00	1.1790	-5.1767	0.1041	0.0000	-0.0000	0.0000
16.00	1.5721	-6.9029	0.1041	0.0000	-0.0000	-0.0000
20.00	1.9600	-8.6060	0.1041	0.0000	-0.0000	-0.0000
24.00	2.3304	-10.2322	0.1041	0.0000	-0.0000	-0.0000
28.00	2.6810	-11.7717	0.1041	0.0000	-0.0000	-0.0000

1   *****

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

----------FIN SET 1 PANEL CHARACTERISTICS----------

ALPHA	PANEL	AEQ (PANEL AXIS SYS.)	PANEL CN
0.00	1	0.0000	0.0000
0.00	2	0.0000	0.0000
0.00	3	0.0000	0.0000
0.00	4	0.0000	0.0000
4.00	1	5.4522	0.1834
4.00	2	-5.4522	-0.1834
4.00	3	0.0000	0.0000
4.00	4	0.0000	0.0000
8.00	1	10.3870	0.3603
8.00	2	-10.3870	-0.3603
8.00	3	0.0000	0.0000
8.00	4	0.0000	0.0000
12.00	1	14.7699	0.5158
12.00	2	-14.7699	-0.5158
12.00	3	0.0000	0.0000
12.00	4	0.0000	0.0000
16.00	1	18.0288	0.6255
16.00	2	-18.0288	-0.6255
16.00	3	0.0000	0.0000
16.00	4	0.0000	0.0000
20.00	1	20.8387	0.7144
20.00	2	-20.8387	-0.7144
20.00	3	0.0000	0.0000
20.00	4	0.0000	0.0000
24.00	1	24.0898	0.8080
24.00	2	-24.0898	-0.8080
24.00	3	0.0000	0.0000
24.00	4	0.0000	0.0000
28.00	1	27.3206	0.8911
28.00	2	-27.3206	-0.8911
28.00	3	0.0000	0.0000
28.00	4	0.0000	0.0000

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

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

ALPHA	PANEL	AEQ (PANEL AXIS SYS.)	PANEL CN
0.00	1	0.0000	0.0000
0.00	2	0.0000	0.0000
0.00	3	0.0000	0.0000
0.00	4	0.0000	0.0000
4.00	1	0.0000	0.0000
4.00	2	3.9491	0.1967
4.00	3	0.0000	0.0000
4.00	4	-3.9491	-0.1967
8.00	1	-0.0000	-0.0000
8.00	2	7.6781	0.3877
8.00	3	0.0000	0.0000
8.00	4	-7.6781	-0.3877
12.00	1	-0.0000	-0.0000
12.00	2	11.4134	0.5895
12.00	3	0.0000	0.0000
12.00	4	-11.4134	-0.5895
16.00	1	0.0000	0.0000
16.00	2	14.7634	0.7861
16.00	3	0.0000	0.0000
16.00	4	-14.7634	-0.7861
20.00	1	0.0000	0.0000
20.00	2	18.1403	0.9800
20.00	3	0.0000	0.0000
20.00	4	-18.1403	-0.9800
24.00	1	0.0000	0.0000
24.00	2	21.5607	1.1652
24.00	3	0.0000	0.0000
24.00	4	-21.5607	-1.1652
28.00	1	0.0000	0.0000
28.00	2	24.9264	1.3405
28.00	3	0.0000	0.0000
28.00	4	-24.9264	-1.3405

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

         CARRYOVER INTERFERENCE FACTORS - FIN SET 1

ALPHA	K-W(B)	K-B(W)	KK-W(B)	KK-B(W)	XCP-W(B)	XCP-B(W)	Y-CP/(B/2)
0.00	1.4031	0.4360	0.9347	0.3658	0.3555	1.0903	0.4055
4.00	1.3650	0.4360	0.9347	0.3658	0.3555	1.0903	0.3730
8.00	1.3042	0.4360	0.9347	0.3658	0.3555	1.0903	0.3524
12.00	1.2404	0.4360	0.9347	0.3658	0.3555	1.0903	0.3396
16.00	1.1819	0.4360	0.9347	0.3658	0.3555	1.0903	0.3327
20.00	1.1325	0.4360	0.9347	0.3658	0.3555	1.0903	0.3297
24.00	1.0934	0.4360	0.9347	0.3658	0.3555	1.0903	0.3267
28.00	1.0638	0.4360	0.9347	0.3658	0.3555	1.0903	0.3253

1   *****

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

             CARRYOVER INTERFERENCE FACTORS - FIN SET 2

      ALPHA   K-W(B)   K-B(W) KK-W(B) KK-B(W) XCP-W(B) XCP-B(W) Y-CP/(B/2)

