PRELIMINARY DESIGN OF AN FAI CLASS I AIRPLANE AND PLANS FOR ESTABLISHING INTERNATIONAL DISTANCE RECORDS

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INTRODUCTION

The purpose of this paper is to outline very briefly the basic design concepts of an airplane suitable for establishing a non-stop distance record for aircraft of gross weight less than 1102 lbs. (FAI Class 1). The present record of 1766 statute miles was established on July 10, 1957 by a Finnish Heinonen HK-1 during a 17 hour flight from Madrid, Spain, to Turku, Finland.

A survey of existing aircraft reveals that none of the present production models are suitable for efficient long-range flights. The small size of the airplane required for this undertaking makes it a very satisfactory subject for a home-built project, and the author of this paper hopes to actually build the airplane described herein and then with FAI sanction fly it as far as it will go for record purposes. The design requirements are quite simple: the airplane must take-off, carry a man as far as possible, land safely, and the take-off gross weight must not exceed 1102 lbs.

AIRPLANE GENERAL ARRANGEMENT

In the case of a long-range airplane of limited gross weight, it is immediately obvious without recourse to lengthy theory or equations that the more fuel carried, the greater the range and it follows that minimum empty weight is a more serious requirement than usual. A study of long-range airplanes such as the "Spirit of St. Louis," the Russian ANT-25, and the modified Piper Comanche showed that the empty weight of the airplane ranges from 29% to 50% of the gross weight with an average value of 37%. Thus, for a gross weight of 1102 lbs. an empty weight of about 410 lbs. can be expected.

The Piper J-3, commonly regarded as a small airplane, has an empty weight of 700 lbs; the smallest American production airplane, the Mooney M-18L weighs 550 lbs. empty. However, the Goodyear midget racers, almost all of which had an empty weight of 500 lbs. due to a minimum weight rule, offer encouragement that lighter airplanes can be built. With relaxation of the 500 lbs. minimum rule in 1958 several of these airplanes were pared down to 465 lbs. Briefly, the Goodyear Midgets are small aircraft powered by a 110-hp modified Continental C-85 engine. The midgets are high-g, high-speed aircraft capable of approximately 240 mph in straight and level flight. They must prove their structural integrity by completing a 6-g pullout in a flight test demonstration. The airplane proposed by this paper will be very similar to the Goodyear racers but will not be restricted by rules fixing minimum wing area, landing gear design, or strength requirements.

Due to the small size of the airplane and the relatively high cruising speed anticipated it was not feasible to use induced drag and ideal span loading as the criteria for wing design. The more critical design items which determined wing layout were take-off speed, wing internal volume (for fuel) and design cruising speed.

Experience of the author indicates that take-off speeds of 80 mph and 65 mph for airplanes equipped with tricycle and conventional landing gears respectively, are realistic maximums for small aircraft operating from ordinary paved runways. The higher allowable take-off speed of the tricycle gear is due primarily to its vastly superior directional stability and control characteristics. The tricycle gear also makes possible consistently smoother landings which allows the use of lower design landing load factors and makes it possible to virtually eliminate the weight penalty usually associated with a tricycle gear. Since a fairly high wing loading is desired in order to increase airplane efficiency and permit the use of higher cruising speeds the tricycle gear became the logical choice.

A retractable landing gear was a must; a brief calculation shows that in the case of a very clean fixed gear having a drag of only 12 lbs. at 125 mph, the fuel required on a 25-hour flight was:

$$\frac{(\text{Drag})(\text{Velocity})}{375} \quad (\text{SFC})\ (\text{Duration}) = \frac{12\ (125)}{375} \quad (.5)\ (25) = 50 \text{ lbs.}$$

Thus a weight penalty of 50 lbs. for the retracting machanism could be accepted. A carefully designed manually-operated retraction system should not add more than 10 lbs. to the weight of an aircraft of this size.

The proposed landing gear is of the cantilever spring type, utilizing Cessna 180 tailwheels as mainwheels and a Piper J-3 tailwheel as a nosewheel. The main gear folds forward into the fuselage and the nosewheel back into the firewall area. Retraction is manual but a retract-cycle assist in the form of shock cord or pneumatic actuators is being considered, in the event that flight tests show that air loads cause retraction difficulties. The pneumatic system would utilize the steel tubing of the fuselage frame as an air reservoir.

The initial empty weight mentioned earlier indicated that about 80 gallons of fuel would be carried. Because this is a relatively large fuel load as much of it as possible was carried in the wings to obtain minimum wing loads during the fully loaded condition. In order to eliminate fuel in the wings during test flights and to provide a tank in which fuel could be measured, 15 gallons of fuel was carried in a fuselage tank leaving 65 gallons for the wings.

The wing of this airplane was designed to give a take-off speed of 80 mph at a wing C_l of 1.2 and an internal volume of 65 gallons. Using a 15% thick airfoil (NASA 64_2-415) with fuel in the forward 70% of the wing, an area of 56.5 ft^2 was required to obtain the necessary volume. The NASA 64_2-415 airfoil section was selected because it was well rounded at maximum lift indicating gradual stall characteristics. Good stall characteristics were considered to be of primary importance for this airplane since it will be necessary to make all take-offs and landings at speeds only slightly above the stall speed to keep landing gear loads to a minimum.

With the wing area determined, a study was made of the effect of aspect ratio on wing drag and weight. From this it was deduced that an aspect ratio of 10 produces the best combination of induced drag and wing structural weight. A straight tapered wing (easiest to build) with a taper ratio of 0.4 was chosen for optimum span efficiency. Studies by Hoerner indicate that a straight-trailing edge is most desirable from the stand point of tip-effects. The resultant slight sweepback also moves the wing root forward on the fuselage, thereby reducing fuselage interference drag and, in this particular airplane, providing more pilot leg room.

A study of long-range airplanes showed that in general, power loadings are about 20 lbs/bhp. For this airplane a minimum of 55 bhp was required to give this power loading. However, the only suitable American-built engine in production is the Continental C-85-8F J which develops 85 bhp at 2575 rpm. The characteristics which made it first choice were (1) very high reliability, (2) better than average smoothness for a 4-cylinder engine, (3) adaptability to fuel injection, (4) dry-sump oil system whose capacity is easily increased, (5) oil cooler not required, (6) a specific fuel consumption comparable to smaller engines even at low power outputs, and, (7) an increased weight of only 12 lbs. over the lightest competitor. With the C-85 engine the take-off power loading was only 13 lbs/bhp which allowed selection of a fixed-pitch propeller strictly on the basis of cruise performance since take-off and climb were not critical. The aircraft general layout was then completed using the selected wing and engine. A midwing position was preferred for the following reasons; (1) fuselage interference drag would be minimized without resorting to large complex fillets (2) wing fuel would be transferred by gravity and (3) slightly better longitudinal stability characteristics were obtainable.

A conventional tail arrangement was chosen because the fuselage interference drag can be kept nearly as low as the vee tail and the simplified structure and control system should save weight. Further weight reduction was attained by using an all-moveable stabilator in place of the usual stabilizer/elevator combination.

Fuselage dimensions were based on Continental C-85 engine dimensions and a 5' - 10" pilot weighing 165 lbs. Because of the duration of the proposed flight (over 25 hours) some concessions were made to pilot comfort. These included (1) widening the cabin to 23 inches to increase shoulder room (nominal shoulder dimension is 18 inches), (2) making the seat bottom 16 inches long and (3) reclining the pilot 30^{o} from the vertical. Items (2) and (3) serve to prevent pilot discomfort due to compression fatigue by distributing his weight over a greater area of his body. The Continental C-85 engine is 31.5 inches wide and the extra width was accomodated by cylinder fairings which extend back into the wing roots.

