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A STANDARD CLASS SAILPLANE DESIGN

By

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CONTENTS

SUMMARY

The design of a standard class sailplane is outlined here as an iterative procedure. Since the cruising performance of a sailplane does not include powered flight, the overall design was accomplished independent of any powerplant requirements. A preliminary parametric design with a powerplant was accomplished in retrospect, where the design focused only on an engine-propeller combination in the climb configuration. The powerplant was not included in the working drawings for weight and balance. This design emphasizes performance in the cruise configuration between thermal convective air currents. The rising air within these "thermals" provides the powerplant design requirements. Hence considerable attention was focused on approximating lift within a rising thermal air mass. This design only provides an outline for a parametric design procedure. Subsequent iterations will refine this preliminary effort that is shown here as a first approximation.

SYMBOLS*

A	Aspect Ratio (Λ)
b	Wing span
c	Wing chord
$\bar{c}$	Mean aerodynamic chord (MAC)
C_c	Drag coefficient parallel to chord line corrected for Λ
C_D	Drag coefficent parallel to relative wind at $\Lambda = \infty$
C_{D_T}	Total drag coefficient
C_L	Lift coefficient parallel to relative wind
$C_{L_{avg}}$	Lift coefficient at average cruise velocity
C_{L_g}	Lift coefficient in ground effect
C_m	Moment coefficient
C_N	Lift coefficient normal to chord line corrected for Λ
c_R	Wing root chord
c_T	Wing tip chord
C_{HT}	Horizontal tail volume coefficient
C_{VT}	Vertical tail volume coefficient
d_{xxx}	Horizontal distance to centroid of xxx
D	Drag force
e	Span efficiency factor
e_{xxx}	Vertical distance to centroid of xxx
K	Reciprocal of e
l_{HT}	Horizontal tail moment arm from tail to wing center of pressure (c.p.)
l_{VT}	Vertical tail moment arm frow wing c.g. to tail c.p.
L	Lift

* symbols for powerplant calculations not included

n	Load factor
r	Turn radius
R	Thermal radius
R_e	Reynolds number
S	Wing area
S_{HT}	Horizontal tail area
S_{VT}	Vertical tail area
V	True airspeed
V_{avg}	Average cruise velocity including thermals
V_{DD}	Average down draft velocity between thermals
V_L	Landing speed
V_u	Relative climb velocity within a thermal
V_z, V_s	Vertical sinking speed
W_{xxx}	Weight of xxx
α	Angle of attack
μ	Landing coefficient of friction
ρ	Density
ν	ρ / μ
λ	Wing taper ratio
Λ	Wing aspect ratio (A)
Θ	Best glide angle
$\mathcal{K}$	Gust reduction factor

DESIGN REQUIREMENTS

- Standard Class (49.2 Foot Span)
- Single Seat (250 lb maximum passenger and parachute)
- Short Field Landing (Spoilers Top side only)
- Maneuverability (All moving "T" tail)
- Competition oriented, $V_{avg} > V_{@\,L/D\,max}$

DESIGN METHODOLOGY

The total drag of a sailplane is composed of three parts, Ref. 1.

$$C_{D_T} = C_{D_{parasite}} + C_{D_{airfoil}} + C_{D_{induced}} \qquad (1)$$

Choosing Airfoil

The induced drag is larger at low speeds, parasite drag is greater at high speeds, and the airfoil drag stays relatively constant if an airfoil with a wide drag bucket is choosen. The bucket with the lowest C_{D_0} was found in the FX-67-K-150, Ref. 2, airfoil, Figure 1. This particular airfoil section was choosen because it was optimized by Dr. F.X. Wortman for Flaps which the author will later include in an updated design. This

$$C_{D_{0_{wing}}} = 0.0055$$

particular airfoil has a thickness to chord ratio of 17%.

Figure 1

WING COEFFICIENTS

$C_{L(\alpha)}$ $C_{m(\alpha)}$ $C_{L(C_D)}$

FX 67-K-170

Ref. 2

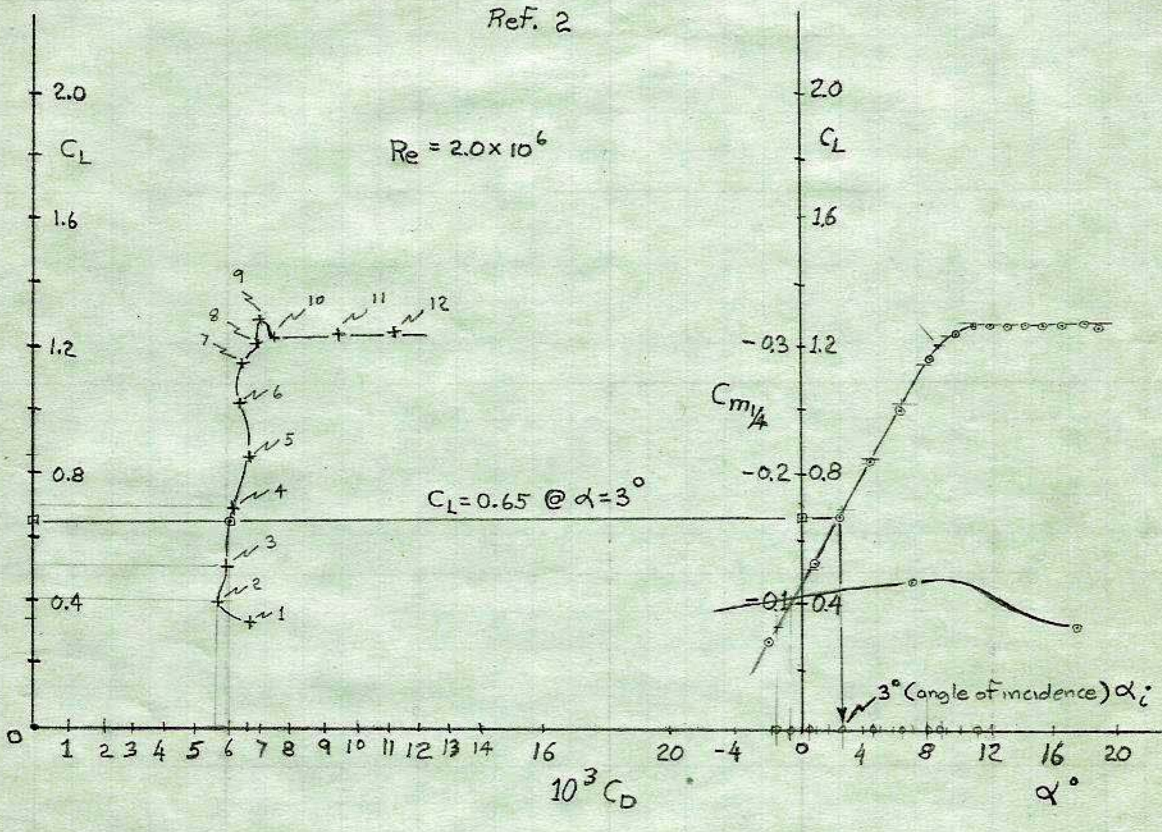


Table I. Calculation of airfoil parameters corrected for aspect ratio (Reference 3).

Aspect ratios
$\Lambda_0 = \infty$
$\Lambda_1 = 22$

		Data points taken from Figure 1, Reference 2											
		1	2	3	4	5	6	7	8	9	10	11	12
1	C_L	0.351	0.400	0.510	0.700	0.850	0.910	1.120	1.210	1.290	1.220	1.230	1.240
2	α_{Λ_0}	-1.8	-1.0	0.07	3.0	4.5	6.4	8.0	8.7	11.0	12.0	13.0	14.0
3	$\frac{57.3}{\pi\Lambda_1} C_L$ *	0.2913	0.332	0.423	0.581	0.706	0.755	0.930	1.004	1.071	1.013	1.021	1.029
4	2+3 α_{Λ_1}	-1.509	-0.668	0.493	3.581	5.206	7.155	8.930	9.704	12.071			
5	$C_{D_{R_0}}$	0.0065	0.0055	0.0060	0.0061	0.0065	0.0062	0.0062	0.0069	0.0070	0.0073	0.0095	0.0110
6	$C_L^2/\pi\Lambda_1$ *	0.00178											
7	5+6 $C_{D\Lambda_1}$	0.00828											
8	$\cos \alpha_{\Lambda_1}$	0.999653											
9	$\sin \alpha_{\Lambda_1}$	-0.02683											
10	$C_L \cos \alpha_{\Lambda_1}$	0.35088											
11	$C_{D\Lambda_1} \sin \alpha_{\Lambda_1}$	-0.000218											
12	10+11 C_N	0.35066	0.3999	0.51006	0.69945	0.84803	0.90517	1.1102	1.1974	1.26797			
13	$C_L \sin \alpha_{\Lambda_1}$	-0.009243											
14	$C_{D\Lambda_1} \cos \alpha_{\Lambda_1}$	0.00827											
15	13+14 C_C	0.01752	0.01247	0.00537	-0.03055	-0.06023	-0.0953	-0.1498	-0.17627	-0.23936			

* equations from reference 3

Choosing Aspect Ratio

For a given wingspan (standard class) the optimum aspect ratio is determined by evaluating the relationship between aspect ratio and C_{D_T}, see figure 2.

Ref. 1

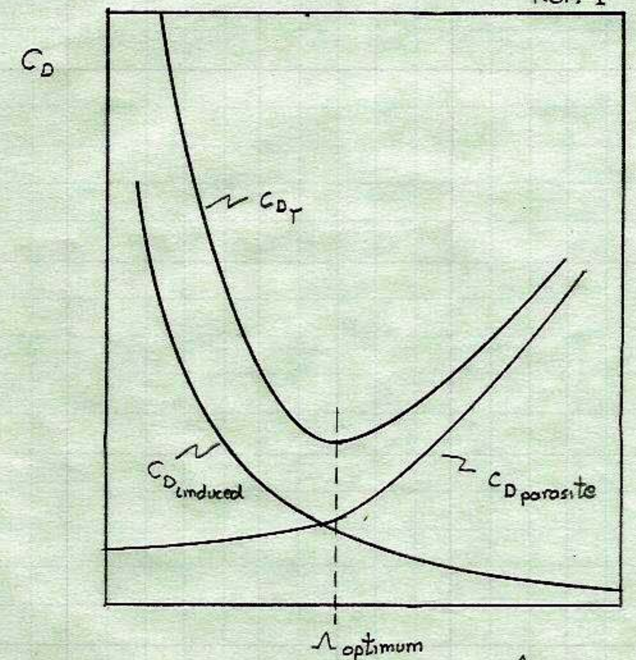


Figure 2

OPTIMUM VALUE OF ASPECT RATIO

For a given span an increase in aspect ratio decreases the wing chord. A decrease in chord causes a lower reynolds number.

$$\downarrow R_e = \frac{V c^{\downarrow}}{\nu} \qquad (2)$$

Lower reynolds numbers produce higher parasite drag coefficients, see figure 3. In this regard also notice that the airfoil parameters shown in figure 1 were choosen at $R_e = 2.0 \times 10^6$.

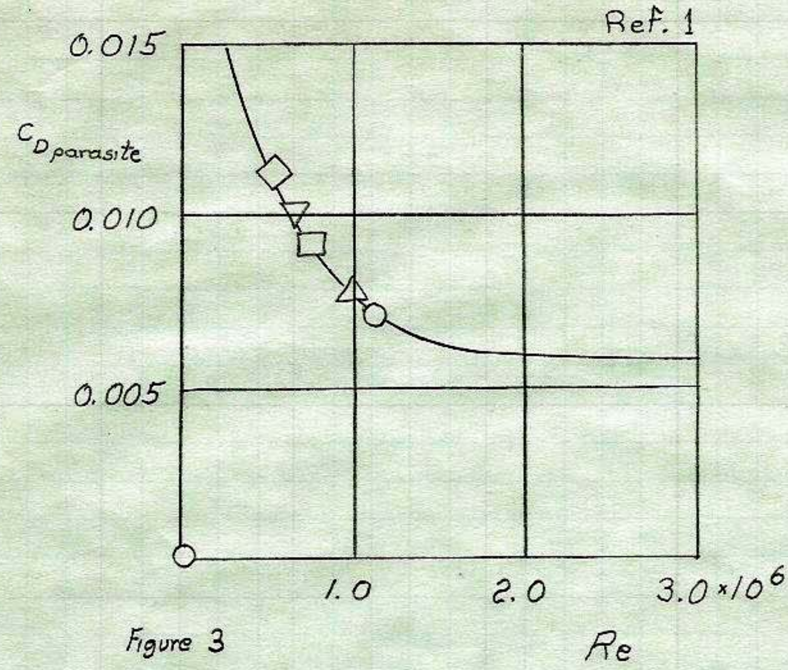
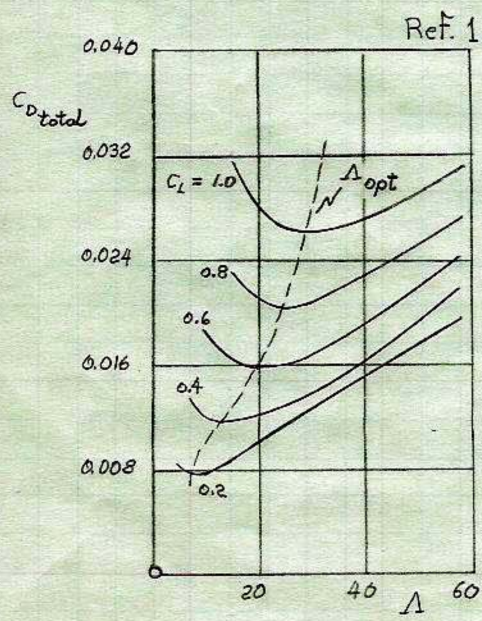


Figure 3

INFLUENCE OF R_e - NUMBER
ON THE AIRFOIL DRAG COEFFICIENT

But higher aspect ratios reduce the induced drag. Therefore there is some optimum aspect ratio which gives the minimum drag, Figure 2. Over a range of lift coefficients a range of optimum aspect ratios can be calculated, Figure 4. The best aspect ratio



OPTIMUM VALUE OF ASPECT RATIO
FOR DIFFERRENT LIFT COEFFICIENTS

Figure 4.

over this range is obtained at cruise condition from figure 5 by arbitrarily choosing $C_{L_{avg}} = 0.7$ as a first approximation.

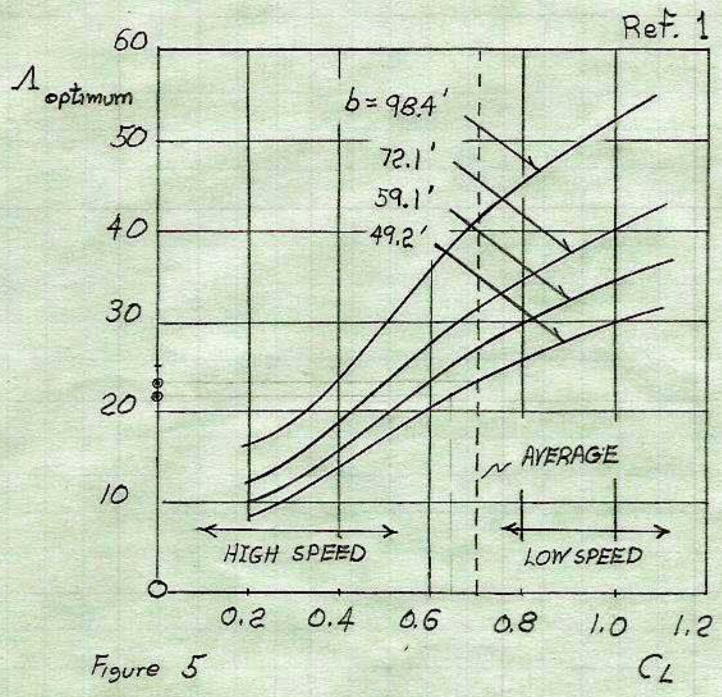


Figure 5

OPTIMUM VALUE OF ASPECT RATIO IN RELATION TO SPAN AND C_L

From figure 5 we obtain $\Lambda = 23$ at optimum cruise condition for a 49.2' span wing. The cruise speed is not a set value for sailplanes but depends on the lift conditions in a thermal. An expression for the average cruise speed, V_{avg}, can be derived from the time required to glide to the next thermal and regain lost altitude,[4] where V is the "speed to fly".

$$V_{avg} = V / (1 + V_s / V_u) \qquad (3)$$

On a strong thermal day V_u is large enough to result in $V_{avg} > (V \text{ at } L/D \text{ max})$. Hence the coefficient of Lift at L/D max should be greater then $C_{L_{avg}}$. From expression given in reference 5 we can calculate C_L at L/D max.

$$e = f(\Lambda) = 1.1 - 0.016\,(\Lambda) \qquad (4)$$

$$C_L @ L/D\,max = \sqrt{C_{D_o} e \pi \Lambda} \qquad (5)$$

$$\Lambda = 23, \quad e = 0.732 \quad C_L @ L/D\,max = 0.790$$

$$\text{recall: } C_{L_{avg}} = 0.7 \text{ and } C_{L_{avg}} < C_{L @ L/D\,max} \qquad (6)$$

Wing Area

Hence, in cruise configuration the choice for $C_{L_{avg}}$ becomes less aribitrary, see eqn. (6). The trend is that higher average cruise velocities require lower $C_{L_{avg}}$. The choice for $C_{L_{avg}}$ still remains somewhat subjective. On this topic the authors opinion is that the potential market for a competitive sailplane requires a larger average cruise speed. Hence we choose $C_{L_{avg}} = 0.65$ and from figure 5 we obtain $\Lambda = 22$. This approximation can be updated after the total design performance has been established. With $\Lambda = 22$ and $b = 49.2$ ft., the wing planform can now be calculated.

$$\Lambda = b^2/S$$

Total wing area required, $S = 110.0\ ft^2$

and from figure 1 at $C_{L_{avg}} = 0.65$

Wing incidence angle, $\alpha_i = 3°$

Wing Geometry

Studies have shown that the optimum shaped wing for stalling characteristics is a rectangular tapered wing with a taper of $\lambda = 0.4$ beginning at the two-thirds semispan with $-3°$ twist included on the tapered portion.[6]

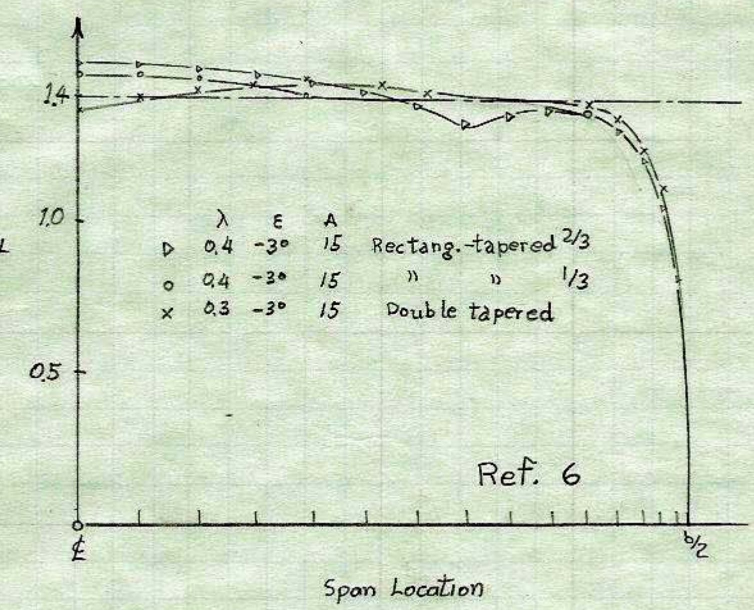


Figure 6

LIFT DISTRIBUTIØN FØR VARIØUS WING GEØMETRIES

Root and Tip Chords

With the wing shape defined and the surface area known the root and tip chords are calculated from two equations for the two unknowns.

$$\frac{2b}{3} C_R + \frac{b}{3} \frac{C_R + C_T}{2} = S \qquad (8)$$

$$\lambda = C_T/C_R = 0.4 \qquad (9)$$

C_R = 36.59 inches

C_T = 14.63 inches

TABLE I Airfoil coordinates for wing root and tip in inches, Ref. 2.

