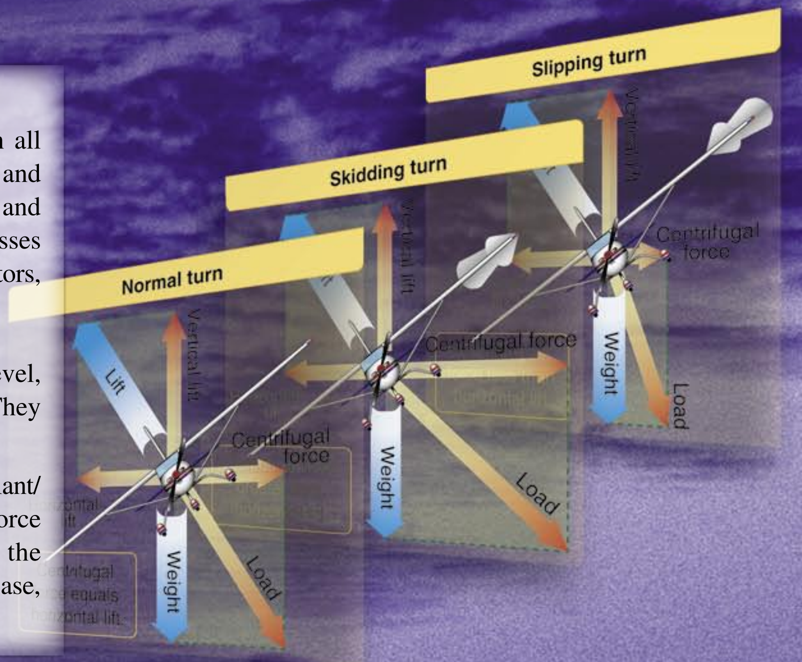
Chapter 4

Aerodynamics of Flight

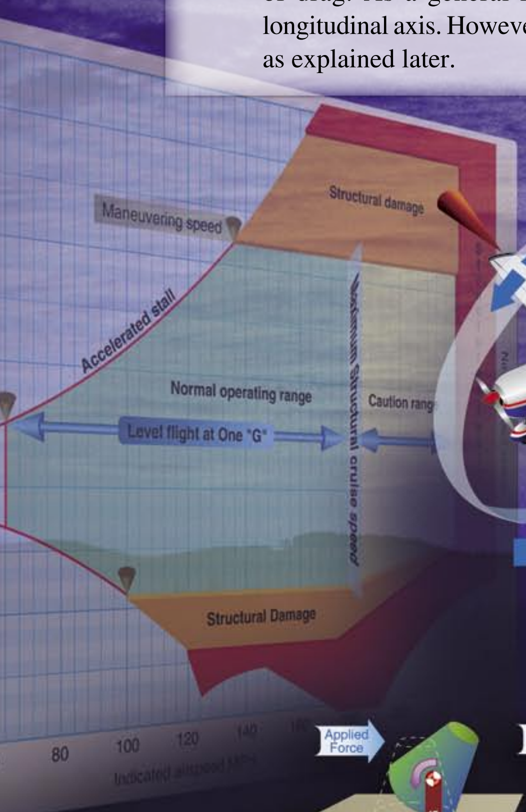


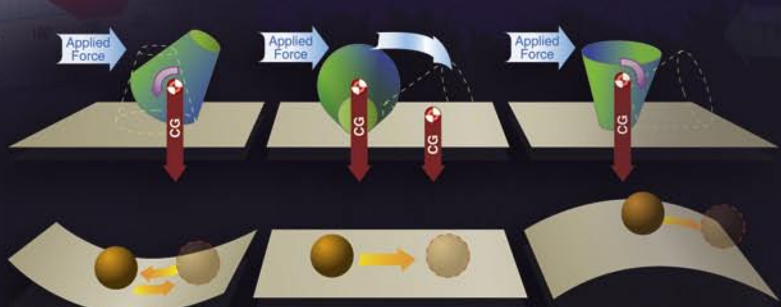
Forces Acting on the Aircraft

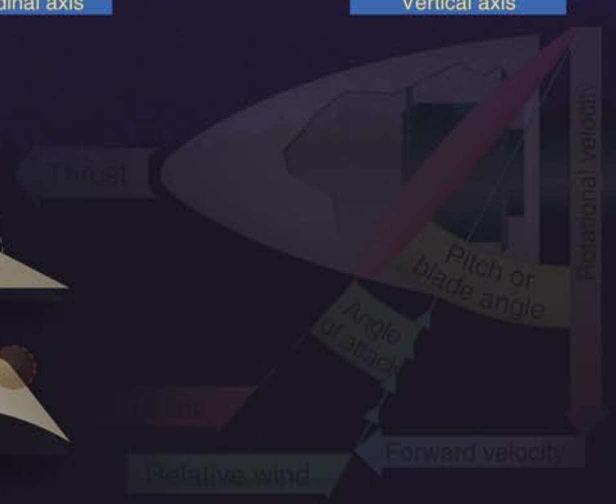
Thrust, drag, lift, and weight are forces that act upon all aircraft in flight. Understanding how these forces work and knowing how to control them with the use of power and flight controls are essential to flight. This chapter discusses the aerodynamics of flight—how design, weight, load factors, and gravity affect an aircraft during flight maneuvers.

The four forces acting on an aircraft in straight-and-level, unaccelerated flight are thrust, drag, lift, and weight. They are defined as follows:

- Thrust—the forward force produced by the powerplant/propeller or rotor. It opposes or overcomes the force of drag. As a general rule, it acts parallel to the longitudinal axis. However, this is not always the case, as explained later.







- Drag—a rearward, retarding force caused by disruption of airflow by the wing, rotor, fuselage, and other protruding objects. Drag opposes thrust, and acts rearward parallel to the relative wind.
- Weight—the combined load of the aircraft itself, the crew, the fuel, and the cargo or baggage. Weight pulls the aircraft downward because of the force of gravity. It opposes lift, and acts vertically downward through the aircraft's center of gravity (CG).
- Lift—opposes the downward force of weight, is produced by the dynamic effect of the air acting on the airfoil, and acts perpendicular to the flightpath through the center of lift.

In steady flight, the sum of these opposing forces is always zero. There can be no unbalanced forces in steady, straight flight based upon Newton's Third Law, which states that for every action or force there is an equal, but opposite, reaction or force. This is true whether flying level or when climbing or descending.

It does not mean the four forces are equal. It means the opposing forces are equal to, and thereby cancel, the effects of each other. In *Figure 4-1* the force vectors of thrust, drag, lift, and weight appear to be equal in value. The usual explanation states (without stipulating that thrust and drag do not equal weight and lift) that thrust equals drag and lift equals weight. Although basically true, this statement can be misleading. It should be understood that in straight, level, unaccelerated flight, it is true that the opposing lift/weight forces are equal. They are also greater than the opposing forces of thrust/drag that are equal only to each other. Therefore, in steady flight:

- The sum of all upward forces (not just lift) equals the sum of all downward forces (not just weight).
- The sum of all forward forces (not just thrust) equals the sum of all backward forces (not just drag).

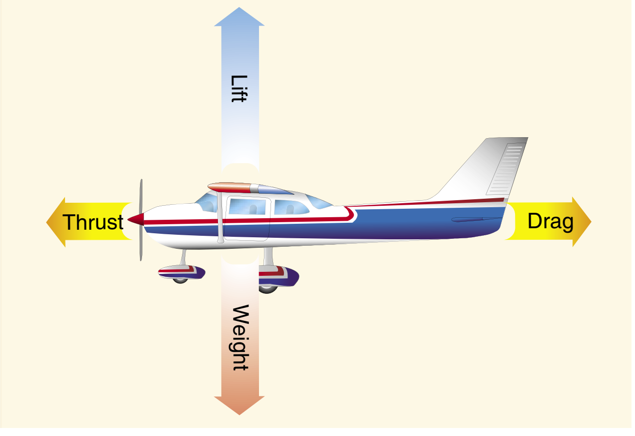


Figure 4-1. *Relationship of forces acting on an airplane.*

This refinement of the old "thrust equals drag; lift equals weight" formula explains that a portion of thrust is directed upward in climbs and acts as if it were lift while a portion of weight is directed backward and acts as if it were drag. *[Figure 4-2]*

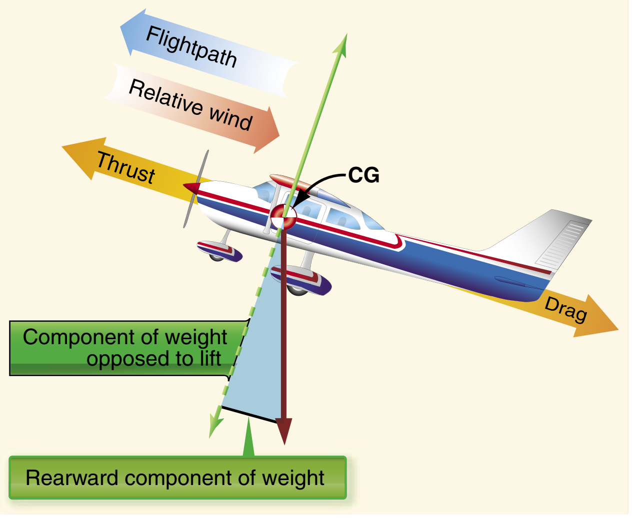


Figure 4-2. *Force vectors during a stabilized climb.*

In glides, a portion of the weight vector is directed forward, and, therefore, acts as thrust. In other words, any time the flightpath of the aircraft is not horizontal, lift, weight, thrust, and drag vectors must each be broken down into two components.

Discussions of the preceding concepts are frequently omitted in aeronautical texts/handbooks/manuals. The reason is not that they are inconsequential, but because the main ideas with respect to the aerodynamic forces acting upon an airplane in flight can be presented in their most essential elements without being involved in the technicalities of the aerodynamicist. In point of fact, considering only level flight, and normal climbs and glides in a steady state, it is still true that lift provided by the wing or rotor is the primary upward force, and weight is the primary downward force.

By using the aerodynamic forces of thrust, drag, lift, and weight, pilots can fly a controlled, safe flight. A more detailed discussion of these forces follows.

Thrust

For an aircraft to move, thrust must be exerted and be greater than drag. The aircraft will continue to move and gain speed until thrust and drag are equal. In order to maintain a constant airspeed, thrust and drag must remain equal, just as lift and weight must be equal to maintain a constant altitude. If in level flight, the engine power is reduced, the thrust is lessened, and the aircraft slows down. As long as the thrust

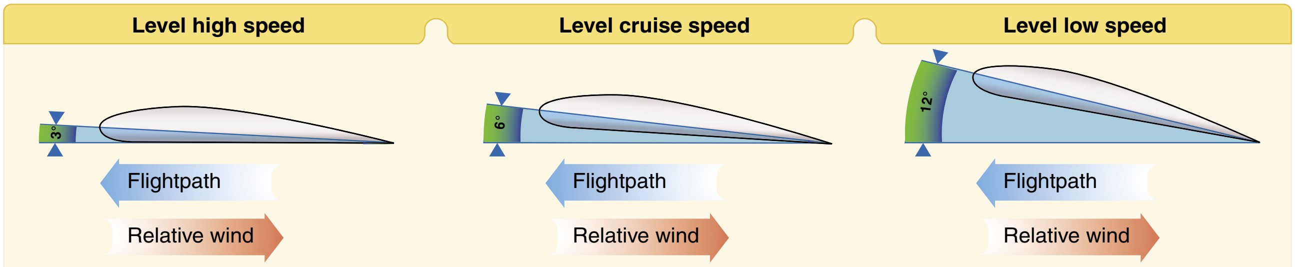


Figure 4-3. *Angle of attack at various speeds.*

is less than the drag, the aircraft continues to decelerate until its airspeed is insufficient to support it in the air.

Likewise, if the engine power is increased, thrust becomes greater than drag and the airspeed increases. As long as the thrust continues to be greater than the drag, the aircraft continues to accelerate. When drag equals thrust, the aircraft flies at a constant airspeed.

Straight-and-level flight may be sustained at a wide range of speeds. The pilot coordinates angle of attack (AOA)—the acute angle between the chord line of the airfoil and the direction of the relative wind—and thrust in all speed regimes if the aircraft is to be held in level flight. Roughly, these regimes can be grouped in three categories: low-speed flight, cruising flight, and high-speed flight.

When the airspeed is low, the AOA must be relatively high if the balance between lift and weight is to be maintained. *[Figure 4-3]* If thrust decreases and airspeed decreases, lift becomes less than weight and the aircraft starts to descend. To maintain level flight, the pilot can increase the AOA an amount which will generate a lift force again equal to the weight of the aircraft. While the aircraft will be flying more slowly, it will still maintain level flight if the pilot has properly coordinated thrust and AOA.

Straight-and-level flight in the slow-speed regime provides some interesting conditions relative to the equilibrium of forces because with the aircraft in a nose-high attitude, there is a vertical component of thrust that helps support it. For one thing, wing loading tends to be less than would be expected. Most pilots are aware that an airplane will stall, other conditions being equal, at a slower speed with the power on than with the power off. (Induced airflow over the wings from the propeller also contributes to this.) However, if analysis is restricted to the four forces as they are usually defined during slow-speed flight the thrust is equal to drag, and lift is equal to weight.

During straight-and-level flight when thrust is increased and the airspeed increases, the AOA must be decreased. That is, if changes have been coordinated, the aircraft will remain in level flight, but at a higher speed when the proper relationship between thrust and AOA is established.

If the AOA were not coordinated (decreased) with an increase of thrust, the aircraft would climb. But decreasing the AOA modifies the lift, keeping it equal to the weight, and the aircraft remains in level flight. Level flight at even slightly negative AOA is possible at very high speed. It is evident then, that level flight can be performed with any AOA between stalling angle and the relatively small negative angles found at high speed.

Some aircraft have the ability to change the direction of the thrust rather than changing the AOA. This is accomplished either by pivoting the engines or by vectoring the exhaust gases. *[Figure 4-4]*

Figure 4-4. *Some aircraft have the ability to change the direction of thrust.*

Drag

Drag is the force that resists movement of an aircraft through the air. There are two basic types: parasite drag and induced drag. The first is called parasite because it in no way functions to aid flight, while the second, induced drag, is a result of an airfoil developing lift.

Parasite Drag

Parasite drag is comprised of all the forces that work to slow an aircraft's movement. As the term parasite implies, it is the drag that is not associated with the production of lift. This includes the displacement of the air by the aircraft, turbulence generated in the airstream, or a hindrance of air moving over the surface of the aircraft and airfoil. There are three types of parasite drag: form drag, interference drag, and skin friction.

Form Drag

Form drag is the portion of parasite drag generated by the aircraft due to its shape and airflow around it. Examples include the engine cowlings, antennas, and the aerodynamic shape of other components. When the air has to separate to move around a moving aircraft and its components, it eventually rejoins after passing the body. How quickly and smoothly it rejoins is representative of the resistance that it creates which requires additional force to overcome. *[Figure 4-5]*

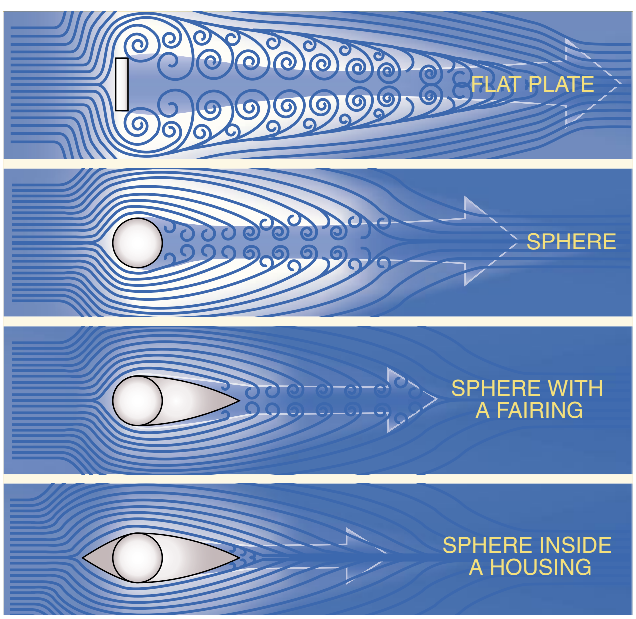


Figure 4-5. *Form drag.*

Notice how the flat plate in *Figure 4-5* causes the air to swirl around the edges until it eventually rejoins downstream. Form drag is the easiest to reduce when designing an aircraft. The solution is to streamline as many of the parts as possible.