       0.00   1.3517   0.1126   0.9359   0.3165   4.3908   4.5036   0.4216
       4.00   1.3142   0.1126   0.9359   0.3165   4.3908   4.5036   0.4264
       8.00   1.2558   0.1126   0.9359   0.3165   4.3908   4.5036   0.4302
      12.00   1.1960   0.1126   0.9359   0.3165   4.3908   4.5036   0.4326
      16.00   1.1430   0.1126   0.9359   0.3165   4.3908   4.5036   0.4326
      20.00   1.1000   0.1126   0.9359   0.3165   4.3908   4.5036   0.4304
      24.00   1.0673   0.1126   0.9359   0.3165   4.3908   4.5036   0.4275
      28.00   1.0438   0.1126   0.9359   0.3165   4.3908   4.5036   0.4246

    NOTE - XCP-W(B) USED FOR STABILITY ONLY DIFFERENT VALUES USED FOR HINGE MOMENTS

         FIN SET 1 PANEL BENDING MOMENTS (ABOUT EXPOSED ROOT CHORD)

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

    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
      4.0 6.35E-02 -6.35E-02
      8.0 1.18E-01 -1.18E-01
     12.0 1.63E-01 -1.63E-01
     16.0 1.93E-01 -1.93E-01
     20.0 2.19E-01 -2.19E-01
     24.0 2.45E-01 -2.45E-01
     28.0 2.69E-01 -2.69E-01
1   *****

        FIN SET 2 PANEL BENDING MOMENTS (ABOUT EXPOSED ROOT CHORD)

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

   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 0.00E+00 9.81E-02 1.53E-08 -9.81E-02
     8.0 -9.49E-10 1.95E-01 3.41E-08 -1.95E-01
    12.0 -1.77E-08 2.98E-01 5.03E-08 -2.98E-01
    16.0 1.94E-08 3.98E-01 7.14E-08 -3.98E-01
    20.0 0.00E+00 4.93E-01 9.52E-08 -4.93E-01
    24.0 0.00E+00 5.82E-01 3.81E-08 -5.82E-01
    28.0 2.35E-08 6.66E-01 1.65E-07 -6.66E-01
1   *****

            FIN SET 1 PANEL HINGE MOMENTS (ABOUT HINGE LINE)

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

   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
     4.0 -4.26E-02 4.26E-02
     8.0 -8.43E-02 8.43E-02
    12.0 -1.22E-01 1.22E-01
    16.0 -1.50E-01 1.50E-01
    20.0 -1.73E-01 1.73E-01
    24.0 -1.98E-01 1.98E-01
    28.0 -2.20E-01 2.20E-01
1   *****

            FIN SET 2 PANEL HINGE MOMENTS (ABOUT HINGE LINE)

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

   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 -0.00E+00 -1.08E-02 -1.63E-09 1.08E-02
     8.0 9.99E-11 -2.20E-02 -3.59E-09 2.20E-02
    12.0 1.85E-09 -3.46E-02 -5.27E-09 3.46E-02
    16.0 -2.04E-09 -4.75E-02 -7.48E-09 4.75E-02
    20.0 -0.00E+00 -6.08E-02 -1.00E-08 6.08E-02
    24.0 -0.00E+00 -7.42E-02 -4.04E-09 7.42E-02
    28.0 -2.50E-09 -8.73E-02 -1.76E-08 8.73E-02
1   *****

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

                  ----- LONGITUDINAL -----   -- LATERAL DIRECTIONAL --
         ALPHA   CN   CM   CA   CY   CLN   CLL

          0.00   0.000   0.000   0.371   0.000   -0.000   0.000
          4.00   1.154   -1.462   0.371   0.000   -0.000   -0.000
          8.00   2.427   -2.967   0.369   0.000   -0.000   -0.000
         12.00   3.920   -4.723   0.366   0.000   -0.000   0.000
         16.00   5.488   -6.531   0.362   0.000   -0.000   -0.000
         20.00   7.038   -8.383   0.356   0.000   -0.000   -0.000
         24.00   8.472   -10.194   0.350   -0.000   0.000   -0.000
         28.00   9.949   -12.031   0.343   0.000   -0.000   -0.000

         ALPHA   CL   CD   CL/CD   X-C.P.

          0.00   0.000   0.371   0.000   -1.316
          4.00   1.126   0.450   2.500   -1.267
          8.00   2.352   0.703   3.345   -1.223
         12.00   3.758   1.173   3.204   -1.205
         16.00   5.176   1.860   2.782   -1.190
         20.00   6.492   2.742   2.367   -1.191
         24.00   7.597   3.765   2.018   -1.203
         28.00   8.624   4.973   1.734   -1.209

     X-C.P. MEAS. FROM MOMENT CENTER IN REF. LENGTHS, NEG. AFT OF MOMENT CENTER

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

                  ---------- DERIVATIVES (PER DEGREE) ----------
         ALPHA   CNA   CMA   CYB   CLNB   CLLB
          0.00   0.2739   -0.3603   -0.1931   0.4605   -0.0000
          4.00   0.3033   -0.3709   -0.2020   0.4512   -0.0105
          8.00   0.3455   -0.4073   -0.2089   0.4212   -0.0187
         12.00   0.3827   -0.4454   -0.2092   0.3321   -0.0110
         16.00   0.3898   -0.4575   -0.1980   0.1993   0.0080
         20.00   0.3729   -0.4579   -0.1841   0.0913   0.0320
         24.00   0.3638   -0.4559   -0.1690   0.0214   0.0486
         28.00   0.3749   -0.4624   -0.1595   -0.0245   0.0615

