
DESIGN
CONSTRUCTION OF
Flying
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THE DESIGN AND CONSTRUCTION OF FLYING MODEL AIRCRAFT

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By

D. A. RUSSELL

A.M.I.Mech.E.

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Author's Note

THIS book in no way deals with full-sized aerodynamics. It was first published in 1937, the intention being to provide a text-book, by the use of which an aero-modeller could work out the complete design and peformance estimation, of medium and large-sized model aircraft, both power and rubber driven.

That the first edition sold out well, and that there has continued to be a demand for the book, has led to the publication of this second, and considerably enlarged, edition.

Whilst the technical part remains essentially as before, the opportunity has been taken to provide a complete set of new sketches, and introduce some further data in the chapter on rubber motors. The practical part of the book has been enlarged by the addition of several new chapters, together with the inclusion of a considerable number of new photographs and sketches.

Except where otherwise stated, the results of researches described in the book and in the Appendices, are the original work of the author, carried out with the aid of a wind tunnel and other equipment specially designed and built for the purpose.

The author realises that model aircraft constructors are keenly individualistic, each with his own theories and ideas, and no attempt has been made to direct the reader into any particular viewpoint. It is hoped that the technical chapters will encourage the reader to design his own 'plane, and that from the photographs and sketches with which the book is illustrated, he may find inspiration sufficient to enable its construction to be undertaken and achieved with success.

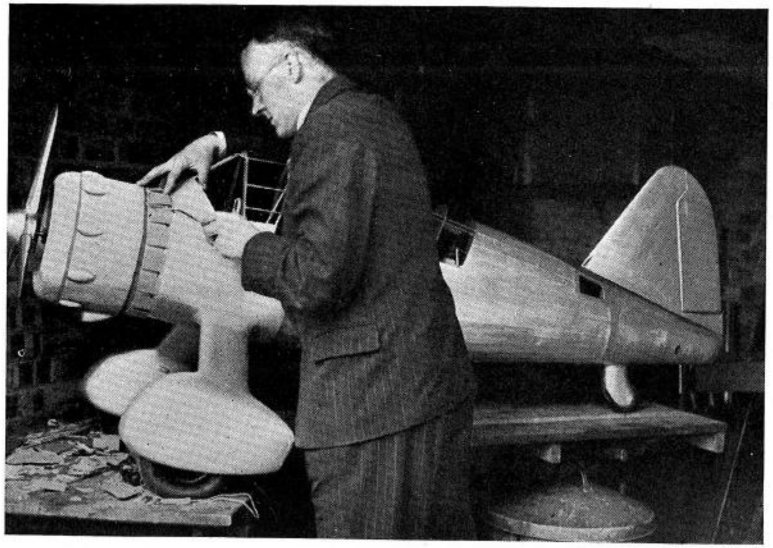
The author acknowledges with grateful thanks the co-operation of Mr. C. A. H. Pollitt in providing a set of new sketches for the book; also the assistance of Mr. J. H. Elwell in correcting proofs and compiling the index.

Acknowledgment is made on page 243 to those model aircraft firms which have kindly loaned blocks and photographs for use in this book, and acknowledgment is also made to the Editors of *The Aero-Modeller*, *Practical Mechanics*, and *Popular Flying*, for permission to quote from the text of several of the author's articles published in these journals, and for the use of several blocks, sketches and photographs which accompanied them.

Market Harborough.
1940. D. A. RUSSELL.

CONTENTS

APPENDIX 1.—FORMULÆ.



The Author at work on his latest 'plane. This is a flying scale model of the Westland " Lysander." It is one-fifth full size, and is thus 10 ft. span and 6 ft. long. It is equipped with a $1\frac{1}{2}$ h.p. 4-cylinder engine, which is described in the chapter on Engine Testing. The machine will be fitted with flaps, slots, and an automatic stabilising control.

CHAPTER I

AIRFOILS

How an airfoil lifts—" Flat plate " and double-surfaced cambered airfoils—Formula for calculating flying speeds—Lift and drag coefficients of airfoils—The boundary layer—Maximum thickness-to-chord ratio of airfoil sections—The geometric chord and angle of attack—The angle of attack of zero lift—Various types of airfoil sections.

(1) In considering the movement of a flat plate airfoil through the air it must be appreciated that if it moves in a horizontal plane it does not generate any lift—but only drag. In this position the airfoil is said to be at the " angle of attack of zero lift."

As soon as the airfoil is tilted, so that the leading edge is higher than the trailing edge, the airfoil is said to be inclined at a " positive angle of attack," and lift is generated. At the same time the drag is increased, due to the greater resistance of the inclined surface of the airfoil. If the angle of attack is increased to much beyond 14 to 16 degrees the ratio of lift to drag falls off to a figure lower than it would be for a smaller angle of attack. If the angle of attack is increased to much beyond 18 to 22 degrees the airfoil is likely to stall.

The aim of the designer, therefore, is to evolve an airfoil section which gives the highest lift/drag ratio—and to know at what angle of attack this section should be set.

In comparing the lift coefficients of different airfoil sections, it is useful to consider them in relation to that of a flat plate—since this may be considered the most elementary airfoil section, and therefore serves as a useful basis for comparison with the various curved sections which have been developed—with the object of obtaining as great a lift as is possible from a given wing area.

The pressure acting on the plate may be ascertained from the formula—

$$P_v = P\frac{2 \text{ Sin } a \text{ Cos } a}{1+\text{Sin}^2 a} \quad \dots \quad \dots \quad \dots \quad \dots \quad (1)$$

i.e., $P = P_v \dfrac{(1+\text{Sin}^2\ a)}{2\ \text{Sin}\ a\ \text{Cos}\ a}$

(As the size and shape of the flat plate would affect its efficiency as an airfoil, this formula, as it stands, would give only an approximate result, and in actual practice other factors would have to be taken into consideration. However, the example here given is sufficiently accurate to show, quite definitely, how low is the lift efficiency of a flat plate airfoil compared with one of normal design).

Assuming the plate to be inclined at an angle of attack of 5 degrees.

$$P = \frac{1\ (1 + \cdot 0872^2)}{2 \times \cdot 0872 \times \cdot 9962}$$
$$= 6 \cdot 1914 \text{ pounds.}$$

The velocity of the wind necessary to support the plate at this angle may be ascertained from the formula—

$$V = \sqrt{\frac{P}{\cdot 0032}} \text{ miles per hour} \quad \ldots \quad \ldots \quad (2)$$
$$= 43 \cdot 99 \text{ miles per hour.}$$

Similarly the speed may be found for any angle of attack by appropriate substitution in the formula.

(2) Now for light model aircraft with wing loadings not exceeding 3 or 4 ounces per square foot of wing area, the simple " flat plate " type of airfoil may be used. For, whilst its efficiency as regards lift is poor, it possesses the advantage of being very easily and lightly constructed, and thus its ratio of weight to surface area is very good.

But, as the minimum flying speed at which an airfoil will support a given weight increases as the square of the loading, it will be appreciated that as soon as the loading increases beyond 4 or 5 ounces per square foot of wing area, relatively high speeds are necessary to sustain the flat plate type of airfoil in flight—and it is for this reason that the double-surfaced " shaped " airfoil has been developed, since its greater efficiency allows of a correspondingly slower minimum flying speed being obtained.

For every given airfoil section, and for every angle of attack at which this airfoil section is fixed in relation to the

horizontal plane, there obtains a definite set of lift and drag conditions—and these are ascertained by means of wind-tunnel tests of models of the airfoil section.

By plotting the results obtained from these tests in the form of " lift to drag " curves, the relative efficiency of the airfoil section, at varying angles of attack, may be readily observed, and comparison made with the results obtained from tests on other airfoil sections.

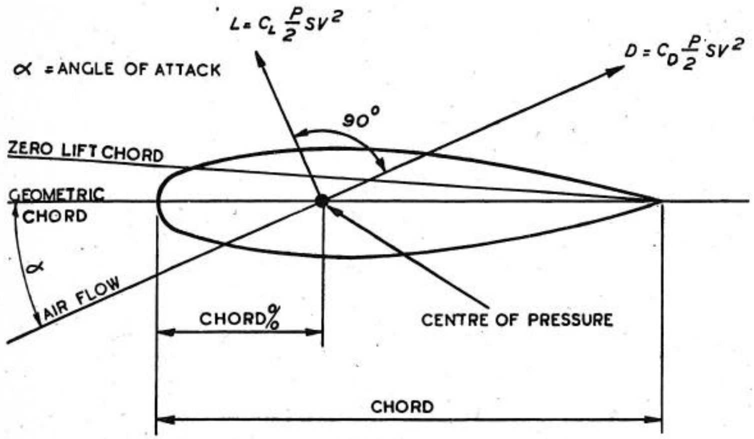


FIG. 1.

For the purpose of calculations, the terms " Lift Coefficient " (C_l) and " Drag Coefficient " (C_d) are introduced, their numerical value depending on the shape, and angle of attack, of the airfoil section to which they refer. Their application in the appropriate formula, allowing of flying speeds being calculated for any set of conditions.

The formula is $L = C_l \frac{\rho}{2} SV^2 \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (3)$

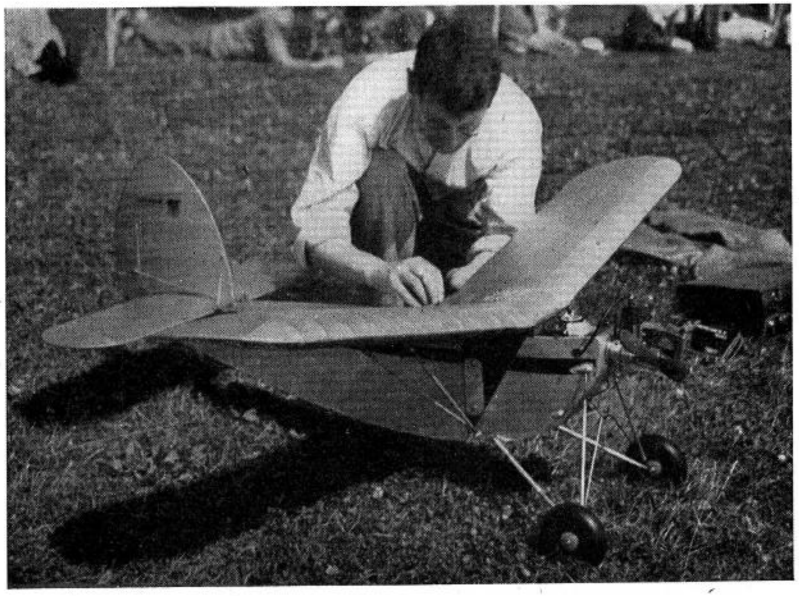
where L = Weight in pounds.
ρ = Mass density of air.
= $\cdot 002378$ in slugs per cubic foot.
S = Wing area in square feet.
V = Velocity in feet per second.
C_l = Lift coefficient.

This formula may be rewritten $C_l = \dfrac{L}{\frac{\rho}{2} SV^2}$

and applied to the " flat plate " type of airfoil section to obtain values of C_l.

Taking the example quoted above (where wing loading = 1 pound per square foot of lifting surface)

$$C_l = \frac{1}{\frac{\rho}{2} \times 1 \times 64{\cdot}5^2} = {\cdot}202 \text{ (at angle of attack of 5 degrees)}$$



Mr. Trevethic, with one of his petrol 'planes. This model has an adjustable fin controlled by time-switch, and is described in the chapter on " Flying the Model."

Since, at the same angle, any well designed airfoil section will have a lift coefficient of approximately ·8, it will readily be seen how relatively inefficient (except for very light wing loadings) is the " flat plate " type of airfoil section.

(3) All well-known airfoil sections published in the Reports and Memoranda of the Air Ministry's Aeronautical Research Committee contain values for C_l at varying angles of attack, for the particular airfoil section covered by each report. And, provided the model aircraft designer keeps rigidly to the airfoil section, when constructing his wing, and

mounts it at the correct angle of attack, he may be assured of reliable results from his calculations.

Suppose it is required to find the lift obtainable from an aircraft designed to the following specification:—

Span = 10 feet.
Wing Area = 10 square feet.
Speed = 30 feet/second.
C_l = 1·6

Substituting in the equation $L = C_l \frac{\rho}{2} SV^2$

$$\text{Then } L = 1{\cdot}6 \times {\cdot}001189 \times 10 \times 30^2$$
$$= 17{\cdot}1 \text{ pounds.}$$

Continuing the example, the same aircraft may be designed with an airfoil with $C_l = 1{\cdot}3$,

In which case

$$L = 1{\cdot}3 \times {\cdot}001189 \times 10 \times 30^2$$
$$= 13{\cdot}9 \text{ pounds.}$$

If, however, the aircraft using this airfoil is desired to lift the same weight, i.e., 17·1 pounds, then the flying speed must be increased, and the formula $L = C_l \frac{\rho}{2} SV^2$ may be rewritten.

$$V^2 = \frac{L}{C_l \frac{\rho}{2} S} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (4)$$

Substituting the appropriate figures from the example

$$V = \sqrt{\frac{17{\cdot}1}{1{\cdot}3 \times {\cdot}001189 \times 10}}$$
$$= \sqrt{\frac{17{\cdot}1}{{\cdot}01545}}$$
$$= \sqrt{1105}$$
$$= 33{\cdot}3 \text{ feet per second.}$$

Unfortunately, whilst a great number of airfoil sections have been made available, together with full particulars of their characteristics, not much use appears to be made of them; instead, certain formulæ have crept into common use, from which it is thought that minimum flying speeds may be calculated.

For instance, a formula often used by model aircraft designers is that which states that the minimum flying speed at which an airfoil will lift may be ascertained from

$$V = \sqrt{840 \times L} \quad \dots \quad \dots \quad \dots \quad \dots \quad (5)$$

when V = minimum flying speed in feet per second, and L = wing loading in pounds per square foot.

Thus for a wing loading of 1 pound per square foot

$$V = \sqrt{840 \times 1} = 29 \text{ feet per second}$$
$$= 19{\cdot}75 \text{ m.p.h.}$$

Another formula commonly used, is that which states

$$V = \sqrt{W} \times 6 \quad \dots \quad \dots \quad \dots \quad \dots \quad \dots \quad (6)$$

where V = minimum flying speed in miles per hour, and W = wing loading in ounces per square foot.

Here, for the *same* wing loading, the minimum flying speed $V = \sqrt{16} \times 6$
$= 24$ miles per hour ! ! !

Such formulæ as these two above quoted can only apply each to one definite airfoil section inclined at a given angle of attack.

The minimum flying speed, generally speaking, varies as the square root of the loading; and whilst, therefore, these formulæ *do* give the speeds for different wing loadings, they can only apply to *one given airfoil section, inclined at one given angle of attack; and in the absence of this vital information, the use of these formulæ can be of little help.*

(4) The drag of an airfoil is dependent partly on the angle of attack, partly on the aspect ratio, and partly on the degree of smoothness of the airfoil surface, and its value may be ascertained from the formula—

$$D = C_d \frac{p}{2} S V^2 \quad \dots \quad \dots \quad \dots \quad \dots \quad \dots \quad (7)$$

where D = Drag in pounds.
p = Mass density of air.
= ·002378 in slugs per cubic foot.
S = Wing area in square feet.
V = Velocity in feet per second.
C_d = Drag coefficient.

As the angle of attack increases so does the drag, but so also does the lift—the ratio remaining fairly constant until

within a few degrees of the stall, when the drag increases very rapidly, this *lowering* the ratio of lift/drag.

Practical research supplemented by theoretical considerations show that an airfoil of infinite span would not have any *induced* drag due to generated lift; but only turbulent drag—due to eddies in the airstream leaving the trailing edge; and frictional drag—due to the airstream flowing over the airfoil surface.

The drag resulting from the eddies leaving the wing tips is considerable, and for this reason the aspect ratio should be as large as possible, since the farther apart are the wing tips the smaller is their percentage effect on the wing as a whole.

For model gliders, aspect ratios up to 20 : 1 may be used. For rubber-driven models the ratio varies from 8 : 1 up to 12 : 1; whilst for large power-driven models the aspect ratio is usually about 7 : 1. (This lower figure is perhaps accounted for by limitations in present day supplies of material, as suitable wood for wing construction is not generally procurable in lengths exceeding 4 feet—thus tending to limit the wing span to some 8 feet.)

To obtain the necessary area of lifting surface, the designer is compelled to use a chord of 14 or 15 inches—and thus an aspect ratio of only 7 : 1 is used.

As will be shown in a later chapter—it is possible to construct wings of the cantilever type of considerably greater span than 10 feet—and in view of the increased efficiency obtained by keeping the wing tips as far apart as possible—it is considered that the aspect ratio of flying model aircraft of any type should not be less than 8 : 1.

(5) When the chord of an airfoil approaches 15 or 16 inches, break up of the "Boundary Layer" near the trailing-edge is likely to occur, with a consequent *increase* in drag due to turbulence, and *decrease* in lift due to the breakaway of the airstream flow from the surface of the airfoil.

J. V. Connoley* states that "In a moving fluid, which flows along a body, there is a thin layer adjacent to the body which is at rest at the surface of the body and has an increasing velocity until it is moving at the speed of the stream."

* "Aerodynamics for the Aero-modeller," *Aero-Modeller,* Jan.-Feb., 1936.

This is the boundary layer—its thickness is small, and is defined as "the distance from the surface at which the air is moving at a velocity from 95 per cent of that of the airstream."

The thickness of the layer can be found from the formula developed by Van der Hegge Zijnen:

$$T = 4{\cdot}5 \sqrt{\frac{KL}{V}} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (8)$$

where T = Thickness of layer, in feet.
K = ·00016.
= Kinematic viscosity of air.
L = Distance from the leading edge of the airfoil in feet.
V = Airspeed, in feet per second.

In the above formula it will be observed that as the distance (L) from the leading edge increases so does the thickness of the boundary layer.

Taking a point 18 inches from the leading edge of an airfoil of say, $18\frac{1}{4}$ inches chord, and assuming an airspeed of 40 feet per second:

$$T = 4{\cdot}5 \sqrt{\frac{{\cdot}00016 \times 1{\cdot}5}{40}}$$

$$= {\cdot}01102 \text{ feet (or about } \tfrac{1}{6} \text{ inch).}$$

(6) Now as the boundary layer increases in thickness, so it tends to form into "ripples"; these grow into "waves," which finally "break" and roll over each other in much the same way as real waves on the sea shore.

This "break up," and consequent increase in drag due to the turbulence created, usually occurs at a Reynold's number of about 3,000.

The Reynold's number of the thickness of the boundary layer may be calculated by multiplying the thickness of the layer by the speed and dividing by the kinematic viscosity.

$$\text{i.e., R.N.} = \frac{TV}{K} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (9)$$

In the example quoted above

$$\text{R.N.} = \frac{{\cdot}01102 \times 40}{{\cdot}00016}$$

$$= 2760$$

Thus it will be seen that for an airfoil moving at a speed in the neighbourhood of 40 feet per second—the chord must

Petrol 'planes are built in considerable numbers in Italy. Here is a biplane built in 1939 at the Parma School of Model Aeronautics.

not exceed 17 or 18 inches, if turbulence of the boundary layer is to be avoided at the trailing-edge of an airfoil unless, of course, special means are adopted to delay the development of the turbulent layer for as long as possible. This can be done, to a certain extent, by ensuring that the surface of the airfoil is made as smooth and even and as free from obstructions as possible.

(7) The aim of the designer is at all times to keep the drag as low as possible—and to do this the natural inclination is to use an airfoil section which is thin and has very little camber. Such an airfoil would have a ratio of maximum thickness to chord of about 1:18; and a ratio of maximum camber to chord of 1:50.

Airfoil sections in this class are suitable for lightly-loaded rubber-driven models of small span; and whilst the thin airfoil section permits of a very light framework (which in some cases need not even be double-surfaced), it does not lend itself to the construction of large spans, due to the impossibility of incorporating a stout " backbone " in the section.

To obtain the highest value for C_l the airfoil section must be " thickened-up," so that the maximum thickness to chord ratio is about 1:9. At a ratio of 1:8 the best L/D ratio is obtained, any further " thickening-up " putting the rear upper surface of the airfoil at such a large angle of incidence as to cause " breakaway " of the airstream.

This class of airfoil section, which should always be used for large power-driven aircraft, allows the designer to incorporate, at the thickest part, a main spar of immense strength, running throughout the span of the wing.

For large rubber-driven and small power-driven aircraft, where the wing loadings may vary from 4 to 10 ounces per square foot of lifting area, there is a choice from any of the well-known airfoil sections published in the Aeronautical Research Committee's Reports and Memoranda; or the development of an airfoil section by the model aircraft designer himself.

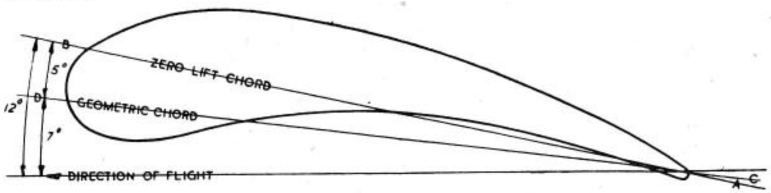


FIG. 2.

(8) Fig. 2 shows a typical " slow-flying " high lift/drag ratio airfoil section in which A B is the " zero lift chord " (i.e., if the airfoil is arranged so that A B is panelled to the direction of the airstream, no lift is generated).

C D is the geometric chord and is usually the further distance (in a straight line) between the leading- and trailing-edges.

All angles of attack are measured from the geometric chord, and thus for the airfoil to be set at an angle of attack

of 0 degrees, the geometric chord must be inclined at a *negative* angle to the direction of the airstream.

For an airfoil of the section illustrated, this angle will be about 6 degrees.

It follows, that when any airfoil is arranged so that the geometric chord is parallel to the airstream, actually it is inclined at an angle of attack equal to the amount known as the " angle of attack of zero lift."

Thus, if a specification calls for an angle of attack of 12 degrees (and the angle of attack of zero lift for the particular airfoil section is 3·5 degrees)—then the angle that the geometric chord subtends to the line of flight is $12 - 3{\cdot}5 = 8{\cdot}5$ degrees.

A method for finding the direction of the zero lift chord which gives results accurate to within ·4 of a degree, is that given by K. D. Wood,* and which consists of drawing a line from the trailing-edge through a point located half-way between the upper and lower surfaces of the airfoil, at a distance ·4 of the chord from the leading-edge—as shown in Fig. 3.

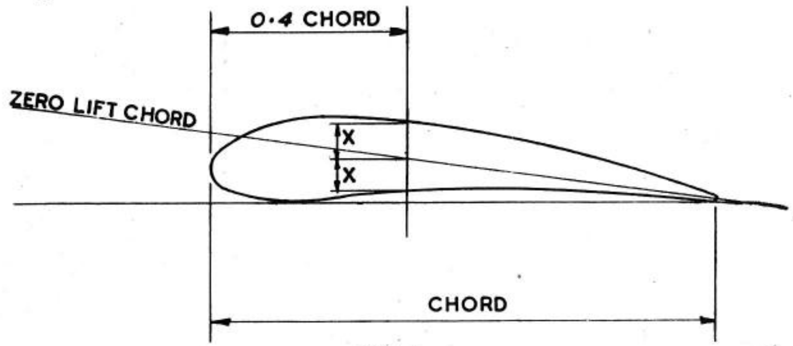


FIG. 3.

If the geometric chord is then drawn, the angle included by the zero lift chord and the geometric chord is the angle of attack of zero lift.

(9) In choosing the most suitable airfoil section to use for a particular aircraft—the following points should be considered:

(*a*) For very light loadings, a thin and nearly flat airfoil

* K. D. Wood, *Technical Aerodynamics*.

section of maximum thickness to chord ratio of 1:18 should be used.

(*b*) For very light loadings, but where duration of flight is of importance, a slightly heavier airfoil section, of maximum thickness-to-chord ratio of 1:15 should be used; care being taken to see that the thickest part of the airfoil section is not more than ·25 of the chord distance from the leading edge.

(*c*) For medium loadings—say from 4 to 10 ounces per square foot of lifting area—there are available a number of well-known sections, choice from which may be made, giving preference to the slightly thicker sections for slow flying combined with heavy loadings; and to the slightly thinner, though not less deeply cambered sections, for faster flying aircraft.

(*d*) For speed record work—since adequate lift is easily obtainable owing to the high speed—a fairly thin airfoil, with a perfectly flat under-surface should be used; great attention being paid to obtaining as smooth and even a finish as possible to the airfoil surfaces, so that the C_d is kept to a minimum.

(*e*) For loadings of 12 ounces and upwards per square foot of lifting surface—i.e., for power-driven aircraft, airfoil sections of maximum thickness to chord ratios of between 1:10 and 1:8 should be used.

Firstly, this section allows of the introduction of substantial spars throughout the span of the airfoil, and secondly, the higher C_d of the thick section keeps the speed down; whilst, at the same time, due to the high C_l, the necessary lift is obtained.

(*f*) It is obvious that any airfoil, given the opportunity, will adjust itself to that angle at which the C_d is at its lowest value, regardless of how large, or small, the C_l may be. Consideration must, therefore, be given not only to its lift and drag characteristics when in power flight, but also when in a free glide; obviously, then, an airfoil which generates very little lift, when moving at its angle of minimum drag, should not be used for a heavily-loaded aircraft.

The minimum drag of the majority of the fairly thick airfoil sections used for power-driven aircraft occurs at an angle of about 4 degrees; whilst the angle of attack of zero lift is usually about 6 degrees.

Thus the airfoil is actually at an angle of 1 or 2 degrees

positive inclination to the line of flight, when in a natural glide.

However, for the same angle of attack, the C_l varies somewhat for different airfoils, and therefore care should be taken, when considering an airfoil section, to see that when the C_d is at its minimum, the C_l is sufficient to give the aircraft a useful lift during the glide.

(*g*) The aspect ratio should be kept as large as possible, as this tends to improve the general efficiency of the airfoil.

(*h*) The chord should be kept as small as possible. As the angle of attack increases, so the centre of pressure of the airfoil moves. The shorter the chord, therefore, the smaller is this movement.

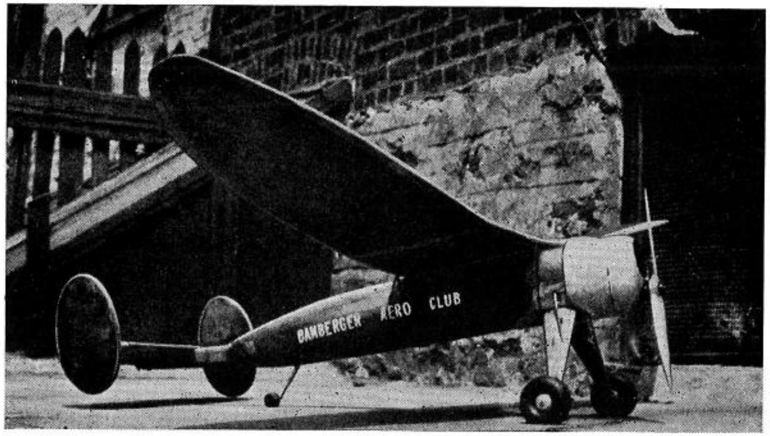
(*i*) The centre of pressure appears to move least in airfoils of thick section, whose angle of attack of zero lift is 5 or 6 degrees, and whose under-surface is well cambered.

The movement being from about 40 per cent chord distance from the leading-edge, to about 33 per cent chord, for angles of attack up to about 13 degrees, and then slightly *backwards* as the stall approaches.

In airfoils with flat under-surfaces the centre of pressure movement appears to be somewhat larger, and to vary from 40 per cent chord at small angles of attack, to 33 per cent chord as the angle of stall is approached.

In airfoils with upswept trailing-edges, the centre of pressure is initially nearer the leading-edge, and moves slightly *backwards*, as the angle of attack increases.

Airfoils of this latter type, whilst having good L/D ratios at small angles of attack, are not so well favoured as the angle increases; until, at the stall, the drag increases very rapidly; they are, however, very stable at most normal angles, the centre of pressure remaining practically stationary.



This interesting machine was built in America, and is a typical example of the type of power-driven model aircraft built in that country. The fuselage is of monocoque construction, and the fully cantilever undercarriage legs are fitted with shock absorbers.