Interference Drag

Interference drag comes from the intersection of airstreams that creates eddy currents, turbulence, or restricts smooth airflow. For example, the intersection of the wing and the fuselage at the wing root has significant interference drag. Air flowing around the fuselage collides with air flowing over the wing, merging into a current of air different from the two original currents. The most interference drag is observed when two surfaces meet at perpendicular angles. Fairings are used to reduce this tendency. If a jet fighter carries two identical wing tanks, the overall drag is greater than the sum of the individual tanks because both of these create and generate interference drag. Fairings and distance between lifting surfaces and external components (such as radar antennas hung from wings) reduce interference drag. *[Figure 4-6]*

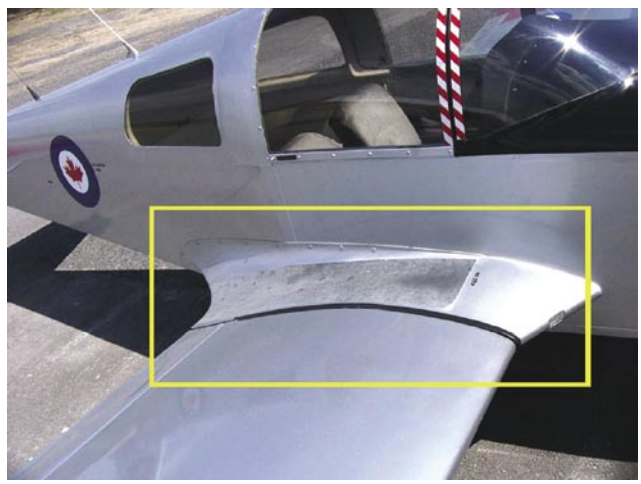


Figure 4-6. *A wing root can cause interference drag.*

Skin Friction Drag

Skin friction drag is the aerodynamic resistance due to the contact of moving air with the surface of an aircraft. Every surface, no matter how apparently smooth, has a rough, ragged surface when viewed under a microscope. The air molecules, which come in direct contact with the surface of the wing, are virtually motionless. Each layer of molecules above the surface moves slightly faster until the molecules are moving at the velocity of the air moving around the aircraft. This speed is called the free-stream velocity. The area between the wing and the free-stream velocity level is about as wide as a playing card and is called the boundary layer. At the top of the boundary layer, the molecules increase velocity and move at the same speed as the molecules outside the boundary layer. The actual speed at which the molecules move depends upon the shape of the wing, the viscosity (stickiness) of the air through which the wing or airfoil is moving, and its compressibility (how much it can be compacted).

The airflow outside of the boundary layer reacts to the shape of the edge of the boundary layer just as it would to the physical surface of an object. The boundary layer gives any object an "effective" shape that is usually slightly different from the physical shape. The boundary layer may also separate from the body, thus creating an effective shape much different from the physical shape of the object. This change in the physical shape of the boundary layer causes a dramatic decrease in lift and an increase in drag. When this happens, the airfoil has stalled.

In order to reduce the effect of skin friction drag, aircraft designers utilize flush mount rivets and remove any irregularities which may protrude above the wing surface. In addition, a smooth and glossy finish aids in transition of air across the surface of the wing. Since dirt on an aircraft disrupts the free flow of air and increases drag, keep the surfaces of an aircraft clean and waxed.

Induced Drag

The second basic type of drag is induced drag. It is an established physical fact that no system that does work in the mechanical sense can be 100 percent efficient. This means that whatever the nature of the system, the required work is obtained at the expense of certain additional work that is dissipated or lost in the system. The more efficient the system, the smaller this loss.

In level flight the aerodynamic properties of a wing or rotor produce a required lift, but this can be obtained only at the expense of a certain penalty. The name given to this penalty is induced drag. Induced drag is inherent whenever an airfoil is producing lift and, in fact, this type of drag is inseparable from the production of lift. Consequently, it is always present if lift is produced.

An airfoil (wing or rotor blade) produces the lift force by making use of the energy of the free airstream. Whenever an airfoil is producing lift, the pressure on the lower surface of it is greater than that on the upper surface (Bernoulli's Principle). As a result, the air tends to flow from the high pressure area below the tip upward to the low pressure area on the upper surface. In the vicinity of the tips, there is a tendency for these pressures to equalize, resulting in a lateral flow outward from the underside to the upper surface. This lateral flow imparts a rotational velocity to the air at the tips, creating vortices, which trail behind the airfoil.

When the aircraft is viewed from the tail, these vortices circulate counterclockwise about the right tip and clockwise about the left tip. *[Figure 4-7]* Bearing in mind the direction of rotation of these vortices, it can be seen that they induce an upward flow of air beyond the tip, and a downwash flow behind the wing's trailing edge. This induced downwash has nothing in common with the downwash that is necessary to produce lift. It is, in fact, the source of induced drag. The greater the size and strength of the vortices and consequent downwash component on the net airflow over the airfoil, the greater the induced drag effect becomes. This downwash over the top of the airfoil at the tip has the same effect as bending the lift vector rearward; therefore, the lift is slightly aft of perpendicular to the relative wind, creating a rearward lift component. This is induced drag.

Figure 4-7. *Wingtip vortex from a crop duster.*

In order to create a greater negative pressure on the top of an airfoil, the airfoil can be inclined to a higher AOA. If the AOA of a symmetrical airfoil were zero, there would be no pressure differential, and consequently, no downwash component and no induced drag. In any case, as AOA increases, induced drag increases proportionally. To state this another way—the lower the airspeed the greater the AOA required to produce lift equal to the aircraft's weight and, therefore, the greater induced drag. The amount of induced drag varies inversely with the square of the airspeed.

Conversely, parasite drag increases as the square of the airspeed. Thus, as airspeed decreases to near the stalling speed, the total drag becomes greater, due mainly to the sharp rise in induced drag. Similarly, as the airspeed reaches the terminal velocity of the aircraft, the total drag again increases rapidly, due to the sharp increase of parasite drag. As seen in *Figure 4-8*, at some given airspeed, total drag is at its minimum amount. In figuring the maximum endurance and range of aircraft, the power required to overcome drag is at a minimum if drag is at a minimum.

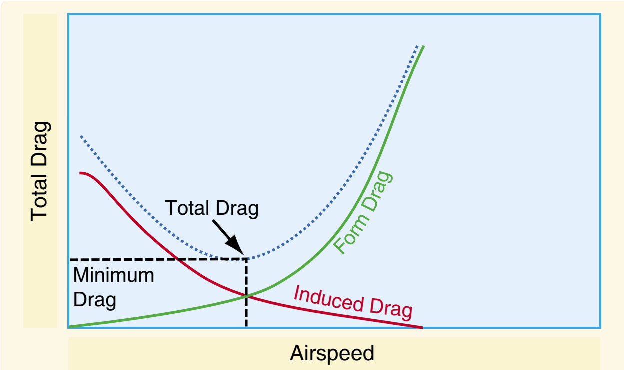


Figure 4-8. *Drag versus speed.*

Lift/Drag Ratio

Drag is the price paid to obtain lift. The lift to drag ratio (L/D) is the amount of lift generated by a wing or airfoil compared to its drag. A ratio of L/D indicates airfoil efficiency. Aircraft with higher L/D ratios are more efficient than those with lower L/D ratios. In unaccelerated flight with the lift and drag data steady, the proportions of the C_L and coefficient of drag (C_D) can be calculated for specific AOA. *[Figure 4-9]*

The L/D ratio is determined by dividing the C_L by the C_D, which is the same as dividing the lift equation by the drag equation. All terms except coefficients cancel out.

L = Lift in pounds

D = Drag

Where L is the lift force in pounds, C_L is the lift coefficient, ρ is density expressed in slugs per cubic feet, V is velocity in feet per second, q is dynamic pressure per square feet, and S is the wing area in square feet.

$C_{D=}$ Ratio of drag pressure to dynamic pressure. Typically at low angles of attack, the drag coefficient is low and small changes in angle of attack create only slight changes in the drag coefficient. At high angles of attack, small changes in the angle of attack cause significant changes in drag.

$$L = \frac{C_L \cdot \rho \cdot V^2 \cdot S}{2}$$

$$D = \frac{C_D \cdot \rho \cdot V^2 \cdot S}{2}$$

The above formulas represent the coefficient of lift (C_L) and the coefficient of drag (C_D) respectively. The shape of an airfoil and other life producing devices (i.e., flaps) effect the production of lift and alter with changes in the AOA. The lift/drag ratio is used to express the relation between lift and drag and is determined by dividing the lift coefficient by the drag coefficient, C_L/C_D.

Notice in *Figure 4-9* that the lift curve (red) reaches its maximum for this particular wing section at 20° AOA, and then rapidly decreases. 15° AOA is therefore the stalling angle. The drag curve (yellow) increases very rapidly from 14° AOA and completely overcomes the lift curve at 21° AOA. The lift/drag ratio (green) reaches its maximum at 6° AOA, meaning that at this angle, the most lift is obtained for the least amount of drag.

Note that the maximum lift/drag ratio (L/D_{MAX}) occurs at one specific C_L and AOA. If the aircraft is operated in steady flight at L/D_{MAX}, the total drag is at a minimum. Any AOA lower or higher than that for L/D_{MAX} reduces the L/D and

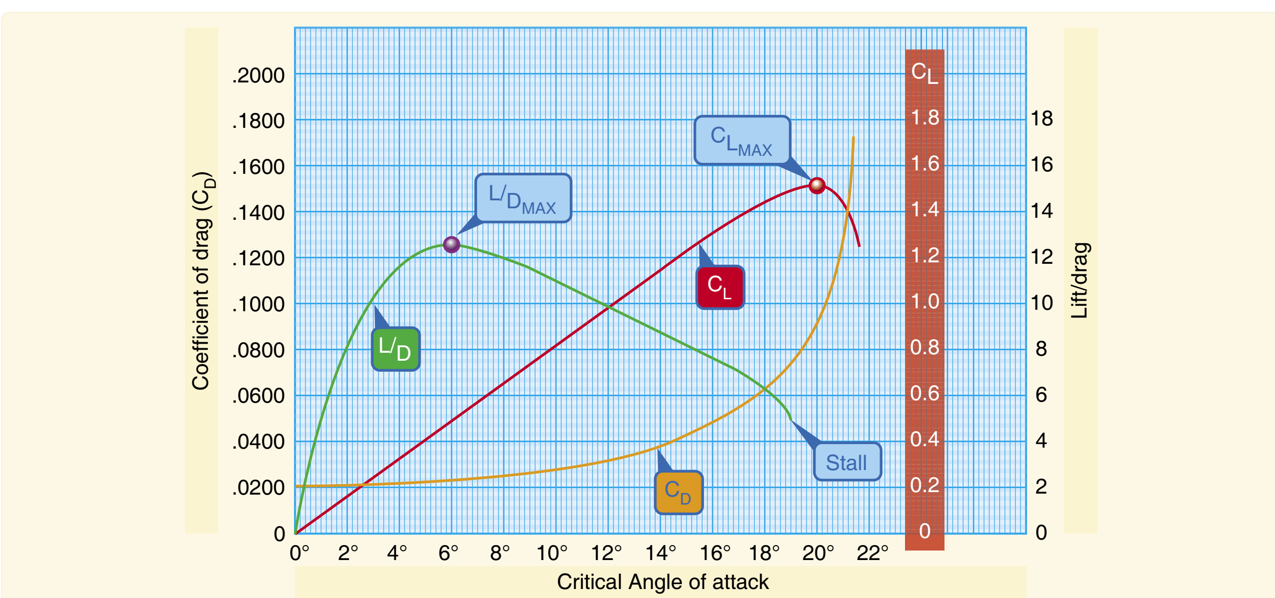


Figure 4-9. *Lift coefficients at various angles of attack.*

consequently increases the total drag for a given aircraft's lift. *Figure 4-8* depicts the L/D_{MAX} by the lowest portion of the orange line labeled "total drag." The configuration of an aircraft has a great effect on the L/D.

Weight

Gravity is the pulling force that tends to draw all bodies to the center of the earth. The CG may be considered as a point at which all the weight of the aircraft is concentrated. If the aircraft were supported at its exact CG, it would balance in any attitude. It will be noted that CG is of major importance in an aircraft, for its position has a great bearing upon stability.

The location of the CG is determined by the general design of each particular aircraft. The designers determine how far the center of pressure (CP) will travel. They then fix the CG forward of the center of pressure for the corresponding flight speed in order to provide an adequate restoring moment to retain flight equilibrium.

Weight has a definite relationship to lift. This relationship is simple, but important in understanding the aerodynamics of flying. Lift is the upward force on the wing acting perpendicular to the relative wind. Lift is required to counteract the aircraft's weight (which is caused by the force of gravity acting on the mass of the aircraft). This weight (gravity) force acts downward through the airplane's CG. In stabilized level flight, when the lift force is equal to the weight force, the aircraft is in a state of equilibrium and neither gains nor loses altitude. If lift becomes less than weight, the aircraft loses altitude. When lift is greater than weight, the aircraft gains altitude.

Lift

The pilot can control the lift. Any time the control yoke or stick is moved fore or aft, the AOA is changed. As the AOA increases, lift increases (all other factors being equal). When the aircraft reaches the maximum AOA, lift begins to diminish rapidly. This is the stalling AOA, known as $C_{L\text{-}MAX}$ critical AOA. Examine *Figure 4-9,* noting how the C_L increases until the critical AOA is reached, then decreases rapidly with any further increase in the AOA.

Before proceeding further with the topic of lift and how it can be controlled, velocity must be interjected. The shape of the wing or rotor cannot be effective unless it continually keeps "attacking" new air. If an aircraft is to keep flying, the lift-producing airfoil must keep moving. In a helicopter or gyro-plane this is accomplished by the rotation of the rotor blades. For other types of aircraft such as airplanes, weight-shift control, or gliders, air must be moving across the lifting surface. This is accomplished by the forward speed of the aircraft. Lift is proportional to the square of the aircraft's velocity. For example, an airplane traveling at 200 knots has four times the lift as the same airplane traveling at 100 knots, if the AOA and other factors remain constant.

Actually, an aircraft could not continue to travel in level flight at a constant altitude and maintain the same AOA if the velocity is increased. The lift would increase and the aircraft would climb as a result of the increased lift force. Therefore, to maintain the lift and weight forces in balance, and to keep the aircraft straight and level (not accelerating upward) in a state of equilibrium, as velocity is increased, lift must be decreased. This is normally accomplished by reducing the AOA by lowering the nose. Conversely, as the aircraft is slowed, the decreasing velocity requires increasing the AOA to maintain lift sufficient to maintain flight. There is, of course, a limit to how far the AOA can be increased, if a stall is to be avoided.

All other factors being constant, for every AOA there is a corresponding airspeed required to maintain altitude in steady, unaccelerated flight (true only if maintaining "level flight"). Since an airfoil always stalls at the same AOA, if increasing weight, lift must also be increased. The only method of increasing lift is by increasing velocity if the AOA is held constant just short of the "critical," or stalling, AOA.

Lift and drag also vary directly with the density of the air. Density is affected by several factors: pressure, temperature, and humidity. At an altitude of 18,000 feet, the density of the air has one-half the density of air at sea level. In order to maintain its lift at a higher altitude, an aircraft must fly at a greater true airspeed for any given AOA.

Warm air is less dense than cool air, and moist air is less dense than dry air. Thus, on a hot humid day, an aircraft must be flown at a greater true airspeed for any given AOA than on a cool, dry day.

If the density factor is decreased and the total lift must equal the total weight to remain in flight, it follows that one of the other factors must be increased. The factor usually increased is the airspeed or the AOA, because these are controlled directly by the pilot.

Lift varies directly with the wing area, provided there is no change in the wing's planform. If the wings have the same proportion and airfoil sections, a wing with a planform area of 200 square feet lifts twice as much at the same AOA as a wing with an area of 100 square feet.

Two major aerodynamic factors from the pilot's viewpoint are lift and velocity because they can be controlled readily and accurately. Of course, the pilot can also control density by adjusting the altitude and can control wing area if the aircraft happens to have flaps of the type that enlarge wing area. However, for most situations, the pilot controls lift and velocity to maneuver an aircraft. For instance, in straight-and-level flight, cruising along at a constant altitude, altitude is maintained by adjusting lift to match the aircraft's velocity or cruise airspeed, while maintaining a state of equilibrium in which lift equals weight. In an approach to landing, when the pilot wishes to land as slowly as practical, it is necessary to increase lift to near maximum to maintain lift equal to the weight of the aircraft.

Wingtip Vortices

Formation of Vortices

The action of the airfoil that gives an aircraft lift also causes induced drag. When an airfoil is flown at a positive AOA, a pressure differential exists between the upper and lower surfaces of the airfoil. The pressure above the wing is less than atmospheric pressure and the pressure below the wing is equal to or greater than atmospheric pressure. Since air always moves from high pressure toward low pressure, and the path of least resistance is toward the airfoil's tips, there is a spanwise movement of air from the bottom of the airfoil outward from the fuselage around the tips. This flow of air results in "spillage" over the tips, thereby setting up a whirlpool of air called a "vortex." *[Figure 4-10]*

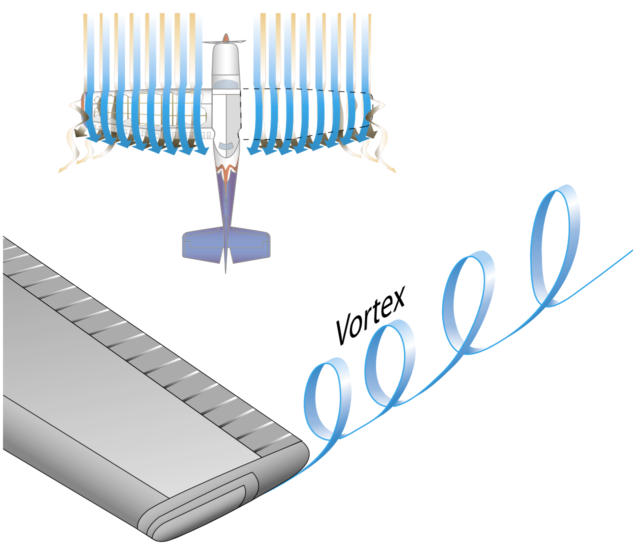


Figure 4-10. *Wingtip vortices.*

At the same time, the air on the upper surface has a tendency to flow in toward the fuselage and off the trailing edge. This air current forms a similar vortex at the inboard portion of the trailing edge of the airfoil, but because the fuselage limits the inward flow, the vortex is insignificant. Consequently, the deviation in flow direction is greatest at the outer tips where the unrestricted lateral flow is the strongest.

