INTRODUCTIN TO USAF DIGITAL DATCOM

By P.Hajipour

What is DATCOM?

A Software that calculates aerodynamic stability and control characteristics

Aerodynamics forces acting on an Airplane during flight

Steady State Flight	Maneuvers	Control Deflections
C_{L_α}	C_{m_q}	$C_{m_{de}}$
C_{n_β}	$C_{n_{\dot{\beta}}}$	$C_{l_{da}}$

How Should I Install It?

- ❑ Run setup file
- ❑ Install Notepad ++
- ❑ Apply these settings in control panel:

1. After installing the Datcom+ package, Press Windows key + r. Type in "SystemPropertiesAdvanced.exe".
2. Click on the Environment Variables box.
3. Change the environment variable for DATCOMROOT, removing the leading and trailing double-quote. Save and get out of this.
4. In the Datcom\bin directory on your desktop, edit the Datcom.bat file. On lines 14, 15, and 16, remove the double-quotes (leading and trailing).

   set PREDAT_PROGRAM="predat"

   set DATCOM_PROGRAM="digdat"

   set DATCOM_MODELER_PROGRAM="datcom-modeler"

5. You should be able to run the programs now.

Who Built it?

Theoretical Methods
Experimental Data

Geometry

Flight Condition

DATCOM STABILITY & CONTROL

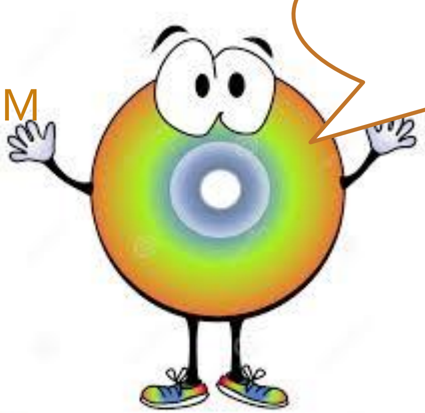
Aerodynamic Forces

MD (1979)

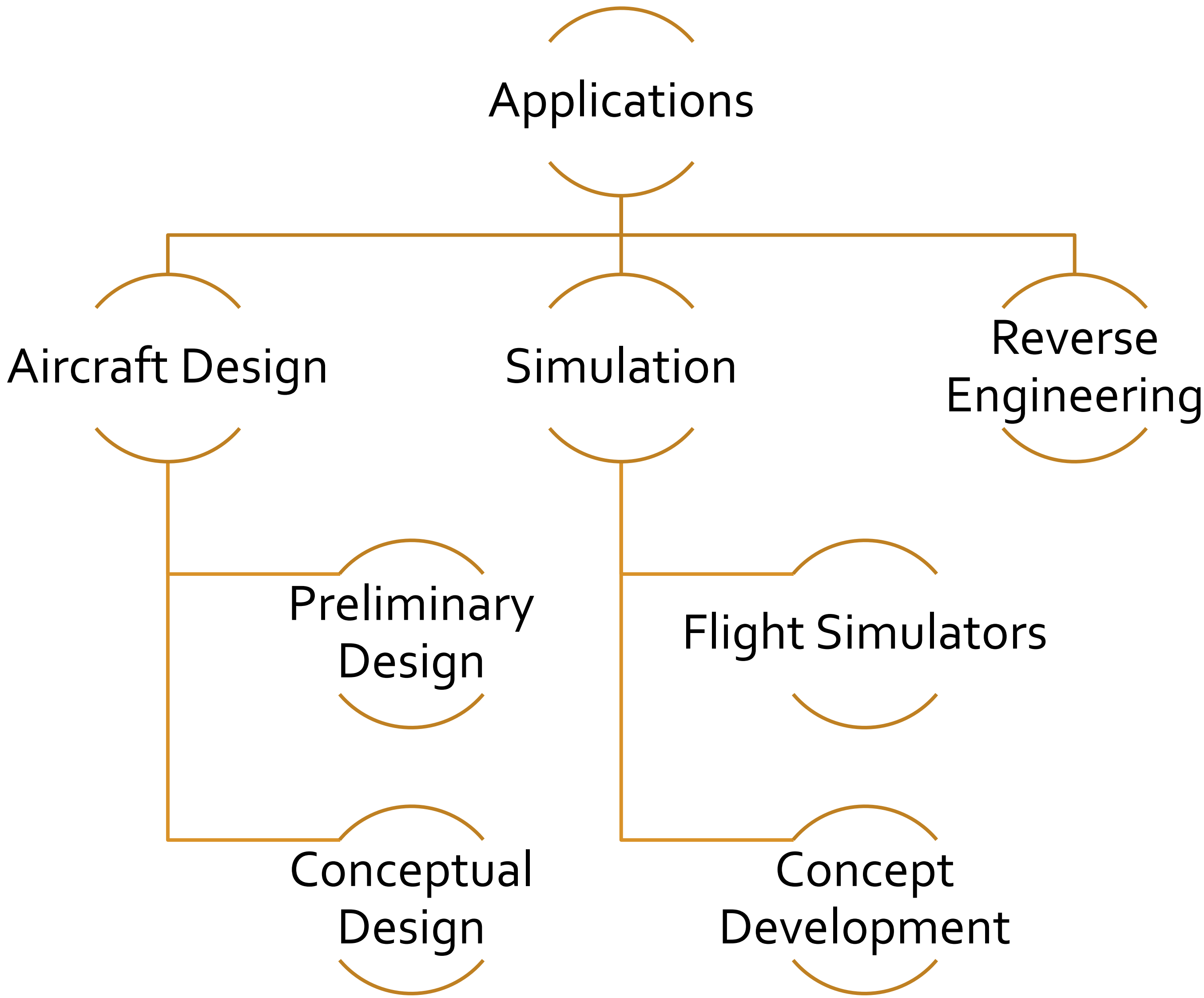
Isn't it too old?

