Chapter 6
Tail Design

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Chapter 6

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6.1. Introduction

As introduced in chapter 2, the next appropriate step after wing design would be the tail design. In this chapter, after describing the tail primary functions, and introducing fundamentals that govern the tail performance, techniques and procedure to design the horizontal tail and vertical tail will be provided. At the end of the chapter a fully solved example that illustrates the implementation of the design technique will be presented.

Horizontal tail and vertical tail (i.e. tails) along with wing are referred to as lifting surfaces. This name differentiates tails and wing from control surfaces namely aileron, elevator, and rudder. Due to this name, several design parameters associated with tails and wing; such as airfoil, planform area, and angle of attack; are similar. Thus, several tails parameters are discussed in brief. The major difference between wing design and tail design originates from the primary function of tail that is different from wing. Primary function of the wing to generate maximum amount of lift, while tail is supposed to use a fraction of its ability to generate lift. If at any instance of a flight mission, tail nears its maximum angle of attack (i.e. tail stall angle); it indicates that there was a mistake in the tail design process. In some texts and references, tail is referred to as empennage.

The ail in a conventional aircraft has often two components of horizontal tail and vertical tail and carries two primary functions:

1. Trim (longitudinal and directional)
2. Stability (longitudinal and directional)

Since two conventional control surfaces (i.e. elevator and rudder) are indeed parts of the tails to implement control, it is proper to add the following item as the third function of tails:

3. Control (longitudinal and directional)

These three functions are described in brief here; however, more details are presented in later sections. The first and primary function of horizontal tail is longitudinal trim; also referred to as equilibrium or balance. But the first and primary function of vertical tail is directional stability. The reason is that an aircraft is usually symmetric about xz plane, while the pitching moment of the wing about aircraft center of gravity must be balanced via a component.

Longitudinal trim in a conventional aircraft is applied through the horizontal tail. Several pitching moment, namely, longitudinal moment of the wing's lift about aircraft center of gravity, wing aerodynamic pitching moment, and sometimes engine thrust's longitudinal moment need to be trimmed about y axis. The summation of these three moments about aircraft center of gravity is often negative; hence the horizontal tail often generates a negative lift to counteract the moment. For this reason, the horizontal tail setting angle is often negative. Since the aircraft center of gravity is moving along x axis; due to fuel burn during flight duration; the horizontal tail is responsible for longitudinal trim throughout flight time. To support the longitudinal trimability of the aircraft, conventional aircraft employ elevator as part of its horizontal tail.

Since conventional aircraft are almost always manufactured symmetrically about xz plane, the trim is not a major function for vertical tail. However, in few instances, vertical tail has the primary function of directional trim or lateral trim. In a multi-engine aircraft, the vertical tail has great responsibility during one engine inoperative (OEI) situation in order to maintain directional trim. The vertical tail must generate a yawing moment to balance the aircraft for the yawing moment generated by active engines. Even in single engine prop-driven aircraft, the vertical has to counteract the rolling moment generated by propeller rotation. This is to maintain aircraft lateral trim and prevent an unwanted roll. For this case, the vertical tail has often installed with few degrees relative to xz plane. The aircraft trim requirement provides the main design requirements in the tail design process. The derivation of design requirements based on the trim will be discussed in details in Section 6.2.

The second function of the tails is to providing stability. The horizontal tail is responsible to maintain the longitudinal stability, while the vertical tail is responsible to maintain the directional stability. Aircraft stability is defined as the tendency of an aircraft to return to the original trim conditions if diverted by a disturbance. The major disturbance source is the atmospheric phenomena such as gust. The stability requirement must also be included in the tail design requirements' list. This topic will be discussed in details in Section 6.3.

The third major function of the tails is "*control*". The elevator as part of the horizontal tail is designed to provide longitudinal control, while the rudder as part of the vertical tail is responsible for providing the directional control. Tails must be powerful enough to control the aircraft such that the aircraft is able to change the flight conditions from one trim condition (say cruise) to another new trim condition (say take-off and landing). For instance, during take-off, the tail must be able to lift up the fuselage nose in a specified pitch rate.

In general, tail is designed based on the trim requirements, but later revised based on stability and control requirements. The following are the tail parameters which need to be determined during the design process:

1. Tail configuration
2. Horizontal tail horizontal location with respect to fuselage (aft tail or canard)

➢ **Horizontal tail**

3. Planform area (S_h)

4. Tail arm (l_h)
5. Airfoil section
6. Aspect ratio (AR_h)
7. Taper ratio (λ_h)
8. Tip chord (C_{h_tip})
9. Root chord (C_{h_root})
10. Mean Aerodynamic Chord (MAC_h or C_h)
11. Span (b_h)
12. Sweep angle (Λ_h)
13. Dihedral angle (Γ_h)
14. Tail installation
15. Incidence (i_h)

- **Vertical tail**

16. Planform area (S_v)
17. Tail arm (l_v)
18. Airfoil section
19. Aspect ratio (AR_v)
20. Taper ratio (λ_v)
21. Tip chord (C_{t_v})
22. Root chord (C_{r_v})
23. Mean Aerodynamic Chord (MAC_v or C_v)
24. Span (b_v)
25. Sweep angle (Λ_v)
26. Dihedral angle (Γ_v)
27. Incidence (i_v)

All 26 tail parameters listed above must be determined in the tail design process. The majority of parameters are finalized through technical calculations, while a few parameters are decided via an engineering selection approach. There are few other intermediate parameters such as downwash angle, sidewash angle, and effective angle of attack which will be used to calculate some tail parameters. These are determined in the design process, but not employed in the manufacturing period.

As discussed in Chapter 2, the "Systems Engineering" approach has been adopted as the basic technique to design the tail. The tail design technique has been developed by this approach to satisfy all design requirements while maintaining low cost in an optimum fashion. Figure 6.1 illustrates the block diagram of the tail design process. As it was explained in Chapter 2, the aircraft design is an iterative process; therefore this procedure (tail design) will be repeated several times until the optimum aircraft configuration has been achieved. The design of vertical and horizontal tails might be performed almost in parallel. However, there is one step in the vertical tail design (i.e. spin recovery) that the effect of horizontal tail into vertical tail is investigated. The details on each step will be introduced in the later sections. The purpose of this chapter is to provide design considerations, design technique, and design examples for the preliminary design of the aircraft tail.

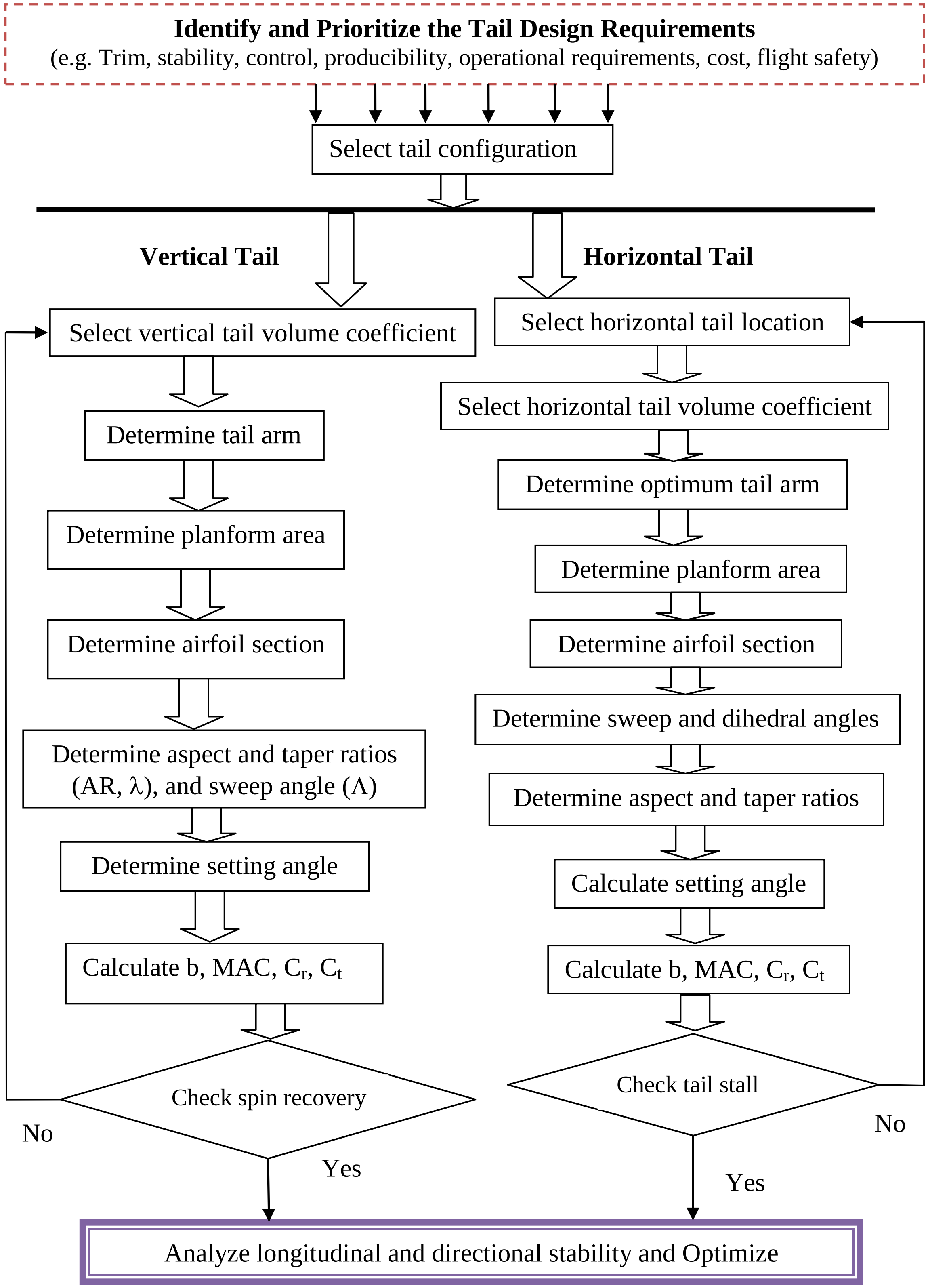


Figure 6.1. The tail design procedure

6.2. Aircraft Trim Requirements

Trim is one of the inevitable requirements of a safe flight. When an aircraft is at trim, the aircraft will not rotate about its center of gravity (cg), and aircraft will either keep moving in a desired direction or will move in a desired circular motion. In another word, when the summations of all forces and moments are zero, the aircraft is said to in trim.

$$\sum F = 0 \tag{6.1}$$

$$\sum M = 0 \tag{6.2}$$

The aircraft trim must be maintained about three axes (x, y, and z): 1. lateral axis (x), 2. longitudinal axis (y), and 3. directional axis (z). When the summation of all forces in x direction (such as drag and thrust) is zero; and the summation of all moments including aerodynamic pitching moment about y axis is zero, the aircraft is said to have the longitudinal trim.

$$\sum F_x = 0 \tag{6.3}$$

$$\sum M_{cg} = 0 \tag{6.4}$$

The horizontal tail is responsible to maintain longitudinal trim and make the summations to be zero, by generating a necessary horizontal tail lift and contributing in the summation of moments about y axis. Horizontal tail can installed behind the fuselage or close to the fuselage nose. The first one is called conventional tail or aft tail, while the second one is referred to as the first tail, foreplane or canard. The equation 6.4 will be used in the horizontal tail design. When the summation of all forces in y direction (such as side force) is zero; and the summation of all moments including aerodynamic yawing moment about z axis is zero, the aircraft is said to have the directional trim.

$$\sum F_y = 0 \tag{6.5}$$

$$\sum N_{cg} = 0 \tag{6.6}$$

The vertical tail is responsible to maintain directional trim and make the summations to be zero, by generating a necessary vertical tail lift and contributing in the summation of moments about y axis. The equation 6.6 will be used in the vertical tail design. When the summation of all forces in z direction (such as lift and weight) is zero; and the summation of all moments including aerodynamic rolling moment about x axis is zero, the aircraft is said to have the directional trim.

$$\sum F_z = 0 \tag{6.7}$$

$$\sum L_{cg} = 0 \tag{6.8}$$

The vertical tail is responsible to maintain directional trim and make the summation of moment to be zero, by generating a necessary vertical tail lift and contributing in the summation of moments about z axis. The equation 6.8 will also be used in the vertical tail design. More details could be found in most flight dynamics textbook. As an example, the interested reader is referred to [1], [2], and [3].

A major design requirements' reference is the Federal Aviation Administration [4]. The following is reproduced from Section 161 of PAR 23 of Federal Aviation Regulations (FAR) which concerns about lateral-directional and longitudinal trim of a General Aviation aircraft:

(a) General. Each airplane must meet the trim requirements of this section after being trimmed and without further pressure upon, or movement of, the primary controls or their corresponding trim controls by the pilot or the automatic pilot. In addition, it must be possible, in other conditions of loading, configuration, speed and power to ensure that the pilot will not be unduly fatigued or distracted by the need to apply residual control forces exceeding those for prolonged application of §23.143(c). This applies in normal operation of the airplane and, if applicable, to those conditions associated with the failure of one engine for which performance characteristics are established.

(b) Lateral and directional trim. The airplane must maintain lateral and directional trim in level flight with the landing gear and wing flaps retracted as follows:

(1) For normal, utility, and acrobatic category airplanes, at a speed of 0.9 V_H, V_C, *or* V_{MO}/M_O, *whichever is lowest; and*

(2) For commuter category airplanes, at all speeds from 1.4 V_{S1} *to the lesser of* V_H *or* V_{MO}/M_{MO}.

(c) Longitudinal trim. The airplane must maintain longitudinal trim under each of the following conditions: (1) A climb, (2) Level flight at all speeds, (3) A descent, (4) Approach

(d) In addition, each multiple airplane must maintain longitudinal and directional trim, and the lateral control force must not exceed 5 pounds at the speed used in complying with §23.67(a), (b)(2), or (c)(3),

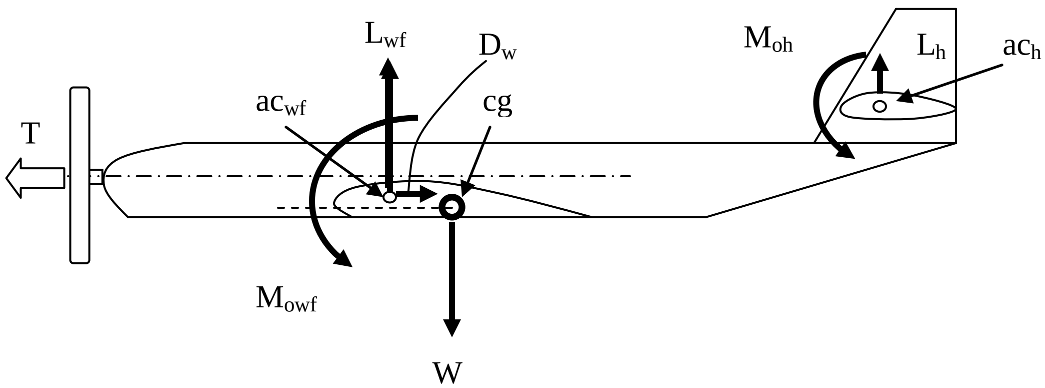
For other types of aircraft, the reader is encouraged to refer to other parts of FAR; for instance, for transport aircraft; the reference is Part 25.

6.2.1. Longitudinal Trim

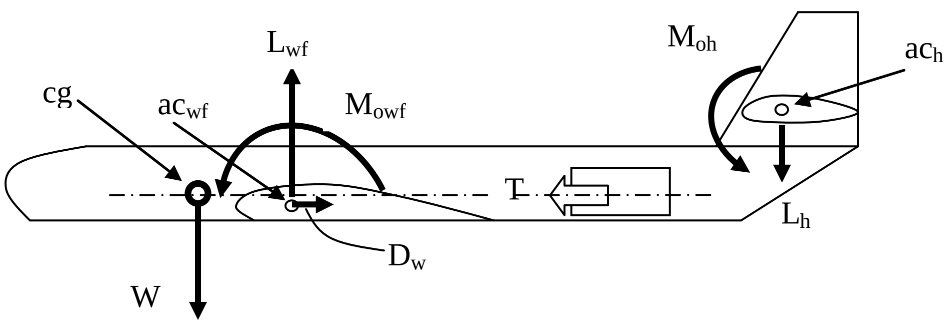
For the horizontal tail design process, we need to develop a few equations; hence the longitudinal trim will be described in more details. Consider the side view of a conventional aircraft (i.e. with aft tail) in figure 6.2 that is in longitudinal trim. Figure 6.2a depicts the aircraft when the aircraft center of gravity (cg) is behind the wing-fuselage aerodynamic center (ac_{wf})[1]. In figure 6.2b, the aircraft is depicted when the aircraft center of gravity is forward of the wing-fuselage aerodynamic center. There are several moments about y axis (cg) that must be balanced by the horizontal tail's lift; two of which are: 1. wing-fuselage aerodynamic pitching moment, 2. the moment of lift about aircraft center of gravity. Other source of moments about cg could be engine thrust, wing drag, landing gear drag, and store drag. For the sake of simplicity, those

[1] The wing-fuselage aerodynamic center is simply the wing aerodynamic center when the contribution of the fuselage is added. The fuselage contribution for most conventional aircraft is usually about ±5% $\overline{C}$. Since the wing aerodynamic center is often located at about quarter mean aerodynamic chord (i.e. 25% $\overline{C}$); hence the wing-fuselage aerodynamic center is often located between 20 percent of MAC to 30 percent of MAC or $\overline{C}$. The reader is referred to [1] for more information.

moments are not included in this figure. The reader is expected to be able to follow the discussion, when other moments are present and/or the aircraft has a canard instead of aft tail.



a. cg aft of ac_{wf}



b. cg forward of ac_{wf}

Figure 6.2. A conventional aircraft in longitudinal trim

The wing-fuselage lift (L_{wf}) is the wing lift (L_w) when the contribution of fuselage lift (L_f) is included. The fuselage lift is usually assumed to be about 10 percent of the wing lift. Ref. [1] can be consulted for the exact calculation. When the cg is aft of the ac_{wf} (as in Fig 6.2a), this moment of the wing-fuselage lift (L_{wf}) is positive, while when the cg is forward of the ac_{wf} (as in Fig 6.2a), this moment of the wing-fuselage lift is negative. Recall from flight dynamics, that the clockwise direction is assumed to be positive, and the y-axis is located at the cg and is directed into the page.

Anther moment is referred to as the wing-fuselage aerodynamic pitching moment (i.e. M_{owf}). The wing-fuselage aerodynamic pitching moment (M_{owf}) is the wing aerodynamic pitching moment (M_{ow}) when the contribution of the fuselage (M_f) is included. The subscript "o" denotes that the aerodynamic moment is measured relative to the wing aerodynamic center. This aerodynamic moment is often negative (as sketched in figure 6.2); so it is often called a nose-down pitching moment; due to its desire to pitch down the fuselage nose. Often times, the summation of these two moments (i.e. the wing-fuselage aerodynamic pitching moment and the wing-fuselage lift generated moment) is not zero. Hence, the horizontal tail is employed to generate a lift in order to balance these moments and make the summation to be zero. This function maintains the aircraft longitudinal trim.

In a similar fashion, a discussion about the directional trim can be addressed. In this case, despite the symmetricity of the conventional aircraft about xz plane, there are forces such as asymmetric engine thrust (when one engine is inoperative in multi-engine aircraft) that disturb the directional trim of an aircraft. In such a situation, the vertical tail is required to generate a lift force in the y direction (i.e. side force) to maintain the directional trim about z axis. The details of this case are left to the reader.

Now, consider the aircraft in figure 6.3 at which the tail aerodynamic pitching moment is neglected. Please note that in this case, the thrust-line is passing through the aircraft cg, so the engine thrust tends to impose no influence on the aircraft longitudinal trim. Although the wing-fuselage lift is positive in a normal flight situation, but the moment of the lift about cg might be positive or negative due to the relationship between cg and ac_{wf}. Thus, the horizontal tail could be negative or positive. The application of the trim equation leads to the following[2]:

$$\sum M_{cg} = 0 \Rightarrow M_{owf} + M_{L_{wf}} + M_{L_h} = 0 \tag{6.9}$$

Recall, the aircraft weight generates no moment about aircraft cg. If the engine thrust line is not passing through the aircraft cg, the equation 6.9 must be modified. To make this equation more convenient to apply, we need to non-dimensionalize it. In order to non-dimensionalize the parameters, it is often customary to measure the distances in the x direction as a factor of mean aerodynamic chord ($\overline{C}$ or simply C). Moreover, a reference line (or point) must be selected to measure all distances with respect to it. Here, we select the fuselage nose as the reference line. Hence, the distance between ac_{wf} to the reference line is h_o times the $\overline{C}$, (i.e. $h_o\overline{C}$), while the distance between cg to the reference line is h times the $\overline{C}$, (i.e. $h\overline{C}$). Both parameters are shown in figure 6.3. The distance between horizontal tail aerodynamic center to the wing-fuselage aerodynamic center is denoted as l, while the distance between horizontal tail aerodynamic center to the aircraft center of gravity is denoted as l_t. Now, we can substitute the values of two moments into the equation 6.9:

$$M_{owf} + L_{wf}\left(h\overline{C} - h_o\overline{C}\right) - L_h \cdot l_h = 0 \tag{6.10}$$

To expand the equation, we need to define the variables of wing-fuselage lift (L_{wf}), horizontal tail lift (L_h), and wing-fuselage aerodynamic pitching moment (M_{owf}).

$$L_{wf} = \frac{1}{2}\rho V^2 S C_{L_{wf}} \tag{6.11}$$

$$L_h = \frac{1}{2}\rho V^2 S_h C_{L_h} \tag{6.12}$$

[2] The horizontal tail aerodynamic pitching moment is ignored, due to its small value.

$$M_{owf} = \frac{1}{2}\rho V^2 S C_{m_{owf}} \overline{C} \tag{6.13}$$

where C_{Lwf} denotes wing-fuselage lift coefficient, C_{Lh} denotes horizontal tail lift coefficient, C_{mowf} denotes wing-fuselage aerodynamic pitching moment coefficient, S denotes wing planform area, S_h denotes horizontal tail planform area, V denotes the aircraft airspeed, and ρ denotes the air density.

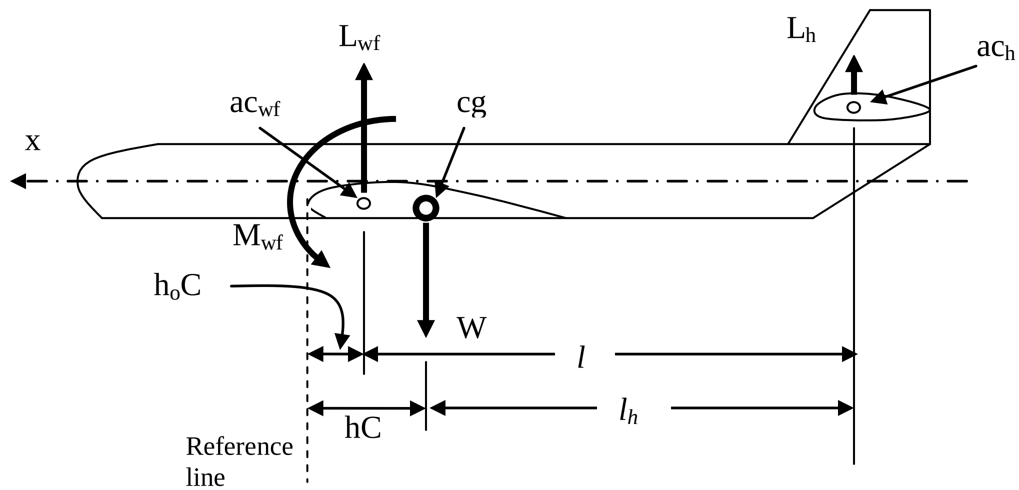


Figure 6.3. The distance between cg, ac_t, and ac_{wf} to the reference line

By substituting equations 6.11, 6.12, and 6.13 into equation 6.10, we will have the following:

$$\frac{1}{2}\rho V^2 S C_{m_{owf}} \overline{C} + \frac{1}{2}\rho V^2 S C_{L_{wf}} \left(h\overline{C} - h_o \overline{C}\right) - \frac{1}{2}\rho V^2 S_h C_{L_h} \cdot l_h = 0 \tag{6.14}$$

This equation is then non-dimensionalized by dividing it to $\frac{1}{2}\rho V^2 S \overline{C}$. Thus, the following is obtained:

$$C_{m_{owf}} + C_{L_{wf}}\left(h - h_o\right) - \frac{l_h}{\overline{C}} C_{L_h} \frac{S_h}{S} = 0 \tag{6.15}$$

Now return to figure 6.3. The distance between horizontal tail aerodynamic center to the reference line can be written in two ways:

$$l + h_o \overline{C} = l_h + h\overline{C} \tag{6.16}$$

or

$$\frac{l_h}{\overline{C}} = \frac{l}{\overline{C}} - \left(h_o - h\right) \tag{6.17}$$

Substituting equation 6.17 into equation 6.15 yields:

$$C_{m_{owf}} + C_{L_{wf}}(h - h_o) - \left[\frac{l}{\overline{C}} - (h_o - h)\right] C_{L_h} \frac{S_h}{S} = 0 \tag{6.18}$$

This can be further simplified as:

$$C_{m_{owf}} + \left(C_{L_{wf}} + C_{L_h}\frac{S_h}{S}\right)(h - h_o) - \frac{l}{\overline{C}}\frac{S_h}{S} C_{L_h} = 0 \tag{6.19}$$

In contrast, the aircraft total lift is the summation of wing-fuselage lift and the horizontal tail lift:

$$L = L_{wf} + L_h \tag{6.20}$$

which leads to:

$$\frac{1}{2}\rho V^2 S C_L = \frac{1}{2}\rho V^2 S C_{L_{wf}} + \frac{1}{2}\rho V^2 S_h C_{L_h} \tag{6.21}$$

This equation is non-dimensionalized as follows:

$$C_L = C_{L_{wf}} + C_{L_h}\frac{S_h}{S} \tag{6.22}$$

Now, the equation 6.22 can be substituted into equation 6.19.

$$C_{m_{owf}} + C_L(h - h_o) - \frac{l}{\overline{C}}\frac{S_h}{S} C_{L_h} = 0 \tag{6.23}$$

The combination $\frac{lS_t}{\overline{C}S}$ in equation 6.23 of is an important non-dimensional parameter in the horizontal tail design, and is referred to as the "*Horizontal tail Volume Coefficient*". The name originates from the fact that both numerator and denominator have the unit of volume (e.g. m^3). The numerator is a function of horizontal tail parameters, while the denominator is a function of wing parameters. Thus, the parameter is the ratio of horizontal tail geometries to wing geometries. It is shown with the symbol of $\overline{V}_H$:

$$\overline{V}_H = \frac{lS_h}{\overline{C}S} \tag{6.24}$$

Thus, the equation 6.23 is further simplified as followed:

$$C_{m_{owf}} + C_L(h - h_o) - \overline{V}_H C_{L_h} = 0 \tag{6.25}$$

This non-dimensional longitudinal trim equation provides a critical tool in the design of the horizontal tail. The importance of this equation will be explained later, and its application will be described in later sections of the chapter. This non-dimensional parameter $\overline{V}_H$ has a limited range in values and also it is not a function of aircraft size or weight. From a small aircraft such as Cessna 172 (Figure 11.15) to a jumbo jet large aircraft such as Boeing 747 (Figures 3.7, 3.12, and 9.4) all have similar tail volume coefficient. Table 6.1 illustrates the tail volume coefficients for several aircraft.

Table 6.4 shows typical values for tail volume coefficient for several aircraft types. The tail volume coefficient is an indication of handling quality in longitudinal stability and longitudinal control. As $\overline{V}_H$ increases, the aircraft tends to be more longitudinally stable, and less longitudinally controllable. The fighter aircraft that are highly maneuverable tend to have a very low tail volume coefficient, namely about 0.2. On the other hand, the jet transport aircraft which must be highly safe and stable tend to have a high tail volume coefficient, namely about 1.1. This parameter is a crucial variable in horizontal tail design and must be selected at the early stages of tail plane design. Although the primary function of the horizontal tail is the longitudinal stability, but the tail volume coefficient serves as a significant parameter both in the longitudinal stability and longitudinal trim issues.

