Preliminary Design Calculation for a Business Jet Aircraft

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ABSTRACT
The purpose of this paper is to calculate the preliminary design for a business jet aircraft in an existing formulas and procedures. The equations are obtained to form a Aircraft design methodology. This methodology is used to obtain the fundamental parameter for a business jet aircraft such as gross weight, lift estimation, lift and drag calculations, power, and performance parameters.

INTRODUCTION
The purpose of this chapter is to describe the preliminary design phase of an aircraft. Based on the Systems Engineering approach, an aircraft will be designed during three phases: 1. Conceptual design phase, 2. Preliminary design phase, and 3. Detail design phase. In the conceptual design phase, the aircraft will be designed in concept without the precise calculations. On the other hand, the preliminary design phase tends to employ the outcomes of a calculation procedure. As the name implies, in the preliminary design phase, the parameters that are determined are not final and will be altered later. In addition, in this phase, parameters are essential and will directly influence the entire detail design phase. Therefore the ultimate care must be taken to insure the accuracy of the results of the preliminary design phase.

Three fundamental aircraft parameters that are determined during the preliminary design phase are: 1. Aircraft maximum take-off weight (WTO), 2. Wing reference area (SW or Sref or S), and 3. Engine thrust (TE or T) or engine power (PE or P). Hence, three primary aircraft parameters of WTO, S and T (or P) are the output of the preliminary design phase. These three parameters will govern the aircraft size, the manufacturing cost, and the complexity of calculation. If during the conceptual design phase, a jet engine is selected, the engine thrust is calculated during this phase. But, if during the conceptual design phase, a prop-driven engine is selected, the engine power is calculated during this phase. A few other non-important aircraft parameters such as aircraft zero-lift drag coefficient and aircraft maximum lift coefficient are estimated in this phase too.

The preliminary design phase is performed in two steps: Step 1: Estimate aircraft maximum take-off weight Step 2: Determine wing area and engine thrust (or power) simultaneously

WEIGHT ESTIMATION

2.1 Gross Weight
The gross weight of the aircraft will be given by the equation

\[ W_o = W_c + W_p + W_f + W_d \]

Empty weight includes structures, landing gears, lift equipment avionic instruments. To simplify fuel weight & empty weight calculation take fraction of them based on total weight.

From equation 1, where \( W_o \) is Gross weight of an aircraft, \( W_c \) is Weight of the crew, \( W_p \) is Weight of payloads, \( W_f \)is Weight of the fuel, \( W_d \)is Weight of the aircraft

\[ W_o = \frac{W_c + W_p}{1 - \frac{W_f}{W_o} - \frac{W_p}{W_o}} \]

\( \frac{W_f}{W_o} \rightarrow \text{Weight to fuel ratio} \)

The mission profile for a business aircraft is shown in the figure 1

![FIG: 1 Mission Profile](image)

Mission Profile Segment is defined as the ratio of Aircraft weight at end of the mission segment to the aircraft weight at the start of the mission.

Mission 1-2
The mission (1-2) is the engine start warm-up & take off from the historical data, it is found to be

\( \frac{W_f}{W_1} = 0.9 \)

Mission 2-3
The mission( 2-3) is the climb. The fuel fraction from historical data was

\( \frac{W_f}{W_2} = 0.9 \)

Mission 3-4
The mission( 3-4) is cruise. The fuel fraction for this mission was found from the range equation

\[ \frac{W_4}{W_3} = 0.866 \sqrt{\frac{R \cdot C}{0.866 \cdot W_c \cdot \frac{L}{D_{max}}}} \]

\( \frac{W_4}{W_3} = 0.99 \)

Mission 4-5
The mission (4-5) is loitering. The fuel fraction for this mission was found from the Brequet equation.
\[
\frac{W_5}{W_4} = e^{\frac{E.C}{(L/D)_{max}}}
\]
\[
W_5/W_4 = 0.71
\]

**Mission 5-6**
The mission(5-6) is the landing. The fuel fraction of the mission is found to
\[
W_6/W_5 = 0.997
\]
The fuel fraction is found from the product of all the values
\[
\frac{W_6}{W_1} = \frac{W_2}{W_3} \times \frac{W_3}{W_4} \times \frac{W_4}{W_5} \times \frac{W_5}{W_6}
\]
\[
\frac{W_f}{W_o} = 1 - \frac{W_o}{W_1} = 0.334
\]

