THESIS

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toward the Degree of Master of Science.

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Subject:-

THE DESIGN OF A TRANS-ATLANTIC AIRPLANE.

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head of the Department of Naval Architecture and
Aeronautical Engineering.

CONTENTS.

INTRODUCTION.

The question of trans-Atlantic flight is at this time very much in the public mind. The difficulties of the attempt have been forcibly called to attention by the protracted disappearance of the Sopwith plane after its courageous start. The success of one of our Navy-Curtiss flyingboats is quite gratifying, and the experiences of the N-C-1 and N-C-3 at least prove the value of being able to navigate on the surface of the sea. Otherwise the N-C-3, which came into port under its own power after weathering a storm for two days, would certainly have been lost. But the destruction of the N-C-1, and the damage to the N-C-3 by the seas, admittedly heavy, makes one skeptical of the actual sea worthiness of ships of this type. It should be borne in mind that none of the three boats were forced down by motor trouble.

The preparations which are going forward at this time for the trans-Atlantic flight are practically one-chance spurts with no idea of establishing air transportation. For air transportation the design laid out in this thesis allows reasonable margins for a non-stop flight of the 2000 miles from New Foundland to Ireland. The problem is, of course, to carry sufficient fuel.

From present indications of contemporary dirigible airship design, it cannot be imagined that the airplane will be used for long-distance weight-carrying work between such countries as England and the United States. Airships such as the

the R-34 and of the Zeppelin type are veritable ocean liners, having, for a given increase of size, a greater increase of lifting capacity. However, this great carrier like the railroad train is limited to fixed terminal stations, whereas the seaplane though smaller is not limited in choice of destination, much as the motor truck, and, seaplanes of the trans-Atlantic type can be of great practical use in flights of perhaps 1000 miles carrying a freight load equal to half the fuel weight of trans-Atlantic requirements.

Some startling predictions have been illustrated and advocated, leaving the solution to the aeronautical engineer. Some of these predictions may come true, but for the present it is certainly most sensible to develop, on an engineering basis of accepted facts, a plane that can be built now and will unquestionably perform as estimated.

ESTIMATE DESIGN.

There are in general three types of airplanes; namely, the monoplane, with internal structure, which acts as a single wing with no parasite resistance, the modern trussed biplane or triplane, and the proposed tandem plane.

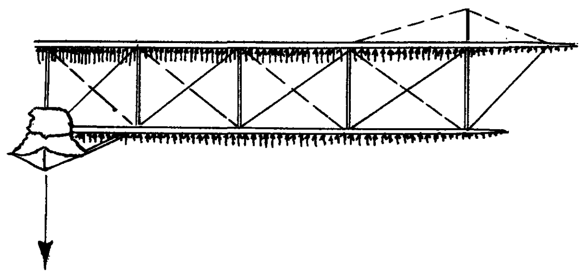
The internal wing truss aerofoil is very attractive as for a given weight to be carried the resistance is practically the Drag/Lift ratio of that weight regardless of the speed. This type, however, has not yet been developed beyond the stage of omitting the lift and landing wires in some of the scout planes. Hence, one cannot consider the 'all-wing' plane as a practical machine at this time.

The modern biplane and triplane we are quite familiar with and have no hesitancy in developing new airplanes along these lines.

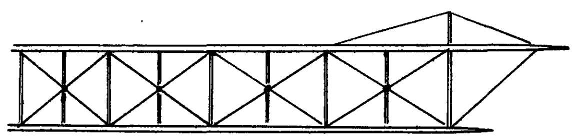
The tandem airplane is a combination of two planes one behind the other attached to a common fuselage. Its advantage lies in the apparent increase of lifting capacity without disproportionate increase of weight, but its objections are, the questionable air conditions behind the leading set of wings, the difficulty of maneuvering, and the failure of past machines of this type.

Since it is not necessary to make use of questionable types of airplanes, although possibly of advantage, we shall consider at once the modern biplane.

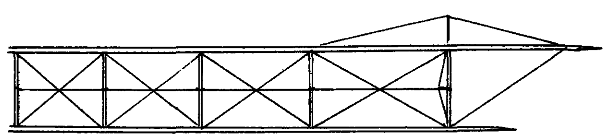
To support the weight of the machine the wings derive an equal lift from the motion of the air, and practically all of this lift is transmitted back to the fuselage through the structural truss composed of the upper wing, the lower wing and the lift wires.



The lift for each bay can be resolved into loads at the strut hinges where it is divided into compression in the upper wing beam and tension in the lift cables. Now the strength of a beam in compression varies as the square of its length so if we use the SPAD system and insert half struts from the junction of the lift and landing wires to the wing beams



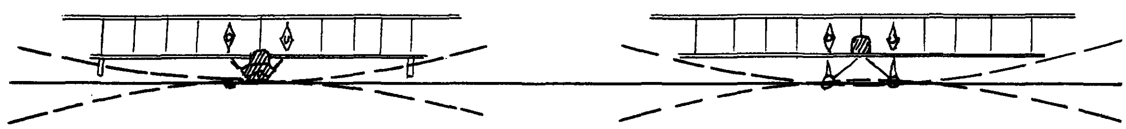
we can decrease the stress in the wing beams to nearly 1/4 of the regular biplane stress. Furthermore, since the full struts have similar characteristics it would be worthwhile to connect their centers as is done in the Caproni and N-C planes:



Combining these two we find sufficient support for the insertion of a center aerofoil without additional external structure. Thus we find that a triplane has a distinct advantage over the biplane from considerations of stress and parasite resistance.

Also since we are designing for a trans-Atlantic flight there are certain limitations which are imediately apparent: two pilots and a navigator are the minimum crew, two motors on a plane which can fly with one is the practical minimum power plant. The biggest convenient motor not requiring the attention of more than one man is at presemt the Liberty motor. Furthermore, this ship should be a seaplane without question.

There are two types of seaplanes: flyingboats using one hull as a combination fuselage and planing surface, with two wing pontoons to keep its balance, and the hydroairplane similar to a land machine except for the hydroplane pontoon in place of the landing gear.This hydroairplane may have a single pontoon with wingtip pontoons as the flying boat or two pontoons like a catamaran.The double pontoon system certainly has an advantage over the single float with wingtip pontoons in a heavy sea,for the wing



pontoons will alternately lift and hang on the wing structure and in extreme cases cannot lift the lower plane clear of the sea.A double pontoon system fits a two-wotored plane very well so for the present we will consider our design limited to a triplane,double-pontoon seaplane carrying a mimimum of three men and two Liberty motors.

A severe opposition was raised against the Handley-Page machines on the basis of airplane weight increasing as the cube of the dimension, whereas the lifting capacity increases only as the square of the dimension. The truth of this assertion is quite apparent for the lift depends directly on the wing surface which is proportional to the square of the dimention and obviously in similar structures the weights will be as the cube of the dimension. That the unit stress remains unchanged during a symmetrical expansion of an airplane cam be seen from the following brief calculations for a wing beam:

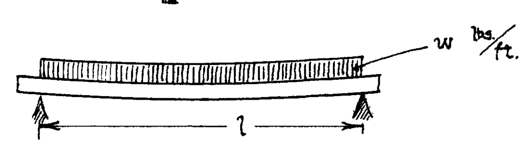
First case:	Loading per square foot	= 10 lbs.
	Chord	= 6 ft.
	Distance between struts	= 6 ft.
	Wing beam	= 4"x2" solid
Second case:	Loading per square foot	= 10 lbs.
	Chord	= 12 ft.
	Distance between struts	= 12 ft.
	Wing beam	= 8" x 4" solid

For bending stress due to air load we have:

wt/ft. $total_1 = 60\# = 30\# / ft.$ on front $beam_1$

wt/ft. $total_2 = 120\# = 60\# / ft.$ on front $beam_2$

Maximum bending moment $= \frac{wl^2}{8}$



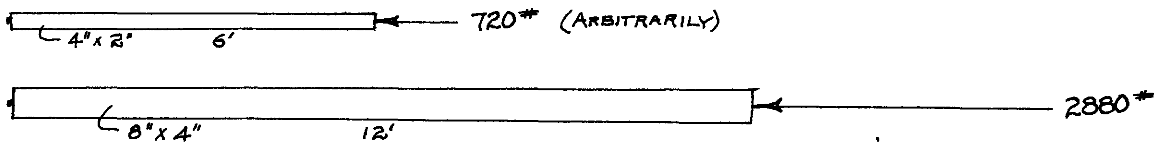
$M_1 = \frac{30 \times 6^2}{8} = 135$ ft, lbs. $\qquad I_1 = \frac{bd^3}{12} = \frac{2 \times 4^3}{12}$

$M_2 = \frac{60 \times 12^2}{8} = 8\ M_1 \qquad I_2 = \quad = \frac{4 \times 8^3}{12} = 16\ I_1$

But $M = \frac{S\ I}{c}$ therefore $S = \frac{Mc}{I}$

$S_1 = \frac{M_1 2}{I_1}$ $S_2 = \frac{8M_1 4}{16 I_1} = \frac{2M_1}{I_1}$ Therefore $S_1 = S_2$

which shows equal strength for air load with weights as the cube of the dimension. For compression we find:



Considering compression with bending about the major axis we have an equal slenderness ratio for the two cases and an equal compression load per square inch, hence the fibre stress in the two cases is the same. Therefore, if an airplane is expanded symmetrically, its weight will increase faster than its lift. As a result of this it is customary to change structural systems, in some such manner as described heretofore, from monoplane to biplane, biplane to triplane and possibly triplane to tandem planes, but it is evident that beyond the point of possible detail design lightening, an increase of size in the airplane is not advantageous for carrying capacity.

In an 800-horsepower airplane no piece will be so small from strength considerations as to be too delicate to work in an American shop, hence we shall use the extraneous minimum limitation as the basis of design for a trans-Atlantic airplane. This calls for a triplane with:-

Double pontoons

3 men

2 Liberty motors (400 HP. each)

2000 mile range.

General Estimate for Choice of Plane.

Let W = the weight of machine full except fuel, in pounds.

G = the weight of fuel, in pounds.

D/L = the wing efficiency ratio for triplane at cruising (speed.

V = the velocity of flight, in miles per hour.

t = time, in hours.

s = distance, in miles.

T = thrust, in pounds.

K_p = coefficient of parasite resistance, in lb.per mi,hr.

Then

$$T = \frac{D}{L}(W + G) + K_p V^2$$

$$HP_{required} = \frac{TV}{375} = \frac{V}{375}\left\{\frac{D}{L}(W + G) + K_p V^2\right\}$$

$$HP_{motor} = \frac{1}{.75}\frac{V}{375}\left\{\frac{D}{L}(W + G) + K_p V^2\right\} \quad \text{(assuming 75\% propeller eff.)}$$

$$HP\text{-hrs.} = HPt = \frac{4}{3}\frac{Vt}{375}\left\{\frac{D}{L}(W + G) + K_p V^2\right\}$$

but

$$Vt = s = 2000$$

therefore

$$HP\text{-hrs.} = 7.12\left\{\frac{D}{L}(W + G) + K_p V^2\right\}.$$

A gasoline consumption of 1/2 pound per HP-hr. is used as a standard estimate figure. Applying this we find

$$G = 3.56\left\{\frac{D}{L}(W + G) + K_p V^2\right\}.$$

Or

$$\frac{G}{(W + G)} = 3.56\left(\frac{D}{L} + \frac{K_p V^2}{W + G}\right).$$

This forces an assumption of:- the live load percentage, the overall wing efficiency and the ratio of parasite resistance to weight at some definite speed.

For the assumption of a live load percentage let us consider the following table of actual planes:

Table I.

