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A SIMPLE WING WEIGHT ESTIMATION EQUATION

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ABSTRACT

A form is derived for a wing weight estimation equation on the basis of a simple beam representation of an airplane semi-wing. Using configuration data available to WRRC, a statistical determination of the unspecified constants is made which minimizes the squares of the percentage errors. The errors in wing weight estimation using this equation are shown for samples of fighter, bomber, and cargo aircraft, and a comparison is made with the results obtainable from four different estimation techniques.

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GLOSSARY OF SYMBOLS

Symbol	Units	Definition
A_b	ft^2	Cross-section area of bending material
A_s	ft^2	Cross-section area of shear material
b	ft	Structural semi-span
c(x)	-	Spanwise chord distribution
e_i	-	Percent error in any given estimate
f(t)	-	Function of thickness
L(x)	lbs/ft	Spanwise load distribution
V(x)	lbs/ft	Spanwise shear distribution
M(x)	ft-bls/ft	Spanwise moment distribution
(b)	-	Conditions at free end of beam
(0)	-	Conditions at fixed end of beam
Q	lbs	Design load on semi-wing
r_b	-	Relief factor in bending
r_s	-	Relief factor in shear
rms	-	Root mean square error $= \left[\frac{1}{n} \Sigma e_i^2\right]^{\frac{1}{2}}$
R	ft	Radius of gyration
S	ft^2	Gross wing area
W_w	lbs	Weight of wing
W_{w_b}	lbs	Weight of bending material
W_{w_s}	lbs	Weight of shear material
x	ft	Station on the structural span measured from the aircraft centerline
y	ft	Distance from elastic axis to extreme fiber of beam

GLOSSARY OF SYMBOLS
(Continued)

Greek Symbols

Symbol	Units	Definition
δ		Thickness taper ratio
λ		Planform taper ratio
η		Ultimate load factor
Ω_K	degrees	Sweepback angle of K% chord line
ρ	lbs/in^3	Material density
σ_b	lbs/in^2	Allowable bending stress
σ_s	lbs/in^2	Allowable shear stress
$\theta(\ )$	-	Of the order of magnitude of ( )

INTRODUCTION

This is the first of a series of three papers concerned with the development of a wing weight estimation equation. Emphasis is placed upon a method of approach, derivation of the form of the equation, and a discussion of the estimations obtainable compared with those obtained from other established estimation equations. Reference 1 outlines the statistical techniques employed in fitting the derived equation to available configuration data. Reference 2 gives a method for determining the confidence limits which can be established on predictions made with the equation. The results presented in these papers are derived from work conducted under USAF contract AF 33(616)-199. This contract is under the jurisdiction of the Weights Section, Structures Branch, Aircraft Laboratory, Wright Air Development Center. Throughout these papers, all reference to security or proprietory information handled under the contract has been eliminated.

During the course of the contract for the development of feasible techniques for the estimation of aircraft structural and equipment weights, it was found that several stringent requirements were imposed upon the techniques which were developed. The feasibility and value of any structural weight estimation technique was judged on the basis of its ability to satisfy the following conditions:

1. Structural weight estimation must be possible on a wide variety of configurations with an rms error of less than ten percent.

2. The computation time required should be only a few minutes, once the necessary data are available, and a minimum of configuration data should be called for.

3. Judgment decisions should be eliminated, insofar as possible, to allow the technique to be used with a minimum indoctrination period.

4. The technique should have sufficient analytic foundation to allow extrapolation to new or unorthodox configurations.

There is a wide variety of weight estimation techniques which could be used, and which satisfy many of these requirements. They range from a simple statement of "wing weight equals 12% of design gross weight", to an elaborate detailed design of the wing structure. Some compromise between these extremes is necessary for the fulfillment of all the requirements listed. For the purposes of the contract work, it was decided to

derive a simple expression for the weight of wing bending and shear material using a simple beam model for the airplane semi-wing. Such an approach has been used many times with reasonable success. The major differences which will be found in this development are:

1. Airload distribution is assumed to be the average of an elliptical distribution and a linear one,

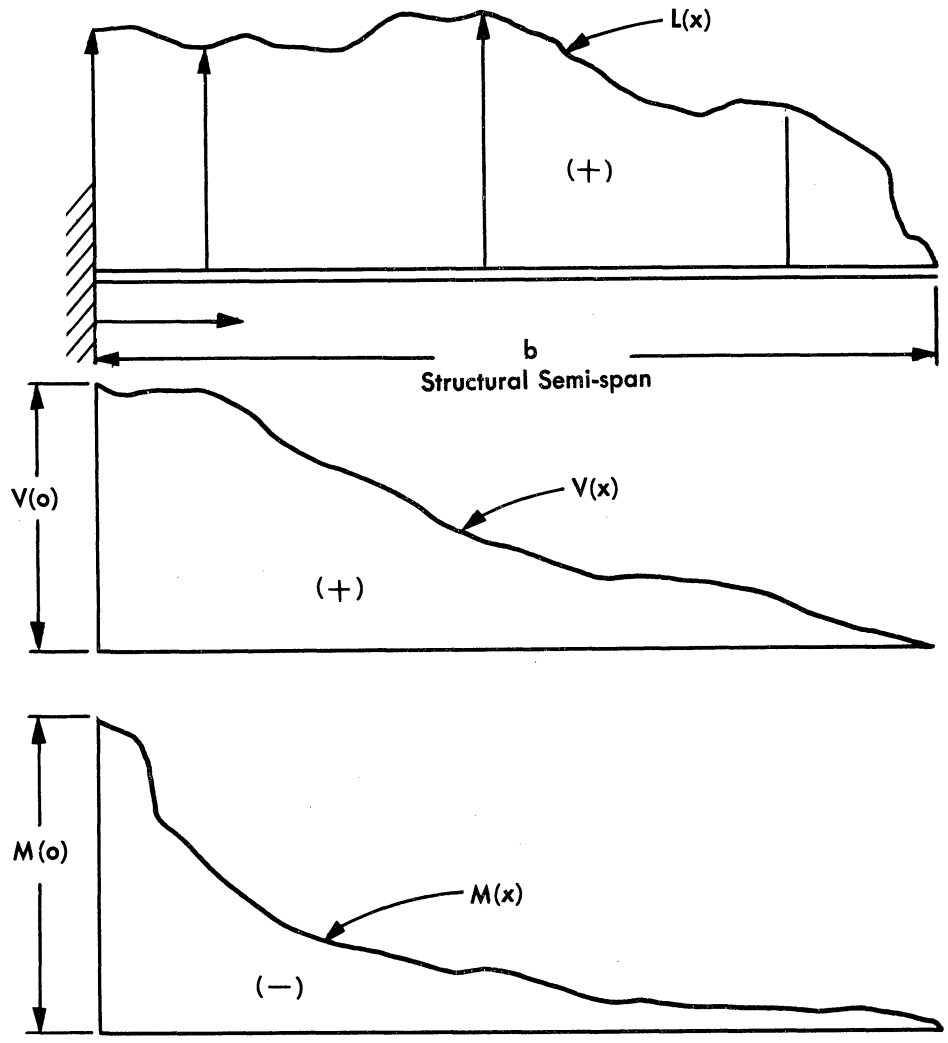
2. Structural and equipment dead weights are included in "relief factors" applied separately to estimates of shear and bending material weights.

