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CRANFIELD

POST NUBES LUX

INITIAL AIRCRAFT WEIGHT PREDICTION

by

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Initial Aircraft Weight Prediction.

-by-

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SUMMARY

Although much has recently been published concerning methods for the weight prediction of individual structural components, it is some considerable time since a review of the subject as a whole has appeared. This note is an attempt to remedy the situation, the scope of the work not being limited to aircraft structures but including also equipment and systems.

Various stages in the design process are considered, commencing with simple formulae suitable for use in the operations system stage, passing through the more detailed project stage and finally recommending methods suitable for weight prediction as the design becomes established.

In some instances the suggested formulae are well known, but an effort has been made to utilise the latest available information in bringing techniques up to date and where possible to fill in gaps, particularly with respect to equipment and systems.

CONTENTS

Notation.

1.0 Introduction.

2.0 Structure.

- 2.1 Wings
- 2.2 Fuselage
- 2.3 Tail Unit
- 2.4 Nacelles and Pods
- 2.5 Tail Booms
- 2.6 Undercarriage
- 2.7 Floats
- 2.8 Flaps

3.0 Power Plant.

4.0 Systems.

- 4.1 Flying Controls
- 4.2 Hydraulics, Electrics and Pneumatics
- 4.3 De-icing
- 4.4 Pressurisation and Air Conditioning
- 4.5 Fuel System

5.0 Equipment and miscellaneous items.

- 5.1 Seats and Furnishings
- 5.2 Intruments
- 5.3 Radio and Radar
- 5.4 Fire Precautions
- 5.5 Oxygen Equipment
- 5.6 Paint

References.

Appendix - Standard weights for crew and passengers.

Notation.

A	Aspect Ratio.
b	Wing Span (feet).
B	Maximum Fuselage Width (feet).
C	Terms in wing weight formula, see Eq.(1).
d	Tyre diameter (inches).
D	Term in formula for wing weight, see Eq.(2).
F	Ratio of Ultimate Tensile Strength of wing material at 15°C to that at temperature corresponding to M.
f	Drag factor in brake weight formula, see paragraph 2.6.
g	Location of inertia axis aft of leading edge as ratio of chord but not less than 0.4.
h	Maximum fuselage depth (feet).
H	$\dfrac{\cos \Lambda}{0.9(1+\frac{0.8}{A})}$ for $A < 3$ or $\cos^{3/2} \Lambda$ for $A > 3$
k	Taper Ratio correction, see Fig. 1.
l	Distance between the ground line with undercarriage leg extended and mean point at which load is transferred to the airframe, parallel to the leg (inches).
L	Ratio of the Shear Modulus of the wing material at 15°C to that at temperature corresponding to M.
L_T	Tail Arm (feet).
M	Mach Number corresponding to V_D.
N	Maximum factored normal acceleration.
n	Number of brakes.
p	Tyre pressure (lbs/sq.in).
P	Max. factored resultant reaction, normal to undercarriage leg (lbs).
Q	Factor in Wing formula, see paragraph 2.1.
r	Relief Factor = 1 - R/W.
R	First Moment of Relief Loads divided by 0.2b (lbs).
s	Number of engines.
S	Wing Area (sq.ft).
S_F	Gross fuselage surface area (sq.ft).
S_H	Gross horizontal tail area (sq.ft).

Notation (continued).

S_V	Gross vertical tail area (sq.ft).
t	Flight time in hours.
T	Engine thrust (lbs).
(t/c)	Thickness chord ratio of wing at root.
u	Wheel rim diameter (inches).
U	Maximum dynamic wheel reaction.
V	Tank Volume (gallons).
V_D	Design diving speed (f.p.s.).
V_S	Stalling Speed (f.p.s.).
w	Tyre width (inches).
W	Take off weight (excluding overload or boosts).
W_B	Brake Weight (lbs).
W_F	Fuselage Weight (lbs).
W_H	Weight of Horizontal Tail (lbs).
W_L	Landing Weight (lbs).
W_{MU}	Weight of Main Undercarriage (lbs).
W_{NU}	Weight of Nose Undercarriage (lbs).
W_T	Tyre Weight (lbs)
W_{TU}	Weight of Tail Undercarriage (lbs).
W_V	Weight of Vertical Tail (lbs).
W_W	Wing Weight (lbs).
$\left.\begin{array}{l}\alpha\\ \beta\end{array}\right\}$	Terms in fuselage weight formula, see Eq.(3).
γ	Sweepback of Structure.
Λ	Leading edge sqeepback.
ϕ	Sweepback of 0.25 chord.
λ	Taper ratio - tip chord/root chord.

1. Introduction.

It is not necessary to emphasise the importance of correct weight estimation in the early design stages of an aircraft. Although a number of contributions to the subject have been forthcoming in recent years, these have been mainly applicable to a particular structural component and it is some considerable time since the overall aspect has been analysed. The purpose of the present work is to review recent contributions to the problem of weight estimation, to comment upon their applicability, and particularly to suggest simple methods for use during the operations system investigation and initial project design stages.

