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Loads and VSAERO

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Lecturer

Introduction

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- Complete Aircraft wing, tail (loft)
- V-n Diagrams
- **Wing Design Loads**
- Wing Structural Sizing

Reading:

1. Nicolai, CH 19
2. Niu, Airframe Structural Design Chapter 3
3. Bruhn, Analysis and Design of Flight Vehicle Structures, Chapter A5

4. NACA TN-921, THEORETICAL SYMMETRIC SPAN LOADING AT SUBSONIC SPEEDS FOR WINGS HAVING ARBITRARY PLAN FORM

5. NACA TN 2282, AN IMPROVED APPROXIMATE METHOD FOR CALCULATING LIFT DISTRIBUTION DUE TO TWIST

Exercise

- Complete Wing Loads and plots net shears, bending moments, and torsion vs. wing span
  - Nicolai, CH 19 – very old school, but gets a good answer
  - Bruhn Method – similar to Nicolai
  - Nui Method – more detailed

- Load Types
  - Basic Aero Loading
    - Loading at C_L=0 (load due to wing twist)
  - Additional Aero Loading
    - Due to AoA
    - Section of the wing (2D) $C_{l\alpha}$
    - Used to obtain spanwise lift distribution
  - Engine/Prop Air Loads (if applicable)
  - Inertia Loading
    - Wing has its own weight

a.c. and Basic Air Loading (From Nui)

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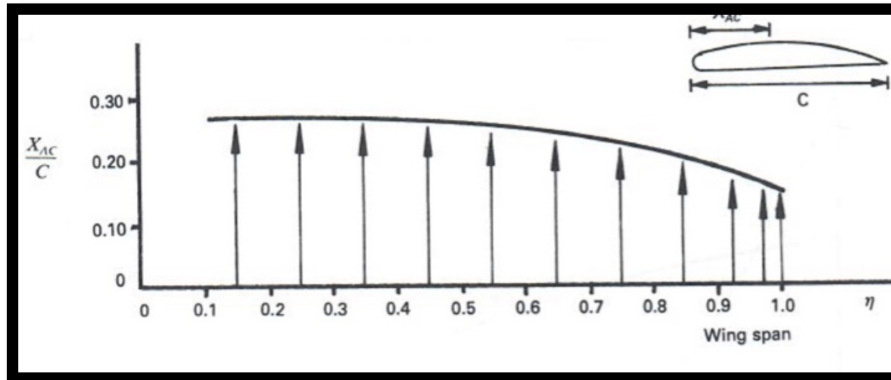
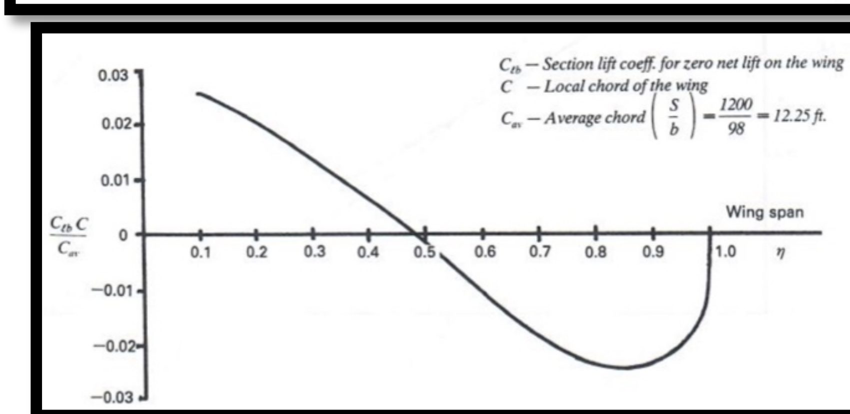
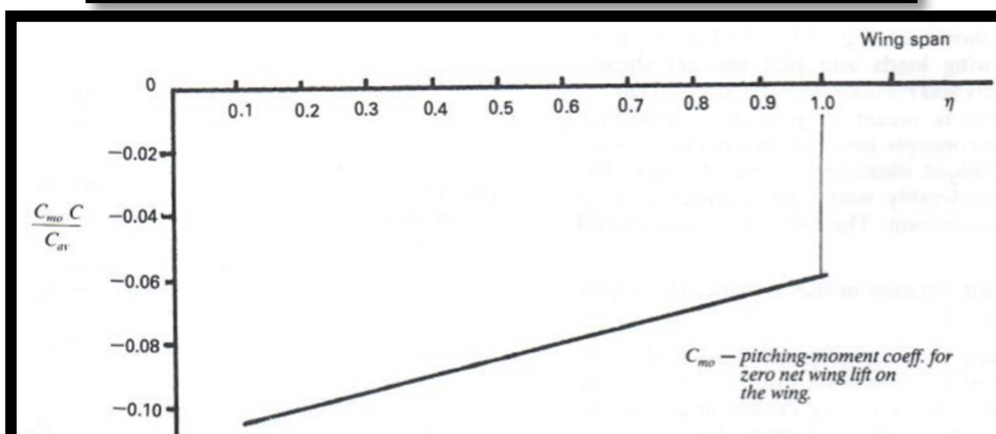
a.c. location vs span