I

DERIVATION OF A GENERAL FORM OF THE EQUATION

The following assumptions were made to serve as the basis for the derivation of a form for a wing weight estimation equation:

1. A semi-wing can be represented by a simple cantilever beam.
2. Load distribution is arbitrary.
3. Shear and Bending material are separable.
4. Same material is used throughout the wing, with allowable stresses constant over the semi-span.



The design load on the semi-wing, the shear distribution, and the bending-moment distribution can be expressed as follows in terms of the illustrated load distribution, L(x):

$$Q = \int_0^b L(x)\, dx \tag{1}$$

$$V(x) = \int_x^b L(x)\, dx \tag{2}$$

$$M(x) = -\int_x^b V(x)\, dx \tag{3}$$

From these definitions, the cross section areas of the shear and bending materials required at any station x are:

$$A_s(x) = \frac{|V(x)|}{\sigma_s} \tag{4}$$

$$A_b(x) = \frac{|M(x)\ y(x)|}{\sigma_b\ R^2(x)} \tag{5}$$

where:

σ_s = allowable shear stress

σ_b = allowable bending stress

y = distance from elastic-axis to extreme fiber

R = section radius of gyration.

The sum of these, multiplied by the appropriate material density, is the weight of shear and bending material per unit span at station x. Integrating over the span gives, for the weight of the semi-wing:

$$W_w = \frac{\rho}{\sigma_s} \int_0^b |V(x)|\, dx + \frac{\rho}{\sigma_b} \int_0^b \frac{|M(x)\ y(x)|}{R^2(x)}\, dx \tag{6}$$

This can be expressed as:

$$W_w = 2\rho \left[\frac{M(0)}{\sigma_s} + \frac{Y(0)}{\sigma_b} \right] \tag{7}$$

where

$$M(0) = \int_0^b V(x)\, dx \tag{8}$$

and

$$Y(0) = \int_0^b \frac{|M(x)\, y(x)|}{R^2(x)} \tag{9}$$

The remaining weight of the semi-wing consists of such items as stiffeners, ribs and formers, joining devices, and non-structural members. The weights of these items can be estimated from structural, geometric, or fabrication considerations, but are normally highly dependent upon the design and production practices used by any given company. Further, in weight data which are available, it is seldom possible to collect a large sample in which these weights are broken down in accurate and sufficient detail. For the purposes of this study, it will be assumed that this "miscellaneous" weight is proportional to the nominal area of the wing, multiplied by some function of thickness. The expression for the weight of the wing can then be written in the form:

$$W_w = a_1 \left[\frac{M(0)}{\sigma_s} + \frac{Y(0)}{\sigma_b} \right] + a_2\, S\, f(t) \tag{10}$$

A wing is required to carry not only applied airloads, but also inertial reactions. These are due to the presence of mass in the wing, when it is in an accelerated condition. At any station, the inertial reaction due to mass (or dead weight) acts opposite in direction to the vector sum of the applied forces. Using a simple beam model, in which the principle of superposition is valid, the beam required to carry the combination of these loads is equivalent to the beam required to carry the airloads minus the beam required to carry the inertial reactions. To show the effect of this on the derived equations, define:

$$L_a(x) = \text{applied airload distribution}$$

$$L_d(x) = \text{inertial load distribution} \tag{11}$$

from which:

$$L(x) = L_a(x) - L_d(x) \tag{12}$$

assuming that a positive inertial load acts opposite in direction to a positive airload. Substituting this relation in equations (1), (2), and (3) gives, respectively:

$$Q = \int_0^b \left[L_a(x) - L_d(x) \right] dx = Q_a - Q_d \tag{13}$$

$$V(x) = \int_x^b \left[L_a(x) - L_d(x) \right] dx = V_a(x) - V_d(x) \tag{14}$$

$$M(x) = - \int_x^b \left[V_a(x) - V_d(x) \right] dx = M_a(x) - M_d(x) \tag{15}$$

and, similarly,

$$\begin{aligned} M(0) &= M_a(0) - M_d(0) \\ Y(0) &= Y_a(0) - Y_d(0) \end{aligned} \tag{16}$$

From these, the expressions for the shear and bending material weights may be written as:

$$W_{w_s} = \frac{\rho}{\sigma_s} \left[M_a(0) - M_d(0) \right] = \frac{\rho r_s}{\sigma_s} M_a(0) \tag{17}$$

$$W_{w_b} = \frac{\rho}{\sigma_b} \left[Y_a(0) - Y_d(0) \right] = \frac{\rho r_b}{\sigma_b} Y_a(0) \tag{18}$$

where the relief function in shear is defined as:

$$r_s = 1 - \frac{M_d(0)}{M_a(0)} \tag{19}$$

and relief function in bending as:

$$r_b = 1 - \frac{Y_d(0)}{Y_a(0)} \tag{20}$$

This enables us to write the expression for wing weight in the form:

$$W_w = a_1 \left[\frac{r_s M_a(0)}{\sigma_s} + \frac{r_b Y_a(0)}{\sigma_b} \right] + a_2 \, S \, f(t) \tag{21}$$

Beginning with this equation, we can look at techniques for computing $M_a(0)$, $M_d(0)$, $Y_a(0)$, and $Y_d(0)$ on an approximate basis. Let us assume that the total aerodynamic load to be carried by the semi-wing is one-half the design gross weight of the airplane multiplied by the ultimate load factor, or:

$$Q = \frac{\eta W_g}{2} = \int_0^b L(x) \, dx \tag{22}$$

In general, the bending moment at the wing root due to this loading is:

$$M_a(0) = \frac{\eta W_g}{2} \bar{x}_a = \frac{\eta W_g b}{2} \bar{y}_a \tag{23}$$

where $\bar{y}_a$ is the distance to the centroid of the load distribution, expressed as a fraction of the wing semi-span. The general expression for this load centroid is:

$$\bar{y}_a = \frac{\int_0^b L_a(x) \, x \, dx}{b \int_0^b L_a(x) \, dx} \tag{24}$$

In the calculation of the root bending moment due to inertial loads, the important dead weights and their centroids are usually known. The root bending moment due to each dead weight can then be calculated, and the moments summed to give the effective value of $M_d(0)$. In equation form:

$$M_d(0) = \sum_i W_i x_i = W_d b \bar{y}_d \tag{25}$$

where

$$W_d = \sum_i W_i$$

and

$$\bar{y}_d = \frac{\sum_i W_i \left[\frac{x_i}{b} \right]}{\sum_i W_i}$$

In attempting to compute the wing root bending moment due to the applied airloads, one is faced with the fact that airload distribution is seldom specified at the early stages of the design in which a preliminary weight estimation is called for. The airload distribution on the wing is often assumed to be elliptical, and sometimes assumed to be proportional to the chord distribution. Neither of these assumptions is precise, and there is reason to believe that the actual airload lies somewhere between these two. The advantage of the elliptical airload assumption is its invariance with configuration changes other than span. The lift distribution proportional to chord does, however, allow for the influence of changes in taper and chord distribution on the strength requirements of the wing. For the sake of simplicity, and with no attempt at additional justification, it will be assumed that the design aerodynamic load on a semi-wing is the average of these two types, each of which is assumed equal to the total design load.