No	**Aircraft**	**Type**	**Mass (kg)**	**Wing area (m^2)**	**Overall length (m)**	$\overline{V}_H$
1	Cessna 172	Light GA (Piston)	1,100	16.2	7.9	0.76
2	Piper PA-46-350P	Light transport (Piston)	1,950	16.26	8.72	0.66
3	Alenia G222	Turboprop transport	28,000	82	22.7	0.85
4	Fokker 100	Jet transport	44,000	93.5	35.5	1.07
5	Lake LA-250	Amphibian	1,424	15.24	9.04	0.8
6	Boeing 747-400	Jet transport	362,000	541	73.6	0.81
7	Airbus 340-200	Jet transport	257,000	363.1	59.39	1.11
8	Pilatus PC-12	Turboprop transport	4,100	25.81	14.4	1.08
9	Eurofighter 2000	Fighter	21,000	50	15.96	0.063
10	F/A-18	Fighter	29,937	46.45	18.31	0.49

Table 6.1. Tail volume coefficients of several aircraft [5]

The wing-fuselage pitching moment coefficient ($C_{m_{owf}}$) in equation 6.25 can be estimated via the following equation [6]:

$$C_{m_{owf}} = C_{m_{af}} \frac{AR\cos^2(\Lambda)}{AR + 2\cos(\Lambda)} + 0.01\alpha_t \qquad (6.26)$$

where $C_{m_{af}}$ is the wing airfoil section pitching moment coefficient, AR is wing aspect ratio, Λ is wing sweep angle, and α_t is the wing twist angle (in degrees). Please note that α_t is often a negative number. The value of $C_{m_{owf}}$ can be determined using the airfoil graphs which an example is shown in figure 5.21 for NACA 63_2-615 airfoil section. For instance, the value of $C_{m_{af}}$ for this airfoil is -0.11.

The parameter C_L in equation 6.25 is the aircraft cruise lift coefficient that is determined by the following equation:

$$C_L = \frac{2W_{avg}}{\rho V_C^2 S} \tag{6.27}$$

where V_C is the cruising speed and the W_{avg} is the average aircraft weigh during the cruising flight. If the wing has been designed prior to the design of horizontal tail, and the aircraft center of gravity (h) was decided, the equation 5.25 has only two unknowns; namely C_{Lh} and $\overline{V}_H$. However, in practice, the design of the wing and the location of the cg are not independent of the tail design. Hence, this is an ideal case, and the tail design is indeed an iteration process. The longitudinal trim equation (i.e. equation 5.26) must be valid in every possible flight conditions. This includes all aircraft allowable load weights, all feasible flight speeds, all aircraft designated configurations (e.g. flap and landing gear, up and down), all allowable cg locations, and all possible flight altitude. These various flight possibilities can be summarized to be between the following two extreme critical conditions:

1. The first unknown flight condition at which the horizontal tail is required to generate the greatest positive pitching moment about aircraft cg.
2. The second unknown flight condition at which the horizontal tail is required to generate the greatest negative pitching moment about aircraft cg.

These two critical flight conditions for the horizontal tail are unknown at this moment, but will be clear later on in the design process. The change in the sign of the tail pitching moment about aircraft cg indicates the necessity of a change in the tail lift coefficient from positive to negative. Two possible solutions are:

1. The application of a moving horizontal tail
2. The application of a fixed horizontal tail, plus a control surface (i.e. elevator).

In the early stage of horizontal tail, the design is performed without considering the elevator. The criterion is to design a horizontal tail to satisfy the cruising flight longitudinal trim requirements. The reason is that the aircraft spends the majority of the flight mission time in the cruising flight.

Due to the effect of wing and fuselage on the horizontal tail (i.e. downwash and sidewash), a new parameter is added to the equation 6.25. The new parameter is the ratio between the dynamic pressure at the tail to the aircraft dynamic pressure, and is called the tail efficiency (η_h) and is defined as follows:

$$\eta_h = \frac{q_t}{q} = \frac{0.5\rho V_h^2}{0.5\rho V^2} = \left(\frac{V_h}{V}\right)^2 \tag{6.28}$$

where the V is the aircraft airspeed, and the V_h is the effective airspeed at the horizontal tail region. The typical value of the tail efficiency for an aircraft with a conventional tail is varied from 0.85 to 0.95. For an aircraft with a T-tail, the tail efficiency can be considered to be 1, which means the wing and fuselage have no effect on the tail dynamic pressure. The horizontal tail of a T-tail is usually out the region of wing wake and downwash during cruising flight. Applying the tail efficiency into the equation 6.25 yields a revised version:

$$C_{m_{owf}} + C_L(h - h_o) - \eta_h \overline{V}_H C_{L_h} = 0 \tag{6.29}$$

This is the most important equation in the design of horizontal tail and implies the requirements for the longitudinal trim. It will be used in both conventional aft tail and canard configuration. The equation is derived in this section, but its application technique will be presented in the Sections 6.6 and 6.8. One of the four parameters in the tail volume coefficient is the distance from wing aerodynamic center to the horizontal tail aerodynamic center (*l*). This distance has statistically a relationship with the aircraft overall length (L). The ratio between the distance from the wing aerodynamic center to the horizontal tail aerodynamic center and aircraft overall length is illustrated in Table 6.2 for several aircraft configurations. It may be employed in the early stage of the horizontal tail design as a starting point. The value will be revised and optimized in the later design steps when more data are available.

No	Aircraft configuration/ type	*l*/L
1	An aircraft whose engine is installed at the nose and has an aft tail	0.6
2	An aircraft whose engine(s) installed above the wing and has an aft tail	0.55
3	An aircraft whose engine installed at the aft fuselage and has an aft tail	0.45
4	An aircraft whose engine installed under the wing and has an aft tail	0.5
5	Glider (with an aft tail)	0.65
6	Canard aircraft	0.4
7	An aircraft whose engine is inside the fuselage (e.g. fighter) and has an aft tail	0.3

Table 6.2. Typical values for the l/L for various aircraft configurations

6.2.2. Directional and Lateral Trim

One of the primary functions for the vertical tail is directional trim. Moreover, the vertical tail tends to have a considerable contribution in the aircraft lateral trim. In this section, the role of the vertical tail in the aircraft directional and lateral trim is examined. Two aircraft are illustrated in figure 6.4; one in directional trim, and another one in directional trim. In figure 6.4-1 the top view of an aircraft is shown where the vertical tail is generating a yawing moment to nullify the yawing moment created by asymmetric thrust of the right engine. In addition, figure 6.4-2 the front view of an aircraft is shown where the vertical tail is generating a rolling moment to nullify the rolling moment created by the rotation of the propeller of the engine. In both cases, the primary production of the vertical tail is an aerodynamic lift in the direction of y-axis.

When an aircraft is in directional trim, the summation of all moments about z-axis must be zero.

$$\sum N_{cg} = 0 \tag{6.6}$$

When an aircraft is in lateral trim, the summation of all moments about x-axis must be zero.

$$\sum L_{cg} = 0 \tag{6.8}$$

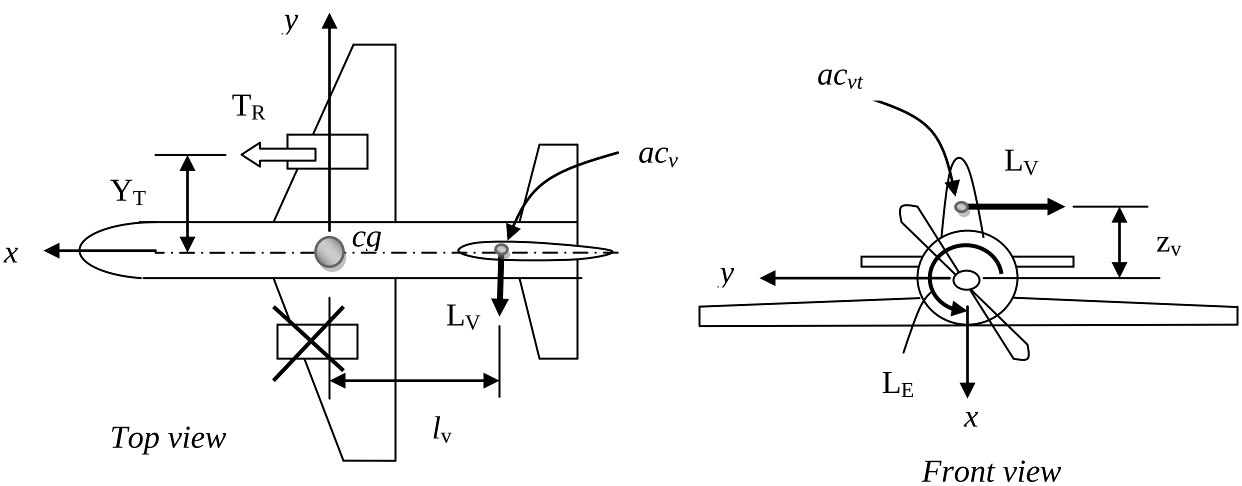
In maintaining the directional and lateral trim, an aerodynamic force along y-axis (lift; L_V) needs to be created by the vertical tail. Thus, the directional and lateral trim equations are:

$$\sum N_{cg} = 0 \Rightarrow T_R Y_T + L_V l_{Vt} = 0 \quad (6.30)$$

$$\sum L_{cg} = 0 \Rightarrow L_E + L_V z_V = 0 \quad (6.31)$$

where T_R denotes the right engine thrust, Y_T is the distance between thrust line and the aircraft cg in the xy plane, l_v is the distance between the vertical tail aerodynamic center and the aircraft cg, L_E is the yawing moment generated by the prop rotation and z_v denotes the distance between vertical tail aerodynamic center and the aircraft cg in the yz plane. The vertical tail lift is obtained from:

$$L_V = \frac{1}{2}\rho V^2 S_V C_{L_v} \quad (6.32)$$



1. One Engine Inoperative (directional trim) *2. Single Propeller Engine (lateral trim)*

Figure 6.4. Vertical tail role in the aircraft lateral and directional trim

where S_V is the vertical tail area, and the C_{LV} is the vertical tail lift coefficient. Four unknowns of S_V, C_{LV}, l_v and z_v are the bases for the design of the vertical tail. Section 6.8 examines application of the technique and procedure for the design of the vertical tail to satisfy the directional and lateral trim requirements.

6.3. A Review on Stability and Control

Stability and control are two requirements of a safe flight. Both horizontal tail and vertical tail has a strong role in aircraft stability and control. Although horizontal tail and vertical tail are initially designed to satisfy the longitudinal and directional trim requirements, but in the later stages of design, the longitudinal and directional stability and control requirements must also be implemented. Thus, the initial design of horizontal tail and vertical tail will be revised to make sure that longitudinal and directional stability and control requirements have been satisfied. In this section, a brief introduction to aircraft stability and control will be provided. This will pave and clarify the path to the design of horizontal tail and vertical tail. Due to the stability requirements by tail, the horizontal tail is sometimes referred to as horizontal stabilizer and vertical tail to as vertical stabilizer.

6.3.1. Stability

The second function of the tail is stability, and the third function of the tail is control. Due to this role, the tail is sometimes referred to as the stabilizer or stabilator. Stability is defined as the tendency of an aircraft to oppose a disturbance (e.g. gust) and return to its initial steady state trim condition if disturbed. Stability is often divided into two branches:

1. Static stability
2. Dynamic stability

Static stability is defined as the initial tendency of an aircraft; without plot assistance; to develop forces and/or moments which oppose an instantaneous perturbation of a motion variable from a steady state flight condition. Dynamic stability is defined as the tendency of an aircraft; without plot assistance; to return to initial steady state trim condition after a disturbance disturbs the trim values. Dynamic stability concerns the entire history of the motion, in particular the rate at which the motion damps out. As a general rule, an aircraft must have some form of dynamic stability even though certain mild disabilities can be tolerated under certain conditions. When an aircraft is dynamically stable, it definitely has static stability. However, if an aircraft is statically stable, there is no guarantee that it has dynamic stability.

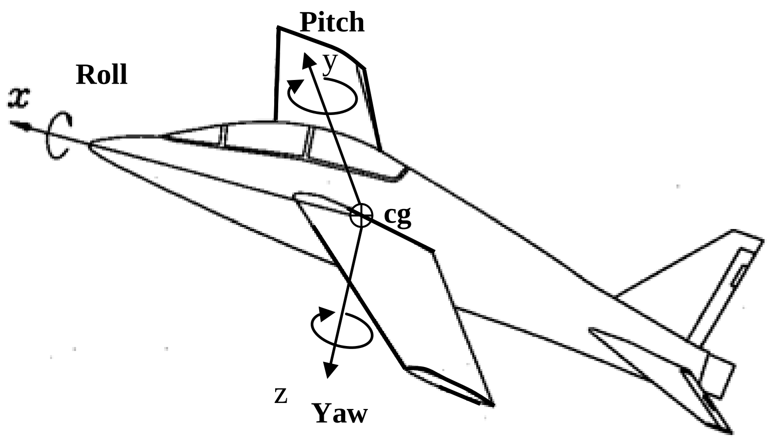


Figure 6.5. Body coordinate system and three rotational motions of roll, pitch, and yaw

An aircraft motion (flight) has six Degrees-Of-Freedom (6 DOF), due to two types of freedom (one linear and one rotational) about each three axis of x, y and z. Therefore, stability is measured about these three axes:

1. Lateral stability
2. Longitudinal stability
3. Directional stability

Lateral stability is defined as the stability of any rotational motion about x axis (i.e. roll) and any corresponding linear motion along yz plane (i.e. side motion). Longitudinal stability is defined as the stability of any rotational motion about y axis (i.e. pitch) and any linear motion along xz plane (i.e. forward and aft, and up and down). Directional stability is defined as the stability of any rotational motion about z axis (e.g. yaw) and any corresponding linear motion along xy plane (e.g. sideslip). Figure 6.5 provides aircraft body coordinate system, plus three rotational motions of roll, pitch, and yaw. The convention is that the clockwise rotation about any axis; when you look from pilot seat; is assumed as the positive rotation.

The requirements for aircraft static and dynamic stability (longitudinal, lateral, and directional) are different. When the aircraft derivative Cm_{α} is negative, the aircraft is said to statically longitudinally stable. An aircraft is said to statically laterally stable, when the aircraft derivative $C_{l\beta}$ (known as dihedral effect) is negative. When the aircraft derivative Cn_{β} is positive, the aircraft is said to statically directionally stable. For an aircraft to be dynamically longitudinally stable, both short-period and long-period (phugoid) modes must be damped (damping ratio greater than zero). When all modes and oscillations (including dutch-roll, spiral, and roll) are damped, an aircraft is said to be dynamically laterally-directionally stable. Some dynamic longitudinal, lateral, and directional stability are tabulated in Chapter 12 (Section 12.3).

Among major aircraft components, the horizontal tail has the largest contribution to the aircraft longitudinal stability. The reason is that the horizontal tail is able to generate the counter pitching moment in order to restore the longitudinal trim position. On the other hand, the vertical tail has the largest contribution to the aircraft directional stability. The vertical tail is able to generate the counter yawing moment in order to restore the directional trim position. Both horizontal tail and vertical tail has significant contributions to the aircraft lateral stability, since both are capable of generating counter rolling moment in order to restore the lateral trim position. Since the chapter is concerned with tail design, only longitudinal and directional stability requirements are emphasized.

The following is reproduced from Section 173 of PAR 23 of Federal Aviation Regulations [4] which concerns about static longitudinal stability of a General Aviation aircraft:

Under the conditions specified in §23.175 and with the airplane trimmed as indicated, the characteristics of the elevator control forces and the friction within the control system must be as follows:

(a) A pull must be required to obtain and maintain speeds below the specified trim speed and a push required to obtain and maintain speeds above the specified trim speed. This must be shown at any speed that can be obtained, except that speeds requiring a control force in excess of 40 pounds or speeds above the maximum allowable speed or below the minimum speed for steady unstalled flight need not be considered.

(b) The airspeed must return to within the tolerances specified for applicable categories of airplanes when the control force is slowly released at any speed within the speed range specified in paragraph (a) of this section. The applicable tolerances are—

(1) The airspeed must return to within plus or minus 10 percent of the original trim airspeed; and

(2) For commuter category airplanes, the airspeed must return to within plus or minus 7.5 percent of the original trim airspeed for the cruising condition specified in §23.175(b).

(c) The stick force must vary with speed so that any substantial speed change results in a stick force clearly perceptible to the pilot.

The following is reproduced from Section 177 of PAR 23 of Federal Aviation Regulations [4] which concerns about static directional stability of a General Aviation aircraft:

a)The static directional stability, as shown by the tendency to recover from a wings level sideslip with the rudder free, must be positive for any landing gear and flap position appropriate to the takeoff, climb, cruise, approach, and landing configurations. This must be shown with symmetrical power up to maximum continuous power, and at speeds from 1.2 V_{S1}up to the maximum allowable speed for the condition being investigated. The angel of sideslip for these tests must be appropriate to the type of airplane. At larger angles of sideslip, up to that at which full rudder is used or a control force limit in §23.143 is reached, whichever occurs first, and at speeds from 1.2 $V_{S1}$to V_O, the rudder pedal force must not reverse.

b) The static lateral stability, as shown by the tendency to raise the low wing in a sideslip, must be positive for all landing gear and flap positions. This must be shown with symmetrical power up to 75 percent of maximum continuous power at speeds above 1.2 V_{S1}in the take-off configuration(s) and at speeds above 1.3 V_{S1} in other configurations, up to the maximum allowable speed for the configuration being investigated, in the takeoff, climb, cruise, and approach configurations. For the landing configuration, the power must be that necessary to maintain a 3 degree angle of descent in coordinated flight. The static lateral stability must not be negative at 1.2 V_{S1}in the takeoff configuration, or at 1.3 V_{S1}in other configurations. The angle of sideslip for these tests must be appropriate to the type of airplane, but in no case may the constant heading sideslip angle be less than that obtainable with a 10 degree bank, or if less, the maximum bank angle obtainable with full rudder deflection or 150 pound rudder force.

The following is reproduced from Section 181 of PAR 23 of Federal Aviation Regulations [4] which concerns about dynamic lateral-directional-longitudinal stability of a General Aviation aircraft:

(a) Any short period oscillation not including combined lateral-directional oscillations occurring between the stalling speed and the maximum allowable speed appropriate to the configuration of the airplane must be heavily damped with the primary controls—

(1) Free; and

(2) In a fixed position.

(b) Any combined lateral-directional oscillations ("Dutch roll") occurring between the stalling speed and the maximum allowable speed appropriate to the configuration of the airplane must be damped to 1/10 amplitude in 7 cycles with the primary controls—

(1) Free; and

(2) In a fixed position.

(c) If it is determined that the function of a stability augmentation system, reference §23.672, is needed to meet the flight characteristic requirements of this part, the primary control requirements of paragraphs (a)(2) and (b)(2) of this section are not applicable to the tests needed to verify the acceptability of that system.

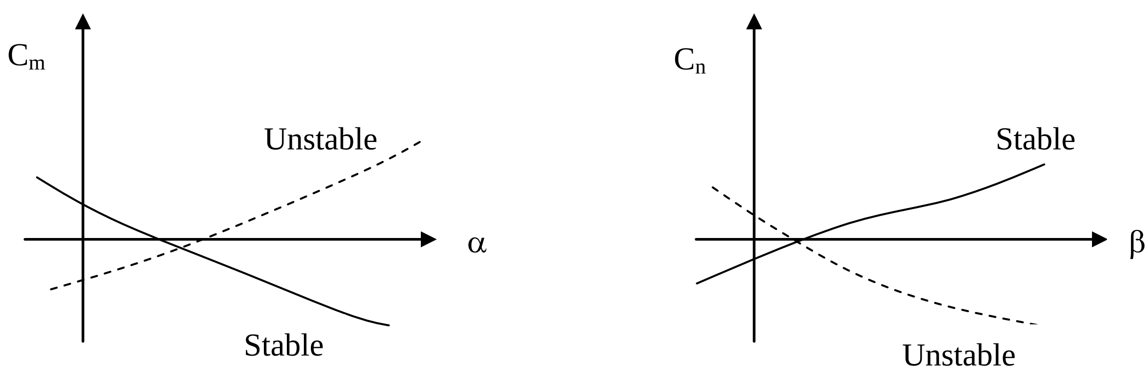
(d) During the conditions as specified in §23.175, when the longitudinal control force required to maintain speeds differing from the trim speed by at least plus and minus 15 percent is suddenly released, the response of the airplane must not exhibit any dangerous characteristics nor be excessive in relation to the magnitude of the control force released. Any long-period oscillation of flight path, phugoid oscillation, that results must not be so unstable as to increase the pilot's workload or otherwise endanger the airplane.

Since the longitudinal stability is concerned with a motion in pitch, the pertinent dynamic characteristic is the variation of the pitching moment with respect to the angle of attack (α). Thus, the primary stability derivative that determines the static longitudinal stability is the Cm_α. Moreover, the primary stability derivative that influences the dynamic longitudinal stability is the Cm_q. The derivative Cm_α is the rate of change of pitching moment coefficient (C_m) with respect to change in the angle of attack (α). The derivative Cm_q is the rate of change of pitching moment coefficient (C_m) with respect to change in the pitch rate (q).

$$C_{m_\alpha} = \frac{\partial C_m}{\partial \alpha} \tag{6.33}$$

$$C_{m_q} = \frac{\partial C_m}{\partial q} \tag{6.34}$$

These two stability derivatives are most influential in the design of horizontal tail. A statically longitudinally stable aircraft requires C_{m_α} to be negative. The typical value for most aircraft is about -0.3 to -1.5 1/rad. A dynamically longitudinally stable aircraft requires that the real parts of the roots of the longitudinal characteristic equation to be negative. One of the major contributor to this requirement is C_{m_q}; such that a negative value has a strong stabilizing impact. The typical value of C_{m_q} for most aircraft is about -5 to -30 1/rad.



a. C_m versus α curve *b. C_n versus β curve*

Figure 6.6. Graphical representations of derivatives C_{m_α} .and C_{n_β}

It is interesting to note that the horizontal tail volume coefficient ($\overline{V}_H$) is the most important parameter affecting both C_{m_α} and C_{m_q}. Figure 6.6-1 provides graphical representation of stability derivative C_{m_α}. The detail of the technique to determine derivatives C_{m_α} and C_{m_q} are available in [6]. Another very important parameter that can be employed to determine the aircraft longitudinal static stability is the aircraft neutral point. Some textbooks refer to this point as the aircraft aerodynamic center (ac_A). If the aircraft neutral point is behind the aircraft center of gravity, the aircraft is said to have longitudinal static stability. At this situation, the static margin (i.e. the non-dimensional distance between the aircraft neutral point to the aircraft cg) is said to be positive. The details of the technique to determine aircraft neutral point and static margin may be found in [1] and [6].

The directional stability is mainly concerned with motion in yaw, so the pertinent dynamic characteristic is the variation of the yawing moment with respect to the sideslip angle (β). Thus, the primary stability derivative that determines the static directional stability is the Cn_β. Moreover, the primary stability derivative that influences the dynamic directional stability is the Cn_r. The derivative Cn_β is the rate of change of yawing moment coefficient (C_n) with respect to change in the sideslip angle (β). The derivative Cn_r is the rate of change of yawing moment coefficient (C_n) with respect to change in the yaw rate (r).

$$C_{n_\beta} = \frac{\partial C_n}{\partial \beta} \tag{6.35}$$

$$C_{n_r} = \frac{\partial C_n}{\partial r} \tag{6.36}$$

No	Requirements	Stability derivatives	Symbol	Typical value (1/rad)
1a	Static longitudinal stability	Rate of change of pitching moment coefficient with respect to angle of attack	C_{m_α}	-0.3 to -1.5
1b	Static longitudinal stability	Static margin	$h_{np} - h_{cg}$	0.1 to 0.3
2	Dynamic longitudinal stability	Rate of change of pitching moment coefficient with respect to pitch rate	C_{m_q}	-5 to -40
3	Static directional stability	Rate of change of yawing moment coefficient with respect to sideslip angle	C_{n_β}	+0.05 to +0.4
4	Dynamic directional stability	Rate of change of yawing moment coefficient with respect to yaw rate	C_{n_r}	-0.1 to -1

Table 6.3. The static and dynamic longitudinal and directional stability requirements

These two stability derivatives are most influential in the design of vertical tail. A statically directionally stable aircraft requires C_{n_β} to be positive. The typical value for most aircraft is about +0.1 to +0.4 1/rad. A dynamically directionally stable aircraft requires that the real parts of the roots of the lateral-directional characteristic equation to be negative. One of the major contributor to this requirement is C_{n_r}; such that a negative value has a strong stabilizing impact. The typical value for most aircraft is about -0.1 to -1 1/rad. These two derivatives are among the influential parameters in the design of the vertical tail. Table 6.3 summarizes the requirements for static and dynamic longitudinal and directional stability. Figure 6.6-2 provides graphical representation of the stability derivative C_{n_β}. The technique to determine derivatives C_{n_β} and C_{n_r} is available in [6].

Almost all General Aviation and transport aircraft are longitudinally and directionally stable. Of military aircraft, only advanced fighters are exception; which means fighters are the only military aircraft that may not be longitudinally and/or directionally stable. The reason lies behind their tough mission of fighting. In order to provide a highly maneuverable fighter aircraft, the stability requirements are relaxed, and safety of flight are left to the pilot plus fighter advanced automatic control system. Thus, we primarily design the horizontal and vertical tail to satisfy longitudinal and directional requirements.

6.3.2. Control

Control is defined as the ability of an aircraft to vary the aircraft condition from trim condition 1 (say cruise) to trim condition 2 (say climb). Due to three axes in the aircraft coordinate system, there are three branches in aircraft control:

1. Lateral control;
2. Longitudinal control;
3. Directional control.

Lateral control is the control of an aircraft about x-axis; longitudinal control is the control of an aircraft about y-axis; and directional control is the control of an aircraft about z-axis. In a conventional aircraft, the lateral control is applied though aileron; the longitudinal control is applied though elevator; and the directional control is applied though rudder. Since the elevator is part of the horizontal tail, and rudder is part of the vertical tail; the tail designer must make sure that horizontal tail and vertical tail are large enough to satisfy longitudinal and directional controllability requirements.

Based on the Section 145 of PAR 23 of Federal Aviation Regulations [4] which concerns about longitudinal control of GA aircraft:

With the airplane as nearly as possible in trim at 1.3 V_{S1}, it must be possible, at speeds below the trim speed, to pitch the nose downward so that the rate of increase in airspeed allows prompt acceleration to the trim speed.

The following is reproduced from Section 147 of PAR 23 of Federal Aviation Regulations [4] which concerns about directional and lateral control of GA aircraft:

(a) For each multiengine airplane, it must be possible, while holding the wings level within five degrees, to make sudden changes in heading safely in both directions. This ability must be shown at 1.4 V_{S1} with heading changes up to 15 degrees, except that the heading change at which the rudder force corresponds to the limits specified in §23.143 need not be exceeded,

(b) For each multiengine airplane, it must be possible to regain full control of the airplane without exceeding a bank angle of 45 degrees, reaching a dangerous attitude or encountering dangerous characteristics, in the event of a sudden and complete failure of the critical engine, making allowance for a delay of two seconds in the initiation of recovery action appropriate to the situation, with the airplane initially in trim.

(c) For all airplanes, it must be shown that the airplane is safely controllable without the use of the primary lateral control system in any all-engine configuration(s) and at any speed or altitude within the approved operating envelope. It must also be shown that the airplane's flight characteristics are not impaired below a level needed to permit continued safe flight and the ability to maintain attitudes suitable for a controlled landing without exceeding the operational and structural limitations of the airplane. If a single failure of any one connecting or transmitting link in the lateral control system would also cause the loss of additional control system(s), compliance with the above requirement must be shown with those additional systems also assumed to be inoperative.

