TECHNICAL MEMORANDUMS

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

No. 326

THE LIGHT AIRPLANE

By Ivan H. Driggs.

MODERN THEORETICAL AERODYNAMICS AS APPLIED TO LIGHT AIRPLANE DESIGN WITH A SERIES OF CHARTS - III.

DESIGN OF AN AIRPLANE WITH REFERENCE TO PHYSICAL DIMENSIONS, COMPONENT WEIGHTS AND DISPOSITION OF SURFACES - IV.

DESIGN OF AN AIRPLANE WITH REFERENCE TO BALANCE, DISTRIBUTION OF WEIGHTS AND MOMENTS - V.

Reprinted from "The Slipstream Monthly,"
February, April and July, 1925.

August, 1925.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

TECHNICAL MEMORANDUM NO. 326.

THE LIGHT AIRPLANE.

MODERN THEORETICAL AERODYNAMICS AS APPLIED TO LIGHT AIRPLANE DESIGN WITH A SERIES OF CHARTS.*

By Ivan H. Driggs.

PART III.

Technical Memorandum No. 311 gave a short outline of modern theoretical aerodynamics as applied to light airplane design. This discussion may have been somewhat obscure to the nontechnical reader. A series of charts or curves should serve to clear up such obscurity as well as to more definitely emphasize those quantities most important for each flight characteristic.

Accordingly a series of light airplanes is chosen for investigation as given below:

Weight - 500 pounds in each case.

Span - 15' 20' 30' 40'.

Power - 16-2/3 20 25 33-1/3 horsepower.

W/b = 33-1/3 25 16-2/3 12-1/2 lb. per ft.

W/P_M = 20 25 20 15 lb. per HP.

S_P = 2 sq.ft. in every case.

Revolutions of all engines 1750 R.P.M. for maximum power.

From this data Fig. 1 is calculated using equation (5) (T.M. No.311, p.19) and ordinary values for propeller efficiency.

* Reprinted from "The Slipstream," Feb., 1925, pp. 11-13;

These curves are sufficiently labeled to be self-explanatory. They themselves are interesting only in the conclusions to be drawn from them in the development of further charts.

Items of importance for light airplane performance, or the performance of any airplane are as follows:

I. Run along the ground before taking off.

II. Climbing angle after taking off.

III. Time to a specified altitude.

IV. Comfort in gusty weather.

From Fig. 1 each of the above characteristics will be investigated and shown by suitable curves.

I. Run to take off.- The length of the run along the ground is influenced by a great number of quantities, namely, the thrust available, the resistance of the airplane, its weight, the friction between ground and running gear, and the minimum speed at which the airplane can fly. In order to simplify the calculation the friction due to running over the ground is assumed constant. This in a measure may be controlled by the designer by the load imposed upon the tail skid. This friction is of short duration since the thrust tends to raise the tail quickly. Also the parasite resistance is taken as constant as the designer has exerted every effort to reduce this quantity to the absolute minimum. This series does not attempt a method of performance calculation but rather it is designed to show certain rules or laws that govern airplane performance. The induced drag has been eliminated also since this quantity does not appear until the airplane is sustaining some weight. The pilot runs the airplane along the ground at such an angle that the lift is nearly zero until he has reached flying speed, when he quickly pulls back the stick, increases the angle of attack and leaves the ground.

Fig. 2 shows the results of the calculation for take-off run for the series of light airplanes at various values of minimum velocity or take-off speed. From these curves may be deduced the following theorems:

III. Length of run to take-off varies directly as the Power Loading - pounds per horsepower, or expressed conversely, with constant weight and take-off speed the length of run varies inversely as the horsepower.

IV. Length of run with constant power varies as the take-off speed squared. Since with constant airfoil the minimum speed varies inversely as the square root of the wing area it follows:

V. Length of run varies inversely with wing area, power and wing section being unchanged.
This leads to:

VI. For the same length of run to be maintained the wing loading should be decreased in the same ratio as the power loading is increased.

This explains why a light airplane with a wing loading of 6 pounds per sq.ft. and a power loading of 20 pounds per horsepower will run the same distance as a larger airplane loaded 10 pounds and 12 pounds per sq.ft. and per horsepower respectively. By using one of the new efficient high lift wings there is no reason why the light airplane with its high power loading should not do equally as well in this respect as a great number of larger airplanes considered very satisfactory.

<u>II. Climbing angle after taking off</u>.- The angle of climb is of very great importance since it controls the height of obstructions that may be cleared in a given distance. This with the run before taking off determine the size and condition of a field from which the airplane may fly. Climbing angle is the resultant of two other quantities, the rate of climb or climbing speed and the velocity at which the climb occurs. This is a very good rate of climb as desired at a very low forward speed. A high rate of climb alone is of little use if this occurs at a high velocity.

The extension of the calculation to the determination of the maximum rate of climb and the velocity at which such climb occurs gives Fig. 3. This chart shows the relation existing between power and span for various slopes of the angle of climb. This slope is given as the height that will be reached in a space of 1000 feet. It represents the obstruction that may be cleared 1000 feet away from the point at which the airplane leaves the ground.

This chart is very interesting and instructive. If the curve marked 100 in 1000 be taken for analysis as a fair average it is found that for this slope to be obtained with a given power on the chosen airplane a very definite span is required. Also that as the power decreases to its lower values the span should increase at a very rapid rate. This shows the fallacy of attempting to construct a low powered light airplane with spans of from 10 to 15 feet as is sometimes suggested. This series of curves also show that if a practicable climbing angle is to be maintained a definite relationship may be found between power and span that will be the most effective and the cheapest as to cost and maintenance. To illustrate this: An angle of 1 in 10 calls for 31-1/2 horsepower at a span of 16 feet. If the span be raised to 20 feet the power is reduced to 24. It probably will be found cheaper and lighter to use the lower powered engine and increase the span to 20 feet. On the other hand, for the same slope a span of 28 feet requires 19 horsepower; if the span be increased to 32 feet the power required reduces to 18-1/4. A designer in this case would more than likely choose the higher power and the lower span.

Good judgment must be relied upon to dictate the correct balance between power and span. No mathematical treatment can be given.

In order to make this section somewhat useful, Fig. 4 is given, which shows the climbing angle plotted against span loading for various values of power loading. This chart may be used as a rather rough means of estimating the slope of the climb for any new design.

III. Time to 5000 feet.- This particular altitude is chosen for convenience and because it represents a fair height for cross country flying. The time spent in reaching any definite altitude is dependent upon two quantities, the rate of climb at the ground and the height at which the rate of climb becomes zero, or absolute ceiling, as it is called.

Charts 5 and 6 are plotted on a basis of the time to 5000 feet against span loading. Again the very marked influences of span are demonstrated, especially at lower powers. These curves are both useful and instructive. Fig. 5 may be used for a rough estimate of the time to 5000 feet for any new design. In one example chosen a very good check is obtained. The D-J-1 climbs 5000 feet in 11 minutes actual test. Its span loading is 18.9 and power loading 22.7. Fig. 5 gives the time as 10.8 minutes. This is well within the accuracy of the test observation.

Fig. 5 also demonstrates the very great rapidity with which the time to altitude decreases with increased span, especially for the higher power loadings. This again shows the fallacy of low power with small span. It simply serves to reiterate: Keep the span large.

Fig. 6, in which span is plotted against power for different values of the time to 5000 feet, illustrates a fact very similar to that which was pointed out in the case of climbing angle.

For any design there is a balance between power and span that will be most effective and the most efficient as to cost. If the designer wishes his airplane to climb to 5000 feet in 12-1/2 minutes, he may use an infinite number of combinations of span and power. As the span is increased from its lowest values the power necessary decreases quite rapidly. At the higher part of the range of span, however, just the opposite occurs, a great increase of span is necessary to lower the necessary engine power but little. The correct solution of the problem is left to the individual.

IV. Comfort in gusty weather. To a person traveling by airplane, this is very important. No one enjoys being tossed about like a feather in a breeze. This comfort cannot be expressed mathematically or shown by curves, because there is no definite coefficient that can be used to express the idea. Comfort may be properly divided into two separate conceptions, one of actual physical comfort of riding smoothly, and one of mental comfort in the feeling of safety. Physical comfort depends upon how much the airplane is affected by bumps. The feeling of safety is present when the airplane answers controls readily and is not easily stalled, that is, when the reserve power is ample for all emergencies.

The property of an airplane that makes it ride bumps comfortably is dependent upon its landing speed. The lower the landing speed the greater will be the effect of bumps. This may be demonstrated as follows:

The effect felt in a bump at any given speed is dependent upon the ratio of weight supported by the wings at any instant to the possible weight that the wings can support at that speed and instant. If two 500-pound airplanes both are flying at 80 miles per hour, one with a landing speed of 40 and the other with 20, and if a gust strikes each with the same intensity and at such an angle that their attitude to the relative wind is the same as for landing: The first will experience a force of $\left(\frac{80}{40}\right)^2$ or four times its own weight; the latter a force of $\left(\frac{80}{20}\right)^2$ or eight times its weight. These forces are the maximum that may happen due to gusts and very rarely occur. This shows, however, that the relative force which a gust exerts upon any airplane is dependent upon the minimum speed at which it may fly. From this is drawn the conclusion that the landing speed should be high. If the consideration of comfort were the only criterion for airplane design this conclusion would be justified. In reality, however, a balance must be reached between comfort and ability to take off and land, both of which demand a low minimum speed. It is often heard that light airplanes should land at 20 to 25 miles per hour. This can be done with present developments but only at the sacrifice of comfort and utility. Such an airplane becomes useful only in fair weather.

The feeling of safety as suggested above is due to relative reserve power. The absolute ceiling offers an excellent measure of this relative power. If it be arbitrarily taken that no airplane is safe to fly that has an absolute ceiling less than 15,000 feet, Fig. 7 shows the dependence of such safety upon power and span. The extremely rapid decrease in engine power required for a given ceiling as the span increases is remarkable. For a ceiling of 15,000 feet to be attained, a light airplane of 500 pounds weight and 16-foot span calls for 29.8 horsepower. If the span be increased to 26 feet, the power necessary becomes 15. The power loading in the first case is 16.78; in the second, 33-1/3, or practically double.

If it be admitted that the theory is correct that ceiling is an indication of safety and maneuverability at the ground, Fig. 7 demonstrates very definitely that high power is not necessary if the airplane has sufficient span. The light airplane is the only airplane that can use this fact without excessive span and wing weight as pointed out in the previous section.

Fig. 8 is appended for the estimation of ceiling for a new design.

