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Flight loads analysis of horizontal tail for single propeller engine aircraft

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Abbreviations and acronyms

Abbreviation	Definition
CFR	Code of Federal Regulations
CG	Aircraft center of gravity
α	Aircraft angle of attack
β	Aircraft sideslip angle
P	Roll rate
Q	Pitch rate
R	Yaw rate
$\dot{P}$	Rolling acceleration
$\dot{Q}$	Pitching acceleration
$\dot{R}$	Yawing acceleration
HT	Horizontal tail
S	Reference wing area
S_H	Horizontal tail area
S_p	Area of the Propeller
L_H	X-distance from 1/4 of wing MAC to 1/4 of HT MAC
L_p	X-distance from propeller aerodynamic center to 1/4 of wing MAC
L_{CG}	X-distance from 1/4 of wing MAC to CG of aircraft
CL_{wb}	Lift coefficient for wing body
CL_H	Horizontal tail lift coefficient
CZ_p	Propeller lift coefficient
α_p	Propeller angle of attack
α_{WB}	Wing body angle of attack
α_H	The horizontal tail angle of the attack
α_{RF}	Rear fuselage angle of attack
ε_0	Downwash angle
δe	Elevator angle

$\frac{dCL}{d\delta_e}$	Elevator lift coefficient derivative
q	Dynamic pressure
Iyy	Mass moment of inertia along the Y-direction
SLZ	Shear load in Z-direction
BMX	Bending moment in X-direction
TMY	Torsion moment in Y-direction
MAC	Mean Aerodynamic Chord

Abstract

Airframe load analysis is a critical aspect of aircraft development, especially while designing light and high-strength aircraft structures, as it requires accurate load evaluation. Once determined, loads are applied to size-respective structures, making the loads group the heart of the design cycle. This team works in a multidisciplinary environment, collaborating with various groups such as aerodynamics, weight and balance, flying qualities, systems, structures, airworthiness, etc.

This paper explores a simple hand calculation-based approach for load calculation, offering both speed and accuracy, making it particularly useful in the early stages of aircraft development. This methodology for the load computation is generic, but an example of a horizontal tail for a single propeller engine aircraft is considered to illustrate the process. The load calculations are in accordance with 14 CFR Part 23 amendment 63, Ref [1].

Introduction

In an aircraft, the horizontal tail is used for longitudinal stability and loads applied on it affect the design of various parts of the aircraft structure, necessitating careful consideration of the various design requirements and resulting conditions. In general, the structures that are designed by horizontal tail loads are:

- The horizontal tail stabilizer and elevator
- The aft fuselage monocoque structure
- The fuselage center section (overwing) structure
- The horizontal stabilizer actuator

Figure 1. Typical single engine propeller aircraft with horizontal tail highlighted, Ref [3]

Objective

This paper aims to provide a comprehensive overview of the methodology used for load analysis used for analyzing horizontal tail loads in compliance with 14 CFR Part 23 regulations. By utilizing simplified aerodynamic and inertia models, the load analysis can be conducted more efficiently, facilitating a faster and more integrated process during the conceptual and preliminary design phase of aircraft development.

Challenges

Several challenges arise during the load analysis process:

- Due to its multidisciplinary nature, it involves interactions like aerodynamics and structure, requiring systematic work. This process demands the ability to handle massive amounts of data, perform data reduction and maintain a deep understanding of both aerodynamic and structural issues.

- Communication with other disciplines within the company and with certification authorities is also challenging.

Analysis methodology

The horizontal tail loads are evaluated by a flight physics-based approach for symmetrically balanced maneuver, checked maneuver, unchecked (abrupt) maneuver, vertical gust and unsymmetrical loads as per 14 CFR Part 23 amendment 63, Ref [1].

A simple flowchart for the load analysis methodology employed in this paper is shown in Figure 2.