Old but still very useful...

Digital DATCOM



When does it come in handy?



Any Development?

DATCOM +

- Improved Input file
- Graphical Representation
- XML & CVS Output

MATLAB Toolbox

- Direct Import

Is DATCOM Applicable for my case?

CONFIGURATION	SPEED REGIME	STATIC AERODYNAMIC CHARACTERISTIC OUTPUT														DYNAMIC STABILITY OUTPUT								
		C_{D_0}	C_D	C_L	C_m	C_N	C_A	C_{L_α}	C_{m_α}	C_{Y_β}	C_{n_β}	C_{l_β}	q/q_∞	ϵ	$\frac{d\epsilon}{d\alpha}$	C_{L_q}	C_{m_q}	$C_{L_{\dot\alpha}}$	$C_{m_{\dot\alpha}}$	C_{l_p}	C_{Y_p}	C_{n_p}	C_{n_r}	C_{l_r}
BODY	SUBSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●					●
	△5 TRANSONIC	●	●					●	●	●	●	●				●	●	●	●					●
	SUPERSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●					●
	HYPERSONIC	●	●	●	●																			
WING	SUBSONIC	●	●	●	●																			
	TRANSONIC	□	▲	▲																				
	SUPERSONIC	●	□	□																				
	HYPERSONIC	●	□	□																				
HORIZONTAL TAIL	SUBSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●	●	●	●	●	
	TRANSONIC	□	▲	▲		▲	▲	□	□			▲				□	□	□	□					
	SUPERSONIC	●	□	□		□	□	●	●	□	□	□				□	□	●	●	●	●	●		
	HYPERSONIC	●	□	□		□	□	●	●	□	□	□												
VERTICAL TAIL OR VENTRAL FIN	SUBSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●	●	●	●	●	
	TRANSONIC	□						●	●							●	●	●	●					
	△3 SUPERSONIC	●		●	●	●	●	●	●	□	□	□				●	●	●	●					
	HYPERSONIC	●		●	●	●	●	●	●	□	□	□												
WING-BODY	△1 SUBSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●	●	●	●	●	●
	TRANSONIC	□	▲	●		▲	▲	□	□	●	●	●				□	□	□	□					
	SUPERSONIC	●	□	□		□	□	●	●	●	●	□				●	●	●	□	●	●			
	HYPERSONIC	●	□	□		□	□	●	●	●	●	□												
HORIZONTAL TAIL-BODY	SUBSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●					
	TRANSONIC	□	●	□		▲	▲	□	□	●	●	●				□	□	□	□					
	SUPERSONIC	●	□	□		□	□	●	●	●	●	□				●	●	●	□					
	HYPERSONIC	●	□	□		□	□	●	●	●	●	□												
VERTICAL TAIL-VENTRAL FIN-BODY	SUBSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●					
	TRANSONIC	□	□	▲		▲	▲	●	●							●	●	●	●					
	△3 SUPERSONIC	●	●	●	●	●	●	●	●	□	□	□				●	●	●	●					
	HYPERSONIC	●	●	●		●	●	●	●	□	□	□												
WING-BODY-HORIZONTAL TAIL	△1 SUBSONIC	□	□	□	□	□	□	□	□	●	●	●	□	□	□	□	□	□	□	□	□	□	□	
	TRANSONIC	□	▲	▲		▲	▲	□	□				□	□	□	□	□	□	□					
	SUPERSONIC	□	□	□		□	□	□	□	●	●	□	□	□	□	□	□	□	□	□	□			
	HYPERSONIC	□	□	□		□	□	□	□	●	●	□	□	□	□									
WING-BODY-VERTICAL TAIL-VENTRAL FIN	△1 SUBSONIC	●	●	●	●	●	●	●	●	●	●	●				●	●	●	●	●	●	●	●	
	△2 TRANSONIC	□	▲	□		▲	▲	□	□	●	●					□	□	□	□					
	△3 SUPERSONIC	●	□	□		□	□	●	●	●	●	□				●	●	●	●					
	HYPERSONIC	●	□	□		□	□	●	●	●	●	□												
WING-BODY-HORIZONTAL TAIL-VERTICAL TAIL-VENTRAL FIN	△1 SUBSONIC	□	□	□	□	□	□	□	□	●	●	●	□	□	□	□	□	□	□	□	□	□	□	□
	TRANSONIC	□	▲	▲		▲	▲	□	□				□	□	□	□	□	□	□					
	△3 SUPERSONIC	□	□	□		□	□	□	□	●	●	□	□	□	□	□	□	□	□					
	HYPERSONIC	□	□	□		□	□	□	□	●	●	□	□	□	□									

● OUTPUT AVAILABLE
□ OUTPUT ONLY FOR CONFIGURATIONS WITH STRAIGHT TAPERED SURFACES
▲ OUTPUT ONLY WITH EXPERIMENTAL DATA INPUT

What Are The Limitations?