Since the design of control surfaces are covered in details in Chapter 12, more information about controllability requirements can be found there. In a case where a horizontal tail design satisfies the longitudinal trim and stability requirements, but is unable to satisfy the longitudinal control requirements, the horizontal tail parameters must be revised. In a similar fashion, if a vertical tail design satisfies the directional trim and stability requirements, but is unable to satisfy the directional control requirements, the vertical tail parameters must be revised.

6.3.3. Handling Qualities

Stability and control are at odd with each other. The reinforcement of stability in an aircraft design weakens the aircraft controllability, while the improvement of controllability of an aircraft has negative effect on the aircraft stability. As the stability features of an aircraft is improved, its controllability features is degraded. A highly stable aircraft (such as passenger aircraft) tends to be less controllable, while a highly maneuverable aircraft (such as a fighter or a missile) tends to be less stable or even not stable. The decision about the extent of stability and controllability is very hard and crucial to make for an aircraft designer. The provision of longitudinal and directional stability are almost straight forward, compared with lateral stability that tends to negatively influence other desired aspects of an aircraft. In majority of cases, the provision of lateral stability is very hard to achieve such that majority of aircraft, even transport aircraft, suffer from the lack of sufficient lateral stability.

The determination of the borderline between stability and control of an aircraft is executed through a topic referred to as "*handling qualities*". The degree of stability and the degree of controllability have been investigated and established by the standards such as Federal Aviation Regulations (FAR) standards, or Military Standards (MIL-STD). The handling qualities (or sometimes called flying qualities) are determined to guarantees the comfort of the pilot and passengers as well as the airworthiness standards. The handling qualities requirements largely influence several aspects of the horizontal and vertical tail. The initial selection of tail parameters (such as tail volume coefficient) must include a satisfactory achievement of handling quality requirements. If your customer has net requested for specific and unique handling qualities, you can trust and follow the published standards such as FAR and MIL-STD. More details of handling qualities are presented in Chapter 12, Section 12.3. The technique outlined in this chapter considers the public aviation standards that are available to aircraft designers in libraries and official government websites.

6.4. Tail configuration

6.4.1. Basic Tail Configuration

The purpose of this section is to present design requirements, and design information related to the selection of the tail configuration. The term "*tail*" in this section means the combination of horizontal and vertical tail. The first step in the tail design is the selection of the tail configuration. The choice of the tail configuration is the output of a selection process, not the result of a mathematical calculation. The decision for the selection of the tail configuration must be made based on the reasoning, logic and evaluation of various configurations against design requirements.

The list of design requirements that must be considered and satisfied in the selection of tail configurations are as follows:

1. Longitudinal trim
2. Directional trim
3. Lateral trim
4. Longitudinal stability
5. Directional stability
6. Lateral stability
7. Manufacturability and controllability
8. Handling qualities (e.g. passenger comfort)
9. Stealth (only in some specific military aircraft)
10. Operational requirements (e.g. pilot view)
11. Airworthiness (e.g. safety, tail stall, and deep stall)
12. Survivability (e.g. spin recovery)
13. Cost
14. Competitiveness (in the market)
15. Size limits (for example, an aircraft may be required to have a limited height; for instance, for the hangar space limits. This will influence the vertical tail configuration)

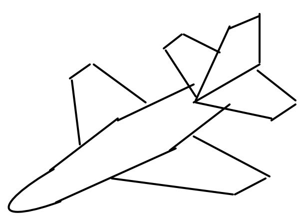
The technical details of these requirements must be established prior to the selection of the tail configuration. Often times, no single tail configuration can satisfy all design requirements; hence, a compromise must be made. After a few acceptable candidates are prepared, a table based on the systems engineering approach must be provided to determine the final selection; i.e. best choice. Sometimes a design requirement (such as lateral stability) is completely ignored (i.e.

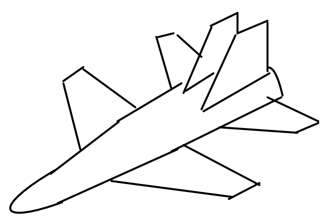
sacrificed), in order to satisfy other more important design requirements (such as maneuverability or stealth requirements).

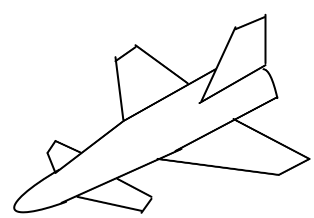
In general, the following tail configurations are available that are capable of satisfying the design requirements in one way or another:

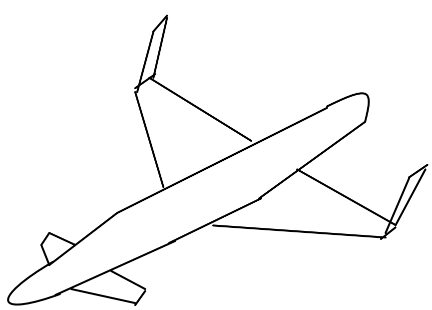
1. Aft tail and one aft vertical tail
2. Aft tail and twin aft vertical tail
3. Canard and aft vertical tail
4. Canard and twin wing vertical tail
5. Triplane (i.e. aft tail as aft plane, and canard as fore-plane plus wing as the third plane)
6. Tailless (delta wing with one vertical tail)
7. No formal tail (also known as "*flying wing*", such as B-2 Spirit (Figure 6.8).

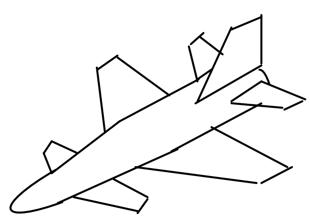
Figure 6.7 depicts these configurations. Based on the statistics, majority of aircraft designers (about 85 percent) are selecting the aft tail configuration. About 10 percent of current aircraft have canard. And about 5 percent of today's aircraft have other configurations that could be called as unconventional tail configuration. The general characteristics of the canard will be described in Section 6.5.


1. Aft tail and one aft vertical tail


2. Aft tail and two aft vertical tails


3. Canard and aft vertical tail


4. Canard and two wing vertical tail


5. Triplane

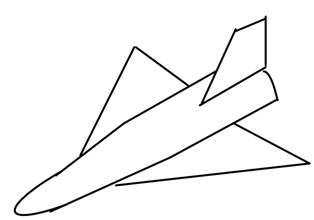

6. Delta wing with one vertical tail

Figure 6.7. Basic tail configurations

The first configuration (aft tail and one aft vertical tail) has several sub-configurations that will be examined in Section 6.4.2. In the first three configurations (see figures 6.7-1 through 6.7-3), vertical tail is installed at the aft of fuselage, while in the fourth configuration (see figure

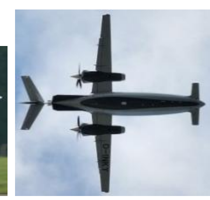
6.7-5); two vertical tails are installed at the wing tips. The features of the canard configuration will be examined in Section 6.5. The selection of twin vertical tail (VT) largely originates mainly from the fact that it provides high directional control, while does not degrades the roll control. Since two short-span vertical tails (see figure 6.7-4) tend to have a lower mass moment of inertia about x axis; compared with one long-span vertical tail. Figure 6.8-6 illustrates the aircraft Piaggio P-180 with a Triplane configuration.

The primary functions of the tail in aircraft with no tail configuration are performed via other component or automatic control system. For instance, in hang gliders, the longitudinal trim of the aircraft is employed by pilot via moving his/her body in order to vary the cg of the aircraft. Furthermore, the longitudinal stability requirements are satisfied through a particular wing airfoil section that has a negative camber at the trailing edge (i.e. reflexed trailing edge) as sketched in figure 6.9. Moreover, the pilot is able to continuously control and make a considerable change to the wing airfoil section via a manual mechanism provided for him/her. This technique is typically employed in hang gliders.



1. *Aero Designs Pulsar (aft tail) 2. Dassault Rafale (canard)   3. B-2 Spirit (flying wing)*
(Courtesy of Jenny Coffey)   (Courtesy of Antony Osborne)





4. Lockheed F-117 Nighthawk (V-tail) 5. Velocity 173 Elite (canard and twin VT) 6. Piaggio P-180 (triplane)
(Courtesy of Antony Osborne)   (Courtesy of Jenny Coffey) (Courtesy of Hansueli Krapf)

7. De Havilland DH-110 Sea Vixen (unconventional twin VT) 8. PZL-Mielec M-28 Bryza (H-tail)
(Courtesy of Jenny Coffey)   (Courtesy of Jenny Coffey)

Figure 6.8. Several aircraft with various tail configurations

Majority of GA aircraft have a conventional aft horizontal tail, and an aft vertical tail configuration. Majority of fighter aircraft have one aft tail and twin vertical tails, due to their maneuverability requirements. Some European fighters (mainly French fighters; such as Dassault Rafale) have canard configuration (see figure 6.8-2). The primary reason for the Bomber aircraft B-2 Spirit flying (figure 6.8-3) wing is the stealth requirements. Most hang gliders do not employ a horizontal tail, they however, satisfy the longitudinal stability requirements through a wing reflex trailing edge.

In some cases, some aircraft configurations impose some limits on the tail configuration. For instance, when a prop-driven engine is considered to be installed inside aft fuselage (i.e. pusher aircraft as seen in MQ-9 Reaper UAV (Figure 6.12), the aft horizontal is not a proper option. The reason is that the horizontal tail will be under continuous wake effect of the engine, and its efficiency will be degraded. By the same reasoning, a canard is not a good option, if a prop-driven engine is inside fuselage nose (e.g. Aero Designs Pulsar as shown in figure 6.8-1)). The main disadvantage for a higher number of tails; such as tri-plane (figure 6.8-6) or two vertical tails (see figure 6.8-7 and 6.8-8); are the higher cost of manufacturing and the complexity of the design. Figure 6.8-8 illustrates the PZL-Mielec M-28 Bryza with an H-tail.

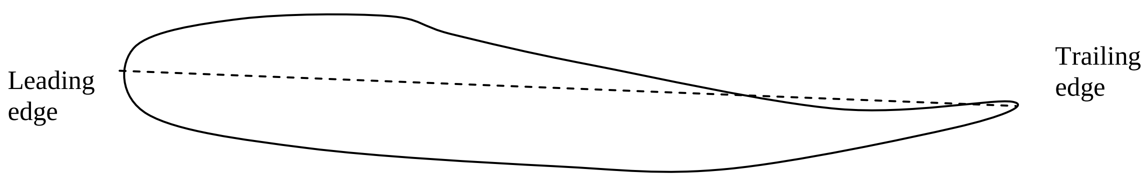


Figure 6.9. A wing airfoil section with reflexed trailing edge

The basic rule for the selection of the tail configuration is as follows: In general, the conventional aft tail configuration (Figure 6.7-1) is often able to satisfy all design requirements, unless one or more requirements imply for another configuration. Thus, it is recommended to begin with conventional aft tail configuration and then to evaluate its features against the design requirements. If one or more requirements are not satisfies, change to a new configuration nearest with the current configuration, until all requirements could be satisfied. If the aircraft is in the manufacturing phase and a change is needed to improve the longitudinal and directional stability, one can utilize a smaller auxiliary horizontal tail (sometimes referred to as stabilon) and ventral stake. These tricks are employed in the twin-turboprop regional transport aircraft Beech 1900D.

6.4.2. Aft Tail Configuration

Aft tail has several configurations that all are able to satisfy the design configurations. Each has unique advantages and disadvantages. The purpose of this section is to provide a comparison between these configurations to enable an aircraft designer to make decision and to select the best one. The aft tail configurations are as follows: 1. Conventional, 2. T-shape, 3. Cruciform (+), 4. H-shape, 5. Triple-tail, 6. V- tail, 7. Inverted V-tail, 8. Improved V-tail 9. Y-tail, 10. Twin vertical tail, 11. Boom-mounted, 12. Inverted boom-mounted, 13. Ring-shape, 14. Twin T, 15. half T, 16. U-tail. Figure 6.10 provides several aft tail configurations.

1. Conventional

The conventional tail or inverted T-shape configuration (see figure 6.10-1) is the simplest configuration and the most convenient to perform all tail functions (i.e. trim, stability, and control). The analysis and evaluation of the performance of a conventional tail is straight forward. This configuration includes one horizontal tail (two left and right sections); located on the aft fuselage; and one vertical tail (one section); located on top of the aft fuselage. Both horizontal and vertical tails are located and mounted to the aft of fuselage. The horizontal tail is mainly employed to satisfy the longitudinal trim and stability requirements, while vertical tail is mainly used to satisfy the directional trim and stability requirements. If the designer has low experience, it is recommended to initially select the conventional tail configuration.

Almost all flight dynamics textbook examine the features of a conventional tail, but not every flight dynamics textbook discuss the characteristics of other tail configurations. The designer must be professional and skillful on the area of the trim analysis, stability analysis, and control analysis, if other configurations are selected. This is one of the reasons that about 60 percent of current aircraft in service have conventional tail. Furthermore it has light weight, efficient, and performs at regular flight conditions. GA aircraft such Cessna 172 (Figure 11.15), Cessna 560 Citation, Beech King Air C90B, Learjet 60, Embraer EMB-314 Super Tucano (Figure 10.6), Socata TBM 700, and Pilatus PC-9; large transport aircraft such as Fokker 60, Boeing 747 (Figures 3.7, 3.12, and 9.4), Boring 777 (see figure 6.12-1), Airbus 340 (Figure 8.20), and fighter aircraft such as F-16 Eagle (Figure 3.12), Harrier GR. Mk 7 (Figure 4.19), and Panavia Tornado F. Mk3 (Figure 5.61**)** all have conventional tail. Figure 6.8-2 illustrates the aircraft Aero Designs Pulsar with a conventional tail configuration.

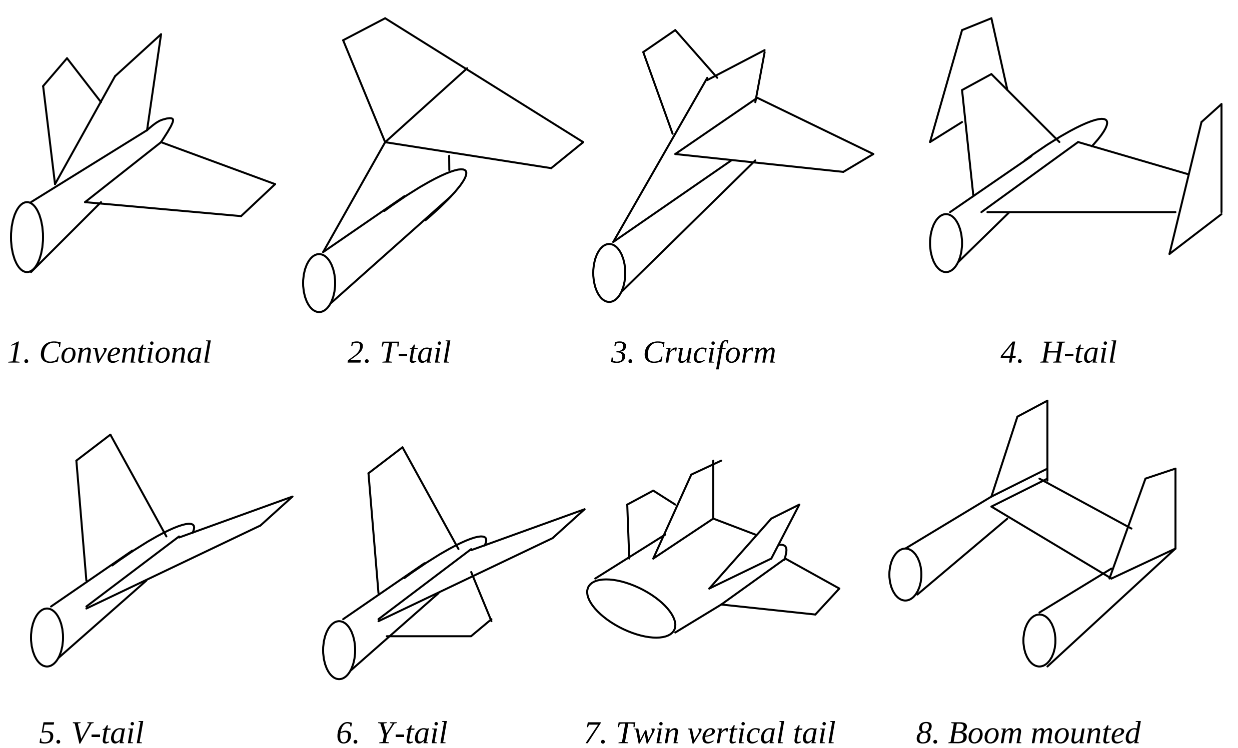


Figure 6.10. Several aft tail configurations

2. **T-tail**

A T-tail is an aft tail configuration (see figure 6.10-2) that looks like the letter "T"; which implies the vertical tail is located on top of the horizontal tail. The T-tail configuration is another aft tail configuration that provides a few advantages, while it has a few disadvantages. The major advantage of a T-tail configuration is that it is out of the regions of wing wake, wing downwash, wing vortices, and engine exit flow (i.e. hot and turbulent high speed gas). This allows the horizontal tail to provide a higher efficiency, and a safer structure. The lower influence from the wing results in a smaller horizontal tail area; and the lower effect from the engine leads in a less tail vibration and buffet. The less tail vibration increases the life of the tail with a lower fatigue problem. Furthermore, another advantage of the T-tail is the positive influence of horizontal tail on the vertical tail. It is referred to as the end-plate effect and results in a smaller vertical tail area.

In contrast, the disadvantages that associated with a T-tail are: 1. heavier vertical tail structure, 2. deep stall. The bending moment created by the horizontal tail must be transferred to the fuselage through the vertical tail. This structural behavior requires the vertical tail main spar to be stronger; which cause the vertical tail to get heavier.

Aircraft with T-tail are subject to a dangerous condition known as the deep stall [7]; which is a stalled condition at an angle of attack far above the original stall angle. T-tail Aircraft often suffer a sever pitching moment instability at angles well above the initial stall angle of about 13 degrees, without wing leading edge high lift device, or about 18 degrees, with wing leading edge high lift device. If the pilot allows the aircraft to enter to this unstable region, it might rapidly pitch up to a higher angle of about 40 degrees. The causes of the instability are fuselage vortices, shed from the forward portion of the fuselage at high angles of attack, and the wing and engine wakes. Thus the horizontal tail contribution on the longitudinal stability is largely reduced. Eventually, at a higher angle of attack, the horizontal tail exits the wing and nacelle wakes and the aircraft become longitudinally stable (see figure 6.11).

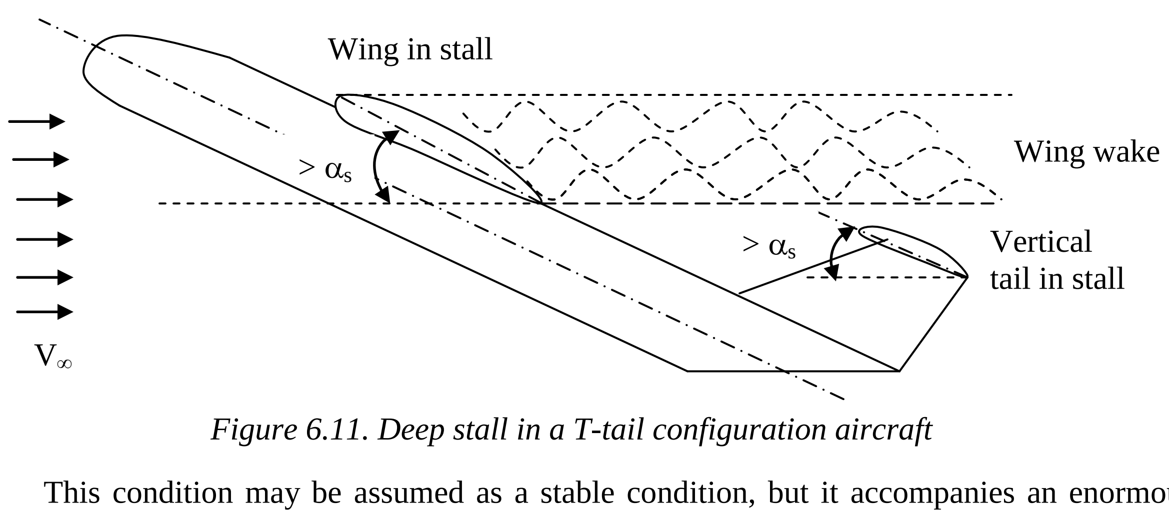


Figure 6.11. Deep stall in a T-tail configuration aircraft

This condition may be assumed as a stable condition, but it accompanies an enormous drag along with a resulting high rate of descent. At this moment, the elevator and aileron effectiveness have been severely reduced because both wing and horizontal tail are stalled at the very high angle of attack. This is known as a locked-in deep stall, a potentially fatal state. The design solutions to avoid a deep stall in a T-tail configuration are to: 1. Ensure a stable pitch down at the initial stall, 2. Extend the horizontal tail span substantially beyond the nacelles, and 3. Employ a mechanism to enable full down elevator angles if a deep stall occurs. In addition, the aircraft must be well protected from the initial stall by devices such as stick shaker, lights, and stall horn.

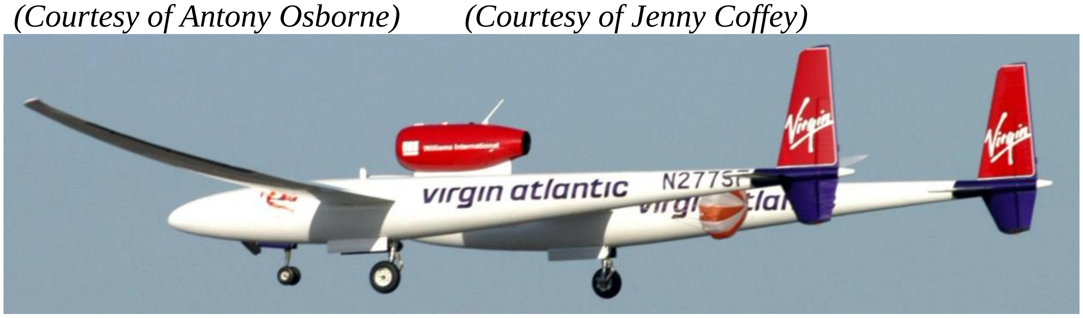

1. Boeing 737 (Conventional) 2. Sky Arrow 1450L (T-tail) 3. Dassault Falcon 900 (Cruciform)
(Courtesy of Anne Deus) (Courtesy of Jenny Coffey) (Courtesy of Jenny Coffey)



4. Fairchild A-10 Thunderbolt (H-tail) 5. Global hawk UAV (V-tail) 6. MQ-9 Reaper UAV (Y-tail)
(Courtesy of Antony Osborne)


7. F-18 Hornet (twin VT) 8. Reims F337F Super Skymaster (Boom mounted)

(Courtesy of Antony Osborne) (Courtesy of Jenny Coffey)



9. Global Flyer (unconventional tail); (Courtesy of NASA)
Figure 6.12. Several aircraft with various aft tail configurations

Despite above mentioned disadvantages of T-tail, it becomes more and popular among aircraft designers. About 25 percent of today's aircraft employ T-tail configuration. It is interesting to note that the GA aircraft Piper Cherokee has two versions; Cherokee III (Figure 7.6) with conventional tail, and Cherokee IV with T-tail. The aircraft has a single piston engine at the nose and a low wing configuration. Several GA and transport aircraft such as Grob Starto 2C, Cessna 525 CitaionJet, Beech Super King Air B200, Beechjet T-1A Jayhawk, Learjet 60, Gulfstream IV (Figure 11.15), MD-90, Boeing 727, Fokker 100 (Figure 10.6), AVRO RJ115, Bombardier BD 701 Global Express, Dassault Falcon 900 (Figure 6.12), Sky Arrow 1450L (see Figure 6.12-3), Embraer EMB-120, Airbus A400M (Figure 8.3), and Boeing (formerly McDonnell Douglas) C-17 Globemaster III (Figure 9.9) employ T-tail configuration.

3. Cruciform

Some tail designers have combined the advantages of conventional tail and T-tail and came up with a new configuration known as cruciform (see figure 6.10-3). Thus, the disadvantages of both configurations are considerably released. The cruciform; as the name implies; is a combination of horizontal tail and vertical tail such that it looks like a cross or "+" sign. This means that the horizontal tail is installed at almost the middle of the vertical tail. The location of the horizontal tail (i.e. its height relative to the fuselage) must be carefully determined such that the deep stall does not occur and at the same time, the vertical tail does not get heavy. Several aircraft such as Thurston TA16, Dassault Falcon 2000, ATR 42-400 (Figure 3.8), Dassault Falcon 900B (see figure 6.12-3), Jetstream 41, Hawker 100, Mirage 2000D (Figure 9.12) employ the cruciform tail configuration.

4. H-tail

The H-tail (see figure 6.10-4), as the name implies, looks like the letter "H". H-tail comprised of one horizontal tail in between two vertical tails. The features associated with an H-tail are as follows:

1. At high angles of attack, the vertical tail is not influenced by the turbulent flow coming from fuselage.
2. In a multiengine turboprop aircraft, vertical tails are located behind the prop-wash region. This causes the vertical tail to have higher performance in the inoperative engine situation.
3. The vertical tail end-plate effect improves the aerodynamic performance of the horizontal tail.
4. In military aircraft, the engine very hot exhaust gasses could be hidden from radars or infrared missiles. This technique has been employed the close support aircraft Fairchild A-10 Thunderbolt (se figure 6.12-4).
5. The H-tail allows the twin vertical tail span to be shorter. The aircraft "*Lockheed constellation*" had to employ an H-tail configuration to be able to park inside short height hangars.
6. The lateral control of the aircraft will be improved due to the shorter vertical tail span.
7. The H-tail allows the fuselage to be shorter, since the tail can be installed on a boom.
8. The H-tail is slightly heavier than conventional; and T-tail configuration. The reason is that the horizontal tail must be strong enough to support both vertical tails.
9. The structural design of the H-tail is more tedious than conventional tail.

As can be noticed, an H-tail configuration tends to offer several advantages and disadvantages; hence, the selection of an H-tail must be the result of a compromise process. Several GA and military aircraft such as Sadler A-22 Piranha, T-46, Short Skyvan, and Fairchild A-10 Thunderbolt (see figure 6.12-4) utilize H-tail configuration.

5. **V-tail**

When the major goal of the tail design is to reduce the total tail area, the V-tail (see figure 6.10-5) is a proper candidate. As the name implies, the V-tail configuration has two sections, which forms a shape that looks like the letter "V". In another word, a V-tail is similar to a horizontal tail with high anhedral angle and without any vertical tail. Two sections of a V-tail act as both horizontal and vertical tails. Due to the angle of each section, the lift perpendicular to each section has two components; one in y-direction, and one in z-direction. If no controller is deflected, two components in the y-direction cancel each other, while two lift components in the z-direction are added together. The V-tail may perform the longitudinal and directional trim role satisfactorily, but it has deficiencies in maintaining the aircraft longitudinal and directional stability. In addition, the V-tail design is more susceptible to Dutch roll tendencies than a conventional tail, and total reduction in drag is minimal.

The V-tail design utilizes two slanted tail surfaces to perform the same functions as the surfaces of a conventional elevator and rudder configuration. The movable surfaces, which are usually called ruddervator, are connected through a special linkage that allows the control wheel to move both surfaces simultaneously. On the other hand, displacement of the rudder pedals moves the surfaces differentially, thereby providing directional control. When both rudder and elevator controls are moved by the pilot, a control mixing mechanism moves each surface the appropriate amount. The control system for the V-tail is more complex than that required for a conventional tail. Ruddervator induce the undesirable phenomenon of the adverse roll-yaw coupling. The solution could be an inverted V-tail configuration that has other disadvantages. Few aircraft such as Beechcraft Bonanza V35, Robin ATL Club, Aviation Farm J5 Marco, high-altitude, long-endurance unmanned aerial reconnaissance vehicle Global Hawk (see figure 6.12-5), and Lockheed F-117 Nighthawk (Figure 6.8-4) employ a V-tail. Unmanned aircraft General Atomic MQ-1 Predator has an inverted V-tail plus a vertical tail under the aft fuselage.