It must be appreciated that the more simple a method for weight estimation is the less accurate it is likely to prove, and hence, in general, a more elaborate method should be used to confirm initial predictions as soon as sufficient information becomes available.

2. Structure.

The aircraft structure constitutes a relatively large proportion of the total weight and is to a large extent under the control of the design team. Although various methods are used for structural weight estimation, probably the most useful is that sometimes known as the 'weight penalty' concept. Briefly this consists of an estimate of the weight of a basic or 'ideal' structure, together with the penalties incurred in transforming it to an actual structure, i.e. allowance for cutouts, joints, controls, etc. This method requires a fairly detailed knowledge of the aircraft layout, and can only be applied satisfactorily in the later project design stages. Previous to this it is necessary to rely upon empirical, or semi-empirical formulae based upon past experience.

2.1. Wings.

In a conventional aircraft the largest individual contribution to the structural weight comes from the wings. Because of this, and the fact that the structural function can be defined fairly simply, wing weight estimation has received much more attention than any other component. The bibliography provides reference to numerous works concerned with wing weight estimation ranging from simple results derived from statistical analysis to theoretical methods requiring a detailed knowledge of design.

For a given all up weight, wing weight depends upon various factors. The most important are wing loading, thickness to chord ratio, aspect ratio, taper ratio, sweepback, design diving speed, maximum normal acceleration and relief loads. During investigation of the operations system stage of an aircraft design, where the main object is to decide upon the best type of aircraft to suit the system and to formulate requirements, only a vague idea of these parameters is likely to be available. Probably the most important single parameter is the wing loading and the following simple formula gives wing weight in terms of wing area and all up weight.

$$W_W = C(25 + 0.08W) \qquad \ldots \quad (1)$$

C is a function of the 'design efficiency', and has the following approximate values.

$C = 0.8$ for a structure having no discontinuity or extreme $t/_c$ or aspect ratio, but having a large relief load.
$C = 1.0$ for an 'average' wing.
$C = 1.15$ for a wing structure with numerous cut outs, low $t/_c$ or very high aspect ratio, folding wings, etc.

It will be immediately appreciated that the accuracy of the formula, which is based upon an analysis of a large number of existing aircraft, is dependent upon the value of C used. As is always true in the art of weight estimation, experience must play a large part.

It is interesting to note, however, that an adverse value of one parameter usually occurs with a correspondingly improved value of another parameter which, partially at least, tends to offset the first effect. For example, the association of low aspect ratio with very thin wings. Because of this, variations from the 'average' are not generally unduly great.

The influence of the various parameters is such that the wing structure may be designed by either strength or stiffness requirements, or in certain cases by both over different parts of the span. In this last case it appears that the structural weight is very similar to that necessary to meet the more dominant requirement over the whole span. A more accurate formula than that given in Eq.(1) which allows for variation of the important parameters when these become known in the early project stage is :-

$$W_W = k\left\{\frac{0.16b.S.(^t/c)}{A} + QW. \times 10^{-2} + D + 0.9S\right\} \quad . . . (2)$$

$$\text{where } D = F.b.WNr \times 10^{-6}\left\{\frac{3A\sec^2\phi}{(^t/c)} + 70\sec\phi + 15\right\} \text{ if}$$

the wing is designed on strength

$$\text{or } D = 3L.b^3. \cos\gamma \times 10^{-10}\left\{\frac{V_D. g . H}{(^t/c)(1-0.166M\cos\Lambda)}\right\}^2 \text{ if}$$

the wing is designed on torsional stiffness.

This formula is based on one suggested by Grinsted[1] in 1948 but has been modified to incorporate a later flutter stiffness requirement[19] and certain other refinements. F and L are functions incorporated to make an approximate allowance for kinetic heating effects, H is a function of aspect ratio and sweepback, and g of inertia axis location.

The first term in Eq.(2) estimates rib weight and the last the weight of leading and trailing edge fairings etc. The term incorporating the function Q is an allowance for discontinuities, joints, cut outs etc.

$Q = 1.5$ for an unbroken structural box, incorporating bonding etc.

$Q = 2.0$ for an average design with no large cutouts.

$Q = 2.5 - 3.0$ for a design with large cutouts, folding wing etc. (The higher value if the fold is near to the root).

On low speed aircraft where the spars react most of the bending loads and the skins only provide stiffness, it is necessary to add both terms for D into the formula.

The other symbols are defined in the notation.

Application of the formula to a number of recent aircraft wings has resulted in agreement of the order of $\pm$ 5% being obtained, but more evidence of accuracy is desirable, especially as it does not cover the aileron reversal case.

It should be noted that both Eqs. (1) and (2) assume that the wing bending moment is not transferred to the body, i.e. the structural box is unbroken across the centre section. If this is not the case, the wing itself will be lighter but the fuselage will suffer a consequent penalty, the value of which is discussed later, (see Eq.(5)). The total wing and fuselage weight is approximately independent of the type of joint.

As soon as the wing design and layout has been finalised, it is desirable to check the estimated wing weight by a more elaborate method. Such a method consists of estimating the weight of each structural component required, together with the weight penalty allowance. Examples of such methods are those of Ripley (5), Burt (8), Micks (3), Hyatt (6) and Hamitt (7). Of these, that proposed by Burt (8) is probably the most suitable for aircraft designed to meet British requirements.