CHAPTER II

AIRFOILS AND FUSELAGES FOR MONOPLANES AND BIPLANES

Trailing-edge vortices—The lift distribution of an airfoil —Streamlines—Plan form of airfoils—Tapered airfoils— " Down-wash "—The effect of trailing-edge vortices on the drag of (A) low-wing fuselages, (B) high-wing fuselages— Suitable fuselage sections—Prevention of " breakaway " and stall at the junction of an airfoil with a fuselage—The low wing versus the high-wing monoplane—The lift of a biplane—Advantages of biplane design—The distribution of lift over the wings of a biplane—Position of the stabiliser in a biplane—The position of thrust-line in a biplane.

(1) In the preceding chapter mention has been made of the drag due to the trailing vortices which originate at the wing tips of an airfoil. Vortices originate also all across the trailing-edge of an airfoil due to the fact that the lift is not constant across the span, but generally is proportionate to the ordinate of an ellipse whose major axis is equal to the span,

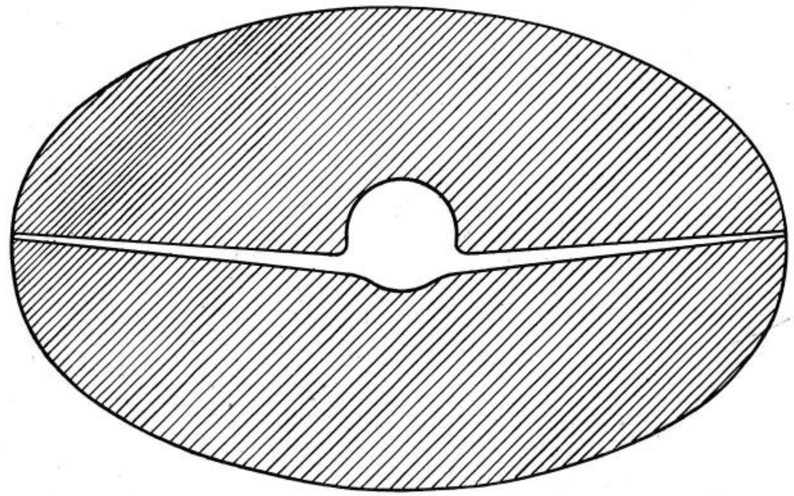


FIG. 4.

Diagram illustrating the empirical assumption that the air affected by an airfoil is contained within an ellipse whose major axis is the span of the wing, and the minor axis four-fifths of the span. The mass of air contained within this ellipse may be regarded as the region of air disturbed by the passage of the aircraft.

and whose minor axis is about equal to four-fifths of the span.

In Fig. 4 the shaded portion indicates the mass of air affected by an airfoil, from which it will be seen that the lift is greatest at the centre of the span, and decreases to zero at the tips. This is known as " elliptic loading."

The effect of this unequal distribution of lift is that there is a considerable decrease of pressure above the centre of the airfoil, and a considerable increase of pressure underneath the centre, resulting in the " streamlines " above the airfoil tending to flow inwards (due to the partial vacuum formed) and those underneath tending to flow outwards (due to the pres-

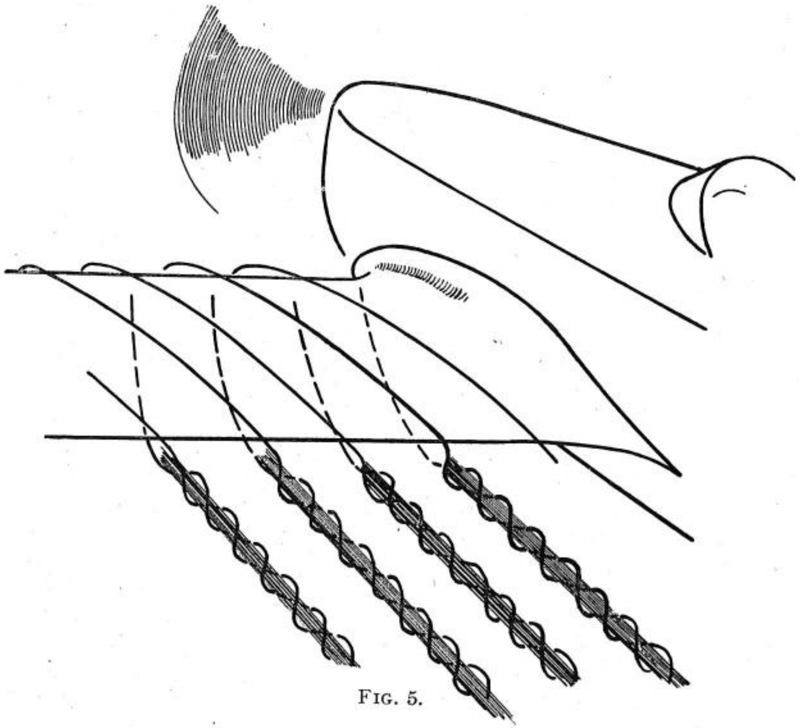


FIG. 5.

Formation of trailing edge vortices.
The sketch above illustrates the flow of air round a wing. It will be seen that the air passing over the top surface of the wing is flowing slightly inboard towards the fuselage, while the air on the underside of the wing tends to flow outboard and away from the fuselage.

sure), as shown in Fig. 5. At the trailing-edge the two streams meet, and since their paths are crossing, rotating vortices are

formed, those on one side of the fuselage rotating in the opposite direction to those on the other side.

Finally, as the vortices of each system are unstable, they roll up into a pair of vortex tubes and pass downstream, one on either side of the fuselage, at a distance apart somewhat less than the span of the airfoil.

Now several points of interest arise from a consideration of the foregoing :

(2) Firstly, there is the effect on the plan form of the airfoil—certainly it should not be rectangular in shape. As the strength of the tip vortex depends on the chord, the shorter this is, the smaller is the drag, and the degree of turbulence which will originate at the tip.

(As a logical deduction from this it follows that in an airfoil which tapered to a *point*, the strength of the tip vortex would be at a minimum; but on account of considerations noted later, this is not feasible in practice).

However, the indication definitely is for the airfoil to be tapered to a reasonable degree, such that a good aerodynamic balance is preserved. If the chord is constant the " downwash " from the tips is relatively large, and the incoming airflow to the leading edge is also given an induced " downwash," thus reducing the effective angle of incidence of the tip; resulting in the *centre* portion of the airfoil stalling first.

If the airfoil is tapered, the " downwash " is reduced, due to the shorter tip chord, and there is less reduction in the effective angle of incidence so that, if the taper is correct, the airfoil will stall all along its span at the same moment. If the degree of taper is excessive, the " downwash " at the tips becomes less even than that normally present at the centre of the airfoil, and the tips stall *first*—since they are now at a relatively larger angle of incidence than the centre section of the airfoil.

Whilst, therefore, a constant chord may tend towards better lateral stability, it does so only at the expense of an increase in drag due to the large tip vortices it originates.

Further, since the greater portion of the lift of an airfoil is generated about its centre section, large surfaces at the wing tips should be avoided; and the careful introduction of a certain degree of taper will have the effect of transferring useful lifting surface from where it is least effective, to a position where it is most effective.

The tip chord may be made equal to from $\frac{7}{8}$ to $\frac{3}{4}$ of the centre section chord, or even $\frac{5}{8}$, provided, in the case of a low-wing monoplane, a fair degree of dihedral angle is used.

In all cases the taper should mainly be effected by bringing forward the trailing-edge, and keeping the leading-edge at right-angles to the fuselage. If, for the sake of appearance, it is desired to set back the leading-edge, the amount of this " set-back " should not be more than 25 per cent of the total chord reduction. For example, if the chord is to be reduced from 10 inches at the centre of the span to 6 inches at the tip, the leading-edge should not be set back more than 1 inch; and the trailing-edge be brought forward by the other 3 inches.

(3) Secondly, there is the effect on the drag of the aircraft as a whole, consequent on the position of the fuselage in relation to the airfoil.

Considering, for a moment, the horizontal " streamlines " of the airflow round the front of a fuselage, it will be appreciated that there is a parting of the air, to either side, as it were; resulting in a local increase of pressure along each side of the fuselage.

Now if the fuselage is mounted *above* the airfoil, these " streamlines " from the nose of the fuselage will be meeting those passing over the top of the airfoil, and which, as was previously pointed out, are converging inwards.

Thus there is a region of high pressure along the upper side of the fuselage, which will persist to the tail of the aircraft. Meanwhile, since the streamlines beneath the airfoil are tending to diverge, there is a tendency to create a region of low pressure along the underside of the fuselage, tending to pull it downwards.

For this reason, the fuselage of a low-wing monoplane should be " egg-shaped "—wide at the bottom, and tending to be pointed at the top. The rudder dimensions (for the moment, taking no account of its area) should be such that it is rather high, and of not too large a chord, neither should it be highly tapered. Thus a useful operating surface is positioned *above* and away from the region of turbulence running along the upper sides and top of the fuselage.

When the aircraft is of the high-wing type the position is more or less reversed.

The streamlines parting from either side of the fuselage tend to follow a path approximately parallel to the diverging (lower) airfoil streams, and whilst the *strength* of the resultant pair of opposing rotating vortex tubes may be somewhat greater than in the case of the low-wing type of aircraft, the degree of turbulence will be less.

The converging upper " streamlines " will follow an uninterrupted path; and except that in meeting over the centre of the fuselage they tend to create a narrow stream of increased pressure, will not cause any great degree of turbulence.

The formation of a downward slope to the top of the fuselage, from the trailing-edge of the airfoil, will tend to keep this increased pressure at a minimum.

It will readily be appreciated that it is over its upper surface that the air finds the greatest difficulty in following the contour of an airfoil section, and consequently, when the fuselage is laid on the top, there can easily occur at its junction with the top surface of the airfoil a break-away of the air-stream, and the creation of large trailing vortices. In effect, the trailing-edge of that portion of the airfoil adjacent to either side of the fuselage becomes stalled.

Despite this, it is, of course, noteworthy that in recent years the low-wing monoplane has been developed to a very large extent; and this has only been made possible by very careful attention to streamlining, and " flaring " the upper surface of the airfoil into the sides of the fuselage, which, to a very large extent, prevents the formation of trailing vortices and " break-away."

(4) In full-sized practice, and from the point of view of manufacturing costs, the low-wing monoplane is at an advantage in certain respects. It is cheaper, and easier, to build a landing chassis which retracts into a low wing instead of into a high wing; whilst from the point of view of landing, the proximity of the ground to the low wing promotes a certain beneficial " cushioning " effect, resulting in a slightly shorter " pull up." Whether this benefit is obtainable with model aircraft, and at speeds in the neighbourhood of 15 to 20 m.p.h. is somewhat doubtful.

The general question, as to whether the high- or low-wing type of aircraft is the more efficient, is seemingly one which is much debated amongst model aircraft designers.

As regards efficiency, however, there is very little to choose between either type, *provided* each has been properly designed for the performance it is expected to give.

A fuselage designed for use in conjunction with a low wing will not be so efficient if used with a high wing, and vice versa —and each type of aircraft should always be considered entirely on its own merits, with the realisation that the problems of design require different angles of approach, and those of flying a different technique.

(5) The positioning of one wing above another, as in a biplane, is simply a means of arranging the required amount of lifting surface in the most convenient position in relation to the fuselage.

This arrangement, however, will not produce the same lift for a given wing area as that of a monoplane of a similar wing area—and it may be generally understood that, in so far as full-sized aircraft are concerned, the type is mainly used where structural or storage conditions set a limit to the span.

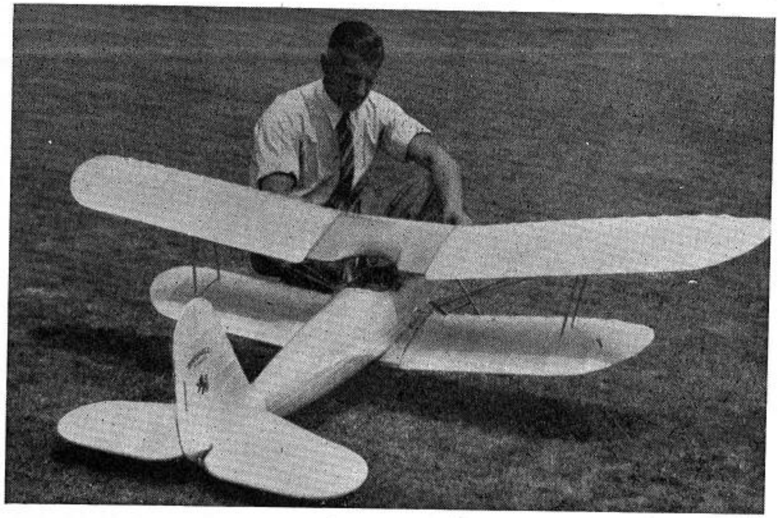
In model work, no such limits need apply, and the biplane design, in fact, possesses certain structural advantages which may be said to balance the disadvantages of its somewhat lower " lift factor."

A biplane of a given wing area will produce between ·7 and ·9 of the lift that would be produced from a monoplane of equal wing area; the actual figure depending mainly on the gap chord ratio; and, to a lesser extent, on the degree of stagger of the two wings.

H. Glauert* gives the undernoted " correction factors " for an *un*staggered biplane compared with a monoplane of the same wing area inclined at the same angle of incidence.

Gap/Chord Ratio.	Correction Factor.
·5	·730
·75	·800
1·0	·855
1·25	·895
1·50	·920

* H. Glauert, *Aerofoil and Airscrew Theory.*



Here is one of the finest power-driven model aircraft that has been built in England. It is a 7 ft. span biplane, with fuselage of monocoque construction, and is powered with a 9 cc. engine. The designer and builder, Mr. Sharvell, is about to start up the engine.

Thus is immediately seen the disadvantage of a small gap. To a limited extent this may be offset by giving the wings an exaggerated " stagger," but this calls for somewhat more complicated inter-wing struts, and does not allow of true scale reproduction; a certain small amount of " stagger " does, however, improve the longitudinal stability; and this may be regulated by moving the top wing ahead of the bottom wing by an amount equal to about one-sixth of the chord.

(6) The advantage accruing from the biplane design lies in the fact that, due to the bracing of the inter-wing struts, a much lighter wing structure may be used; so that whilst the lift produced, compared with a monoplane of the same size, is a little less, the weight of the wings is considerably less; resulting in the wing loading being kept at the same value for both types of aircraft.

In a normal type of biplane, where both wings have the same span and chord, the top wing appears to produce rather more than half the total lift, and for this reason special care should be taken to see that its surfaces should be kept as free as possible from disturbances, and the formation of vortices.

From a consideration of the previous explanation of the direction of the streamlines above and below the wings, it might be thought that the best position for the fuselage was close up to the under surface of the top wing, and raised clear of the upper surface of the lower wing, and to a certain extent this is correct. In full-size practice this design is not often seen, since the arrangement does not allow of sufficient depth to the fuselage, unless a very large gap is used (although, on occasions, it has been used in large military aircraft of the " night bomber " type).

In recent years, such great advances have been made in the " art " of streamlining that it is possible to arrange the fuselage so as to fully occupy the gap between the two wings, and still obtain a low drag figure, and comparative freedom from the formation of " break-away " or objectionable vortices.

(7) In flying models, other than those which are built to exact scale, the gap/chord ratio should not be less than 1·25 to 1, when the " lift factor " will be approximately ·9.

The fuselage should be kept as narrow as possible, and carefully " flared " into the wings where it joins them.

The disposition of the stabiliser in relation to the two wings of a biplane calls for special consideration—it has already been shown how the stream of vortices leaving the trailing-edge of a wing gradually converge, and form into two oppositely rotating vortices, passing down either side of the fuselage—and in the case of the biplane *two* such systems exist, one for each wing. Thus there are two vortices on each side of the fuselage; the pair on one side rotating in the *same* direction, but oppositely to the pair on the other side of the fuselage. Each pair of vortices eventually combines, the upper one dropping rather more than half-way, due to the " down wash," to meet the lower one.

Unless the design of the whole aircraft is essentially a very stable one, it is considered undesirable to place the stabiliser directly in the path of these trailing vortices, and since their path lies nearer to the bottom wing, it should be placed so as to " sit " on the top of the rear of the fuselage.

The secret of a successful model biplane design lies in the correct positioning of the thrust-line in relation to the centre of drag of the whole aircraft. The centre of resistance of each wing must be carefully ascertained, and a " mean "

for the two found. Then, if this position can be arranged to coincide with the centre of resistance of the fuselage, landing gear, etc., and also the thrust-line as well, the aircraft will possess inherent stability.

But it is not often possible to arrange such a happy state of affairs.

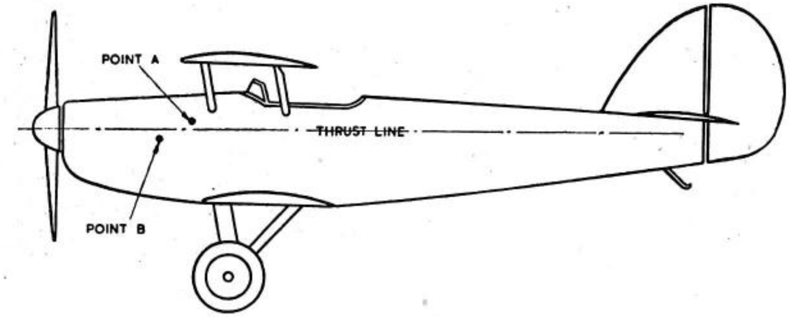


FIG. 6A.

Note that thrust line is arranged to pass *below* point A—the centre of resistance of the fuselage, etc., and *above* point B, the mean centre of resistance of the top and bottom main planes.

Fig. 6a shows the centre of resistance of the fuselage, etc., (*a*) above the centre of resistance of the mean of two wings, (*b*) which tends to pull the nose up—rotating about (*b*). The thrust-line must therefore be arranged to pass through a point between (*a*) and (*b*). Similarly Fig. 6b shows the centre of resistance of the fuselage *below* the centre of resistance of the wings, and the thrust-line must now pass *below* (*b*) to counteract the tendency of (*a*) to pull the nose down.

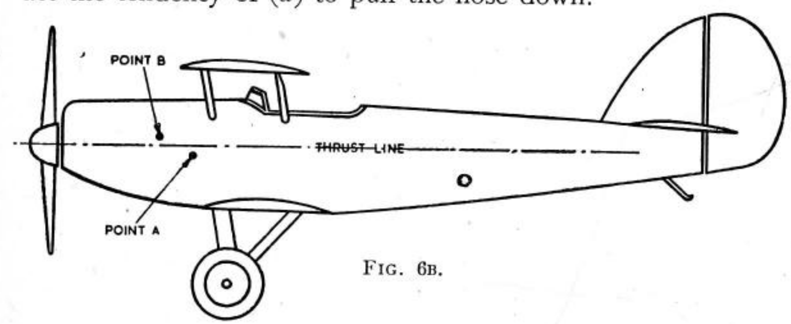


FIG. 6B.

Observe how the thrust line is still arranged to pass *between* points A and B. In this case point A is *below* the trust line and point B *above* it.

At the time the author built this low-wing monoplane of 10 ft. span it was one of the largest, and certainly the heaviest, of models to be built. The span of the 'plane is 10 ft., and the weight is 14 lb. The model is powered with an 18 cc. " Comet " engine, and it was awarded a " Highly Commended " diploma at the *Model Engineer* Exhibition in London, 1935. The 'plane was also awarded first prize at The Concours d'Elegance at the Northern Rally, organised by the Lancs M.A.S. in Manchester in 1936.

The actual position of the thrust-line will, of course, depend on the relative values of (*a*) and (*b*), and their distance apart—and must generally be found by experiment.

On no account should the engine of a biplane be given " down " or " up " thrust to obtain the correct trim, and if the designer remains in doubt as to the accuracy of his calculations, he should provide for a small " up-and-down " movement of the engine-mounting to enable him to bring the thrust-line to the same level as the centre line of resistance.

CHAPTER III

DRAG

Parasite and induced drag—Values of K for fuselage drag —Circular and rectangular section fuselages—Engine positions—Values of K for: tail-planes, rudders, landing chassis, struts, wheels, engines and—flat plates—Advantages of circular or elliptical section fuselages—Calculations for parasite drag of various parts of an aircraft.

(1) The drag of an aircraft may be divided into two parts —wing drag and parasite drag.

Wing drag consists of two kinds: " Profile," which is dependent only on the particular wing section used; and " Induced," which varies with the lift and aspect ratio.

As the aspect ratio *decreases*, so the " downwash " *increases*, thus reducing the effective angle of attack. Conversely, as the aspect ratio increases, so the " downwash " decreases, and the wing works at a better angle—another point in favour of as high an aspect ratio as is possible.

Parasite drag is also of two kinds: that which varies with the angle of attack, and that which does not. Instances of the former kind are square section fuselages, wing sections, and tail surfaces; whilst those of the second kind include fuselages and components having, in general, *good " Streamline " shapes*.

The drag of a fuselage of circular or elliptical section does not vary very much with a change of angle of inclination; but with rectangular section fuselages, any appreciable variation in the angle results in a considerable increase in drag. For instance, at an angle of inclination of 10 degrees, the increase in drag will be about 40 per cent in the case of a rectangular section fuselage, and only about 5 per cent in the case of a circular section fuselage.

(2) The drag of a fuselage may be calculated from the formula

$$D = KAV^2 \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (10)$$

where K = the drag coefficient of the fuselage, and depends on its particular characteristics.

A = the projected cross-sectional area, in square feet, at the largest section.
V = the speed—in miles per hour.
and D = is given in pounds.

K varies from about ·0002 to about ·0009, and averages about ·0004 for totally-enclosed fuselages of approximately circular cross-section.

For rectagular section fuselages of the type shown in Fig. 7—a type in fairly common use owing to its being easily and quickly constructed—

K may be taken as approximately ·0009.

In Fig. 8 is shown a fuselage of the same overall dimensions, and the same cross-sectional area, but which has been streamlined, and provided with a slightly rounded nose, resulting in the drag being reduced by half.

K = ·00046.

AVERAGE VALUE FOR K = ·0009 FIG 7

AVERAGE VALUE FOR K = ·00046 FIG 8

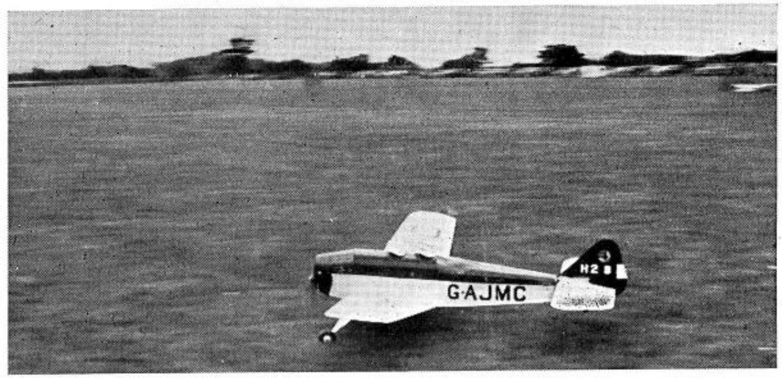
AVERAGE VALUE FOR K = ·00025 FIG 9

Fuselages of the shape shown in Fig. 9, where the section throughout is nearly circular and which have a fairly fine taper to the tail, have the lowest drag.

K = approximately ·00025.

As an example of the great reduction which may be obtained by proper care and attention to streamlining, the drag of the fuselages shown in Figs. 7 and 9 may be calculated for a speed of 15 m.p.h. The cross-sectional area being taken as 48 square inches in each case.

Substituting in the formula $D = KAV^2$
the drag of fuselage, Fig. $7 = \cdot0009 \times \cdot33 \times 15^2 = \cdot0668$ pound
and the drag of fuselage, Fig. $9 = \cdot00025 \times \cdot33 \times 15^2$
$= \cdot0186$ pound.



In this photo is shown the winner of the 1939 Bowden Trophy taking-off on one of its flights. The 'plane was built by Mr. T. M. Coxall, who is shown holding the model in another photo on page 92.

Considering now the effect of a variation in the angle of inclination of 10 degrees for these two fuselages—the drag of fuselage, Fig. 7, is increased (by 40 per cent) to ·0935 pound, and that of fuselage, Fig. 9, is increased (by 5 per cent) to ·01953 pound.

Thus it is seen that, under conditions which may not only occur in flight, but *do* occur when the aircraft is taking off, and until the tail has lifted, the drag of the rectangular section fuselage is nearly five times that of the fuselage of circular cross-section. Surely a strong enough argument in favour of the latter type, in spite of the added time required for its construction?

(3) The thickest part of a fuselage should be at a point about one-quarter to one-third of the overall length distant from

This shoulder-wing petrol 'plane, designed and built by the Author, is of 7 ft. span, weighs 6 lb., and is powered by a 6 cc. "Baby Cyclone" engine. It is of fairly typical design to the aircraft described in Figs. 10 and 11.

the nose, and should be as nearly circular in section as is possible. That portion from the nose to this thickest part should be kept free from obstructions such as control knobs, "domed" inspection doors, etc., since it is over this portion of the surface of the fuselage that the air pressure is greatest.

On rubber-driven models, dummy motors of the radial type, if of overall diameter exceeding that of the nose of the fuselage, can easily double the drag; and in power-driven aircraft the disturbance caused by a cylinder projecting above the top of the fuselage can cause a noticeable increase in drag.

If the design of the aircraft allows of the thrust-line being placed below the centre line of the fuselage (as in a high-wing monoplane with a long and heavy landing chassis), the engine may be fitted in the "upright" position. But if the thrust-line requires to be above the fuselage centre line, then the engine should be inverted, so as to avoid any part of it extending beyond the fuselage.

The coefficients for K which are given apply to fuselages which are totally enclosed, and in which there are no large cracks between detachable panels, doors, etc.

The increase in drag due to failure to appreciate the importance of this last point is considerable—particularly if there should be two openings, one at each end of the fuselage which

would allow the resultant flow of air to set up all kinds of disturbances at the point of exit.

When flying scale models of the type with open or semi-open cockpits are under consideration, the value of K should be increased by from 50 to 100 per cent, according to the "openness" of the cockpit, or the degree of turbulence it is considered likely to be set up behind such erections as wind-screens, machine-gun "cupolas," etc.

(4) An average value for K for tail-planes which have a flat under-surface and a thickness-to-chord ratio of about 1 : 18 may be taken as ·000075. But if the thickness-to-chord ratio is greater, or the section is cambered, then the tail-plane will be of the "lifting" type, and the drag must be calculated from the formula $D = C_d \frac{p}{2} SV^2$.

For fins and rudders the value of K may be taken as approximately ·00006 per square foot of area.

The drag of a landing chassis is as much due to "interference" as to direct resistance offered to the flow of air. This "interference" consists of vortices and cross-currents caused by the streamlines from one strut encountering another strut before they have had time to reform into a uniform air-stream. "Interference" is also caused at the junction of the struts with axle-plates, fuselages, etc.

An average value for K for struts is ·00025—the drag being in pounds per square foot of projected area: but thus must be increased by from 50 to 100 per cent, according to the amount of " interference " which may be thought to exist.

The drag of wheels varies with the ratio of diameter to tyre width, and is also dependent on the degree of " fairing " between the tyre and the hub; also the value of K is relatively greater for small diameter wheels—say 2 to 4 inches—than it is for those of from 6 to 9 inches diameter.

For wheels of from 2 to 4 inches diameter—with a diameter/width ratio of about 2·5 to 1, the value for K is about ·0029, whilst for wheels of from 6 to 9 inches diameter, with a diameter/width ratio of about 4 to 1, the value of K drops to about ·0015.

The drag of an engine installation will depend to a large extent on the type of mountings to which it is affixed. Aluminium " cones " which enclose the petrol tank, and are of tapered form, have low drag values, whereas the type of mounting which consists of two brackets extending from the front of the fuselage to form a platform on which the engine " sits," will create a certain amount of interference.

For average conditions the drag of a single-cylinder engine may be taken as being equal to $\cdot 0006\ V^2$ pound, where V is in m.p.h.

Certain parts of some flying model aircraft—particularly those of the rectangular section fuselage type, may present flat surfaces at right-angles to the direction of the airflow; and, in pointing out that in such cases the value of K is ·003, emphasis is given to the great reduction in drag which is made possible due to good " streamlining," since it has already been shown that the drag of a well-designed fuselage is about ·0004, which is approximately one-eighth of that of a rectangular section and " flat-nosed " fuselage of the same projected cross-section area.