Machine	Power	Full weight	Live load	$\frac{G}{W + G}$
Curtiss T	1500 HP	22000 lbs.	6354 lbs.	28.9%
Curtiss H-12	750	6197	2044	33.2
Curtiss H-16	800	17000	6000	35.3
Curtiss H-16a	660	10900	3500	32.1
Curtiss F-5L	660	13000	4750	36.5
Curtiss NC-1	990	21560	7750	36.0
Curtiss NC-4	1320	29000	10800	37.3
Martin Bomber	800	9663	3801	39.4
Caproni	990	12340	4640	37.6
Handley Page	800	14300	6406	44.7
U.S. D-9A	400	4987	2200	44.2
D. H. 10A	397	8500	2900	34.1
Curtiss N-6	100	1800	700	38.9
Curtiss 18-2 (tri.)	400	2901	1126	37.1
Loening Monoplane	340	2368	1040	44.0
Breguet	225	2142 kg.	800 kg.	37.3
Caproni	240	2900 "	1000 "	34.5
Caudron	160	1235 "	500 "	40.5
SPAD	150	730 "	280 "	38.3
Nieuport	80	660 "	250 "	37.8
		Average load percentage	=	37.4

This will justify an assumption of 40% for live load ratio, and it should be noticed that three of these planes operate at 44 percent.

For the assumption of ratio of parasite resistance to wing-drag at say 100 miles an hour we can use the data in table II calculating the resistance coefficient for each plane as in the case of the NC-1 as follows:

The reduction coefficient for biplane and triplane lift can be based on Hunsaker's triplane experiments from which it is reasonable, for high speed machines, to use an L/D factor of 1.00 for monoplanes, 0.75 for biplanes and 0.70 for triplanes. Combining this with an aerofoil L/D of 14 (considering at low incidence the RAF 6 which is used on flying boats) we get an approximate wing efficiency of 14 for monoplanes, 10.5 for biplanes and 9.8 for triplanes.

Then the wing drag of the NC-1 is $\frac{21560}{10.5} = 2050$ pounds.

The thrust of 990 HP at 81 mph is $\frac{990}{81} x 375 x 0.75 = 3430$ lbs. where the factor 0.75 is the propeller efficiency.

This leaves 3430 - 2050 = 1380 pounds used for parasite resistance. Raising this to its equivalent at 100 mph:

$$1380 x \frac{100^2}{81^2} = 2110 \text{ pounds.}$$

And the ratio is $\frac{2110}{2050} = 1.03$.

Table II.

Machine	Weight	WingDrag	HP	V	Thrust	Par.Res.	K_p	Ratio
Curtiss NC-1	21560	2050	990	81	3430	1380	2110	1.03
Curtiss F-5L	13000	1237	660	87	2130	893	1180	.96
Curtiss H-16A	10900	1040	660	95	1950	910	1010	.97
Curtiss HS-2L	6432	612	330	91	1020	408	493	.81
Martin Bomber	9663	920	800	113	1990	1070	840	.91
Caproni	12340	1175	990	103	2700	1525	1440	1.22
Handley Page	14300	1360	800	93	2420	1060	1230	.91

Average ratio of parasite resistance to wing drag at 100 $\frac{m}{h}$ = .97

These heavy machines were chosen as being nearest the design of this thesis. The parasite resistance of these planes averaging less than their wing drag and since further refinement could undoubtedly be made we can safely consider the parasite resistance equal to the wing drag of the new design at 100 miles an hour.

Then $\frac{D}{L}(W + G) = K_p V^2$ at 100mph. or $K_p = \frac{D(W + G)}{L\ 10000}$.

Therefore $$\frac{G}{W + G} = 3.56\left(\frac{D}{L} + \frac{D}{L}\frac{V^2}{10000}\right).$$

Now assuming an L/D of 17 corresponding to the maximum efficiency of an aerofoil and a triplane factor of 0.76 at maximum L/D, we have an overall wing efficiency of 0.76 x 17 = 13 on the basis of designing primarily for maximum efficiency at maximum velocity.

Hence $$\frac{G}{W + G} = 3.56 x \frac{1}{13}\left(1 + \frac{V^2}{10000}\right)$$

$$= 0.274\left(1 + \frac{V^2}{10000}\right).$$

The consideration of Table I. justified an assumption of $\frac{G}{W + G} = .4$

And $$0.4 = 0.247\left(1 + \frac{V^2}{10000}\right)$$

$$1.46 = 1 + \frac{V^2}{10000}$$

Therefore $V^2 = 46000$ and $V = 68$ mph.

This speed is too slow for comfort asthe trip across the Atlantic would require thirty hours. The gasoline weight would be

$$30 \times 2x35 \times 6 = 12600 \text{ pounds}$$

and this means a machine weight of 31,500 pounds. When this is compared with the NC-4 which weighs 29,000 pounds full and has difficulty in getting off the water with four Liberty motors as against two for this plane, this estimate will fail on account of water resistance previous to complete air sustentation. However, let us investigate the effect of changing the live load ratio from 40% to 50%. This is not an impossible condition and a calculation similar to the one just previous will give us a speed of 91 miles per hour. This means twenty two hours flight, that is 9250 pounds of gasoline with a total machine weight of 18,500 pounds. The load per horsepower is 23 lbs. which is not impossible judging by the performance of present land airplanes.

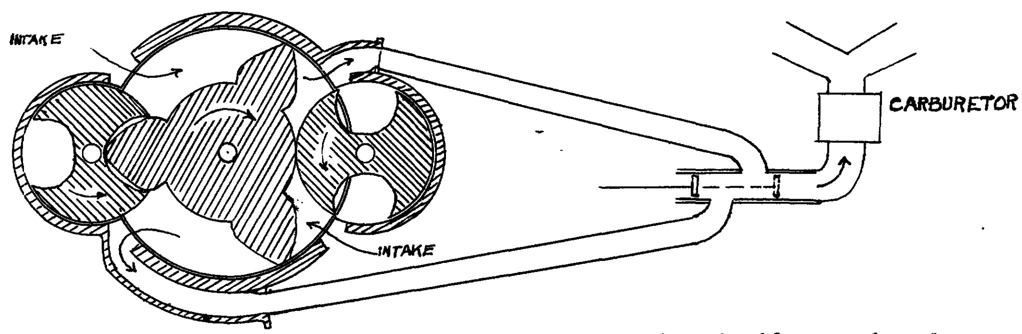
Altitude of Flight.

The remaining general question is that of desirable altitude for trans-Atlantic flight. The advantageous trade winds for the Eastern trip usually increase with altitude above three thousand feet.

The decrease of resistance with increase of altitude due to the falling off of the atmospheric density also encourages altitude work except for the comfort of the crew and the natural loss of power with altitude.

This loss of power can be overcome by supercharging the motor to maintain full atmospheric pressure in the engine. This principle has long been recognized but so far has not been used because of purely mechanical failure.

The power delivered by an internal combustion engine varies as the mass of gas burned per unit of time. As the atmosphere becomes less dense with altitude the actual weight of gas sucked into each cylinder is correspondingly less and the power falls off practically as the density of the air. Now, if a pump be attached to the air intake and a volume of air in excess of the actual displacement of the pistons be forced into the motor it is obvious that the air pressure in the cylinders can be maintained constant regardless of the external pressure by simply making the displacement of the pump inversely proportional to the decrease of atmospheric pressure. So far, centrifugal blowers have not been a success on account of the difficulty of driving them at the requisite high velocities, and the decrease of weight of air on which the action of the centrifugal blowers depend. A standard piston pumpis quite out of the question because of size: for if we would maintain full pressure at one-half density the pump must be half as large as the full motor even if double acting. For this reason I have developed a true rotary pump from the impeller and gear pump types, with enormous displacement and no external forces other than the torque of compression.



If the heat used in the motor remains constant through change of altitude, the decrease of atmospheric pressure would cause an increase in the power delivered by the motor on account of the decrease of back pressure. This normal increase of power will offset the power required to drive the supercharging pump. Hence it may be considered practical to calculate on full power regardless of altitude.

The effect of altitude on the resistance is evidently directly proportional to the density, for let us consider the fundamental formula where $R = K\rho AV^2$. This applies to either the lift or the resistance. But the lift must remain constant, so, if we consider the machine to maintain a constant attitude, the the decrease in density ρ must be balanced by an increase of velocity squared (V^2). And this same equation applies to resistance so with unchanged attitude the resistance will also be constant as the velocity varies inversly as the square root of the density of the air. Now the power required for that resistance at sea level was KR_oV_o and the new power will be KR_oV_h where $V_h = V_o\sqrt{\frac{\rho_o}{\rho_h}}$. Hence the power required to drive the plane at the velocity V_h at an altitude corresponding to ρ_h is $KR_oV_o\sqrt{\frac{\rho_o}{\rho_h}}$, whereas the power required to drive the plane at sea level with the velocity V_x is KR_1V_x and $R_1:R_o = V_x^2:V_o^2$ assuming constant coefficient of resistance.

Hence $$HP_x = KR_o\frac{V_x^2}{V_o^2}V_x = KR_o\frac{V_x^3}{V_o^2}$$

and equals $$HP_h = KR_oV_o\sqrt{\frac{\rho_o}{\rho_h}}$$ since the power is constant.

Or $$V_o^3\sqrt{\frac{\rho_o}{\rho_h}} = V_x^3 \qquad \text{but } V_o = V_h\sqrt{\frac{\rho_h}{\rho_o}}$$

Therefore $$V_h^3\frac{\rho_h}{\rho_o} = V_x^3$$

And $$V_h^3:V_x^3 = \rho_o:\rho_h \ .$$

Thus, with full power at all altitudes, the possible increase of speed varies inversely as the cube root of the decrease of density.

A standard graph of density variation with altitude is presented herewith and on that is plotted the percentage gain of velocity with altitude made possible by having full power available. This speed calculation is inaccurate insofar as the