2.2. Fuselage.

The fuselage presents a much more complex problem than the wing as the weight depends to a far greater extent upon detail considerations, particularly cut outs, windows and doors. In order to obtain a weight estimate with the desired accuracy it is necessary to analyse a fuselage in detail.

It would appear that there is a connection between fuselage weight and volume, or surface area, for a given all up weight, although this relationship is much less defined than in the case of wings. The following relationship is suggested as being suitable for an initial estimate of fuselage weight.

$$W_F = \alpha S_F + \beta W \qquad \ldots (3)$$

where $\alpha = 0.4$ and $\beta = 0.062$ for a passenger aircraft.

$\alpha = 0.9$ and $\beta = 0.062$ for a freighter or transport aircraft.

$\alpha = 0.65$ and $\beta = 0.062$ for fighters, bombers and trainers.

$\alpha = 0.65$ and $\beta = 0.085$ for flying boats.

$\alpha = 0.4$ and $\beta = 0.038$ for aircraft with nose piston engines and no large cut outs.

Again it will be seen that experience is desirable in the application of this formula, and at best only a rough approximation suitable for the operations system analysis is obtained.

A more accurate method of analysis, which requires a more detailed knowledge of the fuselage as well as considerable experience is that proposed by Ripley (12). The weight of a 'basic fuselage' is calculated from the formula :-

$$W_F = 6.5\, V_D^{0.5} \left\{ 1.85 + \frac{L_T}{B+h} \right\} S_F^{1.2} \,.\, 10^{-3} \text{ lbs.} \qquad \ldots (4)$$

An additional weight penalty must be added for cutouts, and special features such as engine mounting, ducting and reinforcement for arresting etc. The penalty for a cutout is given as 0.5 to 2.0 times the weight of the uncut surface, the higher value being for small cutouts. The weight penalties for engines and arresting reinforcement are each 0.003W.

Other simple methods of fuselage weight estimation are referred to in the bibliography.

The weights estimated from Eqs. (3) and (4) allow for shear attachment of the wings, but not for a bending attachment. The additional penalty is given by :-

$$8W.N. \times 10^{-4} \text{ lbs.} \qquad \ldots (5)$$

The wing weight will be correspondingly less.

When sufficient information is available, the fuselage weight should be estimated by a more detailed analysis of the type suggested by Micks (10), Hammitt (7), or Burt and Phillips (13), the latter probably being the most suitable. This method analyses the fuselage in detail and has given very good results in practice, although it is a fairly lengthy process.

2.3. The Tail Unit.

The weight contribution of the tail unit is relatively small, but as it occurs at an extremity of the aircraft it is critical in balance calculations. For initial work it is sufficient merely to express tail unit weight as a function of the all up weight, the following relations being typical :-

$$\left.\begin{array}{ll} \text{Tailplane and Elevator :-} & W_H = 0.015W \\ \text{Fin :-} & W_V = 0.008W \\ \text{or Fin with high mounted tailplane :-} & W_V = 0.013W \end{array}\right\} \qquad \ldots (6)$$

A rather more accurate estimation than Eq.(6), which can be used when the design is more detailed has been suggested by Ripley (15) :-

$$\left.\begin{aligned} W_H &= 1.04S_H^{1.2}\left(0.4 + \frac{V_D}{840}\right) \\ W_V &= 1.03S_V^{1.2}\left(0.4 + \frac{V_D}{1000}\right) \end{aligned}\right\} \quad \ldots \quad (7)$$

(when tail is not mounted on the fin)

These formulae are based essentially on bending strength requirements and so may be in error for thin high speed tail units. It is possible that in this case the application of the wing formula Eq.(2) using the torsion value of D would give better results, although there is no evidence to confirm this. The bending term could also be used by replacing the value of W.N.r in Eq.(2) by the maximum tail load derived from initial balance calculations.

Very little work has been carried out on this problem of tail unit weights but an alternative value based purely on statistical evidence has been given by Driggs (14) in a very comprehensive analysis of aircraft performance. The tail unit weight is based on speed and area only as in Ripley's work.

2.4 Nacelles and Pods.

Nacelles can be either simple fairing structure when the engines are mounted off the wing or in the case of forward mounted engines, load carrying. In either instance, the problem of weight estimation is difficult, but the total weight involved is not normally great.

Nacelle fairings usually amount to about 0.02W or 2 to 3 lbs per sq.ft of nacelle. Long nacelles carrying the engines usually contribute about 0.03W or 4 lb/sq.ft.

Driggs (14) makes an attempt to correlate nacelle weight with engine performance, but the evidence is very scanty.

Pod mountings usually weigh about 0.18 of the engine weight.

2.5. Tail Booms.

Although tail booms are structurally good, they have a small depth in comparison to the load they are required to carry and are consequently heavy. A figure of 0.025W to 0.03W or 5 - 6 lb/sq.ft is typical. The fuselage associated with the tail booms will be lighter than the conventional configuration and the total weight of booms and fuselage will not vary greatly from a conventional arrangement, providing the fuselage does not extend an appreciable distance behind the wing.