Since this airplane will be flown primarily straight and level on cross-country flights the stability calculations have been limited to those required to ensure adequate static longitudinal and directional stability margins. The acceptability of the stability and control characteristics will be determined during flight tests. The tail area of 8.5 ft^2 gave adequate stability and elevator power at both forward and aft c.g. positions.

The author's flying experience has been mostly in aircraft with very light control forces. For this reason, ailerons of the sealed-overhang type were chosen and the stabilator was pivoted close to the quarter-chord point to obtain very nearly zero control forces. Desired handling characteristics will then be obtained by use of adjustable "artificial feel" bungees, which also incorporate a 3-way trim system , a very desirable feature on long flights.

A fixed-pitch metal propeller was selected because it provided maximum efficiency in cruising flight and adequate take-off performance. The initial propeller will probably be either a Sensenich 76 AM or McCauley 1A90 shortened to 64 inches and re-twisted to an 82 inch pitch with final setting to be determined by flight testing.

STRUCTURAL DESIGN

With only highly experienced pilots flying this airplane it was possible to use design load factors considerably less than those required by Part 3 of the Civil Air Regulations. A survey of long range airplanes showed that design maneuver limit load factors varied from +2.0 to +3.3. The airplanes at the lower load factors, however, were modified commercial types whose structural strength had been well established prior to modification for long-range flight. "The Spirit of St. Louis," a notable one-of-a-kind example, was designed for a 3.3 limit load factor.

The gust-load condition in cruising flight was considered the most critical loading condition in this case since it is not pilot-imposed. Civil Air Regulations require that the aircraft be able to withstand a sharp-edged gust of 30 feet/second at cruising speed. However, a survey of gusts encountered by U.S. Civil transport aircraft indicated that 25 feet/second would be satisfactory.

The complete design V-n Diagram is shown as Fig. 2. The maximum load factor of +3.25 resulted from the 25 feet/second gust of 140 mph with full fuselage fuel and no wing fuel. This condition was critical for wing bending.

Only two construction techniques were deemed practical for this project: (1) all metal riveted structure with 3-M EC 801 sealant in all seams or (2) an all-metal primary structure with a fibreglass skin bonded to the primary structure using a cold-setting epoxy resin. If the fibreglass structure is formed in a plaster lay-up mold and then applied to the structure no sealing problems should occur as epoxies are impervious to fuel. In the case of the metal-covered wing the leak qualities of the completed structure can be checked by means of the ammonia test used by plastic balloon manufacturers. The decision between metal and metal/fibreglass wing structure depends upon the outcome of future study and testing.

During construction the surface finish will be kept as smooth as possible without resorting to paint. The question of surface finish is largely academic since on long flights at low altitudes a large number of insects will be encountered, smashed and carried on all leading edges for the remainder of the flight. Weather and heating difficulties rule out a winter time flight as a possible solution to the insect problem.

Steel tubing and fabric were used for the fuselage structure for ease of construction since none of the other techniques offered significant weight differences when reduced to practical terms.

Instrumentation and navigation equipment will be the very minimum required for a successful flight. Navigation instruments selected were an airspeed indicator, a sensitive altimeter, a 2-inch compass, and a 4-1/2 volt Askania electric turn and bank indicator. Engine instruments were limited to tachometer, an oil temperature and pressure gage and a cylinder-head temperature gage. The possibility of using an exhaust gas temperature gage to determine when the engine is operating on the best power (and hottest burning) fuel/air mixture will be investigated during flight tests. A set of appropriate World Aeronautical Charts and a Batori B-26 computer complete the navigation equipment.

For night operations, lights pose a rather difficult problem; the present proposed solution lies in modifying a tachometer generator to provide 3 amps at 12 volts to provide electrical power. If the FAA requires a battery, one will be built up of the nickel-cadmium cells used by radio-controlled model airplanes. These give approximately 3 ampere-hours which is sufficient for an emergency landing in the event of generator failure.

Although radio equipment is a luxury item on a venture of this type it might possibly pay its way by furnishing navigation assistance and weather information. A radio suitable for this flight is the transistorized Heath L/F direction finder weighing 4-1/2 pounds complete. Another possibility is a Narco Superhomer which would provide both 2-way VHF communications and VOR navigation facilities at a weight of only 10-1/2 lbs. including cables and antennas. This weight can be reduced somewhat by eliminating the case, substituting a lighter antenna and modifying the power supply to a transistorized version. Principal drawback is the current drain of 4-1/2 amps. although this can be reduced to about 2-1/2 amps. by going to the transistorized power supply and removing the transmitting tubes except when transmission is necessary.

Taking all the aforementioned design items into consideration the resulting airplane is shown in the 3-view drawing. Upon completing the lay-out drawing a weight and balance estimate was tabulated using actual known weights wherever possible. The initial weight estimate of 438.5 lbs. is 7-1/2% over the desired weight mentioned earlier. This is a weight penalty that is inherent in all very small airplanes which must use practical construction techniques and material sizes designed for use by larger aircraft. At first glance the wing weight of 77.0 lbs. (representing only 7.0% of the airplane gross weight) may appear somewhat low. However, as far as the wing is concerned the airplane weighs only 730 lbs. (since the fuel is uniformly distributed in the wing) and for this condition the wing weight totals 10.6% which is not unreasonable for a wing carrying only a 3.25 limit load factor at a wing loading of 13 psf.

The watchword during construction of this project will be borrowed from Donald Douglas -- "simplicate and add lightness."

FLIGHT PLANNING

The parasite drag area was determined using a technique outlined by Hoerner. The power required curve for this airplane was then plotted as Figure 1. Assuming a propeller efficiency of 85% and that the propeller limits the engine to 80 bhp the maximum speed was found to be 198 mph. Sea level rate of climb at 1102 lbs. was 730 feet per minute at 120 mph.

For comparison purposes, a constant airspeed cruise control was considered and using average L/D ratios the resultant maximum ranges are computed and plotted as Fig.~~5~~6. Maximum practical still-air range is about 4250 miles at 100 mph. However, at 140 mph a 3760 mile flight is still possible and since this requires only 26 hour and 54 minutes this speed is generally more satisfactory.

There are only two overland routes of suitable length available in the Western Hemisphere. The first stretches 3540 miles from Los Angeles, California, to St. Johns, Newfoundland, and the second runs 4070 miles from Fairbanks, Alaska, to Miami, Florida. Both of these routes start out over mountainous terrain, but, since this flight is contemplated at speeds greater than the best L/D speed, increased operating altitude should improve range. At 10,000 feet for example, a true airspeed of 140 mph can be attained at an indicated airspeed of 123 mph which is only 7% above the best L/D speed at full gross weight. The Fairbanks-Miami flight in late June could be planned to virtually eliminate night flying. Another possibility is a Goose Bay-L.A. flight in March or April when it is possible to catch strong easterly winds below 10,000 feet. However, the overall best flight route from a standpoint of terrain and facilities appears to be from L.A. to St. Johns or Goose Bay. A layout of these routes is included as Fig. 7.

Although this airplane has been designed specifically for establishing a Class I distance record, it is also capable of bettering the Class II distance record of 2470 miles. Overloading it very slightly to a gross weight of 1105 lbs will satisfy Class II requirements. (Class II: 1102 to 2204 lbs. gross weight).

This project has been undertaken to fulfill the author's longstanding desire to design, build, and fly an airplane embodying his own ideas. Since the studies undertaken to date tend to prove the feasability of the whole project (as herein outlined) the author feels justified in planning to begin construttion of the airplane as soon as time and circumstances permit.