6. **Y-tail**

The Y-tail (see figure 6.10-6) is an extension to the V-tail, since it has an extra surface located under the aft fuselage. This extra surface reduces the tail contribution in the aircraft dihedral effect. The lower section plays the role of vertical tail, while the two upper sections play the role of the horizontal tail. Therefore, the lower surface has rudder, and the control surface of the upper section plays the role of the elevator. Thus, the complexity of the Y-tail is much lower than the V-tail. One of the reasons this tail configuration is used is to keep the tail out of effect of the wing wake at high angles of attack. The lower section may limit the performance of the aircraft during take-off and landing, since the tail hitting the ground must be avoided. This configuration is not popular, and few old aircraft had this configuration. Unmanned aircraft General Atomic MQ-9 Reaper (see figure 6.12-6) employ Y-tail configuration.

7. **Twin vertical tail**

A twin vertical tail configuration (see figure 6.10-7) has a regular horizontal tail, but two separate and often parallel vertical tails. The twin vertical tail largely improves the directional controllability of an aircraft. Two short span vertical tails have smaller mass moment of inertia about x-axis, compared with a long span vertical tail. Thus a twin tail has the same directional control power, while it has a less negative effect of the roll control. In addition, both rudders are almost out of the fuselage wake region, since they are not located along fuselage center line. A disadvantage of this configuration is that they have slightly heavier weight compared with the conventional tail. Several modern fighter aircraft such as F-14 Phantom (Figure 5.46), McDonnell Douglas F-15 Eagle (Figure 4.21), and F/A-18 Hornet (Figure 6.7-4, 2.11, and 6.12) employ the twin tail configuration.

8. Boom-mounted

Sometime some specific design requirements do not allow the aircraft designer to select the conventional tail configuration. For instance, if a prop-driven engine must be installed at the rear of the fuselage, a conventional tail will tend to have a low efficiency. The reason is the interference between the propeller flow and the tail. One of the options is to use two booms and install the tail at the end of the booms (see figure 6.10-8). This option in turn, allows using a shorter fuselage, but overall aircraft weight would be slightly heavier. Two options are: 1. U-tail, 2. Inverted U-tail. The reconnaissance aircraft Reims F337F Super Skymaster (Figure 6.12-8) and Rutan Voyager (Figure 4.20) employs a boom mounted U-tail. The twin turboprop light utility aircraft Partenavia PD.90 Tapete Air Truck employs a boom mounted inverted U tail configuration which allows for an integrated loading ramp/air-stair.

9. Other configurations

There is variety of other unconventional tail configurations which are usually the forced options to a designer. For instance, sometimes some specific mission requirements such as loading, operational, structural, and engine requirements removes the conventional or T-tail configuration from the list of possible options. Thus, the designer must come up with a new configuration to make an aircraft trimmed and stable throughout flight. Few invented unconventional configurations are as follows: 1. Boom mounted twin vertical tails plus canard (e.g. Rutan Voyager), 2. Boom mounted twin vertical tails plus two separated horizontal tail (e.g. Space Ship One (figure 6.12-9)), 3. Twin T-tail (e.g. Global Flyer (figure 6.12-9)), 4. T-tail plus two fins and an auxiliary fixed horizontal tail (e.g. Beech 1900 D of Continental Express), 4. Ring tail (e.g. Cagny 2000), and 5. Triple vertical tail.

6.5. Canard or Aft Tail

One of the critical issues in the design of the horizontal tail is the selection of the location of horizontal tail. The options are: 1. Aft tail (or sometimes referred to as tail aft), and 2. Fore plane or Canard[3] (sometimes referred to as tail-first). As discussed before, the primary function of the horizontal tail is longitudinal trim, and then, longitudinal stability. Both aft tail and canard are capable of satisfactorily fulfilling both mission requirements. However, there are several aspects of flight features that are influenced differently by either of these two options. It is interesting to note that the first aircraft in history (i.e. Wright Flyer) had canard configuration. Canard configuration is not as popular as aft tail, but several GA and military and few transport aircraft employ canard. Examples are Rutan VariEze (Figure 3.12), Rutan Voyager (Figure 4.20),

[3] Canard is originally a French word which means “duck”. Some early aircraft such as French Canard Vision had a tail-first configuration which was seen by observers to resemble a flying duck.

Mirage 2000, Dassault Rafale (Figure 6.8), Eurofighter Typhoon (Figure 3.7), B-1B Lancer, Saab Viggen, Grumman X-29, Piaggio P-180 Avanti (figure 6.8-6), XB-70 Valkyrie, and Beechcraft Starship (Figure 6.18).

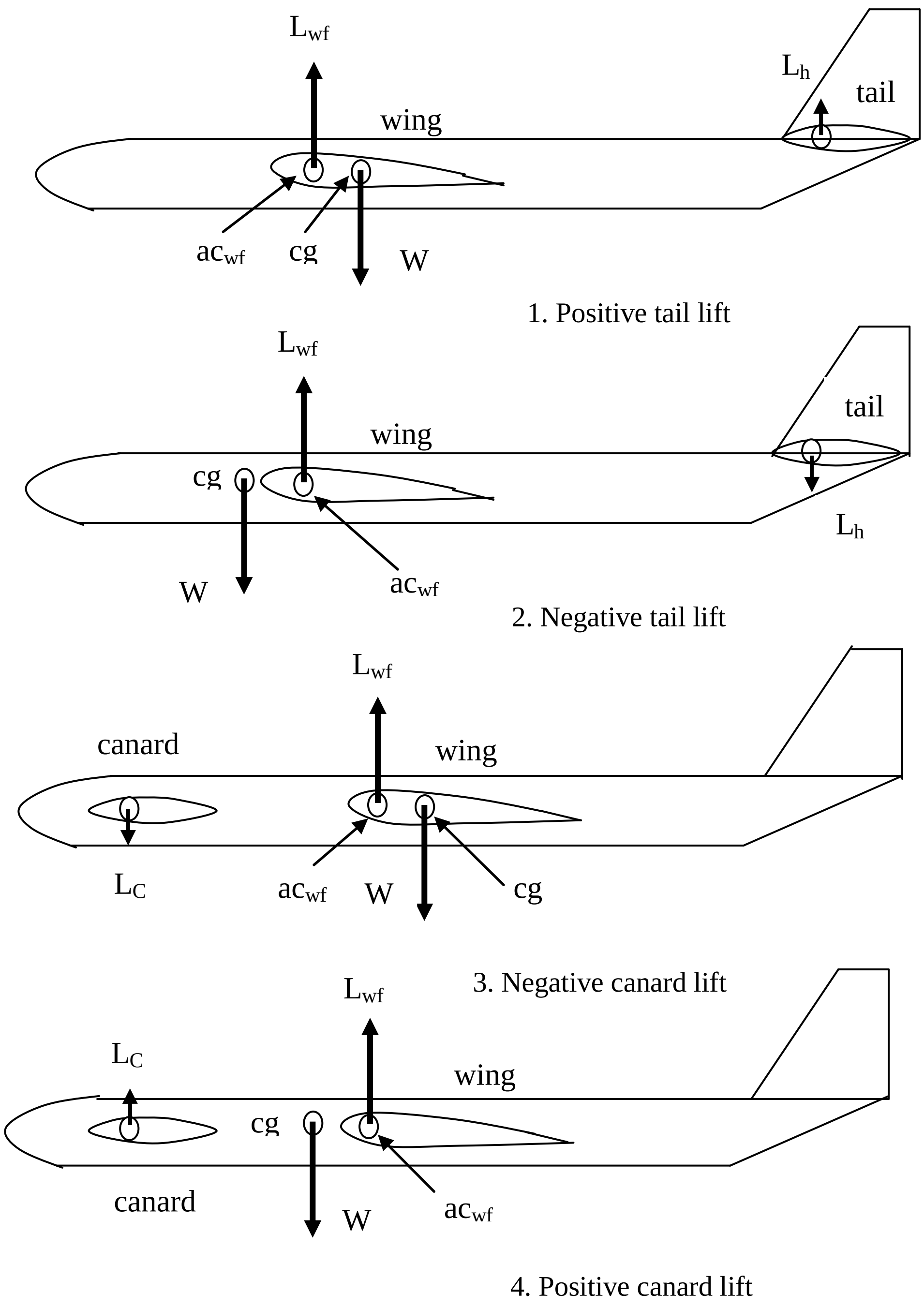


Figure 6.13. The lift of the tail (or canard) in four configurations

To comprehend the fundamental differences between an aft tail and a canard, consider four aircraft configurations as shown in Figure 6.12 which two aircraft have aft tail while the other two have canard. In this figure, the wing nose-down pitching moment is not shown for simplicity. The difference between each two figure is the location of the cg compared with the wing-fuselage aerodynamic center. This simple difference causes a variety of advantages and disadvantages for canard over the conventional aft tail. In all four configurations, the longitudinal trim must hold:

$$\sum M_{cg} = 0 \Rightarrow M_{o_{wf}} + L_h \cdot l_h + L_{wf}(h - h_o)\overline{C} = 0 \qquad \text{(aft tail configuration)} \qquad (6.37a)$$

$$\sum M_{cg} = 0 \Rightarrow M_{o_{wf}} + L_C \cdot l_c + L_{wf}(h - h_o)\overline{C} = 0 \qquad \text{(canard configuration)} \qquad (6.37b)$$

$$\sum F_z = 0 \Rightarrow W = L_{wf} + L_h \qquad \text{(aft tail configuration)} \qquad (6.38a)$$

$$\sum F_z = 0 \Rightarrow W = L_{wf} + L_C \qquad \text{(canard configuration)} \qquad (6.38b)$$

where L_C denotes the canard lift. Equations 6.37 and 6.38 indicate that the aft tail lift or canard lift might be positive, or negative, depending upon the location of aircraft cg relative to wing-fuselage aerodynamic center (see figure 6.13). Equations 6.37b and 6.6.38b are utilized to determine the value and the direction of the canard lift to satisfy trim requirements. It is obvious that the canard lift is sometimes negative (see figure 6.13-3). Keeping in mind the above basic difference between aft tail and canard, a comparison between features of canard as compared with aft tail is presented.

The canard avoids deep stall 100%. This gets interesting, when we note that about 23 percent of world aircraft crash relates to deep stall. Consider a pilot who intends to increase the wing angle of attack in order to either take-off, or climb, or land, or land. Since canard is located forward of the wing, the canard will stall first (i.e. before wing stalls). This causes the canard to drop and exits out of stall before the wing enters to stall. The canard drop is due to the fact that when it stalls, its lift is reduced and causes the aircraft nose to drop. This is regarded as of the major advantages of canard and makes the canard configuration mush safer that aft tail configuration.

Since the canard stalls before the main wing, the wing can never reach its maximum lift capability. Hence, the main wing must be larger than on the conventional configuration, which increases its weight and also zero-lift drag.

1. A canard has a higher efficiency when compared with aft tail. The reason is that it is located in front of wing, so the wing wake does not influence the canard aerodynamic characteristics. Wing, however, is located aft of canard; hence, it is negatively affected by the canard wake. Thus a wing in a canard configuration has a lower aerodynamic efficiency (i.e. lower lift) when compared with an aircraft with aft tail configuration.
2. It is not appropriate to employ canard when the engine is pusher and located at the fuselage nose. The reason is that the aircraft nose will be heavy and the cg adjustment is difficult. Moreover, the structural design of fuselage nose is somewhat complicated, since it must hold both engine and canard.
3. An aircraft with a canard configuration tends to have a smaller static margin compared with an aircraft with a conventional aft tail configuration. In another word, the distance between aircraft neutral point and aircraft center of gravity is shorter. This makes the

canard aircraft longitudinally statically less stable. This feature is regarded as a disadvantage for canard configuration.

4. The center of gravity range in an aircraft with a canard configuration tends to be wider; hence, it is more flexible in the load transportation area.
5. Due to the forward location of a canard, the aircraft cg moves slightly forward compared with an aircraft with a conventional aft tail configuration. This feature requires a slightly larger vertical tail for directional trim and stability.
6. A canard tends to generate a lower "trim drag" compared with an aft tail. In another word, a canard aircraft produces less lift-dependent drag to longitudinally trim the aircraft. However, this feature may leads in a larger wetted area (S_{wet}).
7. One of the potential design challenges in a canard aircraft is to optimally locate the fuel tank. The general rule is to place the fuel tank near the aircraft center of gravity as close as possible, in order to avoid a large movement of cg during the flight operation. The aircraft cg in a canard configuration, if fuel tank is inside the wing, is often forward of the fuel tank. To improve the cg location, designers would rather to place the fuel tank into the fuselage, which in turn increases the possibility of aircraft fire. Another solution is to considerably increase the wing root chord (i.e. employing strake) and to place the fuel tank in wing root. But this technique increases the wing wetted area and reduces the cruise efficiency. The canard aircraft Beechcraft Starship (Figure 6.18) has a wing strake and utilizes this technique.
8. A canard obscures the view of the pilot. This is another disadvantage of the canard configuration.
9. Often times the canard generates a positive lift (see figure 6.13-4) while a conventional tail often produces a negative lift (see figure 6.13-2). The reason is that the aircraft cg in a canard configuration is often forward of the wing-fuselage ac. The aircraft cg in a conventional tail configuration is typically aft of the wing-fuselage ac. Recall that the cg move during flight as the fuel burns. The cg range, in a modern aircraft with a conventional tail or a canard is usually determined such that the cg is most of the times forward of the wing aerodynamic center. However, in a fewer instances of cruising flight, the cg is aft of the wing aerodynamic center. Thus, in an aircraft with a conventional tail, during the cruising flight, the cg usually moves from the most forward location toward the most aft location. However, in an aircraft with a canard, during the cruising flight, the cg often moves from the most aft location toward the most forward location.

   Thus, a canard often generates part of the aircraft lift, while a tail most of the times cancels part of the lift generated by the wing. This feature tends to reduce the aircraft weight and increases the aircraft cruising speed. In addition, during a take-off which the wing nose-down pitching moment is large, the canard lift is higher. Using the same logic, it can be shown that the canard lift is higher during supersonic speeds. Recall that in a supersonic speed, the wing aerodynamic center move aft toward about 50 percent of the mean aerodynamic chord. This is one of the reasons that some European supersonic fighters, such as Mirage 2000 (Figure 9.12), have employed the canard configuration.
10. Item 9 results in the following conclusion: An aircraft with a canard is slightly lighter than an aircraft with a conventional tail.
11. In general, the canard aerodynamic and stability analysis techniques are considerably more complicated than the technique to evaluate the aerodynamic feature and stability

analysis of the conventional tail configuration aircraft. Literature surveys include a variety of published materials regarding conventional tail, while much less papers and technical reports are available for canard analysis. Thus the design of a canard is more time intensive and more complicated than the conventional tail design.

12. A canard configuration seems to be more stylish and more attractive than a conventional tail.
13. A canard is more efficient for fulfilling the longitudinal trim requirements, while a conventional tail tends to be more efficient for fulfilling the longitudinal control requirements.

In general, canard designs fall into two main categories: the lifting-canard and the control-canard. As the name implies, in a lifting-canard the weight of the aircraft is shared between the main wing and the canard wing. The upward canard lift tends to increase the overall lift capability of the configuration. With a lifting-canard, the main wing must be located further aft of the cg range than with a conventional aft tail, and this increases the pitching moment caused by trailing-edge flaps. The first airplane to fly, the *Wright Flyer*, and *X-29* had a lifting-canard. Figure 6.18 depicts two (Beech Starship and Saab Gripen (Figures 6.18 and 2.7)) aircraft with canard configuration. In is interesting to know that about 98% of American aircraft are conventional, not canard.

In the control-canard, most of the weight of the aircraft is carried by the main wing and the canard wing serves primarily as the longitudinal control device. A control-canard could be all moving or could have a large elevator. The control-canard has often higher aspect ratio and employs a thicker airfoil section than a lifting-canard. A control-canard mostly operates at zero angle of attack. Fighter aircraft with a canard configuration, such as Eurofighter Typhoon (Figure 3.7), typically have a control-canard. One benefit obtainable from a control-canard is avoidance of pitch-up. An all-moving canard capable of a significant nose-down deflection will protect against pitch-up. Control canards have poor stealth characteristics, because they present large moving surfaces forward of the wing.

The pros and cons of the canard versus a conventional tail configuration are numerous and complex and it is hard to say which is superior without considering a specific design requirement. One must systems engineering technique to compromise and to decide the tail configuration. In the preliminary design phase, the suggestion is to begin with a conventional tail, unless the designer has a solid reasoning on to employ a canard configuration.

6.6. Optimum Tail Arm

One of the tail parameters that must be determined during the tail design process is the tail arm (l_t), which is the distance between tail aerodynamic center to the aircraft center of gravity (see figure 6.3). Tail arm serves as the arm for the tail pitching moment (i.e. tail lift multiplied by tail arm) about aircraft cg to maintain the longitudinal trim. To determine the tail arm one must establish the criteria based on the design requirements. Two basic tail parameters which interact most are tail arm and tail area, the latter is responsible for generation of the tail lift. As the tail arm is increased, the tail area must be decreased, while as the tail arm is reduced, the tail area must be increased. Both short arm (as in fighters), or long arm (as in most transport aircraft) are capable of satisfying longitudinal trim requirements, given the appropriate necessary tail area.

But the question is that what tail arm is the optimum one. To answer this question, one must look at the other design requirements.

Two very significant aircraft general design requirements are aircraft low weight and low drag. Both of these may be combined and translated as the requirement for a low aircraft wetted area. As the horizontal tail arm is increased, the fuselage wetted area is increased, but horizontal tail wetted area is decreased. Also, as the horizontal tail arm is decreased, the fuselage wetted area is decreased, but horizontal tail wetted area is increased. Hence, we are looking to determine the optimum tail arm to minimize drag; which in turn means to minimize the total wetted area of the aft portion of the aircraft. The following is a general educational approach to determine the optimum tail arm; hence, one must develop his/her own technique and derive more accurate equation based on the suggested approach. The approach is based on the fact that the aircraft zero-lift drag is essentially a function of the aircraft wetted area.

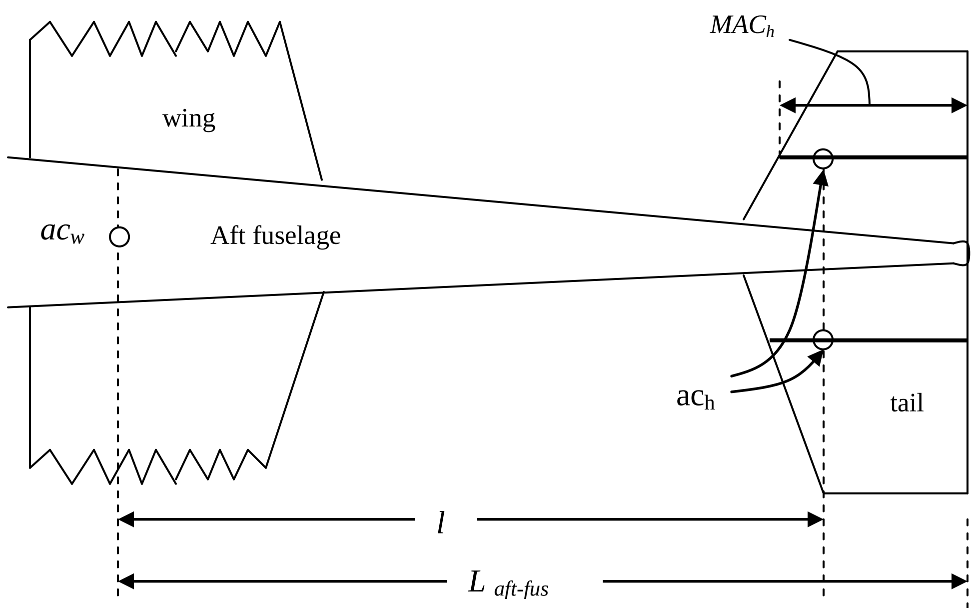


Figure 6.14. Top view of aft portion of the aircraft

Therefore, if the total wetted area is minimized, the aircraft zero-lift drag will be minimized. Moreover, the technique will influence the fuselage length, since the aft portion of the fuselage must structurally support the tail.

Consider the top view of aft aircraft (see figure 6.14) that includes aft portion of the fuselage plus the horizontal tail. The wetted area of the aft portion of the aircraft is the summation of the wetted area of the aft portion of the fuselage ($S_{wet_{aft\,fus}}$) plus the wetted area of the horizontal tail ($S_{wet_{ht}}$).

$$S_{wet_{aft}} = S_{wet_{aft\,fus}} + S_{wet_h} \tag{6.39}$$

Here we assume that aft portion of the fuselage is conical. Hence, the wetted area of the aft portion of the fuselage is

$$S_{wet_{aft\,fus}} = \frac{1}{2}\pi \cdot D_f L_{fus_{aft}} \qquad (6.40)$$

where D_f is the maximum fuselage diameter and $L_{fus_{aft}}$ is the length of the aft portion of the fuselage. At the moment, it is assumed that $L_{fus_{aft}}$ is equal to half of the fuselage length (L_f). On the other hand, the wetted area of the horizontal tail is about twice the tail planform area:

$$S_{wet_t} \approx 2S_h \qquad (6.41)$$

But, the tail volume coefficient is defined as in equation 6.24, so:

$$\overline{V}_H = \frac{lS_h}{\overline{C}S} \Rightarrow S_h = \frac{\overline{C} \cdot S \cdot \overline{V}_H}{l} \qquad (6.42)$$

So

$$S_{wet_h} \approx 2\frac{\overline{C}S\overline{V}_H}{l} \qquad (6.43)$$

Substituting equation 6.41 and 6.43 into 6.39 yields:

$$S_{wet_{aft}} = \frac{1}{2}\pi D_f L_{f_{aft}} + 2\frac{\overline{C}S\overline{V}_H}{l} \qquad (6.44)$$

The relationship between $L_{fus_{aft}}$ and l depends upon the location of the horizontal tail (see figure 6.14). We simply assume they are equal ($L_{fus_{aft}} = l$). This assumption is not accurate for every aircraft configuration, but it is a reasonable assumption based on the data on Table 6.2. This assumption will be modified later. In order to minimize zero-lift drag of the aft part of the aircraft, we have to differentiate the wetted area of the aft part of the aircraft with respect to tail arm (see figure 6.15) and then set it equal to zero. The differentiation yields:

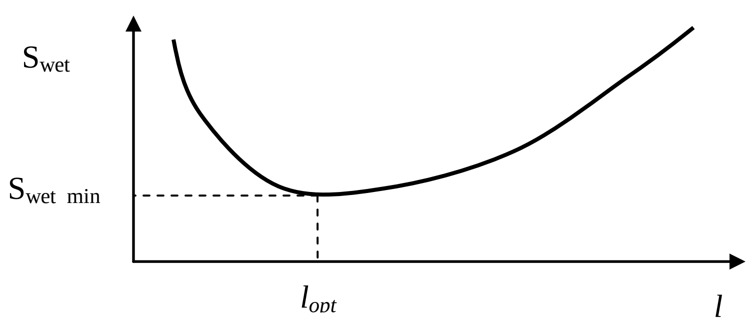


Figure 6.15. The variation of wetted area with respect to tail arm

$$\frac{\partial S_{wet_{aft}}}{\partial l} = \frac{1}{2}\pi D_f + 2\frac{\overline{C}S\overline{V}_H}{l^2} = 0 \qquad (6.45)$$

The optimum tail arm is obtained by solving this equation as follows:

$$l_{opt} = \sqrt{\frac{4\overline{C}S\overline{V}_H}{\pi D_f}} \qquad (6.46)$$

To compensate for our inaccurate assumption, we add a fudge factor as follows:

$$l_{opt} = K_c\sqrt{\frac{4\overline{C}S\overline{V}_H}{\pi D_f}} \qquad (6.47)$$

where K_c is a correction factor and varies between 1 and 1.4 depending on the aircraft configuration. The K_c = 1 is used when the aft portion of the fuselage has a conical shape. As the shape of the aft portion of the fuselage goes further away from a conical shape, the K_c factor is increased up to 1.4. As a general rule, for a single-seat single engine prop-driven GA aircraft, the factor K_c is assumed to be 1.1, but for a transport aircraft, K_c will be 1.4. Note that in a large transport aircraft, the most of the fuselage shape is cylindrical, and only its very aft portion has a conical shape. Therefore if the horizontal tail is located at l_{opt}, the wetted area of the aft part of the aircraft will be minimized, so the drag of the aft part of the aircraft will be minimized. When the horizontal tail arm is less than three time the wing MAC ($3\overline{C}$), the aircraft is said to be short-coupled. An aircraft with such tail configuration possesses the longitudinal trim penalty (e.g. fighters). Example 6.1 provides a sample calculation.

Example 6.1

Consider a twin-seat GA aircraft whose wing reference area is 10 m^2 and wing mean aerodynamic chord is 1 m. The longitudinal stability requirements dictate the tail volume coefficient to be 0.6. If the maximum fuselage diameter is 117 cm, determine the optimum tail arm and then calculate the horizontal tail area. Assume that the aft portion of the fuselage is conical.

Solution:

The aircraft is a GA and has two seats, so the factor K_c is assumed to be 1.4. Using equation 6.47, we have

$$l_{opt} = K_c\sqrt{\frac{4\overline{C}S\overline{V}_H}{\pi D_f}} = 1.4\times\sqrt{\frac{4\times 0.6\times 1\times 10}{\pi\times 1.17}} \Rightarrow l_{opt} = 3.577 \quad m \qquad (6.47)$$

The horizontal tail area is calculated by employing tail volume coefficient equation as follows:

$$\overline{V}_H = \frac{lS_h}{\overline{C}S} \Rightarrow S_h = \frac{\overline{V}_H\overline{C}S}{l} = \frac{0.6\times 1\times 10}{3.577} = 1.677 \quad m^2 \qquad (6.24)$$

6.7. Horizontal Tail Parameters

After the tail configuration is determined, the horizontal tail and vertical tail can be designed almost independently. This section presents the technique to design the horizontal tail and the method to determine horizontal tail parameters. Since the horizontal tail is a lifting surface and also several characteristics of wing and tail are similar (as discussed in Chapter 5), some aspects of the horizontal tail such as taper ratio, sweep angle, dihedral angle and airfoil section, are discussed in brief. The horizontal tail design is also an iterative process and is strongly functions of several wing parameters and few fuselage parameters. Hence, as soon as the major wing and fuselage parameters are changed, the tail must be redesigned and its parameters need to be updated.

6.7.1. Horizontal Tail Design Fundamental Governing Equation

Horizontal tail design fundamental governing equation must be driven based on the primary function of the horizontal tail (i.e. longitudinal trim). Figure 6.2 depicts a general case of an aircraft along with the sources of forces along x and z axes, and moments about y axis which are influencing the aircraft longitudinal trim. The longitudinal trim requires that the summation of all moments about y axis must be zero:

$$\sum M_{cg} = 0 \Rightarrow M_{o_{wf}} + M_{L_{wf}} + M_{L_h} + M_{o_h} + M_{T_{eng}} + M_{D_w} = 0 \tag{6.48}$$

where $M_{o_{wf}}$ denotes nose-down wing-fuselage aerodynamic pitching moment, $M_{L_{wf}}$ denotes the pitching moment generated by the wing-fuselage lift, M_{L_h} denotes the pitching moment generated by the horizontal tail lift, M_{o_h} denotes nose-down horizontal tail aerodynamic pitching moment, $M_{T_{eng}}$ denotes the pitching moment generated by the engine thrust, and M_{D_w} denotes the pitching moment generated by the wing drag. The sign of the each pitching moment depend upon the location of the source force relative to the aircraft center of gravity. This equation must hold at all flight conditions, but the horizontal tail is designed for the cruising flight, since the aircraft spends much of its flight time in cruise. For other flight conditions, a control surface such as the elevator will contribute.