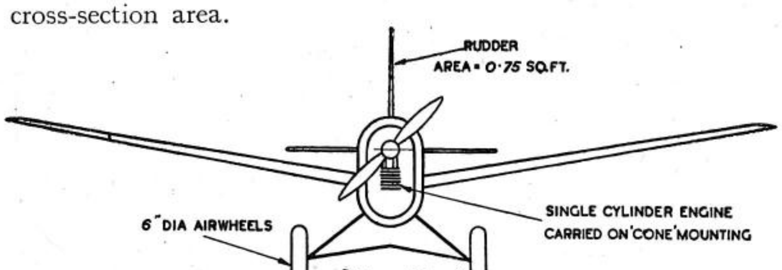


FIG. 10.

(5) As an example of how the total parasite drag of an aircraft is arrived at, calculations, for the drag at 15 m.p.h., may be made for a typical power-driven low-wing monoplane, as shown in Figs. 10 and 11, and built to the undernoted specification and dimensions.

(*a*) Fuselage—flat sides, tapering to a point at the tail. Top and bottom of fuselage rounded with a radius equal to half its width. Projected area at largest cross-section

= 1 rectangle 4 inches × 3 inches.
= 12 square inches.
and 2 half circles = 1 circle 3 inches diameter.
= 7·1 square inches.
Total = 19·1 square inches.
= ·132 square foot.

(*b*) Stabiliser—section thin, flat under-surface, area
= 2·25 square feet.

(*c*) Rudder area = ·75 square foot.

(*d*) Landing chassis, built from streamline section 1·5 inches × ·5 inches. Total length 5 feet 6 inches, projected area
= 5·5 × ·042 square feet.
= ·23 square foot.

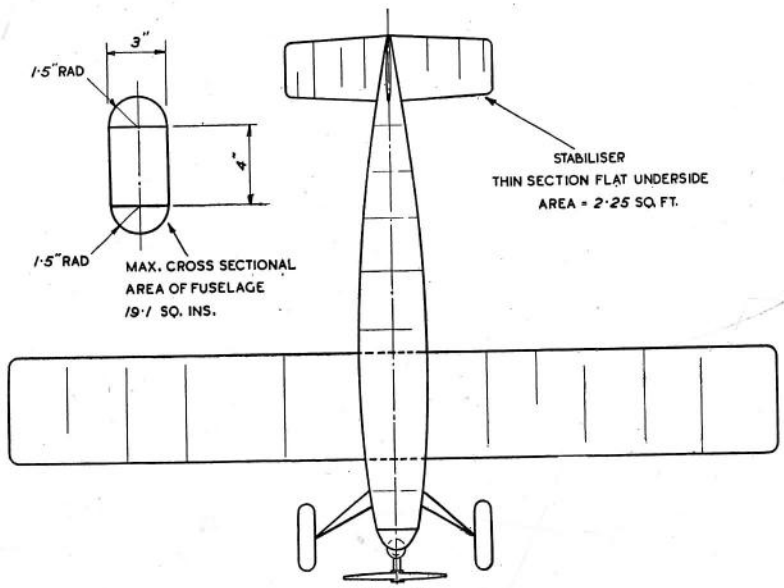


FIG. 11.

(*e*) Two wheels 6 inches diameter × 1·5 inches wide, total projected area = 2 × 6 × 1·5 inches.
= ·125 square foot.

(*f*) Single-cylinder engine—height about 5 in., width about 2 in.. mounted on tapered metal " cone " 1 in. of cylinder extending beyond the fuselage.

(6) Proceeding to estimate:

(*a*) The fuselage is a " cross " between the circular and " square section " type—the taper to the tail is good, but the nose is somewhat " blunt." The value for K would be arrived at by averaging the values of ·0002 for a perfect streamline, and ·0009 for a "square section," giving ·00055, which in view of the blunt nose might be fairly increased to ·0007. The drag of the fuselage is therefore $\cdot 0007 \times \cdot 132 \times 15^2$
= ·0208 pound.

(*b*) The value of K for the stabiliser is taken as ·00007; the drag is therefore $\cdot 00007 \times 2{\cdot}25 \times 15^2$
= ·0355 pound.

(*c*) The value of K for the rudder is taken as ·00006; the drag is therefore $\cdot 00006 \times \cdot 75 \times 15^2$
= ·0101 pound.

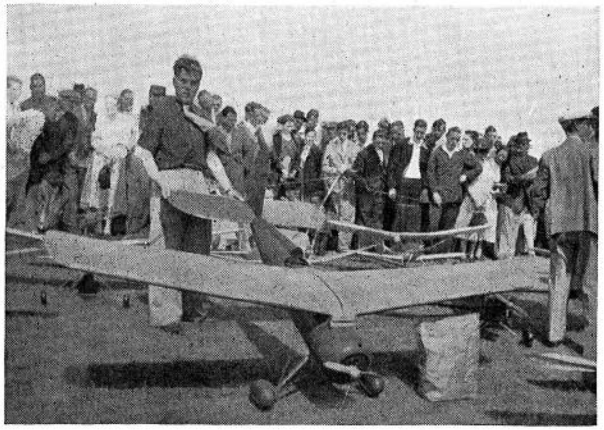
(*d*) The projected area of the chassis struts is ·23 sq. ft. There are four points of attachment to the fuselage, at which " interference " will occur, and at the two at the front there will also be turbulence due to the deflections of the airstream from the " blunt " nose. There will also be " interference " where the wheel axles join the lower ends of the struts, and where the horizontal "tie-bar" strut meets the two front struts. The total drag of such an arrangement will probably be doubled. Assuming, therefore, a projected area of twice ·23 = ·46 square feet, and taking the value of K as ·00025 the chassis drag $= \cdot 46 \times \cdot 00025 \times 15^2$
= ·0259 pound.

(*e*) The total projected area of the two wheels is ·125 sq. ft.—the hubs do not project, so the drag
$= \cdot 125 \times \cdot 0015 \times 15^2$
= ·0422 pound.

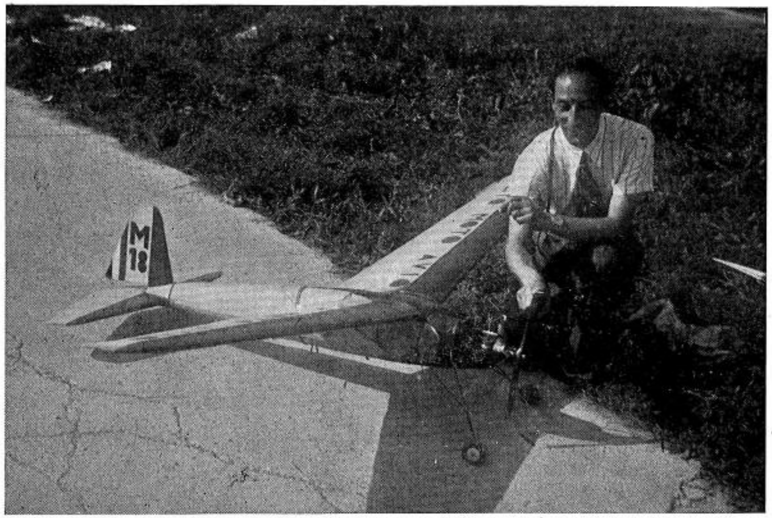
(*f*) The drag of the engine will be
$(\cdot 0006\ V^2) = \cdot 0006 \times 225$
= ·135 pound.

The total parasite drag of the aircraft is therefore found to amount to the sum of ·0208 + ·0355 + ·0101 + ·0259 + ·0422 + ·135 = ·2695 pound.

To which must be added the drag of the wings, which may be calculated from Formula (7).



Mr. " Bunny " Ross, with a semi-scale petrol 'plane of his own design and construction.



Here is another Italian petrol 'plane, built by Signor Clerici, of Milan.

CHAPTER IV

CONTROL SURFACES

Lift of tail-planes with controlled elevators—Stabilisers—Relation of aspect ratios to tail-plane areas—Formula for stabiliser areas—Calculations for correct disposition of main wing and tail-plane—Rear fins—Rudder areas—Formulas for fin areas for power and rubber-driven aircraft.

THE control surfaces of a flying model aircraft consist of horizontal and vertical airfoils—the former being either of the " lifting " or " non-lifting " (stabiliser) type; and the latter consisting of a vertical fin, part of which may or may not be hinged to form a rudder.

Unless the model aircraft is equipped with some form of automatic stabilising device such as a gyroscope or pendulum connected to elevators, the tail-plane should be of the " non-lifting " type, i.e. its axis should lie parallel to the thrust-line, and its function should be purely that of a stabiliser.

If it is intended that the tail-plane of an aircraft shall provide lift, as well as function as a control, it may be of any well-known airfoil section provided that it is not of exaggerated form. Clark Y and R.A.F. 25 are suitable sections.

To have the tail-plane forming part of the total lifting surface of the aircraft is, of course, a very useful feature, but one not to be introduced without certain reservations.

The " lifting " tail-plane is, in effect, a " mixed blessing " and sometimes a very definite handicap, due to the fact that as its lift will increase with the speed of the aircraft, the greater this is, the greater will be the tendency for the tail to rise; thus the degree of longitudinal control varies with the speed of the aircraft—not a good feature, but one which, to a large extent, can be balanced by providing automatically-controlled elevators.

If controlled elevators are fitted, their area should be about 40 per cent of the total tail-plane area, and their movement comparatively small; i.e. a large area with a small up-and-down movement, rather than a small area with a large movement.

If the tail-plane is arranged as a stabiliser it does no work as a lifting agent, but serves solely to keep the aircraft flying in a horizontal position; but since its axis lies parallel to the thrust-line, it only commences to control the flight of the aircraft *after* a diversion has been made from the horizontal; then, if the nose of the aircraft drops, the stabilizer is tilted at a negative angle, and the air pressure on the top surface forces it downwards and so brings the aircraft back to an even keel. Similarly, if the nose rises, the air pressure then acts on the underside of the stabiliser, and forces the tail of the aircraft upwards, and once again the balance is restored. Thus it is seen that, in actual theory, the aircraft must first lose its fore-and-aft balance before it can regain it—and because of this, the stabilisers of flying model aircraft require to be comparatively large, so that they are very sensitive to changes in direction of flight, and exercise a degree of control which, in practice, may be considered as instantaneous.

The determination of the most suitable area for the tail-plane depends on a number of factors, all influencing each other, and chief of which is the ratio of the distance from the centre-of-lift of the main wing to the centre-of-chord of the tail-plane, compared with the span of the main wing. The greater this distance, the greater is the leverage exerted by the tail-plane, and the smaller may be its area.

The higher the aspect ratio of the main wing, the smaller need be the area of the stabiliser. Thus a stabiliser which is suitable for an aircraft with a main-wing aspect ratio of 7 : 1 would require to be increased by about 25 per cent for an aspect ratio of 5 : 1.

The aspect ratio of the stabiliser *itself* has an influence on its size relative to that of the main wing; and a stabiliser with an aspect ratio of 5 : 1, which was suitable for a certain-sized aircraft, would require to be increased in area by about 12 per cent if its aspect ratio were reduced to 3 : 1.

As a general rule the stabiliser area should be about 38 per cent of the main wing area for a rubber-driven aircraft, and about 33 per cent for power-driven machines.

The distance from the centre-of-lift of the main wing to the centre-of-chord of the stabiliser is known as the " moment arm," and should be taken as equal to ·6 of the overall length

in the case of rubber-driven, and ·65 in the case of power-driven aircraft. Being a percentage of the overall length, it will naturally vary with it in relation to the main wing span, and, as already has been pointed out, will have a considerable influence on the area of the stabiliser according to whatever the span to overall length ratio is. But, in relation to the overall length *itself*, the ratios of 60 per cent and 65 per cent should never be departed from by more than 4 or 5 per cent.

An empirical formula for calculating the area of a stabiliser, which takes account of all desiderata mentioned above is

$$Sa = \cdot 257W \times \frac{15}{AR + 6\cdot 4} \times \frac{3\cdot 9}{ar + \cdot 05} \times \frac{\cdot 65}{M} \times \sqrt{\frac{S}{L}} \qquad (11)$$

Where W = Main wing area in square inches.
AR = Main wing aspect ratio.
ar = Stabiliser aspect ratio.
M = Moment arm divided by overall length.
S = Main wing span in inches.
L = Overall length of aircraft, in inches.
and Sa = Required area of stabiliser in square inches.

For example:

Consider the characteristics of an aircraft of which the main wing area is to be 1,250 square inches, and of an aspect ratio of 9 : 1.

The stabiliser aspect ratio is to be 3·1 : 1; the ratio of overall length to moment arm is to be ·62; and the ratio of span divided by overall length is to be 1·7. Calculations may be made as follows:

(1) By substitution in formula (11)

$$Sa = \cdot 257 \times 1250 \times \frac{15}{9 + 6\cdot 4} \times \frac{3\cdot 9}{3\cdot 1 + \cdot 05} \times \frac{\cdot 65}{\cdot 62} \times \sqrt{1\cdot 7}$$

$$= 321 \times \cdot 975 \times 1\cdot 24 \times 1\cdot 05 \times 1\cdot 3$$

= 528 square inches.

(2) Assuming that the main wing is of rectangular plan form, its chord may be calculated from the formula

$$C = \sqrt{\frac{W}{AR}} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (12)$$

Where AR = Aspect ratio
and W = Wing area in square inches.

Thus $C = \sqrt{\frac{1250}{9}}$

$= 11 \cdot 8$ inches.

and the span = 106 inches.

(3) As the aspect ratio of the stabiliser is given as 3·1, and its area has already been calculated to be 528 square inches, then, assuming it to be of rectangular plan form, its chord

$= \sqrt{\frac{528}{3 \cdot 1}}$

= 13·1 inches.

and its span = 40·4 inches.

(4) As the ratio of span divided by overall length is given as 1·7, the overall length is calculated to be

$\frac{106}{1 \cdot 7} = 62 \cdot 5$ inches.

Thus it is seen that the fuselage length is only a little more than half the span, and this accounts for the fairly large stabiliser surface of 528 square inches compared with the main wing area of 1,250 square inches.

(5) The moment arm ratio is given as ·62, and since the overall length has been calculated to be 62·5 inches, the moment arm distance is calculated to be $62 \cdot 5 \times \cdot 62 = 38 \cdot 8$ inches.

(6) The chord of the stabiliser has been calculated to be 13·1 inches, the rearmost location of the moment arm is therefore one-half of the chord distant from the tail of the aircraft, i.e. $13 \cdot 1 \times \cdot 5$

$= 6 \cdot 55$ inches.

(7) As the moment arm has been calculated to be 38·8 inches, its foremost location is at a point $(38 \cdot 8 + 6 \cdot 55) = 45 \cdot 35$ inches distant from the tail, and thus is found the *centre of gravity of the whole aircraft*, as also the centre of lift of the main wing.

(8) Since the main wing chord has been calculated to be 11·8 inches, its leading edge will be one-third of this distance ahead of the centre of gravity, i.e. $(45 \cdot 35 + 3 \cdot 93)$ inches $= 49 \cdot 28$ inches from the tail; or $(62 \cdot 5 - 49 \cdot 28) = 13 \cdot 22$ inches back from the *nose* of the aircraft.

If, when an aircraft is completed, it should be found that the centre of gravity does not coincide with the centre of lift

This 8 ft. span high-wing petrol 'plane, designed and built by the author, made many successful flights in 1939.

of the main wing, a redistribution of weight must be made to effect the desired balance. This may be done, in the case of a rubber-driven machine, by swinging the landing chassis backwards or forwards as may be necessary; and in the case of power-driven aircraft, by an alteration in the position of the battery or coil.

The main wing should *not* be moved, as doing this would alter the length of the "moment arm," which is one of the factors controlling the area of the stabiliser.

Stabilisers should be of uniform section, i.e. without camber, and fairly thin, the maximum thickness-to-chord ratio not exceeding 8 or 9 per cent. They should be mainly of rectangular plan form with rounded tips, and should be without dihedral.

The primary function of the rear fin is to effect directional control, but in a flying model aircraft it has also to provide lateral control in the event of a side slip; and not only its area, but also its *shape*, is of fundamental importance.

Considering the fin as an element designed purely for the obtaining of directional control, it should consist of a plane surface, erected vertically at the tail end of the machine, with its axis parallel to the centre line of the fuselage; and be of such an area that, in conjunction with the leverage obtainable according to the length of its "moment arm," it will keep the nose of the aircraft pointing into the wind.

The area of the fin is also partly dependent on the shape of the forward portion of the fuselage and type of landing chassis used. It is obvious that, in any machine of the tractor type, the centre of drag is behind the airscrew, and thus there is a natural tendency for the aircraft to be "self steering," due to the "castoring" effect introduced.

In the case of an aircraft of which the fuselage is thin at the front, and thick and unstreamlined at the rear, the centre of drag would be at such a distance behind the centre of gravity that this "castoring" effect would be quite pronounced, and only a small fin would be required. But in the vast majority of cases, the reverse is the case, and on account of the drag of the engine and a fairly blunt (though streamlined) nose, the drag of the main wings, and also the landing chassis, the centre of drag is not so very far behind the airscrew, consequently the "castoring" effect is not very pronounced, and thus a fin of fairly large size is introduced to create the necessary degree of drag, or side resistance, as soon as the tail of the aircraft swings or "yaws."

But it is in regard to its area and shape, when being considered as a *vertical stabiliser*, that most careful thought must be given to the design of the fin, so as to ensure that an equal proportion of side area of the whole aircraft is presented in front of, as well as behind, the centre of gravity. The area of the fin should bear a relation to the length of the "moment arm" (in this case the distance from the centre of the chord of the main wing to the centre of the chord of the fin), the main wing span, and the total weight, and an average value, for power-driven aircraft, may be calculated from the empirical formula

$$A = K \times \sqrt{\frac{WS}{M}} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (13)$$

Where W = Weight of aircraft in pounds.
S = Span of aircraft in feet.
M = Moment arm in feet.
and A = is given in square feet.
and K = ·25 for high-wing monoplanes.
= ·30 for mid-wing monoplanes.
and = ·35 for low-wing monoplanes and biplanes.

The several values for K are necessary because the lower

the wing is, in relation to the fuselage, the greater is the dihedral required; and the larger must be the fin to equal the increased projected area of the main wing.

This projected area may easily be ascertained by drawing out to scale a side elevation of the fuselage and projecting the main wing; if this is done on "squared" paper, it is then an easy matter to "count up" the area, and sketch in a fin of equal size.

In all cases this graphical method should be used as a check on the figure obtained from the formula, and in the event of a difference being found, the *larger* figure should be used.

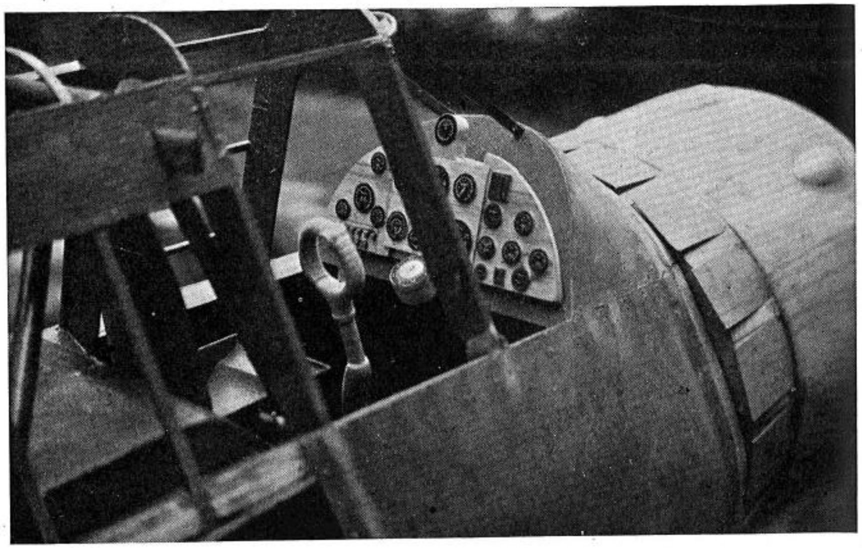
In the case of rubber-driven model aircraft the fin area should be equal to about 10 per cent of the main wing area—but if it is desired to take into consideration all the factors affecting the fin, and to obtain an exact result, use may be made of a formula by C. H. Grant, which states that

$$AF = 0{\cdot}1 \frac{A}{(M)} (3 + N + 0{\cdot}58 \sqrt{ST}) \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (14)$$

Where A = Main wing area.
M = Moment arm.
N = The distance from the centre of gravity to the airscrew bearing face.
S = The wing span.
T = The tip rise of the main wing—i.e. the distance the tip is above the centre section of the wing.
and AF = The required fin area.
(All the values being either in inches or square inches).

The exact shape of the fin is not of very great importance, provided the height is approximately equal to the chord.

The fin should be double-surfaced, of symmetrical section, and with a maximum thickness to chord ratio of not more than 1 : 20 in the case of rubber-driven models, and not more than 1 : 10 in the case of power-driven aircraft.



Aircraft performance of full-size aircraft is recorded by an elaborate set of instruments. Here is the set of instruments in front of the pilot in the one-fifth full size flying scale model of the " Lysander " built by the Author. The joy stick may be noted, and in front of it the compass.

CHAPTER V

AIRCRAFT PERFORMANCE

The value of performance calculations—Formula for (*a*) " Stalling speed," (*b*) H.P. required for steady horizontal flight—Minimum value of VD—A series of calculations for ascertaining the performance of aircraft—Formulæ for (*a*) Rate of climb (*b*) Tractive resistance, (*c*) H.P. required to overcome tractive resistance, (*d*) Tractive effort available for acceleration —(*e*) Distance travelled during take-off.

In so far as light rubber-driven aircraft are concerned, the obtaining of increased duration of flight is ever the aim of the aero-modellist, and since this type of machine has a comparatively slow flying speed, it may with perfect safety be launched innumerable times in the course of its trials without suffering damage. Thus the process of arriving at its best performance is essentially the practical one of " trial and error "—the ultimate object being the obtaining of as high a power/weight ratio as is possible.

In the case of large multi-spindle rubber-driven machines, and more particularly in the case of power-driven aircraft, the " trial and error " method generally results in so many " errors " that the " trials " of the aero-modeller are increased enormously ! Especially are increased the number of hours spent in the workshop on *repairs!* The cost of these large machines often amounts to several pounds, whilst the time spent in their construction extends to several months of the aero-modeller's spare time.

The investigation, therefore, by a series of calculations of the probable performance of the proposed aircraft, *before the design leaves the drawing board*, will do much to prevent disappointment arising from a performance which does not come up to expectation.

In addition, the calculations may often disclose that a small alteration in the design will make all the difference between the performance being a success instead of a failure.

For instance, the use of the highest lift/drag ratio would seem to be an obvious rule to work to, and it *is*, for very light, slow-flying rubber-driven aircraft, but for wing loadings exceeding 8 ounces per square foot—requiring speeds of from 10 to 12 miles per hour and upwards—this rule does not hold good; as the highest lift-drag ratio always occurs at a fairly low C_l which naturally calls for a relatively high speed.

The adoption of a lift/drag ratio, somewhat lower than the highest available, will enable a considerably higher C_l to be used, *at a considerably lower speed;* and since the drag varies as the square of the speed, it follows conversely that if the speed can be reduced by half, the drag will be reduced to one-quarter of its previous value, obviously calling for a very much smaller power output.

The following series of calculations clearly demonstrate the valuable information which may be obtained from such an investigation.

The minimum flying, or "stalling speed," of an aircraft may be calculated from the formula—

$$V = 19{\cdot}77 \sqrt{\frac{W}{C_l \text{ Max.} \times S}} \quad \ldots \quad \ldots \quad (15)$$

Where W = Weight of aircraft, in pounds.
S = Main wing area, in square feet.
and V is given in miles per hour.

In Chapter III calculations were made to ascertain the parasite drag of a particular aircraft at a speed of 15 miles per hour.

Assuming this same aircraft to have

(1) A wing area of 7 square feet;
(2) A wing section R.A.F. 32;
(3) A total weight of 6 pounds; and
(4) Noting from the Aeronautical Research Committee's R. and M. No. 928, that

C_l max. of R.A.F. 32 = 1·308
at which C_d = ·1518

calculations may now be made to ascertain the minimum flying speed of this particular aircraft.

(1) Substituting in the formula (15);

$$V = 19{\cdot}77 \sqrt{\frac{6}{1{\cdot}308 \times 7}}$$

= 16 miles per hour,
or 23·5 feet per sec.

(2) Next, by using formula (7)—in which, be it noted, V is given in *feet per second*—calculations may be made to ascertain the drag of the wing of this particular aircraft.

$$D = {\cdot}1518 \times \frac{{\cdot}002378}{2} \times 7 \times (23{\cdot}5)^2$$

$$= {\cdot}698 \text{ pound.}$$

(3) The total parasite drag of this aircraft, at a speed of 15 miles per hour, has already been calculated to be ·2695 pound. This is equivalent to $\cdot 2695 \times \left(\frac{16}{15}\right)^2 = \cdot 307$ pound at 16 f.p.s., to which must be added the wing drag—making a total of 1·005, say 1 pound.

(4) The power required to maintain an aircraft in steady horizontal flight may be calculated from the formula—

$$\text{H.P.} = \frac{DV}{375} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (16)$$

where D = Total drag, in pounds.
and V = The speed, in miles per hour.

Continuing the example, and taking the value of V as 15·6 miles per hour for the whole aircraft.

$$\text{H.P.} = \frac{1 \times 16}{375}$$

$$= {\cdot}0427 \text{ (or roughly} \tfrac{1}{24} \text{ H.P.)}$$

Now the stalling speed, apart of course from being the minimum flying speed, is also the speed at which the lift/drag ratio is at its lowest value—in this case 6:1.

The highest lift/drag ratio of R.A.F. 32 is 18:1, and occurs when the airfoil is set at an angle of ·8 degrees—when C_l = ·49 and C_d = ·0272.

(5) Under these conditions the minimum flying speed is now found to be equal to

$$19{\cdot}77 \sqrt{\frac{6}{{\cdot}49 \times 7}}$$

= 26·2 miles per hour.
or 39·4 f.p.s.

(6) Since the C_d is now only ·0272, the wing drag is found to be equal to $·0272 \times \frac{·002378}{2} \times 7 \times (39·4)^2$

$$= ·35 \text{ pound.}$$

(7) The total parasite drag has been calculated to be ·2695 pound at 15 miles per hour—and since the drag increases as the square of the speed—at 26·2 miles per hour it will be equal to

$$·2695 \left(\frac{26·2}{15}\right)^2$$

$$= ·822 \text{ pound.}$$

Which gives a total drag of 1·172 pounds for the aircraft, flying at a speed of 26·2 miles per hour.

(8) Substituting in the formula $H.P. = \frac{DV}{375}$

The H.P. required to maintain steady horizontal flight is found to be $\frac{1·172 \times 26·2}{375}$

$$= ·082 \text{ (or roughly) } \frac{1}{12} \text{ H.P.)}$$

Examining the results so far obtained, it may be noted that when the wing is set at the angle of incidence which gives the highest lift/drag ratio, the power required to fly the aircraft is nearly double that required to maintain flight at the "stalling speed."

The difference in speeds between 16 miles per hour and 26·2 miles per hour may also be noted. Somewhere between these limits lies a speed which, multiplied by the drag, gives a minimum value for VD; when obviously the required H.P. will also be at a minimum.

It has already been pointed out that drag is of two kinds —parasite and induced, parasite drag being divided into that which varies with the angle of attack, and that which does not.

Now parasite drag varies directly as the square of the speed, but induced (wing) drag varies *inversely* as the square of the speed; and thus is explained the reason why, at some intermediate speed between the "stalling speed" and the minimum flying speed with maximum lift/drag ratio, the

drag is at its lowest value, lower than at either of the two limits given. As the speed increases above the "stalling speed," the induced (wing) drag at first decreases, and then increases rapidly, whilst the parasite drag increases all the time. It is at a speed approximately 15 per cent greater than the "stalling speed" that the drag is found to be at its lowest value; and thus, to obtain the best power/weight ratio the aircraft should be designed to fly at this speed.

This photo was taken on Sussex Downs in August, 1939, at a meeting organised by the Brighton Model Aircraft Club. The Author is on the extreme right, standing behind his high-wing 'plane.

In the case of the aircraft under consideration the designed flying speed would be equal to 16 + (15 per cent of 16) = 18·4 miles per hour, or 27 f.p.s.