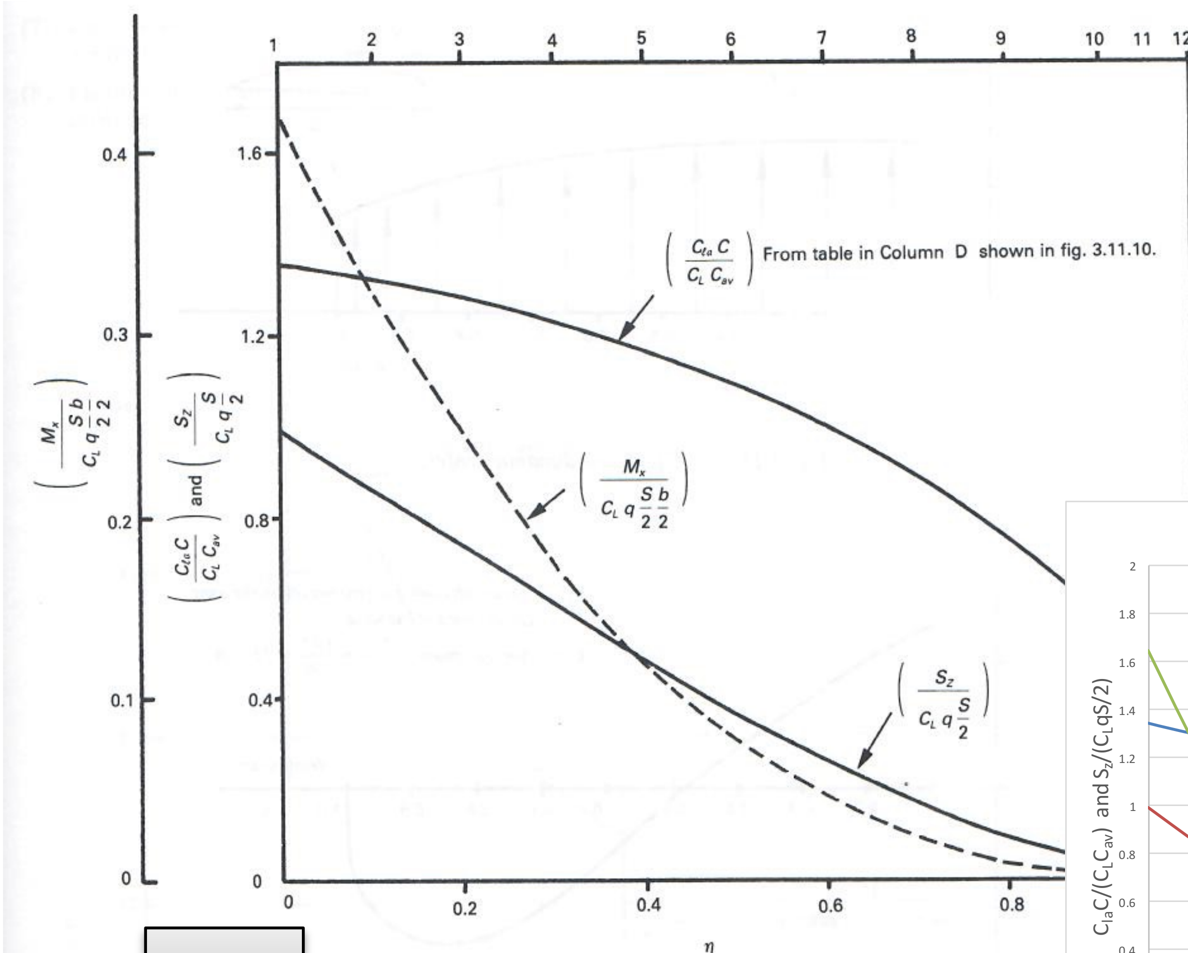
Section Lift Coefficient vs. span for C_L=0