2.6. Undercarriage.

The performance of an undercarriage is defined by relatively simple parameters and hence it is amenable to theoretical treatment when knowledge of the geometry and details are available. Unfortunately however, this information does not become available until the rest of the aircraft is decided upon and for purposes of system evaluation it is necessary to base assumptions upon statistical results. The most important factor in the design of the undercarriage apart from the all up weight is the vertical velocity of descent. This is particularly high for naval aircraft and hence these undercarriages are relatively heavier.

$$\left.\begin{aligned} W_{MU} &= 0.037W \quad (0.044W \text{ for naval aircraft}) \\ W_{NU} &= 0.007W \quad (0.01W \text{ for naval aircraft}) \\ W_{TU} &= 0.003W \quad (0.005W \text{ for naval aircraft}) \end{aligned}\right\} \quad \ldots \quad (8)$$

These values are only approximate and will vary with parameters such as undercarriage length, reaction factor, rake and tyre pressure, and in this connection experience must play a big part.

An attempt by Burt and Ripley (16) to evaluate undercarriage weights more thoroughly has proved unreliable in recent applications due to the fact that it was based on low tyre pressures and outdated requirements. The situation has been remedied by a recent report of Phillips (17). It will be appreciated that at the initial project design stage, tyre and wheel sizes are often not available and thus the first part of the weight estimation process is to decide upon these details. Phillips presents some curves giving tyre load carrying capacity in terms of the product of tyre diameter and width for an 'average' range of tyres ($w = 0.29d$), and various tyre pressures. Tyre plus tube weight follows as a function of the same parameters. The wheel size is estimated from the average value $u = 0.48d$ and the weight of the wheel is given in terms of this. The brake weight is given as a function of the kinetic energy per brake, and the total wheel unit weight is the sum of the three quantities. Normal and emergency cases are considered.

As an alternative to this procedure Figs. 2 - 6 can be used. Fig. 2 gives the required tyre diameter for a given load and tyre pressure and is based on a particular family of tyres which makes allowance for the tendency of high pressure tyres to have reduced widths. The values of these widths and the rim diameters are given in Figs. 3 and 4. Should it be necessary to use a wheel of reduced width the new diameter should be estimated on the basis of a constant product of width and diameter but the subsequent operations should be based on the original diameter. The static load must not exceed one third of the maximum dynamic load U.

The weight of the tyre plus tube is estimated from Fig. 5 and the brake weight from Fig. 6. This last figure is given as a function of aircraft stalling speed and a drag factor f.

$f = 1.0$ for a very clean nosewheel aircraft with no drag flaps.

$f = 0.7$ for a tailwheel aircraft with drag flaps, with intermediate values in between.

The total wheel unit weight is then given by

$$\left.\begin{array}{ll} \text{For braked wheels :-} & 1.43\ (W_T + W_B) \text{ per wheel} \\ \text{For unbraked wheels :-} & 1.6\ W_T \text{ per wheel} \end{array}\right\} \quad \ldots (9)$$

Phillips gives the weight per inch of the undercarriage structure in terms of the resultant factored load normal to the undercarriage leg, P. Thus this value takes account of reaction factor and rake. The curve given is approximately defined by :-

$$0.00127\,\ell\,.\,P^{0.78} \quad \ldots (10)$$

where ℓ is the distance parallel to the leg from the ground line with the leg extended to the point where the undercarriage load is transferred to the airframe (in inches).

The total weight of the undercarriage is the sum of the wheel and structure weights plus an allowance of 2% for miscellaneous parts and a further 5% if the unit is retractable. The estimate does not include the weight of bogies which amount to approximately 0.003W, and when these are used this must also be added in to get the total undercarriage weight.

2.7. <u>Floats.</u>

There is very little evidence available on float weight, but what there is indicates the following values for flying boats :-

$$\left.\begin{array}{ll} \text{For fixed floats :-} & 0.01W \\ \text{For retractable floats : -} & 0.015W \end{array}\right\} \quad \ldots (11)$$

2.8. <u>Flaps.</u>

This is included in the wing weight, paragraph 2.1 but for detailed estimates see Burt (18).

3.0. <u>Power Plant.</u>

It is normal for project designs to be based on engines which, at least, are fairly advanced paper studies and thus a power plant weight is available, together with certain accessories. During the operations system analysis however, it is convenient to use more general information and therefore typical values of power plant weight are given as functions of performance for recently developed engines.

Turbojet :-	0.2T	
Propjet :-	0.5(E.H.P.)lbs (excludes propellers)	... (12)
Piston Engine :-	200 + 1.04 (H.P.)lbs (excludes propellers)	

These figures are based on maximum performance, but exclude special boosting, e.g. methanol injection and reheat.

Propeller weight is normally approximately given by :-

$$0.24\ (\text{H.P.})\ \text{lbs.} \qquad \ldots (13)$$

The engine installation weight is usually of the order of 10% - 20% of the power plant weight.