To find the C_l of the wing at this speed, the formula $L = C_l \frac{\rho}{2} SV^2$ may be re-written

$$C_l = \frac{L}{\frac{\rho}{2} SV^2} \quad \dots \quad \dots \quad \dots \quad \dots \quad \dots \quad (17)$$

(9) Substituting the appropriate figures in the example,

$$C_l = \frac{6}{·001189 \times 7 \times 27^2}$$

$$= ·99$$

Reference to the coefficients for R.A.F. 32 shows that when $C_l = ·99$, $C_d = ·066$; and the angle of incidence of the wing = 5·9 degrees.

Repeating the previous steps—with

$$V = 18{\cdot}4 \text{ miles per hour}$$
$$= 27 \text{ feet per second}$$
and $C_d = {\cdot}066$

(10) The wing drag $= {\cdot}066 \times {\cdot}001189 \times 7 \times 27^2 = {\cdot}4$ pound.

(11) As the parasite drag at 15 miles per hour was ·2695 pound, at 18·4 miles per hour it will be

$$\cdot 2695 \left(\frac{18{\cdot}4}{15}\right)^2$$
$$= {\cdot}406 \text{ pound.}$$

which gives a total drag of ·806 pound.

(12) Substituting in the formula H.P. $= \frac{DV}{375}$

The H.P. required to maintain the aircraft in steady horizontal flight at the speed of 18·4 miles per hour.

$$= \frac{{\cdot}806 \times 18{\cdot}4}{375}$$
$$= {\cdot}0396 \text{ as against } {\cdot}0427 \text{ for the stalling speed.}$$

Now the H.P. of ·0396 is only sufficient to maintain steady horizontal flight, and allows no margin for climb; and a further calculation may be made to ascertain the performance when the stalling speed H.P. (·0427) is available.

(13) The formula H.P. $= \frac{DV}{375}$ may be re-written DV= 375 H.P. and thus, in the example under consideration,

$$DV = 375 \times {\cdot}0427 = 16$$

(whereas, when the available H.P. was ·0396, DV = 14·85.

The difference between these figures is a measure of the increased speed, or rate of climb, available due to the extra H.P.

(14) Keeping to the same angle of wing incidence (5·9 degrees), a simple calculation shows that at a speed of 18·9 miles per hour the total drag will be ·85 pound, absorbing the ·0427 H.P. available.

Tabulated, the results from the foregoing calculations are as follows:—

Speed m.p.h.	Angle inci-dence	C_l	C_d	Para-site drag	Wing drag	Total drag	L/D ratio	H.P.
16·0	14·7	1·308	·1518	·307	·698	1·0	6 : 1	·0427
18·4	5·9	·99	·066	·406	·4	·806	7·46 : 1	·0396
18·9	5·9	·99	·066	·428	·422	·85	7·06 : 1	·0427
26·2	·8	·49	·0272	·822	·35	1·172	5.11 : 1	·082

By analysis of these figures, the following information is obtained:

(1) The minimum horizontal flying speed is 16 miles per hour; at which the H.P. required is ·0427; and the wing angle of incidence=14·7 degrees.

(2) The minimum H.P. required to maintain steady horizontal flight is ·0396; when the flying speed is 18·4 miles per hour, and the wing angle of incidence = 5·9 degrees.

(3) The maximum horizontal flying speed—with H.P.= ·0427, and the wing angle of incidence = 5·9 degrees—is 18·9 miles per hour.

(4) The most effective lift/drag ratio is 7·06:1; with a wing angle of incidence of 5·9 degrees.

All the above calculations are aimed at ascertaining the minimum, or near minimum H.P. required to fly the aircraft; but before a final decision is made as to the engine size, calculations should be made for the rate of climb, to check that the excess H.P. available is sufficient to meet the requirements of the designer.

In the present example the E.H.P. is equal to ·0427—·0396 =·0031. Not a very large margin.

The rate of climb of an aircraft may be calculated from the formula:

$$R/C = E.H.P. \times \frac{33{,}000}{W} \quad \ldots \quad \ldots \quad \ldots \quad (18)$$

where EHP = Excess H.P. available.
W = Total weight of the aircraft.
and R/C = is given in feet per minute.

(15) Thus in the example quoted above

$$R/C = {\cdot}0031 \times \frac{33{,}000}{6}$$
$$= 17{\cdot}1 \text{ feet per minute.}$$

This rate of climb is on the low side, and a more reasonable minimum figure would be 100 feet per minute; which, at a flying speed of 18·4 miles per hour, gives a rate of climb of approximately 1 in 16, i.e., a 5-foot hedge would be cleared at a distance of about 80 feet from the point of take-off.

(16) The formula (18) may be re-written in the form

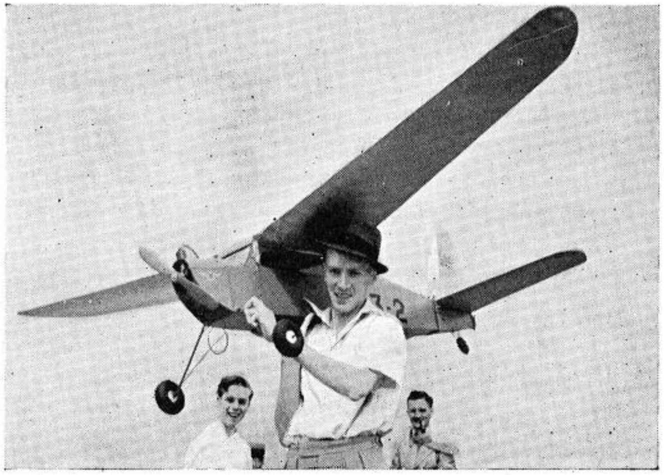
$\text{E.H.P.} = \frac{R/C \times W}{33{,}000}$ and thus, for the aircraft under consideration, the H.P. required for climb at the rate of 100 feet per minute

$$= \frac{100 \times 6}{33{,}000}$$

$= \cdot018$—which, added to that required for minimum flight ($\cdot0396$), gives a total of $\cdot0576$ H.P.

Finally, a simple calculation shows that this H.P. would make possible a maximum speed, in steady horizontal flight, of 20·9 miles per hour. The total drag would be 1·04 pounds; the angle of incidence of the wing being, of course, the same as before = 5·9 degrees.

This final figure of ·0576 H.P. is of considerable interest—in short, it represents the power required to fly an aircraft,



A well-known designer and builder of petrol 'planes, Mr. C. R. Jefferies, is shown in this photo carrying a 'plane of his own construction. Span is some 8 feet.

weighing 6 pounds, at a maximum speed of about 21 miles per hour; or to enable it to climb at the rate of 1 in 16, at a speed of about 18 miles per hour.

With the reservations that the fuselage cross-section is slightly on the small side, the type of aircraft taken for this example is similar to that often powered by a 10-cc. " Brown Junior " engine, advertised as delivering about ·2 H.P. which is about four times as much as that required for the performance above mentioned.

This difference is, of course, accounted for by the fact that the rate of climb of 1 in 16, taken in the example, is much below that actually obtained by present-day models.

Assuming the aircraft in the example to be powered with an engine of ·2 H.P., then the E.H.P. available for climb = $\cdot2 - \cdot0396 = \cdot1604$.

$$\text{Thus the rate of climb now} = \frac{\cdot1604 \times 33{,}000}{6}$$

= 884 feet per minute, which, at a speed of 18·4 miles per hour, gives a rate of climb of about 1 to 1·83, or a climbing angle of approximately 30 degrees. Actually, due to the increased drag resulting from the increased angle of attack of the main wing, the rate of climb would be somewhat less, but an approximate figure of 750-850 feet per minute is in general accord with that obtainable, under good conditions, from the type of aircraft under consideration.

It must be appreciated that, throughout this series of calculations demonstrating how the flight performance of an aircraft may be ascertained, settled air conditions have been assumed; and *given* these conditions, ·0576 H.P. would just fly the 6 pounds weight aircraft taken for the example, and give it a very small E.H.P. for climb. Whilst ·2 H.P. would be sufficient for climb at an angle of approximately 30 degrees, or an increased speed in horizontal flight. However, when calculations are made with the object of ascertaining the *minimum* H.P. required for flight, consideration must be given to the power likely to be required to overcome tractive resistance during the take-off—as under certain conditions an appreciable addition may be required.

During the take-off, the tractive resistance becomes less as the machine becomes air-borne; but also the drag increases;

and it is quite possible for there to be a speed at which the sum of the two kinds of resistances is greater than the total drag. In which case, if the H.P. was only sufficient for flight, the aircraft would be unable to take off.

(19) The tractive resistance may be calculated from the formula

$$R = \frac{W \times F}{2{,}240} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (19)$$

where W = the weight of the aircraft, in pounds
F = Co-efficient of friction, which is expressed in pounds per ton weight—it is equal to about 60 pounds/ton for rubber tyres on a good macadam surface.

And R is given in pounds.

Thus for the aircraft under consideration

$$R = \frac{6 \times 60}{2{,}240}$$
$$= \cdot 16 \text{ pound.}$$

On the closely-mown grass of a tennis court F = about 220 pounds/ton; and on the average aerodrome field F = about 350 pounds/ton. For average conditions F may be taken as 300 pounds/ton; when, in the above example, R would be equal to ·8 pound—roughly equal to that of the whole machine when in flight.

Of course, this value would be dropping as the aircraft was becoming air-borne; but, at the same time, it must be borne in mind that the retarding effect of tufts of grass, small hillocks, etc., can be considerable; and, on balance, it is as well to consider the tractive resistance as being fully effective, until the aircraft has actually left the ground.

(20) The H.P. required to overcome this tractive resistance may be calculated from the formula

$$\frac{R \times S \times 5{,}280}{60 \times 33{,}000} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (20)$$

where R = Tractive resistance in pounds
and S = Speed in miles per hour.

Thus, in the example

$$\text{H.P.} = \frac{\cdot 8 \times 18 \cdot 4 \times 5{,}280}{60 \times 33{,}000}$$
$$= \cdot 04,$$ approximately equal to that

required to overcome the total drag of the aircraft, when flying at about 18 miles per hour.

The general conclusion, therefore, is that to provide a minimum effective performance the machine taken in the example should be powered by an engine of not less than about $\frac{1}{10}$ H.P., whilst $\frac{1}{5}$ H.P. would enable the machine to take off and fly under practically all conditions.

(21) A final performance calculation, and one of more than passing interest, is that to find the length of take-off run.

Continuing the example, and assuming a total H.P. of ·2, the tractive effort available for acceleration may be obtained from the formula

$$F = \frac{\text{H.P.} \times 33{,}000 \times 60}{D \times 5{,}280}$$

where D = speed, in miles per hour.
and F = Tractive effort available for acceleration in pounds.

$$\text{Thus } F = \frac{\cdot 2 \times 33{,}000 \times 60}{18 \cdot 4 \times 5{,}280}$$
$$= 4 \cdot 06 \text{ pounds.}$$

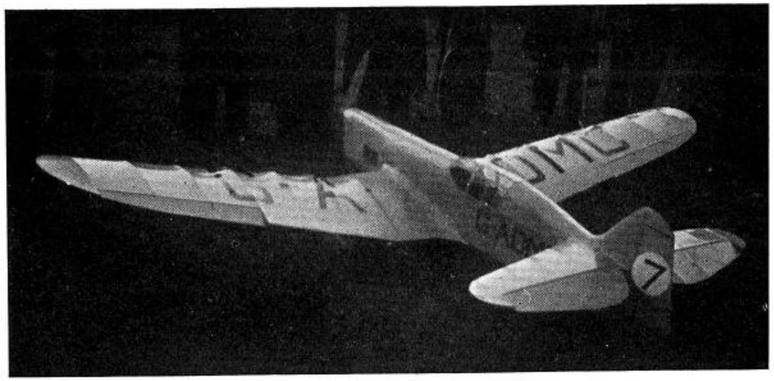
(22) Now to accelerate a mass of 1 pound at a rate of 1 foot per second, requires a force of ·0312 pound—thus in the example (·0312 × 6) = ·187 pound force would be required.

As the force available for acceleration is 4·06 pounds, the rate of acceleration is equal to $\frac{4 \cdot 06}{\cdot 187} = 21 \cdot 8$ feet per second, per second.

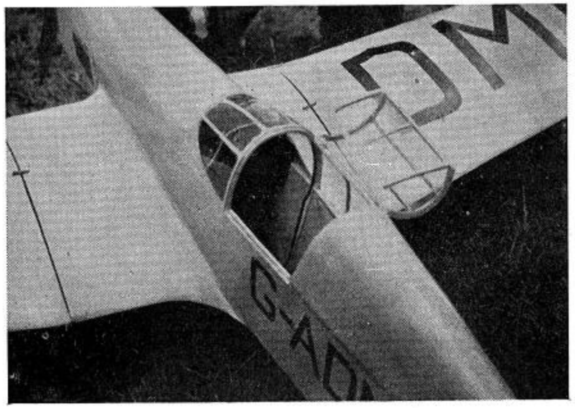
(23) The distance travelled during the take-off may be calculated from the formula $S = \frac{V^2}{2a}$ … … … (22)

where V = Take-off speed in feet per second.
a = Rate of acceleration, in feet per second, per second.

$$\text{Thus, in the example } S = \frac{27^2}{43 \cdot 6}$$
$$= 16 \cdot 7 \text{ feet.}$$



The petrol 'plane is of 11 ft. span and 6 ft. length. It is a scale model of the Percival "Mew Gull," and was built by Mr. Newman. Weight of the 'plane is 8 pounds. and it is powered by a 15 cc. engine, a "close-up" of which is on page 150. Below is a "close-up" of the pilot's cockpit.



CHAPTER VI

AIRSCREW DESIGN

General considerations affecting the design of airscrews—Blade airfoils—Blade width—Blade thickness—Thrust grading lines of airscrew blades—Boss drag—Pitch-Diameter ratio—Metal *v.* wooden airscrews—Formula for calculating the pitch and diameter of power-driven airscrews.

In designing an airscrew for use on a power-driven model aircraft, it must always be borne in mind that the limiting factor is the maximum power available from the motor which will be used to drive the airscrew; since in the absence of any throttle control, or speed limiting device, the airscrew will always revolve at that speed at which it absorbs the full power available from the motor.

The power from a rubber motor, however, gradually falls off from the moment of release, and thus the airscrew cannot be considered by itself, but must always be considered in relation to the power available at any given moment from the rubber motor; and its characteristics must be such that the average of the resultant of the airscrew speeds meets the requirements of the overall design.

Airscrews for use with rubber motors are therefore dealt with under a separate heading in Chapter VIII.

In full-size practice an airscrew will be designed to produce the required amount of thrust at the required forward speed; and a suitable power unit, under the control of a pilot, is then provided.

In model work, however, the designer has available but a limited number of engines, the majority of which are two-strokes, from which to make his choice; and in the absence of a pilot, must make use of such "gadgets," usually adapted clockwork motors, as he can devise and arrange, to effect some measure of variable control, during and after the take-off.

Appreciating, then, that in the absence of any such throttle control, or speed-limiting device, a petrol engine will "rev-up" to that speed at which its full power is being absorbed by the airscrew, it follows that for a given airscrew diameter the finer the pitch, the higher will be the engine speed; and if extremes are considered it will readily be seen that too fine a pitch will allow the engine to " rev " at a speed higher than it is designed for; whilst too coarse a pitch will " hold down " the engine speed to such an extent that it is, to all intents, " stalled," and unable to develop its full power.

Whilst, therefore, the aim of the designer should always be to so " match " the airscrew size with the engine speed, that the power available equals the power required; this may not always be possible, and gearing must then be introduced between the airscrew and engine shafts, a matter not easily arranged and to be avoided if at all possible.

Thus it is seen that the best airscrew, from the aerodynamic point of view, may not necessarily be the best airscrew from the operating point of view, and a compromise may have to be effected. It must, therefore, be appreciated that the following considerations, set out as affecting airscrew design, and the subsequent formulæ, by means of which the dimensions may be ascertained, relate only to the *best* airscrew for any given set of aerodynamic conditions: and that when considered in relation to the characteristics of the engine, some revision in regard to diameter or pitch of the airscrew may have to be made.

(1) The blades of an airscrew are simply airfoils, and operate on the air in exactly the same way as the wings of an aircraft—producing both lift and drag; but whereas the wings are required to produce only lift, the airscrew must produce the required lift (thrust) at a *certain required forward speed*. That is to say, the function of an airscrew is two-fold—firstly, to displace a certain volume of air, thereby producing thrust (which must at least equal the total drag of the aircraft), and secondly, to produce this thrust at a certain forward speed.

(2) The most important factor influencing the performance of an airscrew is the total width of *all* the blades, regardless of the width of each blade, or the number of blades.

Thus a two-blade airscrew will have the same performance as a four-blade airscrew of similar diameter and pitch, if the blades of the former are twice the width of those of the latter —i.e. making the total width the same in each case.

(3) The usual width for the blade of an airscrew is ·05 of the diameter, and the usual thickness about ·125 of the width, in the case of wooden airscrews, and about ·075 of the width in the case of metal airscrews.

The width and thickness being measured at a point ·75 of the radius distant from the centre of the airscrew.

(4) The plan form of airscrew blades is not of very great importance, and provided the blades are tapered, and their tips are rounded, the exact shape and degree of taper has little effect on the efficiency.

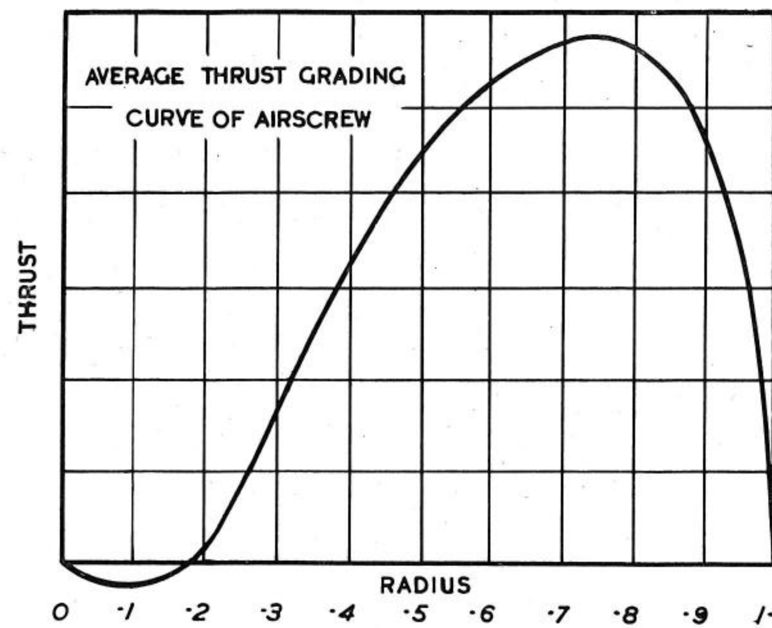


FIG. 11.

That this is quite in the " natural order of things " will be appreciated from a study of Fig. 11 which shows, graphically, the thrust grading of an average airscrew.

It will be seen that by far the greater portion of the thrust is developed by that portion of the blade which lies between ·6 and ·8 of the radius distant from the centre of the airscrew.

(5) It will also be noted that the thrust-grading curve has a slightly *negative* value at a point near the boss, which is

indicative of the drag caused by the uneven airflow through the airscrew at its centre.

It will be appreciated that for the airflow through an airscrew to be constant across the diameter, the *angle* of the blades must increase towards the centre, but from a point about ·15 of the radius distant from the boss, and inwards, the shape is such that the angle is far too large for that portion of the blade to operate efficiently; consequently the airflow is slower at the boss than across the rest of the airscrew diameter, resulting in a decrease of propulsive effort at this point, or in effect, the creation of a small amount of drag.

The introduction of a " spinner " tends to divert the airflow away from the hubs; and by bringing up a small diameter of the fuselage close behind the airscrew, the creation of an area of relatively lower pressure is prevented.

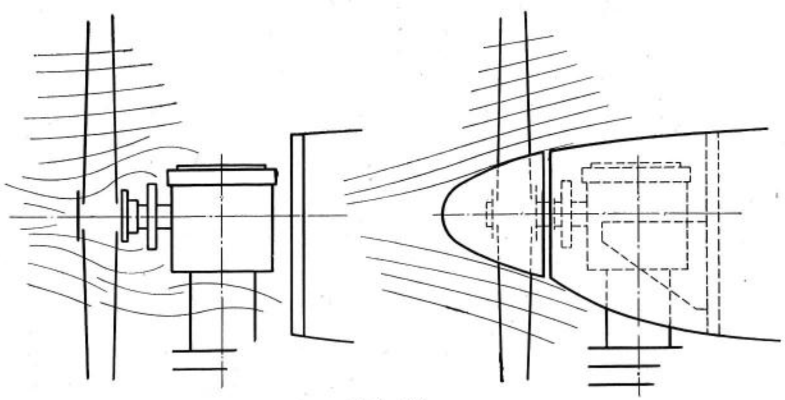


Fig. 12.

Fig. 12 shows, diagrammatically, the airflow through an airscrew with, and without, a spinner, and " backing " of fuselage.

The proportion, as shown, is generally considered the best; where the fuselage diameter at the point immediately behind the airscrew is ·3 of the airscrew diameter.

Whilst the presence of a small diameter of the fuselage in close proximity to the rear of the airscrew hub will prevent a negative thrust at this point, the effect of the fuselage as a whole is to reduce the efficiency value by an amount depending on the ratio of fuselage diameter to airscrew diameter.

This reduction in efficiency value varies from about 3 per cent for a fuselage/airscrew diameter ratio of ·4 to about 13 per cent of a ratio of ·75.

(6) It will also be evident from the thrust-grading curve, that variations in the pitch along the blade of an airscrew will not be of great moment; the *average* pitch of the whole blade, and in particular the pitch at the point 75 per cent of the radius distant from the airscrew centre, being the important thing.

The ratio of pitch to diameter has an influence on the efficiency of the airscrew; *full-size* tests having shown that it will increase from 68 per cent in the case of a P/D ratio of ·45—up to 82 per cent in the case of a P/D ratio of 1 : 1. In practice P/D ratios do not usually exceed ·8.

In model aircraft practice the P/D ratio should not exceed about ·7 when the wing loading is 16 ounces per square foot and over; whilst for wing loadings below that figure the P/D ratio should not exceed ·8.

(7) Regarding the choice between metal and wood airscrews, the matter is more a question of practical consideration (and finance!) rather than design.

Owing to the thinner blade section which may be used with metal airscrews, the drag is a little less, resulting in a slightly higher efficiency (70 per cent for metal, 65 per cent for wood, being about the best values obtainable in model practice).

The blades of wooden airscrews *may*, of course, be made as thin as those of metal, and speeds of 4,000 r.p.m. with diameters of 18-20 inches have been obtained during tests, without signs of splitting.

Wooden airscrews are, of course, much cheaper than those of metal, and thus may more easily be replaced in the event of a breakage; on the other hand, a metal airscrew, whilst costing about four times as much as a wooden one of similar size, will often outlive several wooden airscrews, as quite severe bends to the blades may fairly easily be straightened out, provided care is taken; and within reasonable limits the pitch may be varied.

A model aircraft airscrew, for power work, will be working at its highest efficiency when the value of "J" is approximately ·5; and by use of the formula

$$D = \frac{88 \times MPH}{RPM \times \left(\frac{V}{ND}\right)} \qquad \ldots \quad \ldots \quad \ldots \quad (23)$$

An approximate figure for the airscrew diameter may be calculated, *provided that*, under "in flight" conditions, the value of "J" *is* ·5.

(NOTE.—"J" is defined as "the rate of advance per revolution expressed as a fraction of the diameter"—and is dealt with fully in Chapter VII.)

Suppose it is required to find the diameter and pitch of an airscrew for an aircraft to fly at a speed of 25 m.p.h., the airscrew speed to be 3,000 r.p.m.

$$\text{Then } D = \frac{88 \times 25}{3{,}000 \times \cdot 5}$$

$$= 1{\cdot}46 \text{ feet.}$$

For a metal airscrew, if "J" = ·5, the efficiency will be 70 per cent. Thus the pitch will require to be such as to give a theoretical forward speed of 35·7 m.p.h., or 52·2 feet per second.

As the airscrew speed is 50 revolutions per second

$$\text{pitch} = \frac{52{\cdot}2}{50}$$

$$= 1{\cdot}04 \text{ feet.}$$

$$\text{and the P/D ratio} = \frac{1{\cdot}04}{1{\cdot}46}$$

$$= \cdot 712 \text{ foot.}$$

Suppose, however, that it is desired to find the diameter and pitch of a metal airscrew for an aircraft to fly at a speed of 18 m.p.h., the airscrew speed to be 4,000 r.p.m.

$$\text{Then } D = \frac{88 \times 18}{4{,}000 \times \cdot 5}$$

$$= \cdot 79 \text{ feet.}$$

With the value of "J" taken as ·5, the efficiency will be 70 per cent—thus the theoretical forward speed will be 18 ÷ ·7 = 25·8 m.p.h., or 37·7 feet per second.

As the airscrew speed is 66·6 revolutions per second

$$\text{pitch} = \frac{37{\cdot}7}{66{\cdot}6}$$

$$= \cdot 565 \text{ foot.}$$

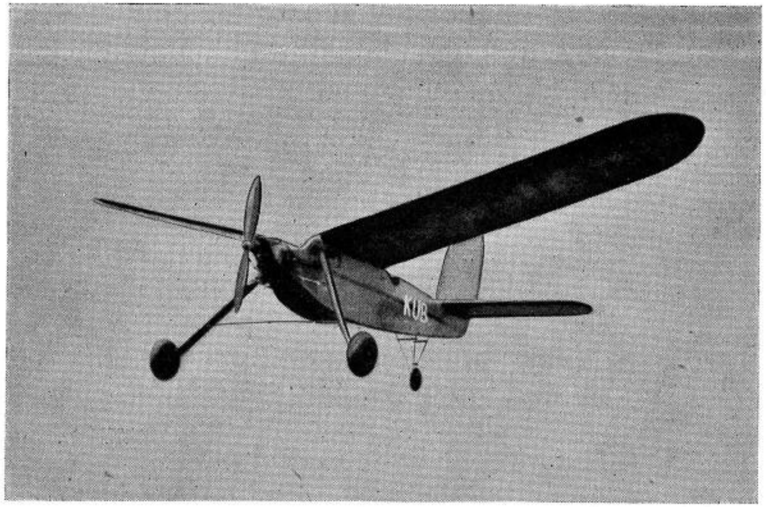
Now the thrust available from such an airscrew, running at 4,000 r.p.m., would be approximately 1·7 pounds, requiring about ·16 b.h.p. from the engine, which would be of about 5-6 cc. capacity.

Suppose, however, a 10-cc. engine were the only one available; then, if given full throttle, it would run up to well over 4,000 r.p.m.

Thus the value of "J" would be decreased, indicating that the airscrew was not working under such good conditions; any increase in aircraft speed which might be thought to accrue, due to the higher airscrew revolutions, would be cancelled out by the lower efficiency.

Here then would be a definite case for some form of throttle control, or a slight increase in the airscrew diameter or pitch, so as to absorb the full power of the engine, and keep the revolutions down to 4,000.

In short, nothing is gained by overdriving the airscrew, and serious damage may result to the engine due to the increased speed.



One of the most successful designers of petrol 'planes is Lt.-Col. C. E. Bowden. Here is the "Kanga Kub," which won the "Sir John Shelley" Cup in 1937. It was built and flown by Mr. C. R. Jeffries, to Lt.-Col. Bowden's design.

CHAPTER VII

AIRSCREW PERFORMANCE

Formula for calculating ideal thrust of airscrews—Static thrust—Value of J—Actual thrust—Method of estimating airscrew performance—Empirical formula for estimating the static thrust of power-driven airscrews.

THEORETICALLY—that is assuming 100 per cent efficiency—the static thrust developed by an airscrew may be calculated by use of the formula

$$T = 3{\cdot}142 \times r^2 \times p \times n \times {\cdot}076 \quad \ldots \quad (24)$$

where r = Airscrew radius, in feet.
p = ,, pitch, in feet.
n = ,, revolutions, per second.
·076 = Weight in pounds, of 1 cubic foot of air,

and T is given pounds.

Actually, of course, 100 per cent efficiency cannot be obtained as, due to the "fluidity" of the air, a certain amount of "slip" inevitably occurs; and the resultant figure of thrust obtained by use of this formula must be multiplied by an "efficiency" factor—which may be anything between ·8 and ·88, according to the type of airscrew under consideration.