Section Moment Coefficent vs. span for C_L=0

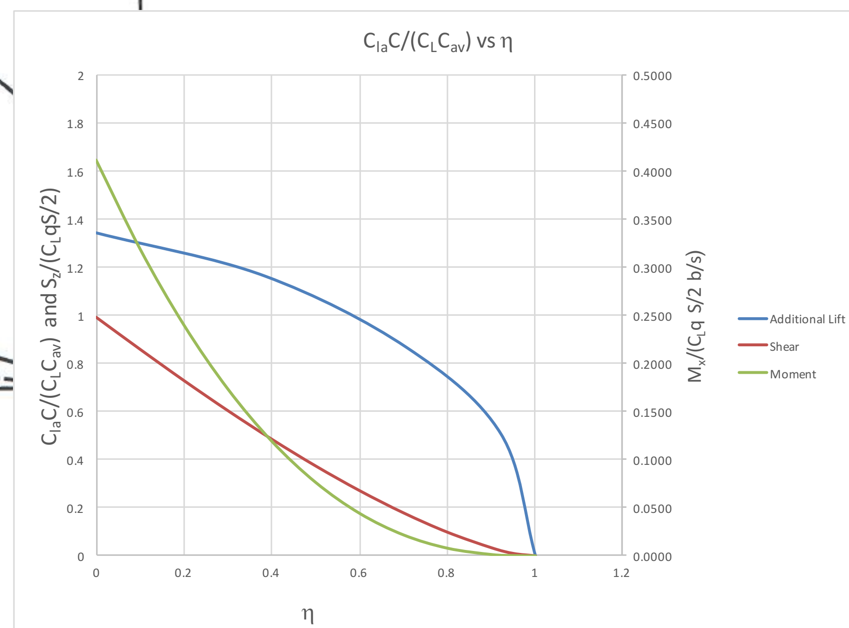


Spanwise Additional Lift Distribution



Nui

Fig. 3.11.12 Spanwise additional lift distribution curves.



Inertia Data

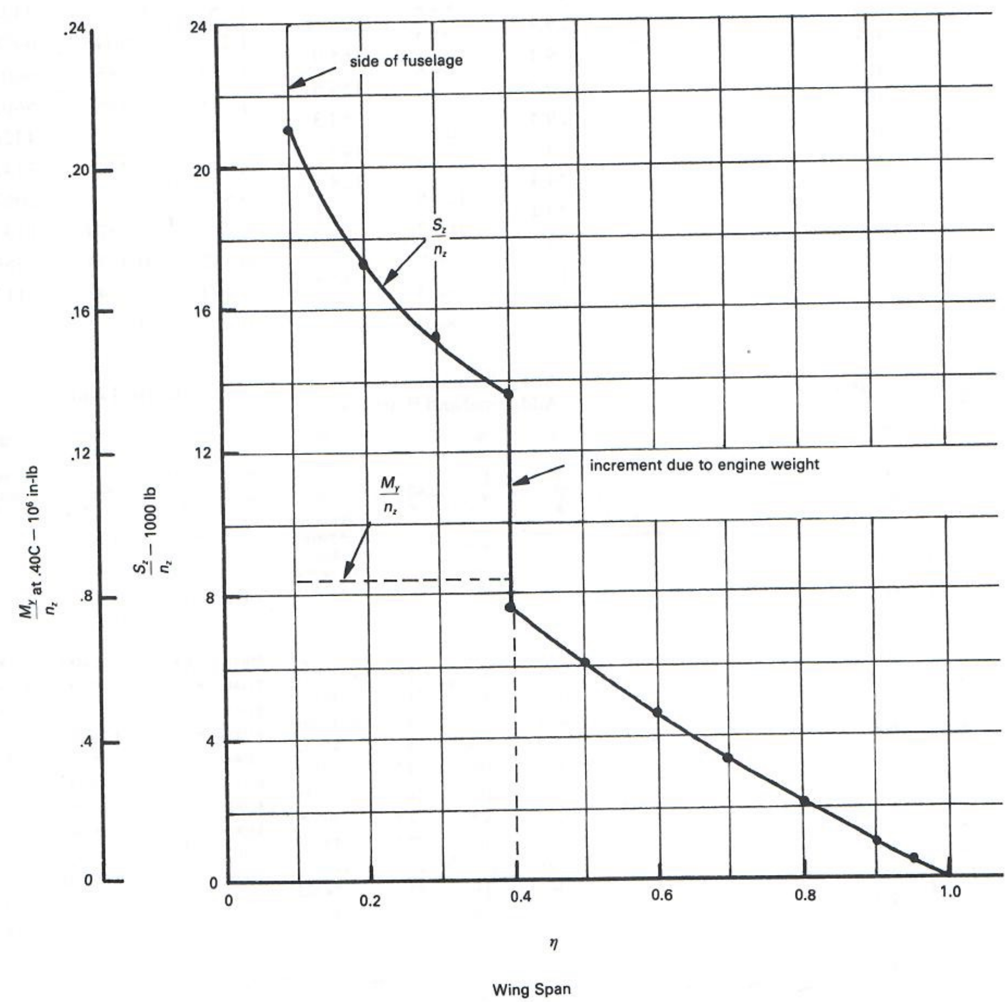


Fig. 3.11.16 Given wing inertia data.