4.0. Systems.

The aircraft systems, i.e. flying controls, hydraulics, electrics, de-icing, pressurisation and air conditioning, and fuel system vary considerably from one aircraft to another, and it is a very difficult problem to attempt an accurate weight estimate. The best method is to use the values of an existing similar type, but unfortunately these are not always available. Certain very approximate suggestions can be made to cover this case.

4.1. Flying Controls.

The flying control system is taken to include flap operation. The weight of this system is given approximately by :-

For large airliners and transport aircraft :- $35 + 0.008W$

For other types :- $35 + 0.005W$ . . . (14)

These figures do not include powered operation of the controls. The little evidence available indicates that when this is used the term independent of W should be increased to about 100. Large transport aircraft are usually equipped with complicated flaps and this accounts for the higher weight.

4.2. Hydraulics, Pneumatics and Electrics.

The weight of a hydraulic system naturally depends upon the number of services operated. The same applies to pneumatic and electrical systems. To some extent they are inter-related in that for a given type of aircraft the total number of services operated will be similar. For this reason they are considered together here, but even so there is a large variation in apparently similar types. Exclusive of de-icing and assuming that every effort is made to reduce weight - e.g. high pressure hydraulic system, lightweight cables etc., the total weight of these power systems usually falls in the range 2% - 4% of the total aircraft weight. Any special features, such as a complex radar installation, will of course tend to increase this value.

4.3. De-icing.

Again there can be considerable variation, but for full de-icing of the airframe, the penalty is about 0.8% of the total weight or :-

$$500 + 0.003W \qquad \ldots (15)$$

which ever is the lower.

4.4. Pressurisation and Air Conditioning.

Insufficient evidence is available to enable any recommendations to be made for high speed aircraft which experience kinetic heating. Each case must be treated on its merits.

For passenger aircraft the weight of the equipment is approximately 2% of the aircraft weight or :-

$$700 + 0.002W + 100t \qquad \ldots (16)$$

which ever is the lower.

4.5. Fuel System.

The weight of the fuel system is very dependent upon the number of power plants and tanks and their relative locations. Hence it is very difficult to generalise. The following figures indicate the order of fuel system weight to be expected :-

$$\left.\begin{array}{ll} \text{For civil airliners and transport aircraft :-} & \\ & 0.016W - 0.02W \\ \text{For other types up to:-} & 0.03W \end{array}\right\} \quad \ldots (17)$$

depending upon the complexity of the system.

These figures include tank weight.

The weight of the tanks themselves can be estimated from :-

$$\left.\begin{array}{ll} \text{For plain metal tanks} & 10 + 0.69V \\ \text{Crash proof tanks (military flexible)} & 6.5 + 0.38V \\ \text{Flexible civil tanks} & 10 + 0.12V \\ \text{Drop tanks} & 1.0V \end{array}\right\} \quad \ldots (18)$$

where V is tank capacity in gallons.

The tank plating for flexible tanks usually weighs about the same as the tank.

Residual fuel and oil average at 5% of tank weight.

5.0. Equipment and miscellaneous items.

These items can amount to a substantial proportional of the total weight and are often very difficult to predict. In the case of military aircraft it is essential to use experience gained from a comparable type or preferably the actual requirements when these are available. It is possible to be a little more definite in the case of civil aircraft.

5.1. Seats and Furnishings.

Ejector seats of the older type weigh about 200 lbs and the more recent lightweight ones about 100 lbs. The ordinary crew seats have a weight of approximately 30 lbs and the special lightweight military transport type 18 lbs.

The weight of passenger seats in civil aircraft varies according to the type and duration of flight. On aircraft intended for short flights, the seat weight is usually about 25 lbs but for longer flights 35 lbs is nearer the average value. Special lightweight seats of recent design weigh as little as 21 lbs per passenger for reasonable luxury.

It is often convenient however to consider seat and furnishing weight together. A n analysis of recent large airliners suggests the following values for total furnishing weight :-

$$\left.\begin{array}{ll}\text{First class accommodation :-} & 60 + 12t \text{ lb/passenger} \\ \text{Tourist accommodation : -} & 25 + 12t \text{ lb/passenger}\end{array}\right\} \quad . . . (19)$$

The higher first class figure is not only a function of greater luxury, but also of lower seating density. The exclusion of seat weight appears to have little effect upon the time variable term and yields :-

$$\left.\begin{array}{ll}\text{First class accommodation : -} & 3 + 12t \text{ lb/passenger} \\ \text{Tourist accommodation :-} & 35 + 12t \text{ lb/passenger}\end{array}\right\} \quad . . . (20)$$

5.2. Instruments.

The weight of instruments on a small trainer or communications aircraft is of the order of 40 lbs rising to about 60 lbs on the more advanced types.

In the case of airliners the weight is usually in the region of 350 lbs - 450 lbs mainly dependent upon range.

5.3. Radio and Radar.

A simple radio installation incurs a direct weight penalty of about 35 lbs. On more advanced aircraft of the trainer type the radio installations will probably weigh of the order of 200 lbs. The installation in airliners is more extensive and accounts for 800 lbs - 1000 lbs, the higher value applying if some form of radar is used.