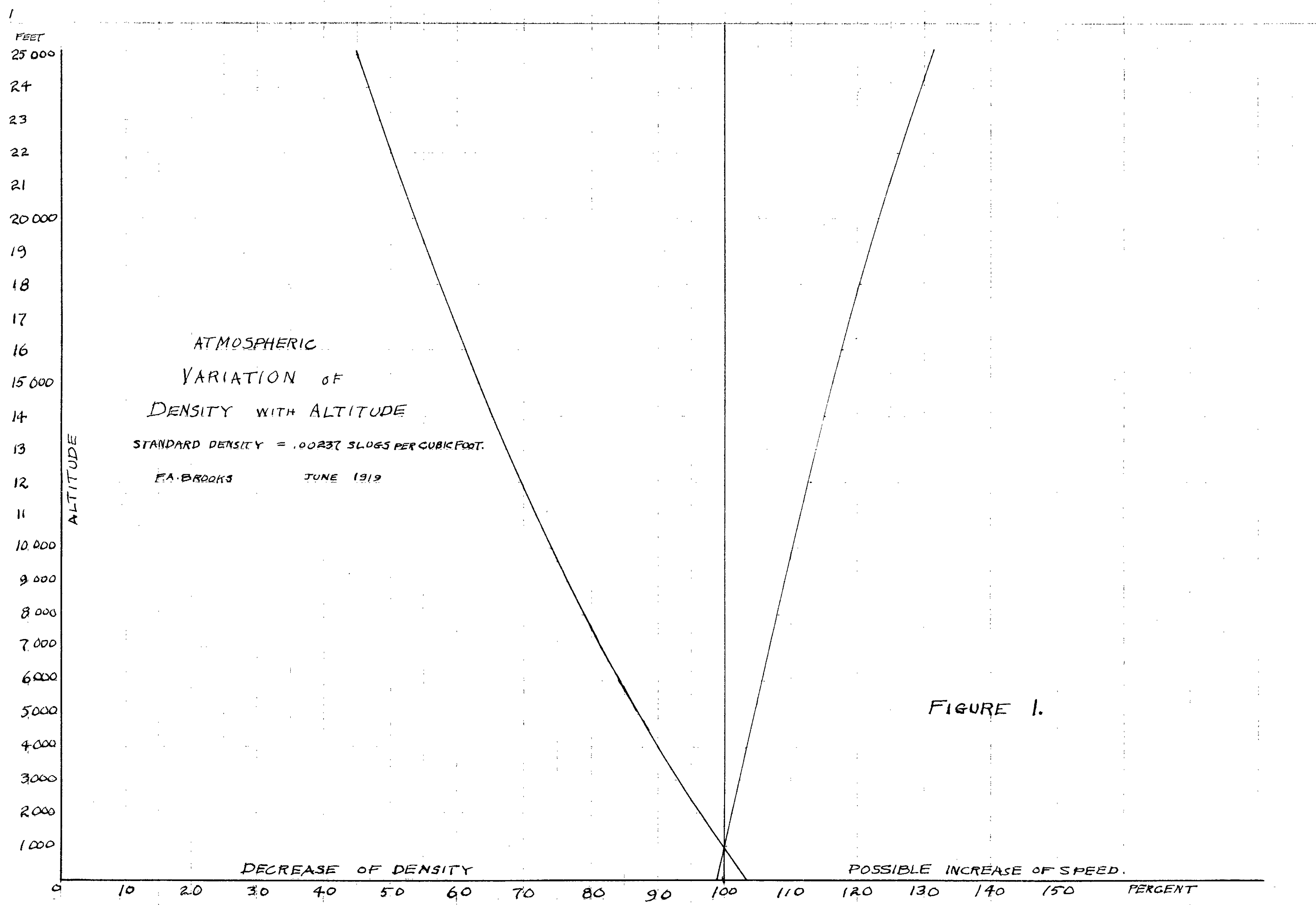

ATMOSPHERIC
VARIATION OF
DENSITY WITH ALTITUDE
STANDARD DENSITY = .00237 SLUGS PER CUBIC FOOT.
F.A. BROOKS
JUNE 1919
FEET
25 000
24
23
22
21
20 000
19
18
17
16
15 000
14
13
12
11
10 000
9 000
8 000
7 000
6 000
5 000
4 000
3 000
2 000
1 000
ALTITUDE
DECREASE OF DENSITY
POSSIBLE INCREASE OF SPEED.
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
PERCENT

FIGURE 1.

coefficient for wing drag is not a constant, but applying it to small changes of velocity the error is negligible. Figure 1.

The limitations of altitude is again fixed by extraneous considerations. It is obviously undesirable to place the crew in an air-tight cabin and otherwise the altitude is limited by their comfort. Hawker was piloting the Sopwith plane at 15,000 feet in his trans-Atlantic attempt, but this strain is too great to be considered practical for air transportation. We can arbitrarily assume an altitude of 10,000 feet as being a reasonable choice between possible shorter time of flight and greater personal comfort.

Conclusion.

Thus the estimated specifications for a trans-Atlantic seaplane are:

Total weight	=	18,500 lbs.
Gasoline (22 hours)	=	9250 lbs.
Motors, 2 Liberty 400	=	800 HP.
Maximum speed (sea level)	=	91 mph.
Range (sea level)	=	2000 miles.
Crew	=	3 men.
Wing structure	=	Triplane .
Pontoons	=	double .
Desirable altitude	=	10,000 ft.

An airplane of this type is quite practical and will meet the requirements of trans-Atlantic flight if the detail design keeps within the limits of the foregoing estimate. As a matter of fact the gasoline load is heavily overestimated as it is based on the assumption that the motors will operate continually at fullpower carrying a full weight plane the whole time at maximum speed. All of these assumptions are on the safe side and are left as safety factors. However, the final performance curves may show these allowances to be excessive and thus permit the reduction of live load and machine weight giving a faster airplane with better climbing ability and maneuvering qualities.

GENERAL DESIGN.

Wings.

The most important factor of an aerofoil is the Lift to Drag ratio. From the viewpoint of air power requirement it is obvious that with an overall machine efficiency of say 9 to 1 it is advisable to save one pound of resistance even if it cost 8 pounds in weight. If we had the choice of two equally efficient aerofoils with maximum L/D at different lift coefficients it would save weight to use the wing whose maximum L/D is at the higher lift as this would mean less wing area required. However, none of the high lift aerofoils come near the efficiency of the RAF 15 which is a widely used high speed aerofoil. Even this section in comparison with the RAF 14 or the Sloane wing has its maximum L/D at a relatively high lift coefficient. Hence, judging from the information available, the RAF 15 is most advantageous for this design. The characteristic coefficients are shown in figure 2.

For the determination of wing area we have:

$$L = K_y \rho A V^2 x\ K_t$$

where

L = total lift = 18,500 lbs.

K_y= lift coefficient = .00085 at 2°5, L/D=19.

ρ = density %standard= .74 at 10,000 ft.

A = wing area

V = air speed = 100 mph, at 10,000 ft.

K_t= triplane Lfactor = .76

or

$$18{,}500 = .00085 x .74 x A x 10000 x .76$$

and

$$A = \frac{18{,}500}{4.78} = 3{,}880 \text{ square feet.}$$

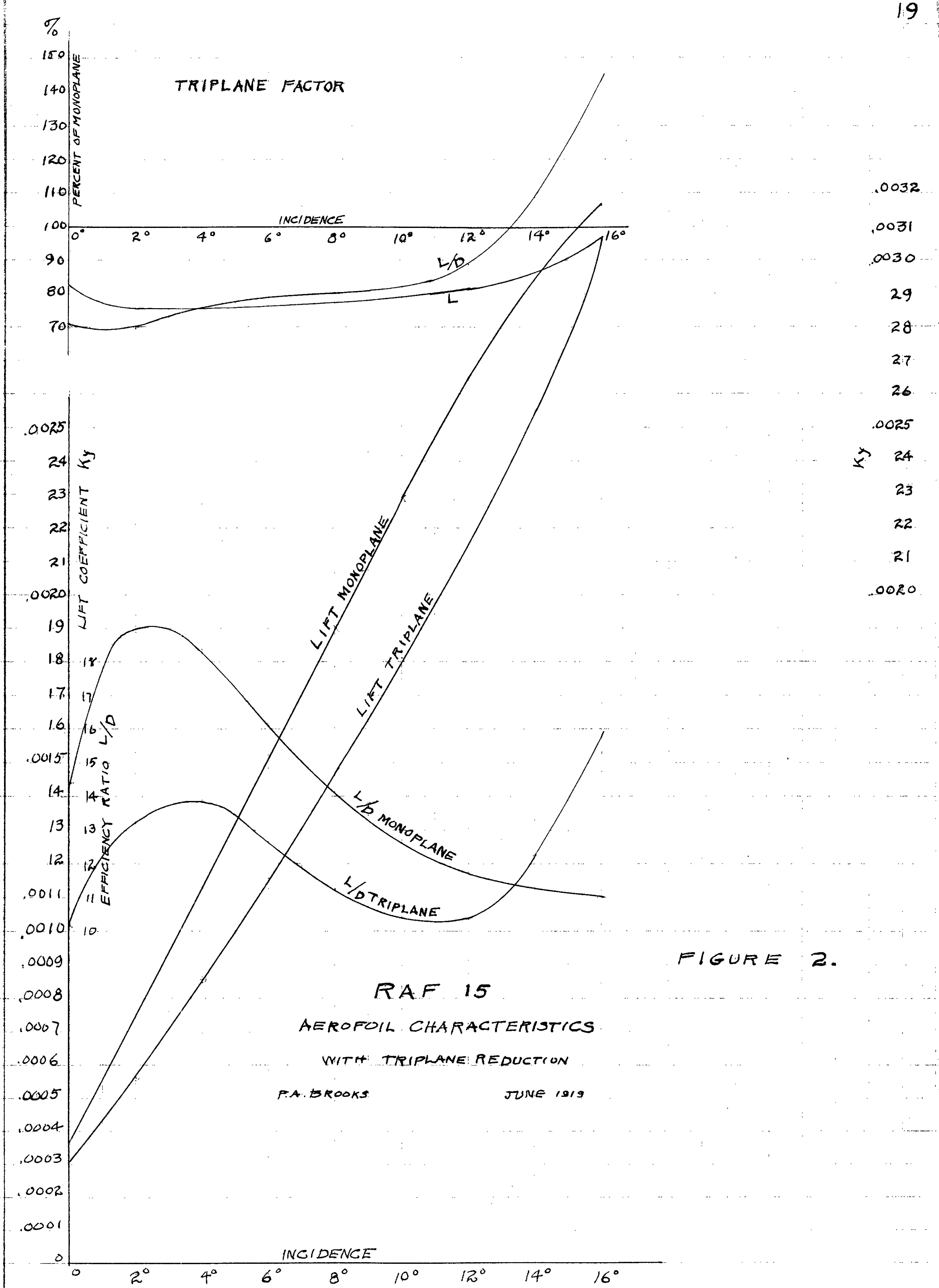


FIGURE 2.

RAF 15

AEROFOIL CHARACTERISTICS

WITH TRIPLANE REDUCTION

F.A. BROOKS JUNE 1919

With a wing loading of only 5 lbs. per square foot 20% of the lifting capacity would be used just to support the wings according to the rough estimate of 1 lb./sq.ft. for wing weight. Using an L/D of 18 which is still above the 17 assumed in the estimate design, we find a K_y = .0012 at 4°.3 then

$$18{,}500 = .0012 \times .74 \times A \times 10000 \times .76$$

$$A = \frac{18{,}500}{6.75} = 2750 \text{ sq.ft.}$$

which is more reasonable. The difference of wingweight ought nearly be 1130 lbs. and the increase of resistance $\frac{18{,}500}{18x.76} - \frac{18{,}500}{19x.76}$ which amounts to 68,5 lbs. which corresponds to nearly 615 pounds in terms of weight, so this saving of 515 lbs. is justified. Going a step further and using an L/D of 17 the result is 2470 sq.ft. or a saving of 280 pounds.in the wings and an increase of resistance weight of 711 pounds making a net loss of 431 lbs. On this basis an L/D of 17.5 should yeild the most economical wing area

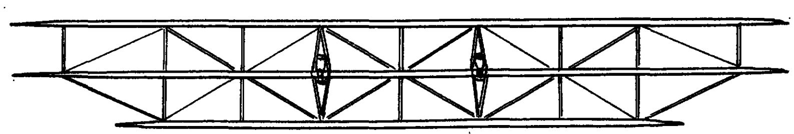
Hence

$$18{,}500 = .00128 \times .74 \times A \times 10000 \times .76 \qquad \text{at } 4°.75$$

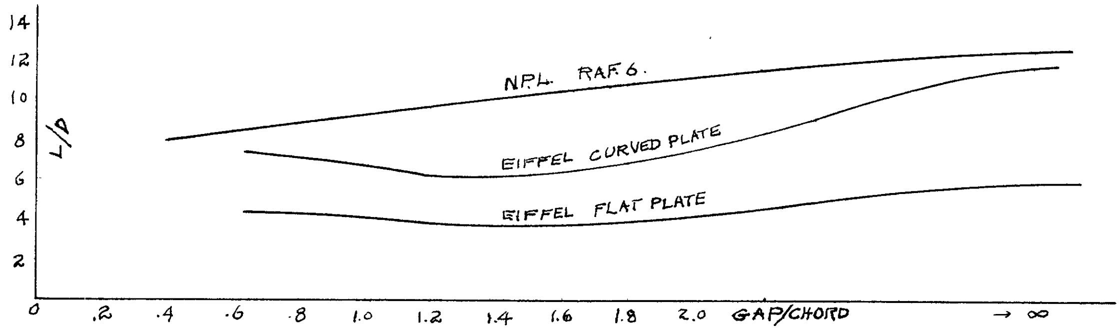
$$A = \frac{18{,}500}{7.2} = 2{,}570 \text{ square feet.}$$

Roughly this calls for three wings each say 9ft. x 95ft.

