The fighter and attack airplanes category contains the following airplane types:

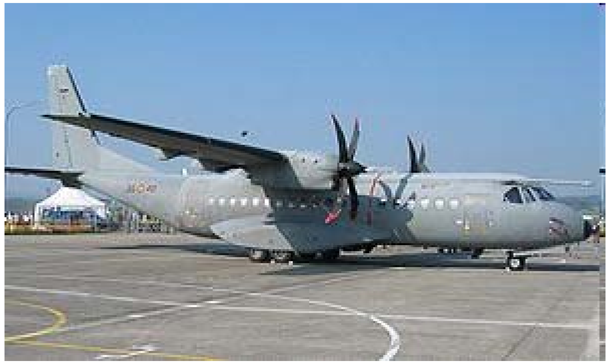
- High speed military trainers
- Fighters
- Supersonic fighters and attack airplanes

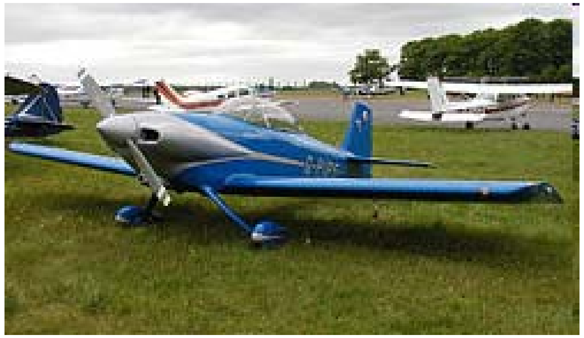
Estimación Pesos DAR Corp Fighter and Attack Airplanes

Sergio Esteban Roncero
Departamento de Ingeniería Aeroespacial
Y Mecánica de Fluidos

Estimación de Pesos (DAR Corp)

- Method for estimating the components of the airplane empty weight (DAR Corporation - Raymer).
- Las ecuaciones para la estimación de los pesos del ala se diferencian en función del tipo de avión al que se hace referencia siendo estos:
  - General Aviation Airplanes
  - Commercial Transport Airplanes
  - Military Patrol, Bomb and Transport Airplanes
  - Fighter and Attack Airplanes

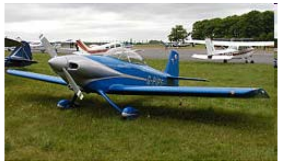




Structure Weight - II

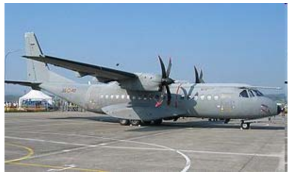
The General Aviation Airplanes category contains the following airplane types:

- Homebuilts
- Single engine props
- Multi-engine props
- Agricultural
- Regional turboprops below 12,500 pounds
- Low speed military trainers
- Small, low speed flying boats, amphibious airplanes, and float airplanes



The Military Patrol, Bomb and Transport Airplanes category contains the following types of airplanes:

- Military patrol airplanes
- Bombers
- Military transports
- Supersonic military patrol airplanes
- Supersonic bombers
- Supersonic military transports



The Commercial Transport Airplanes category contains the following types of airplanes:

- Business jets
- Regional turboprops above 12500 pounds
- Jet transports
- Large, high speed flying boats, amphibious airplanes, and float airplanes
- Commercial supersonic cruise airplanes (use Fighter inlet data)

The fighter and attack airplanes category contains the following airplane types:

- High speed military trainers
- Fighters
- Supersonic fighters and attack airplanes

Weight Estimation - Class II Method

- Class II Method for estimating the components of the airplane empty weight (DAR Corporation - Raymer).

$$W_E = W_{structure} + W_{pp} + W_{fix}$$

where:

$W_{structure}$	is the Class II airplane structure weight.
W_{pp}	is the Class II airplane powerplant weight.
W_{fix}	is the Class II airplane fixed equipment weight.

$$W_{TO} = \frac{W_E + W_{PL} + W_{crew} + W_{PL_{exp}} - W_{F_{refuel}}}{M_{ff}\left(1 + M_{F_{res}}\right) - M_{F_{res}} - M_{tfo}}$$

where:

W_E	is the airplane empty weight.
W_{PL}	is the payload weight.
W_{crew}	is the crew weight.
$W_{PL_{exp}}$	is the total expenditure payload weight.
$W_{F_{refuel}}$	is the total refueled fuel weight.
M_{ff}	is the mission fuel fraction.
$M_{F_{res}}$	is the reserved fuel weight as a fraction of fuel weight used in the mission.
M_{tfo}	is the trapped fuel and oil weight as a fraction of airplane take-off weight.

- Divididos en:
  - Class II Structure Weight
  - Class II Powerplant Weight
  - Class II Fixed Equipment Weight

Class II – Structure Weight

- Organizado según:
  - Wing
  - Horizontal Tail
  - Vertical Tail
  - V-tail
  - Canard
  - Fuselage
  - Landing gear
  - Nacelles
  - Tailboom

$$W_{structure} = W_w + W_h + W_v + W_{vee} + W_c + W_f + W_{tboom} + W_n + W_{gear}$$

where:

W_w	is the wing weight.
W_h	is the horizontal tail weight.
W_v	is the vertical tail weight.
W_{vee}	is the V-Tail weight.
W_c	is the canard weight.
W_f	is the fuselage weight.
W_{tboom}	is the tail boom weight.
W_n	is the nacelle weight.
W_{gear}	is the landing gear weight.

- Las ecuaciones para la estimación de los pesos de los diferentes elementos se diferencian en función del tipo de avión al que se hace referencia:
  - General Aviation Airplanes
  - Commercial Transport Airplanes
  - Military Patrol, Bomb and Transport Airplanes
  - Fighter and Attack Airplanes
- Para el diseño de aviones de patrulla y transporte militar se utilizan los métodos GD y Torenbeek para aviones de transporte comercial con la salvedad que el factor de carga último (ultimate load factor) usado es de 4.5.
- El peso de diseño empleado para la estimación no es el de despegue (takeoff) sino el peso de la misión (flight design weight).
- **Military Patrol, Bomb and Transport Airplanes**
  - Los métodos GD y Torenbeek methods empleados para la estimación de los pesos del Commercial Transport Airplanes son los utilizados para determinar los pesos para los aviones de patruya militar y de transporte con la salvedad de que el factor de carga último empleado es de 4.5.
  - Mientras que el peso de diseño para tanto aviones de transporte comerciales como de aviones legeros se emplea el peso en despegue, para aviones militares, el peso empleado para el diseño del ala se emplea el peso de diseño para la misión (flight design gross).