Airframe Structural Design 83

Based on Distributed Weight of the Wing

Based on the wing estimated weight, use chord length at each wing station to distribute the weight

Nui

Nicolai Method

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- Great cookbook method for spar cap sizing
  - CH 19, example 19.2

Bruhn Method (use your aircraft design)

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Use Section Chord to ratio which weight

Use .40c for reference axis

Use VSAERO Data instead

SHOW UNITS!

Plot using spanwise distance at 1.0

TABLE A5.1

CALCULATION OF WING SHEAR V_z AND WING MOMENTS M_x AND M_y DUE TO TOTAL UNIT DISTRIBUTED HALF WING LOAD OF 17760 LBS., ACTING UPWARD IN Z DIRECTION AND APPLIED AT AERODYNAMIC CENTERS OF WING SECTIONS. (See Figs. A5.37, 38 for Wing Layout)

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Station = y = Distance From Wing Root Section	Chord Length C Inches	$C_L/\bar{C}_L$ Ratio, Assumed Unity.	Running Load Per Inch of Wing (W). Lbs.	Average Running Load. W(ave) (lbs.)	Δy = Distance Between Stations	Strip Load $\Delta P_z = \Delta y$ (col. 5)	d = Arm to Centroid of ΔP_z Strip Load	Shear $V_z = \Sigma \Delta P_z$	$V_z \Delta y$ = (Col. 9) (Col. 6)	$\Delta P_z d$ = (Col. 7) (Col. 8)	M_x = Col. 12 previous + Col. 11 + Col. 10.	x = distance from aerodynamic center to Y ref. axis	x_{ave} or strip load distance from Y axis.	$\Delta P_z x_{ave}$ = (Col. 7) (Col. 14)	$M_y = \Sigma \Delta P_z x_{ave}$ = summation of Col. 15.
240	48	1.0	48					0	0	0	0	7.2			0
				48.54	5.0	242.7	2.49						7.28		
235	49.09	1.0	49.09					242.7	0	605	605	7.36		1767	1767
				49.63	5.0	248.1	2.49						7.44		
230	50.18	1.0	50.18					490.8	1213	618	2436	7.53		1846	3613
				51.27	10.00	512.7	4.97						7.69		
220	52.36	1.0	52.36					1003.5	4908	2540	9884	7.85		3940	7553
				54.00	15.00	810.0	7.43						8.09		
205	55.64	1.0	55.64					1813.5	15052	6020	30956	8.34		6540	14093
				57.27	15.00	859.0	7.43						8.59		
190	58.92	1.0	58.92					2672.5	27020	6490	64466	8.85		7390	21483
				60.56	15.00	908.4	7.43						9.09		
175	62.20	1.0	62.20					3580.9	40087	6760	111313	9.33		8260	29743
				63.84	15.00	957.6	7.43						9.57		
160	65.48	1.0	65.48					4538.5	53713	7100	172126	9.81		9170	38913
				67.13	15.00	1007.0	7.43						10.06		
145	68.78	1.0	68.78					5545.5	68077	7480	247683	10.31		10100	49013
				70.39	15.00	1055.5	7.43						10.55		
130	72.00	1.0	72.00					6601.0	83782	7840	338705	10.80		11100	60113
				73.62	15.00	1104.3	7.43						11.04		
115	75.23	1.0	75.23					7705.3	99015	8200	445920	11.28		11480	71593
				76.89	15.00	1153.4	7.43						11.54		
100	78.55	1.0	78.55					88587	115579	8570	570060	11.80		13270	84863
				80.18	15.00	1202.7	7.43						12.04		
85	81.81	1.0	81.81					10061	132880	8940	702949	12.28		14480	99343
				83.46	15.00	1252.2	7.43						12.52		
70	85.10	1.0	85.10					11313	150915	9310	863174	12.76		15600	114943
				86.73	15.00	1300.9	7.43						13.01		
55	88.35	1.0	88.35					12614	169700	9660	1042534	13.27		16930	131873
				89.98	15.00	1349.7	7.42						13.49		
40	91.62	1.0	91.62					13964	189210	10000	1241744	13.71		18200	150073
				93.81	20.00	1876.3	9.92						14.05		
20	96.00	1.0	96.00					15840	279280	18600	1539624	14.40		26350	176423
				96.00	20.00	1920.0	10.00						14.40		
0	96.00	1.0	96.00					17760	316806	19200	1875630	14.40		27648	214071

Sum = 17760 Checks Total Limit Load Assumed on Half Wing.

