```
 * REV 3/99 *****
               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 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  $FLTCON NALPHA=8.,NMACH=1.,MACH=0.15,REN=3031800.,
     3          ALPHA=0.,4.,8.,12.,
     4          ALPHA(5)=16.,20.,24.,28.,$
     5  $REFQ XCG=0.513,$
     6  $AXIBOD LNOSE=0.18,DNOSE=0.06,LCENTR=0.668,DEXIT=0,$
        ** ERROR **  0*A  0*B  0*C  0*D  0*E  1*F
     7  $AXIBOD BASE=.FALSE.,BETAN=10.,JMACH=2.5,PRAT=4.,TRAT=4.,$
     8  $FINSET1 XLE=0.542,NPANEL=4.,PHIF=0.,90.,180.,270.,SWEEP=0.,STA=1.,
     9           CHORD=0.05,0.05,SSPAN=0.03,0.08,
    10           ZUPPER=0.025,LMAXU=0.5,
    11           LFLATU=0,0,LER=0,$
        ** ERROR **  0*A  0*B  0*C  0*D  0*E  3*F
    12  $FINSET2 XLE=0.748,NPANEL=4.,PHIF=0.,90.,180.,270.,LER=0,
        ** ERROR **  0*A  0*B  0*C  0*D  0*E  1*F
    13           SWEEP=0.,STA=1.,SSPAN=0.03,0.086,CHORD=0.048,0.01,
    14           ZUPPER=0.025,LMAXU=0.5,LFLATU=0.0,$
    15 PART
    16 DAMP DBODY
    17 DAMP DB1
    18 DAMP DB12
    19 PLOT
    20 PRESSURES
    21 SAVE
    22 NEXT CASE
    23  $TRIM SET=2.,$
    24 PRINT AERO TRIM PLOT
    25 NEXT CASE
1         *****
```

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

 DIM M
  $FLTCON NALPHA=8.,NMACH=1.,MACH=0.15,REN=3031800.,
          ALPHA=0.,4.,8.,12.,
          ALPHA(5)=16.,20.,24.,28.,$
  $REFQ XCG=0.513,$
  $AXIBOD LNOSE=0.18,DNOSE=0.06,LCENTR=0.668,DEXIT=0,$
  $AXIBOD BASE=.FALSE.,BETAN=10.,JMACH=2.5,PRAT=4.,TRAT=4.,$
  $FINSET1 XLE=0.542,NPANEL=4.,PHIF=0.,90.,180.,270.,SWEEP=0.,STA=1.,
           CHORD=0.05,0.05,SSPAN=0.03,0.08,
           ZUPPER=0.025,LMAXU=0.5,
           LFLATU=0,0,LER=0,$
  $FINSET2 XLE=0.748,NPANEL=4.,PHIF=0.,90.,180.,270.,LER=0,
           SWEEP=0.,STA=1.,SSPAN=0.03,0.086,CHORD=0.048,0.01,
           ZUPPER=0.025,LMAXU=0.5,LFLATU=0.0,$
 PART
 DAMP DBODY
 DAMP DB1
 DAMP DB12
 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 | 0.180 | 0.668 | 0.000 | 0.848 | M |
| FINENESS RATIO | 3.000 | 11.133 | 0.000 | 14.133 | |
| PLANFORM AREA | 0.007 | 0.040 | 0.000 | 0.047 | M**2 |
| AREA CENTROID | 0.112 | 0.514 | 0.000 | 0.453 | M |
| WETTED AREA | 0.023 | 0.126 | 0.000 | 0.149 | M**2 |
| VOLUME | 0.000 | 0.002 | 0.000 | 0.002 | M**3 |
| VOL. CENTROID | 0.123 | 0.514 | 0.000 | 0.465 | M |

MOLD LINE CONTOUR

```
 LONGITUDINAL STATIONS    0.0000     0.0180     0.0360     0.0540     0.0720
    0.0900     0.1080     0.1260     0.1440     0.1620     0.1800     0.2468
    0.3136     0.3804     0.4472     0.5140     0.5808     0.6476     0.7144
    0.7812     0.8480*

            BODY RADII    0.0000     0.0058     0.0110     0.0155     0.0194
    0.0227     0.0253     0.0274     0.0288     0.0297     0.0300     0.0300
    0.0300     0.0300     0.0300     0.0300     0.0300     0.0300     0.0300
    0.0300     0.0300*
```