In setting these wings up, bearing in mind the triplane wiring system, it is desirable to have one wiring bay between motors, one outside,and then for added area the two upper wings can be conveniently extended further.



The choice of wiring angle must be chosen between the advantages of the desirable 45° for structural reasons, and the extended spacing of struts to reduce their number. This forces a consideration of gap-chord ratio. Unfortunately the tests at the British National Physical Laboratory do not agree with the tests made by Eiffel in Paris. A rough comparison of their results is given herewith:



There is not much choice to make between them because Eiffel ran his test at a higher velocity although with poorer wing sections. Hunsaker's triplane tests show an actual increase at high angles of incidence over a monoplane wing, hence an improvement in a combined action between wings may offset the mutual interference. On the basis of the design of one American battleplane with a gap-chord ratio of .65, in collaboration with Orville Wright, which flew well, handled easily and performed finely, the design of this trans-Atlantic plane will be carried forward on the assumption of a gap-chord ratio = 2/3.

Bearing in mind that a large aspect ratio for the wings is advantageous aerodynamically but heavier for structural reasons the complete wing structure can planned in general.

The consideration as to the setting of the wings can be discussed under incidence, stagger and dihedral.

The question of incidence is difficult concerning possible decalage between wings. The center surface exerting only half the lift of the upper wing leads one to consider the effect of increasing the incidence of the center wing only. This might be done to bring its lift up to that of the upper surface or to the value corresponding to the maximum L/D for a single wing. The latter change would be made if it were assumed that the interference of the wings was in the nature of change of relative wind direction, which is partially true, but for the machine under consideration the total effective lift coefficient is more that twice the maximum efficiency lift, hence the center plane is already at too high an angle of incidence for the application of this theory. The raising of the lift to that of the upper plane is used to equalize the panel loading and would be disadvantageous for tha total wing efficiency, so for the sake of simplicity we shall set the wings all at the same angle. This angle should be 4°.75 according to our previous calculations, but it may be found desirable to operate the airplane below its maximum speed and then a higher incidence found advisable.

If the wing structure has a stagger the wing beam stress will increase as the secant of the angle of stagger, because the weight of the machine acts vertically and if this be supported by an inclined truss the weight will be only the vertical component of the force in the truss. Furthermore the advantage of the stagger was found by the N.P.L. in case of gap-chord ratio of unity or more, the stagger aiding the lower wing to act more independently of the upper. Inthe case of the narrow gap here used where the wings must act as a combination rather than individually, our information is so scarce that one is left purely to guess. For instance, the Eiffel tests were made with aerofoils of uniform thickness throughout and hence were parallel bottom to top, whereas the RAF 6 tested by the N. P. L. has considerable thicknessand must cause some venturi effect between planes. In fitting the top of the RAF15 to the bottom they are found to coincide pretty well at a stagger of .07 of the chord when the wings are at zero incidence. With an incidence of nearly five degrees we get the same relation between surfaces at practically no stagger as it is measured along the wind. Hence, for simplicity we shall use no stagger.

As for dihedral, the War produced some very peculiar differences in fundamental design of the various Nations. The British used an excessive dihedral angle on most of their planes, wheras the French used none,and the Germans little if any. The American flying boats have practically no dihedral as the weight is carried pretty low which has some inherent stability. The main value of a dihedralis in coming out of a side slip and the American planes have generally used about a 2° angle. The action of such a dihedral is quite noticeable though not bothersome while maneuvering in mild fashion. Since a pontoon seaplane does not carry the main load below the wings as with the flying boat, its center of gravity is not very low in comparison with the land planes. So to offset the side resistance of the pontoons below the center of gravity of the plane it will be very desirable to use some dihedral in the wings. A small dihedral on the lower plane only would not produce much effect against the straight upper wings, hence, by inclining all three planes we can save complications in strut length and erection. With a small dihedral of say 2° the wings will have that sweep which si so attractive on boats, but it will be of advantage to use a larger dihedral on account of wing tip water clearance. The loss of lift of a four degree dihedral is quite noticeable while gliding in a Curtiss JN-4A as against the one degree angle of the JN-4D, but the three degree dihedral of the Curtiss R machine has been quite satisfactory and will be used here.

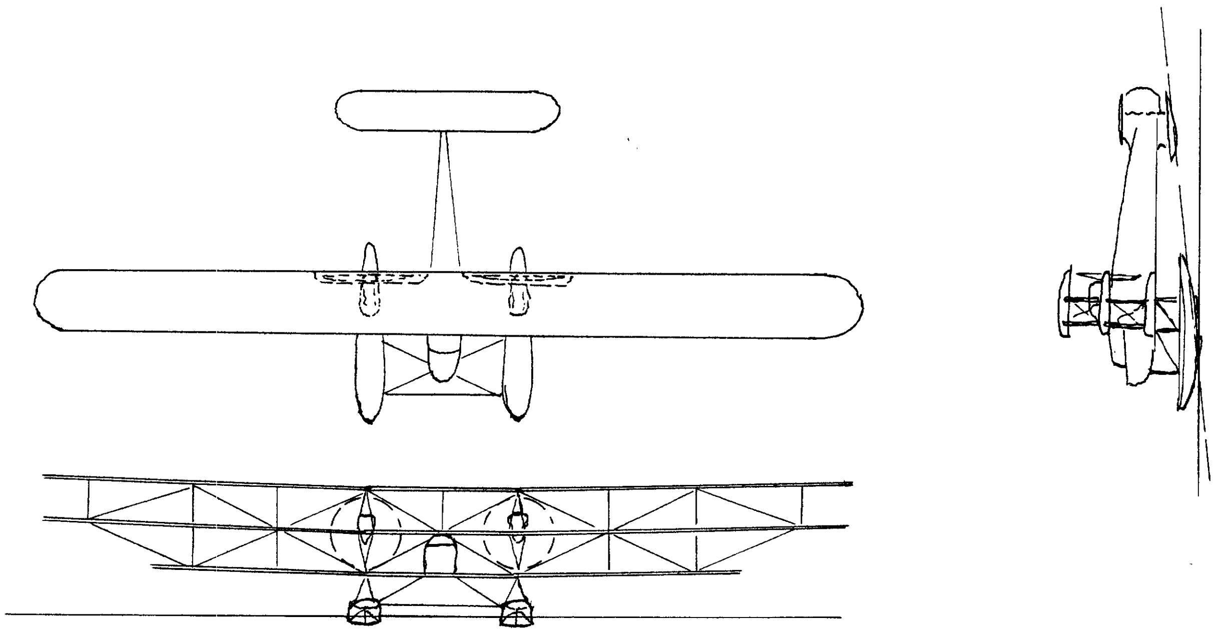
The ailerons on such a large craft as this must have plenty of surface and must be balanced in order to permit hand operation. The ailerons are used to offset the effect of gusts tending to tip the plane, and to tip the plane up into a bank for a turn. For the first use they act against aerodynamic forces which are proportional to the surface of themachine and hence the ailerons on a large plane should be the same fraction of the total wing aera as that of successful smallplanes. For the second use they shift the center of lift to produce tilting. If the ailerons are of constant proportion to the wing area the center of lift will also move proportionally and the moment produced will be increased by the weight ratio and the ratio of linear dimensions. This acts against the increased moment of inertia which varies as the weight ratio and the square of the linear dimension ratio. Thus the aileron area proportional to the wing area will be quite satisfactory for aerodynamic equalization but will act a little slower in banking. The banking, however, is cared for by the generous dihedral angle so we shall an aileron area of 10% based on the following data:

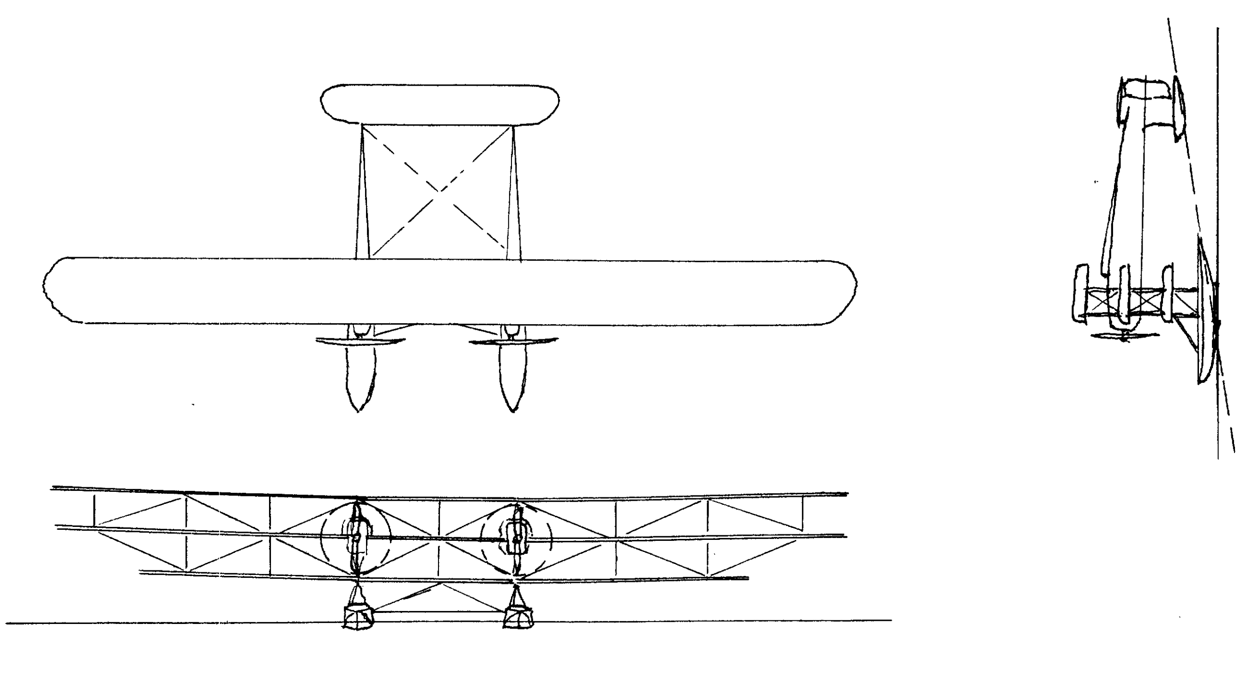
Machine		Wing area	aileron%
Curtiss	H-12	1120 sq,ft.	10.3
Curtiss	HS-1	576	9.4
Curtiss	JN-4A	325	10.8
Curtiss	N-9	496	11.0
Curtiss	R-4	504	10.7
Curtiss	T	2813	6.9
SPAD		225	9.3
Bristol		416	7.0

Fuselage.

The placing of the motors is a real question. First: pusher or tractor? The advantages of the pusher are: no slip-stream resistance, convenient radiator mounting and short stream line cowl for the motors. Its disadvantage is in the loss of twenty feet of trailing edge of the center plane. The tractor, commonly used on account of all War breakage falling away from the propeller, is inefficient on account of the motor being right in the slip-stream and hence requiring fine cowling, the difficulty of properly locating a large enough radiator, and the tearing of the wing covering by the pulsation of the air pressure directly behind the propeller. Its advantage lies in slight increase of propeller efficiency by keeping the air stream out from the hub. Thus a trans-Atlantic plane might be built with either two tractor motors set in separate fuselages as in one of the Caproni types, or with two pusher motors outside a single fuselage as in the german Gotha.

The double fuselage machine is stronger for the same weight and the center bay of wings, which is the most efficient, is left free. The nacelle type keeps the crew together, permits of better weight disposal, gives the observers an ideal range of vision and is away from the noise of the motors. Hence the question reduces to one of personal preference. Sketching the two types, (see next page) we find that the tail unit for the bi-fuselage plane works out finely, but the desirability of having the pilot forward and in the center can hardly be over-ridden. So, one is tempted to combine the two types adding a simple nacelle to the bi-fuselage airplane. This is certainly highly favored by pilots and is according to current practice. The center nacelle invites the installation of a third motor to be available for getting off the water and as a spare engine. This additional weight of 1000lbs. will only be bothersome in starting from the water at full load, and a 50% increase of power at that critical time will yield a large net advantage.





