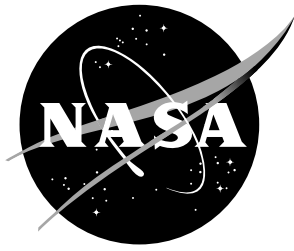# The Flight Optimization System Weights Estimation Method

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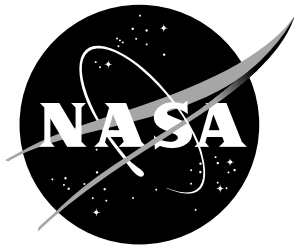
The Flight Optimization System Weights Estimation Method

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Errata

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The Flight Optimization System Weights Estimation Method

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Summary of Changes:

Equation (10) has been corrected to appear as follows:

$$BT = \frac{0.215 \times (0.37 + 0.7 \times TR) \times \left(\frac{SPAN^2}{SW}\right)^{EMS}}{CAYL \times TCA} \qquad (10)$$

The definition of the variable W1NIR has been corrected to read as follows:

W1NIR = Wing bending material weight factor

The paragraph preceding Equation (33) has been revised to read as follows:

Total Wing Bending Material Weight: The first term of the wing weight equation (W1) loosely represents the total bending material weight of the wing. This variable is calculated in two steps. First, Equation 33 is used to calculate the wing bending material weight factor without any adjustment for the effects of inertia relief (W1NIR). Second, after more of the wing weights have been calculated the wing bending material weight is calculated using Equation 37. The wing bending material weight factor is calculated with the following equation.

Acknowledgments

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Abstract

FLOPS has been the primary aircraft synthesis software used by the Aeronautics Systems Analysis Branch at NASA Langley Research Center. It was created for rapid conceptual aircraft design and advanced technology impact assessments. FLOPS is a single computer program that includes weights estimation, aerodynamics estimation, engine cycle analysis, propulsion data scaling and interpolation, detailed mission performance analysis, takeoff and landing performance analysis, noise footprint estimation, and cost analysis. It is well known as a baseline and common denominator for aircraft design studies. FLOPS is capable of calibrating a model to known aircraft data, making it useful for new aircraft and modifications to existing aircraft.

The weight estimation method in FLOPS is known to be of high fidelity for conventional tube with wing aircraft and a substantial amount of effort went into its development. This report serves as a comprehensive documentation of the FLOPS weight estimation method. The process used to develop the FLOPS weight estimation equations is presented along with the FLOPS weight estimation procedure. The component weight estimation relationships, tail sizing method, hybrid wing body sizing method, and fuselage sizing method are documented as support for the aircraft weight estimation.

1 Introduction

The National Aeronautics and Space Administration (NASA) mission is, "Drive advances in science, technology, and exploration to enhance knowledge, education, innovation, economic vitality, and stewardship of the Earth." [1] Furthermore, NASA's aeronautics research vision, "encompasses a broad range of technologies to meet future needs of the aviation community, the Nation, and the world for safe, efficient, flexible, and environmentally sustainable air transportation." [2] One of NASA's current capabilities that supports the mission and Aeronautics Research Mission Directorate (ARMD) vision is the analysis and synthesis of aircraft. This innovative research strives to enable revolutionary transformations in aviation through the exploration of advanced concepts and technologies. The Flight Optimization System (FLOPS) is one software application that is used to enable the aircraft analysis and synthesis capability.

FLOPS was developed at the NASA Langley Research Center (LaRC). It was created to design and synthesize the configuration of new aircraft at the conceptual design level and assess the impacts of advanced technologies [3]. FLOPS is a single computer program with an execution control module that executes eight other modules. Figure 1 shows the relationships between the modules. The modules include weights estimation, aerodynamics estimation, engine cycle analysis, propulsion data scaling and interpolation, detailed mission performance analysis, takeoff and landing performance analysis, noise footprint estimation, and cost analysis.

This manuscript discusses the development process used for the weight estimation equations, how the equations are used in the FLOPS software to estimate the total weight of the aircraft, the methods and weight equations for each component including some helpful utilities, and finally ends with a list of the variables in Appendix D.
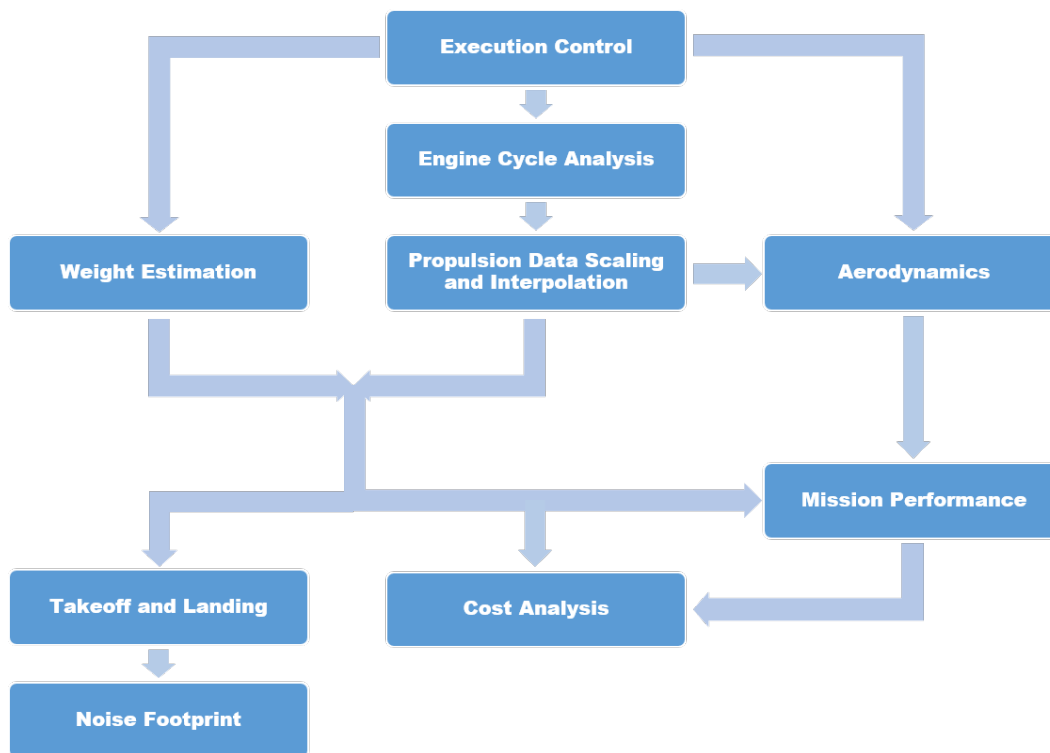
Figure 1. Relationships between FLOPS modules.

2   Purpose

The weights estimation method developed for FLOPS has very little publicly available documentation. Reference 3 is often used, but contains little detail. This report is meant to serve as comprehensive documentation of the FLOPS weights estimation method. To maintain traceability between this document and the source code, where it is practical the original FORTRAN variable names are used. Large amounts of information from two major unpublished sources were used to accomplish this task: the FLOPS WATE Module Documentation, "WateDoc.txt" and the FLOPS User's Guide, "FlopsMan.txt", updated July 8, 2011. Both documents are written by the creator of FLOPS, L. A. McCullers, and are included in the FLOPS software release packages.

3   Development of the Weight Equations

The weight equations in FLOPS are based on aircraft data from 17 transport aircraft and 25 fighter/attack aircraft. [3] The equations were curve fit to that data using a form that is based on physical characteristics. The curve fit was created by optimization using nonlinear programming techniques.

The primary advantage of optimization over fixed-form curve fitting techniques for weight equation development is that an equation form may be selected that mirrors the physical characteristics of the system. Therefore, reasonable predictions can be made for new designs that are outside of the database. A fixed-form equation developed from regression analysis (exponential) or polynomial

techniques may fit existing data somewhat. They are, however, of little predictive value for new designs or sensitivity analyses that lie outside of the database. The first step in the development of the FLOPS weight equations using optimization was to create an equation that utilized all appropriate variables in multiple relationships. The constants and exponents in the equations were optimized with a curve fit program, which eliminated many inappropriate variables and terms by setting constants and exponents equal to zero. The next step was to try new relationships. The resulting equation was then massaged, altered, tested, and then re-optimized. The goal of the last step was to simplify the equation as much as possible and eliminate variables that are not easy to obtain. The process continued until an equation was found that best met the intended objectives: makes physical sense, passes all tests, and is the best fit to the data.

An example of the tests performed is provided here using the relationship between the area and weight of the horizontal tail. A reasonableness test is: "as the area increases, the weight increases." Some limit tests are: "as the area goes to zero, the weight goes to zero" and "as the area goes to infinity, the weight goes to infinity." A sensitivity test is: "the relationship between the area and weight should be nearly linear, meaning that the exponent on area should be between 0.5 and 1.5." These types of tests were applied to each independent variable in each of the weight equations developed for FLOPS.

An optimization problem was set up to curve fit the equations to the data. The objective function was to minimize the sum of squares percent error between the input data and the generated curve. A Davidon-Fletcher-Powell algorithm, an approximate second order method, was used to determine the search directions. Quadratic interpolation was then used in those search directions for minimization. This method enables all constants, coefficients, and exponents of almost any form of equation with an arbitrary number of independent variables to be optimized. Reasonable or physical limits can be applied to the independent variables through upper bounds, lower bounds, or both. Logarithmic or tangent variable transformations are enforced so that optimized constants approach their bounds asymptotically. This method allows for the form of the equation to mirror the physical characteristics of the system.

The FLOPS weight equations were developed for four different aircraft types: transport, fighter/attack, general aviation, and hybrid wing body (HWB). Transport aircraft are those whose primary objective is to carry cargo or passengers. Fighter/attack aircraft are high performance military aircraft. General aviation aircraft are typically small aircraft that serve in various non-combat roles. HWB aircraft are a special type of aircraft that merge the wing and body into one shape. FLOPS was modified to enable HWB analysis [4] which will be discussed more in Appendix B. Unless otherwise indicated, the transport aircraft weight equations are used for HWB aircraft.

4   Aircraft Design Gross Weight Estimation

FLOPS's mission analysis uses the weight estimation routine in one of two modes: (1) fixed gross weight where the mission range or endurance is calculated or (2) fixed mission range or endurance where the vehicle size and gross weight is calculated [3]. One of the more innovative features of FLOPS is the fixed range/endurance mode. The benefit of this approach is that the range/endurance constraint is always satisfied and the gross weight is a calculated value. This method more closely fits with "clean sheet" aircraft design where aircraft are designed to satisfy a

mission requirement like range.

In the fixed range/endurance mode, with the mission range/endurance fixed and the gross weight calculated, the weights are iterated until the resulting range/endurance is equal to the design range/endurance specified. The FLOPS mission performance analysis uses an initial weight estimate to calculate the range/endurance. The weights are then adjusted appropriately and the mission performance analysis is run again. This continues until convergence is reached on the mission range/endurance.

FLOPS uses a weight build-up method instead of a more preliminary weight fraction method. Weight is estimated for each component in the weight statement and then summed together to calculate the maximum zero fuel weight. The fuel weight results from the difference between the current estimate for gross weight and maximum zero fuel weight. This build-up is performed once for a fixed gross weight analysis and the resulting weight statement is output. For the fixed range or endurance mode, this process is repeated for each new guess of gross weight until the fuel weight available equals the fuel weight required.

The weight estimation process begins by initializing the gross weight based on the user input for design gross weight and ramp weight. Next, the total number of passengers and engines are summed from user input values and the wing trapezoidal area, fuselage planform area, and wing span are calculated. Next, weight penalty factors are calculated for distributed propulsion aircraft with a large number of engines. Cruise dynamic pressure for general aviation aircraft is calculated followed by the maximum fuel. The maximum fuel is calculated from user inputs and fuselage and wing geometric parameters. FLOPS's weight estimation routine then starts the component build-up by stepping through the weight statement. Figure 2 shows this process. The groups are shown on the left with their components listed to the right.
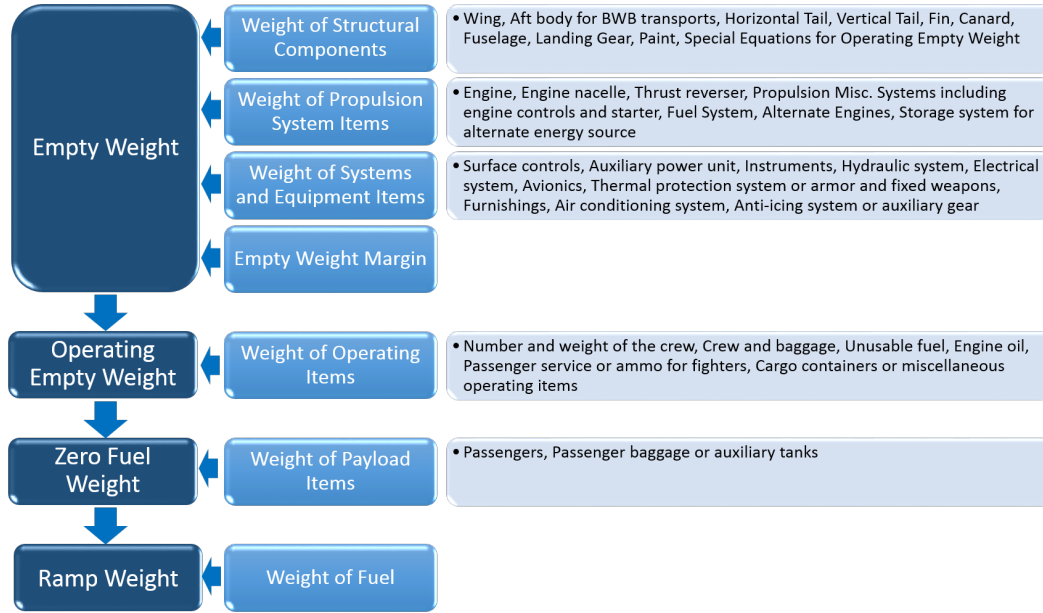
Figure 2. FLOPS weight estimation build-up process flow diagram.

5 Weight Estimation Methods

This section contains the details of the FLOPS weight estimation equations. The variable definitions are included for each component based on the FLOPS User's Guide definitions. Each component of the group weight statement is presented in its own sub-section below.

5.1 Design Gross Weight

The design gross weight (DG) is used throughout the weight estimation process. In FLOPS it can be calculated using three different methods which are applied based on the value of the variable DGW. Design gross weight is the total weight of the aircraft. It can refer to any moment in time during the mission including cruise or ground operations. The takeoff weight is the total weight of the aircraft at takeoff. Ramp weight is the total weight of the aircraft authorized for maneuvering on the ground including taxi and towing.

If DGW is between 0 and 5 then it is assumed to be a fraction of ramp weight. In this case the following equation is used to calculate DG.

$$DG = DGW \times GW \tag{1}$$

where
DG   =   Design gross weight, lb
DGW   =   Design gross weight input variable. This variable has
                 two different units depending on its value. If it is less
                 than 5, it is assumed to be a fraction of ramp weight.
                 Otherwise if it is greater than 5 it is assumed to be
                 equal to the design gross weight and has units of lbs.
GW   =   Ramp weight, lb

If DGW is greater than 5 it is assumed that the user is specifying the design gross weight and the following equation is used.

$$DG = DGW \tag{2}$$

Finally, if the variable DGW is not provided by the user the design gross weight is set equal to the ramp weight.

$$DG = GW \tag{3}$$

5.2   Structural Components

5.2.1   Wing

FLOPS was developed for sizing and synthesis of advanced aircraft concepts which commonly have unconventional wings. Many concepts take advantage of advanced technologies and configurations such as higher aspect ratios, aeroelastic tailoring, forward sweep, and strut bracing. Insufficient statistical data was available to predict aeroelastic tailoring, forward sweep, and strut bracing effects on wing weight when the FLOPS wing weight equations were being developed. Furthermore, the relationship between sweep angle and flutter and divergence weight penalties for high aspect ratio wings was not available. Therefore the Aeroelastic Tailoring and Structural Optimization (ATSO) program [5] was used to generate optimum wing designs to aid in the prediction of those effects. Some trends from the ATSO wing weight study are shown in Reference 3. The study data was used along with existing aircraft weights data to generate an equation that accurately predicts existing and unconventional aircraft wing weights.

The wing weight estimation technique was developed to be sensitive to local changes in wing geometry because many unconventional wings can have complex thickness-to-chord ratios (t/c), break locations, sweeps, and reference areas. Many wing weight equations depend on average values for these variables and assume a trapezoidal planform. Simplifying the geometries and the variation in the reference values can cause significant variations in the weight estimates.

The wing weight estimation technique uses four terms, W1, W2, W3, and W4, which are summed to determine the total wing weight. The most complex is W1, which includes a wing bending material factor that will be explained first. Two methods are available to estimate the wing bending material factor. The simplified method is intended to be used when little detail is known about the wing geometry. The detailed method is used when the wing geometry is well defined.

The wing weight estimation section starts with definitions of reference variables, then discusses the two methods for estimating the bending material weight factor. Next the bending material, shear material and control surfaces, and miscellaneous items weights are discussed including the required wing weight equation constants. Next, the wing bending material weight adjustment for inertia relief is shown followed by the fourth term of the wing weight, which is the aft body for HWB aircraft. Finally, the total wing weight calculation is shown by summing the four wing weight terms.

Wing Reference Area: Three methods are available in FLOPS to estimate the wing reference area. These methods are presented in the following equation and described in the following

paragraph.

$$SW = \begin{cases} GW/WSR, & \text{if wing loading is given} \\ \frac{SPAN^2}{AR+GLOV}, & \text{for simple wing method} \\ \sum(ETA_{i+1} - ETA_i) \times (C_i + C_{i+1}), & \text{for detailed wing method} \end{cases} \quad (4)$$

where

SW	=	Reference wing area, ft^2
GW	=	Ramp weight, lb
WSR	=	Required wing loading, lb/ft^2
GLOV	=	Total glove and bat area beyond theoretical wing area, ft^2
SPAN	=	Wing span, ft
ETA_i	=	Local wing station location, ft. Calculated as the distance from fuselage centerline.
C_i	=	Local chord length, ft

If the wing loading is set by the user, then it is used to calculate the total wing area. The simple wing method uses several common wing geometry variables to calculate the wing area. When more details about the wing geometry are known, the detailed wing method can be used. The detailed wing method discretizes the wing spanwise. The method calculates values as it moves along the span and "i" is used to specify the values at each local spanwise station location.

Wing Trapezoidal Area: The aircraft wing trapezoidal area is estimated using the following equation.

$$SX = SW - GLOV \quad (5)$$

where

SX   =   Wing trapezoidal area, ft

Wing Span: Two variables are used for span in FLOPS, "B" and "SPAN". Both are calculated similarly, but are used in different equations. FLOPS has a special option to fix the span during wing area variations or optimization. It is applied to both variables "B" and "SPAN". To accomplish this the wing aspect ratio is adjusted to maintain a constant span. If the fixed wing span method is used, "SPAN" and "B" are calculated with the following equation.

$$B = SPAN = FIXSPN \quad (6)$$

where

B	=	Wing span, ft
SPAN	=	Wing span, ft
FIXSPN	=	Fixed wing span, ft

If the fixed wing span method is not used, "B" is calculated from the aspect ratio and wing trapezoidal area using the following equation.

$$B = \sqrt{AR \times SX} \quad (7)$$

where

AR   =   Wing aspect ratio

If the fixed wing span method is not used, "SPAN" is calculated from the maximum fuselage width and outboard wing semispan for HWB aircraft and the aspect ratio and wing trapezoidal area for transport, fighter/attack, and general aviation aircraft using the following equation.

$$SPAN = \begin{cases} WF + OSSPAN \times 2, & \text{for HWB aircraft} \\ \sqrt{AR \times (SW - GLOV)}, & \text{otherwise} \end{cases} \tag{8}$$

where
SPAN   = Wing span, ft
WF   = Maximum fuselage width, ft
OSSPAN = Outboard wing semispan of HWB aircraft, ft

Wing Aspect Ratio: FLOPS requires that nonzero values for two of wing aspect ratio, wing reference area, and wing span must be input. The other value is calculated. If all three are input, the total glove and bat area beyond theoretical wing area is calculated. The aircraft wing aspect ratio is estimated using the following equation.

$$AR = \begin{cases} FIXSPN^2/SX, & \text{for fixed wing span method} \\ \frac{SPAN^2}{SW - GLOV}, & \text{otherwise} \end{cases} \tag{9}$$

Simplified Wing Weight Estimation Method: The simplified wing weight estimation uses an equivalent bending material factor assuming a trapezoidal wing area and average thickness to chord ratio. The equivalent bending factor is estimated using the following equation.

$$BT = 0.215 \times (0.37 + 0.7 \times TR) \times \left(\frac{SPAN^2}{SW} \right)^{\frac{EMS}{CAYL \times TCA}} \tag{10}$$

where
BT   = Wing equivalent bending material factor
TR   = Taper ratio of the wing
SPAN   = Wing span, ft
SW   = Reference wing area, ft^2, estimated using Equation 4
EMS   = Wing strut bracing factor, defined below in Equation 11
CAYL   = Wing sweep factor including aeroelastic tailoring, defined below in Equation 12
TCA   = Weighted average of the wing thickness to chord ratio

The wing strut bracing factor, EMS is calculated with the following equation.

$$EMS = 1.0 - 0.25 \times FSTRT \tag{11}$$

where
FSTRT   = Wing strut bracing factor ranging from 0.0 for no wing strut to 1.0 for full benefit from strut bracing

The wing sweep and aeroelastic factor, CAYL, is calculated using the following equation.

$$CAYL = (1.0 - SLAM^2) \times (1.0 + C6 \times SLAM^2 + 0.03 \times CAYA \times C4 \times SLAM) \qquad (12)$$

where
SLAM   =   Sine of the 3/4 chord wing sweep angle, calculated
          using Equation 13
C4   =   Factor defined below in Equation 15
C6   =   Factor defined below in Equation 16
CAYA   =   Factor defined below in Equation 17

The sine of the 3/4 chord wing sweep angle (SLAM) is calculated using the following equation.

$$SLAM = \frac{TLAM}{\sqrt{1.0 + TLAM^2}} \qquad (13)$$

where
TLAM   =   Tangent of the 3/4 chord wing sweep angle, calculated
          using Equation 14

The tangent of the 3/4 chord wing sweep angle is calculated based on the quarter chord wing sweep, aspect ratio, and taper ratio in the following equation.

$$TLAM = tan(SWEEP) - \frac{2.0 \times (1.0 - TR)}{AR \times (1.0 + TR)} \qquad (14)$$

where
SWEEP   =   Quarter chord sweep angle of the wing, deg
TR   =   Taper ratio of the wing
AR   =   Wing aspect ratio

The factor C4 is defined with the following equation.

$$C4 = 1.0 - 0.5 \times FAERT \qquad (15)$$

where
FAERT   =   Aeroelastic tailoring factor used in the design of the
          wing ranging from 0.0 for no aeroelastic tailoring to
          1.0 for maximum aeroelastic tailoring

The factor C6 is defined with the following equation.

$$C6 = 0.5 \times FAERT - 0.16 \times FSTRT \qquad (16)$$

The factor CAYA is defined with the following equation.

$$CAYA = \begin{Bmatrix} 0, & \text{for } AR \leq 5 \\ AR - 5, & \text{for } AR > 5 \end{Bmatrix} \qquad (17)$$

Detailed Wing Weight Estimation Method: The detailed wing weight estimation uses a theoretically derived wing bending factor. It was developed to enable wing weight sensitivity to local geometry changes. The wing is assumed to be an idealized beam with dimensions proportional to the local wing chord and thickness and to carry a user specified load distribution. The bending factor is calculated by numerical integration along a specified load path to estimate the bending material required. Aspect ratio, sweep angle, degree of aeroelastic tailoring, and strut bracing can be used to modify the bending factor for the aeroelastic effects of flutter, divergence, and aeroelastic loads. The modifications are based on a curve fit from the ATSO wing weight study [3].