General Aviation Airplane

- Unidades Imperiales en todos los parámetros
- Tres métodos:
  - Cessna Method
    - Ecuaciones válidas para avionres pequeños, de actuaciones moderadas y con velocidades por deba
    - El peso de los carenados de punta de ala y las superficies de controestá incluido
  - USAF Method
    - Ecuaciones válidas para pequeños aviones utilitarios con actuaciones de hasta 300 knots.
  - Torenbeek Method
    - Ecuaciones válidas para aviones de transporte ligero con pesos al despegue por debajo de 12,500 lbs (55,603 N).

where:

W_w	is the wing weight.
W_h	is the horizontal tail weight.
W_v	is the vertical tail weight.
W_{vee}	is the V-Tail weight.
W_c	is the canard weight.
W_f	is the fuselage weight.
W_{tboom}	is the tail boom weight.
W_n	is the nacelle weight.
W_{gear}	is the landing gear weight.

The General Aviation Airplanes category contains the following airplane types:

- Homebuilts
- Single engine props
- Multi-engine props
- Agricultural
- Regional turboprops below 12,500 pounds
- Low speed military trainers
- Small, low speed flying boats, amphibious airplanes, and float airplanes

Structure Weight – Wing - I

Structure Weight for: Transport Airplanes

Military transport airplanes use a load factor of 4.5

Structure Weight – Wing - I

- GD method:
  - El peso de las superficies hipersustentadoras y de los alerones está incluido

$$W_{wGD} = (1+F_{corr})\frac{0.00428\, S_w^{0.48} AR_w M_H^{0.43} (W_{TO} n_{ult})^{0.84} \lambda_w^{0.14}}{\left(100\left(\frac{t}{c}\right)_{r_w}\right)^{0.76} \left(\cos\Lambda_{c/2_w}\right)^{1.54}}$$

where:

F_{corr}	is the wing weight correction factor.
S_w	is the wing area.
AR_w	is the wing aspect ratio.
M_H	is the maximum Mach number at sealevel.
W_{TO}	is the airplane take-off weight.
n_{ult}	is the airplane ultimate load factor.
λ_w	is the wing taper ratio.
$\left(\frac{t}{c}\right)_{r_w}$	is the wing root thickness ratio.
$\Lambda_{c/2_w}$	is the wing half chord sweep angle.

The maximum Mach number at sea-level is defined as: $M_H = \frac{V_{H_{eas}}}{a_{@SL}}$

where:

$V_{H_{eas}}$	is the equivalent maximum level speed.
$a_{@SL}$	is the speed of sound at sea-level.

Note: Both variables should have the same unit.

Válido solo en los siguientes rangos

M_H	0.4 to 0.8
AR_w	4 to 12
$\left(\frac{t}{c}\right)_{r_w}$	8% to 15%

Structure Weight – Wing - II

- **Torenbeek method:**
  - Aplicable para aviones de transporte ligero con un peso de despegue inferior a las 12500 lbs (55,603 N)
  - El peso de las superficies hipersustentadoras y de los alerones está incluido
  - Unidades Imperiales en todos los parámetros

$$W_{w_{Torenb}} = 0.0017(1+F_{corr})W_{MZF}\, n_{ult}^{0.55}\left(\frac{b_w}{\cos\Lambda_{c/2_w}}\right)^{0.75} X$$

where:

F_{corr}	is the wing weight correction factor.
W_{MZF}	is the maximum zero-fuel weight.
n_{ult}	is the airplane ultimate load factor.
b_w	is the wing span.
$\Lambda_{c/2_w}$	is the wing half chord sweep angle.

The variable X is:

$$X = \left\{1+\left(\frac{6.3\cos\Lambda_{c/2_w}}{b_w}\right)^{0.5}\right\}\left(\frac{b_w S_w}{t_{r_w} W_{MZF}\cos\Lambda_{c/2_w}}\right)^{0.30}$$

where:

b_w	is the wing span.
S_w	is the wing area.
t_{r_w}	is the wng root maximum thickness.
W_{MZF}	is the maximum zero-fuel weight.
$\Lambda_{c/2_w}$	is the wing half chord sweep angle.

The maximum zero-fuel weight is found from:

$$W_{MZF} = W_E + W_{crew} + W_{PL} + W_{PL_{exp}} + W_{tfo}$$

where:

W_E	is the airplane empty weight.
W_{crew}	is the crew weight
W_{PL}	is the payload weight.
$W_{PL_{exp}}$	is the expended payload weight.
W_{tfo}	is the trapped fuel and oil weight.

Structure Weight – Wing - III

The wing span is calculated from:

$$b_w = \sqrt{S_w AR_w}$$

where:

S_w is the wing area.

AR_w is the wing aspect ratio.

The wing root maximum thickness

$$t_{r_w} = \left(\frac{t}{c}\right)_{r_w} \left(\frac{2S_w}{b_w(1+\lambda_w)}\right)$$

where:

$\left(\frac{t}{c}\right)_{r_w}$ is the wing root thickness ratio.

S_w is the wing area.

b_w is the wing span.

λ_w is the wing taper ratio.

wing weight correction factor

$$F_{corr} = F_{corr_s} + F_{corr_e} + F_{corr_g} + F_{corr_f}$$

$F_{corr_s} = 2\%$ if the airplane has spoilers and speed brakes;

$F_{corr_e} = -5\%$ if the airplane has 2 wing-mounted engines;

$F_{corr_e} = -10\%$ if the airplane has 4 wing-mounted engines;

$F_{corr_g} = -5\%$ if the landing gear is not mounted under the wing;

$F_{corr_f} = 2\%$ if the wing has fowler flaps.

- Para “braced wings” se reduce el peso de las alas un 30%.
- La estimación del peso de las alas no incluye el peso de las riostras las cuales representan aproximadamente el 10% del peso del ala.

Structure Weight – Canard - I

- GD method:

The canard weight is found from:

$$W_{c_{GD}} = \left(1 - F_{c_{load}}\right) X_{GD} + F_{c_{load}} Y_{GD}$$

where:

$F_{c_{load}}$	is the canard load weight factor.
X_{GD}	is the first intermediate calculation parameter.
Y_{GD}	is the second intermediate calculation parameter.

