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WEIGHT ESTIMATION TECHNIQUES FOR COMPOSITE AIRPLANES IN GENERAL AVIATION INDUSTRY

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ABSTRACT

The purpose of the present study was to investigate the currently available weight estimation methods used for conventional aluminum airplanes and to extend the basic methodology to develop a procedure to estimate the weight of future non-conventional composite airplanes in the general aviation industry. Basic aircraft component weight estimation equations containing explicit material properties were developed. Regression analysis was applied to the basic equations for a data base of twelve airplanes to determine the coefficients. The resulting equations can be used to predict the component weights of either metallic or composite airplanes. Aircraft component weights predicted using the method developed at WSU for both aluminum and composite airplanes in the general aviation industry show much greater accuracy compared to the existing methods. However, the present data base should be updated for improved accuracy as and when more composite airplanes are produced and certified.

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INTRODUCTION

Light weight fiber reinforced composite materials with their greater strength and modulus-to-weight ratio are rapidly replacing the conventional metallic materials in aerospace applications to improve the performance of flight vehicles. Early applications of composite materials were limited to space missions and fighter aircraft because of the lack of a history of performance of the material. Consequently, the financial risk involved with this new material prohibited its applications in general aviation structures in any significant way. The ever increasing experience with composite materials and the continued demand for lighter weight and stronger aircraft materials has renewed the interest of the general aviation industry in the use of fiber reinforced composite materials in both primary and secondary structures. An additional benefit of these new composite materials is the possibility of greatly improved aerodynamic surface smoothness as well as integral intersection fairings which would not be practical with conventional riveted metal fabrication. This resulting clean aerodynamic surface is indicative of the results in synergistic performance improvements that could be possible with extensive application of advanced composite technology.

Technological advances in aircraft design, materials and computers enable the conceptual and preliminary design processes to provide quick information regarding the selection and feasibility of various design approaches to aircraft mission requirements. The required weight of an aircraft that meets the performance requirements is one of several parameters necessary in the design selection process. The consideration of composite materials for advanced general aviation airplane designs is presently hampered by the lack of accurate weight estimation methodology. The weight of a new metal aircraft may be predicted from the design charts and tables established through years of experience with similar conventional metal aircraft. On the other hand, it is difficult to predict the weight of an all-composite airplane, as there are only a few all-composite airplanes constructed to date to use as a data base.

The purpose of the present research was to investigate some of the currently available weight estimation methods [1,2] used for conventional metallic airplanes and to extend the basic methodology to develop a procedure to estimate the weight of future non-conventional composite airplanes in the general aviation industry. Basically, regression equations for estimating weights of metallic aircraft components are given by Torenbeek [1] and Nicolai [2] for different categories of airplanes. In addition, Nicolai [2] also utilizes "knockdown" factors to determine the weight of a composite aircraft component equivalent to a corresponding metallic counterpart. Neither formulation allows for explicit introduction of the physical properties of the material into the weight estimation equations. The methods and procedures presented in References 3 and 4 were based largely upon a rational approach to the problem as opposed to the empirical approaches of Torenbeek and Nicolai. The weight estimation procedures developed in this study were modeled after those of

References [3] and [4]. Specifically, a rational approach was used to develop weight equations associated with general aircraft structural requirements. Material properties were introduced into this rational weight formulation. Dimensional analysis techniques were used to combine the parameters in the weight equations into non-dimensional groups. Then linear regression analyses were applied to these non-dimensional equations using the actual aircraft component weight data obtained for the general aviation category of airplanes. These equations are used to predict the component weights of aircraft in this category. Computer codes were written by WSU investigators based on this approach which has been outlined herein. The WSU programs were designed for flexibility and easy use, allowing the user to easily update the data base and resulting equations.

2. WEIGHT ESTIMATION TECHNIQUES FOR CONVENTIONAL METALLIC GENERAL AVIATION AIRPLANES

Weight estimation techniques have been developed for the preliminary design of any category of aircraft. Currently available weight estimation methods for the general aviation category of airplanes were surveyed. Most of these methods assume metallic (aluminum) structures. Several methods were investigated to provide an insight in the development of comparable weight equations with explicit material properties. Roland [5] describes three fundamental techniques of weight estimation, (1) the fixed-fraction method, (2) the statistical correlation method, and (3) the "point stress" analysis method. In the fixed-fraction method, the weights of the airplane components are assumed to be a fixed-fraction of the empty weight or takeoff weight. The statistical correlation method relies upon statistical correlation with an appropriate data base to determine the component weights. The method assumes an estimation equation of the form,

$$W = \sum A_i x_i^{Bi}$$

where

W is the component weight

A_i is an empirically determined weight coefficient

B_i is an empirically determined exponent

x_i is a parameter

The selection of the parameter, x_i, is the key to the success of the correlation. The point stress analysis method is only applicable for estimating the weight of major structural components of an airplane, i.e., the wing, tail, fuselage, and landing gear while the weights of the non-structural components are normally calculated with statistical correlation equations. The weight estimate is based on the material required to carry the loads at representative "points" in the component. This method requires the specification of both the component loads and the allowable stresses. Due to the complexity of this method, a computer program is a necessity.

Aircraft companies usually follow one of the methods above or develop their own methods based on their specific airplane data. This investigation was focused on the general prediction equations rather than those unique to a particular company. The simple textbook methods of Torenbeek [1] and Nicolai [2], as well as the computer code GASP based on NASA CR-152303 [6] and the Society of Aircraft Weight Engineers (SAWE) computer code [7] were among those given serious consideration. Most of these methods predict the weights of propulsion and airframe structural component using regression equations derived from data of similar aircraft. Typically a large number of parameters were required to describe the aircraft.

A computer code using Torenbeek's [1] equations was written and investigated. Appendix-A gives Torenbeek's equations as listed in [8] while Appendix-B lists the computer program for Torenbeek's equations. Computer codes for Nicolai [2] and the SAWE method [7] were developed previously [9]. A listing of typical geometric and performance data obtained from the manufacturers for each airplane model and the input data required for each of the computer programs investigated are presented in Table 1 . Typical input and output statements associated with the Torenbeek and Nicolai computer codes are given in Tables 2 and 3 respectively. These programs are interactive and require geometric and performance data inputs of each airplane model.

Torenbeek's [1] equations are the most general of the methods coded. These equations are valid for subsonic aluminum aircraft in both the general aviation and commercial transport class of airplanes. The equations from Nicolai [2] are valid for light general aviation aircraft with maximum airspeed of up to 300 knots. The equations of Society of Aircraft Weight Engineer's paper number 158 [7] are limited to conventional aluminum type general aviation aircraft with twin engines.

The chief advantage of the SAWE [7] is that the weight for each of the aircraft components can be estimated. However, it was found during this study these predicted component weights were quite different from those predicted by both Torenbeek and Nicolai programs. Hence, for the purpose of comparative studies, only Torenbeek and Nicolai methods of weight estimation were adopted. The weights estimated by these two methods were compared to the actual weights for aircraft components such as wing, fuselage, empennage and landing gears. The results of this analysis are presented in terms of normalized factors 'k' as well as in the form of graphical charts in section 4.

Geometry, performance data and component weight breakdowns for both aluminum and composite airplane models from Gates Learjet, Beech Aircraft, Cessna Aircraft, Avtek. DeVore Aviation Corporation and Lear Fan Ltd. were compiled. These data were obtained from the manufacturers and supplemented by Janes All the Worlds Aircraft [10] and "Aviation Week" [11].

Table 1 . TYPICAL DATA FOR AN AIRPLANE MODEL

Symbols	Description	Input required 'x' for Torenbeek	Nicolai	SAWE	WSU
W_{T_O}	TAKEOFF WEIGHT (LBS)	X	X	X	X
N	ULTIMATE LOAD FACTOR	X	X	X	X
A	ASPECT RATIO		X		
C	MEAN AERODYNAMIC CHORD (FT)			X	X
b	WING SPAN (FT)	X		X	X
S_W	WING AREA (FT^2)	X	X	X	X
S_f	AREA OF ALL CONTROL SURFACES ON WING (FT^2)				
λ_w	TAPPER	X	X		X
$(t/c)_w$	MAXIMUM THICKNESS RATIO		X		X
Λ	QUARTER CHORD SWEEP ANGLE (DEG)	X	X		X
t_R	WING ROOT THICKNESS (IN)			X	X
R_f	CONTROL SURFACE RATIO ($= S_f/S_W$)				
L	FUSELAGE LENGTH (FT)		X	X	X
W	MAXIMUM FUSELAGE WIDTH (IN)	X	X		X
D	FUSELAGE MAXIMUM DEPTH (IN)		X	X	X
P_{FUSE}	FUSELAGE PERIMETER (FT)			X	
$(AREA)_{FL}$	FLOOR AREA (FT^2)			X	
$(AREA)_{FS}$	SURFACE AREA OF FUSELAGE (FT^2)	X			
S_H	HORIZONTAL TAIL AREA (FT^2)	X	X	X	X
ℓ_T	DISTANCE FROM WING 1/4 MAC TO TAIL 1/4 MAC	X	X		
b_H	HORIZONTAL TAIL SPAN (FT)	X	X		X
t_{HR}	HORIZONTAL TAIL MAXIMUM ROOT THICKNESS (IN)		X		X
A_{HT}	HORIZONTAL TAIL ASPECT RATIO			X	X
Λ_{HT}	HORIZONTAL TAIL SWEEP ANGLE (DEG)	X		X	X

Table 1. TYPICAL DATA FOR AN AIRPLANE (Cont.)

Symbols	Description	Input required 'x' for Torenbeek	Nicolai	SAWE	WSU
$(t/c)_{HT}$	HORIZONTAL TAIL THICKNESS RATIO			X	X
λ_{HT}	HORIZONTAL TAIL TAPER RATIO			X	X
S_V	VERTICAL TAIL AREA (FT^2)	X	X	X	X
b_V	VERTICAL TAIL SPAN (FT)	X	X		X
t_{VR}	VERTICAL TAIL MAXIMUM ROOT THICK-NESS (IN)		X		X
A_{VT}	VERTICAL TAIL ASPECT RATIO			X	
$(t/c)_{VT}$	VERTICAL TAIL THICKNESS RATIO			X	X
λ_{VT}	VERTICAL TAIL TAPER RATIO			X	X
Λ_{VT}	VERTICAL TAIL SWEEP ANGLE (DEG)	X		X	X
L_{NG}	LENGTH OF NOSE LANDING GEAR (IN)			X	
L_{LG}	LENGTH OF MAIN LANDING GEAR (IN)		X	X	
W_{LAND}	LANDING WEIGHT (LBS)		X	X	
N_{LAND}	MAXIMUM LOAD FACTOR AT LANDING		X		
W_{ENG}	BARE ENGINE WEIGHT (LBS)			X	
N_E	NUMBER OF ENGINES	X		X	
H.P.	TAKEOFF H.P. PER ENGINE	X		X	
V_E	EQUIVALENT MAXIMUM AIRSPEED (KNOTS/MPH)	X	X		
W_{OIL}	OIL WEIGHT (LBS)			X	
W_{FUEL}	FUEL WEIGHT (LBS)			X	
F_G	TOTAL FUEL IN GALLONS		X		
INT	PERCENT OF FUEL TANKS THAT ARE INTEGRAL		X		
N_{PASS}	NUMBER OF PASSENGERS			X	
W_{FS}	WEIGHT OF FUEL SYSTEMS (LBS)		X		
W_{TRON}	WEIGHT OF INSTALLED ELECTRONICS (LBS)		X		
N_{CR}	NUMBER OF CREW		X		

Table 2. Typical Input And Output For 'Torenbeek' Method

INPUT DATA

title: airplane xyz
gross weight (lb): 15,000
wing span (ft): 35.58
wing sweep angle (deg): 13^{o}
wing area (sq ft): 231.77
load factor: 4.5
taper ratio: 0.509
no. of engines each wing: 0
spoilers and speed breaks -1; otherwise -0: 1
landing gears fus. mtd -1; otherwise -0: 0
strut braced wing -1; otherwise -0: 0
fowler flaps -1; otherwise -0: 0
h. tail span (ft): 14.67
h. tail area (sq ft): 54
design dive speed (dts): 290
h. tail sweep angle (deg): 25^{o}
v. fin span (ft): 5.48
v. fin area (sq ft): 37.37
v. fin sweep angle (deg): 48o
fus. mt. tail - 1; fing mt. - 2: 2
fus. shell area (sq ft): 685.2
fus. depth (ft): 5.25
fus. width (ft): 5.25
pressurized fuselage -1; otherwise -0: 1
engines mtd on fuselage -1; otherwise -0: 1
cargo floor -1; otherwise -0: 0
wing 1/4 MAC to tail 1/4 MAC length (ft): 20.553
low wing - 1; other - 2: 1
small jet - 1; other civil - 2: 1
landing gears tricycle -1; tailwheel -2: 1
jet - 0; fxt gr - 1; rtrct gr - 2: 0
lt. arcft - .23: .23
lt. 1; multi 2; tprp 3; tjet/fan 4: 4
thrust or hp.: 2950

OUTPUT DATA

the wing weight = 930.80
the horizontal tail weight = 117.05
the vertical tail weight = 139.35
the fuselage weight = 1421.34
the main landing gear weight = 402.22
the nose landing gear weight = 93.32
the tail landing gear weight = 0
the surface control weight = 139.94
the nacelle weight = 162.25

Table 3. Typical Input And Output For 'Nicolai' Method

Airplane: Airplane xyz

Take-off weight: 15,000
Load factor: 4.5
Aspect ratio: 5.01
Wing sweep: 13^{o}
Wing area: 231.77
Taper ratio: 0.509
Max. thk. ratio: 0.09
Equiv. max. spd: 290

Wing weight: 848.6
Comp. wng. wt.: 636.5

Fuselage length: 45.33
Max. fus. width: 5.25
Max. fus. depth: 5.25
Take-off weight: 15,000
Load factor: 4.5
Max. equiv. spd: 290

Fuselage weight: 1151.7
Composite fus. wt: 863.8

Horizontal tail area: 54
Wg .25mac to tail .25mac: 20.553
Horizontal tail span: 14.67
Nz tail root thickness: 4.80
Take-off weight: 15,000
Load factor: 4.5

Horizontal tail weight: 117.2
Composite hz. tail wt: 87.9

Vertical tail area: 37.37
Vertical tail span: 5.48
Vt. tail root thickness: 9.14
Take-off weight: 15,000
Load factor: 4.5

Vertical tail weight: 16.6
Composite vt. tail wt.: 12.5

Main strut length: 25.4
Landing weight: 13,300
Landing load factor: 3.0

Landing gear weight: 383.1
Composite lndg. gr. wt.: 337.1

3. WEIGHT ESTIMATION THROUGH DIMENSIONAL ANALYSIS TECHNIQUES FOR ALUMINUM AND COMPOSITE GENERAL AVIATION AIRPLANES

Several existing weight estimating techniques were evaluated, including beam strength modeling using dimensional analysis. A method developed by the Grumman Aerospace Corporation [3] (NASA-CR-166173: "Aircraft Wing Weight Build-Up Methodology With Modification for Materials and Construction Techniques") was thoroughly investigated and the analysis of that report was adopted for use in this project.

The advantage of the Grumman method relative to other weight estimation methods investigated stems from the inclusion of material property factors such as density and strength. Thus new materials, such as composites, may be rationally analyzed. As a result, a regression tool similar to that developed by Grumman investigators was developed by Wichita State University (WSU) investigators, in the form of an interactive Fortran program. It included material properties in the weight estimation of each of four general aircraft components: Wing, fuselage, vertical tail and horizontal tail. The wing formulation was very similar to the Grumman procedure and empennage weights were modeled in a similar manner. A fuselage weight equation was developed at WSU which considers bending stiffness as well as strength.

A program "GENREG FORTRAN" has been written at WSU for the purpose of computing the parameters of the regression equations for general aviation airplane major component (wing, fuselage, empennage) weight estimations. The program then uses a general regression analysis method (least-squares curve fit) which generates the regression equations and calculates estimated weight values and errors of the components for those airplanes used in the data base (see flow-chart of Fig. 1).

The program calls subroutines WING, FUSLGE, HZTAIL, and VTAIL. Each subroutine is described in detail below.

WING

The wing weight calculation subroutine is adapted from the formulation presented in NASA CR-166173. The general equations of that report, derived on the basis of assumed rib and spar construction, have been adapted for regression analysis in a way slightly different from that used by the Grumman investigators. The Grumman investigators divided the formulation into an equation for the wing cover (skin) and a separate equation for the wing substructure with subsequent regression on each. The Grumman report contains weight and geometry data for 50 military and commercial airplanes. In contrast, the WSU project is focused on general aviation airplanes for which only total wing component weights are known. As a result, the WSU method combines both of the Grumman equations into a single equation which describes the weight of the entire wing. The equations for the cover weight and the substructure weight, from Figures 5 and 6 of CR-166173, were the following:

$$W_{CVR} = C \frac{b_W (C_R + 2C_T) Bn\, S_W\, \rho}{\cos^2 \Lambda (C_R + C_T)(2C_R + C_T)(2T_R + T_T) F} \qquad (1)$$

$$W_{SUB} = C \frac{\rho}{F_S} Bn\, b_W \qquad (2)$$

FLOW CHART, GENREG/PHSEII

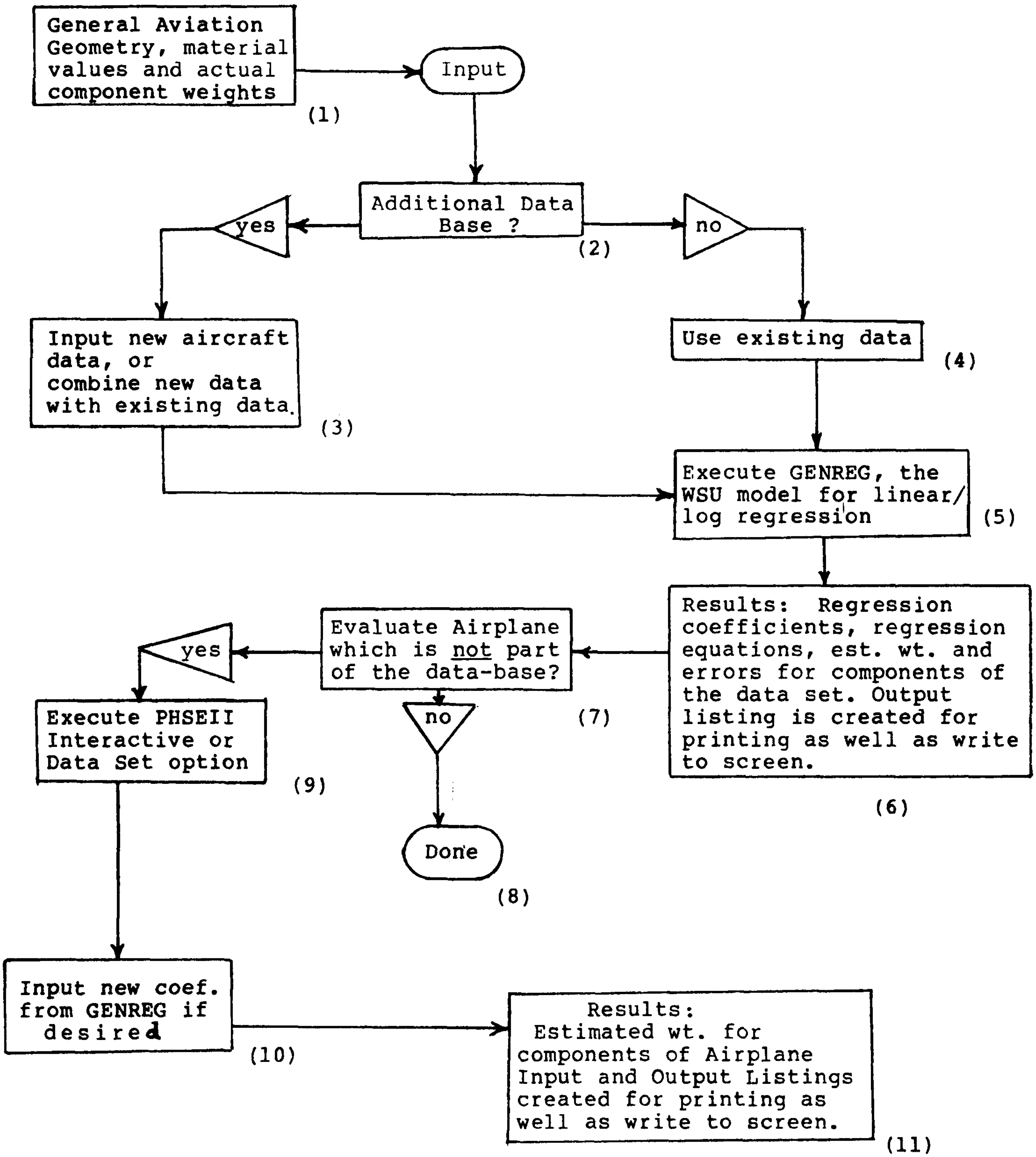


Figure 1. Flow chart of the GENREG computer code.