NOTE - * INDICATES SLOPE DISCONTINUOUS POINTS

1         *****

FIN SET NUMBER 1 AIRFOIL SECTION

NACA S-1-50.0-05.0-00.0

| 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 | 0.00005 | 0.00100 | -0.00005 | 0.00000 | 0.00010 |
| 0.00200 | 0.00200 | 0.00010 | 0.00200 | -0.00010 | 0.00000 | 0.00020 |
| 0.00300 | 0.00300 | 0.00015 | 0.00300 | -0.00015 | 0.00000 | 0.00030 |
| 0.00400 | 0.00400 | 0.00020 | 0.00400 | -0.00020 | 0.00000 | 0.00040 |
| 0.00500 | 0.00500 | 0.00025 | 0.00500 | -0.00025 | 0.00000 | 0.00050 |
| 0.00600 | 0.00600 | 0.00030 | 0.00600 | -0.00030 | 0.00000 | 0.00060 |
| 0.00800 | 0.00800 | 0.00040 | 0.00800 | -0.00040 | 0.00000 | 0.00080 |
| 0.01000 | 0.01000 | 0.00050 | 0.01000 | -0.00050 | 0.00000 | 0.00100 |
| 0.02000 | 0.02000 | 0.00100 | 0.02000 | -0.00100 | 0.00000 | 0.00200 |
| 0.03000 | 0.03000 | 0.00150 | 0.03000 | -0.00150 | 0.00000 | 0.00300 |
| 0.04000 | 0.04000 | 0.00200 | 0.04000 | -0.00200 | 0.00000 | 0.00400 |
| 0.05000 | 0.05000 | 0.00250 | 0.05000 | -0.00250 | 0.00000 | 0.00500 |
| 0.06000 | 0.06000 | 0.00300 | 0.06000 | -0.00300 | 0.00000 | 0.00600 |
| 0.08000 | 0.08000 | 0.00400 | 0.08000 | -0.00400 | 0.00000 | 0.00800 |
| 0.10000 | 0.10000 | 0.00500 | 0.10000 | -0.00500 | 0.00000 | 0.01000 |
| 0.12000 | 0.12000 | 0.00600 | 0.12000 | -0.00600 | 0.00000 | 0.01200 |
| 0.14000 | 0.14000 | 0.00700 | 0.14000 | -0.00700 | 0.00000 | 0.01400 |
| 0.16000 | 0.16000 | 0.00800 | 0.16000 | -0.00800 | 0.00000 | 0.01600 |
| 0.18000 | 0.18000 | 0.00900 | 0.18000 | -0.00900 | 0.00000 | 0.01800 |
| 0.20000 | 0.20000 | 0.01000 | 0.20000 | -0.01000 | 0.00000 | 0.02000 |
| 0.22000 | 0.22000 | 0.01100 | 0.22000 | -0.01100 | 0.00000 | 0.02200 |
| 0.24000 | 0.24000 | 0.01200 | 0.24000 | -0.01200 | 0.00000 | 0.02400 |
| 0.26000 | 0.26000 | 0.01300 | 0.26000 | -0.01300 | 0.00000 | 0.02600 |
| 0.28000 | 0.28000 | 0.01400 | 0.28000 | -0.01400 | 0.00000 | 0.02800 |
| 0.30000 | 0.30000 | 0.01500 | 0.30000 | -0.01500 | 0.00000 | 0.03000 |
| 0.32000 | 0.32000 | 0.01600 | 0.32000 | -0.01600 | 0.00000 | 0.03200 |
| 0.34000 | 0.34000 | 0.01700 | 0.34000 | -0.01700 | 0.00000 | 0.03400 |
| 0.36000 | 0.36000 | 0.01800 | 0.36000 | -0.01800 | 0.00000 | 0.03600 |
| 0.38000 | 0.38000 | 0.01900 | 0.38000 | -0.01900 | 0.00000 | 0.03800 |
| 0.40000 | 0.40000 | 0.02000 | 0.40000 | -0.02000 | 0.00000 | 0.04000 |
| 0.42000 | 0.42000 | 0.02100 | 0.42000 | -0.02100 | 0.00000 | 0.04200 |
| 0.45000 | 0.45000 | 0.02250 | 0.45000 | -0.02250 | 0.00000 | 0.04500 |
| 0.50000 | 0.50000 | 0.02500 | 0.50000 | -0.02500 | 0.00000 | 0.05000 |
| 0.55000 | 0.55000 | 0.02250 | 0.55000 | -0.02250 | 0.00000 | 0.04500 |
| 0.60000 | 0.60000 | 0.02000 | 0.60000 | -0.02000 | 0.00000 | 0.04000 |
| 0.65000 | 0.65000 | 0.01750 | 0.65000 | -0.01750 | 0.00000 | 0.03500 |
| 0.70000 | 0.70000 | 0.01500 | 0.70000 | -0.01500 | 0.00000 | 0.03000 |
| 0.75000 | 0.75000 | 0.01250 | 0.75000 | -0.01250 | 0.00000 | 0.02500 |
| 0.80000 | 0.80000 | 0.01000 | 0.80000 | -0.01000 | 0.00000 | 0.02000 |
| 0.82000 | 0.82000 | 0.00900 | 0.82000 | -0.00900 | 0.00000 | 0.01800 |
| 0.84000 | 0.84000 | 0.00800 | 0.84000 | -0.00800 | 0.00000 | 0.01600 |
| 0.86000 | 0.86000 | 0.00700 | 0.86000 | -0.00700 | 0.00000 | 0.01400 |
| 0.88000 | 0.88000 | 0.00600 | 0.88000 | -0.00600 | 0.00000 | 0.01200 |
| 0.90000 | 0.90000 | 0.00500 | 0.90000 | -0.00500 | 0.00000 | 0.01000 |
| 0.92000 | 0.92000 | 0.00400 | 0.92000 | -0.00400 | 0.00000 | 0.00800 |
| 0.94000 | 0.94000 | 0.00300 | 0.94000 | -0.00300 | 0.00000 | 0.00600 |
| 0.96000 | 0.96000 | 0.00200 | 0.96000 | -0.00200 | 0.00000 | 0.00400 |
| 0.98000 | 0.98000 | 0.00100 | 0.98000 | -0.00100 | 0.00000 | 0.00200 |
| 1.00000 | 1.00000 | 0.00000 | 1.00000 | 0.00000 | 0.00000 | 0.00000 |

