AEROPLANE PERFORMANCE ESTIMATES.

Paper read by Mr. R. Chadwick, A.F.R.Ae.S. before the Institution at the Engineers' Club, Coventry Street, W., on the 8th February, 1924. Mr. S. T. G. Andrews in the Chair.

In calling upon the lecturer the CHAIRMAN remarked that Mr. Chadwick needed no formal introduction, as his work with Messrs. A. V. Roe and Co. was well known. The CHAIRMAN then read a telegram from Mr. A. V. Roe regretting his inability to be present at the meeting.

Mr. CHADWICK said :—

There are several methods for estimating the probable performance of a new aeroplane design, and we will consider three methods.

1st Method.—Prediction from formulæ or graphs obtained by averaging the performance test results of a large number of aeroplanes.

2nd Method.—Estimation of the performance by calculating the wing lift and drag from model tests and the drag of the remainder of the aeroplane from tests on component parts.

The propeller efficiency has also to be calculated and the performance obtained from the resulting values for h.p. required and h.p. available at all speed within the flying range.

3rd Method.—Estimation of the performance from the results of Model Tests in the Wind Tunnel on a complete scale model of the proposed aeroplane.

The first method is usually employed to make a preliminary estimate of the performance, and is particularly useful when considering the possibility of meeting a prospective purchaser's requirements.

By the first method a close approximation can be made of the probable performance without the necessity of preparing any drawings.

The graphs used in the first method are also useful in considering the effect on the performance of an existing aeroplane, of varying the load carried.

The second method is more laborious, and is usually employed to make a detailed estimate when the design has been more or less settled.

It is necessary to use this method when looking into the effect on the performance of varying the Aerofoil Section, propeller dimensions, wing bracing arrangement, etc.

The Aerofoil Model Test figures should be taken from models as like the arrangement and shape of the full-scale wing as possible.

For example, if it is proposed to use a biplane arrangement of wings with rounded tips and aspect ratio 7, then the model should have the same proportions, so as to reduce the amount of correction from model to full scale.

Unfortunately, it is seldom possible to do this, and we have to make numerous corrections when using existing data, as will be seen later.

The third method is not frequently resorted to, unless the proposed design is a considerable departure from the average type, or it is required to obtain special information regarding the proposed design.

The method is expensive, and takes a considerable time before the model test results are available.

In any case it is necessary to make an estimate by the first or second method, and a general arrangement drawing must be prepared before the model can be made.

Perhaps the best way of considering the first and second methods of performance estimation will be to take an example and work it out.

In order to save time we will take a straightforward proposition in which the paying load consists of mails or goods, and we will state only that part of the specification which concerns the performance.

We will also assume that the aeroplane will be of average type and proportions.

The specification may be something as follows :—

Specification of Required Performance.

Paying Load	700 lbs.
Crew (2 @ 180 lbs.)	360 lbs.
Fuel (70 galls.)	510 lbs.
Oil (4 galls.)	40 lbs.
Instruments	25 lbs.
	1,635 lbs.

Performance with Full Load.

Speed at Sea Level not less than	110 m.p.h.
Speed at 10,000 ft. not less than	105 m.p.h.
Landing Speed not to exceed	50 m.p.h.
Service Ceiling not less than	17,500 feet.
Time to 10,000 ft. not more than	16 mts.

Engine.

The engine to be used is a Rolls-Royce "Eagle," of which the following particulars can be taken :—

Weight dry	900 lbs.
Normal h.p. at 1,650 r.p.m. (crankshaft)	330 b.h.p.
Max. h.p. at 1,800 r.p.m. (crankshaft)	350 b.h.p.
Normal Engine Speed...	1,650 r.p.m.
Max. Engine Speed	1,800 r.p.m.
Reduction Gear Ratio	.6 to 1
Fuel Consumption (normal power)	23 galls./hr.
Oil Consumption	.025 pts./h.p. hr.

Having the requirements of the specification before us we proceed as follows, using the *First Method* :—

First of all we visualise the type of aeroplane which we consider most suitable for the work it is intended to perform, and generally our visions are influenced by thoughts of machines we have had experience of and which had round about the required performance.

Let us suppose we conclude that the requirements can be met by either a monoplane or a biplane, then we will choose the biplane as it offers a better example for the purpose of this paper.

The next step is to estimate the weight of the complete aeroplane, and to assist us in this we make use of collected data such as is shown in :—

*Fig. 1.—Weight of Honeycomb Radiators. Weight of Tanks.

Fig. 2.—Weight of Struts with Fork Ends. Weight of Struts with Wooden Fairings.

Fig. 3.—Weight of Wooden Propellers.

Fig. 4.—Weight of Oleo Undercarriages.

Fig. 5.—Weight of Aeroplane Components in percentage of Gross Weight of Aeroplane. Weight of Power Plants.

It is important to collect and tabulate as much information as possible on the subject of our component weights, as this class of information is invaluable when considering a new design.

For example, it is useful to plot variation of wing % weight against wing loading, etc.

In our machine the known weights are :—

(a) Load	... ...	1,635 lbs.
(b) Engine	... ...	900 lbs.

From Fig. 5 we see that the average weight of the engine accessories for a water-cooled engine is 1 lb/h.p. We therefore get

(c) Weight of Engine Accessories	... ...	350 lbs.

From Fig. 1 we find the weight of the tanks is

(d) Fuel Tank (70 galls. capacity)	... ...	60 lbs.
Oil Tank (4 galls. capacity)	... ...	16 lbs.

We now know the weights of the power plant and accessories and the tanks.

It remains for us to find the structure weight.

We see from Fig. 5 that the average structure weight for aeroplanes with standard load factors is 33 per cent. of the gross weight, and we will take this value for our example, so that we get

$$\begin{aligned}
& && \text{Load.} \quad \text{P.P.+Accs.} \quad \text{Tanks.} \quad \text{Structure.} \\
&\text{(e) Gross Weight} = W &&= 1{,}635 + 900 + 350 + 76 + .33W. \\
& &&= 2{,}961 + .33W. \\
&\text{Therefore } .67W &&= 2{,}961. \\
&\text{and } \quad W &&= \text{Gross Weight} = 4{,}421 \text{ lbs.}
\end{aligned}$$

* The figures indicated in the text will be found in numerical order commencing at page 29.

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The structure weight is therefore :—

(f) $.33W = .33 \times 4{,}421 = 1{,}460$ lbs.

We now have a complete weight estimate for the aeroplane.

(g)	Load	1,635 lbs.
	Structure	1,460 lbs.
	Power Plant	900 lbs.
	Power Plant Accessories	350 lbs.
	Fuel Tanks	60 lbs.
	Oil Tanks	16 lbs.
	Gross Weight	4,421 lbs.

The weights of the power plant accessories and also the structure weight are based on the assumption that the machine and engine installation is to be of average type.

If any marked departure from standard practice is contemplated, we should here make an allowance for it if we consider that a saving in weight will be effected.

We now proceed to the question of performance, and to assist us in arriving at a close estimate of this we have collected and plotted the results of a large number of performance tests on different aeroplanes.

These results can be shown on graphs of the type in following slides :—

Fig. 6.—Speeds with different wing and engine loadings.

Fig. 7.—Rate of climb and ceiling with different combinations of wing and engine loading.

Fig. 8.—Rate of climb curves.

Fig. 9.—Climb curves for average aeroplanes with good carburation at all heights.

Fig. 10.—Speed variation with altitude.

Fig. 11.—Landing speed and wing loading.

Using these curves we can obtain a close approximation of the probable performance of the aeroplane at all altitudes below the ceiling.

It should be noted that the curves in Figs. 7 and 9 assume Aerofoil Section R.A.F. 15. If a different aerofoil section is employed an *equivalent wing loading* should be found for use with these curves.

This *equivalent loading* is

$$\text{Actual Loading} \times \frac{K_L \text{ Max. (R.A.F. 15).}}{K_L \text{ Max. (Section used).}}$$

It is also assumed that if a different aerofoil section is used it will have round about the same efficiency as R.A.F. 15, otherwise the curves will not apply.

For the speed curves (Fig. 6) the actual wing loading is taken.

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It will be seen from the curves that the performance of an aeroplane is dependent on the combination of engine loading and wing loading and also on the fineness ratio.

For the preliminary estimate we assume that the fineness ratio of our machine will be about average.

The engine loading is of course

$$\frac{4{,}421}{350} = 12.65 \text{ lb./H.P.}$$

We have now to find the wing loading, which in combination with an engine loading of 12.65 lbs./h.p. will give us the required performance.

We will first try R.A.F. 15 Section, and from Fig. 11 we see at once that the wing loading must not exceed 6.6 lbs./ft.^{2} with a K_L max. of .513 (which is the figure for R.A.F. 15 Aerofoil Model uncorrected) in order to obtain a landing speed of 50 m.p.h.

Before going further we make a check on the speed at 10,000 ft. with this wing loading and engine loading 12.65 lbs./h.p., using Fig. 6, from which we find that with this combination we shall be low on speed and will require to increase the wing loading slightly.

We will therefore try Aerofoil Section No. 64, which has a higher K_L max., and will enable us to use a higher wing loading for the same landing speed.

This section is also a very efficient one, and enables deeper spars to be employed than does R.A.F. 15.

No. 64 Section has a max. K_L of .617 uncorrected, and again using *Fig.* 11 we find that we can increase the wing loading to 7.8 lbs./sq.ft. without exceeding the specified landing speed.

Note.—The speeds given in Fig. 11 are *stalling* speeds, and we should allow a slightly lower stalling speed than the specified max. landing speed.

However, we can safely assume that the K_L max. of the full scale wing will be higher than the uncorrected model value, and also we will rely on the lift of the fuselage and tail plane at large angles of incidence to help in reducing the landing speed.

With this in mind we will assume that a loading of 7.8 lbs./ft.^{2} will be satisfactory.

Again checking the speed at 10,000 ft. from Fig. 6 we see that it will be 106 m.p.h., which is just above the specified speed at that altitude.

We now look into the climb figures, but first of all we must find the *equivalent* wing loading as we propose to use No. 64 Section.

$$\text{Equivalent Loading} = 7.8 \times \frac{.513}{.617} = 6.5 \text{ lbs./ft.}^2$$

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The combined equivalent loading is therefore :—

$$12.65 \times 6.5 = 81.2.$$

From Fig. 7 we see that with a combined loading of 81.2 we may expect a

Service Ceiling of	17,800 ft.
Absolute Ceiling	20,000 ft.
Rate of Climb at Sea Level	950 ft./mt.

These values meet the specification, so that we now proceed to obtain a few more details of performance, using Fig. 9, from which we see that we shall obtain the following climb figures :—

Time to Height.

5,000 ft. in 6.25 mts.
10,000 ft. in 14.75 mts.
15,000 ft. in 29.50 mts.
17,500 ft. in 42.50 mts. (Service Ceiling).

The rate of climb at any altitude can be found by drawing on squared paper a graph with altitude in feet as ordinate and rate of climb in ft./mt. as absica.

If we draw a straight line between ordinate 0 and absica 950 to ordinate 20,000 and absica 0, this will represent fairly accurately the rate of climb curve. (See Fig. 8.)

We now only require to know the speed at all altitudes below the ceiling, and to obtain this information we make use of Fig. 10, which is a very useful graph.

We have already found from Fig. 6 that the speed at 10,000 ft. = 106 m.p.h., and we therefore make a table (Fig 12).

Altitude in feet.	Distance below Absolute Ceiling.	Fraction of Ceiling Speed.	Equivalent Speed.
Sea Level	20,000	1.367	$\frac{1.367}{1.298} = 111.6$
5,000	15,000	1.340	$\frac{1.340}{1.298} = 109.5$
10,000	10,000	1.298	$1 = 106$
15,000	5.000	1.215	$\frac{1.215}{1.298} = 99.2$
17,500	2,500	1.136	$\frac{1.136}{1.298} = 92.8$
20,000	0		

We may draw a complete performance chart as shown in Fig. 12.