Nui Method (use your aircraft design)

①	②	③	④	⑤	⑥	⑦	Additional Airload ⑧	⑨	⑩
n	η	$d\eta = \frac{\Delta \eta}{2}$	y (inches)	$\frac{\Delta y}{2}$	c (inches)	c Mid. strip (inches)	$\frac{C_{\ell a} C}{C_L C_{av}}$	$\frac{S_z}{C_L q \frac{S}{2}}$	S_z Additional airload
		$\frac{②_{n+1} - ②_n}{2}$	$\left[\frac{b}{2}\right][\eta]$ 588 × ②	$\frac{④_{n+1} - ④_n}{2}$	$C_R - (C_R - C_T)\eta$ 210 − 126 × ②	$\frac{C_n + C_{n-1}}{2}$ $\frac{⑥_n + ⑥_{n+1}}{2}$	From Fig. 3.11.11	$\int \frac{C_{\ell a} C}{C_L C_{av}} d\eta$ ↑∫ ⑧ dη	$\frac{L_w}{2}$ ⑨ 115500 ⑨
2	0.1	0.05	58.8	29.4	197.4	191.1	1.325	0.86	99330
3	0.2	0.05	117.6	29.4	184.8	178.5	1.28	0.73	84320
4	0.3	0.05	176.4	29.4	172.2	165.9	1.223	0.604	69760
5	0.4^-	0	235.2	0	159.6	159.6	1.155	0.485	56020
5	0.4^+	0.05	235.2	29.4	159.6	153.3	1.155	0.485	56020
6	0.5	0.05	294.0	29.4	147.0	140.7	1.072	0.374	43200
7	0.6	0.05	352.8	29.4	134.4	128.1	0.98	0.272	31420
8	0.7	0.05	411.6	29.4	121.8	115.5	0.877	0.179	20670
9	0.8	0.05	470.4	29.4	109.2	102.9	0.738	0.0979	11310
10	0.9	0.025	529.2	14.7	96.6	93.45	0.552	0.0334	3860
11	0.95	0.025	558.6	14.7	90.3	87.15	0.392	0.0098	1130
12	1.0		588.0		84.0		0	0	0

①	Basic Airload ⑪	⑫	⑬	⑭	Airload-Torsion due to Additional and Basic Loads ⑮	⑯	⑰	⑱	Inertia Loads ⑲	⑳	㉑	㉒
n	$\frac{C_{lb} C}{C_{av}}$	$\frac{S_z}{q \frac{S}{2}}$	S_z Basic airload	S_z additional + basic load ⑩+⑬	S_z ⑭$_n$ −⑭$_{n+1}$	$\frac{X_{AC}}{C}$	ΔX (0.40−⑯) × ⑦	M_y additional + basic load ↑Σ ⑮ × ⑰	$\frac{S_z}{n_z}$	$\frac{M_y}{n_z}$	S_z Inertia −2.5 ⑲	M_y Inertia −2.5 ⑳
	Fig. 3.11.14	$\int \frac{C_{lb} C}{C_L C_{av}} d\eta$ ↑∫ ⑪ dη	$q \frac{S}{2}$ ⑫ 334000 ⑫			From Fig. 3.11.13			From Fig. 3.11.16			
2	0.025	−0.00325	−1086	98244	15771	.261	26.5	2337639	21000	.84 × 10⁶	−52400	−2.1 × 10⁶
3	0.0205	−0.00553	−1847	82473	14804	.260	25	1919707	17100	.84 × 10⁶	−42700	−2.1 × 10⁶
4	0.014	−0.00626	−2091	67669	14074	.257	23.7	1549607	15100	.84 × 10⁶	−37700	−2.1 × 10⁶
5	0.006	−0.00726	−2425	53595	200	.255	23.2	1216053	13500	.84 × 10⁶	−33700	−2.1 × 10⁶
5	0.006	−0.00786	−2625	53395	12887	.248	23.3	1211413	7500	0	−18750	0
6	−0.002	−0.00806	−2692	40508	11580	.240	22.5	911146	6000	0	−15000	0
7	−0.1	−0.00746	−2492	28928	10289	.225	22.5	650596	4700	0	−11750	0
8	−0.0175	−0.00608	−2031	18639	8675	.205	22.5	419093	3400	0	− 8480	0
9	−0.0235	−0.00403	−1346	9964	6648	.182	22.5	223905	2100	0	− 5240	0
10	−0.024	−0.00163	− 544	3316	2363	.160	22.5	74325	1000	0	− 2500	0
11	−0.021	−0.00053	− 177	953	953	.145	22.2	21157	500	0	− 1250	0
12	0	0	0	0					0	0	0	0

Note: $\frac{b}{2}$ = 49 ft or 588 in
C_R = 17.5 ft or 210 in
C_T = 7.0 ft or 84 in
$C_R - C_T = 210 - 84 = 126$ in
$C_{av} = \left(\frac{C_R + C_T}{2}\right) = 147$ in

Δ Propellor load (including power plant weight), $\frac{0.05\,(220000)}{2} = 5500$ lb/side
and assume the above load is located 200 in forward of 0.40C
Where $M_y = 5500 \times 200 = 110000$ in-lb

Fig. 3.11.17 Tabulation of wing loads.