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PARASITE DRAG AREA

Item	Area (ft^2)	C_{D_f}	Flat Plate Area (ft^2)
Aileron longitudinal gap	.005	1.3	.0065
Aileron lateral gap	.117	.025	.0029
Canopy seams	.075	.010	.0008
Gear door seams	.075	.010	.0008
Rudder gap	.0042	.025	.0001
Wing in slipstream	10.5	.0080	.0840
Elevator and rudder	12.0	.0080	.0960
Elevator and rudder in slipstream	10%		.0096
Fuselage	~~.086~~ 6.2	.086	.5333
Fuselage in slipstream	10%		.0533
Tail interference	4%		.0042
Total Parasite Area			.7957 ft^2

POWER REQUIRED AT 1102 LBS. GROSS WEIGHT

Vmph	C_L	C_D	s_w	s_f	s
73.6	1.40	.1010	5.700	.796	6.496
80	1.19	.0730	4.120	.796	4.916
100	.761	.0320	1.806	.796	2.602
120	.528	.0180	1.016	.796	1.812
140	.387	.0133	.751	.796	1.547
160	.296	.0107	.604	.796	1.400
180	.235	.0095	.537	.796	1.333
200	.190	.0088	.498	.796	1.294

Vmph	s	q	D	Pr	L/D
73.6	6.496	13.91	90.4	17.7	12.20
80	4.916	16.40	80.6	17.2	13.68
100	2.602	25.65	66.7	17.8	16.53
120	1.812	37.00	67.1	21.4	16.42
140	1.547	50.3	78.0	29.1	14.13
160	1.400	65.7	92.0	39.3	11.97
180	1.333	83.2	110.8	53.2	9.95
200	1.294	102.5	132.7	70.8	8.31

POWER AVAILABLE

V/nD	C_T	C	C_{P_o}/C_P	$\frac{\eta}{\eta_o}$	RPM	BHP
0.8	.102	.115	.496	.704	1640	49.0
1.0	.082	.100	.570	.755	1810	55.0
1.2	.053	.075	.762	.873	2100	65.0
1.29	.038	.057	1.000	1.000	2400	80.0

V/nD	C_T/C_P	η	nD	THP	Vmph
0.8	.888	.710	9030	34.8	82.3
1.0	.820	.820	9660	45.0	110
1.2	.706	.848	11200	55.2	153
1.29	.667	.860	12800	68.8	188

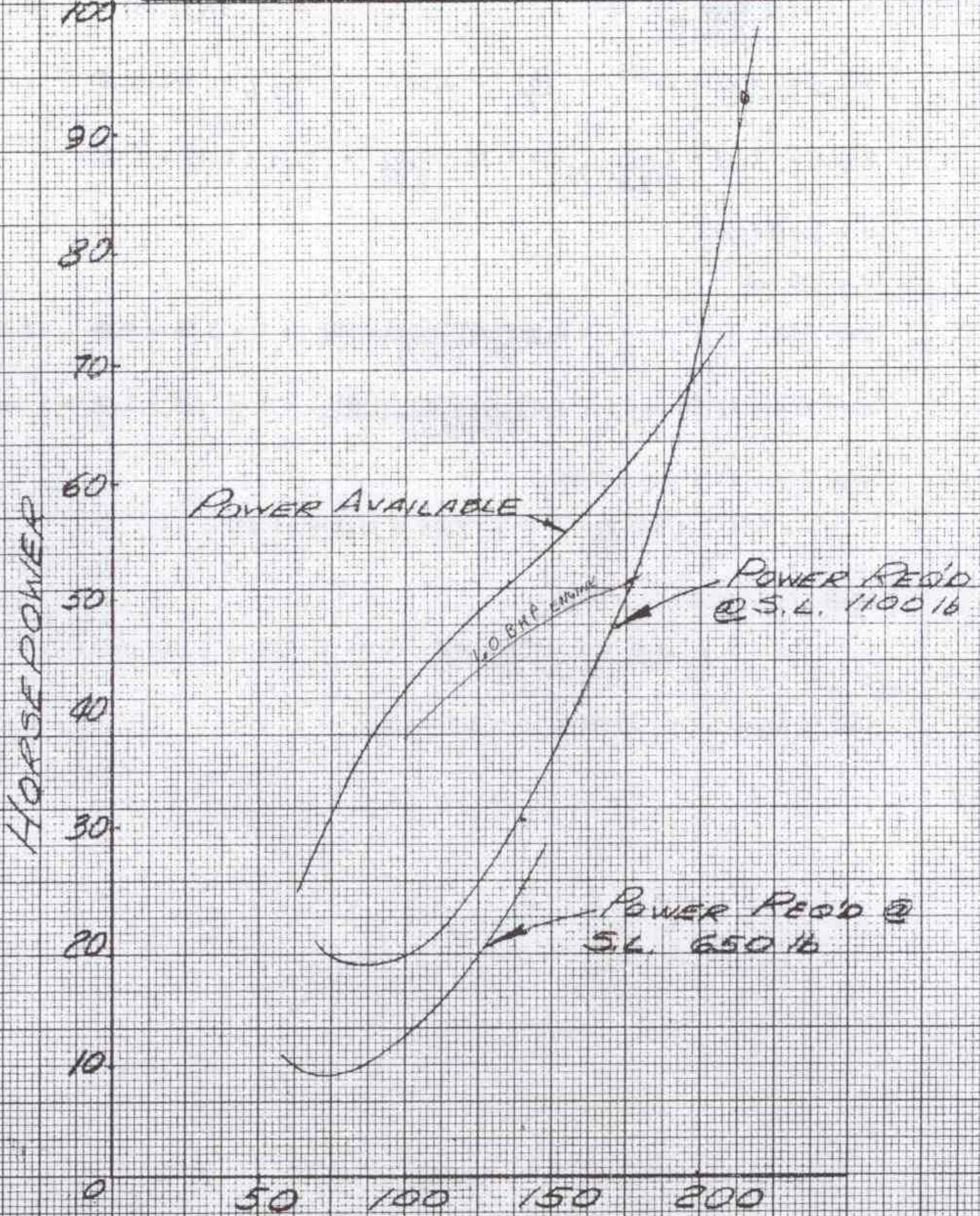
RATE OF CLIMB

Vmph	P_a	P_r	P_e	Rate of Climb
80	34.3	17.3	17.0	510
100	41.5	19.0	22.5	675
120	47.5	23.1	24.4	732
140	52.4	30.0	22.4	672
160	57.0	39.8	17.2	516
180	62.7	52.8	9.9	297
197	68.6	68.6	0	0

$$\text{Rate of Climb} = \frac{33000 P_e}{1100} = 30\ P_e$$

FIG. 1 POWER REQUIRED

AND POWER AVAILABLE



LOAD FACTORS[5]

$$n = 1 + \frac{KUVm}{575(W/S)} \qquad m = 57.3(.087) = 4.98\ C_L/RAD$$

$$K_{11.2} = 1/2(11.2)^{.25} = \frac{187}{2} = 0.935 \qquad K_{12.6} = 1/2(12.6)^{.25} = 0.940$$

$$K_{19.5} = 1.33 - \frac{2.67}{(19.5)^{.75}} = 1.33 - \frac{2.67}{9.25} = 1.33 - 0.29 = 1.04$$

$U = 30$ ft/sec

$V = 140$ mph

when $w/s = 11.2$ $\quad n = 1 + \frac{0.935(30)(140)(4.98)}{575\ (11.2)} = 1 \pm 3.04 = 4.04 = -2.04$

when $w/s = 19.5$ $\quad n = 1 + \frac{1.04(30)(140)(4.98)}{575(19.5)} = 1 \pm 1.95 = 2.95 = -.95$

when $w/s = 11.2$ and $V = 22$

$$n = 1 + 3.04\ \frac{22}{30} = 1 \pm 2.23 = 3.23 = -1.23$$

when $w/s = 12.6$ and $V = 30$

$$n = 1 \pm \frac{.940\ (30)(140)(4.98)}{575\ (12.6)} = 1 \pm 2.71 = 3.71 \text{ or } -1.71$$