Rudder derivatives

Modeling three-surface aircrafts

Drag Estimation

Multi Control Input

Incomplete Dynamic Derivatives

Modeling V-Tail

How Can I Write The Input File?

Use Notepad ++

Save Input file like : filename.dcm

Start By editing sample file

Input numbers in Float Format: (8.23   7.0)

All Characters are UPPERCASE

Arrays are defined like : MACH(1) = 0.1,0.2,0.3,

How Can I Write The Input File?

Control Cards

```
DIM M
DERIV DEG
DAMP
```

Derivatives Unit : 1/Deg
Dynamic Derivatives

TABLE 8 INPUT UNIT OPTIONS

UNITS SYSTEM (LENGTH-FORCE-TIME, ℓ-F-T)	CONTROL CARD	GEOMETRY UNITS (ℓ)	SURFACE ROUGHNESS RØUGFC	PRESSURE P_∞ (F/A)	TEMPERATURE T_∞ (DEG)	REYNOLDS NUMBER PER UNIT LENGTH
FOOT-POUND-SECOND	DIM FT	FOOT	INCH	lb/ft^2	^oR	1/FT
INCH-POUND-SECOND	DIM IN	INCH	INCH	lb/in^2	^oR	1/FT
METER-NEWTON-SECOND	DIM M	METER	CM	N/M^2	^oK	1/M
CENTIMETER-NEWTON-SECOND	DIM CM	CM	CM	N/CM^2	^oK	1/M

Structure of the Input File

- Each NAMELIST starts and ends with “$”
- Order of Input is not important

NAMELISTS

Flight Condition	FLTCON
Options	OPTINS
Synthesis	SYNTHS
Body	BODY
.... Planform	WGPLNF - HTPLNF - VTPLNF
...Characteristics	WGSCHR - HTSCHR - VTSCHR
Symmetric Flap	SYMFLP
Asymmetric Flap	ASYFLP

Flight Condition

Variable	Definition
WT	Take-off Weight
NMACH	Number of Mach numbers
MACH	Array of Mach numbers
NALT	Number of altitudes
ALT	Array of altitudes
NALPHA	Number of angle of attacks
ALSCHD	Array of angle of attacks
LOOP	Program Looping Control