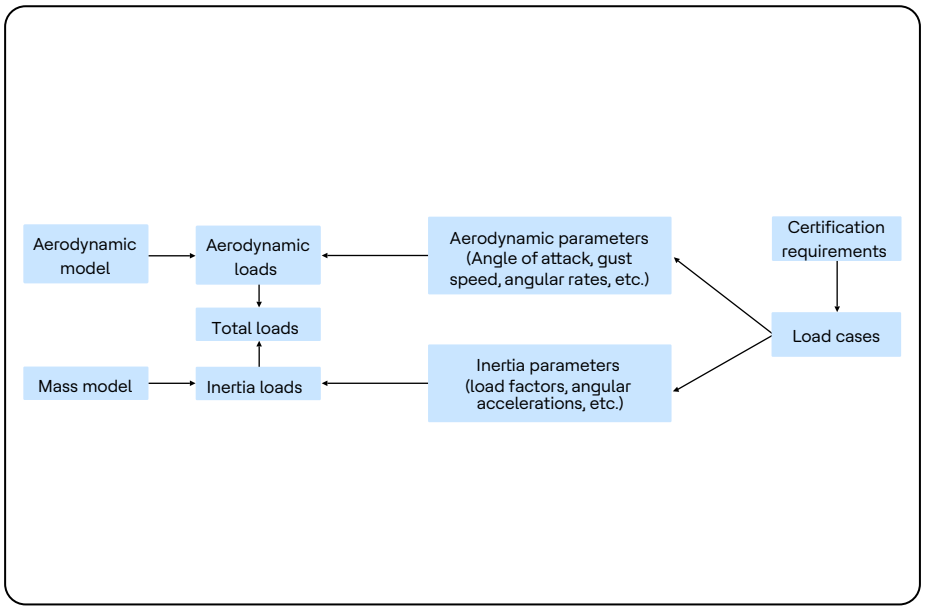

Figure 2. Load analysis methodology

Aerodynamic loads

Symmetric balanced maneuver loads

As per part 23 regulation §23.421, balancing tail load is necessary to maintain the aircraft under equilibrium (steady flight) with no pitching acceleration. Following are the formulations used for the balanced maneuver loads analysis from Ref [2].

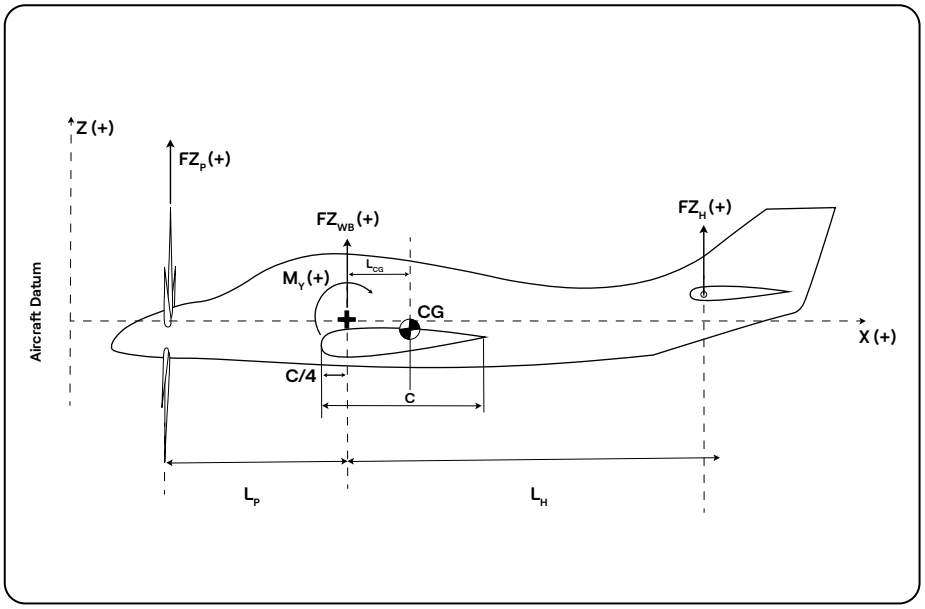

Figure 3. Schematic diagram of symmetrically balanced maneuver loads analysis setup

Note: Variable 'C' in Figure 3 is the same as MAC (Mean Aerodynamic Chord of Wing)

Summation of forces equal to zero: Σ Fz = 0

$$S.CL_{WB} + S_H.CL_H + S_P.CZ_P = \frac{M.g.Nz}{q}$$

$$\boldsymbol{CL_H = \frac{M.g.Nz}{S_H.q} - \frac{S.CL_{WB}}{S_H} - \frac{S_P.CZ_P}{S_H}} \qquad \langle 1 \rangle$$