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Thus we obtain a complete estimate of the performance of the proposed design without having to make a single drawing.

The next step in the procedure is to lay out the general arrangement of the aëroplane; this we do with the assistance of Figs. 13 and 14, which give the average proportions of a large number of aeroplanes in terms of the wing area and wing (mean) chord.

As our machine is to be of average proportions we may use the values given in the tables.

(a) From Fig. 14 we find that the average aspect ratio for two-seater biplanes is 7.3 to 1.

(b) The required wing area is $\frac{4,421}{7.8}=566$ sq. ft.

(c) The wing dimensions will be: $A=2\ (C\times 7.3\ C)=566$ sq. ft. where $C=$Chord (mean).

(This is neglecting any allowance for tip rounding and body gap in lower plane.)

We will allow, say, 50 sq. ft. for body gap and tip rounding, and so get $A=566+50=616=14.6\ C^2$.

Therefore $C=$chord$=6.56$ ft.—say 6 ft. 6 in.

(d) Span (mean)$=7.3\times 6.56=48$ ft.

(e) The C.G. is assumed to be at .33 of the equivalent chord.

(f) The equivalent chord is taken at .55 of the gap above the lower wing.

(g) Tail and elevator area combined
$=12$ per cent. of m.p. area$=.12\times 566=68$ sq. ft.

(h) Aileron area (total)
$=12$ per cent. of m.p. area$=.12\times 566=68$ sq. ft.

And so on, using the values given in Figs. 13 and 14. We can fix all the dimensions of the machines and produce a general arrangement drawing as in Fig. 15.

It is necessary to make a rough calculation of the lift reactions at this stage in order to fix fairly accurately the sizes of the various interplane struts and bracing wires, these sizes being used when calculating the parasite resistance of the second method.

2nd Method.

We will now check the performance, using the second method.

The first step is to decide on the propeller diameter, and this is quickly done by the use of Dr. H. C. Watt's useful Nomogram, shown in Fig. 16, from which we find that the propeller must be either a 2-blader 12 ft. dia. or a 4-blader 10 ft. 3 in. dia.

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We will decide to use the 12 ft. dia. 2-blader, as we have ample ground clearance, and the 2-bladed prop. is simple and easy to transport.

The reason for fixing the propeller dimensions first is that we have to calculate the velocity in the propeller slipstream before attempting to estimate the resistance of parts of the aeroplane structure which are placed in the slipstream, and also we must know the area of the propeller circle to see what parts come in the slipstream.

Calculation of Thrust h.p. available.

We must now calculate the thrust h.p. at various forward speeds of the aeroplane, and from this the thrust in lbs. as the slipstream velocity depends upon the thrust in lb./ft.^{2} of the propeller.

The curves in Fig. 17 which are taken from C.I.M. 704 by Mr. H. Bolas, provide a ready means of obtaining the *thrust h.p. available* at all speeds.

We will decide to have the max. propeller efficiency at 100 m.p.h. for cruising, and this fixes the following data to be used in the calculations :—

V = forward speed considered.

V_0 = designed forward speed = 100 m.p.h.

N_0 = designed max. revs./sec. of propeller = 18 r.p.s.

D = dia. of propeller in feet = 12 feet.

h = thrust h.p. available at V ft./sec.

H_0 = b.h.p. of engine at designed r.p.s. = 350 b.h.p.

$$\frac{V_0}{N_0 D} = \text{Pitch/dia. ratio} = \frac{146.7}{18 \times 12} = .68.$$

The calculation may be tabulated as in Fig. 18.

In Fig. 18 the values in Column 3 are read direct from the curves in Fig. 17.

Column 4, which shows the thrust h.p. available, is obtained by multiplying values in Column 3 by the max. b.h.p. of the engine = 350 b.h.p.

Column 5, which gives the thrust in lbs. of the propeller, is obtained by multiplying the values in Column 4 by $\frac{33{,}000 \times 60}{5{,}280 \times V} = \frac{375}{V}$ The values in Column 4 are used in calculating the thrust per sq. ft. of effective propeller disc area, which determines the slip velocity, as will be seen later.

Note.—The thrust horsepower available at any altitude for a given forward speed is assumed to vary directly as the variation of b.h.p. of engine with altitude.

(The correction factors to convert b.h.p. at sea level to b.h.p. at an altitude are given in Fig. 23. This assumes that the engine revs. remain constant for a given forward speed at any altitude.)

Working on this assumption we calculate the values of thrust h.p. at various altitudes and fill in Columns 6, 7, 8 and 9, multiplying the values in Column 4 by the appropriate correction factor from Fig. 28.

Propeller Effective Disc Area.

To estimate the slipstream velocity we require to know the thrust per sq **ft.** of effective disc area of the propeller, upon which the slipstream velocity depends, and so we proceed to estimate what the effective disc. area is.

The *effective* area of the propeller depends on the blade form and the shape and size of the fuselage nose.

The ineffective diameter of the blade near the boss varies from 30 per cent. to 40 per cent. of the propeller diameter.

We will assume that in the case of the propeller under consideration the ineffective portion of the blade = 33 per cent. = 4 ft.

Therefore effective disc area $= \pi/4\ (12^{\circ} - 4^{2})$

$$= \pi/4 \times 128 = 100 \text{ sq. ft.}$$

Having found the effective disc area of the propeller we now proceed to make the tables and curve shown in Fig. 18A. The curve shows the relationship between the horizontal flying speed and the slipstream velocity.

In Fig. 18A, Columns 1 and 2 are taken direct from Fig. 18.

Column 3 shows the thrust in lb./ft.^{2} of effective disc area, which is obtained by dividing the values in Col. 2 by 100 ft.^{2}, the effective disc area for our propeller.

Column 4.—The outflow velocity in ft./sec. is obtained from Fig. 19,* which is another of Dr. Watts' curves, and which can be used to find the added velocity which must be imparted to the air by the propeller in order to obtain a given thrust in lb./ft.^{2} of effective disc area at any forward speed.

Column 5 shows the values in Col. 4 converted to m.p.h., and Column 6, which shows the slipstream velocity in m.p.h., is obtained by adding together the values in Columns 1 and 5, thus giving us the actual slipstream velocity at the various flying speeds considered.

Parasite Resistance.

Having now obtained the slipstream velocity at all speeds of flight we are in a position to estimate the parasite resistance of the aeroplane.

We may make use of charts and tables such as are shown in Fig. 20 and Fig. 21. These figures show the resistance in lb./sq. ft. of frontal area at 100 m.p.h. of various forms and components which form part of aeroplane structures.

These resistances can be stated in various ways, but the most convenient way for performance estimation is to give them in lb./ft.^{2} of frontal area at 100 m.p.h.

It is assumed that the resistance of all these forms varies as V^2, which is near enough for our purpose.

Taking the values of resistance per sq. ft. of frontal area from Figs. 20 and 21, we make up the tables shown in Fig. 22, which shows the resistance

*** Design of Screw Propellers. H. C. Watts, D.Sc., A.M.Inst.C.E., A.Inst.N.A., F.R.Ae.S. (Longmans, Green & Co.)**

at 100 m.p.h. of the parts out of the slipstream and the resistance of parts *in* the slipstream at a slipstream velocity of 100 m.p.h.

The resistance of any part at 100 m.p.h. air speed is of course found by multiplying its frontal area in sq. ft. by the value of resistance/sq. ft. for the particular part, given in Fig. 20.

As we know from the curve in Fig. 18A the relationship between flying speed and slipstream velocity, it is an easy matter to obtain the parasite resistance both in and out of the slipstream at various forward speeds within the flying range and at ground level.

This is done by multiplying the total resistance at 100 m.p.h. *out of slip* taken from Fig. 22 by $\frac{V^2}{100^2}$ where V is the flying speed considered and the total resistance *in slip* at a slipstream velocity of 100 m.p.h. by $\frac{Vs^2}{100^2}$ where Vs is the slipstream velocity concerned.

The results of this calculation are shown in Columns 3 and 4, Fig. 22A.

The resistances in and out of slip are then added together in Column 5, Fig. 22A, and give us the total parasite resistance at the various flying speeds at sea level.

To obtain the parasite resistance at altitudes we assume that the slipstream velocity is the same for any given flying speed at all altitudes, that is to say, the curve in Fig. 18A holds good for all altitudes. Also we know that the resistance of a body at any speed considered varies with altitude directly as the relative air density.

Hence we obtain the parasite resistances given in Columns 6, 7, 8 and 9, Fig. 22A, by multiplying the total parasite resistance at sea level for any given flying speeds by (ρ/ρ_0) = the relative air density at the altitude concerned.

The values of relative density may be taken from Fig. 23, which shows the density at altitudes as a fraction of Standard Ground Level density.

Fig. 23 also shows the variation of density ρ/s with standard height, and variation of $\sqrt{\rho/\rho_0}$ with altitude and the variation with altitude of the power developed by rotary and stationary engines.

All these curves are very useful in performance calculations and are used later in this example.

Having found the total parasite resistance at all speeds and heights we must now calculate the aerofoil drag.

Aerofoil Drag.

Most of the available data on aerofoil characteristics is obtained from wind tunnel tests on monoplane models with square tips and aspect ratio 6.

The model characteristics must be converted to full scale before they can be used in the performance calculations.

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The corrections required to convert model monoplane values to full scale biplane with rounded tips and staggered planes are :—

1. Correction of scale/speed (from vl 10 to vl 30).
2. Correction of wing tip.
3. Correction of biplane effect.
4. Correction of aspect ratio.
5. Correction of stagger.

These correction factors are shown in Figs. 24, 25, 26, 27 and 28, and represent an average of a considerable amount of data collected from various sources, but principally from results of experiments carried out at the National Physical Laboratory. They are arranged for correcting model values obtained from models with aspect ratio 6 and square tips. If the model varies from these proportions then the test results should be first corrected to standard by *dividing* by the appropriate correction factor from the curves.

They are plotted in bases KL/KL max. and L/D/L/D max., which makes them applicable to aerofoil test figures for aerofoils which have their lift and drag curves spread over varying ranges of angle of incidence.

Unfortunately, the correction factors published up-to date are not strictly applicable to any and all wing sections, but as we have no other information on the subject at present we must apply the data which we *do* possess.

In practice the factors shown above appear to give good results, so we can carry on until such time as the existing data is amplified.

Whenever possible we should use test figures for models of as nearly the same form as the full scale wing, as we can obtain and test at a high speed, as this reduces the number of corrections to be applied.

For example, if a biplane test on the section to be employed is available we should use the figures from this test when calculating the full-scale figures for a biplane.

The characteristics of the main planes in our machine are as follows :—

Plane Arrangement	Biplane
Span (mean)	48 ft. 0 in.
Chord (mean)	6 ft. 6 in.
Aspect Ratio	7.3 to 1
Area	566 sq. ft.
Wing Tips	Rounded No. 3
Stagger	$+15^\circ$
Gap/Chord ratio	1 to 1
Aerofoil Section	No. 64

The model aerofoil characteristics for No. 64 section monoplane are shown on Fig. 29, which also shows the corrected characteristics for the full-scale wings

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Fig. 29 also shows the tabulated calculation for the correction of K_L and L/D from model to full-scale.

The full-scale models for K/L are obtained by multiplying the values of K_L at various values of model K_L/K_L max. by the appropriate correction factors from Figs. 24, 25, 26 and 28, as shown above.

The same procedure is followed in correction L/D.

The resulting values are then plotted and give us the characteristic curves for the full-scale biplane wing.

Having obtained these we are in a position to calculate the drag of the aerofoil of our machine at all speeds and altitudes.

The lift and drag of an aerofoil is found from the formulæ :—

$$\text{Lift} = K_L \rho/g\ A.V.^2$$

$$\text{Drag} = K_D\ \rho/g\ A.V.^2$$

Where K_L = Absolute Lift coefficient.
K_D = Absolute Drag coefficient.
Lift is in lbs.
Drag is in lbs.
ρ = density of air in lbs./cu. ft. at the altitude considered.
g = 32.2 ft./sec./sec.
A = Wing Area in sq. ft.
V = Speed in ft./sec.