Pontoons.

The main pontoons can be set so that the plane will ride level on them with the tail free of the water, or they may be set ahead of the center of gravity and necessitate tail support. For heavy seas naturally accompanied by high winds, it is obviously unwise to try to balance a tall plane by the movement of the center of buoyancy forward and aft along the pontoons as the plane pitches when we have the opportunity to provide a tail pontoon forming a system of three point support with the center of gravity well in the center of the system.

To support this tail pontoon between the two fuselages which are 18 feet apart, it is evident that a biplane tail structure is called for and this again is in accord with present practice.

Tail Unit.

The location of the tail unit is determined mainly by comparison with successful planes. A long tail gives better stability both for correction of attitude and damping of oscillations, but it is heavier, and requires more housing space. Since a biplane tail structure encourages excess tail area, the accepted distance of twice the eing chord aft of the trailing edge of tha main wing can be used without question.

The type of aerofoil used for the tail , and its setting is governed by its effect on maneuvering. First,the incidence, considering the down draft from the wings , must be less than that of the main wings in order that for a given change of incidence the change of lift on the tail will overbalance the change of center of pressure on the lifting surface. Second, for diving the

the tail surface should have a natural negative lift in order to automatically bring the plane out of a dive right side up. This is allied with the previous requisites for stability but the war planes used for this latter reason a no-lift aerofoil. But this means that the tail weight must be carried by the main surfaces whereas it has plenty of lifting capacity to at least sustain itself were it allowed sufficient incidence. Furthermore, since the resistance of an aerofoil at low incidence is nearly constant this lift might be procured at no expense and actually relieve the load of the main surfaces thus reducing their drag. Both Eiffel and the N.P.L. agree that the downwash behind the wings is more than half their incidence, hence, if the tail surface be set at 3° it may be assumed to lie in the air stream. This would give an effective decalage of 4°.75 in normal flight and at least 1°.75 in a straight dive, so this may be considered satisfactory pending further calculations. By using a lifting section, the tail lift even at zero incidence will undoubtedly be sufficient to carry its own weight.

Tail areas always require actual experimentation for correct proportioning. An approximatio can be made for the control surfaces in the same manner as ailerons. A calculation can be made to determine the area required for the horizontal stabilizer to overbalance the change of center of lift. The vertical surface can be judged by visualizing the whole plane as a weathervane pivoted at the center of gravity.

For rudder and elevator proportions we have a choice of 3% for the rudder area, and 5.5% for elevator area based on the following table:-

Machine		Wing area	Rudder %	Elevator %
Curtiss	H-12	1120	2.89	4.94
Curtiss	HS-1	576	3.41	7.92
Curtiss	JN-4A	325	3.7	6.78
Curtiss	N-9	496	2.42	4.44
Curtiss	R-4	504	3.27	4.27
Curtiss	T	2813	2.43	4.72
SPAD		225	3.2	5.7
Bristol		416	1.73	5.5

With a three-bay tail structure four rudders can be mounted conveniently. These rudders can all be balanced as the use of 'K' frames will adequately care for the support of the rear beams. For the elevator the neatest design is that for a single plane with changeable incidence. Practically this is impossible but an enlargement of the Albatross elevator will give very nearly the same effect, and by setting this on the top plane of the tail unit it will be quite safe from damage by the sea.

On the lower surface it will be desirable to add another control surface similar to an elevator but moved by a mechanism which can be locked. In other words provide a secondary locking elevator to act as the present adjustable horizontal stabilizer of British planes. It is used to take up any unbalance that may arise from change of load or of flying attitude. This will enable the main elevator to be manipulated always from a neutral starting point and thus relieve the strain of operation.

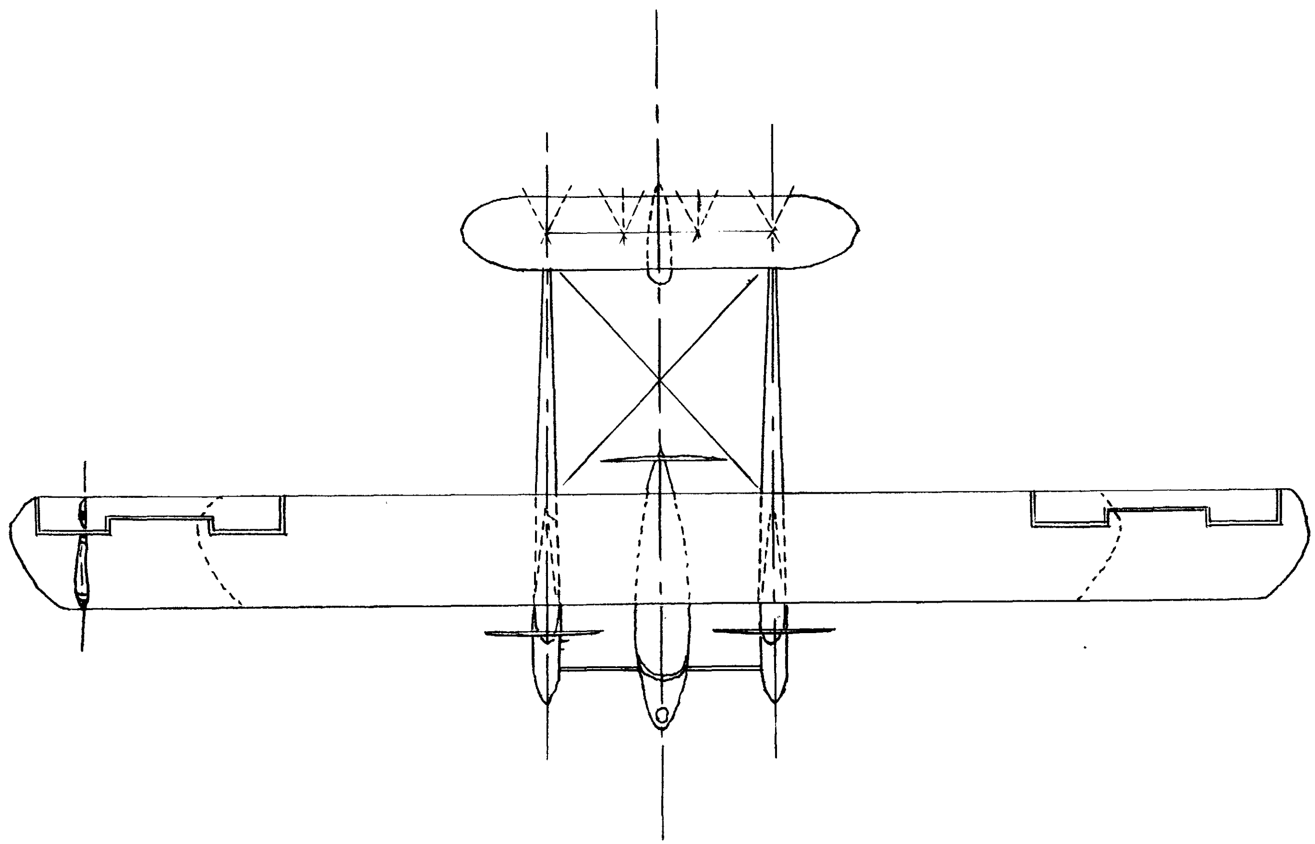
Motor Incidence.

The question of tail incidence immediately brings up the examination of the effect of the propeller blast upon that tail. With the engines normally horizontal the tail surfaces are at greater incidence to the propeller slip stream than to the down-wash from the wings. Then if the motors are shut down the lift of the tail will decrease producing a tendency to stall which is not desired, especially at that critical time. But to set the tail negative to produce a dive would simply load up the main surfaces during normal flight. The other way to counteract this stall is to set the motors below the center of resistance or to incline them at an angle greater than the doun-wash. An angle such as 4° having a cosine of .9976 and a sine of .0698 will reduce the horizontal thrust only 1/4 of one percent. The lift component of 7 % of the thrust may actually save more wing drag than there is decrease of horizontal thrust. For instance, at 100mph the thrust of 800 HP is 2250 pounds and the thrust lift =.07x2250 or 157 pounds. At the L/D of 13.3 this means a decrease of drag of 11.8 pounds, whereas the decrease of thrust is .0024x2250 or 5.4 pounds, leaving a net gain of 6.4 lbs.!

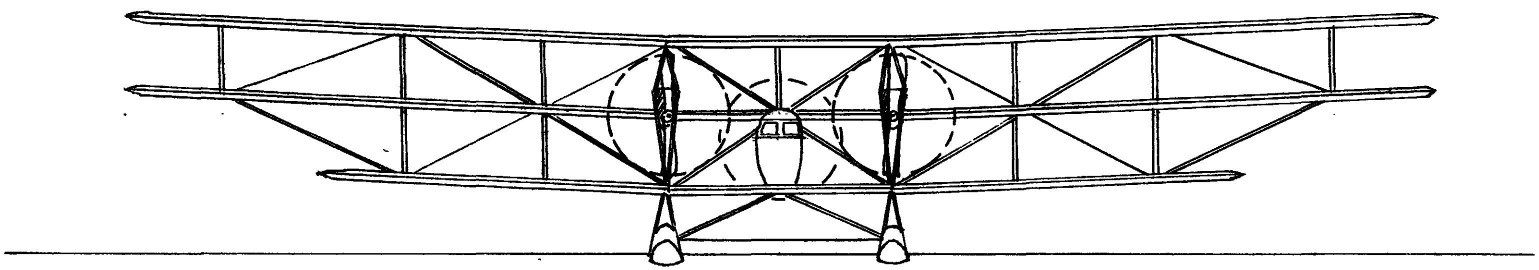
Such a calculation cannot be contradicted in spite of present machines never having such a motor setting. Of course, the gain of six pounds is not worth anything if it were not the bonus of procuring valuable maneuvering qualities.

Conclusion.

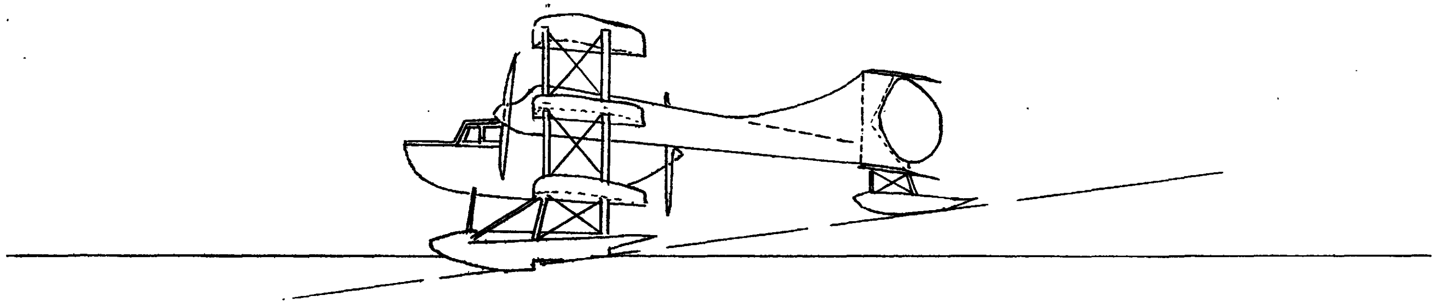
Summing up the General Design we have the following layout for a trans-Atlantic airplane.



~ PLAN VIEW ~



~ FRONT ELEVATION ~



- SIDE ELEVATION -

PERFORMANCE.

The performance of the airplane will depend primarily on the total weight and the parasite resistance.