The first intermediate parameter is given by:

$$X_{GD} = 0.0034 \left\{ \left(W_{TO} n_{ult}\right)^{0.813} S_c^{0.584} \left(\frac{b_c}{t_{r_c}}\right)^{0.033} \left(\frac{\bar{c}_w}{l_c}\right)^{0.28} \right\}^{0.915}$$

where:

W_{TO}	is the airplane take-off weight.
n_{ult}	is the airplane ultimate load factor.
S_c	is the canard area.
b_c	is the canard span.
t_{r_c}	is the canard root maximum thickness.
$\bar{c}_w$	is the wing mean geometric chord.
l_c	is the X-distance between the canard and wing mean geometric chord quarter chord points.

The canard root maximum thickness is found from:

$$t_{r_c} = \left(\frac{t}{c}\right)_{r_c} \left(\frac{2 S_c}{b_c \left(1 + \lambda_c\right)}\right)$$

where:

$\left(\frac{t}{c}\right)_{r_c}$	is the canard root thickness ratio.
S_c	is the canard area.
b_c	is the canard span.
λ_c	is the canard taper ratio.

Structure Weight – Canard - II

The X-distance between the canard and wing mean geometric chord quarter chord points

$$l_c = X_{apex_w} + x_{mgc_w} + \frac{\bar{c}_w}{4} - X_{apex_c} - x_{mgc_c} - \frac{\bar{c}_c}{4}$$

where:

X_{apex_w} is the X-coordinate of the wing apex.

x_{mgc_w} is the X-location of the wing mean geometric chord leading edge relative to the wing apex.

$\bar{c}_w$ is the wing mean geometric chord.

X_{apex_c} is the X-coordinate of the canard apex.

x_{mgc_c} is the X-location of the canard mean geometric chord leading edge relative to the canard apex.

$\bar{c}_c$ is the canard mean geometric chord.

The X-location of lifting surface mean geometric chord leading edge relative to the lifting surface apex is given by:

$$x_{mgc_{l.s.}} = y_{mgc_{l.s.}} \tan \Lambda_{LE_{l.s.}}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail or 'c' for canard.

$y_{mgc_{l.s.}}$ is the Y-distance between the liftng surface apex and the lifting surface mean geometric chord.

$\Lambda_{LE_{l.s.}}$ is the lifting surface leading edge sweep angle.

The Y-distance between the lifting surface apex and the lifting surface mean geometric chord is given by:

$$y_{mgc_{l.s.}} = \frac{b_{l.s.}}{6} \frac{(1+2\lambda_{l.s.})}{(1+\lambda_{l.s.})}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail or 'c' for canard.

$b_{l.s.}$ is the lifting surface span.

$\lambda_{l.s.}$ is the lifting surface taper ratio.

Structure Weight – Canard - III

The lifting surface leading edge sweep angle is computed from:

$$\Lambda_{LE_{l.s.}} = \tan^{-1}\left\{\tan\Lambda_{c/4_{l.s.}} + \frac{(1-\lambda_{l.s.})}{AR_{l.s.}(1+\lambda_{l.s.})}\right\}$$

where:

l.s.	stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail or 'c' for canard.
$\Lambda_{c/4_{l.s.}}$	is the lifting surface quarter chord sweep angle.
$\lambda_{l.s.}$	is the lifting surface taper ratio.
$AR_{l.s.}$	is the lifting surface aspect ratio.

The lifting surface mean geometric chord is given by:

$$\bar{c}_{l.s.} = \frac{4}{3}\frac{\left(1+\lambda_{l.s.}+\lambda_{l.s.}^2\right)}{(1+\lambda_{l.s.})^2}\sqrt{\frac{S_{l.s.}}{AR_{l.s.}}}$$

where:

l.s.	stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail, 'v' for vertical tail or 'c' for canard.
$S_{l.s.}$	is the lifting surface area.
$AR_{l.s.}$	is the lifting surface aspect ratio.
$\lambda_{l.s.}$	is the lifting surface taper ratio.

The lifting surface span is given by:

$$b_{l.s.} = \sqrt{AR_{l.s.}S_{l.s.}}$$

where:

l.s.	stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail, 'v' for vertical tail or 'c' for canard.
$S_{l.s.}$	is the lifting surface area.
$AR_{l.s.}$	is the lifting surface aspect ratio.

Structure Weight – Canard - IV

The second intermediate calculation parameter is computed from:

$$Y_{GD} = \frac{0.00428\, S_c^{0.48} AR_c M_H^{0.43} (W_{TO} n_{ult})^{0.84} \lambda_c^{0.14}}{\left(100\left(\frac{t}{c}\right)_{r_c}\right)^{0.76} \left(\cos \Lambda_{c/2_c}\right)^{1.54}}$$

where:

S_c	is the canard area.
AR_c	is the canard aspect ratio.
M_H	is the maximum Mach number at sea-level.
W_{TO}	is the airplane take-off weight.
n_{ult}	is the airplane ultimate load factor.
λ_c	is the canard taper ratio.
$\left(\frac{t}{c}\right)_{r_c}$	is the canard root thickness ratio.
$\Lambda_{c/2_c}$	is the canard half chord sweep angle.

Note: The above equation is only valid in the following parameter ranges:

M_H	0.4 to 0.8
AR_c	4 to 12
$\left(\frac{t}{c}\right)_{r_c}$	8% to 15%

The maximum Mach number at sealevel is defined as:

$$M_H = \frac{V_{H_{eas}}}{a_{@SL}}.$$

where:

$V_{H_{eas}}$	is the equivalent maximum level speed.
$a_{@SL}$	is the speed of sound at sea-level.

Note: Both variables should have the same unit.

Structure Weight – Canard - V

- **Torenbeek method:**
  - Aplicable para aviones de transporte ligero con un peso de despegue inferior a las 12500 lbs (55,603 N)

The canard weight is obtained from:

$$W_{c_{Torenb}} = \left(1 - F_{c_{load}}\right) X_{Torenb} + F_{c_{load}} Y_{Torenb}$$

where:

$F_{c_{load}}$ is the canard load weight factor.

X_{Torenb} is the first intermediate calculation parameter.

Y_{Torenb} is the second intermediate calculation parameter.

The first intermediate calculation parameter is computed from:

$$X_{Torenb} = K_c S_c \left\{ 3.81 \frac{S_c^{0.2} V_{D_{eas}}}{1000\sqrt{\cos \Lambda_{c/2_c}}} - 0.287 \right\}$$

$K_c = 1.0$ for fixed incidence canard.