Nui

Propellor air loads Δ ㉓	㉔	Torsion load due to C_{mo} ㉕	㉖	㉗	Net wing loads plotted vs. wing span (η) ㉘	㉙	㉚
S_z Propeller air load	M_y Propeller air load	$\frac{C_{mo} C}{C_{av}}$ From Fig. 3.11.15	$\frac{d M_y}{q \frac{S}{2}}$ Δ ↑∫ ㉕ ⑥ dη	M_y (10⁶) due to C_{mo} 334000 × ㉖ (in-lb)	S_z (net) ⑭+㉑+㉓ (lb)	M_x (10⁶) (net) ↑∫ ㉘ dy (in-lb)	M_y (10⁶) (net) ⑱+㉒+㉔+㉗ (in-lb)
5500	1100000	−.105	−10.847	−3.623	51324	11.434	−2.286
5500	1100000	−.10	− 8.886	−2.986	45273	8.594	−2.05
5500	1100000	−.095	− 7.144	−2.386	35469	6.219	−1.836
5500	1100000	−.089	0	0	25395	4.43	
0	0	−.089	− 5.616	−1.876	34645	4.43	−0.665
0	0	−.084	− 4.288	−1.432	25508	2.662	−0.521
0	0	−.08	− 3.133	−1.046	17178	1.407	−0.386
0	0	−.076	− 2.132	−0.712	10159	0.604	−0.293
0	0	−.072	− 1.282	−0.428	4724	0.166	−0.204
0	0	−.066	− 0.57	−0.19	816	0.0033	−0.116
0	0	−.063	− 0.268	−0.09	− 297	−0.004	−0.069
0	0	−.06			0	0	

Δ $d M_y = C_{mo}\, C\, dy\, C\, q$
$= \left(\frac{C_{mo} C}{C_{av}}\right)\left(\frac{dy}{\frac{b}{2}}\right)(C)\left(\frac{S}{2}\right)(q)$
$= \left(\frac{C_{mo} C}{C_{av}}\right)(C)\left(d\eta\right)\left(\frac{S}{2}\right)(q)$

therefore $\frac{d M_y}{q \frac{S}{2}} = \left(\frac{C_{mo} C}{C_{av}}\right)(C)\,(d\eta)$

(Fig. 3.11.17 continued)