<u>Summary</u>.- It is well to go back at this stage and summarize the ideas that have been brought forward. But two of the above requirements depend in any way upon wing area or landing speed, namely, run before take-off and comfort in gusty air. Unfortunately, these two requirements are diametrically opposed. A small run calls for a low landing speed or high wing area, while comfort demands a low wing area or high landing speed. This is a case in

which a compromise must be effected until such a time as a means of varying the lifting capacity of the wings is devised.

For high climbing angle, rapid climb and good ceiling, on the other hand, there is no disagreement. Increasing either power or span has a similar effect upon each of these characteristics. An increase of span lowers power required very rapidly at first for a chosen climbing angle, or time to 5000 feet. As the span increases very greatly, however, the power ceases to drop off so rapidly. This point suggests that for any design a proper balance may be obtained that will make for the greatest all-round efficiency. However, the span is shown to be much more important in the case of ceiling. It may be demonstrated mathematically that such a variation with span is more rapid than is the case with power. Fig. 7 illustrates this point.

This section serves to more clearly fix in the mind the rules laid down in the previous article with this addition:

Rule V.- Do not try to make the landing speed too low. To do so will make a fair weather airplane.

PART IV.

Design of an Airplane with Reference to Physical Dimensions, Component Weights and Disposition of Surfaces.

The previous Parts of this series have developed the theory of aerodynamics. This material was academic in character rather than directly applicable to the layout of a light airplane. This section takes up the design of an airplane with reference to its physical dimensions, component weights and disposition of surfaces.

The first step in the layout of an airplane is to decide upon the type to be built. The purpose for which the design is intended will largely determine this type. The builder will be given race conditions to meet or his own ideas will dictate the uses to which his design may be put. He also must decide whether he will build a monoplane or biplane. This question may not be dismissed as one of individual preference.

In nearly every case, structural considerations will point to the monoplane for single seaters. The span and wing area will be determined by the performance desired. If a biplane be under consideration it will be found that the wing chord will be so short that the internal bracing in the plane of the wing will be very weak and the wing cell will lack torsional rigidity. To illustrate this point: The desired performance may demand a span of 30 feet for a monoplane or 28 feet for a biplane, the wing area in both cases to be 90 square feet. The chord of the monoplane wing works out to be 36 inches while that of the biplane will be 19-1/4 inches. This small chord wing could not be properly braced without the addition of a great amount of weight and parasite resistance. If, however, the wing chord of the biplane be increased to a more practical length the area will be increased with a consequent loss of comfort in bumpy air. This analysis shows that for such a design the monoplane will give the best results. Present light airplane practice bears out this idea. The greatest number of successful light airplanes of the single-seater class have been monoplanes. The two-seater has not been sufficiently developed to warrant any conclusions being drawn at this time. Further experience may show that the monoplane has less advantage over the biplane for this type than in the case of the single seater.

Table II.

Name	Type	No. Cyl.	Power	R.P.M.	Wt. dry lb.	Displacement cu.in.
Anzani	Air-cooled	3	25	1500	110	122.0
Anzani	"	3	35	1600	128	190.0
Wright	"	3	63 72	1800 2000	175	223.0
Morehouse	"	2	20	3000	50	42.5
Morehouse	"	2	30	2500	85	80.0
Henderson	"	4	23	2750	127.5	79.4
Bristol	"	2	22 34	2500 4000	81 Direct 105 Gr.	67.0
Harley	"	2	9	3800	72	37.5
Sargant	Water-cooled	4	16	3200	99	46.4
Haacke	Air-cooled	2	30	1500	143.5	193.0
Haacke	"	3	48	1400	132	217.0
Siemens	"	5	55	1500	225	287.0
Siemens	"	7	75	1500	278	402.0

The next question that must be decided is one of construction and materials to be employed. Here the personal ability of the builder and the availability of materials will have weight. Welded steel tubing for fuselage, tail surfaces and landing gear is very cheap, strong, and light. This construction should not be attempted without the aid of a welder experienced in this class of work. It is much safer for the amateur builder to use older methods of construction with which any good cabinet maker is familiar. Spruce, plywood and aircraft wire with turnbuckles will make a satisfactory structure. It is always safe to follow the practice that is in use on large airplanes.

The type of wing bracing to be employed is dependent upon the preferences of the designer. Light airplane wings may be internally braced with but small increases in weight. This type requires slightly more labor in rib and spar construction due to tapering than an externally braced wing; however, fittings and wires or struts are eliminated. Internal bracing gives a much cleaner airplane and reduces the parasite resistance considerably. Care should be exercised in providing torsional rigidity in this type. This may be obtained by covering the wing between spars with very thin plywood for a large portion of the span.

The next step is to choose the engine. Table II has been prepared from all data at hand on engines suitable for light one- or two-seaters. Nothing need be said regarding a method of choosing an engine. It is obvious that a designer will endeavor to use an engine that calls for the least expenditure, both of money and of weight, for the power he desires to use.

Table III.

Airplane	Wt.	lb./ HP.	S_w sq.ft.	lb./ sq.ft.	Span ft.	lb./ ft.	f	f/c
Avro 558	480	26.7	166.0	2.89	30.0	14.6	13.5	4.66
Avro 560	471	23.5	138.0	3.41	36.0	13.1	16.0	3.7
A.N.E.C.	465	23.2	145.0	3.21	32.0	14.5	10.5	2.33
Wren	408		150.0	2.72	37.0	11.0	15.0	3.75
D.H.53	490		120.0	4.08	30.1	16.3	13.5	3.00
Viget	575	26.0	200.0	2.88	25.0	21.0	11.0	2.6
D-J-1	510	22.7	70.0	7.3	27.0	18.9	12.0	4.37
Farman	518	20.75	107.6	4.82	23.0	22.5	10.5	2.33
Pander H-2	650	26.0	110.0	5.9	25.2	25.8	11.0	2.2
Kolibri-U7	364		134.5	4.3	33.0	11.0	12.0	2.92
Roter-Vogel	397		146.5	4.5	33.0	12.0	10.0	2.25
Brownie I	870	29.0	178.0	4.3	36.5	23.9	15.0	2.27
Pander H-1	705	24.2	150.7	6.2	25.3	25.6		
Wee Bee	837	25.6	187.0	4.47	38.0	22.0	14.0	2.80
Daimler L 15			258.0		41.3			
A.N.E.C.	730	24.3	185.0	3.94	38.0	19.2	13.5	2.7
Avis	810	28.6	255.0	3.2	30.0	24.7	16.0	3.55
Vagabond	887	27.1	235.0	4.78	28.0	28.9	13.5	3.68
Blue Bird	875	26.7	243.0	3.6	2.8	28.5	14.0	3.0
Cygnet	730	23.7	165.0	4.4	28.0	23.8	12.0	2.82
Caspar C-17	716	23.8	168.0	4.25	39.4	18.2		
Udet	904	25.8	94.7	9.5	29.2	31.0		

Table III (Cont.)

Airplane	S_s sq.ft.	S_e sq.ft.	S_r sq.ft.	S_f sq.ft.	S_a sq.ft.	Remarks
Avro 558	9.5	11.0	8.5		31.7	Biplane
Avro 560	9.0	11.0	8.0		19.0	Monoplane
A.N.E.C.						"
Wren						"
D.H.53	9.0	13.0	7.5	2.25	30.0	"
Viget	11.5	8.77	4.84	3.40		Biplane
D-J-1	3.9	3.3	2.4	2.0	7.0	Monoplane
Farman						"
Pander H-2						"
Kolibri-U7						"
Roter-Vogel						"
Brownie I	16.4	15.1	6.7	13.5	26.0	"
Pander H-1						Biplane
Wee Bee	6.5	13.5	7.0	4.3	21.5	Monoplane
Daimler L15						"
A.N.E.C.	6.25	12.75	6.0		25.0	"
Avis	15.2	13.5	9.0		75.0	Biplane
Vagabond	14.0	8.3	5.5	2.5	47.0	"
Blue Bird	17.6	15.6	9.0	4.1	41.6	"
Cygnet	10.0	9.0	7.0		48.2	"
Caspar C-17						Monoplane
Udet						"

Note.- In the case of all biplanes the span loading has been reduced by 1/1.095 to give direct comparison with the monoplanes.

After the power plant has been determined the builder must estimate the total weight of his airplane. Table I (T.M. No.311, p. 12) has been repeated here with additional data as Table III. This material will serve as a guide in making an approximate weight estimate. Such a figure will be close enough for preliminary purposes. One will see at once that a single-seater will weigh from 450 to 500 pounds and a two-seater from 750 to 800 pounds. There will be some variation from these figures with different engines and types of construction but estimates based upon Table III are sufficiently close for the present. If the detail weight estimate to be made later shows too great a variation from this first estimate the work may have to be repeated and the design of the airplane revised.

The designer is now in a position to make a sketch of his airplane. The power he intends using and the estimated weight give the power loading, pounds per horsepower. Fig. 2 shows the length of run to take off for the above power loading at various values of the minimum speed, and thus determines the required wing area with any airfoil. Formula (7) (T.M. No. 311, p.26) gives the wing area required to obtain any minimum speed. This equation is repeated here for convenience.

$$S_W = \frac{Wt}{Ky_{max.}} \quad (V_{min.})^z$$

The table of airfoils given in T.M. No. 311, p. 27, will indicate a good section to use. For cantilever wings the U.S.A.35 and U.S.A.45 give good results. For thin braced wings U.S.A.16

and R.A.F.15 are satisfactory while U.S.A.27, U.S.A.35-B, Göttingen 430 and Clark Y make good thick braced wings.

An estimate of span loading and consequently span may be made by referring to Figs. 4, 5 and 7, and also to Table III. Seven single-seaters show an average span loading of 18.6 pounds per foot. The average for eleven two-seaters is 25 pounds per foot. The power loadings for the single-seaters average 24 and for the two-seaters 26 pounds per horsepower. In deciding upon the span to use it is wise to hold rather close to these figures. If the power loading is much lower than the above averages the span loading should be reduced in the same proportion.

With the span and area of the wing given it is a simple matter to find the chord. In the case of a biplane the span determined above should be reduced in dividing by 1.095. This takes care of the interference experienced between the wings. The biplane will have a slightly smaller span for the same induced power (see equation 5a, T.M. No. 311, p.19). The tail length required follows after the determination of the wing chord. Table III gives the ratio of tail length to average wing chord for the series of light airplanes. By tail length is meant the distance from the center of gravity of the airplane to the rudder post. In general this length should be from three to four times the average wing chord.

The area of all control and stabilizing surfaces are given by the following formulas:

Let S_s = Area of horizontal stabilizer in sq.ft.