$K_c = 1.1$ for variable canard.

where:

K_c is the canard weight constant.

S_c is the canard area.

$V_{D_{eas}}$ is the equivalent flight design dive speed.

$\Lambda_{c/2_c}$ is the canard half chord sweep angle.

The second intermediate calculation parameter is computed from:

$$Y_{Torenb} = 0.0017 W_{MZF} \left(\frac{b_c}{\cos \Lambda_{c/2_c}} \right)^{0.75} \left\{ 1 + \left(\frac{6.3 \cos \Lambda_{c/2_c}}{b_c} \right)^{0.5} \right\} n_{ult}^{0.55} \left(\frac{b_c S_c}{t_{r_c} W_{MZF} \cos \Lambda_{c/2_c}} \right)^{0.30}$$

where:

W_{MZF} is the maximum zero-fuel weight.

b_c is the canard span.

$\Lambda_{c/2_c}$ is the canard half chord sweep angle.

n_{ult} is the airplane ultimate load factor.

S_c is the canard area.

t_{r_c} is the canard root airfoil maximum thickness.

The maximum zero-fuel weight is found from:

$$W_{MZF} = W_E + W_{crew} + W_{PL} + W_{PL_{exp}} + W_{tfo}$$

where:

W_E is the airplane empty weight.

W_{crew} is the crew weight

W_{PL} is the payload weight.

$W_{PL_{exp}}$ is the expended payload weight.

W_{tfo} is the trapped fuel and oil weight.

Structure Weight – Horizontal Tail - I

- **GD method:**
  - Unidades Imperiales en todos los parámetros

The horizontal tail weight is found from:

$$W_{h_{GD}} = 0.0034\left\{\left(W_{TO}n_{ult}\right)^{0.813} S_h^{0.584}\left(\frac{b_h}{t_{r_h}}\right)^{0.033}\left(\frac{\bar{c}_w}{l_h}\right)^{0.28}\right\}^{0.915}$$

where:

W_{TO}	is the airplane take-off weight.
n_{ult}	is the airplane ultimate load factor.
S_h	is the horizontal tail area.
b_h	is the horizontal tail span.
t_{r_h}	is the horizontal tail root maximum thickness.
$\bar{c}_w$	is the wing mean geometric chord.
l_h	is the X-distance between the horizontal tail and wing mean geometric chord quarter chord points.

The horizontal tail root maximum thickness is found from:

$$t_{r_h} = \left(\frac{t}{c}\right)_{r_h}\left(\frac{2S_h}{b_h(1+\lambda_h)}\right)$$

where:

$\left(\frac{t}{c}\right)_{r_h}$	is the horizontal tail root thickness ratio.
S_h	is the horizontal tail area.
b_h	is the horizontal tail span.
λ_h	is the horizontal tail taper ratio.

The X-distance between the horizontal tail and wing mean geometric chord quarter chord points is determined from:

$$l_h = X_{apex_h} + x_{mgc_h} + \frac{\bar{c}_h}{4} - X_{apex_w} - x_{mgc_w} - \frac{\bar{c}_w}{4}$$

where:

X_{apex_h}	is the X-coordinate of the horizontal tail apex.
x_{mgc_h}	is the X-location of the horizontal tail mean geometric chord leading edge relative to the horizontal tail apex.
$\bar{c}_h$	is the horizontal tail mean geometric chord.
X_{apex_w}	is the X-coordinate of the wing apex.
x_{mgc_w}	is the X-location of the wing mean geometric chord leading edge relative to the wing apex.
$\bar{c}_w$	is the wing mean geometric chord.

Structure Weight – Horizontal Tail - II

The X-location of lifting surface mean geometric chord leading edge relative to the lifting surface apex is given by:

$$x_{mgc_{l.s.}} = y_{mgc_{l.s.}} \tan\Lambda_{LE_{l.s.}}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail or 'c' for canard.

$y_{mgc_{l.s.}}$ is the Y-distance between the lifitng surface apex and the lifting surface mean geometric chord.

$\Lambda_{LE_{l.s.}}$ is the lifting surface leading edge sweep angle.

The Y-distance between the lifting surface apex and the lifting surface mean geometric chord is given by:

$$y_{mgc_{l.s.}} = \frac{b_{l.s.}}{6}\frac{(1+2\lambda_{l.s.})}{(1+\lambda_{l.s.})}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail or 'c' for canard.

$b_{l.s.}$ is the lifting surface span.

$\lambda_{l.s.}$ is the lifting surface taper ratio.

The lifting surface leading edge sweep angle is computed from:

$$\Lambda_{LE_{l.s.}} = \tan^{-1}\left\{\tan\Lambda_{c/4_{l.s.}} + \frac{(1-\lambda_{l.s.})}{AR_{l.s.}(1+\lambda_{l.s.})}\right\}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail or 'c' for canard.

$\Lambda_{c/4_{l.s.}}$ is the lifting surface quarter chord sweep angle.

$\lambda_{l.s.}$ is the lifting surface taper ratio.

$AR_{l.s.}$ is the lifting surface aspect ratio.

The lifting surface mean geometric chord is given by:

$$\bar{c}_{l.s.} = \frac{4}{3}\frac{(1+\lambda_{l.s.}+\lambda_{l.s.}^2)}{(1+\lambda_{l.s.})^2}\sqrt{\frac{S_{l.s.}}{AR_{l.s.}}}$$

The lifting surface span is given by::

$$b_{l.s.} = \sqrt{AR_{l.s.}S_{l.s.}}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail, 'v' for vertical tail or 'c' for canard.

$S_{l.s.}$ is the lifting surface area.

$AR_{l.s.}$ is the lifting surface aspect ratio.

$\lambda_{l.s.}$ is the lifting surface taper ratio.

Structure Weight – Horizontal Tail - III

- **Torenbeek method:**
  - Aplicable solo para aviones de transporte y business jets con una velocidad de picado (dive speed) superior a 250 knots.