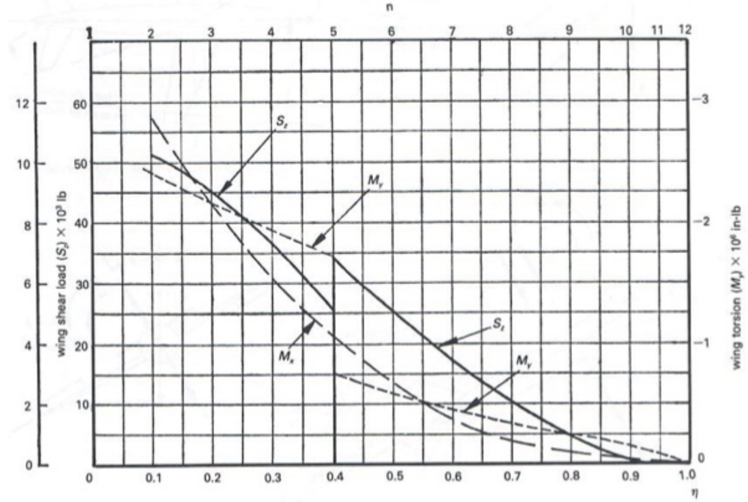
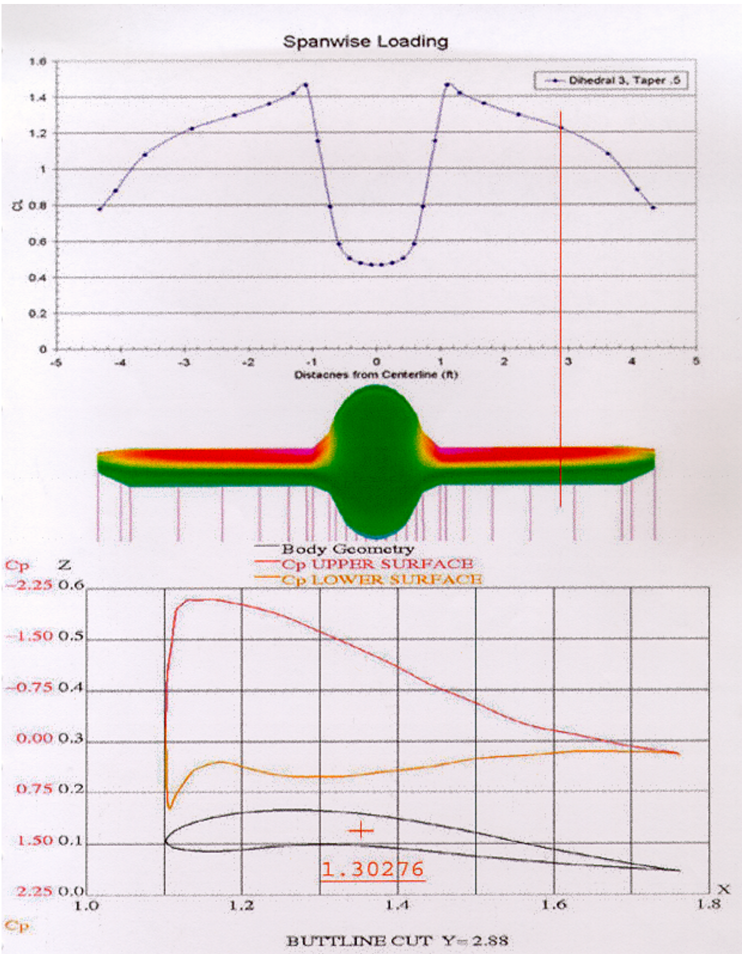
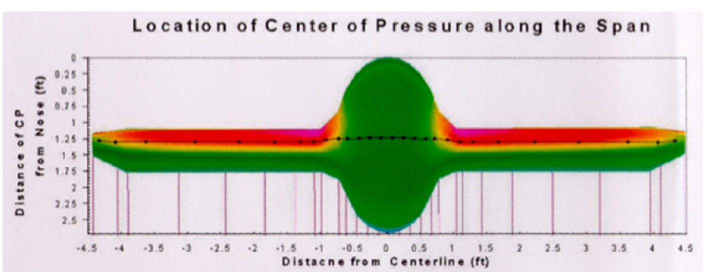
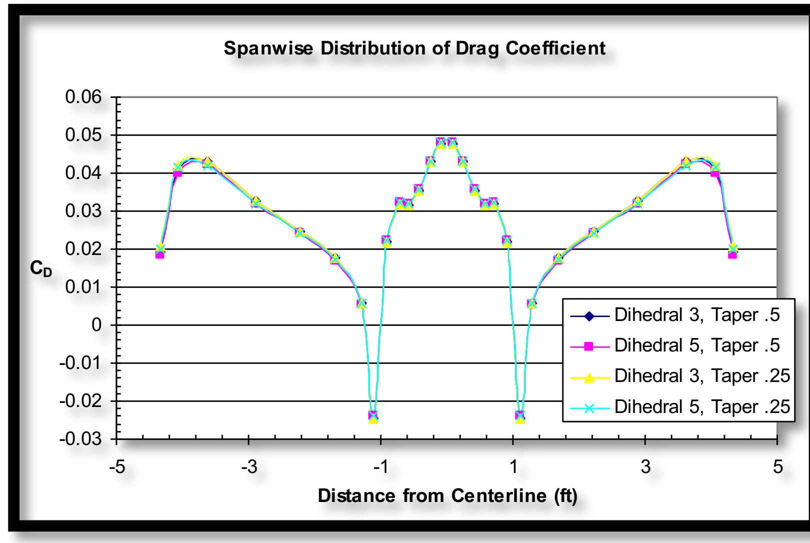


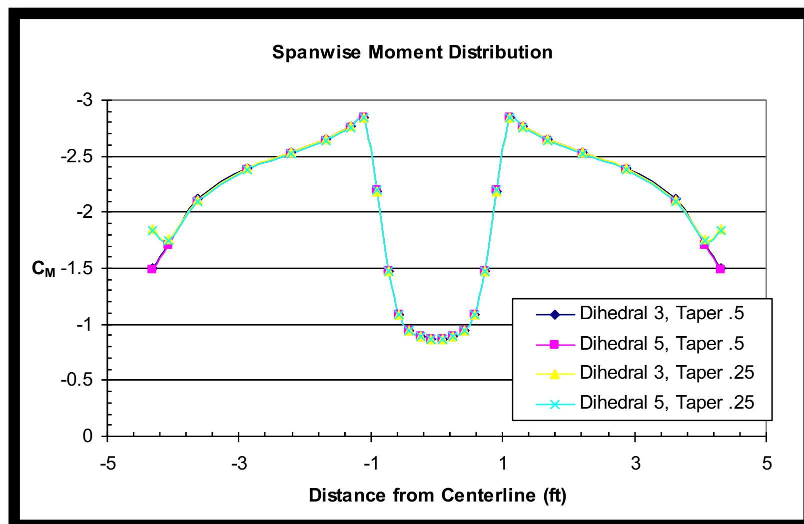
Fig. 3.11.18 Wing spanwise load distribution curves (data from Fig. 3.11.17).