5.4. Fire Precautions.

The fire precaution weight naturally depends very much on the engine and fuel system layout, but an average value for civil airlines is :-

$$(0.1 + 0.1s)W \times 10^{-2} \qquad \ldots (21)$$

where s is the number of engines.

5.5. Oxygen Equipment.

The following figures are based on recent lightweight oxygen containers :-

$$\left.\begin{array}{ll} \text{Wt. for one crew member for four hours} & \text{55 lbs.} \\ \text{Wt. for one passenger for four hours} & \text{20 lbs.} \end{array}\right\} \quad \ldots (22)$$

5.6. Paint.

Paint weight can vary considerably according to type, number of coats, etc., but an average value is 0.035 lb/sq.ft. of surface.

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Appendix 1.

Standard Weights of Crew and Passengers.

Military 200 lbs per man (including parachute but excluding pressure suit, etc.).

Civil

			Baggage for routes		
			Internal	Continental	Overseas
Passenger:	Male	165 lbs plus	33 lbs	or 44 lbs	or 66 lbs
	Female	143 lbs plus	33 lbs	or 44 lbs	or 66 lbs
	Child (2-12 years)	85 lbs plus	33 lbs	or 44 lbs	or 66 lbs
	Child (Under 2 years)	17 lbs plus	33 lbs	or 44 lbs	or 66 lbs
Crew:	Male	165 lbs plus	22 lbs	or 33 lbs	or 44 lbs
	Female	143 lbs plus	22 lbs	or 33 lbs	or 44 lbs

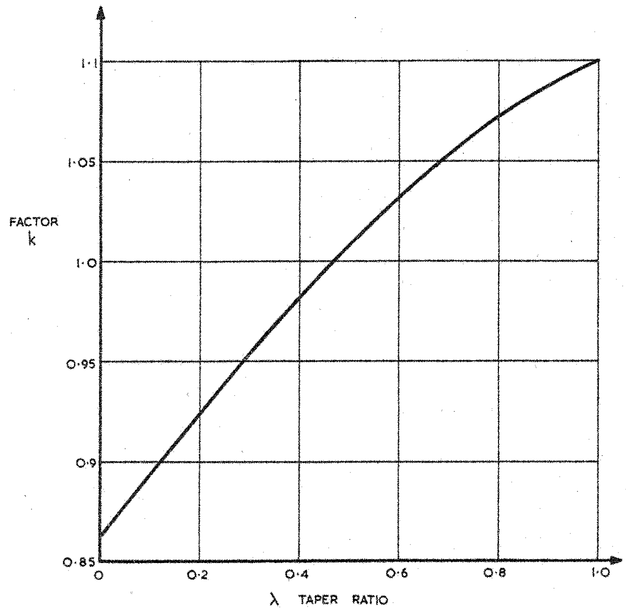

FIG. 1. TAPER CORRECTION FACTOR.