(Note :—ρ/g at Standard Ground Level = .00237.) Slugs/cu. ft.

Fig. 30 shows the tabulated calculation for the aerofoil resistance at all speeds and altitudes within the flying range.

The aerofoil drag is found as follows :—

The required K_L at any speed V.ft./sec. at sea-level is obtained from the formula :

$$K_L = \frac{L}{\rho/gV^2} = \frac{\text{Lift in lb./ft.}^2}{.00237 \times V^2}$$

The value for L/D at this K_L is taken from the wing characteristic curve and the drag of the wing is obtained by dividing the *total lift* by this value of L/D.

For example, in our case the required K_L at 100 m.p.h. or 146.7 ft./sec. is

$$K_L = \frac{7.8}{.00237 \times 146.7^2} = .155$$

The value of L/D corresponding to K_L = .155, is found from Fig. 29 to be 14.2, and therefore the drag of the aerofoils at 100 m.p.h. and sea-level, is

$$\frac{4{,}421}{14.2} = 311 \text{ lbs.}$$

It will be noted from Fig. 30 that we have calculated the Aerofoil resistance at convenient speeds within the flying range and at sea level, these resistances being shown in column 5.

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For convenience, in columns 6, 7, 8 and 9 we have shown the speeds at altitudes 5,000, 10,000, 15,000, and 17,500 ft., corresponding to the aerofoil resistance given in column 5.

We have done this because :—
for any given wing lift and drag the speed varies with altitude inversely as the square root of the relative air density $= \sqrt{\rho/\rho_0}$

The speeds given in columns 6, 7, 8 and 9 are therefore obtained by dividing the speeds in column 1 by values of $\sqrt{\rho/\rho_0}$ appropriate to the altitudes considered.

Values of $\sqrt{\rho/\rho_0}$ are given in Fig. 23.

Having found the total parasite resistance and aerofoil drag of our machine at all speeds and altitudes, we proceed to add these values together to find the resistance of the *complete aeroplane* at all speeds and altitudes. This is done in Fig. 31.

We are now in a position to calculate what thrust horse power will be required to fly the machine.

The H.P. required at any speed V.

$$= \text{H.P.}_{\text{R}} = \frac{\text{R} \times \text{V}}{550}$$

H.P.R. = Thrust H.P. required.

Where R = Total resistance in lbs.

V = Flight velocity in ft./sec.

Therefore taking the values of total R from fig. 31, we multiply them by the corresponding flying speed in ft./sec. and divide by 550, thus obtaining the horse power required to maintain the aeroplane in horizontal flight at the various speeds and altitudes considered, these values as shown in Fig. 32.

We next make Fig. 33, which shows the values of thrust h.p. required taken from Fig. 32, plotted against thrust h.p. available, taken from Fig. 18, which we found earlier in the calculations.

These values were plotted for each altitude considered and, of course, show the excess of thrust h.p. available from the propeller over that required for horizontal flights.

This excess of thrust h.p. is a measure of the rate of climb at the various altitudes, which is

$$\text{Rate of Climb in ft./mt.} = \frac{\text{H.P.e} \times 33{,}000.}{\text{Gross Weight of Aeroplane.}}$$

where H.P.e = Excess H.P. available for climbing.

The curves shown in Fig. 33 also give us the speed range at the various altitudes, the intersections of the thrust h.p. required and thrust h.p. available curves being the limits of high and low speed.

The tabulated calculation for the best rate of climb at the various altitudes is shown in Fig. 34; for this we take only the max. excess. h.p. at each altitude considered.

The rate of climb curve is then plotted as shown in Fig. 34.

We now wish to know the time required to climb to any height, and this is obtained as follows :—

The rate of climb is plotted in minutes/1,000 ft. against altitude in feet, as shown in curve 2, Fig. 34, so that the shaded area to the left of the curve up to the height considered, represents to scale the time required to reach that height, and we can thus draw the time/height curve shown on Fig. 34.

The speed/height curve is drawn using values from Fig. 33.

We have now obtained all the information necessary to enable us to draw the complete performance chart for the aeroplane, as shown in Fig. 35.

This chart gives us the max. speed, min. : speed and best climbing speed for all altitudes, also the rate of climb at all altitudes and the time required to reach any given altitude below the ceiling.

As a matter of interest, the performance as estimated by the first method is shown dotted, and the close agreement between the two estimates will be noted.

3rd Method.

The third method of estimation is by utilising the results of the wind tunnel tests on a model of the aeroplane, and is very similar to the 2nd method as regards the calculations to be made.

The calculations are, however, simplified, as we obtain from the wind tunnel tests a table of lift and drag coefficients for the complete machine at various angles of incidence, so that it is not necessary to estimate the parasite resistance or correct the wing characteristics before calculating the lift and drag of the complete aeroplane.

Models are now tested at values of vl of 30 and over so that no scale/speed correction is necessary, and the model values for K_L and K_D may be used direct for the calculation of the full-scale lift and drag.

It is preferable to test first the fuselage complete with undercarriage and empannage, so as to obtain the resistance co-efficient of the parts in the slipstream.

The model wings can then be added and the tests made on the complete model.

The performance calculations are then carried out in the following order :—

1. First calculate the K_L required for various flying speeds, as in the 2nd method.

2. Then read from the characteristic curves of the model, the values of L/D corresponding to the calculated values of K_L for the various speeds considered, and using these values of L/D calculate the resistance of the complete aeroplane throughout the flying range.

3. Having obtained the resistance at the different speeds of flight, proceed to calculate the thrust h.p. required to fly the aeroplane level at these speeds, as in 2nd method.

4. The thrust h.p. available is then calculated as in the 2nd method.

5. The remainder of the calculations are carried out as in the 2nd method, and therefore call for no further comment.

After a brief discussion the Chairman proposed a very hearty vote of thanks to Mr. Chadwick for an extremely interesting and valuable paper. This was passed with acclamation, and the meeting then closed.

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In the discussion following Mr. Chadwick's paper I made reference to the possibility of replacing methods of piecemeal performance estimates by generalised mathematical methods. It was quite impossible to make any detailed suggestions in this direction, and it therefore seems worth while to give a fuller explanation of my meaning and some indication of the way in which the estimation of performance might be simplified.

The "ceiling" of an aeroplane is a function of the ratio of airscrew h.p. available at standard density to the airscrew h.p. required for level flight at the same density. For any aeroplane this ratio varies with the indicated air speed, and the absolute ceiling is determined by the maximum ratio of h.p. available to h.p. required. But for every indicated air speed there is a particular value for this ratio, and there is corresponding "ceiling" for each indicated air speed. Obviously if the "ceiling" for each indicated air speed can be computed it is only necessary to correct the indicated air speeds for density at the corresponding "ceiling" to obtain complete information as to the machine's maximum speed at all altitudes within its range.

If, therefore, a general expression giving the ceiling in terms of the power required/power available ratio can be obtained, the ceiling at any value of indicated air speed can be computed directly from the value of this ratio as shown by the h.p. required and h.p. available curves at standard density.

The usual method of determining both absolute ceiling and speed at heights, as exemplified by Mr. Chadwick, also depends on the assumption that there is a general relation between ceiling and the power ratio referred to. The precise nature of this general relation is concealed in a series of curves of engine performance at varying altitudes, airscrew characteristics and so forth, and its application in this form involves successive recomputation of both engine and aeroplane characteristics at a number of different heights—which is a somewhat laborious process. If the engine output varied directly with density, and if airscrew efficiency were constant at constant indicated air speed, then the density at the "ceiling" for any indicated air speed would be given by

$$\rho_c = \left(\frac{\text{H.P.r.o.}}{\text{H.P.a.o.}}\right)^{\frac{2}{3}}$$

where ρ_c = density at ceiling (standard density = 1) and H.P.r.o., H.P.a.o. are respectively airscrew h.p. required for level flight, and available at full throttle, at unit density.

Actually the engine output falls more rapidly than does density, and the airscrew efficiency is not constant at constant indicated air speed. The fact that the engine output varies rather less rapidly with altitude than does the

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air pressure, and that in general the airscrew efficiency at a given indicated air speed improves with altitude, suggested that the substitution of pressure for density in the above expression giving

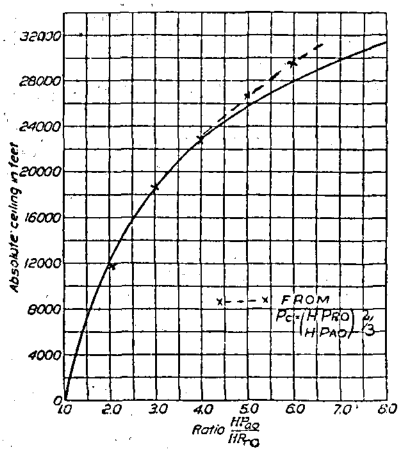
$$\frac{\rho_c}{\rho_o} = \left(\frac{\text{H P r.o.}}{\text{H.P.a.o.}}\right)^{\frac{2}{3}}$$

(where ρ_c and ρ_o are pressures at ceiling and sea-level) would give reasonably accurate results at least for rough predictions. Experience with this expression during the War indicated that it was at least as reliable as the method based on average curves for engine output at heights, and the airscrew designer's estimate of his airscrew characteristics.

Recently Mr. Walter S. Diehl, in Report No. 171 of the United States National Advisory Committee for Aeronautics has investigated the relation between the ceiling of an aeroplane and the ratio $\frac{\text{H.P.a o}}{\text{H.P r.o.}}$ taking into account in great detail the data at present available as to engine output variation with altitude, and airscrew characteristics, and has embodied the results in two curves which give ceiling in feet as a function of this power ratio. One of these curves applies to the normal aeroplane, the other to the special case of an engine which, by supercharging or other means, can maintain constant r.p.m. independently of altitude. The first of these curves, which is attached herewith, gives figures for ceiling which do not differ appreciably from those arrived at by the expression

$$\frac{\rho_c}{\rho_o} = \left(\frac{\text{H.P.r.o.}}{\text{H.P.a o}}\right)^{\frac{2}{3}}$$

for heights of less than 23,000 ft. Above that height the suggested expression is too optimisitic.



It would thus appear that having once obtained curves of power required and airscrew power available at unit density over the whole range of flying speed, the ceiling at every air speed within that range, and consequently both the absolute ceiling and the speed at heights, can be derived at once from the values of the sea-level power ratio at various air speeds, without computation of the complete performance curve at any other altitude.

It should be pointed out that both Mr. Diehl's curve and the suggested approximate expression for ceiling assume that airscrew efficiency at the same indicated air speed rises with increase of altitude. This is generally the case over the part of the speed range which is of material consequence. But it should be pointed out that in cases where the airscrew reaches maximum efficiency at a speed appreciably below the maximum level speed at sea-level, this general method of dealing with the problem will exaggerate the ceiling for high air speeds—or the maximum speed at low altitudes. This error is not of very great magnitude in the case of normal aircraft with airscrews giving maximum efficiency at a speed reasonably close to the maximum speed at sea-level, but it is obvious that if the airscrew gives maximum efficiency at an abnormally low speed—as may be the case if maximum rate of climb at sea-level is desired—a more detailed analysis of performance at altitudes is necessary.

W. H. Sayers.

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Diagrams referred to in the Paper.

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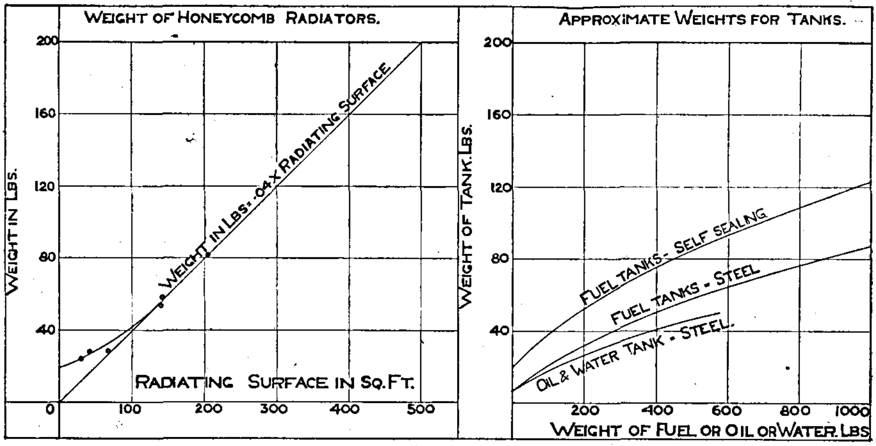

Fig. 1.