1 *****

FIN SET NUMBER 2 AIRFOIL SECTION

NACA S-1-50.0-05.0-00.0

| 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 | 0.00005 | 0.00100 | -0.00005 | 0.00000 | 0.00010 |
| 0.00200 | 0.00200 | 0.00010 | 0.00200 | -0.00010 | 0.00000 | 0.00020 |
| 0.00300 | 0.00300 | 0.00015 | 0.00300 | -0.00015 | 0.00000 | 0.00030 |
| 0.00400 | 0.00400 | 0.00020 | 0.00400 | -0.00020 | 0.00000 | 0.00040 |
| 0.00500 | 0.00500 | 0.00025 | 0.00500 | -0.00025 | 0.00000 | 0.00050 |
| 0.00600 | 0.00600 | 0.00030 | 0.00600 | -0.00030 | 0.00000 | 0.00060 |
| 0.00800 | 0.00800 | 0.00040 | 0.00800 | -0.00040 | 0.00000 | 0.00080 |
| 0.01000 | 0.01000 | 0.00050 | 0.01000 | -0.00050 | 0.00000 | 0.00100 |
| 0.02000 | 0.02000 | 0.00100 | 0.02000 | -0.00100 | 0.00000 | 0.00200 |
| 0.03000 | 0.03000 | 0.00150 | 0.03000 | -0.00150 | 0.00000 | 0.00300 |
| 0.04000 | 0.04000 | 0.00200 | 0.04000 | -0.00200 | 0.00000 | 0.00400 |
| 0.05000 | 0.05000 | 0.00250 | 0.05000 | -0.00250 | 0.00000 | 0.00500 |
| 0.06000 | 0.06000 | 0.00300 | 0.06000 | -0.00300 | 0.00000 | 0.00600 |
| 0.08000 | 0.08000 | 0.00400 | 0.08000 | -0.00400 | 0.00000 | 0.00800 |
| 0.10000 | 0.10000 | 0.00500 | 0.10000 | -0.00500 | 0.00000 | 0.01000 |
| 0.12000 | 0.12000 | 0.00600 | 0.12000 | -0.00600 | 0.00000 | 0.01200 |
| 0.14000 | 0.14000 | 0.00700 | 0.14000 | -0.00700 | 0.00000 | 0.01400 |
| 0.16000 | 0.16000 | 0.00800 | 0.16000 | -0.00800 | 0.00000 | 0.01600 |
| 0.18000 | 0.18000 | 0.00900 | 0.18000 | -0.00900 | 0.00000 | 0.01800 |
| 0.20000 | 0.20000 | 0.01000 | 0.20000 | -0.01000 | 0.00000 | 0.02000 |
| 0.22000 | 0.22000 | 0.01100 | 0.22000 | -0.01100 | 0.00000 | 0.02200 |
| 0.24000 | 0.24000 | 0.01200 | 0.24000 | -0.01200 | 0.00000 | 0.02400 |
| 0.26000 | 0.26000 | 0.01300 | 0.26000 | -0.01300 | 0.00000 | 0.02600 |
| 0.28000 | 0.28000 | 0.01400 | 0.28000 | -0.01400 | 0.00000 | 0.02800 |
| 0.30000 | 0.30000 | 0.01500 | 0.30000 | -0.01500 | 0.00000 | 0.03000 |
| 0.32000 | 0.32000 | 0.01600 | 0.32000 | -0.01600 | 0.00000 | 0.03200 |
| 0.34000 | 0.34000 | 0.01700 | 0.34000 | -0.01700 | 0.00000 | 0.03400 |
| 0.36000 | 0.36000 | 0.01800 | 0.36000 | -0.01800 | 0.00000 | 0.03600 |
| 0.38000 | 0.38000 | 0.01900 | 0.38000 | -0.01900 | 0.00000 | 0.03800 |
| 0.40000 | 0.40000 | 0.02000 | 0.40000 | -0.02000 | 0.00000 | 0.04000 |
| 0.42000 | 0.42000 | 0.02100 | 0.42000 | -0.02100 | 0.00000 | 0.04200 |
| 0.45000 | 0.45000 | 0.02250 | 0.45000 | -0.02250 | 0.00000 | 0.04500 |
| 0.50000 | 0.50000 | 0.02500 | 0.50000 | -0.02500 | 0.00000 | 0.05000 |
| 0.55000 | 0.55000 | 0.02250 | 0.55000 | -0.02250 | 0.00000 | 0.04500 |
| 0.60000 | 0.60000 | 0.02000 | 0.60000 | -0.02000 | 0.00000 | 0.04000 |
| 0.65000 | 0.65000 | 0.01750 | 0.65000 | -0.01750 | 0.00000 | 0.03500 |
| 0.70000 | 0.70000 | 0.01500 | 0.70000 | -0.01500 | 0.00000 | 0.03000 |
| 0.75000 | 0.75000 | 0.01250 | 0.75000 | -0.01250 | 0.00000 | 0.02500 |
| 0.80000 | 0.80000 | 0.01000 | 0.80000 | -0.01000 | 0.00000 | 0.02000 |
| 0.82000 | 0.82000 | 0.00900 | 0.82000 | -0.00900 | 0.00000 | 0.01800 |
| 0.84000 | 0.84000 | 0.00800 | 0.84000 | -0.00800 | 0.00000 | 0.01600 |
| 0.86000 | 0.86000 | 0.00700 | 0.86000 | -0.00700 | 0.00000 | 0.01400 |
| 0.88000 | 0.88000 | 0.00600 | 0.88000 | -0.00600 | 0.00000 | 0.01200 |
| 0.90000 | 0.90000 | 0.00500 | 0.90000 | -0.00500 | 0.00000 | 0.01000 |
| 0.92000 | 0.92000 | 0.00400 | 0.92000 | -0.00400 | 0.00000 | 0.00800 |
| 0.94000 | 0.94000 | 0.00300 | 0.94000 | -0.00300 | 0.00000 | 0.00600 |
| 0.96000 | 0.96000 | 0.00200 | 0.96000 | -0.00200 | 0.00000 | 0.00400 |
| 0.98000 | 0.98000 | 0.00100 | 0.98000 | -0.00100 | 0.00000 | 0.00200 |
| 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 | 0.0025 | 1.000 | 1.000 | 0.000 | 0.000 | 0.050 | 0.050 |
| TOTAL | 0.0025 | 1.000 | 1.000 | 0.000 | 0.000 | 0.050 | 0.050 |

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 | 0.0016 | 1.931 | 0.208 | 34.160 | 0.000 | 0.033 | 0.050 |
| TOTAL | 0.0016 | 1.931 | 0.208 | 34.160 | 0.000 | 0.033 | 0.050 |