S_e = Area of elevators in sq.ft.

c_w = Chord of wing in feet.

f = Tail length in feet.

S_f = Area of fin in sq.ft.

S_r = Area of rudder in sq.ft.

b_w = Span of wing in feet.

$$S_s = \frac{.27\ c_w\ S_w}{f} \tag{9}$$

$$S_e = \frac{.25\ c_w\ S_w}{f} \tag{10}$$

$$S_f = \frac{.005\ b_w\ S_w}{f} \tag{11}$$

$$S_r = \frac{.015\ b_w\ S_w}{f} \tag{12}$$

The area of the ailerons should be from 15 to 18 percent of the total wing area except when the ailerons are used as flaps for reducing the landing speed.

After all areas and dimensions are calculated and the sketch is complete and satisfactory as to appearance and arrangement a detail weight estimate must be made up and the balance checked. Such a weight estimate is best made to a standard form which includes all items. A convenient form is given below:

Power Plant - This group includes all items of weight incident to the engine and fuel installation and is made up by:

A. The Engine - (See Table II).

B. The Propeller - Mr. H. C. Watts gives a formula for weight of wood propellers:

$W_{prop} = .04D^3$

D = diameter of propeller in feet.

If a spinner is used its weight may be calculated from the size to be used.

C. Radiators.- Probably will not be present on light airplanes.

D. Radiator Pipes and Expansion Tank - (See C).

E. Radiator and Tank Water - (See C).

F. Engine Water - (See C).

G. Gasoline Tank - The tank weight will depend upon the capacity to be carried. The sketch will give the dimensions from which the weight may be calculated using data from Table IV. About one pound should be added for filler and brackets.

H. Gasoline Piping - The length may be measured from the sketch and weight calculated. Add about one pound for fittings and cock.

I. Oil Tank - (See G).

J. Oil Pipes - (See H).

K. Engine Controls - A very light control to the carbureter and magneto may be built for one and one-half pounds by using wires to actuate the levers.

L. Exhaust Manifolds - A simple calculation will give the weight after measuring the length from the sketch. Short stacks may be made for one-quarter pound per cylinder.

The sum of all the above items gives the weight of the power plant.

II. Furnishings - This group is made up of:

A. Flooring - One-quarter inch three-ply suitable for flooring weights, .6-pound per square foot. The dimensions necessary may be found from the preliminary drawing.

B. Firewall - With the area known, it is easy to calculate the weight using data from Table IV on the material to be used.

C. Surface Controls - The rudder bar or pedals and the stick unit make up the surface controls. An allowance of three or four pounds is sufficient.

D. Instrument Board - Allow about one-half pound.

E. Control Wires - May be calculated directly by referring to Table IV and the drawing.

F. Seats - A small seat may be made for two and one-half to three pounds if a standard seat is not to be used.

G. Cushions - Allow two and one-half to three pounds.

H. Miscellaneous - Items of furnishings not listed should be estimated and inserted here. A small safety belt will weigh about one and one-half pounds. Map cases, tools and tool boxes should all be estimated and allowances made if they are present.

The sum of the above items gives the total weight of the furnishings group.

III. Equipment -

A. Instruments - Weights of some instruments are as follows:

Switch	.5	pound
Oil pressure gage . .	.4	"
Altimeter	1.0	"
Airspeed indicator .	.75	"
Pitot tube	.5	"
Watch	.6	"
Compass	2.7	"
Tachometer	1.6	"
Shaft, per foot	.2	"
Airspeed aluminum tube per ft.	.03	

B. Parachute - A seat type parachute, weighs 18 pounds.

C. Electrical Equipment - If present the weight should be estimated and entered here.

The sum of the items A, B and C, gives the total equipment.

IV. Crew - Allow 150 pounds per man, or use a known individual weight.

V. Fuel and Oil - Gasoline weighs 6 pounds per gallon, and oil 7.5 pounds per gallon. The oil capacity should be 10 percent of the gasoline capacity by volume.

VI. Body Group - The weight of the body is the most difficult to estimate when no detailed data is available on similar types.

A. Fuselage - The only data available on a single-seater is that of the DJ-1, where the complete fuselage weighs 30 pounds; a two-seater fuselage will weigh approximately twice as much. The following formula gives weights that are not difficult to meet.

Let W_f = Weight of fuselage covered in pounds.

W_p = Weight of power plant (See I).

W_u = Weight of useful load - III + IV + V.

$$W_f = 10 (W_p + W_u)$$

B. Cowling - Here again the preliminary sketch will help in making a reasonable estimate. An allowance of a pound or so should be made for bolts, clips, etc. A small windshield will weigh one and one-half pounds.

C. After-deck - Allow about 4 or 5 pounds or calculate the weight, knowing the dimensions and materials used.

D. Engine Mounting - Considered as part of the fuselage.

The sum of the above items gives the total weight of the body group.

VII. Landing Gear -

A. Chassis - (Struts, axle, etc., without wheels). Allow about one and one-half percent of the total airplane weight for the chassis.

B. Wheels - See Table IV for standard sizes.

C. Tail Skid - Allow two or three pounds or calculate the weight from the dimensions.

The sum of these items equals the total landing gear weight.

VIII. Wings -

A. Airplanes Exclusive of Ailerons - The weight of the panels depends upon the type of construction. An internally braced wing will be heavier than one externally braced. Data on either type is lacking. The DJ-1 cantilever wing weighs 1.2 pounds per square foot. This panel is covered on both surfaces from the leading edge to the rear spar with one-sixteenth inch birch plywood for torsional rigidity. There is no data available on light airplane wings of the braced type, but it should be possible to build for six- to eight-tenths pound per square foot.

B. Ailerons - Allow seven- to eight-tenths pound per square foot.

C. Struts - If present, estimate sizes and calculate weight from the sketch.

D. Wires - (See C).

The sum of these items gives the total wing weight.

IX. Empennage -

A. Elevator - Allow six- to seven-tenths pound per square foot.

B. Rudder - Allow six- to seven-tenths pound per square foot.

C. Stabilizer - Allow seven- to eight-tenths pound per square foot.

D. Fin -

E. Brace Wires - Allow one pound or calculate from sketch.

The sum of the items, A to E inclusive, gives the total weight of the empennage.

Now add up groups I to IX for the total weight of the airplane. This total should be compared with the previous estimate. If there is a large discrepancy between these two figures, it may be necessary to revise the preliminary design and repeat the work until a satisfactory agreement is obtained. As each item is constructed it should be carefully weighed and compared with the estimate. It is often possible to eliminate a great deal of weight by lightening up after a part is constructed. The greatest efforts should be exerted to keep the final weight equal to the estimate or if possible, under it.

After the detail weight schedule has been completed the balance should be checked. For proper stability the center of gravity is found by first locating on the sketch each item of weight from the schedule. A small circle with the weight of the part marked on it will be found convenient. Next choose some convenient base lines, say the rear face of the propeller flange for longitudinal and the ground line for vertical position of the center of gravity. Multiply the weight in each circle by the distance from that circle to the chosen base line and find the sum of these products. This sum divided by the total weight of the airplane will give the distance from the base to the center of gravity. Two calculations, one for longitudinal and one for vertical position will locate a point which should be put in on the sketch and marked c.g. These calculations will be clear after the example to be given in Part V.

If the center of gravity as above located does not lie in the proper place, the wings may have to be shifted and the balance calculation repeated. A line at right angles to the wing chord through the center of gravity should intersect the wing at a point from 28 to 30 percent back from the leading edge. Similarly the landing gear may have to be shifted. The wheels should be located 15 degrees forward of a vertical line through the c.g. to insure safety on the ground.

If a staggered biplane is used the two wings must be replaced by a mean aerodynamic chord when balancing. This imaginary chord line is located between the wings closer to the wing of large area in proportion to the respective areas. The leading and trailing edges of this m.a.c. lie on lines connecting the leading and trailing edges of the upper and lower wings. The position of the center of gravity should be the same as given above. The biplane is imagined to be replaced by a monoplane of equivalent lifting capacity and location.

Table IV.

Weight of Materials of Airplane Construction.

Material	lb./cu.in.
Steel	.284
Duralumin	.102
Al. Bronze	.278
Cast Iron	.261
Brass	.309
Bronze	.295
Copper	.322
Spruce	.016
Ash	.023
Birch	.026
Basswood	.0154
Hickory	.0295
Mahogany	.019
Poplar	.0189
Walnut	.0226
Nut	**lb.**
8-32	.0046
10-32	.0062
1/4-28	.0077
5/16-24	.0125
Washer	**lb.**
No. 8	.0007
" 10	.0007
1/4	.0031
5/16	.0031
Turnbuckles, Short	**lb.**
800 lb.	.031
1600 "	.062
2100 "	.077
3200 "	.108
Turnbuckles, Long	**lb.**
1600 lb.	.093
2100 "	.125
3200 "	.187
4600 "	.233

Table IV (Cont.)

Weight of Materials of Airplane Construction.

Clevis Pin		lb.
5/32	.0025 +	.0044L
3/16	.004 +	.0056L
1/4	.005 +	.012 L
5/16	.0076 +	.02 L

Bolt - Hex. Head		lb.
8-32	.004 +	.0053L
10-32	.0056 +	.0074L
1/4-28	.007 +	.0134L
5-16	.011 +	.0208L

Sheet	lb./sq.in.
Steel	.284t
Duralumin	.102t
Brass	.309t
Copper	.322t

Tubes		lb./ft.
Steel	10.7	$(Dt-t^2)$
Duralumin	3.85	$(Dt-t^2)$
Brass	11.66	$(Dt-t^2)$
Copper	12.15	$(Dt-t^2)$

Wires		lb.
Strm. No. 6	.033 +	.002 L
No. 10	.064 +	.035 L
1/4	.132 +	.066 L
5-16	.237 +	.0104L
3-8	.392 +	.0157L

Galv. Wire	lb./100 ft.
No. 20	.273
" 18	.434
" 16	.690
" 14	1.097
" 12	1.744
" 10	2.77
" 8	4.00

Galv. Cable	lb./100 ft.
1-16	.78
3-32	1.21
1-8	3.5
5-32	5.5
3-16	7.7

t=Thickness in inches; L=Length in inches; D=Diameter in inches.

Table IV (Cont.)

Weight of Materials of Airplane Construction.