CAM -3 Appendix B Suggest $n = 3.3$ at V_c

FIG. 2 V-n DIAGRAM AT WEIGHT = 1102 POUNDS

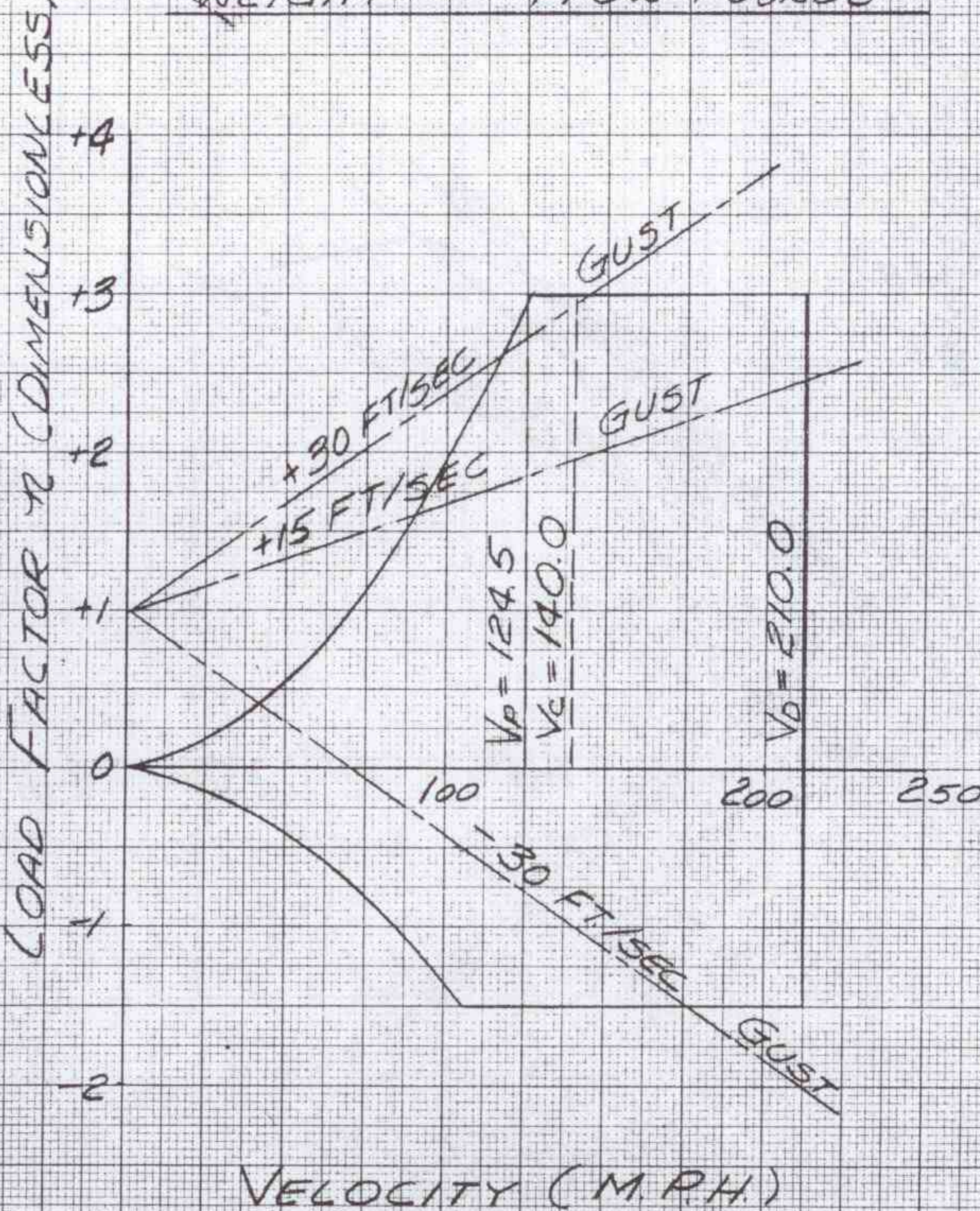
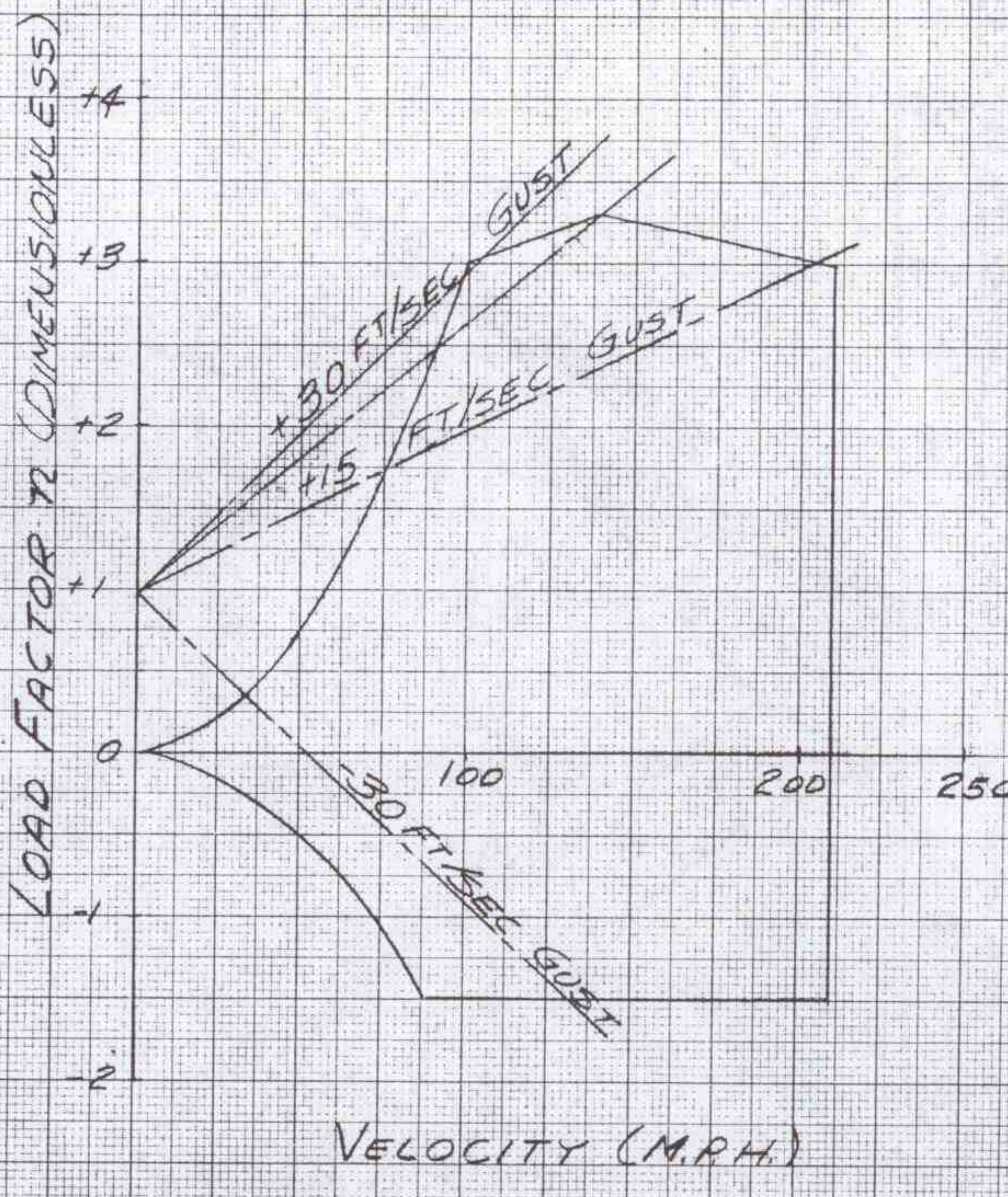