The following terms are defined:

W_{CVR} = weight of the skin plus stringers

W_{SUB} = weight of the spars

C = a generalized constant representing the product of all numerical values in the term

b_W = wing span

S_W = wing planform area

CR = wing root chord length

C_T = wing tip chord length

B = gross weight (GW) - wing weight (WW)

n = load factor

ρ = material density

Λ = sweep angle of the wing

T_R = wing root thickness

T_T = wing tip thickness

F = allowable cover stress (psi)

F_S = ultimate developed shear (psi)

For this project, Eqns. (1) and (2) were combined to form the general equation for the weight of the entire wing in the following form:

$$W_{wing} = C_1 + C_2 \frac{\rho}{F} \frac{n}{\cos^2 \Lambda} \frac{b_W S_W (C_R + 2C_T) B}{(C_R + C_T)(2C_R + C_T)(2T_R + T_T)} + C_3 \frac{\rho}{F_S} n b_W B \quad (3)$$

Where: C1, C2 and C3 are generalized constants for each term.

The equation was made non-dimensional with respect to aircraft gross weight:

$$\frac{W_{wing}}{GW} = C_1 + C_2 \frac{\rho}{F} \frac{n}{\cos^2 \Lambda} \frac{b_W S_W (C_R+2C_T) B/GW}{(C_R+C_T)(2C_R+C_T)(2T_R+T_T)} + C_3 \frac{\rho}{F_S} n b_W \frac{B}{GW} \quad (4)$$

(4)

Since B = GW - WW, B/GW = (GW - WW)/GW = 1 - $WW_{/GW}$ where WW is the known wing weight used in the regression. Hence, the normalized wing weight was cast in the general form:

$$Y_i = WWi\ /\ GWi = C1 + C2 * Xi + C3 * Zi \tag{5a}$$

$$\text{where} \quad Xi = \frac{\rho}{F}\,\frac{n}{\cos^2\Lambda}\,\frac{b_W S_W (C_R+2C_T)\left(1-\frac{WW}{GW}\right)}{(C_R+C_T)(2C_R+C_T)(2T_R+T_T)} \tag{5b}$$

$$\text{and} \quad Z_i = \frac{\rho}{F_S}\, n b_W \left(1-\frac{WW}{GW}\right) \tag{5c}$$

for the ith aircraft.

FUSLGE

The fuselage structural weight estimation equation was developed on the basis of conventional construction methods used by many general aviation manufacturers. A non-dimensional equation for the aircraft's structual fuselage weight based on several parameters was developed as follows.

The fuselage was assumed to be constructed as a thin-wall cylinder with longitudinal stiffeners and trasverse ribs, or bulkheads, as shown below:

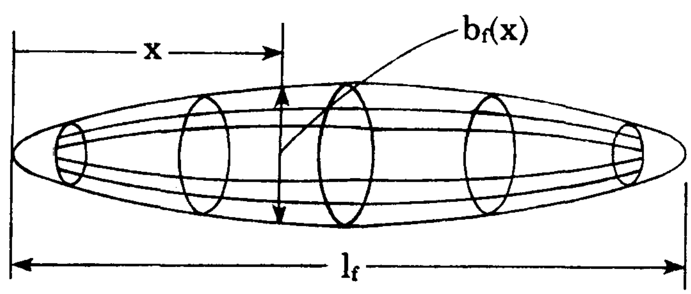


Figure 2: Typical Fuselage Model

This general shape and construction was then approximated by an equivalent monocoque, prismatic beam with transverse ribs such as the following:

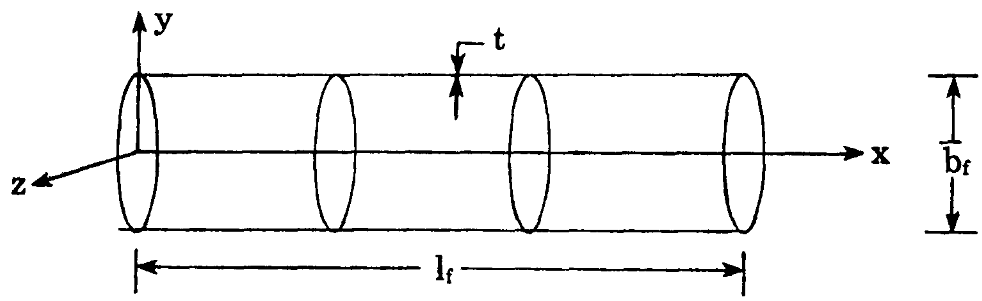


Figure 3: Idealized Fuselage Model

where $\bar{b}_f$ is the average fuselage diameter and t is the equivalent panel thickness to be determined later. In this model, the cylinder skin and longitudinal stringers were replaced with an appropriate panel of thickness, t. The total running weight of the fuselage was then expressed as the sum of the weight of the equivalent panel and the weight of the ribs, i.e.,

$$w_f = W_p + W_r \quad \text{(weight/unit length)} \tag{6}$$

<u>Panel Weight / Unit Length</u>

For the equivalent uniform cross-section that we have used to model the fuselage (Fig. 3) the running weight of the panel material was computed to be

$$W_p = \pi \bar{b}_f t \rho_p \tag{7}$$

where ρ_p is the density of the panel (skin/stringer) material. The equivalent panel thickness was computed as that thickness required to transmit the bending loads of the fuselage. Thus, for the beam with uniform symmetric cross-section and primary transverse loads in the plane of symmetry, as shown below,

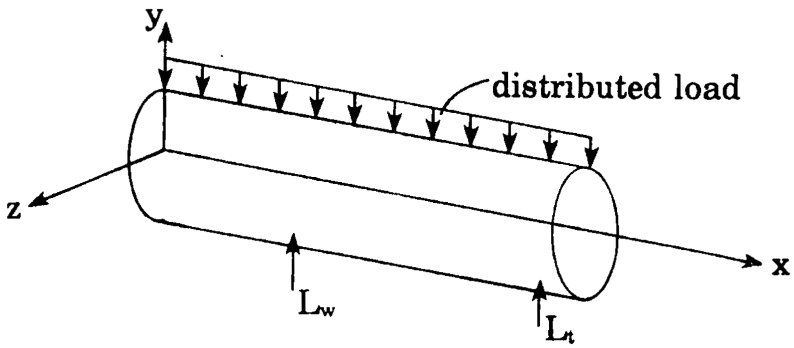


Figure 4: Distributed Load on the Idealized Fuselage

the normal stress on the cross-section is given by

$$\sigma = \frac{My}{I} \tag{8}$$

where: M = M(x) the bending moment about the z axis at station x

y = distance from the neutral axis

I = the moment of inertia about the z-axis.

The load condition of the fuselage was appproximated as shown by the following drawing:

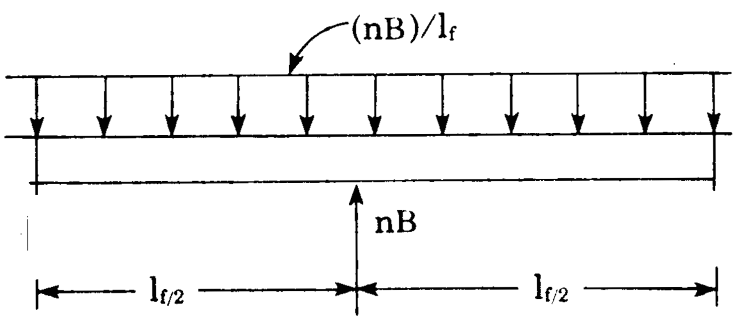


Figure 5: Beam Loading on the Fuselage

This assumed loading leads to a maximum bending moment of

$$M = \frac{nB\ell_f}{8} \tag{9}$$

The moment of inertia is approximated as

$$I = \frac{\pi}{8}\,\bar{b}_f^{\,3}t \tag{10}$$

and the fuselage was designed for the maximum allowable stress of F, so that

$$F = \frac{\frac{nB\ell_f}{8}\,\frac{\bar{b}_f}{2}}{\frac{\pi}{8}\,\bar{b}_f^{\,3}\,t} = \frac{nB\ell_f}{2\pi\bar{b}_f^{\,2}\,t} \tag{11}$$

or the required thickness of:

$$t = \frac{nB\ell_f}{2\pi\bar{b}_f^{\,2}\,F} \tag{12}$$

The weight of the panel is

$$W_p = \pi\bar{b}_f\rho_p\,\frac{nB\ell_f}{2\pi\bar{b}_f^{\,2}\,F} \tag{13a}$$

or

$$W_p = \frac{nB\ell_f\rho_p}{2\bar{b}_f F} \tag{13b}$$

<u>Rib Weight</u>

The transverse ribs were assumed to be of uniform circular shape, as indicated by the following drawing:

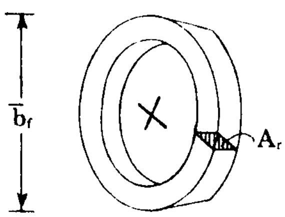


Figure 6: Idealized Fuselage Rib

The weight of ribs per unit length was approximated by

$$W_r = \pi\bar{b}_f A_r\rho_r\frac{n_r}{\ell_f} \tag{14}$$

where: ρ_r = the density of the rib material

n_r = number of ribs

The rib stiffness required to prevent a general instability was obtained from reference [4] in the form:

$$E_r I_r = \frac{C_b M\bar{b}_f^{\,2} n_r}{\ell_f} \tag{15a}$$

where: E_r = Young's modulus of the rib material.

I_r = moment of inertia of the rib cross-section

C_b = a dimensional coefficient

M = maximum bending moment

The moment of inertia, I_r, can be related to the cross-sectional area of the rib, A_r, by a dimensionless factor c_r, such that

$$I_r = c_r A_r^2 \tag{15b}$$

Therefore,
$$E_r c_r A_r^2 = \frac{c_b M \bar{b}_f^2 n_r}{\ell_f} \tag{16}$$

or
$$A_r = \left(\frac{c_b M \bar{b}_f^2 n_r}{E_r c_r \ell_f}\right)^{1/2} \tag{17a}$$

or
$$A_r = k\left(\frac{M \bar{b}_f^2}{E_r \ell_f}\right)^{1/2} \tag{17b}$$

where k is a dimensionless constant, $(c_b/c_r)^{1/2}$.

Therefore, the weight of the ribs per unit length can be written as,

$$W_r = \pi \bar{b}_f^2 \rho_r \frac{n_r}{\ell_f} \; k\left(\frac{M \bar{b}_f^2}{E_r \ell_f}\right)^{1/2} \tag{18}$$

The maximum bending moment was approximated previously by

$$M = \frac{nB\ell_f}{8} \tag{19}$$

Therefore
$$W_r = \tilde{c} \bar{b}_f^2 \rho_r \frac{n_r}{\ell_f} \left(\frac{nB}{E_r}\right)^{1/2} \tag{20}$$

where $\tilde{c}$ is a dimensionless constant.

The weight of the fuselage, W_f, is equal to the weight per unit length, w_f, times the length, ℓ_f. Therefore,

$$W_f = \ell_f (W_p + W_r) \tag{21}$$

$$W_f = \frac{nB\ell_f^2 \rho_p}{2b_f F} + c\bar{b}_f^2 \rho_r n_r \left(\frac{nB}{E_r}\right)^{\frac{1}{2}} \quad (22)$$

This equation was then written in the non-dimensional form:

$$\frac{W_f}{nB} = \frac{1}{2} \frac{\rho_p \ell_f}{F} \frac{\ell_f}{b_f} + \tilde{c} n_r \frac{\rho_r \bar{b}_f}{E_r} \left(\frac{E_r \bar{b}_f^2}{nB}\right)^{\frac{1}{2}} \quad (23)$$

where:

W_f = weight of the fuselage

ρ_p = skin panel density

l_f = fuselage length

$\bar{b}_f$ = mean fuselage width

n_r = number of ribs

$\tilde{c}$ = an empirical dimensionless constant

ρ_r = rib material density

E_r = Young's Modulus of the ribs

This equation has been adapted for regression by replacing the analytical coefficients with coefficients C1, C2, and C3 in the following manner:

$$W_f = C_1 + C_2 \frac{\rho_p}{F} n \frac{B\ell_f^2}{\bar{b}_f} + C_3 \frac{\rho_r}{\sqrt{E_r}} \frac{1}{\sqrt{n}} \frac{\bar{b}_f^2}{\sqrt{B}} \quad (24)$$

Then, as before, $B = GW - WW$ and

$$Y_i = W_f / GW = C1 + C2 * X_i + C3 * Z_i \quad (25a)$$

where: $$X_i = \frac{\left(\frac{\rho_p}{F}\right) n \left(1 - {}^{WW}/_{GW}\right) \ell_f^2}{\bar{b}_f} \quad (25b)$$

and $$Z_i = \frac{\left(\frac{\rho_r}{\sqrt{E_r}}\right) \frac{1}{\sqrt{n}} \bar{b}_f^2}{\sqrt{1 - {}^{WW}/_{GW}}} \quad (25c)$$

HZTAIL

The regression equation for estimating the horizontal tail weight was very similar to that for the wing weight. The major difference between the two equations was that whereas the wing load was B * n, the tail load was considered to be a fraction of the wing load in proportion to the surface areas of each.

$$\frac{L_t}{S_t} = k \frac{L_w}{S_w} \tag{26a}$$

$$L_t = k \frac{S_t}{S_w} Bn \tag{26b}$$

$$L_t = k \frac{S_t}{Sw} n \left(1 - \frac{WW}{GW}\right) GW \tag{26c}$$

Then, the same general form of the regression equation was used; that is:

$$Y_i = W_{hz} / GW = C1 + C2 * Xi + C3 * Zi \tag{27a}$$

$$\text{where: } x_i = \frac{\rho_t}{F} \frac{n}{\cos^2 \Lambda_t} \frac{b_t S_t^2 \left(C_{Rt} + 2C_{Tt}\right)\left(1 - {}^{WW}/_{GW}\right)}{\left(C_{Rt} + C_{Tt}\right)\left(2C_{Rt} + C_{Tt}\right)\left(2T_{Rt} + T_{Tt}\right)S_w} \tag{27b}$$

$$\text{and } z_i = \frac{\rho_t}{F_S} nb_t \frac{S_t}{S_W} \left(1 - \frac{WW}{GW}\right) \tag{27c}$$

The constant k has been combined with C2 and C3.

VTAIL

The regression equation for the vertical tail was developed in a manner similar to that for the horizontal tail. The design load of the vertical tail is proportional to the load on the horizontal tail as their surface areas are proportional.

$$L_v = k \frac{S_v}{S_t} L_t \tag{28a}$$

$$L_v = k \frac{S_v}{S_t} \frac{S_t}{S_w} Bn \tag{28b}$$

$$L_v = k \frac{S_v}{S_w} n \left(1 - \frac{WW}{GW}\right) GW \quad (28c)$$

Again, the regression equation was of the general form

$$Y_i = Wvt / GW = C1 + C2 * Xi + C3 * Zi \quad (29a)$$

where: $$X_i = \frac{\rho_v}{F} \frac{n}{\cos^2 \Lambda_v} \frac{b_v S_v^2 \left(1 - \frac{WW}{GW}\right)(C_{RV} + 2C_{TV})}{(C_{RV}+C_{TV})(2C_{RV}+C_{TV})(2T_{RV}+T_{TV}) S_w} \quad (29b)$$

and $$Z_i = \frac{\rho_v}{F_s} n\, b_v \frac{S_v}{S_W}\left(1 - \frac{WW}{GW}\right) \quad (29c)$$

The input values for these regression equations were obtained from a variety of sources. Certain approximations and assumptions were required in order to make use of these equations and these are listed as follows:

1. Panel weight density was assumed .1 lb/in^3 for all models evaluated.
2. Rib weight density was assumed .1 lb/in^3 for all models evaluated.
3. Allowable stress F was assumed to be 65000 lb/in^2.
4. Young's Modulus Eribs was assumed 10.6(106) psi.
5. Load factor was 1.5(3.8) = 5.7
6. The aircraft total length was used as the length of the fuselage. This could represent an error of 5% to 10% for those aircraft with swept tail and fin.
7. Average fuselage diameter $\bar{b}_f$ (BBF in the program) was approximated by using 0.67 maximum fuselage width.

Equation Form

Equations of the form $Y = C1 + C2 * X + C3 * Z$ and of the form $Y = C1 * X^{C_2} * Z^{C_3}$ were generated by this program.

The coefficients of both equations were determined by a linear least squares method in which the sum of the squares of the error (Y_{actual} - Ycalculated) is minimized. The same procedure was used for both forms of the regression with the second equation being cast in the form of the first equation by taking the natural log of both sides of the equation. The Grumman investigators proposed this second method to accomodate the vast range of weights in their data set (the T-37 weighs 4889 pounds at take-off, while the C-5A weighs 728,000 pounds). The same method was included in this study but the results indicated that the logarithmic method was not required, since improved accuracy was not achieved. Development of the general parameters Xi, Yi, and Zi was the same for both methods.

The coefficients of the equation of the form

$$Y_i = C1 + C2 * Xi + C3 * Zi$$

can be computed for use in the least-squares curve fit by solving the set of linear algebraic equations shown below:

$$\Sigma Y = nC_1 + C_2\Sigma X + C_3\Sigma Z \tag{30a}$$

$$\Sigma XY = C_1\Sigma X + C_2\Sigma X^2 + C_3\Sigma XZ \tag{30b}$$

$$\Sigma ZY = C_1\Sigma Z + C_2\Sigma XZ + C_3\Sigma Z^2 \tag{30c}$$

where:

$\Sigma Y = \sum_{i=1}^{n} Yi$, the sum of the known weight parameter, Yi

$\Sigma X = \sum_{i=1}^{n} Xi$, the sum of the known independent variable, Xi

$\Sigma Z = \sum_{i=1}^{n} Zi$, the sum of the known independent variable, Zi

$\Sigma XY = \sum_{i=1}^{n} X_i Y_i$

$$\Sigma ZY = \sum_{i=1}^{n} Z_i Y_i$$

$$\Sigma X^2 = \sum_{i=1}^{n} X_i^2$$

$$\Sigma XZ = \sum_{i=1}^{n} X_i Z_i$$

$$\Sigma Z^2 = \sum_{i=1}^{n} Z_i^2$$

n = number of aircraft in the data set.