From the empirical relation
\[
\frac{W_f}{W_o} = aW_o + b
\]
\[
a = 1.13 \times 10^6
\]
\[
b = 0.48
\]
\[
\frac{W_f}{W_o} = 0.48
\]

**Actual weight**
From the above all values substituted in equation, the actual gross weight obtained is
\[
W_o = 6164.5 \text{ kg}
\]

**2.2 Weight of the fuel**
\[
W_f = \frac{W_f}{W_o} \times W_o = 2058.9 \text{ kg}
\]

**2.3 Tank capacity**
Weight of the activation fuel = 5.64 lb/ gal

Tank capacity = weight of fuel/weight of aviation gasoline

Tank capacity = 804.2 gal

**THRUST CALCULATION**
From the thrust to weight ratio, the thrust produced can be calculated. The obtained thrust value is
\[
T = 175 \text{N}
\]

**3.1 Thrust Matching**
\[
(L/D)_{max} \text{ for loitering} = 12
\]
\[
(L/D)_{max} \text{ for cruise} = 0.866 \times (L/D)_{max} \text{ for loiter}
\]
\[
(L/D)_{max} = 10.3
\]

**3.2 (T/W) \text{cruise}**
\[
(T/W)_{cruise} \text{ is the thrust to velocity ratio at cruising speed. It was derived by following formulae.}
\]
\[
(T/W)_{cruise} = \frac{1}{(L/D)_{max \text{ cruise}}}
\]
\[
(T/W)_{cruise} = 0.0962
\]

**3.3 (T/W) \text{loiter}**
It is the thrust to velocity ratio at loitering condition. It was derived by
\[
(T/W)_{loiter} = 1/(L/D)_{loiter}
\]
\[
(T/W) \text{ Loiter} = 5037.05
\]

**3.4 (T/W) \text{climb}**
\[
(T/W)_{climb} \text{ is the thrust to velocity ratio at climbing condition. It was derived by formula}
\]
\[
(T/W)_{climb} = (1/(L/D)_{max}) + (V_{vertical}/V)
\]
\[
V_{vertical} \rightarrow \text{Velocity at vertical condition}
\]
\[
V_{stall} \rightarrow \text{Velocity at stalling condition}
\]
\[
1.2 V_{stall}
\]
\[
V_{cruise} \rightarrow 0.25 V_{cruise}
\]
\[
\text{where}
\]
\[
V_{stall} = 194.07
\]
\[
V_{cruise} = 232.89
\]
\[
(T/W)_{climb} = 0.377
\]

**3.5 (T/W) \text{Takeoff}**
\[
(T/W)_{Takeoff} \text{ is thrust to velocity ratio at takeoff condition. It was derived}
\]
\[
(T/W)_{Takeoff} = (T/W)_{cruise} \times (W_{cruise} \times W_{takeoff}) \times (T_{Takeoff} / T_{cruise})
\]
\[
(T/W)_{Takeoff} = 0.337
\]
\[
\text{Average } C_{L_{max}} = (2.6 + 2.6) / 2
\]
\[
C_{L_{max}} = 2.6
\]
\[
\text{with flap deflection 45°} \text{ [Ref : Lofting ref 13]}
\]
\[
C_L = \text{Average } C_{L_{max}} + 0.9
\]
\[
C_L = 3.5
\]
\[
C_{L_{max}} = 0.9 \times C_L
\]
\[
C_{L_{max}} = 3.15
\]

The ratio of the fully loaded airplane weight to the wing plan form area is called wing loading

**WING LOADING**

Wing loading is defined as the ratio of gross weight to the gross area
\[
\text{Wing loading} = \frac{\text{Gross weight}}{\text{Gross area}}
\]
\[
C_{L_{max}} = 2 \times W/S \times \frac{1}{\rho \frac{V_{stall}^2}{2}}
\]
\[
w/s = C_{L_{max}} \times \rho \times V_{stall}^2 / 2
\]

**4.1 Wing area**
Wing area is defined as the Ratio of Gross weight to wing loading.
\[
\text{Wing area} = \frac{\text{W}_{o}}{\text{W}_{o}/S}
\]
\[
S = 0.0845 \text{ m}^2
\]

**4.2 Reynolds number**
Reynolds number is a dimensional less parameter. It is defined as the ratio of inertia to viscous force

\[
Re = \frac{\rho V D}{\mu}
\]

\[
Re = 103.37 \times 10^6
\]