In the absence of complete detail design and calculations which are out of place in such a thesis as this, we can estimate from experience the weights of the main units:-

Wings, 2570 sq.ft.	=	2400. lbs.
Fuselages, at 900	=	1800.
Main pontoons, at 500	=	1000.
Tail unit, (self supporting)	=	(800.)
Nacelle	=	600.
Motors, 3 at 825 + extras	=	3000.
Gasoline	=	9250.
Crew	=	500.
Total weight	=	19350. lbs.
Total wing lift	=	18550. lbs.

This can undoubtedly be cut down but we shall assume the first estimate of 18,500 pounds as a satisfactory basis for calculation.

The parasite resistance also depends on detail work which is omitted here, but the triplane wiring gives such a cleaner machine than the biplane of the same size that the average ratio, of one to one for parasite resistance to wing drag at 100mph, is certainly higher than necessary.

Hence, proceeding on the original estimates we calculate the sea level performance curves as follows:

Weight = 18,500 #

Wing Area = 2570 sq.ft, RAF 15

$$K_p = \frac{D}{L}\frac{W + G}{10000} = .1423 \text{ \#/mi,hr.}$$

$$W = K_y \rho A V^2$$

$$D = \frac{D}{L} \times W$$

$$R = K_p V^2$$

Total resistance = D + R

i°	K_y	L/D	V^2	V	Drag	Par. R.	Total R.	HP.req.
0°	.000307	10.2	24900	158.	1815	3540	5355	2255
2°	.000565	13.2	13550	116.	1395	1930	3325	1030
4°	.000855	13.9	8750	94.5	1330	1275	2605	655
6°	.00116	12.6	6600	81.3	1460	940	2400	520
8°	.00147	11.2	5200	72.2	1645	740	2395	461
10°	.00181	10.4	4230	65.	1780	603	2383	413
12°	.00215	10.4	3560	59.6	1780	507	2287	364
14°	.00255	12.3	3000	54.8	1505	427	1932	283
16°	.00306	15.9	2500	50.	1163	356	1519	203

These results are plotted in Figure 3.

The density at 10,000 feet is 74 % of standard, hence the velocity at that altitude, considering constant incidence, is

$$V_h = V_o\sqrt{.74} = \frac{V_o}{.86}$$

The power required is also increased by the same ratio. This curve and the required motor power including a propeller efficiency of 80 % are shown in Figure 4. Along with this is the Gasoline consumption curve per HorsePower from which the curve of gasoline

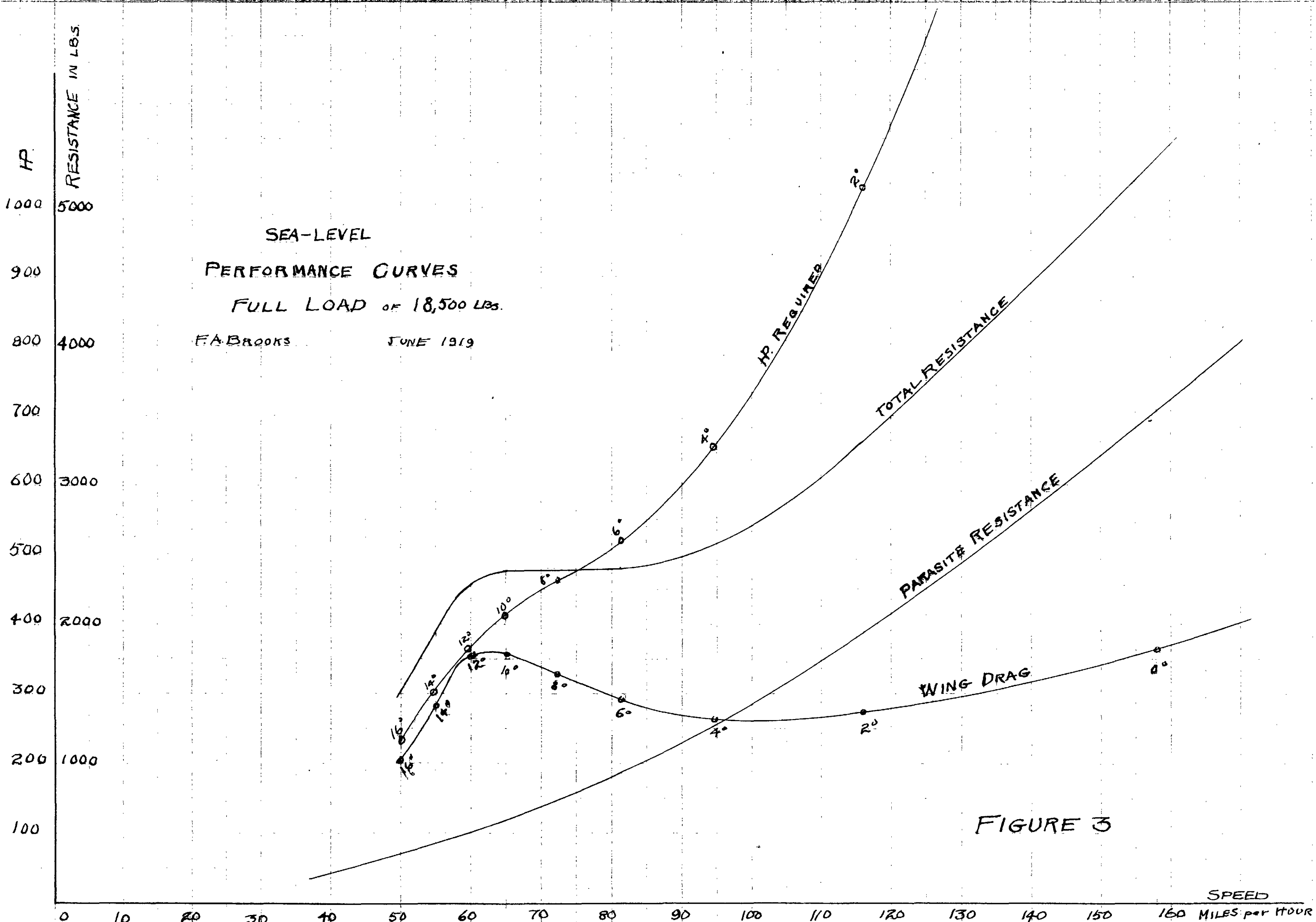

SEA-LEVEL
PERFORMANCE CURVES
FULL LOAD OF 18,500 LBS.
F.A. BROOKS
JUNE 1919
HP
RESISTANCE IN LBS.
1000 5000
900
800 4000
700
600 3000
500
400 2000
300
200 1000
100
0 10 20 30 40 50 60 70 80 90 100 110 120 130 140 150 160
SPEED
MILES per HOUR
HP. REQUIRED
TOTAL RESISTANCE
PARASITE RESISTANCE
WING DRAG
16° 14° 12° 10° 8° 6° 4° 2° 0°

FIGURE 3

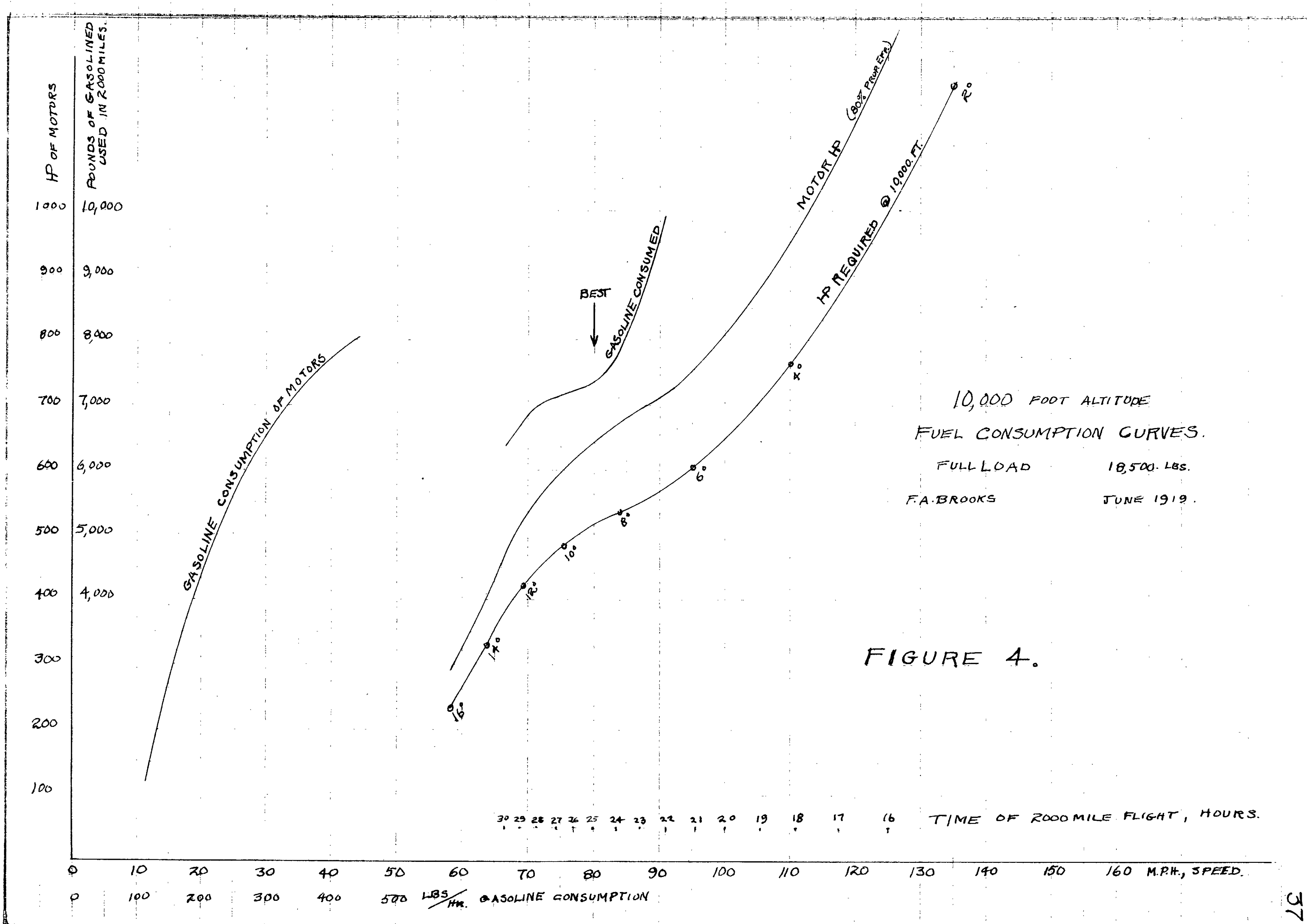

10,000 FOOT ALTITUDE
FUEL CONSUMPTION CURVES.
FULL LOAD 18,500. LBS.
F.A. BROOKS JUNE 1919.
HP OF MOTORS
POUNDS OF GASOLINE USED IN 2000 MILES.
1000 10,000
900 9,000
800 8,000
700 7,000
600 6,000
500 5,000
400 4,000
300
200
100
GASOLINE CONSUMPTION OF MOTORS
GASOLINE CONSUMED
BEST
MOTOR HP (80% PROP EFF.)
HP REQUIRED @ 10,000 FT.
30 29 28 27 26 25 24 23 22 21 20 19 18 17 16 TIME OF 2000 MILE FLIGHT, HOURS.
0 10 20 30 40 50 60 70 80 90 100 110 120 130 140 150 160 M.P.H., SPEED.
0 100 200 300 400 500 LBS/HR. GASOLINE CONSUMPTION

FIGURE 4.

consumed has been derived. The gasoline used is overestimated because the plane is considered fully loaded all the time and the speed uniform for the whole flight of 2000 miles.