1          *****

FIN SET 1 SECTION AERODYNAMICS

```
                 IDEAL ANGLE OF ATTACK =    0.0000 DEG.
             ZERO LIFT ANGLE OF ATTACK =    0.0000 DEG.
                IDEAL LIFT COEFFICIENT =    0.0000
 ZERO LIFT PITCHING MOMENT COEFFICIENT =    0.0000
            MACH ZERO LIFT-CURVE-SLOPE =    0.0973 /DEG.
                   LEADING EDGE RADIUS =    0.0000 FRACTION CHORD
             MAXIMUM AIRFOIL THICKNESS =    0.0500 FRACTION CHORD
                               DELTA-Y =    0.2925 PERCENT CHORD


MACH = 0.150   CL-ALPHA = 0.0984 /DEG.   XAC = 0.2692   CL MAX = 0.7487
```

1         *****

```
 * REV 3/99 *****     CASE   1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   7
                          FIN SET 2 SECTION AERODYNAMICS

                        IDEAL ANGLE OF ATTACK =    0.0000 DEG.
                    ZERO LIFT ANGLE OF ATTACK =    0.0000 DEG.
                       IDEAL LIFT COEFFICIENT =    0.0000
        ZERO LIFT PITCHING MOMENT COEFFICIENT =    0.0000
                   MACH ZERO LIFT-CURVE-SLOPE =    0.0973 /DEG.
                          LEADING EDGE RADIUS =    0.0000 FRACTION CHORD
                    MAXIMUM AIRFOIL THICKNESS =    0.0500 FRACTION CHORD
                                      DELTA-Y =    0.2925 PERCENT CHORD


    MACH = 0.150   CL-ALPHA = 0.0984 /DEG.   XAC = 0.2692   CL MAX = 0.7482

1         *****
```

BODY ALONE PARTIAL OUTPUT

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

| ALPHA | CA-FRIC | CA-PRES/WAVE | CA-BASE | CA-PROT | CA-SEP | CA-ALP |
|---|---|---|---|---|---|---|
| 0.00 | 0.1988 | 0.0114 | 0.1370 | | | -0.0000 |
| 4.00 | 0.1979 | 0.0113 | 0.1375 | | | 0.0005 |
| 8.00 | 0.1950 | 0.0112 | 0.1373 | | | 0.0013 |
| 12.00 | 0.1902 | 0.0109 | 0.1364 | | | 0.0006 |
| 16.00 | 0.1837 | 0.0105 | 0.1348 | | | -0.0059 |
| 20.00 | 0.1756 | 0.0100 | 0.1325 | | | -0.0242 |
| 24.00 | 0.1659 | 0.0095 | 0.1295 | | | -0.0623 |
| 28.00 | 0.1550 | 0.0089 | 0.1259 | | | -0.1299 |

CROSS FLOW DRAG PROPORTIONALITY FACTOR = 0.72597

| ALPHA | CN-POTEN | CN-VISC | CN-SEP | CM-POTEN | CM-VISC | CM-SEP | CDC |
|---|---|---|---|---|---|---|---|
| 0.00 | 0.000 | 0.000 | | 0.000 | 0.000 | | 0.740 |
| 4.00 | 0.181 | 0.044 | | 0.894 | 0.044 | | 0.746 |
| 8.00 | 0.358 | 0.177 | | 1.768 | 0.179 | | 0.753 |
| 12.00 | 0.526 | 0.399 | | 2.601 | 0.402 | | 0.759 |
| 16.00 | 0.682 | 0.706 | | 3.374 | 0.712 | | 0.765 |
| 20.00 | 0.823 | 1.096 | | 4.070 | 1.105 | | 0.771 |
| 24.00 | 0.945 | 1.563 | | 4.673 | 1.575 | | 0.777 |
| 28.00 | 1.046 | 2.097 | | 5.172 | 2.113 | | 0.783 |

1 *****

FIN SET 1 CA PARTIAL OUTPUT

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

| | | | |
|---|---|---|---|
| MACH NO = | 0.15 | REYNOLDS NO = | 3.032E+06 /M |
| SIDESLIP = | 0.00 DEG | ROLL = | 0.00 DEG |
| REF AREA = | 0.003 M**2 | MOMENT CENTER = | 0.513 M |
| REF LENGTH = | 0.06 M | LAT REF LENGTH = | 0.06 M |

SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS

| | |
|---|---|
| SKIN FRICTION | 0.0115 |
| SUBSONIC PRESSURE | 0.0007 |
| TRANSONIC WAVE | 0.0000 |
| SUPERSONIC WAVE | 0.0000 |
| LEADING EDGE | 0.0000 |
| TRAILING EDGE | 0.0000 |
| TOTAL CAO | 0.0122 |

FIN AXIAL FORCE DUE TO ANGLE OF ATTACK

| ALPHA | CA DUE TO LIFT (SINGLE PANEL) | CA-TOTAL (4 FINS) |
|---|---|---|
| 0.00 | 0.0000 | 0.0489 |
| 4.00 | 0.0000 | 0.0488 |
| 8.00 | 0.0000 | 0.0484 |
| 12.00 | 0.0000 | 0.0478 |
| 16.00 | 0.0000 | 0.0470 |
| 20.00 | 0.0000 | 0.0460 |
| 24.00 | 0.0000 | 0.0447 |
| 28.00 | 0.0000 | 0.0432 |