Based on the aerodynamics fundamentals, two aerodynamic pitching moments of wing and horizontal tail are always nose down (i.e. negative). The sign of wing drag moment depends on the wing configuration. For instance, a high-wing generates a nose up pitching moment, while a low-wing generates a nose down pitching moment. The sign of engine thrust moment depends on the thrust line and engine incidence. If the engine has a setting angle other than zero, both horizontal and vertical components will contribute to the longitudinal trim. The major unknown in this equation is the horizontal tail lift. Another requirements for longitudinal trim is that the summations of all forces along x and z-axes must be zero. Only the summation of forces along the z axis contributes to the tail design:

$$\sum F_z = 0 \Rightarrow L_{wf} + T\sin(i_T) + L_h = 0 \tag{6.49}$$

where T is the engine thrust and i_T is the engine thrust setting angle (i.e. the angle between the thrust line and the x-axis). This angle almost always is not zero. The reason is the engine thrust contribution to the aircraft longitudinal stability. The typical engine setting angle is about 2 to 4 degrees. The horizontal tail designer should expand two equations of 6.48 and 6.49 and solve simultaneously for two unknowns of wing lift and horizontal tail lift. The latter is employed in the horizontal tail design. The derivation is left to the reader.

It is presumed that the horizontal tail designer is familiar with the flight dynamics principles and is capable of deriving the complete set of longitudinal trim equations based on the aircraft configuration. Since the goal of this textbook is educational, so a simple version of longitudinal trim equation is employed. If the pitching moments of engine thrust, wing drag, and horizontal tail pitching moment are ignored (as shown in figure 6.3), the non-dimensional horizontal tail design principle equation is as derived earlier:

$$C_{m_{owf}} + C_L\left(h - h_o\right) - \eta_h \overline{V}_H C_{L_h} = 0 \tag{6.29}$$

The full derivation has been introduced is Section 6.2. This equation has three terms, the last of which is the horizontal tail contribution to the aircraft longitudinal trim. The cruising flight is considered for horizontal tail design application. The equation has only two unknowns (i.e. $\overline{V}_H$ and C_{Lh}). The first unknown (horizontal tail volume coefficient; $\overline{V}_H$) is determined primarily based on the longitudinal stability requirements. The longitudinal flying qualities requirements govern this parameter. The reader is encouraged to consult with References [1] and [6] for a full guidance. However, Chapter 12 presents a summary of longitudinal flying qualities requirements. A higher value for $\overline{V}_H$ results in a longer fuselage, and/or smaller wing and and/or a larger horizontal tail.

As the value of $\overline{V}_H$ is increased, the aircraft becomes longitudinally more stable. On the other hand, a more stable aircraft means a less controllable flight vehicle. Hence, a lower value for $\overline{V}_H$ causes the aircraft to become longitudinally more controllable and less stable. If the horizontal tail design is at the preliminary design phase; and the other aircraft components have not yet been designed; a typical value for $\overline{V}_H$ must be selected. Table 6.4 illustrates the typical values for horizontal and vertical tail volume coefficients. The values are driven from the current successful aircraft statistics. A number from this table based on the aircraft mission and configuration is recommended at the early design phase. When the other aircraft components are designed and their data are available, a more accurate value for $\overline{V}_H$ may be determined.

The variable "h_o" denotes the non-dimensional wing-fuselage aerodynamic center ($\frac{X_{ac_{wf}}}{\overline{C}}$) position. A typical value for "h_o" is about 0.2 to 0.25 for majority of aircraft configurations. [6] and [1] introduce a precise technique to evaluate the value of h_o. Another significant parameter in equation 6.29 is "h". The parameter "h" denotes the non-dimensional aircraft center of gravity (cg) position ($\frac{X_{cg}}{\overline{C}}$). The value for "h" must be known prior to the horizontal tail design.

No	Aircraft	Horizontal tail volume coefficient ($\overline{V}_H$)	Vertical tail volume coefficient ($\overline{V}_V$)
1	Glider and motor glider	0.6	0.03
2	Home-built	0.5	0.04
3	GA-single prop-driven engine	0.7	0.04
4	GA-twin prop-driven engine	0.8	0.07
5	GA with canard	0.6	0.05
6	Agricultural	0.5	0.04
7	Twin turboprop	0.9	0.08
8	Jet trainer	0.7	0.06
9	Fighter aircraft	0.4	0.07
10	Fighter (with canard)	0.1	0.06
11	Bomber/military transport	1	0.08
12	Jet Transport	1.1	0.09

Table 6.4. Typical values for horizontal and vertical tail volume coefficients

Chapter 11 is dedicated to the techniques and methods to determine the aircraft cg position, provided the details of geometries of all aircraft components. However, if at the early stages of the horizontal tail design, the other aircraft components such as fuselage, engine, and landing gear have not yet been designed, the only option is to pick a value for "h". The best value is a mid-value between the most forward and the most aft position of the aircraft cg. This minimized the aircraft trim drag while in cruise. This is based on a logical assumption that the aircraft cg is at it one end of the extreme position (say most forward) at the beginning of the cruise, and moves to another end of the extreme position (say most aft) at the end of the cruise.

In contrast, in order to reduce the longitudinal control effort during a cruising flight, the aircraft cg is recommended to be close to the wing-fuselage aerodynamic center. The aircraft non-dimensional center of gravity limit (Δh) is the difference between the most forward and the most aft position of the aircraft cg. The typical values for the aircraft non-dimensional center of gravity limit are:

$$\Delta h = 0.1 \quad to \quad 0.3 \tag{6.50}$$

This means that a typical value for the most forward of the aircraft cg is about 10 percent of the wing mean aerodynamic chord. In addition, a typical value for the most aft of the aircraft cg is about 30 percent of the wing mean aerodynamic chord. Therefore, a proper assumption for the value of h at the early stage of the horizontal tail design would be about 0.2. As soon as a more realistic value for the aircraft cg position (h) is available, the horizontal tail design must be updated. The value for the aircraft lift coefficient (C_L) in equation 6.29 is determined based on the cruising velocity, cruise altitude, and the aircraft average weight (equation 5.10). Finally, by solving the equation 6.29, the only unknown (C_{Lh}), is determined.

At this moment, three horizontal tail parameters are decided (i.e. $\overline{V}_H$, C_{Lh} and l). On the other hand, since the tail volume coefficient is a function of horizontal tail area (S_h), the horizontal tail area is readily determined using equation 6.24. By the technique that has just been introduced, three horizontal tail parameters that have been determined are as follows:

1. horizontal tail planform area (S_h)
2. horizontal tail moment arm (l)
3. horizontal tail cruise lift coefficient (C_{Lh})

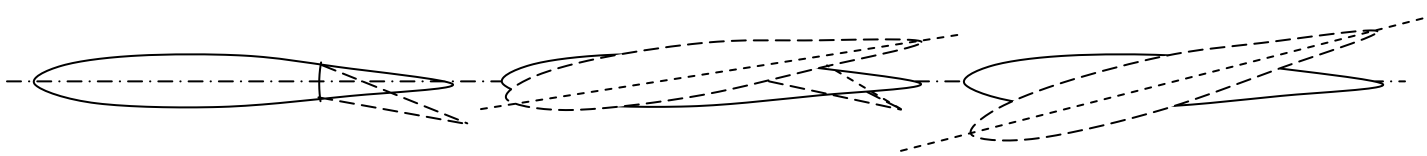
It is important to remember that the design is an iterated process, so as soon as any assumption (such as aircraft cg) is changed; the horizontal tail design must be revised.

6.7.2. Fixed, All Moving, or Adjustable

Due to the fact that the aircraft has numerous flight conditions such as various speeds, cg locations, weights, and altitudes, the longitudinal trim requirements are satisfied only through change in horizontal tail lift. Since the horizontal tail has a fixed planform area and fixed airfoil section, the only way to change the tail lift is to vary its angle of attack (α_h). There are three tail setting configurations (as sketched in figure 6.16) to fulfill a change the angle of attack:

1. Fixed horizontal tail;
2. Adjustable tail;
3. All-moving tail.

A fixed tail is permanently attached to the fuselage by some joining techniques such as screw and nut or welding. A fixed tail angle of attack cannot be varied unless by pitching up or down the fuselage nose. On the other hand, the angle of attack of an all moving tail is easily changed by the pilot using the forward or aft motion of the stick inside the cockpit.



a. Fixed b. Adjustable c. All moving

Figure 6.16. Three horizontal tail setting configurations

There are several basic differences between these options. First of all, a fixed tail is much lighter, cheaper and structurally easier to design compared with an all moving tail. Moreover, a fixed tail is safer than all moving tail, due to the possibility of failure of a moving mechanism. On the other hand, an aircraft with all moving tail (such as in fighter aircraft Dassault Rafale as shown in figure 6.8) is more controllable and maneuverable than an aircraft with a fixed tail. One difference between these two tails is that a fixed tail is equipped with a longitudinal control surface (i.e. elevator); while an all moving tail does not have any separate deflectable section. In general, the trim drag of a fixed tail is higher than that of an all moving tail. An all moving tail is sometimes referred to as a variable incidence tailplane.

1. Adjustable horizontal tail in Fairchild C-26A Metro III 2. All moving tail in Panavia Tornado
(Courtesy of Luis David Sanchez) (Courtesy of Antony Osborne)
Figure 6.17. An adjustable tail and an all moving tail

A tail option which has some advantages of a fixed tail and some advantages of the moving tail is referred to as adjustable tail (such as in Fairchild C-26A Metro III as shown in figure 6.17-1). As the name implies, an adjustable tail allows the pilot to adjust its setting angle for a long time. The adjustment process usually happens before the flight; however, a pilot is allowed to adjust the tail setting angle during the flight operation. An adjustable tail employs an elevator, but a major between an adjustable tail and all moving tail is in the tail rotation mechanism. An all moving tail is readily and rapidly (in a fraction of a second) rotated about its hinge by the pilot. But, the angle of attack adjustment process for an adjustable tail takes time (few or even several seconds). The range of deflections of an adjustable tail (about +5 to -12 degrees) is considerably less than that of an all moving tail (about +15 to -15 degrees). For instance, the tailplane deflection for transport aircraft Boeing 777 is 4° up and 11° down.

If the longitudinal maneuverability is not a desired design requirement, it is recommended to employ a fixed tail configuration. But the aircraft is required to be able to perform fats maneuver, the appropriate option is an all moving tail. On the other hand, if the flight cost is a significant issue in the design requirements list, it is better to employ an adjustable tail. In general, most GA and small transport aircraft (e.g. Cessna 172 (Figure 11.15), Jetstream 41) have fixed tail, most large transport aircraft (e.g. Boeing 767 (Figure 5.4), Airbus 340 (Figure 8.20)) utilize the adjustable tail, and most fighter aircraft (e.g. F/A-18 Hornet (Figures 2.11, 6.12), F-16 Falcon (Figure 3.12), and Harrier GR. Mk 7 (Figure 4.19) employ all moving tail. Table 6.5 shows the setting configuration of horizontal tail for several aircraft. Figure 6.17 demonstrates the adjustable horizontal tail of Fairchild C-26A Metro III, and all moving horizontal tail of Panavia Tornado.

6.7.3. Airfoil Section

Horizontal tailplane is a lifting surface (similar to the wing) and requires a special airfoil section. The basic fundamentals about airfoil section (definition, parameters, selection criteria, and related calculation) has been presented in Section 5.4, hence they are not repeated here. In summary, tailplane requires an airfoil section that is able to generate the required lift with minimum drag and minimum pitching moment. The specific horizontal tail airfoil requirements are described in this section.

No	Aircraft	m_{TO} (kg)	Tail type	Airfoil	$(t/C)_{max}$ (%)	$\overline{V}_H$	S_h/S	AR_h	λ_h	Λ_h (deg)	Γ_h (deg)	i_h (deg)	
												+	-
1	Wright Flyer	420	Moving	Cambered plate	low	-0.36	0.16	5.7	1	0	0	-	-
2	Cessna 177	1,100	Fixed	NACA 0012/0009	10.5	0.6	0.2	4	1	0	0	-	-
3	Cessna Citation I	5,375	Fixed	NACA 0010/0008	9	0.75	0.26	5.2	0.5	-	9	-	-
4	Beech Starship	6,759	Fixed	-	-	-0.96	0.22	10.2	0.5	33	3	-	-
5	Fokker F-27	19,773	Fixed	NACA 63A-014	14	0.96	0.23	6	0.4	0	6	-	-
6	Boeing 737-100	50,300	Adjustable	12%-9%	10.5	1.14	0.32	4.16	0.38	30	7	-	-
7	Boeing 707-320	151,320	Adjustable	BAC 317	11.6	0.63	0.216	3.37	0.42	35	7	0.5	14
8	Boeing 747-100	333,390	Adjustable	-	9	1	0.267	3.6	0.26	37	8.5	1	12
9	DC-8-10	141,000	Adjustable	DSMA-89-90	8.75	0.59	0.203	4.04	0.33	35	10	2	10
10	Airbus 300B	165,000	Adjustable	-	-	1.07	0.27	4.13	0.5	32.5	6	3	12
11	Lockheed C-130 Hercules	70,305	Fixed	Inverted NACA	12	1	0.313	5.2	0.36	7.5	0	-	-
12	Lockheed L-1011	211,000	Adjustable	-	8	0.928	0.37	4	0.33	35	3	0	14
13	Lockheed C-5A	381,000	Adjustable	-	10	0.7	0.156	4.9	0.36	24.5	-4.5	4	12
14	Eurofighter 2000	21,000	Movable	-	-	-0.1	0.048	3.4	0.34	45	17	-	-
15	F-15 Eagle	36,741	Movable	-	-	0.24	0.183	2.3	0.36	48	0	-	-

Table 6.5. Horizontal tail characteristics for several aircraft

a. Beech Starship
(Courtesy of Ken Mist)

b. Saab JAS-39B Gripen
(Courtesy of Antony Osborne)

Figure 6.18. Two aircraft with canard configuration

Basically, the tailplane airfoil lift curve slope ($C_{L_{\alpha_t}}$) must be as large as possible along with a considerably wide usable angles of attack. Since the aircraft center of gravity moves during the cruising flight, the airfoil section must be able to create sometimes a positive (+L_h) and sometimes a negative lift (-L_h). This requirement necessitates the tailplane to behave similar in both positive and negative angles of attack. For this reason, a symmetric airfoil section is a suitable candidate for horizontal tail.

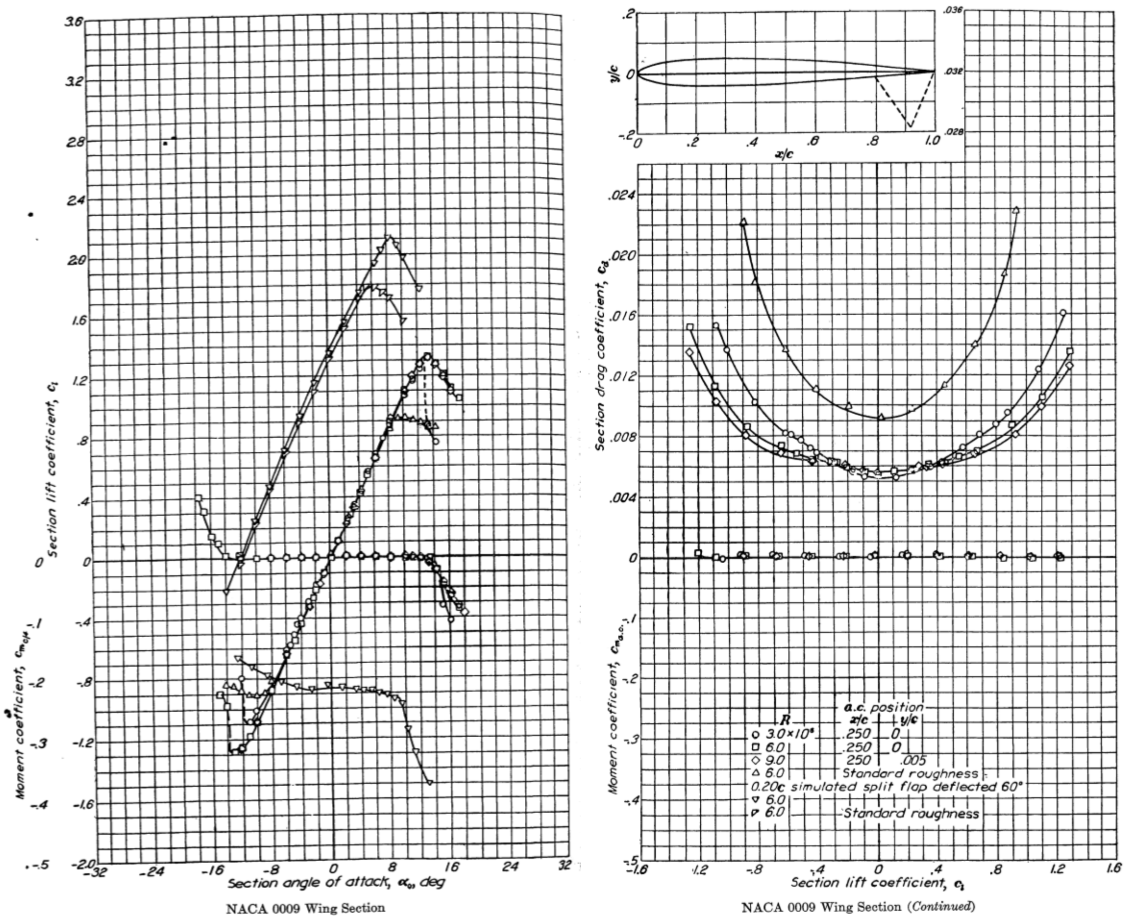


Figure 6.19. Characteristics graphs of NACA 0009 airfoil section [8]

Recall from Chapter 5 that the indication of a symmetric airfoil is that the second digit in a 4-digit and the third digit in a 5-digit and 6-series NACA airfoil sections is zero. This denotes that the airfoil design lift coefficient and zero-lift angle of attack are both zero. NACA airfoil sections such as 0009, 0010, 0012, 63-006, 63-009, 63-012, 63-015, 63-018, 64-006, 64-012, 64A010, 65-009, 65-015, 66-012, 66-018, and 66-021 are all symmetric airfoils. Reference [8] is a rich collection for NACA airfoil sections.

In several GA aircraft, NACA airfoil sections 0009 or 0012 (with 9% or 12% maximum thickness-to-chord ratio) are employed for horizontal tail. Both of this NACA airfoil sections are symmetric. Moreover, it is desired that the horizontal tail never stalls, and the wing must stall before the tail. Hence, the stall feature of the tail airfoil section (sharp or docile) is not significant.

In addition, another tail requirement is that horizontal tail must be clean of compressibility effect. In order the tail to be out of the compressibility effect, the tail lift coefficient is determined to be less than the wing lift coefficient. To insure this requirement, the flow Mach number at the tail must be less than the flow Mach number at the wing. This objective will be realized by selecting a horizontal tail airfoil section to be thinner (say about 2 percent of MAC) than the wing airfoil section. For instance, if the wing airfoil section is NACA 23015 (i.e. $(t/C)_{max}$ = 0.15 or 15%), the horizontal tail airfoil section can be selected to be NACA 0009 (i.e. $(t/C)_{max}$ = 0.9 or 9%). Figure 6.19 shows the characteristics graphs of the NACA 0009 airfoil section.

In an aircraft with an aft tail configuration, when the center of gravity, most of the time, is behind the wing-fuselage aerodynamic center, the horizontal tail must produce a negative lift to longitudinally trim the aircraft. If the aircraft center of gravity range is such that the tail must produce a negative lift coefficient most of the time, an inverted non-symmetric airfoil section may be utilized. This is the case for the cargo aircraft Lockheed C-130B tail airfoil section.

6.7.4. Tail Incidence

When a fixed tail configuration is adopted, the horizontal tail setting angle (i.e. tail incidence); i_t; must be determined. The tail setting angle (i_t) primary requirement is to nullify the pitching moment about cg at the cruising flight. This is the longitudinal trim requirement through which the tail is generating a lift to counteract all other aircraft pitching moments. Tail incidence is determined to satisfy trim design requirement when no control surface (i.e. elevator) is deflected. Although this fixed setting angle satisfies only one flight condition, but it must be such that a mild change (through the application of elevator) is necessary to trim the aircraft on other flight situations.

Looking at the C_L-α graph of the tail airfoil section (such as in figure 6.19), it is noticed that the tail angle of attack is simply a function of the tail lift coefficient. Therefore, as soon as the tail lift coefficient is known, the tail incidence is readily determined by using this graph as the corresponding angle. As already discussed in section 6.2, the tail lift coefficient is obtained from the non-dimensional longitudinal trim equation such as equation 6.29:

$$C_{m_{owf}} + C_L\left(h - h_o\right) - \eta_h \overline{V}_H C_{L_h} = 0 \tag{6.29}$$

In summary, the desired tail lift coefficient is calculated through equation 6.29, and then the tail incidence will be determined by using the C_L-α graph of the tail airfoil section.

$$C_{L_{\alpha_h}} = \frac{C_{L_h}}{\alpha_h} \Rightarrow \alpha_h = \frac{C_{L_h}}{C_{L_{\alpha_h}}} \tag{6.51}$$

This is an initial value for the setting angle and will be revised in the later design phases. The typical value would be about -1 degrees. In case, the tail configuration is adjustable, the highest incidence (usually positive angles) and lowest incidence (usually a negative angle) must be determined. For instance, the large transport aircraft Boeing 727 has an adjustable tail with +4 degrees for most positive incidence and -12.5 degrees for most negative incidence. Table 6.5 introduces the horizontal tail setting angles for several aircraft. So the horizontal tail angle of attack in this aircraft is negative most of the time.

Another factor influencing the value of the tail setting angle is the requirement for longitudinal static stability. Several parameters will affect the aircraft longitudinal static stability, but it can be shown that the "longitudinal dihedral" will have a positive impact on the longitudinal static stability. The term "longitudinal dihedral" is invented by tail designers to transfer the technical meaning of the wing dihedral angle (Γ) form y-z plane to a similar angle in the aircraft x-z plane. As the aircraft lateral stability is benefited from the wing and tail dihedral angles, the aircraft longitudinal stability will be improved by a geometry referred to as the aircraft longitudinal dihedral angle. When the horizontal tail chord line and wing chord line can form a V-shape, it is said that the aircraft has the longitudinal dihedral.

There are a few other technical interpretations for longitudinal dihedral as follows:

1. When the wing (or foreplane such as canard) setting angle is positive and the horizontal tail (or aft plane such as the wing in a canard configuration) angle is negative, the aircraft is said to have longitudinal dihedral.

$$i_w > i_h$$

2. When the wing (or foreplane) lift coefficient is higher than that of the horizontal tail (or foreplane), the aircraft is said to have longitudinal dihedral.

$$C_{Lw} > C_{Lh}$$

3. When the wing (or foreplane) zero-lift angle of attack is higher than that of the horizontal tail (or aft plane), the aircraft is said to have longitudinal dihedral.

$$\alpha_{ow} > \alpha_{oh}$$

4. When the wing (or foreplane) effective angle of attack is higher than that of the horizontal tail (or aft plane), the aircraft is said to have longitudinal dihedral.

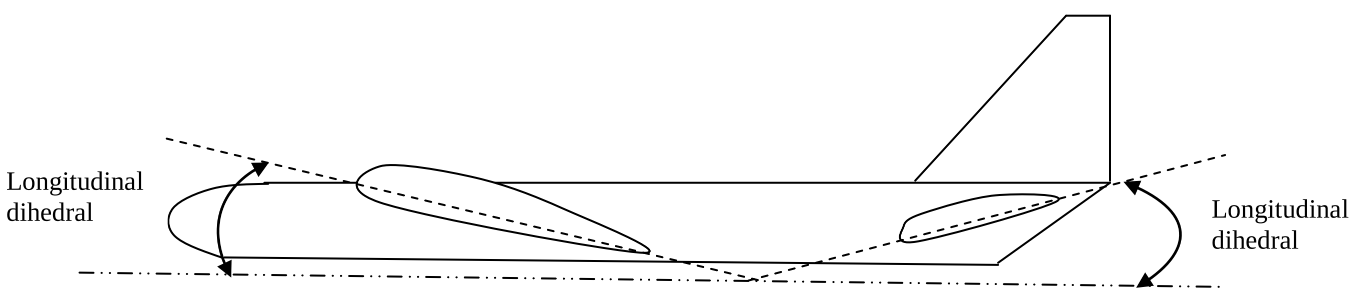


Figure 6.20. Longitudinal dihedral (angle is exaggerated)

These four above mentioned definitions are very similar, but it seems that the last one (see figure 6.20) is technically more accurate. Hence, in determining the horizontal tail setting angle, make sure that the aircraft has longitudinal dihedral. So this requirement is as follows:

$$\begin{aligned} &\alpha_{eff_w} > \alpha_{eff_t} \quad (Conventional \quad configuration) \\ &\alpha_{eff_c} > \alpha_{eff_w} \quad (Canard \quad configuration) \end{aligned} \tag{6.52}$$

The difference between tail setting angle and the effective tail angle of attack needs to be clarified. Due to the presence of the downwash at the horizontal tail location, the tail effective angle of attack is defined as follows:

$$\alpha_h = \alpha_f + i_h - \varepsilon \tag{6.53}$$

where α_f is the fuselage angle of attack and ε is the downwash at the tail (see figure 6.21).

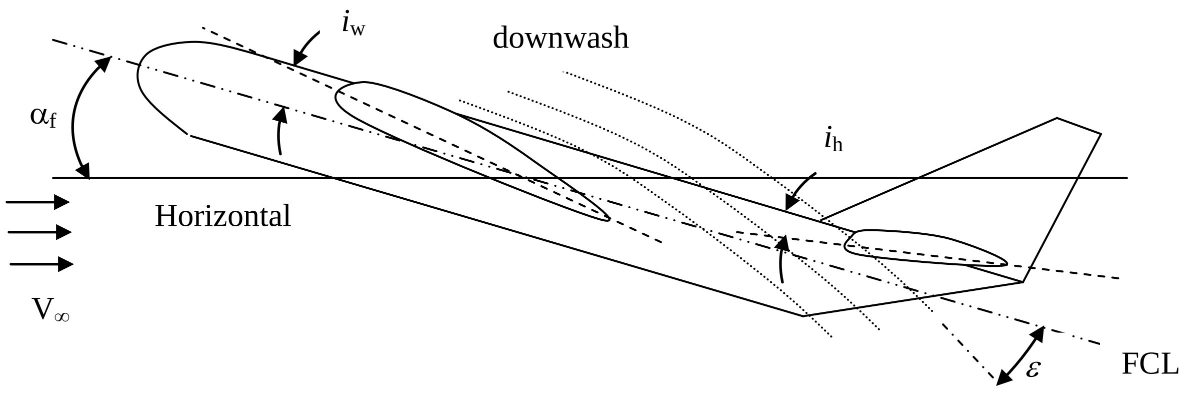


Figure 6.21. Horizontal tail effective angle of attack (downwash is exaggerated)

The fuselage angle of attack is defined as the angle between the fuselage center line and the aircraft flight path (V_∞). The downwash is the effect of the wing trailing vortices on the flow field after passing through the wing airfoil section. Each trailing vortex causes a downflow at and behind the wing and an upflow outboard of the wing. The downwash is constant along the span of a wing with elliptical lift distribution. The downwash is a function of wing angle of attack (α_w) and is determined [2] as follows:

$$\varepsilon = \varepsilon_o + \frac{\partial \varepsilon}{\partial \alpha}\alpha_w \tag{6.54}$$

where ε_o (downwash angle at zero angle of attack) and $d\varepsilon/d\alpha$ (downwash slope) are found as:

$$\varepsilon_o = \frac{2C_{L_w}}{\pi \cdot AR} \tag{6.55}$$

$$\frac{\partial \varepsilon}{\partial \alpha} = \frac{2C_{L_{\alpha_w}}}{\pi \cdot AR} \tag{6.56}$$

The wing lift curve slope ($C_{L_{\alpha_w}}$) is in 1/rad and ε is in rad. The parameter C_{Lw} is the wing lift coefficient. The typical value for ε_o is about 1 degree and $d\varepsilon/d\alpha$ is about 0.3 rad/rad. The ideal value for the horizontal tail setting angle (i_h) is zero; however, it is usually a few degrees close to zero (+ or -). The exact value for i_h is obtained in the calculation process as described in this section.