For the airload distribution in which the lift force at any station is proportional to the chord at the station we can write, assuming a linear planform taper:

$$L_a(x) = K\, c(x)$$

$$c(x) = \frac{c_R}{b}\left[b - (1 - \lambda)\, x \right]$$

$$\lambda = \frac{c_T}{c_R} \qquad (26)$$

$$L_a(x) = \frac{\eta W_g}{b^2(\lambda + 1)} \left[b - (1 - \lambda)\, x \right]$$

The centroid of this load distribution is readily found to be:

$$\bar{y}_a = \frac{2\lambda + 1}{3(\lambda + 1)} \qquad (27)$$

Assuming an elliptical lift distribution, and setting the area under one quarter of an ellipse equal to the design load on a semi-wing gives us:

$$L_a(x) = \frac{2\, \eta W_g}{\pi b^2} \sqrt{b^2 - x^2} \qquad (28)$$

whose centroid is:

$$\bar{y}_a = \frac{4}{3\pi} \tag{29}$$

The aerodynamic bending moment at the wing root can then be estimated by multiplying the design load by the average of these load centroids, giving:

$$M_a(0) = \frac{\eta W_g\, b}{12}\, f(\lambda)$$
$$f(\lambda) = \left[\frac{4}{\pi} + \frac{(2\lambda + 1)}{(\lambda + 1)}\right] \tag{30}$$

In a similar fashion, the expression for $Y_d(0)$ can be written as:

$$Y_d(0) = \xi\, b\, W_d\, \bar{z}_d \tag{31}$$

where

$$\bar{z}_d = \frac{\sum_i W_i \left[\frac{x_i}{b}\right]^2}{\sum_i W_i}$$

and the expression for $Y_a(0)$ as:

$$Y_a(0) = \xi \int_0^b M_a(x)\, dx \tag{32}$$

Evaluating this definite integral for the linear and elliptical airloads described previously, and taking the average of the two results gives:

$$Y_a(0) = \frac{\xi \eta W_g\, b^2}{24}\, g(\lambda)$$
$$g(\lambda) = \frac{3\lambda + 2}{\lambda + 1} \tag{33}$$

where the parameter ξ is defined as:

$$\xi = \frac{\int_0^b \frac{\left| M_a(x)\ y(x) \right|}{R^2(x)}\, dx}{\int_0^b M_a(x)\, dx} \tag{34}$$

Any attempt to evaluate this expression for ξ rapidly becomes messy for other than the simplest of airloads. The expressions obtained are not well suited to our objective of an equation which can be used for reasonably rapid computations. Certain numerical approximations appear to be valid, however, in determining an expression for this quantity. If we assume that the radius of gyration of any wing section is not radically different from the distance between the elastic axis and the extreme fiber, and that this distance is not appreciably different from half the maximum wing thickness, then the function ξ becomes proportional to some representative thickness along the wing semi-span. As defined in the equations above, the contribution made by each station to this representative value is weighted by a function which approximates the moment distribution due to the assumed airload. A numerical average of thickness distribution has been tried, and does not appear to give consistent results. As a possibly improved estimate, it will be assumed that the root mean square of the root and tip thicknesses can be taken as a first approximation to this representative thickness, or:

$$\bar{t} = \sqrt{t_R^2 + t_T^2} = t_R \sqrt{1 + \delta^2}$$

$$\delta = \frac{t_T}{t_R} \tag{35}$$

Using these approximations, the equation for the weight of an aircraft wing becomes:

$$W_w = a_1 \left[(x_1 - x_2) + (x_3 - x_4) \right] + a_2\, S\, f(t) \tag{36}$$

where:

$$x_1 = \eta W_g \, b \, f(\lambda)$$

$$f(\lambda) = \left[\frac{4}{\pi} + \frac{(2\lambda + 1)}{(\lambda + 1)} \right]$$

$$x_2 = \eta W_d \, b \, \bar{y}_d$$

$$\bar{y}_d = \frac{\sum_i W_i \left[\frac{x_i}{b} \right]}{\sum_i W_i}$$

$$x_1 - x_2 = \eta W_g \, b \left[f(\lambda) - \frac{W_d \, \bar{y}_d}{W_g} \right]$$

$$x_3 = \frac{\eta W_g \, b^2 \, g(\lambda)}{t_R \sqrt{1 + \delta^2}} \qquad g(\lambda) = \left[\frac{3\lambda + 2}{\lambda + 1} \right]$$

$$x_4 = \frac{\eta W_d \, b^2 \, \bar{z}_d}{t_R \sqrt{1 + \delta^2}} \qquad \bar{z}_d = \frac{\sum_i W_i \left[\frac{x_i}{b} \right]^2}{\sum_i W_i}$$

$$x_3 - x_4 = \frac{\eta W_g \, b^2}{t_R \sqrt{1 + \delta^2}} \left[g(\lambda) - \frac{W_d}{W_g} \, \bar{z}_d \right]$$

II

DISCUSSION OF THE EQUATION

On the basis of the derivation in the previous section, Equation (36) gives the weight of an airplane wing if the representation of the wing as a simple beam is valid. The first term in this equation should be proportional to the weight of the shear and bending material in the wing, while the second term represents the miscellaneous weight. If no statistical constants are derived for the first term of Equation (36), the constant a_1 is equal to the density of the wing material divided by an effective stress (which is the same in compression as in tension). As an estimate of the order of magnitude of a_1, we know that:

$$\rho = \theta(10^{-1}) \text{ lbs per in}^3 \quad *$$

$$\sigma = \theta(10^{4}) \text{ lbs per in}^2$$

$$a_1 = \frac{\rho}{\sigma} = \frac{\theta(10^{-1}) \times 12 \times 144}{\theta(10^{4}) \times 144} = \theta(10^{-4})$$

Let us assume, therefore, that

$$W_{(B+S)} \cong 10^{-4} \left[(x_1 - x_2) + (x_3 - x_4) \right] \qquad (37)$$

is approximately the weight of shear and bending material in an airplane wing. Using this equation, shear and bending material weight estimates were computed for a variety of configurations for which detailed statements concerning shear and bending weights were available. In Figure 1 is plotted estimated versus actual shear plus bending material weights for fighter aircraft. Similarly, Figure 2 shows the estimated versus actual shear plus bending material weights for bomber and cargo aircraft. For the configurations shown, it is possible to fit a curve to the data points which will allow prediction of shear plus bending weight with an rms error of less than ten percent.

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* $\theta(\ \ )$ = of the order of ( )

If these points are plotted on log-log paper and a best straight line fit to them, it is found that the exponent for the fighter equation will be approximately one, while the exponent for the bomber and cargo equation will be approximately 0.8. Thus, Equation (37) modified by these exponents can be used for an accurate estimate of the shear plus bending material weight. A more accurate fit could readily be obtained by using separate effective stresses for the shear and bending weights, but no attempt is made to explore this refinement here.