=1 VARY ALTITUDE AND MACH TOGETHER, DEFAULT
=2 VARY MACH, AT FIXED ALTITUDE
=3 VARY ALTITUDE, AT FIXED MACH

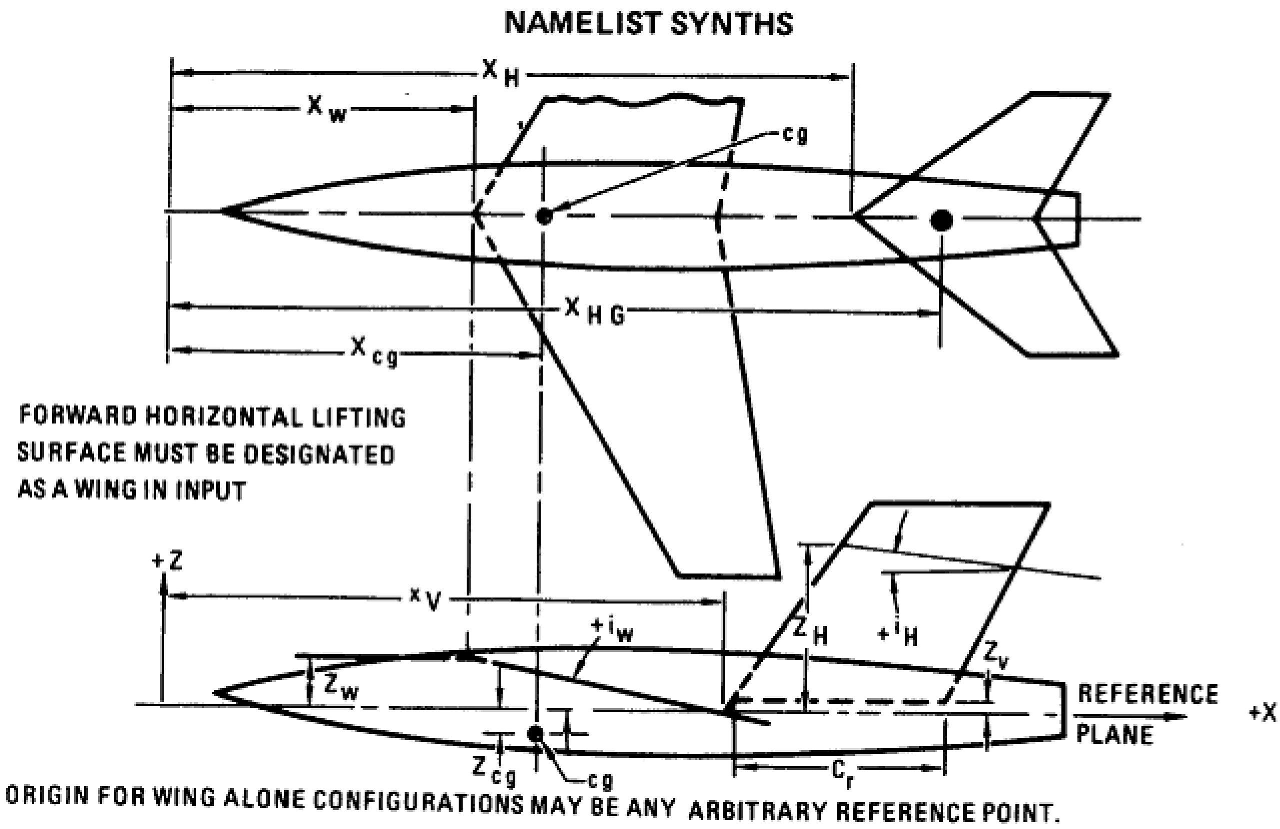
OPTIONS

Variable	Definition
SREF	Reference area
CBARR	Mean Aerodynamic chord
BLREF	Wing Span
ROUGFC	Surface Roughness

TYPE OF SURFACE	EQUIVALENT SAND ROUGHNESS	
	INCHES	cm
AERODYNAMICALLY SMOOTH	0	0
POLISHED METAL OR WOOD	$0.02 - 0.08 \times 10^{-3}$	$0.051 - 0.203 \times 10^{-3}$
NATURAL SHEET METAL	0.16×10^{-3}	0.406×10^{-3}
SMOOTH MATTE PAINT, CAREFULLY APPLIED	0.25×10^{-3}	0.635×10^{-3}
STANDARD CAMOUFLAGE PAINT, AVERAGE APPLICATION	0.40×10^{-3}	1.016×10^{-3}
CAMOUFLAGE PAINT, MASS-PRODUCTION SPRAY	1.20×10^{-3}	3.048×10^{-3}
DIP-GALVANIZED METAL SURFACE	6×10^{-3}	15.240×10^{-3}
NATURAL SURFACE OF CAST IRON	10×10^{-3}	25.400×10^{-3}

Synthesis

Define a coordinate system at A/C nose



Synthesis

ENGINEERING SYMBOL	VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS
x_{cg}	XCG	–	LONGITUDINAL LOCATION OF CG, (MOMENT REF. CENTER)	ℓ
z_{cg}	ZCG	–	VERTICAL LOCATION OF CG RELATIVE TO REFERENCE PLANE	ℓ
x_W	XW	–	LONGITUDINAL LOCATION OF THEORETICAL WING APEX	ℓ
z_W	ZW	–	VERTICAL LOCATION OF THEORETICAL WING APEX RELATIVE TO REFERENCE PLANE	ℓ
i_W	ALIW	–	WING ROOT CHORD INCIDENCE ANGLE MEASURED FROM REFERENCE PLANE	DEG
$\triangle 2$ x_H	XH	–	LONGITUDINAL LOCATION OF THEORETICAL HORIZONTAL TAIL APEX	ℓ
$\triangle 2$ z_H	ZH	–	VERTICAL LOCATION OF THEORETICAL HORIZONTAL TAIL APEX RELATIVE TO REFERENCE PLANE	ℓ
i_H	ALIH	–	HORIZONTAL TAIL ROOT CHORD INCIDENCE ANGLE MEASURED FROM REFERENCE PLANE	DEG
x_V	XV	–	LONGITUDINAL LOCATION OF THEORETICAL VERTICAL TAIL APEX	ℓ
x_{V_F}	XVF	–	LONGITUDINAL LOCATION OF THEORETICAL VENTRAL FIN APEX	ℓ
z_V	ZV	–	VERTICAL LOCATION OF THEORETICAL VERTICAL TAIL APEX	ℓ
z_{V_F}	ZVF	–	VERTICAL LOCATION OF THEORETICAL VENTRAL TAIL APEX	ℓ
	SCALE	–	VEHICLE SCALE FACTOR (MULTIPLIER TO INPUT DIMENSIONS)	–
	VERTUP	–	VERTUP = .TRUE. VERTICAL PANEL ABOVE REF PLANE (DEFAULT)	–
		–	VERTUP = .FALSE. VERTICAL PANEL BLEOW REF PLANE	–
$\triangle 1$ x_{HG}	HINAX	–	LONGITUDINAL LOCATION OF HORIZONTAL TAIL HINGE AXIS	ℓ