The horizontal tail weight is calculated from:

$$W_{h_{Torenb}} = K_h S_h \left\{ 3.81 \frac{S_h^{0.2} V_{D_{eas}}}{1000\sqrt{\cos \Lambda_{c/2_h}}} - 0.287 \right\}$$

where:

$K_h = 1.0$	for fixed incidence stabilizers.
$K_h = 1.1$	for variable incidence stabilizers.

where:

K_h	is a horizontal tail weight constant.
S_h	is the horizontal tail area.
$V_{D_{eas}}$	is the equivalent flight design dive speed.
$\Lambda_{c/2_h}$	is the horizontal tail half chord sweep angle.

Structure Weight – Vertical Tail - I

- **GD method:**
  - Unidades Imperiales en todos los parámetros

The vertical tail weight is determined from:

$$W_{v_{GD}} = 0.19\left\{X\left(1+\frac{Z_h}{b_v}\right)^{0.5}\left(W_{TO}n_{ult}\right)^{0.363} S_v^{1.089} M_H^{0.601} l_v^{-0.726}\right\}^{1.014}$$

where:

Z_h is the Z-distance from vertical tail root to horizontal tail root.

b_v is the vertical tail span.

W_{TO} is the airplane take-off weight.

n_{ult} is the airplane ultimate load factor.

S_v is the vertical tail area.

M_H is the maximum Mach number at sea-level.

l_v is the X-distance between the vertical tail and wing mean geometric chord quarter chord ponts.

The variable X is:

$$X = \left(1+\frac{S_r}{S_v}\right)^{0.217} AR_v^{0.337}\left(1+\lambda_v\right)^{0.363}\left(\cos\Lambda_{c/4_v}\right)^{-0.484}$$

where:

S_r is the rudder area.

S_v is the vertical tail area.

AR_v is the vertical tail aspect ratio.

λ_v is the vertical tail taper ratio.

$\Lambda_{c/4_v}$ is the vertical tail quarter chord sweep angle.

The rudder to vertical tail area ratio is defined as:

$$\frac{S_r}{S_v} = \frac{\eta_{o_r}-\eta_{i_r}}{1+\lambda_v}\left[2-\left(1-\lambda_v\right)\left(\eta_{o_r}-\eta_{i_r}\right)\right]\frac{c_r}{c_v}$$

where:

η_{o_r} is the rudder outboard station in terms of vertical tail span.

η_{i_r} is the rudder inboard station in terms of vertical tail span.

λ_v is the vertical tail taper ratio.

$\frac{c_r}{c_v}$ is the average rudder chord to vertical tail chord raio.

Structure Weight – Vertical Tail - II

The Z-distance from vertical tail root to horizontal tail root is determined from:

$$Z_h = \left| Z_{c_r/4_h} - Z_{apex_v} \right|$$

where:

$Z_{c_r/4_h}$	is the Z-coordinate of horizontal tail root quarter chord point.
Z_{apex_v}	is the Z-coordinate of vertical tail apex.
D_{fh}	is the average fuselage diameter in region of horizontal tail.

However,

$$\text{If } Z_h < \frac{D_{fh}}{2}, \quad \text{then } Z_h = \frac{D_{fh}}{2}$$

where:

$Z_{c_r/4_h}$	is the Z-coordinate of horizontal tail root quarter chord point.
Z_{apex_v}	is the Z-coordinate of vertical tail apex.
D_{fh}	is the average fuselage diameter in region of horizontal tail.

Note: If there is no horizontal tail, $Z_h = 0$

The maximum Mach number at sealevel is defined as:

$$M_H = \frac{V_{H_{eas}}}{a_{@SL}}$$

where:

$V_{H_{eas}}$	is the equivalent maximum level speed.
$a_{@SL}$	is the speed of sound at sea-level.

Note: Both variables should have the same unit.

The X-distance between the vertical tail and wing mean geometric chord quarter chord points is given by:

$$l_v = X_{apex_v} + x_{mgc_v} + \frac{\bar{c}_v}{4} - X_{apex_w} - x_{mgc_w} - \frac{\bar{c}_w}{4}$$

where:

X_{apex_v}	is the X-coordinate of the vertical tail apex.
x_{mgc_v}	is the X-location of the vertical tail mean geometric chord leading edge relative to the vertical tail apex.
$\bar{c}_v$	is the vertical tail mean geometric chord.
X_{apex_w}	is the X-coordinate of the wing apex.
x_{mgc_w}	is the X-location of the wing mean geometric chord leading edge relative to the wing apex.
$\bar{c}_w$	is the wing mean geometric chord.

Structure Weight – Vertical Tail - III

The X-location of vertical tail mean geometric chord leading edge relative to the vertical tail apex is computed from:

$$x_{mgc_v} = z_{mgc_v} \tan \Lambda_{LE_v}$$

where:

z_{mgc_v} is the Z-distance between the vertical tail apex and the vertical tail mean geometric chord.

Λ_{LE_v} is the vertical tail leading edge sweep angle.

The Z-distance between the vertical tail mean geometric chord and the vertical tail apex is calculated from:

$$z_{mgc_v} = \frac{b_v}{3} \frac{(1+2\lambda_v)}{(1+\lambda_v)}$$

where:

b_v is the vertical tail span.

λ_v is the vertical tail taper ratio.

The vertical tail leading edge sweep angle is obtained from:

$$\Lambda_{LE_v} = \tan^{-1}\left\{\tan \Lambda_{c/4_v} + \frac{(1-\lambda_v)}{2AR_v(1+\lambda_v)}\right\}$$

where:

$\Lambda_{c/4_v}$ is the vertical tail quarter chord sweep angle.

λ_v is the vertical tail taper ratio.

AR_v is the vertical tail aspect ratio.

The X-location of wing mean geometric chord leading edge relative to the wing apex is computed from:

$$x_{mgc_w} = y_{mgc_w} \tan \Lambda_{LE_w}$$

where:

y_{mgc_w} is the Y-distance between the wing apex and the wing mean geometric chord.

Λ_{LE_w} is the wing leading edge sweep angle.

Structure Weight – Vertical Tail - IV

The Y-distance between the wing apex and the wing mean geometric chord is given by:

$$y_{mgc_w} = \frac{b_w}{6}\frac{(1+2\lambda_w)}{(1+\lambda_w)}$$

where:

b_w is the wing span.

λ_w is the wing taper ratio.

The wing leading edge sweep angle is obtained from:

$$\Lambda_{LE_w} = \tan^{-1}\left\{\tan\Lambda_{c/4_w} + \frac{(1-\lambda_w)}{AR_w(1+\lambda_w)}\right\}$$

where:

$\Lambda_{c/4_w}$ is the wing quarter chord sweep angle.