In an attempt to explain the difference remaining between this estimate of shear plus bending weight and the actual weight of the wing, the difference was plotted against a variety of combinations of nominal wing area and representative thickness. The combination for Sf(t) which was finally chosen cannot be described here in detail since it is based upon the modification of another formula which is proprietory. In Figure 3, however, is plotted the difference between actual wing weight and one half the shear plus bending material weight as calculated by Equation (37). Examination of the points shows that the various classifications of aircraft group well enough to allow a satisfactory fit to be made by a linear predicting equation in each classification. Further examination of these points and those shown on Figures (1) and (2) has shown that large negative errors in the prediction of shear and bending material weight tend to be accompanied by large positive errors in the estimation of the miscellaneous weight. This situation is violated for only a few configurations, most of which are experimental in nature.

Figures 4, 5, and 6 present the results of combining these two equations, for fighters, bombers, and cargo craft respectively. Figure 7 summarizes the predictions for all categories. The rms error in using the equation is not significantly different from ten percent for any of the airplane classifications. More exact fits can and have been made with these basic equations which have achieved rms errors, for cargo aircraft for example, of less than five percent. The improved fits were possible by an expansion of Equation (36) in a form which provided more undetermined constants, and hence allowed improved statistical fits to the data. A further discussion of this point can be found in Reference 1. No attempt has been made to use the improved fits here, since the changes in sign and magnitude which often result from a statistical fit make it impossible to assign physical significance to any given term in the equation.

Equation (36) has been used to predict the wing weight on a reasonably large sample of aircraft, containing more configurations than were used in the statistical determination of the constant a_2. Computations were limited to those configurations for which the data were readily available in a convenient form. In Tables I, II, and III are listed the percentage

prediction errors for the fighter, bomber, and cargo classifications respectively. The numbers listed under "Configuration" are a reference code used between WRRC and WADC. Because of the presence of proprietory data in some of the tabulations, the configurations are not identified further. Also included in the tables are the prediction errors obtained using four other wing weight estimation equations.

Method A is an empirical equation which has been formalized by WRRC. In wing weight studies up to the present date it has been the most consistent equation available, as well as one of the simplest. Method B is used by an airframe manufacturer, and is a rather elaborate method based upon detailed statistical and analytical studies. Method C is an exponential fit of an analytically derived expression to configuration data. Method D is an empirical equation used by an airframe manufacturer.

III

CONCLUSIONS

On the basis of the data presented in the Figures, and of the common root mean square errors listed in Tables I, II, and III, the following conclusions are drawn:

1. Equation (36) is satisfactory for preliminary wing weight estimation of fighter, bomber, or cargo aircraft.

2. Acceptable estimates of the weight of shear plus bending material can be obtained by fitting Equation (37) to the appropriate classification of aircraft.

3. For fighter aircraft, Method B has the smallest common rms error, and is not significantly different from Method A and Equation (36).

4. For bombers, Equation (36) has the smallest common rms error, and is not significantly different from Methods A and C.

5. For cargo aircraft, Method D has the smallest common rms error, and is not significantly different from Equation (36).

Equation (36) has been applied to many configurations by inexperienced personnel. It has been found that the average computing time is somewhat less than five minutes per configuration, using a desk calculator. This computing time is less than or equal to the computing time required for any of the other methods investigated during the course of this contract.

REFERENCES

1. Tysver, Joseph B., "Statistical Constants in Predicting Equations", University of Michigan, Willow Run Research Center Memorandum 2063-4-J.

2. Green, Lyle D., "Probability Confidence Belts", University of Michigan Willow Run Research Center Memorandum 2063-5-J.

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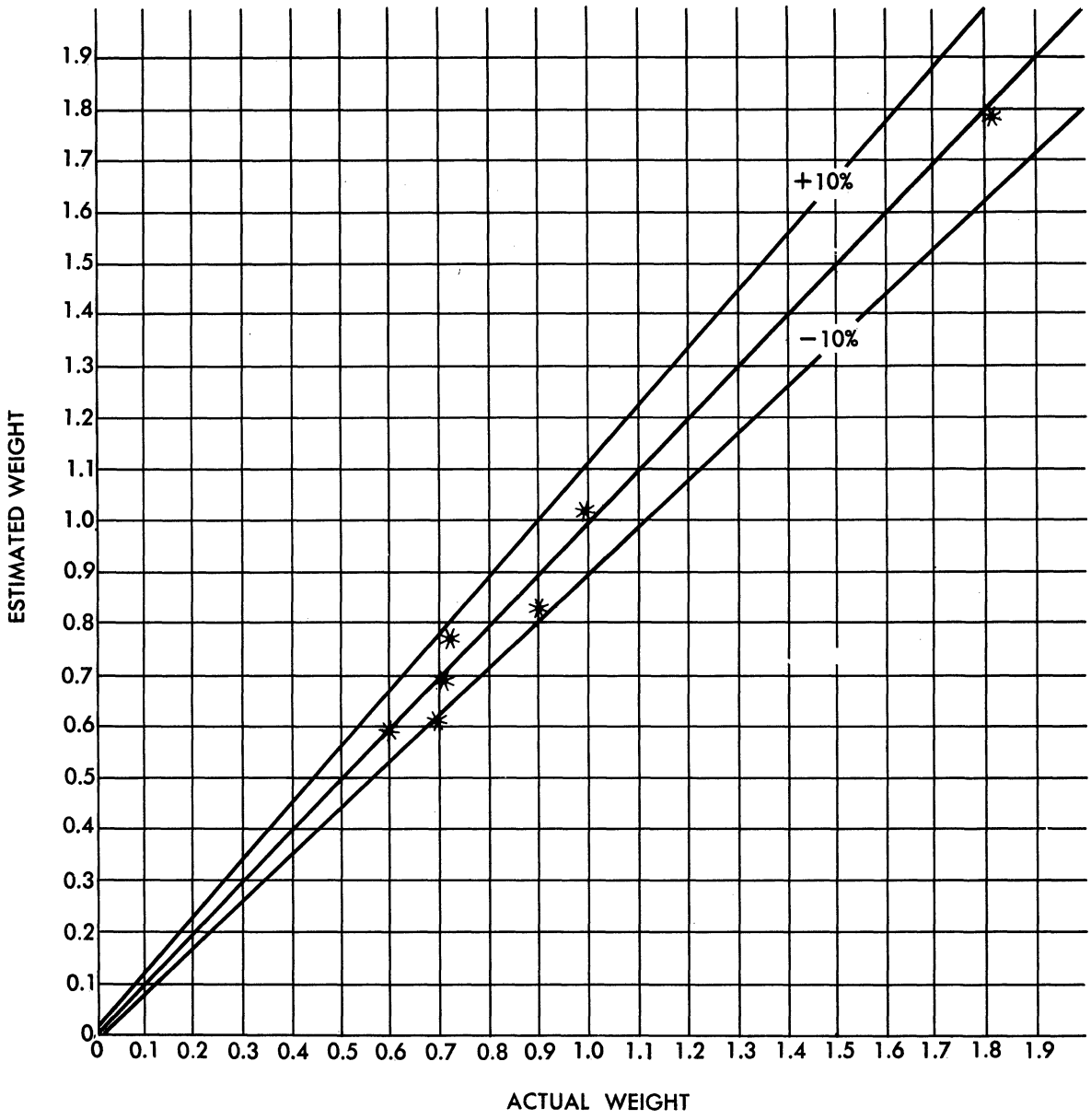