For a well-designed metal airscrew working under its "best" conditions, this factor will be about ·85, and for a similar wooden airscrew, about ·8.

Now static thrust is that which is developed when an airscrew is revolving in a fixed plane: when it is, in effect, acting as a fan.

Under this condition the efficiency is said to be zero, since although the thrust will be quite large, as the forward velocity is zero, no useful work is being done.

A measure of the efficiency of the conditions under which an airscrew is working may be obtained by use of the formula

$$J = \frac{V}{ND} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (25)$$

where V = Aircraft velocity in feet per second.
N = Airscrew revolutions per second.
and D = ,, diameter in feet.

Thus when the efficiency is zero, " J " is zero. As the value of " J " increases (with increase in V) so the efficiency increases until it reaches a maximum point, after which it falls very quickly.

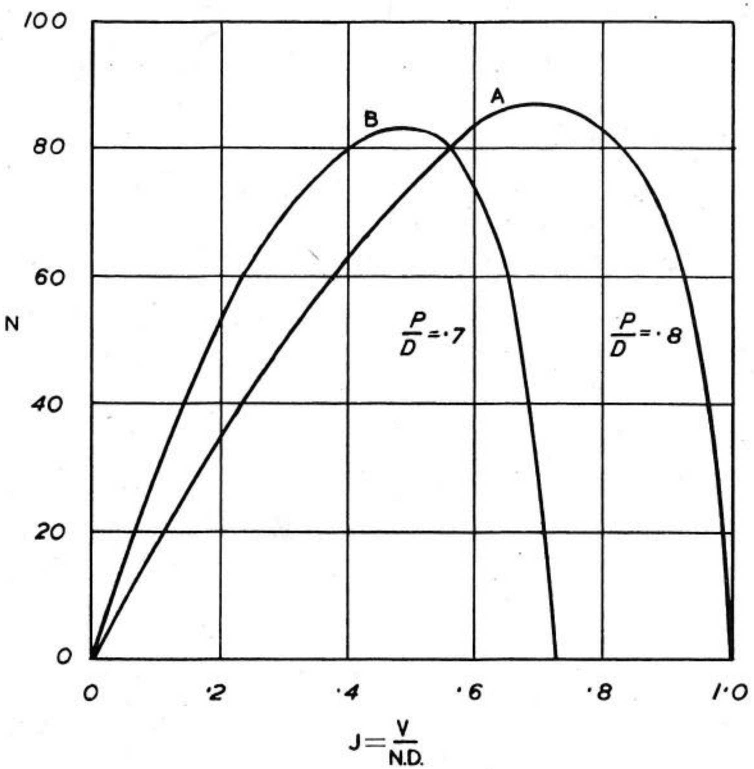
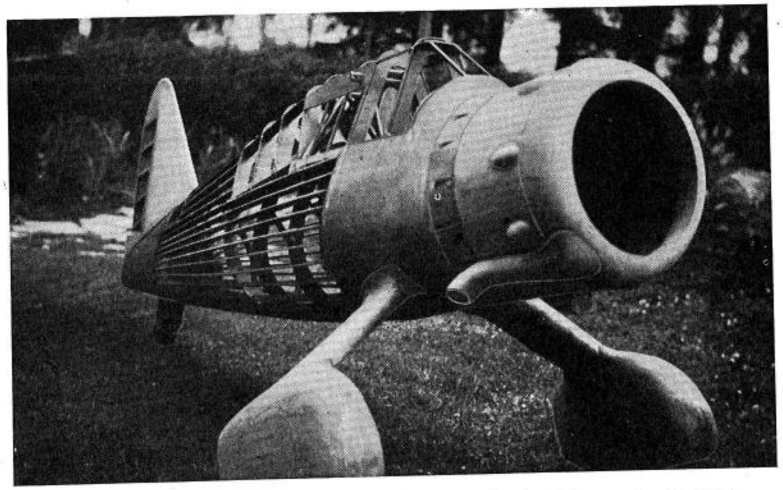


FIG. 13.

Curve A, Fig. 13, is typical of average full-size airscrews —from which it will be seen that the efficiency reaches its maximum of ·88 when the value of " J " is a little over ·75. In model airscrew practice, however, the efficiency appears to reach a maximum of ·83 when the value of " J " is approximately ·5, and curve B is typical of metal airscrews of diameter of from 14 inches to 18 inches.

As the forward velocity of an airscrew increases, as in



Here is the fuselage, 6 ft. long, of the Author's " Lysander." This fine photograph shows how the construction of the full size machine has been faithfully imitated. There are some forty longerons in the fuselage. The front part, and the front portion of the fin, which are of metal construction, are in the model built of balsa and three-ply, $\frac{1}{32}$ in. thick. The spats, which house 8 in. diameter pneumatic-tyred wheels, are carved from solid blocks of balsa.

flight, the thrust developed becomes less, due to the blades meeting the air at a reduced angle of attack; and in model practice this reduction is of the order of about 17 per cent of the static thrust developed when the value of " J " is zero.

The net result of these reductions is to indicate that the maximum working efficiency of a metal airscrew is approximately 70 per cent of the theoretical value as calculated by means of formula (24).

Size for size, wood airscrews are usually from 4 per cent to 6 per cent less efficient than those of metal, thus the maximum working efficiency may be taken as approximately 65 per cent of the theoretical value obtained by means of the same formula.

These efficiencies, of course, only obtain when the value of " J " is approximately ·5; and compare with maximum values of 85 per cent and 78 per cent—with a value for " J " of approximately ·6—obtained in full-size practice.

Generally speaking, large airscrews revolving at relatively slow speeds are the most effective, and full-size airscrews rarely turn at much more than 2,200 r.p.m., compared with 3,000-4,000 r.p.m. commonly obtained in model practice.

Steps in the performance estimation of an airscrew are as follows:—

Consider a metal airscrew 1·5 feet in diameter, of 1-foot pitch, running at 3,500 r.p.m.

(1) If the airscrew is to operate under the best conditions "J" must equal approximately ·5; when the efficiency will be 70 per cent.

In the case of the example

$$J = \frac{1 \times 58{\cdot}5 \times {\cdot}7}{58{\cdot}5 \times 1{\cdot}5}$$

$= {\cdot}466$ which may be considered quite satisfactory. If the result of this calculation had been to yield an answer much above or below this figure—one or other of the three factors—pitch, diameter, or airscrew revolutions—would have required to have been altered accordingly.

(2) By formula (24)

$$T = 3{\cdot}142 \times {\cdot}75 \times {\cdot}75 \times 1 \times 58{\cdot}5 \times {\cdot}076$$

$= 7{\cdot}86$ pounds thrust (100 per cent efficiency).

Allowing for slip $7{\cdot}86 \times {\cdot}85$

$= 6{\cdot}68$ pounds thrust (static). Allowing for reduced thrust, due to reduced angle of attack of blades $6{\cdot}68 \times {\cdot}83$

$= 5{\cdot}55$ pounds thrust (actual).

[NOTE.—Ratio of 5·55 pounds (actual working thrust) to 7·86 pounds (assuming 100 per cent efficiency) = ·705.]

For ordinary purposes these two calculations may be consolidated into one—by an overall reduction to 70 per cent in the case of metal airscrews, and to 65 per cent in the case of wood airscrews, of the value obtained by use of formula (24).

(3) The actual velocity may now be calculated by multiplying together the pitch, the number of revolutions, and the efficiency of the airscrew.

In the case of the example quoted above

$$V = 1 \times 58{\cdot}5 \times {\cdot}7$$

$= 41$ feet per second.

An empirical formula often used in full-size practice for

calculating static thrust is that developed by W. S. Diehl,* which states that

$$T = 6{,}000 \left[18{\cdot}7 - 9{\cdot}5 \left(\frac{P}{D} \right) \right] \frac{\text{B.H.P.}}{\text{r.p.m.} \times D} \quad \ldots \quad (26)$$

where P = Airscrew pitch, in feet.
D = Airscrew diameter, in feet.
and T = Actual static thrust developed.

This formula may be re-written

$$\text{B.H.P.} = \frac{T \times \text{r.p.m.} \times D}{6{,}000 \left(18{\cdot}7 - 9{\cdot}5 \left(\frac{P}{D} \right) \right)} \quad \ldots \quad \ldots \quad (27)$$

and used to ascertain the power required to drive a given airscrew, *provided* $J = {\cdot}5$.

For use with model airscrews, it would seem that a coefficient of 1·15 must be introduced, when the resultant answer fairly accurately agrees with results obtained from a series of tests carried out with a number of airscrews of the sizes commonly used in model aircraft practice.

In the case of the example quoted above

$$\text{B.H.P.} = 1{\cdot}15 \left(\frac{6{\cdot}68 \times 3{,}500 \times 1{\cdot}5}{6{,}000 \times (18{\cdot}7 - (9{\cdot}5 \times {\cdot}667))} \right)$$

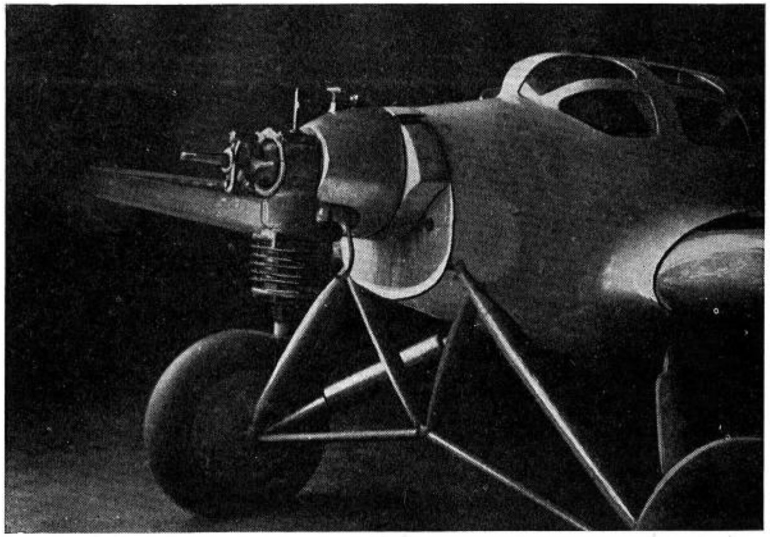
$= {\cdot}543$

Finally, reference must be made to the engine power curve, to ascertain whether the required power will be delivered at the specified number of airscrew revolutions.

In the case of the example ·543 B.H.P. *must* be available at 3,500 r.p.m.

If this were not the case, either the diameter or pitch of the airscrew—or perhaps both—would require to be altered accordingly.

* W. S. Diehl, *Engineering Aerodynamics.*



This photo gives a very good idea of the construction of the front of the fuselage and undercarriage of the 10 ft. span low-wing 'plane designed by the Author. Note the smooth flaring of the wing root into the fuselage.

CHAPTER VIII

RUBBER MOTORS

The energy stored in twisted rubber—Number of turns available in different types of rubber motors—Equipment for measuring thrust of rubber-driven airscrews—Curves showing typical results obtained with this apparatus.

A TWISTED rubber motor suffers from the handicap that its power output is not constant, but decreases—very rapidly in the first few seconds—during the time of unwinding.

The energy stored in a rubber motor, wound up nearly to breaking strain, may be quite accurately calculated on the basis of 2,000 foot-pounds of energy per 1 pound weight of rubber; but the estimation of the amount of power available at any given moment during the unwinding is not possible except by means of very complicated calculations.

Further, since the strands of rubber in a motor can be arranged in a great number of different ways, all of which will give different results—and any of which might be described as the " best," depending on the conditions under which it has to work—it follows that it is quite impossible to lay down " hard and fast " any one rule as to how best the strands of a rubber motor shall be arranged.

Only by testing out various arrangements of the strands of a motor can an idea of the power available be obtained—and then only as applying to the particular airscrew used for the tests—since the rate of unwinding of *any* rubber motor will be controlled by the diameter and pitch of the airscrew it is driving.

An equipment for testing out various arrangements of rubber motors may be easily and cheaply built, and will yield very useful results. The aero-modellist will find that, after carrying out a number of tests with a series of certain arrangements of the strands of rubber, he will be able to predict, with a quite fair degree of accuracy, the probable performance of a series of somewhat different arrangements of the strands of rubber.

Following is a description of the apparatus built by the author, and with which many tests have been carried out; the results of some of them, in the form of thrust power and airscrew revolution curves, being given.

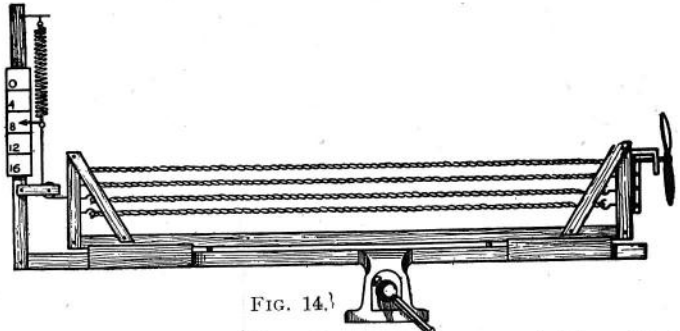


FIG. 14.

Fig. 14 is a sketch of the apparatus, and Fig. 15 is a photo showing the 4-spindle gearbox and airscrew mounting.

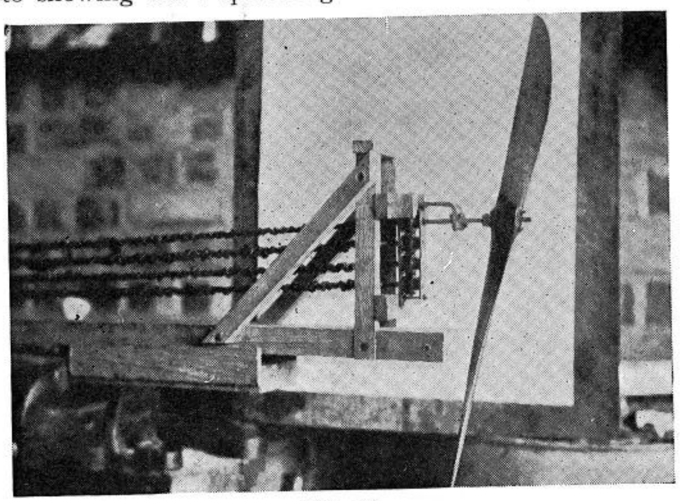


FIG. 15.

The four-spindle motor shown on the test rig in the above photograph was specially built for airscrew testing. All spindles are mounted with ball thrust bearings, and the propeller shaft also carries a set of roller bearings to reduce friction.

Essentially the apparatus consists of a "carriage" on which is mounted the rubber motor and airscrew to be tested, the carriage being free to move on rollers in a fore and aft direction along the sloping runway. This runway is held for convenience in a bench vice, and is thus easily adjustable

as to its angle of inclination. When a given motor and airscrew have been assembled, the runway is set so that the carriage will just *not* run down the incline. This has the effect of reducing the "tractive resistance" (in this case the friction of the rollers on which the carriage runs) to a negligible quantity.

To the back end of the carriage is attached a thread which passes under a pulley and up to the lower end of a vertically-mounted spring which, up to a certain limit, has a constant rate of extension per ounce of added weight. To the lower end of the spring, where the thread joins it, is fixed a pointer which moves across a scale marked in ounces. (This scale has, of course, been previously calibrated by weights hung direct on to the spring.)

In operation, after the motor and airscrew to be tested have been mounted on the carriage and wound up, the thread is detached from the spring, and the runway adjusted as already described. The thread is then fixed to the spring and the motor released; immediately the carriage moves forward, the thrust in ounces being indicated by the pointer on the scale. As the power falls off, the spring pulls the carriage backwards, until finally, when the motor has completely unwound, the pointer is back at zero.

During the test, 5-second (and with practice, 3-second) readings are taken with the aid of a stop-watch, and thus accurate curves may be drawn showing the rate at which the power falls off. Readings are also taken with a revolution-counter and the results compared with the thrust, and this information allows the rubber motor to be so arranged that it will (say for a duration flight) deliver as constant an output for as long as possible, at a figure just in excess of the minimum at which the machine will fly.

Fig. 16 shows curves of two rubber motors of average arrangement, in each case driving a 16-inch diameter × 14-inch pitch airscrew. Curve "B," representing the 10-strand motor, showing how the extra power, at the take-off, is obtained, though at the expense of shortening the length of flight a little. Fig. 17 shows the power output of the same rubber motors recorded in ounces of thrust, as compared with r.p.m., as recorded in Fig. 16.

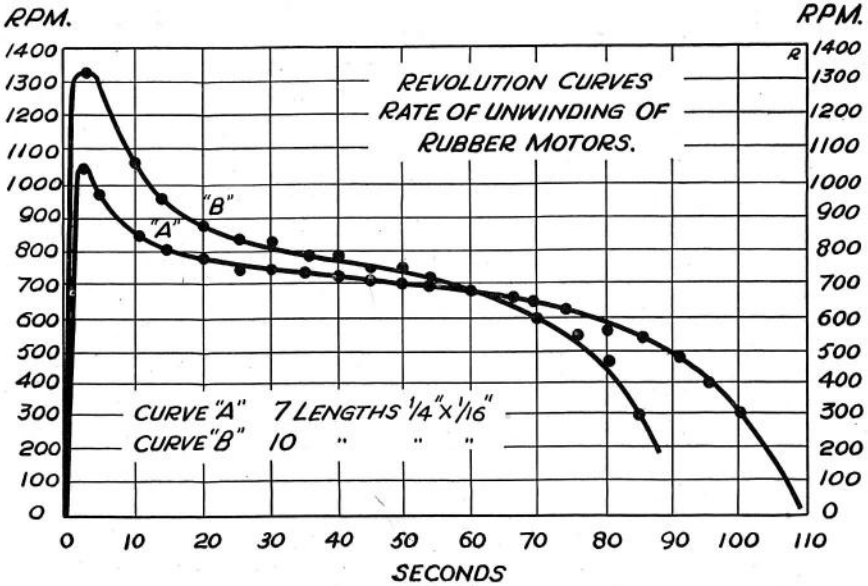


FIG. 16.

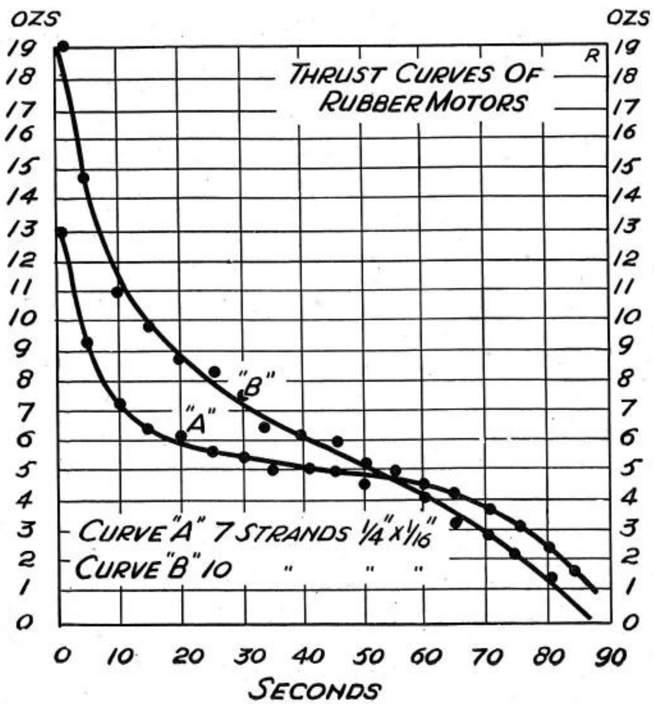


FIG. 17.

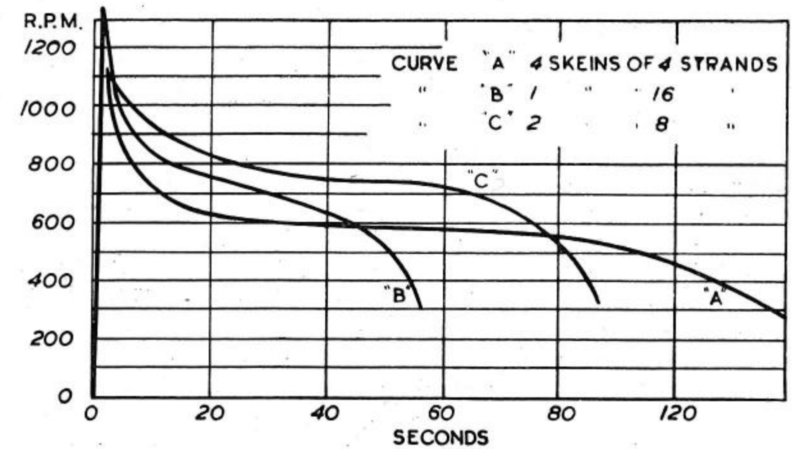


FIG. 18.

Fig. 18 shows curves of results obtained with 48 feet of $\frac{1}{4}$-inch × $\frac{1}{30}$-inch rubber arranged in three different ways, in each case driving an 18-inch diameter × 14-inch pitch airscrew. Curve "A" 4—36-inch skeins, each of 4 strands, on the 4 hooks of the motor. Curve "B"—with *all* the rubber on one hook, i.e. 16—36-inch strands; and Curve "C" showing the results obtained by using the rubber arranged as 2—36-inch skeins, each containing 8 strands.

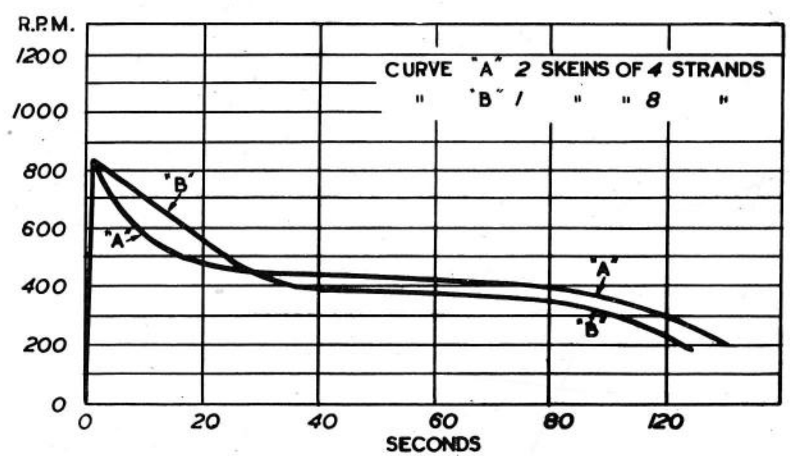


FIG. 19.

Fig. 19 shows results obtained by using 24 feet of $\frac{1}{4}$-inch × $\frac{1}{30}$-inch rubber. Curve " A " with 2—36-inch skeins, each of 4 strands, and Curve " B " with all the 8 strands on one hook.

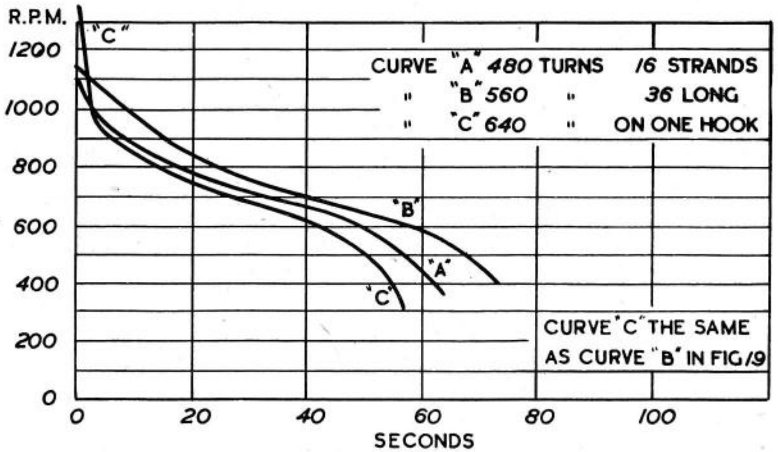


FIG. 20.

Fig. 20 shows the results obtained by winding up the same motor to different numbers of turns—to 480, 560 and 640 respectively, out of a permissible maximum of approximately 700.

The interesting feature is that *both* A and B show a " peak " of 1,100 revolutions per minute, with B, as one would expect, running for a longer period than A. But curve C shows how the power rapidly increases with the last 100 turns, giving a " peak " of 1,400 revolutions per minute, but falling off so rapidly that the total duration is a little *less* than A. From this it is seen that when a motor has a great number of strands on it, too great a number of turns will produce a motor quite unsuitable for flying.

Fig. 21 shows results obtained using very long rubber motors, specially arranged for duration flying, driving the same airscrew.

Fig. 22 shows two torque curves of different rubber motors, each showing the rapid rise in torque as the number of twists imparted to the rubber approaches breaking point.

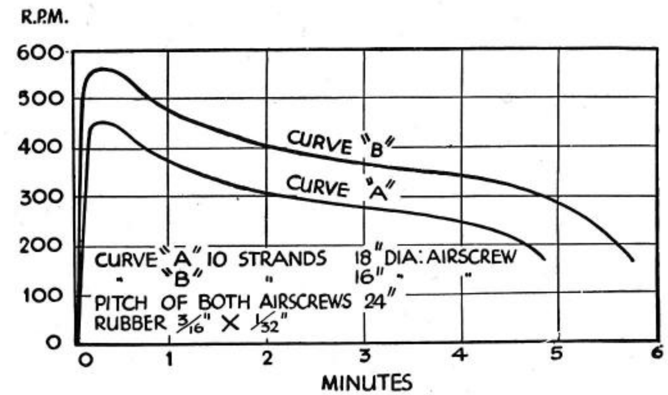


FIG. 21.

Here again is indicated the value of " bench tests " of the power unit, since by careful experimenting it is possible to so arrange the strands of rubber that the extra power is available for just as long as it is wanted, after which it falls off to the minimum required to fly the machine, this giving as long a flight as possible.

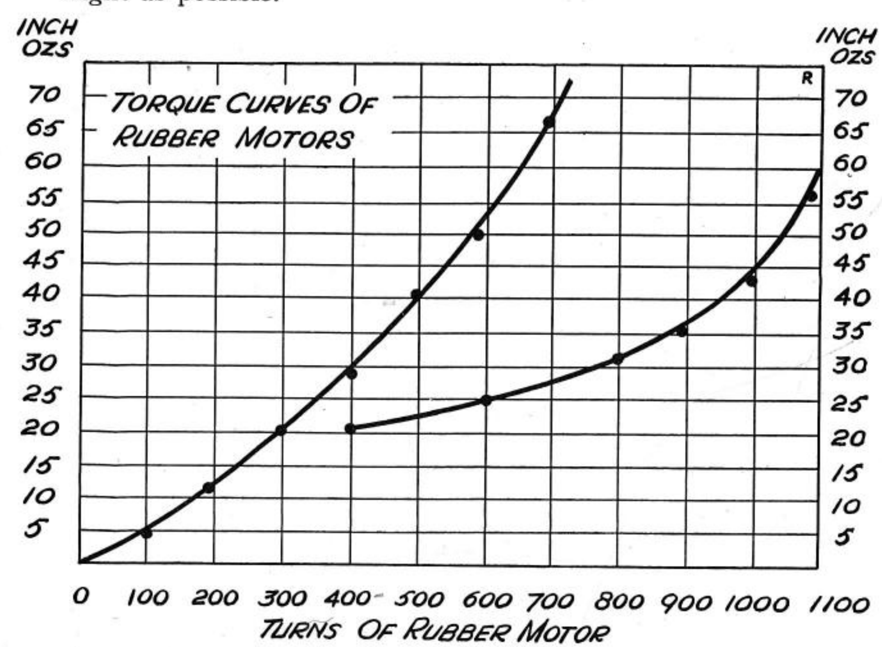


FIG. 22.

The following formulæ may be used in estimating the performance of rubber-driven model aircraft:—

(1) The distance a model will travel under power =

$$D = \frac{K\ WR}{W}\ \text{feet} \quad \ldots \quad \ldots \quad (28)$$

where WR = Weight of rubber motor.
W = Total weight of model with motor.
K = Approx. 3,000 for models with high lift wing sections and not specially streamlined—4,000—5,000 for streamlined models.

(2) The number of turns a rubber motor will stand =

$$N = \frac{KL\sqrt{L}}{\sqrt{W}} \quad \ldots \quad \ldots \quad \ldots \quad (29)$$

where L = Length of skein unstretched in inches.
W = Weight of rubber per skein in ounces.
K is usually taken as 4, but if the motor is stretch wound may be safely increased to 5.