An intermediate horizontal tail parameter that must be determined is its lift curve slope ($C_{L_{\alpha_h}}$). Since the horizontal tail is a lifting surface; similar to the wing; the horizontal tail lift curve slope (3D) is determined [9] as follows:

$$C_{L_{\alpha_h}} = \frac{dC_{L_h}}{d\alpha_h} = \frac{C_{l_{\alpha_h}}}{1 + \dfrac{C_{l_{\alpha_h}}}{\pi \cdot AR_h}} \tag{6.57}$$

where $C_{l_{\alpha_h}}$ denotes the horizontal tail airfoil section lift curve slope (2D).

6.7.5. Aspect Ratio

The definition, the benefits and the parameters affecting the aspect ratio was explained in Section 5.6 in Chapter 5, so they are not repeated here. The tail aspect ratio has influences on the aircraft lateral stability and control, aircraft performance, tail aerodynamic efficiency, and aircraft center of gravity. Most of the tail aspect ratio benefits are very similar to those of the wing benefits, but in a smaller scale. The tail designer is encouraged to consult with section 5.6 for more information. Similar to the wing, tail aspect ratio is defined as the ratio between tail span to the tail mean aerodynamic chord.

$$AR_h = \frac{b_h}{\overline{C}_h} \tag{6.58}$$

The tail aspect ratio (AR_h) tends to have a direct effect on the tail lift curve slope. As the tail aspect ratio is increased, the tail lift curve slop is increased. There are several similarities between wing and horizontal tail in terms of aspect ratio, but in a smaller scale. The differences are as follows:

1. The elliptical lift distribution is not required for the tail.

2. A lower aspect ratio is desirable for tail, compared with that of the wing. The reason is that the deflection of the elevator creates a large bending moment at the tail root. Hence, the lower the aspect ratio results in a smaller bending moment.

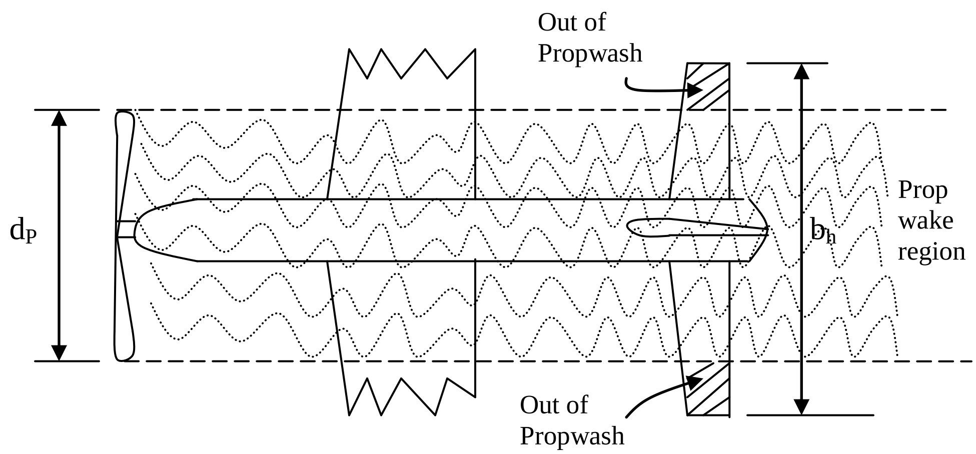


Figure 6.22. The tail span and propwash

3. In a single engine prop-driven aircraft, it is recommended to have an aspect ratio such that the tails span (b_h) is longer than the propeller diameter (d_P) (see figure 6.22). This provision insures that the tail flow field is fresh and clean of wake and out of propwash area. Therefore, the efficiency of the tail (η_h) will be increased.

Based on the above reasoning, an initial value for the tail aspect ratio may be determined as follows:

$$AR_h = \frac{2}{3} AR_w \tag{6.59}$$

A typical value for the horizontal tail aspect ratio is about 3 to 5. Table 6.5 illustrates the horizontal tail aspect ratio for several aircraft. The final value for tail aspect ratio will be determined based on the aircraft stability and control, cost, and performance analysis evaluations after the other aircraft components have been designed.

6.7.6. Taper Ratio

The definition, the benefits and the parameters affecting the taper ratio was explained in Section 5.7 in Chapter 5, so they are not repeated here. The tail taper ratio has influences on the aircraft lateral stability and control, aircraft performance, tail aerodynamic efficiency, and aircraft weight and center of gravity. Most of the tail taper ratio benefits are very similar to those of the wing benefits, but in a smaller scale. The tail designer is encouraged to consult with section 5.7 for more information. Similar to the wing, tail taper ratio (λ_h) is defined as the ratio between the tail tip chord to the tail root chord.

$$\lambda_h = \frac{C_{h_{tip}}}{C_{h_{root}}} \tag{6.60}$$

Thus the value is between zero and one. The major difference with wing taper ratio is that the elliptical lift distribution is not a requirement for tail. Thus the main motivation behind the value for the tail taper ratio is to lower the tail weight.

For this reason, the tail taper ratio is typically smaller than the wing taper ratio. The tail taper ratio is typically between 0.7 and 1 for GA aircraft and between 0.4 and 0.7 for transport aircraft. For instance, transport aircraft Boeing B-727 and Boeing B-737 (Figure 6.12) has a tail taper ratio of 0.4 and Airbus A-300 has a tail taper ratio of 0.5. Table 6.5 shows the horizontal tail taper ratio for several aircraft. The final value for tail taper ratio will be determined based on the aircraft stability and control, cost, and performance analysis evaluations after the other aircraft components have been designed.

6.7.7. Sweep Angle

The definition, the benefits and the parameters affecting the sweep angle was explained in Section 5.9 in Chapter 5, so they are not repeated here. Sweep angle is normally measured either relative to the leading edge or relative to the quarter chord line. Similar to the wing, tail leading edge sweep angle (Λ_{h_LE}) is defined as the angle between the tail leading edge and the y-axis in the x-y plane. The horizontal tail sweep angle has influences on the aircraft longitudinal and lateral stability and control, aircraft performance, tail aerodynamic efficiency, and aircraft center of gravity. Most of the tail sweep angle effects are very similar to those of the wing effects, but in a smaller scale. The tail designer is encouraged to consult with Section 5.9 for more information. The value of the horizontal tail sweep angle is often the same as wing sweep angle.

Table 6.5 shows the horizontal tail sweep angle for several aircraft. As an initial selection in preliminary design phase, select the value of the tail sweep angle to be the same as the wing sweep angle. The final value for tail sweep angle will be determined based on the aircraft stability and control, cost, and performance analysis evaluations after the other aircraft components have been designed.

6.7.8. Dihedral Angle

The definition, the benefits and the parameters affecting the dihedral angle was explained in Section 5.11 in Chapter 5, so they are not repeated here. Similar to the wing, tail dihedral angle (Γ_h) is defined as the angle between each tail half section and the y-axis in the y-z plane. The horizontal tail dihedral angle has contribution to the aircraft lateral stability and control, aircraft performance, and the tail aerodynamic efficiency. Most of the tail dihedral angle contributions are very similar to those of the wing effects, but in a smaller scale. The tail designer is encouraged to consult with Section 5.11 for more information.

The value of the horizontal tail dihedral angle is often the same as wing sweep angle. In some cases, the tail dihedral angle is totally different than the wing dihedral angle. There are several reasons for this difference including a need for the aircraft lateral stability adjustment (e.g. few transport aircraft such as tail dihedral of -3 degrees for Boeing 727); a need for lateral control adjustment (e.g. fighters such as McDonnell Douglas F-4 Phantom); and a need for a reduction in aircraft height and operational requirements (e.g. unmanned aircraft Predator). Table 6.5 shows the tail dihedral angle for several aircraft. In some aircraft instances, the manufacturing limits and considerations force the designer not to employ any dihedral for the wing. So the need for lateral stability requires a large dihedral for the tail. As an initial selection in preliminary design phase, select the value of the tail dihedral angle to be the same as the wing dihedral angle. The final value for tail dihedral angle will be determined based on the aircraft stability and control, and performance analysis evaluations after the other aircraft components have been designed.

6.7.9. Tail Vertical Location

In an aircraft with aft tail configuration, the height of the horizontal tail relative to the wing chord line must be decided. In a conventional aircraft, the horizontal tail has two options for the installation: 1. At the fuselage aft section, 2. At the vertical tail. Beside the structural considerations and complexities, the horizontal tail efficiency and its contribution to aircraft longitudinal and lateral stability must be analyzed. Unlike wing vertical location, there are no locations for tail such as low tail, mid tail or high tail. However, the low tail implies a conventional tail, the high tail implies a T-tail and the mid tail implies a cruciform tail.

A complete aircraft computational fluid dynamic model allows the designer to find the best location in order to increase the effectiveness of the tail. There are few components that are sources of interference with the tail effectiveness. They include wing, fuselage, and engine.

The wing influences the horizontal tail via downwash, wake and tailing vortices. In general, wing downwash decreases the tail effective angle of attack. Moreover, the wing wake degrades the tail efficiency, reduces the tail efficiency (h_t), and decreases the tail dynamic pressure. Most important considerations about the location of the horizontal tail relative to the wing are the prevention of deep stall. The horizontal tail location must not be in the wing wake region when wing stall happens. As the figure 6.23 illustrates, there are three major regions for tail installation behind the wing: 1. out of wake region and downwash, 2. inside wake region but out of wing downwash, 3. out of wake region but affected by downwash. In terms of deep stall avoidance criterion, the region 1 is the best and safest. The region 3 is safe from deep stall and pitch up, but tail is not efficient. The region 2 is not safe and not recommended for the horizontal tail installation.

The decision about the vertical height of the horizontal tail must be made after a thorough analysis, since a variety of parameters including wing airfoil, tail airfoil, wing-fuselage aerodynamic pitching moment, and tail arm plus manufacturing considerations are contributing. The following experimental equations are recommended for the initial approximation of the horizontal tail vertical height:

$$h_t > l \cdot \tan(\alpha_s - i_w + 3) \tag{6.61}$$

$$h_t < l \cdot \tan(\alpha_s - i_w - 3) \tag{6.62}$$

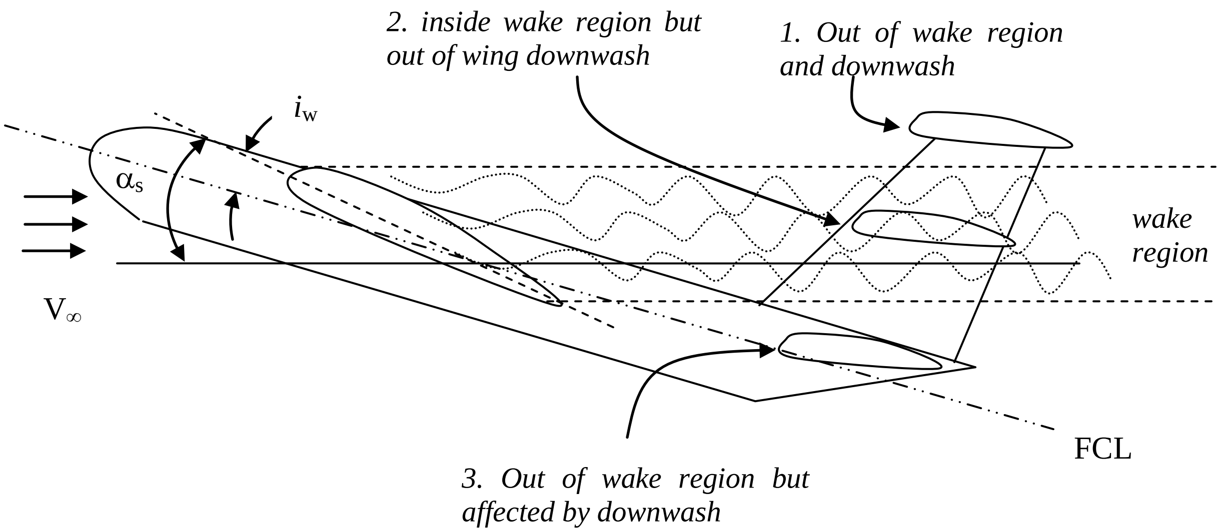


Figure 6.23. An aircraft with three tail installation locations when wing stalls

where h_t is the vertical height of the horizontal tail relative to the wing aerodynamic center, l is the horizontal tail moment arm, α_s is the wing stall angle (in degrees), and i_w denote the wing incidence (in degrees).

The fuselage interferes with the tail through fuselage wake and sidewash. The reader is referred to aerodynamic text for the details. In a multi-engine jet aircraft, the engine hot and fast speed gas has both positive and negative effects. The high speed gas increase the tail dynamic pressure, while the hot gas creates a fatigue problem for tail structure. If the tail is made of composite materials, make sure that the tail is out of engine exhaust area. Hence, the horizontal tail location is the output of a compromise process to satisfy all design requirements.

6.7.10. Other Tail Geometries

Other horizontal tail geometries include tail span (b_h), tail tip chord ($C_{h_{tip}}$), tail root chord ($C_{h_{root}}$), and tail mean aerodynamic chord ($\overline{C}_h$ or MAC_h). These four tail parameters are sketched in Figure 6.24 that shows the top view of an aircraft aft section. These unknowns are determined by solving the following four equations simultaneously:

$$AR_h = \frac{b_h}{\overline{C}_h} \tag{6.63}$$

$$\lambda_h = \frac{C_{h_{tip}}}{C_{h_{root}}} \tag{6.64}$$

$$\overline{C}_h = \frac{2}{3} C_{h_{root}} \left(\frac{1 + \lambda_h + \lambda_h^2}{1 + \lambda_h} \right) \tag{6.65}$$

$$S_h = b_h \cdot \overline{C}_h \tag{6.66}$$

The first two equations have been introduced previously in this section, but the last two equations are reproduced from wing geometry governing equations (see Chapter 5). The required data to solve these equations are the tail planform area, tail aspect ratio, and tail taper ratio.

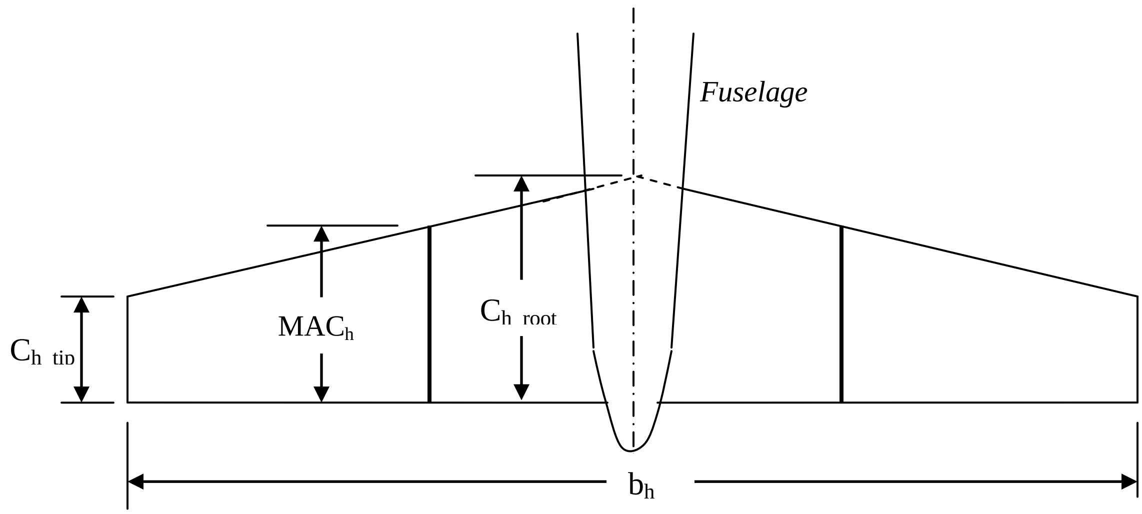


Figure 6.24. Horizontal tail geometry

6.7.11. Control Provision

One of the secondary functions of the horizontal tail is the aircraft longitudinal control. The horizontal tail must generate a variety of tail lift forces in various flight conditions to longitudinally trim the aircraft and create the new trim conditions. For this purpose, a fixed and an adjustable horizontal tail have movable sections; which in a conventional aircraft; is called elevator. Therefore, in designing the horizontal tail, one must consider some provisions for future control applications. The provisions include insuring the sufficient space for elevator's area, span, and chord as well as elevator deflection angle to allow for an effective longitudinal control. The design of the aircraft control surfaces including the elevator design is examined in Chapter 12.

6.7.12. Final Check

When all horizontal tail parameters have been determined, two design requirements must be examined: 1. aircraft longitudinal trim, 2. aircraft static and dynamic longitudinal stability. In the analysis of the longitudinal trim, the tail lift coefficient needs to be calculated. The generated horizontal tail lift coefficient should be equal to the required cruise tail lift coefficient. There are several aerodynamic software packages and tools to calculate the horizontal tail lift coefficient. In the early stage of design, it is recommended to employ the lifting line theory as described in Chapter 5. When whole aircraft is designed, modern CFD software is utilized to determine aerodynamic feature of the aircraft including horizontal tail. If the longitudinal trim requirements are not satisfied, horizontal tail parameters such as tail incidence must be adjusted.

The static longitudinal stability is examined through the sign of the longitudinal stability derivative Cm_α or the location of the aircraft neutral point. For an aircraft with a fixed aft tail, the aircraft static longitudinal stability derivative is determined [6] as:

$$C_{m_\alpha} = C_{L_{\alpha_{wf}}}\left(h - h_o\right) - C_{L_{\alpha_h}}\eta_h \frac{S_h}{S}\left(\frac{l}{\overline{C}} - h\right)\left(1 - \frac{d\varepsilon}{d\alpha}\right) \tag{6.67}$$

When the derivative Cm_α is negative or when the neutral point is behind the aircraft cg, the aircraft is said to be statically longitudinally stable.

The dynamic longitudinal stability analysis is performed after all aircraft components are designed and the roots (λ) of the longitudinal characteristic equation are calculated. A general form of the aircraft longitudinal characteristic equation looks like the following:

$$A_1\lambda^4 + B_1\lambda^3 + C_1\lambda^2 + D_1\lambda + E_1 = 0 \tag{6.68}$$

where coefficients A_1, B_1, C_1, D_1, and E_1 are functions of the several stability derivatives such as Cm_α and Cm_q. An aircraft is dynamically longitudinally stable, if the real parts of all roots of longitudinal characteristic equation are negative. Another way to analyze dynamic longitudinal stability is to make sure that longitudinal modes (i.e. short period and long period (Phugoid)) are damped.

The reader is encouraged to consult with [1] to see to see how to derive the aircraft longitudinal characteristic equation. The longitudinal stability derivatives cannot be determined unless all aircraft components including wing and fuselage have been designed. This is why resort to a simplifying criterion that could be a base for the horizontal tail preliminary design. When the horizontal tail volume coefficient ($\overline{V}_H$) is in ballpark number (see Table 6.5), we are 90 percent confident that the longitudinal stability requirements have been satisfied. When other aircraft components such as fuselage and wing have been designed, the horizontal tail design will be revised and optimized in the longitudinal stability analysis process.

6.8. Vertical Tail Design

6.8.1. Vertical Tail Design Requirements

The third lifting surface in a conventional aircraft is the vertical tail; which is sometimes referred to as vertical stabilizer or fin. The vertical tail tends to have two primary functions: 1. directional stability, 2. directional trim. Moreover, the vertical tail is a major contributor in maintaining directional control; which is the primary function of the rudder. These three design requirements are described briefly in this section:

1. The primary function of the vertical tail is to maintain the aircraft directional stability. The static and dynamic directional stability requirements were discussed in Section 6.3. In summary, the stability derivatives Cn_β must be positive (to satisfy the static directional stability requirements), but the stability derivatives Cn_r must be negative (to satisfy the dynamic directional stability requirements). Two major contributors to the value of these stability derivatives are vertical tail area (S_V) and vertical tail moment arm (l_V). If vertical tail

area is large enough and vertical tail moment arm is long enough, the directional stability requirements could be easily satisfied. The directional stability analysis is performed after all aircraft components are designed and the roots (λ) of the lateral-directional characteristic equation are calculated. A general form of the aircraft lateral-directional characteristic equation looks like the following:

$$A_2\lambda^4 + B_2\lambda^3 + C_2\lambda^2 + D_2\lambda + E_2 = 0 \tag{6.69}$$

where coefficients A_2, B_2, C_2, D_2, and E_2 are functions of the several stability derivatives such as Cn_{β} and Cn_r.

An aircraft is dynamically directionally stable, if the real parts of all roots of lateral-directional characteristic equation are negative. Another way to analyze dynamic directional stability is to make sure that directional modes (i.e. dutch roll, and spiral) are damped.

The reader is encouraged to consult with [1] to see how to derive the aircraft lateral-directional characteristic equation. The directional stability derivatives cannot be determined unless all aircraft components including wing and fuselage have been designed. Hence, we have to resort to some other simplifying criterion that could be a base for the vertical tail preliminary design. Similar to horizontal tail volume coefficient, a new parameter that is referred to as vertical tail volume coefficient (V_V) is defined. If the value of this parameter is in ballpark number, we are 90 percent sure that the directional stability requirements have been satisfied. When other aircraft components have been designed, the vertical tail design will be revised and optimized in the directional stability analysis process. The vertical tail volume coefficient will be introduced in Section 6.8.2.

2. The second function of the vertical tail is to maintain the aircraft directional trim. As discussed in Section 6.3, the summation of all forces along the y-axis and the summation of all moments about z-axis must be zero.

$$\sum F_y = 0 \tag{6.5}$$

$$\sum N_{cg} = 0 \tag{6.6}$$

An aircraft is normally manufactured symmetrical about x-z plane, so the directional trim is naturally maintained. Although this is an ideal case and is considered in the production of components such as right and left wing sections, but in several cases, there is a slight asymmetricity in the aircraft x-y plane. One source for the asymmetricity could be a difference between manufacturing jigs and fixtures of right and left sections (wing and tail). Another reason for directional asymmetricity lies in the internal components inside fuselage such as fuel system, electrical wiring, and even load and cargo inside load compartment.

However, in a single engine prop-driven aircraft, the aircraft directional trim is disturbed by the rotation of the engine propeller. In a multi-engine prop-driven aircraft, with odd number of engines, a similar problem exists. Hence, the vertical tail is responsible for maintaining the directional trim by providing an opposing yawing moment about z-axis. One of the critical parameters influencing the directional trim in such aircraft is the vertical tail incidence angle relative to the x-z plane.

Another directional trim case is in multi-engine aircraft, where one engine in inoperative. In such situation, the operative engines create a disturbing yawing moment and the only way to balance this asymmetric moment is the counteracting yawing moment generated by the vertical tail. A control surface (e.g. rudder) must be deflected to directionally trim the aircraft.

Although the vertical tail is contributing to the aircraft lateral stability and control, but this item is not considered as a base for the design of the vertical tail. However, in the analysis of the vertical tail performance, the lateral stability must be studied. This is to make sure that the vertical tail is improving the aircraft lateral stability and not having a negative impact. Recall that the aircraft lateral stability is primarily a function of the wing parameters. The static and dynamic directional trim requirements were discussed in Section 6.2.

3. The third aircraft design requirement in which the vertical tail is a major contributor is the directional control. Maneuvering operations such as turning flight and spin recovery are successfully performed by using a movable section of the vertical tail which is called rudder. The design of the rudder is examined in Chapter 12, but the spin recovery requirements will be discussed in Section 6.8.3.

6.8.2. Vertical Tail Parameters

Basically, the vertical tail parameters must be initially determined such that the directional stability requirements are satisfied. In the second and third stage of the vertical tail design process, the directional trim requirements and directional control requirements will be examined.

In the design of the vertical tail, the following parameters must be determined:

1. Vertical tail location
2. Planform area (S_v)
3. Tail arm (l_{vt})
4. Airfoil section
5. Aspect ratio (AR_v)
6. Taper ratio (λ_v)
7. Tip chord (C_{t_v})
8. Root chord (C_{r_v})
9. Mean Aerodynamic Chord (MAC_v or C_v)
10. Span (b_v)
11. Sweep angle (Λ_v)
12. Dihedral angle (Γ_v)
13. Incidence (i_v)

Several of these vertical tail parameters are illustrated in figure 6.25. The vertical tail is a lifting surface, whose aerodynamic force of lift is generated in the direction of y-axis. In maintaining the directional stability, control and trim, an aerodynamic force along y-axis needs to be created by the vertical tail (i.e. vertical tail lift; L_V).

$$L_V = \frac{1}{2}\rho V^2 S_V C_{L_v} \tag{6.70}$$

where S_V is the vertical tail area, and the C_{LV} is the vertical tail lift coefficient. The vertical tail lift is generating a yawing moment about z-axis:

$$N_{cg} = L_V l_V \tag{6.71}$$

This moment must be large enough to maintain directional trim and must have a positive contribution to the directional stability. As explained in Section 6.8.1, a preliminary evaluation of the directional stability is applied through a parameter called vertical tail volume coefficient ($\overline{V}_V$):

$$\overline{V}_V = \frac{l_V S_V}{bS} \tag{6.72}$$

where l_v is the distance between vertical tail aerodynamic center (ac_v) and the wing-fuselage aerodynamic center (see figure 6.25), S_V is the vertical tail planform area, b is the wing span, and S denotes the wing reference area. The vertical tail aerodynamic center is located at the quarter chord of the vertical tail mean aerodynamic chord.

The vertical tail volume coefficient is a non-dimensional parameter which is directly functions of two significant vertical tail parameters: vertical tail area (S_V) and vertical tail moment arm (l_v). Two parameters of l_v and l_{vt} are very close, such that one can be determined from another one. The vertical tail volume coefficient is an indirect representative for the aircraft directional stability. The typical value for the vertical tail volume coefficient is between 0.02 and 0.12. Table 6.6 illustrates the vertical tail parameters including the vertical tail volume coefficient for several aircraft. Remember that the vertical tail planform area includes both the fixed section and the movable section (i.e. rudder).

Since the definitions and features of lifting surface basic parameters such as aspect ratio, taper ratio, and airfoil section have been presented in Chapter 5 and also in horizontal tail design section (Section 6.7), they are introduced briefly here.

1. Vertical tail location

In order to maintain the directional stability, the only location for the vertical tail is aft of the aircraft center of gravity. Three possible candidates are 1. aft of fuselage, 2. wingtips, and 3. boom(s). If a single aft horizontal tail has been selected, the only place for the vertical tail is on the top of the aft fuselage. The vertical tail cannot be placed in front of the fuselage (i.e. forward of the aircraft cg), since it makes the aircraft directionally unstable. Other two options, namely wingtips and boom, are appropriate for some special purposes that have been described earlier in Section 6.4.

2. Vertical tail moment arm (l_{vt})

The vertical tail moment arm (see figure 6.25) must be long enough to satisfy directional stability, control, and trim requirements. In a spinnable aircraft, the vertical tail must also satisfy the spin recovery requirements. Increasing the vertical tail moment arm increases the values of the derivatives Cn_β and Cn_r and thus makes the aircraft directionally more stable. The major contributor to the static directional stability derivative (Cn_β) is the vertical tail [1]:

$$C_{n_\beta} \approx C_{n_{\beta V}} = K_f C_{L_{\alpha V}} \left(1 - \frac{d\sigma}{d\beta}\right) \eta_V \frac{l_{Vt} S_V}{bS} \tag{6.73}$$

where $C_{L_{\alpha V}}$ denotes the vertical tail lift curve slope, $\frac{d\sigma}{d\beta}$ is the vertical tail sidewash gradient, and η_V is the dynamic pressure ratio at vertical tail. The parameter K_{f1} represents the contribution of fuselage to aircraft Cn_β and depends strongly on the shape of the fuselage and its projected side area. The fuselage contribution to directional static stability tends to be strongly negative. The typical value of K_{f1} for a conventional aircraft is about 0.65 to 0.85. The value of C_{n_β} for a statically directionally stable aircraft is positive. A higher value for Cn_β implies a more directionally statically stable aircraft. The parameter l_{vt} in equation 6.65 is in the numerator which implies the longer moment arm is desirable.