FIG. 1 ESTIMATED vs. ACTUAL WEIGHT OF SHEAR AND BENDING MATERIAL — FIGHTERS
(Expressed as Percentage of a Reference Weight)

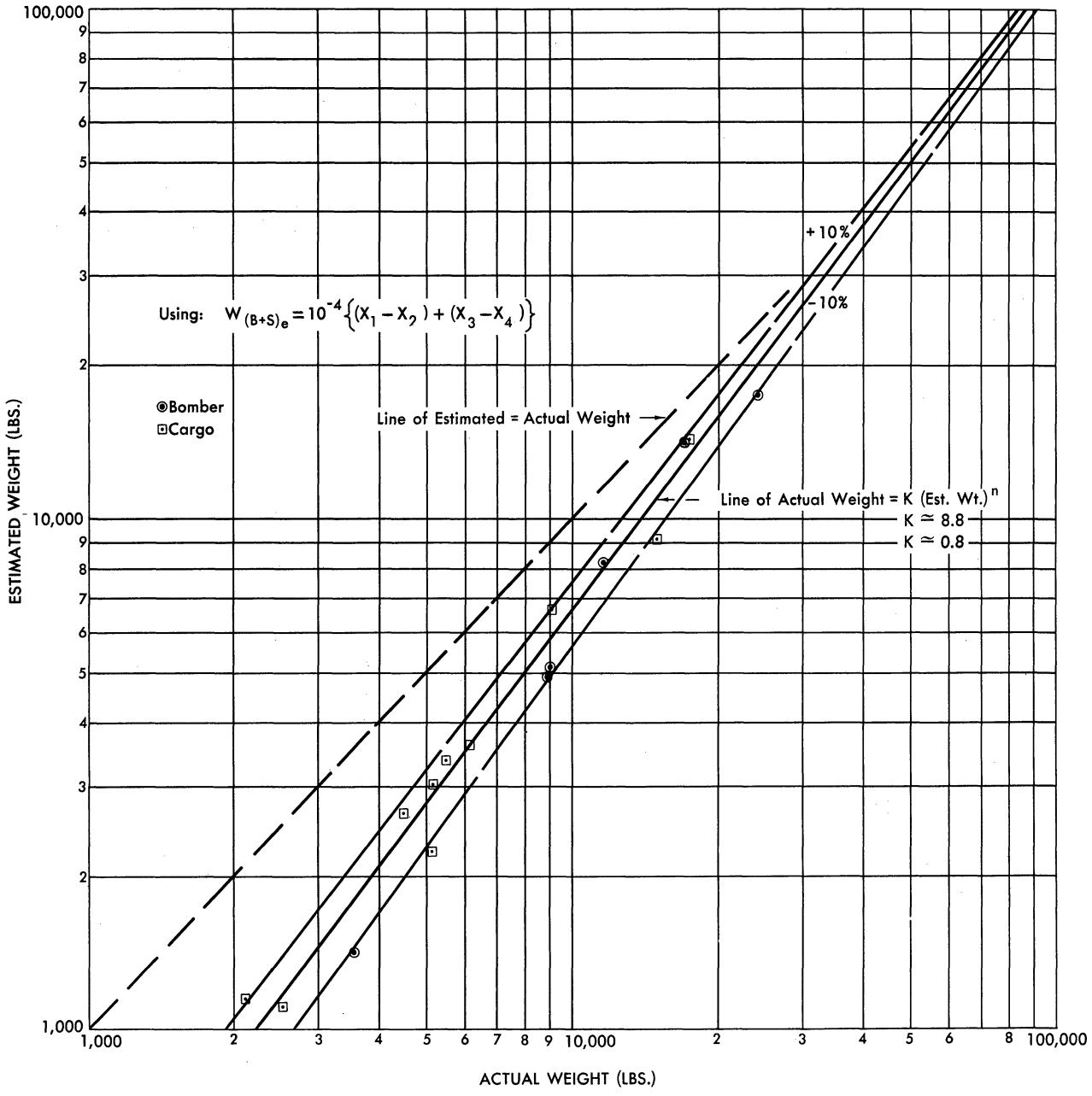


FIG. 2 ESTIMATED vs. ACTUAL WEIGHT OF SHEAR AND BENDING MATERIAL