(3) The propeller pitch, allowing 25 per cent slip =

$$\frac{D}{N \times R \times \cdot 75}\ \text{feet}$$

R = gear ratio = 1 if the propeller is driven direct.

Taking as an example a non-streamlined model with 200 square inches wing area, and weighing 3 ounces + 1 ounce of rubber, with a single skein 30 inches long.

$$D = \frac{3{,}000 \times 1}{4} = 750\ \text{feet.}$$

$$N = \frac{5 \times 30 \times 5 \cdot 5}{1} = 825$$

$$\text{Propeller pitch} = \frac{750}{825 \times \cdot 75} = 1 \cdot 21\ \text{feet} = 14\tfrac{1}{2}\ \text{inches.}$$

The propeller pitch should be from one to one-and-a-half its diameter for rubber driven models.

In order that a given rubber motor shall store the maximum amount of energy it is obvious that all the particles of rubber in that motor shall be equally stressed. Now if a dry

rubber skein be twisted, i.e. wound up, there will be a tendency for the strands on the surface of the skein to be stretched considerably more than those at the centre. Since each strand in the length of the skein passes several times from the surface through the centre it will be seen that if the strands are lubricated and allowed to slide over each other, the strain will be more equally divided, and hence the energy stored will be increased.

The energy can be still further increased by stretching the skein to about five times its original length before starting to wind, and gradually decreasing the length when approaching the full number of turns.

The following tables, compiled by R. M. Glass from some recent experiments he has made, show the torque at various turns on motors unlubricated, lubricated unstretched, and lubricated stretch wound.

Each sample consisted of 8 strands of $\frac{1}{8}$ inch × 1/30 inch rubber 16 inches long, 12 inches between hooks. All the samples were cut from the same hank, and each weighed $\frac{1}{4}$ ounce.

Turns, Torques, Breaking Points.

Unlubricated—Unstretched.

A.

Turns.	Torque.
100	2
200	4
298	Breaking point.

Lubricated—Unstretched.

B.		*C.*		
			Torque	
Turns.	Torque	Turns.	Winding.	Unwinding
100	2	100	2	$\frac{1}{2}$
200	$2\frac{1}{2}$	200	$2\frac{1}{2}$	1
300	3	300	3	1
350	$3\frac{1}{2}$	350	$3\frac{1}{2}$	$1\frac{1}{4}$
400	5	400	4	2
450	$6\frac{1}{2}$	450	$5\frac{1}{2}$	—
486	Breaking point.			

Lubricated—Stretched.

D. Prewound.		*E. First Wind.*		
			Torque	
Turns.	Torque	Turns.	Winding.	Unwinding
200	1	200	2	—
400	2	400	3	—
500	2	500	3	—
600	$2\frac{1}{2}$	600	$3\frac{1}{2}$	—
700	$3\frac{1}{2}$	700	6	—
800	$4\frac{1}{2}$	800	8	1
900	$6\frac{1}{2}$	900	10	2
1000	$8\frac{1}{2}$	950	$12\frac{1}{2}$	4
1025	$10\frac{1}{2}$	975	13	—
1040	12	1000	15	—
1050	13	On rewind breaking point		
1070	13		=1128	
1075	Breaking point.			

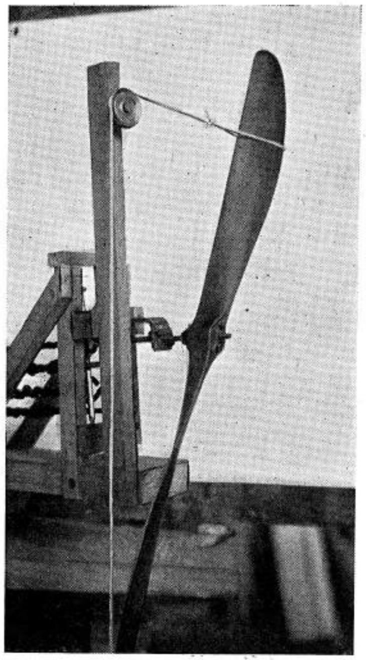


FIG. 23.

In this photo is shown the set-up for measuring the torque of a rubber motor. At the end of the piece of string, which passes round the pulley, is a hook on which are hung weights to balance—and so measure—the force which is being exerted by the rubber motor to turn the airscrew round.

As an examination of the results shows, the number of turns required to break a stretch wound skein is more than twice that for a lubricated unstretched skein, also the torque

at breaking point is much higher, although the average torque as the motor is unwinding is slightly less for the stretch wound motor. The thick lines on the curves show the torque when winding, and the dotted lines when unwinding. The area below the dotted curves represents the energy delivered by the motor.

Other interesting facts concerning rubber motors are:—

(1) When a motor is wound up nearly to breaking point and held in this condition, the torque decreases very rapidly.

(2) A motor will stand considerably more turns after it has been wound up and run out a few times, although the torque at breaking point is considerably reduced. The total energy which can be obtained from a motor does not vary much, if well lubricated, during the first ten or more flights; it is usually considered to be a maximum on the third.

(3) If a motor is wound up slowly it will stand more turns and give less torque than it would if wound quickly.

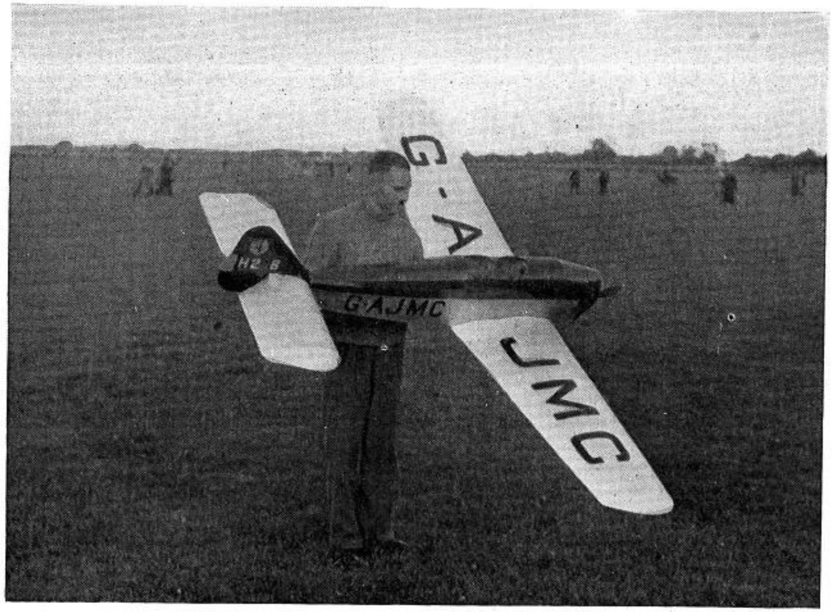
A skein consisting of a large number of small strands will stand more turns than one of the same length and weight consisting of a few thick strands.

When rubber is first produced in the form of strand for model aircraft motors it is of a soft, sticky nature. If this rubber is exposed to air, or more particularly, to light, it gradually dries and hardens. When sold by model firms it is usually in about the best condition for flying, and to prevent deterioration stock should be kept in an airtight tin.

A week's exposure to sunlight will completely ruin a motor, as the rubber becomes brittle. For this reason it is necessary to frequently replace rubber bands when they are used in exposed places, such as to hold wings or tail unit in position.

A motor should be lubricated and the lubricant well rubbed in at least an hour, and preferably a day before winding. If it is going to be out of use for a period of more than a fortnight it is advisable to rinse the lubricant off, dry the skein and return it to the airtight tin.

As has already been pointed out, a quantity of rubber may be arranged in a very great number of ways on the hooks



The " Bowden Trophy "—an International Competition for petrol 'planes —was won in 1939 by Mr. J. M. Coxall, who is shown here with his winning plane. A photo showing the model taking-off on one of its competition flights is on page 39.

of a single or multi-spindle motor—but from a study of the curves and tables given in this chapter, the aero-modeller will gain a general idea of how the power output is delivered during the time of unwinding, whilst from a test apparatus as here described he may obtain power outputs to suit practically any set of conditions.

CHAPTER IX.

TESTING POWER-DRIVEN AIRSCREWS

Method of ascertaining thrust of power-driven airscrews —Value of K for electric motors—Curves of test results of metal airscrews.

THE " Carriage and Spring Balance " equipment described in Chapter VIII cannot be used for the testing of the larger and much faster-revolving airscrews used on petrol-driven aircraft, since these may require anything up to $\frac{3}{4}$ h.p. to drive them, and the weight of the necessary motor, some 25-35 pounds, introduces so much friction on the carriage bearings that accurate results are difficult to obtain.

By slinging the driving motor from a suitable support about 6 or 8 feet above the airscrew centre, and measuring the distance forward which the motor moves when driving the airscrew, a direct measurement of the static thrust may be obtained, allowance of course being made for the air resistance offered by the electric motor.

Fig. 23 shows diagrammatically the method of slinging the motor for test; the distance from the support to the motor

A motor slung from the workshop ceiling to measure the distance forward which the motor moves when driving the propeller at various speeds.

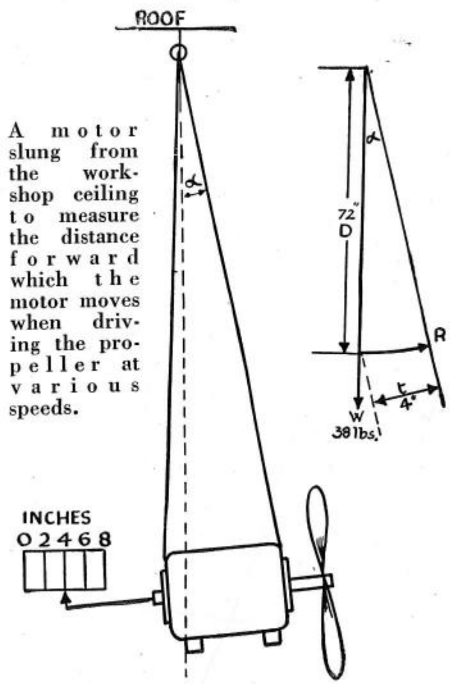


FIG. 23.

shaft being not less than 6 feet. Care should be taken to see that the motor leads are flexible, and that they are arranged to hang freely from the support; they must *not* be led to the motor from the side or their weight may have a restraining effect on the movement of the motor.

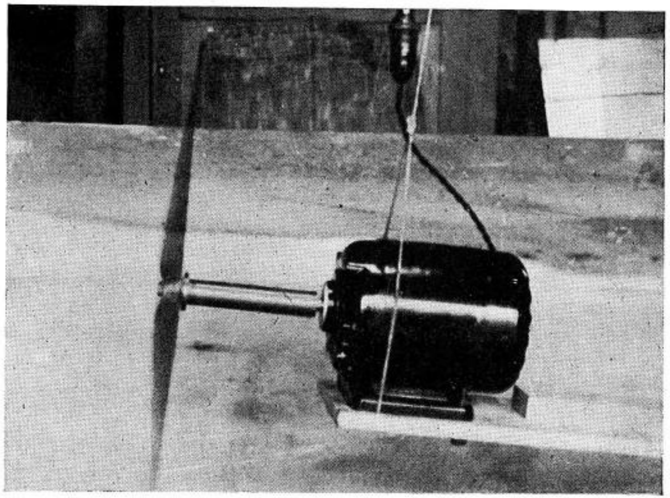


FIG. 24

This photograph shows a metal airscrew being tested in the Author's workshop. The motor and board weigh nearly 30 lb., and as can be clearly seen, it is being pulled forward several inches by the airscrew.

To the back end of the motor is fixed a pointer, so arranged as to pass across a scale, graduated in inches, as the motor swings forward under the *pull* of the airscrew. It is important to see that the direction of rotation of the motor and the " hand " of the airscrew, are such that the motor is *pulled* and not *pushed* forward, i.e. the arrangement *must* be as shown in Fig. 24. To have the motor being *pushed* forward is not safe, as the arrangement is not stable.

During test there will be a tendency for the motor to swing round, due to the torque reaction of the airscrew, and this may be counteracted by means of a fine wire led from the back of the motor to a point at the side some 3 or 4 feet away. This then allows the motor to swing backwards and forwards, with a practically straight motion.

Having measured the exact distance from the point of

suspension to the centre of the airscrew, and obtained also the exact weight of the motor, airscrew, supporting wires, and electric cables—in fact, *all* the suspended weight—tests may proceed.

Firstly, the motor should be run up to say 1,000 r.p.m., this being checked by a revolution-counter, whilst the motor is held steady by hand. Secondly the motor is released, and allowed to move forward under the influence of the pull of the airscrew thrust, the distance moved being measured on the scale. The motor speed is then increased by 200-300 r.p.m., and the process repeated, until a series of readings over the airscrew speed range has been taken.

The value of the actual static thrust developed is then calculated from the formula $T = \frac{WC}{D}$ ... ... (28)

where W = Total suspended weight, in pounds.

C = Distance the motor moves forward in inches.

D = Distance from point of suspension to airscrew centre in inches.

and T = Static thrust in pounds.

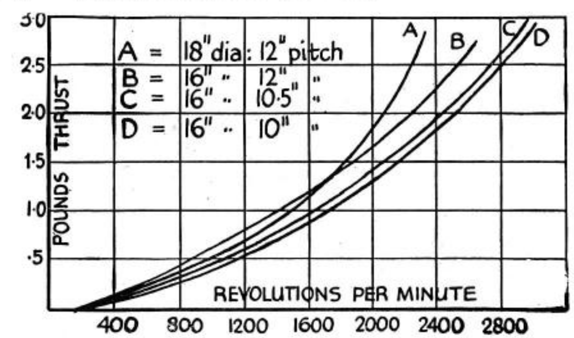


FIG. 25.

Fig. 25 shows a typical set of readings obtained from testing a number of airscrews in this manner. It must not be forgotten that the thrust measured is *static* thrust, at zero advance; and thus, as explained in Chapter VII, a reduction must be made to estimate the thrust available actually in flight. All static thrust readings obtained in this manner should be multiplied by ·83.

Allowance must also be made for the drag of the motor driving the airscrew, if the actual net thrust value is required, and this, of course, depends on the shape of the motor.

In the case of those in use by the author, it has been found, by experiments in the wind tunnel, that the value for K varies between ·001 and ·002, according to the individual shape characteristics of each motor.

A value of ·0015 may be taken for the average circular section type of motor of some 6-7-inch diameter.

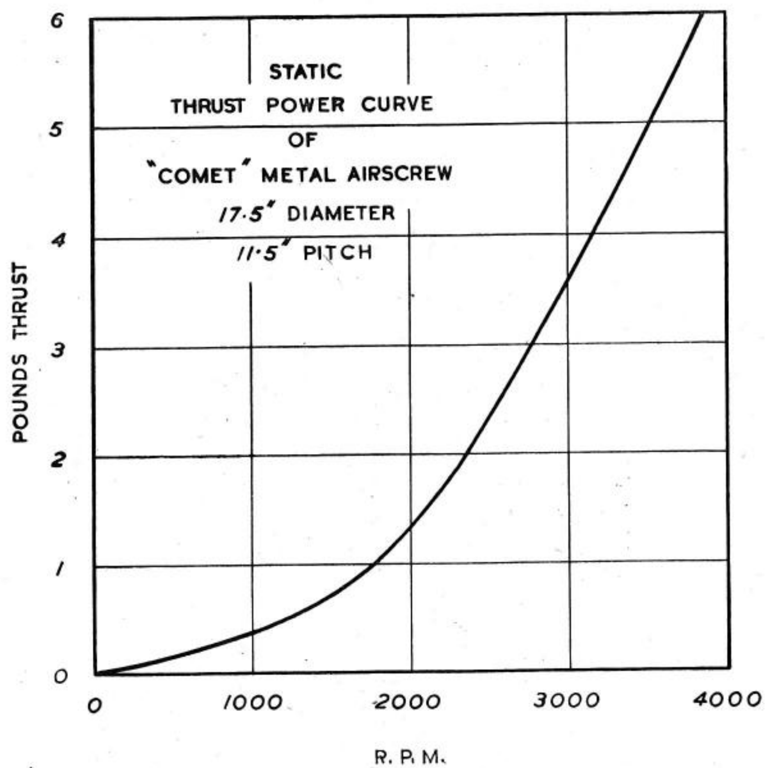


FIG. 26.

Fig. 26 shows an actual result obtained by this method of testing; using a motor 6-inch diameter, driving an airscrew of 17·5-inch diameter × 11·5-inch pitch.

The thrust delivered by this airscrew at 3,800 r.p.m. is seen to be 5·7 pounds, and an example may be taken of this figure to check up how it compares with the estimated performance as calculated by formula 24.

(1) 3,800 r.p.m. = 63·5 revolutions per second, which,

with a pitch of 11·5 inches (·96 foot) and an assumed efficiency of 70 per cent, gives a rate of actual forward advance of 42·7 feet per second.

$$\text{Therefore } J = \frac{42{\cdot}7}{63{\cdot}5 \times 1{\cdot}46} = {\cdot}46$$

The efficiency will more likely be about 67 per cent, indicating a flying speed of approximately 41 feet per second.

(2) Taking the value of K = ·0015 the drag of the motor may be calculated to be equal to

$$\begin{aligned} &{\cdot}0015 \times {\cdot}196 \times 28^2 \\ &= {\cdot}231 \text{ pound.} \end{aligned}$$

(3) Thus the actual static thrust delivered by the airscrew at 3,800 r.p.m. = 5·7 + ·231

= 5·931 pounds.

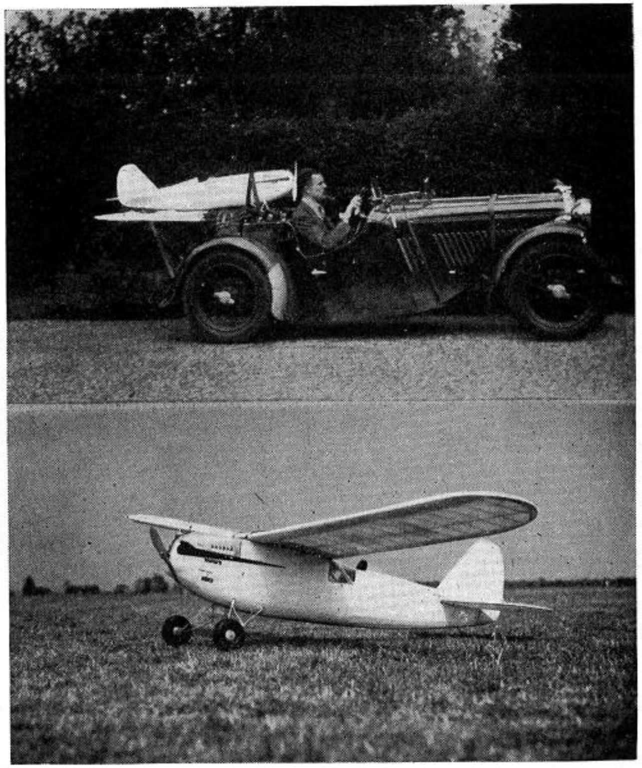
(4) Taking 83 per cent of this figure gives the actual thrust delivered during flight of 4·93 pounds.

(5) Using formula 24, the theoretical thrust delivered by the airscrew would be equal to

$$\begin{aligned} T &= 3{\cdot}142 \times {\cdot}73^2 \times {\cdot}96 \times 63{\cdot}5 \times {\cdot}076 \\ &= 7{\cdot}75 \text{ pounds.} \end{aligned}$$

(6) Taking 67 per cent of this figure gives 5·2 pounds as the actual thrust delivered during flight, which compares with the figure of 4·93 pounds ascertained by the test.

The conclusion is therefore justified that the airscrew is working at an efficiency of approximately 67 per cent when J = ·46, and that, at a slightly slower rate of revolutions, the airscrew would be operating under its maximum conditions of efficiency.



Petrol 'planes may easily be carried to and from the local flying field if a small bracket is fitted to the rear of a car!

CHAPTER X

WIND-TUNNEL TESTING

Description of a wind tunnel—Method of operation—The visual observation of airflow—The measurement of lift and drag—Examples of results obtained from wind-tunnel tests.

WITH power-driven model aircraft flying at speeds of from 15-40 m.p.h., the effect of wind resistance can no longer be ignored, and to obtain the best results it is essential that proper attention be paid to " streamlining," and the reducing of drag and " interference " to a minimum.

As in full-size practice, so in the sphere of model aeronautics the wind tunnel provides the means of making tests and observations of parts of an aircraft, under similar conditions to those which operate during flight.

A wind tunnel consists esentially of a large tube, in which is suitably suspended the object about which it is desired to

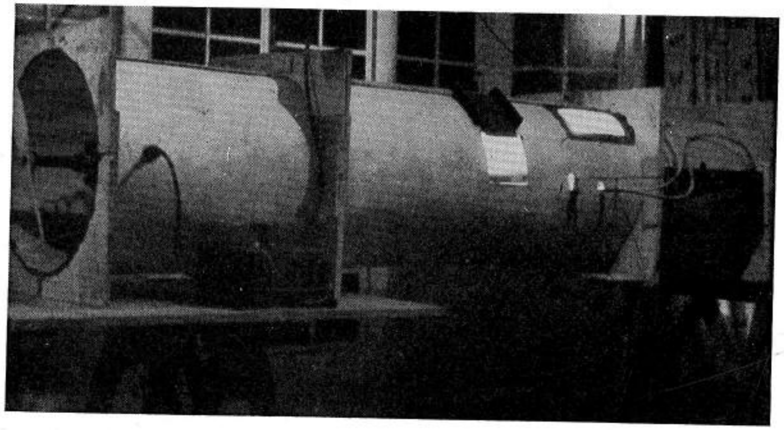


FIG. 27.

In the middle of the centre section is an opening, through which units to be tested can be inserted into the tunnel. To the right of the flaps is one of the two celluloid-covered inspection windows. On the table in the foreground is the rheostat for controlling the speed of the fan motor.

obtain information, and a means of providing a " wind," or flow of air, past the object at a speed similar to that at which it would move when passing through the air when in flight.

Fig. 27 shows a general view (less the section at the inlet end) of the wind tunnel constructed by the author and used in his research work. It is 10 feet long by 20 inches bore, and is divided into 4 sections for convenience in storing. There are one 4-foot section and three 2-foot sections, the flanges of each section being concentric and held together by short screws.

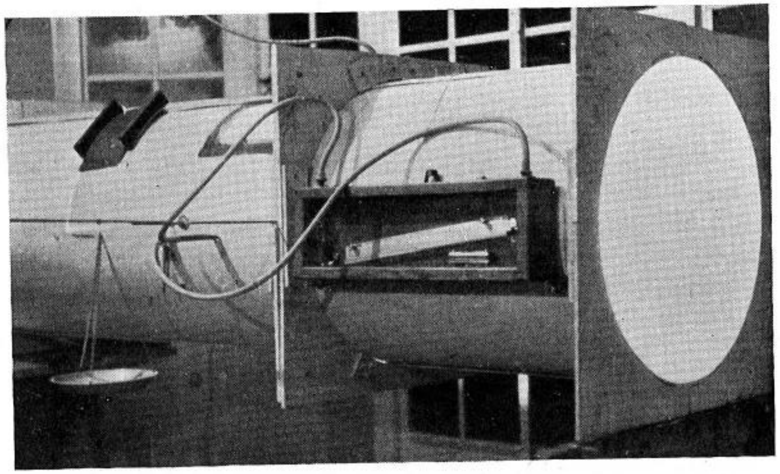


FIG. 28.

In the centre of the photograph is shown the manometer for measuring the air pressure. The rubber tubes lead to the pilot tube which is inserted in the side of the tunnel.

The tube forming the tunnel is made of $\frac{3}{16}$-inch three-ply—each section of " tube " being inserted into a pair of flanges made from $\frac{3}{8}$-inch three-ply—these being each 2 feet square.

Fig. 28 shows the balance arm and the manometer at the discharge end. Vanes are arranged inside the tunnel, in both vertical and horizontal planes, to prevent rotation of the column of air as it passes through.

Celluloid " windows " are provided for observations of parts under test, and electric light is installed inside the tunnel to enable photographs to be taken.

The motor used will deliver up to $1\frac{1}{2}$ h.p. and, driving an 18-inch diameter airscrew, will produce airspeeds up to 30

m.p.h. in the tunnel, a regulator being provided to enable any speed below this figure to be obtained.

The *exact* efficiency of the airscrew is known over its full working range, and the calculated airspeeds are checked by means of a pitot tube. The tunnel has also been calibrated by means of readings with an anemometer.

Broadly speaking, the information which may be obtained from wind-tunnel testing is of two kinds—that obtained by direct observation of the flow of air around the object being tested, and that obtained by means of direct readings with the aid of a balance or other mechanical apparatus.

Direct observation of the flow of air is made possible by attaching very thin " streamers," consisting of 3—5-inch lengths of wool, to the object to be tested. By this means not only the direction of the airflow may be observed, but also its condition as regards stability and tendency to form vortices or " whirls." If the airflow is steady the pieces of wool will remain stable or " rigid "; whereas if the airflow is uneven the ends of the " streamers " will waver.

Figs. 29, 30 and 31 show an airfoil section being tested in the tunnel, the " breakaway " as the stalling angle is approached, being indicated by the spreading of the wool streamers. Figs. 32, 33, and 34 are views looking up the tunnel taken during the same test.

Measurements of drag are obtained by means of a balance, the actual value of the drag being obtained direct.

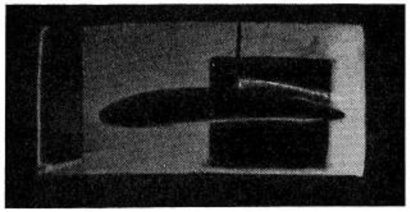
Supported on a pivot fixed to the outside of the tunnel is a horizontally-mounted " balance arm," which passes through a clearance hole in the side of the tunnel. This arm carries the test object at one end (inside the tunnel), and a suitable counterbalance weight at the other end (outside the tunnel)—this consisting of a scale pan in which are placed the necessary counterweights, so keeping the " balance arm " horizontal.

At the outer end of the " balance arm " also is fixed a length of fine thread which is led over a pulley mounted on a spindle which projects horizontally, and at the same level as the " balance arm," from the side of the tunnel. To the other end of this thread weights are attached to balance the drag of the test object.

The pivot rod is mounted on a thrust ball-bearing which

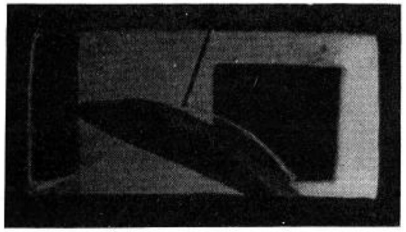
(a) In Figs. 29, 30 and 31, end views of the airfoil sections are shown photographed through the inspection window described in Fig. 27. Figs. 32, 33 and 34 are photographs which were taken looking up the tunnel from the outlet end.

FIG. 29.



The airfoil section is normal to the air stream, and the white streamers can be seen following the curve of the upper portion of the airfoil.

FIG. 30



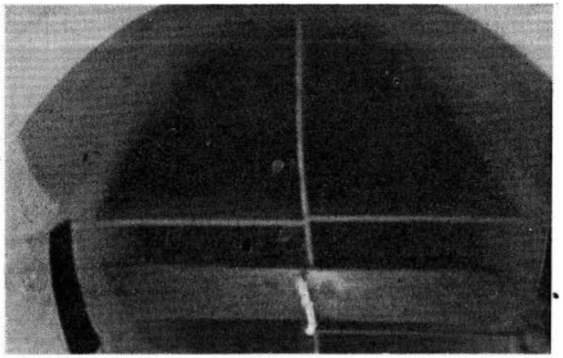
The airfoil has now been tilted and the streamers are starting to lift away from the top surface.

FIG. 31.

The airfoil has now been tilted until it is stalled, and the streamers are spreading out due to the "breakaway" of the air stream.

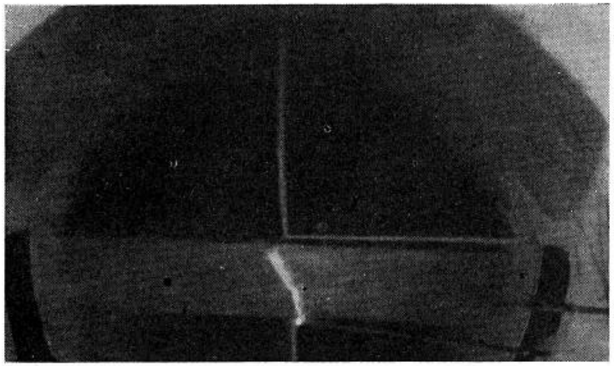
Airfoil Section R.A.F. 32. 10-inch chord.
Wind speed approx. 30 m.p.h.
(all six photographs taken by J. C. Eck, Esq.)

