4.10. Contribution of Running Propellers

Running propellers can have a profound effect on the trim and static stability of an airplane. The contributions of a running propeller to trim and static stability can be divided into two main categories. These are the direct effects, arising from the aerodynamic forces and moments on the propeller itself, and the indirect effects, which result from the interaction of the propeller slipstream with other surfaces of the airplane.

The effects of the propeller slipstream on other airplane surfaces such as the wing, tail, and fuselage are very complex and do not lend themselves to accurate analytical treatment. These effects can be accurately evaluated only from powered wind tunnel tests. Such tests are commonly performed in the final phase of the airplane design process. Simply because of complexity, the slipstream effects are usually neglected in the preliminary design phase. It should be remembered, however, that the slipstream effects can be significant.

The direct effects of the propeller forces and moments, on the other hand, can be estimated analytically using the method presented in Chapter 2. When a rotating propeller is advancing through the air so that the axis of rotation make some angle of attack with the freestream, in addition to the thrust, a normal force is produced that is proportional to the angle of attack. This is shown in Figs. 4.10.1 and 4.10.2. The pitching moment that the propeller produces about the airplane's center of gravity is a result of the thrust and normal force, each multiplied by the appropriate moment arm,

$$m_{cg_p} = -h_p T - l_p N_p \tag{4.10.1}$$

where T and N_p are, respectively, the propeller thrust and normal force. For consistency, we are defining l_p as the axial distance that the propeller is mounted aft of the CG and h_p is the vertical distance that the propeller axis is above the CG. For a conventional tractor prop, l_p is typically negative, and for a pusher prop, l_p is usually positive.

Figure 4.10.1. Forces generated by an aft propeller. (Photograph by Barry Santana)

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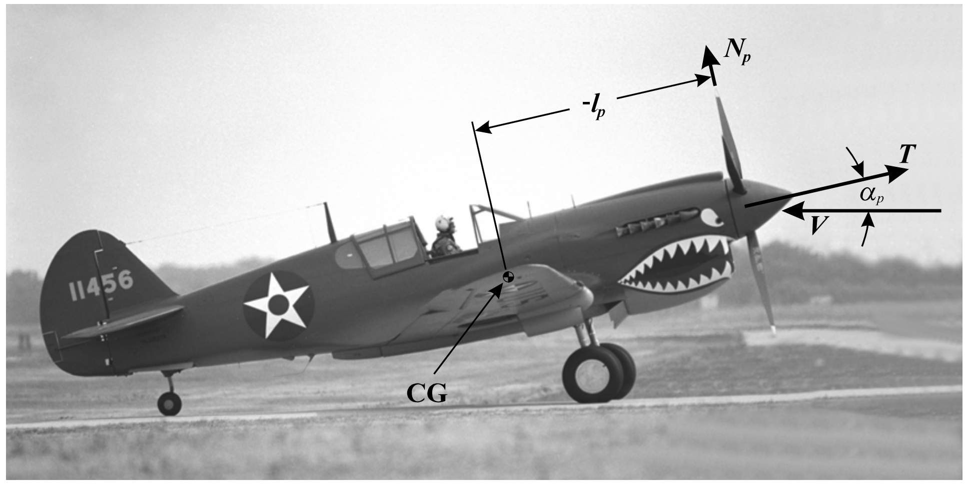

Figure 4.10.2. Forces generated by a forward propeller. (Photograph by Barry Santana)

The pitching moment computed from Eq. (4.10.1) can be added to the pitching moment obtained for all of the other airplane components to determine the pitching moment for the complete airplane. Thus, the contribution of the propeller to the total airplane pitching moment coefficient is found by nondimensionalizing Eq. (4.10.1),

$$(\Delta C_m)_p = \frac{m_{cg_p}}{\frac{1}{2}\rho V_\infty^2 S_w \bar{c}_w} = -\frac{h_p}{\bar{c}_w}\frac{T}{\frac{1}{2}\rho V_\infty^2 S_w} - \frac{l_p}{\bar{c}_w}\frac{N_p}{\frac{1}{2}\rho V_\infty^2 S_w} \tag{4.10.2}$$