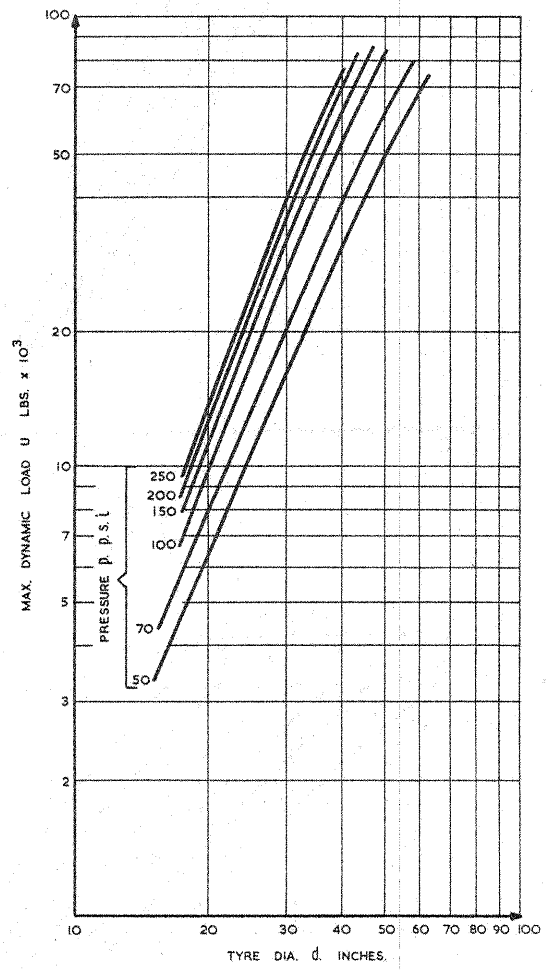

FIG. 2. PERMISSIBLE TYRE LOAD

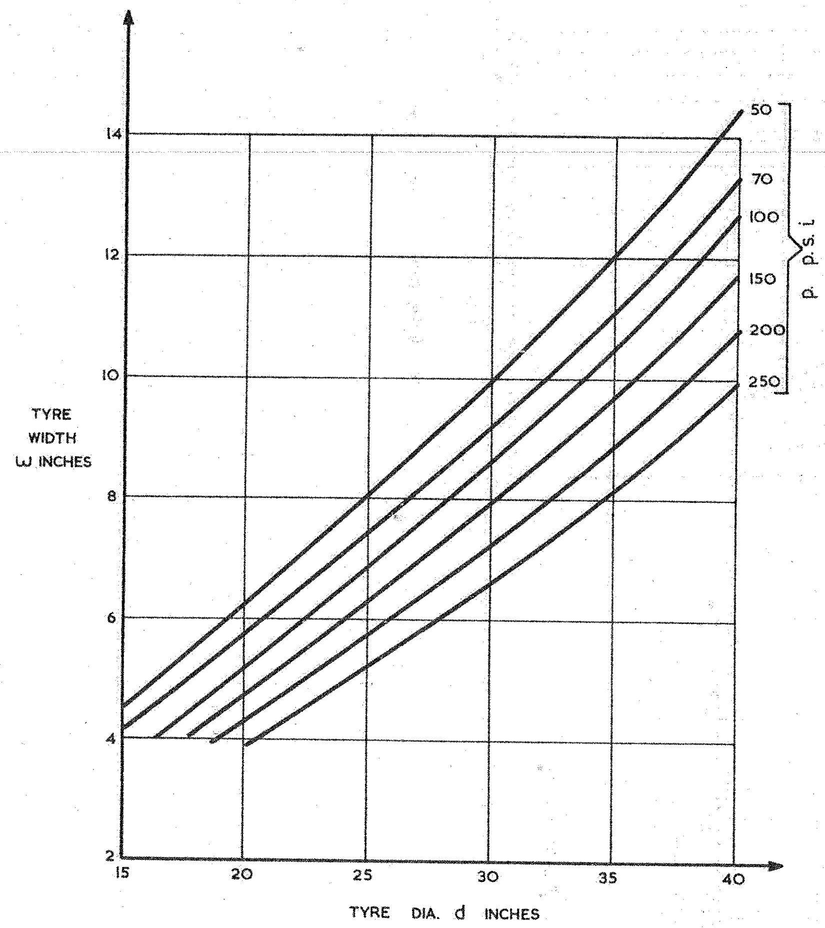

FIG. 3. TYRE WIDTH.