As the air curls upward around the tip, it combines with the wash to form a fast-spinning trailing vortex. These vortices increase drag because of energy spent in producing the turbulence. Whenever an airfoil is producing lift, induced drag occurs, and wingtip vortices are created.

Just as lift increases with an increase in AOA, induced drag also increases. This occurs because as the AOA is increased, there is a greater pressure difference between the top and bottom of the airfoil, and a greater lateral flow of air; consequently, this causes more violent vortices to be set up, resulting in more turbulence and more induced drag.

In *Figure 4-10,* it is easy to see the formation of wingtip vortices. The intensity or strength of the vortices is directly proportional to the weight of the aircraft and inversely proportional to the wingspan and speed of the aircraft. The heavier and slower the aircraft, the greater the AOA and the stronger the wingtip vortices. Thus, an aircraft will create wingtip vortices with maximum strength occurring during the takeoff, climb, and landing phases of flight. These vortices lead to a particularly dangerous hazard to flight, wake turbulence.

Avoiding Wake Turbulence

Wingtip vortices are greatest when the generating aircraft is "heavy, clean, and slow." This condition is most commonly encountered during approaches or departures because an aircraft's AOA is at the highest to produce the lift necessary to land or take off. To minimize the chances of flying through an aircraft's wake turbulence:

- Avoid flying through another aircraft's flightpath.
- Rotate prior to the point at which the preceding aircraft rotated, when taking off behind another aircraft.
- Avoid following another aircraft on a similar flightpath at an altitude within 1,000 feet. *[Figure 4-11]*
- Approach the runway above a preceding aircraft's path when landing behind another aircraft, and touch down after the point at which the other aircraft wheels contacted the runway. *[Figure 4-12]*

A hovering helicopter generates a down wash from its main rotor(s) similar to the vortices of an airplane. Pilots of small aircraft should avoid a hovering helicopter by at least three rotor disc diameters to avoid the effects of this down wash. In forward flight this energy is transformed into a pair of strong, high-speed trailing vortices similar to wing-tip vortices of larger fixed-wing aircraft. Helicopter vortices should be avoided because helicopter forward flight airspeeds are often very slow and can generate exceptionally strong wake turbulence.

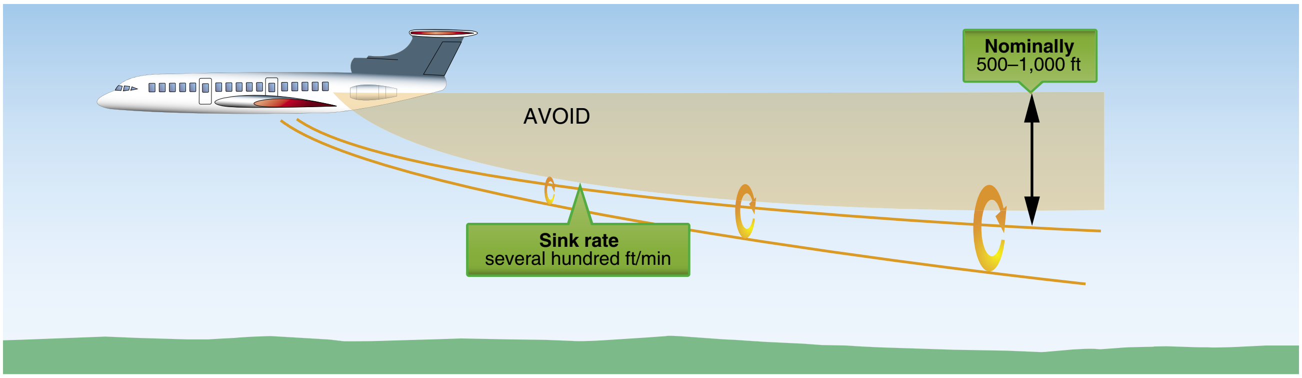


Figure 4-11. *Avoid following another aircraft at an altitude within 1,000 feet.*

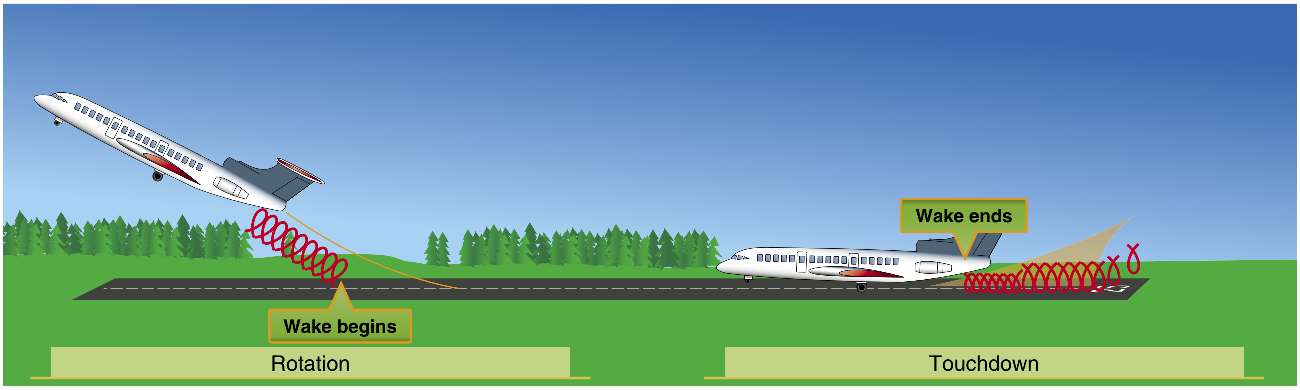


Figure 4-12. *Avoid turbulence from another aircraft.*

Wind is an important factor in avoiding wake turbulence because wingtip vortices drift with the wind at the speed of the wind. For example, a wind speed of 10 knots causes the vortices to drift at about 1,000 feet in a minute in the wind direction. When following another aircraft, a pilot should consider wind speed and direction when selecting an intended takeoff or landing point. If a pilot is unsure of the other aircraft's takeoff or landing point, approximately 3 minutes provides a margin of safety that allows wake turbulence dissipation. For more information on wake turbulence, see Advisory Circular 90-23.

Ground Effect

It is possible to fly an aircraft just clear of the ground (or water) at a slightly slower airspeed than that required to sustain level flight at higher altitudes. This is the result of a phenomenon better known of than understood even by some experienced pilots.

When an aircraft in flight comes within several feet of the surface, ground or water, a change occurs in the three-dimensional flow pattern around the aircraft because the vertical component of the airflow around the wing is restricted by the surface. This alters the wing's upwash, downwash, and wingtip vortices. *[Figure 4-13]* Ground effect, then, is due to the interference of the ground (or water) surface with the airflow patterns about the aircraft in flight.

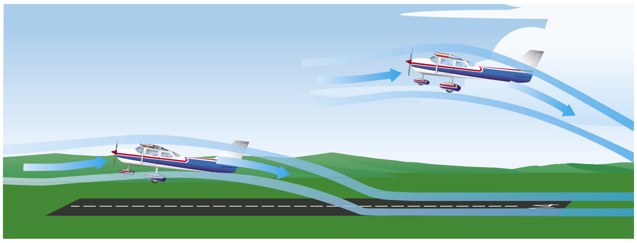


Figure 4-13. *Ground effect changes airflow.*

While the aerodynamic characteristics of the tail surfaces and the fuselage are altered by ground effect, the principal effects due to proximity of the ground are the changes in the aerodynamic characteristics of the wing. As the wing encounters ground effect and is maintained at a constant lift coefficient, there is consequent reduction in the upwash, downwash, and wingtip vortices.

Induced drag is a result of the airfoil's work of sustaining the aircraft, and a wing or rotor lifts the aircraft simply by accelerating a mass of air downward. It is true that reduced pressure on top of an airfoil is essential to lift, but that is only one of the things contributing to the overall effect of pushing an air mass downward. The more downwash there is, the harder the wing pushes the mass of air down. At high angles of attack, the amount of induced drag is high; since this corresponds to lower airspeeds in actual flight, it can be said that induced drag predominates at low speed.

However, the reduction of the wingtip vortices due to ground effect alters the spanwise lift distribution and reduces the induced AOA and induced drag. Therefore, the wing will require a lower AOA in ground effect to produce the same C_L. If a constant AOA is maintained, an increase in C_L results. *[Figure 4-14]*

Ground effect also alters the thrust required versus velocity. Since induced drag predominates at low speeds, the reduction of induced drag due to ground effect will cause the most significant reduction of thrust required (parasite plus induced drag) at low speeds.

The reduction in induced flow due to ground effect causes a significant reduction in induced drag but causes no direct effect on parasite drag. As a result of the reduction in induced drag, the thrust required at low speeds will be reduced. Due to the change in upwash, downwash, and wingtip vortices, there may be a change in position (installation) error of the airspeed system, associated with ground effect. In the majority of cases, ground effect will cause an increase in the local pressure at the static source and produce a lower indication of airspeed and altitude. Thus, an aircraft may be airborne at an indicated airspeed less than that normally required.

In order for ground effect to be of significant magnitude, the wing must be quite close to the ground. One of the direct results of ground effect is the variation of induced drag with wing height above the ground at a constant C_L. When the wing is at a height equal to its span, the reduction in induced drag is only 1.4 percent. However, when the wing is at a height equal to one-fourth its span, the reduction in induced drag is 23.5 percent and, when the wing is at a height equal to one-tenth its span, the reduction in induced drag is 47.6 percent. Thus, a large reduction in induced drag will take place only when the wing is very close to the ground. Because of this variation, ground effect is most usually recognized during the liftoff for takeoff or just prior to touchdown when landing.

During the takeoff phase of flight, ground effect produces some important relationships. An aircraft leaving ground effect after takeoff encounters just the reverse of an aircraft entering ground effect during landing; i.e., the aircraft leaving ground effect will:

- Require an increase in AOA to maintain the same C_L.
- Experience an increase in induced drag and thrust required.
- Experience a decrease in stability and a nose-up change in moment.
- Experience a reduction in static source pressure and increase in indicated airspeed.

Ground effect must be considered during takeoffs and landings. For example, if a pilot fails to understand the relationship between the aircraft and ground effect during takeoff, a hazardous situation is possible because the recommended takeoff speed may not be achieved. Due to the reduced drag in ground effect, the aircraft may seem capable of takeoff well below the recommended speed. As the aircraft rises out of ground effect with a deficiency of speed, the greater induced drag may result in marginal initial climb performance. In extreme conditions, such as high gross weight, high density altitude, and high temperature, a deficiency of airspeed during takeoff may permit the aircraft to become airborne but be incapable of sustaining flight out of ground effect. In this case, the aircraft may become airborne initially with a deficiency of speed, and then settle back to the runway.

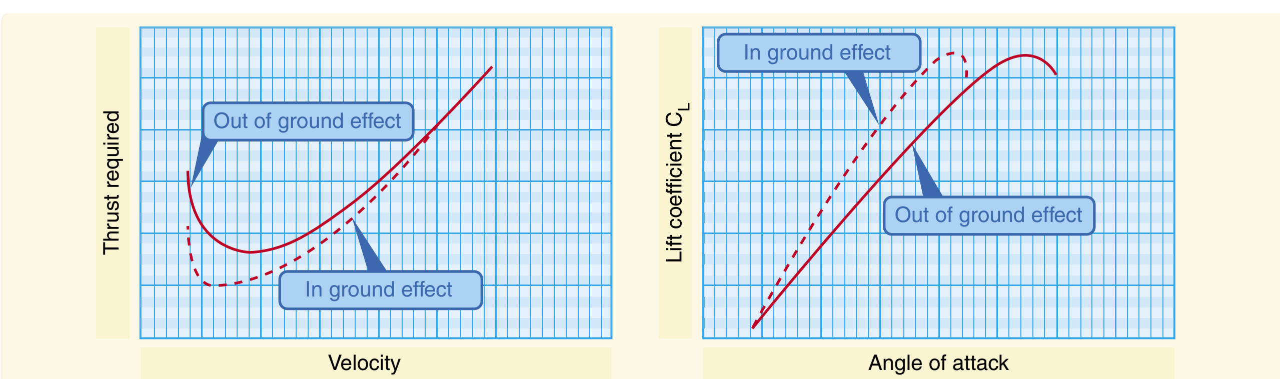


Figure 4-14. *Ground effect changes drag and lift.*

A pilot should not attempt to force an aircraft to become airborne with a deficiency of speed. The manufacturer's recommended takeoff speed is necessary to provide adequate initial climb performance. It is also important that a definite climb be established before a pilot retracts the landing gear or flaps. Never retract the landing gear or flaps prior to establishing a positive rate of climb, and only after achieving a safe altitude.

If, during the landing phase of flight, the aircraft is brought into ground effect with a constant AOA, the aircraft experiences an increase in C_L and a reduction in the thrust required, and a "floating" effect may occur. Because of the reduced drag and power-off deceleration in ground effect, any excess speed at the point of flare may incur a considerable "float" distance. As the aircraft nears the point of touchdown, ground effect is most realized at altitudes less than the wingspan. During the final phases of the approach as the aircraft nears the ground, a reduced power setting is necessary or the reduced thrust required would allow the aircraft to climb above the desired glidepath (GP).

Axes of an Aircraft

The axes of an aircraft are three imaginary lines that pass through an aircraft's CG. The axes can be considered as imaginary axles around which the aircraft turns. The three axes pass through the CG at 90° angles to each other. The axis from nose to tail is the longitudinal axis, the axis that passes from wingtip to wingtip is the lateral axis, and the axis that passes vertically through the CG is the vertical axis. Whenever an aircraft changes its flight attitude or position in flight, it rotates about one or more of the three axes. *[Figure 4-15]*

The aircraft's motion about its longitudinal axis resembles the roll of a ship from side to side. In fact, the names used to describe the motion about an aircraft's three axes were originally nautical terms. They have been adapted to aeronautical terminology due to the similarity of motion of aircraft and seagoing ships. The motion about the aircraft's longitudinal axis is "roll," the motion about its lateral axis is "pitch," and the motion about its vertical axis is "yaw." Yaw is the horizontal (left and right) movement of the aircraft's nose.

The three motions of the conventional airplane (roll, pitch, and yaw) are controlled by three control surfaces. Roll is controlled by the ailerons; pitch is controlled by the elevators; yaw is controlled by the rudder. The use of these controls is explained in Chapter 5, Flight Controls. Other types of aircraft may utilize different methods of controlling the movements about the various axes.

For example, weight-shift control aircraft control two axes, roll and pitch, using an "A" frame suspended from the flexible wing attached to a three-wheeled carriage. These aircraft are controlled by moving a horizontal bar (called a control bar) in roughly the same way hang glider pilots fly. *[Figure 4-16]* They are termed weight-shift control aircraft

Figure 4-16. *A weight-shift control aircraft.*

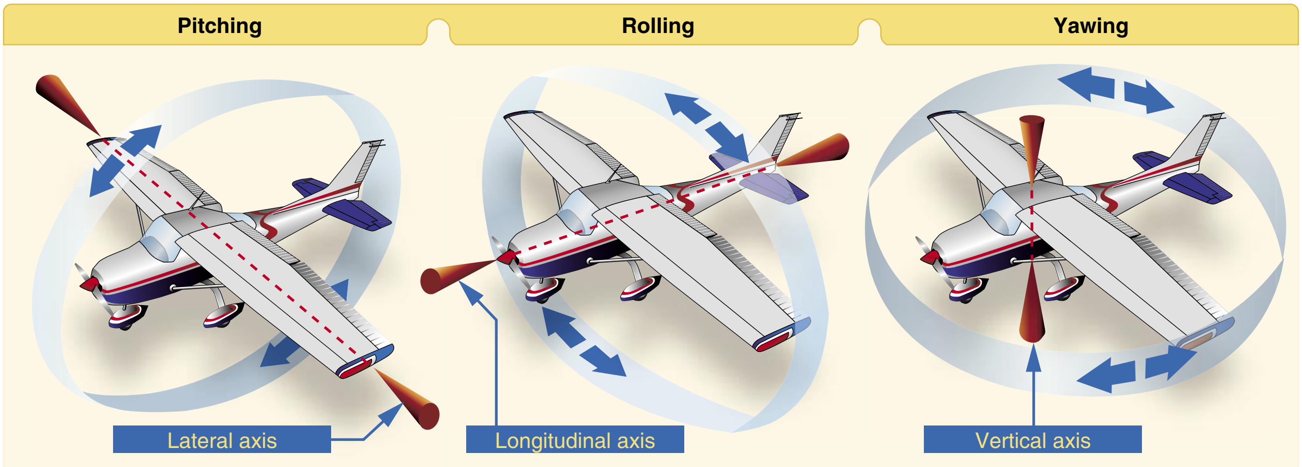


Figure 4-15. *Axes of an airplane.*

because the pilot controls the aircraft by shifting the CG. For more information on weight-shift control aircraft, see the Federal Aviation Administration (FAA) Weight-Shift Control Flying Handbook, FAA-H-8083-5. In the case of powered parachutes, aircraft control is accomplished by altering the airfoil via steering lines.