The process starts by dividing the wing into integration stations along the span. The span stations are assigned to the idealized beam, modifying properties to account for sweep. A load intensity factor, relative y-location and distance, chord length, thickness to chord ratio, and load path sweep angle are calculated for each integration station along the wing. Note that the y-axis for the detailed wing estimation method is assumed to be along the straight trapezoidal wing. The load intensity factor is a relative load, ranging from zero to one that is calculated based on the wing pressure load distribution at each integration station. Additionally, the average sweep weighted by distance from the centerline is calculated using the following equation.

$$ASW = \sum_{i=tip}^{root} (DY_i + 2 \times ETA_i) \times DY_i \times SWP_i \tag{18}$$

where
ASW $=$ Average sweep angle weighted by distance from the centerline, deg
DY_i $=$ Y-distance from current to previous integration station location, ft
ETA_i $=$ Local wing station location, ft. Calculated as the distance from fuselage centerline.
SWP_i $=$ Sweep of load path, deg

The pressure load is calculated for the area between each integration station using the following equation.

$$DELP_i = DY_i \times \frac{C_{(i-1)} \times (2 \times P_{(i-1)} + P_i) + C_i \times (2 \times P_i + P_{(i-1)})}{6} \tag{19}$$

where
$DELP_i$ $=$ Local pressure load used in the detailed wing weight estimation method
C_i $=$ Local chord length, ft
P_i $=$ Local load intensity factor, ranging from 0.0 to 1.0

The total pressure load along the load path is estimated using the following equation.

$$EL_{tot} = \sum_{i=tip}^{root} DELP_i \tag{20}$$

where
EL_{tot} $=$ Total pressure load along the load path used in the calculation of the wing equivalent bending material factor

The moment of the pressure load is calculated about each integration station using the following equation.

$$DELM_i = DY_i^2 \times \frac{C_{(i-1)} \times (3 \times P_{(i-1)} + P_i) + C_i \times (P_i + P_{(i-1)})}{12} \tag{21}$$

where

$DELM_i$ = Local moment of the pressure load used in the detailed wing weight estimation method

The total moment is modified by the local load path sweep angle using the following equation.

$$EM_{tot} = \sum_{i=tip}^{root} (DELM_i + DY_i \times EL_i) \times CSW_i \tag{22}$$

where

EM_{tot} = Total wing bending moment modified by the local load path sweep angle used in the calculation of the wing equivalent bending material factor

EL_i = Partial summation from the tip to the current integration station of the pressure load along the load path, psi

CSW_i = Secant of the load path sweep angle

The required bending material area is calculated at each integration station using the following equation.

$$BMA_i = \frac{EM_i \times CSW_i}{C_i \times T_i} \tag{23}$$

where

BMA_i = Local required bending material area used in the detailed wing weight estimation method

EM_i = Partial summation from the tip to the current integration station of the wing bending moment modified by the local load path sweep angle

T_i = Local thickness to chord ratio

The local area moment and wing area are summed along the load path to calculate the total area moment and wing area using the following equations.

$$PM_{tot} = \sum_{i=tip}^{root} (BMA_{(i-1)} + BMA_i) \times \frac{DY_i}{2} \tag{24}$$

$$S_{tot} = \sum_{i=tip}^{root} (C_{(i-1)} + C_i) \times \frac{DY_i}{2} \tag{25}$$

where

PM_{tot} = Total area moment along the wing load path used in the detailed wing weight estimation method

S_{tot} = Total wing area along the wing load path used in the detailed wing weight estimation method, ft^2

The total area moment is then nondimensionalized with respect to the total load to arrive at the bending factor. The bending factor, before adjusting the sweep angle bucket for high aspect ratio, strut braced wing, and/or aeroelastic tailoring, is calculated using the following equation.

$$BTB = 4 \times \frac{PM_{tot}}{EL_{tot}} \tag{26}$$

where

BTB $=$ Wing equivalent bending material factor without the sweep angle adjustment

The engine inertia relief is calculated at the wing integration stations that are inboard of the engine locations. The engine inertia relief factor is calculated using the following equation.

$$BTE = 8 \times \sum_{i=tip}^{root} (EA_{(i-1)} + EA_i) \times \frac{DY_i}{2} \tag{27}$$

where

BTE $=$ Wing inertia relief factor due to engines, used in the calculation of CAYE in Equation 39

EA_i $=$ Factor defined below in Equation 28

The factor EA_i is defined with the following equation.

$$EA_i = EEM_i \times \frac{CSW_i}{C_i \times T_i} \tag{28}$$

where

EEM_i $=$ Factor defined below in Equation 29

The factor EEM_i is defined with the following equation.

$$EEM_i = \sum_{i=tip}^{root} EEM_{(i-1)} + DELME_i \times CSW_i \tag{29}$$

where

$DELME_i$ $=$ Local moment of the pressure load used for the engine inertia relief calculation in the detailed wing weight estimation method. Defined in Equation 30.

The moment of the pressure load used for the engine inertia relief is calculated about each integration station using the following equation.

$$DELME_i = DELME_{(i-1)} - Y_i + EETA_i \tag{30}$$

where

Y_i $=$ Y-location of the integration station location, ft

$EETA_i$ $=$ Engine position factor, ft. At each station it is equal to the distance between the next outboard engine and the fuselage center line. At stations where there are no additional outboard engines it is equal to zero.

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The bending factor is then modified to shift the sweep angle bucket for high aspect ratio strut braced wings and aeroelastic tailoring using the following equation.

$$BT = \frac{BTB}{AR^{(0.25 \times FSTRT)}} \times (1.0 + (0.5 \times FAERT - 0.16 \times FSTRT)$$
$$\times SA^2 + 0.03 \times CAYA \times (1.0 - 0.5 \times FAERT) \times SA) \quad (31)$$

where

BT	$=$	Wing equivalent bending material factor
AR	$=$	Wing aspect ratio
FAERT	$=$	Aeroelastic tailoring factor used in the design of the wing ranging from 0.0 for no aeroelastic tailoring to 1.0 for maximum aeroelastic tailoring
FSTRT	$=$	Wing strut bracing factor ranging from 0.0 for no wing strut to 1.0 for full benefit from strut bracing
SA	$=$	Sine of the average sweep weighted by distance from the centerline. Defined below using Equation 32.
CAYA	$=$	Factor defined in Equation 17

The sine of the average sweep weighted by distance from the centerline is calculated using the following equation.

$$SA = sin(ASW) \quad (32)$$

where

ASW	$=$	Average sweep angle weighted by distance from the centerline, *deg*. Calculated using Equation 18.

Total Wing Bending Material Weight: The first term of the wing weight equation (W1) loosely represents the total bending material weight of the wing. This variable is calculated in two steps. First in Equation 33 the total bending material is calculated without accounting for the effects of inertia relief. Then after more of the wing weights have been calculated the bending material weight is adjusted to account for the effects of inertia relief in Equation 37. The wing bending material weight without the effects of inertia relief is calculated with the following equation.

$$W1NIR = A1 \times BT \times \left(1.0 + \sqrt{\frac{A2}{SPAN}}\right) \times ULF \times SPAN$$
$$\times (1.0 - 0.4 \times FCOMP) \times (1.0 - 0.1 \times FAERT)$$
$$\times CAYF \times VFACT \times \frac{PCTL}{1,000,000} \quad (33)$$

where

W1NIR	$=$	Wing bending material weight, not including the effects of inertia relief, *lb*
A1	$=$	Constant which depends on the type of aircraft. Defined using Table 1 below.
A2	$=$	Constant which depends on the type of aircraft. Defined using Table 1 below.

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ULF	=	Structural ultimate load factor. The default value is 3.75.
SPAN	=	Wing span, ft
FCOMP	=	Composite utilization factor for wing structure ranging from 0.0 for no composites to 1.0 for maximum use of composites
CAYF	=	Multiple fuselage factor. $CAYF = 1.0$ for a single fuselage and $CAYF = 0.5$ for multiple fuselages.
VFACT	=	Variable sweep factor defined below in Equation 34
PCTL	=	Fraction of load carried by the defined wing

The factor $VFACT$ is defined with the following equation.

$$VFACT = \begin{cases} 1, & VARSWP \leq 0 \\ 1 + VARSWP \times \left(\dfrac{0.96}{cos(SWEEP)} - 1 \right), & VARSWP > 0 \end{cases} \tag{34}$$

where

VARSWP	=	Wing variable sweep weight penalty factor ranging from 0.0 for fixed-geometry wing to 1.0 for full variable-sweep wing

Total Wing Shear Material and Control Surface Weight: The second term of the wing weight equation is W2, which loosely represents the total shear material and control surface weight of the wing. During the weight estimation techniques development, structural and statistical studies showed that the weight of the shear material depends almost entirely on the control surface sizes and design gross weight.

$$W2 = A3 \times (1.0 - 0.17 \times FCOMP) \times SFLAP^{A4} \times DG^{A5} \tag{35}$$

where

W2	=	Second term of the wing weight equation, lb. W2 loosely represents the total shear material and control surface weight of the wing.
A3	=	Constant which depends on the type of aircraft. Defined using Table 1 below.
A5	=	Constant which depends on the type of aircraft. Defined using Table 1 below.
SFLAP	=	Total movable wing surface area including flaps, elevators, spoilers, etc., ft^2
DG	=	Design gross weight, lb

Total Wing Miscellaneous Items Weight: The third term of the wing weight equation is W3, which loosely represents the weight of miscellaneous items.

$$W3 = A6 \times (1.0 - 0.3 \times FCOMP) \times SW^{A7} \tag{36}$$

where

W3 = Third term of the wing weight equation, *lb*. Loosely represents the weight of the miscellaneous items.

A6 = Constant which depends on the type of aircraft. Defined using Table 1 below.

A7 = Constant which depends on the type of aircraft. Defined using Table 1 below.

SW = Reference wing area, ft^2, estimated using Equation 4

Wing Weight Equation Constants: The A1 through A7 constants that depend on the aircraft type are shown in Table 1 below.

Table 1. FLOPS Wing Weight Constants for Each Aircraft Type

	A1	A2	A3	A4	A5	A6	A7
Transport and HWB	8.80	6.25	0.68	0.34	0.60	0.035	1.50
Fighter/Attack	6.80	0.00	0.12	0.65	0.62	0.800	1.20
General Aviation	30.00	0.00	0.25	0.50	0.50	0.160	1.20

Wing Bending Material Weight Inertia Relief Adjustment: The first term of the wing weight equation is adjusted for inertia relief for any engines on the wing and the wing itself.

$$W1 = \frac{DG \times CAYE \times W1NIR + W2 + W3}{1.0 + W1NIR} - W2 - W3 \qquad (37)$$

where

W1 = First term of wing weight equation, *lb*. It loosely represents the total bending material weight of the wing.

DG = Design gross weight, *lb*

CAYE = Propulsion system pod inertia relief factor. Calculated using either Equation 38 if the simplified wing method is used or Equation 39 if the detailed wing method is used.

The propulsion system pod inertia relief factor (CAYE) is calculated differently for the simplified wing and detailed wing methods. For the simplified wing weight estimation method it is defined as follows.

$$CAYE = 1.0 - 0.03 \times NEW \qquad (38)$$

where

NEW = Number of wing mounted engines

For the detailed wing estimation method CAYE is defined as follows.

$$CAYE = 1.0 - \frac{BTE}{BT} \times \frac{WPOD}{DG} \qquad (39)$$

where

BTE $\quad=\quad$ Wing inertia relief factor due to engines. Calculated using Equation 27.

BT $\quad=\quad$ Wing equivalent bending factor. Calculated using Equation 31 for the detailed wing weight method.

WPOD $\quad=\quad$ Weight of engine pod including the nacelle, *lb*. Defined using Equation 40.

The weight of the engine pod (WPOD) is calculated using the following equation.

$$WPOD = \frac{WTNFA}{NENG} + \frac{WNAC}{TNAC} \tag{40}$$

where

WPOD $\quad=\quad$ Weight of engine pod including the nacelle, *lb*

WTNFA $\quad=\quad$ Total weight of engine pod(s) not including the nacelle, *lb*. Defined using Equation 41.

NENG $\quad=\quad$ Total number of engines

WNAC $\quad=\quad$ Weight of nacelle or air induction system. Defined later in this document in Equations 69 and 72, *lb*.

TNAC $\quad=\quad$ Total number of nacelles plus 0.5 if there is a center-mounted engine, calculated with Equation 74

The total weight of engine pod(s) not including the nacelle is defined with the following equation.

$$WTNFA = WENG \times NENG + WTHR + WSTART + 0.25 \times WEC + 0.11 \times WIN$$
$$+ 0.13 \times WELEC + 0.13 \times WHYD + 0.25 \times WFSYS \tag{41}$$

where

WENG $\quad=\quad$ Weight of each scaled baseline engine, *lb*

WTHR $\quad=\quad$ Weight of the thrust reversers, *lb*

WSTART $\quad=\quad$ Weight of the engine starter system, *lb*

WEC $\quad=\quad$ Weight of the engine controls system, *lb*

WIN $\quad=\quad$ Weight of the instruments system group, *lb*

WELEC $\quad=\quad$ Weight of the electrical system group, *lb*

WHYD $\quad=\quad$ Weight of the hydraulic system group, *lb*

WFSYS $\quad=\quad$ Weight of the fuel system (including tanks and plumbing), *lb*

Total Aft Body Weight for Hybrid Wing Body Aircraft: The fourth term of the wing weight equation is W4, which loosely represents the total aft body weight for hybrid wing body (HWB) configurations. This term is only used when the aircraft type selected is HWB. The HWB cabin area is included in the fuselage weight, so this fourth term in the wing weight equation was added to account for the aft body weight. The equation below is based on the work in Reference 6.

$$W4 = (1.0 + 0.05 \times FNEF) \times 0.53 \times SAFTB \times DG^{0.2}$$
$$\times (0.5 + TRAFTB) \times (1 - 0.17 \times FCOMP) \tag{42}$$

where

W4   =   Fourth term of the wing weight equation, *lb*. Loosely represents the total aft body weight for HWB configurations.

FNEF   =   Number of fuselage mounted engines, scaled to account for distributed propulsion if applicable

SAFTB   =   Area of the aft body, ft^2. Defined in Equation 44.

DG   =   Design gross weight, *lb*

TRAFTB   =   Taper ratio of the aft body. Calculated below with Equation 43.

FCOMP   =   Composite utilization factor for wing structure ranging from 0.0 for no composites to 1.0 for maximum use of composites

The taper ratio of the aft body (TRAFTB) is calculated with the following equation.

$$TRAFTB = \frac{(1.0 - RSPSOB) \times \frac{XLW}{RSPSOB}}{(1.0 - RSPCHD) \times XL} \tag{43}$$

where

RSPSOB   =   Percent chord of the HWB fuselage rear spar at the side of body. By default $RSPSOB = RSPCHD$. Where RSPCHD is the percent chord of the HWB fuselage rear spar at the fuselage center line.

XLW   =   Fixed length of side wall, ft. The side wall refers to the outboard wall of the passenger cabin which is also used to define the wing root chord. User input or estimated using Equation 197.

RSPCHD   =   Percent chord of the HWB fuselage rear spar at the fuselage centerline. The default value is 70%.

XL   =   Total fuselage length, ft

The area of the aft body is calculated with the following equation.

$$SAFTB = FPAREA - ACABIN \tag{44}$$

where

SAFTB   =   Area of the aft body, ft^2

FPAREA   =   Fuselage planform area, ft^2, estimated using Equation 62

ACABIN   =   Passenger cabin floor area, ft^2

Total Wing Weight: The wing weight is calculated by summing the four wing weight terms.

$$WWING = W1 + W2 + W3 + W4 \tag{45}$$

where
WWING $=$ Wing weight, lb

5.2.2 Horizontal Tail

Transport aircraft horizontal tail weight is estimated from the tail area, design gross weight, and taper ratio using the following equation.

$$WHT = 0.53 \times SHT \times DG^{0.2} \times (TRHT + 0.5) \tag{46}$$

where
WHT $=$ Horizontal tail weight, lb
SHT $=$ Horizontal tail theoretical area, ft^2
DG $=$ Design gross weight, lb
TRHT $=$ Horizontal tail theoretical taper ratio

Fighter/attack aircraft horizontal tail weight is estimated from the tail area, ultimate load factor, and design gross weight using the following equation.

$$WHT = 0.002 \times SHT^{0.87} \times (ULF \times DG)^{0.66} \tag{47}$$

where
ULF $=$ Structural ultimate load factor. The default value is 3.75.