In addition, an increase in the vertical tail moment arm improves the directional and lateral control. In the early stage of the vertical tail design; where other aircraft components have not been designed; the vertical tail moment arm is selected to be equal to the horizontal tail moment arm (l). This assumption means that the vertical tail is located at the same distance to the wing as the horizontal tail. The assumption will be modified in the later design stage when other aircraft components are designed and the aircraft directional and lateral stability, control and trim are analyzed.

Another phenomenon that influences the vertical tail moment arm is spin. When an aircraft is spinnable, the aircraft is required to be able to recover from spin safely. Spin is a dangerous flight if the aircraft is not designed to recover safely from it. Some aircraft, however, are not spinnable by design. Most transport aircraft are not spinnable (i.e. spin resistant), while most fighters and maneuverable aircraft are spinnable.

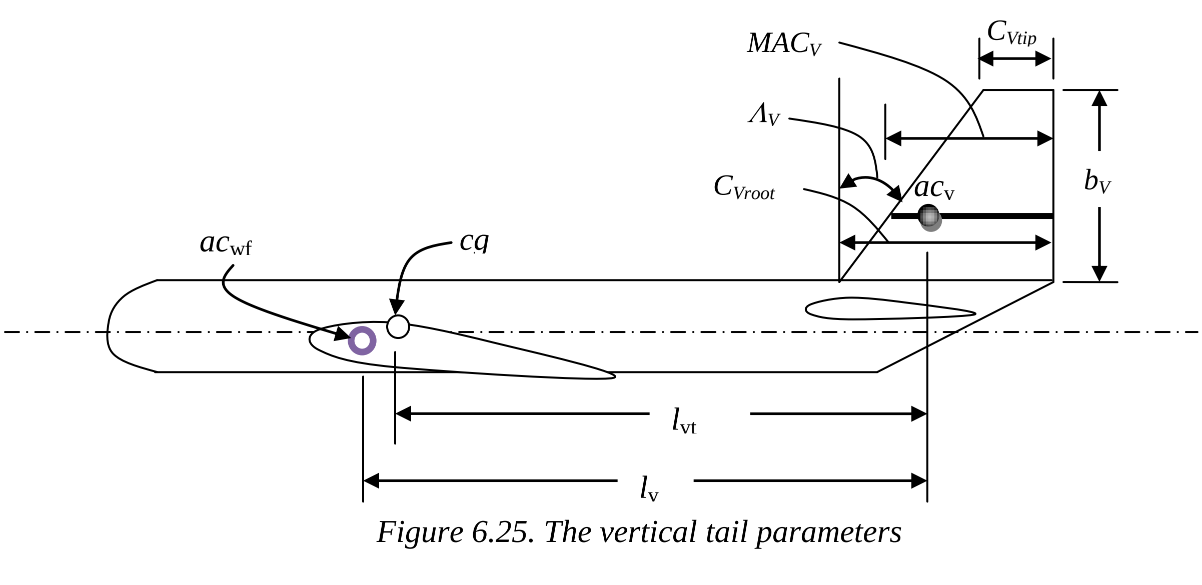


Figure 6.25. The vertical tail parameters

A spin is an aggravated stall resulting in autorotation about the spin axis wherein the aircraft follows a screw path. Spins is characterized by high angle of attack, low airspeed, high sideslip angle, and high rate of descent. In a spin, both wings are in a stalled condition; however one wing will be in a deeper stall than the other. This causes the aircraft to autorotate due to the non-symmetric lift and drag. Spins can be entered unintentionally or intentionally. In either case, a specific and often counterintuitive set of actions are needed to influence recovery. If the aircraft exceeds published limitations regarding spins, or is loaded improperly, or if the pilot uses incorrect technique to recover, the spin may lead to a crash.

The following is reproduced from Section 221 of PAR 23 of Federal Aviation Regulations [4] which concerns about spinning of GA aircraft:

(a) Normal category airplanes. A single-engine, normal category airplane must be able to recover from a one-turn spin or a three-second spin, whichever takes longer, in not more than one additional turn after initiation of the first control action for recovery, or demonstrate compliance with the optional spin resistant requirements of this section.

(b) Utility category airplanes. A utility category airplane must meet the requirements of paragraph (a) of this section. In addition, the requirements of paragraph (c) of this section and §23.807(b)(7) must be met if approval for spinning is requested.

(c) Acrobatic category airplanes. An acrobatic category airplane must meet the spin requirements of paragraph (a) of this section and §23.807(b)(6). In addition, the following requirements must be met in each configuration for which approval for spinning is requested:

(1) The airplane must recover from any point in a spin up to and including six turns, or any greater number of turns for which certification is requested, in not more than one and one-half additional turns after initiation of the first control action for recovery. However, beyond three turns, the spin may be discontinued if spiral characteristics appear.

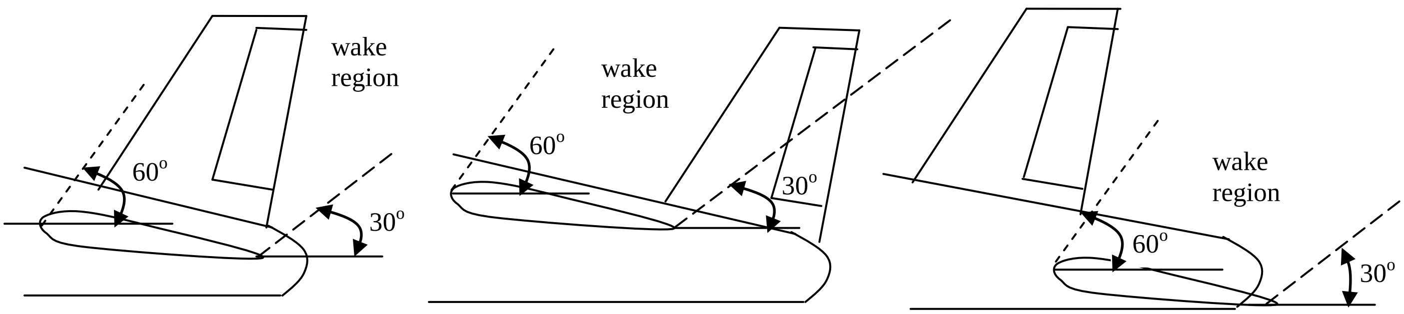
(2) The applicable airspeed limits and limit maneuvering load factors must not be exceeded. For flaps-extended configurations for which approval is requested, the flaps must not be retracted during the recovery.

(3) It must be impossible to obtain unrecoverable spins with any use of the flight or engine power controls either at the entry into or during the spin.

(4) There must be no characteristics during the spin (such as excessive rates of rotation or extreme oscillatory motion) that might prevent a successful recovery due to disorientation or incapacitation of the pilot.

When a spin occurs, all that is mainly required is a sufficient yaw rate while an aircraft is stalled. Hence the vertical tail must be able to generate the yawing moment to stop autorotation. Thus, the vertical tail plays a vital role in spin recovery. The vertical tail may have a long moment arm, but there is a situation that could negatively influence the effectiveness of the vertical tail. If the vertical tail is in the horizontal tail wake region, it will lose its effectiveness. Therefore, the vertical tail moment arm needs to be determined such that provide a wake free region for the vertical tail.

An experimental rule for the vertical tail effectiveness to achieve a recoverable spin is as follows: At least 50 percent of the vertical tail planform area must be out of the horizontal tail wake region to be effective in the case of a spin. The horizontal tail wake region is considered to lie between two lines. The first line is drawn at the horizontal tail trailing edge by the orientation of 30 degrees. The second line is drawn at the horizontal tail leading edge by the orientation of 60 degrees.



1. VT is in the wake region 2. Part of the VT is in the wake region 3. VT is out of the wake region

Figure 6.26. The vertical tail effectiveness and the wake region of the horizontal tail

So, even if the vertical tail moment arm is theoretically calculated to be sufficient, but if the vertical tail is graphically located to be inside the horizontal tail wake region, the moment arm needs to be adjusted. It is clear that if the moment arm needs to be decreased, the vertical tail area must be increased. However, if the adjustment of the vertical tail arm leads to a larger arm, the vertical tail area could be decreased. Another technique to move the vertical tail out of the horizontal tail wake region is to employ a dorsal fin. A graphical method is illustrated in figure 6.26. Figure 6.26-1 shows a vertical tail that is completely inside the wake region. This configuration does not satisfy spin recovery requirements. Figure 6.26-2 demonstrates a vertical tail that is completely out of the wake region. This configuration does satisfy the spin recovery requirements. Figure 6.26-3 depicts a vertical tail that is partly inside the wake region. Although the moment arm of the vertical tail (l_v) in figure 6.26-3 is shorter than that of the two other vertical tails, but the advantage is that is wake free.

3. Planform area (S_v)

The parameter S_v in equation 6.65 is in the numerator which implies the larger vertical tail area is desirable. The vertical tail area must be large enough to satisfy lateral-directional stability, control, and trim requirements. Increasing the vertical tail area increases the values of the derivatives Cn_β and Cn_r and thus makes the aircraft lateral-directionally more stable. In addition, an increase in the vertical tail area improves the directional and lateral control ($Cn_{\delta R}$, $C_{l\delta R}$). If the vertical tail area is too small, the lateral-directional stability requirements will not be satisfied. On the other hand, when the vertical tail area is too large, the aircraft will be lateral-directionally too stable, but the directional control requirements are not satisfied. Thus, the middle value is very hard to determine. For this reason, the vertical tail design is utilizing a backward design technique. It means that we select a combination of vertical tail area and vertical tail moment arm in a ballpark area through a parameter called vertical tail volume coefficient. Another criterion for the vertical tail area is that it must be small such that to minimize the manufacturing cost and the aircraft weight.

It is interesting to note that a typical value for the ratio between vertical tail area and the wing area for a conventional GA aircraft is about 0.1 to 0.15. The vertical tail planform area is preliminary determined based on the selection of the vertical tail volume coefficient ($\overline{V}_V$). The typical value for the vertical tail volume coefficient for several aircraft type is introduced in Table 6.4. Hence the vertical tail area is determined as:

$$S_V = \frac{b \cdot S \cdot \overline{V}_V}{l_V} \tag{6.74}$$

where it is initially assumed that the parameter l_v is equal to the vertical tail moment arm (l_{vt}). This area will be adjusted in the later design stage after other aircraft components are designed and the aircraft directional and lateral stability, control and trim are analyzed. The design of the vertical is one of the difficult tasks for aircraft designers, since theoretical and experimental results may not match concerning the features of the vertical tail. It is often the case for several aircraft that the vertical tail area is found; in flight test; insufficient to satisfy lateral-directional stability requirements.

If the aircraft is in the manufacturing stage, and the initial vertical tail design may not be changed, one solution to increase the vertical tail area is to employ the dorsal fin. A dorsal fin[4] (see figures 6.27-1 and 6.27-2) is generally a flat plate (i.e. no airfoil section) installed in front of the original vertical tail with a greater sweep angle. The other benefit of a dorsal fin is to reduce the minimum control speed (V_{mc}) during take-off operation (as employed in Piper Arapaho PA-40). In addition, it provided a hidden antenna feature that allows the com antennas to be located under the fin for further drag reduction.

Another approach to solve the small vertical tail area problem is to employ the ventral fin. A ventral fin[5] (see figure 6.27-3) is simply a flat plate (i.e. no airfoil section) installed under the aft fuselage (almost in the same longitudinal location of the vertical tail). It is also possible and useful to consider the airfoil section for dorsal and ventral fins to improve their aerodynamic characteristics. These two techniques improve the lateral-directional stability of an aircraft, while they do not touch the original vertical tail geometry. Table 6.6 shows the value for the ratio between vertical tail area and the wing area for several aircraft. Figure 6.27 illustrates the dorsal and ventral fin of Beech 200 Super King Air; the ventral fin of Gates Learjet 35A; and the ventral fin of General Atomics Predator.

The wing and the horizontal tail have two right and left sections. But, unlike the wing and horizontal tail, the vertical tail has normally one section. Thus, the vertical tail span (b_v) is the distance between the vertical tail tip chord and root chord (see figure 6.25). For this reason, the vertical tail aerodynamic center in a conventional aircraft is normally above the fuselage center line (and most of the time above the aircraft center of gravity).

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[4] This term has been borrowed from fish anatomy. A "*dorsal fin*" is a polyphyletic fin located on the backs of some fish, whales, and dolphins.

[5] This term has been borrowed from fish anatomy.

4. Airfoil section

The vertical tail airfoil section is responsible for the generation of the vertical tail lift coefficient (C_{LV}). The airfoil must generate the required lift coefficient with a minimum drag coefficient. Recall that a nonsymmetrical airfoil section creates an aerodynamic pitching moment. One of the basic aircraft design requirement is the symmetricity about the x-z plane. Therefore, to insure the symmetricity of the aircraft about x-z plane, the vertical airfoil section must be symmetric. Moreover, if the engines, wing, horizontal tail and fuselage are designed to be symmetric about x-z plane, the vertical tail is not required to produce any lift to maintain directional trim in a normal flight condition.

Recall from Chapter 5 that the indication of a symmetric airfoil is that the second digit in a 4-digit and the third digit in a 5-digit and 6-series NACA airfoil sections is zero. This denotes that the airfoil design lift coefficient and zero-lift angle of attack are both zero. NACA airfoil sections such as 0009, 0010, 0012, 63-006, 63-009, 63-012, 63-015, 63-018, 64-006, 64-012, 64A010, 65-009, 65-015, 66-012, 66-018, and 66-021 are all symmetric airfoils. In several GA aircraft, NACA airfoil sections 0009 or 0012 (with 9% or 12% maximum thickness-to-chord ratio) are employed for vertical tail. Both of this NACA airfoil sections are symmetric.

In addition, another tail requirement is that the vertical tail must be clean of compressibility effect. To satisfy this requirement, the flow Mach number at the vertical tail must be less than the flow Mach number at the wing. This objective will be realized by selecting a vertical tail airfoil section to be thinner (say about 2 percent of MAC) than the wing airfoil section. For instance, if the wing airfoil section is NACA 23015 (i.e. $(t/C)_{max}$ = 0.15 or 15%), the vertical tail airfoil section can be selected to be NACA 0009 (i.e. $(t/C)_{max}$ = 0.9 or 9%). Figure 6.18 shows the characteristics graphs of the NACA 0009 airfoil section. Table 6.5 illustrates the airfoil section for vertical tail of several aircraft.

The third desired feature for the vertical tail airfoil section is the high value for the lift curve slope ($C_{L_{\alpha V}}$), since the static directional stability derivative (Cn_{β}) is directly a function of $C_{L_{\alpha V}}$ (equation 6.72). Thus, as a general rule, a symmetric airfoil section with a high lift curve slope is desirable for the vertical tail. Recall that the theoretical value for an airfoil section is about 2π 1/rad. Table 6.6 shows airfoil section of the vertical tail for several aircraft.

5. Incidence (i_v)

The vertical tail incidence is defined as the angle between the vertical tail chord line and the aircraft x-z plane (when look at the aircraft from top view). The vertical tail is responsible for the generation of the vertical tail lift coefficient (C_{LV}). One of the basic aircraft design objective is the symmetricity about the x-z plane. Hence, if the engines, wing, horizontal tail and fuselage are designed to be symmetric about x-z plane, the vertical tail is not required to produce any lift to maintain directional trim in a normal flight condition. For this reason, the vertical tail incidence must be initially zero.

No	Aircraft	Type	m_{TO} (kg)	Airfoil	$(t/C)_{max}$ (%)	$\overline{V}_V$	S_V/S	AR_V	Λ_V (deg)
1	Wright Flyer	First aircraft in history	420	Flat plate	low	0.013	0.045	6.3	0
2	Cessna 177	GA single prop engine	1,100	NACA 0009/0006	7.5	0.14	0.107	1.41	35
3	C-130 Hercules	Large turboprop cargo	70,305	NACA 64A-015	15	0.06	0.18	1.84	18.8
4	DC-9/10	Large jet transport	41,100	DSMA	11	0.08	0.19	0.95	43.5
5	Cessna Citation I	Business jet	5,375	NACA 0012/0008	10	0.0806	0.191	1.58	33
6	Fokker F-27	Turboprop transport	19,773	modified NACA	15	0.07	0.203	1.55	33
7	Boeing 737-100	Large jet transport	50,300	-	12	0.11	0.27	1.88	35
8	Beechjet 400A	Business jet transport	7,303	-	12	0.123	0.263	1	55
9	DC-8-10	Large jet transport	141,000	DSMA-111/-112	9.85	0.05	0.122	1.91	35
10	Airbus 300B	Large jet transport	165,000	-	12.5	0.102	0.204	1.623	40
11	C-17A	Heavy jet cargo	265,352	-	9	0.08	0.195	1.36	36
12	Eurofighter 2000	Fighter	21,000	-	7	0.035	0.096	1.3	45
13	F-15 Eagle	Fighter	36,741	-	7	0.06	0.346[6]	1.3	35

Table 6.6. Vertical tail characteristics for several aircraft

1. Beech 200 Super King Air (dorsal and ventral fin) (Courtesy of Jenny Coffey) *2. Gates Learjet 35A (ventral fin) (Courtesy of Antony Osborne)* *3. General Atomics Predator (ventral fin)*

Figure 6.27. Dorsal fin and ventral fin in three aircraft

[6] The aircraft has a twin vertical tail, so the areas of both vertical tails are included in the calculation.

However, in a prop-driven aircraft with one single engine (or with odd number of prop-driven engines), the lateral trim is disturbed by the revolution of the propeller and engine shaft about x-axis. The aircraft body is going to roll as a reaction to the rotation of the propeller and its shaft (recall the third law of Newton). Although this rolling moment is not large, but the safety requirements requires the trim to be maintained and aircraft roll be avoided. To nullify this yawing moment, the vertical tail is required to generate a lift and cancels this rolling moment. One solution for this problem is to consider a few degrees of incidence for the vertical tail. The vertical tail in most single engine prop-driven aircraft have about 1-2 degrees of incidence to insure the prevention of aircraft roll in a reaction to propeller revolution. Another solution is to select a non-symmetric airfoil for the vertical tail, but this technique has several disadvantages. The exact value for the vertical tail incidence is determined by calculating the propeller rotation's rolling moment. An experimental approach would be more accurate.

6. Aspect ratio (AR_v)

The vertical tail aspect ratio is defined as the ratio between vertical tail span; b_y (see figure 6.25) and the vertical tail mean aerodynamic chord ($\overline{C}_V$).

$$AR_V = \frac{b_V}{\overline{C}_V} \tag{6.75}$$

The general characteristics of the aspect ratio are introduced in Chapter 5 (see Section 5.6), so they are not repeated here. The vertical tail aspect ratio has several other features than impact various aircraft characteristics. These must be noticed in determining the vertical tail aspect ratio[7].

1. First of all, a high aspect ratio results in a tall vertical tail that causes the aircraft overall height to be increased. Many aircraft especially large transport aircraft and fighter aircraft have parking limitations in the hangar space. Thus, an aircraft is not allowed to have an overall height beyond a pre-specified value.
2. A high tail aspect ratio weakens the aircraft lateral control, since the vertical tail mass moment of inertia about x-axis is increased.
3. A vertical tail with a high aspect ratio has a longer yawing moment arm compared with a low aspect ratio vertical tail. Hence, an aircraft with high aspect ratio has a higher directional control.
4. As the vertical aspect ratio is increased, the bending moment and bending stress at the vertical tail root are increase which causes the aft portion of the aircraft to be heavier.
5. A high aspect ratio vertical tail is prone to fatigue and flutter.
6. A high aspect ratio vertical tail is longitudinally destabilizing, since the vertical tail drag generates a nose-up pitching moment.
7. As the aspect ratio of the vertical tail is increased, the aircraft directional stability is improved, due to an increase in the yawing moment arm.
8. As the aspect ratio of the vertical tail is increased, the vertical tail induced drag is increased.
9. If the aircraft has a T-tail configuration, the horizontal tail location and efficiency are functions of vertical tail aspect ratio. Thus, if the deep stall is a major concern, the vertical

[7] Reference 10 defines the vertical tail aspect ratio as 1.55(b/C).

aspect ratio must be large enough to keep the horizontal tail out of the wing wake when the wing stalls.
10. A high aspect ratio vertical tail is aerodynamically more efficient (i.e. has a higher $(L/D)_{max}$) than a vertical tail with a low aspect ratio. The reason is the vertical tail tip effect.

The above-mentioned advantages and disadvantages for a high and low aspect ratio are general guidelines for the vertical tail designer. As a starting point, a value between 1 and 2 is recommended for the vertical tail aspect ratio. The final value will be determined in the overall aircraft directional stability analysis. Table 6.6 shows the value for aspect ratio of vertical tail for several aircraft.

7. Taper ratio (λ_v)

As with other lifting surfaces (e.g wing and horizontal tail), the vertical tail taper ratio is defined as the ratio between the vertical tail tip chord; $C_{V_{tip}}$ (see figure 6.25) to the vertical tail root chord; $C_{V_{root}}$.

$$\lambda_V = \frac{C_{V_{tip}}}{C_{V_{root}}} \tag{6.76}$$

General features of the taper ratio are introduced in Chapter 5 (see Section 5.7), so they are not repeated here. The main purposes of the taper ratio are 1: to reduce the bending stress on the vertical tail root and also 2: to allow the vertical tail to have a sweep angle. The application of taper ratio adds a complexity to the tail manufacturing process and also increases the empennage weight. As the taper ratio of the vertical tail in increased, the yawing moment arm is reduced which reduces the directional control of the aircraft. Moreover, an increase in the taper ratio of the vertical tail would reduce the lateral stability of the aircraft. A compromise between these positive and negative features determines the value for the vertical tail taper ratio.

8. Sweep angle (Λ_v)

The general features of the sweep angle are introduced in Chapter 5 (see Section 5.9), so they are not repeated here. As the sweep angle of the vertical tail in increased, the yawing moment arm is increased which improves the directional control of the aircraft. Subsequently, an increase in the vertical tail sweep angle weakens the aircraft directional stability, since the mass moment inertia about z-axis in increased. If the aircraft has a T-tail configuration, an increase in the vertical tail sweep angle increases the horizontal tail moment arm which improves the aircraft longitudinal stability and control.

Another reason for the application of the vertical tail sweep angle is to decrease the wave drag in high subsonic and supersonic flight regime. For this reason, it is suggested to initially adopt a sweep angle similar to the sweep angle of the wing. The final value for the vertical tail sweep angle will be the results of a compromise between these positive and negative features. Table 6.6 shows the value for the sweep angle of vertical tail for several aircraft.

9. Dihedral angle (Γ_v)

Due to the aircraft symmetricity requirement about x-z plane, an aircraft with one vertical tail is not allowed to have any dihedral angle. However, if the aircraft has a twin vertical tail, (such as few fighters), the dihedral angle has positive contributing to the aircraft lateral control. But it reduces the aerodynamic efficiency of the vertical tails, since two vertical tails will cancel part of their lift forces. In addition, the vertical tail dihedral angle will contribute to detectability features of the aircraft. For instance, McDonnell Douglas F-15 Eagle (Figure 9.14) twin vertical tails canted 15 deg to reduce radar cross section. The exact value for the dihedral angles of a twin vertical tail is determined in the overall aircraft lateral- directional stability analysis process.

10. **Tip chord (C_{t_v}), Root chord (C_{r_v}), Mean Aerodynamic Chord (MAC_v or C_v), and Span (b_v)**

The other vertical tail geometries include vertical tail span (b_V), vertical tail tip chord ($C_{V_{tip}}$), vertical tail root chord ($C_{V_{root}}$), and vertical tail mean aerodynamic chord ($\overline{C}_V$ or MAC_V). These unknown parameters (see figure 6.25) are determined by solving the following four equations simultaneously:

$$AR_V = \frac{b_V}{\overline{C}_V} = \frac{b_V^2}{S_V} \tag{6.77}$$

$$\lambda_V = \frac{C_{V_{tip}}}{C_{V_{root}}} \tag{6.78}$$

$$\overline{C}_V = \frac{2}{3} C_{V_{root}} \left(\frac{1 + \lambda_V + \lambda_V^{\ 2}}{1 + \lambda_V} \right) \tag{6.79}$$

$$S_V = b_V \cdot \overline{C}_V \tag{6.80}$$

The first two equations have been introduced previously in this section, but the last two equations are reproduced from wing geometry governing equations (see Chapter 5). The required data to solve these equations are the vertical tail planform area, vertical tail aspect ratio, and vertical tail taper ratio.

6.9. Practical Design Steps

The tail design flowchart was presented in section 6.1. Fundamentals of the tail primary functions and design requirements were reviewed in Sections 6.2 and 6.3. Sections 6.4 through 6.8 introduced the various tail configurations, horizontal tail parameters, vertical tail parameters and the technique to determine each parameter. The purpose of this section is to outline the practical design steps of the tail. The tail design procedure is as follows:

1. Select tail configuration (Sections 6.4 and 6.7)

Horizontal tail

2. Select horizontal tail location (aft, or forward (canard)); Section 6.5

3. Select the horizontal tail volume coefficient; $\overline{V}_H$ (Table 6.4)
4. Calculate optimum tail moment arm (l_{opt}) to minimize the aircraft drag and weight (Section 6.6)
5. Calculate horizontal tail planform area; S_t (equation 6.24).
6. Calculate wing-fuselage aerodynamic pitching moment coefficient (equation 6.26)
7. Calculate cruise lift coefficient (C_{Lc}); equation 6.27
8. Calculate horizontal tail desired lift coefficient at cruise from trim equation (6.29)
9. Select horizontal tail airfoil section (Section 6.7)
10. Select horizontal tail sweep angle and dihedral (Section 6.7)
11. Select horizontal tail aspect ratio and taper ratio (Section 6.7)
12. Determine horizontal tail lift curve slope; $C_{L_{\alpha_h}}$ (Equation 6.57)
13. Calculate horizontal tail angle of attack at cruise; (equation 6.51)
14. Determine downwash angle at the tail (equation 6.54)
15. Calculate horizontal tail incidence angle; i_t (equation 6.53)
16. Calculate tail span, tail root chord, tail tip chord and tail mean aerodynamic chord (equations 6.63 through 6.66)
17. Calculate horizontal tail generated lift coefficient at cruise (e.g. lifting line theory; Chapter 5). Treat the horizontal tail as a small wing.
18. If the horizontal tail generated lift coefficient (step 17) is not equal to the horizontal tail required lift coefficient (step 8), adjust tail incidence
19. Check horizontal tail stall
20. Calculate the horizontal tail contribution to the static longitudinal stability derivative ($C_{m\alpha}$). The value for $C_{m\alpha}$ derivative must be negative to insure a stabilizing contribution. If the design requirements are not satisfied, redesign the tail.
21. Analyze dynamic longitudinal stability. If the design requirements are not satisfied, redesign the tail.
22. Optimize horizontal tail

Vertical Tail

23. Select vertical tail configuration (e.g. conventional, twin vertical tail, vertical tail at swept wing tip, V-tail) (Section 6.8.2-1)
24. Select the vertical tail volume coefficient; $\overline{V}_V$ (Table 6.4)
25. Assume the vertical tail moment arm (l_v) as equal to the horizontal tail moment arm (l)
26. Calculate vertical tail planform area; S_v (equation 6.74)
27. Select vertical tail airfoil section (Section 6.8.2-4)
28. Select vertical tail aspect ratio; AR_v (Section 6.8.2-6)
29. Select vertical tail taper ratio; λ_V (Section 6.8.2-7)
30. Determine the vertical tail incidence angle (Section 6.8.2-5)
31. Determine the vertical tail sweep angle (Section 6.8.2-8)

32. Determine the vertical tail dihedral angle (Section 6.8.2-9)
33. Calculate vertical tail span (b_v), root chord (Cv_{root}), and tip chord(Cv_{tip}), and mean aerodynamic chord (MAC_v) (equations 6.76 through 6.79)
34. Check the spin recovery
35. Adjust the location of the vertical tail relative to the horizontal tail by changing l_v, to satisfy the spin recovery requirements (Section 6.8.2-2)
36. Analyze directional trim (Section 6.8.1)
37. Analyze directional stability (Section 6.8.1)
38. Modify to meet the design requirements
39. Optimize the tail

Reminder: Tail design is an iterative process. When the other aircraft components (such as fuselage and wing) are designed, the aircraft dynamic longitudinal-directional stability needs to be analyzed, and based on that; the tail design may need some adjustments.