A powered parachute wing is a parachute that has a cambered upper surface and a flatter under surface. The two surfaces are separated by ribs that act as cells, which open to the airflow at the leading edge and have internal ports to allow lateral airflow. The principle at work holds that the cell pressure is greater than the outside pressure, thereby forming a wing that maintains its airfoil shape in flight. The pilot and passenger sit in tandem in front of the engine which is located at the rear of a vehicle. The airframe is attached to the parachute via two attachment points and lines. Control is accomplished by both power and the changing of the airfoil via the control lines. *[Figure 4-17]*

Figure 4-17. *A powered parachute.*

Moment and Moment Arm

A study of physics shows that a body that is free to rotate will always turn about its CG. In aerodynamic terms, the mathematical measure of an aircraft's tendency to rotate about its CG is called a "moment." A moment is said to be equal to the product of the force applied and the distance at which the force is applied. (A moment arm is the distance from a datum [reference point or line] to the applied force.) For aircraft weight and balance computations, "moments" are expressed in terms of the distance of the arm times the aircraft's weight, or simply, inch-pounds.

Aircraft designers locate the fore and aft position of the aircraft's CG as nearly as possible to the 20 percent point of the mean aerodynamic chord (MAC). If the thrust line is designed to pass horizontally through the CG, it will not cause the aircraft to pitch when power is changed, and there will be no difference in moment due to thrust for a power-on or power-off condition of flight. Although designers have some control over the location of the drag forces, they are not always able to make the resultant drag forces pass through the CG of the aircraft. However, the one item over which they have the greatest control is the size and location of the tail. The objective is to make the moments (due to thrust, drag, and lift) as small as possible and, by proper location of the tail, to provide the means of balancing an aircraft longitudinally for any condition of flight.

The pilot has no direct control over the location of forces acting on the aircraft in flight, except for controlling the center of lift by changing the AOA. Such a change, however, immediately involves changes in other forces. Therefore, the pilot cannot independently change the location of one force without changing the effect of others. For example, a change in airspeed involves a change in lift, as well as a change in drag and a change in the up or down force on the tail. As forces such as turbulence and gusts act to displace the aircraft, the pilot reacts by providing opposing control forces to counteract this displacement.

Some aircraft are subject to changes in the location of the CG with variations of load. Trimming devices are used to counteract the forces set up by fuel burnoff, and loading or off-loading of passengers or cargo. Elevator trim tabs and adjustable horizontal stabilizers comprise the most common devices provided to the pilot for trimming for load variations. Over the wide ranges of balance during flight in large aircraft, the force which the pilot has to exert on the controls would become excessive and fatiguing if means of trimming were not provided.

Aircraft Design Characteristics

Each aircraft handles somewhat differently because each resists or responds to control pressures in its own way. For example, a training aircraft is quick to respond to control applications, while a transport aircraft feels heavy on the controls and responds to control pressures more slowly. These features can be designed into an aircraft to facilitate the particular purpose of the aircraft by considering certain stability and maneuvering requirements. The following discussion summarizes the more important aspects of an aircraft's stability, maneuverability and controllability qualities; how they are analyzed; and their relationship to various flight conditions.

Stability

Stability is the inherent quality of an aircraft to correct for conditions that may disturb its equilibrium, and to return to or to continue on the original flightpath. It is primarily an aircraft design characteristic. The flightpaths and attitudes an aircraft flies are limited by the aerodynamic characteristics of the aircraft, its propulsion system, and its structural strength. These limitations indicate the maximum performance and maneuverability of the aircraft. If the aircraft is to provide maximum utility, it must be safely controllable to the full extent of these limits without exceeding the pilot's strength or requiring exceptional flying ability. If an aircraft is to fly straight and steady along any arbitrary flightpath, the forces acting on it must be in static equilibrium. The reaction of any body when its equilibrium is disturbed is referred to as stability. The two types of stability are static and dynamic.

Static Stability

Static stability refers to the initial tendency, or direction of movement, back to equilibrium. In aviation, it refers to the aircraft's initial response when disturbed from a given AOA, slip, or bank.

- Positive static stability—the initial tendency of the aircraft to return to the original state of equilibrium after being disturbed *[Figure 4-18]*
- Neutral static stability—the initial tendency of the aircraft to remain in a new condition after its equilibrium has been disturbed *[Figure 4-18]*
- Negative static stability—the initial tendency of the aircraft to continue away from the original state of equilibrium after being disturbed *[Figure 4-18]*

Dynamic Stability

Static stability has been defined as the initial tendency to return to equilibrium that the aircraft displays after being disturbed from its trimmed condition. Occasionally, the initial tendency is different or opposite from the overall tendency, so a distinction must be made between the two.

Dynamic stability refers to the aircraft response over time when disturbed from a given AOA, slip, or bank. This type of stability also has three subtypes: *[Figure 4-19]*

- Positive dynamic stability—over time, the motion of the displaced object decreases in amplitude and, because it is positive, the object displaced returns toward the equilibrium state.
- Neutral dynamic stability—once displaced, the displaced object neither decreases nor increases in amplitude. A worn automobile shock absorber exhibits this tendency.
- Negative dynamic stability—over time, the motion of the displaced object increases and becomes more divergent.

Stability in an aircraft affects two areas significantly:

- Maneuverability—the quality of an aircraft that permits it to be maneuvered easily and to withstand the stresses imposed by maneuvers. It is governed by the aircraft's weight, inertia, size and location of flight controls, structural strength, and powerplant. It too is an aircraft design characteristic.
- Controllability—the capability of an aircraft to respond to the pilot's control, especially with regard to flightpath and attitude. It is the quality of the aircraft's response to the pilot's control application when maneuvering the aircraft, regardless of its stability characteristics.

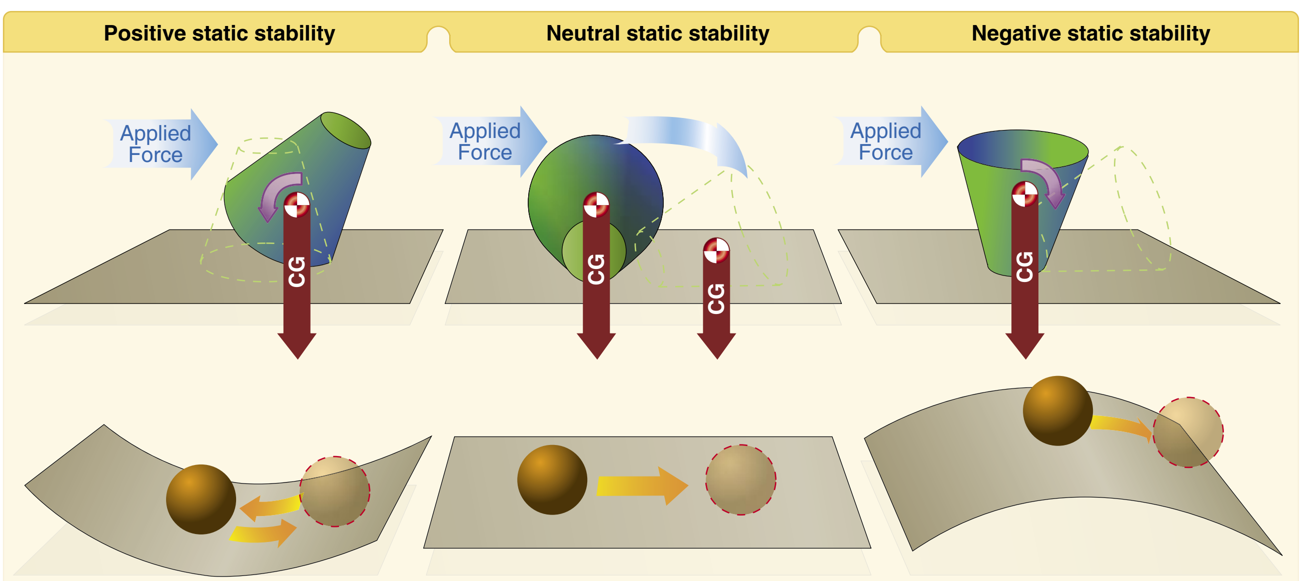


Figure 4-18. *Types of static stability.*

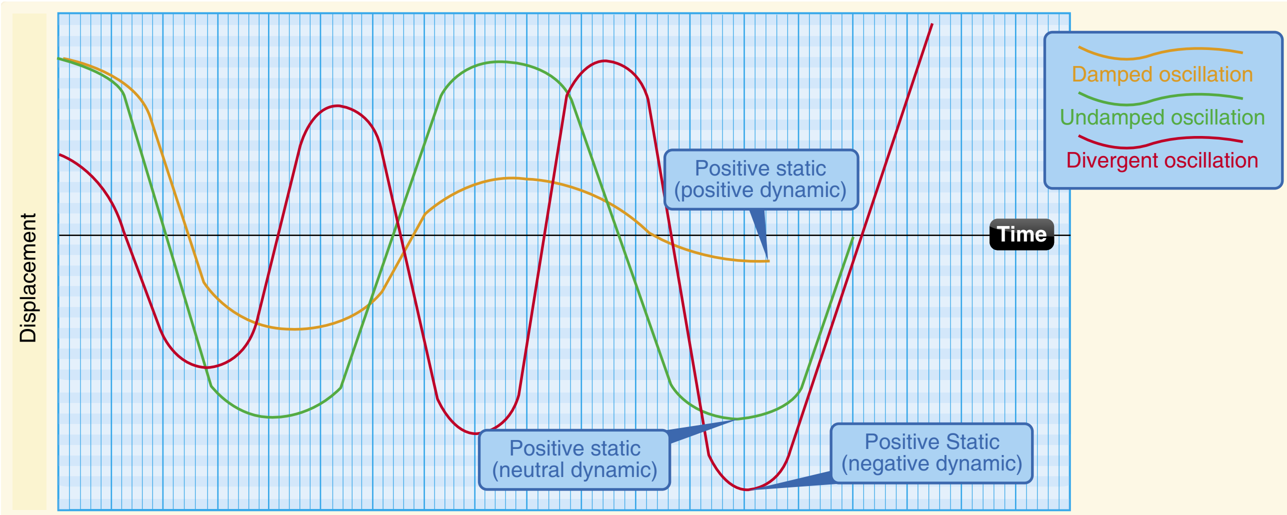


Figure 4-19. *Damped versus undamped stability.*

Longitudinal Stability (Pitching)

In designing an aircraft, a great deal of effort is spent in developing the desired degree of stability around all three axes. But longitudinal stability about the lateral axis is considered to be the most affected by certain variables in various flight conditions.

Longitudinal stability is the quality that makes an aircraft stable about its lateral axis. It involves the pitching motion as the aircraft's nose moves up and down in flight. A longitudinally unstable aircraft has a tendency to dive or climb progressively into a very steep dive or climb, or even a stall. Thus, an aircraft with longitudinal instability becomes difficult and sometimes dangerous to fly.

Static longitudinal stability or instability in an aircraft, is dependent upon three factors:

1. Location of the wing with respect to the CG
2. Location of the horizontal tail surfaces with respect to the CG
3. Area or size of the tail surfaces

In analyzing stability, it should be recalled that a body free to rotate always turns about its CG.

To obtain static longitudinal stability, the relation of the wing and tail moments must be such that, if the moments are initially balanced and the aircraft is suddenly nose up, the wing moments and tail moments change so that the sum of their forces provides an unbalanced but restoring moment which, in turn, brings the nose down again. Similarly, if the aircraft is nose down, the resulting change in moments brings the nose back up.

The CL in most asymmetrical airfoils has a tendency to change its fore and aft positions with a change in the AOA. The CL tends to move forward with an increase in AOA and to move aft with a decrease in AOA. This means that when the AOA of an airfoil is increased, the CL, by moving forward, tends to lift the leading edge of the wing still more. This tendency gives the wing an inherent quality of instability. (NOTE: CL is also known as the center of pressure (CP).)

Figure 4-20 shows an aircraft in straight-and-level flight. The line CG-CL-T represents the aircraft's longitudinal axis from the CG to a point T on the horizontal stabilizer.

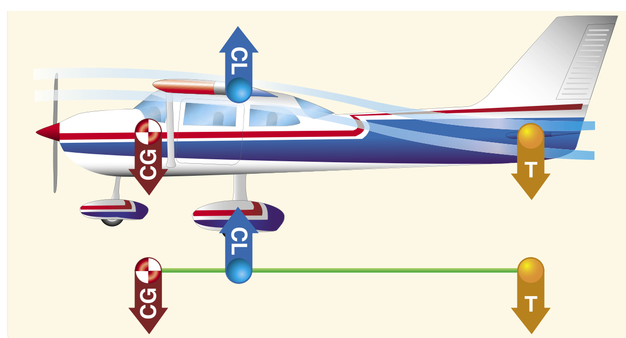


Figure 4-20. *Longitudinal stability.*

Most aircraft are designed so that the wing's CL is to the rear of the CG. This makes the aircraft "nose heavy" and requires that there be a slight downward force on the horizontal stabilizer in order to balance the aircraft and keep the nose from continually pitching downward. Compensation for this nose heaviness is provided by setting the horizontal stabilizer at a slight negative AOA. The downward force thus produced holds the tail down, counterbalancing the "heavy" nose. It is as if the line CG-CL-T were a lever with an upward force at CL and two downward forces balancing each other, one a strong force at the CG point and the other, a much lesser force, at point T (downward air pressure on the stabilizer). To better visualize this physics principle: If an iron bar were suspended at point CL, with a heavy weight hanging on it at the CG, it would take downward pressure at point T to keep the "lever" in balance.

Even though the horizontal stabilizer may be level when the aircraft is in level flight, there is a downwash of air from the wings. This downwash strikes the top of the stabilizer and produces a downward pressure, which at a certain speed is just enough to balance the "lever." The faster the aircraft is flying, the greater this downwash and the greater the downward force on the horizontal stabilizer (except T-tails). *[Figure 4-21]* In aircraft with fixed-position horizontal stabilizers, the aircraft manufacturer sets the stabilizer at an angle that provides the best stability (or balance) during flight at the design cruising speed and power setting.

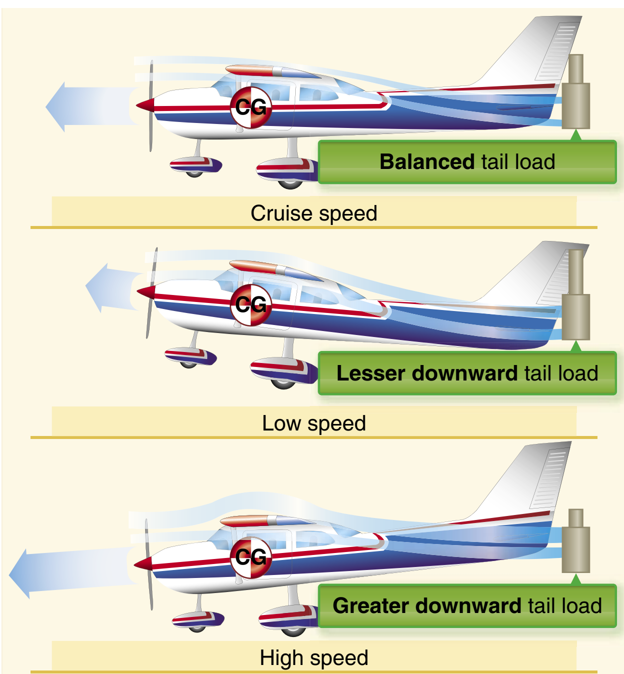


Figure 4-21. *Effect of speed on downwash.*

If the aircraft's speed decreases, the speed of the airflow over the wing is decreased. As a result of this decreased flow of air over the wing, the downwash is reduced, causing a lesser downward force on the horizontal stabilizer. In turn, the characteristic nose heaviness is accentuated, causing the aircraft's nose to pitch down more. *[Figure 4-22]* This places the aircraft in a nose-low attitude, lessening the wing's AOA and drag and allowing the airspeed to increase. As the aircraft continues in the nose-low attitude and its speed increases,

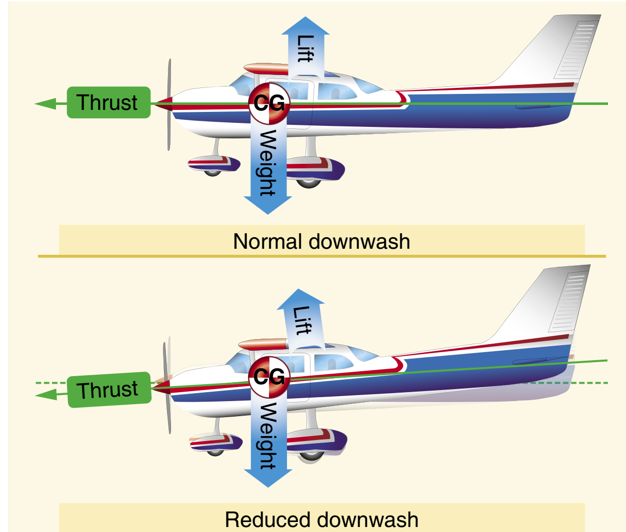


Figure 4-22. *Reduced power allows pitch down.*

the downward force on the horizontal stabilizer is once again increased. Consequently, the tail is again pushed downward and the nose rises into a climbing attitude.