Flex. Cable	lb./100 ft.
1-16	.73
3-32	1.3
1-8	2.88
5-32	4.44
3-16	6.47

Thimbles	lb./1000 ft.
1-16	3.00
3-32	3.00
1-8	4.3
5-32	6.5
3-16	9.0

Ferrules	lb./1000
No. 20	.52
" 18	1.14
" 16	2.09
" 14	4.25
" 12	8.56
" 10	17.56
" 8	4.4

Wing Cover Doped	lb./sq.ft.
Single Surface	.1
3-ply	
3-16	.175
1-16	.2
3-32	.35
1-8	.4
3-16	.5
1-4	.65

Wheels	lb.
20 x 2	7.5
24 x 3	12.25
26 x 3	13.5
26 x 4	17.5
28 x 3	14.9

PART V.

Design of an Airplane with Reference to Balance, Distribution of Weight and Moments.

In the design of the airplane to be undertaken in this division the following points are given preference:

1. The layout must be such that the simplest construction methods and the cheapest material may be employed.

2. The engine must be cheap and easily obtained.

3. The landing speed must be low and the general performance must be the best that can be obtained.

4. The airplane will be a single seater with two hours' fuel capacity.

To meet the above specifications the airplane will be a semi-cantilever monoplane for simplicity of construction. Wood will be used throughout and the number of fittings will be reduced to a minimum. The 4-cylinder Henderson De Luxe motorcycle engine will be used.

A propeller hub and thrust bearing can be built on to the Henderson motor in place of the regular flywheel and housing. Although this engine is quite heavy for its power the price and reliability strongly recommend it.

By inspection of Table III, of Part IV, the first estimate of weight is taken as 525 lb. From Fig. 9 the engine power is $23\frac{1}{2}$ at 2700 R.P.M. The power leading then becomes 22.3 lb. per HP. This is somewhat lower than the average from Table III, so that the span loading may be increased over the average value of 18.6 lb. per ft.

$$\frac{18.6 \times 24}{22.3} = 20 \text{ lb. per ft.}$$

$$\frac{525}{20} = 26.25 \text{ ft. span.}$$

Since the cockpit will not be enclosed and also since struts are to be provided for bracing the wings, the parasite area S_p, will probably be about 3.0 sq.ft. In Technical Memorandum No. 311, Part II, it was demonstrated that the theoretical minimum speed should not exceed the speed of minimum power as given by formula (6)

$$V_{mp} = 10.64 \sqrt[4]{\frac{W^2}{b^2 \;\; S_p}} \tag{6}$$

$$V_{mp} = 36.2 \text{ miles per hour.}$$

If the minimum speed is taken as 40 miles per hour, the above requirement will be approximately fulfilled and the landing speed will be low enough to ensure rapid take-off and ease of handling near the ground. The U.S.A. 45 airfoil is well suited to cantilever wing construction and has the highest ratio $\frac{K_{y\,max}}{K}$ of the sections listed. The maximum lift coefficient is .00331. Referring to formula (7) the wing area is given as

$$S_W = \sqrt{\frac{W}{K_{y_{max}} V_{min}^2}} = 101 \text{ sq.ft.} \tag{7}$$

To find the chord of the wing it is assumed to be tapered from the brace strut outward to a tip chord one-half that at the root. The fuselage is 2 ft. wide and the strut supports the wing 4 ft. from the side of the body.

Root chord = 5 ft.

Tip " = 2.5 ft.

Mean " = 4.17 ft.

If the tail length is 12 ft., the ratio of tail length to chord becomes 3.2. This value is within the limits of the designs given by Table III.

Formulas (9) to (12) inclusive, are used to find the tail surface areas:

$$S_s = .27 \frac{c_W S_W}{L_h} = \frac{.27 \times 4.17 \times 101}{12} = 9.5 \text{ sq.ft.}$$

$$S_e = .25 \frac{c_W S_W}{L_h} = \frac{.25 \times 4.17 \times 101}{12} = 8.8 \text{ sq.ft.}$$

$$S_f = .009 \frac{b_W S_W}{L_h} = 2 \text{ sq.ft.}$$

$$S_r = .03 \frac{b_W S_W}{L_h} = 6.6 \text{ sq.ft.}$$

The data previously computed is summarized below for ready reference in laying out the preliminary sketch.

Engine, Henderson 4, 23.5 HP. at	2700 R.P.M.
Total weight,	525 lb.
Wing Span,	26.25 ft.
Wing Area,	101.00 sq.ft.
Mean Wing Chord	4.17 ft.
Root " "	5.00 "
Tip " "	2.50 "
Tail length - C.G. to tail post,	12.00 "
Area Stabilizer,	9.50 sq.ft.
" Elevators,	8.80 "
" Fin,	2.00 "
" Rudder,	6.60 "
" Ailerons,	15.00 "
Diameter of propeller,	5.50 ft.

Fig. 10 shows a preliminary sketch and balance diagram of the proposed airplane. Table V contains the calculation necessary for finding the position of the center of gravity both longitudinally and vertically. The calculated weight of 539 lb. is sufficiently close to the first estimated for all practical purposes. From Table V, the C.G. is located 47.5 in. from the front face of the propeller and 42 in. from the ground. With reference to the mean chord, the C.G. is located at 29.2% back from the leading edge, and on the thrust line. This position is very favorable for longitudinal stability both power on and off.

Figs. 11, 12 and 13, show the three views of the airplane as designed in the previous paragraphs of this section. These drawings and data conclude this part as well as the whole series.

Table V.

Item	Weight lb.	Horizontal Arm	Horizontal Moment	Vertical Arm	Vertical Moment
Engine	127.5	14	1785	43.75	5580
Propeller	6.5	1.5	10	42	278
Tank & Gas	35.0	33	1155	45	1575
Gas Pipes	3.5	24.5	86	44	154
Engine Control	1.5	36	54	44	66
Manifold	1.0	14	14	43.75	44
Floor	2.0	31.5	63	21	42
Stick	1.5	46.5	70	31	46
Rudder Bar	1.0	28.5	28	24	24
Fire wall	3.0	24.5	73	37.5	113
Inst. Bd.	.5	45.5	23	51	25
Control Wires	3.5	77.5	271	23.5	82
Seat	3.0	58	174	33.5	100
Cushion	2.0	58	116	28.5	57
Belt	1.5	60	90	36	54
Instruments	4.5	45.5	205	51	229
Pilot	150.0	54.5	8180	42	6300
Oil	7.5	14	105	38	285

Table V (Cont.)

Item	Weight lb.	Horizontal Arm	Horizontal Moment	Vertical Arm	Vertical Moment
Fuselage	36	75.5	2700	34	1225
Cowling	7	16	112	38	266
Chassis	8	35.5	284	15	120
Wheels	15	37	555	10	150
Tail Skid	2.5	180	450	34.5	86
Wing	78.0	54	4210	50	3900
Ailerons	12.0	76	912	47	564
Struts	8.0	51.5	412	34.5	276
Stabilizer	6.0	184.0	1105	48.5	291
Elevator	5.5	197.5	1086	48.5	267
Fin	1.5	189.0	284	54.0	81
Rudder	4.0	204.0	816	56.0	224
Tail Wires	1.0	192.0	192	58.0	58
	539.5	47.5	25,629	41.8	22,557

Note.- The front face of the propeller and the ground are taken as base lines for horizontal and vertical moment arms, respectively.