Summation of moments equal to zero: Σ My = 0 (about MAC/4)

$$S_H.CL_H.L_H = CM_{WB}.C.S + \frac{M.g.Nz.L_{CG}}{q} + CZ_P.S_P.L_P$$

$$\boldsymbol{CL_H = \frac{M.g.Nz.L_{CG}}{q.S_H.L_H} + \frac{CM_{WB}.C.S}{S_H.L_H} + \frac{CZ_P.S_P.L_P}{S_H.L_H}}$$

$\langle 2 \rangle$

Equating the above formulas $\langle 1 \rangle$ and $\langle 2 \rangle$

$$\boldsymbol{\frac{M.g.Nz}{S.q}\left(L_H - L_{CG}\right) = CL_{WB}.L_H + CM_{WB}.C + \frac{CZ_P.S_P}{S}\left(L_H + L_P\right)} \qquad \langle 3 \rangle$$

The coefficient of Lift and Moment for the Wing body is calculated as,

$$CL_{WB} = CL_{wb\alpha}.\alpha_{WB} + CL_{wb0}$$

$$CM_{WB} = CM_{wb\alpha}.\alpha_{WB} + CM_{wb0}$$

$$CZ_P = CZ_{P\alpha}.\alpha_P$$

Assuming $\alpha P = \alpha WB$

Alpha Wing body is calculated by equating the above equations in the equation 〈3〉,

$$\frac{M.g.Nz}{S.q}\left(L_H - L_{CG}\right) = CL_{wb\alpha}.\alpha_{WB}.L_H + CL_{wb0}.L_H + CM_{wb\alpha}.\alpha_{WB}.C + CM_{wb0}.C + \frac{CZ_{P\alpha}.\alpha_{WB}S_P}{S}\left(L_H + L_P\right)$$

$$\boldsymbol{\alpha_{WB} = \frac{\frac{M.g.Nz}{S.q}\left(L_H - L_{CG}\right) - CL_{wb0}.L_H - CM_{wb0}.C}{CL_{wb\alpha}.L_H + CM_{wb\alpha}.C + \frac{CZ_{P\alpha}.S_P}{S}\left(L_H + L_P\right)}}$$

The total Lift on the Wing, horizontal tail and propeller is:

$$FZ_{WB} = CL_{WB}.q.S \qquad FZ_{H_eq} = CL_H.q.S_H \qquad FZ_P = CZ_P.q.S_P$$

Total Lift on aircraft is:

$$\boldsymbol{FZ_{aircraft} = FZ_{WB} + FZ_{H_eq} + FZ_P}$$

The formulation for the horizontal tail local angle of attack and the elevator displacement is given by

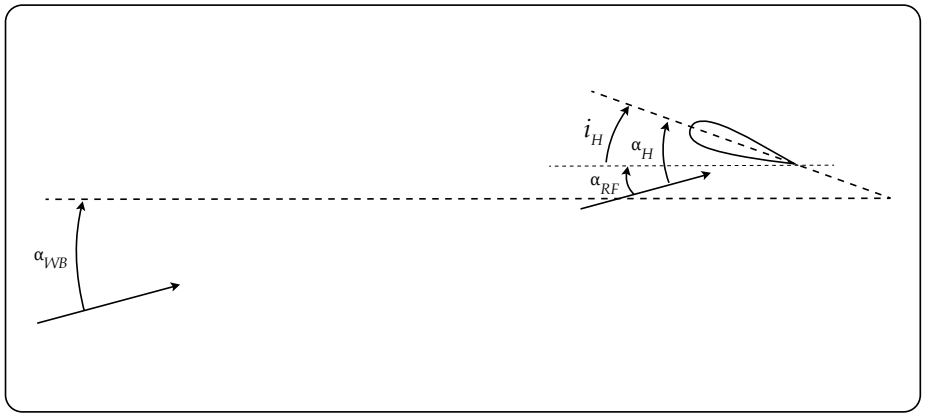

Figure 4. Schematic diagram of horizontal tail angle of attack

$$\alpha_{RF} = \alpha_{WB}\left(1 - \frac{d\varepsilon}{d\alpha}\right) - \varepsilon_0$$