As this climb continues, the airspeed again decreases, causing the downward force on the tail to decrease until the nose lowers once more. Because the aircraft is dynamically stable, the nose does not lower as far this time as it did before. The aircraft acquires enough speed in this more gradual dive to start it into another climb, but the climb is not as steep as the preceding one.

After several of these diminishing oscillations, in which the nose alternately rises and lowers, the aircraft finally settles down to a speed at which the downward force on the tail exactly counteracts the tendency of the aircraft to dive. When this condition is attained, the aircraft is once again in balanced flight and continues in stabilized flight as long as this attitude and airspeed are not changed.

A similar effect is noted upon closing the throttle. The downwash of the wings is reduced and the force at T in *Figure 4-20* is not enough to hold the horizontal stabilizer down. It seems as if the force at T on the lever were allowing the force of gravity to pull the nose down. This is a desirable characteristic because the aircraft is inherently trying to regain airspeed and reestablish the proper balance.

Power or thrust can also have a destabilizing effect in that an increase of power may tend to make the nose rise. The aircraft designer can offset this by establishing a "high thrust line" wherein the line of thrust passes above the CG. *[Figures 4-23* and *4-24]* In this case, as power or thrust is increased a moment is produced to counteract the down

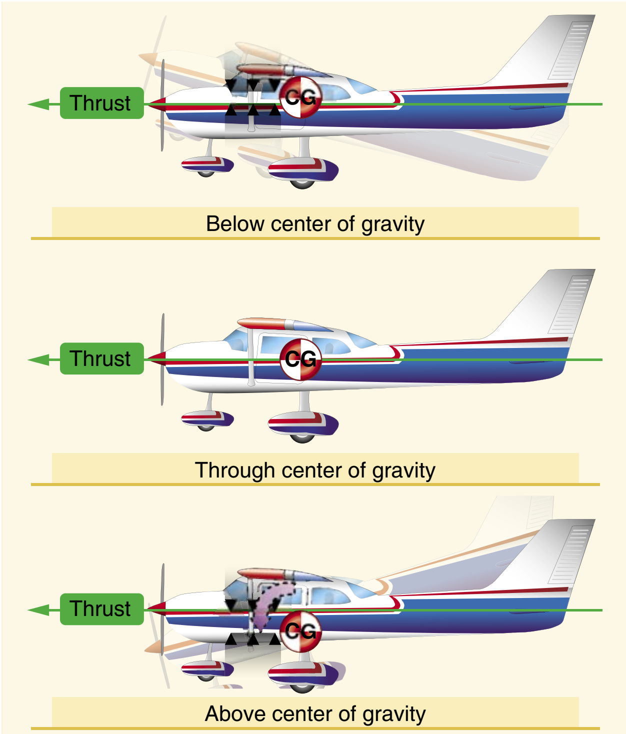


Figure 4-23. *Thrust line affects longitudinal stability.*

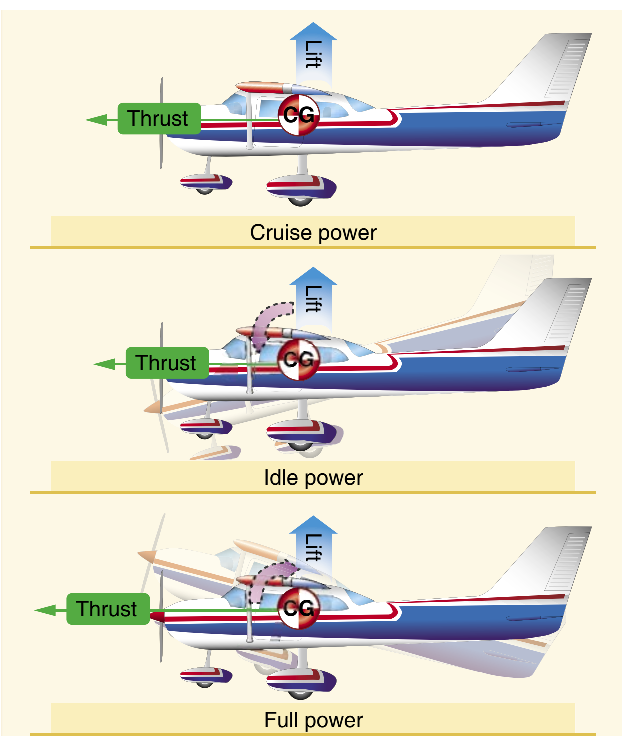


Figure 4-24. *Power changes affect longitudinal stability.*

load on the tail. On the other hand, a very "low thrust line" would tend to add to the nose-up effect of the horizontal tail surface.

Conclusion: with CG forward of the CL and with an aerodynamic tail-down force, the aircraft usually tries to return to a safe flying attitude.

The following is a simple demonstration of longitudinal stability. Trim the aircraft for "hands off" control in level flight. Then, momentarily give the controls a slight push to nose the aircraft down. If, within a brief period, the nose rises to the original position and then stops, the aircraft is statically stable. Ordinarily, the nose passes the original position (that of level flight) and a series of slow pitching oscillations follows. If the oscillations gradually cease, the aircraft has positive stability; if they continue unevenly, the aircraft has neutral stability; if they increase, the aircraft is unstable.

Lateral Stability (Rolling)

Stability about the aircraft's longitudinal axis, which extends from the nose of the aircraft to its tail, is called lateral stability. This helps to stabilize the lateral or "rolling effect" when one wing gets lower than the wing on the opposite side of the aircraft. There are four main design factors that make an aircraft laterally stable: dihedral, sweepback, keel effect, and weight distribution.

Dihedral

The most common procedure for producing lateral stability is to build the wings with an angle of one to three degrees above perpendicular to the longitudinal axis. The wings on either side of the aircraft join the fuselage to form a slight V or angle called "dihedral." The amount of dihedral is measured by the angle made by each wing above a line parallel to the lateral axis.

Dihedral involves a balance of lift created by the wings' AOA on each side of the aircraft's longitudinal axis. If a momentary gust of wind forces one wing to rise and the other to lower, the aircraft banks. When the aircraft is banked without turning, the tendency to sideslip or slide downward toward the lowered wing occurs. *[Figure 4-25]* Since the wings have dihedral, the air strikes the lower wing at a much greater AOA than the higher wing. The increased AOA on the lower wing creates more lift than the higher wing. Increased lift causes the lower wing to begin to rise upward. As the wings approach the level position, the AOA on both wings once again are equal, causing the rolling tendency to subside. The effect of dihedral is to produce a rolling tendency to return the aircraft to a laterally balanced flight condition when a sideslip occurs.

The restoring force may move the low wing up too far, so that the opposite wing now goes down. If so, the process is repeated, decreasing with each lateral oscillation until a balance for wings-level flight is finally reached.

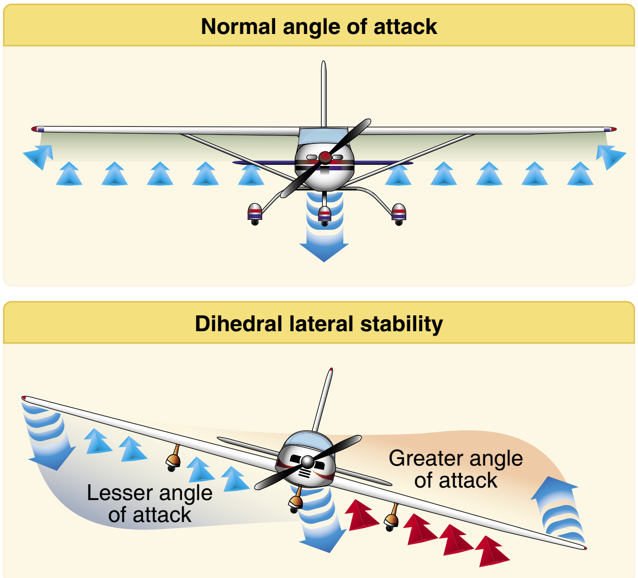


Figure 4-25. *Dihedral for lateral stability.*

Conversely, excessive dihedral has an adverse effect on lateral maneuvering qualities. The aircraft may be so stable laterally that it resists an intentional rolling motion. For this reason, aircraft that require fast roll or banking characteristics usually have less dihedral than those designed for less maneuverability.

Sweepback

Sweepback is an addition to the dihedral that increases the lift created when a wing drops from the level position. A sweptback wing is one in which the leading edge slopes backward. When a disturbance causes an aircraft with sweepback to slip or drop a wing, the low wing presents its leading edge at an angle that is perpendicular to the relative airflow. As a result, the low wing acquires more lift, rises, and the aircraft is restored to its original flight attitude.

Sweepback also contributes to directional stability. When turbulence or rudder application causes the aircraft to yaw to one side, the right wing presents a longer leading edge perpendicular to the relative airflow. The airspeed of the right wing increases and it acquires more drag than the left wing. The additional drag on the right wing pulls it back, turning the aircraft back to its original path.

Keel Effect and Weight Distribution

An aircraft always has the tendency to turn the longitudinal axis of the aircraft into the relative wind. This "weather vane" tendency is similar to the keel of a ship and exerts a steadying influence on the aircraft laterally about the longitudinal axis. When the aircraft is disturbed and one wing dips, the fuselage weight acts like a pendulum returning the airplane to its original attitude.

Laterally stable aircraft are constructed so that the greater portion of the keel area is above and behind the CG. *[Figure 4-26]* Thus, when the aircraft slips to one side, the combination of the aircraft's weight and the pressure of the airflow against the upper portion of the keel area (both acting about the CG) tends to roll the aircraft back to wings-level flight.

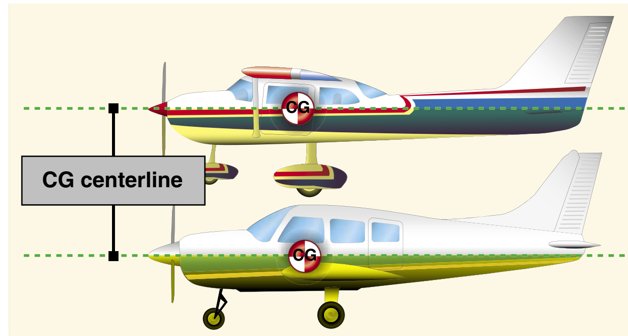


Figure 4-26. *Keel area for lateral stability.*

Vertical Stability (Yawing)

Stability about the aircraft's vertical axis (the sideways moment) is called yawing or directional stability. Yawing or directional stability is the most easily achieved stability in aircraft design. The area of the vertical fin and the sides of the fuselage aft of the CG are the prime contributors which make the aircraft act like the well known weather vane or arrow, pointing its nose into the relative wind.

In examining a weather vane, it can be seen that if exactly the same amount of surface were exposed to the wind in front of the pivot point as behind it, the forces fore and aft would be in balance and little or no directional movement would result. Consequently, it is necessary to have a greater surface aft of the pivot point than forward of it.

Similarly, the aircraft designer must ensure positive directional stability by making the side surface greater aft than ahead of the CG. *[Figure 4-27]* To provide additional positive stability to that provided by the fuselage, a vertical fin is added. The fin acts similar to the feather on an arrow in maintaining straight flight. Like the weather vane and the arrow, the farther aft this fin is placed and the larger its size, the greater the aircraft's directional stability.

If an aircraft is flying in a straight line, and a sideward gust of air gives the aircraft a slight rotation about its vertical axis (i.e., the right), the motion is retarded and stopped by the fin because while the aircraft is rotating to the right, the air is striking the left side of the fin at an angle. This causes pressure on the left side of the fin, which resists the turning motion and slows down the aircraft's yaw. In doing so, it

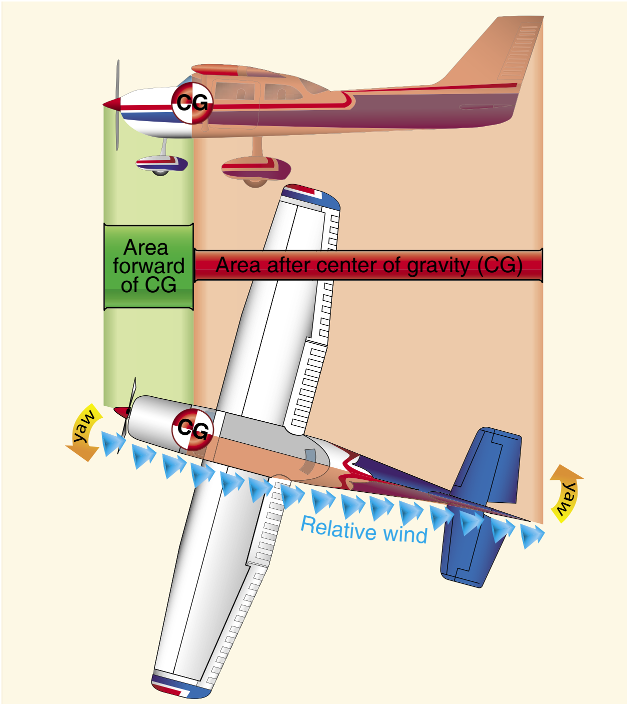


Figure 4-27. *Fuselage and fin for vertical stability.*

acts somewhat like the weather vane by turning the aircraft into the relative wind. The initial change in direction of the aircraft's flightpath is generally slightly behind its change of heading. Therefore, after a slight yawing of the aircraft to the right, there is a brief moment when the aircraft is still moving along its original path, but its longitudinal axis is pointed slightly to the right.

The aircraft is then momentarily skidding sideways, and during that moment (since it is assumed that although the yawing motion has stopped, the excess pressure on the left side of the fin still persists) there is necessarily a tendency for the aircraft to be turned partially back to the left. That is, there is a momentary restoring tendency caused by the fin.

This restoring tendency is relatively slow in developing and ceases when the aircraft stops skidding. When it ceases, the aircraft is flying in a direction slightly different from the original direction. In other words, it will not return of its own accord to the original heading; the pilot must reestablish the initial heading.

A minor improvement of directional stability may be obtained through sweepback. Sweepback is incorporated in the design of the wing primarily to delay the onset of compressibility during high-speed flight. In lighter and slower aircraft, sweepback aids in locating the center of pressure in the correct relationship with the CG. A longitudinally stable aircraft is built with the center of pressure aft of the CG.

Because of structural reasons, aircraft designers sometimes cannot attach the wings to the fuselage at the exact desired point. If they had to mount the wings too far forward, and at right angles to the fuselage, the center of pressure would not be far enough to the rear to result in the desired amount of longitudinal stability. By building sweepback into the wings, however, the designers can move the center of pressure toward the rear. The amount of sweepback and the position of the wings then place the center of pressure in the correct location.

The contribution of the wing to static directional stability is usually small. The swept wing provides a stable contribution depending on the amount of sweepback, but the contribution is relatively small when compared with other components.

Free Directional Oscillations (Dutch Roll)

Dutch roll is a coupled lateral/directional oscillation that is usually dynamically stable but is unsafe in an aircraft because of the oscillatory nature. The damping of the oscillatory mode may be weak or strong depending on the properties of the particular aircraft.

If the aircraft has a right wing pushed down, the positive sideslip angle corrects the wing laterally before the nose is realigned with the relative wind. As the wing corrects the position, a lateral directional oscillation can occur resulting in the nose of the aircraft making a figure eight on the horizon as a result of two oscillations (roll and yaw), which, although of about the same magnitude, are out of phase with each other.

In most modern aircraft, except high-speed swept wing designs, these free directional oscillations usually die out automatically in very few cycles unless the air continues to be gusty or turbulent. Those aircraft with continuing Dutch roll tendencies are usually equipped with gyro-stabilized yaw dampers. Manufacturers try to reach a midpoint between too much and too little directional stability. Because it is more desirable for the aircraft to have "spiral instability" than Dutch roll tendencies, most aircraft are designed with that characteristic.