For small thrust angles the equilibrium thrust is nearly equal to the drag and Eq. (4.10.2) can be closely approximated as

$$(\Delta C_m)_p = \frac{m_{cg_p}}{\frac{1}{2}\rho V_\infty^2 S_w \bar{c}_w} = -\frac{h_p}{\bar{c}_w} C_D - \frac{l_p}{\bar{c}_w}\frac{N_p}{\frac{1}{2}\rho V_\infty^2 S_w} \tag{4.10.3}$$

While the equilibrium thrust developed by an airplane's propeller can be related to the airplane drag, this cannot be said for the propeller normal force. To evaluate the propeller normal force, we return to the material presented in Chapter 2.

Propeller forces are conventionally expressed in terms of the thrust coefficient and the normal force coefficient,

$$C_T \equiv \frac{T}{\rho(\omega/2\pi)^2 d_p^4} \tag{4.10.4}$$

$$C_{N_p} \equiv \frac{N_p}{\rho(\omega/2\pi)^2 d_p^4} \tag{4.10.5}$$

where d_p is the propeller diameter and ω is the angular velocity. The normal force coefficient is linearly proportional to the angle of attack, α_p, that the propeller axis makes with the direction of relative airflow,

$$C_{N_p} = C_{N_p,\alpha}\,\alpha_p \tag{4.10.6}$$

The proportionality constant is the dimensionless normal force gradient, defined as

$$C_{N_p,\alpha} \equiv \frac{1}{\rho(\omega/2\pi)^2 d_p^4}\frac{\partial N_p}{\partial \alpha_p} \tag{4.10.7}$$

For a given propeller geometry both the thrust coefficient and the dimensionless normal force gradient are strong functions of the propeller advance ratio, J, which was defined in Chapter 2 as

$$J \equiv \frac{2\pi V_\infty}{\omega d_p} \tag{4.10.8}$$

With a fixed-pitch propeller of given geometry, to attain some particular level flight airspeed, the pilot must adjust the propeller's rotational speed so that the propeller thrust will balance the drag at the desired airspeed. The advance ratio that is required to balance the drag with the thrust determines the propeller normal force gradient for a particular operating condition. Since the thrust coefficient decreases with increasing advance ratio while the dimensionless normal force gradient increases with increasing advance ratio, the propeller's effect on static stability and trim will vary significantly with propeller advance ratio.

Using Eqs. (4.10.5) through (4.10.8) in Eq. (4.10.3), the contribution of the propeller to the total airplane pitching moment coefficient is

$$\left(\Delta C_m\right)_p = -\frac{h_p}{\bar{c}_w}C_D - \frac{2\,d_p^2\,l_p}{S_w\bar{c}_w}\frac{C_{N_p,\alpha}}{J^2}\alpha_p \tag{4.10.9}$$

The dimensionless normal force gradient, which appears in Eq. (4.10.9), depends on the number and geometry of the propeller blades, as well as the advance ratio. For known propeller geometry, the normal force gradient can be determined as a function of advance ratio using the method presented in Chapter 2. For example, Fig. 4.10.3 shows the dimensionless normal force gradient divided by the advance ratio squared, plotted as a function of advance ratio and pitch-to-diameter ratio, K_c, for one particular propeller blade geometry.

The propeller angle of attack in Eq. (4.10.9) is affected by the airflow around the wing. If the propeller is aft of the wing, it is located in a region of downwash and the propeller angle of attack is decreased. If the propeller is forward of the wing, it is in a region of upwash and the propeller angle of attack is increased. The angle of attack that the propeller axis makes with the local relative airflow can be written as