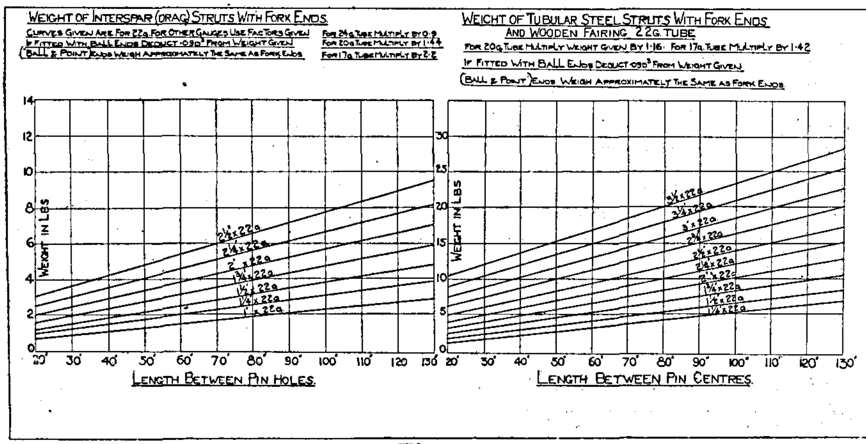

Fig. 2.

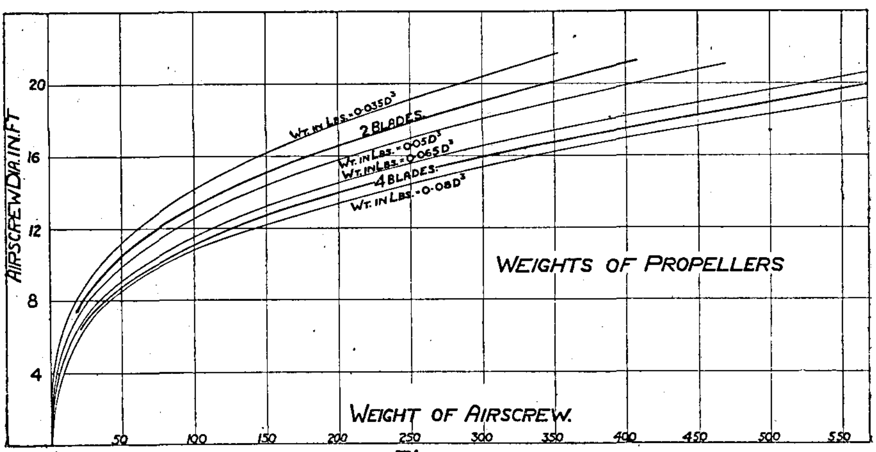

Fig. 3.

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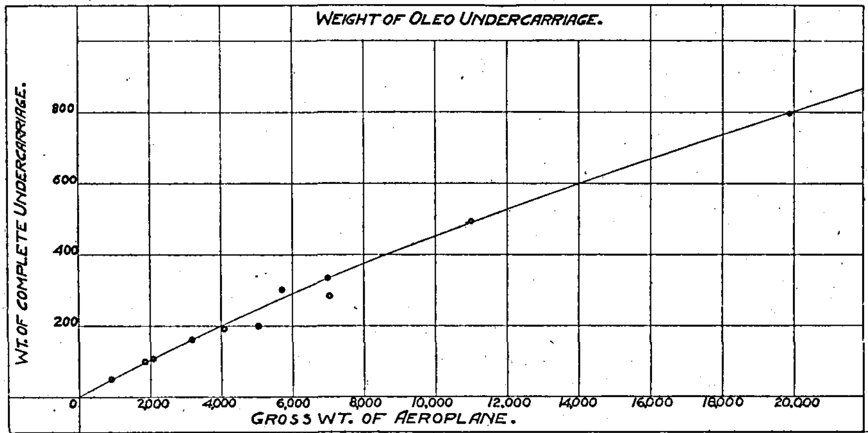

Fig. 4.

WEIGHTS OF AIRCRAFT COMPONENTS IN PERCENTAGE OF GROSS WEIGHT COMPONENT.	AVERAGE VALUE.
WINGS COMPLETE WITH ALL BRACING.	15·00%
TAIL, ELEVATORS, RUDDER, & FIN.	2·00%
UNDERCARRIAGE & TAIL SKID.	4·00%
BODY, WITH ENGINE MOUNTING, SEATING &C.	11·25%
CONTROLS, MACHINE & ENGINE.	·75%
STRUCTURE TOTAL.	33·00%
ENGINE	18·50%
RADIATOR & SHUTTERS & COWLING	2·75%
COOLING WATER	2·75%
FUEL & OIL TANKS	3·00%
PIPING, COCKS & PUMPS &C	0·50%
PROPELLER.	2·50%
STARTING GEAR FOR ENGINES	0·50%
POWER PLANT. TOTAL.	30·00%
WT BARE AS PERCENTAGE OF GROSS WT. TOTAL	63·00%

	POWER PLANTS	LBS/B.H.P.
AVERAGE WEIGHTS OF:-	WATER-COOLED ENGINES	2·50
	AIR-COOLED ENGINES	2·33
	WATER-COOLED ENGINE ACCESSORIES.	1·00.
	TWIN WATER-COOLED ENGINE ACCESSORIES.	1·25.
	AIR-COOLED ENGINE ACCESSORIES.	0·383.
	WATER-COOLED ENGINES & ACCESSORIES	3·50.
	TWIN WATER-COOLED ENGINES & ACCESSORIES.	3·75.
	AIR-COOLED ENGINES & ACCESSORIES.	2·70.

ENGINE ACCESSORIES INCLUDE :- RADIATOR & SHUTTERS. COOLING WATER. PIPING & COCKS FOR FUEL, OIL, & WATER. EXHAUST PIPES. PUMPS. STARTING GEAR. PROPELLER. FUEL & OIL TANKS NOT INCLUDED AS THESE VARY WITH DURATION OF FLIGHT.

WEIGHT AS PERCENTAGE OF ENGINE WEIGHT	W-C ENGINE ACCESSORIES.	43%
	TWIN W-C ENGINE ACCESSORIES	47%
	A-C ENGINE ACCESSORIES.	16%

Fig. 5.

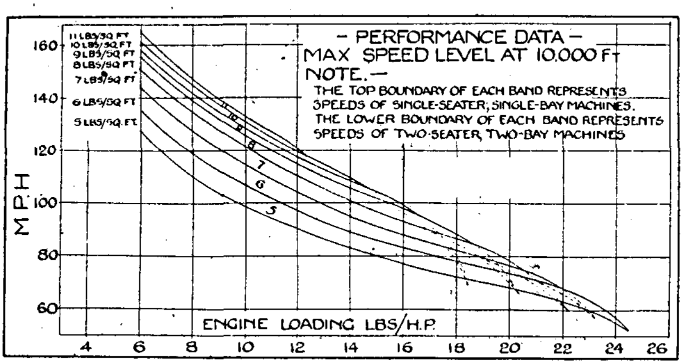

Fig. 6.

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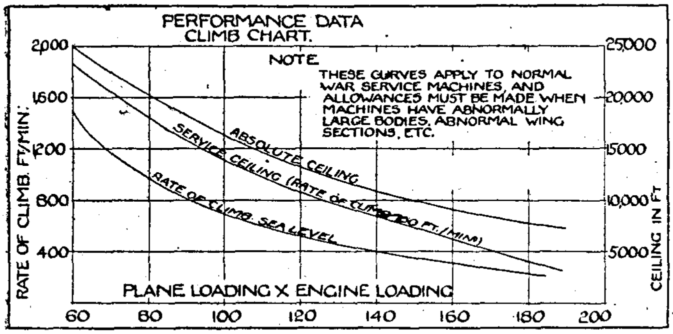

Fig. 7.

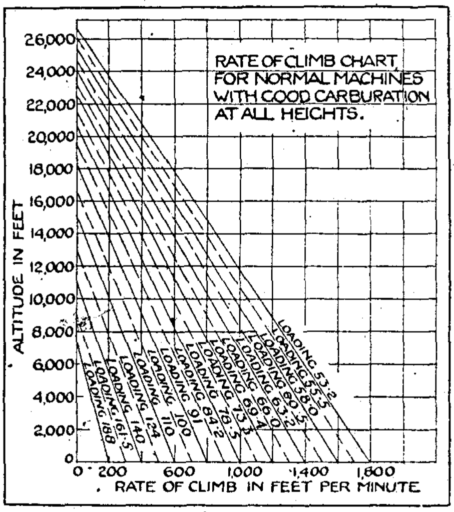

Fig. 8.

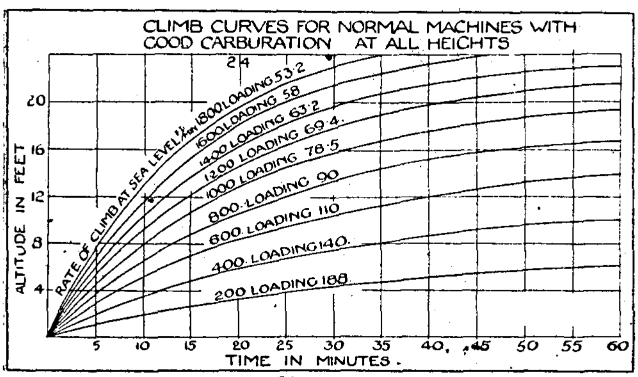

Fig. 9.

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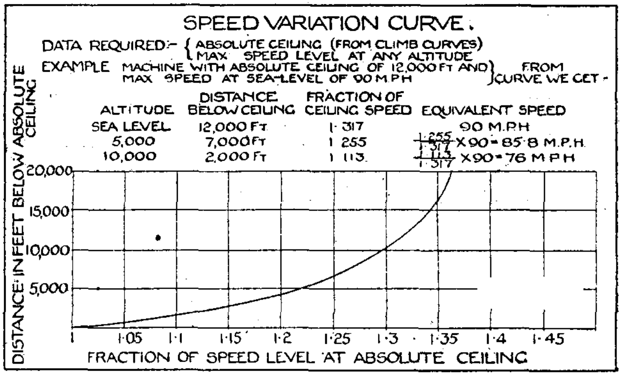

Fig. 10.

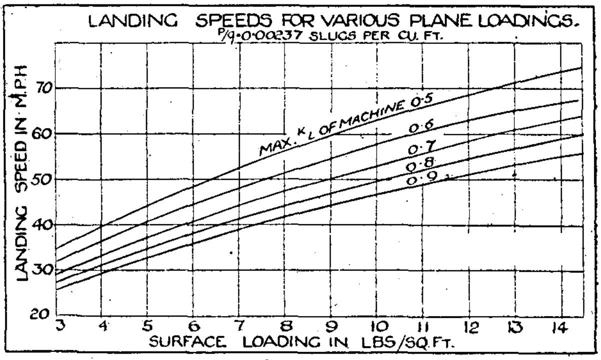

Fig. 11.

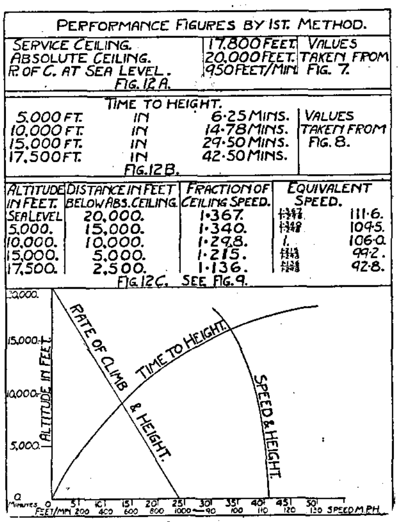

Fig. 12.