Expressed in matrix form:

$$\begin{pmatrix} n & \Sigma X & \Sigma Z \\ \Sigma X & \Sigma X^2 & \Sigma XZ \\ \Sigma Z & \Sigma XZ & \Sigma Z^2 \end{pmatrix} \begin{pmatrix} C_1 \\ C_2 \\ C_3 \end{pmatrix} = \begin{pmatrix} \Sigma Y \\ \Sigma XY \\ \Sigma ZY \end{pmatrix} \tag{30d}$$

In this form we see that the first and last matrix of the equation are matrices of known quantities and thus we can solve for the unknown coefficients C1, C2, and C3. Once the coefficients have been determined we have a weight estimation equation $Y = C1 + C2 * X + C3 * Z$ that can be used to predict the component weight of an aircraft with known parameters X and Z.

A feature of the program GENREG, as indicated in step 9 of the flow-chart, is the following. The geometry and material description of each airplane is resubmitted to sub-programs for the purpose of evaluating each airplane on the basis of the newly generated coefficients (see the program listing for details, Appendix C). Since the wing weight appears on both sides of the wing weight prediction equation (equation 5), a linear interpolation procedure is used to converge on the best estimate of the wing weight. The solution converges rapidly for a reasonably small difference factor EPS (see Appendix C). The normalized ratio of wing weight to aircraft gross weight, WW/GW, is then used in the calculations of the other three aircraft component weights (equations 25,27,29), so the iterative procedure is needed only during the wing weight calculation for each airplane.

For subsequent use of the coefficients generated by GENREG FORTRAN, a flexible program PHSEII FORTRAN (Appendix C) has been written. With this program, airplanes not used in the GENREG regression may be evaluated using the original coefficients of GENREG, or, new coefficents may be supplied interactively. PHSEII makes use of the option to explore alternate material constants, i.e., composites.

Table 4 lists input and output formats along with representative values for the GENREG and PHSEII computer codes. Table 4a contains a chart of the inputs required for the WSU GENREG computer code indicating the units for each parameter. Table 4b contains a listing of the current 12-airplane data base of the GENREG code. The output from the GENREG program is presented in TABLE 4c for the 12-airplane data base mentioned above. Note that the coefficients for both the linear and power series form of the equation have been generated for the four airplane component weights and that the program also computes an estimate of the four components for each airplane of the data set. Table 4d contains a chart of the inputs for the PHSEII code while Tables 4e and 4f contain typical input and output for the PHSEII code for an airplane that is a part of the 12-airplane data set. Tables 4g and 4h contain similar input and output information for an airplane not in the data set.

TABLE 4a. TYPICAL INPUT FOR WSU'S GENREG PROGRAM

MODEL NAME	AIRPLANE GROSS WT (lb)	ACTUAL WING WT (lb)	ACTUAL FUSELAGE WT (lb)	ACTUAL HZ TAIL WT (lb)	ACTUAL VT FIN WT (lb)	DESIGN LOAD FACTOR
	Wing skin Spc. wt (lb/in^3)	Fuse panel Spc. wt (lb/in^3)	Fuse rib Spc. wt (lb/in^3)	Rib modulus E_r 10^{-6} psi	Tail skin Spc. wt (lb/in^3)	Fin skin Spc. wt (lb/in^3)
	Allowable Wing Cover Stress (lb/in^2)	Ultimate Wing Shear Stress (lb/in^2)	Allowable Tail Cover Stress (lb/in^2)	Ultimate Tail Shear Stress (lb/in^2)	Allowable Fin Cover Stress (lb/in^2)	Ultimate Fin Shear Stress (lb/in^2)
	Wing Span (in)	Wing Area (in^2)	Hz Tail Span (in)	Hz Tail Area (in^2)	Vt. Fin Span (in)	Vt. Fin Area (in^2)
	Fuselage* mean dia. (in)	Fuselage length (in)	Fuselage cover stress (lb/in^2)	wing (rad)	tail (rad)	fin (rad)
	Wing root chord (in)	Wing tip chord (in)	Tail root chord (in)	Tail tip chord (in)	Fin root chord (in)	Fin tip chord (in)
	Wing root thickness (in)	Wing tip thickness (in)**	Tail root thickness (in)	Tail tip thickness (in)	Fin root thickness (in)	Fin tip thickness (in)

* assume 2/3 max. width if unknown

** assume thickness ratio equal taper ratio if unknown

Table 4b. GENREG Data-base

12

APLNE A	6850.	642.	680.	83.	78.	5.25
APLNE B	9850.	885.	935.	147.	90.	5.25
APLNE C	11850.	1073.	970.	175.	121.	5.70
APLNE D	18300.	1389.	1650.	222.	148.	4.50
APLNE E	21000.	1959.	2506.	260.	174.	4.50
APLNE F	5100.	445.	712.	54.6	36.4	5.25
APLNE G	8000.	786.	892.	113.6	78.4	5.25
APLNE H	12500.	1212.	1758.	180.	120.	5.25
APLNE I	3800.	345.4	480.	51.2	34.2	5.7
APLNE J	5150.	500.4	368.	73.2	48.8	5.25
APLNE K	6650.	838.0	742.0	120.	80.	5.25
APLNE L	3100.	297.6	349.2	45.6	28.2	5.7
A	.1	.1	.1	10.6	.1	.1
B	.1	.1	.1	10.6	.1	.1
C	.1	.1	.1	10.6	.1	.1
D	.1	.1	.1	10.6	.1	.1
E	.1	.1	.1	10.6	.1	.1
F	.1	.1	.1	10.6	.1	.1
G	.1	.1	.1	10.6	.1	.1
H	.1	.1	.1	10.6	.1	.1
I	.1	.1	.1	10.6	.1	.1
J	.1	.1	.1	10.6	.1	.1
K	.1	.1	.1	10.6	.1	.1
L	.1	.1	.1	10.6	.1	.1
A	65000.	24200.	65000.	24200.	65000.	24200.
B	65000.	24200.	65000.	24200.	65000.	24200.
C	65000.	24200.	65000.	24200.	65000.	24200.
D	65000.	24200.	65000.	24200.	65000.	24200.
E	65000.	24200.	65000.	24200.	65000.	24200.
F	65000.	24200.	65000.	24200.	65000.	24200.
G	65000.	24200.	65000.	24200.	65000.	24200.
H	65000.	24200.	65000.	24200.	65000.	24200.
I	65000.	24200.	65000.	24200.	65000.	24200.
J	65000.	24200.	65000.	24200.	65000.	24200.
K	65000.	24200.	65000.	24200.	65000.	24200.
L	65000.	24200.	65000.	24200.	65000.	24200.
A	529.	32400.	204.	8740.	83.	5940.
B	588.	36395.	229.	8927.	91.	6271.
C	562.	40100.	228.	10300.	108.	7322.
D	458.	36470.	176.	7780.	66.	5520.
E	525.	38090.	176.	7780.	81.	7240.
F	576.	38300.	190.	7400.	87.4	5400.
G	696.	52130.	233.	11730.	115.	8700.
H	780.	60480.	248.	14400.	148.	6910.
I	430.	25060.	116.	3580.	47.	1302.
J	468.	27240.	204.	6730.	85.	3890.
K	622.	40250.	246.	11480.	93.4	4643.
L	430.	25060.	140.	3310.	66.7	1670.

Table 4b. continued

A	40.	405.	65000.	0.	.0653	.6976
B	40.	441.	65000.	0.	.0641	.6974
C	42.7	522.	65000.	.0246	.0930	.5742
D	42.	554.	65000.	.0349	.4887	.8727
E	51.3	627.	65000.	.2269	.4887	.8727
F	59.	380.	65000.	0.	0.	.18
G	45.	444.	65000.	0.	0.	.18
H	42.2	595.	65000.	0.	0.	.44
I	37.	250.	65000.	0.	0.	.7
J	52.6	330.	65000.	0.	0.	.7
K	47.7	451.	65000.	0.	.0873	.6974
L	48.6	336.	65000.	0.	0.	.7
A	69.8	41.2	54.3	34.	98.3	44.2
B	70.	42.7	50.	28.	98.3	38.3
C	106.	36.9	60.	29.3	96.3	44.
D	108.	60.9	53.3	32.	119.	78.1
E	122.2	47.7	53.3	32.	130.	85.4
F	66.5	66.5	53.2	24.7	95.	28.5
G	74.9	74.9	60.	40.8	108.	43.
H	78.	78.	55.2	55.2	108.	61.2
I	64.	43.	38.5	22.	35.2	20.9
J	69.	46.5	46.	26.3	67.5	24.
K	70.	43.1	50.	28.	98.25	38.32
L	64.	43.	52.	32.	50.8	29.
A	12.57	7.41	4.89	3.1	11.79	5.3
B	12.6	7.7	6.0	3.36	11.79	4.6
C	14.84	5.2	6.0	2.93	11.55	5.3
D	9.72	5.5	4.8	2.9	10.7	7.0
E	11.0	4.3	4.8	2.9	11.7	7.7
F	10.5	10.5	4.6	4.6	7.6	2.5
G	10.8	10.8	4.8	4.8	7.7	3.4
H	15.6	15.6	7.2	7.2	10.8	7.2
I	5.5	3.4	3.1	1.5	3.5	1.5
J	12.6	4.2	6.3	3.3	6.9	3.7
K	12.6	7.76	6.	3.36	11.79	4.6
L	7.25	3.6	2.9	1.5	4.5	2.

Table 4c. GENREG Output

REGRESSION VALUES FOR THE MAIN WING

THE LINEAR MATRICES

0.1200E+02	0.4270E+00	0.1304E+00		C1		0.1136E+01
0.4270E+00	0.1656E-01	0.4804E-02	*	C2	=	0.4090E-01
0.1304E+00	0.4804E-02	0.1469E-02		C3		0.1244E-01

Y = C1 + C2 * X + C3 * Z :

C1 = 0.743525E-01

C2 = 0.209865E+00

C3 = 0.118106E+01

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	642.0	635.6	0.99
APLNE B	885.0	940.8	1.06
APLNE C	1073.0	1113.5	1.04
APLNE D	1389.0	1608.4	1.16
APLNE E	1959.0	1879.4	0.96
APLNE F	445.0	488.4	1.10
APLNE G	786.0	814.1	1.04
APLNE H	1212.0	1276.7	1.05
APLNE I	345.4	362.5	1.05
APLNE J	500.4	465.3	0.93
APLNE K	838.0	648.4	0.77
APLNE L	297.6	289.3	0.97

MEAN: 1.0100, DEL PRCT: 1.00%, VARIANCE: 0.0096, STD. DEV.: 9.77%

Table 4c. continued

MAIN WING REGRESSION VALUES (CONT.)

THE LOG MATRICES

$$\begin{bmatrix} 0.1200E+02 & -0.4060E+02 & -0.5448E+02 \\ -0.4060E+02 & 0.1385E+03 & 0.1847E+03 \\ -0.5448E+02 & 0.1847E+03 & 0.2477E+03 \end{bmatrix} * \begin{bmatrix} C1 \\ C2 \\ C3 \end{bmatrix} = \begin{bmatrix} -0.2837E+02 \\ 0.9613E+02 \\ 0.1289E+03 \end{bmatrix}$$

Y = C1 * X ** C2 * Z ** C3 :

C1 = 0.249955E+00

C2 = 0.835551E-01

C3 = 0.153023E+00

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	642.0	633.9	0.99
APLNE B	885.0	941.9	1.06
APLNE C	1073.0	1109.6	1.03
APLNE D	1389.0	1574.5	1.13
APLNE E	1959.0	1853.7	0.95
APLNE F	445.0	489.0	1.10
APLNE G	786.0	811.8	1.03
APLNE H	1212.0	1274.9	1.05
APLNE I	345.4	359.9	1.04
APLNE J	500.4	460.7	0.92
APLNE K	838.0	649.4	0.77
APLNE L	297.6	288.2	0.97

MEAN: 1.0046, DEL PRCT: 0.46%, VARIANCE: 0.0090, STD. DEV.: 9.51%

Table 4c. continued

REGRESSION VALUES FOR THE FUSELAGE

THE LINEAR MATRICES

$$\begin{bmatrix} 0.1200E+02 & 0.3997E+00 & 0.3608E+00 \\ 0.3997E+00 & 0.1601E-01 & 0.1164E-01 \\ 0.3608E+00 & 0.1164E-01 & 0.1171E-01 \end{bmatrix} * \begin{bmatrix} C1 \\ C2 \\ C3 \end{bmatrix} = \begin{bmatrix} 0.1299E+01 \\ 0.4339E-01 \\ 0.3946E-01 \end{bmatrix}$$

Y = C1 + C2 * X + C3 * Z :

C1 = 0.895039E-01

C2 = 0.111449E+00

C3 = 0.500767E+00

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	680.0	713.3	1.05
APLNE B	935.0	1031.8	1.10
APLNE C	970.0	1273.8	1.31
APLNE D	1650.0	1977.3	1.20
APLNE E	2506.0	2413.1	0.96
APLNE F	712.0	591.9	0.83
APLNE G	892.0	858.9	0.96
APLNE H	1758.0	1361.3	0.77
APLNE I	480.0	381.0	0.79
APLNE J	368.0	570.2	1.55
APLNE K	742.0	725.1	0.98
APLNE L	349.2	333.4	0.95

MEAN: 1.0392, DEL PRCT: 3.92%, VARIANCE: 0.0527, STD. DEV.: 22.95%

Table 4c. continued

FUSELAGE REGRESSION VALUES (CONT.)

THE LOG MATRICES

0.1200E+02	-0.4216E+02	-0.4250E+02		C1		-0.2691E+02
-0.4216E+02	0.1510E+03	0.1491E+03	*	C2	=	0.9454E+02
-0.4250E+02	0.1491E+03	0.1514E+03		C3		0.9535E+02

Y = C1 * X ** C2 * Z ** C3 :

C1 = 0.136667E+00

C2 = 0.540585E-02

C3 = 0.658196E-01

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	680.0	715.7	1.05
APLNE B	935.0	1030.1	1.10
APLNE C	970.0	1248.9	1.29
APLNE D	1650.0	1938.5	1.17
APLNE E	2506.0	2284.4	0.91
APLNE F	712.0	559.2	0.79
APLNE G	892.0	849.2	0.95
APLNE H	1758.0	1320.3	0.75
APLNE I	480.0	390.2	0.81
APLNE J	368.0	555.8	1.51
APLNE K	742.0	711.2	0.96
APLNE L	349.2	330.5	0.95

MEAN: 1.0204, DEL PRCT: 2.04%, VARIANCE: 0.0495, STD. DEV.: 22.25%

Table 4c. continued

REGRESSION VALUES FOR THE HORIZONTAL TAIL

THE LINEAR MATRICES

0.1200E+02	0.2586E-01	0.1067E-01		C1		0.1661E+00
0.2586E-01	0.6517E-04	0.2612E-04	*	C2	=	0.3683E-03
0.1067E-01	0.2612E-04	0.1070E-04		C3		0.1511E-03

Y = C1 + C2 * X + C3 * Z :

C1 = 0.114751E-01

C2 = 0.115779E+01

C3 = -0.147074E+00

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	83.0	98.6	1.19
APLNE B	147.0	140.5	0.96
APLNE C	175.0	173.3	0.99
APLNE D	222.0	247.8	1.12
APLNE E	260.0	282.5	1.09
APLNE F	54.6	66.5	1.22
APLNE G	113.6	115.9	1.02
APLNE H	180.0	179.0	0.99
APLNE I	51.2	47.1	0.92
APLNE J	73.2	69.1	0.94
APLNE K	120.0	106.5	0.89
APLNE L	45.6	37.8	0.83

MEAN: 1.0126, DEL PRCT: 1.26%, VARIANCE: 0.0144, STD. DEV.: 11.98%

Table 4c. continued

HZ. TAIL REGRESSION VALUES (CONT.)

THE LOG MATRICES

$$\begin{bmatrix} 0.1200E+02 & -0.7495E+02 & -0.8527E+02 \\ -0.7495E+02 & 0.4712E+03 & 0.5350E+03 \\ -0.8527E+02 & 0.5350E+03 & 0.6081E+03 \end{bmatrix} * \begin{bmatrix} C1 \\ C2 \\ C3 \end{bmatrix} = \begin{bmatrix} -0.5146E+02 \\ 0.3217E+03 \\ 0.3659E+03 \end{bmatrix}$$

Y = C1 * X ** C2 * Z ** C3 :

C1 = 0.317396E-01

C2 = -0.206676E-01

C3 = 0.136166E+00

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	83.0	97.0	1.17
APLNE B	147.0	139.9	0.95
APLNE C	175.0	170.8	0.98
APLNE D	222.0	242.6	1.09
APLNE E	260.0	277.0	1.07
APLNE F	54.6	69.2	1.27
APLNE G	113.6	112.4	0.99
APLNE H	180.0	178.7	0.99
APLNE I	51.2	47.3	0.92
APLNE J	73.2	72.6	0.99
APLNE K	120.0	96.4	0.80
APLNE L	45.6	39.4	0.86

MEAN: 1.0072, DEL PRCT: 0.72%, VARIANCE: 0.0162, STD. DEV.: 12.75%

Table 4c. continued

REGRESSION VALUES FOR THE VERTICAL TAIL

THE LINEAR MATRICES

$$\begin{bmatrix} 0.1200E+02 & 0.2348E-02 & 0.3002E-02 \\ 0.2348E-02 & 0.5370E-06 & 0.6749E-06 \\ 0.3002E-02 & 0.6749E-06 & 0.8808E-06 \end{bmatrix} * \begin{bmatrix} C1 \\ C2 \\ C3 \end{bmatrix} = \begin{bmatrix} 0.1132E+00 \\ 0.2234E-04 \\ 0.2873E-04 \end{bmatrix}$$

Y = C1 + C2 * X + C3 * Z :

C1 = 0.879748E-02

C2 = -0.476586E+01

C3 = 0.628541E+01

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	78.0	66.2	0.85
APLNE B	90.0	94.6	1.05
APLNE C	121.0	117.6	0.97
APLNE D	148.0	169.0	1.14
APLNE E	174.0	197.0	1.13
APLNE F	36.4	48.7	1.34
APLNE G	78.4	76.6	0.98
APLNE H	120.0	125.5	1.05
APLNE I	34.2	33.3	0.97
APLNE J	48.8	46.5	0.95
APLNE K	80.0	63.5	0.79
APLNE L	28.2	27.7	0.98

MEAN: 1.0175, DEL PRCT: 1.75%, VARIANCE: 0.0206, STD. DEV.: 14.36%

Table 4c. continued

VT. TAIL REGRESSION VALUES (CONT.)