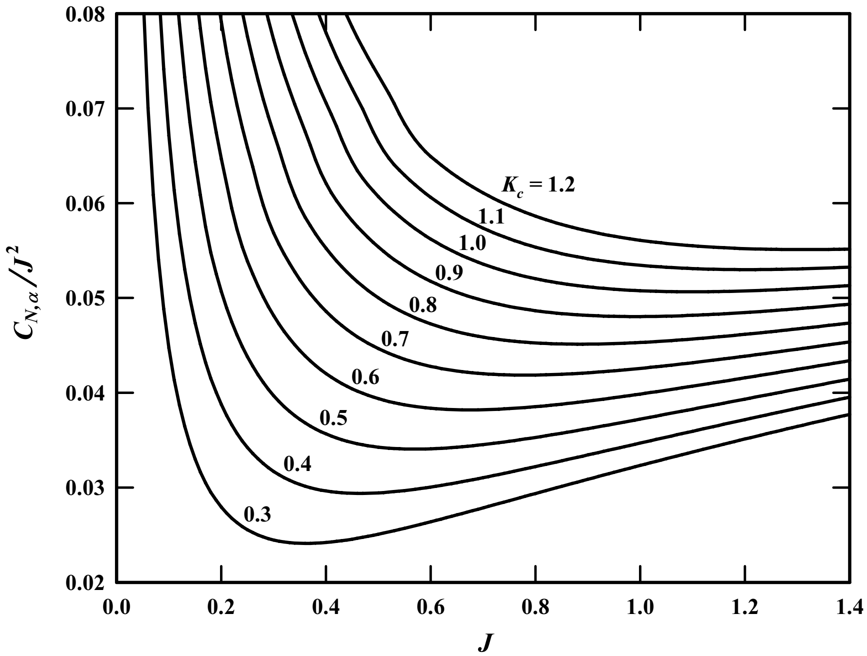

Figure 4.10.3. Dimensionless normal force gradient divided by the advance ratio squared.

$$\alpha_p = \alpha - \varepsilon_{dp} + \alpha_{0p} \tag{4.10.10}$$

where α is the airplane angle of attack relative to the fuselage reference line, ε_{dp} is the local downwash angle at the position of the propeller, and α_{0p} is the angle that the propeller axis makes with the fuselage reference line. It should be noted that in some textbooks, a propeller upwash angle is used in Eq. (4.10.10), because propellers are most commonly mounted ahead of the wing. For consistency with the material presented in Sec. 4.5, in this textbook we are defining ε_{dp} as the propeller downwash angle. With the present definition, if the propeller is forward of the wing, ε_{dp} is negative. If the propeller is aft of the wing, ε_{dp} is positive.

Using Eq. (4.10.10) with Eq. (4.10.9), the contribution of the propeller to the total airplane pitching moment coefficient can be written as

$$\left(\Delta C_m\right)_p = \left(\Delta C_{m0}\right)_p + \left(\Delta C_{m,\alpha}\right)_p \alpha \tag{4.10.11}$$

where

$$\left(\Delta C_{m0}\right)_p = -\frac{h_p}{\bar{c}_w} C_D - \frac{2 d_p^2 l_p}{S_w \bar{c}_w} \frac{C_{N_p,\alpha}}{J^2} (\alpha_{0p} - \varepsilon_{d0p}) \tag{4.10.12}$$

$$\left(\Delta C_{m,\alpha}\right)_p = -\frac{2 d_p^2 l_p}{S_w \bar{c}_w}(1-\varepsilon_{d,\alpha})_p \frac{C_{N_p,\alpha}}{J^2} \tag{4.10.13}$$

ε_{d0p} is the propeller downwash angle with the fuselage reference line at zero angle of attack and $(\varepsilon_{d,\alpha})_p$ is the downwash gradient at the position of the propeller. Since the change in the normal force coefficient with angle of attack is positive, if the propeller is forward of the center of gravity, l_p is negative and the propeller contribution to the pitch stability derivative is destabilizing. If the propeller is aft of the center of gravity, l_p is positive and the propeller contribution to the pitch stability derivative is stabilizing.

EXAMPLE 4.10.1. The airplane that is described in Examples 4.4.1 and 4.9.1 is powered by a piston engine turning a 74-inch propeller at 2,350 rpm and is flying at 80 mph. The change in normal force coefficient with propeller angle of attack for this propeller and operating condition is 0.04. The propeller is mounted in front of the fuselage and is 9 feet forward of the center of gravity. Estimate the static margin in the linear lift range for the complete airplane, including the effect of the fuselage and propeller.