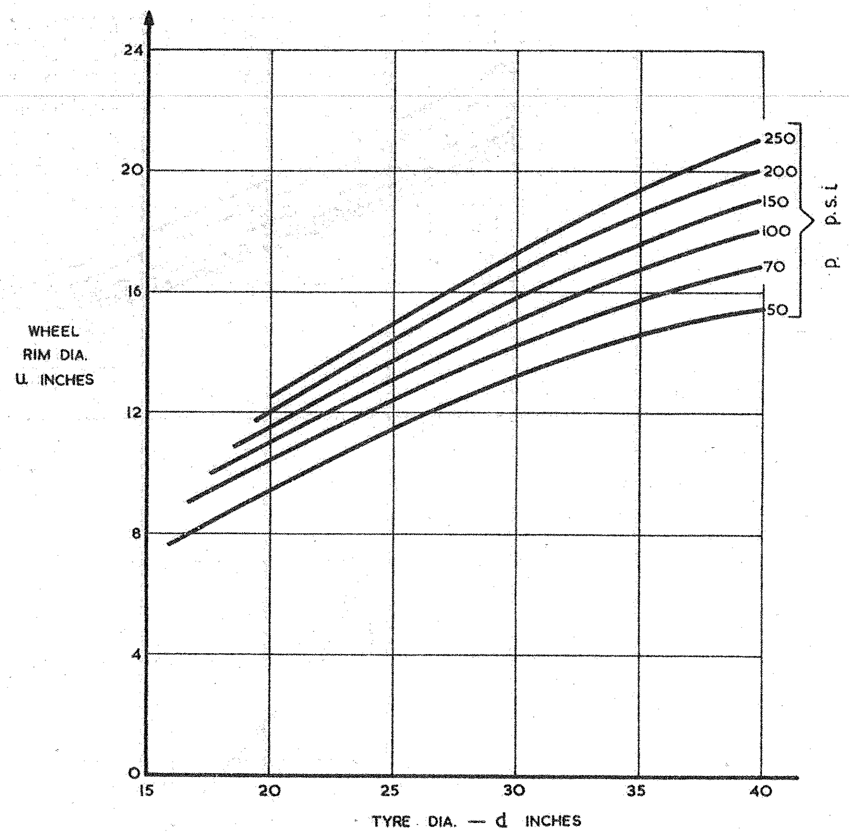

FIG. 4. WHEEL RIM DIAMETER

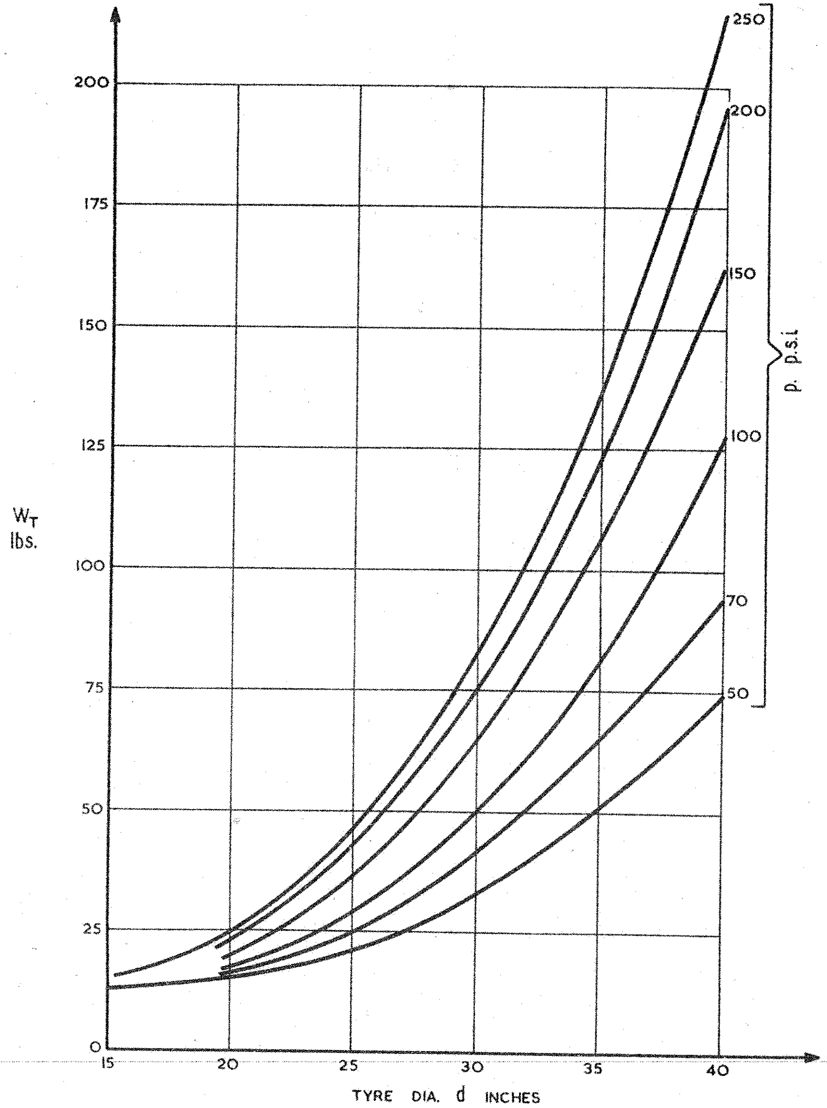

FIG. 5. TYRE WEIGHT W_T

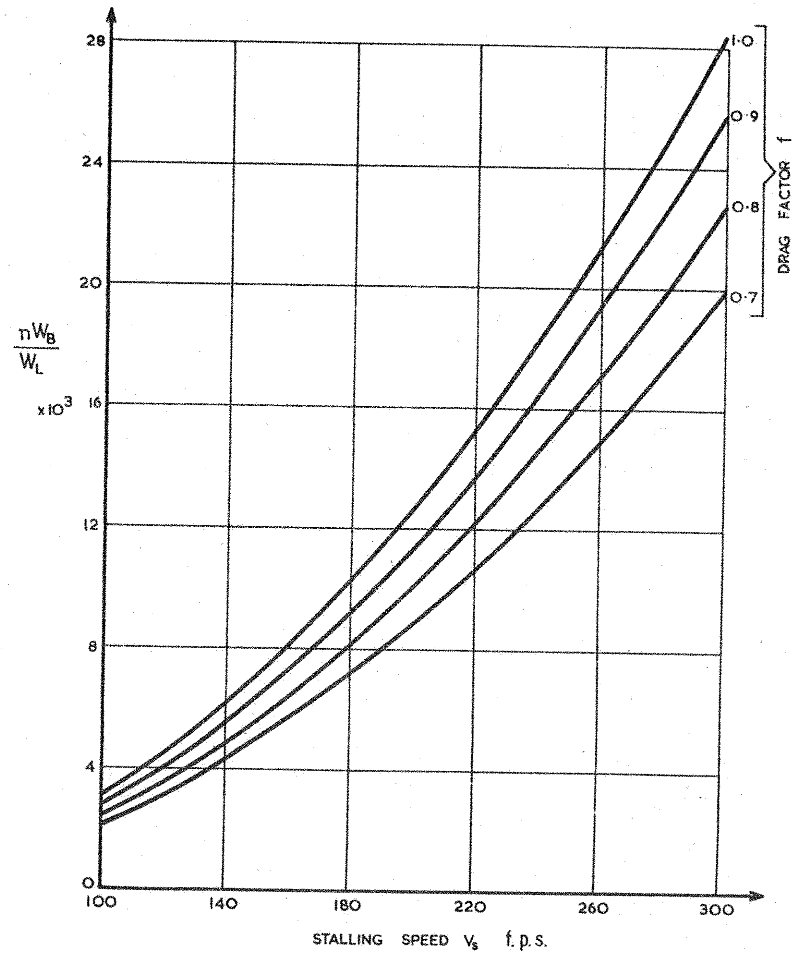

FIG. 6. BRAKE WEIGHT W_B