    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 VAN DYKE HYBRID THEORY

1   *****

                        BODY ALONE DYNAMIC DERIVATIVES

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

                  ------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------
        ALPHA   CNQ   CMQ   CAQ   CNAD   CMAD
         0.00   0.221   -0.838   0.000   0.400   -0.267
         4.00   0.221   -0.838   0.000   0.400   -0.267
         8.00   0.221   -0.838   0.000   0.400   -0.267
        12.00   0.221   -0.838   0.000   0.400   -0.267
        16.00   0.221   -0.838   0.000   0.400   -0.267
        20.00   0.221   -0.838   0.000   0.400   -0.267
        24.00   0.221   -0.838   0.000   0.400   -0.267
        28.00   0.221   -0.838   0.000   0.400   -0.267

    PITCH RATE DERIVATIVES NON-DIMENSIONALIZED BY Q*LREF/2*V
1   *****

                        BODY ALONE DYNAMIC DERIVATIVES

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

                  ------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------
         ALPHA   CYR   CLNR   CLLR   CYP   CLNP   CLLP
          0.00   0.242   -0.929   0.000   0.000   0.000   0.000
          4.00   0.242   -0.929   0.000   0.000   0.000   0.000
          8.00   0.242   -0.929   0.000   0.000   0.000   0.000
         12.00   0.242   -0.929   0.000   0.000   0.000   0.000
         16.00   0.242   -0.929   0.000   0.000   0.000   0.000
         20.00   0.242   -0.929   0.000   0.000   0.000   0.000
         24.00   0.242   -0.929   0.000   0.000   0.000   0.000
         28.00   0.242   -0.929   0.000   0.000   0.000   0.000

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

                       BODY + 2 FIN SETS DYNAMIC DERIVATIVES

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

                   ------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------
         ALPHA   CNQ   CMQ   CAQ   CNAD   CMAD
          0.00   1.601   -6.584   0.000   0.833   -1.652
          4.00   1.622   -6.652   0.000   0.833   -1.652
          8.00   1.570   -6.442   0.000   0.833   -1.652
         12.00   1.491   -6.125   0.000   0.833   -1.652
         16.00   1.379   -5.667   0.000   0.833   -1.652
         20.00   1.270   -5.219   0.000   0.833   -1.652
         24.00   1.171   -4.819   0.000   0.833   -1.652
         28.00   1.075   -4.418   0.000   0.833   -1.652

    PITCH RATE DERIVATIVES NON-DIMENSIONALIZED BY Q*LREF/2*V

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

                  ------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------
         ALPHA   CYR   CLNR   CLLR   CYP   CLNP   CLLP
          0.00   1.566   -6.755   0.000   -0.000   0.000   -0.422
          4.00   1.533   -6.609   0.000   0.004   -0.019   -0.419
          8.00   1.486   -6.403   0.000   0.009   -0.039   -0.420
         12.00   1.443   -6.217   0.000   0.012   -0.055   -0.429
         16.00   1.413   -6.085   0.000   0.015   -0.067   -0.437
         20.00   1.398   -6.018   0.000   0.016   -0.072   -0.410
         24.00   1.399   -6.023   0.000   0.016   -0.068   -0.396
         28.00   1.416   -6.098   0.000   0.014   -0.062   -0.379

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

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 * REV 3/99 *****     CASE   2
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   1
    CASE INPUTS
    FOLLOWING ARE THE CARDS INPUT FOR THIS CASE

  $TRIM SET=2.,$
 PRINT AERO TRIM PLOT
 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 METERS, THE SCALE FACTOR IS   1.0000
1         *****
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 * REV 3/99 *****   CASE   2

STATIC AERODYNAMIC COEFFICIENTS TRIMMED IN PITCH

******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******

MACH NO =	2.36	REYNOLDS NO =	3.000E+06 /M
SIDESLIP =	0.00 DEG	ROLL =	0.00 DEG
REF AREA =	11.045 M**2	MOMENT CENTER =	18.750 M
REF LENGTH =	3.75 M	LAT REF LENGTH =	3.75 M

ALPHA	DELTA	CL	CD	CN	CA
0.00	0.00	0.000	0.371	0.000	0.371
4.00	-3.50	0.794	0.430	0.822	0.373
8.00	-6.95	1.685	0.599	1.752	0.359
12.00	-10.69	2.716	0.917	2.847	0.332
16.00	-14.20	3.764	1.401	4.004	0.309
20.00	-17.89	4.725	2.027	5.134	0.289
24.00	-21.67	5.517	2.743	6.156	0.262
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
*** END OF JOB ***