FIG. 3 V-n DIAGRAM

AT WEIGHT = 712 POUNDS



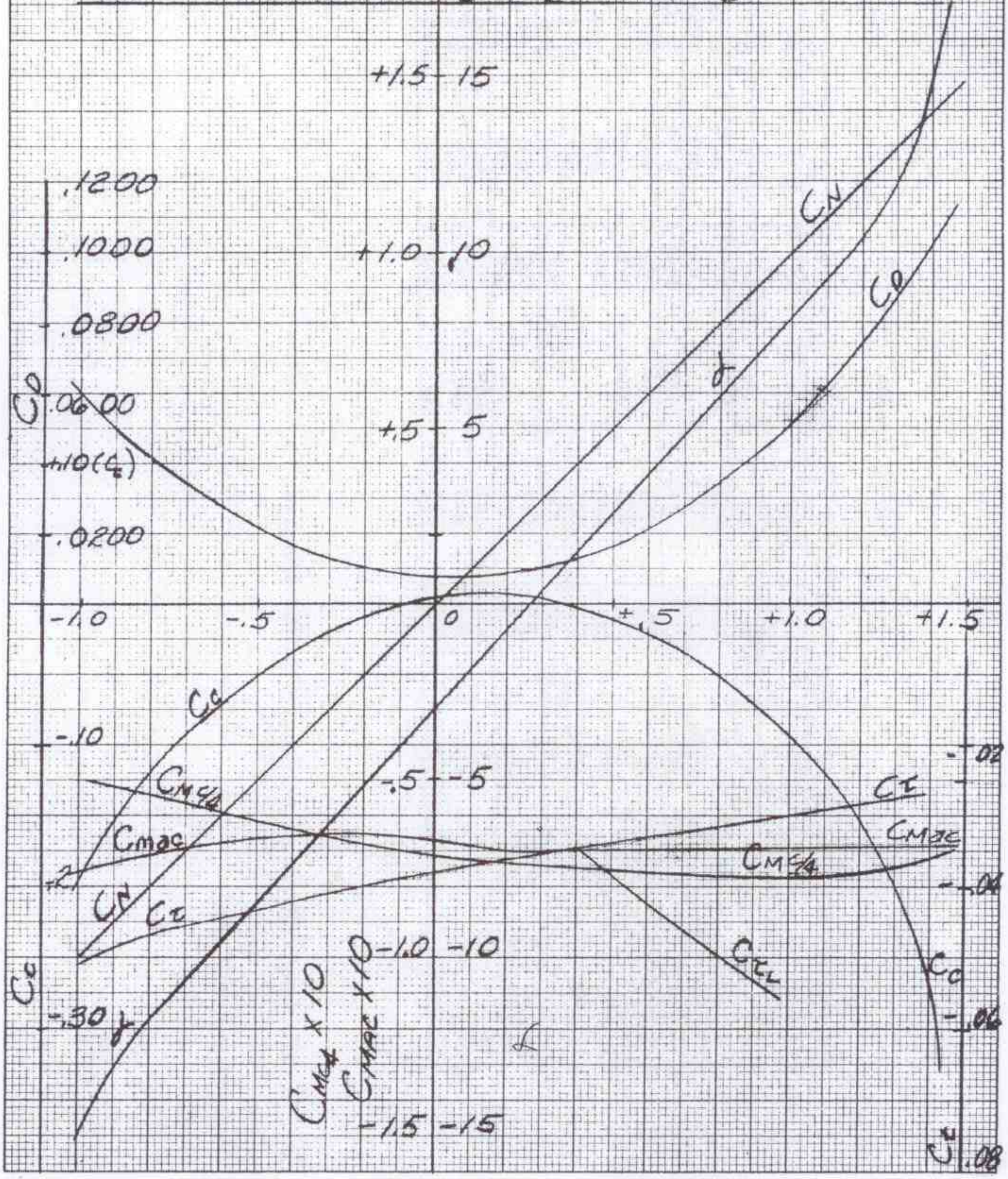
AIRFOIL CHARACTERISTICS[4]

1.	C_L	- 1.0	- .6	- .2	0	.2	.6	1.0	1.47
2.	α_o	-12.58	- 8.17	- 4.65	- 2.90	- 1.15	2.37	5.85	14.00
3.	$\Delta\alpha = 2.22\ C_L$	- 2.22	- 1.33	- .44	0	.44	1.33	2.22	3.26
4.	$\alpha = 2 + 3$	-14.80	- 9.50	- 5.09	- 2.90	- .71	3.70	8.07	17.26
5.	Cd_o	.0220	.0140	.0096	.0085	.0073	.0076	.0130	.0290
6.	$Cd_i = \frac{C_L^2}{25}$	.0386	.0139	.0016	0	.0016	.0139	.0386	.0835
7.	$C_D = 5 + 6$	.0606	.0279	.0112	.0085	.0089	.0215	.0516	.1125
8.	$\text{Cos}\ \alpha = \text{Cos}\ 4$	.965	.986	.995	.998	.999	.997	.989	.953
9.	$\text{Sin}\ \alpha = \text{Sin}\ 4$	- .256	- .165	- .089	- .0505	- .0124	+ .0535	.140	.297
10.	$C_L\ \text{Cos}\ \alpha = 1\times8$	- .965	- .592	- .199	0	.200	.598	.989	1.405
11.	$C_D\ \text{Sin}\ \alpha = 7\times9$	- .0155	- .005	- .001	0	0	.001	.007	.033
12.	$C_N = 10+11$	- .981	- .597	- .200	0	.200	.600	.996	1.438
13.	$C_L\ \text{Sin}\ \alpha = 1\times9$	.256	.099	.018	0	.002	.032	.140	.437
14.	$C_D\ \text{Cos}\ \alpha = 7\times8$	.058	.028	.011	.009	.009	.022	.051	.107
15.	$C_c = 14 - 13$	- .198	- .071	- .007	.009	.007	- .010	- .089	- .330
16.	$Cm_{c/4}$	- .050	- .060	- .065	- .068	- .071	- .075	- .075	- .067
17.	$Cmac$	- .075	- .068	- .067	- .067	- .068	- .068	- .068	- .068

$$a = \frac{1.4 - (.-8)}{13.61 - (-11.70)} = \frac{2.2}{25.31} = 0.0870\ C_L/\text{Degree}$$

$$m = 4.98\ C_L/\text{RAD}$$

FIG 4 AIRFOIL SECTION PROPERTIES CORRECTED TO ASPECT RATIO OF 10 — NACA 64_2 - 415



PROPELLER DESIGN[1]

(1) Assume engine develops 80 BHP at 2400 RPM at 187 MPH

(2) Assume $\eta = 0.86$

(3) Then $C_s = \frac{0.638V}{P^{.2}\ N^{.4}} = \frac{0.638\ (187)}{80^{.2}\ 2400^{.4}} = 2.21$

(4) Ref: TR 640 For Clark Y Section, Two Blades:

$\beta = 30^{\circ}$ at 0.75 RAD

and $\frac{V}{nD} = 1.29$ also $\eta = 0.86$ (as assumed)

(5) Diameter $= \frac{V}{1.29n} = \frac{187\ (1.467)(60)}{2400\ (1.29)} = 5.32$ ft. $= 64$ inches