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AEROPLANE DESIGN CONSTANTS.

$\frac{\text{C.G. TO HINGE.}}{\text{MEAN CHORD M.P.}}$ — 2·5 MONOPLANE; 3·3 BIPLANE; 4·0 TRIPLANE; 4·75 QUADPLANE

$\frac{\text{TAIL AREA.}}{\text{M.P. AREA.}}$ — 15% MONOPLANE; 12% BIPLANE; 9% TRIPLANE; 6% QUADPLANE

TAIL AREAS

h - TAIL AREA IN % OF MAIN PLANE AREA.
v - FIN & RUDDER IN % OF MAIN PLANE AREA
t - LENGTH. C.G. TO ELEVATOR HINGE IN CHORDS M.P.
th - AVERAGE FOR AEROPLANES & SEAPLANES.......................37·5.
th - AVERAGE FOR FLYING-BOATS.......................41·5.
tv - AVERAGE FOR AEROPLANES.......................14·15.
tv - AVERAGE FOR FLYING-BOATS.......................18·2.

C.G. TO C.P. TAIL VARIES FROM ·5 TO ·66 M.P. SPAN.

AILERON AREA. AVERAGE 12% M.P. AREA. (GERMAN AVERAGE 8% M.P. AREA.)

Fig. 13.

AVERAGE ASPECT RATIO

$\frac{\text{MAX: SPAN}}{\text{MAX: CHORD}}$

MONOPLANES	5·0
SINGLE SEATERS	5·9
TWO SEATERS (NON TRAINING)	7·3
SEAPLANES TWO SEATERS	8·7
TRIPLANES	9·6
TWO ENGINE BIPLANES	9·2
FLYING BOATS	10·4

AVERAGE CONTROL SURFACE COEFFICIENTS

	t	h	v	a	th	tv	AR
MONOPLANES	2·5	14·8	6·1	12·5	37	15·2	5·0
SINGLE SEATERS	3·0	11·8	4·4	12·9	36·4	13·2	5·9
TWO SEATERS. (NON TRAINING)	3·2	11·7	4·3	12·6	37·4	13·8	7·3
SEAPLANES TWO SEATERS	3·3	11·4	4·0	11·3	37·6	13·2	8·7
TRIPLANES	4·0	9·2	3·7	11·2	36·8	14·8	9·6
TWO ENGINE BIPLANES	3·5	11·1	4·2	11·3	38·8	14·7	9·2
FLYING BOATS	3·0	13·8	6·3	11·2	41·4	18·9	10·4
AVROS	3·35	12·8	4·0	12·7	43·0	13·4	7·0

Fig. 14.

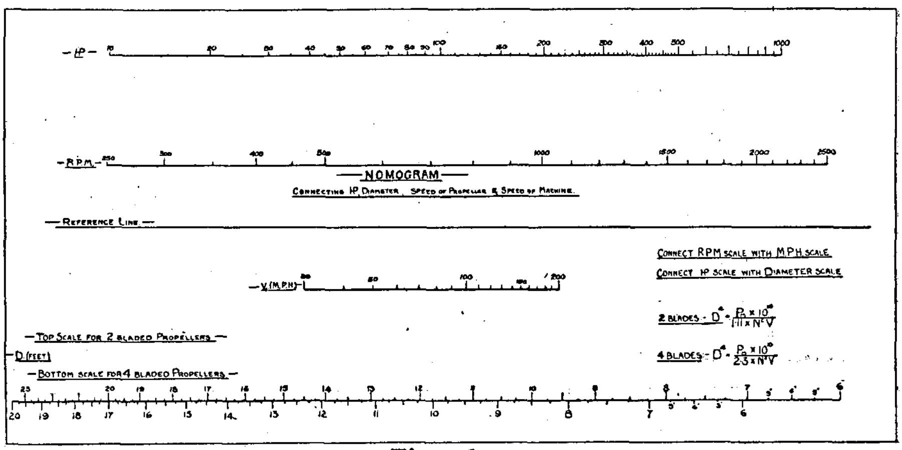

Fig. 16.

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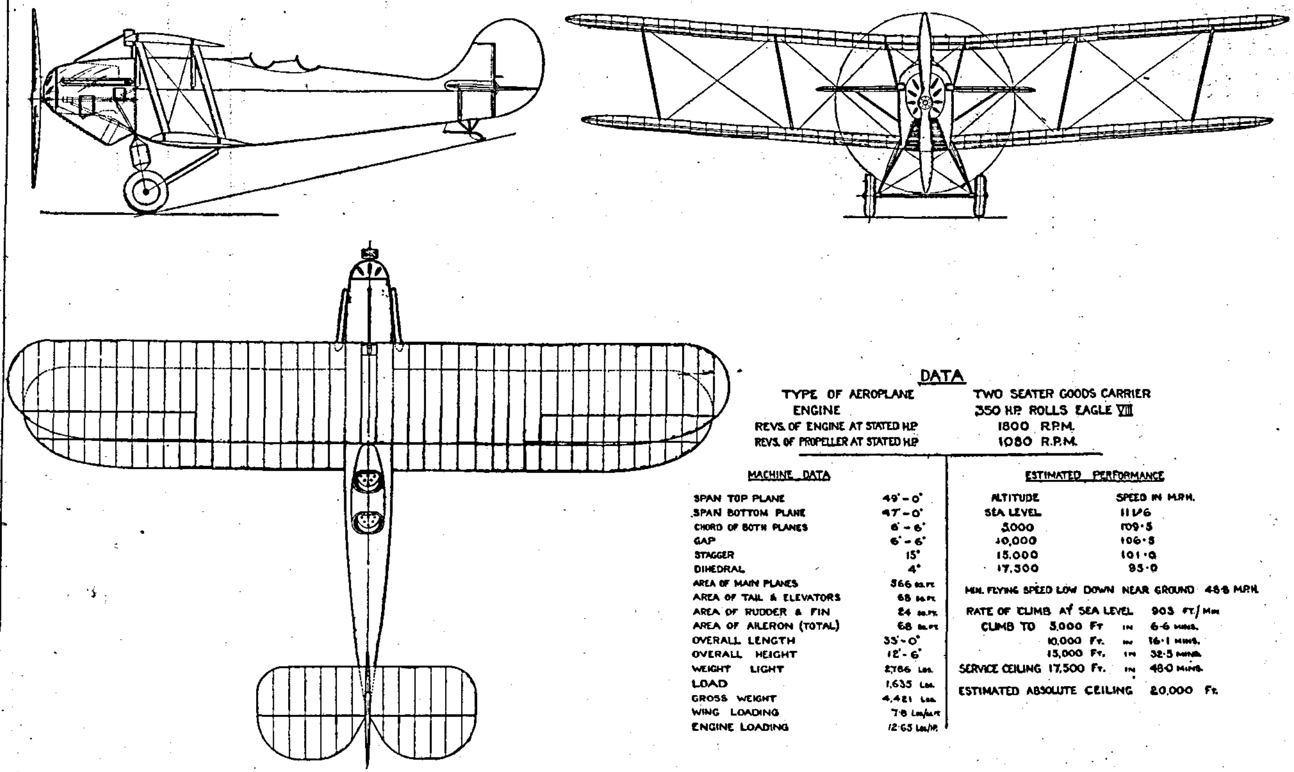
DATA
TYPE OF AEROPLANE TWO SEATER GOODS CARRIER
ENGINE 350 H.P. ROLLS EAGLE VIII
REVS. OF ENGINE AT STATED H.P. 1800 R.P.M.
REVS. OF PROPELLER AT STATED H.P. 1080 R.P.M.
MACHINE DATA
SPAN TOP PLANE 49'-0"
SPAN BOTTOM PLANE 47'-0"
CHORD OF BOTH PLANES 6'-6"
GAP 6'-6"
STAGGER 15"
DIHEDRAL 4°
AREA OF MAIN PLANES 566 SQ.FT.
AREA OF TAIL & ELEVATORS 68 SQ.FT.
AREA OF RUDDER & FIN 24 SQ.FT.
AREA OF AILERON (TOTAL) 68 SQ.FT.
OVERALL LENGTH 33'-0"
OVERALL HEIGHT 12'-6"
WEIGHT LIGHT 2,786 LBS.
LOAD 1,635 LBS.
GROSS WEIGHT 4,421 LBS.
WING LOADING 7·8 LBS/SQ.FT.
ENGINE LOADING 12·65 LBS/H.P.
ESTIMATED PERFORMANCE
ALTITUDE SPEED IN M.P.H.
SEA LEVEL 111·6
5,000 109·5
10,000 106·5
15,000 101·0
17,300 93·0
MIN. FLYING SPEED LOW DOWN NEAR GROUND 48·8 M.P.H.
RATE OF CLIMB AT SEA LEVEL 903 FT./MIN.
CLIMB TO 5,000 FT. IN 6·6 MINS.
10,000 FT. IN 16·1 MINS.
15,000 FT. IN 32·5 MINS.
SERVICE CEILING 17,500 FT. IN 48·0 MINS.
ESTIMATED ABSOLUTE CEILING 20,000 FT.

Fig. 15.

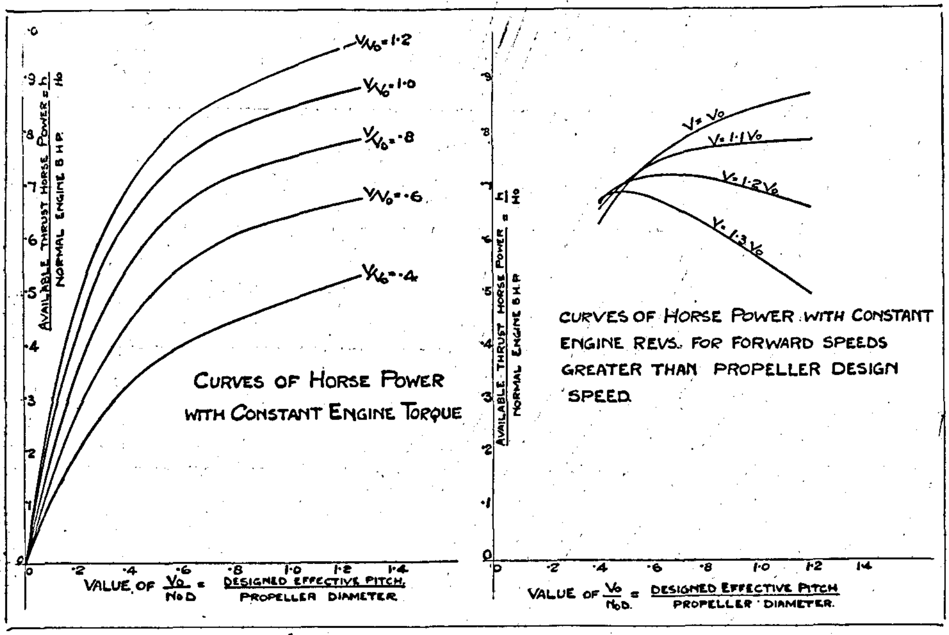

Fig. 17.

THRUST HP AVAILABLE AT VARIOUS SPEEDS & ALTITUDES.

1.	2.	3.	4.	5.	6.	7.	8.	9.
V/V_0	FORWARD SPEED. M.P.H.	h/H_0	THRUST HP AVAILABLE = h. SEA LEVEL	THRUST IN LBS. SEA LEVEL	THRUST HP AVAILABLE AT 5,000 FT. ·82h.	10,000 FT. ·67h.	15,000 FT. ·54h.	17,500 FT. ·48h.
·4	40	·435	152	1427	124·6	102	82	73
·6	60	·585	205	1280	168·0	137·2	110·6	98·6
·8	80	·700	245	1147	200·8	164	132	117·7
1·0	100	·785	275	1030	225·5	184	148·3	132
1·1	110	·763	267	913	218·8	179	144	128
1·2	120	·730	255	796	209	171	137·7	122·5

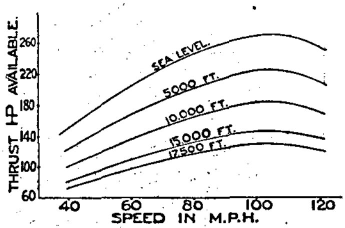

Fig. 18.