THE LOG MATRICES

$$\begin{bmatrix} 0.1200E+02 & -0.1036E+03 & -0.1011E+03 \\ -0.1036E+03 & 0.8968E+03 & 0.8753E+03 \\ -0.1011E+03 & 0.8753E+03 & 0.8553E+03 \end{bmatrix} * \begin{bmatrix} C1 \\ C2 \\ C3 \end{bmatrix} = \begin{bmatrix} -0.5607E+02 \\ 0.4842E+03 \\ 0.4725E+03 \end{bmatrix}$$

Y = C1 * X ** C2 * Z ** C3 :

C1 = 0.122276E-01

C2 = -0.618665E-01

C3 = 0.952747E-01

MODEL	ACTUAL WEIGHT	CALCULATED WEIGHT	NORM
APLNE A	78.0	65.2	0.84
APLNE B	90.0	93.4	1.04
APLNE C	121.0	113.7	0.94
APLNE D	148.0	170.1	1.15
APLNE E	174.0	196.9	1.13
APLNE F	36.4	48.4	1.33
APLNE G	78.4	75.8	0.97
APLNE H	120.0	121.6	1.01
APLNE I	34.2	32.6	0.95
APLNE J	48.8	47.3	0.97
APLNE K	80.0	63.5	0.79
APLNE L	28.2	27.8	0.99

MEAN: 1.0090, DEL PRCT: 0.90%, VARIANCE: 0.0207, STD. DEV.: 14.40%

TABLE 4d. TYPICAL INPUT FOR WSU'S PHSEII PROGRAM

MODEL NAME	AIRPLANE GROSS WT (lb)	DESIGN LOAD FACTOR				
	Wing skin Spc. wt (lb/in^3)	Fuse panel Spc. wt (lb/in^3)	Fuse rib Spc. wt (lb/in^3)	Rib modulus E_r 10^{-6} psi	Tail skin Spc. wt (lb/in^3)	Fin skin Spc. wt (lb/in^3)
	Allowable Wing Cover Stress (lb/in^2)	Ultimate Wing Shear Stress (lb/in^2)	Allowable Tail Cover Stress (lb/in^2)	Ultimate Tail Shear Stress (lb/in^2)	Allowable Fin Cover Stress (lb/in^2)	Ultimate Fin Shear Stress (lb/in^2)
	Wing Span (in)	Wing Area (in^2)	Hz Tail Span (in)	Hz Tail Area (in^2)	Vt. Fin Span (in)	Vt. Fin Area (in^2)
	Fuselage* mean dia. (in)	Fuselage length (in)	Fuselage cover stress (lb/in^2)	wing (rad)	tail (rad)	fin (rad)
	Wing root chord (in)	Wing tip chord (in)	Tail root chord (in)	Tail tip chord (in)	Fin root chord (in)	Fin tip chord (in)
	Wing root thickness (in)	Wing tip thickness (in)**	Tail root thickness (in)	Tail tip thickness (in)	Fin root thickness (in)	Fin tip thickness (in)

* assume 2/3 max. width if unknown
** assume thickness ratio equal taper ratio if unknown

Table 4e. PHSEII data, using Airplane A

AIRPLANE A

6850.	5.25				
.1	.1	.1	10.6	.1	.1
65000.	24200.	65000.	24200.	65000.	24200.
529.	32400.	204.	8740.	83.	5940.
40.	405.	65000.	0.	.0653	.6976
69.8	41.2	54.3	34.	98.3	44.2
12.57	7.41	4.89	3.1	11.79	5.3

Table 4f. PHSEII Output for Airplane A

TITLE: AIRPLANE A

WING WEIGHT:	635.60	POUNDS
FUSELAGE WEIGHT:	713.28	POUNDS
HZ. TAIL WEIGHT:	98.68	POUNDS
VT. TAIL WEIGHT:	66.22	POUNDS

Table 4g. PHSEII Data, Using a Typical General Aviation Airplane (Airplane M) Which was not part of the GENREG Data-base

AIRPLANE M, USING COEF. OF A-L DATA BASE

12500.0000	5.2500				
0.1000	0.1000	0.1000	10.6000	0.1000	0.1000
65000.0000	24200.0000	65000.0000	24200.0000	65000.0000	24200.0000
654.0000	43632.0000	221.0000	9791.0000	91.1000	7525.0000
42.0000	525.0000	65000.0000	0.0	0.2094	0.5585
94.3200	39.2400	49.1700	24.5800	12.1200	95.6400
17.4900	7.2900	5.9000	2.9500	8.8100	69.4000

Table 4h. PHSEII Output for Airplane M

TITLE: AIRPLANE M, USING COEF. OF A-L DATA BASE

WING WEIGHT:	1192.31	POUNDS
FUSELAGE WEIGHT:	1341.22	POUNDS
HZ. TAIL WEIGHT:	181.11	POUNDS
VT. TAIL WEIGHT:	121.25	POUNDS

4. NORMALIZED WEIGHT ESTIMATES FOR GENERAL AVIATION CONVENTIONAL AND COMPOSITE AIRPLANES

The basic concept of the WSU weight estimation methods is applicable for both conventional metallic and non-conventinal composite airplanes since the equations contain explicit material physical properties. The WSU weight estimation methods were developed to predict the wing, fuselage and empennage weight of general aviation airplanes. Torenbeek's methods are applicable to estimate the aircraft component weights of only metallic airplanes, whereas, Nicolai and WSU methods are applicable for both conventional and composite airplanes. The Nicolai code uses some knock-down factors to evaluate the weights of aircraft component made of composite materials. In order to protect the proprietary nature of the component actual weight data supplied to WSU by the manufacturers for this study, the estimated aircraft component weights were normalized with respect to their actual weights. The normalized factor 'k' for any aircraft component is defined as follows:

$$\text{Normalized factor, } k = \frac{\text{Estimated weight}}{\text{Actual weight}}$$

Tables 5 thru 8 contain the normalized factors 'k' for the wings, fuselages, wing and fuselage and empennage for some of the conventional and composite airplanes in the general aviation industry, using these three different methods described. The accuracy of these methods in predicting the weights of aircraft components is indicated by the proximity of 'k' to unity. Normalized factor $k>1$, indicates that the components predicted weight is greater than the actual weight and vice versa for $k<1$.

Caution should be exercised to carefully assess the results in any particular situation. For example, most of the Cessna models have wings attached to the fuselage and the actual wing weight, supplied by Cessna, does not include the carry-through structure. Cessna includes carry-through structures in fuselage weight, while most of the equations used to estimate the wing weight assume the carry through structure as part of the wing. Hence, for most of the Cessna models, the wing normalized factors k are greater than 1 and the k for the fuselage less than 1. Note that $k \approx 1$ for the wing and fuselage combination as would be appropriate.

The estimated aircraft component's weights vs. normalized gross take off weights are presented graphically in figures 7 thru 10 using Torenbeek, Nicolai and WSU methods. The actual weights of aircraft components were also indicated in these charts for different airplane models. Again, the names of airplane models were omitted to preserve the confidentiality of the data.

The WSU method seemed to predict the component weights much better than the other two methods. One of the reasons for the apparent accuracy of the WSU method was that the aircraft for the comparison study were in the data base used in the regression analysis. However, aircraft component's weights of other model not within the data base, such as Learjet 25D, were predicted using the WSU method and these estimated weights were also comparable with Torenbeek and Nicolai estimates. Aircraft component weights for LearFan 2100, Avtek 400 and DeVore V-100 were also estimated by both Nicolai and WSU methods. These predicted weights by the three different methods and the normalized aircraft component weight factors are shown in Tables 5 thru 8.

The chief advantage of the WSU method is its applicability for both conventional and composite airplanes. Also, the normalized data seemed to indicate that the WSU method predicts weight better than the other two existing methods for various aircraft models considered in this investigation.

Table 5. Normalized Wing Weight Factors

AIRPLANES	TORENBEEK	NICOLAI	WSU
Cessna Chancellor 414A	1.350	0.870	0.960
Cessna Conquest II	1.425	0.900	1.030
Cessna Citation I	1.720	0.950	1.030
Cessna Caravan 208	1.280	0.786	0.780
Learjet Model 35A	0.812	0.791	0.986
Learjet Model 55	0.818	0.669	0.910
Learjet Model 25D	0.650	0.590	0.900
DeVore 'Sundancer'* Model 100	N/A	0.810**	0.880**
LearFan 2100*	N/A	0.750	1.410
Avtek 400*	N/A	0.510	1.240

Note: Normalized weight factor $k = \frac{\text{Estimated Weight}}{\text{Actual Weight}}$

* Normalized weight factors for these airplanes are based on WSU's estimates for the material properties such as density and allowable strengths.

** Normalized weight factors are based on the 'designed weights' instead of 'actual weights' of the components.

Table 6. Normalized Fuselage Weight Factors

AIRPLANES	TORENBEEK	NICOLAI	WSU
Cessna Chancellor 414A	0.710	0.880	0.970
Cessna Conquest II	0.596	0.770	0.950
Cessna Citation I	0.960	1.230	1.287
Cessna Caravan 208	0.613	0.920	1.020
Learjet Model 35A	0.855	0.756	0.960
Learjet Model 55	0.623	0.728	0.970
Learjet Model 25D	0.900	0.730	1.040
DeVore 'Sundancer' * Model 100	N/A	1.040 **	1.100 **
LearFan 2100 *	N/A	0.610	0.780
Avtek 400 *	N/A	0.970	1.260

Note: Normalized weight factor $k = \frac{\text{Estimated Weight}}{\text{Actual Weight}}$

* Normalized weight factors for these airplanes are based on WSU's estimates for the material properties such as density and allowable strengths.

** Normalized weight factors are based on the 'designed weights' instead of 'actual weights' of the components.

Table 7. Normalized Wing & Fuselage Weight Factors

AIRPLANES	TORENBEEK	NICOLAI	WSU
Cessna Chancellor 414A	1.020	0.870	0.966
Cessna Conquest II	0.999	0.83	0.987
Cessna Citation I	1.360	1.080	1.154
Cessna Caravan 208	0.970	0.848	0.894
Learjet Model 35A	0.835	0.772	0.970
Learjet Model 55	0.664	0.702	0.942
Learjet Model 25D	0.780	0.670	0.970
DeVore 'Sundancer' * Model 100	N/A	0.910 **	0.960 **
LearFan 2100 *	N/A	0.660	1.001
Avtek 400 *	N/A	0.750	1.250

Note: Normalized weight factor $k = \frac{\text{Estimated Weight}}{\text{Actual Weight}}$

* Normalized weight factors for these airplanes are based on WSU's estimates for the material properties such as density and allowable strengths.

** Normalized weight factors are based on the 'designed weights' instead of 'actual weights' of the components.

Table 8. Normalized Empennage Weight Factors

AIRPLANES	TORENBEEK	NICOLAI	WSU
Cessna Chancellor 414A	0.894	0.510	0.992
Cessna Conquest II	0.624	0.540	1.060
Cessna Citation I	1.080	0.540	0.960
Cessna Caravan 208	0.584	0.620	0.849
Learjet Model 35A	0.716	0.390	1.040
Learjet Model 55	0.676	0.388	0.980
Learjet Model 25D	1.040	0.540	1.360
DeVore 'Sundancer' * Model 100	N/A	0.630 **	0.640 **
LearFan 2100 *	N/A	0.400	1.180
Avtek 400 *	N/A	0.300	1.090

Note: Normalized weight factor $k = \frac{\text{Estimated Weight}}{\text{Actual Weight}}$

* Normalized weight factors for these airplanes are based on WSU's estimates for the material properties such as density and allowable strengths.

** Normalized weight factors are based on the 'designed weights' instead of 'actual weights' of the components.

Figure 7. Statistical and Actual Wing Weight vs. Normalized Gross Take-off Weight

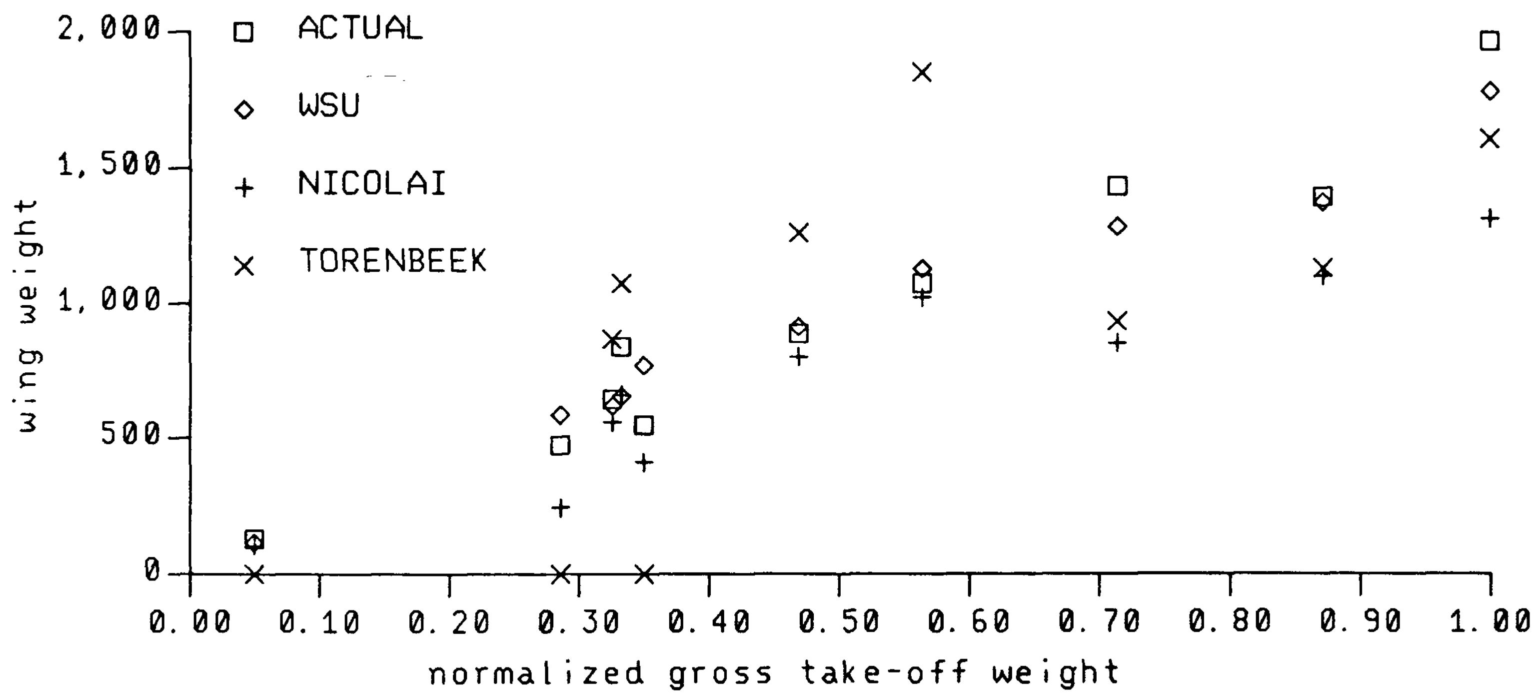


Figure 8. Statistical and Actual Fuselage Weight Vs. Normalized Gross Take-off Weight

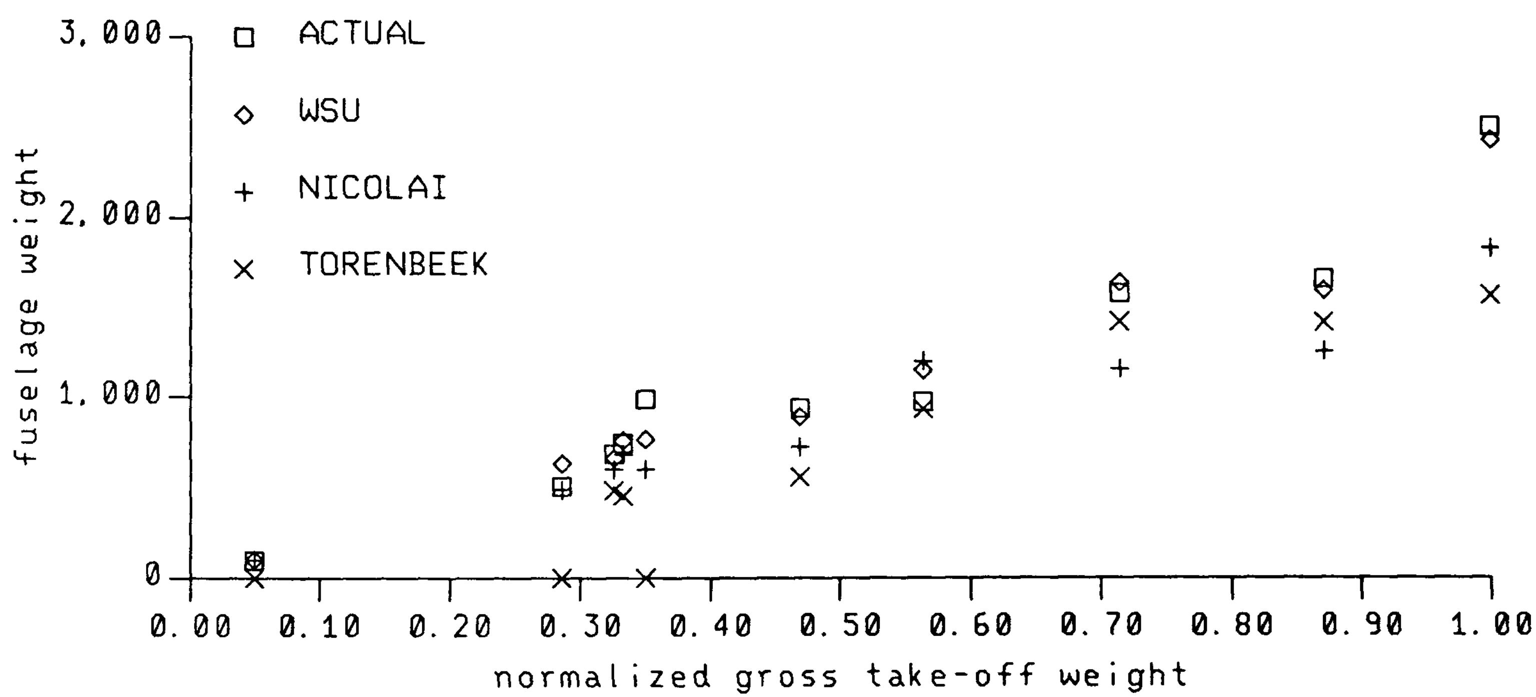


Figure 9. Statistical and Actual Wing+Fuselage Weight Vs. Normalized Gross Take-off Weight