Body

Variable	Definition
NX	Number of Sections
X	Array of longitudinal Distance
ZU	Upper body
ZL	Lower Body
R	Section Equivalent Radius
S	Section Area
P	Section Periphery
ITYPE	= 1 Straight Wing
	=2 Swept Wing

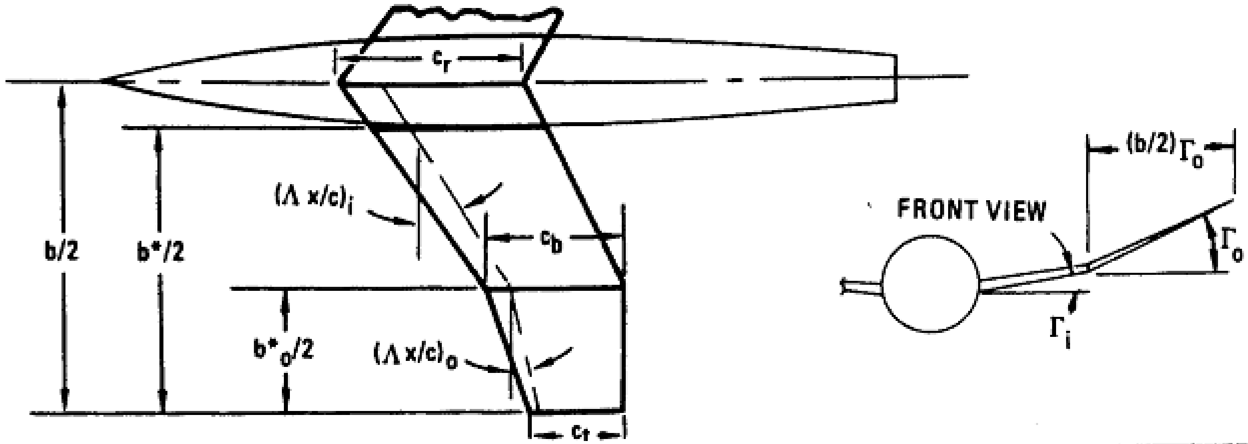
At Least One is Required
(2 At most!)

Maximum Array Dimension is 20

WGPLNF - HTPLNF - VTPLNF

Variable	Definition
CHRDTP	Tip Chord
CHRDR	Root Chord
SSPNE	Semi Span (Exposed)
SSPN	Semi Span (Theoretical)
SAVSI	Sweep Angle
CHSTAT	=0.0 Leading edge sweep =1.0 Trailing Edge sweep
TWISTA	Twist angle (negative L.E rotated down)
DHDADI	Dihedral Angle
TYPE	=1.0 Straight Taperd Planform

WGPLNF - HTPLNF - VTPLNF



INPUT DATA FOR			ENGINEERING SYMBOL	VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS
WGPLNF	HTPLNF	VTPLNF VFPLNF					
●	●	●	c_t	CHRDTP	–	TIP CHORD	ℓ
●	●	●	$b^*_o/2$	△1 SSPNØP	–	SEMI-SPAN OUTBOARD PANEL	ℓ
●	●	●	$b^*/2$	SSPNE	–	SEMI-SPAN EXPOSED PANEL	ℓ
●	●	●	$b/2$	SSPN	–	SEMI-SPAN THEORETICAL PANEL FROM THEORETICAL ROOT CHORD	ℓ
●	●	●	c_b	△1 CHRDBP	–	CHORD AT BREAKPOINT	ℓ
●	●	●	c_r	CHRDR	–	ROOT CHORD	ℓ
●	●	●	$(\Lambda_{x/c})_i$	SAVSI	–	INBOARD PANEL SWEEP ANGLE	DEG
●	●	●	$(\Lambda_{x/c})_o$	△1 SAVSØ	–	OUTBOARD PANEL SWEEP ANGLE	DEG
●	●	●	x/c	CHSTAT	–	REFERENCE CHORD STATION FOR INBOARD AND OUTBOARD PANEL SWEEP ANGLES, FRACTION OF CHORD	–
●	●		Θ	TWISTA	–	TWIST ANGLE, NEGATIVE LEADING EDGE ROTATED DOWN (FROM EXPOSED ROOT TO TIP)	DEG
●	●		$(b/2)_{\Gamma_o}$	△1 SSPNDD	–	SEMI-SPAN OF OUTBOARD PANEL WITH DIHEDRAL	ℓ
●	●		Γ_i	DHDADI	–	DIHEDRAL ANGLE OF INBOARD PANEL (IF $\Gamma_i = \Gamma_o$ ONLY INPUT Γ_i)	DEG
●	●		Γ_o	DHDADØ	–	DIHEDRAL ANGLE OF OUTBOARD PANEL	DEG
●	●	●		TYPE	–	= 1.0 STRAIGHT TAPERED PLANFORM = 2.0 DOUBLE DELTA PLANFORM (ASPECT RATIO ≤ 3) = 3.0 CRANKED PLANFORM (ASPECT RATIO > 3)	–