$\frac{x}{c} c_{Root}$	$\left(\frac{y}{c}\right)_{upper} c_{Root}$	$\left(\frac{y}{c}\right)_{lower} c_{Root}$
36.550848	0.009879	0.001829
36.238370	0.088913	0.016099
35.619267	0.230882	0.038419
34.703419	0.439445	0.045371
33.506560	0.728506	0.013538
32.049912	1.11233[illegible]	-0.072082
31.231394	1.349805	-0.141237
30.357625	1.623498	-0.228687
29.432264	1.934513	-0.334066
28.459336	2.279191	-0.452252
27.442500	2.646554	-0.575194
26.386512	3.021968	-0.693746
25.296130	3.389331	-0.800223
24.175744	3.735107	-0.891698
23.033770	4.047951	-0.971098
21.864354	4.320547	-1.040619
20.682863	4.547771	-1.102090
19.491493	4.727062	-1.154414
18.295000	4.856956	-1.197224
17.098507	4.935991	-1.231253
15.907136	4.965628	-1.256866
14.725645	4.949163	-1.273332
13.559888	4.892083	-1.281015
12.414255	4.800242	-1.280284
11.293869	4.677299	-1.271136
10.203487	4.524353	-1.253207
9.147500	4.343233	-1.227228
8.130663	4.136499	-1.193199
7.157735	3.906714	-1.151121
6.232374	3.656804	-1.101724
5.358605	3.389331	-1.045010
4.540087	3.106491	-0.981343
3.780478	2.811941	-0.911091
3.083439	2.508610	-0.834983
2.451164	2.199424	-0.754485
1.886580	1.887312	-0.668499
1.392615	1.576663	-0.578122
0.970732	1.275893	-0.483353
0.623493	1.011713	-0.386756
0.351629	0.736190	-0.298208
0.156605	0.472742	-0.188072
0.039151	0.238932	-0.079400

TABLE II

$\left(\frac{x}{c}\right)_{C_{Tip}}$	$\left(\frac{y}{c}\right)_{upper}^{C_{Tip}}$	$\left(\frac{y}{c}\right)_{lower}^{C_{Tip}}$
14.614345	0.003803	0.001024
14.489405	0.034087	0.008778
14.241866	0.087633	0.020774
13.875677	0.165319	0.028235
13.397129	0.271532	0.022969
12.814709	0.410956	-0.000877
12.487436	0.496542	-0.021652
12.138072	0.594709	-0.048864
11.768079	0.706190	-0.081781
11.379067	0.829521	-0.118941
10.972500	0.961191	-0.158004
10.550278	1.095787	-0.195895
10.114304	1.228188	-0.230422
9.666333	1.353128	-0.260999
9.208268	1.466657	-0.287772
8.742156	1.565702	-0.311619
8.269753	1.648654	-0.332686
7.793401	1.714050	-0.350973
7.315000	1.761598	-0.366335
6.836599	1.790565	-0.378624
6.360246	1.802123	-0.388133
5.887843	1.797295	-0.394717
5.421731	1.777837	-0.398521
4.963666	1.745066	-0.399399
4.515695	1.699859	-0.397497
4.079721	1.642949	-0.392669
3.657500	1.575797	-0.385061
3.250932	1.498989	-0.374820
2.861920	1.413696	-0.361946
2.491927	1.320650	-0.346438
2.142563	1.221312	-0.328589
1.815290	1.116122	-0.308254
1.511571	1.006836	-0.286016
1.232870	0.893893	-0.261584
0.980063	0.779340	-0.235689
0.754322	0.663616	-0.208038
0.556817	0.548917	-0.179071
0.388133	0.437729	-0.148494
0.249295	0.337953	-0.117625
0.140594	0.237883	-0.088804
0.062616	0.144690	-0.054862
0.015654	0.067590	-0.021213

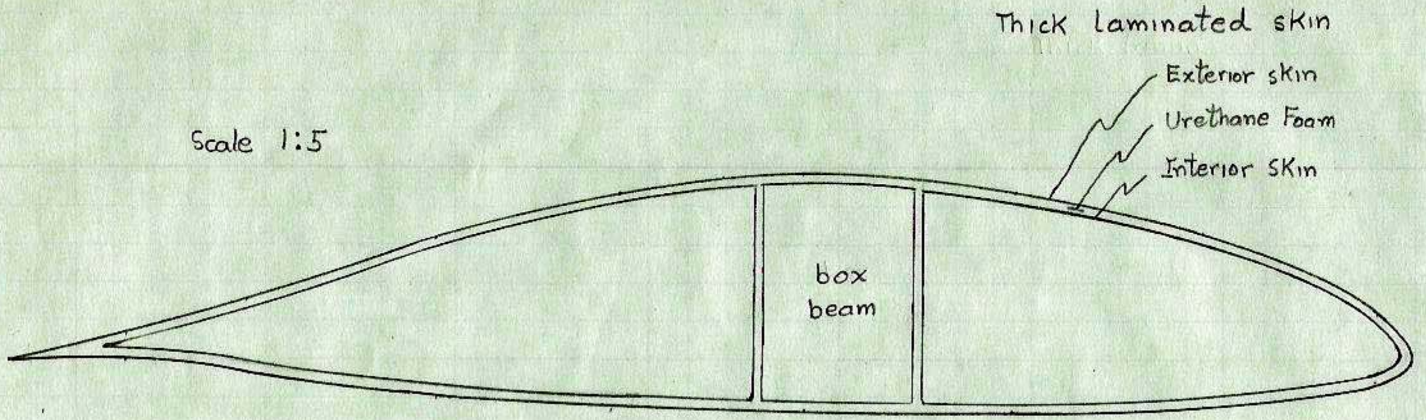


Figure 7

CROSSECTION OF WING CONSTRUCTION

Shear bracing?

Note: There are no ribs from the root to the tip of the wing. The pressure loads on the top and bottom wing surfaces are transfered to the box beam through the thick laminate skin.

With the given geometry the mean aerodynamic center (MAC) can be calculated for the wing shape shown in figure 8.

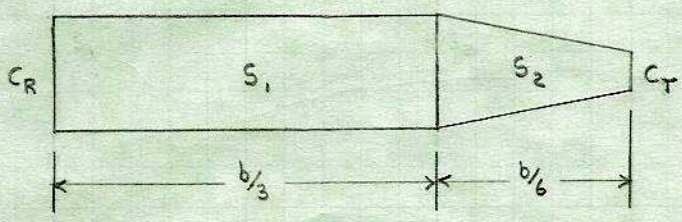


Figure 8
WING SHAPE

$$\bar{C} = \frac{C_R S_1 + \frac{2S_2}{3}\left[C_R + C_T - \frac{C_R C_T}{C_R + C_T}\right]}{S_1 + S_2} \qquad (10)$$

$\bar{C}$ = 34.146 in. (M.A.C.)

Drag Polar

A) Wing
- Surface Planform Area
- C_{D_w} average over drag bucket = 0.0055

B) Fuselage
- Frontal Area = 5.945 FT^2
- C_{D_f} based on Frontal Area = 0.08, Ref. 5

C) Tail
- Surface Planform Area, Ref. 5
- C_{D_t} based on planform area = 0.006

Component	Area FT^2 ×	C_D	= ΔF
Fuselage	5.945	0.08	0.4756
Tail	25.4	0.006	0.1524
Wing	110.0	0.0055	0.6050
			$1.2330 = \sum \Delta F$

Drag build-up calculations assume 5% interference drag

$$f = 1.05 \sum \Delta F = 1.295$$

$$C_{D_0} = \frac{f}{S} = \frac{1.295}{110} = 0.0118$$

recall eqn. (4), $e = 1.1 - 0.016(\Lambda)$

$$\Lambda = 22 \qquad e = 0.748$$

$$C_D = C_{D_0} + \frac{C_L^2}{e \pi \Lambda} \tag{11}$$

$$C_D = 0.0118 + 0.019343\, C_L^2$$

Data plotted
in Figure 9 C_D .vs. C_L
C_D .vs. C_L^2

C_L ⊡	C_L^2 ⊙	C_D
0.2	0.04	0.0126
0.4	0.16	0.0149
0.6	0.36	0.0188
0.8	0.64	0.0242
1.0	1.00	0.0311
1.2	1.44	0.0397
1.4	1.96	0.0497

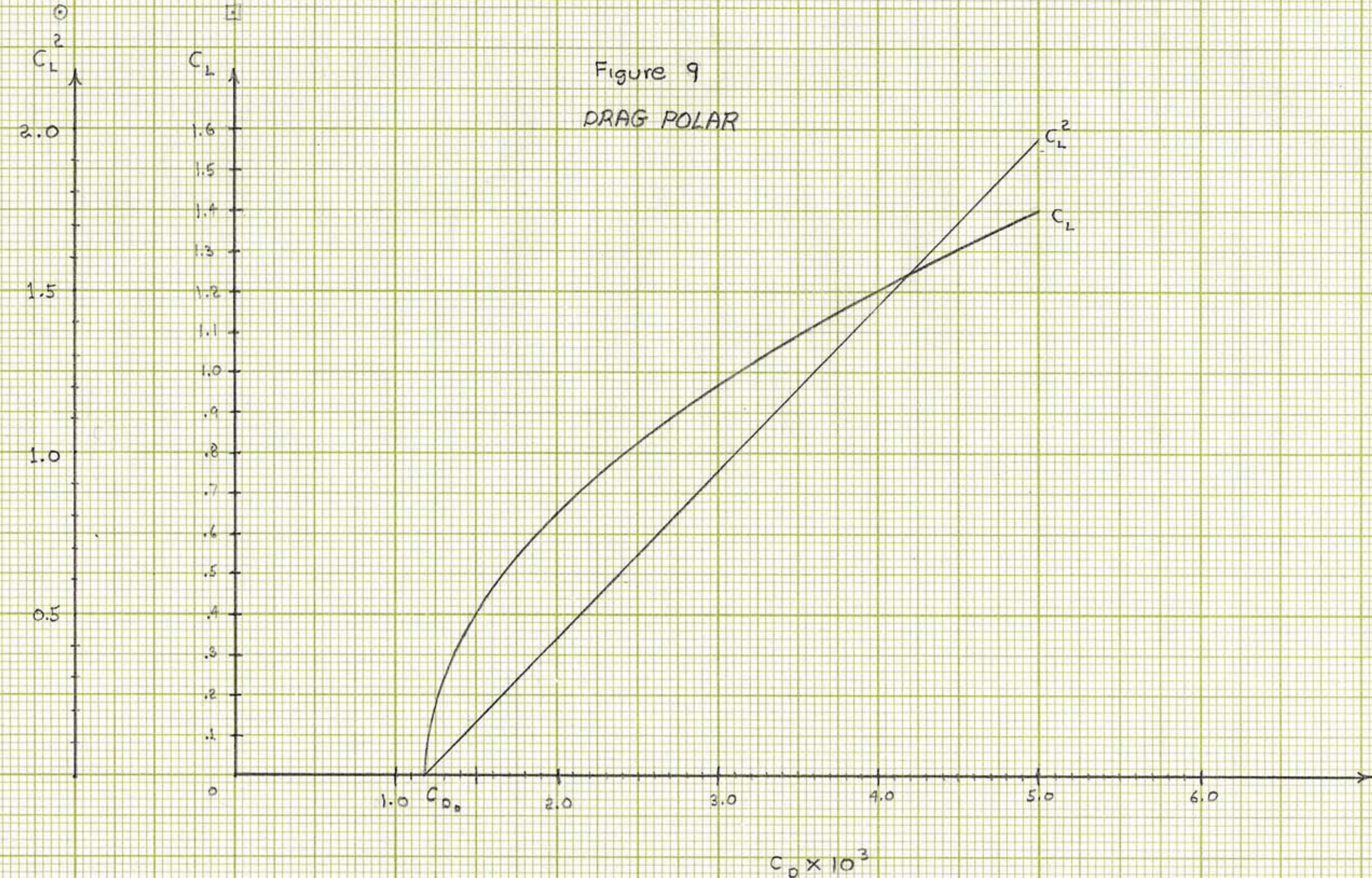


Figure 9
DRAG POLAR

The following parameters are calculated for figure 9.

$$L/D_{max} = \frac{1}{2}\sqrt{\frac{e\pi\Lambda}{C_{D_0}}} = 33.14 \qquad (12)$$

$$\theta = \operatorname{Tan}^{-1}\left(\frac{1}{L/D_{max}}\right) = 1.728^\circ \qquad (13)$$

$$C_L @ V_{z\,min} = \sqrt{3C_{D_0}e\pi\Lambda} = 1.351 \qquad (14)$$

$$C_L @ L/D_{max} = \sqrt{C_{D_0}e\pi\Lambda} = 0.780 \qquad (15)$$

Stability and Control : Vertical/Horizontal tails and Ailerons

As a first approximation for stability and control we calculate required areas for the wing, tail, and aileron surfaces and moment arms. The distance between the center of pressure of the wing (aerodynamic center) and the center of pressure of tail surfaces, l_{VT} and l_{HT}, were approximated from working drawings. Tail surface areas, S_{HT} and S_{VT}, were calculated from known tail volume coefficients. Tail volume coefficents C_{HT} and C_{VT} were approximated from working drawings of the standard cirrus. Aileron areas were choosen the same as the standard cirrus.

$C_{HT} = 0.432$, $l_{HT} = 143.3$ in.
$C_{VT} = 0.0304$, $l_{VT} = 140.0$ in.
Aileron area = 4920 in^2

$$S_{HT} = C_{HT}\,S(MAC)/l_{HT} = 1627\text{ in}^2 \qquad (16)$$

$$S_{VT} = C_{VT}\,S(MAC)/l_{VT} = 2031\text{ in}^2 \qquad (17)$$

Wing Materials and Properties

The wing will be constructed of foam core and epoxy-glass reinforced laminated skins as shown in Figure 9.

Exterior Skin

- 5 layers of 0.005 in. thick laminates oriented $\pm 45°$ with respect to the wing lateral axis
- 181 style glass fabric treated with silane Z-8-009

Interior Skin

- 2 layers of 0.005 in thick laminates oriented $\pm 45°$ with respect to the wing lateral axis
- 181 style glass fabric treated with silane Z-8-009

Rigid Urethane Foam

- CPR 9005-2 Upjohn Division of CPR
- $E_c = 1419$ psi, $\sigma_{ult_{Ten}} = 35$ psi ; $\sigma_{ult_{Comp}} = 500$ psi
- $\tau_{ult} = 20$ psi
- $\rho_c = 0.00116$ lb/in^3

Epoxy Resin and Glass Wet layup

procedure Ref. 7

	5 parts	Hycor CTBN	cook for one
1)	84 parts	Ciba 6005	hour @ 175°C
	3 parts	Methylon 75108	while stirring

2) Cool to room temperature, add 16 parts Ciba RD-4
3) To cure add 21 parts AEP
4) Post cure 12 hrs @ 50°C

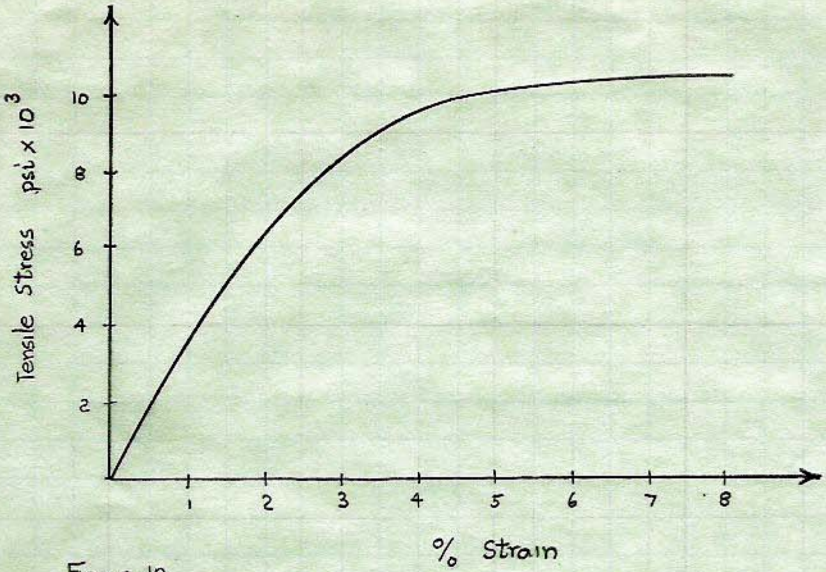


Figure 10

EPOXY RESIN STRESS-STRAIN CURVE
CIBA 6005/CTBN system

Vol % Glass 65
Vol % Resin 35
Wt % Glass 81
Wt % Resin 19
Material Cost 1.73 \$/lb
$\rho_s = 0.074\ \mathrm{lb/in^3}$

The wing weight can be estimated from the geometry and material densities given. The wing is approximated as two flat laminated skins seperated by two shear webs that form a box beam.

$$W_{wing} = \left[\rho_s\,(0.07 + 0.75\gamma) + 0.75\rho_c(1+\gamma)\right]\,S \qquad (18)$$

$$W_{wing} = 238.69\ \mathrm{lbs.}$$

Wing Centroid Location

The wing was approximated as a thin skin with a thin shear web located 0.475C and 0.383C from the leading edge. The wing C.G. location was determined by constructing a flat plate approximation of the wing crossection.

Horizontal 17.7" from leading edge or ≈ MAC/2
Vertical 1.3 upward from chord line

Fuselage, Tail, and Rudder Weight and Centroid Calculation

From a recent interview an experienced sailplane structure repairman† recommended 1/8 in. thick skin for fuselage and wings. The fuselage was divided into 22 sections. Each section was approximated as a constant crossectional cylinder from which centroids were calculated in Table III.

Horizontal, $d_{struct} = 139.7''$
Vertical, $e_{struct} = 10.79''$

$$W_{struct} = 149.1 \text{ lbs.}$$

Skin density Woven 181 fabric + epoxy $= \rho_s = 0.074 \text{ lb/in}^3$
Core density CPR 9005-2 Urethane $= \rho_c = 0.00116 \text{ lb/in}^3$

Fuselage Weight $= S(t_s + \rho_s) = 0.00925S$
Vertical Stablizer Weight $= S(2\, t_{s_{V.S.}} \rho_s) = 0.02775S$
Rudder and Horizontal Stabilizer Weight $= S(2t_{s_{RHS}} \rho_s + t_c \rho_c) = 0.0055S$

† private communication, Fred Giran, Mojave Airport, Mojave, California, 1972.

TABLE III. Fuselage Weight and Centroid Location

Station	Circumference in.	Distance Between Stations in.	Surface Area Between Stations in.^{2}	Moment Arm d in.	Moment Arm e in.	Weight lbs.	d x w in-lb	e x W in-lb
1	0	9.0	183.6	4.5	-1.2	1.608	7.641	- 2.038
2	40.8	10	354.0	13.8	-1.5	3.275	45.195	- 4.913
3	30.0	10	326.5	23.8	-1.9	3.020	71.876	- 5.738
4	35.3	10	669.5	33.8	-2.1	6.193	209.323	- 13.005
5	38.6	10	400.5	43.8	-2.2	3.705	162.279	- 8.151
6	41.5	10	425.0	53.8	-2.4	3.931	211.488	- 9.434
7	43.5	10	443.0	63.8	-2.4	4.100	261.580	- 9.840
8	45.1	10	673.5	73.8	+1.3	6.230	459.774	8.100
9	89.6	10	875.5	83.8	+4.7	8.100	678.780	38.070
10	85.5	10	823.5	93.8	+4.3	7.617	714.475	32.753
11	79.2	10	771.0	103.8	+3.7	7.132	740.302	26.388
12	75.0	10	711.0	113.8	+3.7	6.577	748.463	24.335
13	67.2	10	621.0	123.8	+3.2	5.744	711.107	18.381
14	57.0	10	525.0	133.8	+2.3	4.856	649.733	11.169
15	48.0	10	462.0	143.8	+1.9	4.274	614.601	8.121
16	44.5	10	417.5	153.8	+1.7	3.862	593.976	6.565
17	39.0	10	367.0	163.8	+1.6	2.470	404.586	3.952
18	34.4	10	326.0	173.8	+1.5	3.016	524.181	4.524
19	30.8	10	287.5	183.8	+1.5	2.660	488.908	3.990
20	26.7	10	251.3	193.8	+1.5	2.325	450.585	3.488
21	23.6	13.6	292.6	205.3	+1.7	2.707	555.747	4.602
22	19.5	—	—	—	—	—	—	—
Horizontal Stabilizer	—		1627.0	227.6	+53.7	8.949	2036.792	480.561
Rudder	—		785.9	242.8	+26.6	4.322	1049.382	114.965
Vertical Stabilizer	—		1245.5	225.2	+26.0	34.563	7783.588	898.638
Cockpit seat molding	—			45.0	-4.0	6.000	270.000	-24.00
Skid (Tail)	—			230.0	-1.0	1.780	410.000	-1.780
						Σ 149.099	Σ 20854.360	Σ 1609.702

Canopy Weight and Centroid Location

Material Acrylic clear plastic

$$\rho_{canopy} = 0.0435 \ lb/in^3 \qquad t_{canopy} = 0.125 \ in$$

$$W_{canopy} = 0.00544 \ S_{canopy}$$

The canopy was divided into 8 sections and approximated as constant crossectional cylinders from which the centroid is calculated in Table IV.

