

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 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 DB14
17 PLOT
18 PRESSURES
19 SAVE
20 NEXT CASE
21 $TRIM SET=2.,$
22 PRINT AERO TRIM PLOT
23 NEXT CASE

```

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 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

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
AXISYMMETRIC BODY DEFINITION

SHAPE	NOSE OGIVE	CENTERBODY CYLINDER	AFT BODY -----	TOTAL	
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
0.3136	0.3804	0.4472	0.5140	0.5808	0.6476
0.7812	0.8480*				0.7144

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*				

NOTE - * INDICATES SLOPE DISCONTINUOUS POINTS

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

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

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
GEOMETRIC RESULTS FOR FIN SETSFIN 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

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

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

***** 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

***** 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

***** 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

***** 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

***** 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

***** 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

***** 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

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
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

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
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

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
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

***** 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

***** 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				

***** 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				

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
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				

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
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				

***** 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	----- LONGITUDINAL -----			-- LATERAL DIRECTIONAL --		
	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

***** 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				

***** 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

***** 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

***** 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

***** 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

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

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS
 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 ***