Spiral Instability

Spiral instability exists when the static directional stability of the aircraft is very strong as compared to the effect of its dihedral in maintaining lateral equilibrium. When the lateral equilibrium of the aircraft is disturbed by a gust of air and a sideslip is introduced, the strong directional stability tends to yaw the nose into the resultant relative wind while the comparatively weak dihedral lags in restoring the lateral balance. Due to this yaw, the wing on the outside of the turning moment travels forward faster than the inside wing and, as a consequence, its lift becomes greater. This produces an overbanking tendency which, if not corrected by the pilot, results in the bank angle becoming steeper and steeper. At the same time, the strong directional stability that yaws the aircraft into the relative wind is actually forcing the nose to a lower pitch attitude. A slow downward spiral begins which, if not counteracted by the pilot, gradually increases into a steep spiral dive. Usually the rate of divergence in the spiral motion is so gradual the pilot can control the tendency without any difficulty.

All aircraft are affected to some degree by this characteristic, although they may be inherently stable in all other normal parameters. This tendency explains why an aircraft cannot be flown "hands off" indefinitely.

Much research has gone into the development of control devices (wing leveler) to correct or eliminate this instability. The pilot must be careful in application of recovery controls during advanced stages of this spiral condition or excessive loads may be imposed on the structure. Improper recovery from spiral instability leading to inflight structural failures has probably contributed to more fatalities in general aviation aircraft than any other factor. Since the airspeed in the spiral condition builds up rapidly, the application of back elevator force to reduce this speed and to pull the nose up only "tightens the turn," increasing the load factor. The results of the prolonged uncontrolled spiral are inflight structural failure or crashing into the ground, or both. The most common recorded causes for pilots who get into this situation are: loss of horizon reference, inability to control the aircraft by reference to instruments, or a combination of both.

Aerodynamic Forces in Flight Maneuvers

Forces in Turns

If an aircraft were viewed in straight-and-level flight from the front *[Figure 4-28]*, and if the forces acting on the aircraft could be seen, lift and weight would be apparent: two forces. If the aircraft were in a bank it would be apparent that lift did not act directly opposite to the weight, rather it now acts in the direction of the bank. A basic truth about turns: when the aircraft banks, lift acts inward toward the center of the turn, as well as upward.

Newton's First Law of Motion, the Law of Inertia, states that an object at rest or moving in a straight line remains at rest or continues to move in a straight line until acted on by some other force. An aircraft, like any moving object, requires a sideward force to make it turn. In a normal turn, this force is supplied by banking the aircraft so that lift is exerted inward, as well as upward. The force of lift during a turn is separated into two components at right angles to each other. One component, which acts vertically and opposite to the weight (gravity), is called the "vertical component of lift." The other, which acts horizontally toward the center of the turn, is called the "horizontal component of lift," or centripetal force. The horizontal component of lift is the force that pulls the aircraft from a straight flightpath to make it turn. Centrifugal force is the "equal and opposite reaction" of the aircraft to the change in direction and acts equal and opposite to the horizontal component of lift. This explains why, in a correctly executed turn, the force that turns the aircraft is not supplied by the rudder. The rudder is used to correct any deviation between the straight track of the nose and tail of the aircraft. A good turn is one in which the nose and tail of the aircraft track along the same path. If no rudder is used in a turn, the nose of the aircraft yaws to the outside of the turn. The rudder is used to bring the nose back in line with the relative wind.

An aircraft is not steered like a boat or an automobile. In order for an aircraft to turn, it must be banked. If it is not banked, there is no force available to cause it to deviate from a straight flightpath. Conversely, when an aircraft is banked, it turns, provided it is not slipping to the inside of the turn.

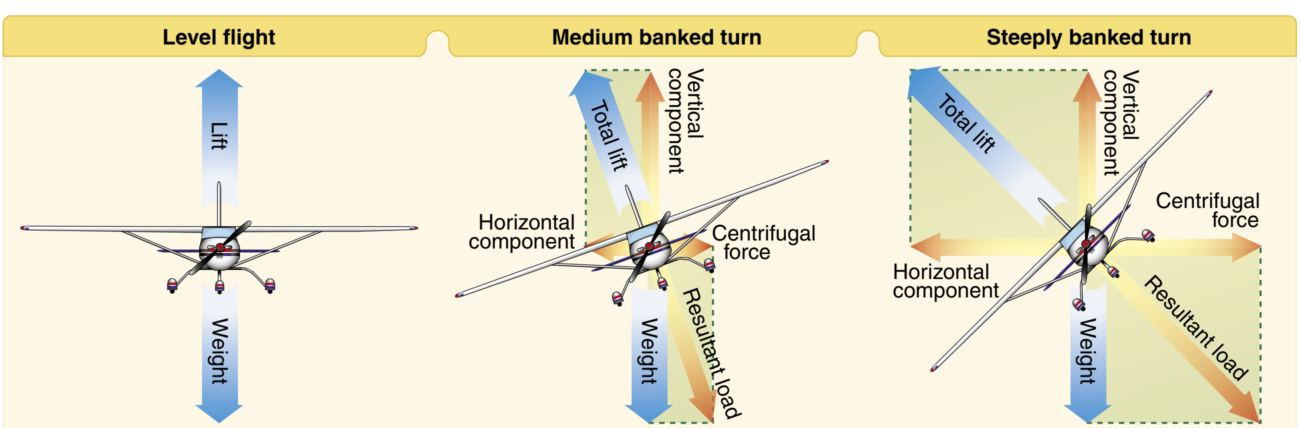


Figure 4-28. *Forces during normal coordinated turn.*

Good directional control is based on the fact that the aircraft attempts to turn whenever it is banked. Pilots should keep this fact in mind when attempting to hold the aircraft in straight-and-level flight.

Merely banking the aircraft into a turn produces no change in the total amount of lift developed. Since the lift during the bank is divided into vertical and horizontal components, the amount of lift opposing gravity and supporting the aircraft's weight is reduced. Consequently, the aircraft loses altitude unless additional lift is created. This is done by increasing the AOA until the vertical component of lift is again equal to the weight. Since the vertical component of lift decreases as the bank angle increases, the AOA must be progressively increased to produce sufficient vertical lift to support the aircraft's weight. An important fact for pilots to remember when making constant altitude turns is that the vertical component of lift must be equal to the weight to maintain altitude.

At a given airspeed, the rate at which an aircraft turns depends upon the magnitude of the horizontal component of lift. It is found that the horizontal component of lift is proportional to the angle of bank—that is, it increases or decreases respectively as the angle of bank increases or decreases. As the angle of bank is increased, the horizontal component of lift increases, thereby increasing the ROT. Consequently, at any given airspeed, the ROT can be controlled by adjusting the angle of bank.

To provide a vertical component of lift sufficient to hold altitude in a level turn, an increase in the AOA is required. Since the drag of the airfoil is directly proportional to its AOA, induced drag increases as the lift is increased. This, in turn, causes a loss of airspeed in proportion to the angle of bank. A small angle of bank results in a small reduction in airspeed while a large angle of bank results in a large reduction in airspeed. Additional thrust (power) must be applied to prevent a reduction in airspeed in level turns. The required amount of additional thrust is proportional to the angle of bank.

To compensate for added lift, which would result if the airspeed were increased during a turn, the AOA must be decreased, or the angle of bank increased, if a constant altitude is to be maintained. If the angle of bank is held constant and the AOA decreased, the ROT decreases. In order to maintain a constant-ROT as the airspeed is increased, the AOA must remain constant and the angle of bank increased.

An increase in airspeed results in an increase of the turn radius, and centrifugal force is directly proportional to the radius of the turn. In a correctly executed turn, the horizontal component of lift must be exactly equal and opposite to the centrifugal force. As the airspeed is increased in a constant-rate level turn, the radius of the turn increases. This increase in the radius of turn causes an increase in the centrifugal force, which must be balanced by an increase in the horizontal component of lift, which can only be increased by increasing the angle of bank.

In a slipping turn, the aircraft is not turning at the rate appropriate to the bank being used, since the aircraft is yawed toward the outside of the turning flightpath. The aircraft is banked too much for the ROT, so the horizontal lift component is greater than the centrifugal force. *[Figure 4-29]* Equilibrium between the horizontal lift component and centrifugal force is reestablished by either decreasing the bank, increasing the ROT, or a combination of the two changes.

A skidding turn results from an excess of centrifugal force over the horizontal lift component, pulling the aircraft toward the outside of the turn. The ROT is too great for the angle of bank. Correction of a skidding turn thus involves a

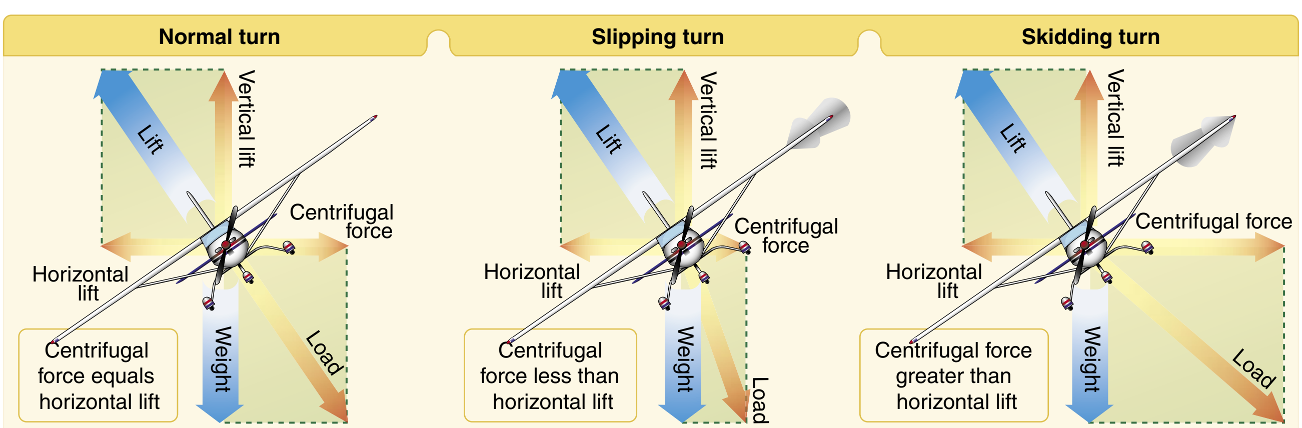


Figure 4-29. *Normal, slipping, and skidding turns.*

reduction in the ROT, an increase in bank, or a combination of the two changes.

To maintain a given ROT, the angle of bank must be varied with the airspeed. This becomes particularly important in high-speed aircraft. For instance, at 400 miles per hour (mph), an aircraft must be banked approximately 44° to execute a standard-rate turn (3° per second). At this angle of bank, only about 79 percent of the lift of the aircraft comprises the vertical component of the lift. This causes a loss of altitude unless the AOA is increased sufficiently to compensate for the loss of vertical lift.

Forces in Climbs

For all practical purposes, the wing's lift in a steady state normal climb is the same as it is in a steady level flight at the same airspeed. Although the aircraft's flightpath changed when the climb was established, the AOA of the wing with respect to the inclined flightpath reverts to practically the same values, as does the lift. There is an initial momentary change as shown in *Figure 4-30.* During the transition from straight-and-level flight to a climb, a change in lift occurs when back elevator pressure is first applied. Raising the aircraft's nose increases the AOA and momentarily increases the lift. Lift at this moment is now greater than weight and starts the aircraft climbing. After the flightpath is stabilized on the upward incline, the AOA and lift again revert to about the level flight values.

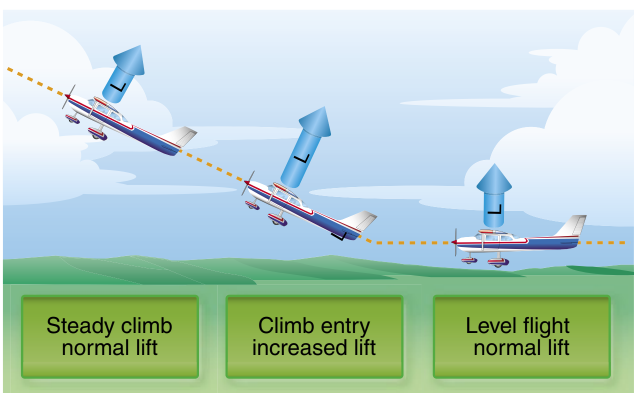


Figure 4-30. *Changes in lift during climb entry.*

If the climb is entered with no change in power setting, the airspeed gradually diminishes because the thrust required to maintain a given airspeed in level flight is insufficient to maintain the same airspeed in a climb. When the flightpath is inclined upward, a component of the aircraft's weight acts in the same direction as, and parallel to, the total drag of the aircraft, thereby increasing the total effective drag. Consequently, the total drag is greater than the power, and the airspeed decreases. The reduction in airspeed gradually results in a corresponding decrease in drag until the total drag (including the component of weight acting in the same direction) equals the thrust. *[Figure 4-31]* Due to momentum, the change in airspeed is gradual, varying considerably with differences in aircraft size, weight, total drag, and other factors. Consequently, the total drag is greater than the thrust, and the airspeed decreases.

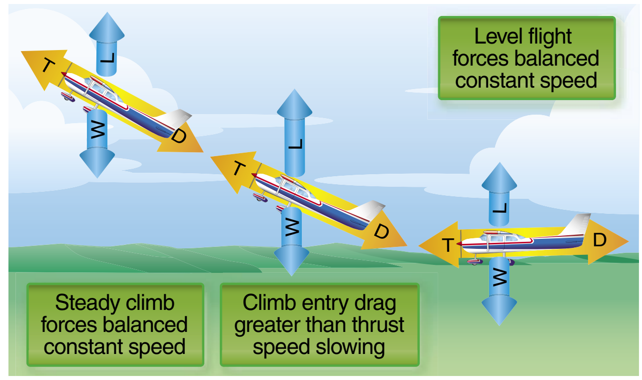


Figure 4-31. *Changes in speed during climb entry.*

Generally, the forces of thrust and drag, and lift and weight, again become balanced when the airspeed stabilizes but at a value lower than in straight-and-level flight at the same power setting. Since the aircraft's weight is acting not only downward but rearward with drag while in a climb, additional power is required to maintain the same airspeed as in level flight. The amount of power depends on the angle of climb. When the climb is established steep enough that there is insufficient power available, a slower speed results.

The thrust required for a stabilized climb equals drag plus a percentage of weight dependent on the angle of climb. For example, a 10° climb would require thrust to equal drag plus 17 percent of weight. To climb straight up would require thrust to equal all of weight and drag. Therefore, the angle of climb for climb performance is dependent on the amount of excess power available to overcome a portion of weight. Note that aircraft are able to sustain a climb due to excess thrust. When the excess thrust is gone, the aircraft is no longer able to climb. At this point, the aircraft has reached its "absolute ceiling."

Forces in Descents

As in climbs, the forces which act on the aircraft go through definite changes when a descent is entered from straight-and-level flight. For the following example, the aircraft is descending at the same power as used in straight-and-level flight.

As forward pressure is applied to the control yoke to initiate the descent, the AOA is decreased momentarily. Initially, the momentum of the aircraft causes the aircraft to briefly continue along the same flightpath. For this instant, the AOA decreases causing the total lift to decrease. With weight now being greater than lift, the aircraft begins to descend. At the same time, the flightpath goes from level to a descending flightpath. Do not confuse a reduction in lift with the inability to generate sufficient lift to maintain level flight. The flightpath is being manipulated with available thrust in reserve and with the elevator.

To descend at the same airspeed as used in straight-and-level flight, the power must be reduced as the descent is entered. The component of weight acting forward along the flightpath increases as the angle of rate of descent increases and, conversely, decreases as the angle of rate of descent decreases. The component of weight acting forward along the flightpath increases as the angle of rate of descent increases and, conversely, decreases as the angle of rate of descent decreases.

Stalls

An aircraft stall results from a rapid decrease in lift caused by the separation of airflow from the wing's surface brought on by exceeding the critical AOA. A stall can occur at any pitch attitude or airspeed. Stalls are one of the most misunderstood areas of aerodynamics because pilots often believe an airfoil stops producing lift when it stalls. In a stall, the wing does not totally stop producing lift. Rather, it can not generate adequate lift to sustain level flight.

Since the C_L increases with an increase in AOA, at some point the C_L peaks and then begins to drop off. This peak is called the $C_{L\text{-}MAX}$. The amount of lift the wing produces drops dramatically after exceeding the $C_{L\text{-}MAX}$ or critical AOA, but as stated above, it does not completely stop producing lift.

In most straight-wing aircraft, the wing is designed to stall the wing root first. The wing root reaches its critical AOA first making the stall progress outward toward the wingtip. By having the wing root stall first, aileron effectiveness is maintained at the wingtips, maintaining controllability of the aircraft. Various design methods are used to achieve the stalling of the wing root first. In one design, the wing is "twisted" to a higher AOA at the wing root. Installing stall strips on the first 20–25 percent of the wing's leading edge is another method to introduce a stall prematurely.

The wing never completely stops producing lift in a stalled condition. If it did, the aircraft would fall to the Earth. Most training aircraft are designed for the nose of the aircraft to drop during a stall, reducing the AOA and "unstalling" the wing. The "nose-down" tendency is due to the CL being aft of the CG. The CG range is very important when it comes to stall recovery characteristics. If an aircraft is allowed to be operated outside of the CG, the pilot may have difficulty recovering from a stall. The most critical CG violation would occur when operating with a CG which exceeds the rear limit. In this situation, a pilot may not be able to generate sufficient force with the elevator to counteract the excess weight aft of the CG. Without the ability to decrease the AOA, the aircraft continues in a stalled condition until it contacts the ground.