FIG. 32.



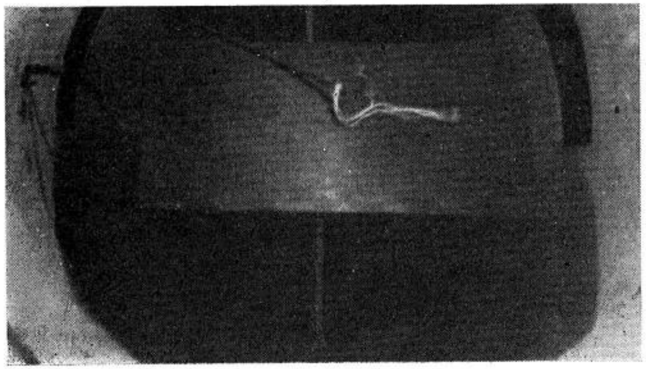
There are several streamers of wool, but in the normal position they are lying close together.

FIG. 33.



With the tilt of the airfoil approaching stalling point, the wool streamers are starting to waver.

FIG. 34.



With the airfoil in the stalled position the air flow has broken down, and the streamers are flowing about all over the top surface.

takes the weight of the test object and its counter-balance, and the " balance arm " itself is mounted on a universal bearing which enables it to move in any direction.

The whole apparatus is sufficiently sensitive and friction free for a weight of $\frac{1}{10}$ ounce, at the end of the fine thread, to swing the " balance arm " round.

Fig. 35 shows a pneumatic-tyred wheel mounted inside the tunnel ready for testing; the counter-balance, as a matter of convenience, being a similar wheel. The arrangement of the mounting of the " balance arm " on the pivot may be

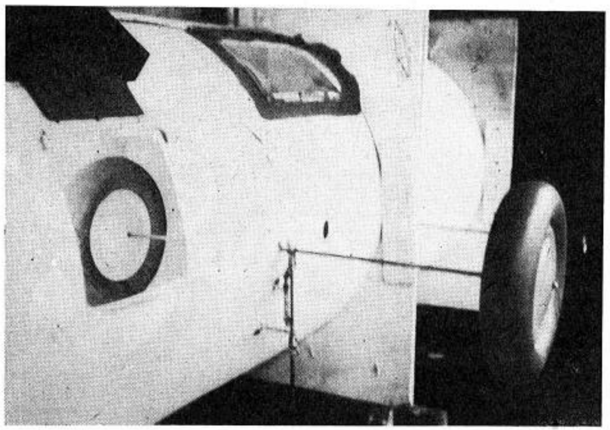


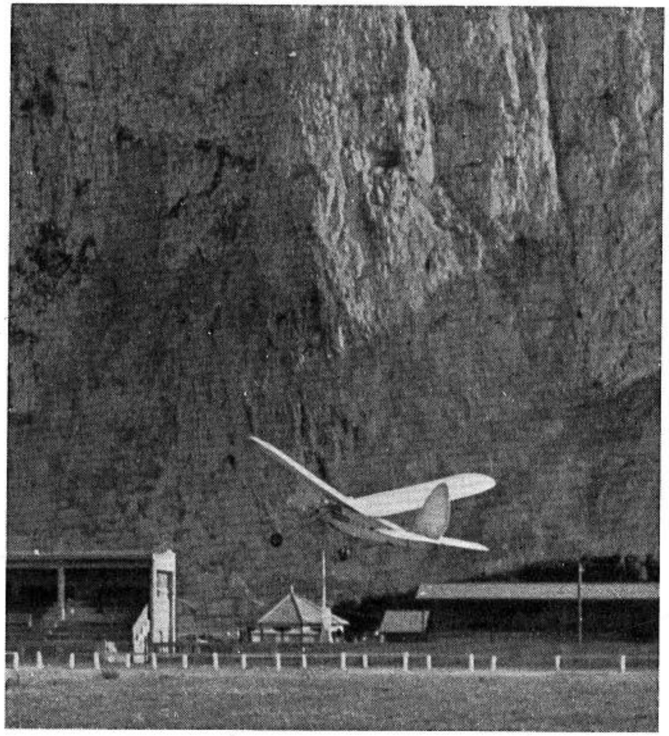
FIG. 35.

Here is shown one of the 8 in. diameter pneumatic power wheels described in another chapter, mounted inside the tunnel at one end of the balance arm, with its mate at the other end to statically balance it. The supporting bracket at the side of the tunnel carries a ball-bearing universal mounting, on which the balance arm can move.

noted, as also the spindle and pulley (above and to the right of the wheel used as the counter-balance), over which the thread is passed, and to which are attached the weights balancing the drag of the test object.

To carry out a test, the motor is run up to the desired speed and, due to the drag, the wheel in the tunnel swings " downstream." Weights are then attached to the thread, and are so adjusted that when the " balance arm " is moved back and is placed exactly at right-angles to the axis of the tunnel, it stays there.

It will be appreciated that as soon as the " balance arm " swings the smallest distance away from this position, the wheel is no longer lying with its diameter coincident with the axis of the tunnel, but is tending to lie " across " it. Thus a greater area is presented to the airstream, and consequently it swings still more obliquely across the tunnel. It will thus be seen that unless the wheel is positioned with its diameter *exactly* coincident with the axis of the tunnel (and it can only remain so if the weight adjustment to balance the drag is exact) the " balance arm " will not " stay put " when the operator releases it, but will swing either up- or downstream.



This photo shows one of Lt.-Col. Bowden's 'planes just after taking-off from the sand at the foot of the Rock of Gibraltar.

To measure the lift and drag of an airfoil, a section of suitable size is fixed at its centre of lift, and at the desired angle of incidence, to the inner end of the " balance arm ";

counter-balance weights being placed in the scale pan at the outer end so as to balance the arm horizontally.

When the desired airspeed has been obtained in the tunnel, weights necessary to approximately balance the drag are attached to the end of the thread passing over the pulley.

During this operation, the " balance arm," which due to the lift has been trying to rise inside the tunnel, has been prevented from doing so by a light touch from the operator. As soon as the drag has been roughly balanced, weights are *removed* from the scale pan, until the " balance arm " will remain in a horizontal position.

Readjustments are then made to the drag balance weights, and, if necessary, to the lift counter-balance weights, until the test piece will remain exactly horizontal, and with the balance arm at right-angles to the axis of the tunnel; when the drag is as indicated by the weights attached to the thread, and the lift is as indicated by the value of the weights *removed* from the scale **pan.**

A fine wire, stretched horizontally across, and at right-angles to the axis of, the tunnel, just in front of the leading-edge, serves as an excellent guide for correctly aligning the position of the airfoil section in relation to the airstream.

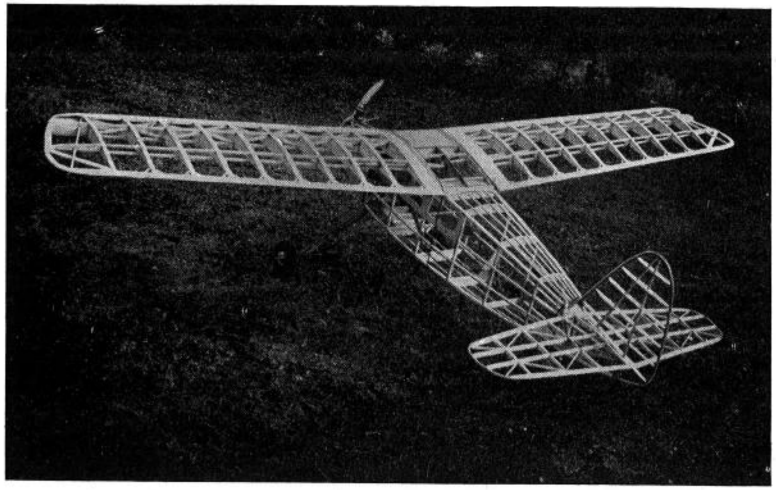
Below is given a set of readings (typical of many) obtained with this apparatus. The values are for drag of a wheel fitted with an 8-inch diameter × 2-inch wide pneumatic tyre—actually the wheel shown in the tunnel in Fig. 35.

Airspeed	Position of wheel			
Miles per Hour	Normal to wind Projected area = ·105 sq. ft. Ounces drag	K =	At right-angles to wind Projected area = ·345 sq. ft. Ounces drag	K =
17·3	·571	·00113	3·219	·00193
19·2	·781	·00126	4·395	·00215
22·6	1·020	·00119	5·895	·00205
	Average =	·0012	Average =	·002

The actual drag of many small model aircraft parts is not of great amount, and it might be thought that several parts, with drag values as small as ·5 ounce, could not much matter on a machine powered perhaps with an engine giving 2 or 3 pounds thrust. Apart, however, from the fact that these small amounts have an awkward habit of " adding up " to a total higher than at first might be expected, the losses in aerodynamic efficiency due to turbulence and " interference " may be of a much greater value than the actual drag itself.

For example, the drag of a pair of wheels might be 2 ounces, and the drag of the landing-chassis struts 1 ounce, and the drag of the fuselage 6 ounces; each figure being ascertained with the unit tested separately, but of course at the same speed in the tunnel.

When, however, these parts are all assembled together, the measured drag will exceed, sometimes by a large margin, the sum total of their respective drag values—in the case of this example, 9 ounces—due to the high degree of turbulence and mutual interference set up. The value of a wind tunnel, as a means of testing out various arrangements and assemblies, will therefore be appreciated, in addition to its normal use as an apparatus for measuring the lift and drag of aircraft components.



This photograph is of the Author's 1939 high-wing cabin petrol 'plane. Span is 8 ft. and overall length 4 ft. 6 in. The engine is a 9 cc. " Dennymite."

CHAPTER XI

WING CONSTRUCTION

A suitable assembling board—Method of simple wing construction—Detailed method of large wing construction—The pressure distribution over the surface of an airfoil—Detailed method of stressed skin construction.

It is essential when building wings of *any* size that the work shall be carried out on a perfectly level board, so that accuracy of construction and uniformity of shape may be assured. Such a board should be at least 6 inches longer than the wing (or half-wing as the case may be) and at least 6 inches wider than the greatest chord. It should not be less than $\frac{3}{4}$-inch thick, and should be well battened at each end and in the middle. When finished it should be dead level for the full length and breadth.

A piece of wood 1-inch square, the same length as the board, and dead straight, should be screwed down to the board to serve as a guide up against which the leading edge of the wing may be held during construction.

Fig. 36 shows a typical assembling board as used for a half-wing some 5 feet long.

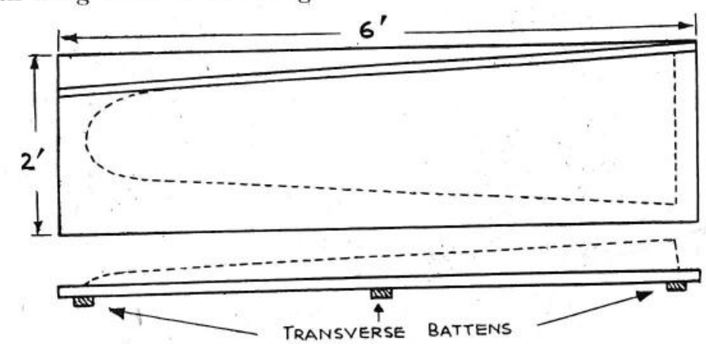


Fig. 36.

The simplest type of wing construction is that shown in Fig. 37, in which ribs of three-ply or balsa are spaced at fairly large intervals, and have a number of longitudinals set into notches cut as shown.

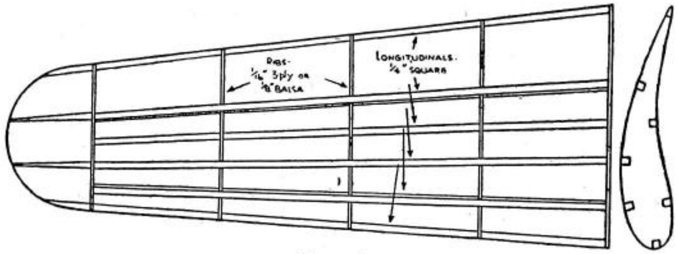


FIG. 37.

This design, whilst simple and easily and quickly built up, should not be used for wings of above about 5 feet span, as it allows of comparatively large areas being unsupported, and the tautness of the fabric alone is relied upon to keep its surface to the required shape; also the type of construction does not lend itself to large spans due to the absence of a vertical "backbone" running down the length of the span, without which the wing will tend to "droop."

For spans above 5 feet the wings should be built in halves, and either fixed into the sides of the fuselage or joined together, according to the design of the aircraft, by means of dowel rods of birch passing into each wing. These rods should not be glued in position, but should slide into a series of holes cut in the first 3 or 4 ribs, as shown in Fig. 38. Thus, in the

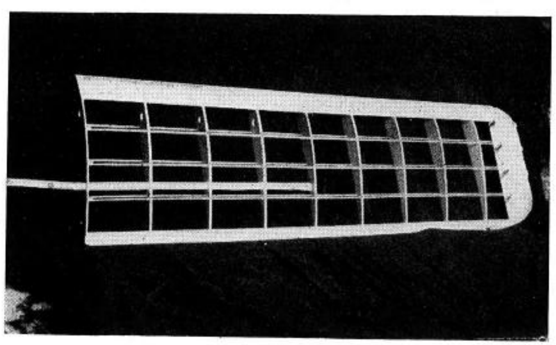


FIG. 38.

event of a bad crash, if a rod should break, the broken piece may easily be removed from the tube and a fresh rod inserted.

Two rods should always be used, and they should be of

$\frac{3}{8}$-inch or $\frac{1}{2}$-inch diameter, according to the size of the wing. Not be less than 15 inches long for a 3—4-foot half-wing, and about 20 inches by $\frac{1}{2}$-inch diameter for wings above 8-foot span.

There is one disadvantage to the method of fixing the two half-wings direct into the side of the fuselage, and that is that the angle of incidence cannot too easily be altered. If the machine has been carefully designed no great variation should be necessary. However, to allow of major adjustment being made in the angle of incidence, the fuselage must be constructed in such a way that the wing unit rests on the top, and it is best in this case to construct the centre section the width of the fuselage, into which the two half-wings are jointed. This method of construction is shown in Fig. 39.

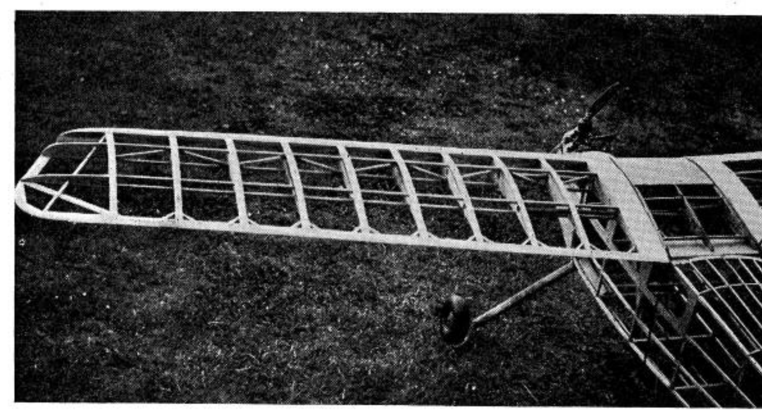


FIG. 39.

The centre section is held to the fuselage by rubber bands, and the two half-wings are joined to it by the rods, as already described. These can be seen in the photograph projecting into the wing through the first three ribs.

The construction of this type of wing is of interest. It is entirely of balsa, and each half-wing is 3 feet 8 inches long with a chord at the root of 14 inches, and the area is just under 4 square feet. Weight of each half-wing covered and doped is 10 ounces. The ribs are built from $\frac{1}{8}$-inch balsa, overlaid with strips of balsa $\frac{1}{2}$-inch wide and $\frac{1}{32}$-inch thick. These protect the edges of the ribs and also the silk stretched over them. (See

Fig. 40). The leading edge is covered with $\frac{1}{16}$-inch balsa. This construction allows the wing unit to be shifted backwards and forwards, and also the angle of incidence to be varied. In a severe crash the whole unit can be knocked off without either it or the fuselage suffering a great amount of damage.

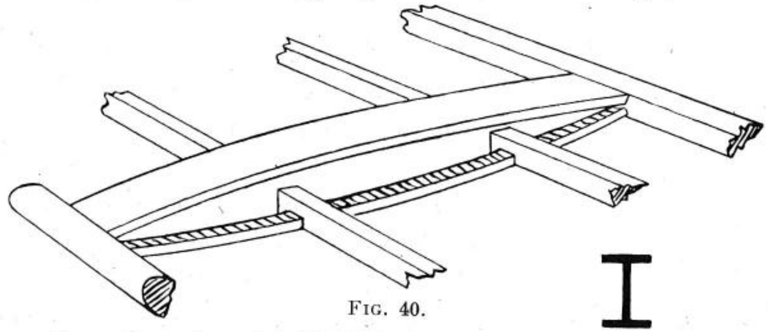


FIG. 40.

For wings of models of 10 feet span and over, it is essential that the design allows for the incorporation of a "backbone"—in effect a solid vertical panel—running throughout the length of the span at its deepest section. The construction recommended is that in which a number of wing-ribs are linked together by a number of longitudinals; these latter consisting of relatively thin, but fairly deep sections, arranged vertically in the ribs in such a manner that the top and bottom pair at the deepest section may be joined by a series of panels of three-ply, inserted in between each wing-rib—thus forming a vertical panel or "backbone" running throughout the length of the span.

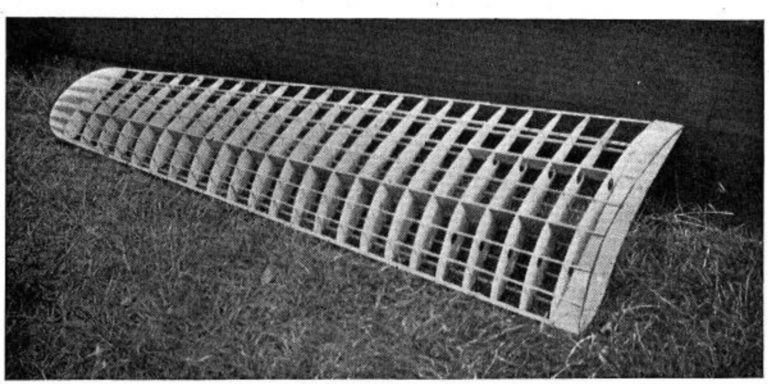


FIG. 41.

Fig. 41 shows an example of this method of construction, the half-wing being 4 feet 6 inches long by 18-inch chord at the widest part.

The series of holes for the dowel "tubes" may be clearly seen, as also the panels between each wing-rib, forming the "backbone."

This type of wing is immensely strong, permits of assembly in a reasonable length of time, and leaves no part of the wing covering unsupported for more than 2 or 3 inches in either direction. The ribs may be of $\frac{1}{32}$-inch three-ply or $\frac{1}{16}$-inch balsa, and the longitudinals of $\frac{3}{16}$-inch × $\frac{1}{16}$-inch birch, with a ileading edge of $\frac{1}{4}$-inch × $\frac{1}{8}$-inch and a trailing edge of $\frac{3}{8}$-inch × $\frac{1}{8}$-inch.

The method of construction is as follows:—

First, the wing-ribs should all be cut out, each one smaller than its neighbour by the same amount, depending on the degree of taper given to the wing. They should then be assembled as shown in Fig. 42, whereby the lines of centre of lift are all superimposed and the top edges are all flush; a saw-cut is then made to accommodate one of the longitudinals.

Next, while the lines of centre of lift are still superimposed, the bottom edges of all the ribs are brought flush and another saw-cut made. Similarly two more cuts are made in the leading and trailing edges—the work now being as shown in Fig. 43.

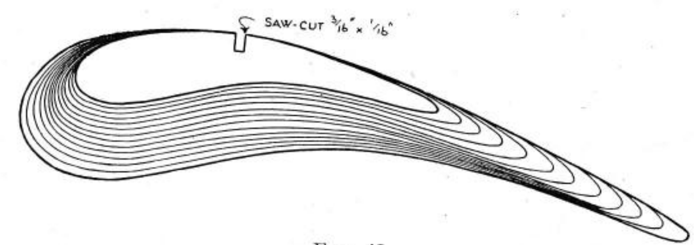


FIG. 42.

Having now decided on the number of intermediate longitudinal spars to be inserted, these positions should be marked off on the largest and smallest ribs, A1, A2, A3, etc. (See Fig. 44).

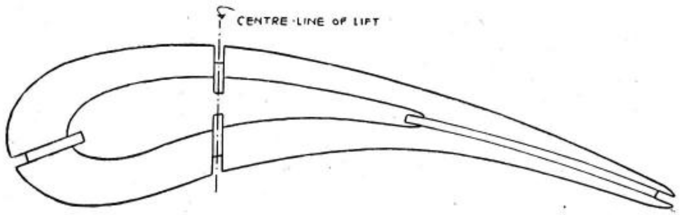


Fig. 43.

If the ribs are now assembled as originally—the points A.1, A1 : A.2, A2 : etc., can be joined up—thus all the intermediate ribs are marked in their correct places. When cutting these remaining saw-cuts, care must be taken that all edges, at the point of the cut, are flush to ensure the same depth.

This work completed, the assembly of the wing may be proceeded with. The piece of 1-inch square wood is screwed to the large board to give the required angle of " rake " which the wing design calls for, and the ribs are set out at 2-inch intervals. They are held upright and equally spaced by suitable pieces of wood provided for the purpose, as shown in Fig. 45. The top longitudinals are inserted first, then the

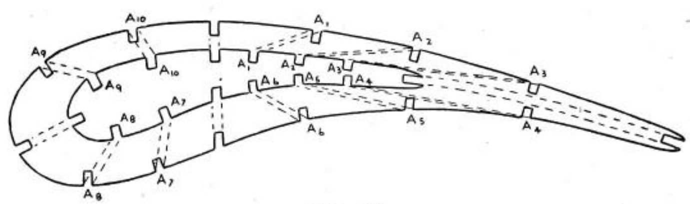


Fig. 44.

wing turned over, and the bottom one inserted in the saw-cuts. These should be such that the longitudinals are a nice tight push-fit in the notches cut in the ribs.

All joints should be glued, but this is best done after *all* the longitudinals have been inserted and the wood blocks removed. The longitudinals may then be withdrawn one by one, dabbed with glue and reinserted.

Pieces of $\frac{1}{32}$-inch three-ply should now be nailed between each rib, joining the top and bottom longitudinals which are over the line of centre of lift, as shown in Fig. 46.

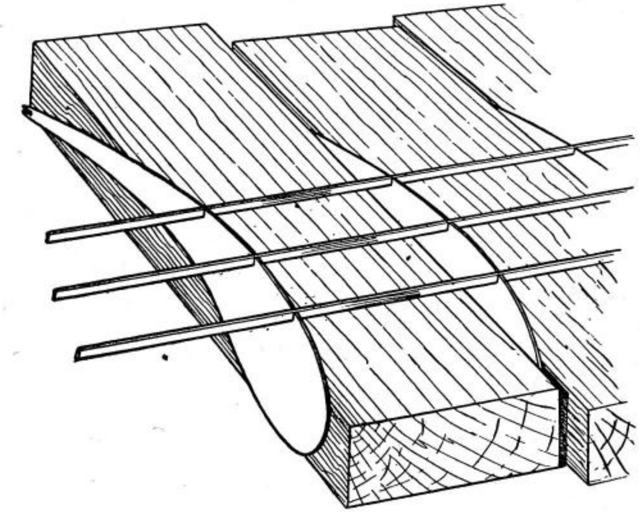


Fig. 45.

Thus are formed the rigidly-braced girders, which run the whole length of the wings, and the cantilevers (formed by the $\frac{1}{32}$-inch three-ply ribs) which project from each side. On these cantilevers are carried the intermediate longitudinals, which thus form an extremely strong framework on which the fabric, when stretched, is at no place unsupported for more than $2\frac{1}{2}$ inches in either direction.

To prevent any possible warping while the glue is drying,

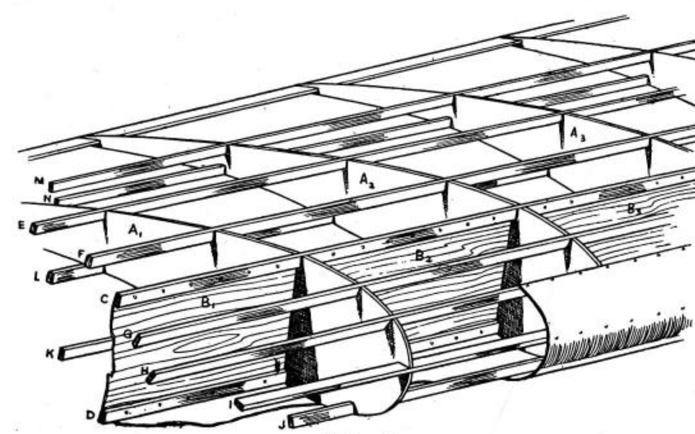


Fig. 46.

long pieces of perfectly straight wood may be laid along the top of the wing.

Finally, three-ply is soaked in very hot water, and bent to fit round the leading edge of each rib. If glue is smeared over the edge of each rib, and the three-ply laid over it and held in position by a length of elastic tied round and round the three-ply as shown in Fig. 47, it will be found that when dry it has firmly stuck to the ribs.

When covering the wing the under-surface should be fixed first. The fabric should be sewn up to the underside of each rib for its full length—a tedious job, but very necessary if the proper shape is to be maintained. The upper surface may then be fixed in place and the wing doped and painted. During construction the wing should be kept as much as is possible on the board and held in position by long lengths of wood. When both half-wings have been completed they should always be held together, under-surface to under-surface, by elastic bands, and stored in a vertical position.

The question of dihedral angle between the two half-wings is best dealt with by setting the first rib at the required angle from the vertical, so that when the two half-wings are joined these two ribs will be flush together.

It is very important that the leading edge of a wing should be given a perfectly smooth surface, and that its shape should be uniform throughout the span; and this can only be done by covering the front portion with thin three-ply or balsa.

Fig. 48 gives a general idea of the pressure distribution over the upper and lower surfaces of a wing, from which it will be seen that the highest pressure occurs at the nose, and

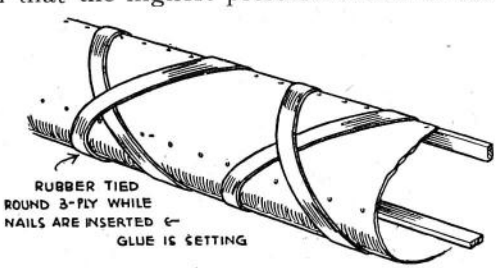


FIG. 47.

at the front of the upper surface—hence the reason for recommending that this portion should always be covered with

$\frac{1}{32}$-inch three-ply or balsa, so as to ensure that the profile is symmetrical and regular throughout the span of the wing.

Another method of construction best applied to large wings, is that in which the wing covering is of $\frac{1}{32}$-inch balsa, overlaid with silk, and finally painted. If the ribs are made of $\frac{1}{16}$-inch or $\frac{1}{8}$-inch balsa, and spaced not more than 3 inches apart, large half-wings of 5 or 6 feet length may be satisfactorily built by this method.

Steps in the construction are as follows:—

First the complete set of ribs is cut out in the usual way—and three sets of saw-cuts made to accommodate 3 longitudinals of $\frac{3}{16}$-inch × $\frac{3}{16}$-inch birch; cuts are also made for the leading and trailing edges as shown in Fig. 49. These and

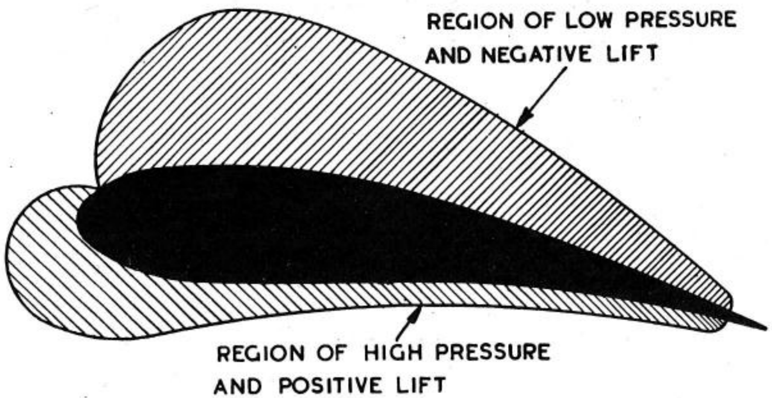


FIG. 48.

the 3 longitudinals are then glued in positon, and the framework allowed to set quite dry—long pieces of wood being laid along the wing to keep it perfectly flat on the assembling board.