WGSCHR - HTSCHR - VTSCHR

Happy NACA ☺

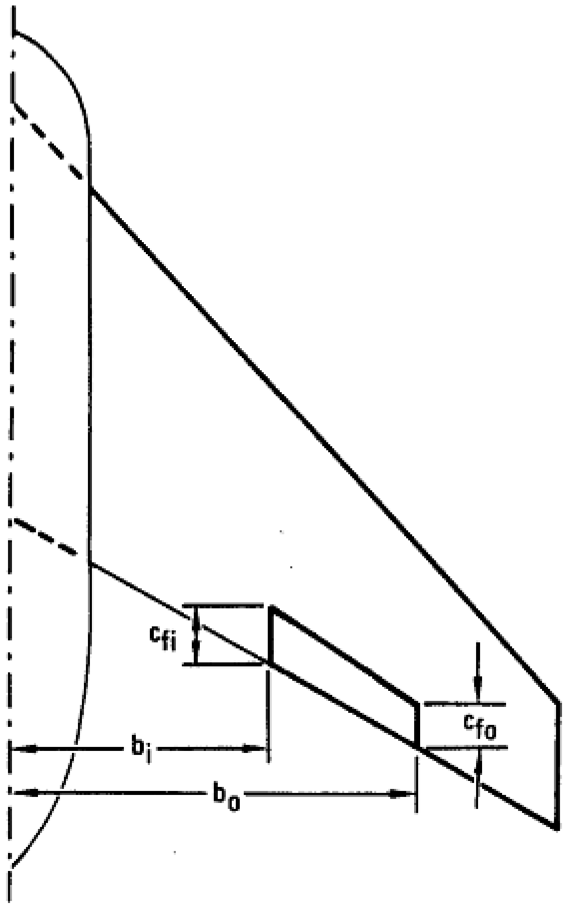
```
NACA-W-6-65-210
NACA-V-4-0012
NACA-H-4-0012
```

Custom Airfoil ☹

```
$WGSCHR TYPEIN=1.0, NPTS=49.0, DWASH=1.0,
 XCORD=0.00000, 0.00102, 0.00422, 0.00960, 0.01702, 0.02650,
     0.03802, 0.05158, 0.06694, 0.08422, 0.10330, 0.12403,
     0.14643, 0.17037, 0.19558, 0.22221, 0.24998, 0.27891,
     0.30861, 0.33933, 0.37056, 0.40243, 0.43469, 0.46733,
     0.49997, 0.53274, 0.56525, 0.59750, 0.62938, 0.66074,
                  0.69133, 0.72115, 0.74995, 0.77773, 0.80435, 0.82970,
     0.85350, 0.87590, 0.89644, 0.91571, 0.93299, 0.94848,
     0.96192, 0.97344, 0.98291, 0.99034, 0.99571, 0.99891,
     1.00000,
 YUPPER=0.00000, 0.00850, 0.01791, 0.02600, 0.03369, 0.04047,
     0.05044, 0.05781, 0.06739, 0.07483, 0.08384, 0.09093,
     0.09914, 0.10557, 0.11262, 0.11794, 0.12347, 0.12732,
     0.13099, 0.13306, 0.13444, 0.13385, 0.13176, 0.12776,
     0.12294, 0.11695, 0.11046, 0.10334, 0.09602, 0.08828,
     0.08058, 0.07274, 0.06512, 0.05754, 0.05032, 0.04338,
     0.03695, 0.03094, 0.02551, 0.02060, 0.01627, 0.01244,
     0.00918, 0.00643, 0.00415, 0.00235, 0.00107, 0.00026,
     0.00000,
 YLOWER=0.00000, -0.00350, -0.00800, -0.01293, -0.01603,
     -0.02164, -0.02484, -0.02993, -0.03315, -0.03790,
     -0.04104, -0.04540, -0.04826, -0.05208, -0.05447,
     -0.05747, -0.05906, -0.06103, -0.06162, -0.06225,
     -0.06143, -0.06026, -0.05699, -0.05322, -0.04826,
     -0.04337, -0.03793, -0.03278, -0.02744, -0.02259,
     -0.01783, -0.01362, -0.00970, -0.00633, -0.00333,
     -0.00093, 0.00111, 0.00258, 0.00363, 0.00425,
      0.00450, 0.00438, 0.00395, 0.00324, 0.00236,
      0.00147, 0.00070, 0.00019, 0.00000$
```

These data can be easily obtained
Using airfoil databases.