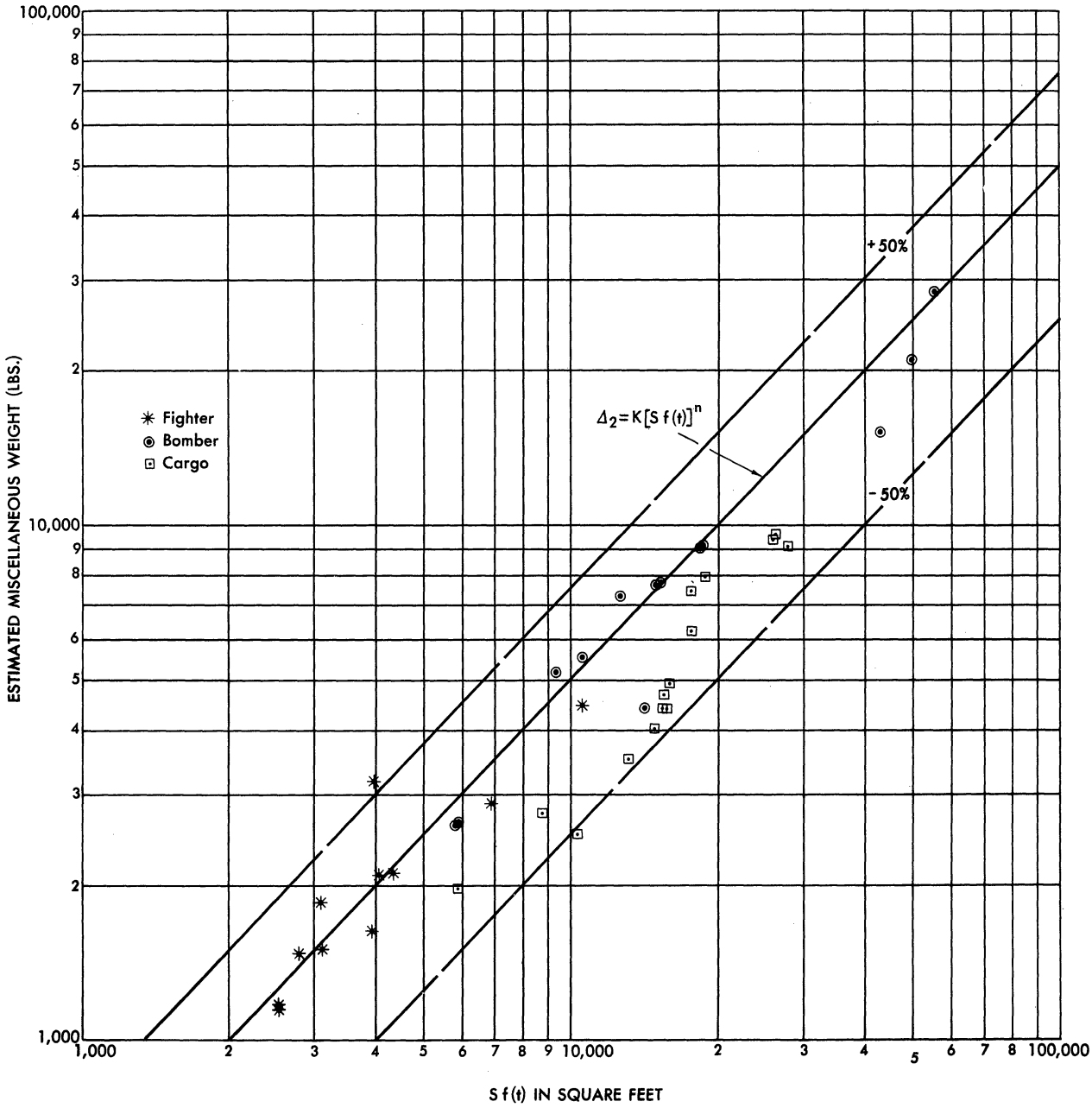


FIG. 3 ESTIMATED MISCELLANEOUS WEIGHT vs. S f(t)

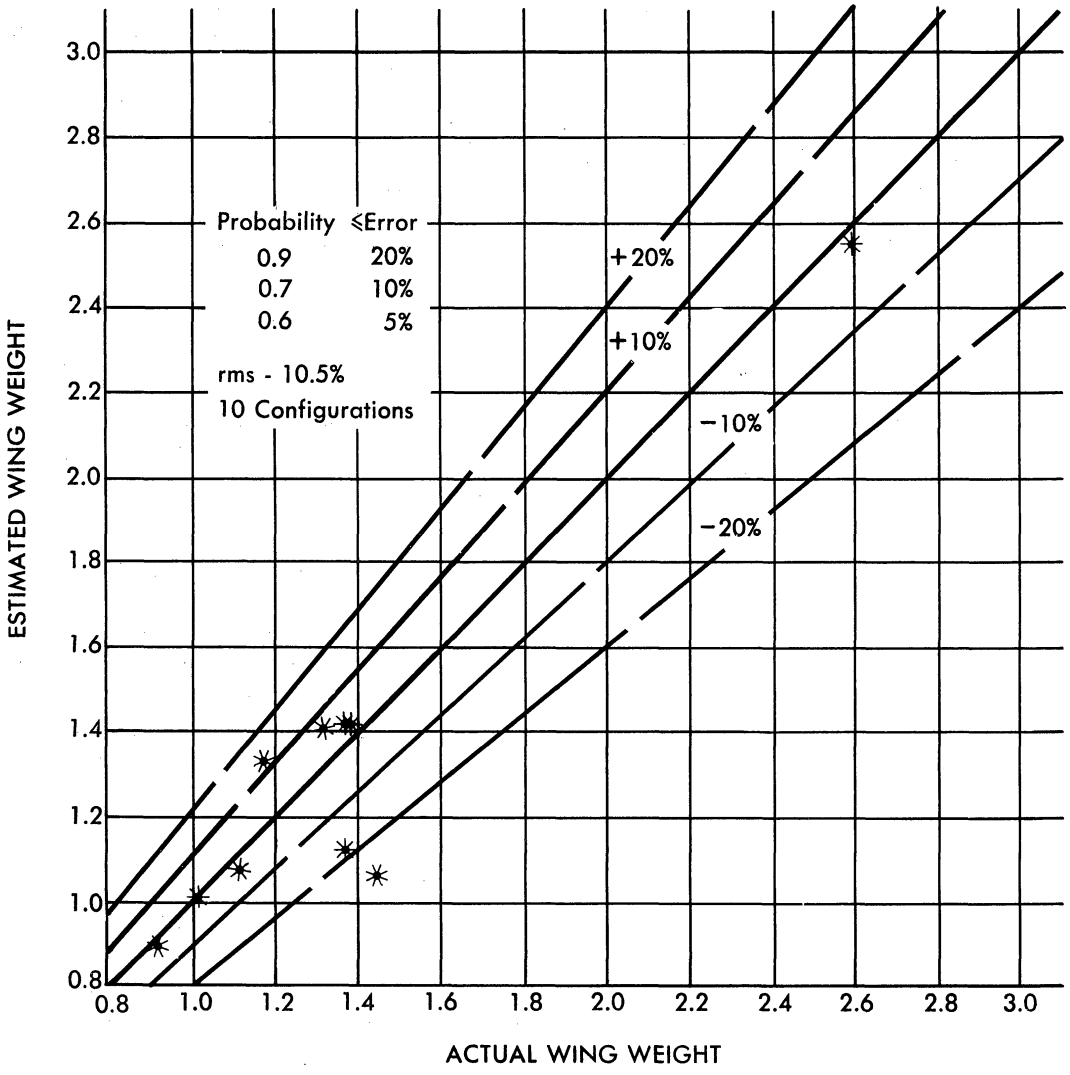


FIG. 4 ESTIMATED vs. ACTUAL WING WEIGHT
FIGHTERS
(Expressed as Percentage of a Reference Weight)