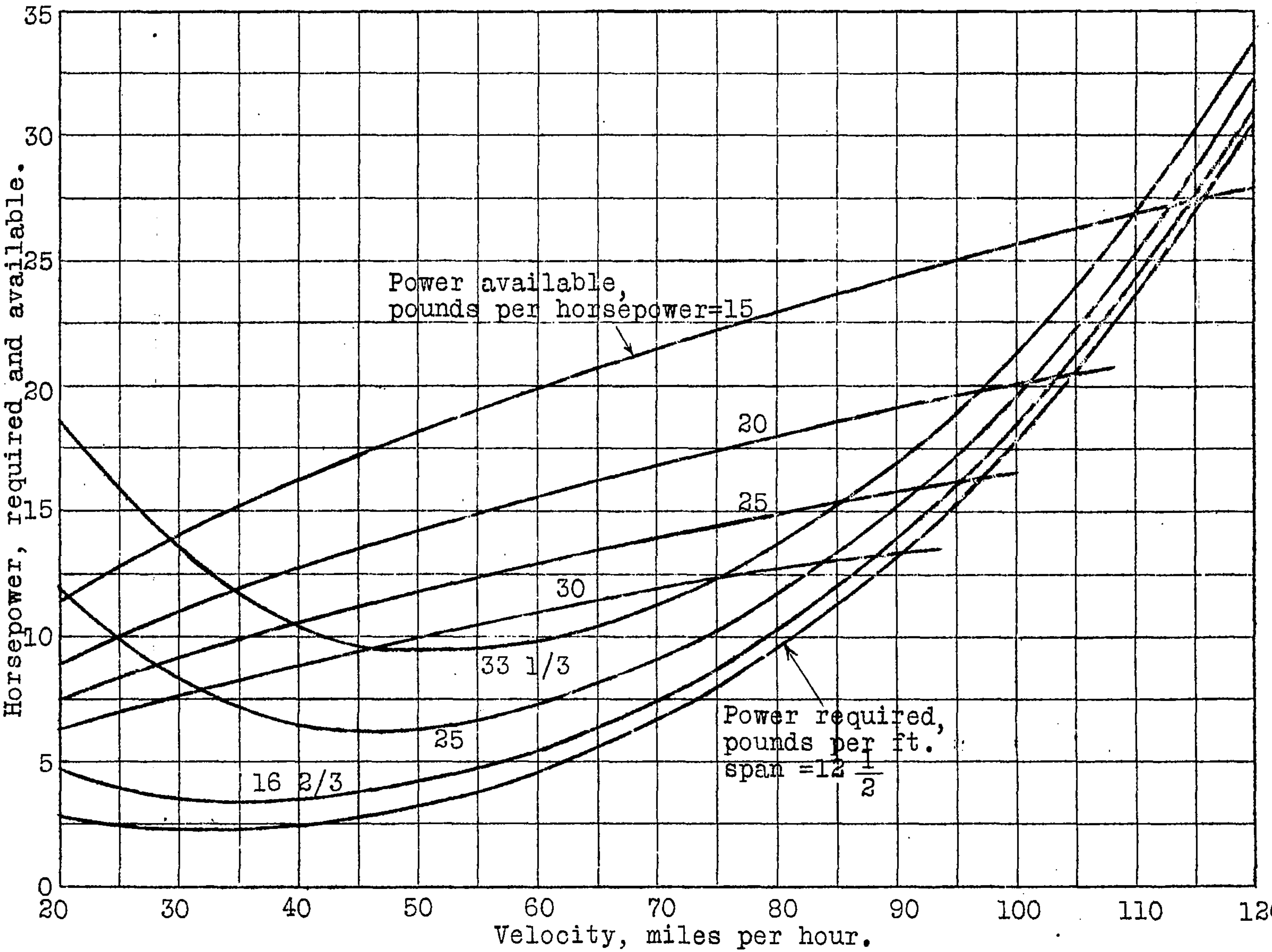

Fig.1 .Curves of power available and required, 500 pound light airplane

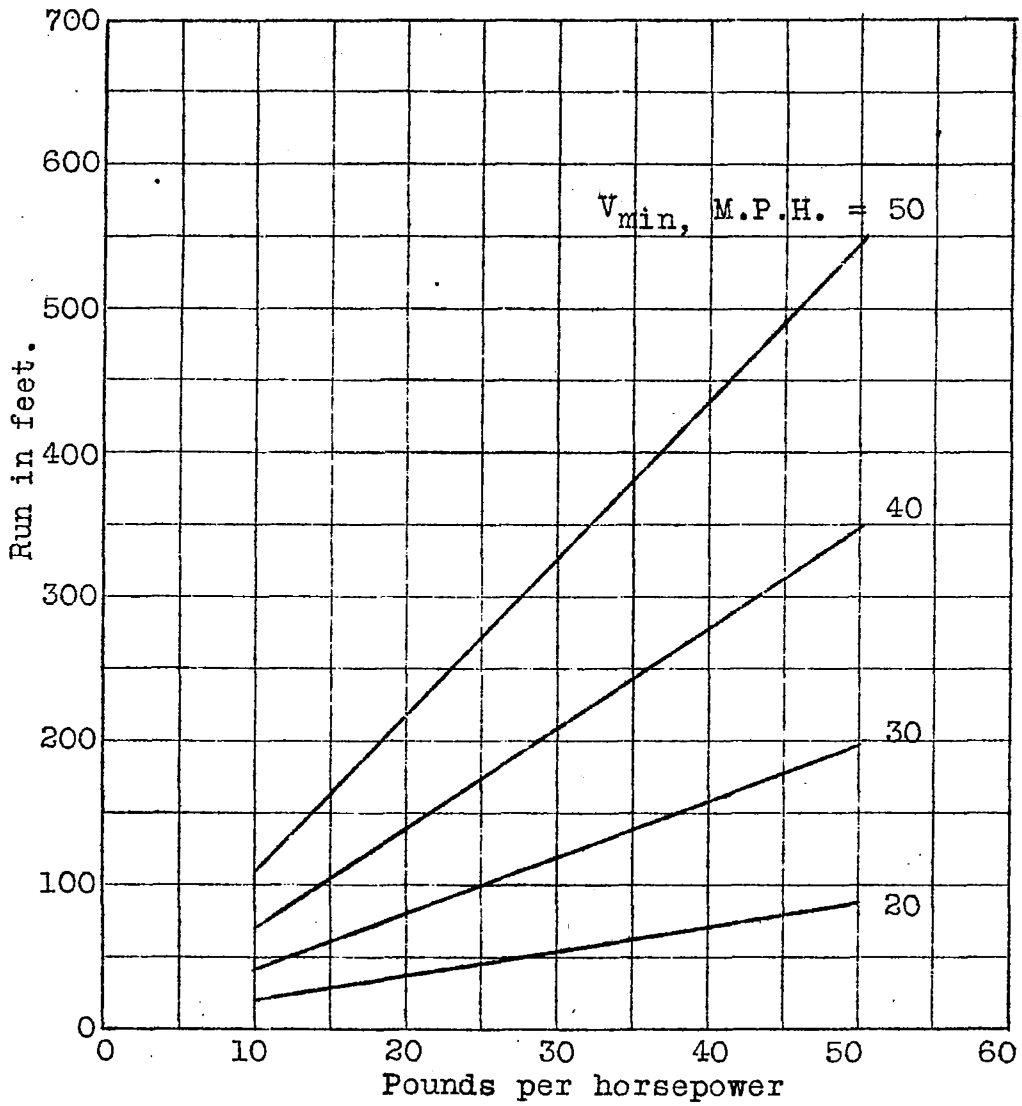

Fig.2 Run for take-off for 500 pound light airplane.

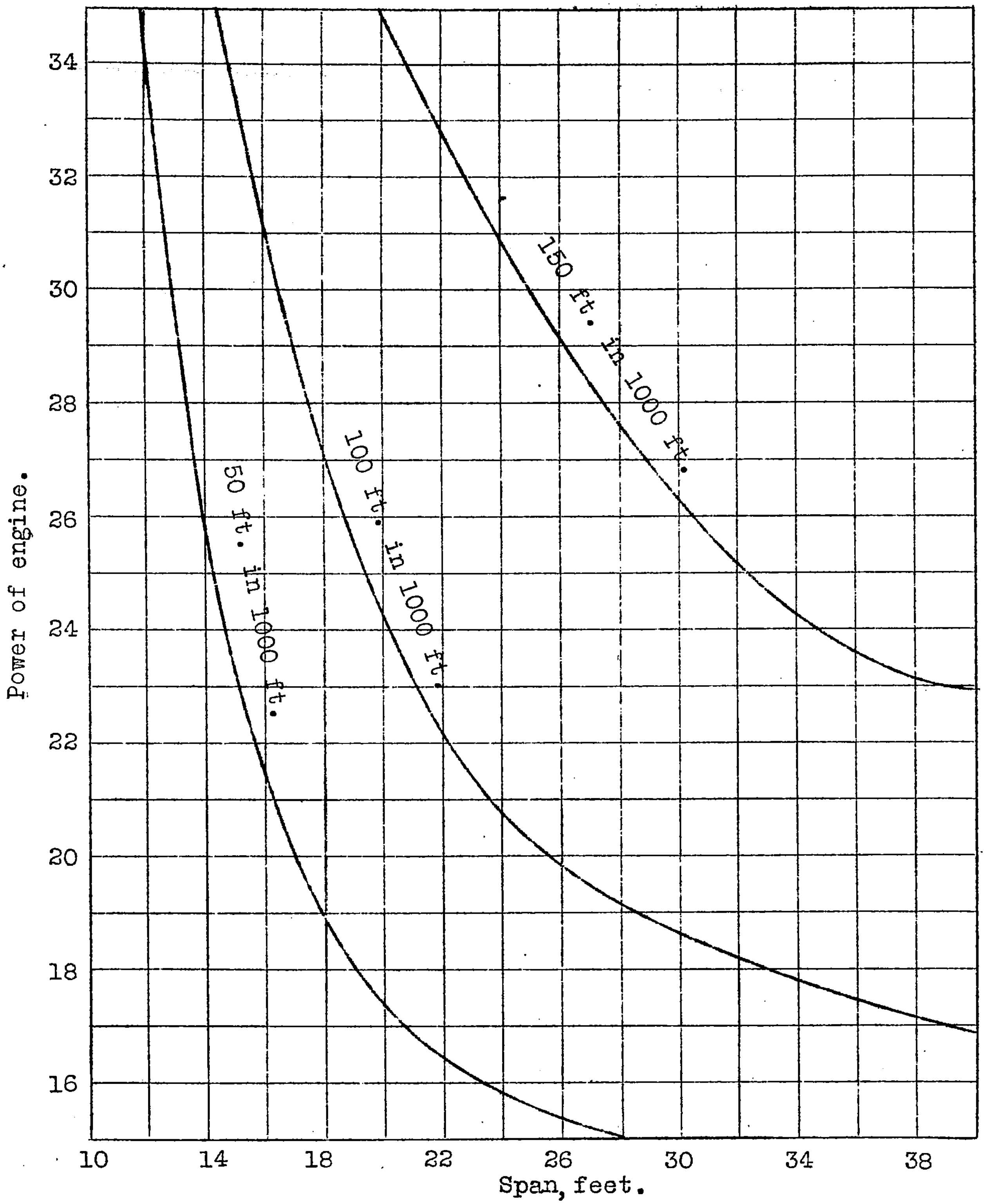

Fig.3 Angle of climb, 500 pound light airplane.

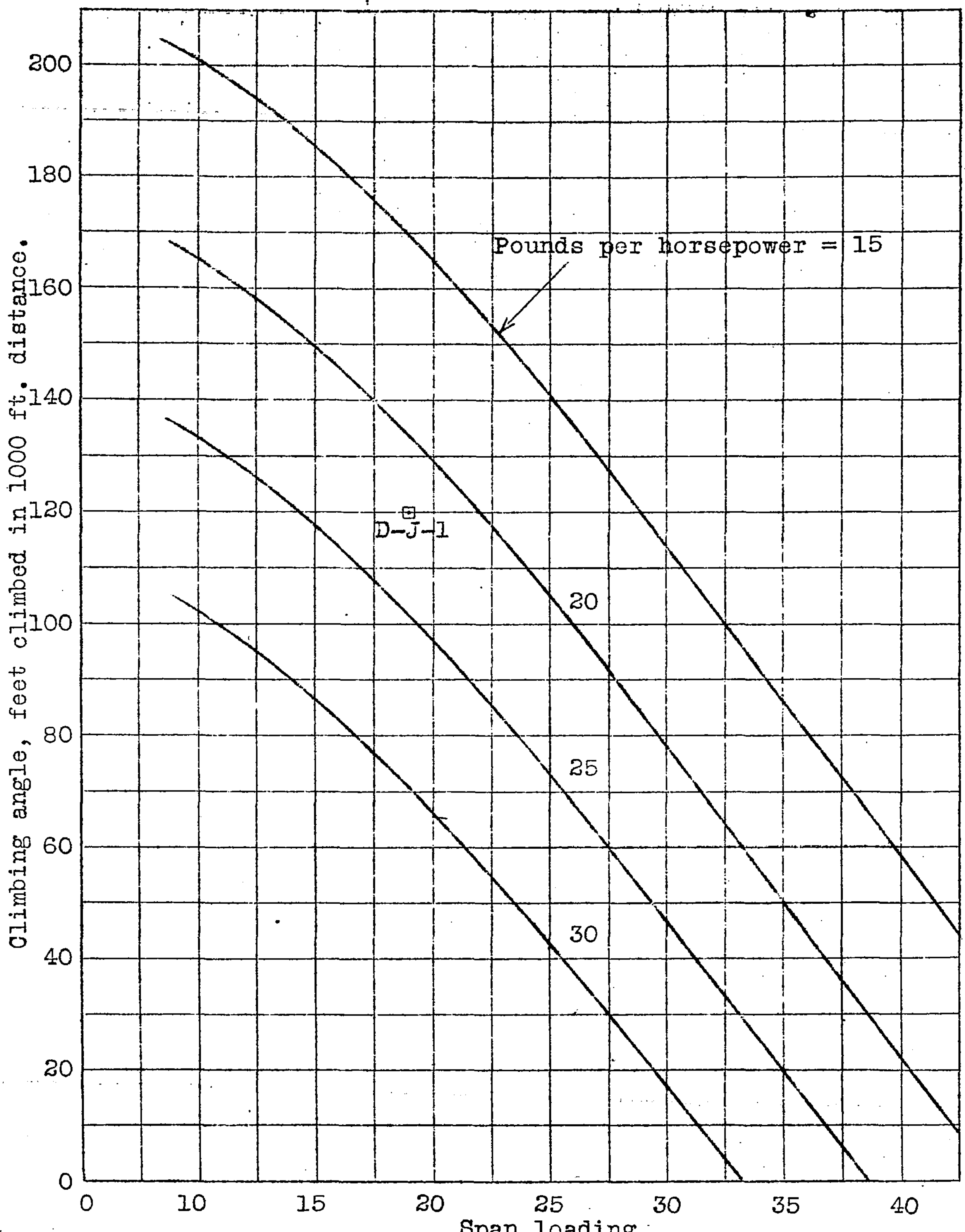

Fig.4. Angle of climb vs span loading.