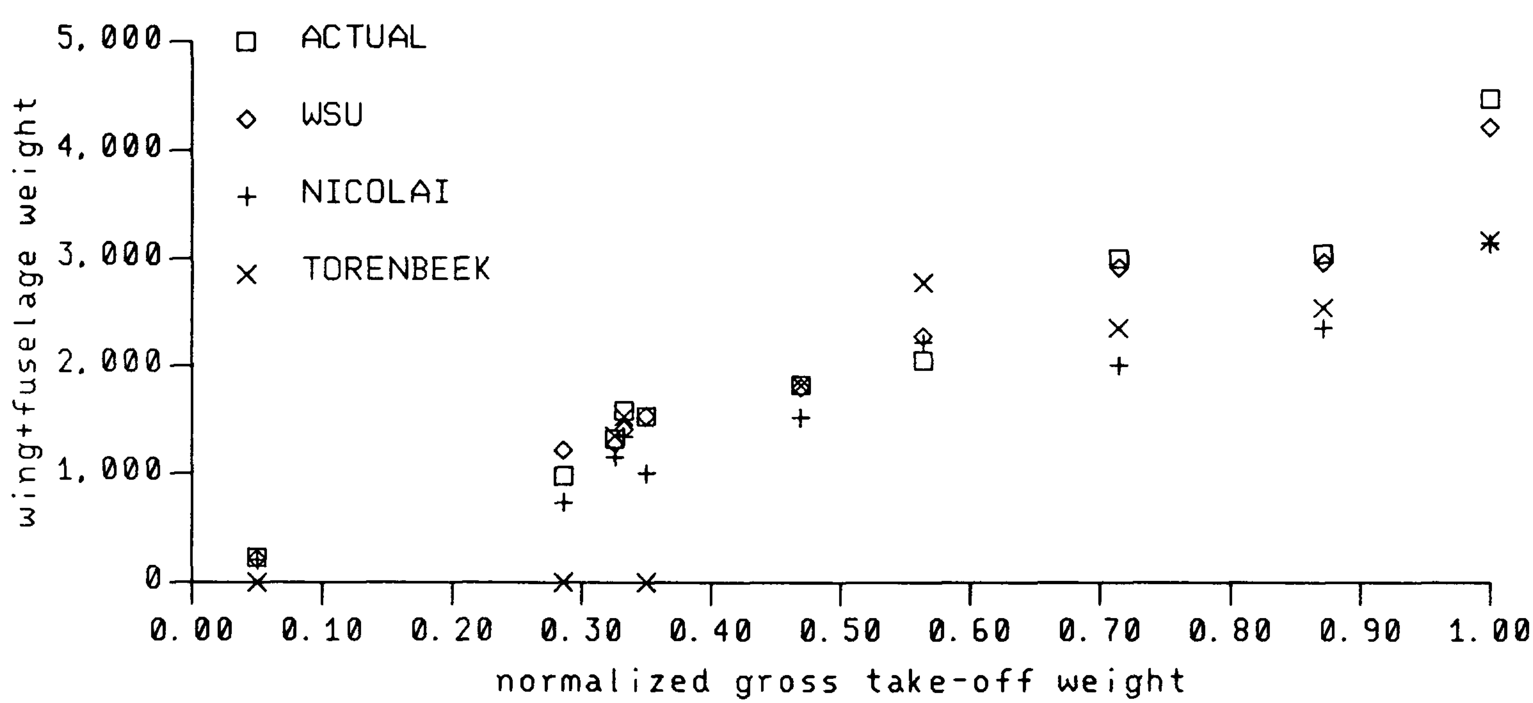
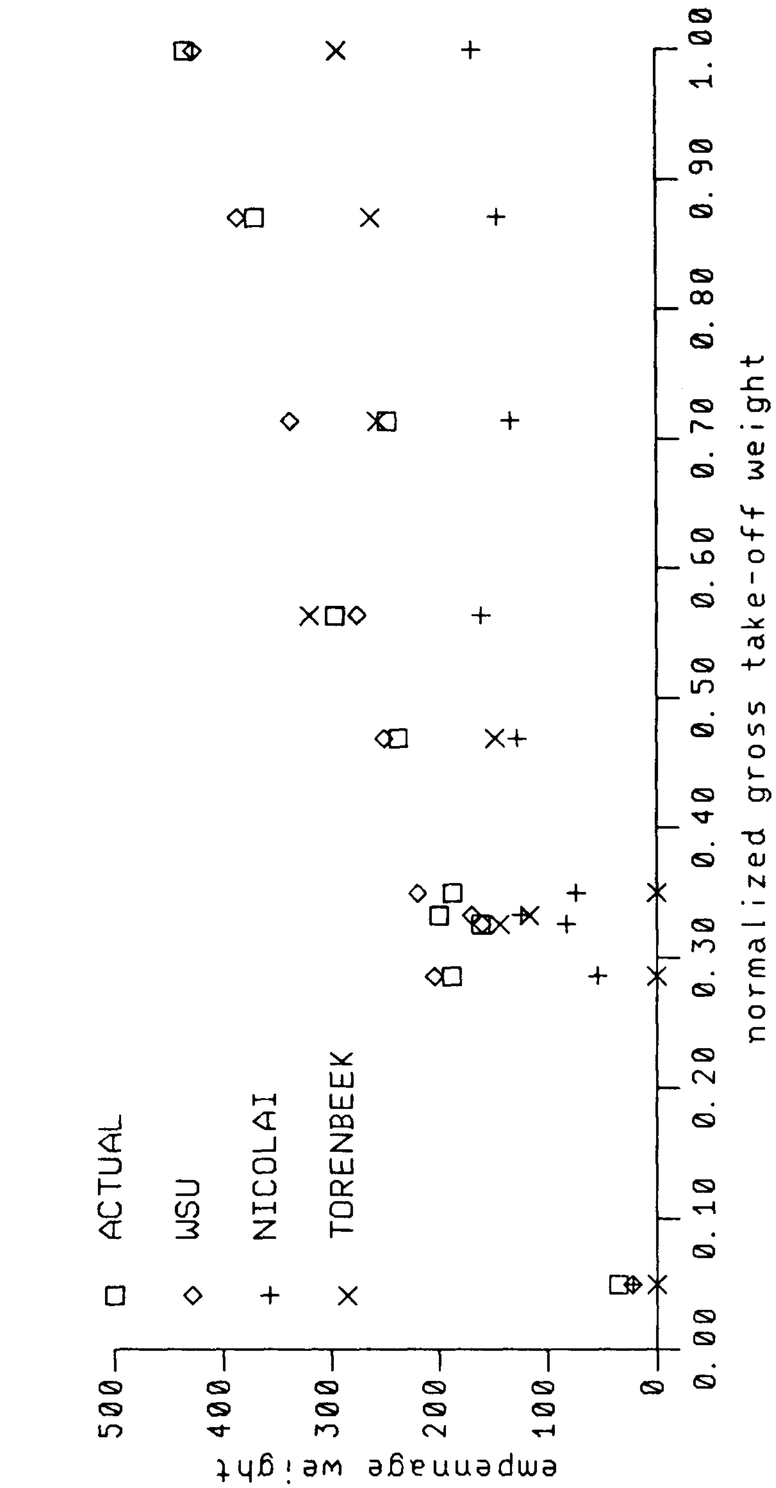


Figure 10. Statistical and Actual Empennage Weight Vs. Normalized Gross Take-off Weight



5. CONCLUSION AND RECOMMENDATIONS

Most of the aircraft companies contacted for participation in this project were reluctant to disseminate any proprietary information such as aircraft component weights breakdowns and their present weight estimation techniques. Hence a limited data base is used to determine the regression coefficents associated with the WSU method for the aircraft weight prediction techniques.

The following conclusions and recommendations are made as a result of this study.

1. The principal advantage of the WSU method is that it can be used for aircraft component weight predictions of both conventional metallic and composite airplanes. Also, the WSU method seems to predict aircraft component weights with a greater accuracy compared to the existing methods as obvious from figures 7 thru 10 for most of the general aviation airplanes considered.
2. The original database for the WSU method is always retained after execution. The user has an option for improved accuracy should he decide to merge 'new data' of general aviation airplanes with the existing data base.
3. The WSU program should be updated periodically with a permanent data base for improved accuracy in the weight prediction of aircraft components when more composite airplanes are produced and certified. Also, more detailed weight analysis methods are needed to reflect i) the 'composite construction methods' of manufacturing and ii) the 'anisotropy' of the strength and stiffness of these new materials [12-15] besides being 'light weight'.

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APPENDICES

APPENDIX - A: WEIGHT ESTIMATION EQUATIONS

BY 'TORENBEEK'

The airplane structure weight, W_{struct} will be assumed to consist of the following components:

5.1 Wing, W_w 5.2 Empennage, W_{emp}

5.3 Fuselage, W_f 5.4 Nacelles, W_n

5.5 Landing gear, W_g Therefore:

$$W_{struct} = W_w + W_{emp} + W_f + W_n + W_g \qquad \text{(A.1)}$$

Equations for structure weight estimation are presented for the following types of airplanes:

1. General Aviation Airplanes
2. Commercial Transport Airplanes

1. Wing Weight Estimation

A.1.1

The following equation applies to light transport airplanes with take-off weights below 12,500 lbs:

$$W_w =$$

$$= 0.00125 W_{TO}(b/\cos\Lambda_{1/2})^{0.75}[1 + \{6.3\cos(\Lambda_{1/2})/b\}^{1/2}]\text{x}$$

$$\text{x}(n_{ult})^{0.55}(bS/t_r W_{TO}\cos\Lambda_{1/2})^{0.30} \qquad \text{(A.1.1)}$$

Definition of new terms:

b = wing span in ft

$\Lambda_{1/2}$ = wing semi-chord sweep angle

t_r = maximum thickness of wing root chord in ft

A.1.2

The following equation applies to transport airplanes with take-off weights above 12,500 lbs:

$W_w =$

$$= 0.0017W_{MZF}(b/\cos\Lambda_{1/2})^{0.75}[1 + \{6.3\cos(\Lambda_{1/2})/b\}^{1/2}]x$$

$$x(n_{ult})^{0.55}(bS/t_r W_{MZF}\cos\Lambda_{1/2})^{0.30} \quad \text{(A.1.2)}$$

Definition of new term:

$$W_{MZF} = \text{maximum zero fuel weight} = W_{TO} - W_F \quad \text{(A.1.3)}$$

Special notes:

1. Equation (A.1.2) includes the weight of normal high lift devices as well as ailerons.
2. For spoilers and speed brakes 2 percent should be added.
3. If the airplane has 2 wing mounted engines reduce the wing weight by 5 percent.
4. If the airplane has 4 wing mounted engines reduce the wing weight by 10 percent.
5. If the landing gear is not mounted under the wing reduce the wing weight by 5 percent.
6. For braced wings reduce the wing weight by 30 percent. The resulting wing weight estimate does include the weight of the strut. The latter is roughly 10 percent of the wing weight.
7. For Fowler flaps add 2 percent to wing weight.

2. Empennage Weight Estimation

Empennage weight, W_{emp} will be expressed as follows:

$$W_{emp} = W_h + W_v + W_c, \quad \text{(A.2)}$$

where: W_h = horizontal tail weight in lbs

W_v = vertical tail weight in lbs

W_c = canard weight in lbs

Equations for empennage weight components are presented in the remainder of this section.

A.2.1

The following equation applies to light transport airplanes with design dive speeds up to 250 kts and with conventional tail configurations:

$$W_{emp} = 0.04\{n_{ult}(S_v + S_h)^2\}^{0.75}, \quad \text{(A.2.1)}$$

l_v = dist. from wing $\bar{c}/4$ to vert. tail $\bar{c}_v/4$ in ft

S_r = rudder area in ft^2

λ_v = vertical tail taper ratio

A.2.2

The following equation applies to transport airplanes and to business jets with design dive speeds above 250 kts.

<u>Horizontal tail:</u>

$$W_h = \qquad \text{(A.2.2)}$$

$$= K_h S_h [3.81\{(S_h)^{0.2} V_D\}/\{1{,}000(\cos\ _{1/2_h})^{1/2}\} - 0.287]$$

where K_h takes on the following values:

K_h = 1.0 for fixed incidence stabilizers

K_h = 1.1 for variable incidence stabilizers

<u>Vertical tail:</u>

$$W_v = \qquad \text{(A.2.3)}$$

$$= K_v S_v [3.81\{(S_v)^{0.2} V_D/1{,}000(\cos\ _{1/2_v})^{1/2}\} - 0.287]$$

where K_v takes on the following values:

K_v = 1.0 for fuselage mounted horizontal tails

for fin mounted horizontal tails:

$$K_v = \{1 + 0.15(S_h z_h / S_v b_v)\} \qquad \text{(A.2.4)}$$

<u>Definition of new terms:</u>

V_D = design dive speed in KEAS

$_{1/2_h}$ horizontal tail semi-chord sweep angle

$_{1/2_v}$ vertical tail semi-chord sweep angle

3. Fuselage Weight Estimation

A.3.1

The following equation applies to transport airplanes and to business jets with design dive speeds above 250 kts.

$$W_f = 0.021 K_f \{(V_D l_h/(w_f + h_f)\}^{1/2} (S_{fgs})^{1.2} \qquad \text{(A.3.1)}$$

The constant K_f takes on the following values:

K_f = 1.08 for a pressurized fuselage

= 1.07 for a main gear attached to the fuselage.

= 1.10 for a cargo airplane with a cargo floor

These effects are multiplicative for airplanes equipped with all of the above.

Definition of new terms:

V_D = design dive speed in KEAS

l_h = distance from wing $\bar{c}/4$ to hor. tail $\bar{c}_h/4$ in ft

S_{fgs} = fuselage gross shell area in ft^2

4. Nacelle Weight Estimation

The nacelle weight is assumed to consist of the following components:

1. For podded engines: the structural weight associated with the engine external ducts and or cowls. Any pylon weight is included.

2. For propeller driven airplanes: the structural weight associated with the engine external ducts and or cowls plus the weight due to the engine mounting trusses.

3. For buried engines: the structural weight associated with special cowling and or ducting provisions.

A.4.1

For single engine propeller driven airplanes with the nacelle in the fuselage nose:

$$W_n = 2.5(P_{TO})^{1/2} \qquad \text{(A.4.1)}$$

This weight includes the entire engine section forward of the firewall.

For multi-engine airplane with piston engines:

$W_n = 0.32 P_{TO}$ for horizontally opposed engines (A.4.2)

$W_n = 0.045(P_{TO})^{5/4}$ for radial engines (A.4.3)

$W_n = 0.14(P_{TO})$ for turboprop engines (A.4.4)

Notes: 1. Since P_{TO} is the total required take-off horsepower, these weight estimates include the weights of **all** nacelles.

2. If the main landing gear retracts into the nacelles, add 0.04 lbs/hp to the nacelle weight

3. If the engine exhausts over the wing, as in the Lockheed Electra, add 0.11 lbs/hp to the nacelle weight.

A.4.2

For turbojet or low bypass ratio turbofan engines:

$$W_n = 0.055T_{TO} \tag{A.4.5}$$

For high bypass ratio turbofan engines:

$$W_n = 0.065T_{TO} \tag{A.4.6}$$

Since T_{TO} is the total required take-off thrust, these equations account for the weight of all nacelles.

5. Landing Gear Weight Estimation

A.5.1

The following equation applies to transport airplanes and to business jets with the main gear mounted on the wing and the nose gear mounted on the fuselage:

$$W_g = K_{g_r}\{A_g + B_g(W_{TO})^{3/4} + C_gW_{TO} + D_g(W_{TO})^{3/2}\} \tag{A.5.1}$$

The factor K_{g_r} takes on the following values:

$K_{g_r} = 1.0$ for low wing airplanes

$K_{g_r} = 1.08$ for high wing airplanes

Table A.1 Constants in Landing Gear Weight Eqn. (A.5.1)

Airplane Type	Gear Type	Gear Comp.	A_g	B_g	C_g	D_g
Jet Trainers and Business Jets	Retr.	Main	33.0	0.04	0.021	0.0
		Nose	12.0	0.06	0.0	0.0
Other civil airplanes	Fixed	Main	20.0	0.10	0.019	0.0
		Nose	25.0	0.0	0.0024	0.0
		Tail	9	0.0	0.0024	0.0
	Retr.	Main	40.0	0.16	0.019	$1.5x10^{-5}$
		Nose	20.0	0.10	0.0	$2.0x10^{-6}$
		Tail	5.0	0.0	0.0031	0.0