(6) and Pitch $= \frac{V}{n} = \frac{187\ (1.467)12}{40} = 82$ inches

LONGITUDINAL STABILITY[3]

$$\frac{dC_m}{dC_L} = \frac{x_a}{c} + \left[\frac{dC_m}{dC_L}\right]_{fus.} - \frac{a_t}{a_w} \bar{V}_{\eta t} \left[1 - \frac{d\epsilon}{d\alpha}\right]$$

$$\frac{Xa.c.}{c} = 0.2860 - .2640 = 0.0220$$

$Xa.c. = .2640c$
$Xc.g. = .2860c$

$$\frac{dC_m}{dC_L}\ fus. = \frac{K_f w_f^2 L_f}{S_w C a_w}$$

$$= \frac{.0133\ (4)(15.5)}{56.5\ (2.5)(.091)}$$

$$= 0.0640$$

$K_f = .0133$
$w_f = 2.0$ ft.
$L_f = 15.5$ ft.
$S_w = 56.5$ ft.^{2}
$c = 2.5$ ft.
$a_w = 0.091$
$A_t = 4.46$
$a_t = .075$
$l_t = 9.0$
$S_t =$

$$\bar{V} = \frac{S_t l_t}{S_c}$$

$$\bar{V} = \frac{8.5\ (9.0)}{56.5\ (2.5)} = .532$$

$$\eta_t = .90$$

$$\frac{d\epsilon}{d\alpha} = .38 \quad \text{(p.224 Perkins \& Hage)}$$

$$\frac{a_t}{a_w} \bar{V} \eta_t \quad 1 - \frac{d\epsilon}{d\alpha} = \frac{.075}{.091} (.532)(.90)(.62) = .2446$$

$$N_o = .2640 - .0640 + .2446 = .4446$$

Propeller Contribution

$$\Delta N_o = \frac{-N \frac{dC_n}{d\alpha} \frac{d\beta}{d\alpha} l_p S_p}{S_w C a_w} - \frac{a_t}{a_w} \frac{\bar{V}\eta_t}{.07} \frac{d\beta}{d\alpha} \frac{dC_n}{d\alpha}$$

$$= \frac{.00165\ (1.15)(4.65)(1.75)}{56.5\ (.091)(2.5)} - \frac{.091}{.075} \frac{.605}{.07} \underline{(1.15)(.00165)}$$

$$= -.0012 - -.0199$$

$$= -.0211 \quad \text{Power Off}$$

$$= -.0611 \quad \text{Power On}$$

$$No = 0.4446 - .0211 = .4235 \quad \text{Power Off}$$

$$No = 0.4235 - 0.0400 = .3835 \quad \text{Power On}$$

DIRECTIONAL STABILITY[2]

(1) Wing Contribution:

$$C_{n\psi_w} = -.00006 \quad (\Lambda)^{.5}$$

$$= -\ .00006\ (.75)^{.5} = -.0001$$

(2) Fuselage Contribution:

$$C_{n\psi_f} = \frac{.96\ K_\beta}{57.3} \quad \frac{S_s}{S_w} \quad \frac{L_f}{b}\left[\frac{h_1}{h_2}\right]^{.5}\left[\frac{w_2}{w_1}\right]^{.33}$$

$S_s = 28.4$
$S_w = 56.5$
$L_f = 15.3$
$b = 23.75$
$h_1 = 26$
$h_2 = 20.5$
$w_1 = 32.5$
$w_2 = 14.0$
$d = 64$
$h = 36$

$$\frac{d}{L_f} = \frac{64}{184} = 0.347$$

$$K_\beta = 0.15$$

$$\frac{L_f}{h} = \frac{184}{36} = 5.12$$

$$C_{n\psi_f} = \frac{0.96\ (.15)}{57.3} \quad \frac{28.4}{56.5} \quad \frac{15.3}{23.75} \quad \left[\frac{26}{20.5}\right]^{.5} \quad \left[\frac{14.0}{32.5}\right]^{.33}$$

$$= \frac{.96(.15)}{57.3} \quad .503 \quad .644 \quad 1.25 \quad .755$$

$$= \frac{.0441}{57.3} = .000769$$

Interference Effect = - .0001 for Midwing

Then $C_{n\psi_f}$ = .000669

(3) Propeller Contribution

$$C_{n\psi_p} = \frac{\pi\ D^2\ l_p\ \frac{dC\psi_p}{dx}\ N}{4\ S_w\ b}$$

$D = 5.33$
$l_p = 4.65$
$\frac{dC\psi_p}{dx} = .00165$
$N = 1$
$S_w = 56.5$
$b = 23.75$

$$C_{n\psi_p} = \frac{\pi\ (28.45)\ 4.65(.00165)}{4(56.5)\ (23.75)}$$

$C_{n\psi_p}$ = .000128 Power Off

= .000184 Power On

$$C_n\psi = -a_v \frac{S_v}{S_w} \frac{l_v}{b} \eta_v + \Delta_2 C_n\psi$$

$$C_n\psi = -.059 \frac{4.167}{56.5} \frac{9.0}{23.75} .98 + -.00013$$

$$C_n\psi = -.00161 - .00013$$

$$C_n\psi = -.00174$$

$A_v = .059$
$N_v = .98$
$l_v = 9.0$
$S_v = 4.167$
$S_n = 56.5$
$L = 23.75$
$\Delta_2 C_n\psi = -.00013$
$D_f = 38$
$d = 20$
$\frac{d}{D_f} = .527$

STATIC DIRECTIONAL STABILITY

Item	$C_n\psi$ (Power Off)	$C_n\psi$ (Power On)
Wing	-.000100	-.000100
Fuselage	.000669	.000667
Propeller	.000128	.000184
Tail	-.001740	-.001740
	-.001043	-.000983

Desired Range $\leq -.0005$

RANGE SUMMARY

(avg) Vmph	L/D avg	η	C	Range
160	9.74	.86	.525	3360
140	11.78	.86	.565	3760
120	14.29	.86	.615	4170
100	15.87	.85	.665	4250
95	16.40	.84	.715	4050

$$\text{Range} = 375 \ \frac{L}{D}_{avg} \ \frac{\eta}{c} \ \ln \frac{GW}{EW}$$ (Breguets' Equation)