The stalling speed of a particular aircraft is not a fixed value for all flight situations, but a given aircraft always stalls at the same AOA regardless of airspeed, weight, load factor, or density altitude. Each aircraft has a particular AOA where the airflow separates from the upper surface of the wing and the stall occurs. This critical AOA varies from 16° to 20° depending on the aircraft's design. But each aircraft has only one specific AOA where the stall occurs.

There are three flight situations in which the critical AOA can be exceeded: low speed, high speed, and turning.

The aircraft can be stalled in straight-and-level flight by flying too slowly. As the airspeed decreases, the AOA must be increased to retain the lift required for maintaining altitude. The lower the airspeed becomes, the more the AOA must be increased. Eventually, an AOA is reached which results in the wing not producing enough lift to support the aircraft which starts settling. If the airspeed is reduced further, the aircraft stalls, since the AOA has exceeded the critical angle and the airflow over the wing is disrupted.

Low speed is not necessary to produce a stall. The wing can be brought into an excessive AOA at any speed. For example, an aircraft is in a dive with an airspeed of 100 knots when the pilot pulls back sharply on the elevator control. *[Figure 4-32]* Gravity and centrifugal force prevent an immediate alteration

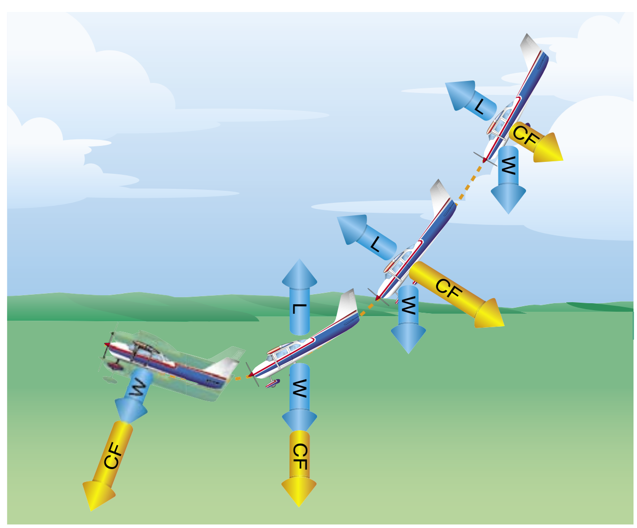


Figure 4-32. *Forces exerted when pulling out of a dive.*

of the flightpath, but the aircraft's AOA changes abruptly from quite low to very high. Since the flightpath of the aircraft in relation to the oncoming air determines the direction of the relative wind, the AOA is suddenly increased, and the aircraft would reach the stalling angle at a speed much greater than the normal stall speed.

The stalling speed of an aircraft is also higher in a level turn than in straight-and-level flight. *[Figure 4-33]* Centrifugal force is added to the aircraft's weight and the wing must produce sufficient additional lift to counterbalance the load imposed by the combination of centrifugal force and weight. In a turn, the necessary additional lift is acquired by applying back pressure to the elevator control. This increases the wing's AOA, and results in increased lift. The AOA must increase as the bank angle increases to counteract the increasing load caused by centrifugal force. If at any time during a turn the AOA becomes excessive, the aircraft stalls.

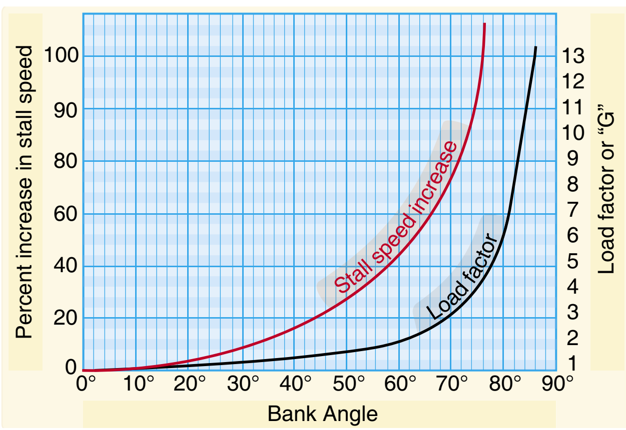


Figure 4-33. *Increase in stall speed and load factor.*

At this point, the action of the aircraft during a stall should be examined. To balance the aircraft aerodynamically, the CL is normally located aft of the CG. Although this makes the aircraft inherently nose-heavy, downwash on the horizontal stabilizer counteracts this condition. At the point of stall, when the upward force of the wing's lift and the downward tail force cease, an unbalanced condition exists. This allows the aircraft to pitch down abruptly, rotating about its CG. During this nose-down attitude, the AOA decreases and the airspeed again increases. The smooth flow of air over the wing begins again, lift returns, and the aircraft is again flying. Considerable altitude may be lost before this cycle is complete.

Airfoil shape and degradation of that shape must also be considered in a discussion of stalls. For example, if ice, snow, and frost are allowed to accumulate on the surface of an aircraft, the smooth airflow over the wing is disrupted. This causes the boundary layer to separate at an AOA lower than that of the critical angle. Lift is greatly reduced, altering expected aircraft performance. If ice is allowed to accumulate on the aircraft during flight *[Figure 4-34],* the weight of the aircraft is increased while the ability to generate lift is decreased. As little as 0.8 millimeter of ice on the upper wing surface increases drag and reduces aircraft lift by 25 percent.

Figure 4-34. *Inflight ice formation.*

Pilots can encounter icing in any season, anywhere in the country, at altitudes of up to 18,000 feet and sometimes higher. Small aircraft, including commuter planes, are most vulnerable because they fly at lower altitudes where ice is more prevalent. They also lack mechanisms common on jet aircraft that prevent ice buildup by heating the front edges of wings.

Icing can occur in clouds any time the temperature drops below freezing and super-cooled droplets build up on an aircraft and freeze. (Super-cooled droplets are still liquid even though the temperature is below 32 °Fahrenheit (F), or 0 °Celsius (C).

Basic Propeller Principles

The aircraft propeller consists of two or more blades and a central hub to which the blades are attached. Each blade of an aircraft propeller is essentially a rotating wing. As a result of their construction, the propeller blades are like airfoils and produce forces that create the thrust to pull, or push, the aircraft through the air. The engine furnishes the power needed to rotate the propeller blades through the air at high speeds, and the propeller transforms the rotary power of the engine into forward thrust.

A cross-section of a typical propeller blade is shown in *Figure 4-35.* This section or blade element is an airfoil comparable to a cross-section of an aircraft wing. One surface of the blade is cambered or curved, similar to the upper surface of an aircraft wing, while the other surface is flat like the bottom surface of a wing. The chord line is an imaginary line drawn through the blade from its leading edge to its trailing edge. As in a wing, the leading edge is the thick edge of the blade that meets the air as the propeller rotates.

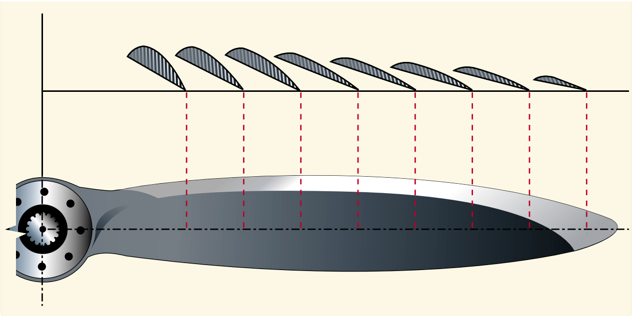


Figure 4-35. *Airfoil sections of propeller blade.*

Blade angle, usually measured in degrees, is the angle between the chord of the blade and the plane of rotation and is measured at a specific point along the length of the blade. *[Figure 4-36]* Because most propellers have a flat blade "face," the chord line is often drawn along the face of the propeller blade. Pitch is not blade angle, but because pitch is largely determined by blade angle, the two terms are often used interchangeably. An increase or decrease in one is usually associated with an increase or decrease in the other.

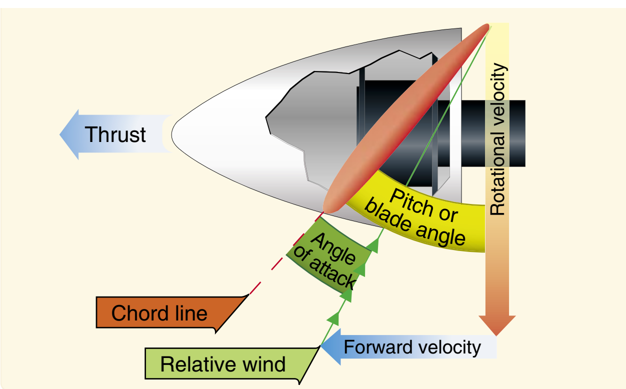


Figure 4-36. *Propeller blade angle.*

The pitch of a propeller may be designated in inches. A propeller designated as a "74-48" would be 74 inches in length and have an effective pitch of 48 inches. The pitch is the distance in inches, which the propeller would screw through the air in one revolution if there were no slippage.

When specifying a fixed-pitch propeller for a new type of aircraft, the manufacturer usually selects one with a pitch that operates efficiently at the expected cruising speed of the aircraft. Every fixed-pitch propeller must be a compromise because it can be efficient at only a given combination of airspeed and revolutions per minute (rpm). Pilots cannot change this combination in flight.

When the aircraft is at rest on the ground with the engine operating, or moving slowly at the beginning of takeoff, the propeller efficiency is very low because the propeller is restrained from advancing with sufficient speed to permit its fixed-pitch blades to reach their full efficiency. In this situation, each propeller blade is turning through the air at an AOA that produces relatively little thrust for the amount of power required to turn it.

To understand the action of a propeller, consider first its motion, which is both rotational and forward. As shown by the vectors of propeller forces in *Figure 4-36,* each section of a propeller blade moves downward and forward. The angle at which this air (relative wind) strikes the propeller blade is its AOA. The air deflection produced by this angle causes the dynamic pressure at the engine side of the propeller blade to be greater than atmospheric pressure, thus creating thrust.

The shape of the blade also creates thrust because it is cambered like the airfoil shape of a wing. As the air flows past the propeller, the pressure on one side is less than that on the other. As in a wing, a reaction force is produced in the direction of the lesser pressure. The airflow over the wing has less pressure, and the force (lift) is upward. In the case of the propeller, which is mounted in a vertical instead of a horizontal plane, the area of decreased pressure is in front of the propeller, and the force (thrust) is in a forward direction. Aerodynamically, thrust is the result of the propeller shape and the AOA of the blade.

Thrust can be considered also in terms of the mass of air handled by the propeller. In these terms, thrust equals mass of air handled multiplied by slipstream velocity minus velocity of the aircraft. The power expended in producing thrust depends on the rate of air mass movement. On average, thrust constitutes approximately 80 percent of the torque (total horsepower absorbed by the propeller). The other 20 percent is lost in friction and slippage. For any speed of rotation, the horsepower absorbed by the propeller balances the horsepower delivered by the engine. For any single revolution of the propeller, the amount of air handled depends on the blade angle, which determines how big a "bite" of air the propeller takes. Thus, the blade angle is an excellent means of adjusting the load on the propeller to control the engine rpm.

The blade angle is also an excellent method of adjusting the AOA of the propeller. On constant-speed propellers, the blade angle must be adjusted to provide the most efficient AOA at all engine and aircraft speeds. Lift versus drag curves, which are drawn for propellers, as well as wings, indicate that the most efficient AOA is small, varying from +2° to +4°. The actual blade angle necessary to maintain this small AOA varies with the forward speed of the aircraft.

Fixed-pitch and ground-adjustable propellers are designed for best efficiency at one rotation and forward speed. They are designed for a given aircraft and engine combination. A propeller may be used that provides the maximum efficiency for takeoff, climb, cruise, or high-speed flight. Any change in these conditions results in lowering the efficiency of both the propeller and the engine. Since the efficiency of any machine is the ratio of the useful power output to the actual power input, propeller efficiency is the ratio of thrust horsepower to brake horsepower. Propeller efficiency varies from 50 to 87 percent, depending on how much the propeller "slips."

Propeller slip is the difference between the geometric pitch of the propeller and its effective pitch. *[Figure 4-37]* Geometric pitch is the theoretical distance a propeller should advance in one revolution; effective pitch is the distance it actually advances. Thus, geometric or theoretical pitch is based on no slippage, but actual or effective pitch includes propeller slippage in the air.

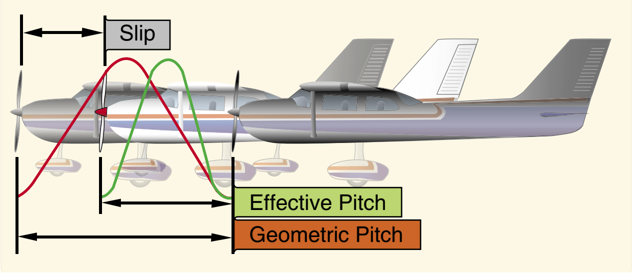


Figure 4-37. *Propeller slippage.*

The reason a propeller is "twisted" is that the outer parts of the propeller blades, like all things that turn about a central point, travel faster than the portions near the hub. *[Figure 4-38]* If the blades had the same geometric pitch throughout their lengths, portions near the hub could have negative AOAs while the propeller tips would be stalled at cruise speed. Twisting or variations in the geometric pitch of the blades permits the propeller to operate with a relatively constant AOA along its length when in cruising flight. Propeller blades are twisted to change the blade angle in proportion to the differences in speed of rotation along the length of the propeller, keeping thrust more nearly equalized along this length.

Usually 1° to 4° provides the most efficient lift/drag ratio, but in flight the propeller AOA of a fixed-pitch propeller varies—normally from 0° to 15°. This variation is caused by changes in the relative airstream, which in turn results from changes in aircraft speed. Thus, propeller AOA is the product of two motions: propeller rotation about its axis and its forward motion.

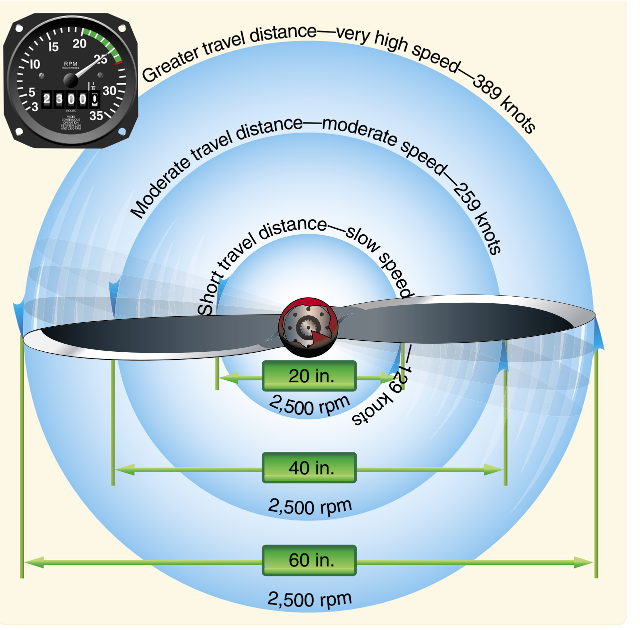


Figure 4-38. *Propeller tips travel faster than the hub.*

A constant-speed propeller automatically keeps the blade angle adjusted for maximum efficiency for most conditions encountered in flight. During takeoff, when maximum power and thrust are required, the constant-speed propeller is at a low propeller blade angle or pitch. The low blade angle keeps the AOA small and efficient with respect to the relative wind. At the same time, it allows the propeller to handle a smaller mass of air per revolution. This light load allows the engine to turn at high rpm and to convert the maximum amount of fuel into heat energy in a given time. The high rpm also creates maximum thrust because, although the mass of air handled per revolution is small, the rpm and slipstream velocity are high, and with the low aircraft speed, there is maximum thrust.

After liftoff, as the speed of the aircraft increases, the constant-speed propeller automatically changes to a higher angle (or pitch). Again, the higher blade angle keeps the AOA small and efficient with respect to the relative wind. The higher blade angle increases the mass of air handled per revolution. This decreases the engine rpm, reducing fuel consumption and engine wear, and keeps thrust at a maximum.

After the takeoff climb is established in an aircraft having a controllable-pitch propeller, the pilot reduces the power output of the engine to climb power by first decreasing the manifold pressure and then increasing the blade angle to lower the rpm.

At cruising altitude, when the aircraft is in level flight and less power is required than is used in takeoff or climb, the pilot again reduces engine power by reducing the manifold pressure and then increasing the blade angle to decrease the rpm. Again, this provides a torque requirement to match the reduced engine power. Although the mass of air handled per revolution is greater, it is more than offset by a decrease in slipstream velocity and an increase in airspeed. The AOA is still small because the blade angle has been increased with an increase in airspeed.

Torque and P-Factor

To the pilot, "torque" (the left turning tendency of the airplane) is made up of four elements which cause or produce a twisting or rotating motion around at least one of the airplane's three axes. These four elements are:

1. Torque reaction from engine and propeller,
2. Corkscrewing effect of the slipstream,
3. Gyroscopic action of the propeller, and
4. Asymmetric loading of the propeller (P-factor).