General aviation aircraft horizontal tail weight is estimated from the cruise dynamic pressure, horizontal tail area, ultimate load factor, and the design gross weight using the following equation.

$$WHT = 0.016 \times SHT^{0.873} \times (ULF \times DG)^{0.414} \times QCRUS^{0.122} \tag{48}$$

where
QCRUS $=$ Cruise dynamic pressure, psf, calculated using Equation 49

The cruise dynamic pressure is estimated from half of the product of standard atmosphere pressure at sea level in lb/ft^2 and the specific heat of air, the cruise pressure ratio, and the cruise Mach number using the following equation.

$$QCRUS = 1481.35 \times DELTA \times VCMN^2 \tag{49}$$

where
QCRUS $=$ Cruise dynamic pressure, psf
DELTA $=$ Atmospheric pressure ratio, equal to the ratio of the pressure at cruise altitude to pressure at sea level
VCMN $=$ Cruise Mach number

5.2.3 Vertical Tail

Transport aircraft vertical tail weight is estimated from the design gross weight, taper ratio, number of vertical tails, and tail area using the following equation.

$$WVT = 0.32 \times DG^{0.3} \times (TRVT + 0.5) \times NVERT^{0.7} \times SVT^{0.85} \tag{50}$$

where
WVT	=	Vertical tail weight, lb
DG	=	Design gross weight, lb
TRVT	=	Vertical tail theoretical taper ratio
NVERT	=	Number of vertical tails
SVT	=	Vertical tail theoretical area per tail, ft^2

Fighter/attack aircraft vertical tail weight is estimated from the design gross weight, taper ratio, number of vertical tails, tail area, aspect ratio, and sweep angle using the following equation.

$$WVT = 0.212 \times DG^{0.3} \times (TRVT + 0.5) \times NVERT^{0.7}$$
$$\times SVT^{0.94} \times ARVT^{0.5}/cos(SWPVT)^{1.5} \tag{51}$$

where
ARVT	=	Vertical tail theoretical aspect ratio
SWPVT	=	Vertical tail sweep angle at 25% chord, deg

General aviation aircraft vertical tail weight is estimated using the following equation.

$$WVT = 0.073 \times (1.0 + 0.2 \times HHT) \times (ULF \times DG)^{0.376} \times QCRUS^{0.122}$$
$$\times SVT^{0.873} \times \left(\frac{ARVT}{CSVT^2}\right)^{0.357} / \left(\frac{100.0 \times TCVT}{CSVT}\right)^{0.49} \tag{52}$$

where
HHT	=	Horizontal tail mounting location indicator ranging from 0.0 for body mounted to 1.0 for a T-tail
ULF	=	Structural ultimate load factor. The default value is 3.75.
QCRUS	=	Cruise dynamic pressure, psf, calculated using Equation 49
CSVT	=	Factor defined in Equation 53
TCVT	=	Vertical tail thickness to chord ratio

The factor CSVT is defined with the following equation.

$$CSVT = cos(SWPVT) \tag{53}$$

5.2.4 Fin

The fin weight depends on the design weight of the aircraft, the number of fins, and basic geometric properties of the fins. The fin weight is estimated with the following equation.

$$WFIN = 0.32 \times DG^{0.3} \times SFIN^{0.85} \times (TRFIN + 0.5) \times NFIN \tag{54}$$

where
WFIN   =   Fin weight, lb
DG   =   Design gross weight, lb
SFIN   =   Fin theoretical area, ft^2
TRFIN =   Fin theoretical taper ratio
NFIN   =   Number of fins

5.2.5 Canard

The canard weight depends on the canard area, taper ratio, and the design gross weight of the aircraft. The canard weight is estimated with the following equation.

$$WCAN = 0.53 \times SCAN \times DG^{0.2} \times (TRCAN + 0.5) \tag{55}$$

where
WCAN   =   Canard weight, lb
SCAN   =   Canard theoretical area, ft^2
DG   =   Design gross weight, lb
TRCAN =   Canard theoretical taper ratio

5.2.6 Fuselage

The fuselage weight for a transport aircraft depends on the dimensions of the fuselage, number of fuselage mounted engines, and whether the aircraft is used to transport military cargo. The fuselage weight for transport aircraft is estimated with the following equation.

$$WFUSE = 1.35 \times (XL \times DAV)^{1.28} \times (1 + 0.05 \times FNEF)$$
$$\times (1 + 0.38 \times CARGF) \times NFUSE \tag{56}$$

where
WFUSE =   Fuselage weight, lb
XL   =   Total fuselage length, ft
DAV   =   Average fuselage diameter, ft, estimated with Equation 57
FNEF   =   Number of fuselage mounted engines, scaled to account for distributed propulsion if applicable
CARGF =   Cargo aircraft floor factor, where 0.0 is for passenger transport and 1.0 is for military cargo transport
NFUSE =   Number of fuselages

The average fuselage diameter is estimated with the following equation.

$$DAV = \frac{WF + DF}{2} \tag{57}$$

where

WF = Maximum fuselage width, ft

DF = Maximum fuselage depth, ft

The fuselage weight for HWB aircraft is estimated from the design gross weight and cabin area with the following equation.

$$WFUSE = 1.8 \times DG^{0.167} \times ACABIN^{1.06} \tag{58}$$

where

DG = Design gross weight, lb

ACABIN = Passenger cabin floor area, ft^2

The fuselage weight for fighter/attack aircraft depends on the dimensions of the fuselage, the design gross weight, the number of fuselage mounted engines, and whether or not the aircraft has variable sweep wings. Fighter/attack aircraft fuselage weight is defined using the following equation.

$$WFUSE = 0.15 \times XL^{0.9} \times DG^{0.61}$$
$$\times (1 + 0.3 \times FNEF) \times (1 + 0.33 \times VARSWP) \times NFUSE^{0.3} \tag{59}$$

where

VARSWP = Wing variable sweep weight penalty factor ranging from 0.0 for fixed-geometry wing to 1.0 for full variable-sweep wing

The fuselage weight for general aviation aircraft is a function of the fuselage wetted area, the design gross weight, ultimate load factor, and cruise dynamic pressure. This weight is defined using the following equation.

$$WFUSE = 0.052 \times SWFUS^{1.086} \times (ULF \times DG)^{0.177} \times QCRUS^{0.241} \tag{60}$$

where

SWFUS = Fuselage wetted area ft^2. User input or estimated with Equation 61.

ULF = Structural ultimate load factor. The default value is 3.75.

QCRUS = Cruise dynamic pressure, psf, calculated using Equation 49

Fuselage Wetted Area: The fuselage wetted area can be estimated from the fuselage length and average diameter with the following equation.

$$SFWFUS = 3.14159 \times \left(\frac{XL}{DAV} - 1.7 \right) \times DAV^2 \tag{61}$$

where

XL = Total fuselage length, ft

DAV = Average fuselage diameter, ft, estimated with Equation 57

Fuselage Planform Area: The fuselage planform area is estimated from the maximum fuselage width, fuselage length, and number of fuselages using the following equation.

$$FPAREA = \begin{cases} WF \times \frac{XL+XLW}{2 \times RSPSOB}, & \text{for HWB aircraft} \\ NFUSE \times XL \times WF, & \text{otherwise} \end{cases} \quad (62)$$

where
FPAREA $=$ Fuselage planform area, ft^2
WF $=$ Maximum fuselage width, ft
XLW $=$ Fixed length of side wall, ft. The side wall refers to the outboard wall of the passenger cabin which is also used to define the wing root chord. User input or estimated using Equation 197.
RSPSOB $=$ Percent chord of the HWB fuselage rear spar at the side of body. By default $RSPSOB = RSPCHD$. Where RSPCHD is the percent chord of the HWB fuselage rear spar at the fuselage center line.
NFUSE $=$ Number of fuselages

5.2.7 Landing Gear

The weight of the main and nose landing gear are a function of the landing weight, gear length, the type of aircraft, and whether or not the aircraft is carrier based. If the landing weight or gear length is not input, estimates are computed by FLOPS using the equations provided below.

The weight of the main landing gear is estimated with the following equation.

$$WLGM = (0.0117 - 0.0012 \times DFTE) \times WLDG^{0.95} \times XMLG^{0.43} \quad (63)$$

where
WLGM $=$ Main landing gear weight, lb
DFTE $=$ Aircraft type. Equal to 1.0 for fighter/attack, 0.0 for all others.
WLDG $=$ Aircraft design landing weight, lb. User input or estimated using Equation 65.
XMLG $=$ Length of the extended main landing gear oleo, in. User input or estimated with Equation 66.

The weight of the nose landing gear is estimated with the following equation.

$$WLGN = (0.048 - 0.0080 \times DFTE) \times WLDG^{0.67} \times XNLG^{0.43}$$
$$\times (1 + 0.8 \times CARBAS) \quad (64)$$

where

WLGN   =   Nose landing gear weight, lb

CARBAS   =   Carrier based aircraft switch, where 1.0 is for carrier-based aircraft and 0.0 is for land-based aircraft

XNLG   =   Length of the extended nose landing gear oleo, $in.$ User input or estimated with the method presented below in Equation 67.

The aircraft landing weight (WLDG) can be estimated based on the ramp weight, cruise speed, and the design range using the following equation.

$$WLDG = GW \times (1 - RFACT \times DESRNG) \tag{65}$$

where

WLDG   =   Aircraft design landing weight, lb

GW   =   Ramp weight, lb

RFACT   =   Supersonic cruise factor. Equal to 0.00004 for subsonic cruise, 0.00009 for supersonic cruise.

DESRNG   =   Design range, nmi

The following two equations can be used to estimate the main landing gear length. If the aircraft has wing mounted engines the main landing gear length is a function of the engine nacelle diameter, the wing dihedral angle, the location of the most outboard engine, and the fuselage width. Otherwise, if there are no engines on the wing, the length is estimated based on the fuselage length.

$$XMLG = \begin{cases} 12 \times FNAC + (0.26 - \tan(DIH)) \\ \qquad \times (YEE - 6 \times WF), & \text{engines on wing} \\ 0.75 \times XL, & \text{otherwise} \end{cases} \tag{66}$$

where

XMLG   =   Length of the extended main landing gear oleo, in

FNAC   =   Average diameter of each scaled engine, scaled to account for distributed propulsion if applicable, ft

DIH   =   Wing dihedral angle, deg

YEE   =   Location of outboard engine, measured from centerline, in

WF   =   Maximum fuselage width, ft

XL   =   Total fuselage length, ft

If the nose landing gear length (XNLG) is not input, it is estimated with the following equation.

$$XNLG = 0.7 \times XMLG \tag{67}$$

where

XNLG   =   Length of the extended nose landing gear oleo, in

5.2.8 Paint

The weight of the paint depends on the surface area of the aircraft and the area density of the paint. It is computed with the following equation.

$$WTPNT = WPAINT$$
$$\times (SWTWG + SWTHT + SWTVT + SWTFU + SWTNA + SWTCN) \quad (68)$$

where
WTPNT	=	Weight of aircraft paint, lb
WPAINT	=	Area density of paint for all wetted areas, lb/ft^2
SWTWG	=	Wetted area of wings, ft^2
SWTHT	=	Wetted area of horizontal tail, ft^2
SWTVT	=	Wetted area of vertical tail, ft^2
SWTFU	=	Wetted area of fuselage, ft^2
SWTNA	=	Wetted area of nacelles, ft^2
SWTCN	=	Wetted area of canards, ft^2

5.2.9 Nacelles and Air Induction Systems

Transport aircraft nacelle weight is based on the number of nacelles, engine thrust, and nacelle dimensions. Center-mounted engines have an additional 50% nacelle weight penalty. This is applicable to aircraft such as the Boeing 727 and Lockheed L-1011. Transport aircraft nacelle weight is estimated using the following equation.

$$WNAC = 0.25 \times TNAC \times DNAC \times XNAC \times THRUST^{0.36} \quad (69)$$

where
WNAC	=	Weight of nacelle or air induction system, lb
TNAC	=	Total number of nacelles plus 0.5 if there is a center-mounted engine, calculated with Equation 74
DNAC	=	Average diameter of the scaled engine nacelles, ft. Either calculated by scaling the baseline engine nacelles by $\sqrt{\frac{THRUST}{THRSO}}$ or with Equation 71.
XNAC	=	Average length of the scaled engine nacelles, ft. Either calculated by scaling the baseline engine nacelles by $\sqrt{\frac{THRUST}{THRSO}}$ or with Equation 70.
THRUST	=	Rated thrust of each scaled engine, lb
THRSO	=	Rated thrust of each baseline engine, lb

If XNAC or DNAC are not input by the user, they are estimated with the following equations.

$$XNAC = 0.07 \times \sqrt{THRSO} \quad (70)$$

$$DNAC = 0.04 \times \sqrt{THRSO} \quad (71)$$

Few fighter/attack aircraft have engines located on the wings, most have air induction systems for engines embedded in the fuselage. For fighter/attack aircraft with engines on the wings, the nacelle weight is similar to the transports. For fighter/attack aircraft with engines embedded in the fuselage, the air induction weight is based on the fuselage dimensions, maximum speed, and the specific design. Weight estimates for the air induction systems should only be considered as rough values. Fighter/attack aircraft nacelle and air induction system weight is estimated using the following equation.

$$WNAC = 0.25 \times NEW \times DNAC \times XNAC$$
$$\times THRUST^{0.36} + 1.06 \times (THRUST \times NEF)^{0.23}$$
$$\times (WF + DF)^{1.4} \times VMAX^{0.83} \quad (72)$$

where
NEW	=	Number of wing mounted engines
NEF	=	Number of fuselage mounted engines
WF	=	Maximum fuselage width, ft
DF	=	Maximum fuselage depth, ft
VMAX	=	Maximum Mach number

5.3 Propulsion System Items

The propulsion system items group includes the engines, thrust reversers, miscellaneous propulsion systems, fuel system, tanks, and plumbing, alternate engines, and alternate energy source storage system. This section also includes the calculations for the reference number of engines and adjustments for more than four engines.

Total Number of Engines: The total number of engines is calculated by summing the user specified values using the following equation.

$$NENG = NEW + NEF \quad (73)$$

where
NENG	=	Total number of engines
NEW	=	Number of wing mounted engines
NEF	=	Number of fuselage mounted engines

The factor TNAC is the total number of engine nacelles plus 0.5 if there is a center-mounted engine. TNAC is calculated using the following equation.

$$TNAC = NENG + 0.5 \times \left(NENG - 2 \times \left\lfloor \frac{NENG}{2} \right\rfloor \right) \quad (74)$$

where
$\left\lfloor \frac{NENG}{2} \right\rfloor$	=	Floor function of $\frac{NENG}{2}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor \frac{3}{2} \right\rfloor = 1$, $\left\lfloor \frac{1}{2} \right\rfloor = 0$, $\left\lfloor \frac{6}{2} \right\rfloor = 3$, $\left\lfloor \frac{0}{2} \right\rfloor = 0$

5.3.1 Engines

Engine weight is based on a baseline engine weight that is either user supplied, calculated using the methods presented in Reference 7, or calculated using Equation 76. The inlet and nozzle weights can either be included in the engine weight or calculated separately. All components are scaled with thrust and a scaling parameter. The engine weight is calculated with the following equations.

$$WENGP = \begin{cases} WENGB \times \left(\frac{THRUST}{THRSO}\right)^{EEXP}, & EEXP \geq 0.3 \\ WENGB + (THRUST - THRSO) \times EEXP, & EEXP < 0.3 \end{cases} \tag{75}$$

where
WENGP   =   Intermediate variable used in the calculation of the weight of each engine, lb. If no separate inlet or nozzle weights are provided then it is equal to the engine weight.
WENGB   =   Weight of baseline engine, lb. Includes inlet and nozzle weight if they are not specified separately. User input or estimated using Equation 76.
THRUST   =   Rated thrust per scaled engine, lb
THRSO   =   Rated thrust of each baseline engine, lb
EEXP   =   Engine weight scaling parameter. The default value is 1.15.

If the baseline engine weight (WENGB) is neither input, nor calculated using Reference 7, it is estimated for each aircraft type with the following equation.

$$WENGB = \begin{cases} \frac{THRSO}{5.5}, & \text{Transport and HWB} \\ \frac{THRSO}{8}, & \text{Fighter/Attack} \\ \frac{THRSO}{10.5}, & \text{General Aviation} \end{cases} \tag{76}$$

If the inlet weight is not included in the baseline engine weight (WENGB), it is calculated with following equation.

$$WINL = WINLB \times \left(\frac{THRUST}{THRSO}\right)^{EINL} \tag{77}$$

where
WINL   =   Weight of the engine inlet, lb
WINLB   =   Inlet weight for baseline engine, lb
EINL   =   Engine inlet weight scaling exponent. The default value is 1.

If the nozzle weight is not included in the baseline engine weight (WENGB), it is calculated with following equation.

$$WNOZ = WNOZB \times \left(\frac{THRUST}{THRSO}\right)^{ENOZ} \tag{78}$$

where

WNOZ	=	Weight of the engine nozzle, *lb*
WNOZB	=	Nozzle weight for baseline engine, *lb*
ENOZ	=	Engine nozzle weight scaling exponent. The default value is 1.

If the nozzle weight and inlet weight are specified separately from the baseline engine weight the following equation is used to calculate the weight of each scaled engine.

$$WENG = WENGP + WINL + WNOZ \tag{79}$$

where

WENG   =   Weight of each scaled engine, *lb*

Otherwise if the nozzle weight and inlet weights are included in the baseline engine weight then WENG is simply set equal to WENGP.

$$WENG = WENGP \tag{80}$$

5.3.2   Distributed Propulsion

FLOPS considers aircraft that have more than four engines to have distributed propulsion. It was assumed that technologies would be developed along with the distributed concepts to alleviate the prohibitive weight penalties for the systems. The arctangent function was used to model this trend because it reaches a limit asymptotically. The limit was an assumption based on acceptable penalties for distributed propulsion technology to be considered. The penalties increase the weight estimates for unusable fuel, engine oil, engine nacelle, engine controls and starter, fuel system, instruments, hydraulic system, electrical system, air conditioning system, anti-icing system, fuselage, and landing gear. This is accomplished though scaled variables. The scaling for total number of engines, wing mounted engines, fuselage mounted engines, thrust, and nacelle diameter are estimated using the following equations. Note that the distributed propulsion scaling is separate from the scaling used to resize the propulsion based on a baseline engine.

Scaled Number of Engines: The scaled total number of engines is calculated with the following equation.

$$FNENG = \begin{cases} NENG, & \text{for } NENG \leq 4 \\ 4 + 2.0 \times \arctan\left(\frac{NENG-4.0}{3.0}\right), & \text{for } NENG > 4 \end{cases} \tag{81}$$

where

FNENG	=	Total number of engines, scaled to account for distributed propulsion if applicable
NENG	=	Total number of engines
arctan	=	Arctangent function

If there are four or less engines no scaling is applied. The advantage of this approach is that the number of engines variable (NENG) can be replaced with a new variable, FNENG, that is identical to NENG in aircraft without distributed propulsion and still valid for aircraft with distributed propulsion. The relationship between NENG and FNENG is shown in Figure 3.

The scaled number of the wing mounted engines is calculated with the following equation. As with
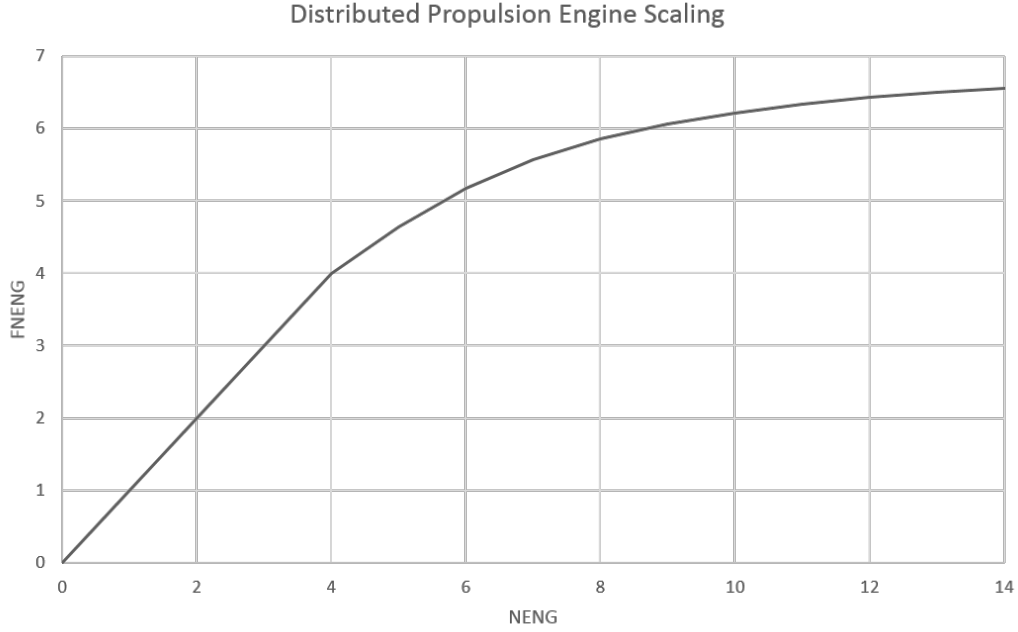
Figure 3. Relationship between the number of engines and the distributed propulsion number of engines.

the total number of engines, no scaling is applied if there are less than four engines.

$$FNEW = \begin{cases} NEW, & \text{for } NEW \leq 4 \\ 4 + 2.0 \times \arctan\left(\frac{NEW-4.0}{3.0}\right), & \text{for } NEW > 4 \end{cases} \tag{82}$$

where

FNEW = Number of wing mounted engines, scaled to account for distributed propulsion if applicable

NEW = Number of wing mounted engines

The scaled number of fuselage mounted engines is calculated with the following equation. As with the total number of engines, no scaling is applied if there are less than four engines.

$$FNEF = \begin{cases} NEF, & \text{for } NEF \leq 4 \\ 4 + 2.0 \times \arctan\left(\frac{NEF-4.0}{3.0}\right), & \text{for } NEF > 4 \end{cases} \tag{83}$$

where

FNEF = Number of fuselage mounted engines, scaled to account for distributed propulsion if applicable

NEF = Number of fuselage mounted engines

Scaled Thrust: The distributed propulsion thrust is calculated from the number of engines, the rated thrust per engine, and the scaled number of engines.

$$FTHRST = \begin{cases} THRUST, & \text{for } NENG \leq 4 \\ \frac{NENG \times THRUST}{FNENG}, & \text{for } NENG > 4 \end{cases} \tag{84}$$

32

where

FTHRST   =   Rated thrust of each scaled engine, scaled to account
for distributed propulsion if applicable, *lb*

THRUST   =   Rated thrust of each scaled engine, *lb*

Scaled Nacelle Diameter: The distributed propulsion scaled nacelle diameter is calculated from the scaled engine nacelle diameter and number of engines with the following equation. FNAC is only used in starter weights, anti-icing, and landing gear lengths for tip over. When it is used for the starter and anti-icing weights, it is a complexity factor to bring those systems up to a reasonable value. The same factor was used to increase the impact of distributed propulsion on the landing gear length which otherwise would be negligible.

$$FNAC = \begin{cases} DNAC, & \text{for } NENG \leq 4 \\ \frac{DNAC \times \sqrt{NENG}}{2.0}, & \text{for } NENG > 4 \end{cases} \tag{85}$$

where

FNAC   =   Average diameter of each scaled engine, scaled to account for distributed propulsion if applicable, *ft*

DNAC   =   Average diameter of the scaled engine nacelles, *ft*. Either calculated by scaling the baseline engine nacelles by $\sqrt{\frac{THRUST}{THRSO}}$ or with Equation 71.

5.3.3   Thrust Reversers

Thrust reverser weight is based on the engine thrust, the number of engines, and the type and design of the thrust reverser. Center-mounted engines have an additional 50% weight penalty. Weight estimates for the thrust reversers should only be considered as rough values because there are many variations. The thrust reverser weight is estimated using the following equation.

$$WTHR = 0.034 \times THRUST \times TNAC \tag{86}$$

where

WTHR   =   Weight of the thrust reversers, *lb*

THRUST   =   Rated thrust of each scaled engine, *lb*

TNAC   =   Total number of nacelles plus 0.5 if there is a center-mounted engine, calculated using Equation 74

5.3.4   Miscellaneous Propulsion Systems

The miscellaneous propulsion systems weight is composed of engine controls, starters, and any miscellaneous weight input by the user.