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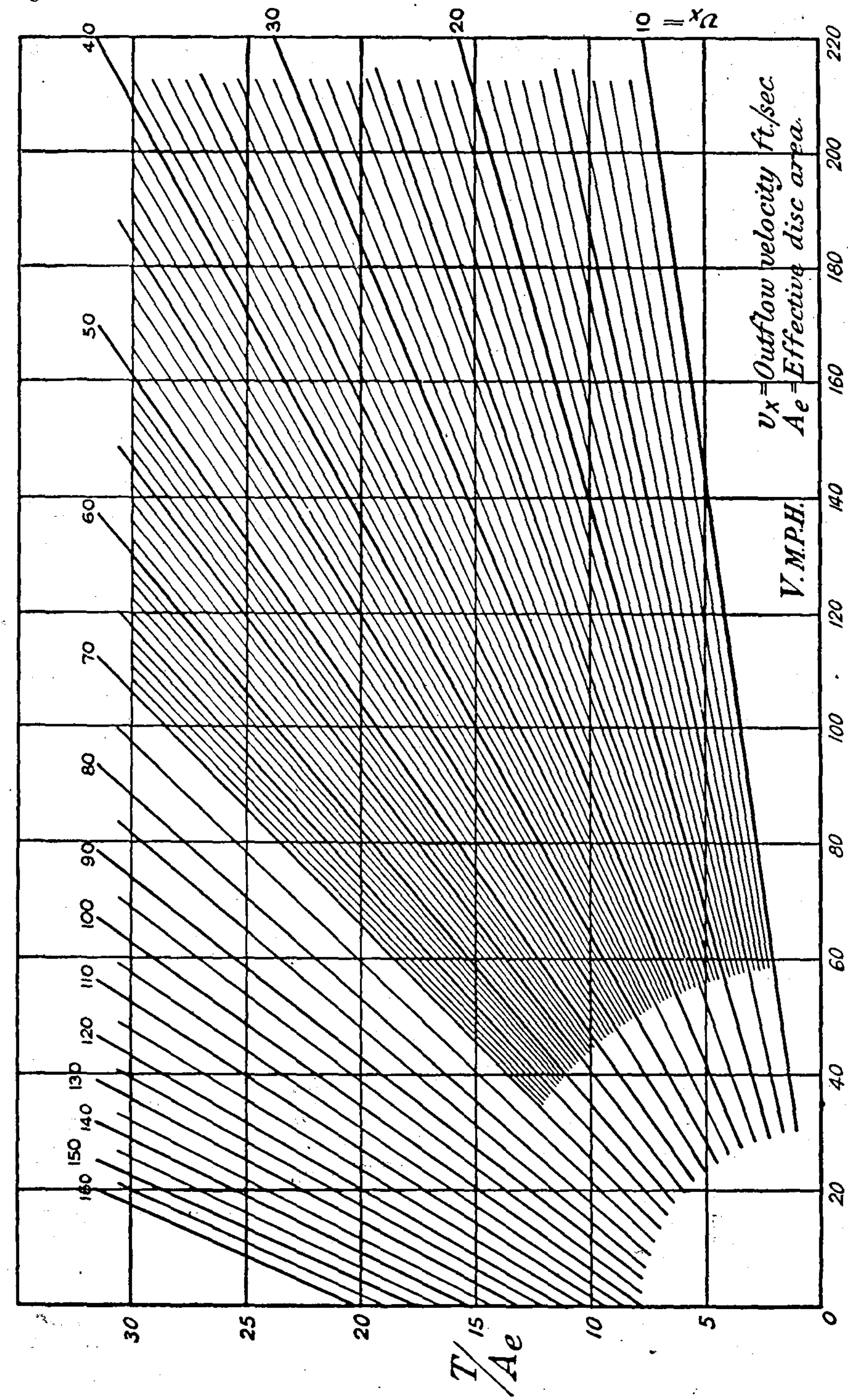

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Fig. 19.

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CALCULATION OF SLIP STREAM VELOCITY.

1	2	3	4	5	6
FORWARD SPEED IN M.P.H.	THRUST IN LBS.	THRUST IN LBS/FT²	OUTFLOW VELOCITY FT/SEC	OUTFLOW VELOCITY M.P.H	SLIP STREAM VELOCITY.
40	1427	14·27	72	49	89
60	1280	12·80	52	35·5	95·5
80	1147	11·47	37·5	25·6	105·6
100	1030	10·30	28·0	19	119·0
110	913	9·13	23·0	15·7	125·7
120	796	7·96	18·0	12·3	132·3

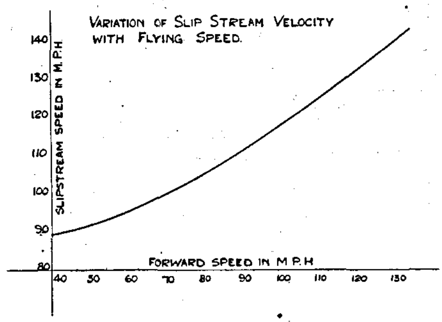

Fig. 18a.

RESISTANCE OF BODIES IN LBS PER SQ. FT. AT 100 M.P.H.

SKETCH	DESCRIPTION	RESISTANCE			
	FUSELAGE. POINTED NOSE ENTIRELY FAIRED	3·0		SEAPLANE FLOAT WITH STEP FLAT STERN	7·0
	FUSELAGE ROUND NOSE ENTIRELY FAIRED	3·5		WHEELS RESISTANCE IN TERMS OF NOMINAL TYRE DIMENSIONS IN %M. D= WHEEL DIAMETER d=TYRE DIAMETER UNSYMMETRICAL UNFAIRED WHEEL	·00017 D.d.
	FUSELAGE 1 COCKPIT. POINTED NOSE	3·8		UNSYMMETRICAL FAIRED WHEEL. (TYRE FAIRINGS)	·0001 D.d.
	FUSELAGE. 1 COCKPIT ROUND NOSE	4·3		UNSYMMETRICAL FAIRED WHEEL (RIM FAIRINGS)	·00008 D.d.
	FUSELAGE. 1 COCKPIT & ROTARY ENGINE	6·0		RADIATOR. FREE AIR FLOW	15
	FUSELAGE 1 COCKPIT & NOSE RADIATOR	7·0 TO 9·0		WINDSCREEN OR OTHER FLAT PLATES.	29·0
	FUSELAGE 2 COCKPITS & ROTARY ENGINE	7·0		RUDDER & FIN IN LBS PER SQ FT. SUPERFICIAL AREA	·61
	FUSELAGE 2 OR 3 COCKPITS & NOSE RADIATOR	7·6 TO 10		TAIL PLANE & ELEVATOR IN LBS PER SQ. FT. SUPERFICIAL AREA	·77
	FUSELAGE 2 COCKPITS, RADIAL AIR COOLED ENGINE WITH PROJECTING CYLINDERS	12·0		STRUT CIRCULAR OR ANY CYLINDRICAL BODY	31
	FLYING BOAT HULL. 2 STEPS. HULL LEVEL	4·2		STRUT DE HAVILLAND, FINENESS RATIO 1·6	18·2
	FLYING BOAT HULL WITH 2 COCKPITS	5·5	R=4·0	STRUT ROUND TUBE. FAIRED AT REAR R & M 381 FINENESS RATIO 3·0	4·3
	SEAPLANE FLOAT WITHOUT STEPS POINTED STEM	5·0		STRUT ROUND TUBE FAIRED FRONT & REAR FINENESS RATIO 4·0	2·6
	SEAPLANE FLOAT WITH STEP POINTED STERN	6·0			

Fig. 20.

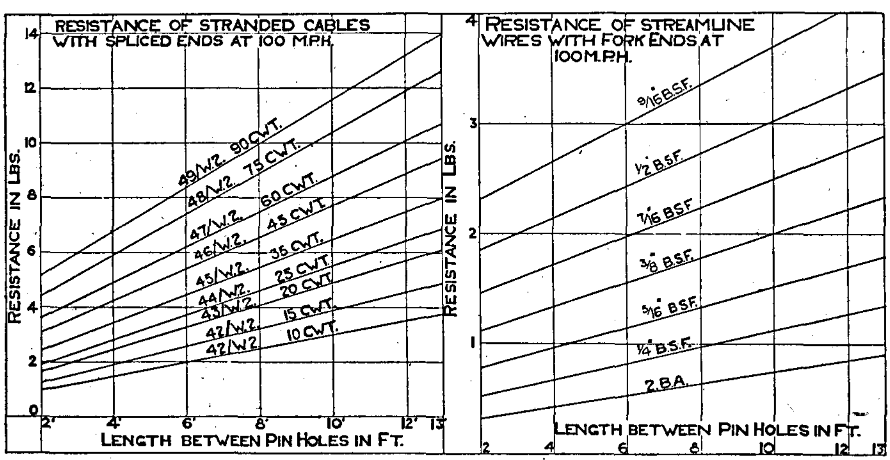

Fig. 21.

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PARASITE RESISTANCE DETAILS.

TABLE 1.

PARASITE RESISTANCE.
DETAILS OUT OF SLIPSTREAM AT 100 M.P.H. AND SEA LEVEL.

DETAIL.	FRONTAL AREA SQ. FT.	RESISTANCE PER SQ. FT. AT 100 M.P.H. LBS.	RESISTANCE OF DETAIL AT 100 M.P.H. LBS.	DISTANCE FROM DATUM.	MOMENT IN FT. LBS.
OUTER FRONT INTERPLANE STRUTS	1·7	4·3	7·30		
OUTER REAR INTERPLANE STRUTS	1·7	4·3	7·30		
INNER FRONT INTERPLANE STRUTS	2·0	4·3	8·60		
INNER REAR INTERPLANE STRUTS	2·0	4·3	8·60		
OUTER FRONT LIFT WIRES	5/32" B.S.F. × 144" × 2 OFF		4·0		
OUTER REAR LIFT WIRES	9/32" B.S.F. × 144" × 2 OFF		3·0		
OUTER INCIDENCE WIRES	1/4" B.S.F. × 78" × 4 OFF		3·44		
OUTER FRONT ANTI-LIFT WIRES	9/32" B.S.F. × 114" × 2 OFF		2·6		
OUTER REAR ANTI-LIFT WIRES	1/4" B.S.F. × 114" × 2 OFF		2·2		
INNER FRONT LIFT WIRES. (PART)	9/32" B.S.F. × 126" × 2 OFF		2·3		
INNER REAR LIFT WIRES. (PART)	5/16" B.S.F. × 126" × 2 OFF		1·6		
INNER INCIDENCE WIRES	1/4" B.S.F. × 76" × 4 OFF		3·44		
INNER FRONT ANTI-LIFT WIRES. (PART)	9/32" B.S.F. × 108" × 2 OFF		1·2		
INNER REAR ANTI-LIFT WIRES (PART)	1/4" B.S.F. × 108" × 2 OFF		1·0		
CHASSIS AXLE	1·15	4·3	4·95		
CHASSIS WHEELS 800 × 150			25·0		
			86·53		

TABLE 2.

PARASITE RESISTANCE
DETAILS IN SLIPSTREAM OF 100 M.P.H. AND SEA LEVEL.