$$\ln \frac{1102}{630} = \ln 1.75 - .560$$

$$375 \ln 1.75 = 210$$

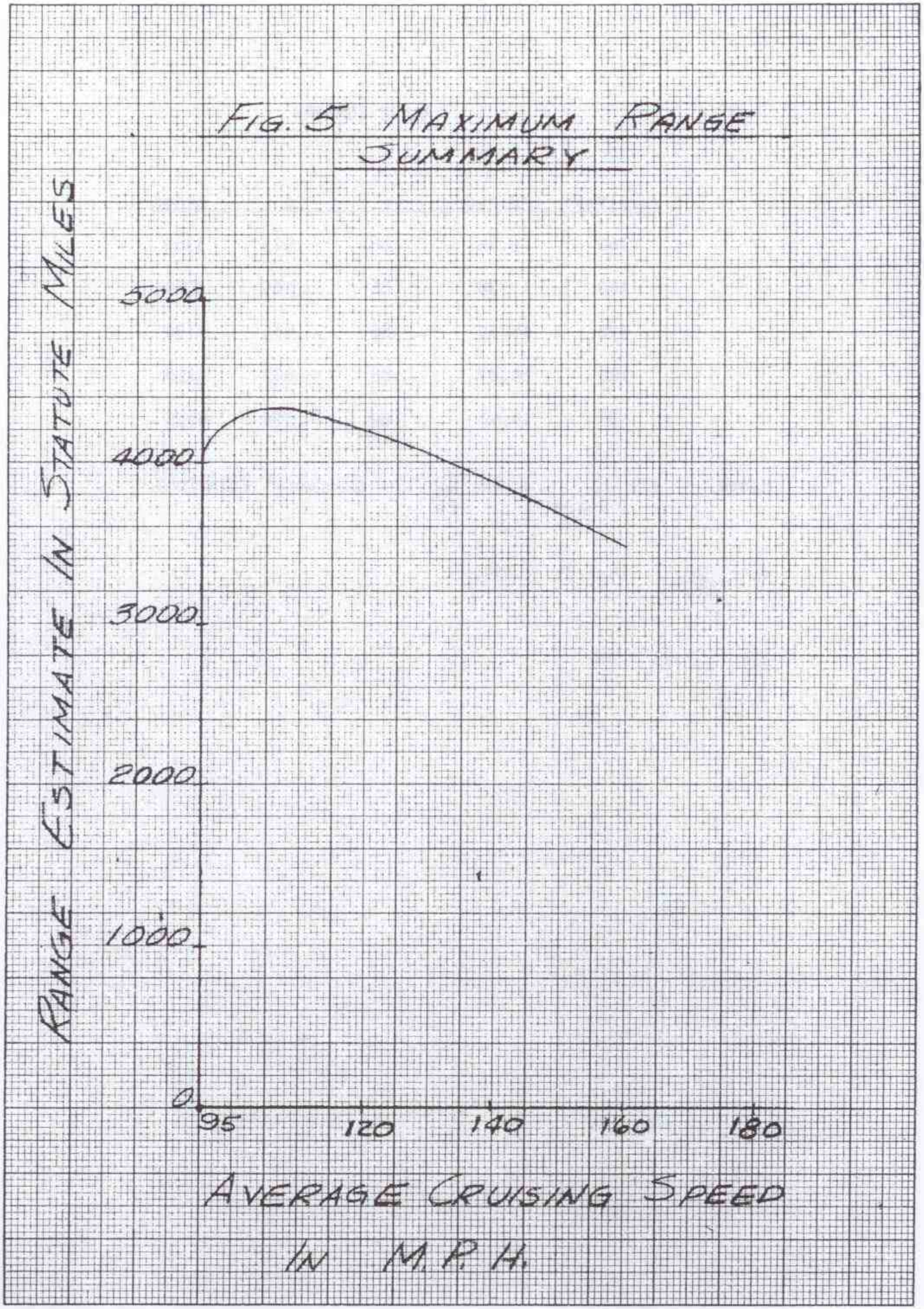
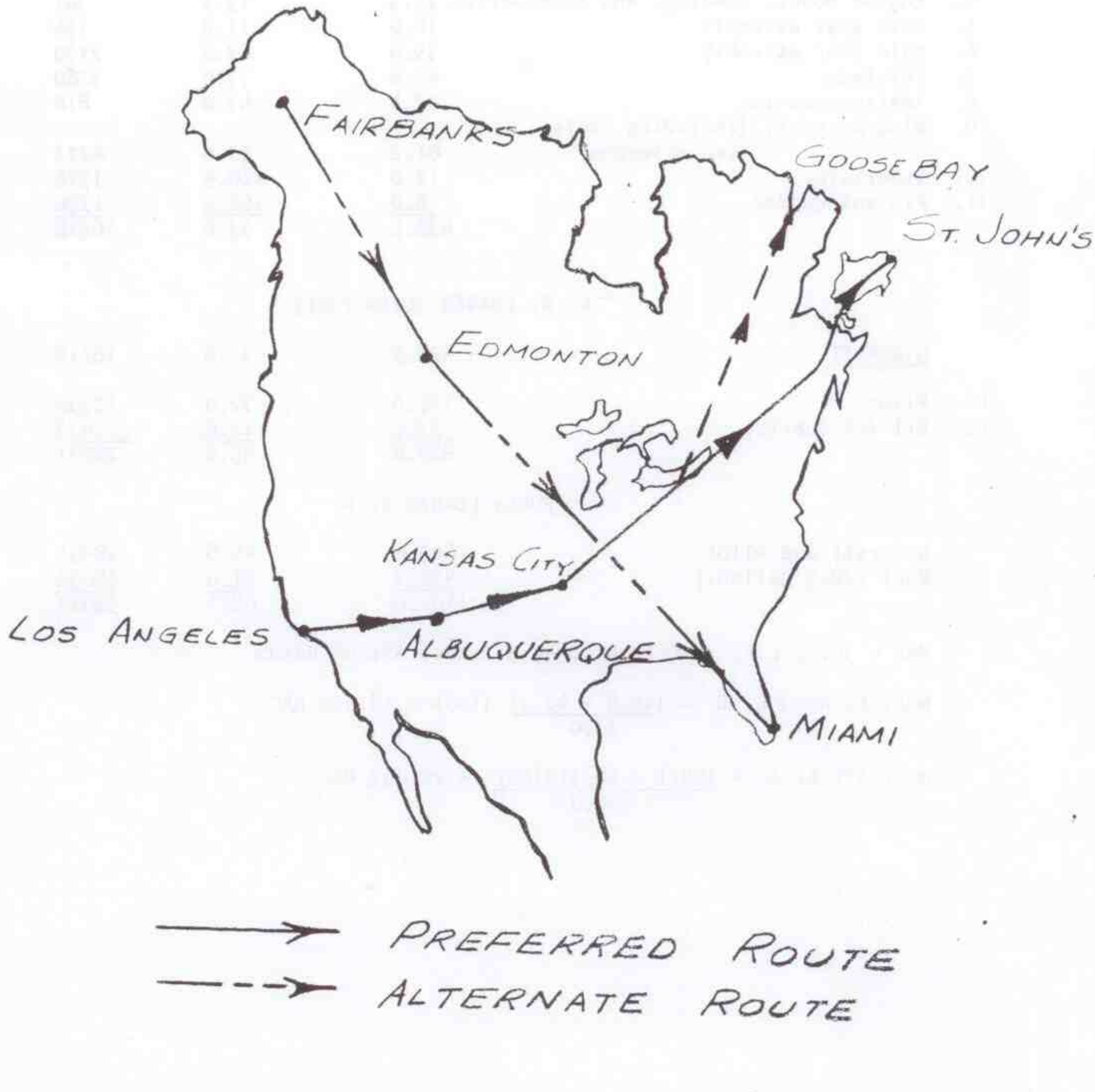


FIG. 5 MAXIMUM RANGE SUMMARY
RANGE ESTIMATE IN STATUTE MILES
5000
4000
3000
2000
1000
0
95
120
140
160
180
AVERAGE CRUISING SPEED IN M.P.H.

FIG. 6 ROUTE MAP



WEIGHT AND BALANCE CALCULATIONS

	ITEM	WEIGHT	ARM	MOMENT
1.	Propeller	22.0	-1.5	-33
2.	Spinner	2.0	-5.5	-11
3.	Engine	182.0	11.5	2093
4.	Engine Mount, Cowling, and Accessories	22.3	13.5	301
5.	Nose Gear Assembly	14.0	11.0	154
6.	Main Gear Assembly	35.0	62.0	2170
7.	Fuselage	42.6	77.0	3280
8.	Instrumentation	17.4	47.0	818
9.	Wing Assembly (including cables and ailerons)	81.2	52.0	4222
10.	Stabilator	12.0	160.5	1926
11.	Fin and Rudder	8.0	162.0	1296
		438.5	37.8	16218

C. G. LOADED (LESS FUEL)

	AIRCRAFT	438.5	37.8	16218
12.	Pilot	170.0	72.0	12240
13.	Oil (12 Quarts)	22.5	21.0	473
		629.8	45.8	28931

FULLY LOADED C. G.