TABLE IV CANOPY WEIGHT AND CENTROID LOCATION

Station	in. Circumference	in. Distance Between Stations	in.² Surface Area Between Stations	in. Moment Arm d	e	lbs. Weight	in.lb. d×W	in.lb. e×W
$2_{modified}$	24.2	7	195.3	15	7.4	1.0624	15.94	7.862
3	31.6	10	359.0	23.8	7.4	1.9530	46.48	14.452
4	40.2	10	419.0	33.8	8.8	2.2794	77.04	20.059
5	43.6	10	445.5	43.8	10.0	2.4235	106.15	24.235
6	45.5	10	453.0	53.8	11.0	2.4643	132.58	27.107
7	45.1	10	443.0	63.8	11.4	2.4100	153.76	27.474
8	43.5	5	417.5	71.4	16.8	2.2712	162.16	38.156
$9_{modified}$	40.0					14.8633	694.11	159.345

$$W_{canopy} = 14.8633$$

$d_{canopy} = 46.70$ From nose

$e_{canopy} = 10.72$ From datum

OXYGEN BOTTLE

$$W_{oxy} = 8 \ lbs$$

$d_{oxy} = 78''$ From nose

$e_{oxy} = 17''$ From datum

Instrument Panel, Weight, and Centroid Location

Two variometers

-0 ± 10 Knots 2.5 lbs.
-0 ± 5 Knots 2.5 lbs.

One Altimeter 2.0 lbs

One Airspeed indicator, Winter MPH + Knots, 4 lbs.

One turn and bank indicator, electric, 4 lbs.

One radio, 360 communication frequencies, 10 lbs.

Metal enclosure 3 lbs.

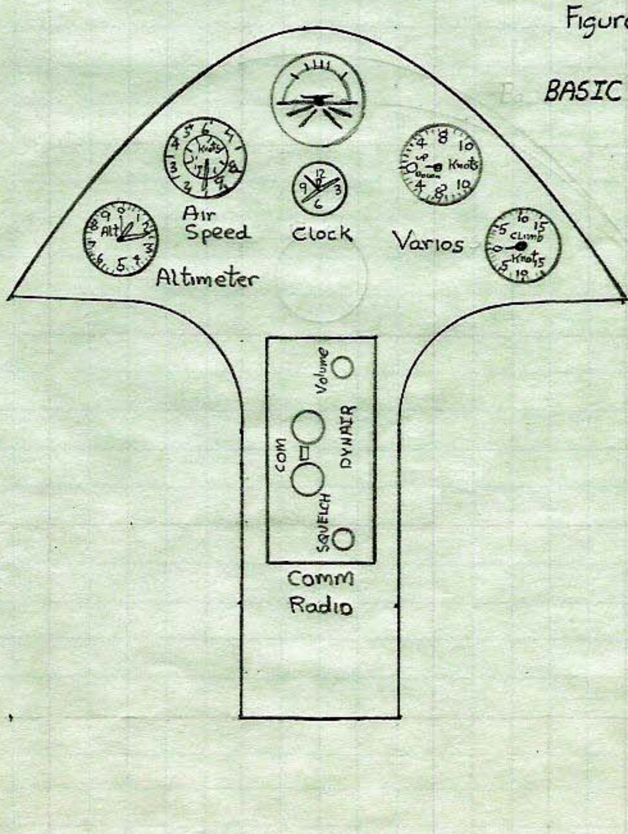
Total Weight = 28 lbs.

d_{inst} = 33" From nose

e_{inst} = +4" From datum

Figure 11

BASIC INSTRUMENT PANEL



Landing Gear Structure, Weight, and Centroid Location.

Wheel: $\Delta C_{D_0} = 0.003$

Manually - retractable mono wheel standard.

Tost wheel with drum brake and Continental

4.00 x 4 tire, psi 50.

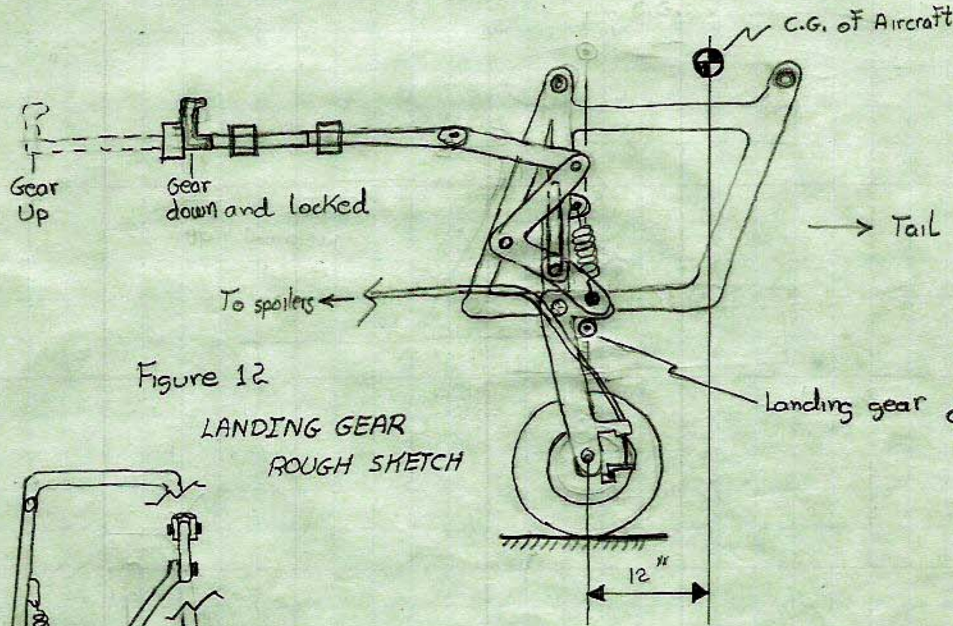
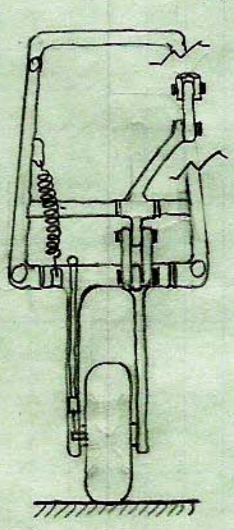


Figure 12

LANDING GEAR ROUGH SKETCH



$W_{estimated} = 50$ lbs

12" forward of the plane's C.G. Location is a reasonable distance when compared with most tail dragers.

According to "F.A.A. Basic glider criteria", Ref. 8, the minimum vertical limit load factor should be 4.0 with a horizontal component equal to ¼ the vertical component, for the detailed structural design.

TABLE V CENTROID LOCATION FOR FUSELAGE AND ATTACHMENTS

Item	Description	Weight, lbs	Moment arm d	Moment arm e	Moment in.lb. $d \times W$	Moment in.lb. $e \times W$
1	Fuselage Structure	149.10	139.7	+10.70	20854.36	106.702
2	Canopy	14.86	46.7	+10.72	694.11	159.200
3	Control System	15.00	120.0	0	1800.00	0
4	Instruments	28.00	33.0	+4.00	924.00	112.000
5	Oxygen	5.00	78.0	+17.00	390.00	85.000
6	Pilot + Chute	250.00	60.0	+1.00	15000.00	250.000
	$W_{fuselage}$ =	461.96			20854.36	712.902

$$d_1 = \frac{\sum Wd}{\sum W} = 85.8 \qquad e_1 = \frac{\sum We}{\sum W} = 1.54$$

$$W_{total} = W_{fuselage} + W_{wing} + W_{\substack{landing\\gear}} = 461.96 + 238.69 + 50.0 = 750.65 \text{ lbs}$$

$$W_{total}\left(d_3 + \frac{MAC}{4}\right) = W_{fuselage}\, d_1 + W_{wing}\left(\frac{MAC}{2} + d_3\right) + W_{\substack{landing\\gear}}\left(d_3 + \frac{MAC}{4} - 12\right)$$

For payload 250 lbs $d_3 = 81.56''$

For payload 170 lbs $d_3 = 86.45''$

$d_{3\,average} = 84''$

Centroid Location for Total Aircraft

TABLE VI CENTROID LOCATION TOTAL AIRCRAFT

Item	Description	Weight lbs.	Moment Arm, in. e	Moment, in-lb $e \times W$
1	Fuselage	461.96	1.54	712.902
2	Wing	238.69	12.80	3280.000
3	Landing Gear	50.00	-10.00	-500.000
		750.65		3492.902

$$e_3 = \frac{\Sigma eW}{\Sigma W} = 4.64''$$

Airbrakes and Landing Distance

For a detailed structural design of the airbrakes it is recommended by the F.A.A. "Basic Glider Criteria", Ref.8, that spoilers and their attachment structures be premised on the limit loading obtained by the formula below, where this approximation

$$W_{sp} = 0.0052\, V_{sp}^2$$

W_{sp} = limit loading, lb/FT^2

V_{sp} = IAS at which maximum operation of spoilers is assumed, m.p.h.

assumes the load is uniformly distributed over the wing.

The increase in drag coefficient's and efficiency factors for loading will be approximated from parameters given in reference 10 for landing.

Air Brakes	$\Delta C_{D_0} = 0.048$	$\Delta K = 4.09$
Landing Gear	$\Delta C_{D_0} = 0.003$	$\Delta K = 0$
	$C_{D_0} = 0.00628$	$K = 0.784$

$$C_D = 0.0628 + 4.10934\, C_L^2 \qquad (20)$$

From reference 10

Nonlinear differential equation for landing run

$$\frac{dV}{dt} = -g\mu - \frac{g\rho S}{2W}\left(C_D - \mu C_{L_g}\right)V^2 \qquad (21)$$

$$a = -g\mu$$

$$b = \frac{g\rho S}{2W}\left(C_D - \mu C_{L_g}\right)$$

Solution where $V = V_L$ at $x = 0$

$$\text{Landing distance} = +\frac{1}{2b}\ln\left(1 - \frac{b}{a}V_L^2\right) \qquad (22)$$

Given

W = Airplane weight = 750.87 lb (maximum) / 670.87 lb (minimum)
S = Wing area = 110 FT^2
V_2 = Aircraft speed just before touch down = 88 FT/sec
μ = Coefficient of Friction = 0.12 without brakes
(Ref. 8) = 0.25 with brakes
C_{D_0} = Coefficient of Drag a zero lift = 0.0118
Λ = Aspect Ratio = 22
K = constant = $1/e\pi\Lambda$ (with spoilers & Landing Gear) = 4.10934
ρ = airdensity = 0.002377 slugs/FT^3
g = 32.2 FT/sec^2

An approximation for C_{L_g} is derived as follows. The distance will be a minimum when the deacceleration is a maximum. If equation (22) is differentiated with respect to C_{L_g} and set equal to zero then C_{L_g} can be calculated as follows.

$$\frac{d}{dC_{L_g}}\left(C_D + \mu C_{L_g}\right) = 0 \qquad (23)$$

where $$C_D = C_{D_0} + K C_{L_g}^2 \qquad (24)$$

$$\frac{d}{dC_{L_g}}\left(C_{D_0} + K C_{L_g}^2 + \mu C_{L_g}\right) = 0 \qquad (25)$$

$$2KC_{L_g} - \mu = 0, \quad C_{L_g} = \frac{\mu}{2K}$$

$$\text{recall, } b = \frac{g\rho S}{2W}\left(C_D - \mu C_{L_g}\right) = \frac{g\rho S}{2W}\left(C_{D_0} - \frac{\mu^2}{4K}\right)$$

	with brakes	without brakes
recall, $-g\mu = a =$	-8.05	$= -3.864$
$b =$	-0.00105	$= -0.000191$

Ground run

Landing distance with out brakes = 726.2 Feet
Landing distance with brakes = 307.4 Feet

Performance: Speed Polar (equations Ref. 3)

$$\frac{C_L}{C_D} = \frac{\dfrac{2W}{\rho S C_{D_0}}}{V^4 + \dfrac{4W^2}{C_{D_0}\rho^2 S^2 \pi A e}} V^2 \qquad (26)$$

$$V_Z = \frac{V}{C_L/C_D} \qquad (27)$$

where $A=\Lambda$ and $e=0.748$

$$\frac{C_L}{C_D} = \frac{\dfrac{2(750.87)}{0.002377(110)\,0.0118}}{V^4 + \dfrac{4(750.87)^2}{0.0118(0.002377)^2(110)^2\,3.1416(22)\,0.748}} V^2$$

$$\frac{C_L}{C_D} = \frac{486733.09}{V^4 + 54074137.98} V^2$$

TABLE VII SPEED POLAR

Pnt	$V_{FT/sec}$	V_{Knots}	$V_{Z\ FT/sec}$	$V_{Z\ FT/min}$	C_L/C_D
1	50.67	30	2.46	147.59	20.60
2	67.56	40	2.28	136.68	29.66
3	84.45	50	2.55	153.18	33.08
4	101.34	60	3.23	194.07	31.33
5	118.23	70	4.34	260.10	27.27
6	135.12	80	5.89	353.43	22.94
7	152.01	90	7.95	476.84	19.13
8	168.90	100	10.55	633.41	16.00
9	185.79	110	13.77	826.42	13.48
10	202.68	120	17.65	1059.23	11.48

$$V_{@V_{Z_{min}}} = \sqrt{\frac{2W}{\rho S\sqrt{3C_{D_0}\pi A e}}} \qquad (28)$$

$$V_{@V_{Z_{min}}} = \sqrt{\frac{2(750.87)}{0.002377(110)\sqrt{3(0.0118)\,3.1416(22)\,0.748}}} = 65.16\ \text{Ft/sec}$$

44.42 mph

Wing Loading

$W = 750.87$ $\quad$ $W = 670.87$

$W/S = 6.826$ $\quad$ $W/S = 6.099$

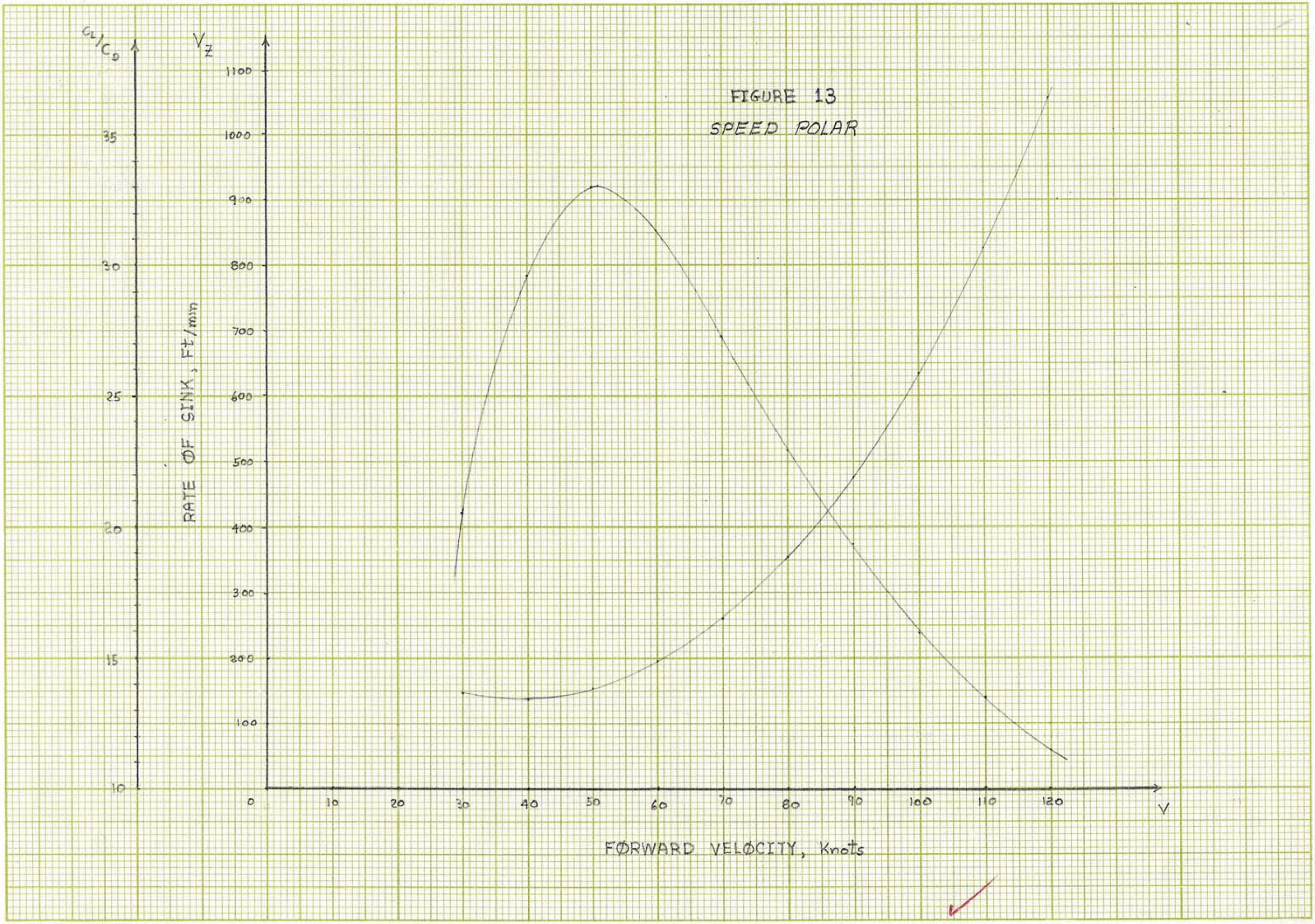


FIGURE 13
SPEED POLAR
C_L/C_D
35
30
25
20
15
10
V_Z
RATE OF SINK, Ft/min
1100
1000
900
800
700
600
500
400
300
200
100
0
10
20
30
40
50
60
70
80
90
100
110
120
V
FORWARD VELOCITY, Knots

$$V_{z_{min}} = \frac{4\,C_{D_0}^{\,0.25}}{(3\pi A e)^{0.75}} \sqrt{\frac{2W}{\rho S}}$$

$$V_{z_{min}} = \frac{4\,[0.0118]^{0.25}}{[3(3.1416)\,22\,(0.748)]^{0.75}} \sqrt{\frac{2(750.87)}{0.002377(110)}} = 2.27 \text{ Ft/sec}$$

136.2 Ft/min

$$V_{@\frac{L}{D}max} = \sqrt{\frac{2W}{\rho S\sqrt{\pi A C_{D_0} e}}}$$

$$V_{@\frac{L}{D}max} = \sqrt{\frac{2(750.87)}{0.002377(110)\sqrt{3.1416\,(22)\,0.0118\,(0.748)}}} = 85.75 \text{ Ft/sec}$$

58.47 mph

TURNING FLIGHT

$$V_z = \frac{C_D^{\,2}}{C_L^{\,1.5}} \sqrt{\frac{2W}{\rho S}}\ \frac{1}{\cos\phi^{1.5}} \qquad (29)$$

$$C_L = \frac{W}{S}\frac{2}{\rho g}\frac{1}{r\sin\phi}$$

$$V_{z_{\phi,r}} = \frac{C_{D_0} + \dfrac{\left(\frac{W}{S}\frac{2}{\rho g}\frac{1}{r\sin\phi}\right)^{2.0}}{e\pi A}}{\left(\frac{W}{S}\frac{2}{\rho g}\frac{1}{r\sin\phi}\right)^{1.5}} \sqrt{\frac{2W}{\rho S}}\ \frac{1}{\cos\phi^{1.5}} \qquad (30)$$

TURNING IN A THERMAL

$$C_{L@V_{z_{min}}} = \sqrt{3C_{D_0} e\pi A} \qquad (31)$$

$$\sqrt{3C_{D_0} e\pi A} = \frac{W}{S}\frac{2}{\rho g}\frac{1}{r\sin\phi} \qquad (32)$$