Engine Controls: The transport aircraft engine controls weight is estimated from the number of engines and engine thrust using the following equation.

$$WEC = 0.26 \times FNENG \times FTHRUST^{0.5} \tag{87}$$

where

WEC $\quad$ = $\quad$ Weight of the engine controls system, lb

FNENG $\quad$ = $\quad$ Total number of engines, scaled to account for distributed propulsion if applicable

FTHRUST $\quad$ = $\quad$ Rated thrust of each scaled engine, including adjustment for distributed propulsion if applicable, lb

The fighter/attack aircraft engine controls weight is estimated from the number of engines, scaled thrust per engine, and the number of flight crew using the following equation.

$$WEC = 0.106 \times (NENG \times THRUST \times NFLCR)^{0.55} \tag{88}$$

where

NENG $\quad$ = $\quad$ Total number of engines

THRUST $\quad$ = $\quad$ Rated thrust of each scaled engine, lb

NFLCR $\quad$ = $\quad$ Number of flight crew. For fighter/attack aircraft it is assumed to be 1.

Engine Starters: The transport aircraft engine starters weight is estimated from the maximum Mach number, scaled number of engines, and scaled engine nacelle diameter using the following equation.

$$WSTART = 11.0 \times FNENG \times VMAX^{0.32} \times FNAC^{1.60} \tag{89}$$

where

WSTART $\quad$ = $\quad$ Weight of the engine starter system, lb

VMAX $\quad$ = $\quad$ Maximum Mach number

FNAC $\quad$ = $\quad$ Average diameter of each scaled engine, scaled to account for distributed propulsion if applicable, ft

The fighter/attack aircraft engine starter weight is estimated from the thrust per engine and number of engines using the following equation.

$$WSTART = 0.0072 \times THRUST \times NENG \tag{90}$$

Miscellaneous Propulsion System Weight: The total miscellaneous propulsion systems weight is estimated by summing engine controls weight, starter weight, and user provided miscellaneous propulsion system weight using the following equation.

$$WPMSC = WEC + WSTART + WPMISC \tag{91}$$

where

WPMSC $\quad$ = $\quad$ Weight of miscellaneous propulsion systems such as engine controls, starter, and electrical, lb

WPMISC $\quad$ = $\quad$ Additional miscellaneous propulsion system weight, lb

5.3.5 Fuel System, Tanks, and Plumbing

The fuel system weight (including the fuel tanks and plumbing) is calculated from the number of tanks, fuel capacity, maximum speed, number of engines and aircraft type. For transport aircraft the fuel system weight is calculated with the following equation.

$$WFSYS = 1.07 \times FMXTOT^{0.58} \times FNENG^{0.43} \times VMAX^{0.34} \tag{92}$$

where

WFSYS   =   Weight of the fuel system (including tanks and plumbing), *lb*

FMXTOT   =   Aircraft maximum fuel capacity, *lb*. Includes wing, fuselage and auxiliary tanks.

FNENG   =   Total number of engines, scaled to account for distributed propulsion if applicable

VMAX   =   Maximum Mach number

For fighter/attack aircraft the fuel system weight is calculated with the following equation.

$$WFSYS = 36 \times FMXTOT^{0.2} \times NTANK^{0.5} \times FNENG^{0.4} \tag{93}$$

where

NTANK   =   Number of fuel tanks

For general aviation aircraft the fuel system weight is calculated with the following equation.

$$WFSYS = 1.07 \times FMXTOT^{0.58} \times FNENG^{0.43} \tag{94}$$

5.3.6   Alternate Engines

A second set of engine data can be used in FLOPS. The alternate engines weight is estimated using the following equation.

$$AEWT = NEALT \times ALTEWT \times ALTTHR \tag{95}$$

where

AEWT   =   Weight of alternate engines, *lb*

NEALT   =   Number of engines for the alternate propulsion system

ALTEWT   =   Weight-to-thrust ratio per engine for the alternate propulsion system, *lb*

ALTTHR   =   Rated thrust per engine for the alternate propulsion system, *lb*

5.3.7   Alternate Energy Source Storage System

Alternate energy source storage systems can be used in FLOPS. The alternate energy source storage system weight is estimated using the energy density of the storage system and the required energy system capacity using the following equation. The unit used for energy is user specified, but must be consistent for both BATWT and POWMAX.

$$WTBAT = BATWT \times POWMAX \tag{96}$$

where

WTBAT   =   Weight of alternate energy source storage system, *lb*

BATWT   =   Storage system density for the alternate energy source, *lb/energy*. Energy unit is user specified and must be consistent with POWMAX.

POWMAX   =   Storage system capacity for alternate energy source, *energy*. Energy unit is user specified and must be consistent with BATWT.

5.4 Systems and Equipment Items

The systems and equipment items group includes the surface controls, auxiliary power unit, instruments, hydraulics, electrical, avionics, furnishings and equipment, air conditioning, and anti-icing systems.

5.4.1 Surface Controls

The surface controls, often called flight controls, includes the items involved in controling the surface controls. These items include actuators, hydraulic and/or electrical lines, cockpit controls, etc. For transport aircraft the surface controls weight is a function of the maximum Mach number, wing control surface area, and design gross weight. Surface controls weight is estimated with the following equation.

$$WSC = 1.1 \times VMAX^{0.52} \times SFLAP^{0.6} \times DG^{0.32} \tag{97}$$

where

WSC	=	Weight of the surface control systems, lb
VMAX	=	Maximum Mach number
SFLAP	=	Total movable wing surface area including flaps, elevators, spoilers, etc., ft^2
DG	=	Design gross weight, lb

For fighter/attack aircraft the surface controls weight no longer depends on the maximum Mach number. It is estimated with the following equation.

$$WSC = 2.95 \times SFLAP^{0.45} \times DG^{0.36} \tag{98}$$

The surface controls weight equation for general aviation aircraft was obtained from the General Aviation Synthesis Program (GASP) and therefore has a different format from the transport and fighter/attack equations. [8]

$$WSC = 0.404 \times SW^{0.317} \times \left(\frac{DG}{1000}\right)^{0.602} \times ULF^{0.525} \times QDIVE^{0.345} \tag{99}$$

where

SW	=	Reference wing area, ft^2, estimated using Equation 4
ULF	=	Structural ultimate load factor. The default value is 3.75.
QDIVE	=	Dive maneuver dynamic pressure, psf, estimated using Equation 100

The dynamic pressure in a dive maneuver is estimated using the following equation.

$$QDIVE = 1481.35 \times DELTA \times VMAX^2 \tag{100}$$

where

QDIVE	=	Dive maneuver dynamic pressure, psf
DELTA	=	Atmospheric pressure ratio, equal to the ratio of the pressure at cruise altitude to pressure at sea level
VMAX	=	Maximum Mach number

5.4.2 Auxiliary Power Unit

The auxiliary power unit weight for transport aircraft varies with the number of passengers and the fuselage dimensions. It is estimated with the following equation.

$$WAPU = 54 \times FPAREA^{0.3} + 5.4 \times NPASS^{0.9} \tag{101}$$

where

WAPU	=	Weight of the auxiliary power unit, lb
FPAREA	=	Fuselage planform area, ft^2, estimated using Equation 62
NPASS	=	Total number of passengers

5.4.3 Instruments

The instrument weight varies with the fuselage planform area, maximum speed, number of engines, and the number of flight crew. The following equation is used to estimate the instrument weight for transport and general aviation aircraft.

$$WIN = 0.48 \times FPAREA^{0.57} \times VMAX^{0.5}$$
$$\times (10 + 2.5 \times NFLCR + FNEW + 1.5 \times FNEF) \tag{102}$$

where

WIN	=	Weight of the instruments system group, lb
FPAREA	=	Fuselage planform area, ft^2, estimated using Equation 62
VMAX	=	Maximum Mach number
NFLCR	=	Number of flight crew. User input for transport aircraft or estimated using Equation 118 and for fighter/attack aircraft it is assumed to be 1.
FNEW	=	Number of wing mounted engines, scaled to account for distributed propulsion if applicable
FNEF	=	Number of fuselage mounted engines, scaled to account for distributed propulsion if applicable

The following equation is used for fighter/attack aircraft.

$$WIN = 0.09 \times NFUSE \times XL \times DF$$
$$\times (1 + 2.5 \times NFLCR + 0.1 \times FNEW + 0.15 \times FNEF) \tag{103}$$

where

NFUSE	=	Number of fuselages
XL	=	Total fuselage length, ft
DF	=	Maximum fuselage depth, ft

5.4.4 Hydraulics

The hydraulic system weight is a function of the wing area, number of engines, hydraulic system pressure, whether or not the aircraft has variable sweep wings, and the maximum speed of the

aircraft. The hydraulic system weight for transport and general aviation aircraft is estimated with the following equation.

$$WHYD = 0.57 \times (FPAREA + 0.27 \times SW) \times (1 + 0.03 \times FNEW + 0.05 \times FNEF)$$
$$\times \left(\frac{3000}{HYDPR}\right)^{0.35} \times (1 + 0.04 \times VARSWP) \times VMAX^{0.33} \tag{104}$$

where

WHYD	=	Weight of the hydraulic system group, lb
FPAREA	=	Fuselage planform area, ft^2, estimated using Equation 62
SW	=	Reference wing area, ft^2, estimated using Equation 4
FNEW	=	Number of wing mounted engines, scaled to account for distributed propulsion if applicable
FNEF	=	Number of fuselage mounted engines, scaled to account for distributed propulsion if applicable
HYDPR	=	Hydraulic system pressure, psi. The default value is 3,000.
VARSWP	=	Wing variable sweep weight penalty factor ranging from 0.0 for fixed-geometry wing to 1.0 for full variable-sweep wing
VMAX	=	Maximum Mach number

The hydraulic system weight for fighter/attack aircraft is estimated with the following equation.

$$WHYD = 0.55 \times (FPAREA + 0.27 \times SW)$$
$$\times (1 + 0.03 \times FNEW + 0.05 \times FNEF) \times \left(\frac{3000}{HYDPR}\right)^{0.35}$$
$$\times (1 + 0.04 \times VARSWP) \times VMAX^{0.01} \tag{105}$$

5.4.5 Electrical

For the transport and general aviation aircraft the electrical system weight was found to be a function of fuselage dimensions, the number of passengers and crew, and the number of engines. This weight is estimated with the following equation.

$$WELEC = 92 \times XL^{0.4} \times WF^{0.14} \times NFUSE^{0.27} \times FNENG^{0.69}$$
$$\times (1 + 0.044 \times NFLCR + 0.0015 \times NPASS) \tag{106}$$

where

WELEC	=	Weight of the electrical system group, lb
XL	=	Total fuselage length, ft
WF	=	Maximum fuselage width, ft
NFUSE	=	Number of fuselages

FNENG	=	Total number of engines, scaled to account for distributed propulsion if applicable
NFLCR	=	Number of flight crew. User input or estimated using Equation 118 for transport aircraft and assumed to be 1 for fighter/attack aircraft.
NPASS	=	Total number of passengers

For fighter/attack aircraft the electrical system weight is a function of basic geometry of the aircraft, the number of crew, and the maximum velocity. This weight is estimated with the following equation.

$$WELEC = 10 \times (XL + B)^{0.85} \times NFUSE^{0.27} \times VMAX^{0.1} \times (1 + 0.1 \times NFLCR) \qquad (107)$$

5.4.6 Avionics

For transport and general aviation aircraft the avionics weight varies with the design range, number of flight crew, and fuselage planform area.

$$WAVONC = 15.8 \times DESRNG^{0.1} \times NFLCR^{0.7} \times FPAREA^{0.43} \qquad (108)$$

where

WAVONC	=	Weight of the avionics system group, lb
DESRNG	=	Design range, nmi
NFLCR	=	Number of flight crew. User input or estimated using Equation 118 for transport aircraft and assumed to be 1 for fighter/attack aircraft.
FPAREA	=	Fuselage planform area, ft^2, estimated using Equation 62

For fighter/attack aircraft the avionics weight varies with basic fuselage dimensions, whether or not the aircraft is used on aircraft carriers and the maximum Mach number.

$$WAVONC = 0.41 \times (NFUSE \times XL \times DF)^{1.3}$$
$$\times (1 + 0.37 \times CARBAS) \times VMAX \qquad (109)$$

where

NFUSE	=	Number of fuselages
XL	=	Total fuselage length, ft
DF	=	Maximum fuselage depth, ft
CARBAS	=	Carrier based aircraft switch, where 1.0 is for carrier-based aircraft and 0.0 is for land-based aircraft
VMAX	=	Maximum Mach number

5.4.7 Furnishings and Equipment

The furnishings and equipment weight varies with the geometry of the cabin, use of the cabin, number of passengers, type of passengers, number of crew, and the maximum Mach number of the aircraft. Separate equations are provided for transport and general aviation aircraft, fighter/attack

aircraft, and HWB aircraft.

The furnishings and equipment weight for transport and general aviation aircraft is estimated with the following equation.

$$
\begin{aligned}
WFURN = {} & 127 \times NFLCR + 112 \times NPF + 78 \times NPB + 44 \times NPT \\
& + 2.6 \times XLP \times (WF + DF) \times NFUSE
\end{aligned}
\tag{110}
$$

where
WFURN   =   Weight of the furnishings group, lb
NFLCR   =   Number of flight crew. User input or estimated using
                  Equation 118 for transport aircraft and assumed to
                  be 1 for fighter/attack aircraft.
NPF   =   Number of first class passengers
NPB   =   Number of business class passengers
NPT   =   Number of tourist class passengers
XLP   =   Length of passenger compartment, ft
NFUSE   =   Number of fuselages

The furnishings and equipment weight for fighter/attack aircraft is estimated with the following equation.

$$
WFURN = 80 \times NFLCR \times VMAX^{0.38} \times XL^{0.25}
\tag{111}
$$

where
VMAX   =   Maximum Mach number
XL   =   Total fuselage length, ft

The furnishings and equipment weight for HWB aircraft is estimated with the following equation.

$$
\begin{aligned}
WFURN = {} & 127 \times NFLCR + 112 \times NPF + 78 \times NPB \\
& + 44 \times NPT + 2.6 \times (1 - CARGF) \\
& \times \left[\frac{ACABIN \times (WF + DF \times NBAY)}{WF} \right. \\
& \left. + WF \times DF \times \left(1 + \frac{1}{\cos(SWPLE)} \right) \right]
\end{aligned}
\tag{112}
$$

where
CARGF   =   Cargo aircraft floor factor, where 0.0 is for passenger
                  transport and 1.0 is for military cargo transport
ACABIN   =   Passenger cabin floor area, ft^2
WF   =   Maximum fuselage width, ft
DF   =   Maximum fuselage depth, ft
NBAY   =   Number of passenger bays
SWPLE   =   Sweep angle of the passenger cabin, deg

5.4.8 Air Conditioning

For transport and general aviation aircraft the air conditioning system weight varies with the dimensions of the fuselage, maximum Mach number, and avionics system weight as is shown in the

following equation.

$$WAC = \left(3.2 \times (FPAREA \times DF)^{0.6} + 9 \times NPASS^{0.83}\right)$$
$$\times VMAX + 0.075 \times WAVONC \quad (113)$$

where

WAC	=	Weight of the air conditioning system group, lb
FPAREA	=	Fuselage planform area, ft^2, estimated using Equation 62
DF	=	Maximum fuselage depth, ft
NPASS	=	Total number of passengers
VMAX	=	Maximum Mach number
WAVONC	=	Weight of the avionics system group, lb

For fighter/attack aircraft the weight varies with the avionics system weight, number of engines, thrust per engine, and maximum Mach number as is shown in the following equation.

$$WAC = 0.075 \times WAVONC + 0.37 \times FNENG \times FTHRST^{0.6} \times VMAX^{0.57} \quad (114)$$

where

FNENG	=	Total number of engines, scaled to account for distributed propulsion if applicable
FTHRST	=	Rated thrust of each scaled engine, scaled to account for distributed propulsion if applicable, lb

5.4.9 Anti-Icing

For transport aircraft the anti-icing system weight varies with wingspan, sweep, number of engines, engine dimensions, and fuselage width. This weight is estimated with the following equation. For fighter/attack aircraft, this variable is a user input used for the auxiliary gear system.

$$WAI = \frac{B}{\cos{(SWEEP)}} + 3.8 \times FNAC \times FNENG + 1.5 \times WF \quad (115)$$

where

WAI	=	Weight of the anti-icing system for transport aircraft, lb
B	=	Wing span, ft
SWEEP	=	Quarter chord sweep angle of the wing, deg
FNAC	=	Average diameter of each scaled engine, scaled to account for distributed propulsion if applicable, ft
FNENG	=	Total number of engines, scaled to account for distributed propulsion if applicable
WF	=	Maximum fuselage width, ft

5.5 Operating Items

The operating items include the crew and baggage, unusable fuel, engine oil, passenger service, ammunition and non-fixed weapons, and cargo containers.

5.5.1 Crew and Baggage

The weight of the crew and baggage is based on Federal Aviation Administration (FAA) regulations. The crew includes flight attendants, galley crew, and flight crew.

Number of Crew: The default values for the number of crew are calculated using the following equations.

The number of flight attendants is calculated using the following equation.

$$
NSTU = \begin{cases} 0, & \text{for } NPASS \leq 0 \\ 1, & \text{for } 0 < NPASS < 51 \\ 1 + \left\lceil \frac{NPASS}{40} \right\rceil, & \text{for } NPASS \geq 51 \end{cases} \tag{116}
$$

where
NSTU $\quad=\quad$ Number of flight attendants
NPASS $\quad=\quad$ Number of passengers
$\left\lceil \frac{NPASS}{40} \right\rceil \quad=\quad$ Ceiling function of $\frac{NPASS}{40}$. Equal to the nearest integer rounded up. Examples: $\left\lceil \frac{1}{40} \right\rceil = 1$, $\left\lceil \frac{40}{40} \right\rceil = 1$, $\left\lceil \frac{150}{40} \right\rceil = 4$

The number of galley crew is calculated using the following equation.

$$
NGALC = \begin{cases} 0, & \text{for } NPASS < 151 \\ 1 + \left\lceil \frac{NPASS}{250} \right\rceil, & \text{for } NPASS \geq 151 \end{cases} \tag{117}
$$

where
NGALC $\quad=\quad$ Number of galley crew
$\left\lceil \frac{NPASS}{250} \right\rceil \quad=\quad$ Ceiling function of $\frac{NPASS}{250}$. Equal to the nearest integer rounded up. Examples: $\left\lceil \frac{150}{250} \right\rceil = 1$, $\left\lceil \frac{250}{250} \right\rceil = 1$, $\left\lceil \frac{251}{250} \right\rceil = 2$

The number of flight crew is calculated using the following equation.

$$
NFLCR = \begin{cases} 2, & \text{for transport and HWB aircraft with } NPASS < 151 \\ 3, & \text{for transport and HWB aircraft with } NPASS \geq 150 \\ 1, & \text{for fighter/attack and general aviation aircraft} \end{cases} \tag{118}
$$

where
NFLCR $\quad=\quad$ Number of flight crew

Crew Weight: The crew weight is calculated using the following equations.

The flight attendant and galley crew and baggage weight is calculated using the following equation.

$$WSTUAB = NSTU \times 155 + NGALC \times 200 \tag{119}$$

> where
> WSTUAB $=$ Weight of flight attendants and galley crew and baggage, *lb*

The flight crew and baggage weight is calculated using the following equation.

$$WFLCRB = \begin{cases} NFLCR \times (215 - 35 \times CARBAS), & \text{for fighter/attack aircraft} \\ NFLCR \times (225 - 35 \times CARBAS), & \text{otherwise} \end{cases} \tag{120}$$

> where
> WFLCRB $=$ Weight of flight crew and baggage, *lb*
> CARBAS $=$ Carrier based aircraft switch, where 1.0 is for carrier-based aircraft and 0.0 is for land-based aircraft

5.5.2 Unusable Fuel

For transport and general aviation aircraft the weight of the unusable fuel is a function of the number of engines, thrust per engine, wing area, total fuel weight, and number of tanks. The weight of the unusable fuel for transport aircraft is estimated with the following equation.

$$WUF = 11.5 \times FNENG \times FTHRST^{0.2} + 0.07 \times SW$$
$$+ 1.6 \times NTANK \times FMXTOT^{0.28} \tag{121}$$

> where
> WUF $=$ Weight of unusable fuel, *lb*
> FNENG $=$ Total number of engines, scaled to account for distributed propulsion if applicable
> FTHRST $=$ Rated thrust of each scaled engine, scaled to account for distributed propulsion if applicable, *lb*
> SW $=$ Reference wing area, ft^2, estimated using Equation 4
> NTANK $=$ Number of fuel tanks
> FMXTOT $=$ Aircraft maximum fuel capacity, *lb*. Includes wing, fuselage and auxiliary tanks.