1 *****

FIN SET 1 CN, CM PARTIAL OUTPUT

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

NORMAL FORCE SLOPE AT ALPHA ZERO, CNA = 0.03882/DEG (1 PANEL)
CENTER OF PRESSURE FOR LINEAR CN = -0.70833 (CALIBERS FROM C.G.)
CENTER OF PRESSURE FOR NON-LINEAR CN = -0.90000 (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 | 0.3096 | 0.0221 | 0.3317 | -0.2193 | -0.0199 | -0.2392 |
| 8.00 | 0.6131 | 0.0881 | 0.7013 | -0.4343 | -0.0793 | -0.5136 |
| 12.00 | 0.9048 | 0.1632 | 1.0679 | -0.6409 | -0.1468 | -0.7877 |
| 16.00 | 1.1788 | 0.2089 | 1.3876 | -0.8350 | -0.1880 | -1.0230 |
| 20.00 | 1.4298 | 0.1967 | 1.6265 | -1.0128 | -0.1770 | -1.1898 |
| 24.00 | 1.6531 | 0.0557 | 1.7088 | -1.1709 | -0.0501 | -1.2211 |
| 28.00 | 1.8441 | -0.2001 | 1.6440 | -1.3063 | 0.1801 | -1.1262 |

1 *****

FIN SET 2 CA PARTIAL OUTPUT

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

| | | | |
|---|---|---|---|
| MACH NO = | 0.15 | REYNOLDS NO = | 3.032E+06 /M |
| SIDESLIP = | 0.00 DEG | ROLL = | 0.00 DEG |
| REF AREA = | 0.003 M**2 | MOMENT CENTER = | 0.513 M |
| REF LENGTH = | 0.06 M | LAT REF LENGTH = | 0.06 M |

SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS

| | |
|---|---|
| SKIN FRICTION | 0.0082 |
| SUBSONIC PRESSURE | 0.0005 |
| TRANSONIC WAVE | 0.0000 |
| SUPERSONIC WAVE | 0.0000 |
| LEADING EDGE | 0.0000 |
| TRAILING EDGE | 0.0000 |
| TOTAL CAO | 0.0087 |

FIN AXIAL FORCE DUE TO ANGLE OF ATTACK

| ALPHA | CA DUE TO LIFT (SINGLE PANEL) | CA-TOTAL (4 FINS) |
|---|---|---|
| 0.00 | 0.0000 | 0.0349 |
| 4.00 | 0.0000 | 0.0348 |
| 8.00 | 0.0000 | 0.0346 |
| 12.00 | 0.0000 | 0.0342 |
| 16.00 | 0.0000 | 0.0336 |
| 20.00 | 0.0000 | 0.0328 |
| 24.00 | 0.0000 | 0.0319 |
| 28.00 | 0.0000 | 0.0308 |

1 *****

FIN SET 2 CN, CM PARTIAL OUTPUT

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

NORMAL FORCE SLOPE AT ALPHA ZERO, CNA = 0.03489/DEG (1 PANEL)
CENTER OF PRESSURE FOR LINEAR CN = -4.31456 (CALIBERS FROM C.G.)
CENTER OF PRESSURE FOR NON-LINEAR CN = -4.44042 (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 | 0.2782 | 0.0044 | 0.2827 | -1.2005 | -0.0196 | -1.2201 |
| 8.00 | 0.5511 | 0.0176 | 0.5687 | -2.3776 | -0.0782 | -2.4558 |
| 12.00 | 0.8132 | 0.0362 | 0.8494 | -3.5085 | -0.1607 | -3.6691 |
| 16.00 | 1.0594 | -0.0656 | 0.9939 | -4.5710 | 0.2912 | -4.2799 |
| 20.00 | 1.2851 | -0.2719 | 1.0132 | -5.5446 | 1.2073 | -4.3373 |
| 24.00 | 1.4857 | -0.5218 | 0.9639 | -6.4103 | 2.3170 | -4.0933 |
| 28.00 | 1.6575 | -0.7295 | 0.9279 | -7.1512 | 3.2394 | -3.9118 |

1 *****

AERODYNAMIC FORCE AND MOMENT SYNTHESIS

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

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

| ALPHA | CN | CM | CA | CY | CLN | CLL |
|---|---|---|---|---|---|---|
| 0.00 | 0.0000 | 0.0000 | 0.0489 | 0.0000 | -0.0000 | 0.0000 |
| 4.00 | 0.4754 | -0.3367 | 0.0489 | 0.0000 | -0.0000 | -0.0000 |
| 8.00 | 0.9623 | -0.6816 | 0.0489 | 0.0000 | -0.0000 | -0.0000 |
| 12.00 | 1.3415 | -0.9502 | 0.0489 | 0.0000 | -0.0000 | -0.0000 |
| 16.00 | 1.6083 | -1.1392 | 0.0489 | 0.0000 | -0.0000 | -0.0000 |
| 20.00 | 1.7083 | -1.2100 | 0.0489 | 0.0000 | -0.0000 | -0.0000 |
| 24.00 | 1.6662 | -1.1802 | 0.0489 | 0.0000 | -0.0000 | -0.0000 |
| 28.00 | 1.6439 | -1.1644 | 0.0489 | 0.0000 | -0.0000 | -0.0000 |

1 *****

AERODYNAMIC FORCE AND MOMENT SYNTHESIS

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

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

| ALPHA | CN | CM | CA | CY | CLN | CLL |
|---|---|---|---|---|---|---|
| 0.00 | 0.0000 | 0.0000 | 0.0349 | 0.0000 | -0.0000 | 0.0000 |
| 4.00 | 0.2116 | -0.9132 | 0.0349 | 0.0000 | -0.0000 | 0.0000 |
| 8.00 | 0.4093 | -1.7661 | 0.0349 | -0.0000 | 0.0000 | -0.0000 |
| 12.00 | 0.7043 | -3.0389 | 0.0349 | -0.0000 | 0.0000 | 0.0000 |
| 16.00 | 0.9483 | -4.0914 | 0.0349 | 0.0000 | -0.0000 | 0.0000 |
| 20.00 | 1.0110 | -4.3621 | 0.0349 | 0.0000 | -0.0000 | -0.0000 |
| 24.00 | 0.9833 | -4.2424 | 0.0349 | 0.0000 | -0.0000 | 0.0000 |
| 28.00 | 0.9337 | -4.0287 | 0.0349 | 0.0000 | -0.0000 | -0.0000 |