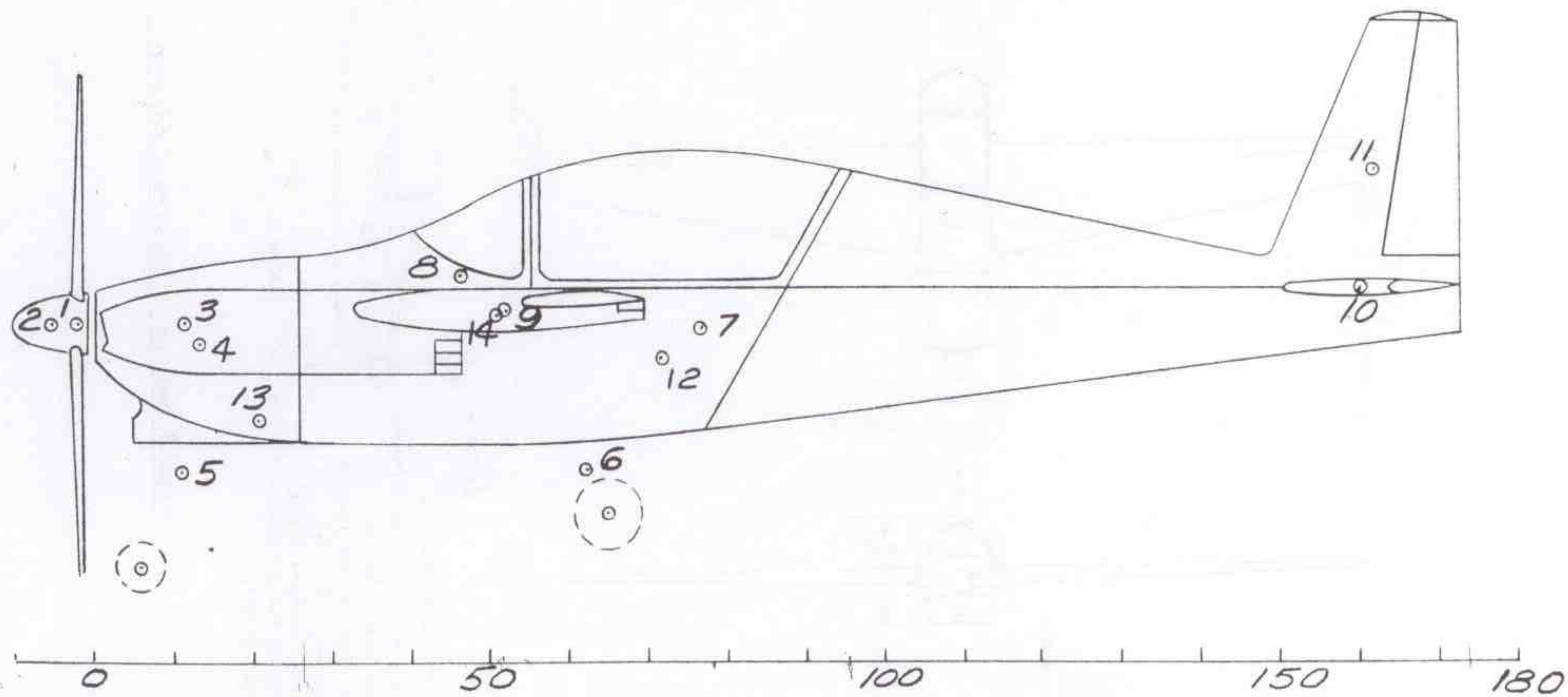
Aircraft and Pilot	629.8	45.8	28931
Fuel (78.7 Gallons)	472.2	51.0	24082
	1102.0	48.1	53013

MAC = 30.0, L.E. MAC located 40.3 inches Aft of Datum

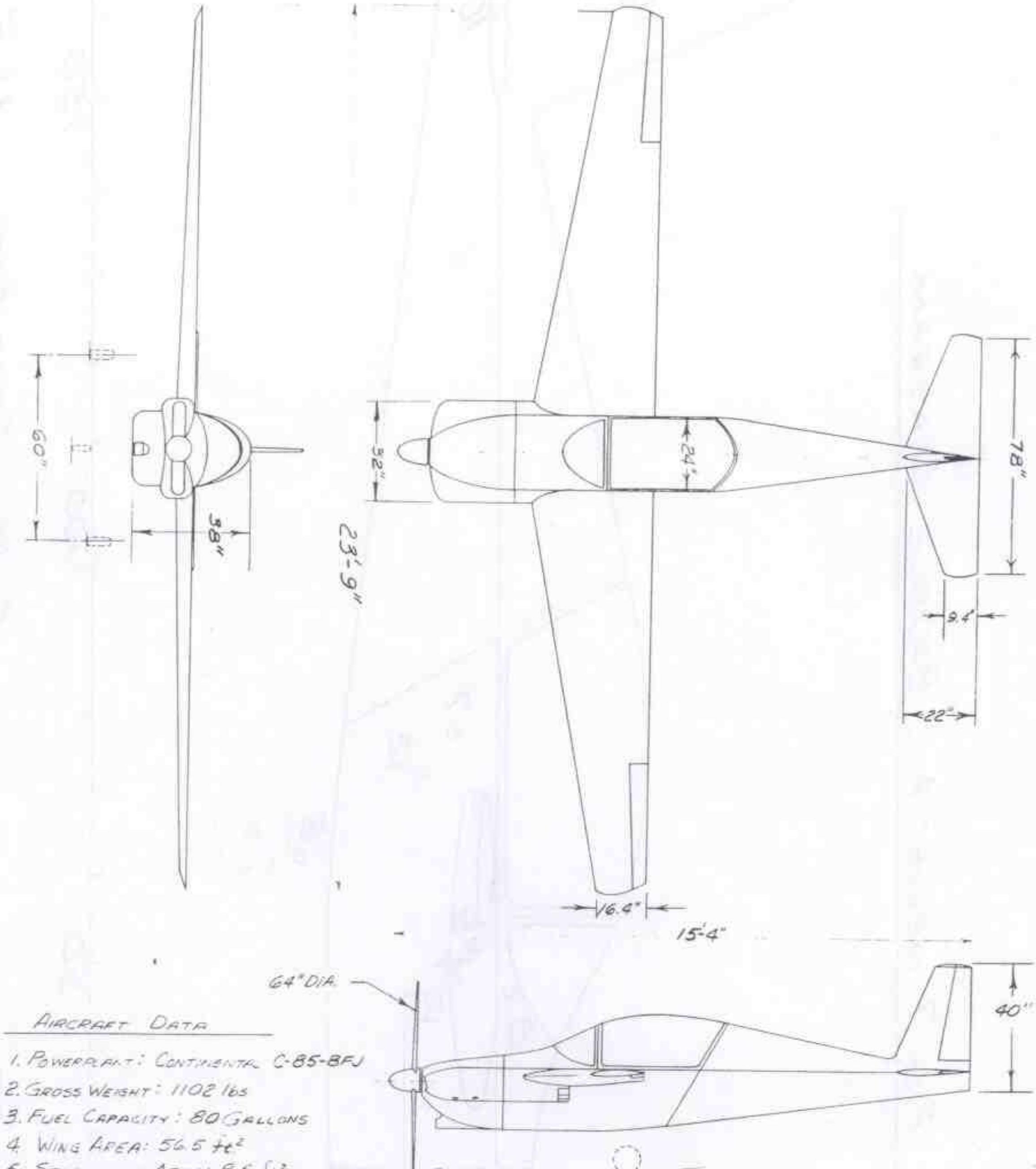
Most Forward C. G. = $\frac{(45.8 - 40.3)}{30.0}(100) = 18.33\%$ MAC

Most Aft C. G. = $\frac{(48.1 - 40.3)(100)}{30.0} = 26.00\%$ MAC

FIG. 7 WEIGHT AND BALANCE DIAGRAM



1. PROPELLER
2. SPINNER
3. ENGINE
4. ENGINE MOUNT, COWL & ACC.
5. NOSE GEAR ASS'Y
6. MAIN GEAR ASS'Y
7. FUSELAGE
8. INSTRUMENTATION
9. WINL ASS'Y
10. STABILATOR
11. FIN AND RUDDER
12. PILOT
13. OIL



AIRCRAFT DATA

1. POWERPLANT: CONTINENTAL C-85-8FJ
2. GROSS WEIGHT: 1102 lbs
3. FUEL CAPACITY: 80 GALLONS
4. WING AREA: 56.5 ft^2
5. STABILATOR AREA: 8.5 ft^2
6. VERTICAL FIN AND RUDDER AREA: 5.25 ft^2

FIG. 8 AIRCRAFT GENERAL LAYOUT