λ_w is the wing taper ratio.

AR_w is the wing aspect ratio.

The lifting surface mean geometric chord is calculated from:

$$\bar{c}_{l.s.} = \frac{4}{3}\frac{(1+\lambda_{l.s.}+\lambda_{l.s.}^2)}{(1+\lambda_{l.s.})^2}\sqrt{\frac{S_{l.s.}}{AR_{l.s.}}}$$

The lifting surface span is given by:

$$b_{l.s.} = \sqrt{AR_{l.s.}S_{l.s.}}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail, 'v' for vertical tail or 'c' for canard.

$S_{l.s.}$ is the lifting surface area.

$AR_{l.s.}$ is the lifting surface aspect ratio.

$\lambda_{l.s.}$ is the lifting surface taper ratio.

Structure Weight – Vertical Tail - V

- **Torenbeek method:**
  - Aplicable solo para aviones de transporte y business jets con una velocidad de picado (dive speed) superior a 250 knots.

The vertical tail weight is estimated from:

$$W_{v_{Torenb}} = K_v\, S_v \left\{ 3.81 \frac{S_v^{0.2}\, V_{D_{eas}}}{1000 \sqrt{\cos \Lambda_{c/2_v}}} - 0.287 \right\}$$

where:

K_h is a horizontal tail weight constant.

S_h is the horizontal tail area.

$V_{D_{eas}}$ is the equivalent flight design dive speed.

$\Lambda_{c/2_h}$ is the horizontal tail half chord sweep angle.

where:

S_h is the horizontal tail area.

Z_h' is the Z-location of horizontal tail relative to the fuselage centerline.

S_v is the vertical tail area.

b_v is the vertical tail span.

$$K_v = 1.0$$ for fuselage mounted horizontal tails or for airplanes without horizontal tail.

$$K_v = 1 + 0.15 \frac{S_h\, Z_h'}{S_v\, b_v}$$ for fin mounted horizontal tails

The Z-location of horizontal tail relative to the fuselage centerline is determined by:

$$Z_h' = \left| Z_{c_r/4_h} - \left(Z_{fc_h} + \frac{D_{f_h}}{2} \right) \right|$$

where:

$Z_{c_r/4_h}$ is the Z-coordinate of horizontal tail root quarter chord point.

Z_{fc_h} is the Z-coordinate of fuselage centerline in region of horizontal tail.

D_{f_h} is the average fuselage diameter in region of horizontal tail.

Structure Weight – Fuselage - I

- **GD method:**
  - Unidades Imperiales en todos los parámetros

The fuselage weight is found from:

$$W_{f_{GD}} = 10.43 K_{inl}^{1.42} \left(\frac{\bar{q}_D}{100}\right)^{0.283} \left(\frac{W_{TO}}{1000}\right)^{0.95} \left(\frac{L_f}{h_{f_{max}}}\right)^{0.71}$$

$K_{inl} = 1.25$ for airplanes with inlets in or on the fuselage for a buried engine installation.

$K_{inl} = 1.0$ for inlets located elsewhere.

where:

K_{inl} is an inlet location constant.

$\bar{q}_D$ is the design dive dynamic pressure.

W_{TO} is the airplane take-off weight.

L_f is the fuselage length.

$h_{f_{max}}$ is the maximum fuselage height.

The design dive dynamic pressure is given by:

$$\bar{q}_D = \frac{1}{2}\rho_{@SL}\left(1.689 V_{D_{eas}}\right)^2$$

where:

$\rho_{@SL}$ is the air density at sea-level.

1.689 is the conversion factor from kts to ft/s.

$V_{D_{eas}}$ is the equivalent design dive speed.

Structure Weight – Fuselage - II

- **Torenbeek method:**
  - Aplicable solo para aviones de transporte y business jets con una velocidad de picado (dive speed) superior a 250 knots.

The fuselage weight for tail-aft airplane is determined from:

$$W_{f_{Torenb}} = 0.021\left(K_f + K_{press} + K_{gear}\right)\left(\frac{V_{D_{eas}}\left(X_{apex_h} + \frac{c_{r_h}}{4} - X_{apex_w} - \frac{c_{r_w}}{4}\right)}{w_{f_{max}} + h_{f_{max}}}\right)^{0.5} S_{wet_f}^{1.2}$$

The fuselage weight for canard airplane is determined from:

$$W_{f_{Torenb}} = 0.021\left(K_f + K_{press} + K_{gear}\right)\left(\frac{V_{D_{eas}}\left(X_{apex_w} + \frac{c_{r_w}}{4} - X_{apex_c} - \frac{c_{r_c}}{4}\right)}{w_{f_{max}} + h_{f_{max}}}\right)^{0.5} S_{wet_f}^{1.2}$$

The fuselage weight for three surface airplane is determined from:

$$W_{f_{Torenb}} = 0.021\left(K_f + K_{press} + K_{gear}\right)\left(\frac{V_{D_{eas}}\left(X_{apex_h} + \frac{c_{r_h}}{4} - X_{apex_c} - \frac{c_{r_c}}{4}\right)}{w_{f_{max}} + h_{f_{max}}}\right)^{0.5} S_{wet_f}^{1.2}$$

where:

K_f	is a fuselage constant.
K_{press}	is a fuselage pressurization correction factor.
K_{gear}	is a fuselage gear attachment correction factor.
$V_{D_{eas}}$	is the equivalent design dive speed.
X_{apex_h}	is the X-coordinate of the horizontal tail apex.
c_{r_h}	is the horizontal tail root chord.
X_{apex_w}	is the X-coordinate of the wing apex.
c_{r_w}	is the wing root chord.
X_{apex_c}	is the X-coordinate of the canard apex.
c_{r_c}	is the canard root chord.
$w_{f_{max}}$	is the maximum fuselage width.
$h_{f_{max}}$	is the maximum fuselage height.
S_{wet_f}	is the fuselage wetted area.

Structure Weight – Fuselage - III

fuselage constant is a user-defined value.

$K_f = 1.10$ for freighter airplanes;

$K_f = 1.04$ for airplanes with rear fuselage mounted engines;

$K_f = 1.14$ for freighter airplanes with rear fuselage mounted engines;

$K_f = 1.00$ for all other airplanes.

The fuselage pressurization correction factor is given by:

$K_{press} = 0.08$ for pressurized airplanes;

$K_{press} = 0.00$ for non-pressurized airplanes.