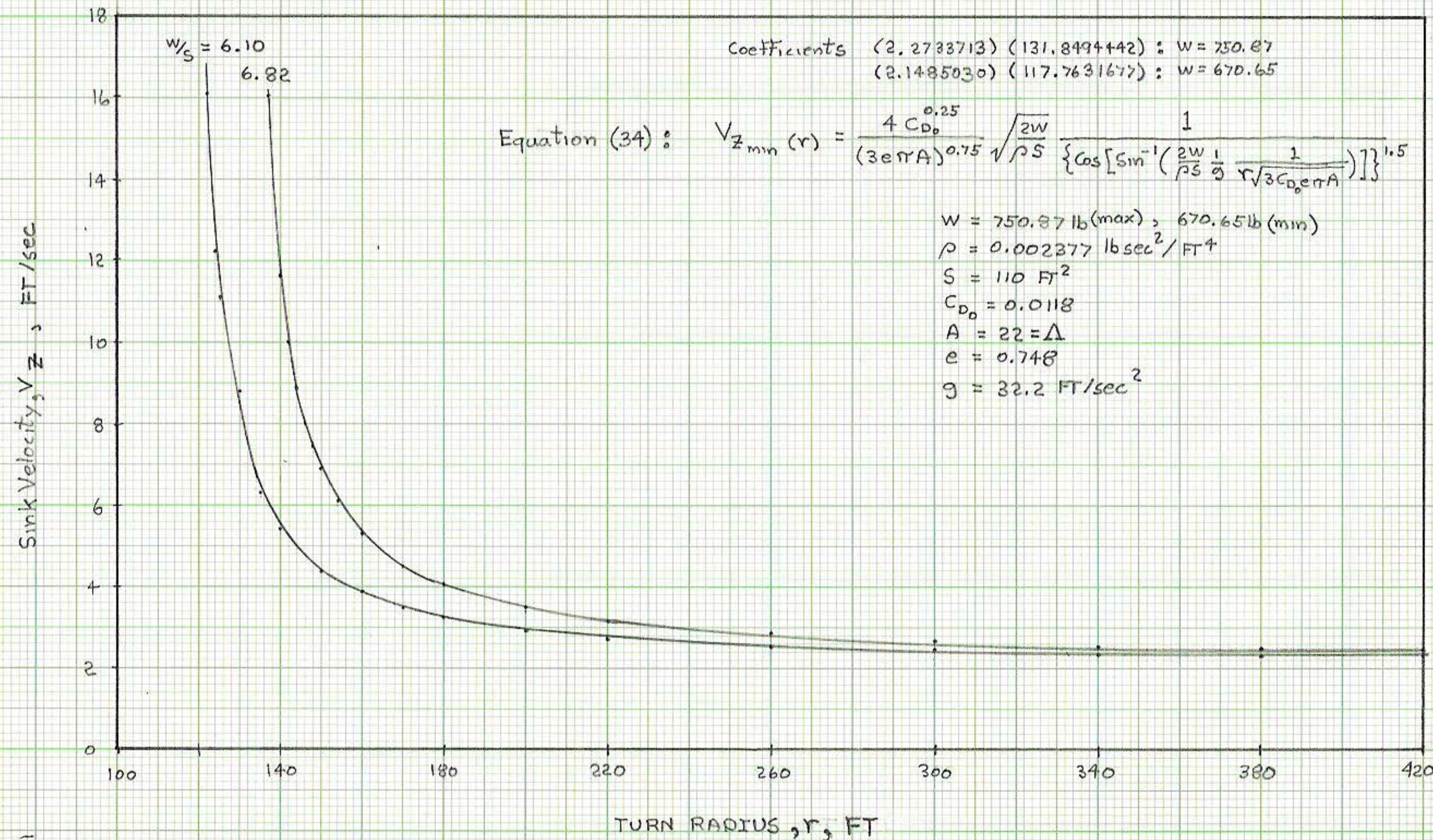
$$\phi = \sin^{-1}\left[\frac{W}{S}\frac{2}{\rho g}\frac{1}{r\sqrt{3C_{D_0} e\pi A}}\right] \qquad (33)$$

Figure 14 V_z .vs. r

$$V_{z_{min}\,f(r)} = \frac{4\,C_{D_0}^{\,0.25}}{(3e\pi A)^{0.75}}\sqrt{\frac{2W}{\rho S}}\ \frac{1}{\left\{\cos\left[\sin^{-1}\left(\frac{W}{S}\frac{2}{\rho g}\frac{1}{r\sqrt{3C_{D_0} e\pi A}}\right)\right]\right\}^{1.5}} \qquad (34)$$

Figure 14

MINIMUM SINK IN A THERMALING TURN



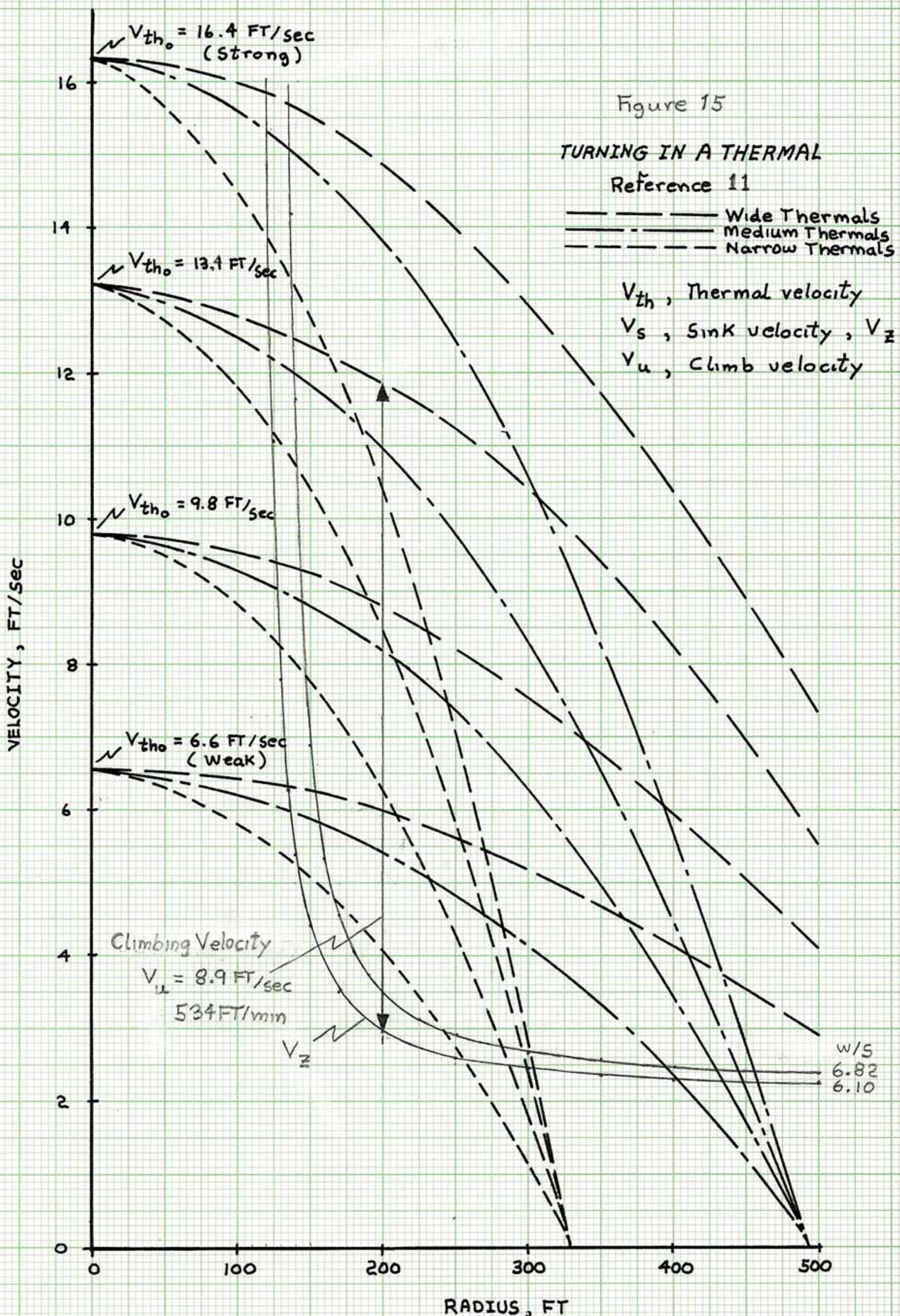
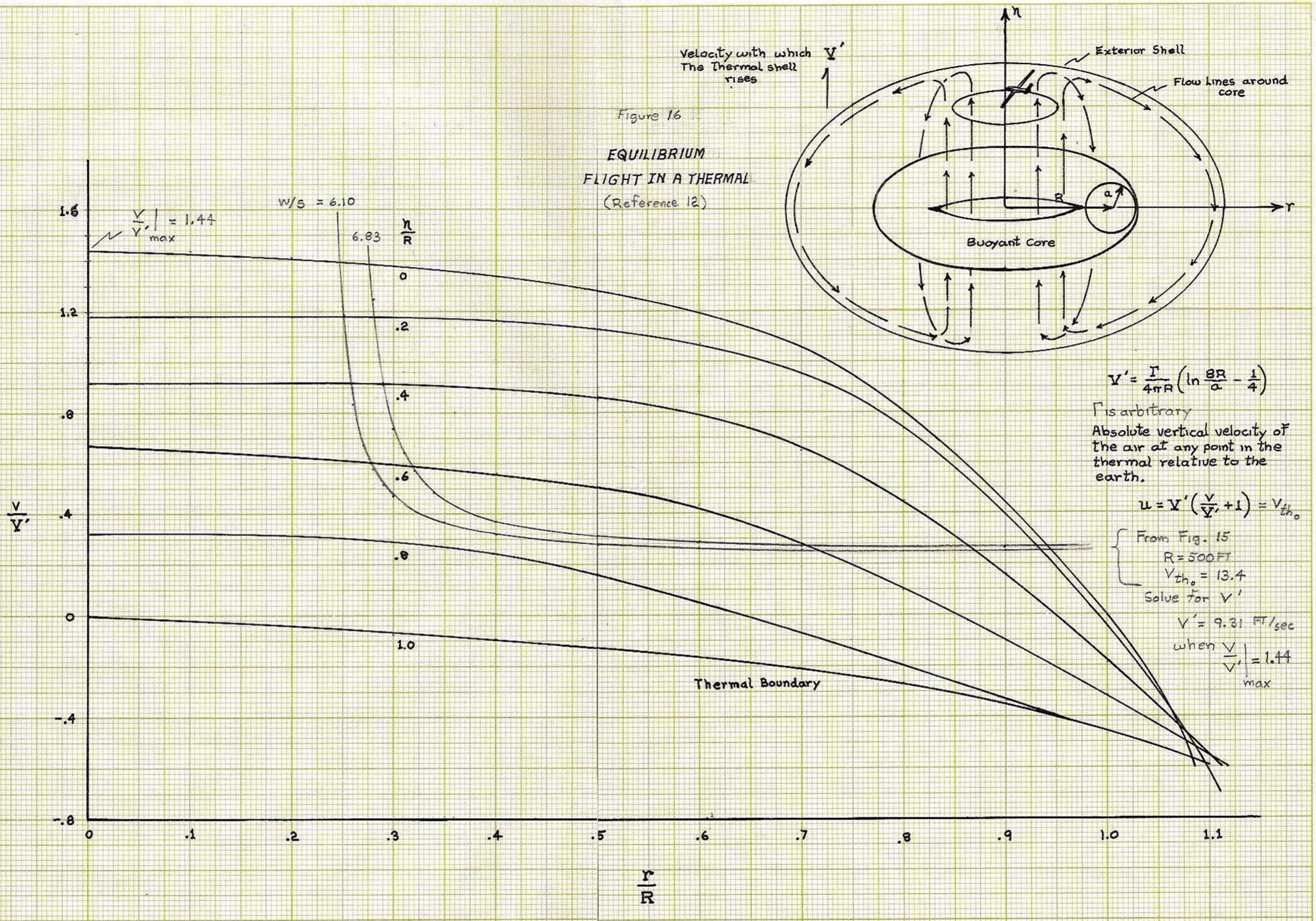


Figure 15
TURNING IN A THERMAL
Reference 11
Wide Thermals
Medium Thermals
Narrow Thermals
V_{th}, Thermal velocity
V_s, Sink velocity, V_z
V_u, Climb velocity
V_{th_o} = 16.4 FT/sec (Strong)
V_{th_o} = 13.4 FT/sec
V_{th_o} = 9.8 FT/sec
V_{th_o} = 6.6 FT/sec (Weak)
Climbing Velocity
V_u = 8.9 FT/sec
534 FT/min
V_z
W/S
6.82
6.10
VELOCITY, FT/SEC
RADIUS, FT
0
2
4
6
8
10
12
14
16
0
100
200
300
400
500

Figure 16

EQUILIBRIUM
FLIGHT IN A THERMAL
(Reference 12)



Thermal diagram for vertical velocity field (Nondimensional, $\frac{R}{a} \approx 5$)

The first design iteration is complete where performance is measured in free and thermalling flight shown in figures 13, 14, 15, and 16. Overall performance in cruising flight between thermals is considered next.

There are two possible cruise configurations that can be considered between thermals. First, is an instantaneous cruise velocity based on the "speed to fly" in a descending air mass which is outlined in Appendix A of Ref. 13. In this design a second method is used which calculates the "speed to fly", V_c, from the speed polar, Fig. 17, and Eqn (35), Ref. 4. For competitive flight two performance diagrams are included in this design: (1) "Speed to fly" for best L/D, Fig. 18, and (2) "speed to fly" between thermals", Fig. 19, where thermals are assumed to have an average climb velocity, V_u, from Fig. 17. These two performance diagrams are calculated using Eqn. (35) from Ref. 4 which is based on the time required to glide to the next thermal and regain lost altitude.

$$V_{avg} = \frac{V_c}{1 + V_z'/V_u} \qquad (35)$$

where V_z' is the summation of the sinking speed, V_z, and the average DownDraft velocity, V_{DD}, between thermals.

$V_z' = V_z + V_{DD}$ and V_{DD} is approximated by $0.2(V_u + V_z)$

where $V_u + V_z$ is the total thermal strength and V_u is the actual climb or vertical airspeed ("variometer") indicator.

Equation (35) is graphically compared with Fig. 17. The average cruise velocity, V_{avg}, is obtained by the intersection of the V axis with the line drawn from V_u to the tangent of the speed polar, Ref. 11. Figure 17 is the same as Fig. 13 except the vertical axis is extended to calculate V_{avg} from V_u. For example the climb velocity $V_u = 534$ Ft/min was estimated in Fig. 15 for a wide thermal at $V_{th_o} = 13.4$ Ft/sec and used in Fig. 17 to find $V_{avg①} = 48$ knots, where $V_{avg①}$ coincides with $V_{@L/D\,max}$.* If V_u was greater than 534 Ft/min, e.g. $V_u = 700$ Ft/min, then the average cruise "speed to fly", $V_{avg②} = 53$ knots, is greater than $V_{@L/D\,max}$. Hence the design requirement for competition, $V_{avg} > V_{@L/D\,max}$, is satisfied at least for wide thermals at $V_{th_o} = 13.4$ Ft/sec from Fig. 15.

Subsequent design iterations requires a subjective evaluation of this design requirement. The author is satisfied with the present design which competes well for wide-medium sized thermals. This trend shows larger L/D design extends competitive performance for weaker thermals.

In summary the cruise performance in competitive flight is shown in Figs. 18 and 19 for different thermal strengths where V_{avg} and L/D are both plotted as functions of best speed to fly.

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* Typically Vu from Fig. 15 does not yield $V_{avg①} = V_{@L/D\,max}$ in Fig. 17 but constructed here as a limiting case for a competitive design.

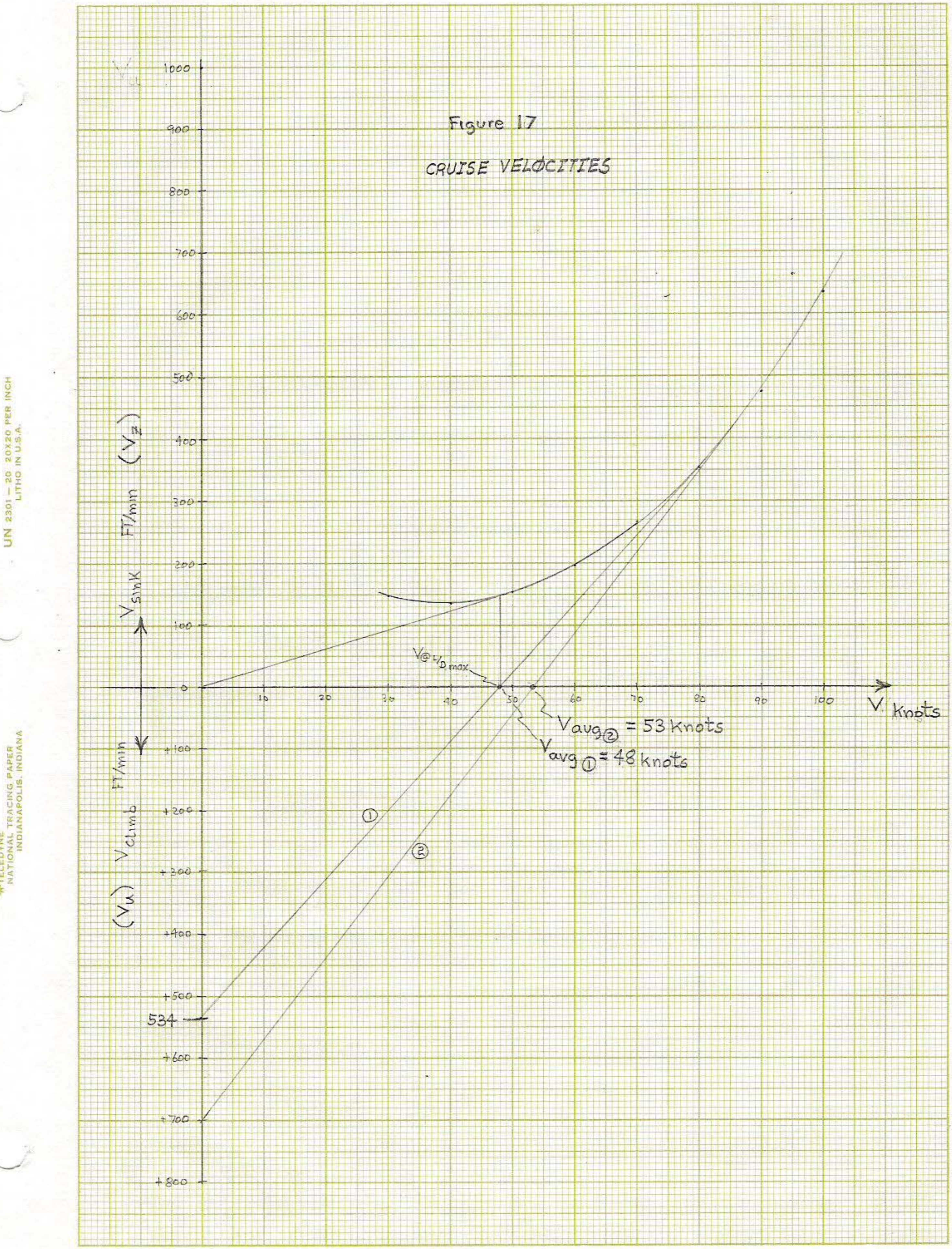


Figure 17
CRUISE VELØCITIES
1000
900
800
700
600
500
400
300
200
100
0
+100
+200
+300
+400
+500
534
+600
+700
+800
10
20
30
40
50
60
70
80
90
100
V Knots
(Vz) FT/min Vsink
(Vu) Vclimb FT/min
V@L/D max
Vavg② = 53 Knots
Vavg① = 48 knots
①
②
UN 2301 — 20 20X20 PER INCH
LITHO IN U.S.A.
TELEDYNE
NATIONAL TRACING PAPER
INDIANAPOLIS, INDIANA