For fighter/attack aircraft the weight of the unusable fuel is a function of only the number of engines, thrust per engine, and wing area. The weight of the unusable fuel for fighter/attack aircraft is estimated with the following equation.

$$WUF = 11.5 \times FNENG \times FTHRST^{0.2} + 0.04 \times SW \tag{122}$$

5.5.3 Engine Oil

The engine oil weight depends on the number of engines and thrust per engine. It is estimated with the following equation for all types of aircraft.

$$WOIL = 0.082 \times FNENG \times FTHRST^{0.65} \tag{123}$$

where

WOIL	=	Weight of the engine oil, *lb*
FNENG	=	Total number of engines, scaled to account for distributed propulsion if applicable
FTHRST	=	Rated thrust of each scaled engine, scaled to account for distributed propulsion if applicable, *lb*

5.5.4 Passenger Service

The weight reserved for passenger service for transport aircraft is estimated from the number and type of passengers, the design range of the aircraft, and the maximum Mach number. The weight reserved for passenger service is estimated with the following equation. For fighter/attack aircraft, this variable is a user input used for ammunition and non-fixed weapons.

$$WSRV = (5.164 \times NPF + 3.846 \times NPB + 2.529 \times NPT) \times \left(\frac{DESRNG}{VMAX}\right)^{0.225} \tag{124}$$

where

WSRV	=	Weight of passenger service for transport aircraft, *lb*
NPF	=	Number of first class passengers
NPB	=	Number of business class passengers
NPT	=	Number of tourist class passengers
DESRNG	=	Design range, *nmi*
VMAX	=	Maximum Mach number

5.5.5 Cargo Containers

If cargo containers are used on transport aircraft the following equations are used to estimate their weight. Each cargo container is assumed to weigh 175 *lb*. One cargo container is assumed for every 950 *lb* of cargo placed in the containers.

$$WCON = 175 \times NCON \tag{125}$$

$$NCON = \left\lceil \frac{WCARGO}{950} \right\rceil \tag{126}$$

where

WCON	=	Weight of cargo containers, *lb*
NCON	=	Number of cargo containers
WCARGO	=	Weight of cargo that will be placed in containers, *lb*
$\lceil \frac{WCARGO}{950} \rceil$	=	Ceiling function of $\frac{WCARGO}{950}$. Equal to the nearest integer rounded up. Examples: $\lceil \frac{0}{950} \rceil = 0$, $\lceil \frac{950}{950} \rceil = 1$, $\lceil \frac{2000}{950} \rceil = 3$

5.6 Payload Items

The payload items group includes passengers, passenger baggage, fighter/attack aircraft auxiliary fuel tanks, and cargo.

5.6.1 Passengers

The total number of passengers is calculated by summing the different types of passengers using the following equation.

$$NPASS = NPF + NPB + NPT \tag{127}$$

where
NPASS   =   Total number of passengers
NPF   =   Number of first class passengers
NPB   =   Number of business class passengers
NPT   =   Number of tourist class passengers

The total passenger weight is calculated with the following equation.

$$WPASS = NPASS \times WPPASS \tag{128}$$

where
WPASS   =   Total passenger weight, lb
NPASS   =   Number of passengers
WPPASS   =   Weight per passenger, lb. The default value is 165 lb.

5.6.2 Passenger Baggage

The weight of passenger baggage is a function of the number of passengers and baggage weight per passenger. It is calculated with the following equation. For fighter/attack aircraft, this variable is a user input used for auxiliary fuel tanks.

$$WPBAG = BPP \times NPASS \tag{129}$$

where
WPBAG   =   Weight of passenger baggage for transport aircraft, lb
BPP   =   Weight of baggage per passenger, lb. User input or
            estimated using Equation 130.

The default weight of baggage per passenger is a function of the design range of the aircraft. It is calculated with the following equation.

$$BPP = \begin{cases} 35, & \text{for } 1 \leq DESRNG \leq 900 \\ 40, & \text{for } 900 < DESRNG \leq 2900 \\ 44, & \text{for } 2900 < DESRNG \end{cases} \tag{130}$$

where
BPP   =   Weight of baggage per passenger, lb
DESRNG   =   Design range, nmi

5.6.3 Fighter/Attack Aircraft Auxiliary Fuel Tanks

FLOPS estimates auxiliary tank weight for the fuselage of fighter/attack aircraft using the following equation.

$$WPBAG = 1.15 \times FULAUX \tag{131}$$

where

WPBAG   =   Weight of auxiliary fuel tanks for fighter/attack aircraft, *lb*

FULAUX   =   Auxiliary (external) fuel tank capacity, *lb*. This is used for fighter/attack aircraft only.

5.6.4 Cargo

The total cargo is calculated by summing the cargo in the wing and fuselage using the following equation.

$$WCARGO = CARGOW + CARGOF \qquad (132)$$

where

WCARGO   =   Total weight of the cargo, *lb*

CARGOW   =   Cargo carried in wing including the weight of wing-mounted external stores for fighter/attack aircraft, *lb*

CARGOF   =   Cargo other than passenger baggage carried in the fuselage including fuselage external stores for fighter/attack aircraft, *lb*

5.7 Fuel Capacity

FLOPS allows fuel to be stored in the wings, fuselage, and external auxiliary tanks. The maximum fuel capacity of the wing is estimated using geometric parameters if it is not input by the user. It will also be the default total aircraft fuel capacity if the total capacity is not input by the user. When the total aircraft fuel capacity is input by the user, the fuselage fuel capacity is the difference of the total and the wing fuel capacities. The auxiliary fuel capacity is calculated if the total aircraft and fuselage fuel capacities are input by the user and all of the fuel cannot be stored in the fuselage and wings. The maximum fuel capacity of the wing is scaled when the wing area, thickness to chord ratio, aspect ratio, or taper ratio is varied or optimized. The maximum fuel capacity of the wing is either input by the user or estimated using the following equation.

$$FULWMX = FULDEN \times FWMAX \times SW^2 \times TCA \times \left(1.0 - \frac{TR}{(1.0 + TR)^2}\right)/SPAN \quad (133)$$

where

FULWMX   =   Total fuel capacity of the wing, *lb*

FULDEN   =   Fuel density ratio for alternate fuels compared to jet fuel (typical density of 6.7 *lb/gal*). The default value is 1.

FWMAX   =   Factor for wing fuel capacity equation. The default value is 23.

SW   =   Reference wing area, ft^2, estimated using Equation 4

TCA   =   Weighted average of the wing thickness to chord ratio

TR   =   Taper ratio of the wing

SPAN   =   Wing span, ft

The maximum fuel capacity of the fuselage can be input by the user or varied to meet the required fuel capacity for the primary mission. It includes the wing carry-through structure and/or fuel

tanks in the fuselage. If the aircraft geometry changes during sizing the following equation is used to recalculate the maximum fuel capacity of the fuselage.

$$FUFU = FUELM - FULWMX - FULAUX \tag{134}$$

where
FUFU	=	Maximum fuel capacity of the fuselage, *lb*
FUELM	=	Total aircraft fuel weight, *lb*
FULAUX	=	Auxiliary (external) fuel tank capacity, *lb*. This is used for fighter/attack aircraft only.

The maximum fuel capacity is either input by the user or estimated using the following equation.

$$FMXTOT = FULWMX + FULFMX + FULAUX \tag{135}$$

where
FMXTOT	=	Aircraft maximum fuel capacity, *lb*. Includes wing, fuselage and auxiliary tanks.
FULFMX	=	Total fuel capacity of the fuselage, *lb*. Includes wing carry through structure and fuselage tanks.

6 Aircraft Weight Build-up

The component weights are used to estimate the total weight of the aircraft. This section shows how the weights are grouped together and summed to estimate the total aircraft weight.

6.1 Empty Weight

The aircraft empty weight is the summation of the structural, propulsion, systems and equipment weights, and empty weight margin.

6.1.1 Structural Weight

The total structural weight is the summation of the wing, horizontal tail, vertical tail, fin, canard, fuselage, landing gear, paint, and nacelles and air induction systems weights. It is calculated using the following equation.

$$WSTRCT = WWING + WHT + WVT + WCAN + WFUSE$$
$$+ WLGM + WLGN + WFIN + WNAC + WTPNT \tag{136}$$

where
WSTRCT	=	Total aircraft structural group weight, *lb*
WWING	=	Wing weight, *lb*
WHT	=	Horizontal tail weight, *lb*
WVT	=	Vertical tail weight, *lb*
WCAN	=	Canard weight, *lb*
WFUSE	=	Fuselage weight, *lb*
WLGM	=	Main landing gear weight, *lb*

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WLGN	=	Nose landing gear weight, *lb*
WFIN	=	Fin weight, *lb*
WNAC	=	Weight of nacelle or air induction system, *lb*
WTPNT	=	Weight of aircraft paint, *lb*

6.1.2 Propulsion Weight

The total propulsion weight is the summation of the engines, thrust reversers, miscellaneous propulsion systems, fuel system, tanks, and plumbing, alternate engines, and alternate energy source storage system weights. Note that the alternate energy source storage system weights are input by the user. The total propulsion weight is calculated using the following equation.

$$WPRO = WENG \times NENG + WTHR + WPMSC + WFSYS$$
$$+ AEWT + POWWT + WTBAT \quad (137)$$

where
WPRO	=	Total aircraft propulsion group weight, *lb*
WENG	=	Weight of each scaled baseline engine, *lb*
NENG	=	Total number of engines
WTHR	=	Weight of the thrust reversers, *lb*
WPMSC	=	Weight of miscellaneous propulsion systems such as engine controls, starter, and electrical, *lb*
WFSYS	=	Weight of the fuel system (including tanks and plumbing), *lb*
AEWT	=	Weight of alternate engines, *lb*
POWWT	=	System weight for alternate propulsion system, *lb*
WTBAT	=	Weight of alternate energy source storage system, *lb*

6.1.3 Systems and Equipment Weight

The total systems and equipment weight is the summation of the surface controls, auxiliary power unit, instruments, hydraulics, electrical, avionics, armament, furnishings and equipment, air conditioning, and anti-icing weights. Note that the armament group weight is input by the user. The total systems and equipment weight is calculated using the following equation.

$$WSYS = WSC + WAPU + WIN + WHYD + WELEC$$
$$+ WAVONC + WARM + WFURN + WAC + WAI \quad (138)$$

where
WSYS	=	Total aircraft systems and equipment group weight, *lb*
WAPU	=	Weight of the auxiliary power unit, *lb*
WIN	=	Weight of the instruments system group, *lb*
WHYD	=	Weight of the hydraulic system group, *lb*

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WELEC	=	Weight of the electrical system group, *lb*
WAVONC	=	Weight of the avionics system group, *lb*
WARM	=	Weight of the armament group, *lb* (Includes thermal protection system or armor and fixed weapons)
WFURN	=	Weight of the furnishings group, *lb*
WAC	=	Weight of the air conditioning system group, *lb*
WAI	=	Weight of the anti-icing system for transport aircraft, *lb* (For fighter/attack aircraft this variable is a user input used for auxiliary gear system)

6.1.4 Empty Weight

The aircraft empty weight is the summation of the structural, propulsion, systems and equipment weights, and empty weight margin. It is calculated using the following equation.

$$WWE = WSTRCT + WPRO + WSYS + WMARG \tag{139}$$

where
| WWE | = | Aircraft empty weight, *lb* |
| WMARG | = | Empty weight margin, *lb*. Defined below. |

Empty Weight Margin Method: A special option exists to add an empty weight margin ($WMARG$). The empty weight margin is specified either as a delta weight or as a fraction of the calculated empty weight that is converted to a delta weight and then added to the calculated aircraft empty weight.

6.2 Operating Empty Weight

The aircraft operating empty weight is the summation of the aircraft empty and operating items weights.

6.2.1 Operating Items Weight

The total operating items weight is the summation of the crew and baggage, unusable fuel, engine oil, passenger service, and cargo containers weights. It is calculated using the following equation.

$$WOPIT = WFLCRB + WSTUAB + WUF + WOIL + WSRV + WCON \tag{140}$$

where
WOPIT	=	Total operating items weight, *lb*
WFLCRB	=	Weight of flight crew and baggage, *lb*
WSTUAB	=	Weight of flight attendants and galley crew and baggage, *lb*
WUF	=	Weight of unusable fuel, *lb*
WOIL	=	Weight of the engine oil, *lb*

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| WSRV | = | Weight of passenger service for transport aircraft, *lb*. (For fighter/attack aircraft this variable is a user input used for ammunition and non-fixed weapons) |
| WCON | = | Weight of cargo containers, *lb* |

6.2.2 Operating Empty Weight

The aircraft operating empty weight is the summation of the aircraft empty and operating items weights. It is calculated using the following equation.

$$WOWE = WWE + WOPIT \tag{141}$$

where
WOWE = Aircraft operating empty weight, *lb*

6.3 Zero Fuel Weight

The aircraft zero fuel weight is the summation of the aircraft operating empty and payload weights.

6.3.1 Payload Weight

The total payload weight is the summation of the passengers, passenger baggage, fighter/attack aircraft auxiliary fuel tanks, and cargo weights. It is calculated using the following equation.

$$WPAYLOAD = WPASS + WPBAG + WCARGO \tag{142}$$

where
WPAYLOAD	=	Aircraft total payload weight, *lb*
WPASS	=	Total passenger weight, *lb*
WPBAG	=	Weight of passenger baggage for transport aircraft or weight of auxiliary fuel tanks for fighter/attack aircraft, *lb*
WCARGO	=	Weight of cargo that will be placed in containers, *lb*

6.3.2 Zero Fuel Weight

The aircraft zero fuel weight is the summation of the aircraft operating empty and payload weights. It is calculated using the following equation.

$$WZF = WOWE + WPAYLOAD \tag{143}$$

where
WZF = Aircraft zero fuel weight, *lb*

6.4 Fuel Weight

FLOPS calculates the component weights based on a ramp weight and then calculates the fuel as the difference between that ramp weight and the zero fuel weight. The aircraft fuel weight is calculated using the following equation.

$$FUELM = GW - WZF \tag{144}$$

where
FUELM = Total aircraft fuel weight, lb
GW = Ramp weight, lb

Excess Fuel Capacity: The excess fuel capacity is the difference between the maximum fuel capacity and the fuel weight. It is calculated using the following equation.

$$THEXF = FMXTOT - FUELM \tag{145}$$

where
THEXF = Aircraft excess fuel capacity, lb
FMXTOT = Aircraft maximum fuel capacity, lb. Includes wing, fuselage and auxiliary tanks.

7 Special Options

FLOPS has three special options for weight estimation. The first enables an alternate set of weight equations for some components. The second enables an alternate calculation of the wing fuel capacity that scales the wing fuel capacity with changing geometry. The third enables an alternate calculation of the operating empty weight. These alternate methods are presented in the following subsections.

7.1 Alternate Weight Equations

A set of alternate weight equations were developed to estimate the weight of most structural elements, the fuel system, systems and equipment items, select operating items, aircraft empty weight, and wing fuel capacity.

These equations were developed separately from the FLOPS development effort and were therefore developed using different methods than those presented previously. They were originally intended for general aviation aircraft. The equations were never fully developed and consequently do not correlate well with manufactured aircraft data. For completeness, however, they are presented below.

7.1.1 Structural Components

The alternate structural equations estimate the weight of the horizontal tail structure, vertical tail structure, fuselage structure, and landing gear.

Horizontal Tail: The alternate method for estimating the horizontal tail weight is based on the horizontal tail area. The horizontal tail weight is estimated with the following equation.

$$WHT = 5.4 \times SHT \tag{146}$$

where
WHT = Horizontal tail weight, lb
SHT = Horizontal tail theoretical area, ft^2

Vertical Tail: The alternate method for estimating the vertical tail weight is based on the vertical tail area. The vertical tail weight is estimated with the following equation.

$$WVT = 6.0 \times SVT \tag{147}$$

where
WVT = Vertical tail weight, lb
SVT = Vertical tail theoretical area per tail, ft^2

Fuselage: The alternate method for estimating the fuselage structural weight is based on the fuselage wetted area, fuselage width, and fuselage depth. The fuselage structural weight is estimated with the following equation.

$$WFUSE = 3.939 \times \frac{SWFUS}{\left(\dfrac{DF}{WF}\right)^{0.221}} \tag{148}$$

where
WFUSE = Fuselage weight, lb
SWFUS = Fuselage wetted area ft^2. User input or estimated
                with Equation 61.
DF   = Maximum fuselage depth, ft
WF   = Maximum fuselage width, ft

Landing Gear: The alternate method for estimating the main and nose landing gear weight is based on the ramp weight and landing gear length. The total landing gear weight is calculated and then split between the main and nose gears in the following equations.

$$WLG = GW \times \frac{30100 + 0.3876 \times XMLG^2 + 0.09579 \times XNLG^2}{1,000,000} \tag{149}$$

$$WLGM = 0.85 \times WLG \tag{150}$$

$$WLGN = 0.15 \times WLG \tag{151}$$

where
WLG   = Total landing gear weight, lb
WLGM = Main landing gear weight, lb
WLGN = Nose landing gear weight, lb

GW	=	Ramp weight, lb
XMLG	=	Length of the extended main landing gear oleo, $in.$ User input or estimated with Equation 66.
XNLG	=	Length of the extended nose landing gear oleo, $in.$ User input or estimated with Equation 67.

7.1.2 Propulsion System Items

Fuel Systems, Tanks, and Plumbing: The alternate method for estimating the weight of the fuel systems, tanks, and plumbing is based on the number of tanks and total fuel capacity. The weight of the fuel systems, tanks, and plumbing is estimated with the following equation.

$$WFSYS = 0.3 \times 3262 \times \frac{NTANK}{13} + 0.7 \times 3262 \times \left(\frac{FMXTOT}{208100}\right)^{\frac{2}{3}}$$
$$+ 350 + 0.00029 \times FMXTOT \quad (152)$$

where

WFSYS	=	Weight of the fuel system (including tanks and plumbing), lb
NTANK	=	Number of fuel tanks
FMXTOT	=	Aircraft maximum fuel capacity, lb. Includes wing, fuselage and auxiliary tanks.

7.1.3 Systems and Equipment Items

The alternate systems and equipment equations estimate the weight of the surface controls, hydraulics system, electrical system, furnishings, and air conditioning system. The total alternate systems and equipment weight is adjusted by adding an additional 1% of the empty weight, shown at the end of this section.

Surface Controls: The alternate method for estimating the weight of the surface controls is based on the wing reference area and tail geometry. The weight of the surface controls is estimated with the following equation.

$$WSC = 480 + 0.99 \times SW + \frac{2.5 \times SWTHT}{2 + 0.387 \times TCHT} + 1.6 \times SVT \quad (153)$$

where

WSC	=	Weight of the surface control systems, lb
SW	=	Reference wing area, ft^2, estimated using Equation 4
SWTHT	=	Wetted area of horizontal tail, ft^2
TCHT	=	Horizontal tail thickness to chord ratio
SVT	=	Vertical tail theoretical area per tail, ft^2

Hydraulics: The alternate method for estimating the weight of the hydraulics system is based on the wing reference area and tail geometry. The weight of the hydraulics system is estimated with the following equation.

$$WHYD = 0.6053 \times \left(SW + 1.44 \times \left(\frac{SWTHT}{2 + 0.387 \times TCHT} + SVT\right)\right) \tag{154}$$

where
WHYD   =   Weight of the hydraulic system group, lb

Electrical: The alternate method for estimating the weight of the electrical system is based on the total number of passengers. The weight of the electrical system is estimated with the following equation.

$$WELEC = 16.3 \times NPASS \tag{155}$$

where
WELEC   =   Weight of the electrical system group, lb
NPASS   =   Total number of passengers

Furnishings: The alternate method for estimating the weight of the furnishings is based on the total number of passengers. The weight of the furnishings is estimated with the following equation.

$$WFURNB = 72.6 \times NPASS + 5.25 \times NPASS + 4.3 \times NPASS + 3600 \tag{156}$$

where
WFURNB   =   Weight of the furnishings group without the additional 1% of the empty weight, lb

The alternate method for estimating the weight of the furnishings includes an additional 1% of the empty weight. The weight of the furnishings is estimated with the following equation.

$$WFURN \quad = \quad WFURNB + 0.01 \times (WSTRCT + WPRO + WSYSB) \tag{157}$$

where
WFURN   =   Weight of the furnishings group, lb
WSTRCT   =   Total aircraft structural group weight, lb
WPRO   =   Total aircraft propulsion group weight, lb
WSYSB   =   Total aircraft systems and equipment group weight without the additional 1% of the empty weight, lb

Air Conditioning: The alternate method for estimating the weight of the air conditioning system is based on the number of passengers. The weight of the air conditioning system is estimated with the following equation.

$$WAC = 26 \times NPASS \tag{158}$$

where
WAC   =   Weight of the air conditioning system group, lb

Total Systems and Equipment: The total weight of the systems and equipment is calculated with the following equation.

$$WSYSB = WSC + WAPU + WIN + WHYD + WELEC$$
$$+ WAVONC + WARM + WFURNB + WAC + WAI \quad (159)$$

where
WSYSB	=	Total aircraft systems and equipment group weight without the additional 1% of the empty weight, *lb*
WAPU	=	Weight of the auxiliary power unit, *lb*
WIN	=	Weight of the instruments system group, *lb*
WHYD	=	Weight of the hydraulic system group, *lb*
WELEC	=	Weight of the electrical system group, *lb*
WAVONC	=	Weight of the avionics system group, *lb*
WARM	=	Weight of the armament group, *lb* (Includes thermal protection system or armor and fixed weapons)
WFURNB	=	Weight of the furnishings group, *lb*
WAC	=	Weight of the air conditioning system group, *lb*
WAI	=	Weight of the anti-icing system for transport aircraft, *lb* (For fighter/attack aircraft this variable is a user input used for auxiliary gear system)

As part of this method an additional 1% of the empty weight is added to the total systems and equipment using the following equation.

$$WSYS = WSYSB + 0.01 \times (WSTRCT + WPRO + WSYSB) \quad (160)$$

where
WSYS	=	Total aircraft systems and equipment group weight with the additional 1% of the empty weight, *lb*

7.1.4 Operating Items

The alternate operating items equations estimate the weight of the unusable fuel, engine oil, and passenger service.

Unusable Fuel: The alternate method for estimating the weight of the unusable fuel is based on the total fuel capacity. The weight of the unusable fuel is estimated with the following equation.

$$WUF = 0.0084 \times FMXTOT \quad (161)$$

where
WUF	=	Weight of unusable fuel, *lb*
FMXTOT	=	Aircraft maximum fuel capacity, *lb*. Includes wing, fuselage and auxiliary tanks.