APPENDIX B

Program Listing for "Torenbeek" Method
of Aircraft Component Weight Estimation

```
C           THIS PROGRAM IS BASED ON TORENBEEK'S  EQUATIONS FOR WEIGHT
C                PREDICTION FOR CONVENTIONAL METAL AIRCRAFT
C
C                         WRITTEN BY RICHARD D. ROE
C                         BOEING MILITARY AIRPLANE CO.
C                         (316) 526-7336
C                         MODIFIED BY JIM RITTER, FEBR. 1986
C                         THE WICHITA STATE UNIVERSITY
C                         (316) 689-3410
C
C
C                  B          WING SPAN IN FEET
C                  BF         WIDTH OF FUSELAGE IN FEET
C                  BS         STRUCTURAL WING SPAN IN FEET
C                  ESHP       ENGINE SHIP HORSEPOWER
C                  HF         DEPTH OF FUSELAGE IN FEET
C                  LAMDA      WING SWEEP ANGLE IN DEGREES
C                  LAMH       HORIZONTAL TAIL SWEEP ANGLE IN DEGREES
C                  LAMV       VERTICAL TAIL SWEEP ANGLE IN DEGREES
C                  NULT       ULTIMATE LOAD FACTOR
C                  PTO        BLOWER HORSEPOWER PER ENGINE
C                  S          WING AREA IN SQUARE FEET
C                  SH         HORIZONTAL TAIL AREA IN SQUARE FEET
C                  SG         TOTAL SURFACE AREA OF FUSELAGE
C                  SV         VERTICAL TAIL AREA IN SQUARE FEET
C                  TR         TAPER RATIO
C                  TTO        TAKEOFF TORQUE
C                  VD         DESIGN DIVE SPEED
C                  WF         WEIGHT OF FUSELAGE IN LBS
C                  WGROSS     MAXIMUM TAKEOFF WEIGHT
C                  WHOR       WEIGHT OF HORIZONTAL TAIL
C                  WMAIN      WEIGHT OF MAIN LANDING GEAR
C                  WNAC       WEIGHT OF NACELLE IN LBS
C                  WNOSE      WEIGHT OF NOSE LANDING GEAR IN LBS
C                  WSC        WEIGHT OF SURFACE CONTROL GROUP IN LBS
C                  WTAIL      WEIGHT OF TAIL LANDING GEAR IN LBS
C                  WVERT      WEIGHT OF VERTICAL TAIL IN LBS
C                  WW8        WEIGHT OF WING IN LBS
C
C
C
C
C     WING WEIGHT
C
      LOGICAL*1 TITLE(60)
      REAL LAMDA,KW,NULT,LAMH,KH,LAMV,KWF,KUC,KSC,KV,LT,
     &KW6,KW7,KW8,KW9,KF1,KF2,KF3,KF4
      DATA NW6/0/,NW7/0/,NW8/0/,NW9/0/,NF1/0/,NF2/0/,NF3/0/,
     &KW6/1./,KW7/1./,KW8/1./,KW9/1./,KF1/1./,KF2/1./,KF3/1./,KF4/1./
C
```

```
C       IN THIS MODIFICATION, TWO FILES ARE CREATED, AN IN-
C       PUT FILE "NASA4IN DATA" AND AN OUTPUT FILE "NASA4OUT LIST-
C       ING". AT THE FIRST STEP OF INTERACTION, IT IS POSSIBLE TO
C       CHOOSE THE PREVIOUSLY CREATED INPUT SET (NASA4IN) THUS BYPASSING
C       THE FOLLOWING PROCEDURE. THE ADVANTAGE IS IN TIME SAVED IF
C       IT IS DESIRABLE TO VARY ONLY A FEW PARAMETERS BETWEEN RUNS.
C       THE OUTPUT SET (NASA4OUT) IS ALWAYS RECREATED.
C
      LLLL=1
      WRITE(6,398)
  398 FORMAT('     IF INTERACTIVE PROCEDURE NOT REQUIRED, ENTER "2" ',
     */,2X,'AND PROGRAM WILL READ FROM A PREVIOUSLY CREATED INPUT LIST',
     */,2X,'"NASA4IN LISTING". IF INTERACTIVE PROCEDURE IS DESIRED, ',
     */,2X,'ENTER "1" AND NASA4IN LISTING WILL BE CREATED ANEW.')
      READ(5,*)LLLL
      IF(LLLL.EQ.1)GO TO 4
      READ(4,400)TITLE,WGROSS,B,LAMDA,S,NULT,TR,NI,
     *NW6,NW7,NW8,NW9,BH,SH,VD,LAMH,BV,SV,LAMV,N6,SG,HF,BF,LT,
     *NF1,NF2,NF3,N3,NN,N2,KLG,KSC,N5,PTO
C     WRITE(4,400)TITLE,WGROSS,B,LAMDA,S,NULT,TR,NI,
C    *NW6,NW7,NW8,NW9,BH,SH,VD,LAMH,BV,SV,LAMV,N6,SG,HF,BF,LT,
C    *NF1,NF2,NF3,N3,NN,N2,KLG,KSC,N5,PTO
  400 FORMAT(60A1/,6(F12.4/),5(11X,I1/),7(F12.4/),11X,I1/,
     *4(F12.4/),7(11X,I1/),F12.4/,11X,I1/,F12.4)
      KW=.0017
      IF(WGROSS.GE.12500.)KW=.00125
      IF(NI.EQ.0)F1=1.0
      IF(NI.EQ.1)F1=.95
      IF(NI.EQ.2)F1=.90
      KV=1.0
      IF(N6.EQ.2)KV=(SH*BH/SV/BV)*.15+1.0
      KUC=1.0
      IF(N3.NE.1)KUC=1.08
      IF(NN.EQ.1)WMAIN=KUC*(33.+.04*WGROSS**.75+.021*WGROSS)
      IF(NN.EQ.1)WNOSE=KUC*(12.+.06*WGROSS**.75)
      IF(NN.EQ.1)WTAIL=0.
      IF(N2.EQ.1)WMAIN=KUC*(20.+.1*WGROSS**.75+.019*WGROSS)
      IF(N2.EQ.1)WNOSE=KUC*(25.+.0024*WGROSS)
      IF(N2.EQ.1)WTAIL=KUC*(9.+.0024*WGROSS)
      IF(N2.EQ.2)WMAIN=KUC*(40.+.16*WGROSS**.75+.019*WGROSS+1.5E-5
     **WGROSS**1.5)
      IF(N2.EQ.2)WNOSE=KUC*(20.+.1*WGROSS**.75+2.E-6*WGROSS**1.5)
      IF(N2.EQ.2)WTAIL=KUC*(5.+.0031*WGROSS)
      IF(N5.EQ.1)WNAC=2.5*(PTO)**.5
      IF(N5.EQ.2)WNAC=.045*PTO**1.25
      IF(N5.EQ.3)WNAC=.14*PTO
      IF(N5.EQ.4)WNAC=.055*PTO
      IF(N5.EQ.5)WNAC=.065*PTO
      IF(NW6.EQ.1)KW6 = 1.02
      IF(NW7.EQ.1)KW7 = .95
      IF(NW8.EQ.1)KW8 = .70
      IF(NW9.EQ.1)KW9 = 1.02
      IF(NF1.EQ.1)KF1 = 1.08
      IF(NF2.EQ.1)KF2 = 1.07
      IF(NF3.EQ.1)KF3 = 1.10
```

```
      IF(NW7.EQ.1)KF4 = 1.Ø4
      IF(LLLL.EQ.2)GO TO 3Ø1
    4 WRITE(6,135)
  135 FORMAT(' INPUT TITLE FOR OUTPUT UP TO 6Ø CHARACTERS')
      READ(5,5Ø) (TITLE(KKK),KKK=1,6Ø)
      WRITE(4,51) (TITLE(KKK),KKK=1,6Ø)
   5Ø FORMAT(6ØA1)
   51 FORMAT(2X,6ØA1)
      WRITE(6,1ØØ)
  1ØØ FORMAT(' INPUT MAXIMUM TAKEOFF WEIGHT')
      READ(5,*)WGROSS
      WRITE(4,*)WGROSS
      WRITE(6,1Ø1)
  1Ø1 FORMAT(' INPUT WING SPAN IN FEET')
      READ(5,*)B
      WRITE(4,*)B
      WRITE(6,1Ø2)
 1Ø2  FORMAT(' INPUT WING SWEEP ANGLE IN DEGREES')
      READ(5,*)LAMDA
      WRITE(4,*)LAMDA
      WRITE(6,1Ø3)
 1Ø3  FORMAT(' INPUT WING AREA IN SQUARE FEET')
      READ(5,*)S
      WRITE(4,*)S
      IF(WGROSS.GE.125ØØ.) GO TO 2ØØ
      KW=.ØØ17
      GO TO 2Ø1
 2ØØ  KW=.ØØ125
 2Ø1  CONTINUE
      WRITE(6,114)
 114  FORMAT(' INPUT THE ULTIMATE LOAD FACTOR')
      READ(5,*)NULT
      WRITE(4,*)NULT
      WRITE(6,123)
 123  FORMAT(' INPUT TAPER RATIO')
      READ(5,*)TR
      WRITE(4,*)TR
      WRITE(6,151)
 151  FORMAT(' INPUT NUMBER OF ENGINES ON EACH WING Ø,1,2')
      READ(5,*)NI
      WRITE(4,*)NI
      NP=NI+1
      GOTO(152,153,154),NP
 152  F1=1.Ø
      GO TO 155
 153  F1=.95
      GO TO 155
 154  F1=.9Ø
 155  CONTINUE
      WRITE(6,5Ø6)
  5Ø6 FORMAT(' IF SPOILERS OR SPEEDBRAKES ARE INCLUDED, ENTER 1.'/,
     &' OTHERWISE, ENTER Ø.')
      READ(5,*)NW6
  6Ø6 IF(NW6.EQ.1)KW6=1.Ø2
      WRITE(4,*)NW6
```

```
      WRITE(6,507)
  507 FORMAT(' IF MAIN LANDING GEAR IS FUSELAGE-MOUNTED, ENTER 1.'/,
     &' OTHERWISE, ENTER 0.')
      READ(5,*)NW7
  607 IF(NW7.EQ.1)KW7=.95
      WRITE(4,*)NW7
      WRITE(6,508)
  508 FORMAT(' THE WING IS STRUT-BRACED, ENTER 1.  OTHERWISE ENTER 0.'/,
     &' (WING WEIGHT WILL NOT INCLUDE STRUT WEIGHT.)')
      READ(5,*)NW8
  608 IF(NW8.EQ.1)KW8=.70
      WRITE(4,*)NW8
      WRITE(6,509)
  509 FORMAT(' IF FOWLER FLAPS ARE USED, ENTER 1.'/,
     &' OTHERWISE, ENTER 0.')
      READ(5,*)NW9
  609 IF(NW9.EQ.1)KW9=1.02
      WRITE(4,*)NW9
 301  BREF=6.25
      BS=B/COS(.017453*LAMDA/2)
      AA=(BREF/BS)**.5+1
      AB=(BS*S/WGROSS/TR)**.3
      WW8=(WGROSS*KW*BS**.75*AA*NULT**.55*AB)*F1*KW6*KW7*KW8*KW9
      IF(LLLL.EQ.2)GO TO 302
C-----------------------------------------------------------------------
C
C     TAIL GROUP
C
      WRITE(6,104)
 104  FORMAT(' INPUT HORIZONTAL TAIL SPAN IN FEET')
      READ(5,*)BH
      WRITE(4,*)BH
      WRITE(6,105)
 105  FORMAT(' INPUT HORIZONTAL TAIL AREA IN SQUARE FEET')
      READ(5,*)SH
      WRITE(4,*)SH
      WRITE(6,106)
 106  FORMAT(' INPUT DESIGN DIVE SPEED IN KNOTS')
      READ(5,*)VD
      WRITE(4,*)VD
      WRITE(6,107)
 107  FORMAT(' INPUT HORIZONTAL TAIL SWEEP ANGLE IN DEGREES')
      READ(5,*)LAMH
      WRITE(4,*)LAMH
  302 KH=1.0
C
C HORIZONTAL TAIL WEIGHT
C
      AC=SH**.2*VD/(COS(LAMH*.017453))**.5/1000.
      WHOR=SH*KH*(3.5*AC-.2)
      IF(LLLL.EQ.2)GO TO 303
      WRITE(6,108)
 108  FORMAT(' INPUT VERTICAL TAIL SPAN')
      READ(5,*)BV
      WRITE(4,*)BV
```

```
      WRITE(6,109)
 109  FORMAT(' INPUT VERTICAL TAIL AREA IN SQUARE FEET')
      READ(5,*)SV
      WRITE(4,*)SV
      WRITE(6,110)
 110  FORMAT(' INPUT VERTICAL TAIL SWEEP ANGLE IN DEGREES')
      READ(5,*)LAMV
      WRITE(4,*)LAMV
      WRITE(6,133)
 133  FORMAT(' INPUT',/,20X,'1 FOR FUSELAGE MOUNTED TAILPLANE'
     $ ,/,20X,'2 FOR FIN MOUNTED TAILPLANE')
      READ(5,*)N6
      WRITE(4,*)N6
      GOTO(213,214),N6
 213  KV=1.0
      GO TO 215
 214  KV=(SH*BH/SV/BV)*.15+1.0
 215  CONTINUE
C
C VERTICAL TAIL WEIGHT
C
  303 AD=SV**.2*VD/(COS(LAMV*.017453))**.5/1000.
      WVERT=SV*KV*(3.5*AD-.2)
      IF(LLLL.EQ.2)GO TO 304
C-----------------------------------------------------------------------
C
C     BODY GROUP
C
      WRITE(6,111)
 111  FORMAT(' INPUT TOTAL SHELL AREA IN SQUARE FEET')
      READ(5,*)SG
      WRITE(4,*)SG
      WRITE(6,112)
 112  FORMAT(' INPUT DEPTH OF FUSELAGE IN FEET')
      READ(5,*)HF
      WRITE(4,*)HF
      WRITE(6,113)
 113  FORMAT(' INPUT WIDTH OF FUSELAGE IN FEET')
      READ(5,*)BF
      WRITE(4,*)BF
      WRITE(6,134)
 134  FORMAT(' INPUT WING 1/4 MAC TO TAIL 1/4 MAC IN FEET')
      READ(5,*)LT
      WRITE(4,*)LT
      WRITE(6,701)
  701 FORMAT(' IF THE FUSELAGE IS PRESSURIZED, ENTER 1.'/,
     &' OTHERWISE, ENTER 0.')
      READ(5,*)NF1
  801 IF(NF1.EQ.1)KF1=1.08
      WRITE(4,*)NF1
      WRITE(6,702)
  702 FORMAT(' IF ENGINES ARE REAR-MOUNTED ON THE FUSELAGE, ENTER 1.'/,
     &' OTHERWISE, ENTER 0.')
      READ(5,*)NF2
  802 IF(NF2.EQ.1)KF2=1.04
```

```
      WRITE(4,*)NF2
      WRITE(6,703)
  703 FORMAT(' IS THE AIRPLANE A CARGO AIRPLANE WITH A CARGO FLOOR?',
     &' ENTER 1.'/,' OTHERWISE, ENTER Ø.')
      READ(5,*)NF3
  8Ø3 IF(NF3.EQ.1)KF3=1.1Ø
      WRITE(4,*)NF3
      IF(NW7.EQ.1)KF4=1.Ø7
C
C  (NW7 REPRESENTS THE STATE OF MAIN GEAR ATTACHMENT.  THE QUESTION HAS
C  BEEN ASKED IN THE WING SECTION, THE INFORMATION IS AGAIN USED HERE.)
C
  3Ø4 KWF=.Ø21
      WF=(KWF*(VD*LT/(BF+HF))**.5*SG**1.2)*KF1*KF2*KF3*KF4
      IF(LLLL.EQ.2)GO TO 3Ø5
C-----------------------------------------------------------------------
C
C     ALIGHTING GEAR
C
      WRITE(6,117)
 117  FORMAT(' INPUT',/,20X,'1 FOR LOW WING AIRCRAFT',/,2ØX,'2 FOR ',
     &'ALL OTHERS')
      READ(5,*)N3
      WRITE(4,*)N3
      IF(N3.NE.1) GO TO 2Ø5
      KUC=1.
      GO TO 2Ø6
 2Ø5  KUC=1.Ø8
 2Ø6  CONTINUE
      WRITE(6,115)
 115  FORMAT(' INPUT',/,2ØX,'1 FOR JET TRAINERS AND EXECUTIVE ',
     $'AIRCRAFT',/,2ØX,'2 FOR ALL OTHER CIVIL AIRCRAFT',/,1ØX,
     $'(A CHOICE OF "1" WILL ASSUME A RETRACTABLE, NOSE GEAR AIRPLANE)')
      READ(5,*)NN
      WRITE(4,*)NN
      IF(NN.NE.1) GO TO 2Ø2
      WMAIN=KUC*(33.+.Ø4*WGROSS**.75+.Ø21*WGROSS)
      WNOSE=KUC*(12.+.Ø6*WGROSS**.75)
      N2=Ø
      WRITE(4,*)N2
      KLG = 1
      GO TO 5Ø5
 2Ø2  WRITE(6,116)
 116  FORMAT(' INPUT',/,2ØX,'1 FOR FIXED LANDING GEAR',/,2ØX,'2 FOR',
     $' RETRACTABLE LANDING GEAR')
      READ(5,*)N2
      WRITE(4,*)N2
      IF(N2.NE.1) GO TO 2Ø4
      WMAIN=KUC*(2Ø.+.1*WGROSS**.75+.Ø19*WGROSS)
      WNOSE=KUC*(25.+.ØØ24*WGROSS)
      WTAIL=KUC*(9.+.ØØ24*WGROSS)
      GO TO 2Ø3
 2Ø4  WMAIN=KUC*(4Ø.+.16*WGROSS**.75+.Ø19*WGROSS+1.5E-5*WGROSS**1.5)
      WNOSE=KUC*(2Ø.+.1*WGROSS**.75+2.E-6*WGROSS**1.5)
      WTAIL=KUC*(5.+.ØØ31*WGROSS)
```

```
 203  CONTINUE
      WRITE(6,217)
 217  FORMAT(' INPUT',/,20X,'1 FOR TRICYCLE GEAR AIRCRAFT'/,
     $20X,'2 FOR CONVENTIONAL (TAILWHEEL) AIRCRAFT')
      READ(5,*)KLG
 505  WRITE(4,*)KLG
C-----------------------------------------------------------------------
C
C     SURFACE CONTROL GROUP
C
      WRITE(6,118)
 118  FORMAT(' INPUT KSC',/,20X,'KSC=.23 FOR LIGHT AIRCRAFT WITHOUT ',
     $'DUPLICATE CONTROLS',/,20X,'KSC=.44 FOR TRANSPORT AIRCRAFT AND ',
     $'TRAINERS, MANUAL CONTROLS',/,20X,'KSC=.64 FOR TRANSPORT ',
     $'AIRCRAFT WITH POWERED CONTROLS AND',/,28X,'TRAILING EDGE ',
     $'HIGH LIFT DEVICES')
      READ(5,*)KSC
      WRITE(4,*)KSC
  305 WSC=KSC*WGROSS**.6667
      IF(LLLL.EQ.2)GO TO 306
C-----------------------------------------------------------------------
C
C     NACELLE GROUP
C
      WRITE(6,119)
 119  FORMAT(' INPUT ',/,20X,'1 FOR LIGHT AIRCRAFT',/,20X,'2 FOR ',
     $'MULTI ENGINE RECIPROCATING',/,20X,'3 FOR TURBOPROP AIRCRAFT',
     $/,20X,'4 FOR TURBOJET OF TURBOFAN AIRCRAFT',/,20X,'5 FOR ',
     $'HIGH BYPASS TURBOFANS')
      READ(5,*)N5
      WRITE(4,*)N5
      GOTO(207,208,209,210,211),N5
C
C LIGHT AIRCRAFT
C
 207  WRITE(6,120)
 120  FORMAT(' INPUT BHP PER ENGINE')
      READ(5,*)PTO
      WRITE(4,*)PTO
      WNAC=2.5*(PTO)**.5
      GO TO 212
C
C MULTIPLE RECIPROCATING ENGINES AIRCRAFT
C
 208  WRITE(6,120)
      READ(5,*)PTO
      WRITE(4,*)PTO
      WNAC=.045*PTO**1.25
      GO TO 212
C
C TURBOPROP
C
 209  WRITE(6,121)
 121  FORMAT(' INPUT ENGINE SHIP HORSEPOWER AT TAKEOFF')
      READ(5,*)ESHP
```

```
      WRITE(4,*)ESHP
      WNAC=.14*ESHP
      GO TO 212
C
C TURBOJET OR TURBOFAN
C
 210  WRITE(6,122)
 122  FORMAT(' INPUT TAKEOFF TORQUE')
      READ(5,*)TTO
      WRITE(4,*)TTO
      WNAC=.055*TTO
      GO TO 212
C
C HIGH BYPASS TURBOFANS
 211  WRITE(6,122)
      READ(5,*)TTO
      WRITE(4,*)TTO
      WNAC=.065*TTO
 212  CONTINUE
 306  WRITE(6,136) TITLE
      WRITE(7,136) TITLE
 136  FORMAT(5X,(40A1))
      WRITE(6,124) WW8
      WRITE(7,124) WW8
 124  FORMAT(' THE WING WEIGHT=',F15.4)
      WRITE(6,125) WHOR
      WRITE(7,125) WHOR
 125  FORMAT( /,' THE HORIZONTAL TAIL WEIGHT =',F15.4)
      WRITE(6,126)WVERT
      WRITE(7,126)WVERT
 126  FORMAT( /,' THE VERTICAL TAIL WEIGHT=',F15.4)
      WRITE(6,127)WF
      WRITE(7,127)WF
 127  FORMAT( /,' THE FUSELAGE WEIGHT =',F15.4)
      WRITE(6,128)WMAIN
      WRITE(7,128)WMAIN
 128  FORMAT( /,' THE MAIN LANDING GEAR WEIGHT=',F15.4)
      IF(KLG.EQ.1)WRITE(6,129)WNOSE
      IF(KLG.EQ.1)WRITE(7,129)WNOSE
 129  FORMAT( /,' THE NOSE LANDING GEAR WEIGHT=',F15.4)
      IF(KLG.EQ.2)WRITE(6,130)WTAIL
      IF(KLG.EQ.2)WRITE(7,130)WTAIL
 130  FORMAT( /,' THE TAIL LANDING GEAR WEIGHT=',F15.4)
      WRITE(6,131)WSC
      WRITE(7,131)WSC
 131  FORMAT( /,' THE SURFACE CONTROL WEIGHT=',F15.4)
      WRITE(6,132)WNAC
      WRITE(7,132)WNAC
 132  FORMAT( /,' THE NACELLE WEIGHT=',F15.4)
      STOP
      END
```

APPENDIX C. WSU Weight Estimation Programs.