Figure 18

BEST L/D FOR CRUISE PERFORMANCE

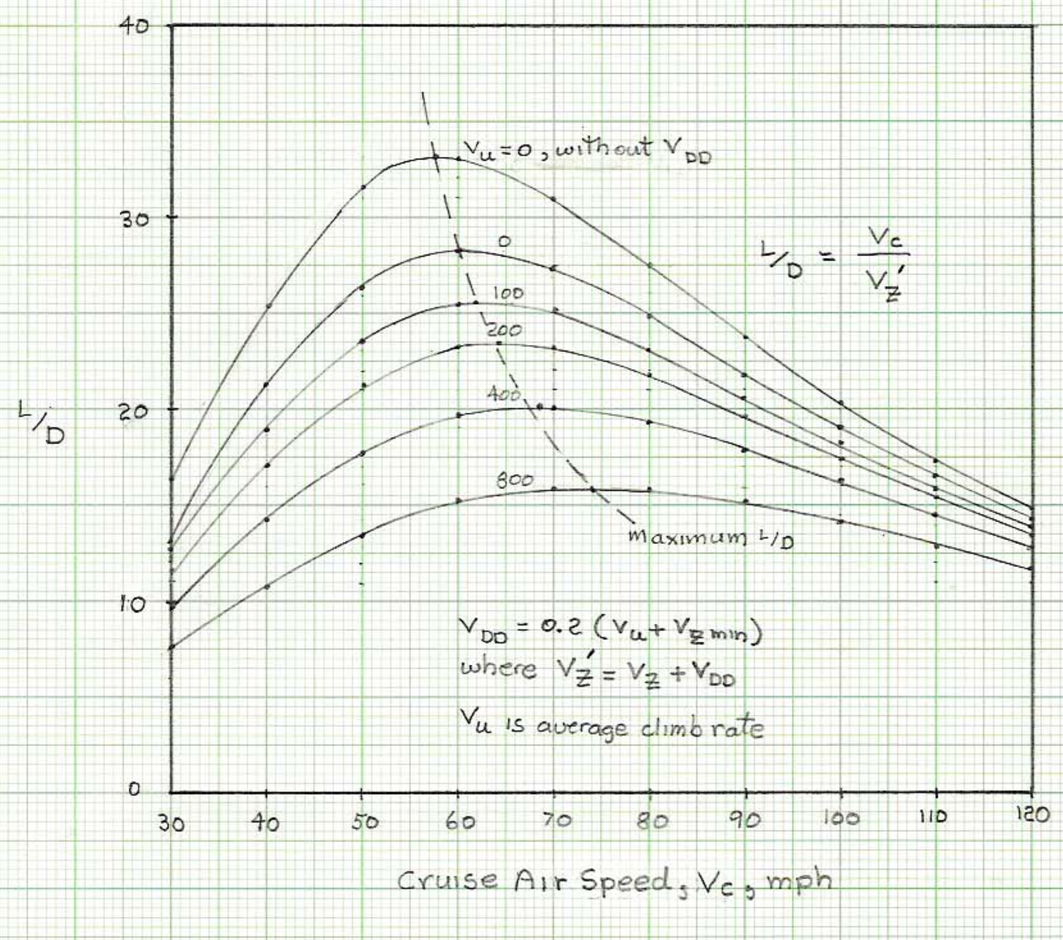
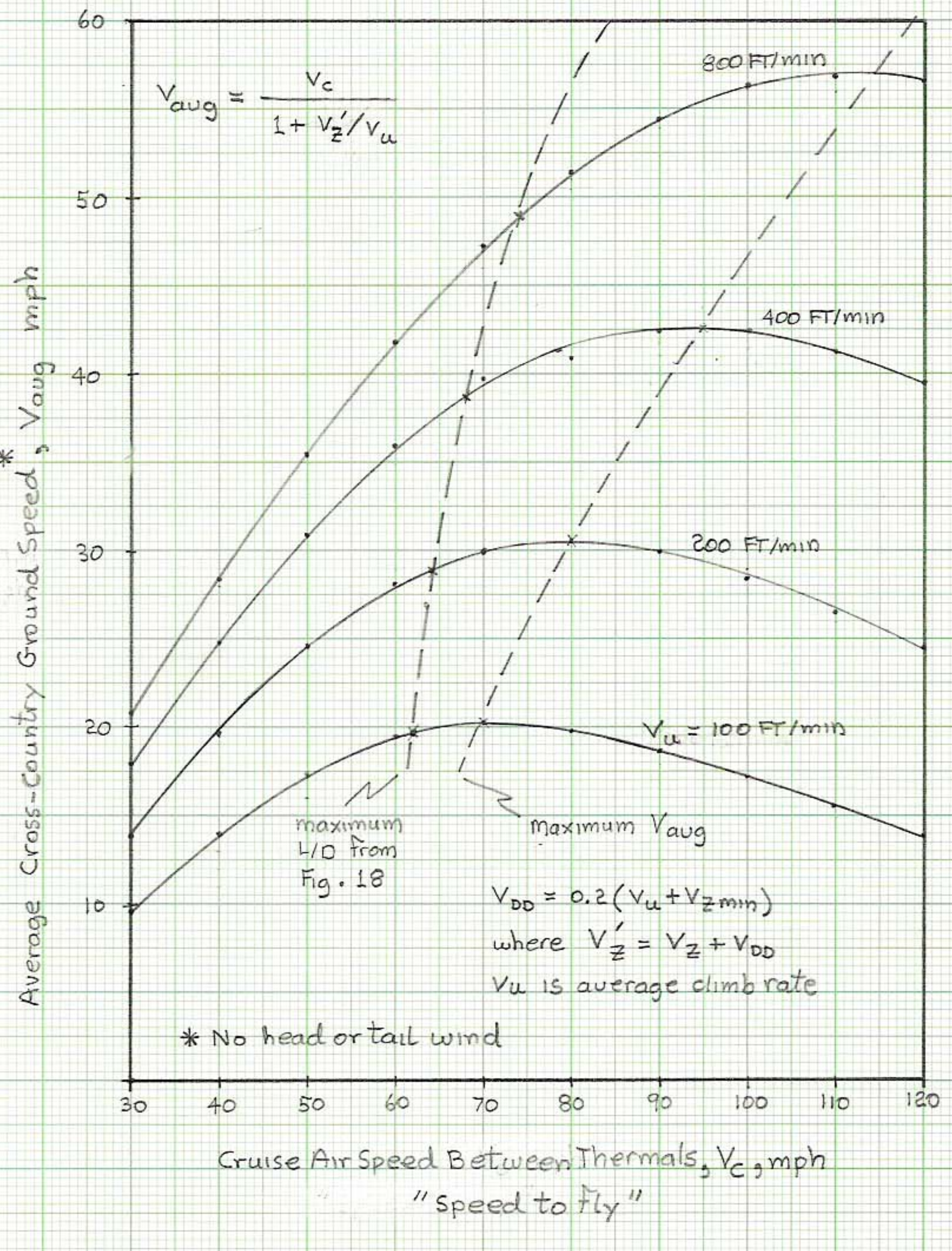


Figure 19

CROSS COUNTRY CRUISE PERFORMANCE



Calculations

line 1 $$n = \frac{V^2 C_{L_{max}}}{391 \frac{W}{S}} = \frac{1.2}{391(6.826)} V^2$$

$$n = 0.00045\, V^2$$

V	n
10	0.045
50	1.125
70	2.201
90	3.650
100	4.50
120	6.48

line 2 $$n = -\frac{V^2}{391 \frac{W}{S}} = -\frac{1}{391(6.826)} V^2$$

$$n = -0.000375\, V^2$$

V	-n
10	-0.0375
20	-0.150
50	-0.937
70	-1.837
100	-3.750

(+)Manuever Load Factor $$n = 1 + \frac{K\,24\,V_g\,m}{575 \frac{W}{S}}$$

V_g m.p.h = 159
$K = 0.81$
$m = 6.77$ units/rad.
W/S = lb/FT²

$$= 1 + \frac{0.81(24)159(6.77)}{575(6.826)}$$

$$= 1 + \frac{20960}{3920} = 1 + 5.34$$

$$n = 6.34$$

(-)Manuever Load Factor $$n = 1 - \frac{K\,24\,V_g\,m}{575 \frac{W}{S}}$$

$$n = 4.34$$

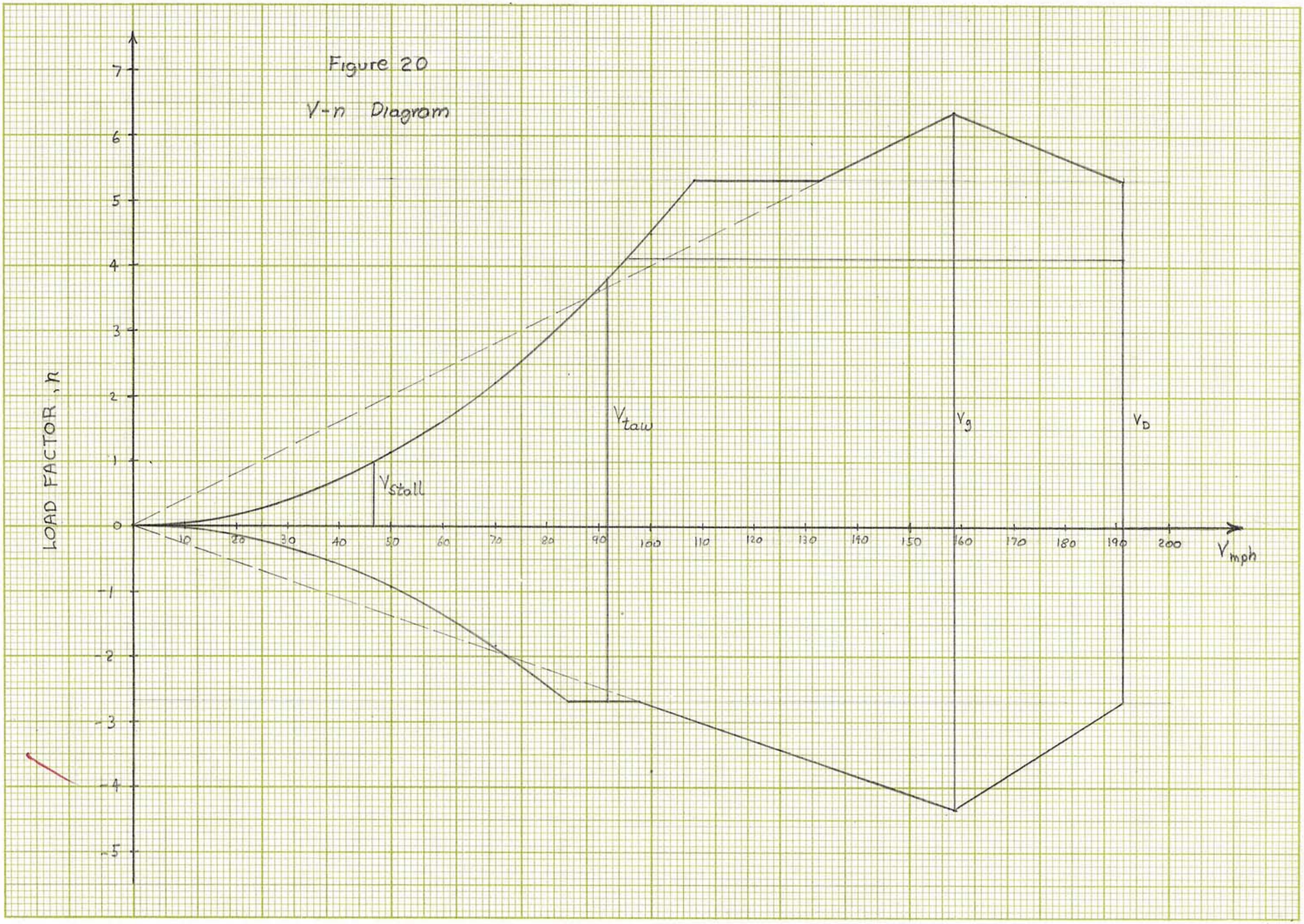


Figure 20
V-n Diagram
LOAD FACTOR, n
7
6
5
4
3
2
1
0
-1
-2
-3
-4
-5
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
160
170
180
190
200
V_{mph}
V_{stall}
V_{taw}
V_g
V_D

Engine Calculations

Engine Given

30 H.P. @S.L.
2400 rpm
68 lbs.

Estimated Frame weight
15 lbs.
Parachute weight
20 lbs.
Fuel weight estimate
25 lbs.

weight updated
$W = W + 128 = 878.87$ lbs with engine

a) Horsepower required calculated

For small angles of climb ref. 7

$$HP_{req} = \frac{\rho C_{D_0} S}{348} V_{mph}^3 + \frac{4W^2}{1610 \pi e \Delta \rho S} \frac{1}{V_{mph}}$$

Curve plotted Figure 21

b) Horsepower avaliable

Must first choose a propeller. The normal method of calculation starts with an educated estimate of the maximum airspeed for which a propeller is choosen. The thrust horsepower available can then be calculated knowing the prop efficiency. The intersection of $THP_{avaliable}$ and $THP_{required}$ gives the real maximum airspeed. If this maximum airspeed doesnot correspond with the initial estimated maximum airspeed then the propeller has been choosen incorrectly. A better estimates could then be made on the second, third or as many iterations as necessary so that the airspeed used in designing the propeller corresponds to the maximum airspeed obtianed from the thrust horsepower curves

This method of calculating propeller performance and thrust horsepower available assumes one is designing an aircraft for maximum efficiency in the cruise configuration. Unfortunately powered sailplanes only utilize power for climbing to altitude and or emergency conditions, and then the power is turned off for cruising in a gliding configuration. Since the increased drag of the power plant, propeller, and frame adversely effects the gliding performance some designs have incorporated retractable power plant. It has been the authors observation that the retractable propeller powered sailplanes where built when sailplanes needed emergency power on days when there was changing lift conditions. Todays sailplanes on an average california desert day can stay aloft all day with only a 2000 foot tow. The advance in sailplane performance has therefore changed the requirement for powered sailplanes. Todays high performance sailplanes need only climb to altitude. The need for power in emergency has been eliminated because of advances in sailplane design. Therefore the need for a retractable power plant is in the authors opinion obsolete but the need for a power plant is not. Therefore the idea of a jettisonable powerplant is a reasonable design requirement. Where the powerplant propeller combination is designed for maximum efficiency in climb.

From representative data on some european powered sailplanes with equivalent horsepower powerplants. The Forward speed for maximum rate of climb averages 110 Km/Hr (68.5 mph) with a top speed of 165 Km/Hr at Sea level. The propeller should be designed not for maximum efficiency for cruising but instead for climbing. Therefore choose propeller with a maximum efficiency at 68.5 mph.

$$V_{design} = 68.5 \text{ mph}$$

$$C_S = \frac{0.638\ V_{mph}}{N_{rpm}^{2/5}\ H.P.^{1/5}} = \frac{0.638\,(68.5)}{(2400)^{0.4}(30)^{0.2}} = \frac{43.7}{22.5\,(1.97)} = 0.99$$

From Figure 15.7 $\eta = 0.719$ $\beta_{@.75R} = 10°$

From Figure 15.8 $\frac{V}{ND} = 0.44$

Note: This is working on the lowest portion of propeller design curves

$$D = \frac{V_{FT/sec}}{N_{\frac{rev}{sec}}\ 0.44} = \frac{100}{40\,(0.44)} = 5.7 \text{ Feet diameter}$$

Next the thrust horsepower available is calculated for three altitudes sealevel (S.L.), 3000 FT, and 5000 Feet

The calculation for table VIII is accomplished as follows

For $\rho_{altitude}$ and $BHP_o = BHP_{SL}\left(\frac{\rho}{\rho_{SL}}\right)^{1.3}$

Procedure:

1. choose V/ND
2. } Find on curve V/ND vs. C_T; C_P
3. }
4. C_{P_o}/C_{P_n}
5. $\frac{N}{N_o} = \sqrt{C_{P_o}/C_{P_n}}$
6. $RPM = N_o\sqrt{C_{P_o}/C_{P_n}}$
7. $BHP = BHP_o\sqrt{C_{P_o}/C_{P_n}}$
8. C_T/C_P
9. $\eta = C_T/C_P\ V/ND$
10. $THP = \eta\, BHP$
11. $V_{mph} = \frac{V}{ND}\ RPM\ \frac{D}{88}$

Figures 15.4, 5*, 6, 7, and 8

Class handouts

* see page 46

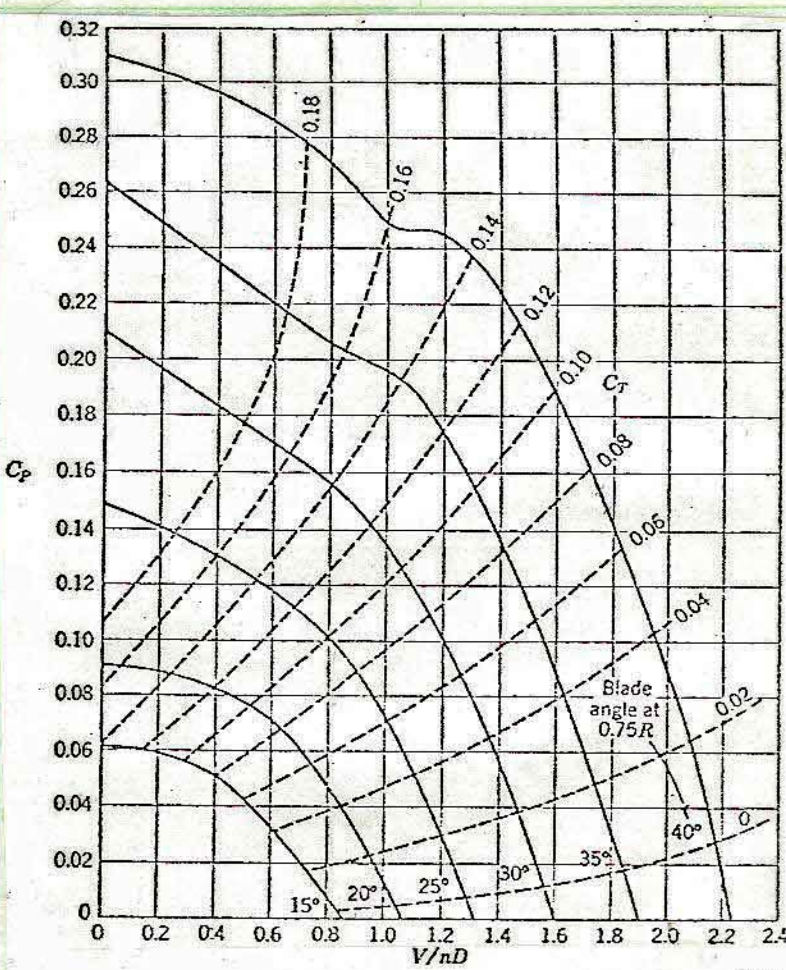


FIG. 15.4. Typical power coefficient curves.

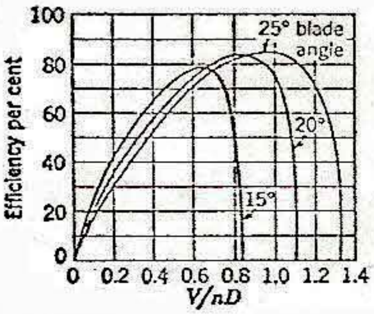


FIG. 15.6. Typical efficiency curves.

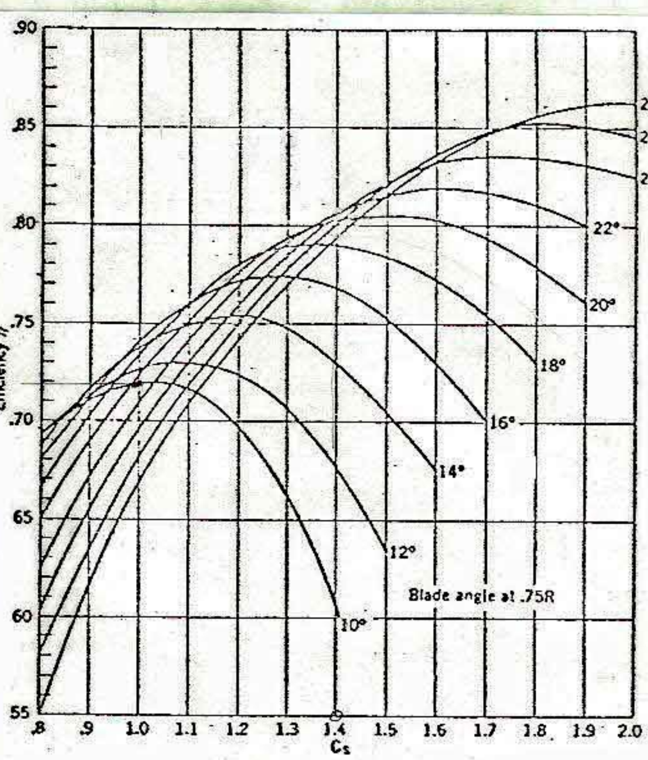


FIG. 15.7. Efficiency versus C_s for various blade angles.