Solution. For this airplane we have

$$S_w = 180\,\text{ft}^2, \quad b_w = 33\,\text{ft}, \quad C_{L_w,\alpha} = 4.44, \quad l_w = -0.71\,\text{ft},$$
$$S_t = 36\,\text{ft}^2, \quad b_t = 12\,\text{ft}, \quad C_{L_t,\alpha} = 3.97, \quad l_t = 14.29\,\text{ft},$$
$$R_{T_w} = 0.40, \quad \Lambda_w = 0°, \quad \eta_t = 1.0, \quad (\varepsilon_{d,\alpha})_t = 0.44,$$
$$S_f = 21\,\text{ft}^2, \quad d_f = 2\sqrt{21\,\text{ft}^2/\pi} = 5.17\,\text{ft}, \quad c_f = 23\,\text{ft}, \quad l_f = -3.5\,\text{ft},$$
$$d_p = 74/12\,\text{ft}, \quad \omega/2\pi = 2{,}350\,\text{rpm}, \quad l_p = -9\,\text{ft}, \quad V_\infty = 80\,\text{mph}, \quad C_{N_p,\alpha} = 0.04$$

For this propeller and airspeed the advance ratio is

$$J = \frac{V_\infty}{(\omega/2\pi)d_p} = \frac{80 \times 5{,}280/3{,}600}{(2{,}350/60)(74/12)} = 0.4858$$

The downwash angle for the propeller can be estimated using the method presented in Sec. 4.5. For this wing and propeller position, we have

$$R_{A_w} = 6.05, \quad R_{T_w} = 0.40, \quad \bar{x} = \frac{l_p - l_w}{b_w/2} = -0.502, \quad \bar{y} = 0.0, \quad \Lambda_w = 0°$$

and the propeller downwash angle is

$$\varepsilon_{dp} = \frac{\kappa_v \kappa_p \kappa_s}{\kappa_b} \frac{C_{L_w}}{R_{A_w}} = \frac{1.035(-0.165)1.0}{0.759} \frac{C_{L_w}}{R_{A_w}} = -0.225 \frac{C_{L_w}}{R_{A_w}}$$

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The propeller downwash gradient is then

$$(\varepsilon_{d,\alpha})_p = -0.225\frac{C_{L_w,\alpha}}{R_{A_w}} = -0.225\frac{4.44}{6.05} = -0.165$$

By definition, the moment slope relative to the neutral point is zero. Thus, using Eqs. (4.9.6) and (4.10.12) to estimate the fuselage and propeller contributions to the moment slope,

$$\begin{aligned}\frac{\partial C_{m_{np}}}{\partial \alpha} &= -\frac{(l_w - l_{np})}{\bar{c}_w}C_{L_w,\alpha} - \frac{S_t(l_t - l_{np})}{S_w\bar{c}_w}\eta_t(1-\varepsilon_{d,\alpha})_t C_{L_t,\alpha}\\ &\quad + \frac{l_f - l_{np}}{l_f}\left(\Delta C_{m,\alpha}\right)_f - \frac{2d_p^2(l_p - l_{np})}{J^2 S_w\bar{c}_w}(1-\varepsilon_{d,\alpha})_p\, C_{N_p,\alpha}\\ &= 0\end{aligned}$$

where

$$\begin{aligned}\left(\Delta C_{m,\alpha}\right)_f &= -2\frac{S_f l_f}{S_w\bar{c}_w}\left[1 - 1.76\left(\frac{d_f}{c_f}\right)^{3/2}\right]\\ &= -2\frac{21(-3.5)}{180(180/33)}\left[1 - 1.76\left(\frac{5.17}{23}\right)^{3/2}\right]\\ &= 0.1216\end{aligned}$$