VSAERO examples – Eye Candy

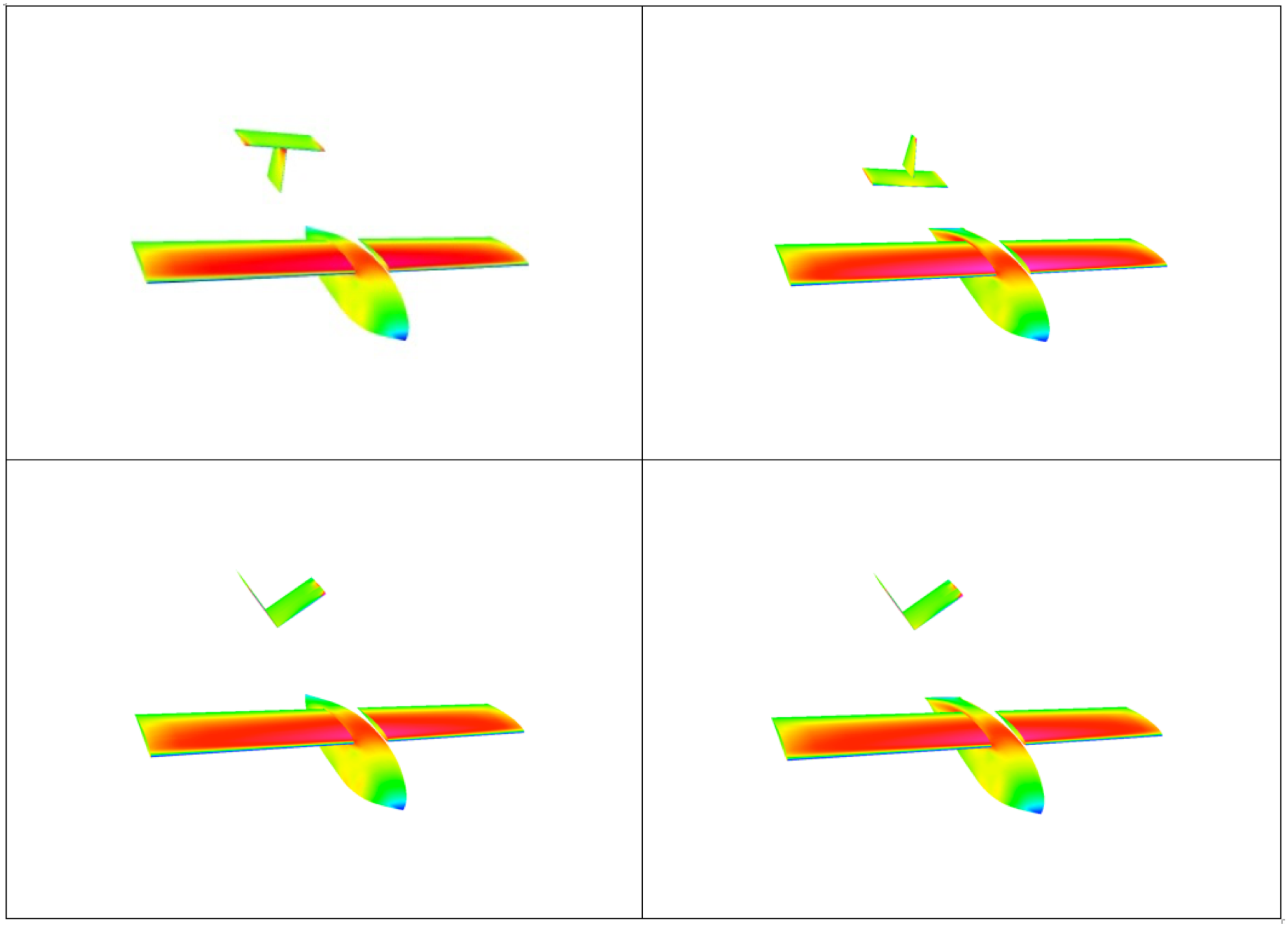








More Eye Candy



Report tasking

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- Develop V-n diagrams
- Develop wing loads
- Size spar caps
- Develop design for wing/fuselage interface
  - Calculate and present loads you are reacting
  - Show load points/reaction loads

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Backup

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