1 *****

AERODYNAMIC FORCE AND MOMENT SYNTHESIS

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 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 | 0.0000 | 0.0000 |
| 4.00 | 2 | 5.6009 | 0.2377 |
| 4.00 | 3 | 0.0000 | 0.0000 |
| 4.00 | 4 | -5.6009 | -0.2377 |
| 8.00 | 1 | -0.0000 | -0.0000 |
| 8.00 | 2 | 10.8000 | 0.4812 |
| 8.00 | 3 | 0.0000 | 0.0000 |
| 8.00 | 4 | -10.8000 | -0.4812 |
| 12.00 | 1 | 0.0000 | 0.0000 |
| 12.00 | 2 | 15.3667 | 0.6707 |
| 12.00 | 3 | 0.0000 | 0.0000 |
| 12.00 | 4 | -15.3667 | -0.6707 |
| 16.00 | 1 | 0.0000 | 0.0000 |
| 16.00 | 2 | 19.6092 | 0.8042 |
| 16.00 | 3 | 0.0000 | 0.0000 |
| 16.00 | 4 | -19.6092 | -0.8042 |
| 20.00 | 1 | -0.0000 | -0.0000 |
| 20.00 | 2 | 23.1469 | 0.8541 |
| 20.00 | 3 | 0.0000 | 0.0000 |
| 20.00 | 4 | -23.1469 | -0.8541 |
| 24.00 | 1 | -0.0000 | -0.0000 |
| 24.00 | 2 | 26.6162 | 0.8331 |
| 24.00 | 3 | 0.0000 | 0.0000 |
| 24.00 | 4 | -26.6162 | -0.8331 |
| 28.00 | 1 | -0.0000 | -0.0000 |
| 28.00 | 2 | 29.9318 | 0.8220 |
| 28.00 | 3 | 0.0000 | 0.0000 |
| 28.00 | 4 | -29.9318 | -0.8220 |

1 *****

AERODYNAMIC FORCE AND MOMENT SYNTHESIS

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 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 | 3.0027 | 0.1058 |
| 4.00 | 3 | 0.0000 | 0.0000 |
| 4.00 | 4 | -3.0027 | -0.1058 |
| 8.00 | 1 | 0.0000 | 0.0000 |
| 8.00 | 2 | 5.7729 | 0.2047 |
| 8.00 | 3 | 0.0000 | 0.0000 |
| 8.00 | 4 | -5.7729 | -0.2047 |
| 12.00 | 1 | -0.0000 | -0.0000 |
| 12.00 | 2 | 9.9057 | 0.3522 |
| 12.00 | 3 | -0.0000 | -0.0000 |
| 12.00 | 4 | -9.9057 | -0.3522 |
| 16.00 | 1 | -0.0000 | -0.0000 |
| 16.00 | 2 | 14.2877 | 0.4741 |
| 16.00 | 3 | 0.0000 | 0.0000 |
| 16.00 | 4 | -14.2877 | -0.4741 |
| 20.00 | 1 | 0.0000 | 0.0000 |
| 20.00 | 2 | 18.4680 | 0.5055 |
| 20.00 | 3 | 0.0000 | 0.0000 |
| 20.00 | 4 | -18.4680 | -0.5055 |
| 24.00 | 1 | -0.0000 | -0.0000 |
| 24.00 | 2 | 22.7479 | 0.4916 |
| 24.00 | 3 | -0.0000 | -0.0000 |
| 24.00 | 4 | -22.7479 | -0.4916 |
| 28.00 | 1 | -0.0000 | -0.0000 |
| 28.00 | 2 | 26.7339 | 0.4669 |
| 28.00 | 3 | 0.0000 | 0.0000 |
| 28.00 | 4 | -26.7339 | -0.4669 |

1 *****

AERODYNAMIC FORCE AND MOMENT SYNTHESIS

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 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 | 1.4267 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4408 |
| 4.00 | 1.4024 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4655 |
| 8.00 | 1.3616 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4719 |
| 12.00 | 1.3148 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4700 |
| 16.00 | 1.2671 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4638 |
| 20.00 | 1.2217 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4615 |
| 24.00 | 1.1805 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4624 |
| 28.00 | 1.1443 | 0.5659 | 0.9348 | 0.3899 | 0.7083 | 0.6917 | 0.4630 |

1 *****

AERODYNAMIC FORCE AND MOMENT SYNTHESIS

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

| | | | |
|---|---|---|---|
| MACH NO = | 0.15 | REYNOLDS NO = | 3.032E+06 /M |
| SIDESLIP = | 0.00 DEG | ROLL = | 0.00 DEG |
| REF AREA = | 0.003 M**2 | MOMENT CENTER = | 0.513 M |
| REF LENGTH = | 0.06 M | LAT REF LENGTH = | 0.06 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 | 1.4019 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.4025 |
| 4.00 | 1.3773 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.3883 |
| 8.00 | 1.3361 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.3753 |
| 12.00 | 1.2894 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.3662 |
| 16.00 | 1.2424 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.3637 |
| 20.00 | 1.1983 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.3639 |
| 24.00 | 1.1588 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.3638 |
| 28.00 | 1.1247 | 0.5201 | 0.9347 | 0.3645 | 4.3146 | 4.2515 | 0.3637 |