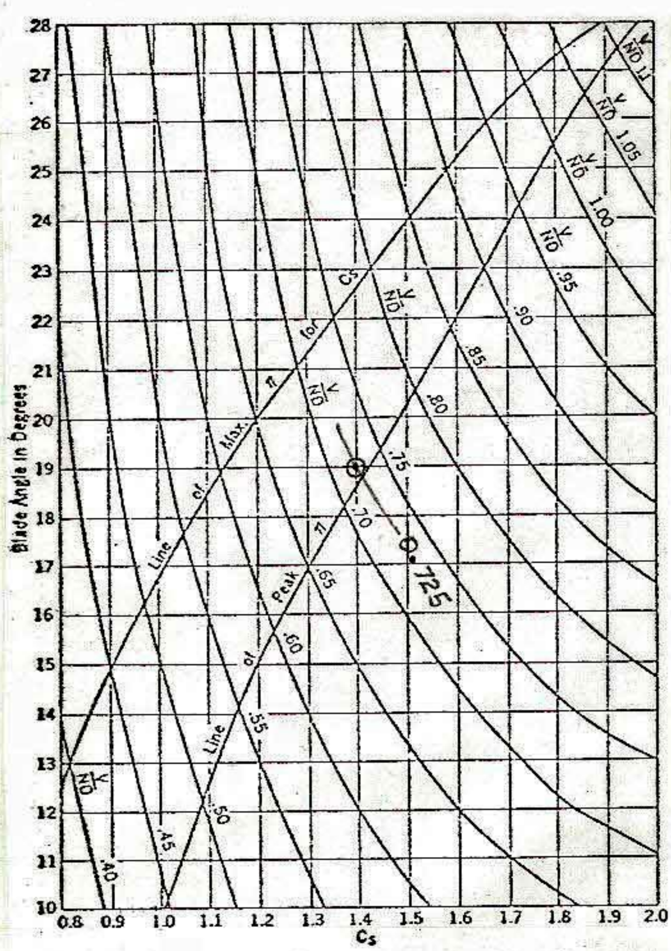


FIG. 15.8. C_s versus blade angle for various values of $V/(nD)$.

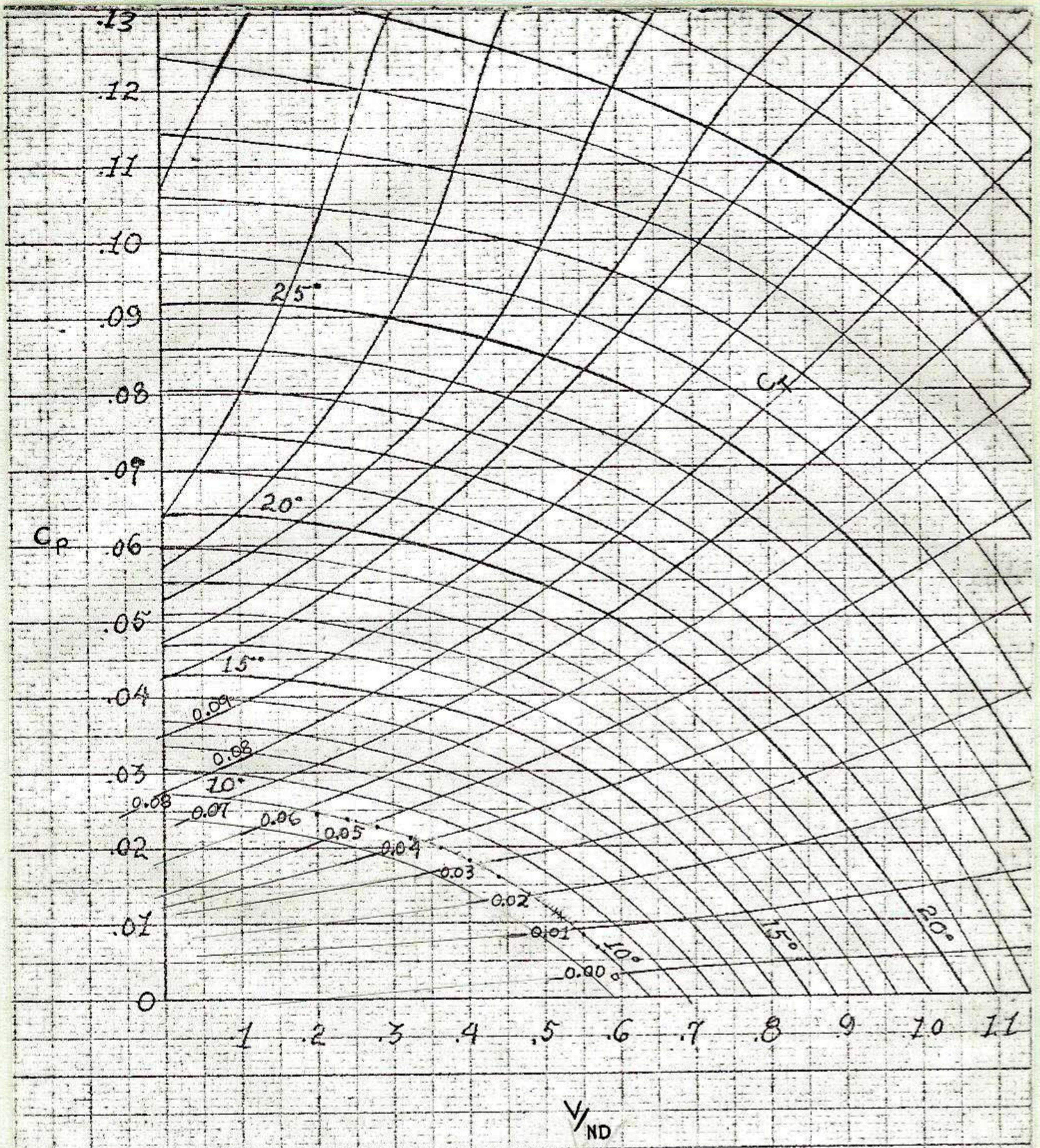


Figure 15.5 NACA Report 1943 Two Blade ΔF = 90

TABLE VIII AVAILABLE THRUST HORSEPOWER FOR THREE ALTITUDES

$\frac{V}{ND}$	C_T	C_P	C_{P_0}/C_{P_n}	$\frac{N_n}{N_0}$	RPM	$BHP = BHP_{S.L.}\sigma^{1.3}$			C_T/C_P	η	$THP_{available}$			V_{mph}
						Sealevel	3000′	9000′			Sealevel	3000′	9000′	
						1.0*	0.892*	0.700*			⊕	△	⊙	
0.57	0.006	0.0070	1.0	1.0	2400	30	26.8	21.0	0.859	0.489	14.69	13.1	10.25	88.6
0.55	0.009	0.0085	1.0	1.0	2400	30	26.8	21.0	1.059	0.582	17.45	15.6	12.22	85.5
0.52	0.013	0.0110	1.0	1.0	2400	30	26.8	21.0	1.180	0.614	18.40	16.4	12.90	80.8
0.50	0.017	0.0130	1.0	1.0	2400	30	26.8	21.0	1.415	0.654	19.60	17.5	13.70	77.8
0.48	0.020	0.0140	1.0	1.0	2400	30	26.8	21.0	1.430	0.685	20.60	18.4	14.38	74.6
0.44	0.026	0.0160	1.0	1.0	2400	30	26.8	21.0	1.625	0.715	21.40	19.2	15.06	68.4
0.40	0.031	0.0181	0.885	0.940	2260	28.2	25.2	19.7	1.712	0.685	19.30	17.2	13.50	58.5
0.36	0.038	0.0200	0.800	0.894	2140	26.8	24.0	18.8	1.900	0.665	17.80	16.0	12.50	56.0
0.25	0.052	0.0235	0.681	0.826	1980	24.8	22.2	17.3	2.210	0.553	13.70	12.3	9.55	38.8
0.00	0.084	0.0307	0.522	0.722	1730	21.6	19.3	15.2	2.740	0	0	0	0	0

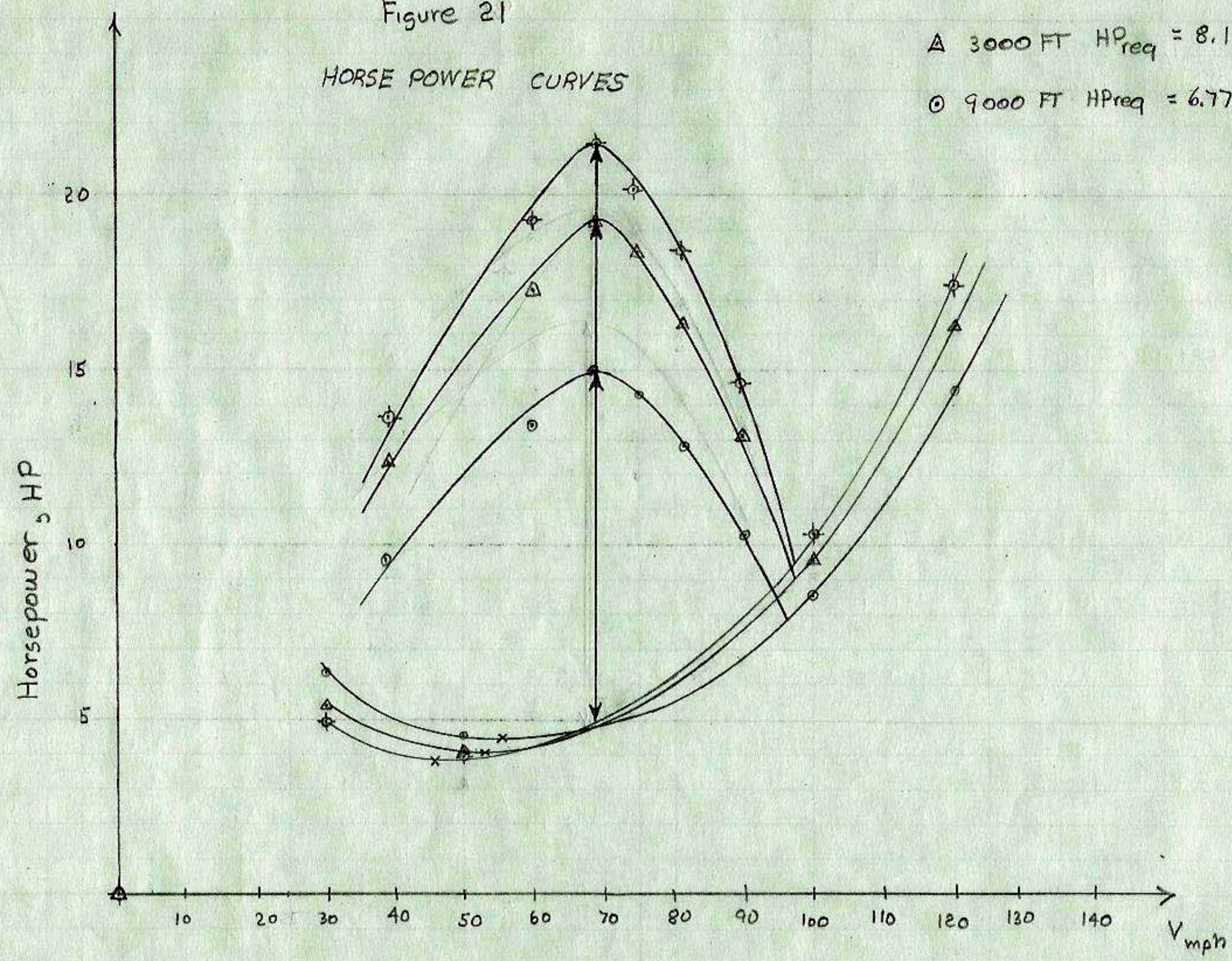
$\rho_{S.L.} = 0.0023769$

$\rho_{3000'} = 0.0021752$, $\left.\frac{V}{ND}\right|_0 = 0.44$, $^*\sigma^{1.3} = (\rho/\rho_{S.L.})^{1.3} = \begin{cases} 1.0 \text{ @ Sealevel} \\ 0.892 \text{ @ } 3000' \\ 0.700 \text{ @ } 9000' \end{cases}$

$\rho_{9000'} = 0.0018113$

Figure 21

HORSE POWER CURVES



Sea level $HP_{req} = 8.87 \times 10^{-6}\, V_{mph}^3 + 141.8 \frac{1}{V_{mph}}$

3000 FT $HP_{req} = 8.11 \times 10^{-6}\, V_{mph}^3 + 154.7 \frac{1}{V_{mph}}$

9000 FT $HPreq = 6.77 \times 10^{-6}\, V_{mph}^3 + 185.5 \frac{1}{V_{mph}}$

Rate of Climb (R.C.)

It would be desirable to design a system with more R.C. but the propeller curves would not allow it.

$$\text{Rate of Climb } \frac{FT}{min} = \frac{\left(HP_{avail} - HP_{req}\right) 33000}{W}$$

@ sea level R.C. = 624 FT/min

@ 3000 FT R.C. = 540 FT/min

@ 9000 FT R.C. = 383 FT/min

Climb corrected for acceleration reference

$$R.C._{corrected} = \frac{R.C.}{1 + \frac{V}{g}\frac{dV}{dh}}$$

troposhpere

$$1 + \frac{V}{g}\frac{dV}{dh} = 1 + 4.55\times10^{-7} V^2 (G)^{-1.235}$$

$V = 100$ FT/sec $= 1.00455$ ∴ negligible

c) Ceiling

Graphically the Absolute and Service Ceiling are determined.

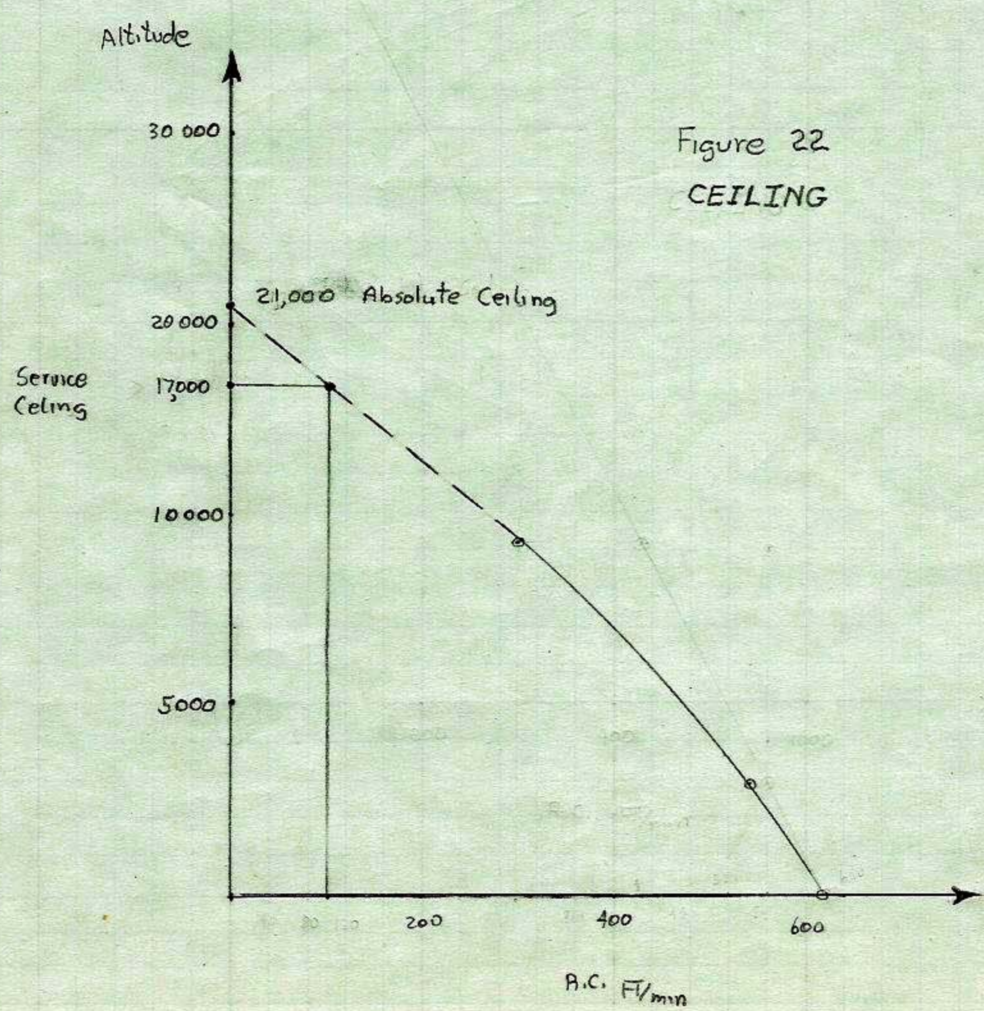


Figure 22
CEILING

d) Static Thrust

From reference 7 an equation for static thrust was derived

$$T_{static} = \left\{ \frac{\sqrt{A}\,\sigma}{21.4} \left[731\ HP - 2.53\ A\,B\,\bar{c}_{p_o} \frac{\bar{b}}{D} \frac{(\pi n D)^3}{10^4} \right\} ^{2/3}$$

$\sigma = \rho/\rho_o = 1.0$ for sea level

A = propeller disk area $= \pi(5.7/2)^2 = 25.5\ FT^2$

$\bar{b}$ = average blade chord $= 0.5$ FT

D = diameter $= 5.7$ FT

n = rev/sec $= 40$/sec

HP = horspower $= 30$ HP

B = number of blades $= 2$

$\bar{c}_{D_o} = 0.020$ (from whirl test conducted by USAF) ref. 7

$$T_{static} = \left\{ \frac{\sqrt{25.5}}{22.4} \left[731(30) - 2.53(25.5)2(0.02)\frac{0.5}{5.7} \frac{(3.1416(40)5.7)^3}{10^4} \right\}^{2/3}$$

$$= \left\{ 0.225 \left[21930 - 7750 \right] \right\}^{2/3}$$

$$= \left\{ 3180 \right\}^{2/3}$$

$$T_{static} = 200\ \text{lbs.}$$

THRUST AT ANY VELØCITY

$$T = T_{static} - a V^2 \qquad V_{TO} = 1.15\sqrt{\frac{W841}{S C_{L_{max}}}} = 74.9\ FT/sec$$

given
$V = V_{TO}$
$N = 40$ rev/sec
$D = 5.7$ FT

calculate $V/ND = 0.328 \quad \beta = 10°$

Figure 15.4 $\quad C_T = 0.041$

$$T = C_T \rho N^2 D^4 = 0.041(0.002377)(40)^2(5.7)^4 = 164\ \text{lbs.}$$

$$a = -(T - T_{static})/(V_{TO})^2 = +0.00644$$

$$T = 200 - 0.00644V^2$$

e) Takeoff Distance

From reference 10

$$\text{Take off distance} = -\frac{1}{2B} \ln\left(1 - \frac{B}{A} V_{TO}^2\right)$$

$$A = g\left(\frac{T_{static}}{W} - \mu\right)$$

$$B = \frac{g}{W}\left\{\frac{\rho S}{2}\left(C_{D_0} - \frac{\mu^2}{4K}\right) + a\right\}$$

given

W = Aircraft weight = 878.87

S = Wing area = 110 FT^2

V_{TO} = Aircraft take off speed = $1.15\,V_{stall}$ = 74.9 Ft/sec

C_{D_0} = Drag Coefficient at zero lift =

$K = 1/e\pi A$ = 0.019343

μ = Coefficient of ground friction = 0.025

T_{static} = Propeller thrust at $V=0$ = 200

a = constant = 0.00644

$$A = 32.2\left(\frac{200}{878.87} - 0.12\right) = 3.48$$

$$B = \frac{32.2}{878.87}\left\{\frac{0.002377(110)}{2}\left[(0.0118) - \frac{(0.025)^2}{4(0.01934)}\right] + 0.00644\right\}$$

$$B = 0.0367\left\{0.131(0.0037) + 0.00644\right\} = 0.0367\left\{0.006925\right\}$$

$$B = 0.000254$$

$$\text{Take of distance} = -\frac{1}{0.000508} \ln\left[1 - 0.000073(5600)\right]$$

$$= -1970 \ln(0.608) = -1970(-0.5)$$

ground run

Take of distance = 985 Feet

$S = 110\ FT^2 \quad \Lambda = 22$

$C_{D_{o\ wing}} = 0.0055$

M.A.C. = 34.146 in.