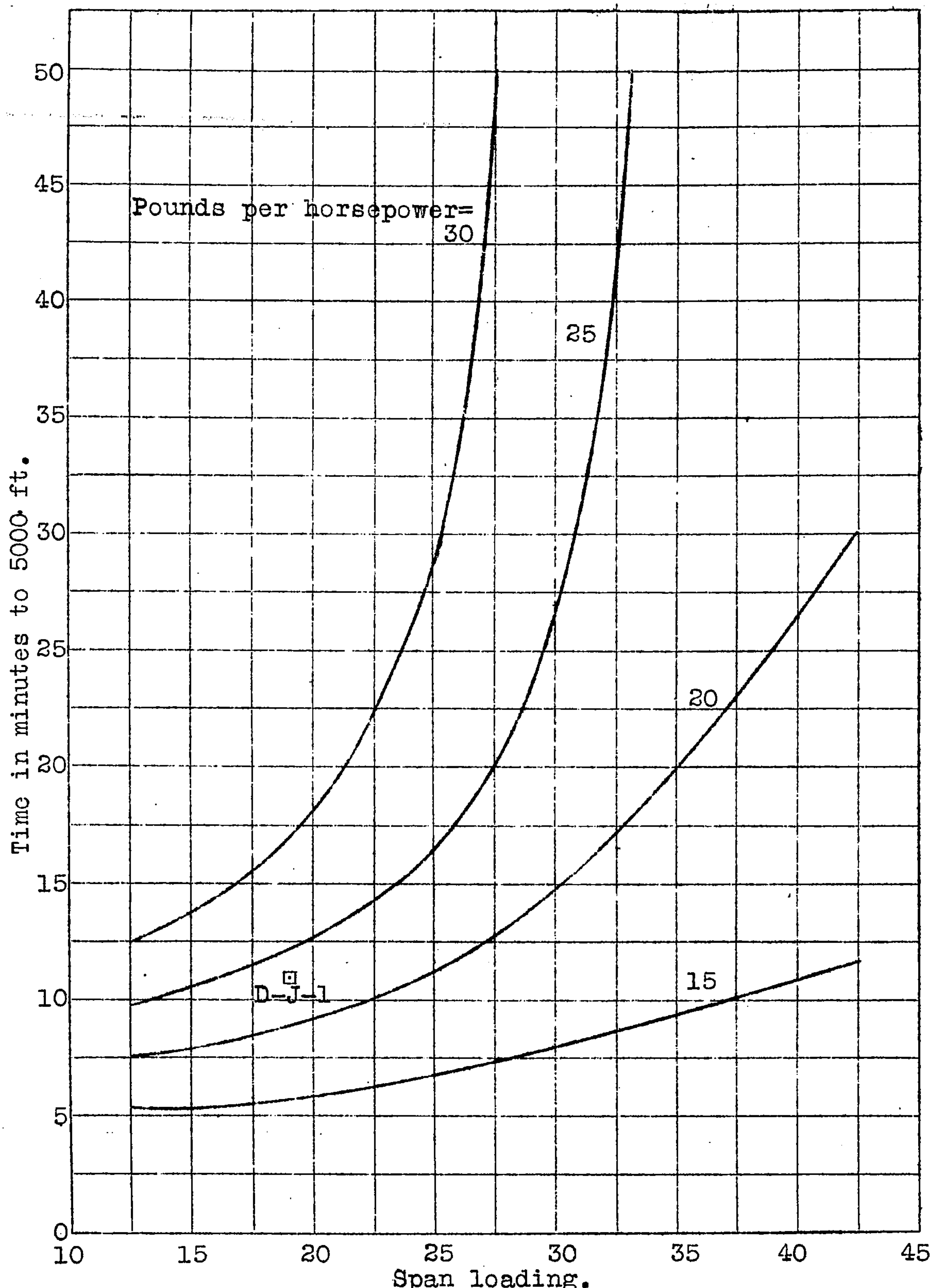

Fig.5 Time to 5000 ft. vs. span loading.

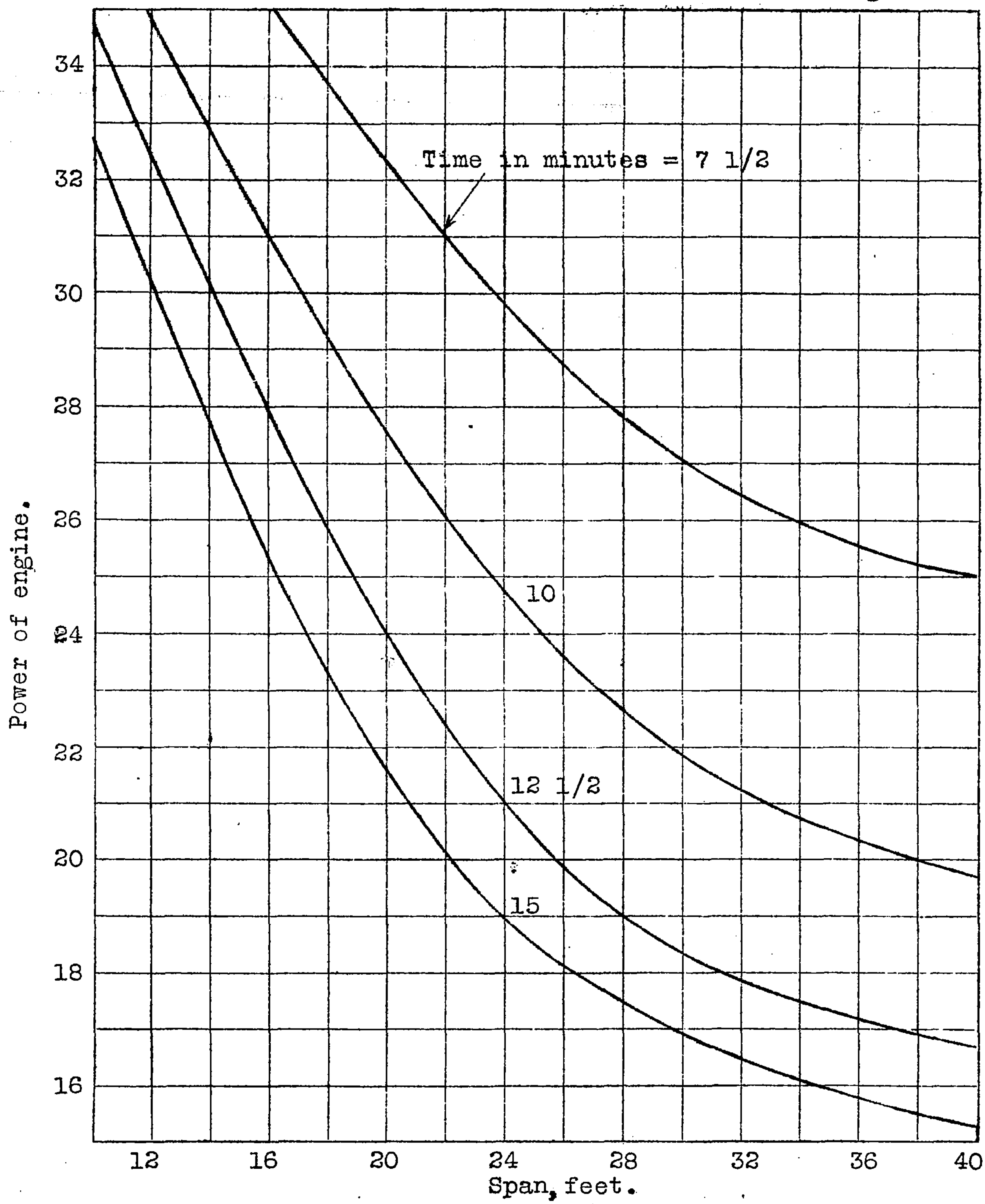

Fig.6 Time to 5000 ft. for 500 pound light airplane.