$$\alpha_H = \alpha_{RF} + i_H$$

For fixed horizontal tail surface $i_H = 0$, hence $\alpha_H = \alpha_{RF}$

$$\Delta CL_H = CL_H - CL_{H1g}$$

$$\Delta\alpha_H = \alpha_H - \alpha_{H1g}$$

Change in elevator angle at equilibrium is given by:

$$\delta_{e_eq} = \frac{\Delta CL_H - CL_{\alpha H} \cdot \Delta\alpha_H}{CL_{\delta e}}$$

Unchecked (abrupt) maneuver loads

According to §23.423(a) Ref [1], unchecked maneuver loads on the horizontal tail are calculated as

$$\Delta FZ_H = q.S_H.\left(\frac{dCL}{d\delta_e}\right).\delta_e$$

Where δ_e is the maximum displacement of the elevator at maneuvering speed, V_A.

The total load on the horizontal tail is the sum of balancing and unchecked maneuver loads given by

$$\boldsymbol{FZ_H = FZ_{H_eq} + \Delta FZ_H}$$

Checked maneuver loads

According to §23.423(b) Ref [1], checked maneuver loads on the horizontal tail are calculated as

$$\Delta FZ_H = \frac{\dot{Q}.Iyy}{L_H}$$

Where $\dot{Q}$ is pitching angular acceleration calculated by equations as shown in Table 1

Condition	Normal acceleration (n_z)	Angular acceleration (rad/sec²)
Nose-up pitching	1.0	$+39 \cdot \frac{n_m}{V} \cdot \left(n_m - 1.5\right)$
Nose-down pitching	n_m	$-39 \cdot \frac{n_m}{V} \cdot \left(n_m - 1.5\right)$

Table 1. Checked maneuver condition as per §23.423(b) Ref [1]

n_m = Maximum load factor at specific airspeed.

V=Specific airspeed in m/s ==> V_A *or* V_C *or* V_D

The total load on the horizontal tail is the sum of balancing and checked maneuver loads given by

$$\boldsymbol{FZ_H = FZ_{H_eq} + \Delta FZ_H}$$

Vertical gust loads

According to §23.425 Ref [1], vertical gust loads on the horizontal tail are calculated as

$$\Delta FZ_H = \frac{K_g \cdot U_{de} \cdot V \cdot a_H \cdot S_H}{498}\left(1 - \frac{d\epsilon}{d\alpha}\right)$$

Where,

Gust alleviation factor, Kg = 0.88μg /5.3+ μg

Airplane mass ratio, μg = 2(W/S)/ (ρo.C.a. g)

The density of air, Rho [Slugs/ft^3]

Mean Geometric Chord, C [ft] = (Root Chord + Tip Chord)/2

Acceleration due to Gravity, g [ft/s^2]

Lift curve slope, a [1/rad]

Airplane Equivalent speed, V [Knots]

Wing Loading, W/S [lb/ft²]

Ude [ft/s] = Derived gust velocities from § 23.333 (c), Ref [1],

Positive and Negative gusts of 50 ft/s at VC must be considered at altitudes between sea level and 20,000 ft. The gust velocity may be reduced linearly from 50 ft/s at 20,000ft to 25 ft/s at 50,000 ft.

Positive and Negative gusts of 25 ft/s at VD must be considered at altitudes between sea level and 20,000 ft. The gust velocity may be reduced linearly from 25 ft/s at 20,000ft to 12.5 ft/s at 50,000 ft.