This curve of gasolined consumed shows it advisable to fly at 80mph. which is rather slow. Also it shows only 7300 lbs. of fuel required whereas 9250 pounds were allowed for in the estimate design. This means that with a plane such as has been indicated herein the gasoline load need only be 40 % of the total weight of the airplane and thus coincides with the results of Table I. Furthermore, even this 7300 pounds of gasoline assumes fullload conditions which obviously allows a large margin of safety in the actual flight.

From the performance curves it will be noticed that at the desirable speed of 80 mph. the plane is operating under very poor conditions such as at nearly the worst wing efficiency and also at too much incidence for ease of control. Hence, it would be advisable to decrease the power loading. In other words by maintaining our improved live load factor of 50 % , which is within reason, and assuming a fuel load of 7,500 pounds we have a machine of total weight 15,000 pounds. This reduces the power loading from 23.1 to 18.75 pounds per horse-power eliminating the necessity of a third motor. The wing area might also be cut down and for a wing loading of 7.5 pounds per square foot the aera will be 2000 sq.ft. This wing loading may seem heavy but when the machine is light the loading goes below 4 $\#/ft^2$, which is extra-ordinary.

Such a reduction in wing area would hardly diminish the parasite resistance. In fact the reduction can best be made by cutting down the chord to give a better aspect ratio and a gap/chord ratio of 3/4 which is not so extreme.

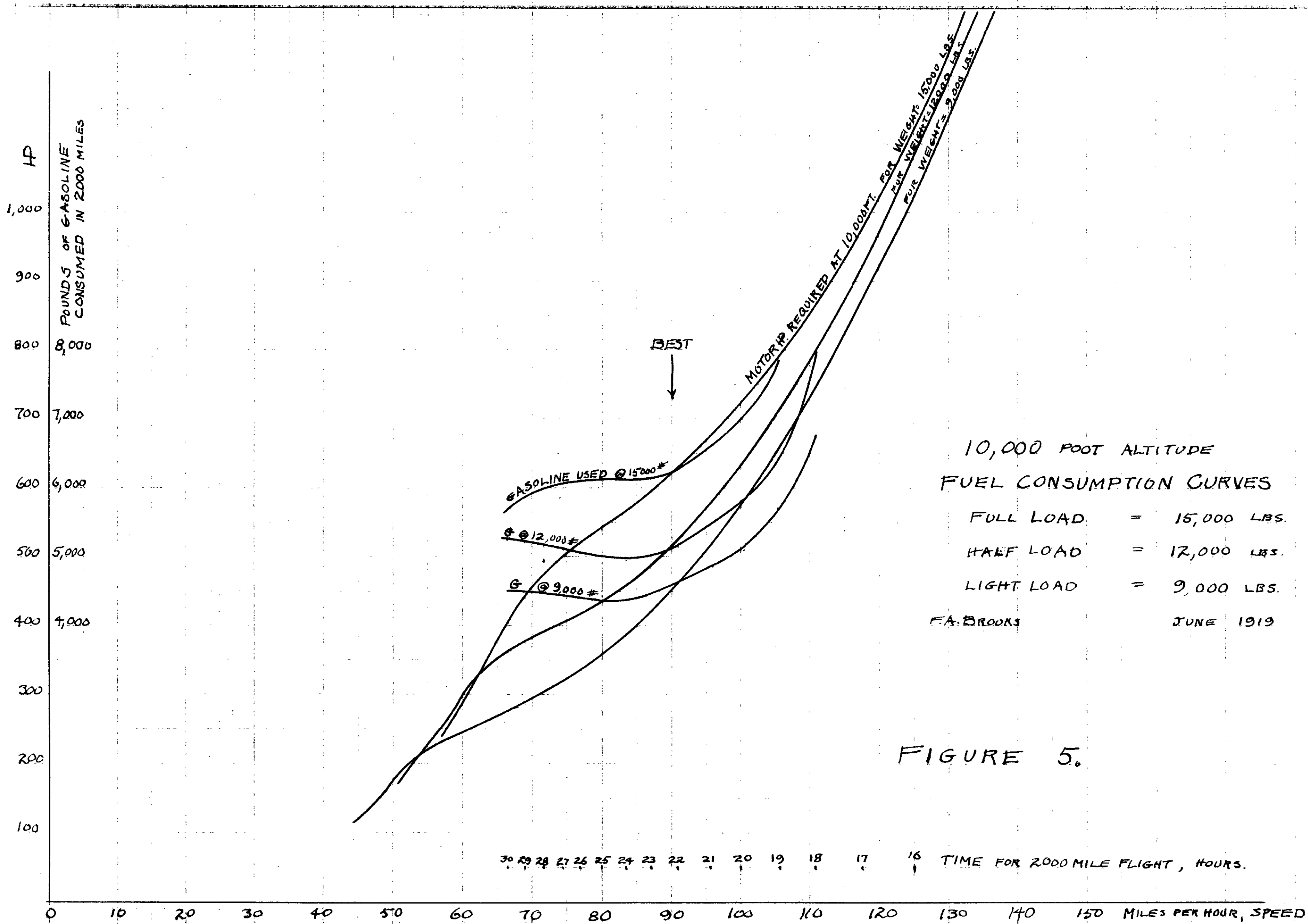
Thus the airplane has been corrected to:-

Total weight	= 15,000 lbs.
Area	= 2,000 sq.ft.
Power	= 800 HP.
Fuel	= 7,500 lbs.
K_p	= .14000 #/mi.hr.

Recalculating:-

i°	K_y	L/D	V^2	V	D	R	Tot.R.	V_h	HP_h	MotHP. (80%)
0°	.000307	10.2	24500	156.5	1470	3430	4900	182	2380	2970
2°	.000565	13.2	13250	115.	1132	1855	2987	134	1068	1335
4°	.000855	13.9	8780	93.7	1080	1230	2310	109	672	840
6°	.00116	12.6	6470	80.2	1185	905	2090	93.3	520	650
8°	.00147	11.25	5100	71.5	1333	714	2047	83.2	455	568
10°	.00181	10.4	4140	64.4	1440	578	2018	75	403	505
12°	.00215	10.4	3490	59.1	1440	487	1927	68.8	354	442
14°	.00255	12.3	2940	54.3	1220	409	1629	63.2	274	342
16°	.00306	15.9	2450	49.5	945	341	1286	57.5	197	246

Applying these results to the gasoline consumption curve per horse-power and converting the speed into the time required for a 2000 mile flight we procure the gasoline consumed curve according to the calculations on the following page. This is also carried out for a total weight of 12,000 and 9,000 pounds to get the characteristics of fuel consumption when the plane has used large quantities of fuel load. The curves are plotted in Figure 5.



10,000 FOOT ALTITUDE

FUEL CONSUMPTION CURVES

FULL LOAD = 15,000 LBS.

HALF LOAD = 12,000 LBS.

LIGHT LOAD = 9,000 LBS.

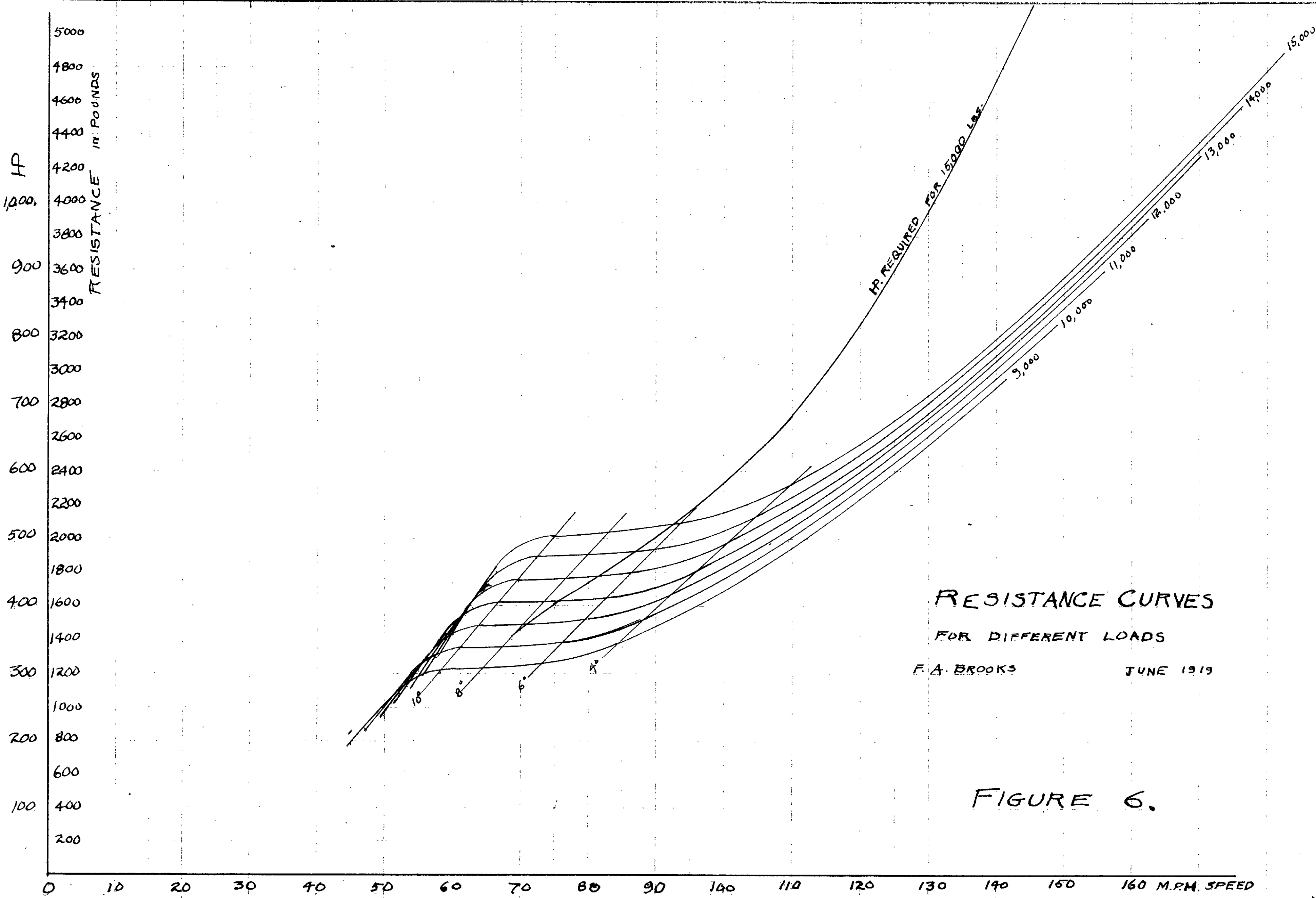
F.A. BROOKS JUNE 1919

FIGURE 5.

Time (hr)	HP 15000	Gas/hr. (lbs)	G 15000	HP 12000	Gas/hr (lbs)	G 12000	HP 9000	GAS/hr (lbs)	G 9000
30	410	190	5700	365	175	5250	275	150	4500
29	450	205	5950	375	180	5220	290	155	4500
28	475	213	5940	390	185	5180	305	160	4470
27	500	225	6070	405	190	5130	320	165	4450
26	525	235	6120	417	195	5070	340	170	4420
25	545	245	6130	435	200	5000	360	175	4370
24	565	255	6130	455	207	4970	385	183	4380
23	595	267	6140	490	220	5060	420	195	4480
22	630	285	6270	525	235	5160	465	210	4620
21	675	315	6620	580	260	5470	515	230	4830
20	720	350	7000	635	290	5800	575	255	5100
19	785	415	7890	710	340	6450	655	300	5700
18	865			800	440	7920	750	375	6750

These gasoline consumed curves show the most desirable speed to be 90 mph. regardless of the decrease of fuel load. This means that if the propeller be designed for maximum efficieny at 90 mph. and at 310 HP. for each motor we shall operate at practically the most efficient point of the plane as a whole, and this point being as high as ninety miles an hour will give a very satisfactory airplane in all respects.