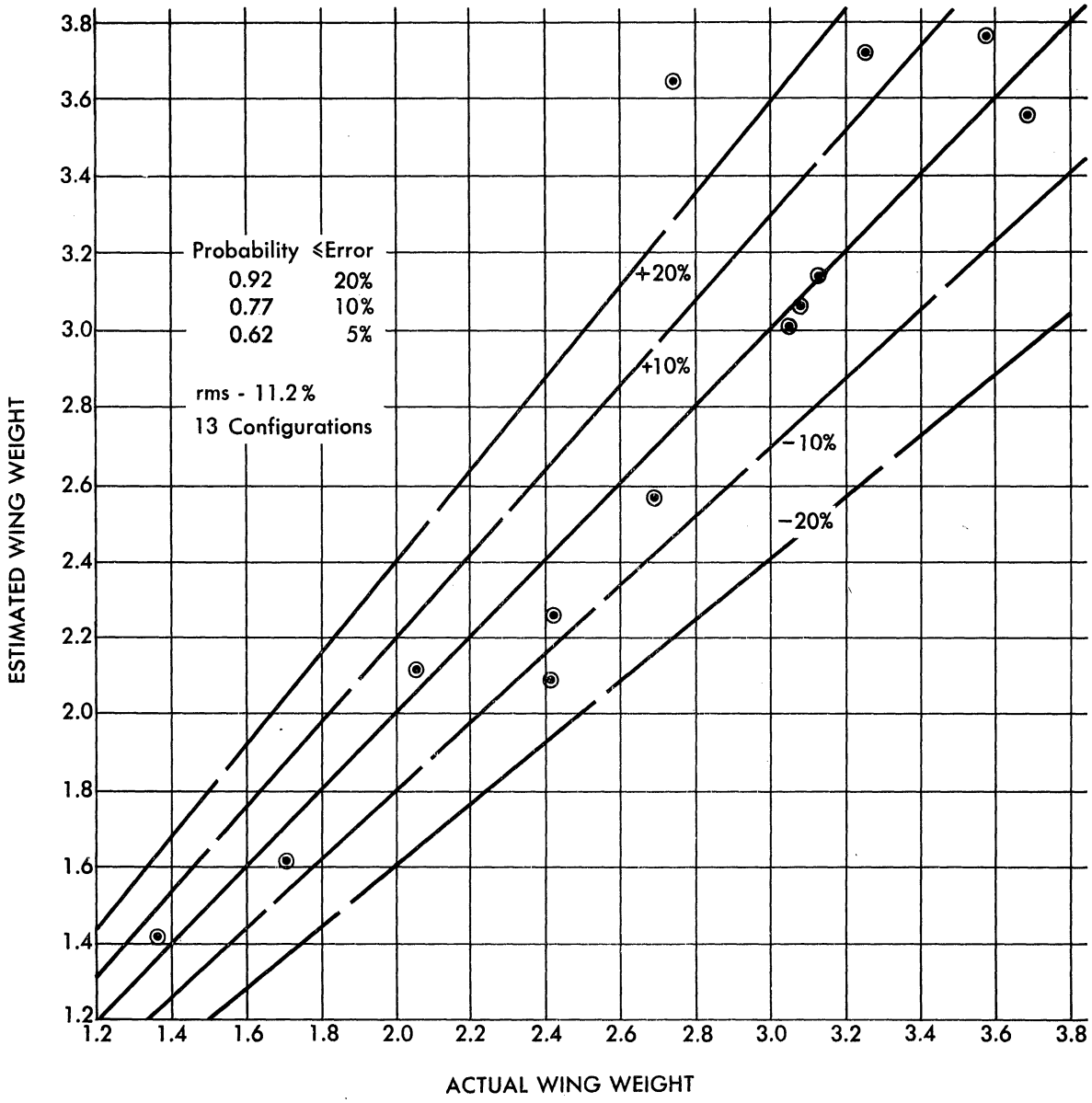


FIG. 5 ESTIMATED vs. ACTUAL WING WEIGHT
BOMBERS
(Expressed as Percentage of a Reference Weight)

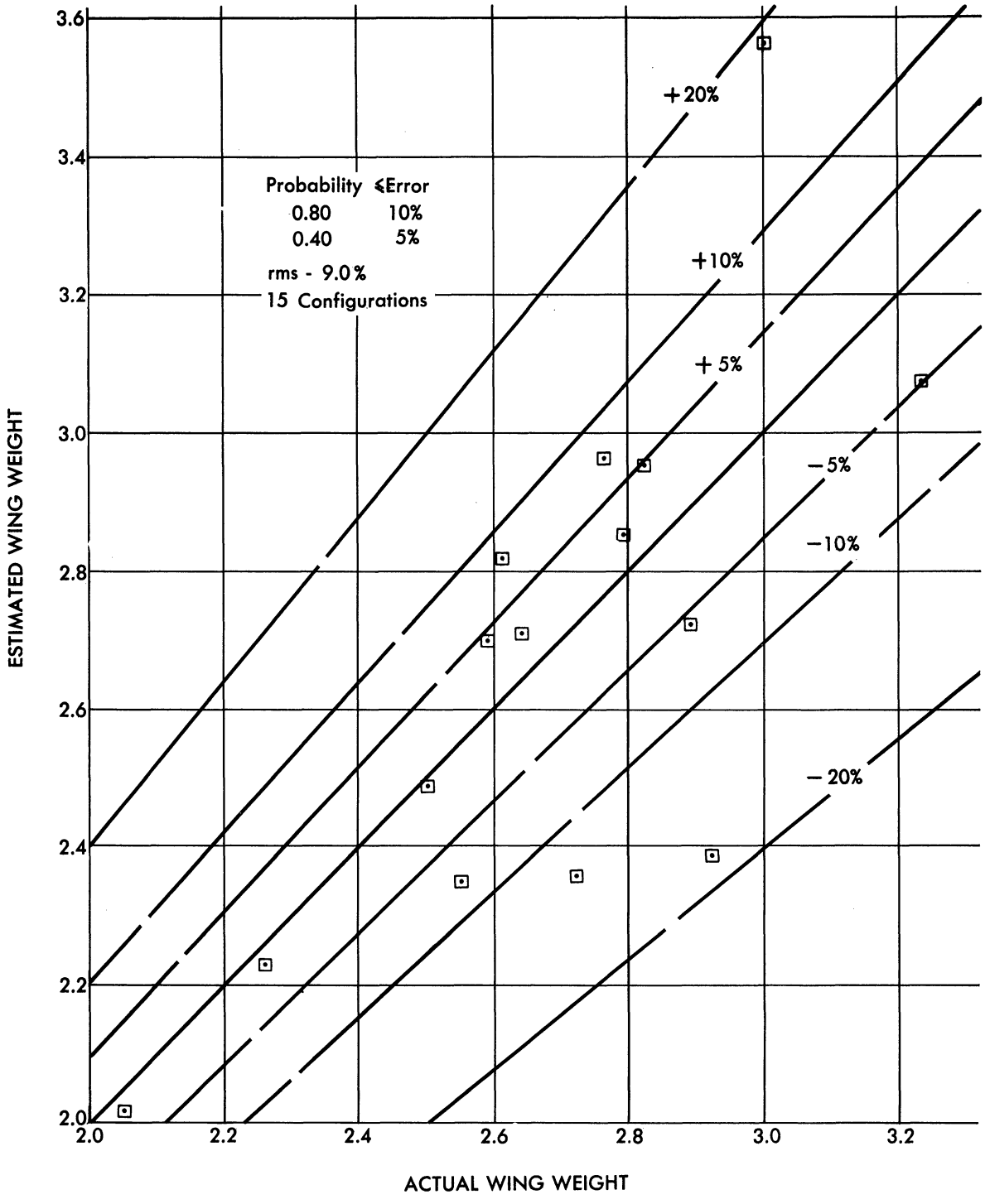


FIG. 6 ESTIMATED vs. ACTUAL WING WEIGHT
CARGOS
(Expressed as Percentage of a Reference Weight)