Asymmetric Flap (Modeling Aileron)



Variable	Definition
STYPE	=4.0 Plain Flap Aileron
SPANFI	Inboard Span
SPANFO	Outboard Span
NDELTA	Number of deflection
DELTAL	Right Aileron Deflection
DELTAR	Left Aileron Deflection
CHRDFI	Inboard Chord
CHRDFO	Outboard Chord

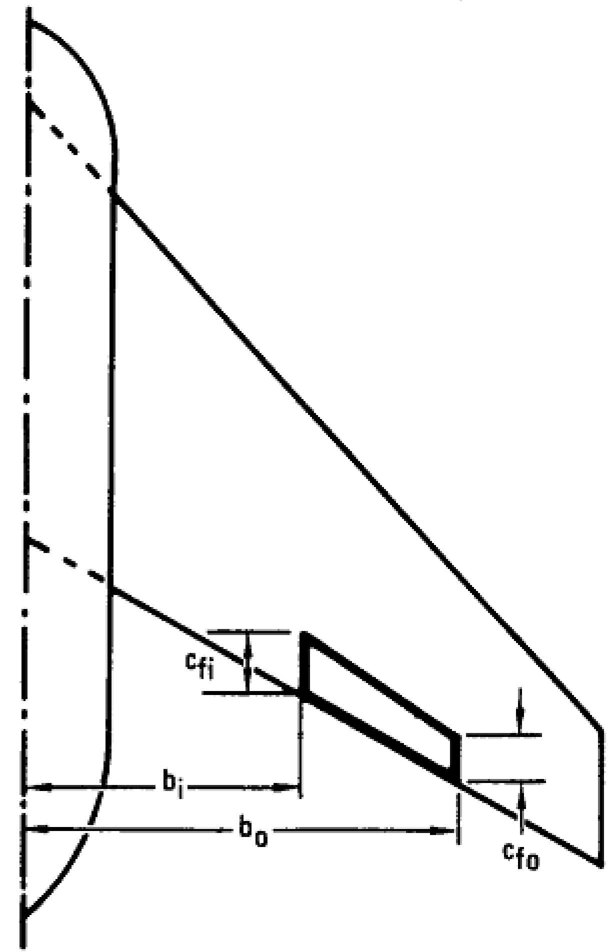
Symmetric Flap (Modeling Flap And Elevator)

You Can’t model both flap and elevator at the same time

Wing-Body-Htail -> SYMFLP= Elevator

Wing-Body -> SYMFLP= Wing Flap

FTYPE	–	= 1.0 PLAIN FLAPS = 2.0 SINGLE SLOTTED FLAPS = 3.0 FOWLER FLAPS = 4.0 DOUBLE SLOTTED FLAPS = 5.0 SPLIT FLAPS = 6.0 LEADING EDGE FLAP = 7.0 LEADING EDGE SLATS = 8.0 KRUEGER

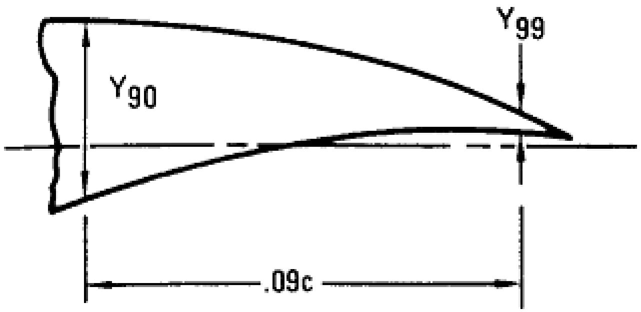


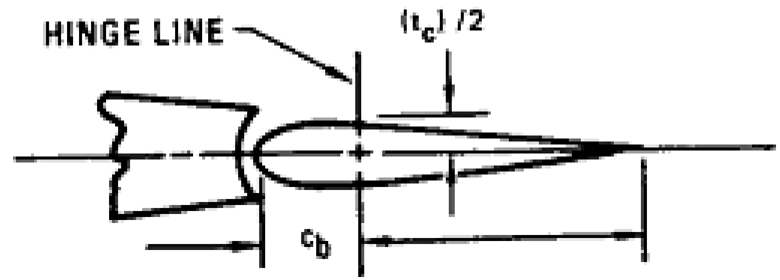
Symmetric Flap (Modeling Flap And Elevator)