Solving for the static margin gives

$$\frac{l_{np}}{\bar{c}_w} = \frac{l_w C_{L_w,\alpha} + \dfrac{S_t l_t}{S_w}\eta_t(1-\varepsilon_{d,\alpha})_t C_{L_t,\alpha} - \bar{c}_w\left(\Delta C_{m,\alpha}\right)_f + \dfrac{2d_p^2\, l_p}{J^2 S_w}(1-\varepsilon_{d,\alpha})_p C_{N_p,\alpha}}{\bar{c}_w\left\{C_{L_w,\alpha} + \dfrac{S_t}{S_w}\eta_t(1-\varepsilon_{d,\alpha})_t C_{L_t,\alpha} - \dfrac{\bar{c}_w}{l_f}\left(\Delta C_{m,\alpha}\right)_f + \dfrac{2d_p^2}{J^2 S_w}(1-\varepsilon_{d,\alpha})_p C_{N_p,\alpha}\right\}}$$

$$= \frac{(-.71)4.44 + \dfrac{36(14.29)}{180}1.(1-.44)3.97 - \dfrac{180}{33}.1216 + \dfrac{2(74/12)^2(-9)}{(.4858)^2 180}(1.165).04}{\dfrac{180}{33}\left\{4.44 + \dfrac{36}{180}1.(1-.44)3.97 - \dfrac{180/33}{-3.5}.1216 + \dfrac{2(74/12)^2}{(.4858)^2 180}(1.165).04\right\}}$$

$$= \underline{6\%}$$

Comparing this result with that from Example 4.9.1, we see that the destabilizing effect of the propeller was to move the neutral point forward by about 3 percent of the wing chord. This is quite typical of what might be expected for an airplane with low thrust-to-weight ratio. However, an even greater destabilizing effect can be encountered for airplanes with higher thrust-to-weight ratio, such as the fighter planes of World War II.

4.11. Contribution of Jet Engines

Turbojet or turbofan engines can also make a significant contribution to the trim and static stability of an airplane. These effects are also divided into two categories, those arising directly from the forces and moments produced by the engine and those which result from the interaction of the exhaust jet with other surfaces of the airplane.

Even if the exhaust jet does not impinge directly upon any surface of the airplane, the jet can still have significant indirect effects resulting from the entrained flow. Because the velocity of the exhaust gas is much greater than that of the surrounding air, a turbulent shear layer develops between the exhaust jet and the freestream. The viscous shear and turbulent mixing that occur between the exhaust jet and the surrounding air decrease the velocity of the exhaust gas and increase the velocity of the adjacent air. This entrainment of freestream air into the exhaust jet causes the jet to broaden as it moves downstream and induces a flow of ambient air toward the axis of the jet, as is shown in Fig. 4.11.1. If the tail or any other surface of the airplane is placed in this entrainment flow field, the local angle of attack will be modified by the entrained flow. For example, if the horizontal tail is above the exhaust jet, as shown in Fig. 4.11.1, the angle of attack will be decreased as a result of the entrainment.

The indirect effects of the exhaust jet on other airplane surfaces, such as the wing, the tail, and the fuselage, are very complex and can only be evaluated analytically with the use of computational methods on a digital computer. Because entrainment is a viscous phenomenon, inviscid methods such as panel codes cannot be used for this analysis. A viscous CFD code must be used. Such methods can be used in the final phase of the airplane design process. However, because of complexity, the indirect effects of jet exhaust are usually neglected in the preliminary design phase. It should be remembered, however, that such effects could be significant.

The direct effects of the forces and moments produced by a jet or turbofan engine are essentially identical to those produced by an engine-propeller combination. Like a propeller, a turbojet or turbofan engine produces both an axial thrust and an off-axis normal force. The normal force is proportional to the angle that the axis of the exhaust jet makes with the local freestream at the engine inlet. As was the case for a propeller, the pitching moment that a jet engine produces about the airplane center of gravity is

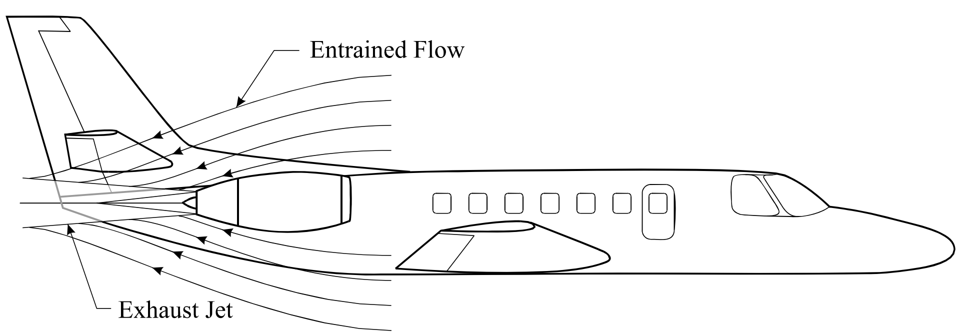

Figure 4.11.1. Flow entrained by the exhaust of a turbojet or turbofan engine.