The fuselage gear attachment correction factor is given by:

$K_{gear} = 0.07$ if the main gears are attached on the fuselage;

$K_{gear} = -0.04$ if the main gears are not attached on the fuselage.

The lifting surface root chord is calculated from:

$$c_{r_{l.s.}} = \frac{2}{1+\lambda_{l.s.}} \sqrt{\frac{S_{l.s.}}{AR_{l.s.}}}$$

where:

l.s. stands for 'lifting surface'. It denotes either 'w' for wing, 'h' for horizontal tail, 'v' for vertical tail or 'c' for canard.

$S_{l.s.}$ is the lifting surface area.

$b_{l.s.}$ is the lifting surface span.

$\lambda_{l.s.}$ is the lifting surface taper ratio.

Structure Weight – Landing Gear - I

- **GD method:**
  - Unidades Imperiales en todos los parámetros

The gear weight is found from:

$$W_{g_{GD}} = 62.21\left(\frac{W_{TO}}{1000}\right)^{0.84}$$

where:

W_{TO} is the airplane take-off weight.

- **Torenbeek method:**
  - **Aplicable para aviones y business jets con el tren de aterrizaje principal montado en el al y el tren delantero montado en el fuselaje. Cada grupo se evalúa de forma distinta**

The gear weight is computed from:

$$W_{g_{Torenb}} = W_{mg_{Torenb}} + W_{ng_{Torenb}} + W_{tg_{Torenb}}$$

where:

$W_{mg_{Torenb}}$ is the main gear according to Torenbeek method.

$W_{ng_{Torenb}}$ is the nose gear according to Torenbeek method.

$W_{tg_{Torenb}}$ is the tail gear according to Torenbeek method.

Structure Weight – Landing Gear - II

rection factor is determined from: y:

$$W_{xg_{Torenb}} = K_{g_r} \{A_{xg_{Torenb}} + B_{xg_{Torenb}} W_{TO}^{0.75} + C_{xg_{Torenb}} W_{TO} + D_{xg_{Torenb}} W_{TO}^{1.5}\}$$

where, xg = mg for main gear,
xg = ng for nose gear,
xg = tg for tail gear.

Note: $B_{xg_{Torenb}}$ and $D_{xg_{Torenb}}$ are zero for the tail gear.

where:

z_f	is the fuselage height at wing root.
Z_{fc_w}	is the Z-coordinate of fuselage centerline in the region of wing.
$Z_{c_r/4_w}$	is the Z-coordinate of wing root quarter chord point.

$$K_{g_r} = 1 + 0.08\left[\frac{0.5z_f - \left(Z_{fc_w} - Z_{c_r/4_w}\right)}{z_f}\right]$$

The above equation yields:

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

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

where:

K_{g_r}	is the landing gear weight wing location correction factor.
xg	denotes 'mg' for main gear, 'n' for nose gear or 't' for tail gear.
$A_{xg_{Torenb}}$	is the Torenbeek constant A for main, nose or tail gear.
$B_{xg_{Torenb}}$	is the Torenbeek constant B for main, nose or tail gear.
$C_{xg_{Torenb}}$	is the Torenbeek constant C for main, nose or tail gear.
$D_{xg_{Torenb}}$	is the Torenbeek constant D for main, nose or tail gear.
W_{TO}	is the airplane take-off weight.

Landing Gear Weight Constants

Airplane Type	Gear Type	Gear Comp.	A_g	B_g	C_g	D_g
Jet Trainers and Business Jets	Retr.	Main	33.0	0.04	0.021	0.0
		Nose	12.0	0.06	0.0	0.0
Other civil airplanes	Fixed	Main	20.0	0.10	0.019	0.0
		Nose	25.0	0.0	0.0024	0.0
		Tail	9.0	0.0	0.0024	0.0
	Retr.	Main	40.0	0.16	0.019	0.000015
		Nose	20.0	0.10	0.0	0.000002
		Tail	5.0	0.0	0.0031	0.0

Structure Weight – Nacelle - I

- El peso de la góndola viene determinado por:
  - Para motores colgados: el peso estructural incluye los conductos externos y/o carenados, incluyendo el peso de cualquier pilon.
  - Para motores con hélice: el peso estructural incluye el peso estructural asociado de los conductos externos y/o carenados, más el peso adicional de las estructuras necesarias para el montaje del motor.
  - Para motores encastrados: el peso estructural incluye los conductos especiales necesarios para realizar toda la instalación interna.

Cessna method

The nacelle weight is found from:

$$W_{n_{Cessna}} = K_{n_{Cessna}} SHP_{TO}$$

where:

$K_{n_{Cessna}}$ is a nacelle weight engine-type constant according to Cessna method.

SHP_{TO} is the total required take-off power.

$$K_{n_{Cessna}} = 0.37 \frac{lbs}{hp}$$ for radial engines and piston engines mounted in the fuselage nose.

$$K_{n_{Cessna}} = 0.24 \frac{lbs}{hp}$$ for horizontally opposed cylinder engines.

Structure Weight – Nacelle - II

Torenbeek method

The nacelle weight is found from:

$$W_{n_{Torenb}} = K_{n_{Torenb}} \left(N_{eng}\right)^{n_{e_{Torenb}}} \left(SHP_{TO}\right)^{n_{p_{Torenb}}}$$

where typical values for the factors are:

Engine Type	$K_{n_{Torenb}}$	$n_{e_{Torenb}}$	$n_{p_{Torenb}}$
Single Engine	2.50	0.0	0.50
Multi Engines			
Horizontally Opposed Piston Engines	0.320	0.0	1.00
Radial Piston Engines	0.045	-0.25	1.25
Turboprop Engines	0.140	0	1.00

where:

$K_{n_{Torenb}}$ is the engine type correction factor for nacelle weight according to Torenbeek method.

N_{eng} is the number of engines.

$n_{e_{Torenb}}$ is the engine quantity exponent for nacelle weight according to Torenbeek method.

SHP_{TO} is the total required take-off power.

$n_{p_{Torenb}}$ is the engine type exponent for nacelle weight according to Torenbeek method.

Notes:

1. These weights include the weights of all nacelles.
2. If the main landing gear retracts into the nacelles, 0.04 lbs/hp is added to the nacelle weight.
3. If the engine exhausts over the wing, as in the Lockheed Electra, add 0.11 lbs/hp to the nacelle weight.