### 4.3 Skin friction drag for turbulence flow

\[
C_{f1} = 0.455 \left( \log_{10} Re \right)^{2.58} + \left( 1 + 0.144 M^2 \right)^{0.05}
\]

\[
C_{f1} = \frac{9.375 \times 10^{-5}}{0.012}
\]

Subs the values in thickness to chord ratio
t/c = 10-40% chord

\[
S_{net} = 2 \times (1 + 0.2 \times t/c) \times 0.0845
\]

\[
S_{net} = 0.1694 \text{ m}^2
\]

Substitute \( S_{net} \) in \( C_{D,0} \) equation

\[
C_{D,0} = C_{le} \times \frac{S_{net}}{S}
\]

\[
= 4.74 \times 10^{-3} \left( \frac{0.1694}{0.0845} \right)
\]

\[
= 9.50 \times 10^{-3}
\]

To find \( C_D \) from the drag polar equation

\[
C_D = C_{D,0} + K CL^2
\]

where

\[
K = \frac{1}{\pi e AR}
\]

From historical values

\[
C_{L,\text{max}} = 3.15
\]

\[
\text{Drag Polar } C_{D,0} = 5.90 \times 10^{-3}
\]

\[
\text{Aspect Ratio } (AR) = 9.5
\]

\[
\text{Ostwald efficiency} = 0.8
\]

\[
K = 0.0419
\]

\[
C_D = C_{D,0} + K CL^2
\]

\[
= 0.4252
\]

### 4.4 To find (L/D)\(_{\text{max}}\)

\[
L/D = \sqrt{\frac{1}{4KC_{D,0}}}
\]

\[
(L/D)_{\text{max}} = 25.6
\]

### 4.5 Wing selection

For this paper the tapered wing is selected.

### 4.6 Wing Span:

Wing span of an aircraft is the distance from one wing tip to another wing tip.

From aspect ratio

\[
AR = \frac{b^2}{s}
\]

where

\[
b \rightarrow \text{wing span}
\]

\[
s \rightarrow \text{wing area}
\]

\[
b^2 = AR \times S
\]
From the calculation
\[ C_{in} = 820.576 \]

PERFORMANCE CALCULATION

6.1 Length of Take-off Run

\[ T - D = \mu(w - L) + \left( \frac{w}{g} \right) + \left( \frac{dv}{dt} \right) \]
\[ = \left( T - D \right) - \mu(w_o - L) \]
\[ S_g = 10.522 \times 10^3 \text{ m} \]

6.1.1. Length of approach

\[ R = \frac{V_f^2}{0.2g} \]
\[ V_f \rightarrow 1.15 V_{stall} \]
\[ R = 22.075 \times 10^3 \text{ m} \]

Flare Height (H_f) \[ H_f = R(1-\cos \theta_a) \]

To find \( \theta_a \)

\[ \text{approach angle } \theta_a = \sin^{-1} \left( \frac{1}{L/D} \right) \left( \frac{R}{W} \right) \]
\[ \theta_a = 1.9098 \]
\[ H_f = 885.97 \text{ m} \]
\[ S_a = 50 \text{ ft} - h_f \]
\[ S_a = 1507.21 \text{ m} \]

Flare distance \[ S_f = R \sin \theta_f \]
\[ \theta_f = \theta_a \]
\[ = 885.97 \text{ m} \]

6.2 Calculation of ground Roll

\[ S_g = j N \left[ \frac{2}{\rho \infty} \times \frac{W}{S} \times \frac{1}{C_{L, max}} + \frac{j^2 (W/S)}{g \rho \infty (C_{L, max} \times \mu_t)} \right] \]

The Touch down Velocity should be less than \( jV_{stall} \)

\[ j = 1.15 \text{ for business jet} \]
\[ \mu_r \rightarrow \text{Assume runway with dry concrete and brake on} \]
\[ \mu_r = 0.4 \]
\[ N \rightarrow \text{Time increment for free roll. Free roll depends partly on pilot.} \]
\[ N = 1-3S \]
\[ N = 1S \]
\[ S_g = 649 \text{ m} \]

6.4 Power Loading

Power loading is defined as the ratio between the weight of the aircraft to the power required to the aircraft.
\[ W/P = 10.45 \text{ N} \]

CONCLUSION

The preliminary design calculation for the business jet aircraft is done. First the comparison data analysis is studied and the basic parameters like the weight estimation, the thrust calculation wing area, lift and drag calculation and performance calculation for the taken business jet is done and the results are taken.

REFERENCES