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simply a result of the thrust and the normal force, each multiplied by the appropriate moment arm. The only difference between computing the pitching moment contribution for a jet engine and that for propeller is the manner in which the normal force is computed.

The velocity of the air passing through a turbojet or turbofan engine can undergo a change in direction as well as a change in magnitude. The thrust and normal forces exerted on the airplane by the engine result directly from these changes in the velocity vector. Consider a control volume that encloses the engine of the airplane that is shown in Fig. 4.11.2. This control volume extends everywhere sufficiently far from the engine so that ambient pressure exists on the entire enclosing surface. Writing the vector form of Newton's second law for this nonaccelerating control volume, we have

$$\mathbf{F}_j = \dot{m}_j(\mathbf{V}_i - \mathbf{V}_j) \tag{4.11.1}$$

where $\mathbf{F}_j$ is the resultant force exerted on the airplane by the engine, $\mathbf{V}_i$ and $\mathbf{V}_j$ are, respectively, the inlet and exit velocities, and $\dot{m}_j$ is the mass flow rate passing through the engine. The axial and normal components of this vector equation result in

$$T = \dot{m}_j(V_j - V_i \cos\alpha_j) \tag{4.11.2}$$

$$N_j = \dot{m}_j V_i \sin\alpha_j \tag{4.11.3}$$

where V_j is the magnitude of the exhaust jet velocity, V_i is the magnitude of the local freestream velocity at the inlet, and α_j is the angle of attack that the axis of the exhaust jet makes with the direction of the local freestream at the inlet. For small angles, Eqs. (4.11.2) and (4.11.3) can be approximated as

$$T = \dot{m}_j(V_j - V_\infty) \tag{4.11.4}$$

$$N_j = \dot{m}_j V_\infty \alpha_j \tag{4.11.5}$$

where V_∞ is the far-field airspeed. Solving Eq. (4.11.4) for the mass flow rate and using the result in Eq. (4.11.5) gives

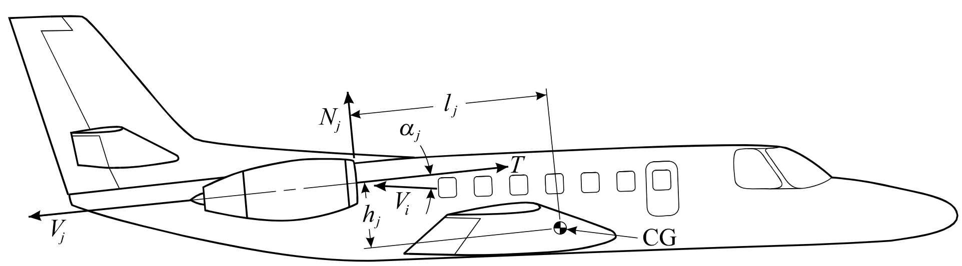

Figure 4.11.2. Forces generated by a turbojet or turbofan engine.

$$N_j = \frac{V_\infty T}{V_j - V_\infty}\alpha_j \tag{4.11.6}$$

From the geometry in Fig. 4.11.2, the pitching moment that the engine produces about the airplane center of gravity is

$$m_{cg_j} = -T h_j - N_j l_j \tag{4.11.7}$$

Nondimensionalizing Eq. (4.11.7) and applying Eq. (4.11.6), the contribution of the jet engine to the total airplane pitching moment coefficient is

$$(\Delta C_m)_j = \frac{m_{cg_j}}{\frac{1}{2}\rho V_\infty^2 S_w \bar{c}_w} = -\frac{T}{\frac{1}{2}\rho V_\infty^2 S_w}\left(\frac{h_j}{\bar{c}_w} + \frac{V_\infty l_j}{(V_j - V_\infty)\bar{c}_w}\alpha_j\right) \tag{4.11.8}$$