Fixed Equipment

- Flight Control System
- Hydraulic and Pneumatic System
- Instrumentation, Avionics and Electronics
- Electrical System
- Air-conditioning, Pressurization, Anti- and De-icing System
- Oxygen System
- Auxiliary Power Unit
- Furnishings
- Baggage and Cargo Handling Equipment
- Operational Items
- Other Items

Fixed Equipment - Flight Control System - I

GD method

The flight control system weight for military transport airplanes is found from:

$$W_{fcs_{GD}} = 15.96\left(\frac{W_{TO}\bar{q}_D}{100000}\right)^{0.815} + W_{cgCtrl}$$

where:

W_{TO} is the airplane take-off weight.

$\bar{q}_D$ is the design dive dynamic pressure.

W_{cgCtrl} is the center of gravity control system weight.

The design dive dynamic pressure is computed from:

$$\bar{q}_D = \frac{1}{2}\rho_{@SL}\left(1.689 V_{D_{eas}}\right)^2$$

where:

$\rho_{@SL}$ is the air density at sea-level.

1.689 is the conversion factor from kts to ft/s.

$V_{D_{eas}}$ is the equivalent design dive speed.

The flight control system weight for bombers is obtained from:

$$W_{fcs_{GD}} = 1.049\left(\frac{S_{cs}\bar{q}_D}{1000}\right)^{1.21} + W_{cgCtrl}$$

where:

S_{cs} is the total area of the control surfaces.

$\bar{q}_D$ is the design dive dynamic pressure.

W_{cgCtrl} is the center of gravity control system weight.

Fixed Equipment - Flight Control System - II

The total area of the control surfaces is given by:

$$S_{cs} = 2S_a + S_e + S_{cv} + N_v S_r$$

where:

S_a	is the aileron area. Note that this is the area of only ONE panel.
S_e	is the elevator area.
S_{cv}	is the canardvator area.
S_r	is the rudder area.
N_v	is the number of vertical tails.

The center of gravity control system weight is calculated from:

$$W_{cgCtrl} = K_{cgCtrl}\left(0.01\frac{W_{F_{max_w}}}{\rho_F}\right)^{0.442}$$

where:

K_{cgCtrl}	is the center of gravity control system weight factor.
$W_{F_{max_w}}$	is the maximum fuel weight limited by the fuel tank volume.
ρ_F	is the fuel density.

Note: These estimates include the weight of all associated hydraulic and/or pneumatic systems.

Hydraulic and Pneumatic System

Military patrol, transport and bombers: $0.0060 - 0.0120$ of W_{TO}

where:

$\frac{W_{paint}}{W_{TO}}$	is the paint weight to take-off weight ratio.
W_{TO}	is the airplane take-off weight.

Instrumentation, Avionics and Electronics

GD method (modified)

The weight of instruments is found from:

$$W_{iae_{GD}} = \left(N_{Captain} + N_{CoPilot} + N_{FltEngr}\right)\left(15 + \frac{0.032W_{TO}}{1000}\right) + N_{eng}\left(5 + \frac{0.006\,W_{TO}}{1000}\right) + \frac{0.15\,W_{TO}}{1000} + 0.012W_{TO}$$

where:

$N_{Captain}$ is the number of captains.

$N_{CoPilot}$ is the number of co-pilots.

$N_{FltEngr}$ is the number of flight engineers.

W_{TO} is the airplane take-off weight.

N_{eng} is the number of engines.

Torenbeek method

The weight of instrumentation, avionics and electronics for propeller driven transports is given from:

$$W_{iae_{Torenb}} = 120 + 20N_{eng} + 0.006W_{TO}$$

where:

N_{eng} is the number of engines.

W_{TO} is the estimated airplane take-off weight.

For jet transports, the instrumentation, avionics and electronics weight is determined from:

$$W_{iae_{Torenb}} = 0.575W_E^{0.556}R_{\max}^{0.25}$$

where:

W_E is the airplane empty weight.

$R_{\max}$ is the airplane maximum range.

Electrical System

GD method

For Transport airplanes, the electrical system weight is found from:

The electrical system weight is obtained from:

$$W_{els_{GD}} = 1163\left(\frac{W_{fs} + W_{iae}}{1000}\right)^{0.506}$$

where:

W_{fs} is the fuel system weight.

W_{iae} is the instrumentation, avionics and electronics weight.

GD method

The air-conditioning, pressurization, anti- and de-icing system weight is found from:

$$W_{api_{GD}} = K_{api} \left(\frac{V_{press}}{100} \right)^{0.242}$$

where:

K_{api} is the air-conditioning, pressurization, anti-icing and de-icing system weight correction factor.

V_{press} is the airplane pressurized volume.

where:

	K_{api}
For subsonic airplanes with anti-icing,	887
For subsonic airplanes without anti-icing,	610
For supersonic airplanes without anti-icing,	748

Note: Low subsonic airplanes are the airplanes with a design cruise Mach number lower than 0.4 whereas high subsonic airplanes have a design cruise Mach number higher than 0.5. For airplanes with a design cruise Mach number in between 0.4 to 0.5, the values of the factors shown in the table above might be interpolated.

Oxygen System

GD method

The oxygen system weight is found from:

$$W_{ox_{GD}} = 7\left(N_{crew} + N_{pax}\right)^{0.702}$$

where:

N_{crew} is the number of crew.

N_{pax} is the number of passengers.

Torenbeek method

The oxygen weight is determined from:

$$W_{ox_{Torenb}} = W_{ox_{fixed}} + K_{ox}\left(N_{crew} + N_{pax}\right)$$

The typical values for the factors are:

	$W_{ox_{fixed}}$	K_{ox}
For flights below 25,000 ft	20	0.5
For short flights above 25,000 ft	30	1.2
For extended overwater flights	40	2.4

where:

$W_{ox_{fixed}}$ is the oxygen system fixed weight depending on airplane flight altitude and range.

K_{ox} is the passenger number correction factor for oxygen system weight.

N_{crew} is the number of crew.

N_{pax} is the number of passengers.

Auxiliary Power Unit

From the detailed weight statements in Appendix A of *Airplane Design Part V*, it is possible to derive weight fractions for auxiliary power units as a function of the take-off weight:

$$W_{apu} = (0.004 \text{ to } 0.013) W_{TO}$$

where:

$\frac{W_{paint}}{W_{TO}}$ is the paint weight to take-off weight ratio.

W_{TO} is the airplane take-off weight.

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