$C_{D_o} = 0.0118$

$K = 0.01934$

$\theta = 1.728^\circ$

$C_{L@V_{Z_{min}}} = 1.351 \qquad C_{L_{cruise}} = 0.650$

$C_{L@L/D_{max}} = 0.780$

<u>Wing Loading</u>

Maximum Weight = 750.65 lbs $\qquad \frac{W_{max}}{S} = 6.826$

Minimum Weight = 670.65 lbs $\qquad \frac{W_{min}}{S} = 6.099$

Landing distance ground run

Without brakes = 726.2 Feet
With brakes = 307.4 Feet

$V_{Z_{min}} = 2.27\ FT/sec$

$V@V_{Z_{min}} = 65.158\ FT/sec$

$L/D_{max} = 33.138$

$V_{@L/D_{max}} = 85.753\ FT/sec$

$V_{stall} = 38$ Knots

$V_g = 159$ mph

$V_{Dive} = 191$ mph

$V_{tow} = 91.3$ mph

With Engine

30 H.P.
2400 rpm

$W_{total} = 878.87$ lbs.

$V_{prop\ design} = 68.5$ mph

$\eta_{prop} = 0.719$

$\beta_{@0.75R} = 10°$

$D_{prop} = 5.7$

$R.C._{S.L.} = 624$ FT/min

$R.C._{3000'} = 540$ FT/min

$R.C._{9000'} = 383$ FT/min

Service Ceiling = 21,000 Ft

static Thrust = 200 lbs

Ground Run Take of distance = 985 Feet

$V_{cruise} = 68.5$ mph

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Appendix A

PERFORMANCE EQUATION DERIVATIONS

Concept: Power Balance

$$DV = W V_z$$

Drag · Velocity = Weight · Sink Velocity

lb · Ft/sec [=] lb · Ft/sec

$$\begin{Bmatrix} \text{Power Loss} \\ \text{moving Forward} \end{Bmatrix} = \begin{Bmatrix} \text{Power Loss} \\ \text{moving Vertical} \end{Bmatrix}$$

For steady state cruise flight

Cruise Flight

From $DV = WV_Z$, $V_Z = \frac{D}{W}V = \frac{C_D \frac{1}{2}\rho V^2 S}{C_L \frac{1}{2}\rho V^2 S}\underbrace{\sqrt{\frac{1}{C_L}}\sqrt{\frac{2W}{\rho S}}}_{V}$

$$\boxed{V_Z = \frac{C_D}{C_L^{1.5}}\sqrt{\frac{2W}{\rho S}}} \quad (1)$$

$$WV_Z = VD = V\left(C_{D_0} + \frac{C_L^2}{e\pi A}\right)\frac{1}{2}\rho V^2 S = \frac{\rho S C_{D_0}}{2}V^3 + \frac{C_L^2 \rho S}{2e\pi A}V^3$$

$$C_L^2 = \frac{W^2}{\frac{1}{4}\rho^2 V^4 S^2}$$

$$WV_Z = \frac{\rho S C_{D_0}}{2}V^3 + \frac{2W^2}{e\pi A\rho S}\frac{1}{V}$$

$$V_Z = \frac{\rho S C_{D_0}}{2W}V^3 + \frac{2W}{e\pi A\rho S}\frac{1}{V}$$

$$\frac{dV_Z}{dV} = 0$$

$$\frac{3}{2}\frac{\rho S C_{D_0}}{2W}V_{\substack{min\\sink}}^2 - \frac{2W}{e\pi A\rho S}\frac{1}{V_{\substack{min\\sink}}^2} = 0$$

$$V_{\substack{min\\sink}}^4 = \frac{4W^2}{3e\pi A\rho^2 S^2 C_{D_0}}$$

$$\boxed{V_{\substack{min\\sink}} = \sqrt{\frac{2W}{\rho S\sqrt{3C_{D_0}e\pi A}}}} \quad (2)$$

$$C_L @ V_{\substack{min\\sink}} = \frac{W}{\frac{1}{2}\rho V_{\substack{min\\sink}}^2 S} = \frac{\cancel{W}}{\frac{1}{\cancel{2}}\cancel{\rho}\frac{\cancel{2W}}{\cancel{\rho}\cancel{S}\sqrt{3C_{D_0}e\pi A}}\cancel{S}}$$

$$\boxed{C_L @ V_{\substack{min\\sink}} = \sqrt{3C_{D_0}e\pi A}} \quad (3)$$

$$V_{Z_{\substack{min\\sink}}} = \frac{\rho S C_{D_0}}{2W}V_{\substack{min\\sink}}^3 + \frac{2W}{e\pi A\rho S}\frac{1}{V_{\substack{min\\sink}}}$$

$$= \frac{\rho S C_{D_0}}{2W}\left(\frac{2W}{\rho S\sqrt{3C_{D_0}e\pi A}}\right)^{3/2} + \frac{2W}{e\pi A\rho S}\frac{1}{\left[\frac{2W}{\rho S(3C_{D_0}e\pi A)^{1/2}}\right]^{1/2}}$$

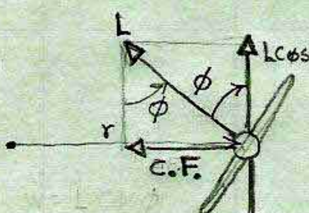
$$V_{z_{\substack{min\\sink}}} = C_{D_0}\sqrt{\frac{2W}{\rho S}}\;\frac{1}{(3C_{D_0}e\pi A)^{0.75}} + \frac{3}{3}\sqrt{\frac{2W}{\rho S}}\;\frac{1}{\frac{1}{C_{D_0}^{0.25}}\frac{e\pi A}{(3e\pi A)^{0.25}}}$$

$$= C_{D_0}\sqrt{\frac{2W}{\rho S}}\;\frac{1}{C_{D_0}^{0.75}(3e\pi A)^{0.75}} + 3C_{D_0}^{0.25}\sqrt{\frac{2W}{\rho S}}\;\frac{1}{\frac{3e\pi A}{(3e\pi A)^{0.25}}}$$

$$= C_{D_0}^{0.25}\sqrt{\frac{2W}{\rho S}}\;\frac{1}{(3e\pi A)^{0.75}} + 3C_{D_0}^{0.25}\sqrt{\frac{2W}{\rho S}}\;\frac{1}{(3e\pi A)^{0.75}}$$

$$\boxed{V_{z_{\substack{min\\sink}}} = \frac{4C_{D_0}^{0.25}}{(3e\pi A)^{0.75}}\sqrt{\frac{2W}{\rho S}}} \quad (4)$$

Turning Flight



Equilibrium, $W = L\cos\theta$

C.F. ≡ Centripetal Force

$$\tan\phi = \frac{C.F.}{L\cos\phi} = \frac{\frac{W}{g}\frac{V^2}{r}}{C_L\frac{1}{2}\rho V^2 S\cos\phi}$$

$$C_L = \frac{W}{S}\,\frac{2}{\rho g}\,\frac{1}{r\sin\phi}$$

From $DV = WV_z$

$$V_z = \frac{D}{W}V = \frac{D}{L\cos\phi}V$$

$$L = C_L\frac{1}{2}\rho V^2 S$$

$$V = \sqrt{\frac{2L}{\rho S}} = \sqrt{\frac{2W}{\rho S\cos\phi}}$$

$$\boxed{V_z = \frac{C_D}{C_L^{1.5}}\sqrt{\frac{2W}{\rho S}}\;\frac{1}{\cos\phi^{1.5}}} \quad (1)$$

$$WV_z = VD = V\left(C_{D_0} + \frac{C_L^2}{e\pi A}\right)\frac{1}{2}\rho V^2 S = \frac{\rho S C_{D_0}}{2}V^3 + \frac{C_L^2\rho S}{2e\pi A}V^3$$

$$C_L^2 = \frac{L^2}{\frac{1}{4}\rho^2V^4S^2} = \frac{4W^2}{\rho^2V^4S^2\cos^2\phi}$$

$$WV_z = \frac{\rho S C_{D_0}}{2}V^3 + \frac{2W^2}{e\pi A\rho S}\,\frac{1}{\cos^2\phi}\,\frac{1}{V}$$

$$V_Z = \frac{\rho S C_{D_0}}{2W} V^3 + \frac{2W}{e\pi A\rho S} \frac{1}{\cos^2\phi} \frac{1}{V}$$

$$\frac{dV_Z}{dV} = 0$$

$$\frac{3}{2} \frac{\rho S C_{D_0}}{W} V^2_{\substack{min\\sink}} - \frac{2W}{e\pi A\rho S} \frac{1}{\cos^2\phi} \frac{1}{V^2_{\substack{min\\sink}}} = 0$$

$$V^4_{\substack{min\\sink}} = \frac{4W^2}{3 e\pi A\rho^2 S^2 \cos^2\phi \, C_{D_0}}$$

$$\boxed{V_{\substack{min\\sink}} = \sqrt{\frac{2W}{\rho S \cos\phi \sqrt{3C_{D_0} e\pi A}}}} \quad (2)$$

$$V_{Z_{\substack{min\\sink}}} = \frac{\rho S C_{D_0}}{2W} V^3_{\substack{min\\sink}} + \frac{2W}{e\pi A\rho S} \frac{1}{\cos^2\phi} \frac{1}{V_{\substack{min\\sink}}}$$

$$\boxed{V_{Z_{\substack{min\\sink}}} = \frac{4 C_{D_0}^{0.25}}{(3e\pi A)^{0.75}} \sqrt{\frac{2W}{\rho S}} \frac{1}{\cos\phi^{1.5}}} \quad (3)$$

$$C_{L_{\substack{min\\sink}}} = \frac{L}{\frac{1}{2}\rho V^2_{\substack{min\\sink}} S} = \frac{\cancel{W}/\cancel{\cos\phi}}{\frac{1}{\cancel{2}} \cancel{\rho} \frac{\cancel{2}\cancel{W}}{\cancel{\rho}\cancel{S}\cancel{\cos\phi}\sqrt{3C_{D_0} e\pi A}} \cancel{S}}$$

$$\boxed{C_{L_{\substack{min\\sink}}} = \sqrt{3C_{D_0} e\pi A}} \quad (4)$$

$$\sqrt{3C_{D_0} e\pi A} = \frac{W}{S} \frac{2}{\rho g} \frac{1}{r \sin\phi}$$

$$\phi = \sin^{-1}\left[\frac{W}{S} \frac{2}{\rho g} \frac{1}{r\sqrt{3C_{D_0} e\pi A}}\right]$$

$$V_{Z_{min}} f(r) = \frac{4 C_{D_0}^{0.25}}{(3e\pi A)^{0.75}} \sqrt{\frac{2W}{\rho S}} \frac{1}{\left\{\cos\left[\sin^{-1}\left(\frac{W}{S}\frac{2}{\rho g}\frac{1}{r\sqrt{3C_{D_0} e\pi A}}\right)\right]\right\}^{1.5}}$$

Appendix B

COMPUTER PROGRAMS

(Compucorp Calculator)
-1973-

$$V_{z_{min}f(r)} = \sqrt{\frac{32\, W\sqrt{C_{D_0}}}{\rho S (3 e \pi A)^{1.5}}}\; \frac{1}{\left\{\cos\left[\sin^{-1}\left(\frac{1}{r}\,\frac{2W}{S\rho g\sqrt{3C_{D_0} e\pi A}}\right)\right]\right\}^{1.5}}$$

Program No. 5 : 4 punched cards
Compucorp Calculator

RESET
GΦ TΦ (0)
LΦAD Latch

	RESET	024
	HALT	401
W	ST_1	440
	HALT	401
ρ	ST_2	441
	HALT	401
S	ST_3	442
	HALT	401
C_{D_0}	ST_4	443
	HALT	401
A	ST_5	444
	HALT	401
e	ST_6	445
	3	003
	×	070
	RCL_6	465
	×	070
	π	017
	×	070
	RCL_5	464
	a^x	074
	1	001
	.	012
	5	005
	=	020
	ST_0	457
	RCL_4	463
	$\sqrt{\ }$	056
	×	070
	3	003
	2	002
	×	070
	RCL_1	460
	÷	072
	RCL_2	461
	÷	072
	RCL_3	462
	÷	072
	RCL_0	477
	=	020
	$\sqrt{\ }$	056
	ST_0	457
	3	003
	×	070
	RCL_4	463
	×	070
	RCL_6	465
	×	070

	π	017
	×	070
	RCL_5	464
	=	020
	$\sqrt{\ }$	056
	ST_6	445
	2	002
	×	070
	RCL_1	460
	÷	072
	RCL_3	462
	÷	072
	RCL_2	461
	÷	072
	3	003
	2	002
	.	012
	2	002
	÷	072
	RCL_6	465
	=	020
	ST_6	445
	HALT	401
V	ST_1	440
	HALT	401
R	ST_2	441
	GΦ TΦ (6)	746

RESET
GΦ TΦ (6)
LΦAD Latch

	RESET	024
	HALT	401
r	ST_3	442
	$1/x$	054
	×	070
	RCL_6	465
	=	020
	$\sin^{-1}$	042
	=	020
	$R\rightarrow^{\circ}$	046
	=	020
	Cos	040
	a^x	074
	1	001
	.	012
	5	005
	=	020

	$1/x$	054
	×	070
	RCL_0	477
	=	020
V_z	PRINT ANS	027
	÷	072
	RCL_1	460
	=	020
$\frac{V_z}{V}$	PRINT ANS	027
	RCL_3	462
	÷	072
	RCL_2	461
	=	020
$\frac{r}{R}$	PRINT ANS	027
	GΦ TΦ 6	746

$$V_{z_{min}(r)} = \frac{4C_{D_0}^{0.25}}{(3e\pi A)^{0.75}}\sqrt{\frac{2W}{\rho S}}\frac{1}{\left\{\cos\left[\sin^{-1}\left(\frac{2W}{\rho S}\frac{1}{g}\frac{1}{r\sqrt{3C_{D_0}e\pi A}}\right)\right]\right\}^{1.5}}$$

$r = 200$

$W = 750.87$	0	ΔR
$\rho = 0.002377$	1	R_1
$S = 110$	2	R_n
$C_{D_0} = 0.0118$	3	
$A = 22$	4	
$e = 0.748$	5	
$2W/\rho S$	6	
$3e\pi A$	7	
$(3e\pi A)^{0.75}$	8	$V_{z_{min}}$
$\frac{4C_{D_0}^{0.25}}{(3e\pi A)^{0.75}}\sqrt{\frac{2W}{\rho S}}$	9	

$$= \frac{1.31835}{(1.55094\ 10^3)^{0.75}}\sqrt{5.74345\ 10^3}\frac{1}{\left\{\cos\left[\sin^{-1}\left(\frac{5.74345\ 10^3}{32.2}\frac{1}{200\sqrt{1.83011}}\right)\right]\right\}^{1.5}}$$

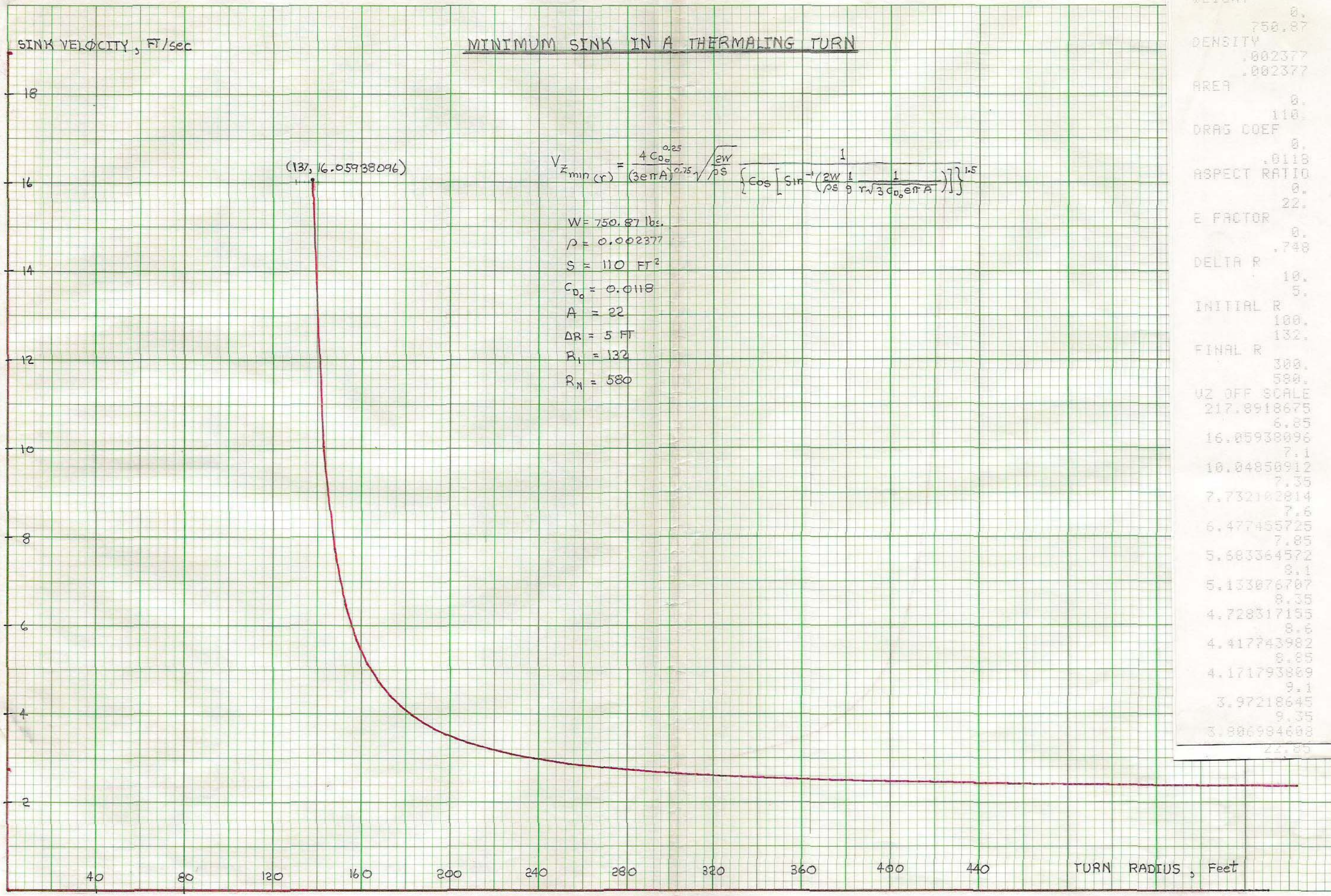
$$= \frac{1.31835}{4.39488\ 10^1}(7.57836\ 10^1)\frac{1}{\left\{\cos\left[\sin^{-1}\left(1.78368\ 10^2\frac{1}{2.70563\ 10^2}\right)\right]\right\}^{1.5}}$$

$$= 2.27337\frac{1}{\{\cos[\sin^{-1}(0.659248)]\}^{1.5}} = 7.18905\ 10^1\frac{1}{\{\cos[4.12425]\}^{1.5}} = \frac{7.18905\ 10^1}{\{7.51926\ 10^{-1}\}^{1.5}}$$

$$= \frac{2.27337}{6.52023\ 10^{-1}} = 3.48665$$

```
Alpha  WEIGHT 0 PRNT STØP = K 0 PRNT CLR DSP
  "    DENSITY 0.002377 PRNT STØP = K 1 PRNT DSP
  "    AREA 0 PRNT STØP = K 2 PRNT CLR DSP
  "    DRAG CØEF 0 PRNT STØP = K 3 PRNT CLR DSP
  "    ASPECT RATIØ 0 PRNT STØP = K 4 PRNT CLR DSP
  "    E FACTØR 0 PRNT STØP = K 5 PRNT CLR DSP
2 * K0 ÷ K1 ÷ K2 = K6 CLR DSP
3 * K5 * π * K4 = K7 CLR DSP
K7 |x|^a 0.75 = K8 CLR DSP
K3 |x|^a 0.25 * 4 ÷ K8 * K6 √x = K9 CLR DSP
Alpha  DELTA R 10 PRNT STØP = K0 PRNT CLR DSP
  "    INITIAL R 100 PRNT STØP = K1 PRNT CLR DSP
  "    FINAL R 300 PRNT STØP = K2 PRNT CLR DSP
→ K7 * K3) √x * K1) 1/x * K6 ÷ 32.2) ARC SIN) CØS) |x|^a 1.5) 1/x * K9 = K8 CLR DSP
K8 - 10)
YES | IF Ø ≥ 1   NØ
    ↓ K1 + K0 = K1     ← insert ALPHA VZ ØFF SCALE K8 PRNT
— GØ TØ
CØNTINUE ←
K1 PRNT ÷ 20) RM31 K8) PRNT RM32        ← RM33
K2 - K1)
YES | IF Ø ≥ 1   NØ
    ↓ K1 + KØ = K1
— GØ TØ
CØNTINUE ←
RESET
```