The local angle of attack in Eq. (4.11.8) is affected by the airflow around the wing. If the engine inlet is aft of the wing, it is located in a region of downwash and the local angle of attack is decreased. If the inlet is forward of the wing, it is in a region of upwash and the local angle of attack is increased. The angle of attack that the exhaust jet axis makes with the local relative airflow at the engine inlet can be written as

$$\alpha_j = \alpha - \varepsilon_{dj} + \alpha_{0j} \tag{4.11.9}$$

where α is the airplane angle of attack relative to the fuselage reference line, ε_{dj} is the local downwash angle at the position of the engine inlet, and α_{0j} is the angle that the exhaust jet axis makes with the fuselage reference line.

Using Eq. (4.11.9) in Eq. (4.11.8), the contribution of a jet engine to the total airplane pitching moment coefficient can be written as

$$\left(\Delta C_m\right)_j = \left(\Delta C_{m0}\right)_j + \left(\Delta C_{m,\alpha}\right)_j \alpha \tag{4.11.10}$$

where

$$(\Delta C_{m0})_j = -\frac{T}{\frac{1}{2}\rho V_\infty^2 S_w}\left[\frac{h_j}{\bar{c}_w} + \frac{V_\infty l_j}{(V_j - V_\infty)\bar{c}_w}(\alpha_{0j} - \varepsilon_{d0j})\right] \tag{4.11.11}$$

$$(\Delta C_{m,\alpha})_j = -\frac{T}{\frac{1}{2}\rho V_\infty^2 S_w}\frac{V_\infty l_j}{(V_j - V_\infty)\bar{c}_w}(1 - \varepsilon_{d,\alpha})_j \tag{4.11.12}$$

ε_{d0j} is the inlet downwash angle with the fuselage reference line at zero angle of attack, and $(\varepsilon_{d,\alpha})_j$ is the downwash gradient at the position of the engine inlet. Since the thrust is positive in normal flight, if the inlet is forward of the center of gravity, l_j is negative and the engine's contribution to the pitch stability derivative is destabilizing. If the inlet is aft of the center of gravity, l_j is positive and the engine's contribution to the pitch stability derivative is stabilizing.

As discussed in Sec. 2.1, the ideal propulsive efficiency for a jet engine is a function only of the exhaust jet velocity and the freestream velocity,

$$\eta_{p_i} = \frac{2V_\infty}{V_\infty + V_j} = \frac{2}{1 + V_j/V_\infty} \tag{4.11.13}$$

Thus, we can express the ratio of the exhaust jet velocity to the freestream velocity as a function of ideal propulsive efficiency. Rearranging Eq. (4.11.13), the velocity ratio for a jet engine can be expressed as

$$\frac{V_j}{V_\infty} = \frac{2}{\eta_{p_i}} - 1 \tag{4.11.14}$$

or

$$\frac{V_\infty}{V_j - V_\infty} = \frac{\eta_{p_i}}{2(1-\eta_{p_i})} \tag{4.11.15}$$

The velocity ratio in Eq. (4.11.15), which also appears in Eqs. (4.11.11) and (4.11.12), is plotted in Fig. 4.11.3. This figure combined with Eqs. (4.11.11) and (4.11.12) shows that increasing the propulsive efficiency of a jet engine will also increase the engine's effect on static stability and trim.