ENGR SYMBOL	VARIABLE NAME	ARRAY DIMENSION	DEFINITION	UNITS	PLAIN FLAPS	SINGLE SLOTTED FLAPS	FOWLER FLAPS	DOUBLE SLOTTED FLAPS	SPLIT FLAPS	LEADING EDGE FLAP	LEADING EDGE SLATS	KRUEGER FLAP	JET FLAPS
	FTYPE	–	= 1.0 PLAIN FLAPS	–	●								●
			= 2.0 SINGLE SLOTTED FLAPS			●							● △2
			= 3.0 FOWLER FLAPS				●						●
			= 4.0 DOUBLE SLOTTED FLAPS					●					●
			= 5.0 SPLIT FLAPS						●				
			= 6.0 LEADING EDGE FLAP							●			
			= 7.0 LEADING EDGE SLATS								●		
			= 8.0 KRUEGER									●	
	NDELTA	–	NUMBER OF FLAP OR SLAT DEFLECTION ANGLES, MAX 9	–	●	●	●	●	●	●	●	●	●
δ_f	DELTA	9	FLAP DEFLECTION ANGLE MEASURED STEAMWISE	DEG	●	●	●	●	●	●	●	●	●
$\tan(\phi_{TE}/2)$	PHETE	–	TANGENT OF AIRFOIL TRAILINE EDGE ANGLE BASED ON ORDINATES AT 90 AND 99 PERCENT CHORD	–	●	●	●	●	●				
$\tan(\phi_{TE}/2)$	PHETEP	–	TANGENT OF AIRFOIL TRAILING EDGE ANGLE BASED ON ORDINATES AT 95 AND 99 PERCENT CHORD	–	●	●	●	●	●				
c_{f_i}	CHRDFI	–	FLAP CHORD AT INBOARD END OF FLAP, MEASURED PARALLEL TO LONGITUDINAL AXIS	ℓ	●	●	●	●	●	●	●	●	●
c_{f_o}	CHRDFØ	–	FLAP CHORD AT OUTBOARD END OF FLAP, MEASURED PARALLEL TO LONGITUDINAL AXIS	ℓ	●	●	●	●	●	●	●	●	●
b_i	SPANFI	–	SPAN LOCATION OF INBOARD END OF FLAP, MEASURED PERPENDICULAR TO VERTICAL PLANE OF SYMMETRY	ℓ	●	●	●	●	●	●	●	●	●
b_o	SPANFØ	–	SPAN LOCATION OF OUTBOARD END OF FLAP, MEASURED PERPENDICULAR TO VERTICAL PLANE OF SYMMETRY	ℓ	●	●	●	●	●	●	●	●	●
c_i'	CPRMEI	9	TOTAL WING CHORD AT INBOARD END OF FLAP (TRANSLATING DEVICES ONLY) MEASURED PARALLEL TO LONGITUDINAL AXIS	ℓ		●	●	●			●	●	●
c_o	CPRMEØ	9	TOTAL WING CHORD AT OUTBOARD END OF FLAP (TRANSLATING DEVICES ONLY) MEASURED PARALLEL TO LONGITUDINAL AXIS	ℓ		●	●	●			●	●	●
c'_{a_I}	CAPINB	9		ℓ				●					
c_{a_O}	CAPØUT	9		ℓ				●					
$(\delta_f)_2$	DØBDEF	9		ℓ				●					
c_{2_I}	DØBCIN	–		ℓ				●					
c_{2_O}	DØBCØT	–		ℓ				●					
△1 Δc_ℓ	SCLD	9	INCREMENT IN SECTION LIFT COEFFICIENT DUE TO DEFLECTING FLAP TO THE ANGLE δ_f										
△1 Δc_{m_f}	SCMD	9	INCREMENT IN SECTION PITCHING MOMENT COEFFICIENT DUE TO DEFLECTING FLAP TO ANGLE δ_f										
c_b	CB	–	AVERAGE CHORD OF THE BALANCE	ℓ	●								
t_c	TC	–	AVERAGE THICKNESS OF THE CONTROL AT HINGE LINE	ℓ	●								
	NTYPE	–	= 1.0 ROUND NOSE FLAP = 2.0 ELLIPTIC NOSE FLAP = 3.0 SHARP NOSE FLAP	–	●								
	JETFLP	–	= 1.0 PURE JET FLAP = 2.0 IBF = 3.0 EBF = 4.0 COMBINATION MECHANICAL AND PURE JET FLAP	–									●
C_μ	CMU	–	TWO-DIMENSIONAL JET EFFLUX COEFFICIENT	–									●
δ_j	DELJET	9	JET DEFLECTION ANGLE	DEG									●
$\delta_{j_{eff}}$	EFFJET	9	EBF EFFECTIVE JET DEFLECTION ANGLE	DEG									●

△1 OPTIONAL FOR ALL FLAP TYPES

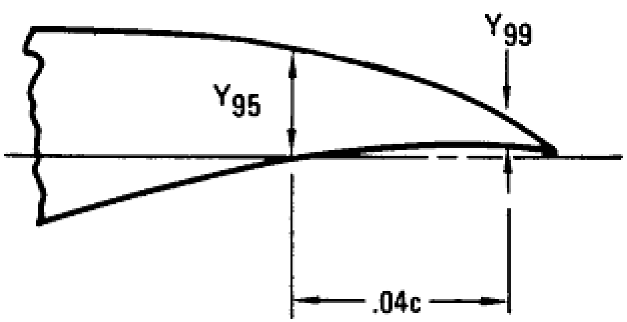
Symmetric Flap (Modeling Flap And Elevator)





CONTROL BALANCE INPUT VARIABLES

$$\tan\left(\phi_{TE}/2\right) = 1/2\left[\frac{Y_{90} - Y_{99}}{9}\right]$$



$$\tan\left(\phi_{TE}/2\right) = 1/2\left[\frac{Y_{95} - Y_{99}}{4}\right]$$

Any Last Words???

Error handling

Output File Structure

Control Derivatives

Other Options…

AVL	http://web.mit.edu/drela/Public/web/avl/
OPENVSP	http://openvsp.org/
Tornado	https://tornado.redhammer.se/
VOGEL	https://www.openvogel.org/