Plotter
X axis scale factor 1" = 20 FT



WEIGHT
0.
750.87
DENSITY
.002377
.002377
AREA
0.
110.
DRAG COEF
0.
.0118
ASPECT RATIO
0.
22.
E FACTOR
0.
.748
DELTA R
10.
5.
INITIAL R
100.
132.
FINAL R
300.
580.
VZ OFF SCALE
217.8918675
6.85
16.05938096
7.1
10.04850912
7.35
7.732182814
7.6
6.477455725
7.85
5.683364572
8.1
5.133076707
8.35
4.728317155
8.6
4.417743982
8.85
4.171793889
9.1
3.97218645
9.35
3.806984608
22.85

Calculate Fig. 18 and 19 curves

$$1\ \frac{FT}{sec}\left(\frac{FT}{sec}\ \frac{60\,sec}{min}\ \frac{60\,min}{Hr}\ \frac{m}{5280\,FT}\right) \rightarrow \frac{45}{66}\ MPH$$

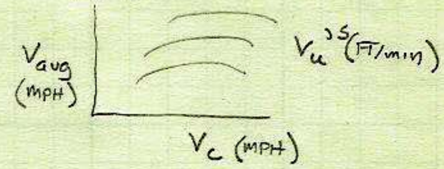
$$V_{Z(V_c)} = \frac{\overset{⑤}{V_c}{}^4 + \overset{①}{54074137.98}}{{}^{②}486733.09\,V_{c\,⑤}}$$

④ $V_{Z\,min} = 2.2733713$ FT/sec

$V_u = 0, 100, 200, 400, 800$ FT/min

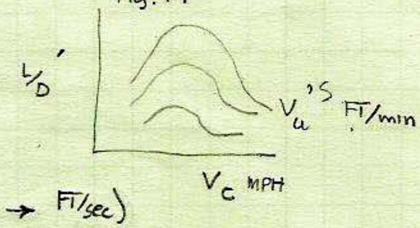
③ $0, 1\frac{2}{3}, 3\frac{1}{3}, 6\frac{2}{3}, 13\frac{1}{3}$ FT/sec

Fig. 18



$$V_{aug(V_c)} = \frac{{}^{⑤}V_c}{1 + \dfrac{{}^{⑤}V_Z + 0.2(\overset{③}{V_u} + \overset{④}{V_{Z\,min}})}{{}_{③}V_u}}$$

Fig. 19



$$L/D' = \frac{{}^{⑤}V_c}{{}_{⑤}V_Z + 0.2(\underset{③}{V_u} + \underset{④}{V_{Z\,min}})}$$

$V_u \rightarrow$ store (FT/sec) in Reg. K_{in} ③ $\left(\frac{FT}{min}/60 \rightarrow FT/sec\right)$

$V_c \rightarrow$ store (FT/sec) in Reg K_{in} ⑤ $\left(mph\ \frac{66}{45} \rightarrow FT/sec\right)$

<u>L/D'</u>

ENT 30 × 66 ÷ 45 = K_{in} ⑤ inv x^y 4 + K_{out} ① ≒ ÷ K_{out} ② ÷ K_{out} ⑤ ≒

+ (K_{out} ③ + K_{out} ④) × .2 = inv $\frac{1}{x}$ × K_{out} ⑤ = L/D'

<u>Vaug</u>

+ (K_{out} ③ + K_{out} ④) × .2 ≒ ÷ K_{out} ③ + 1 = inv $\frac{1}{x}$ × K_{out} ⑤ × 45 ÷ 66 =

$$V_{aug} = \frac{V_c}{1 + V_Z'/V_u}$$

where $V_Z' = V_Z + 0.2(V_u + V_{Z\,min})$

and

$V_Z = V_c / L/D$

and

$$L/D = \frac{\dfrac{2W}{\rho S C_{D0}}\,V_c^2}{V_c^4 + \dfrac{4W^2}{C_{D_0}\rho^2 S^2 \pi A C}}$$

PROGRAM: AIRFØIL DATA @ $A.R._0 = \infty$ CHANGED FØR $A.R._1$ CARD 1 OF 2 No. 000

— INSTRUCTION —

RESET
GØ TØ (0)
LØAD
Input
$C_{L_{A.R.\infty}}$ St_6 08:24
$A.R._1$ St_5
$\alpha_{A.R.\infty}$ 0.318
$C_{D_{A.R.\infty}}$
Output
$\alpha_{A.R._1}$
$C_{D_{A.R._1}}$
C_N C_c
LØAD unlatch

VERIFY 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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PROGRAM: AIR FØIL DATA @ $A.R._0 = \infty$ CHANGED FØR $A.R._1$ CARD 2 OF 2 No. 000

— INSTRUCTION —

RESET
GØ TØ (0)
LØAD
Input
$C_{L_{A.R.\infty}}$
$A.R._1$
$\alpha_{A.R.\infty}$
$C_{D_{A.R.\infty}}$
Output
$\alpha_{A.R._1}$ $C_{D_{A.R._1}}$ C_N C_c
LØAD unlatch

VERIFY 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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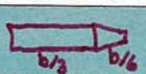
PROGRAM: 1 M.A.C.



CARD 1 OF 2 No. 1

— INSTRUCTION —

Input
C_R
C_t
$b/2$
output
$\bar{C}_2$
$\bar{C}$ = M.A.C.

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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PROGRAM: 1 M.A.C.

CARD 2 OF 2 No. 1

— INSTRUCTION —

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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PROGRAM: 2 V_z + L/D = f(V) CARD 1 OF 3 No. 2

— INSTRUCTION —

RESET
GØ TØ (0)
LØAD Latch
input
W V
ρ
S
C_{D_0}
A
e
output Top
Bottom
L/D
V_z

VERIFY 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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PROGRAM: 2 V_z + L/D = f(V) CARD 2 OF 3 No. 2

— INSTRUCTION —

LØAD unlatch

VERIFY 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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PROGRAM: 2 V_z + L/D = f(V) CARD 3 OF 3 No. 2

— INSTRUCTION —

RESET
GØ TØ (6)
LØAD Latch

LØAD unlatch

VERIFY 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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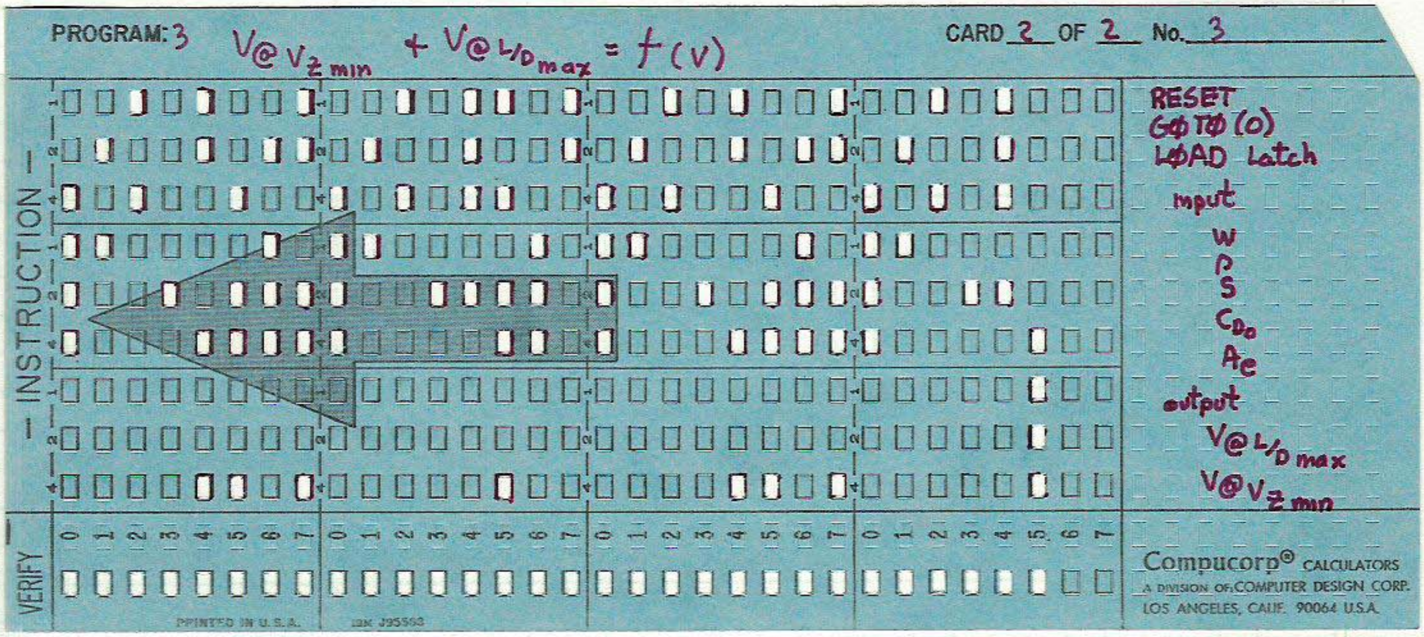

PROGRAM: 3 $V@V_{Z_{min}} + V@_{L/D_{max}} = f(V)$ CARD 2 OF 2 No. 3

RESET
GØTØ (0)
LØAD Latch
input
W
ρ
S
C_{D_0}
A_e
output
$V@_{L/D\,max}$
$V@V_{Z_{min}}$

— INSTRUCTION —

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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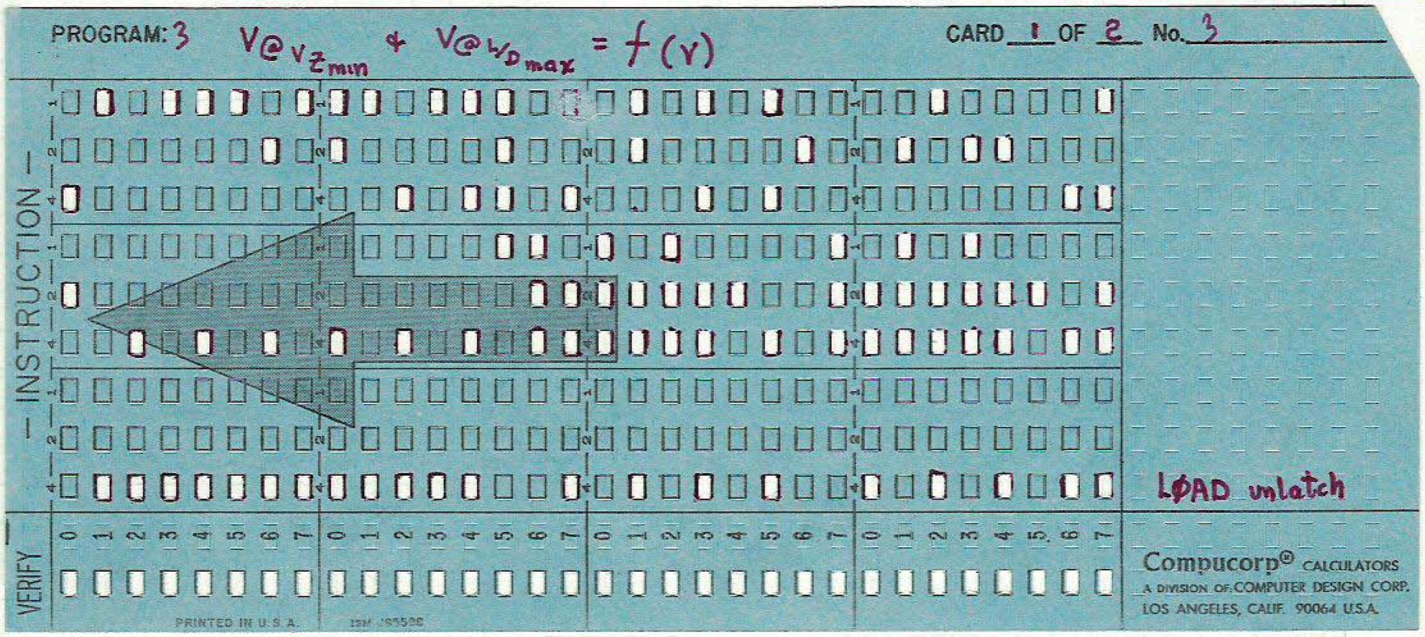

PROGRAM: 3 $V@V_{Z_{min}}$ & $V@_{L/D_{max}} = f(V)$ CARD 1 OF 2 No. 3

LØAD unlatch

— INSTRUCTION —

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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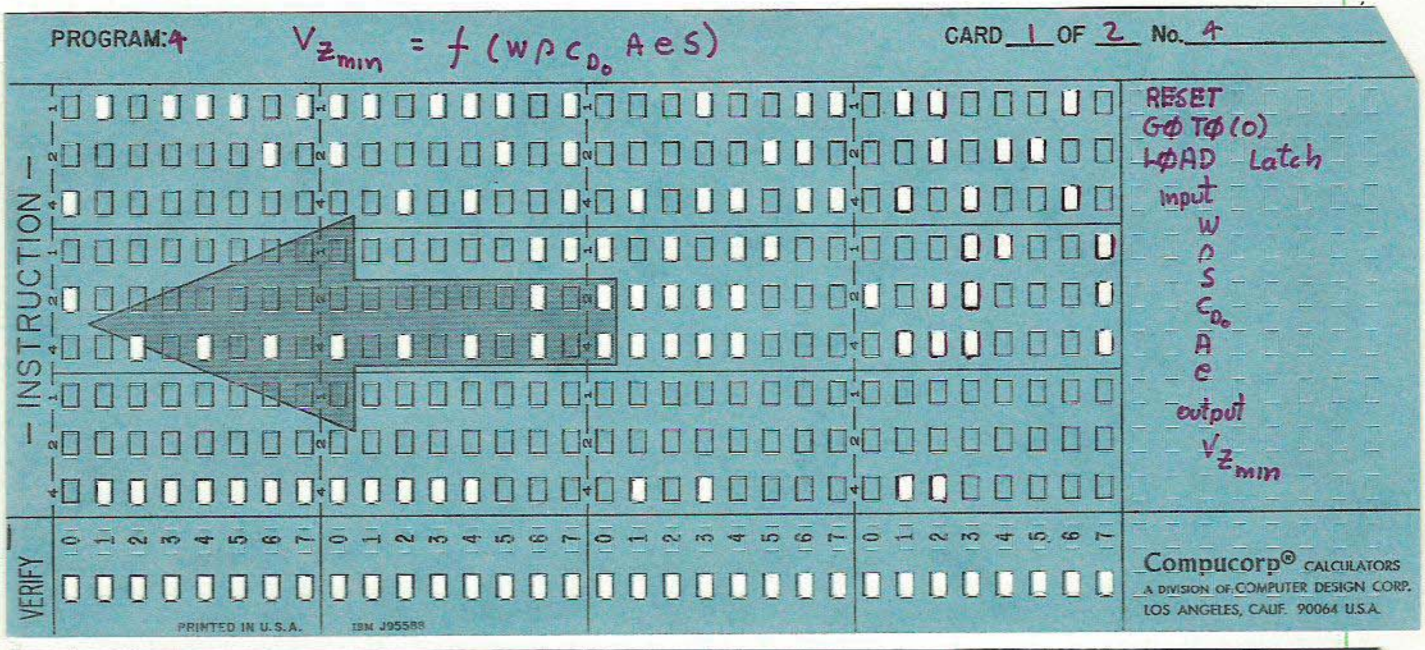
PROGRAM: 4
$V_{z_{min}} = f(w \rho C_{D_0} A e S)$
CARD 1 OF 2 No. 4
INSTRUCTION
RESET
GØ TØ (0)
LØAD Latch
input
w
ρ
S
C_{D_0}
A
e
output
$V_{z_{min}}$
VERIFY
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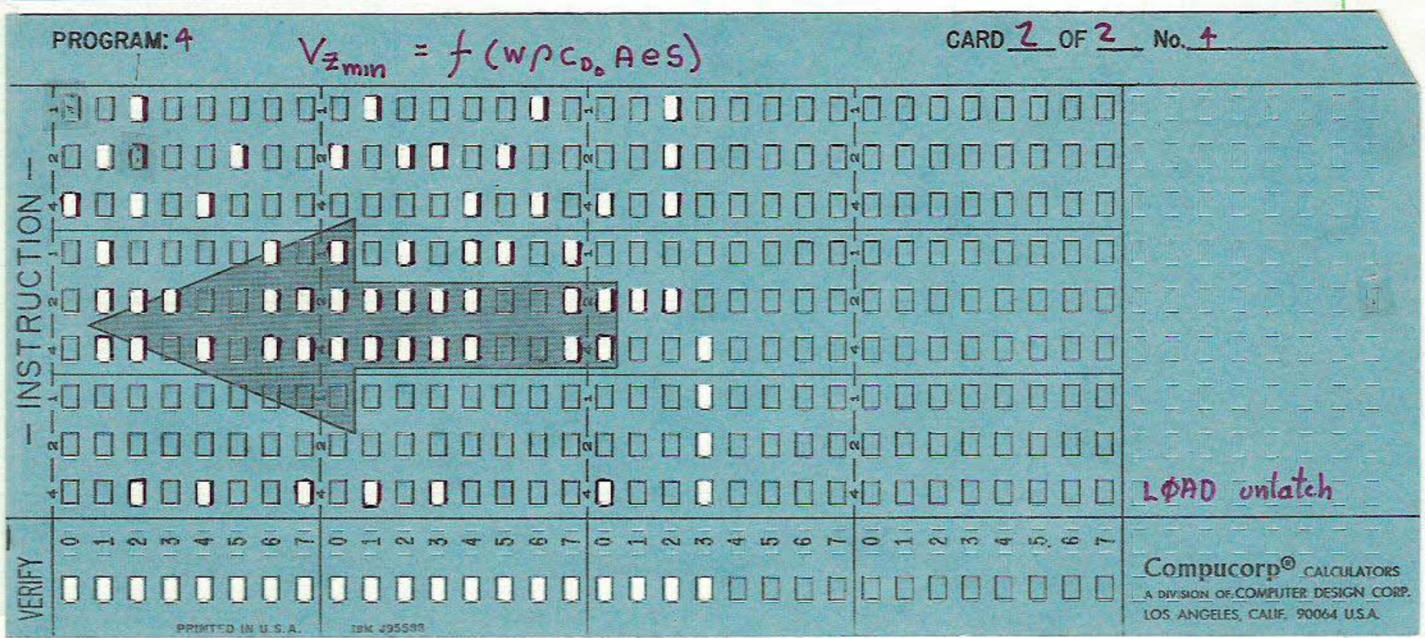
PROGRAM: 4
$V_{z_{min}} = f(w \rho C_{D_0} A e S)$
CARD 2 OF 2 No. 4
INSTRUCTION
LØAD unlatch
VERIFY
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PROGRAM: 5 $V_{z\,min}$ $f(r)$

CARD 1 OF 4 No. 5

— INSTRUCTION —

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

RESET
GØ TØ (0)
LØAD Latch
Input
→ W → V
→ ρ
→ S → R
→ C_{D_0}
→ A
→ e
output
None

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PROGRAM: 5 $V_{z\,min}$ $f(r)$

CARD 2 OF 4 No. 5

— INSTRUCTION —

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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PROGRAM: 5 $V_{z\,min}$ $f(r)$

CARD 3 OF 4 No. 5

— INSTRUCTION —

LØAD Latch

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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PROGRAM: 5 $V_{z_{min}}$ $f_{(r)}$

CARD 4 OF 4 No. 5

— INSTRUCTION —

RESET
GØ TØ (6)
LØAD Latch
Input
→ r
Output
V_z →
V_z/V →
r/R →
[redacted] unlatch

VERIFY

0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7 0 1 2 3 4 5 6 7

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