Engine Oil: The alternate method for estimating the weight of the engine oil is based on the number of passengers. The weight of the engine oil is estimated with the following equation.

$$WOIL = 240 \times \frac{NPASS + 39}{40} \tag{162}$$

where
WOIL   =   Weight of the engine oil, *lb*
NPASS   =   Total number of passengers

Passenger Service: The alternate method for estimating the weight of the passenger service is based on the number of passengers. The weight of the passenger service is estimated with the following equation.

$$WSRV = 31.7 \times NPASS \tag{163}$$

where
WSRV   =   Weight of passenger service for transport aircraft, *lb*

7.1.5   Empty Weight

As part of this method an additional 1% is added to the aircraft empty weight. The following equation is used to estimate the alternate method aircraft empty weight.

$$WWE = 1.01 \times (WSTRCT + WPRO + WSYSB) \tag{164}$$

where
WWE   =   Aircraft empty weight, *lb*
WSTRCT   =   Total aircraft structural group weight, *lb*
WPRO   =   Total aircraft propulsion group weight, *lb*
WSYS   =   Total aircraft systems and equipment group weight, *lb*

7.2   Alternate Wing Fuel Capacity Method

An alternate method is available for scaling the wing fuel capacity with changing geometry. It depends on a reference fuel capacity, the wing area, and two user provided scaling factors.

$$FULWMX = FUELRF + FUSCLA$$
$$\times (SW^{1.5} - FSWREF^{1.5}) + FUSCLB \times (SW - FSWREF) \tag{165}$$

where
FULWMX   =   Total fuel capacity of the wing, *lb*
FUELRF   =   Fuel capacity of wing at reference area FSWREF, *lb*
FUSCLA   =   User specified factor A for the 1.5 power term
SW   =   Reference wing area, ft^2, estimated using Equation 4
FSWREF   =   Reference wing area for FUELRF, ft^2. The default value is SW.
FUSCLB   =   User specified factor B for the linear term

7.3 Alternate Operating Empty Weight Method

A special option exists for an alternate calculation of the FLOPS operating empty weight estimate. The structural component weights are scaled to meet the alternate calculated operating empty weight. The resulting operating empty weight is then used to calculate the empty weight and then the furnishings and equipment and systems and equipment weights. The special option estimates the operating empty weight using the following equation.

$$WOWE = SPWTH \times THRUST + SPWSW \times SW$$
$$+ SPWGW \times (GW - TAXOFL) + SPWCON \quad (166)$$

where
WOWE	=	Aircraft operating empty weight, lb
SPWTH	=	Multiplier for thrust per engine, lb/lb. The default value is 2.2344.
THRUST	=	Rated thrust of each scaled engine, lb
SPWSW	=	Multiplier for wing area, lb/ft^2. The default value is 9.5.
SW	=	Reference wing area, ft^2, estimated using Equation 4
SPWGW	=	Multiplier for gross weight used in the alternate operating empty weight method, lb/lb. The default value is 0.104087.
GW	=	Ramp weight, lb
TAXOFL	=	Fixed taxi out fuel, lb, estimated using Equation 167.
SPWCON	=	Constant weight term, lb. The default value is 38,584.

The fixed taxi out fuel is estimated with taxi out time, taxi fuel flow, thrust, and total number of engines using the following equation.

$$TAXOFL = \frac{TAXOTM \times TXFUFL \times THRUST}{60} \times NENG \quad (167)$$

where
TAXOTM	=	Taxi out time, min
TXFUFL	=	Taxi fuel flow, $lb/hr/engine$
NENG	=	Total number of engines

The aircraft empty weight is adjusted using the following equation. Once calculated, this value replaces the standard aircraft empty weight estimate in subsequent calculations.

$$OWWE = WOWE - WOPIT \quad (168)$$

where
OWWE	=	Aircraft empty weight, lb. Used in alternate operating empty weight method.
WOPIT	=	Total operating items weight, lb

The furnishings and equipment and systems and equipment weights are adjusted using the following equations. Once calculated, these values replace the standard furnishings and equipment and systems and equipment weight estimates in subsequent calculations.

$$OWFURN = WFURN + 0.01 \times OWWE \tag{169}$$

$$OWSYS = WSYS + 0.01 \times OWWE \tag{170}$$

where

OWFURN	=	Weight of the furnishings group, *lb*. Used in alternate operating empty weight method.
WFURN	=	Weight of the furnishings group, *lb*
OWSYS	=	Total systems and equipment weight, *lb*. Used in alternate operating empty weight method.
WSYS	=	Total aircraft systems and equipment group weight, *lb*

8 Summary

The methods developed for the FLOPS software to calculate the weight of conceptual aircraft have been presented in this paper. These methods are applicable to transport aircraft, fighter/attack aircraft, general aviation aircraft, and HWB aircraft and have been in use for many years in the NASA Langley Aeronautics Systems Analysis Branch. These methods are based on the component weight build up method and provide a reasonably accurate estimate of aircraft weight, especially for conventional tube and wing aircraft.

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Appendix A   Development of the Tail Sizing Equations

Tail sizing in the preliminary design phase involves stability and control considerations which are not typically evaluated during the conceptual design phase. The conceptual design phase usually involves sizing the aircraft tails using tail volume coefficients. FLOPS weight estimation method includes a conceptual tail sizing method based on historical data using the same 17 transport aircraft and 25 fighter/attack aircraft that were used to develop the weight equations.

Multiple approaches were evaluated and a method based on tail volume coefficients was selected to provide the most accurate estimates. The traditional tail volume coefficient equations are shown below.

$$SHT = \frac{Vh \times Mac \times SW}{Lh} \tag{171}$$

$$SVT = \frac{Vv \times SPAN \times SW}{Lv} \tag{172}$$

where

SHT	=	Horizontal tail theoretical area, ft^2
SVT	=	Vertical tail theoretical area, ft^2
Vh	=	Horizontal tail volume coefficient
Vv	=	Vertical tail volume coefficient
Mac	=	Mean aerodynamic chord, ft
SPAN	=	Wing span, ft
SW	=	Reference wing area, ft^2, estimated using Equation 4
Lh	=	Horizontal tail moment arm defined as the distance from the aircraft center of gravity to the aerodynamic center of the horizontal tail, ft
Lv	=	Vertical tail moment arm defined as the distance from the aircraft center of gravity to the aerodynamic center of the vertical tail, ft

The locations of the aerodynamic centers are not available in FLOPS, neither as an internal estimate nor as an input. Therefore, geometric factors are used to represent them based on the available geometry. The tail moment arm lengths are replaced with the length of the fuselage as a representative length. The mean aerodynamic chord is also unknown and not an input for FLOPS, so it is replaced with an average chord defined as the wing reference area divided by the span. The theoretical area of each tail is calculated with the following equations. The modified tail volume coefficient equations are discussed later in this section.

$$SHT = HTVC \times HFac \tag{173}$$

$$SVT = VTVC \times VFac \tag{174}$$

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where

HTVC	=	Modified horizontal tail volume coefficient. Calculated using Equation 177 for transport aircraft or Equation 179 for fighter/attack aircraft.
VTVC	=	Modified vertical tail volume coefficient. Calculated using Equation 178 for transport aircraft or Equation 180 for fighter/attack aircraft.
HFac	=	Horizontal tail geometric factor. Defined below in Equation 175.
VFac	=	Vertical tail geometric factor. Defined below in Equation 176.

The horizontal tail geometric factor is calculated with the following equation.

$$HFac = \frac{SW^2}{SPAN \times XL} \tag{175}$$

where

XL	=	Total fuselage length, ft

The vertical tail geometric factor is calculated with the following equation.

$$VFac = \frac{SPAN \times SW}{XL} \tag{176}$$

A.1 Development of the Tail Volume Coefficient Relationships

The process to develop the tail volume coefficient relationship equations was similar to the process used to develop the weight equations. First, all geometric parameters that relate to stability and control and tail size of aircraft were considered from the aircraft reference database. The optimization techniques previously described for the weight equations were used to reduce the number of parameters. The goal was to use the fewest and most meaningful variables to achieve the best fit to the data.

A.2 Transport Aircraft Tail Sizing Equations

The transport aircraft tail sizing equations are used for transport, HWB, and general aviation aircraft. The transport aircraft tail volume coefficient relationships are shown below.

$$HTVC = 56.9 \times WF^{0.5} \times FLAPR^{0.82}$$
$$\times (1.0 - 0.46 \times HHT) \times (1.0 - 0.10 \times NEW)/(VCMN \times SWEEP) \tag{177}$$

$$VTVC = 0.0035 \times SWEEP^{0.57} \times DF^{0.5} \times (1.0 + 0.20 \times HHT)$$
$$\times (1.0 + 0.33 \times NEF)/VCMN \times NVERT^{0.7} \tag{178}$$

where

WF	=	Maximum fuselage width, ft

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FLAPR	=	Ratio of total movable wing surface area (flaps, elevators, spoilers, etc.) to wing area. The default value is 0.333.
HHT	=	Horizontal tail mounting location indicator ranging from 0.0 for body mounted to 1.0 for a T-tail
NEW	=	Number of wing mounted engines
VCMN	=	Cruise Mach number
SWEEP	=	Quarter chord sweep angle of the wing, deg
DF	=	Maximum fuselage depth, ft
NEF	=	Number of fuselage mounted engines
NVERT	=	Number of vertical tails

The tail volume coefficient relationships are used in the previous equations to estimate the aircraft tail areas. Figures 4 and 5 show a comparison of the transport aircraft tail area estimates to the tail area data in the database. Actual values are plotted on the vertical axis and the FLOPS estimated values are plotted on the horizontal axis. The line $y = x$ represents an exact match where the actual value is equal to the estimated value. There are a few aircraft with large discrepancies, however, given the variety of aircraft types, sizes, missions, and requirements the correlation is very good.

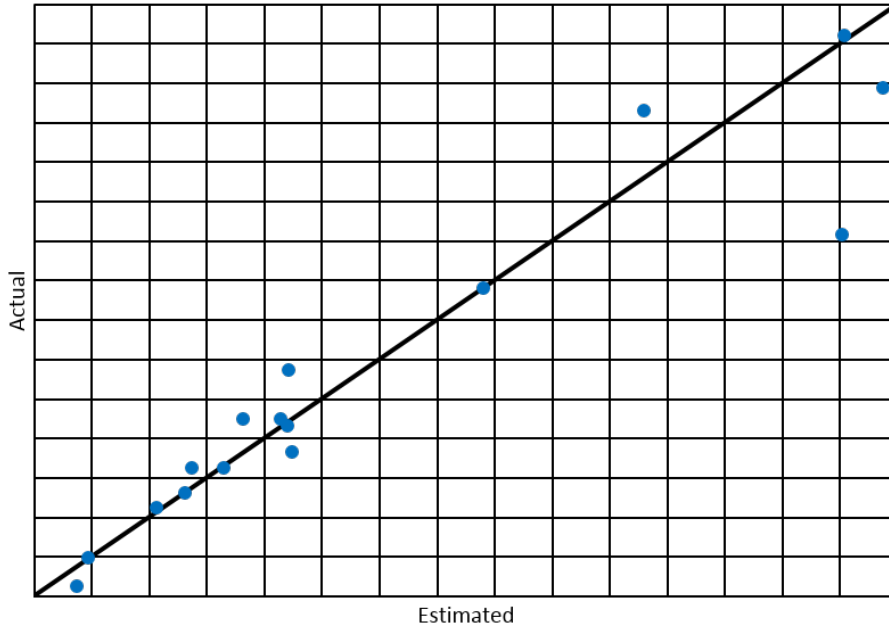## Transport Aircraft Horizontal Tail Area

Figure 4. FLOPS transport horizontal tail area comparison.

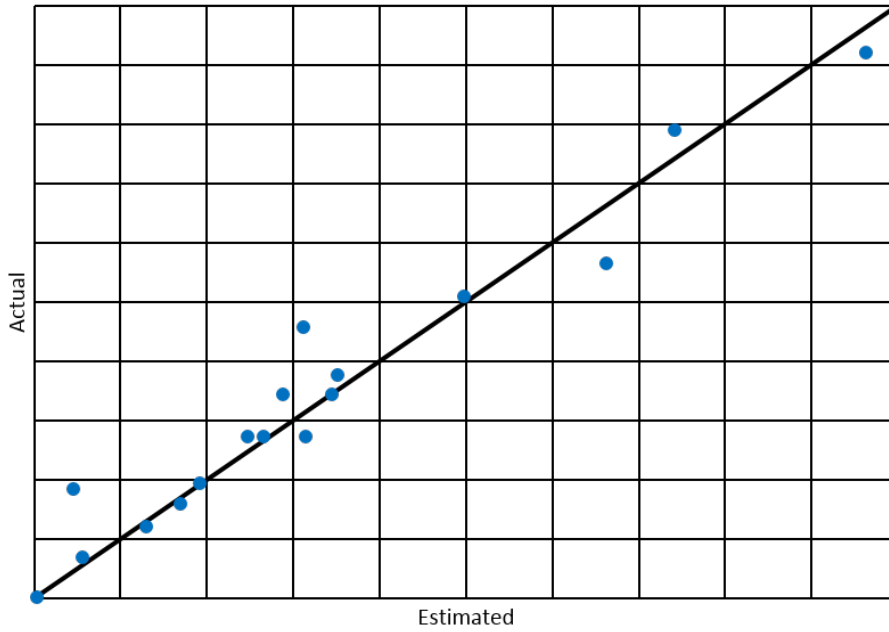## Transport Aircraft Vertical Tail Area

Figure 5. FLOPS transport vertical tail area comparison.

A.3 Fighter/Attack Aircraft Tail Sizing Equations

The fighter/attack aircraft tail volume coefficient relationships are shown below.

$$HTVC = 1.85 \times FLAPR^{0.42}$$
$$\times (1.0 + 0.38 \times HHT) \times (1.0 + 0.25 \times NEW) \times (1.0 + 0.50 \times IVSWP) \quad (179)$$

$$VTVC = 0.227 \times (1.0 + 0.56 \times HHT)$$
$$\times (1.0 - 0.36 \times IEW) \times (1.0 - 0.17 \times IVSWP)/NVERT^{0.7} \quad (180)$$

where

FLAPR	=	Ratio of total movable wing surface area (flaps, elevators, spoilers, etc.) to wing area. The default value is 0.333.
HHT	=	Horizontal tail mounting location indicator ranging from 0.0 for body mounted to 1.0 for a T-tail
NEW	=	Number of wing mounted engines
IVSWP	=	Variable sweep indicator with 1.0 for variable sweep wing and 0.0 for fixed wing
IEW	=	1.0 for wing mounted engines and 0.0 for fuselage mounted engines
NVERT	=	Number of vertical tails

The tail volume coefficient relationships are used in the previous equations to estimate the aircraft tail areas. Figures 6 and 7 show a comparison of the fighter/attack aircraft tail area estimates to the tail area data in the database. Again, actual values are plotted on the vertical axis and the FLOPS estimated values are plotted on the horizontal axis. The line $y = x$ represents an exact match where the actual value is equal to the estimated value.

There are more aircraft in the fighter/attack figures with larger discrepancies than in the transport figures. This was expected based on the wider variety of aircraft types, sizes, missions, and requirements.
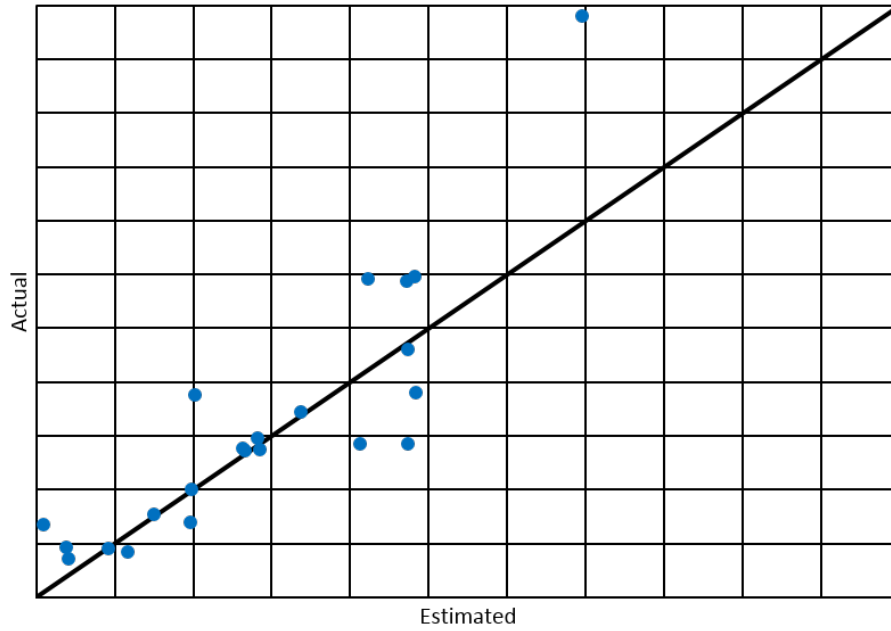
Fighter/Attack Aircraft Horizontal Tail Area

Figure 6. FLOPS fighter/attack aircraft horizontal tail area comparison.

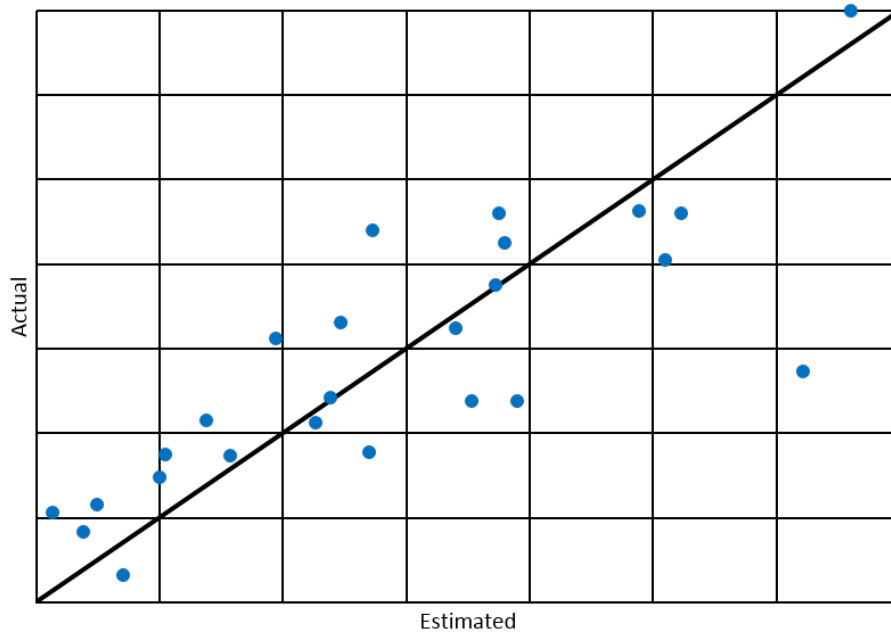**Fighter/Attack Aircraft Vertical Tail Area**

Figure 7. FLOPS fighter/attack aircraft vertical tail area comparison.

Appendix B Development of the Hybrid Wing Body Sizing and Analysis

Since the development of FLOPS began, the hybrid wing body (HWB) configuration has increased in technical feasibility. HWB configurations have significantly different cabins than traditional tube with wing aircraft. Therefore FLOPS was modified to enable HWB mission analysis [4]. The modifications include HWB cabin sizing and layout, weight routines, and propulsion and aerodynamic inputs. Reference 4 contains more information on the HWB geometric definition and weight estimation including a FLOPS HWB comparison to a Boeing Blended Wing Body (BWB) and an application study.

In Reference 6, the fuselage was sized by placing passengers and utility areas in a 12 foot wide tube, considering packaging constraints and guidelines. The tube was then divided and the resulting sections placed side-by-side to form the HWB cabin as 12 foot wide bays. The drawback of this method is that the bays are not always filled, resulting in large discrete changes in the cabin area and weight when bays are added or removed due to passenger changes.

The method that is implemented in FLOPS assigns an area to each passenger based on the passenger class and packing rules, including the number of passengers abreast and the seat pitch. The utility areas, including lavatories, galleys, and closets are calculated based on the number of passengers in each class. The total cabin area is the sum of the passenger areas, utility areas, aisles, and other areas that result from geometry constraints. This method is believed to provide better sensitivity to payload for system studies, however both methods are approximations that should be replaced as a design matures.

B.1 Calculations for Required Cabin Area

The total pressurized cabin area can be calculated using one of two methods. In the first method, the cabin area is calculated by summing the areas for all of the areas that must be in the cabin. For the second method, the area is calculated from the cabin dimensions. These two methods are defined in Equation 195 and 196.

The assumptions and equations used to estimate the cabin area for a HWB are below. The area per passenger including aisle area is defined by class as shown below. The total area required for the passenger seating is then the sum of each class seat area times the number of passengers.

B.1.1 Total Passenger Area

The seat area for each passenger class is estimated with the following equation. This equation is used with inputs from Table 2 to calculate the area per seat of each class.

$$ASEAT = \frac{BAYW \times PITCH}{12 \times NABR} \tag{181}$$

where
ASEAT   =   Area per seat, ft^2
BAYW   =   Passenger bay width, ft. The default value is 12.
PITCH   =   Seat pitch, in
NABR   =   Number of seats abreast

Table 2. FLOPS HWB Cabin Default Values

	Units	First Class	Business Class	Tourist Class
NABR		4	5	6
PITCH	*in*	61	39	32

The default number of seats abreast (NABR) and seat pitch (PITCH) are shown in Table 2.