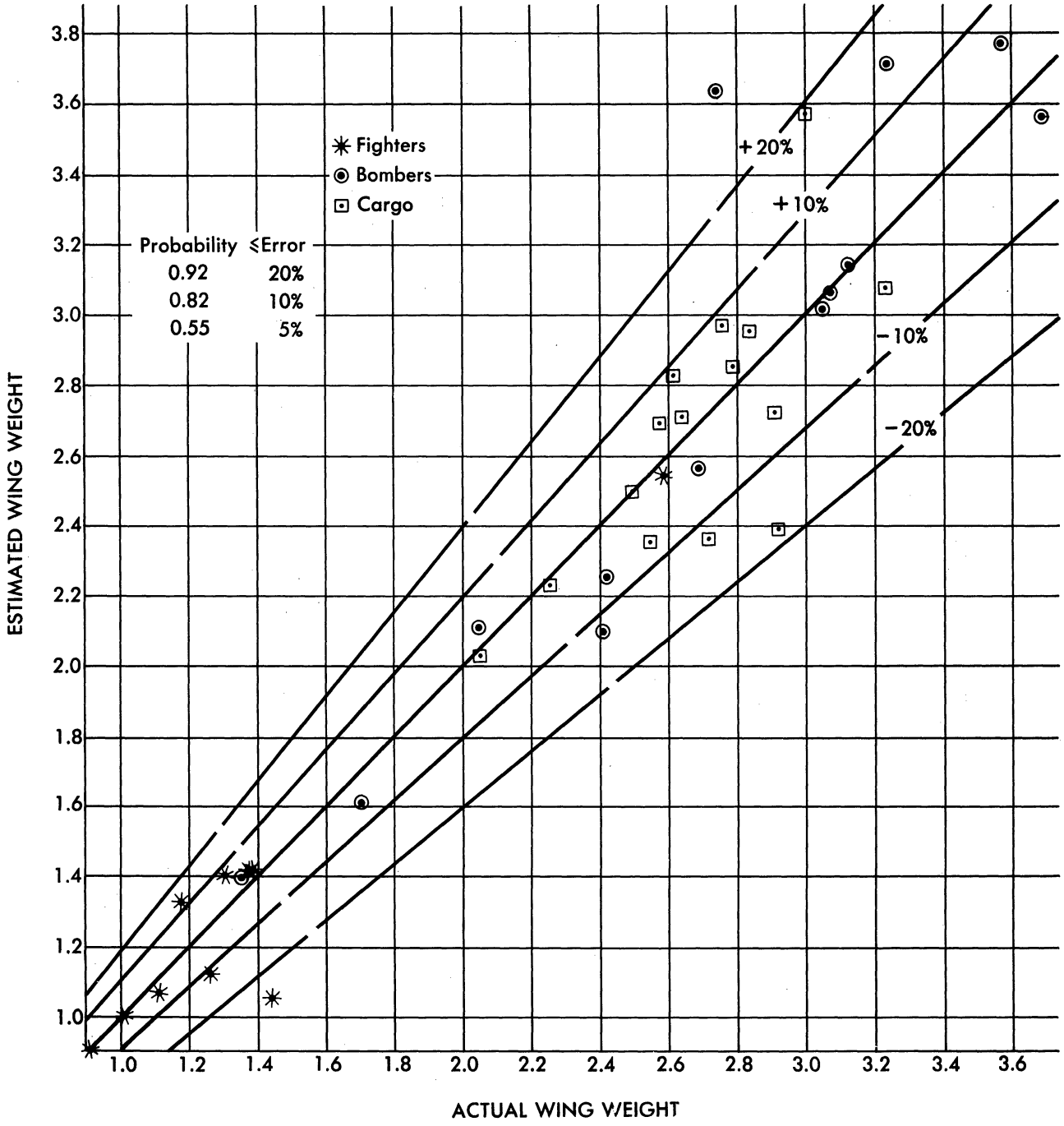


FIG. 7 ESTIMATED vs. ACTUAL WING WEIGHT
ALL CATEGORIES
(Expressed as Percentage of a Reference Weight)

TABLE I

PERCENTAGE PREDICTION ERRORS USING VARIOUS ESTIMATION EQUATIONS
(FIGHTERS)

Configuration	Eqn. (36)	A	B	C	D
101		-11.27	-11.8	-30.5	-25.1
103	12.51	-13.74		1.2	- 6.7
106				9.2	5.0
107				22.1	19.1
108				4.8	- 0.6
109				2.1	- 1.1
110	7.0	6.3	11.1	- 9.9	13.2
111		12.0			
112				5.0	11.0
116		-13.4			
117	- 3.6	- 9.2	- 1.4	- 2.2	- 0.7
120		7.5			
121		5.5		2.2	0.6
122	3.1	- 0.5	19.8		
123	-11.0	-10.2	- 1.4	- 8.0	-12.3
124	13.5	17.1		16.9	23.7
126		-21.0	4.7	17.2	-13.4
127	16.1	-33.2			
128			-38.6	-29.5	-32.4
130	-26.4	-29.4			
131				- 3.3	15.6
132				18.7	15.4
133		9.6	4.5	19.4	10.8
134		10.7			
136	- 0.8	1.5	3.8	11.9	14.4
137	3.9	- 0.1	1.1	6.6	11.8
138	- 1.5	9.2	6.1		
139				- 3.5	- 6.1
140		19.4	- 6.3		
rms error[1]	11.7	14.7	13.3	14.3	14.6
common rms[2]	6.2	6.8	5.3	8.4	11.6

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1. Based on total sample for a given equation.
2. Based on sample common to all equations.

TABLE II

PERCENTAGE PREDICTION ERRORS USING VARIOUS ESTIMATION EQUATIONS
(BOMBERS)

Configuration	Eqn. (36)	A	B	C	D
301				9.2	16.6
302	33.2	7.7	13.1	7.5	16.7
305	3.2			- 1.2	- 3.0
312	- 0.4	- 7.2	-15.7	- 5.5	- 1.3
316				-20.2	-10.9
317	- 4.5	- 6.7	5.0		
318	5.4	3.8	10.3	3.0	10.6
325	-13.3	-31.3	-15.1	10.9	- 6.7
328	0.5	- 5.5	-35.3	- 6.2	-27.9
332	- 1.4	- 9.2	8.2		
333				- 6.5	- 2.2
334	4.2	15.5	- 3.5	34.6	26.9
335	14.5	- 0.8	-13.7	4.6	- 4.0
336	- 6.8	-17.0	-21.0	13.7	10.0
337	- 3.8	3.4	9.8	1.2	6.4
339	- 5.0		- 4.1	17.3	0.5
rms error[1]	11.2	12.8	16.0	13.4	13.4
common rms[2]	13.3	13.7	17.4	13.6	15.2

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1. Based on total sample for a given equation.
2. Based on sample common to all equations.

TABLE III

PERCENTAGE PREDICTION ERRORS USING VARIOUS ESTIMATION EQUATIONS
(CARGO)

Configuration	Eqn. (36)	A	B	C	D
501	- 1.4	- 2.7	6.8	11.8	12.8
505	7.4	- 2.1	3.2	11.2	1.7
506				7.4	- 0.7
507	18.8	11.6	18.3	12.6	4.1
509	4.6	0.9	21.2	20.5	17.6
511	- 7.8	- 7.3	- 8.7	-11.9	- 2.4
512	- 8.4		-11.6	6.1	5.4
513	- 5.1	-44.1	- 5.7	- 6.9	4.6
514	2.0	- 6.7	- 1.9	2.0	6.6
519	-13.6	-13.6	-21.0	-15.7	-11.8
522	- 1.4			9.7	3.6
524		- 7.4	19.4		
526	- 0.5			4.9	7.3
529	-18.1			-10.4	-11.9
531		-17.7	-11.6	- 5.4	- 8.5
532	7.7	5.5	- 5.5	13.4	- 3.8
533	- 5.9	- 2.7	- 1.8	- 8.3	3.4
536		-24.0	-20.9	-18.6	-18.6
538	4.6	6.9	- 5.1	- 4.3	- 4.8
rms error[1]	9.0	15.6	10.1	11.0	8.1
common rms[2]	8.6	14.9	11.5	11.8	8.3

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1. Based on total sample for a given equation.
2. Based on sample common to all equations.