DETAIL.	FRONTAL AREA SQ. FT.	RESISTANCE PER SQ. FT. AT 100 M.P.H. LBS.	RESISTANCE OF DETAIL AT 100 M.P.H. LBS.	DISTANCE FROM DATUM.	MOMENT IN FT. LBS.
FUSELAGE WITH RADIATOR, & TWO COCKPITS	16	8	128		
CENTRE SECTION STRUTS	2	4·3	8·6		
CENTRE SECTION LIFT WIRES	1/4" B.S.F. × 86" × 4 OFF		2·2		
CENTRE SECTION INCIDENCE WIRES	1/4" B.S.F. × 56" × 4 OFF		2·5		
INNER FRONT LIFT WIRES (PART)			2·3		
INNER REAR LIFT WIRES (PART)			1·6		
INNER FRONT ANTI-LIFT WIRES (PART)			1·2		
INNER REAR ANTI-LIFT WIRES. (PART)			1·0		
CHASSIS OLEO LEGS	3	4·0	12·0		
CHASSIS RADIUS RODS.	·4	4·3	1·6		
WIND SCREENS	1·0	29·0	29·00		
TAIL SKID	·4	20	8·0		
WINDMILL PUMP	2 OFF AT 5 LB. R EACH.				
EXHAUST PIPES	·25	20	5·0		
TAIL PLANE. SUPERFICIAL AREA	70·0	·75	52·5		
FIN & RUDDER SUPERFICIAL AREA	24·0	·60	14·4		
TAIL STAYS	1·0	6·0	6		
CONTROL WIRES			4		
CONTROL LEVERS	0·2	10·0	2		
			292·9		

Fig. 22.

PARASITE RESISTANCE AT VARIOUS SPEEDS & ALTITUDES

1	2	3	4	5	6	7	8	9
MACHINE SPEED M.P.H	SLIPSTREAM SPEED M.P.H	BODY & TAIL RESISTANCE GROUNDLEVEL IN SLIPSTREAM	OUT OF SLIPSTREAM	TOTAL	AT 5,000 FT	AT 10,000 FT	AT 15,000 FT	AT 17,500 FT
50	92	247·9	21·6	269·5	232·2	198·9	169·8	156·0
60	95·6	267·8	31·2	299·0	257·8	220·7	188·4	173·1
70	100·2	294·0	42·4	336·4	289·9	248·2	212·0	194·8
80	105·3	324·6	55·4	380·0	327·5	280·5	239·3	220·0
90	110·2	355·8	70·1	425·9	367·0	314·2	268·2	246·6
100	117·9	407·0	86·5	493·5	425·4	364·1	310·9	285·7
110	125·0	457·1	104·7	561·8	484·0	414·4	353·8	325·2
120	132·5	514·0	124·6	638·6	550·8	471·5	402·3	369·8

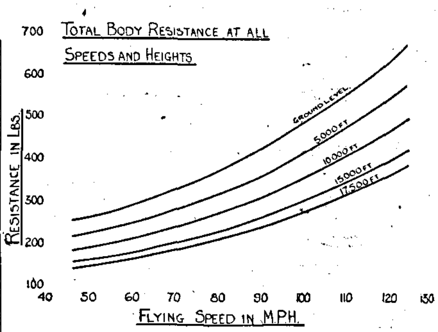

Fig. 22a.

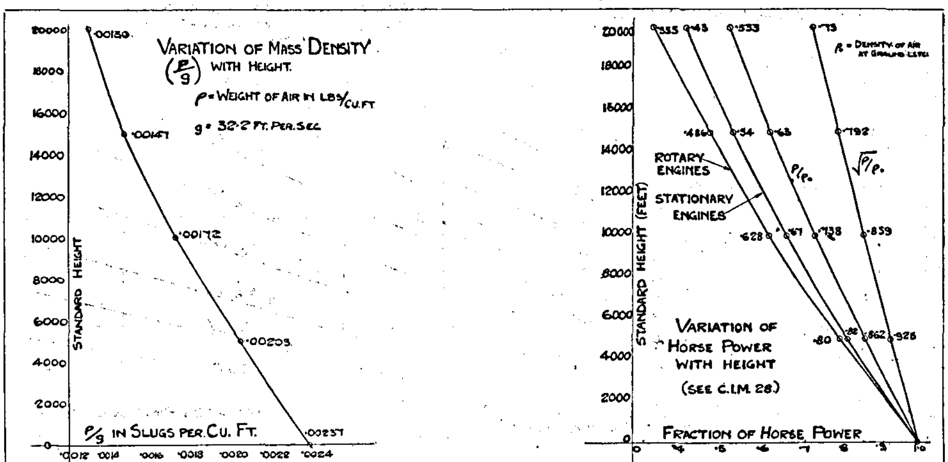

Fig. 23.

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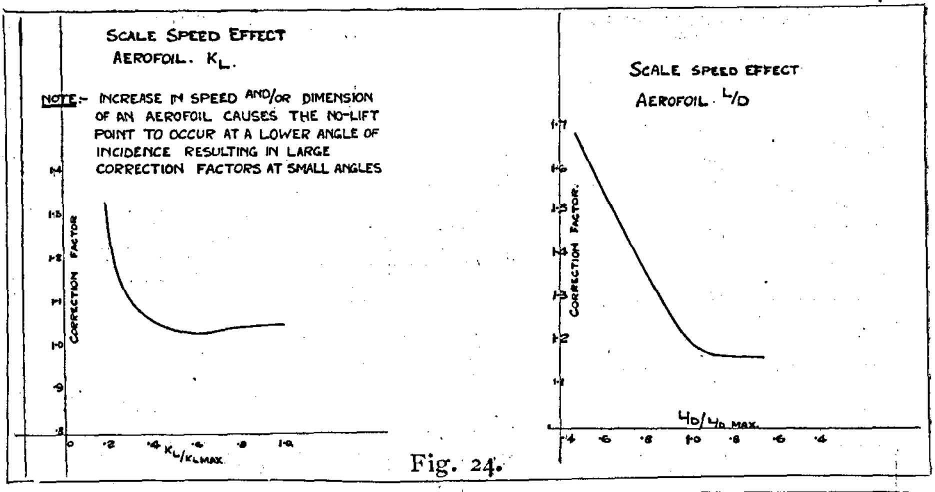

Fig. 24.

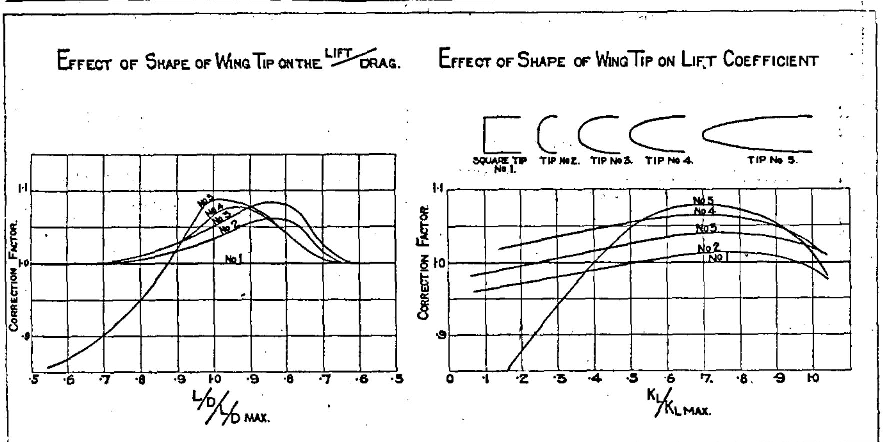

Fig. 25.

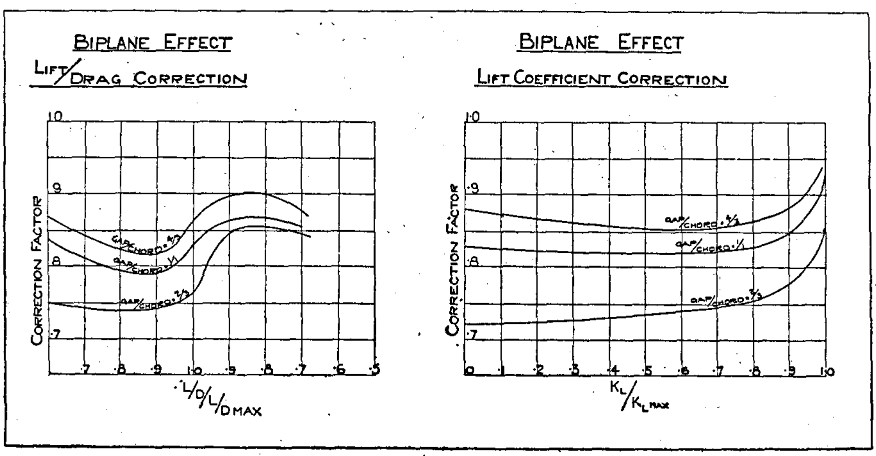

Fig. 26.

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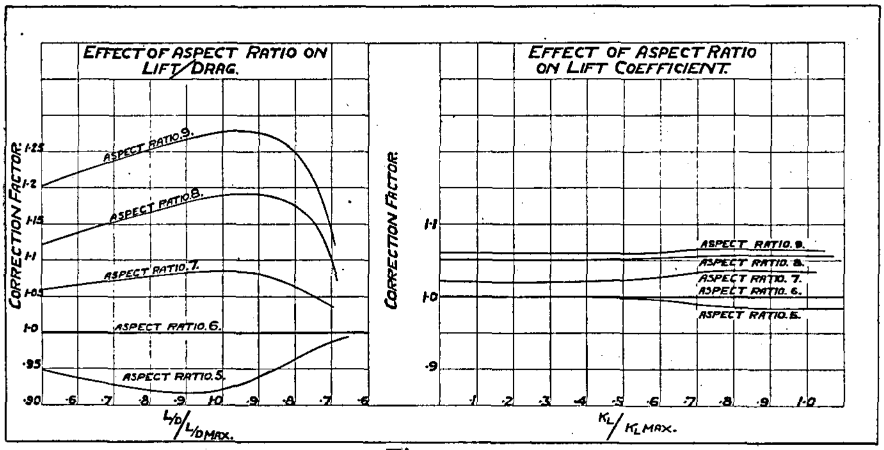

Fig. 27.

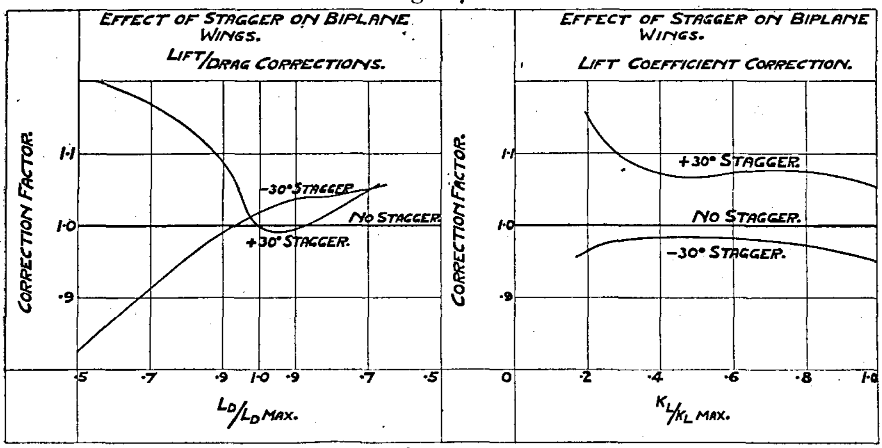

Fig. 28.

LIFT COEFFICIENT CORRECTIONS (K_L) TO MONOPLANE MODEL FIGURES.