Total Passenger Area: The total area required for passengers is the sum of the individual passenger area times the number of passengers for each class.

$$ASEATS = NPT \times ATSEAT + NPB \times ABSEAT + NPF \times AFSEAT \quad (182)$$

where

ASEATS	=	Area for passengers, ft^2
NPT	=	Number of tourist class passengers
ATSEAT	=	Seat area for tourist class passengers, ft^2, estimated from Equation 181
NPB	=	Number of business class passengers
ABSEAT	=	Seat area for business class passengers, ft^2, estimated from Equation 181
NPF	=	Number of first class passengers
AFSEAT	=	Seat area for first class passengers, ft^2, estimated from Equation 181

B.1.2   Total Utility Area

The utility area consists of the lavatories, galleys, and closets. These areas can be estimated using the following equations.

Lavatory Area: The number of lavatories is estimated using the following equation.

$$NLAVA = \left\lfloor 0.99 + \frac{NPF}{16} \right\rfloor + \left\lfloor 0.99 + \frac{NPB}{24} \right\rfloor + \left\lfloor 0.99 + \frac{NPT}{40} \right\rfloor \quad (183)$$

where

NLAVA	=	Number of lavatories
NPF	=	Number of first class passengers
NPB	=	Number of business class passengers
NPT	=	Number of tourist class passengers
$\left\lfloor 0.99 + \frac{NPF}{16} \right\rfloor$	=	Floor function of $0.99 + \frac{NPF}{16}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor 0.99 + \frac{0}{16} \right\rfloor = 0$, $\left\lfloor 0.99 + \frac{10}{16} \right\rfloor = 1$, $\left\lfloor 0.99 + \frac{17}{16} \right\rfloor = 2$
$\left\lfloor 0.99 + \frac{NPB}{24} \right\rfloor$	=	Floor function of $0.99 + \frac{NPB}{24}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor 0.99 + \frac{0}{24} \right\rfloor = 0$, $\left\lfloor 0.99 + \frac{12}{24} \right\rfloor = 1$, $\left\lfloor 0.99 + \frac{25}{24} \right\rfloor = 2$

$$\left\lfloor 0.99 + \frac{NPT}{40} \right\rfloor = \text{Floor function of } 0.99 + \frac{NPT}{40}. \text{ Equal to the nearest integer rounded down. Examples: } \left\lfloor 0.99 + \frac{0}{40} \right\rfloor = 0, \left\lfloor 0.99 + \frac{21}{40} \right\rfloor = 1, \left\lfloor 0.99 + \frac{41}{40} \right\rfloor = 2$$

The area for each lavatory including half of an aisle is estimated with the following equation.

$$ALAVA = \frac{BAYW \times 3.0}{2.0} \tag{184}$$

where
ALAVA   =   Area for each lavatory, included half of an aisle, ft^2
BAYW   =   Passenger bay width, ft. The default value is 12.

Galley Area: The number of galleys is estimated using the following equation.

$$NGALLY = \lfloor 0.99 + 0.6 \times NLAVA \rfloor \tag{185}$$

where
NGALLY   =   Number of galleys
$\lfloor 0.99 + 0.6 \times NLAVA \rfloor$ =   Floor function of $0.99 + 0.6 \times NLAVA$. Equal to the nearest integer rounded down. Examples: $\lfloor 0.99 + 0.6 \times 0 \rfloor = 0$, $\lfloor 0.99 + 0.6 \times 1 \rfloor = 1$, $\lfloor 0.99 + 0.6 \times 6 \rfloor = 4$

The area for each galley including half of an aisle is estimated with the following equation.

$$AGALLY = \frac{BAYW \times 3.0}{2.0} \tag{186}$$

where
AGALLY   =   Area for each galley, included half of an aisle, ft^2

Closet Area: The number of closets is estimated using the following equation.

$$NCLSET = \lfloor 0.99 + 0.4 \times NLAVA \rfloor \tag{187}$$

where
NCLSET   =   Number of closets
$\lfloor 0.99 + 0.4 \times NLAVA \rfloor$ =   Floor function of $0.99 + 0.4 \times NLAVA$. Equal to the nearest integer rounded down. Examples: $\lfloor 0.99 + 0.4 \times 0 \rfloor = 0$, $\lfloor 0.99 + 0.4 \times 1 \rfloor = 1$, $\lfloor 0.99 + 0.4 \times 9 \rfloor = 4$

The area for each closet including half of an aisle is estimated with the following equation.

$$ACLSET = \frac{BAYW}{2.0} \tag{188}$$

where
ACLSET   =   Area for each closet, including half of an aisle, ft^2

Total Utility Area: The total area required for the utilities is then the sum of the total required areas for the closets, lavatories, and galleys.

$$ASERVS = NLAVA \times ALAVA + NGALLY \times AGALLY + NCLSET \times ACLSET \quad (189)$$

where
ASERVS $=$ Area for passenger services, ft^2

B.1.3 Wasted Cabin Area

The area of the cabin that is wasted due to the slanted side wall is estimated with the following equation.

$$AWASTE = \frac{NBAY \times TanLE \times BAYW^2}{4.0} \quad (190)$$

where
AWASTE $=$ Area of the cabin that is wasted due to the slanted side wall, ft^2
NBAY $=$ Number of passenger bays. User input or estimated using Equation 191.
TanLE $=$ Tangent of the cabin leading edge sweep angle, estimated using Equation 193
BAYW $=$ Passenger bay width, ft. The default value is 12.

The number of passenger bays for an HWB aircraft is estimated using the following equation.

$$NBAY = \left\{ \begin{array}{ll} \left\lfloor 0.01 + \frac{WF}{BAYW} \right\rfloor, & \text{if WF is given} \\ \left\lfloor 0.5 + \frac{ASEATS+ASERVS}{550} \right\rfloor, & \text{otherwise} \end{array} \right\} \quad (191)$$

where
WF $=$ Maximum fuselage width, ft, estimated for HWB aircraft using Equation 192
$\left\lfloor 0.01 + \frac{WF}{BAYW} \right\rfloor$ $=$ Floor function of $0.01 + \frac{WF}{BAYW}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor 0.01 + \frac{13}{12} \right\rfloor = 1$, $\left\lfloor 0.01 + \frac{35}{12} \right\rfloor = 2$, $\left\lfloor 0.01 + \frac{50}{10} \right\rfloor = 5$
$\left\lfloor 0.5 + \frac{ASEATS+ASERVS}{550} \right\rfloor$ $=$ Floor function of $0.5 + \frac{ASEATS+ASERVS}{550}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor 0.5 + \frac{100+100}{550} \right\rfloor = 0$, $\left\lfloor 0.5 + \frac{300+250}{550} \right\rfloor = 1$, $\left\lfloor 0.5 + \frac{1750+425}{550} \right\rfloor = 4$

The HWB fuselage width is estimated using the equation below if the user input the number of bays and the bay width.

$$WF = NBAY \times BAYW \quad (192)$$

The tangent of the cabin leading edge sweep angle for an HWB aircraft is estimated using the following equation.

$$TanLE = tan(SWPLE) \quad (193)$$

where
SWPLE = Sweep angle of the passenger cabin, deg

B.1.4 Aisle Area

The estimated area for transverse aisles from center to center of outboard bays is shown below. Note that the outboard half of the outboard bays do not have aisles, hence the $(NBAY - 1)$ term. Horseshoe shaped aisles are assumed to be five feet wide, cross aisles to be two feet wide, and rear aisles to be three feet wide.

$$AAISLE = 10.0 \times (NBAY - 1) \times BAYW \tag{194}$$

where
AAISLE = Area for transverse aisles from center to center of outboard bays, ft^2
NBAY = Number of passenger bays
BAYW = Passenger bay width, ft. The default value is 12.

B.1.5 Total Pressurized Cabin Area

The total pressurized cabin area is the sum the total passenger seat, utility, aisle, and wasted areas, calculated using the following equation.

$$ACABIN = ASEATS + ASERVS + AWASTE + AAISLE \tag{195}$$

where
ACABIN = Passenger cabin floor area, ft^2
ASEATS = Area for passengers, ft^2
ASERVS = Area for passenger services, ft^2
AWASTE = Area of the cabin that is wasted due to the slanted side wall, ft^2
AAISLE = Area for transverse aisles from center to center of outboard bays, ft^2

The total pressurized cabin area can also be calculated with the fuselage length, fuselage width, and passenger cabin length using the following equation.

$$ACABIN = (XLP + XLW) \times \frac{WF}{2} \tag{196}$$

where
XLP = Length of passenger compartment, ft
XLW = Fixed length of side wall, ft. The side wall refers to the outboard wall of the passenger cabin which is also used to define the wing root chord, estimated using Equation 197.
WF = Maximum fuselage width, ft, estimated for HWB aircraft using Equation 192

The fixed length of the side wall for an HWB aircraft is estimated using the following equation.

$$XLW = XLP - TanLE \times \frac{WF}{2} \tag{197}$$

where

TanLE   =   Tangent of the cabin leading edge sweep angle, estimated using Equation 193

Appendix C   Development of Transport Fuselage Design Method

The transport fuselage design method calculates the fuselage length, width, and passenger compartment length based on the number of passengers in each class. It is interesting to note that the number of business class passengers is not used in this method. This method is only used for transport aircraft.

The fuselage is designed to change from a single aisle configuration to a twin aisle configuration if the number of tourist passengers exceeds 200. This limit was derived from the Boeing 757, which is the largest single aisle commercial passenger transport. The configuration will also switch to a twin aisle if the tourist class seating is greater than six abreast. The default number of seats abreast (NABR), aisle width (WAISL), and seat pitch (PITCH) are shown in Table 3 and Table 4.

Table 3. FLOPS Fuselage Design Default Values For Single Aisle Aircraft

	Units	Up to 59 Passengers	Single Aisle	
		First Class	First Class	Tourist Class
NABR		5	4	6
WAISL	*in*	15	20	18
PITCH	*in*	*No default*	38	34

Table 4. FLOPS Fuselage Design Default Values For Twin Aisle Aircraft

	Units	Twin Aisle	
		First Class	Tourist Class
NABR		6	8
WAISL	*in*	18	15
PITCH	*in*	38	34

The number of closets is estimated using the following equation.

$$NCLSET = 1 + \left\lfloor \frac{NPF}{30} \right\rfloor + \left\lfloor \frac{NPT}{60} \right\rfloor \tag{198}$$

where

NCLSET   =   Number of closets

NPF   =   Number of first class passengers

NPT   =   Number of tourist class passengers

$\left\lfloor \frac{NPF}{30} \right\rfloor$ =   Floor function of $\frac{NPF}{30}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor \frac{25}{30} \right\rfloor = 0$, $\left\lfloor \frac{45}{30} \right\rfloor = 1$, $\left\lfloor \frac{60}{30} \right\rfloor = 2$

$$\left\lfloor \frac{NPT}{60} \right\rfloor \quad = \quad \text{Floor function of } \frac{NPT}{60}. \text{ Equal to the nearest integer rounded down. Examples: } \left\lfloor \frac{0}{60} \right\rfloor = 0, \left\lfloor \frac{65}{60} \right\rfloor = 1, \left\lfloor \frac{120}{60} \right\rfloor = 2$$

The number of galleys is estimated using the following equation.

$$NGALLY = 1 + \left\lfloor \frac{NPF + NPT}{100} \right\rfloor \tag{199}$$

where

NGALLY	=	Number of galleys
$\left\lfloor \frac{NPF+NPT}{100} \right\rfloor$	=	Floor function of $\frac{NPF+NPT}{100}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor \frac{20+40}{100} \right\rfloor = 0$, $\left\lfloor \frac{99+100}{100} \right\rfloor = 1$, $\left\lfloor \frac{100+100}{100} \right\rfloor = 2$

The number of lavatories is estimated using the following equation.

$$NLAVA = 2 + \left\lfloor \frac{NPT}{60} \right\rfloor + \left\lfloor \frac{NPF}{100} \right\rfloor \tag{200}$$

where

NLAVA	=	Number of lavatories
$\left\lfloor \frac{NPT}{60} \right\rfloor$	=	Floor function of $\frac{NPT}{60}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor \frac{0}{60} \right\rfloor = 0$, $\left\lfloor \frac{65}{60} \right\rfloor = 1$, $\left\lfloor \frac{120}{60} \right\rfloor = 2$
$\left\lfloor \frac{NPF}{30} \right\rfloor$	=	Floor function of $\frac{NPF}{100}$. Equal to the nearest integer rounded down. Examples: $\left\lfloor \frac{60}{100} \right\rfloor = 0$, $\left\lfloor \frac{199}{100} \right\rfloor = 1$, $\left\lfloor \frac{200}{100} \right\rfloor = 2$

Limits are set for the number of closets, galleys, and lavatories based on the number of passengers and range. The default values are shown in Table 5.

Table 5. FLOPS Fuselage Design Limits

Conditions (All must be met)	Max Closets	Max Galleys	Max Lavatories
NPF = 0, DESRNG $\geq$ 1250, 320 $\geq$ NPT $\geq$ 110	2	2	3
NPF = 0, DESRNG < 1250, 320 $\geq$ NPT $\geq$ 110	2	2	2
NPF = 0, NPT < 110	1	1	1
NPT > 320	4	4	4
NPT > 600	8	8	8

where

DESRNG = Design range, nmi

The passenger compartment length without accounting for the presence of doors in the passenger compartment is estimated using the following equation.

$$XLPB = \frac{FPART}{12} + (NGALLY + NLAVA) \times 3 + \frac{TPART}{12} + NCLSET \tag{201}$$

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where

XLPB   =   Passenger compartment length without additional length needed for doors, ft

FPART   =   Passenger compartment length for the first class passengers, in. Calculated using Equation 202.

TPART   =   Passenger compartment length for the tourist class passengers, in. Calculated using equation 203.

The passenger compartment length needed for the first class passengers is calculated with the following equation.

$$FPART = \begin{cases} \frac{NPF \times FPITCH}{NFABR}, & \text{for } NPF > 0 \\ 0, & \text{otherwise} \end{cases} \qquad (202)$$

where

FPITCH   =   Seat pitch for the first class passengers, in

NFABR   =   Number of first class passengers abreast

The passenger compartment length needed for tourist class passengers is calculated with the following equation.

$$TPART = \begin{cases} \frac{NPT \times TPITCH}{NTABR}, & \text{for } NPT > 0 \\ 0, & \text{otherwise} \end{cases} \qquad (203)$$

where

TPITCH   =   Seat pitch for tourist class passengers, in

NTABR   =   Number of tourist class passengers abreast

The passenger compartment length is then adjusted to account for the presence of the doors using the following equation.

$$XLP = (XLPB + NDOORS \times 2.96) \times FUSMLT \qquad (204)$$

where

XLP   =   Length of passenger compartment, ft

NDOORS   =   Number of doors in the passenger compartment. Estimated using Equation 205.

FUSMLT   =   Factor used in calculation of fuselage passenger compartment length. Defined below in Equation 206.

The number of doors in the passenger compartment is estimated with the following equation.

$$NDOORS = 1 + \frac{XLPB}{50} \times \frac{NTABR}{6} \qquad (205)$$

The factor FUSMLT used in the calculation of the fuselage passenger compartment length is defined in the following equation.

$$FUSMLT = \begin{cases} 0.95, & \text{for } NPF > 0 \text{ and } NTABR < 8 \\ 0.91, & \text{for } NPF = 0 \text{ and } NPT < 180 \\ 1, & \text{otherwise} \end{cases} \qquad (206)$$

where

NPF   =   Number of first class passengers

NTABR   =   Number of tourist class passengers abreast

NPT   =   Number of tourist class passengers

The fuselage length is estimated using the equation shown below.

$$XL = XLP + 25 \times NCEN + 40 \qquad (207)$$

where
XL   =   Total fuselage length, ft
NCEN   =   Factor used in calculation of fuselage length. Calcu-
                lated below using Equation 208.

The factor NCEN used in the calculation of the fuselage length is determined using the following equation.

$$NCEN = NENG - 2 \times \left\lfloor \frac{NENG}{2} \right\rfloor \qquad (208)$$

where
NENG   =   Total number of engines
$\lfloor \frac{NENG}{2} \rfloor$ =   Floor function of $\frac{NENG}{2}$. Equal to the nearest integer
                         rounded down. Examples: $\lfloor \frac{1}{2} \rfloor = 0$, $\lfloor \frac{5}{2} \rfloor = 2$, $\lfloor \frac{8}{2} \rfloor = 4$

The number of rows for each passenger class is calculated using the following equation.

$$NR = \frac{NP}{NABR} \qquad (209)$$

where
NR   =   Number of rows in a passenger class
NP   =   Number of passengers in the given class

The fuselage width can be estimated using the following equation.

$$WF = max(WIDTHT, WIDTHF) \times 1.06 \qquad (210)$$

where
WF   =   Maximum fuselage width, ft
max(x,y)   =   Larger of x and y parameters
WIDTHT   =   Fuselage width parameter based on the tourist class.
                 Defined below using Equation 211.
WIDTHF   =   Fuselage width parameter based on the first class. De-
                 fined below using Equation 212.

The fuselage width parameter WIDTHT is calculated using the following equation.

$$WIDTHT = \frac{NAISL \times WTAISL + NTABR \times 20}{12} \qquad (211)$$

where
NAISL   =   Number of aisles
WTAISL =   Tourist class aisle width, in
NTABR   =   Number of tourist class passengers abreast

The fuselage width parameter WIDTHF is calculated using the following equation.

$$WIDTHF = \frac{NAISL \times WFAISL + NFABR \times 25}{12} \qquad (212)$$

where

WFAISL   =   First class aisle width, in

NFABR   =   Number of first class passengers abreast

The fuselage depth can be estimated using the following equation.

$$DF = WF + 0.9 \qquad (213)$$

where

DF   =   Maximum fuselage depth, ft

Long Passenger Cabins: If the passenger compartment is longer than 190 feet, the following method is used to shorten the compartment when possible. The number of passengers abreast in the tourist class is increased by one when there are tourist class passengers and the number abreast is less than ten. For this same scenario, the first class passengers abreast is calculated using the equation shown below.

$$NFABR = \left\lfloor \frac{NTABR - 2}{2} \right\rfloor \times 2 \qquad (214)$$

where

NFABR   =   Number of first class passengers abreast

NTABR   =   Number of tourist class passengers abreast

$\lfloor \frac{NTABR-2}{2} \rfloor$ =   Floor function of $\frac{NTABR-2}{2}$. Equal to the nearest integer rounded down. Examples: $\lfloor \frac{7-2}{2} \rfloor = 2$, $\lfloor \frac{6-2}{2} \rfloor = 2$, $\lfloor \frac{3-2}{2} \rfloor = 0$

The tourist class seat pitch is decreased by one inch if there are tourist class passengers and the number abreast is ten. The first class seat pitch is decreased by one inch if there are first class passengers and the number abreast is eight and the number of tourist class passengers abreast is ten. When none of the scenarios above are true, rows are added and removed until the passenger compartment is between 180 and 200 feet long. If the passenger compartment is still longer than 200 feet, a second deck is added. The second deck is not allowed to be longer than the first and is a single class.

Appendix D   Variable Definition List

The equations in the FLOPS weight estimation method use many variables. This appendix includes a complete list of the variables used in the equations that are presented in this manuscript. The variables are ordered alphabetically.

$\lceil x \rceil$ =   Ceiling function of x. Equal to the nearest integer rounded up. Examples: $\lceil 4.1 \rceil = 5$, $\lceil 4.9 \rceil = 5$, $\lceil 5 \rceil = 5$, $\lceil -3.4 \rceil = -3$

$\lfloor x \rfloor$ =   Floor function of x. Equal to the nearest integer rounded down. Examples: $\lfloor 2.6 \rfloor = 2$, $\lfloor 2.1 \rfloor = 2$, $\lfloor 11 \rfloor = 11$, $\lfloor -7.3 \rfloor = -8$

A1   =   Constant which depends on the type of aircraft. Defined using Table 1.

A2   =   Constant which depends on the type of aircraft. Defined using Table 1.

A3	=	Constant which depends on the type of aircraft. Defined using Table 1.
A4	=	Constant which depends on the type of aircraft. Defined using Table 1.
A5	=	Constant which depends on the type of aircraft. Defined using Table 1.
A6	=	Constant which depends on the type of aircraft. Defined using Table 1.
A7	=	Constant which depends on the type of aircraft. Defined using Table 1.
AAISLE	=	Area for transverse aisles from center to center of outboard bays, ft^2
ABSEAT	=	Seat area for business class passengers, ft^2, estimated from Equation 181
ACABIN	=	Passenger cabin floor area, ft^2
ACLSET	=	Area for each closet including half of an aisle, ft^2
AEWT	=	Weight of alternate engines, lb
AFSEAT	=	Seat area for first class passengers, ft^2, estimated from Equation 181
AGALLY	=	Area for each galley included half of an aisle, ft^2
ALAVA	=	Area for each lavatory included half of an aisle, ft^2
ALTEWT	=	Weight-to-thrust ratio per engine for the alternate propulsion system, lb
ALTTHR	=	Rated thrust per engine for the alternate propulsion system, lb
AR	=	Wing aspect ratio
arctan	=	Arctangent function
ARVT	=	Vertical tail theoretical aspect ratio
ASEAT	=	Area per seat, ft^2
ASEATS	=	Area for passengers, ft^2
ASERVS	=	Area for passenger services, ft^2
ASW	=	Average sweep angle weighted by distance from the centerline, deg. Calculated using Equation 18.
ATSEAT	=	Seat area for tourist class passengers, ft^2, estimated from Equation 181
AWASTE	=	Area of the cabin that is wasted due to the slanted side wall, ft^2
B	=	Wing span, ft
BATWT	=	Storage system density for the alternate energy source, $lb/energy$. Energy unit is user specified and must be consistent with POW-MAX.
BAYW	=	Passenger bay width, ft. The default value is 12.
BMA_i	=	Local required bending material area used in the detailed wing weight estimation method
BPP	=	Weight of baggage per passenger, lb, estimated using Equation 130.