GENREG FORTRAN

```
C        ******************************************************
C        *   THE FOLLOWING PROGRAM IS A GENERAL REGRESSION    *
C        *   ANALYSIS PROGRAM.  THE 'Y' VALUES (ACTUAL WTS.   *
C        *   ON WHICH THE REGRESSION IS BASED) ARE READ IN    *
C        *   BELOW.  THE 'X' AND 'Z' VALUES ARE DETERMINED    *
C        *   BY SUBROUTINES AT THE END OF THE LISTING. THE    *
C        *   REGRESSION IS DONE TWICE, ONCE IN LINEAR FOR-    *
C        *   MAT AND ONCE IN NATURAL LOG FORMAT FOR THE       *
C        *   PURPOSE OF COMPRESSING THE DATA.  THE MATRICES   *
C        *   ARE DETERMINED AND PRINTED, AND THEN SOLVED      *
C        *   SIMULTANEOUSLY BY IBM LIBRARY SUBROUTINE TO      *
C        *   TO DETERMINE THE COEFFICIENTS.  FINALLY, THE     *
C        *   ORIGINAL 'X' AND 'Z' VALUES ARE USED WITH THE    *
C        *   NEW COEFFICIENTS TO COMPUTE VALUES OF 'Y' AND    *
C        *   THESE NEW VALUES ARE COMPARED WITH THE ACTUAL    *
C        *   VALUES, NORMALIZED, AND STATISTICALLY EVALUATED. *
C        *                                                    *
C        *   THIS PROGRAM IS EXECUTED BY USE OF EXEC "GENREX". *
C        *                                                    *
C        ******************************************************
C
      REAL N(20),LF(20),LAMDA(20),LAMDAT(20),LAMDAV(20),LNX,LNY,LNZ
      DIMENSION A1(20),A2(20),GW(20),WW(20),RHO(20),FW(20),FS(20),
     &BW(20),SW(20),CR(20),CT(20),TR(20),TT(20),X(20),Y(20),Z(20),
     &RHOP(20),RHOR(20),ER(20),FF(20),BF(20),FWTACT(20),RHOT(20)
      DIMENSION BT(20),ST(20),CRT(20),CTT(20),TRT(20),TTT(20),YCAV(20),
     &HWTACT(20),RHOV(20),FV(20),FVS(20),BV(20),SV(20),CRV(20),CTV(20),
     &TRV(20),TTV(20),VWTACT(20),A(3,3),B(3),YCAL(20),YTOT(20),YCCC(20),
     &YNORM(20),FT(20),FTS(20),YYCAL(20),YY(20),WGS(20),WGS2(20),AGW(20)
C
C        ***************************************************
C        * THE FOLLOWING DATA IS READ FROM FILE #4, CALLED *
C        * 'GENDAT DATA'.  IT REPRESENTS WEIGHTS, MATERIAL *
C        * PROPERTIES, AND GEOMETRIES OF THE AIRPLANES IN- *
C        * TENDED FOR THE REGRESSION.                      *
C        ***************************************************
C
      READ(4,1)NCASE
      READ(4,2)(A1(I),A2(I),GW(I),WW(I),FWTACT(I),HWTACT(I),
     &VWTACT(I),N(I),I=1,NCASE)
      READ(4,3)(RHO(I),RHOP(I),RHOR(I),ER(I),RHOT(I),RHOV(I),
     &I=1,NCASE)
      READ(4,3)(FW(I),FS(I),FT(I),FTS(I),FV(I),FVS(I),I=1,NCASE)
      READ(4,3)(BW(I),SW(I),BT(I),ST(I),BV(I),SV(I),I=1,NCASE)
      READ(4,3)(BF(I),LF(I),FF(I),LAMDA(I),LAMDAT(I),LAMDAV(I),
     &I=1,NCASE)
      READ(4,3)(CR(I),CT(I),CRT(I),CTT(I),CRV(I),CTV(I),I=1,NCASE)
      READ(4,3)(TR(I),TT(I),TRT(I),TTT(I),TRV(I),TTV(I),I=1,NCASE)
      DO 6 I = 1,NCASE
    6 ER(I)=ER(I) * 1000000.
```

```
C
C        ********************************************************
C        *  THE FOLLOWING SUBROUTINES ARE SEQUENTIALLY          *
C        *  CALLED FOR THE PURPOSE OF DETERMINING 'X' AND       *
C        *  'Z' FROM THE APPROPRIATE DATA.  THE SUBROUTINE      *
C        *  EQUATIONS ARE IN NON-DIMENSIONAL FORM;  'X', 'Y',   *
C        *  AND 'Z' HAVE BEEN MADE NON-DIMENSIONAL BY DIVIDING  *
C        *  ACTUAL WEIGHT OF EACH COMPONENT BY THE AIRPLANE     *
C        *  GROSS WEIGHT.                                       *
C        ********************************************************
C
      DO 108 L = 1,4
      IF(L.EQ.1)CALL WING(NCASE,GW,WW,N,RHO,FW,FS,LAMDA,BW,SW,CR,CT,
     &TR,TT,X,Y,Z)
      IF(L.EQ.2)CALL FUSLGE(NCASE,GW,WW,N,RHOP,RHOR,ER,FF,LF,BF,
     &FWTACT,X,Y,Z)
      IF(L.EQ.3)CALL HZTAIL(NCASE,GW,WW,N,RHOT,FT,FTS,LAMDAT,BT,SW,
     &ST,CRT,CTT,TRT,TTT,HWTACT,X,Y,Z)
      IF(L.EQ.4)CALL VTAIL(NCASE,GW,WW,N,RHOV,FV,FVS,LAMDAV,BV,SW,
     &SV,CRV,CTV,TRV,TTV,VWTACT,X,Y,Z)
C     WRITE(7,23)(X(I),Y(I),Z(I),I=1,NCASE)
   23 FORMAT(3F20.8)
C
C           ******************
C           * LINEAR VERSION *
C           ******************
C
C      INITIALIZE SUMMATION ROUTINE
C
      SMX = X(1)
      SMX2 = X(1) ** 2.
      SMY = Y(1)
      SMZ = Z(1)
      SMZ2 = Z(1) ** 2.
      SMXZ = X(1) * Z(1)
      SMXY = X(1) * Y(1)
      SMZY = Z(1) * Y(1)
C
C    SUMMATION:
C
      DO 30 I = 2,NCASE
      SMX = SMX + X(I)
      SMX2 = SMX2 + X(I) ** 2.
      SMY = SMY + Y(I)
      SMZ = SMZ + Z(I)
      SMZ2 = SMZ2 + Z(I) ** 2.
      SMXZ = SMXZ + X(I) * Z(I)
      SMXY = SMXY + X(I) * Y(I)
      SMZY = SMZY + Z(I) * Y(I)
      X2 = X(I) ** 2.
      Z2 = Z(I) ** 2.
      XZ = X(I) * Z(I)
      XY = X(I) * Y(I)
      ZY = Z(I) * Y(I)
   30 CONTINUE
```

```
C
C    DEFINE 'A' AND 'B' MATRICES
C
      A(1,1) = FLOAT(NCASE)
      A(1,2) = SMX
      A(1,3) = SMZ
      A(2,1) = SMX
      A(2,2) = SMX2
      A(2,3) = SMXZ
      A(3,1) = SMZ
      A(3,3) = SMZ2
      A(3,2) = SMXZ
      B(1) = SMY
      B(2) = SMXY
      B(3) = SMZY
      IF(L.EQ.1)WRITE(7,96)
      IF(L.EQ.2)WRITE(7,97)
      IF(L.EQ.3)WRITE(7,98)
      IF(L.EQ.4)WRITE(7,99)
      WRITE(7,100)(A(1,J),J=1,3),B(1),(A(2,J),J=1,3),B(2),
     &(A(3,J),J=1,3),B(3)
      CALL SIMQ(A,B,3,0)
C
C      A IS DESTROYED, B IS REPLACED BY X  (A * X = B)
C
      WRITE(7,200)B
C
C   **************************************************************
C   *  THE FOLLOWING LOOPS DETERMINE AND INITIALIZE WING WEIGHT  *
C   *  PER GROSS WEIGHT FACTORS FOR EACH CASE, LINEAR AND LOGR.  *
C   **************************************************************
C
      WGSI = WW(1) / GW(1)
      DO 17 I = 2,NCASE
   17 WGSI = WGSI + WW(I) / GW(I)
      WGSI = WGSI / (FLOAT(NCASE))
      IF(L.EQ.1)WRITE(9,110)NCASE,WGSI
  110 FORMAT(2X,'ARPLNE   WGS(I)    YCAL(WGS)  ACT. WW/GW ',
     &'YCAL(ACT. WW/GW) YCAL(AVG)'//,12X,'(AVERAGE WW/GW FOR',I3,
     &' AIRPLANES IS',F7.4,' )'///)
      DO 18 I = 1,NCASE
      WGS(I) = WGSI
      WGS2(I) = WGSI
   18 AGW(I) = WW(I) / GW(I)
      EPS = .0001
      IF(L.NE.1)GO TO 41
      DO41I = 1,NCASE
   38 CALL WINC(NCASE,WGS,N,RHO,FW,FS,LAMDA,BW,SW,CR,CT,
     &TR,TT,X,Z)
      YCAL(I) = B(1) + B(2) * X(I) + B(3) * Z(I)
      IF(YCAL(I).GT.WGS(I))WGS(I) = WGS(I) + (YCAL(I)-WGS(I))/2.
      IF(YCAL(I).LT.WGS(I))WGS(I) = WGS(I) - (WGS(I)-YCAL(I))/2.
   39 IF(ABS(WGS(I)-YCAL(I)).GT.EPS)GO TO 38
      CALL WINC(NCASE,WGS2,N,RHO,FW,FS,LAMDA,BW,SW,CR,CT,TR,TT,X,Z)
      YCCC(I) = B(1) + B(2) * X(I) + B(3) * Z(I)
```

```
      CALL WINC(NCASE,AGW,N,RHO,FW,FS,LAMDA,BW,SW,CR,CT,TR,TT,X,Z)
      YCAV(I) = B(1) + B(2) * X(I) + B(3) * Z(I)
      WRITE(9,111)I,WGS(I),YCAL(I),AGW(I),YCAV(I),YCCC(I)
  111 FORMAT(2X,I3,5F12.4/)
   41 CONTINUE
      IF(L.EQ.1)GO TO 44
      DO44I = 1,NCASE
      IF(L.EQ.2)CALL FUSLGC(NCASE,WGS,N,RHOP,RHOR,ER,FF,LF,BF,
     &X,Z)
      IF(L.EQ.3)CALL HZTAIC(NCASE,WGS,N,RHOT,FT,FTS,LAMDAT,BT,SW,
     &ST,CRT,CTT,TRT,TTT,X,Z)
      IF(L.EQ.4)CALL VTAIC(NCASE,WGS,N,RHOV,FV,FVS,LAMDAV,BV,SW,
     &SV,CRV,CTV,TRV,TTV,X,Z)
      YCAL(I) = B(1) + B(2) * X(I) + B(3) * Z(I)
   44 CONTINUE
      WRITE(7,300)
      DO 71 I = 1,NCASE
      YYCAL(I) = YCAL(I) * GW(I)
      YY(I) = Y(I) * GW(I)
   71 YNORM(I) = YYCAL(I) / YY(I)
      YMEAN = YNORM(1)
      DO 73 I = 2,NCASE
   73 YMEAN = YMEAN + YNORM(I)
      YMEAN = YMEAN / NCASE
      DELPR = (YMEAN - 1.) * 100.
      S2 = (YNORM(1) - 1.) ** 2
      DO 81 I = 2,NCASE
   81 S2 = S2 + (YNORM(I) - 1.) ** 2
      S2 = S2 / (NCASE - 1)
      SD = SQRT(S2) * 100.
      WRITE(7,400)(A1(I),A2(I),YY(I),YYCAL(I),YNORM(I),I=1,NCASE)
      WRITE(7,404)YMEAN,DELPR,S2,SD
C
C
C                ***********************
C                * LOGARYTHMIC VERSION *
C                ***********************
C
C      INITIALIZE SUMMATION ROUTINE
C
      SMX = ALOG(X(1))
      SMY = ALOG(Y(1))
      SMZ = ALOG(Z(1))
      SMX2 = SMX * SMX
      SMZ2 = SMZ * SMZ
      SMXZ = SMX * SMZ
      SMXY = SMX * SMY
      SMZY = SMZ * SMY
C
C    SUMMATION:
C
      DO 31 I = 2,NCASE
      LNX = ALOG(X(I))
      LNY = ALOG(Y(I))
      LNZ = ALOG(Z(I))
```

```
      X2 = LNX * LNX
      Z2 = LNZ * LNZ
      XZ = LNX * LNZ
      XY = LNX * LNY
      ZY = LNZ * LNY
      SMX = SMX + LNX
      SMY = SMY + LNY
      SMZ = SMZ + LNZ
      SMX2 = SMX2 + X2
      SMZ2 = SMZ2 + Z2
      SMXZ = SMXZ + XZ
      SMXY = SMXY + XY
      SMZY = SMZY + ZY
   31 CONTINUE
C
C     DEFINE 'A' AND 'B' MATRICES
C
      A(1,1) = FLOAT(NCASE)
      A(1,2) = SMX
      A(2,1) = SMX
      A(2,2) = SMX2
      B(1) = SMY
      A(1,3) = SMZ
      A(3,1) = SMZ
      A(3,3) = SMZ2
      A(2,3) = SMXZ
      A(3,2) = SMXZ
      B(2) = SMXY
      B(3) = SMZY
C     IF(L.EQ.1)WRITE(6,196)
      IF(L.EQ.1)WRITE(7,196)
C     IF(L.EQ.2)WRITE(6,197)
      IF(L.EQ.2)WRITE(7,197)
C     IF(L.EQ.3)WRITE(6,198)
      IF(L.EQ.3)WRITE(7,198)
C     IF(L.EQ.4)WRITE(6,199)
      IF(L.EQ.4)WRITE(7,199)
      WRITE(7,101)(A(1,J),J=1,3),B(1),(A(2,J),J=1,3),B(2),
     *(A(3,J),J=1,3),B(3)
      CALL SIMQ(A,B,3,0)
C
C      A IS DESTROYED, B IS REPLACED BY X  (A * X = B)
C
      B(1) = EXP(B(1))
      WRITE(7,201)B
      IF(L.NE.1)GO TO 51
      DO51I = 1,NCASE
   48 CALL WINC(NCASE,WGS2,N,RHO,FW,FS,LAMDA,BW,SW,CR,CT,
     &TR,TT,X,Z)
      YCAL(I) = B(1)  *  X(I) ** B(2) *  Z(I) ** B(3)
      IF(YCAL(I).GT.WGS2(I))WGS2(I) = WGS2(I) + (YCAL(I)-WGS2(I))/2.
      IF(YCAL(I).LT.WGS2(I))WGS2(I) = WGS2(I) - (WGS2(I)-YCAL(I))/2.
   49 IF(ABS(WGS2(I)-YCAL(I)).GT.EPS)GO TO 48
   51 CONTINUE
      IF(L.EQ.1)GO TO 54
```

```
      DO54I = 1,NCASE
      IF(L.EQ.2)CALL FUSLGC(NCASE,WGS2,N,RHOP,RHOR,ER,FF,LF,BF,
     &X,Z)
      IF(L.EQ.3)CALL HZTAIC(NCASE,WGS2,N,RHOT,FT,FTS,LAMDAT,BT,SW,
     &ST,CRT,CTT,TRT,TTT,X,Z)
      IF(L.EQ.4)CALL VTAIC(NCASE,WGS2,N,RHOV,FV,FVS,LAMDAV,BV,SW,
     &SV,CRV,CTV,TRV,TTV,X,Z)
      YCAL(I) = B(1)  *  X(I) ** B(2) *  Z(I) ** B(3)
   54 CONTINUE
      WRITE(7,300)
      DO 75 I = 1,NCASE
      YCAL(I) = YCAL(I) * GW(I)
      Y(I) = Y(I) * GW(I)
   75 YNORM(I) = YCAL(I) / Y(I)
      YMEAN = YNORM(1)
      DO 77 I = 2,NCASE
   77 YMEAN = YMEAN + YNORM(I)
      YMEAN = YMEAN / NCASE
      DELPR = (YMEAN - 1.) * 100.
      S2 = (YNORM(1) - 1.) ** 2
      DO 83 I = 2,NCASE
   83 S2 = S2 + (YNORM(I) - 1.) ** 2
      S2 = S2 / (NCASE - 1)
      SD = SQRT(S2) * 100.
      WRITE(7,400)(A1(I),A2(I),Y(I),YCAL(I),YNORM(I),I=1,NCASE)
      WRITE(7,404)YMEAN,DELPR,S2,SD
  108 CONTINUE
C
C
    1 FORMAT(8X,I2)
    2 FORMAT(2X,2A4,6F10.0)
    3 FORMAT(10X,6F10.0)
   96 FORMAT('1'////,15X,'REGRESSION VALUES FOR THE MAIN WING')
  196 FORMAT('1'////,15X,'MAIN WING REGRESSION VALUES (CONT.)')
   97 FORMAT('1'////,15X,'REGRESSION VALUES FOR THE FUSELAGE')
  197 FORMAT('1'////,15X,'FUSELAGE REGRESSION VALUES (CONT.)')
   98 FORMAT('1'////,15X,'REGRESSION VALUES FOR THE HORIZONTAL TAIL')
  198 FORMAT('1'////,15X,'HZ. TAIL REGRESSION VALUES (CONT.)')
   99 FORMAT('1'////,15X,'REGRESSION VALUES FOR THE VERTICAL TAIL')
  199 FORMAT('1'////,15X,'VT. TAIL REGRESSION VALUES (CONT.)')
  100 FORMAT('0'//,25X,'THE LINEAR MATRICES'////,4X,3E12.4,9X,'C1',
     *9X,E12.4/,4X,3E12.4,4X,'*',4X,'C2',4X,'=',4X,E12.4/,
     *4X,3E12.4,9X,'C3',9X,E12.4)
  101 FORMAT('0'//,25X,'THE LOG MATRICES'////,4X,3E12.4,9X,'C1',
     *9X,E12.4/,4X,3E12.4,4X,'*',4X,'C2',4X,'=',4X,E12.4/,
     *4X,3E12.4,9X,'C3',9X,E12.4)
  200 FORMAT(///10X,'Y = C1  +  C2 * X  +  C3 * Z :'//,
     *15X,'C1 = ',E14.6//,15X,'C2 = ',E14.6//,15X,'C3 = ',E14.6)
  201 FORMAT(///10X,'Y = C1  *  X ** C2  *  Z ** C3 :'//,
     *15X,'C1 = ',E14.6//,15X,'C2 = ',E14.6//,15X,'C3 = ',E14.6)
  300 FORMAT('0'//,8X,' MODEL    ACTUAL WEIGHT   CALCULATED WEIGHT',
     *'     NORM'//)
  400 FORMAT(11X,2A4,2X,F8.1,8X,F8.1,8X,F6.2)
  404 FORMAT('0'//,5X,'MEAN:',F7.4,', DEL PRCT:',F5.2,'%, VARIANCE:',
     *F7.4,', STD. DEV.:',F7.2,'%')
```

```
      STOP
      END
C
C
C
      SUBROUTINE WING(M,GW,WW,N,RHO,F,FS,LAMDA,BSPAN,SW,CR,CT,TR,TT,
     &X,Y,Z)
      REAL GW(20),WW(20),N(20),RHO(20),F(20),FS(20),
     &LAMDA(20),BSPAN(20),SW(20),CR(20),CT(20),TR(20),TT(20),
     &X(20),Y(20),Z(20),A(20),B(20),D(20)
      DO 10 I = 1,M
      A(I) = RHO(I) / F(I)
      B(I) = RHO(I) / FS(I)
      D(I) = WW(I) / GW(I)
      X(I) = A(I) * N(I) * BSPAN(I) * SW(I) * (1.-D(I)) *
     &       (CR(I) + 2. * CT(I)) / ((COS(LAMDA(I)))**2. *
     &       (CR(I) + CT(I)) * (2. * CR(I) + CT(I)) *
     &       (2. * TR(I) + TT(I)))
      Y(I) = WW(I) / GW(I)
   10 Z(I) = B(I) * BSPAN(I) * N(I) * (1.-D(I))
      RETURN
      END
C
      SUBROUTINE FUSLGE(M,GW,WW,N,RHOP,RHOR,ER,FF,LF,BF,WACT,X,Y,Z)
      REAL GW(20),WW(20),N(20),RHOP(20),RHOR(20),ER(20),LF(20),
     &FF(20),BF(20),X(20),Y(20),Z(20),A(20),B(20),D(20),WACT(20)
      DO 10 I = 1,M
      A(I) = RHOP(I) / FF(I)
      B(I) = RHOR(I) / ER(I) ** .5
      D(I) = WW(I) / GW(I)
      X(I) = A(I) * N(I) * (1.-D(I)) * LF(I) ** 2. / BF(I)
      Y(I) = WACT(I) / GW(I)
   10 Z(I) = (B(I)/(N(I) ** .5)) * BF(I) ** 2. / (1.-D(I)) ** .5
      RETURN
      END
C
      SUBROUTINE HZTAIL(M,GW,WW,N,RHO,F,FS,LAMDA,BSPAN,SW,S,CR,CT,
     &TR,TT,WACT,X,Y,Z)
      REAL GW(20),WW(20),N(20),RHO(20),F(20),FS(20),LAMDA(20),BSPAN(20),
     &SW(20),S(20),CR(20),CT(20),TR(20),TT(20),X(20),Y(20),Z(20),
     &WACT(20),A(20),B(20),D(20)
      DO 10 I = 1,M
      A(I) = RHO(I) / F(I)
      B(I) = RHO(I) / FS(I)
      D(I) = WW(I) / GW(I)
      X(I) = A(I) * N(I) * (1. - D(I)) *
     &       BSPAN(I) * S(I) ** 2. * (CR(I) + 2. * CT(I)) /
     &       (SW(I) * (COS(LAMDA(I)))**2. * (CR(I) + CT(I)) *
     &       (2. * CR(I) + CT(I)) * (2. * TR(I) + TT(I)))
      Y(I) = WACT(I) / GW(I)
   10 Z(I) = B(I) * N(I) * BSPAN(I) * (1.-D(I)) * S(I) / SW(I)
      RETURN
      END
C
```

```
      SUBROUTINE VTAIL(M,GW,WW,N,RHO,F,FS,LAMDA,BSPAN,SW,S,CR,CT,
     &TR,TT,WACT,X,Y,Z)
      REAL GW(20),WW(20),N(20),RHO(20),F(20),FS(20),LAMDA(20),BSPAN(20),
     &SW(20),S(20),CR(20),CT(20),TR(20),TT(20),X(20),Y(20),Z(20),
     &WACT(20),A(20),B(20),D(20)
      DO 10 I = 1,M
      A(I) = RHO(I) / F(I)
      B(I) = RHO(I) / FS(I)
      D(I) = WW(I) / GW(I)
      X(I) = A(I) * N(I) * (1. - D(I)) *
     &       BSPAN(I) * S(I) ** 2. * (CR(I) + 2. * CT(I)) /
     &       (SW(I) * (COS(LAMDA(I)))**2. * (CR(I) + CT(I)) *
     &       (2. * CR(I) + CT(I)) * (2. * TR(I) + TT(I)))
      Y(I) = WACT(I) / GW(I)
   10 Z(I) = B(I) * N(I) * BSPAN(I) * (1.-D(I)) * S(I) / SW(I)
      RETURN
      END
C
      SUBROUTINE WINC(M,WGS,N,RHO,F,FS,LAMDA,BSPAN,SW,CR,CT,TR,TT,
     &X,Z)
      REAL WGS(20),N(20),RHO(20),F(20),FS(20),
     &LAMDA(20),BSPAN(20),SW(20),CR(20),CT(20),TR(20),TT(20),
     &X(20),Z(20),A(20),B(20),D(20)
      DO 10 I = 1,M
      A(I) = RHO(I) / F(I)
      B(I) = RHO(I) / FS(I)
      D(I) = WGS(I)
      X(I) = A(I) * N(I) * BSPAN(I) * SW(I) * (1.-D(I)) *
     &       (CR(I) + 2. * CT(I)) / ((COS(LAMDA(I)))**2. *
     &       (CR(I) + CT(I)) * (2. * CR(I) + CT(I)) *
     &       (2. * TR(I) + TT(I)))
   10 Z(I) = B(I) * BSPAN(I) * N(I) * (1.-D(I))
      RETURN
      END
C
      SUBROUTINE FUSLGC(M,WGS,N,RHOP,RHOR,ER,FF,LF,BF,X,Z)
      REAL WGS(20),N(20),RHOP(20),RHOR(20),ER(20),LF(20),
     &FF(20),BF(20),X(20),Z(20),A(20),B(20),D(20)
      DO 10 I = 1,M
      A(I) = RHOP(I) / FF(I)
      B(I) = RHOR(I) / ER(I) ** .5
      D(I) = WGS(I)
      X(I) = A(I) * N(I) * (1.-D(I)) * LF(I) ** 2. / BF(I)
   10 Z(I) = (B(I)/(N(I) ** .5)) * BF(I) ** 2. / (1.-D(I)) ** .5
      RETURN
      END
C
      SUBROUTINE HZTAIC(M,WGS,N,RHO,F,FS,LAMDA,BSPAN,SW,S,CR,CT,
     &TR,TT,X,Z)
      REAL WGS(20),N(20),RHO(20),F(20),FS(20),LAMDA(20),BSPAN(20),
     &SW(20),S(20),CR(20),CT(20),TR(20),TT(20),X(20),Z(20),
     &A(20),B(20),D(20)
      DO 10 I = 1,M
      A(I) = RHO(I) / F(I)
      B(I) = RHO(I) / FS(I)
```

```
      D(I) = WGS(I)
      X(I) = A(I) * N(I) * (1. - D(I)) *
     &       BSPAN(I) * S(I) ** 2. * (CR(I) + 2. * CT(I)) /
     &       (SW(I) * (COS(LAMDA(I)))**2. * (CR(I) + CT(I)) *
     &       (2. * CR(I) + CT(I)) * (2. * TR(I) + TT(I)))
   10 Z(I) = B(I) * N(I) * BSPAN(I) * (1.-D(I)) * S(I) / SW(I)
      RETURN
      END
C
      SUBROUTINE VTAIC(M,WGS,N,RHO,F,FS,LAMDA,BSPAN,SW,S,CR,CT,
     &TR,TT,X,Z)
      REAL WGS(20),N(20),RHO(20),F(20),FS(20),LAMDA(20),BSPAN(20),
     &SW(20),S(20),CR(20),CT(20),TR(20),TT(20),X(20),Z(20),
     &A(20),B(20),D(20)
      DO 10 I = 1,M
      A(I) = RHO(I) / F(I)
      B(I) = RHO(I) / FS(I)
      D(I) = WGS(I)
      X(I) = A(I) * N(I) * (1. - D(I)) *
     &       BSPAN(I) * S(I) ** 2. * (CR(I) + 2. * CT(I)) /
     &       (SW(I) * (COS(LAMDA(I)))**2. * (CR(I) + CT(I)) *
     &       (2. * CR(I) + CT(I)) * (2. * TR(I) + TT(I)))
   10 Z(I) = B(I) * N(I) * BSPAN(I) * (1.-D(I)) * S(I) / SW(I)
      RETURN
      END
```