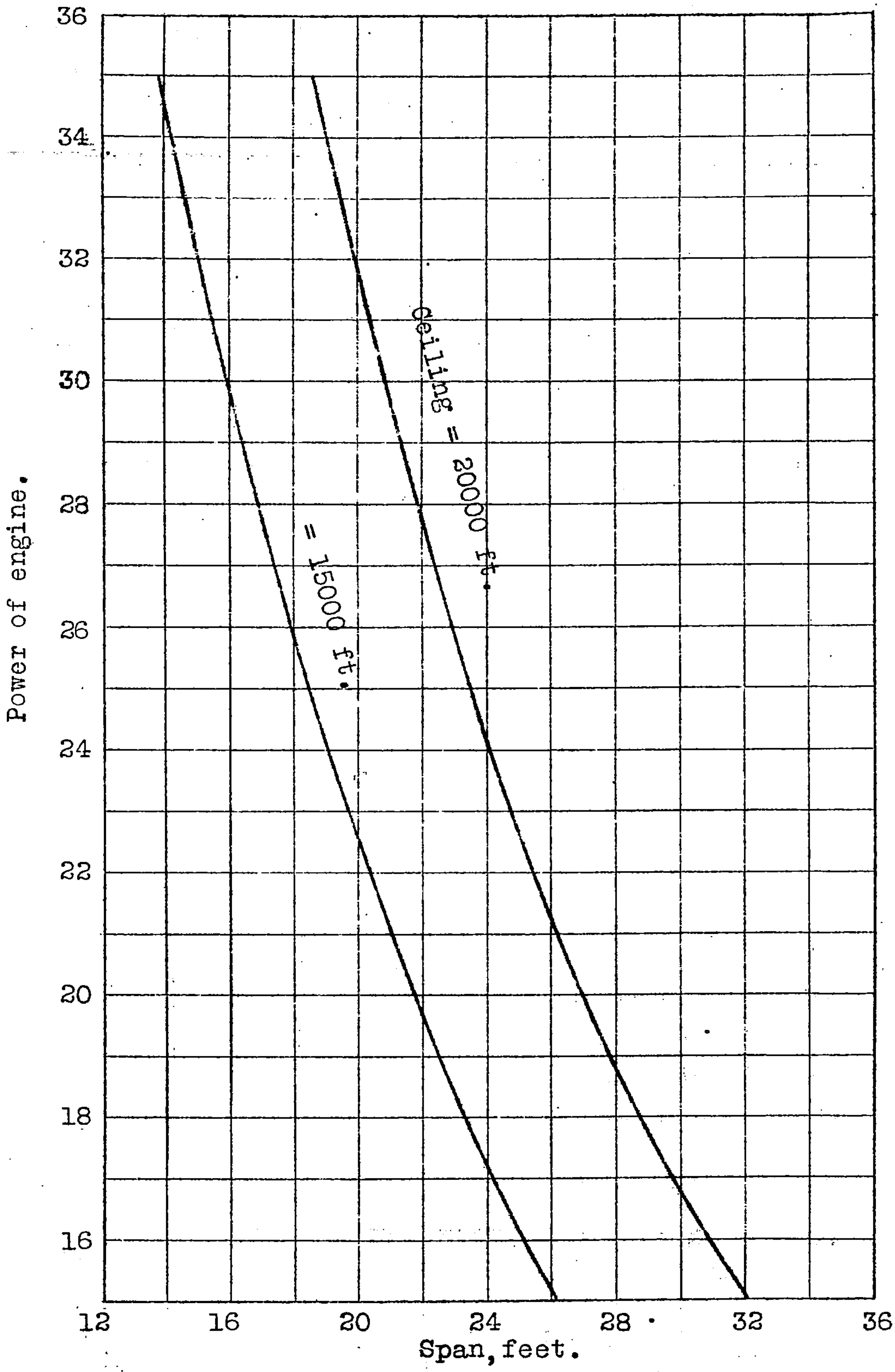

Fig.7 Absolute ceiling of 500 pound light airplane.

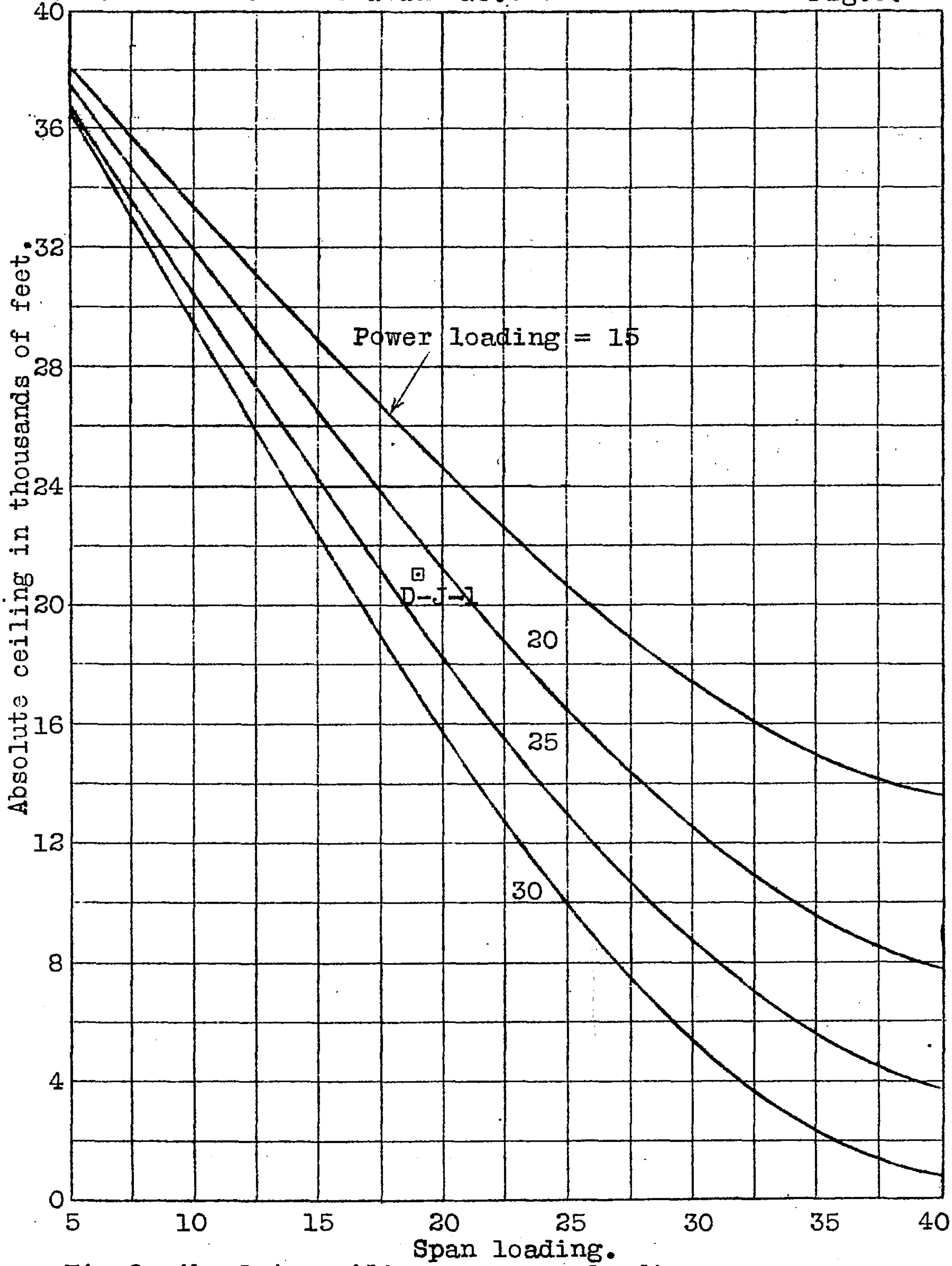

Fig.8 Absolute ceiling vs. span loading.

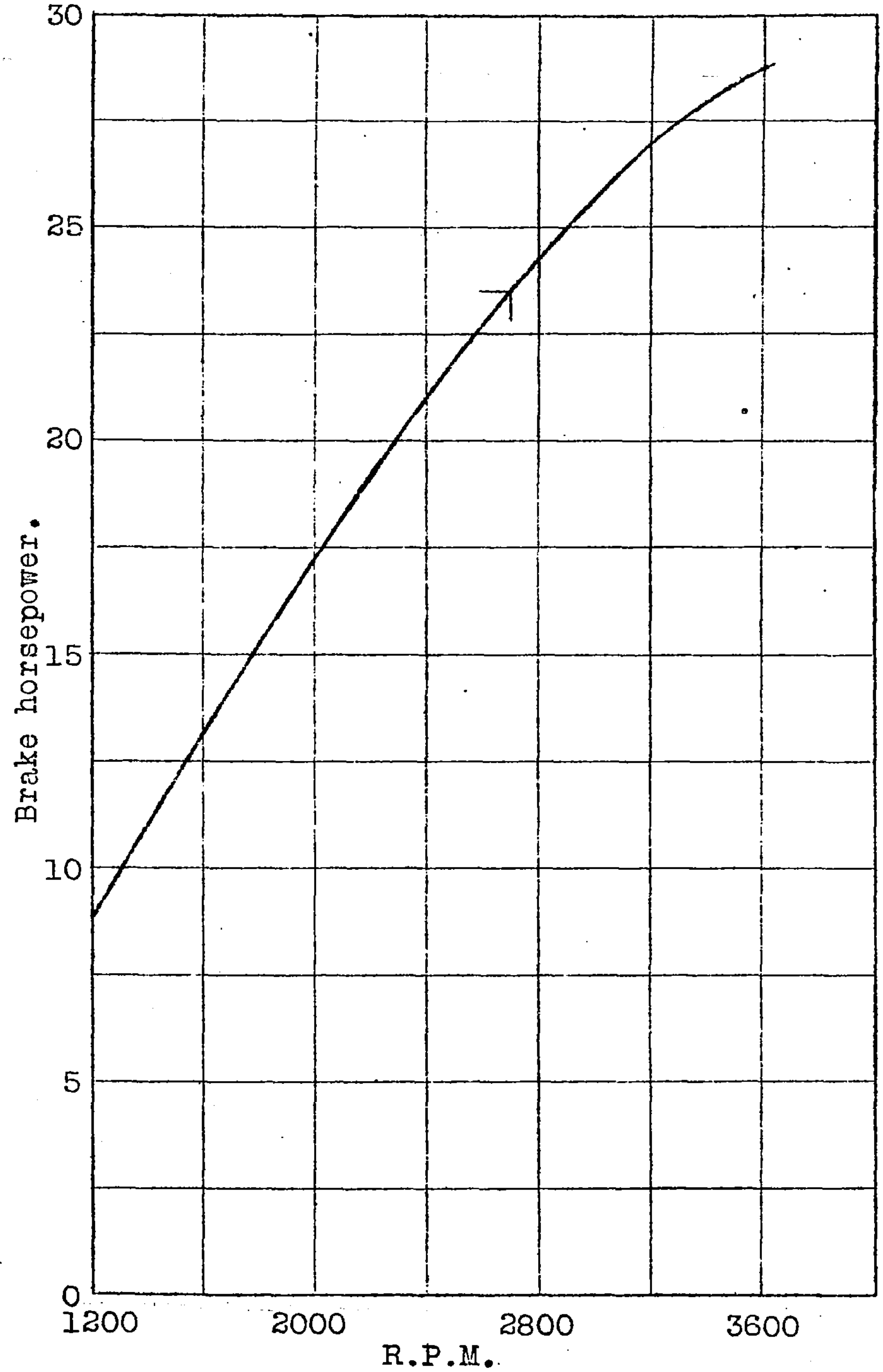

Fig.9 Power curve of Henderson "DeLuxe" engine.