6.10. Tail Design Example

Example 6.2 provides a tail design example.

Example 6.2

Problem statement: Design a horizontal tail for a two-seat motor glider aircraft with the following characteristics:

$$m_{TO} = 850 \text{ kg}, D_{fmax} = 1.1 \text{ m}, V_c = 95 \text{ knot (at 10,000 ft)}, \alpha_f = 1 \text{ deg (at cruise)}$$

The wing has a reference area 18 m^2 of and the following features:

$$\overline{C} = 0.8 \text{ m}, AR = 28, \lambda = 0.8, i_w = 3 \text{ deg}, \alpha_{twist} = -1.1 \text{ deg}, \Lambda_{LE} = 8 \text{ deg}, \Gamma = 5 \text{ deg},$$

$$\text{airfoil: NACA 23012}, C_{L\alpha} = 5.8 \text{ 1/rad}$$

The aircraft has a high wing and an aft conventional tail configuration, and the aerodynamic center of the wing-fuselage combination is located at 23% of MAC. In cruising flight condition, the aircraft center of gravity is located at 32 percent of the fuselage length. Assume that the aircraft cg is 7 cm ahead of the wing-fuselage aerodynamic center.

Then following tail parameters must be determined: airfoil section, S_h, C_{h_tip}, C_{h_root}, b_h, i_h, AR_h, λ_h, Λ_h, Γ_h. At the end, draw a top-view of the aircraft that shows fuselage, wing and horizontal tail (with dimensions).

Solution:

The tail configuration has been already selected and stated, so there is no need to investigate this item. The only parameter that needs to be decided is the type of setting angle. Since the aircraft is not maneuverable and the cost must be low, a fixed tail is selected. Thus, the design begins with the selection of the horizontal tail volume coefficient.

$\overline{V}_H = 0.6$ (Table 6.4)

To determine the optimum tail moment arm (l_{opt}), we set the goal to minimize the aircraft drag. Hence:

$$l = l_{opt} = K_c \sqrt{\frac{4\overline{C}S\overline{V}_H}{\pi D_f}} = 1.2\sqrt{\frac{4\times 0.8\times 18\times 0.6}{\pi \times 1.1}} = 3.795 \quad m \tag{6.47}$$

where the correction factor K_c is selected to be 1.2. Then, the tail planform area is determined as:

$$\overline{V}_H = \frac{lS_h}{\overline{C}S} \Rightarrow S_h = \frac{\overline{C}S\overline{V}_H}{l} = \frac{0.8\times 18\times 0.6}{3.795} = 2.277 \quad m^2 \tag{6.24}$$

The aircraft cruise lift coefficient is:

$$C_L = C_{L_C} = \frac{2W_{avg}}{\rho V_c^2 S} = \frac{2\times 850\times 9.81}{0.905\times (95\times 0.5144)^2 \times 18} = 0.428 \tag{6.27}$$

where the air density at 10,000 ft is 0.905 kg/m^3. The wing-fuselage aerodynamic pitching moment coefficient is:

$$C_{m_{owf}} = C_{m_{af}} \frac{AR\cos^2(\Lambda)}{AR + 2\cos(\Lambda)} + 0.01\alpha_t = -0.013\frac{28\times\cos^2(8)}{28 + 2\cos\ (8)} + 0.01\times(-1.1) = -0.023 \tag{6.26}$$

where the value for the wing airfoil section pitching moment coefficient ($C_{m_{owf}}$) is usually extracted from the airfoil graphs. Based on the Table 5.2, the value of $C_{m_{af}}$ for NACA 23012 airfoil section is -0.013.

In order to use the trim equation, we need to find h and h_o. Referring on Table 6.2, for this type of aircraft, the l_{opt}/L_f is 0.65. So the fuselage length is selected to be:

$L_f = l_{opt} /0.65 = 3.795/0.65 = 5.838$ m

The aerodynamic center of the wing-fuselage combination is located at 23% of MAC, and the aircraft center of gravity is located at the 32% of the fuselage length. This cg is 7 cm ahead of wing-fuselage aerodynamic center. Combining these three data, we have the following relationship regarding the wing:

X_{apex} + 0.23 MAC = 0.32 L_f + 0.07

Thus X_{apex} = – 0.23 MAC + 0.32 L_f + 0.0 7 = 1.754 m

This leads us to find the cg location (X_{cg}) in terms of MAC:

X_{cg} = 0.23 MAC – 0.07 = 0.23 (0.8 m) – 0.07 = 0.114 m (from wing leading edge)

$$\overline{X}_{cg} = h = \frac{0.114}{MAC} = \frac{0.114}{0.8} = 0.142 = 14.2\% \quad MAC$$

So h = 0.142. The tail efficiency is assumed to be 0.98. The horizontal tail required lift coefficient at cruise is calculated by using trim equation.

$$C_{m_{owf}} + C_L(h - h_o) - \eta_h \overline{V}_H C_{L_h} = 0 \Rightarrow C_{L_h} = \frac{C_{m_{owf}} + C_L(h - h_o)}{\overline{V}_H}$$

$$= \frac{-0.023 + 0.428 \times (0.114 - 0.23)}{0.6} \Rightarrow C_{L_h} = -0.121 \tag{6.29}$$

The horizontal tail airfoil section must have several properties that are described in Section 6.7. Two significant properties are: 1. Symmetric, 2. Thinner than wing airfoil. The wing thickness-to-chord ratio is 12 percent. There are several airfoil sections that can satisfy these requirements. But we are looking for one which a low drag coefficient. A symmetric airfoil section with a low drag coefficient (C_{do} = 0.005) and 3% thinner than the wing airfoil section is NACA 0009. Figure 6.19 provides the characteristic graphs for NACA 0009 airfoil section. From this figure, other features of this airfoil are extracted as follows:

C_{l_i}	$C_{d_{min}}$	C_m	$(C_l/C_d)_{max}$	α_o (deg)	α_s (deg)	$C_{l_{max}}$	C_{l_α} (1/rad)	$(t/c)_{max}$
0	0.005	0	83.3	0	13	1.3	6.7	9%

The initial tail aspect ratio is determined to be:

$$AR_h = \frac{2}{3} AR_w = \frac{2}{3} \times 28 = 18.6 \tag{6.59}$$

The tail taper ratio is initially determined to be equal to the wing taper ratio: $\lambda_h = \lambda_w = 0.8$.

The tail sweep angle and the tail dihedral angle are tentatively considered to be the same as those of wing. The reasons are presented in Section 6.7.

Λ_h = 10 deg, Γ_h = 5 deg

Now we need to determine the tail setting angle (i_h) such that it produces the tail coefficient of – 0.121. In order to determine this parameter, we not only need to consider all tail parameters, but also wing downwash. At the beginning, the tail angle of attack is determined based on the tail lift curve slope. In the next step, the lifting line theory is used to calculate the tail generated lift coefficient. If the tail generated lift coefficient is not equal to the tail required lift coefficient, the tail incidence will be adjusted until these two are equal. In the last, downwash is applied to determine the tail incidence. The tail lift curve slope is:

$$C_{L_\alpha} = \frac{C_{l_{\alpha_h}}}{1 + \frac{C_{l_{\alpha_h}}}{\pi \cdot AR_h}} = \frac{6.7}{1 + \frac{6.7}{3.14 \times 18.6}} = 6.1 \quad \frac{1}{rad} \tag{6.57}$$

The tail angle of attack in cruise is:

$$\alpha_h = \frac{C_{L_h}}{C_{L_{\alpha_h}}} = \frac{-0.121}{6.1} = -0.018 \quad rad = -1.02 \quad \deg \tag{6.51}$$

To calculate the tail created lift coefficient, the lifting line theory is employed as introduced in Chapter 5 (Section 5.14). The following MATLAB m-file is utilized to calculate the tail lift coefficient with an angle of attack of -1.02 degrees.

```
==========================================================================

clc
clear
N = 9; % (number of segments-1)
S = 2.277; % m^2
AR = 18.6; % Aspect ratio
lambda = 0.8; % Taper ratio
alpha_twist = 0.00001; % Twist angle (deg)
a_h = -1.02; % tail angle of attack (deg)
a_2d = 6.1; % lift curve slope (1/rad)
alpha_0 = 0.000001; % zero-lift angle of attack (deg)
b = sqrt(AR*S); % tail span
MAC = S/b; % Mean Aerodynamic Chord
Croot = (1.5*(1+lambda)*MAC)/(1+lambda+lambda^2); % root chord
theta = pi/(2*N):pi/(2*N):pi/2;
alpha=a_h+alpha_twist:-alpha_twist/(N-1):a_h; % segment's angle of attack
z = (b/2)*cos(theta);
c = Croot * (1 - (1-lambda)*cos(theta)); % Mean Aerodynamics chord at each
segment
mu = c * a_2d / (4 * b);
LHS = mu .* (alpha-alpha_0)/57.3; % Left Hand Side
% Solving N equations to find coefficients A(i):
for i=1:N
    for j=1:N
    B(i,j) =  sin((2*j-1) * theta(i)) * (1 + (mu(i) * (2*j-1)) /
sin(theta(i)));
    end
end
A=B\transpose(LHS);
for i = 1:N
    sum1(i) = 0;
    sum2(i) = 0;
    for j = 1 : N
        sum1(i) = sum1(i) + (2*j-1) * A(j)*sin((2*j-1)*theta(i));
        sum2(i) = sum2(i) + A(j)*sin((2*j-1)*theta(i));
    end
   end
CL_tail = pi * AR * A(1)
===================================================================
```

The output of this m-file is:

```
CL_tail = -0.0959
```

The tail is expected to generate a C_{Lh} of – 0.121, but it generates a C_{Lh} of – 0.0959. To increase the tail lift coefficient to the desired value, we need to increase the tail angle of attack. With a trial and error and using the same m-file, we find that the tail angle of attack of – 1.29 degrees generates the desired tail lift coefficient. Hence:

α_h = – 1.29 degrees

Now, we need to take into account the downwash. The ε_o (downwash angle at zero angle of attack) and $d\varepsilon/d\alpha$ (downwash slope) are:

$$\varepsilon_o = \frac{2C_{L_w}}{\pi \cdot AR} = \frac{2 \times 0.428}{\pi \cdot 28} = 0.0097 \quad rad = 0.558 \quad \deg \tag{6.55}$$

$$\frac{\partial \varepsilon}{\partial \alpha} = \frac{2C_{L_{\alpha_w}}}{\pi \cdot AR} = \frac{2 \times 5.8}{\pi \cdot 28} = 0.132 \quad \deg/\deg \tag{6.56}$$

Thus:

$$\varepsilon = \varepsilon_o + \frac{\partial \varepsilon}{\partial \alpha}\alpha_w = 0.0097 + 0.132 \times \frac{3}{57.3} = 0.017 \quad rad = 0.954 \quad \deg \tag{6.54}$$

Therefore, the tail setting angle would be:

$$\alpha_t = \alpha_f + i_h - \varepsilon \Rightarrow i_h = \alpha_h - \alpha_f + \varepsilon = -1.29 - 1 + 0.954 = -1.33 \quad \deg \tag{6.53}$$

The other horizontal tail parameters are determined by solving the following four equations simultaneously:

$$AR_h = \frac{b_h}{\overline{C}_h} \tag{6.63}$$

$$\lambda_h = \frac{C_{h_{tip}}}{C_{h_{root}}} \tag{6.64}$$

$$\overline{C}_h = \frac{2}{3}C_{h_{root}}\left(\frac{1 + \lambda_h + \lambda_h^{\,2}}{1 + \lambda_h}\right) \tag{6.65}$$

$$S_h = b_h \cdot \overline{C}_h \tag{6.66}$$

The solution of these four equations simultaneously yields the following results:

$$b_h = 6.52 \quad m, \quad \overline{C}_h = 0.349 \quad m, \quad C_{h_{tip}} = 0.309 \quad m, \quad C_{h_{root}} = 0.386 \quad m$$

The last step is to examine the aircraft static longitudinal stability. The aircraft has a fixed tail, so the aircraft static longitudinal stability derivative is determined as follows:

$$C_{m_\alpha} = C_{L_{\alpha_{wf}}}\left(h - h_o\right) - C_{L_{\alpha_h}}\eta_h \frac{S_h}{S}\left(\frac{l}{\overline{C}} - h\right)\left(1 - \frac{d\varepsilon}{d\alpha}\right) \quad (6.67)$$

$$C_{m_\alpha} = 5.7(0.114 - 0.23) - 6.1 \times 0.98\frac{2.277}{18}\left(\frac{3.795}{0.8} - 0.114\right)(1 - 0.132) = -3.7 \quad \frac{1}{rad} \quad (6.67)$$

where we assumed that the wing-fuselage lift curve slope is equal to the wing lift curve slope. Since the derivative Cm_α is negative, the aircraft is *statically longitudinally stable*. The aircraft longitudinal dynamic stability analysis requires the information about other aircraft components that are not provided by the problem statement. So this analysis is not performed in this example. Figure 6.28 shows top-view of the aircraft with details of the tail geometries.

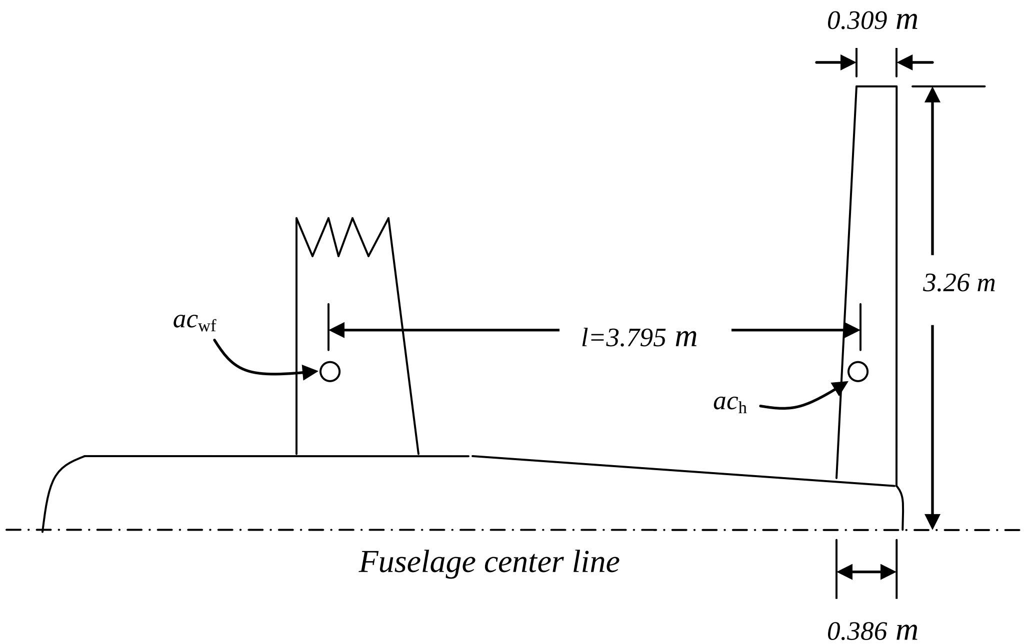

Figure 6.28. Top view of the aircraft in Example 6.2

It is important to note that this is the first phase of the horizontal tail design. If the characteristics of the other aircraft components are known, the complete analysis for the longitudinal dynamic and static stability may be performed and the tail could be optimized.

Problems

1. Using the Reference [5] or other reliable sources, identify the tail configurations of the following aircraft:

   Stemme S10 (Germany), Dassault Falcon 2000 (France), Embraer EMB 145 (Brazil), Canadair CL-415, ATR 42, Aeromacchi MB-339C (Italy), Eagle X-TS (Malaysia), PZL Mielec M-18 Dromader (Poland), Beriev A-50 (Russia), Sukhoi Su-32FN (Russia), Sukhoi S-80, Saab 340B (Sweden), Pilatus PC-12 (Switzerland), An-225 (Ukraine), Jetstream 41 (UK), FLS Optica OA7-300 (UK), Bell/Boeing V-22 Osprey, Boeing E-767 AWACS, Cessna 750 Citation X, Learjet 45, Lockheed F-16 Fighting Falcon, Lockheed F-117A Nighthawk, McDonnell Douglas MD-95, Northrop Grumman B-2 Spirit, Bede BD-10, Hawker 1000, Schweizer SA 2-38, Sino Swearingen SJ30, Visionaire Vantage

2. Using the Reference [5] or other reliable sources, identify an aircraft for each of the following tail configurations:

   Conventional aft tail, V-tail, Canard, T-tail, H-tail, Non-conventional, Cruciform, Tri-plane, Boom-mounted, twin vertical tail, inverted V-tail

3. Using the Reference [5] or other reliable sources, identify an aircraft with a conventional aft tail that the vertical tail is out of wake region of the horizontal tail.

4. An aircraft has a fuselage with a circular cross section. Derive an equation for the optimum horizontal tail moment arm such that the aft portion of the aircraft (including aft fuselage and horizontal tail) has the lowest wetted area.

5. An unmanned aircraft has the following features:

   $$S = 55 \text{ m}^2,\ AR = 25,\ S_h = 9.6 \text{ m}^2,\ l = 6.8 \text{ m}$$

   Determine the horizontal tail volume coefficient.

6. The airfoil section of a horizontal tail in a fighter aircraft is NACA 64-006. The tail aspect ratio is 2.3. Using the Reference [8], calculate the tail lift curve slope in 1/rad.

7. The airfoil section of a horizontal tail in a transport aircraft is NACA 64_1-012. The tail aspect ratio is 5.5. Using the Reference [8], calculate the tail lift curve slope in 1/rad.

8. The airfoil section of a horizontal tail in a GA aircraft is NACA 0012. The tail aspect ratio is 4.8. Using the Reference [8], calculate the tail lift curve slope in 1/rad.

9. The wing reference area of an agricultural aircraft is 14.5 m^2 and wing mean aerodynamic chord is 1.8 m. The longitudinal stability requirements dictate the tail volume coefficient to be 0.9. If the maximum fuselage diameter is 1.6 m, determine the optimum tail arm and then calculate the horizontal tail area. Assume that the aft portion of the fuselage is conical.

10. Consider a single-seat GA aircraft whose wing reference area is 12 m^2 and wing mean aerodynamic chord is 1.3 m. The longitudinal stability requirements dictate the tail volume coefficient to be 0.8. If the maximum fuselage diameter is 1.3 m, determine the optimum tail

arm and then calculate the horizontal tail area. Assume that the aft portion of the fuselage is conical.

11. A 19-seat business aircraft with a mass 6,400 kg is cruising with a speed of 240 knot at 26,000 ft. Assume that the aircraft lift coefficient is equal to the wing lift coefficient. The aircraft has the following characteristics:

$$S = 32\ m^2,\ AR_w = 8.7,\ \text{Wing airfoil: NACA } 65_1\text{-}412$$

Determine the downwash angle (in degrees) at the horizontal tail.

12. Suppose that the angle of attack of the fuselage for the aircraft in problem 11 is 2.3 degrees and the horizontal tail has an incidence of -1.5 degrees. How much is the horizontal tail angle of attack at this flight condition?

13. The horizontal tail of a transport aircraft has the following features:

$$AR_h = 5.4,\ \lambda_h = 0.7,\ S_h = 14\ m^2,\ \Lambda_{h\_LE} = 30 \text{ degrees}$$

Determine span, root chord, tip chord and the mean aerodynamic of the horizontal tail. Then sketch the top-view of the tail with dimensions.

14. The horizontal tail of a fighter aircraft has the following features:

$$AR_h = 3.1,\ \lambda_h = 0.6,\ S_h = 6.4\ m^2,\ \Lambda_{h\_LE} = 40 \text{ degrees}$$

Determine span, root chord, tip chord and the mean aerodynamic of the horizontal tail. Then sketch the top-view of the tail with dimensions.

15. The vertical tail of a transport aircraft has the following features:

$$AR_V = 1.6,\ \lambda_V = 0.4,\ S_V = 35\ m^2,\ \Lambda_{V\_LE} = 45 \text{ degrees}$$

Determine span, root chord, tip chord and the mean aerodynamic of the vertical tail. Then sketch the side-view of the tail with dimensions.

16. The aircraft in problem 11 has other features as follows:

$$h = 0.18,\ h_o = 0.23,\ \eta_h = 0.97,\ l = 12\ m,\ S_h = 8.7\ m^2$$

Determine the aircraft static longitudinal stability derivative (Cm_α) and discuss whether the horizontal tail is longitudinally stabilizing or destabilizing.

17. Design a horizontal tail for a twin jet business aircraft with the following characteristics:

$$m_{TO} = 16{,}000\ kg,\ D_{fmax} = 1.8\ m,\ V_c = 270 \text{ knot (at 30,000 ft)},\ \alpha_f = 1.5 \text{ deg (at cruise)}$$

The wing has a reference area 49 m^2 of and the following features:

$$AR = 8,\ \lambda = 0.6,\ i_w = 2.4 \text{ deg},\ \alpha_{twist} = -1.3 \text{ deg},\ \Lambda_{LE} = 37 \text{ deg},\ \Gamma = 3 \text{ deg, NACA } 65_2\text{-}415$$

The aircraft has a low wing and an aft conventional tail configuration, and the aerodynamic center of the wing-fuselage combination is located at 22% of MAC. In cruising flight condition, the aircraft center of gravity is located at 42% of the fuselage length. Assume that the aircraft cg is 15 cm ahead of the wing-fuselage aerodynamic center.

The following tail parameters must be determined: airfoil section, S_h, C_{h_tip}, C_{h_root}, b_h, i_h, AR_h, λ_h, Λ_h, Γ_h. At the end, draw a top-view of the aircraft that shows fuselage, wing and horizontal tail (with dimensions).

18. A large transport aircraft with a mass of 63,000 kg is supposed to cruise with a speed of 510 knots at 42,000 ft. The maximum fuselage diameter is 3.6 m and fuselage angle of attack at cruise is 3.2 degrees. The wing has a reference area 116 m^2 of and the following features:

$$AR = 11.5,\ \lambda = 0.5,\ i_w = 2.7 \text{ deg},\ \alpha_{twist} = -1.6 \text{ deg},\ \Lambda_{LE} = 30 \text{ deg},\ \Gamma = 6 \text{ deg, NACA } 64_1\text{-}412$$

The aircraft has a low wing and a T-tail configuration, and the aerodynamic center of the wing-fuselage combination is located at 20% of MAC. In cruising flight condition, the aircraft center of gravity is located at 49% of the fuselage length. Assume that the aircraft cg is 18 cm ahead of the wing-fuselage aerodynamic center. Design a horizontal tail to satisfy longitudinal trail and static longitudinal stability requirements. Then determine airfoil section, S_h, C_{h_tip}, C_{h_root}, b_h, i_h, AR_h, λ_h, Λ_h, Γ_h. At the end, draw a top-view of the aircraft that shows fuselage, wing and horizontal tail (with dimensions).

19. Figure 6.29 shows the original design for the empennage of a transport aircraft with a horizontal tail area of 12.3 m^2. The wing reference area is 42 m^2, and wing aspect ratio is 10.5.

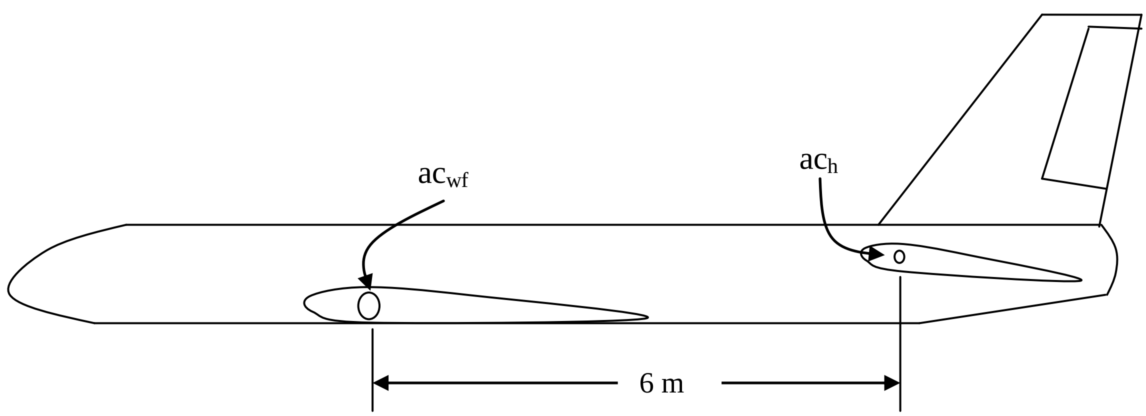

Figure 6.29. Side-view of the aircraft in problem 19

The aircraft is spinnable and the designer found out that the vertical tail is not effective for spin recovery. Move the horizontal tail horizontally such that the vertical tail becomes effective in recovering from spin. Then determine the horizontal tail area such that the horizontal tail volume coefficient remains unchanged. Assume that the sketch in figure 6.29 is scaled.

20. A fighter aircraft has the following features:

$$S = 57 \text{ m}^2,\ AR = 3,\ S_h = 10.3 \text{ m}^2,\ S_v = 8.4 \text{ m}^2,\ l = 6.8 \text{ m},\ l_v = 6.2 \text{ m}$$

Determine the horizontal and vertical tails volume coefficients.

21. Design a vertical tail for the aircraft in problem 18 to satisfy the static directional stability requirements.

22. The airfoil section of the vertical tail for a twin-jet engine aircraft is NACA 66-009. Other features of the aircraft is as follows:

$$S = 32 \text{ m}^2,\ AR = 10.3,\ S_V = 8.1 \text{ m}^2,\ AR_V = 1.6,\ l = 9.2 \text{ m},\ \frac{d\sigma}{d\beta} = 0.32,\ \eta_V = 0.95$$

Determine the aircraft static directional stability derivative (Cn_β). Then analyze the static directional stability of the aircraft.

23. The angle of attack of a horizontal tail for a cargo aircraft is -1.6 degrees. Other tail features are as follows:

$$S_h = 12 \text{ m}^2,\ AR_h = 5.3,\ \lambda_h = 0.7,\ \text{airfoil section: NACA 64-208},\ \eta_h = 0.96$$

If the aircraft is flying at an altitude of 15,000 ft with a speed of 245 knot, determine how much lift is generated by the tail. Assume that the tail has no twist.

24. The sideslip angle of a vertical tail for a maneuverable aircraft during a turn is 4 degrees. Other vertical tail features are as follows:

$$S_h = 7.5 \text{ m}^2,\ AR_V = 1.4,\ \lambda_V = 0.4,\ \text{airfoil section: NACA 0012},\ \eta_V = 0.92$$

If the aircraft is flying at an altitude of 15,000 ft with a speed of 245 knot, determine how much lift (i.e. side force) is generated by the vertical tail. Assume that the tail has no twist.

25. An aft horizontal tail is supposed to be designed for a single piston engine aircraft. The aircraft with a mass of 1,800 kg is cruising with a speed of 160 knot an altitude of 22,000 ft. The aircraft center of gravity is at 19% MAC and the wing-fuselage aerodynamic center is located at 24% MAC.

$$S = 12 \text{ m}^2,\ AR = 6.4,\ S_h = 2.8 \text{ m}^2,\ l = 3.7 \text{ m},\ C_{m_{owf}} = -0.06$$

Determine the horizontal tail lift coefficient that must be produced in order to maintain the longitudinal trim.

26. Redo the problem 25 with the assumption that the aircraft has a canard instead of an aft horizontal tail.

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