PHSEII FORTRAN

```
C           ******************************************************
C           *   THE FOLLOWING PROGRAM USES THE COEFFICIENTS OF    *
C           *   THE GENREG PROGRAM TO CALCULATE AIRCRAFT COM-     *
C           *   PONENT WEIGHTS (WING, FUSELAGE, VERTICAL AND      *
C           *   HORIZONTAL TAIL).  THE PROGRAM MAY BE USED IN-    *
C           *   TERACTIVELY OR WITH A DATA SET (PHSEII DATA).     *
C           *                                                     *
C           *   THIS PROGRAM IS EXECUTED BY USE OF EXEC "PHSEII". *
C           *                                                     *
C           ******************************************************
C
      REAL N,LF,LAMDA,LAMDAT,LAMDAV
      LOGICAL *1 TITLE(60)
      DATA RHO/.1/,RHOP/.1/,RHOR/.1/,ER/10.6/,RHOT/.1/,RHOV/.1/
      DATA FW/65000./,FS/24200./,FT/65000./,FTS/24200./,FV/65000./
      DATA FVS/24200./,FF/65000./
      DATA  C1/.0744046/, C2/.122482/, C3/1.19660/
      DATA C1F/.0876243/,C2F/.217097/,C3F/.432849/
      DATA C1H/.0120291/,C2H/-.234289/,C3H/2.37790/
      DATA C1V/.00835541/,C2V/.0203209/,C3V/3.80514/
      WRITE(6,300)
  300 FORMAT(/2X,'TO USE THIS PROGRAM WITH EXISTING DATA SET "PHSEII',
     &' DATA",'/,' ENTER "1".  FOR INTERACTIVE USAGE, ENTER "2".'/)
      READ(5,*)NTR
      IF(NTR.EQ.1)GO TO 680
      WRITE(6,310)
  310 FORMAT(2X,'NAME OF AIRPLANE OR CASE?')
      READ(5,311)(TITLE(KKK),KKK=1,60)
      WRITE(4,311)(TITLE(KKK),KKK=1,60)
  311 FORMAT(60A1)
      WRITE(6,320)
  320 FORMAT(2X,'ENTER ESTIMATED GROSS WEIGHT AND DESIGN LOAD FACTOR:')
      READ(5,*)GW,N
      WRITE(4,317)GW,N
  317 FORMAT(2F12.4)
      WRITE(6,325)
  325 FORMAT(2X,'IF AIRPLANE IS ALUMINUM AND MATERIAL STRENGTHS ARE '/,
     &' UNKNOWN, ENTER "1".  TO SUPPLY MATERIAL VALUES, ENTER "2"'/,
     &' (DEFAULT VALUES WILL BE SHOWN FOR EACH ENTRY):')
      READ(5,*)MTV
      IF(MTV.EQ.1)GO TO 649
      WRITE(6,330)
  330 FORMAT(' WING MATERIAL DENSITY IN PSI (FOR DEFAULT, ENTER .1):')
      READ(5,*)RHO
      WRITE(6,335)
  335 FORMAT(' FUSELAGE PANEL MATERIAL DENSITY IN PSI (DFLT ENTR .1):')
      READ(5,*)RHOP
      WRITE(6,340)
  340 FORMAT(' FUSELAGE RIB MATERIAL DENSITY IN PSI (DFLT ENTR .1):')
      READ(5,*)RHOR
      WRITE(6,345)
```

```
345 FORMAT(' FUSELAGE RIB YOUNGS MODULUS IN PSI (DFLT ENTR 10.6):')
    READ(5,*)ER
    WRITE(6,350)
350 FORMAT(' HORIZONTAL TAIL MATERIAL DENSITY IN PSI (DFLT ENTR .1):')
    READ(5,*)RHOT
    WRITE(6,355)
355 FORMAT(' VERTICAL TAIL MATERIAL DENSITY IN PSI (DFLT ENTR .1):')
    READ(5,*)RHOV
    WRITE(6,360)
360 FORMAT(' ALLOWABLE WING COVER STRESS IN PSI (DFLT ENTR 65000):')
    READ(5,*)FW
    WRITE(6,365)
365 FORMAT(' ULTIMATE WING SHEAR STRESS IN PSI (DFLT ENTR 24200):')
    READ(5,*)FS
    WRITE(6,370)
370 FORMAT(' ALLOWABLE TAIL COVER STRESS IN PSI (DFLT ENTR 65000):')
    READ(5,*)FT
    WRITE(6,375)
375 FORMAT(' ULTIMATE TAIL SHEAR STRESS IN PSI (DFLT ENTR 24200):')
    READ(5,*)FTS
    WRITE(6,380)
380 FORMAT(' ALLOWABLE FIN COVER STRESS IN PSI (DFLT ENTR 65000):')
    READ(5,*)FV
    WRITE(6,385)
385 FORMAT(' ULTIMATE FIN SHEAR STRESS IN PSI (DFLT ENTR 24200):')
    READ(5,*)FVS
    WRITE(6,390)
390 FORMAT(' ALWABLE FUSELAGE COVER STRESS IN PSI (DFLT ENTR 65000):')
    READ(5,*)FF
649 CONTINUE
    WRITE(4,318)RHO,RHOP,RHOR,ER,RHOT,RHOV
    WRITE(4,318)FW,FS,FT,FTS,FV,FVS
C
C   **********************************************************
C   *  CONTINUE BY ENTERING GEOMETRIC VALUES FOR THIS CASE.  *
C   **********************************************************
C
    WRITE(6,395)
395 FORMAT(' ENTER WING SPAN, WING AREA, TAIL SPAN, TAIL AREA,'/
   &' FIN SPAN, AND FIN AREA, UNITS ARE INCHES OR SQUARE INCHES:')
    READ(5,*)BW,SW,BT,ST,BV,SV
    WRITE(4,318)BW,SW,BT,ST,BV,SV
    WRITE(6,400)
400 FORMAT(' ENTER FUSELAGE WIDTH AND LENGTH, AND 25% CHORD SWEEP'/,
   &' OF THE WING, TAIL AND FIN.  UNITS ARE INCHES AND RADIANS:')
    READ(5,*)BF,LF,LAMDA,LAMDAT,LAMDAV
    WRITE(4,318)BF,LF,FF,LAMDA,LAMDAT,LAMDAV
    WRITE(6,405)
405 FORMAT(' ENTER WING ROOT CHORD, WING TIP CHORD, TAIL ROOT'/,
   &' AND TIP CHORDS, AND FIN ROOT AND TIP CHORDS (INCHES):')
    READ(5,*)CR,CT,CRT,CTT,CRV,CTV
    WRITE(4,318)CR,CT,CRT,CTT,CRV,CTV
    WRITE(6,410)
410 FORMAT(' ENTER WING ROOT MAXIMUM THICKNESS, WING TIP MAXIMUM'/,
   &' THICKNESS, HZ. TAIL ROOT AND TIP MAXIMUM THICKNESSES, AND'/,
```

```
     *' FIN ROOT AND TIP MAXIMUM THICKNESSES (IN INCHES):')
      READ(5,*)TR,TT,TRT,TTT,TRV,TTV
      WRITE(4,318)TR,TT,TRT,TTT,TRV,TTV
  318 FORMAT(6F12.4)
      GO TO 681
  680 CONTINUE
C
C        **************************************************
C        * THE FOLLOWING DATA IS READ FROM FILE #4, CALLED  *
C        * 'PHSEII DATA'.  IT REPRESENTS TITLE, GW, MATR.   *
C        * PROPERTIES, AND GEOMETRIES OF THE AIRPLANE IN-   *
C        * TENDED FOR THE ANALYSIS.                         *
C        **************************************************
C
      READ(4,311) (TITLE(KKK),KKK=1,60)
      READ(4,*)GW,N
      READ(4,*)RHO,RHOP,RHOR,ER,RHOT,RHOV
      READ(4,*)FW,FS,FT,FTS,FV,FVS
      READ(4,*)BW,SW,BT,ST,BV,SV
      READ(4,*)BF,LF,FF,LAMDA,LAMDAT,LAMDAV
      READ(4,*)CR,CT,CRT,CTT,CRV,CTV
      READ(4,*)TR,TT,TRT,TTT,TRV,TTV
  681 CONTINUE
C
C        *******************************************************
C        *  THE FOLLOWING SUBROUTINES ARE SEQUENTIALLY          *
C        *  CALLED FOR THE PURPOSE OF EVALUATING AIRPLANE       *
C        *  COMPONENT WEIGHTS.  THE COEFFICIENTS C1, C2, AND    *
C        *  C3  FOR THE EQUATION "YCAL = C1 + C2 * X + C3 * Z"  *
C        *  HAVE BEEN SUPPLIED.  DIFFERENT COEFFICIENTS MAY BE  *
C        *  USED AND DEFINED INTERACTIVELY.                     *
C        *******************************************************
C
      WRITE(6,450)
  450 FORMAT(' THE COEFFICIENTS C1, C2, AND C3 FOR THE EQUATION'/,
     &' "YCAL = C1 + C2 * X + C3 * Z" HAVE BEEN SUPPLIED IN THIS'/,
     &' PROGRAM (X AND Z ARE DETERMINED PER AIRPLANE).  TO PROVIDE'/,
     &' DIFFERENT COEFFICIENTS, ENTER "1".  TO CONTINUE WITH THE'/,
     &' GIVEN COEFFICIENTS, ENTER "2".')
      READ(5,*)NCOF
      IF(NCOF.EQ.2)GO TO 691
      WRITE(6,415)
  415 FORMAT(' ENTER NEW WING COEFFICIENTS C1, C2, AND C3:')
      READ(5,*)C1,C2,C3
      WRITE(6,420)
  420 FORMAT(' ENTER NEW FUSELAGE COEFFICIENTS C1F, C2F, AND C3F:')
      READ(5,*)C1F,C2F,C3F
      WRITE(6,425)
  425 FORMAT(' ENTER NEW HZ TAIL COEFFICIENTS C1H, C2H, AND C3H:')
      READ(5,*)C1H,C2H,C3H
      WRITE(6,430)
  430 FORMAT(' ENTER NEW VT TAIL COEFFICIENTS C1V, C2V, AND C3V:')
      READ(5,*)C1V,C2V,C3V
  691 CONTINUE
      WGS = .10
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```
      EPS = .0001
      WRITE(6,92)
   92 FORMAT(//2X,'ITERATION FROM WW/GW =.10 (10%):'/)
   38 CALL WINC(WGS,N,RHO,FW,FS,LAMDA,BW,SW,CR,CT,
     &TR,TT,X,Z)
      YCAL = C1 + C2 * X + C3 * Z
      IF(YCAL.GT.WGS)WGS = WGS + (YCAL-WGS)/2.
      IF(YCAL.LT.WGS)WGS = WGS - (WGS-YCAL)/2.
      WRITE(6,93)YCAL,WGS
   39 IF(ABS(WGS-YCAL).GT.EPS)GO TO 38
   41 YCAL = YCAL * GW
   93 FORMAT(2F14.4)
      CALL FUSLGC(WGS,N,RHOP,RHOR,ER,FF,LF,BF,X,Z)
      YCFUS = ( C1F + C2F * X + C3F * Z )  * GW
      CALL HZTAIC(WGS,N,RHOT,FT,FTS,LAMDAT,BT,SW,
     &ST,CRT,CTT,TRT,TTT,X,Z)
      YCHZT = ( C1H + C2H * X + C3H * Z) * GW
      CALL VTAIC(WGS,N,RHOV,FV,FVS,LAMDAV,BV,SW,
     &SV,CRV,CTV,TRV,TTV,X,Z)
      YCVTL = ( C1V + C2V * X + C3V * Z) * GW
   44 CONTINUE
      WRITE(6,500)(TITLE(KKK),KKK=1,60),YCAL,YCFUS,YCHZT,YCVTL
      WRITE(7,500)(TITLE(KKK),KKK=1,60),YCAL,YCFUS,YCHZT,YCVTL
  500 FORMAT('1'//,2X,'TITLE: ',60A1////,2X,'WING WEIGHT: ',
     &T20,F10.2,' POUNDS'//,2X,'FUSELAGE WEIGHT: ',T20,F10.2,
     &' POUNDS'//,2X,'HZ. TAIL WEIGHT: ',T20,F10.2,' POUNDS'//,2X,
     &'VT. TAIL WEIGHT: ',T20,F10.2,' POUNDS')
      STOP
      END
C
C
      SUBROUTINE WINC(WGS,N,RHO,F,FS,LAMDA,BSPAN,SW,CR,CT,TR,TT,
     &X,Z)
      REAL N,LAMDA
      A = RHO / F
      B = RHO / FS
      D = WGS
      X = A * N * BSPAN * SW * (1.-D) *  (CR + 2. * CT) /
     &((COS(LAMDA))**2. * (CR + CT) * (2. * CR + CT) * (2. * TR + TT))
   10 Z = B * BSPAN * N * (1.-D)
      RETURN
      END
C
      SUBROUTINE FUSLGC(WGS,N,RHOP,RHOR,ER,FF,LF,BF,X,Z)
      REAL N,LF
      ER = ER * 1000000.
      A = RHOP / FF
      B = RHOR / ER ** .5
      D = WGS
      X = A * N * (1.-D) * LF ** 2. / BF
      Z = (B/(N ** .5)) * BF ** 2. / ((1.-D) ** .5)
      RETURN
      END
C
      SUBROUTINE HZTAIC(WGS,N,RHO,F,FS,LAMDA,BSPAN,SW,S,CR,CT,
```

```
     &TR,TT,X,Z)
      REAL N,LAMDA
      A = RHO / F
      B = RHO / FS
      D = WGS
      X = A * N * (1. - D) * BSPAN * S ** 2. * (CR + 2. * CT) /
     &(SW*(COS(LAMDA))**2. * (CR + CT) * (2. * CR + CT) * (2.*TR + TT))
      Z = B * N * BSPAN * (1.-D) * S / SW
      RETURN
      END
C
      SUBROUTINE VTAIC(WGS,N,RHO,F,FS,LAMDA,BSPAN,SW,S,CR,CT,
     &TR,TT,X,Z)
      REAL N,LAMDA
      A = RHO / F
      B = RHO / FS
      D = WGS
      X = A * N * (1. - D) *BSPAN * S ** 2. * (CR + 2. * CT) /
     &(SW*(COS(LAMDA))**2. * (CR + CT) * (2. * CR + CT) * (2.*TR + TT))
      Z = B * N * BSPAN * (1.-D) * S / SW
      RETURN
      END
```

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15. Supplementary Notes

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16. Abstract

Currently available weight estimation methods for general aviation airplanes were investigated. New equations with explicit material properties were developed for the weight estimation of aircraft components such as wing, fuselage and empennage. Regression analysis was applied to the basic equations for a data base of twelve airplanes to determine the coefficients. The resulting equations can be used to predict the component weights of either metallic or composite airplanes.

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