$\left(1 - \frac{d\epsilon}{d\alpha}\right)$ = Downwash factor

The total load on the horizontal tail is the sum of balancing and vertical gust loads given by

$$\boldsymbol{FZ_H = FZ_{H_eq} + \Delta FZ_H}$$

Unsymmetrical loads

According to §23.427 Ref [1], the unsymmetrical load on the horizontal tail is calculated. The following procedure is used in the analysis:

- 100% of the maximum loading from the symmetrical flight conditions is assumed on one side of the horizontal tail
- 40% of the same loading is assumed on the other side of the horizontal tail

Inertia loads

For inertia loads calculation horizontal tail may be divided into discrete portions called mass items. The coordinates and mass values of each mass item must be known. The absolute acceleration vector of the i^{th} mass item is obtained as follows:

$$\left\{a_x\ a_y\ a_z\right\} = \left\{a_{CG,x}\ a_{CG,y}\ a_{CG,z}\right\} + \left\{\dot{P}\ \dot{Q}\ \dot{R}\right\} \times \left\{x_i - x_{CG}\ y_i - y_{CG}\ z_i - z_{CG}\right\} + \left\{P\ Q\ R\right\} \times \left(\left\{P\ Q\ R\right\} \times \left\{x_i - x_{CG}\ y_i - y_{CG}\ z_i - z_{CG}\right\}\right)$$

Angular accelerations and velocities due to flight and ground conditions must be considered. The inertia force vector of the ith mass item is then obtained by multiplying the absolute acceleration vector by the item mass.

The Δn_{Zlocal} is due to the pitch acceleration. The below equation gives the calculation of pitch acceleration.

$$\dot{Q} = -\frac{\Delta FZ_H \cdot L_H}{I_{YY}}$$

The below equation gives the incremental local load factor.

$$\Delta n_{Z_{local}} = -\frac{\dot{Q} \cdot L_H}{g}$$

Chordwise pressure distribution

The chordwise distribution of pressure due to a tail surface displacement is shown in Figure 5.

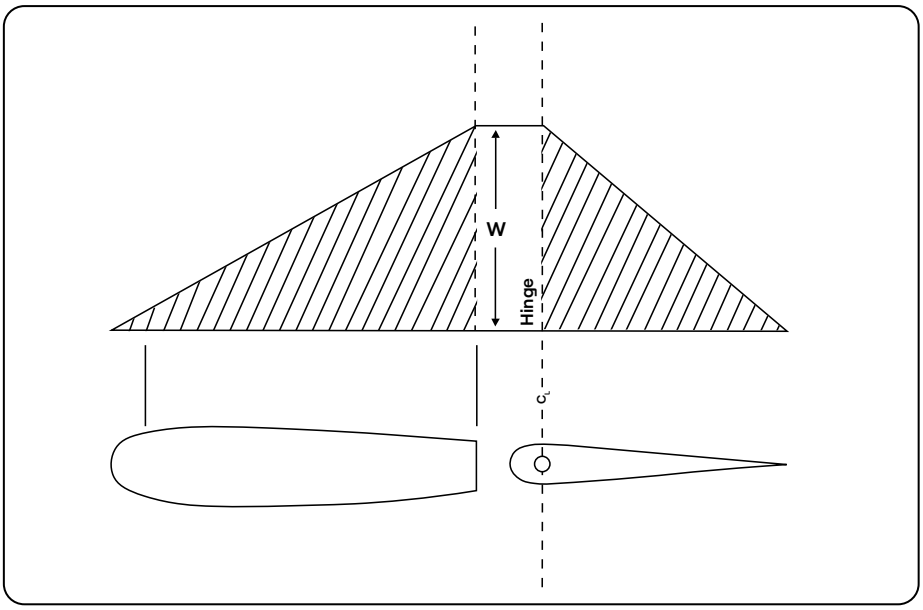

Figure 5. Distribution of pressure due to tail surface displacement, Ref [1]

The employed chordwise distribution of pressure due to a tail angle of attack is shown in Figure 6.