BT	=	Wing equivalent bending factor. Calculated using either Equation 10 if the simple wing weight method is used or Equation 31 if the detailed wing weight method is used.
BTB	=	Wing equivalent bending material factor without the sweep angle adjustment
BTE	=	Wing inertia relief factor due to engines. Calculated using Equation 27.
C_i	=	Local chord length, ft. Used in the detailed wing weight estimation method.
C4	=	Factor used in the calculation of the wing sweep aeroelastic factor. Defined in Equation 15.
C6	=	Factor used in the calculation of the wing sweep aeroelastic factor. Defined in Equation 16.
CARBAS	=	Carrier based aircraft switch, where 1.0 is for carrier-based aircraft and 0.0 is for land-based aircraft
CARGF	=	Cargo aircraft floor factor, where 0.0 is for passenger transport and 1.0 is for military cargo transport
CARGOF	=	Cargo other than passenger baggage carried in the fuselage including fuselage external stores for fighter/attack aircraft, lb
CARGOW	=	Cargo carried in wing including the weight of wing-mounted external stores for fighter/attack aircraft, lb
CAYA	=	Factor used in wing sweep and aeroelastic tailoring factor
CAYE	=	Propulsion system pod inertia relief factor. Calculated using either Equation 38 if the simplified wing method is used or 39 if the detailed wing method is used.
CAYF	=	Multiple fuselage factor. $CAYF = 1.0$ for a single fuselage and $CAYF = 0.5$ for multiple fuselages.
CAYL	=	Wing sweep factor including aeroelastic tailoring
CSVT	=	Factor used for vertical tail weight calculation. Defined in Equation 53.
CSW_i	=	Secant of the load path sweep angle
DAV	=	Average fuselage diameter, ft, estimated with Equation 57.
$DELM_i$	=	Local moment of the pressure load used in the detailed wing weight estimation method
$DELME_i$	=	Local moment of the pressure load used for the engine inertia relief calculation in the detailed wing weight estimation method. Defined in Equation 30.
$DELP_i$	=	Local pressure load used in the detailed wing weight estimation method
DELTA	=	Atmospheric pressure ratio, equal to the ratio of the pressure at cruise altitude to pressure at sea level
DESRNG	=	Design range, nmi
DF	=	Maximum fuselage depth, ft
DFTE	=	Aircraft type. Equal to 1.0 for fighter/attack, 0.0 for all others.
DG	=	Design gross weight, lb

DGW	=	Design gross weight input variable. This variable has two different units depending on its value. If it is less than 5, it is assumed to be a fraction of ramp weight. Otherwise if it is greater than 5 it is assumed to be equal to the design gross weight and has units of lbs.
DIH	=	Wing dihedral angle, deg
DNAC	=	Average diameter of the scaled engine nacelles, ft. Either calculated by scaling the baseline engine nacelles by $\sqrt{\frac{THRUST}{THRSO}}$ or with Equation 71.
DY_i	=	Y-distance from current to previous integration station location used in the detailed wing weight estimation method, ft
EA_i	=	Factor used in the detailed wing weight estimation method. Calculated using Equation 28.
EEM_i	=	Factor used in the detailed wing weight estimation method. Defined in Equation 29.
$EETA_i$	=	Engine position factor, ft. At each station it is equal to the distance between the next outboard engine and the fuselage center line. At stations where there are no additional outboard engines it is equal to zero.
EEXP	=	Engine weight scaling parameter. The default value is 1.15.
EINL	=	Engine inlet weight scaling exponent. The default value is 1.
EL_{tot}	=	Total pressure load along the load path used in the calculation of the wing equivalent bending material factor for the detailed wing weight estimation method
EL_i	=	Partial summation from the tip to the current integration station of the pressure load along the load path, psi
EM_{tot}	=	Total wing bending moment modified by the local load path sweep angle used in the calculation of the wing equivalent bending material factor for the detailed wing weight estimation method
EM_i	=	Partial summation from the tip to the current integration station of the wing bending moment modified by the local load path sweep angle
EMS	=	Wing strut bracing factor, used in the calculation of the wing equivalent bending material factor for the simplified wing weight estimation method. Defined in Equation 11.
ENOZ	=	Engine nozzle weight scaling exponent. The default value is 1.
ETA_i	=	Local wing station location, ft. Calculated as the distance from fuselage centerline.
FAERT	=	Aeroelastic tailoring factor used in the design of the wing ranging from 0.0 for no aeroelastic tailoring to 1.0 for maximum aeroelastic tailoring
FCOMP	=	Composite utilization factor for wing structure ranging from 0.0 for no composites to 1.0 for maximum use of composites
FIXSPN	=	Fixed wing span, ft

FLAPR	=	Ratio of total movable wing surface area (flaps, elevators, spoilers, etc.) to wing area. The default value is 0.333.
FMXTOT	=	Aircraft maximum fuel capacity, *lb*. Includes wing, fuselage, and auxiliary tanks.
FNAC	=	Average diameter of each scaled engine, scaled to account for distributed propulsion if applicable, *ft*
FNEF	=	Number of fuselage mounted engines, scaled to account for distributed propulsion if applicable
FNENG	=	Total number of engines, scaled to account for distributed propulsion if applicable
FNEW	=	Number of wing mounted engines, scaled to account for distributed propulsion if applicable
FPAREA	=	Fuselage planform area, ft^2, estimated using Equation 62
FPART	=	Passenger compartment length for the first class passengers, *in*. Calculated using Equation 202.
FPITCH	=	Seat pitch for the first class passengers, *in*
FSTRT	=	Wing strut bracing factor ranging from 0.0 for no wing strut to 1.0 for full benefit from strut bracing
FSWREF	=	Reference wing area for FUELRF, ft^2. The default value is SW.
FTHRST	=	Rated thrust of each scaled engine, scaled to account for distributed propulsion if applicable, *lb*
FTHRUST	=	Rated thrust of each scaled engine, including adjustment for distributed propulsion if applicable, *lb*
FUELM	=	Total aircraft fuel weight, *lb*
FUELRF	=	Fuel capacity of wing at reference area FSWREF, *lb*
FUFU	=	Maximum fuel capacity of the fuselage, *lb*
FULAUX	=	Auxiliary (external) fuel tank capacity, *lb*. This is used for fighter/attack aircraft only.
FULDEN	=	Fuel density ratio for alternate fuels compared to jet fuel (typical density of 6.7 *lb/gal*). The default value is 1.
FULFMX	=	Total fuel capacity of the fuselage, *lb*. Includes wing carry through structure and fuselage tanks.
FULWMX	=	Total fuel capacity of the wing, *lb*
FUSCLA	=	User specified factor A for the 1.5 power term. Used to scale wing fuel capacity.
FUSCLB	=	User specified factor B for the linear term. Used to scale wing fuel capacity.
FUSMLT	=	Factor used in calculation of fuselage passenger compartment length. Defined in Equation 206.
FWMAX	=	Factor for wing fuel capacity equation. The default value is 23.
GLOV	=	Total glove and bat area beyond theoretical wing area, ft^2
GW	=	Ramp weight, *lb*
HFac	=	Horizontal tail geometric factor. Calculated using Equation 175.
HHT	=	Horizontal tail mounting location indicator ranging from 0.0 for body mounted to 1.0 for a T-tail

HTVC	=	Modified horizontal tail volume coefficient. Calculated using Equation 177 for transport aircraft or Equation 179 for fighter/attack aircraft.
HYDPR	=	Hydraulic system pressure, psi. The default value is 3,000.
IEW	=	1.0 for wing mounted engines and 0.0 for fuselage mounted engines
IVSWP	=	Variable sweep indicator with 1.0 for variable sweep wing and 0.0 for fixed wing
Lh	=	Horizontal tail moment arm defined as the distance from the aircraft center of gravity to the aerodynamic center of the horizontal tail, ft
Lv	=	Vertical tail moment arm defined as the distance from the aircraft center of gravity to the aerodynamic center of the vertical tail, ft
Mac	=	Mean aerodynamic chord, ft
max(x,y)	=	Function that returns larger of x and y parameters
NABR	=	Number of seats abreast
NAISL	=	Number of aisles
NBAY	=	Number of passenger bays
NCEN	=	Factor used in calculation of fuselage length. Calculated using Equation 208.
NCLSET	=	Number of closets
NCON	=	Number of cargo containers
NDOORS	=	Number of doors in the passenger compartment. Estimated using Equation 205.
NEALT	=	Number of engines for the alternate propulsion system
NEF	=	Number of fuselage mounted engines
NENG	=	Total number of engines
NEW	=	Number of wing mounted engines
NFABR	=	Number of first class passengers abreast
NFIN	=	Number of fins
NFLCR	=	Number of flight crew. User input or estimated using Equation 118 for transport aircraft and assumed to be 1 for fighter/attack aircraft.
NFUSE	=	Number of fuselages
NGALLY	=	Number of galleys
NGALC	=	Number of galley crew
NLAVA	=	Number of lavatories
NP	=	Number of passengers in a given class
NPASS	=	Total number of passengers
NPB	=	Number of business class passengers
NPF	=	Number of first class passengers
NPT	=	Number of tourist class passengers
NR	=	Number of rows in a passenger class
NSTU	=	Number of flight attendants
NTABR	=	Number of tourist class passengers abreast
NTANK	=	Number of fuel tanks
NVERT	=	Number of vertical tails

OSSPAN	=	Outboard wing semispan of HWB aircraft, ft
OWFURN	=	Weight of the furnishings group, lb. Used in alternate operating empty weight method.
OWSYS	=	Total systems and equipment weight, lb. Used in alternate operating empty weight method.
OWWE	=	Aircraft empty weight, lb. Used in alternate operating empty weight method.
P_i	=	Local load intensity factor, ranging from 0.0 to 1.0. Used in the detailed wing weight estimation method.
PCTL	=	Fraction of load carried by the defined wing
PITCH	=	Seat pitch, in
PM_{tot}	=	Total area moments along the wing load path used in the detailed wing weight estimation method
POWMAX	=	Storage system capacity for alternate energy source, $energy$. Energy unit is user specified and must be consistent with BATWT.
POWWT	=	System weight for alternate propulsion system, lb
QCRUS	=	Cruise dynamic pressure, psf, estimated using Equation 49
QDIVE	=	Dive maneuver dynamic pressure, psf, estimated using Equation 100
RFACT	=	Supersonic cruise factor. Equal to 0.00004 for subsonic cruise, 0.00009 for supersonic cruise.
RSPCHD	=	Percent chord of the HWB fuselage rear spar at the fuselage centerline. The default value is 70%.
RSPSOB	=	Percent chord of the HWB fuselage rear spar at the side of body. By default $RSPSOB = RSPCHD$. Where RSPCHD is the percent chord of the HWB fuselage rear spar at the fuselage centerline.
S_{tot}	=	Total wing areas along the wing load path used in the detailed wing weight estimation method, ft^2
SA	=	Sine of the average wing sweep angle weighted by distance from the centerline
SAFTB	=	Area of the aft body, ft^2. Defined in Equation 44.
SCAN	=	Canard theoretical area, ft^2
SFIN	=	Fin theoretical area, ft^2
SFLAP	=	Total movable wing surface area including flaps, elevators, spoilers, etc., ft^2
SHT	=	Horizontal tail theoretical area, ft^2
SLAM	=	Sine of the 3/4 chord wing sweep angle, calculated using Equation 13
SPAN	=	Wing span, ft
SPWCON	=	Constant weight term used in the alternate operating empty weight method, lb. The default value is 38,584.
SPWGW	=	Multiplier for gross weight used in the alternate operating empty weight method, lb/lb. The default value is 0.104087.
SPWSW	=	Multiplier for wing area used in the alternate operating empty weight method, lb/ft^2. The default value is 9.5.

SPWTH	=	Multiplier for thrust per engine used in the alternate operating empty weight method, lb/lb. The default value is 2.2344.
SVT	=	Vertical tail theoretical area per tail, ft^2
SW	=	Reference wing area, ft^2, estimated using Equation 4
SWEEP	=	Quarter chord sweep angle of the wing, deg
SWFUS	=	Fuselage wetted area ft^2, estimated with Equation 61.
SWP$_i$	=	Wing load path sweep angle, deg.
SWPLE	=	Sweep angle of the passenger cabin, deg
SWPVT	=	Vertical tail sweep angle at 25% chord, deg
SWTCN	=	Wetted area of canards, ft^2
SWTFU	=	Wetted area of fuselage, ft^2
SWTHT	=	Wetted area of horizontal tail, ft^2
SWTNA	=	Wetted area of nacelles, ft^2
SWTVT	=	Wetted area of vertical tail, ft^2
SWTWG	=	Wetted area of wings, ft^2
SX	=	Wing trapezoidal area, ft
T$_i$	=	Local thickness to chord ratio used in the detailed wing weight estimation method
TanLE	=	Tangent of the cabin leading edge sweep angle, estimated using Equation 193
TAXOFL	=	Fixed taxi out fuel, lb, estimated using Equation 167.
TAXOTM	=	Taxi out time, min
TCA	=	Weighted average of the wing thickness to chord ratio
TCHT	=	Horizontal tail thickness to chord ratio
TCVT	=	Vertical tail thickness to chord ratio
THEXF	=	Aircraft excess fuel capacity, lb
THRSO	=	Rated thrust of each baseline engine, lb
THRUST	=	Rated thrust of each scaled engine, lb
TLAM	=	Tangent of the 3/4 chord wing sweep angle, calculated using Equation 14
TNAC	=	Total number of nacelles plus 0.5 if there is a center-mounted engine, calculated with Equation 74
TPART	=	Passenger compartment length for the tourist class passengers, in. Calculated using equation 203.
TPITCH	=	Seat pitch for tourist class passengers, in
TR	=	Taper ratio of the wing
TRAFTB	=	Taper ratio of the aft body. Calculated with Equation 43.
TRCAN	=	Canard theoretical taper ratio
TRFIN	=	Fin theoretical taper ratio
TRHT	=	Horizontal tail theoretical taper ratio
TRVT	=	Vertical tail theoretical taper ratio
TXFUFL	=	Taxi fuel flow, $lb/hr/engine$
ULF	=	Structural ultimate load factor. The default value is 3.75.
VARSWP	=	Wing variable sweep weight penalty factor ranging from 0.0 for fixed-geometry wing to 1.0 for full variable-sweep wing
VCMN	=	Cruise Mach number

VFac	=	Vertical tail geometric factor. Calculated with Equation 176.
VFACT	=	Variable wing sweep factor
Vh	=	Horizontal tail volume coefficient
VMAX	=	Maximum Mach number
VTVC	=	Modified vertical tail volume coefficient. Calculated using Equation 178 for transport aircraft or Equation 180 for fighter/attack aircraft.
Vv	=	Vertical tail volume coefficient
W1	=	First term of wing weight equation, lb. It loosely represents the total bending material weight of the wing.
W1NIR	=	Wing bending material weight, not including the effects of inertia relief, lb
W2	=	Second term of the wing weight equation, lb. W2 loosely represents the total shear material and control surface weight of the wing.
W3	=	Third term of the wing weight equation, lb. Loosely represents the weight of the miscellaneous items.
W4	=	Fourth term of the wing weight equation, lb. Loosely represents the total aft body weight for HWB configurations.
WAC	=	Weight of the air conditioning system group, lb
WAI	=	Weight of the anti-icing system for transport aircraft, lb (For fighter/attack aircraft this variable is a user input used for auxiliary gear system)
WAISL	=	Width of the aisle, in
WAPU	=	Weight of the auxiliary power unit, lb
WARM	=	Weight of the armament group, lb (Includes thermal protection system or armor and fixed weapons)
WAVONC	=	Weight of the avionics system group, lb
WCAN	=	Canard weight, lb
WCARGO	=	Weight of cargo that will be placed in containers, lb
WCON	=	Weight of cargo containers, lb
WEC	=	Weight of the engine controls system, lb
WELEC	=	Weight of the electrical system group, lb
WENG	=	Weight of each scaled baseline engine, lb
WENGB	=	Weight of baseline engine, lb. Includes inlet and nozzle weight if they are not specified separately, estimated using Equation 76.
WENGP	=	Intermediate variable used in the calculation of the weight of each engine, lb. If no separate inlet or nozzle weights are provided then it is equal to the engine weight.
WF	=	Maximum fuselage width, ft
WFAISL	=	First class aisle width, in
WFIN	=	Fin weight, lb
WFLCRB	=	Weight of flight crew and baggage, lb
WFSYS	=	Weight of the fuel system (including tanks and plumbing), lb
WFURN	=	Weight of the furnishings group, lb
WFURNB	=	Weight of the furnishings group without the additional 1% of the empty weight, lb

WFUSE	=	Fuselage weight, *lb*
WHT	=	Horizontal tail weight, *lb*
WHYD	=	Weight of the hydraulic system group, *lb*
WIDTHF	=	Fuselage width parameter based on the first class defined. Defined using Equation 212.
WIDTHT	=	Fuselage width parameter based on the tourist class. Defined using Equation 211.
WIN	=	Weight of the instruments system group, *lb*
WINL	=	Weight of the engine inlet, *lb*
WINLB	=	Inlet weight for baseline engines, *lb*
WLDG	=	Aircraft design landing weight, *lb*. User input or estimated using Equation 65.
WLG	=	Total landing gear weight, *lb*
WLGM	=	Main landing gear weight, *lb*
WLGN	=	Nose landing gear weight, *lb*
WMARG	=	Empty weight margin, *lb*
WNAC	=	Weight of nacelle or air induction system, *lb*
WNOZ	=	Weight of the engine nozzle, *lb*
WNOZB	=	Nozzle weight for baseline engine, *lb*
WOIL	=	Weight of the engine oil, *lb*
WOPIT	=	Total operating items weight, *lb*
WOWE	=	Aircraft operating empty weight, *lb*
WPAINT	=	Area density of paint for all wetted areas, lb/ft^2
WPASS	=	Total passenger weight, *lb*
WPAYLOAD	=	Aircraft total payload weight, *lb*
WPBAG	=	Weight of passenger baggage for transport aircraft or weight of auxiliary fuel tanks for fighter/attack aircraft, *lb*
WPMISC	=	Additional miscellaneous propulsion system weight, *lb*
WPMSC	=	Weight of miscellaneous propulsion systems such as engine controls, starter, and electrical, *lb*
WPOD	=	Weight of engine pod including the nacelle, *lb*. Defined using Equation 40.
WPPASS	=	Weight per passenger, *lb*. The default value is 165 *lb*.
WPRO	=	Total aircraft propulsion group weight, *lb*
WSC	=	Weight of the surface control systems, *lb*
WSR	=	Required wing loading, lb/ft^2
WSRV	=	Weight of passenger service for transport aircraft, *lb*. (For fighter/attack aircraft this variable is a user input used for ammunition and non-fixed weapons)
WSTART	=	Weight of the engine starter system, *lb*
WSTRCT	=	Total aircraft structural group weight, *lb*
WSTUAB	=	Weight of flight attendants and galley crew and baggage, *lb*
WSYS	=	Total aircraft systems and equipment group weight, *lb*
WSYSB	=	Total aircraft systems and equipment group weight without the additional 1% of the empty weight, *lb*
WTAISL	=	Width of the tourist class aisles, *in*

WTBAT	=	Weight of alternate energy source storage system, lb
WTHR	=	Weight of the thrust reversers, lb
WTNFA	=	Total weight of engine pod(s) not including the nacelle, lb. Defined using Equation 41.
WTPNT	=	Weight of aircraft paint, lb
WUF	=	Weight of unusable fuel, lb
WVT	=	Vertical tail weight, lb
WWE	=	Aircraft empty weight, lb
WWING	=	Wing weight, lb
WZF	=	Aircraft zero fuel weight, lb
XL	=	Total fuselage length, ft
XLP	=	Length of passenger compartment, ft
XLPB	=	Passenger compartment length without additional length needed for doors, ft
XLW	=	Fixed length of side wall, ft. The side wall refers to the outboard wall of the passenger cabin which is also used to define the wing root chord, estimated using Equation 197.
XMLG	=	Length of the extended main landing gear oleo, in. User input or estimated with Equation 66.
XNAC	=	Average length of the scaled engine nacelles, ft. Either calculated by scaling the baseline engine nacelles by $\sqrt{\frac{THRUST}{THRSO}}$ or with Equation 70.
XNLG	=	Length of the extended nose landing gear oleo, in. User input or estimated with Equation 67.
Y_i	=	Y-location of the integration station location used in the detailed wing weight estimation method, ft
YEE	=	Location of outboard on wing engine, measured from centerline, in

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14. ABSTRACT

FLOPS has been the primary aircraft synthesis software used by the Aeronautics Systems Analysis Branch at NASA Langley Research Center. It was created for rapid conceptual aircraft design and advanced technology impact assessments. FLOPS is a single computer program that includes weights estimation, aerodynamics estimation, engine cycle analysis, propulsion data scaling and interpolation, detailed mission performance analysis, takeoff and landing performance analysis, noise footprint estimation, and cost analysis. It is well known as a baseline and common denominator for aircraft design studies. FLOPS is capable of calibrating a model to known aircraft data, making it useful for new aircraft and modifications to existing aircraft. The weight estimation method in FLOPS is known to be of high fidelity for conventional tube with wing aircraft and a substantial amount of effort went into its development. This report serves as a comprehensive documentation of the FLOPS weight estimation method. The development process is presented with the weight estimation process.

15. SUBJECT TERMS

FLOPS, Flight Optimization System, Weight Estimation, Aircraft Design, Conceptual Design

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