NOTE - XCP-W(B) USED FOR STABILITY ONLY DIFFERENT VALUES USED FOR HINGE MOMENTS
1         *****

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

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 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 | 9.22E-02 | 1.30E-08 | -9.22E-02 | | | | |
| 8.0 | -1.28E-09 | 1.89E-01 | 2.64E-08 | -1.89E-01 | | | | |
| 12.0 | 4.75E-09 | 2.63E-01 | 3.75E-08 | -2.63E-01 | | | | |
| 16.0 | 5.08E-09 | 3.11E-01 | 4.81E-08 | -3.11E-01 | | | | |
| 20.0 | -1.09E-08 | 3.28E-01 | 5.86E-08 | -3.28E-01 | | | | |
| 24.0 | -1.49E-08 | 3.21E-01 | 6.94E-08 | -3.21E-01 | | | | |
| 28.0 | -3.85E-08 | 3.17E-01 | 8.05E-08 | -3.17E-01 | | | | |

1 *****

* REV 3/99 ***** CASE 1
AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS 
FIN SET 2 PANEL BENDING MOMENTS (ABOUT EXPOSED ROOT CHORD)

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

| | | | |
|---|---|---|---|
| MACH NO = | 0.15 | REYNOLDS NO = | 3.032E+06 /M |
| SIDESLIP = | 0.00 DEG | ROLL = | 0.00 DEG |
| REF AREA = | 0.003 M**2 | MOMENT CENTER = | 0.513 M |
| REF LENGTH = | 0.06 M | LAT REF LENGTH = | 0.06 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 | -1.51E-09 | 3.83E-02 | 3.13E-09 | -3.83E-02 | | | | |
| 8.0 | 3.10E-09 | 7.17E-02 | 1.39E-09 | -7.17E-02 | | | | |
| 12.0 | -1.39E-08 | 1.20E-01 | -1.34E-08 | -1.20E-01 | | | | |
| 16.0 | -1.02E-08 | 1.61E-01 | 3.06E-08 | -1.61E-01 | | | | |
| 20.0 | 5.68E-09 | 1.72E-01 | 3.08E-08 | -1.72E-01 | | | | |
| 24.0 | -3.85E-08 | 1.67E-01 | -1.87E-08 | -1.67E-01 | | | | |
| 28.0 | -2.95E-08 | 1.58E-01 | 8.15E-08 | -1.58E-01 | | | | |

1 *****

FIN SET 1 PANEL HINGE MOMENTS (ABOUT HINGE LINE)

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 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 | 5.48E-02 | 8.08E-09 | -5.48E-02 | | | | |
| 8.0 | -7.80E-10 | 1.02E-01 | 1.62E-08 | -1.02E-01 | | | | |
| 12.0 | 2.91E-09 | 1.29E-01 | 2.30E-08 | -1.29E-01 | | | | |
| 16.0 | 3.16E-09 | 1.39E-01 | 2.99E-08 | -1.39E-01 | | | | |
| 20.0 | -6.82E-09 | 1.40E-01 | 3.66E-08 | -1.40E-01 | | | | |
| 24.0 | -9.28E-09 | 1.40E-01 | 4.33E-08 | -1.40E-01 | | | | |
| 28.0 | -2.40E-08 | 1.39E-01 | 5.01E-08 | -1.39E-01 | | | | |

1 *****

FIN SET 2 PANEL HINGE MOMENTS (ABOUT HINGE LINE)

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

MACH NO  =       0.15                REYNOLDS NO = 3.032E+06 /M
SIDESLIP =       0.00 DEG                   ROLL =      0.00 DEG
REF AREA =      0.003 M**2         MOMENT CENTER =     0.513 M
REF LENGTH =     0.06 M           LAT REF LENGTH =      0.06 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 | -9.22E-11 | 1.57E-03 | 1.91E-10 | -1.57E-03 | | | | |
| 8.0 | 1.95E-10 | 1.66E-03 | 8.74E-11 | -1.66E-03 | | | | |
| 12.0 | -9.00E-10 | 4.20E-04 | -8.65E-10 | -4.20E-04 | | | | |
| 16.0 | -6.66E-10 | -2.25E-03 | 1.99E-09 | 2.25E-03 | | | | |
| 20.0 | 3.69E-10 | -3.86E-03 | 2.00E-09 | 3.86E-03 | | | | |
| 24.0 | -2.50E-09 | -3.13E-03 | -1.21E-09 | 3.13E-03 | | | | |
| 28.0 | -1.92E-09 | -1.90E-03 | 5.30E-09 | 1.90E-03 | | | | |

1         *****

STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

| | ----- LONGITUDINAL ----- | | | -- LATERAL DIRECTIONAL -- | | |
|---|---|---|---|---|---|---|
| ALPHA | CN | CM | CA | CY | CLN | CLL |
| 0.00 | 0.000 | 0.000 | 0.431 | 0.000 | -0.000 | 0.000 |
| 4.00 | 1.184 | -0.784 | 0.431 | 0.000 | -0.000 | -0.000 |
| 8.00 | 2.466 | -1.455 | 0.429 | 0.000 | 0.000 | -0.000 |
| 12.00 | 3.832 | -2.594 | 0.422 | 0.000 | 0.000 | 0.000 |
| 16.00 | 5.061 | -3.329 | 0.407 | 0.000 | -0.000 | -0.000 |
| 20.00 | 5.869 | -2.810 | 0.378 | 0.000 | -0.000 | -0.000 |
| 24.00 | 6.397 | -1.603 | 0.326 | 0.000 | -0.000 | 0.000 |
| 28.00 | 6.966 | -0.306 | 0.244 | 0.000 | -0.000 | -0.000 |

| ALPHA | CL | CD | CL/CD | X-C.P. |
|---|---|---|---|---|
| 0.00 | 0.000 | 0.431 | 0.000 | -0.740 |
| 4.00 | 1.151 | 0.512 | 2.246 | -0.662 |
| 8.00 | 2.382 | 0.768 | 3.103 | -0.590 |
| 12.00 | 3.661 | 1.209 | 3.027 | -0.677 |
| 16.00 | 4.753 | 1.786 | 2.661 | -0.658 |
| 20.00 | 5.386 | 2.362 | 2.280 | -0.479 |
| 24.00 | 5.712 | 2.900 | 1.969 | -0.251 |
| 28.00 | 6.036 | 3.485 | 1.732 | -0.044 |