Enlarge to 10" Photostat

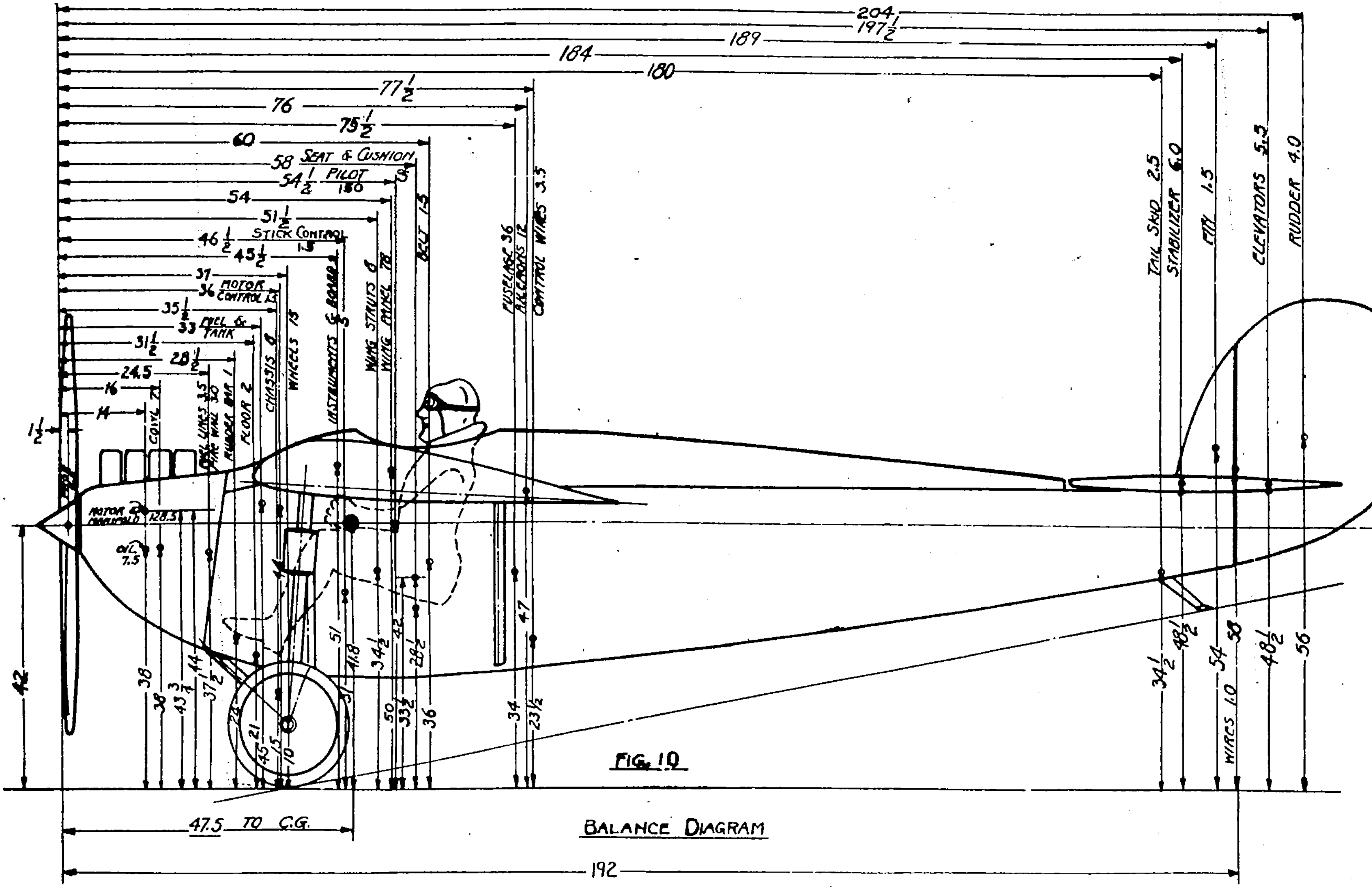

FIG. 10

BALANCE DIAGRAM

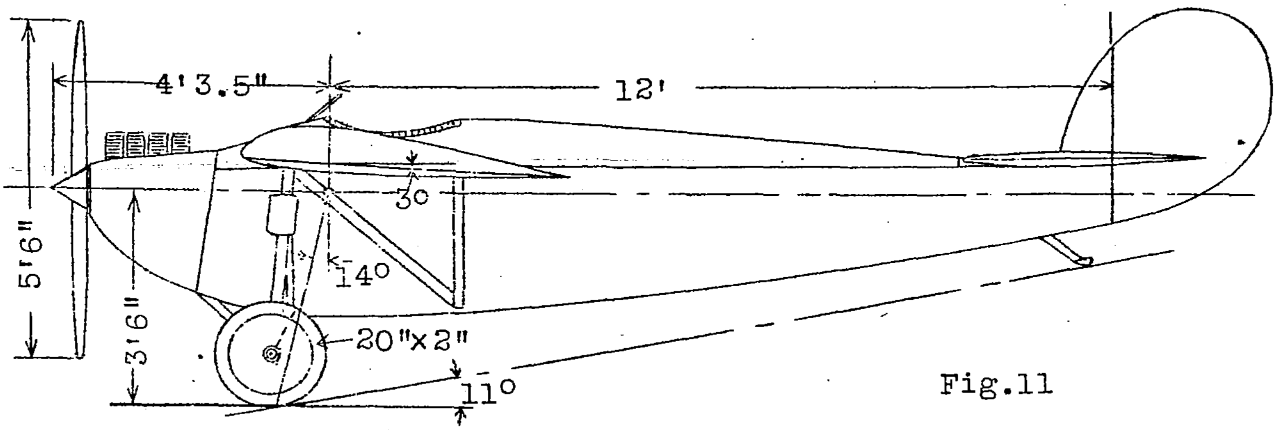

Fig.11

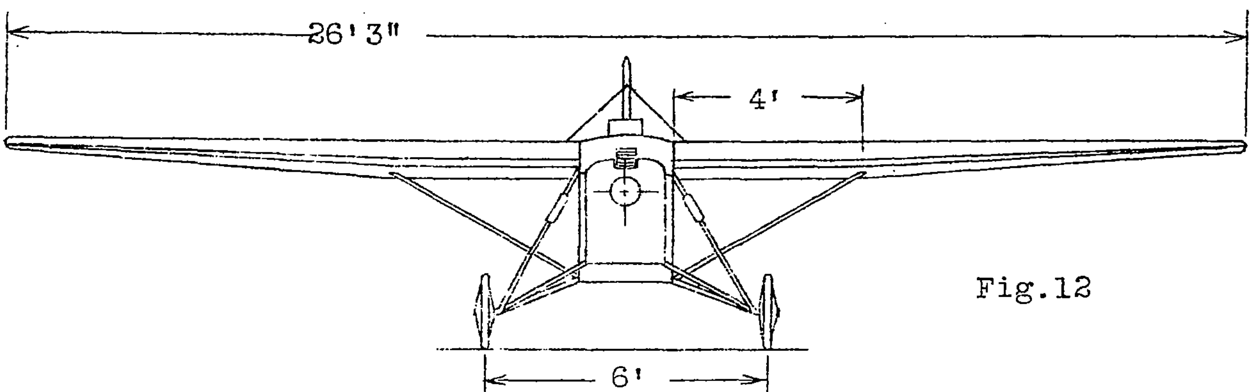

Fig.12

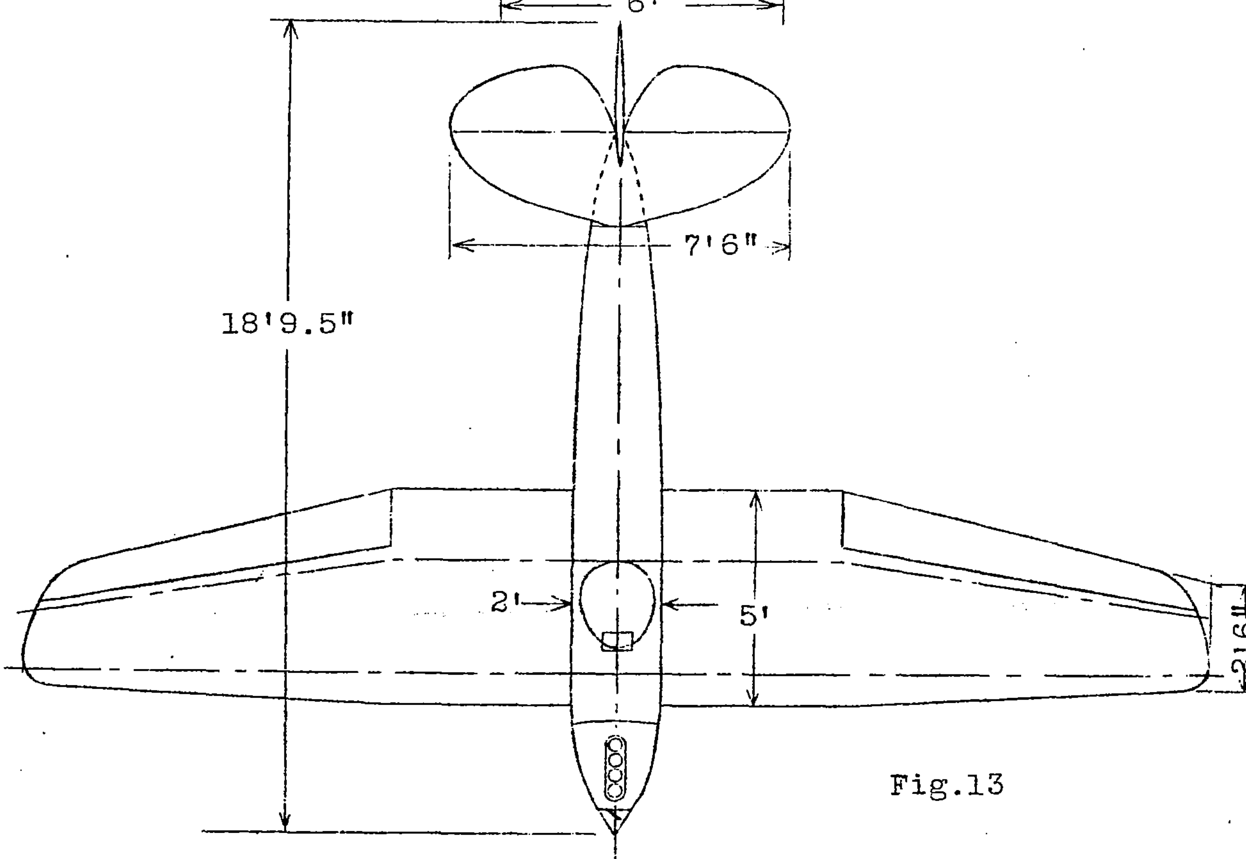

Fig.13