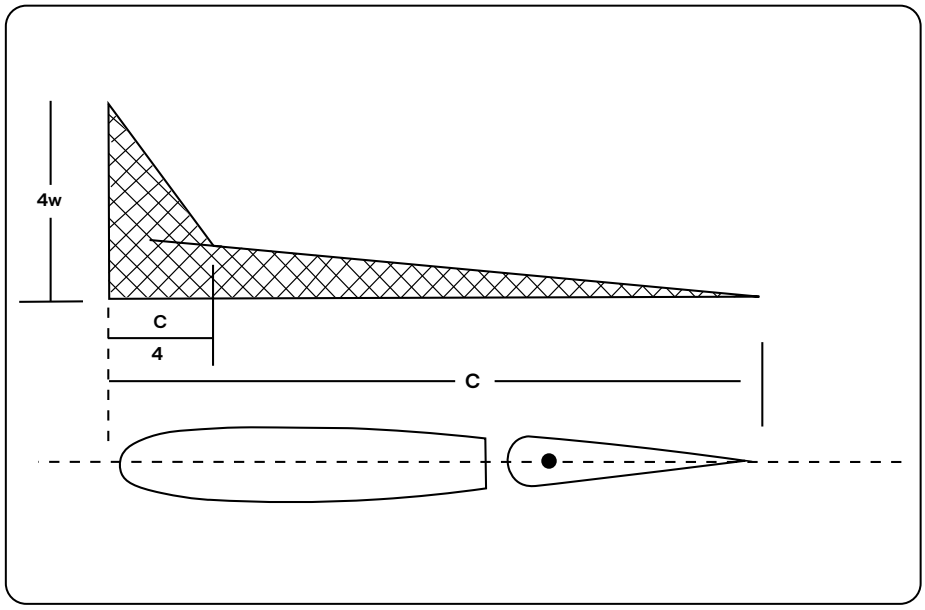

Figure 6. Distribution of pressure due to angle of attack, Ref [1]

The total chordwise distribution of pressure employed in the load calculations is the sum of these two distributions.

Results

Typical shear force, bending moment and torsion moment diagrams along the semi-span of the horizontal tail are shown in Figure 7

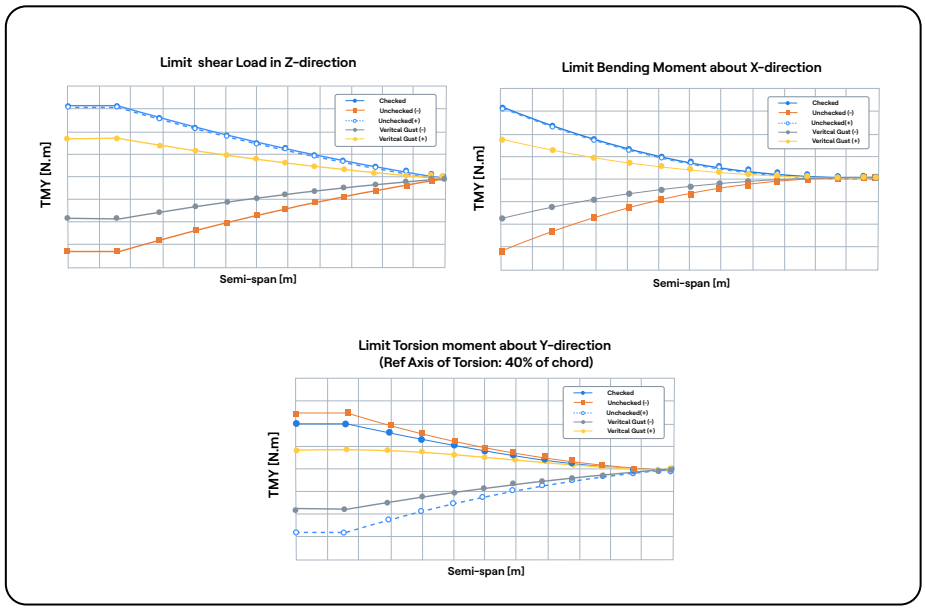

Figure 7. Typical shear force, bending moment and torsion moment diagram along the semi-span of HT

Conclusion

In general, load computation is a complex and time-consuming process. An attempt has been made to show the loads' analysis methodology for horizontal tail with a simplified approach as presented in this paper, considering various flight conditions in accordance with 14 CFR part 23 regulation, which makes the analysis much faster and can be used in the conceptual and preliminary design phase of the aircraft development program.

References

[1] 14 CFR Part 23 - Airworthiness Standards: Normal Utility, Acrobatic and Commuter Category Airplanes, Amendment 63, FAA.

[2] Lomax, T.L. "Structure Loads Analysis for Commercial Transport Aircraft: Theory and practice", AIAA Education Series, 1996.

[3] https://www.shutterstock.com/image-photo/sp-brazil-june-222019-demonstration-squadron-smoke-1431740900.

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