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

1 *****

* REV 3/99 ***** CASE 1

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS 

STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

---------- DERIVATIVES (PER DEGREE) ----------

| ALPHA | CNA | CMA | CYB | CLNB | CLLB |
|---|---|---|---|---|---|
| 0.00 | 0.2837 | -0.2099 | -0.2828 | 0.2091 | -0.0000 |
| 4.00 | 0.3082 | -0.1819 | -0.2900 | 0.2047 | -0.0007 |
| 8.00 | 0.3310 | -0.2255 | -0.2737 | 0.1022 | 0.0086 |
| 12.00 | 0.3243 | -0.2337 | -0.2241 | -0.1117 | 0.0239 |
| 16.00 | 0.2540 | -0.0264 | -0.2320 | -0.0796 | 0.0714 |
| 20.00 | 0.1669 | 0.2142 | -0.1679 | -0.2585 | 0.1094 |
| 24.00 | 0.1371 | 0.3130 | -0.1358 | -0.3238 | 0.1313 |
| 28.00 | 0.1471 | 0.3356 | -0.1083 | -0.3635 | 0.1406 |

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 | 0.00 | 0.00 | | | | |
| 2 | 0.00 | 0.00 | 0.00 | 0.00 | | | | |

1 *****

BODY ALONE DYNAMIC DERIVATIVES

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------

| ALPHA | CNQ | CMQ | CAQ | CNAD | CMAD |
|---|---|---|---|---|---|
| 0.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |
| 4.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |
| 8.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |
| 12.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |
| 16.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |
| 20.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |
| 24.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |
| 28.00 | 0.255 | -0.522 | 0.000 | 0.481 | 0.112 |

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

1 *****

BODY ALONE DYNAMIC DERIVATIVES

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------

| ALPHA | CYR | CLNR | CLLR | CYP | CLNP | CLLP |
|---|---|---|---|---|---|---|
| 0.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |
| 4.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |
| 8.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |
| 12.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |
| 16.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |
| 20.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |
| 24.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |
| 28.00 | 0.270 | -0.522 | 0.000 | 0.000 | 0.000 | 0.000 |

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

1 *****

BODY + 2 FIN SETS DYNAMIC DERIVATIVES

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------

| ALPHA | CNQ | CMQ | CAQ | CNAD | CMAD |
|---|---|---|---|---|---|
| 0.00 | 1.540 | -5.224 | 0.000 | 0.667 | -0.354 |
| 4.00 | 1.433 | -4.651 | 0.000 | 0.667 | -0.354 |
| 8.00 | 1.210 | -3.801 | 0.000 | 0.667 | -0.354 |
| 12.00 | 0.793 | -2.209 | 0.000 | 0.667 | -0.354 |
| 16.00 | 0.343 | -0.575 | 0.000 | 0.667 | -0.354 |
| 20.00 | 0.139 | -0.065 | 0.000 | 0.667 | -0.354 |
| 24.00 | 0.185 | -0.337 | 0.000 | 0.667 | -0.354 |
| 28.00 | 0.370 | -0.914 | 0.000 | 0.667 | -0.354 |

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

1 *****

BODY + 2 FIN SETS DYNAMIC DERIVATIVES

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

MACH NO = 0.15 REYNOLDS NO = 3.032E+06 /M
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 0.003 M**2 MOMENT CENTER = 0.513 M
REF LENGTH = 0.06 M LAT REF LENGTH = 0.06 M

------------ DYNAMIC DERIVATIVES (PER DEGREE) -----------

| ALPHA | CYR | CLNR | CLLR | CYP | CLNP | CLLP |
|---|---|---|---|---|---|---|
| 0.00 | 1.554 | -5.224 | 0.000 | -0.000 | -0.000 | -0.265 |
| 4.00 | 1.548 | -5.206 | -0.010 | 0.008 | -0.035 | -0.275 |
| 8.00 | 1.540 | -5.180 | -0.018 | 0.012 | -0.053 | -0.279 |
| 12.00 | 1.496 | -5.001 | -0.007 | -0.028 | 0.121 | -0.258 |
| 16.00 | 1.553 | -5.255 | -0.035 | 0.040 | -0.171 | -0.271 |
| 20.00 | 1.556 | -5.268 | -0.025 | 0.026 | -0.111 | -0.214 |
| 24.00 | 1.566 | -5.306 | -0.019 | 0.015 | -0.066 | -0.179 |
| 28.00 | 1.584 | -5.375 | -0.021 | 0.022 | -0.095 | -0.207 |

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

1 *****

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

```
 * REV 3/99 *****     CASE   2
               AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS          PAGE   2
                  STATIC AERODYNAMIC COEFFICIENTS TRIMMED IN PITCH

      ******* FLIGHT CONDITIONS AND REFERENCE QUANTITIES *******
    MACH NO  =       0.15                REYNOLDS NO = 3.032E+06 /M
    SIDESLIP =       0.00 DEG                   ROLL =      0.00 DEG
    REF AREA =      0.003 M**2         MOMENT CENTER =     0.513 M
    REF LENGTH =     0.06 M           LAT REF LENGTH =      0.06 M

        ALPHA     DELTA       CL        CD        CN        CA
         0.00      0.00     0.000     0.431     0.000     0.431
         4.00     -2.01     0.969     0.504     1.001     0.435
         8.00     -3.67     2.047     0.715     2.127     0.424
        12.00     -6.44     3.075     1.059     3.228     0.397
        16.00     -9.15     4.024     1.514     4.286     0.346
        20.00    -11.25     4.807     2.040     5.215     0.273
        24.00    -12.91     5.436     2.602     6.024     0.166
        28.00    -13.94     6.076     3.259     6.894     0.025
   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 ***
```