To determine the performance curves on the basis of mileage, we need the individual curves for say every thousand pounds decrease of load from 15,000 total weight , to 9,000 empty weight. The curves are presented herewith in Figure 6 but the calculations are omitted as being uninteresting. The L/D at a



RESISTANCE CURVES
FOR DIFFERENT LOADS

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FIGURE 6.

a particular incidence is constant for all loads so the wing drag will decrease directly as the weight considering equal attitudes. The parasite resistance is proportional to V^2 and, for a particular incidence this V^2 must decrease as the weight the resistance also will decrease directly as the weight. Therefore the total resistance will fall off proportionally with the decrease of total weight at any definite incidence.

In order to maintain constant velocity during the flight the motor must be throttled down as the resistance diminishes from decrease of weight and this will produce a saving of fuel. To determine the fuel consumption it is necessary to calculate the motor revolutions per minute at the various weights. The thrust of the propeller must balance the resistance of the plane. This resistance for any speed has already been found as per Figure 6. From the propeller efficiency curve at 90 mph. shown in Figure 7 one can compute the thrust delivered by the motor at different rates of revolution and this is plotted also in Figure 7. Compar- this with the resistance curve of 90 mph. (Figure 6) we find a fuel consumption according to the following table:

Weight	RPM	HP	Gas/hr	hours $\left(\frac{1000}{G/hr}\right)$	Dist.	Mileage.
15,000	1475	610	274	3.65	328	328
14,000	1450	564	250	4.00	360	688
13,000	1425	550	236	4.24	381	1069
12,000	1405	520	228	4.39	395	1464
11,000	1382	496	222	4.53	408	1872
10,000	1365	476	214	4.67	420	2292
9,000	1348	460	208	4.81	433	2725

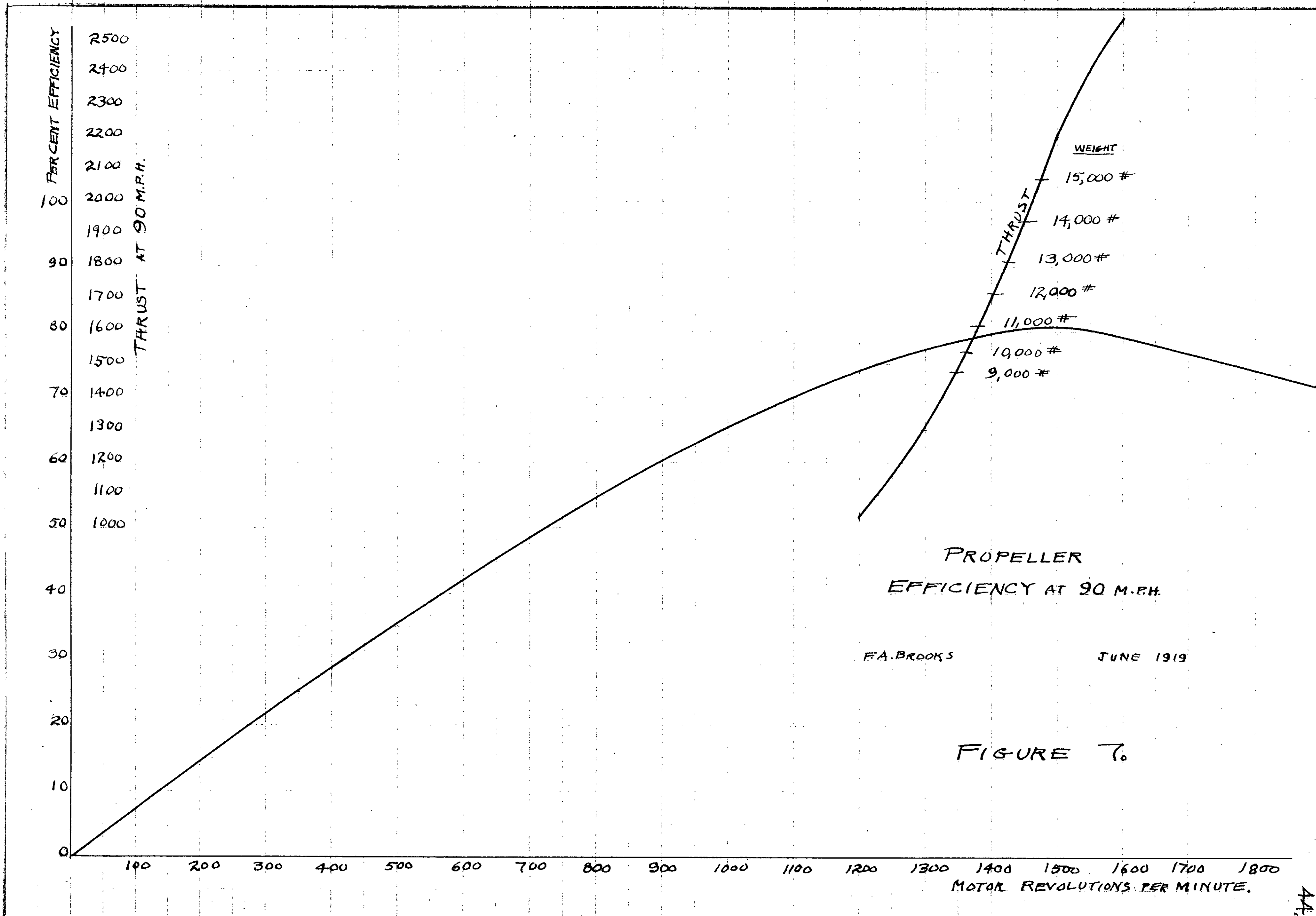
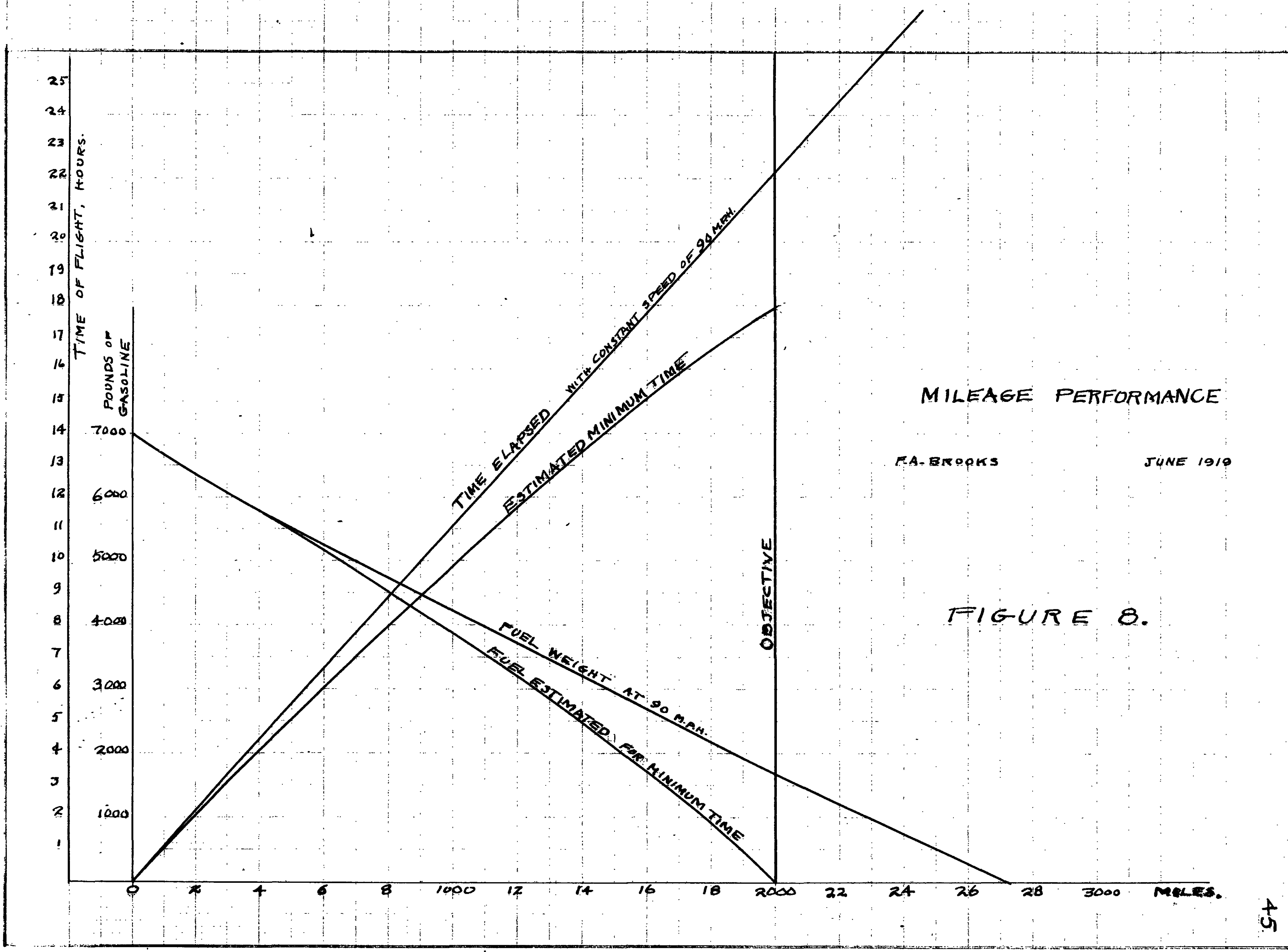

PERCENT EFFICIENCY
THRUST AT 90 M.P.H.
100
90
80
70
60
50
40
30
20
10
0
2500
2400
2300
2200
2100
2000
1900
1800
1700
1600
1500
1400
1300
1200
1100
1000
WEIGHT
15,000 #
14,000 #
13,000 #
12,000 #
11,000 #
10,000 #
9,000 #
THRUST
PROPELLER
EFFICIENCY AT 90 M.P.H.
F.A. BROOKS
JUNE 1919
100 200 300 400 500 600 700 800 900 1000 1100 1200 1300 1400 1500 1600 1700 1800
MOTOR REVOLUTIONS PER MINUTE.

FIGURE 7.



MILEAGE PERFORMANCE

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FIGURE 8.

This data is plotted in Figure 8. The fuel curve can be shifted up and down according to the weight of gasoline at the start provided the total machine weight remains constant. Thus if the primary assumption of 7,500 pounds of gasoline were carried out the plane will have a range of 3000miles . This would cover a straight flight from the United States to England if such a trip might be desired. But if the regular trip of 2000 miles were to be made in a hurry the motor might progressively be opened up, the speed of flight increased as the necessity for safety margins was diminished and the gasoline consumption curve would fall off more rapidly as is shown in the estimate curve in Figure 8. These two curves will normally indicate the extremes of operation for long-distance flying such as the trans-Atlantic flight.

CONCLUSION.

The design developed in this Thesis produces an airplane which meets the requirements of trans-Atlantic flight quite satisfactorily. It has an excess mileage reserve of at least one third of the prescribed distance of two thousand miles. The motors operate at three fourths of full power which is at the point of minimum fuel consumption and excellent reliability. The speed of flight is sufficiently high for overriding any such head winds as ordinarily occur and yet the maximum speed is enough greater than the cruising velocity to give plenty of climbing capacity and maneuvering ability. The wing loading and power loading is well within present day practice, and in no way can this seaplane be considered a freak.

The Vickers-Vimy Bomber which recently won the trans-Atlantic prize is rather similar to the plane outlined in this Thesis although a smaller, biplane, land machine.

The outstanding features of my design are the pontoon arrangement, the triplane structure, the use of the RAF 15 aerofoil, the inclination of the motors and the equipment for altitude operation.

The design has been developed in a general theoretical manner avoiding involved mathematical treatment as inapplicable with the present empirical data, and omitting detail design as not being of sufficient interest. But the complete success of the aerodynamical calculations makes the future prospects of air transportation seem much more promising.

Submitted by [illegible]

June 21, 1919