1	2	3	4	5	6	7	8	9	10
$K_L/K_{L\,MAX}$	SCALE SPEED	WING TIP Nº 3	BIPLANE SPACING GAP/CHORD	ASPECT RATIO 7·28	STAGGER +15°	CORRECTION FACTOR 2.3.4.5.6	MODEL K_L	CORRECTED K_L 7×8	ANGLE OF INCIDENCE
1·0	1·05	1·02	·925	1·04	1·025	1·036	·617	·651	14°
·9	1·05	1·03	·85	1·04	1·032	·987	·555	·548	11·1°
·8	1·045	1·038	·83	1·04	1·04	·974	·494	·481	9·5°
·7	1·056	1·038	·822	1·04	1·04	·956	·432	·413	7·85°
·6	1·03	1·032	·82	1·037	1·038	·938	·37	·347	6·1°
·5	1·04	1·022	·82	1·035	1·036	·935	·309	·289	4·42°
·4	1·065	1·012	·821	1·035	1·04	·952	·247	·235	2·82°
·3	1·116	1·005	·822	1·035	1·048	1·00	·185	·185	1·36°
·2	1·3	·995	·823	1·035	1·06	1·169	·123	·144	0·05°

LIFT/DRAG CORRECTION (L/D) TO MONOPLANE MODEL FIGURES

1	2	3	4	5	6	7	8	9	10
L/D / L/D MAX	SCALE SPEED	WING TIP Nº 3	BIPLANE SPACING GAP/CHORD	ASPECT RATIO 7·28	STAGGER +15°	CORRECTION FACTOR 2.3.4.5.6	MODEL L/D	CORRECTED L/D 7×8	ANGLE OF INCIDENCE
·5	1·65	1·0	·85	1·07	1·1	1·65	8·9	14·69	0·55°
·95	1·23	1·036	·80	1·1	1·01	1·135	16·9	19·15	1·9°
1·0	1·2	1·05	·85	1·11	1·0	1·161	17·8	20·66	2·9°
·95	1·178	1·065	·85	1·1	·995	1·167	16·9	19·72	4·4°
·9	1·17	1·08	·86	1·099	·998	1·192	16·02	19·1	6·3°
·8	1·163	1·08	·866	1·09	1·01	1·198	14·24	17·05	8·62°
·7	1·161	1·026	·858	1·06	1·02	1·106	12·47	13·8	11·0°

PERFORMANCE CALCULATIONS

CHARACTERISTICS OF MAIN PLANES.

NUMBER OF PLANES	TWO
SPAN	44 FT
CHORD	6·56 FT
GAP	6'-6"
STAGGER	+15°
WING TIPS	ROUNDED Nº 3.
AEROFOIL SECTION	Nº 64.
SURFACE	566 SQ FT

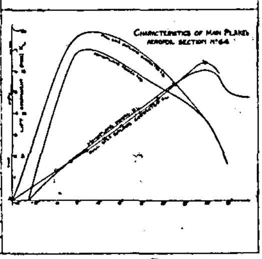

Fig. 29.

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RESISTANCE OF MAIN PLANES AT GROUND LEVEL. $\rho/g = \cdot 00237$ slugs/c.ft.

TOTAL WEIGHT OF MACHINE LOADED. = 4421 LBS.

SURFACE OF MAIN PLANES = 566 SQ.FT

SURFACE LOADING = 4421/566 = 7·8 LBS/SQ.FT

1	2	3	4	5	6	7	8	9
SPEED V_o M.P.H	REQUIRED K_L	ANGLE OF INCIDENCE	CORRESPONDING L/D	RESISTANCE OF MAIN PLANES IN LBS	SPEED AT 5,000 FT $V_o \div \sqrt{\rho/\rho_o}$	SPEED AT 10,000 FT $V_o \div \sqrt{\rho/\rho_o}$	SPEED AT 15,000 FT $V_o \div \sqrt{\rho/\rho_o}$	SPEED AT 17,500 FT $V_o \div \sqrt{\rho/\rho_o}$
50	·615	12·65°	11·7	378·0	53·9	58·2	63·2	65·8
60	·428	8·2°	17·3	255·7	64·65	69·9	75·8	78·95
70	·312	5·1°	19·5	226·8	75·4	81·5	88·4	92·1
80	·241	3·0°	19·82	223·0	86·2	93·2	101·0	105·3
90	·190	1·55°	18·6	237·8	97·0	104·8	113·6	118·4
100	·153	0·5°	16·3	271·2	107·8	116·4	126·3	131·6
110	·127	−0·3°	13·8	320·5	118·5	128·1	138·9	144·8
120	·107	−0·82°	11·7	378·0	129·3	139·8	151·6	157·9

MAXIMUM K_L (FROM FIG. 29) = ·65

MINIMUM FLYING SPEED AT GROUND LEVEL $= \sqrt{\dfrac{7 \cdot 8}{\cdot 00237 \times \cdot 65}}$ = 71·15 FT. PER SEC. OR 48·8 M.P.H.

MINIMUM FLYING SPEED AT ANY OTHER ALTITUDE = $48 \cdot 8 \div \sqrt{\rho/\rho_o}$ WHICH GIVES 52·6 M.P.H. AT 5,000 FT., 56·8 M.P.H. AT 10,000 FT., 61·5 M.P.H. AT 15,000 FT AND 64·2 M.P.H. AT 17,500 FT.

WING RESISTANCE AT MINIMUM FLYING SPEED $= \dfrac{4421}{9 \cdot 5} \doteq 465 \cdot 6$ LBS

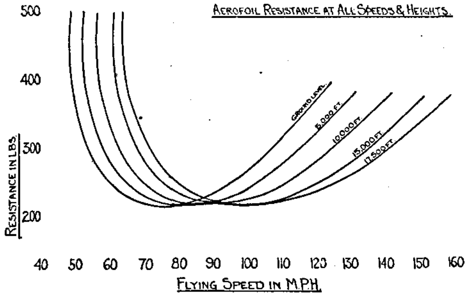

Fig. 30.

TOTAL RESISTANCE OF COMPLETE AEROPLANE AT ALL SPEEDS & ALTITUDES.

SPEED IN MPH	SEA LEVEL			5000 FT			10,000 FT.			15000 FT.			17,500 FT.		
	PARASITE RES. (lbs)	AEROFOIL DRAG (lbs)	TOTAL RES. (lbs)	PARASITE RES. (lbs)	AEROFOIL DRAG (lbs)	TOTAL RES. (lbs)	PARASITE RES. (lbs)	AEROFOIL DRAG (lbs)	TOTAL RES. (lbs)	PARASITE RES. (lbs)	AEROFOIL DRAG (lbs)	TOTAL RES. (lbs)	PARASITE RES. (lbs)	AEROFOIL DRAG (lbs)	TOTAL RES. (lbs)
50	269·5	378·0	647·5	232·0			199·0								
60	299·0	255·7	554·7	258·0	277·0	535·0	220·0	332·0	552·0	188·4			173·0		
70	336·4	227·0	563·4	290·0	237·5	527·5	248·0	250·0	498·0	212·0	285·0	497·0	195·0	305·0	500·0
80	380·0	224·0	604·0	327·5	222·0	549·5	280·5	230·0	510·5	239·3	245·0	484·3	220·0	252·0	472·
90	426·0	238·0	664·0	367·0	225·0	592·0	314·0	222·0	536·0	268·2	226·0	494·2	246·6	230·0	476·6
100	493·5	271·0	764·5	425·4	245·0	670·4	364·0	230·0	594·0	311·0	221·0	532·0	285·7	221·0	506·7
110	562·0	320·5	882·5	484·0	280·0	764·0	414·0	250·0	664·0	354·0	233·0	587·0	325·0	225·0	550·0
120	638·6	378·0	1016·6	551·0	330·0	881·0	471·5	285·0	756·5	402·3	250·0	652·3	370·0	240·0	610·0

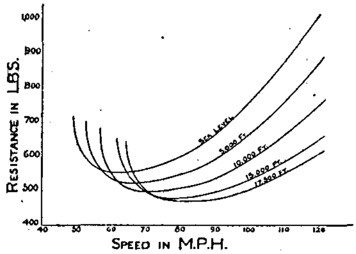

Fig. 31.

THRUST HORSE POWER REQUIRED AT ALL SPEEDS & HEIGHTS.

SPEED M.P.H.	SEA LEVEL.		5,000 FT		10,000 FT.		15,000 FT.		17,500 FT.	
	TOTAL RESISTANCE IN LBS.	THRUST H.P REQD.	TOTAL RESISTANCE IN LBS.	THRUST H.P REQD.	TOTAL RESISTANCE IN LBS.	THRUST H.P. REQD.	TOTAL RESISTANCE IN LBS.	THRUST H.P REQD.	TOTAL RESISTANCE IN LBS.	THRUST H.P REQD.
50	647·5	86·3								
60	554·7	88·7	535·0	85·6	552·0	88·3				
70	563·4	105·1	527·5	98·5	498·0	93·0	497·0	92·9	500·0	93·4
80	604·0	128·9	549·5	117·2	510·5	108·8	484·3	103·3	472·0	100·6
90	664·0	159·4	592·0	142·0	536·0	128·6	494·2	118·6	476·6	114·2
100	764·5	204·0	670·4	178·8	594·0	158·4	532·0	142·0	506·7	135·1
110	882·5	258·2	764·0	223·7	664·0	194·4	587·0	171·8	550·0	161·0
120	1016·6	325·3	881·0	281·9	756·5	242·0	652·3	208·9	610·0	195·2

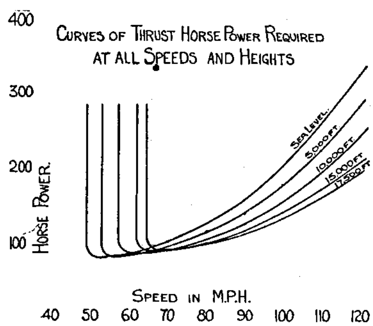

Fig. 32.

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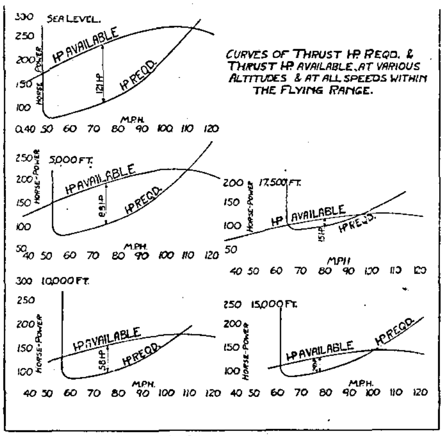

Fig. 33.

ALTITUDE IN FEET.	MAXIMUM SURPLUS H.P.	RATE OF CLIMB IN FT. PER MIN.	CLIMBING SPEED M.P.H.	RATE OF CLIMB IN MINS /1,000 FT
SEA LEVEL	121	903	74	1·107
5,000 FT.	88	658	75	1·52
10,000 FT.	58	433	76	2·31
15,000 FT.	29	216·3	78	4·62
17,500 FT.	15	112	80	8·92

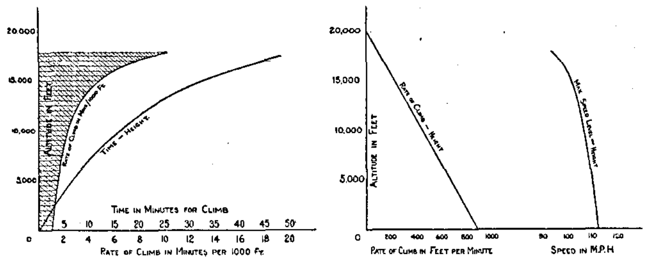

Fig. 34.

COMPLETE PERFORMANCE CURVES.

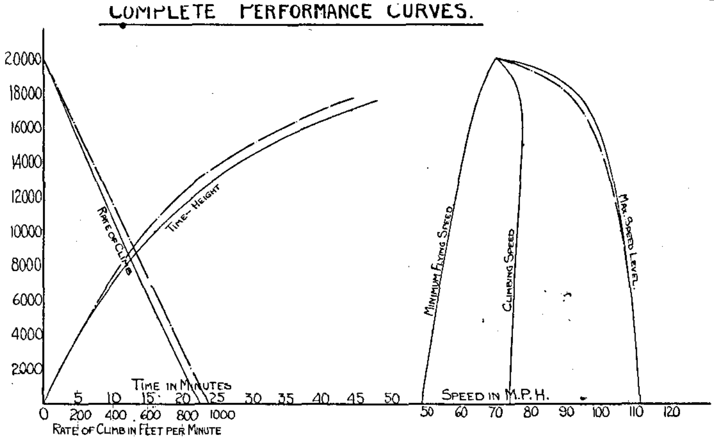

Fig. 35.

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