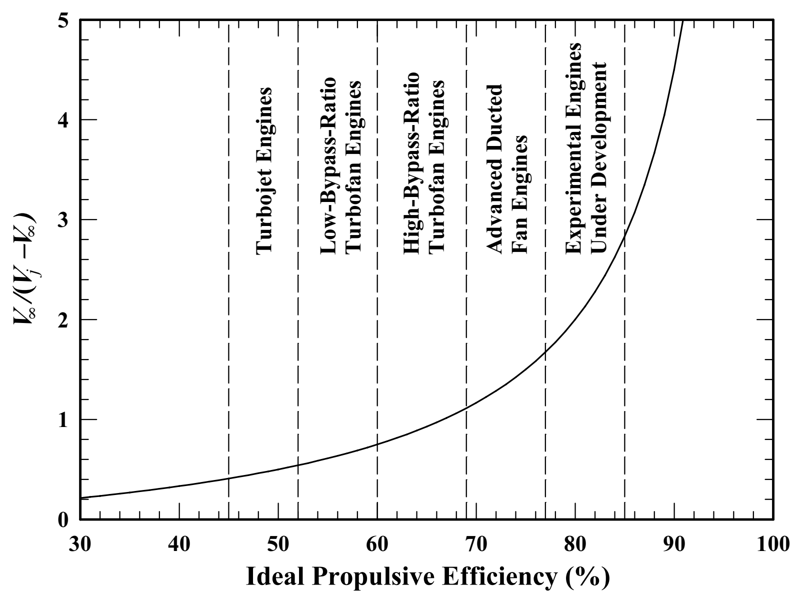

Figure 4.11.3. Velocity ratio for a jet engine as a function of ideal propulsive efficiency.

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Using Eq. (4.11.15) in Eqs. (4.11.11) and (4.11.12), we have

$$(\Delta C_{m0})_j = -\frac{T}{\frac{1}{2}\rho V_\infty^2 S_w}\left[\frac{h_j}{\bar{c}_w} + \frac{\eta_{p_i} l_j}{2(1-\eta_{p_i})\bar{c}_w}(\alpha_{0j} - \varepsilon_{d0j})\right] \tag{4.11.16}$$

$$(\Delta C_{m,\alpha})_j = -\frac{T}{\frac{1}{2}\rho V_\infty^2 S_w}\frac{\eta_{p_i} l_j}{2(1-\eta_{p_i})\bar{c}_w}(1 - \varepsilon_{d,\alpha})_j \tag{4.11.17}$$

For small thrust angles, the thrust is very nearly equal to the drag and Eqs. (4.11.16) and (4.11.17) can be closely approximated as

$$(\Delta C_{m0})_j = -C_D\left[\frac{h_j}{\bar{c}_w} + \frac{\eta_{p_i} l_j}{2(1-\eta_{p_i})\bar{c}_w}(\alpha_{0j} - \varepsilon_{d0j})\right] \tag{4.11.18}$$

$$(\Delta C_{m,\alpha})_j = -C_D\,\frac{\eta_{p_i} l_j}{2(1-\eta_{p_i})\bar{c}_w}(1 - \varepsilon_{d,\alpha})_j \tag{4.11.19}$$

From Eqs. (4.11.18) and (4.11.19) we can estimate the effects of a jet engine on airplane trim and stability if we know the ideal propulsive efficiency for the engine. From knowledge of the engine type, a rough estimate for this efficiency can be made from the information in Fig. 4.11.3. After replacing Eqs. (4.10.12) and (4.10.13) with Eqs. (4.11.18) and (4.11.19), the direct contribution of a turbojet or turbofan engine to the trim and static stability of an airplane is identical to that of an engine-propeller combination, as described in Sec. 4.10.

4.12. Problems

4.1. An airplane is to weigh 2,000 pounds and its fuselage is to be 15 feet long. The center of gravity is to be 7.5 feet aft of its nose. The airplane is to be a wing-tail configuration. It is to be designed to fly at 15,000 feet at 150 mph. For this first analysis assume that all lifting surfaces are symmetric thin airfoils with infinite aspect ratio. Neglecting changes in weight with the size of the wing and tail, determine the following (there are many possible solutions):

a) The size of the wing and tail required to maintain trimmed cruise.
b) The location of the wing and tail in order to place the airplane's neutral point 0.5 ft aft of the CG.
c) The angle that the wing and the tail make with the upstream flow, taking the fuselage to make an angle of 0° with the upstream flow.

4.2. For the airplane in problem 4.1, assume that the wing has an aspect ratio of 6.0, a taper ratio of 0.4, and no sweep. Also assume that the horizontal tail has an aspect ratio of 4.0, a taper ratio of 0.4, and no sweep. Refine your design to account for downwash.