Panels of $\frac{1}{32}$-inch three-ply are then nailed and glued between each wing-rib, forming the triangular-shaped backbone as shown in Fig. 50.

FIG. 49.

The top covering of $\frac{1}{32}$-inch balsa is applied first, as this can be done with the wing lying flat on the assembling board; strips of balsa, as wide as may be obtained, are laid up edge-to-edge and parallel to the leading edge; and are glued direct on to the wing-ribs, after which the wing is allowed to dry out with suitable weights laid along the full length so as to ensure that the covering is glued to the full chord of the ribs. After which the wing is turned over and the underside covered, suitable weights being laid on long strips of wood so as to ensure that the balsa conforms to the concave camber of the wing section.

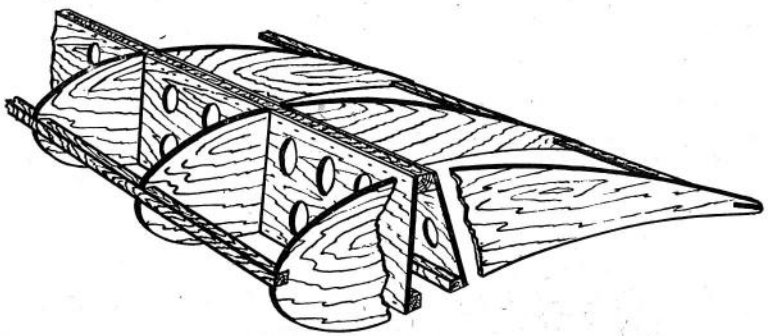


FIG. 50.

Finally, the wing is carefully rubbed down with very fine-grade sandpaper, after which it is covered with silk affixed with photo paste—no stretching is necessary—the material being slightly damped and then laid smoothly over the balsa —the whole of which has been previously smeared with a thin layer of photo paste. Dope must *not* be applied, as it will shrink the wood as well as the silk.

This " stressed skin " method of wing construction has the great advantage that a very rigid wing is formed with a good degree of resistance to torsion—this ensuring that the trailing edge at the tips will not " droop," a common fault found in many large wings.

Whilst quick-drying cement may be used in the building up of a small wing, it should not be used in large wing construction, as, due to its quick-setting properties, one part of the wing will become quite rigid before another part is completed, and as not all of the assembling can be carried out

with the wing actually on the board, there is a risk that it may not dry out perfectly flat.

Instead, one of the several proprietary brands of glue

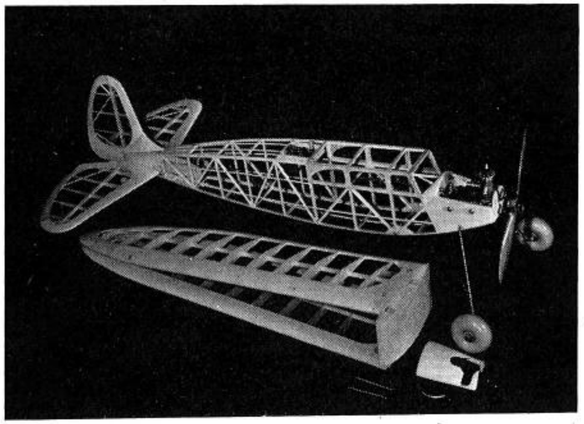
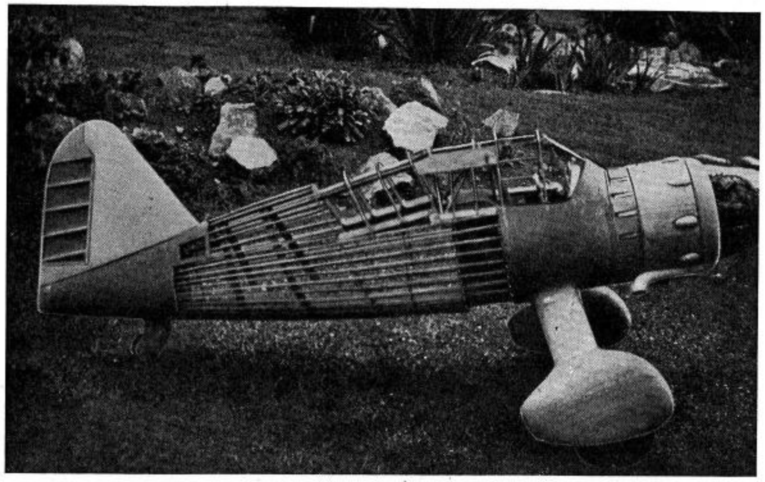


FIG. 51.

Here is an interesting method of construction which makes for easy assembly. Note how the main wing is in two halves hinged to a centre section. The tail assembly is shown in greater detail in Fig. 69.

should be used, as with these a certain degree of flexibility exists for the first 2 or 3 hours; so that, when the wing is completely assembled, any small twists or warps will " fade out " during the subsequent drying and hardening of the glue if weights are placed on the " high " parts.



This photograph shows the fuselage of the Author's 10 ft. span "Lysander." The overall length is 6 ft. The spats are some 14 in. long. Three-ply $\frac{1}{32}$-inch thick is used to represent metal sheeting on the full-size aircraft.

CHAPTER XII

FUSELAGE CONSTRUCTION

Stresses in fuselages—" Compression " struts for use in fuselages—Methods of constructing fuselages of (*a*) rectangular section, (*b*) circular section—Methods of constructing fuselages with " moulded " lines, (*c*) with stressed skin of thin three-ply, (*b*) with laminations of thin balsa, and with stringers.

THE fuselage of a model aircraft, whilst serving no useful *flying* purpose, since it produces no lift, *does* provide a means of linking up the various component parts. Additionally, in the case of the rubber-driven motor, the fuselage has to serve as a " stretcher " for the rubber; a factor which calls for quite different considerations in construction when comparison is made with a fuselage for a petrol-engined aircraft.

A fuselage which is to accommodate a rubber motor will *always* be in compression, whilst the petrol-engined aircraft, strictly speaking, will be in tension; since the motor is pulling at the front, whilst the air resistance offered by the stabiliser and fin is acting as a drag at the rear.

From the point of view of landing strains, both types of fuselage are, of course, subjected to the same kinds of stresses, which tend in the main to break the back of the fuselage as the wheels touch the ground.

Since the longerons are in compression—due to the tension of the rubber motor—the top and bottom tie-bars are in tension, and thus may be flat and of comparatively thin section. The side tie-bars are also in tension *during flight*, but in landing, those forming the anchorage for the rear legs of the landing chassis are in compression—sometimes very much so! —and therefore a considerably stronger section must be used.

A suitable design for " compression " struts is that in which an angle is made up from 2 pieces of birch or spruce, say $\frac{1}{4}$-inch × $\frac{1}{16}$-inch and $\frac{3}{16}$-inch × $\frac{1}{16}$-inch. These should be pinned at intervals of about $1\frac{1}{2}$ inches, after having been thinly coated with glue, and laid up together at right-angles.

Fig. 53 shows this method of construction applied to a large fuselage powered by a multi-size spindle, rubber-driven motor, the distance between hooks being 36 inches, and the maximum cross-section of the fuselage being 10 inches high by $4\frac{1}{2}$ inches wide. The longerons are $\frac{3}{16}$-inch × $\frac{3}{16}$-inch, angle struts at the front built from $\frac{1}{4}$-inch × $\frac{1}{16}$-inch and $\frac{3}{16}$-inch × $\frac{1}{16}$-inch, and those at the back from $\frac{3}{16}$-inch × 1-24 inch and $\frac{1}{8}$-inch × 1-24 inch.

It should be noted how the landing stresses are distributed in several directions from the point where the rear chassis strut is attached to the fuselage. The weight of the fuselage proper, i.e., excluding landing gear and rudder, was 7 ounces.

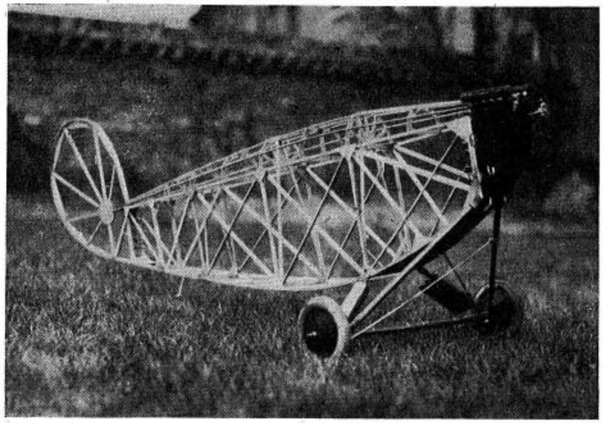


FIG. 53.

It will be noticed that three-ply formers, as often used in small rubber-driven models, have not been used, the reason being that as the longerons are in tension (and therefore trying to " bow outwards ") they could not properly be anchored to the former—whereas a "lapped" joint, braced top and bottom by fishplates, is so strong as to be practically indestructible.

The fuselage of the petrol-engined machine has to contend with a somewhat different set of conditions. It is part of a machine weighing up to 10 or 12 lb., and thus must withstand landing shocks of some considerable magnitude. It also has to accommodate the coil and battery somewhere in its " inside," as well as the engine at its extreme front.

If the fuselage is to be of rectangular cross-section, the principles of construction as illustrated in Fig. 53 should be used—suitable modifications, of course, being made according to the type of aircraft.

In constructing a fuselage which is of circular cross-section, three-ply formers, suitably lightened are, of course, ideal for supporting the longerons. For large machines the formers should not be less than $\frac{1}{16}$-inch thick at the rear and $\frac{1}{8}$-inch at the front.

From the point of view of appearance, the greater the number of longerons the better, as they naturally allow of the truly circular cross-section being retained between the three-ply formers. The disposition of these latter will naturally depend on the positions of the battery, coil, wing, and landing-gear attachments—as the formers should, of course, be so positioned that they provide suitable points of anchorage for the above-mentioned components.

These closely-spaced longerons should be about $\frac{3}{16}$-inch × $\frac{1}{16}$-inch, but the two to which the struts of the landing chassis are attached should be $\frac{3}{16}$-inch × $\frac{3}{16}$-inch. If the aircraft is a high-wing machine, then two of the top longerons should also be of this heavier size.

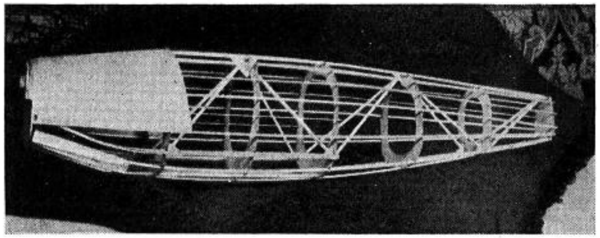


FIG. 54.

Fig. 54 shows a large fuselage built to the above specification—the greatest diameter is 11 inches and the overall length is 5 feet. The 2 longerons to which the landing-chassis struts are anchored consist each of two lengths of $\frac{1}{4}$-inch × $\frac{1}{8}$-inch birch, spaced $\frac{1}{8}$-inch apart. The longeron at the top and 2 main ones at each side are of $\frac{3}{16}$-inch × $\frac{3}{16}$-inch, whilst the remainder are of $\frac{3}{16}$-inch × $\frac{1}{16}$-inch—the material in each case being birch. All the bulkheads are of $\frac{1}{8}$-inch three-ply. This fuselage was built for a high-wing 10-foot span monoplane, and as shown (but, of course, less the landing chassis and engine) weighed $13\frac{1}{2}$ ounces.

When it is desired to construct a fuselage with fully " moulded " lines—i.e. in which truly curved surfaces, and radiused " flarings " of the wing roots into the fuselage, are designed, the whole fuselage may be covered with $\frac{1}{32}$-inch three-ply, or $\frac{1}{16}$-inch balsa.

An example of this type of construction is shown in Fig. 55, which shows the partly-finished fuselage of the 10-foot span low-wing monoplane, described in the last chapter of this book. The overall length is 4 feet 6 inches, and the largest diameter is of nearly circular section—actually 11 inches deep by 10 inches wide.

The framework consists of 8—$\frac{3}{16}$-inch × $\frac{3}{16}$-inch birch longerons, let into notches cut in a series of $\frac{1}{16}$-inch three-ply bulkheads, the whole fuselage then being covered with $\frac{1}{32}$-inch three-plybirch, laid up in sections, after steaming and curving to shape. Wherever joints in the sections occur, the edges are butted, and strips of $\frac{1}{32}$-inch three-ply laid along the joint, and glued and nailed up in position on the inside of the fuselage. Thus the outer surface is quite smooth.

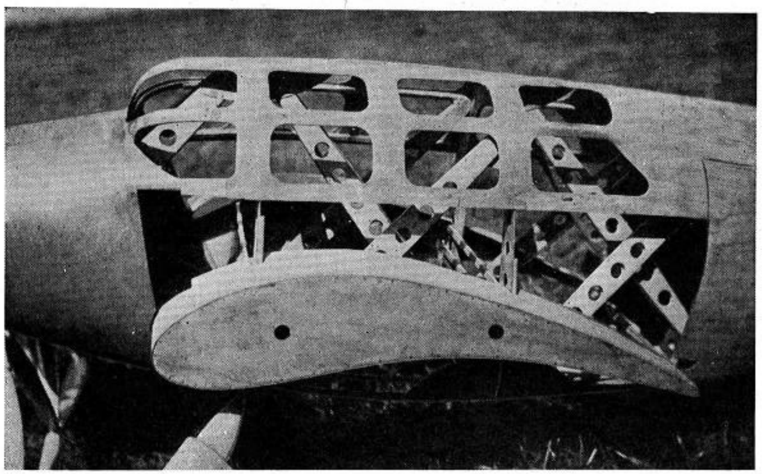


FIG. 55.

Fig. 56 shows how the " flaring " of the wing roots into the fuselage is carried out—this being done by forming a series of " petals " in a sheet of $\frac{1}{32}$-inch three-ply, and steaming up to shape.

The whole secret of making a satisfactory job lies in the fact that the cuts in the three-ply are made in such a manner that they extend past and in between each other as shown in the illustration.

It will also be noted that the sharper the angle, the more close together are they positioned. The cuts are made with a pair of scissors. The three-ply is well soaked in very hot water, glued on the underside, and then carefully formed in position. It will be noted that when coming round a convex surface, as at the front of the wing, the arranging of the three-ply leaves triangular gaps between each section, and these must afterwards be fitted with carefully-cut wedge-shaped pieces.

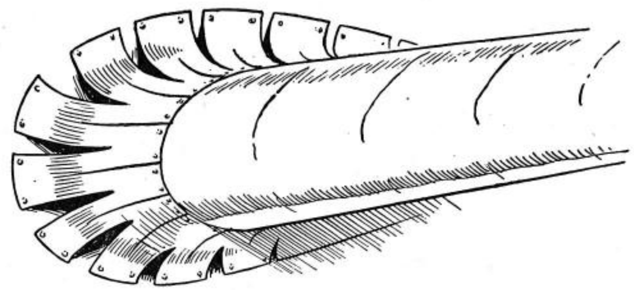


FIG. 56.

When coming round a concave surface, as in the case of the underside of the wing, the sections will overlap, and the overlapping pieces must be carefully cut away.

When dry, plastic wood may be smeared along the joins to fill up tiny cracks and the whole " flare " well sandpapered down.

The photo on page 80 shows the " flaring " completed. This fuselage was finally covered with silk and given several coats of paint—the resultant surface being quite smooth.

The weight, complete with suitable steel fish-plate anchorages for the landing chassis and nose-block for the engine mounting; also suitable supports for the coil and batteries, was 4 pounds 3 ounces; and the strength was such that, when stood on its nose, it would support the weight of a man without showing the slightest sign of breakage.

Compression struts for use in large fuselages may be made in the following way—the type of strut being as shown in Fig. 57.

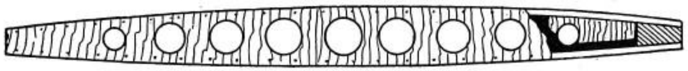


FIG. 57.

From a sheet of $\frac{1}{32}$-inch three-ply, two identical pieces are cut and lightened with suitable holes punched out with the type of punch used for cutting leather washers. These two pieces are separated by 2 pieces of $\frac{1}{8}$-inch × $\frac{1}{8}$-inch, curved to follow the edges of the pieces of three-ply. The maximum width should be $\frac{1}{12}$ of the length, and the width at each end should be $\frac{1}{24}$ of the length, i.e., a 6-inch long strut will be $\frac{1}{2}$-inch wide at the middle and $\frac{1}{4}$-inch at the ends, where the $\frac{1}{8}$-inch × $\frac{1}{8}$-inch ribs touch. For sizes above 6 inches long there will be a gap between the ribs at the ends which must be filled with a piece of $\frac{1}{8}$-inch thick wood neatly shaped to suit. Birch must be used for the ribs and these fillets and the three-ply should be glued and pinned to them.

When introducing the spars into the fuselage frame care

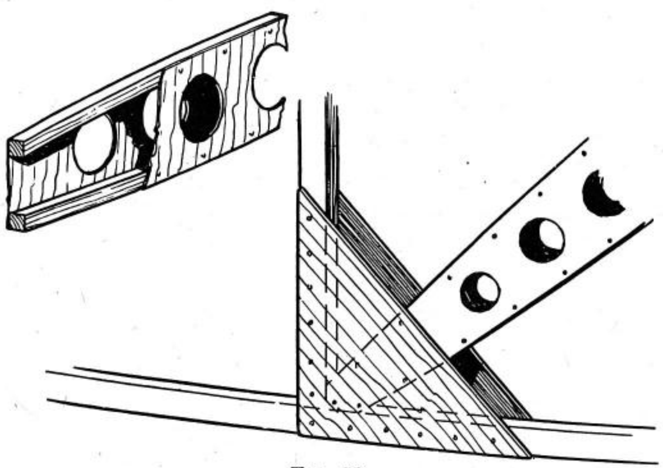


FIG. 58.

should be taken to see that they fit exactly. It is best to make the spars with square ends, and then trim them to a point to fit snugly between the junction of former and longeron. Finally the spar should be glued and pinned into position, and triangular gusset plates, also $\frac{1}{32}$-inch three-ply, fixed on each side as shown in Fig. 58.

The method of construction in which sheets of thin balsa are used for covering the fuselage may be used for both small and large fuselages, and whilst the finished product will not be quite as strong as the fuselage covered with $\frac{1}{32}$-inch three-ply, it will be amply strong enough for all ordinary purposes, and has the advantage of being extremely light.

Essentially the method of construction consists of building up a skin or " shell " of balsa on a wooden former which has been carved and shaped to the exact finished size of the fuselage.

Commencing with the tail end of the fuselage, sheets of $\frac{1}{32}$-inch balsa are laid up edge-to-edge over the surface of the block of wood, and held in position by means of drawing pins and rubber bands. As work proceeds towards the nose, the strips will require to be shaped so that they conform to the contour of the wood block.

When the block has been completely covered with strips of balsa, all butting edges are carefully gone over with fine sandpaper to ensure that there are no " humps " or ridges standing.

Next a second layer of strips of $\frac{1}{32}$-inch balsa is glued over the first layer, these running diagonally across the first.

Commencing at the tail of the fuselage, a few drawing pins are removed, sufficient to allow of the first strip being glued into position, after which the pins are replaced; and so, strip by strip, the outer covering is glued in position.

During this operation great care must be taken to see that no glue creeps down between any cracks between the strips of the inner layer.

Immediately the outer covering has been fixed in position, a length of fairly thick rubber say $\frac{1}{4}$-inch × $\frac{1}{16}$-inch, is tightly wound from end to end of the fuselage, the drawing pins and temporary rubber bands being gradually removed as the taping up proceeds. Thus the outer layer of strips is brought into intimate contact with the inner layer; and if glue, and *not* quick-setting cement, is used, the inherent flexibility possessed by these glues for the first 2 or 3 hours allows of slight " bedding-down " movements taking place between the layers of balsa.

Quick-drying cements must *not* be used.

The fuselage should be allowed to remain in a warm, dry atmosphere for not less than 48 hours, after which the rubber taping may be removed, and the entire surface carefully rubbed down with fine sandpaper. A circular cut, round the largest diameter, is then made with a razor blade, whereupon the two sections may be withdrawn from the wood block.

Suitably shaped bulkheads are then made from $\frac{1}{16}$-inch or $\frac{1}{8}$-inch three-ply and inserted at intervals along the length of each section, care being taken to arrange that they are positioned so as to strengthen the fuselage where the wings and the landing chassis are attached. In large fuselages several longerons of $\frac{3}{16}$-inch × $\frac{3}{16}$-inch may, with advantage, be glued throughout the length of each section.

The nose will require to be strengthened with pieces of solid balsa, and brackets fixed to carry the coil and batteries.

When all interior work has been completed the whole of the inside of each section should be given a good coat of hot glue; after which the two sections may be joined together in the following manner:—

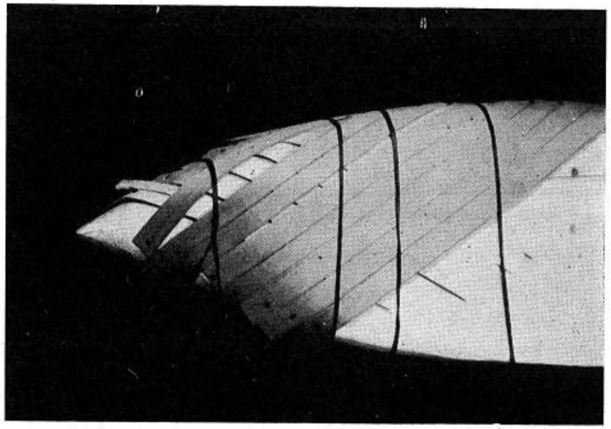


Fig. 59.

A liner of $\frac{1}{16}$-inch three-ply is made by bending a strip about three inches wide into a circle, whose diameter is such that the sleeve so formed just fits into one of the fuselage sections, and having been pushed in half-way, it is glued in position and a bulkhead inserted inside this ring. The other section of the fuselage is then slipped over the projecting portion of the sleeve, and thus the joint is made in exactly the same manner as the two sections of a cardboard Easter egg fit together.

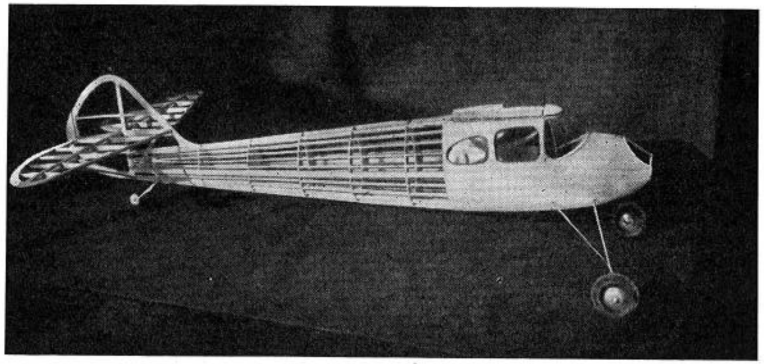


Fig. 60.

As the sleeve is supported by the bulkhead, a length of rubber may be quite tightly wound round the joint to ensure contact between the fuselage sections and the sleeve whilst the glue is drying.

Finally, the whole fuselage is covered with a layer of silk, stuck down with photo paste, and given two coats of cellulose paint.

Fig. 59 shows the nose of a large fuselage in course of construction by this method.

The width of the strips of balsa used in this method of construction will, of course, depend on the size of the fuselage, but normally 2 inches to 3 inches wide strips may be used, although these will require to be narrowed down to conform to the contours at the nose and wing-root " flarings."

With the considerably increased attention being given to the construction of Scale Model Aircraft, in which monocoque construction is not always used, the aero-modeller has had to develop his art by devising suitable constructions, which allow of the appropriate number of longerons being incorporated, so as to imitate the full-sized aircraft. In light-weight models the bulkheads may be cut from $\frac{1}{8}$-inch thick sheet balsa of fairly hard grade, the longerons being notched into them in the usual way. An example of this construction is shown in Fig. 60, which is of a 21-ounce all-up weight petrol 'plane, driven by a 2·5 cc. engine.

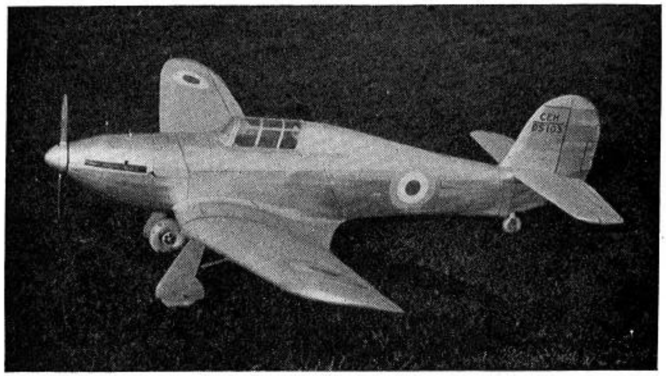


FIG. 61.

In constructing larger models the bulkheads should be of three-ply $\frac{1}{16}$-inch thick. A very interesting model, in which the workmanship is of high order, is a scale model Hawker "Hurricane," built by Mr. D. J. Miller. Fig. 61 shows the completed model. It is of 6 feet 8 inches span and driven by a 9 cc. Ohlsson petrol engine. The engine is totally enclosed, and to achieve this it had to be set back some inches from the nose of the fuselage, and the drive to the airscrew arranged through a shaft with universal coupling.

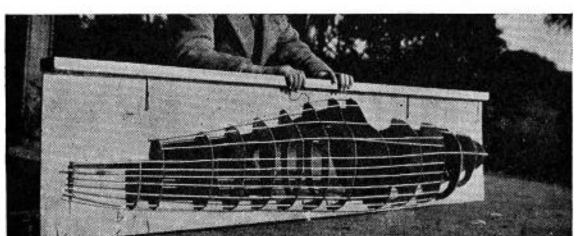
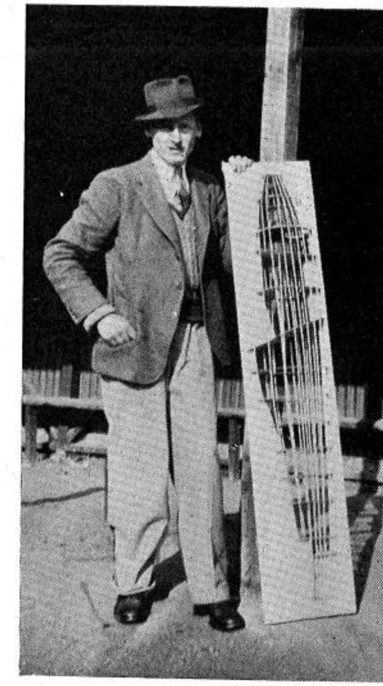


FIG. 62.

FIG. 63.



There is a full description of Mr. Miller's 'plane, and a set of scale drawings spread over three pages, in the May, 1939, issue of *The Aero-Modeller*.

In constructing the fuselage of this model, Mr. Miller mounted the bulkheads in a diaphragm and then let in the longerons on either side.

Fig. 62 shows the bulkheads mounted in the diaphragm, and Fig. 63 Mr. Miller holding the unit. In Fig. 64 is shown the completed fuselage ready for covering.

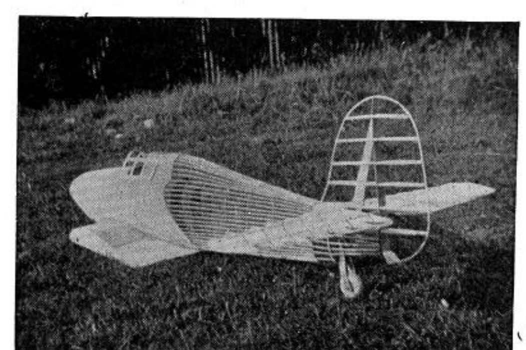


FIG. 64.