Torque Reaction

Torque reaction involves Newton's Third Law of Physics—for every action, there is an equal and opposite reaction. As applied to the aircraft, this means that as the internal engine parts and propeller are revolving in one direction, an equal force is trying to rotate the aircraft in the opposite direction. *[Figure 4-39]*

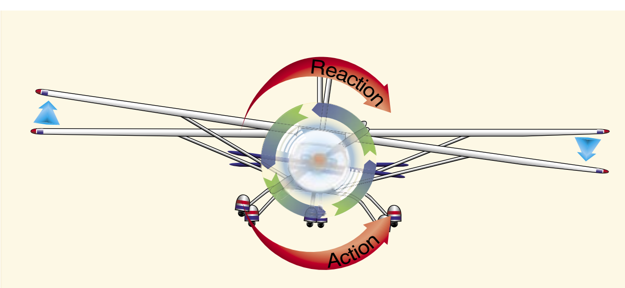


Figure 4-39. *Torque reaction.*

When the aircraft is airborne, this force is acting around the longitudinal axis, tending to make the aircraft roll. To compensate for roll tendency, some of the older aircraft are rigged in a manner to create more lift on the wing that is being forced downward. The more modern aircraft are designed with the engine offset to counteract this effect of torque.

NOTE: Most United States built aircraft engines rotate the propeller clockwise, as viewed from the pilot's seat. The discussion here is with reference to those engines.

Generally, the compensating factors are permanently set so that they compensate for this force at cruising speed, since most of the aircraft's operating lift is at that speed. However, aileron trim tabs permit further adjustment for other speeds.

When the aircraft's wheels are on the ground during the takeoff roll, an additional turning moment around the vertical axis is induced by torque reaction. As the left side of the aircraft is being forced down by torque reaction, more weight is being placed on the left main landing gear. This results in more ground friction, or drag, on the left tire than on the right, causing a further turning moment to the left. The magnitude of this moment is dependent on many variables. Some of these variables are:

1. Size and horsepower of engine,
2. Size of propeller and the rpm,
3. Size of the aircraft, and
4. Condition of the ground surface.

This yawing moment on the takeoff roll is corrected by the pilot's proper use of the rudder or rudder trim.

Corkscrew Effect

The high-speed rotation of an aircraft propeller gives a corkscrew or spiraling rotation to the slipstream. At high propeller speeds and low forward speed (as in the takeoffs and approaches to power-on stalls), this spiraling rotation is very compact and exerts a strong sideward force on the aircraft's vertical tail surface. *[Figure 4-40]*

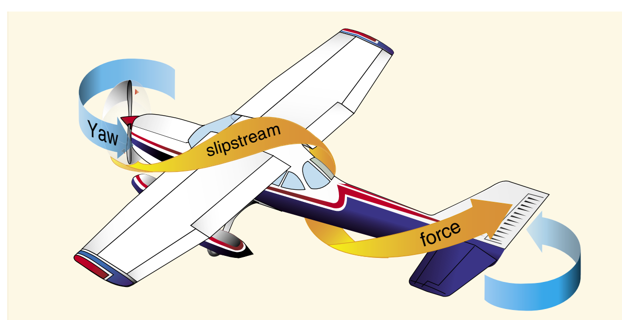


Figure 4-40. *Corkscrewing slipstream.*

When this spiraling slipstream strikes the vertical fin it causes a turning moment about the aircraft's vertical axis. The more compact the spiral, the more prominent this force is. As the forward speed increases, however, the spiral elongates and becomes less effective.The corkscrew flow of the slipstream also causes a rolling moment around the longitudinal axis.

Note that this rolling moment caused by the corkscrew flow of the slipstream is to the right, while the rolling moment caused by torque reaction is to the left—in effect one may be counteracting the other. However, these forces vary greatly and it is the pilot's responsibility to apply proper corrective action by use of the flight controls at all times. These forces must be counteracted regardless of which is the most prominent at the time.

Gyroscopic Action

Before the gyroscopic effects of the propeller can be understood, it is necessary to understand the basic principle of a gyroscope. All practical applications of the gyroscope are based upon two fundamental properties of gyroscopic action: rigidity in space and precession. The one of interest for this discussion is precession.

Precession is the resultant action, or deflection, of a spinning rotor when a deflecting force is applied to its rim. As can be seen in *Figure 4-41,* when a force is applied, the resulting force takes effect 90° ahead of and in the direction of rotation.

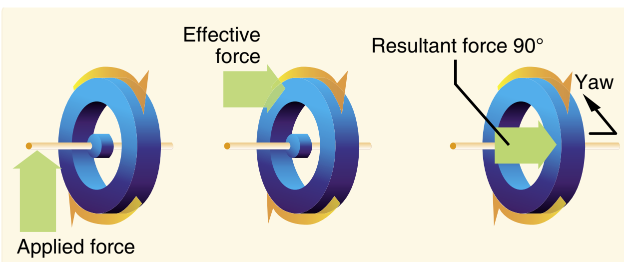


Figure 4-41. *Gyroscopic precession.*

The rotating propeller of an airplane makes a very good gyroscope and thus has similar properties. Any time a force is applied to deflect the propeller out of its plane of rotation, the resulting force is 90° ahead of and in the direction of rotation and in the direction of application, causing a pitching moment, a yawing moment, or a combination of the two depending upon the point at which the force was applied.

This element of torque effect has always been associated with and considered more prominent in tailwheel-type aircraft, and most often occurs when the tail is being raised during the takeoff roll. *[Figure 4-42]* This change in pitch attitude has the same effect as applying a force to the top of the propeller's plane of rotation. The resultant force acting 90° ahead causes a yawing moment to the left around the vertical axis. The magnitude of this moment depends on several variables, one of which is the abruptness with which the tail is raised (amount of force applied). However, precession, or gyroscopic action, occurs when a force is applied to any point on the rim of the propeller's plane of rotation; the resultant force will still be 90° from the point of application in the direction of rotation. Depending on where the force is applied, the airplane is caused to yaw left or right, to pitch up or down, or a combination of pitching and yawing.

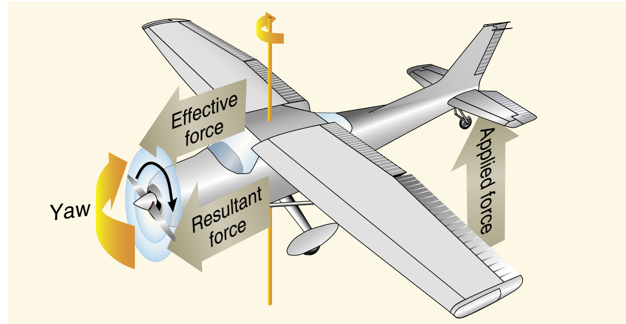


Figure 4-42. *Raising tail produces gyroscopic precession.*

It can be said that, as a result of gyroscopic action, any yawing around the vertical axis results in a pitching moment, and any pitching around the lateral axis results in a yawing moment. To correct for the effect of gyroscopic action, it is necessary for the pilot to properly use elevator and rudder to prevent undesired pitching and yawing.

Asymmetric Loading (P-Factor)

When an aircraft is flying with a high AOA, the "bite" of the downward moving blade is greater than the "bite" of the upward moving blade. This moves the center of thrust to the right of the prop disc area, causing a yawing moment toward the left around the vertical axis. To prove this explanation is complex because it would be necessary to work wind vector problems on each blade while considering both the AOA of the aircraft and the AOA of each blade.

This asymmetric loading is caused by the resultant velocity, which is generated by the combination of the velocity of the propeller blade in its plane of rotation and the velocity of the air passing horizontally through the propeller disc. With the aircraft being flown at positive AOAs, the right (viewed from the rear) or downswinging blade, is passing through an area of resultant velocity which is greater than that affecting the left or upswinging blade. Since the propeller blade is an airfoil, increased velocity means increased lift. The downswinging blade has more lift and tends to pull (yaw) the aircraft's nose to the left.

When the aircraft is flying at a high AOA, the downward moving blade has a higher resultant velocity, creating more lift than the upward moving blade. *[Figure 4-43]* This might be easier to visualize if the propeller shaft was mounted perpendicular to the ground (like a helicopter). If there were no air movement at all, except that generated by the

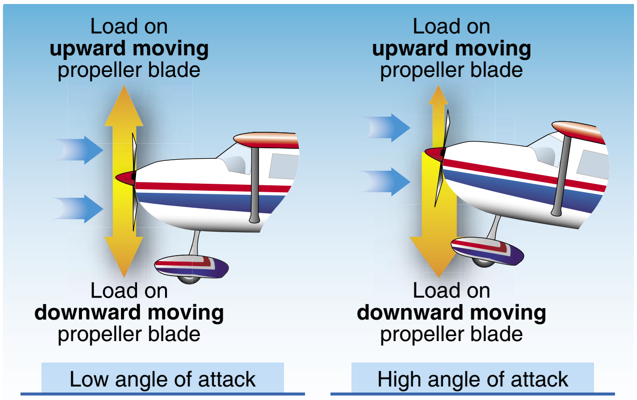


Figure 4-43. *Asymmetrical loading of propeller (P-factor).*

propeller itself, identical sections of each blade would have the same airspeed. With air moving horizontally across this vertically mounted propeller, the blade proceeding forward into the flow of air has a higher airspeed than the blade retreating with the airflow. Thus, the blade proceeding into the horizontal airflow is creating more lift, or thrust, moving the center of thrust toward that blade. Visualize rotating the vertically mounted propeller shaft to shallower angles relative to the moving air (as on an aircraft). This unbalanced thrust then becomes proportionately smaller and continues getting smaller until it reaches the value of zero when the propeller shaft is exactly horizontal in relation to the moving air.

The effects of each of these four elements of torque vary in value with changes in flight situations. In one phase of flight, one of these elements may be more prominent than another. In another phase of flight, another element may be more prominent. The relationship of these values to each other varies with different aircraft—depending on the airframe, engine, and propeller combinations, as well as other design features. To maintain positive control of the aircraft in all flight conditions, the pilot must apply the flight controls as necessary to compensate for these varying values.

Load Factors

In aerodynamics, load factor is the ratio of the maximum load an aircraft can sustain to the gross weight of the aircraft. The load factor is measured in Gs (acceleration of gravity), a unit of force equal to the force exerted by gravity on a body at rest and indicates the force to which a body is subjected when it is accelerated. Any force applied to an aircraft to deflect its flight from a straight line produces a stress on its structure, and the amount of this force is the load factor. While a course in aerodynamics is not a prerequisite for obtaining a pilot's license, the competent pilot should have a solid understanding of the forces that act on the aircraft, the advantageous use of these forces, and the operating limitations of the aircraft being flown.

For example, a load factor of 3 means the total load on an aircraft's structure is three times its gross weight. Since load factors are expressed in terms of Gs, a load factor of 3 may be spoken of as 3 Gs, or a load factor of 4 as 4 Gs.

If an aircraft is pulled up from a dive, subjecting the pilot to 3 Gs, he or she would be pressed down into the seat with a force equal to three times his or her weight. Since modern aircraft operate at significantly higher speeds than older aircraft, increasing the magnitude of the load factor, this effect has become a primary consideration in the design of the structure of all aircraft.

With the structural design of aircraft planned to withstand only a certain amount of overload, a knowledge of load factors has become essential for all pilots. Load factors are important for two reasons:

1. It is possible for a pilot to impose a dangerous overload on the aircraft structures.
2. An increased load factor increases the stalling speed and makes stalls possible at seemingly safe flight speeds.

Load Factors in Aircraft Design

The answer to the question "How strong should an aircraft be?" is determined largely by the use to which the aircraft is subjected. This is a difficult problem because the maximum possible loads are much too high for use in efficient design. It is true that any pilot can make a very hard landing or an extremely sharp pull up from a dive, which would result in abnormal loads. However, such extremely abnormal loads must be dismissed somewhat if aircraft are built that take off quickly, land slowly, and carry worthwhile payloads.

The problem of load factors in aircraft design becomes how to determine the highest load factors that can be expected in normal operation under various operational situations. These load factors are called "limit load factors." For reasons of safety, it is required that the aircraft be designed to withstand these load factors without any structural damage. Although the Code of Federal Regulations (CFR) requires the aircraft structure be capable of supporting one and one-half times these limit load factors without failure, it is accepted that parts of the aircraft may bend or twist under these loads and that some structural damage may occur.

This 1.5 load limit factor is called the "factor of safety" and provides, to some extent, for loads higher than those expected under normal and reasonable operation. This strength reserve is not something which pilots should willfully abuse; rather, it is there for protection when encountering unexpected conditions.

The above considerations apply to all loading conditions, whether they be due to gusts, maneuvers, or landings. The gust load factor requirements now in effect are substantially the same as those that have been in existence for years. Hundreds of thousands of operational hours have proven them adequate for safety. Since the pilot has little control over gust load factors (except to reduce the aircraft's speed when rough air is encountered), the gust loading requirements are substantially the same for most general aviation type aircraft regardless of their operational use. Generally, the gust load factors control the design of aircraft which are intended for strictly nonacrobatic usage.

An entirely different situation exists in aircraft design with maneuvering load factors. It is necessary to discuss this matter separately with respect to: (1) aircraft designed in accordance with the category system (i.e., normal, utility, acrobatic); and (2) older designs built according to requirements which did not provide for operational categories.

Aircraft designed under the category system are readily identified by a placard in the flight deck, which states the operational category (or categories) in which the aircraft is certificated. The maximum safe load factors (limit load factors) specified for aircraft in the various categories are:

CATEGORY	LIMIT LOAD FACTOR
Normal[1]	3.8 to −1.52
Utility (mild acrobatics, including spins)	4.4 to −1.76
Acrobatic	6.0 to −3.00

[1] For aircraft with gross weight of more than 4,000 pounds, the limit load factor is reduced. To the limit loads given above, a safety factor of 50 percent is added.

There is an upward graduation in load factor with the increasing severity of maneuvers. The category system provides for maximum utility of an aircraft. If normal operation alone is intended, the required load factor (and consequently the weight of the aircraft) is less than if the aircraft is to be employed in training or acrobatic maneuvers as they result in higher maneuvering loads.

Aircraft that do not have the category placard are designs that were constructed under earlier engineering requirements in which no operational restrictions were specifically given to the pilots. For aircraft of this type (up to weights of about 4,000 pounds), the required strength is comparable to present-day utility category aircraft, and the same types of operation are permissible. For aircraft of this type over 4,000 pounds, the load factors decrease with weight. These aircraft should be regarded as being comparable to the normal category aircraft designed under the category system, and they should be operated accordingly.

Load Factors in Steep Turns

In a constant altitude, coordinated turn in any aircraft, the load factor is the result of two forces: centrifugal force and gravity. *[Figure 4-44]* For any given bank angle, the ROT varies with the airspeed—the higher the speed, the slower the ROT. This compensates for added centrifugal force, allowing the load factor to remain the same.

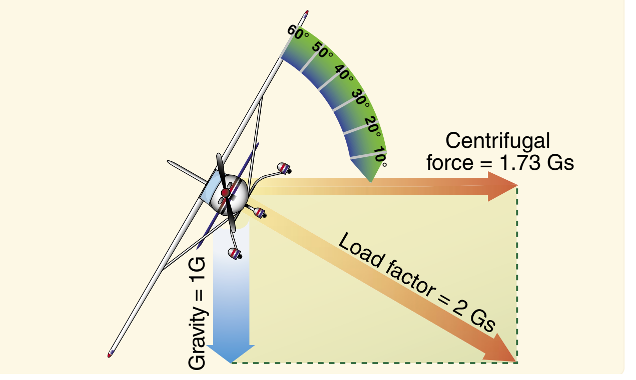


Figure 4-44. *Two forces cause load factor during turns.*

Figure 4-45 reveals an important fact about turns—the load factor increases at a terrific rate after a bank has reached 45° or 50°. The load factor for any aircraft in a 60° bank is 2 Gs. The load factor in an 80° bank is 5.76 Gs. The wing must produce lift equal to these load factors if altitude is to be maintained.

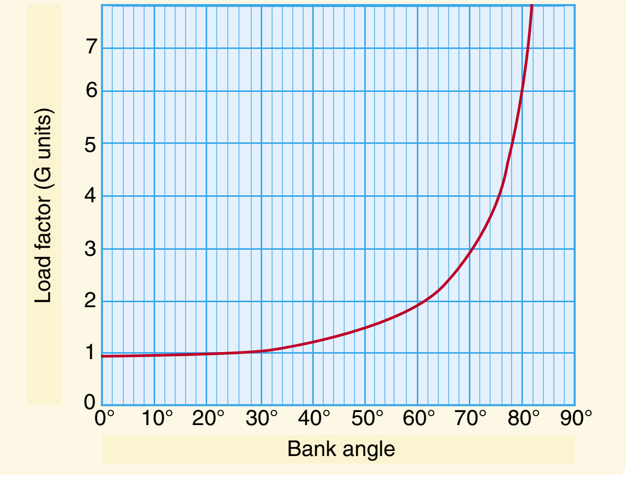


Figure 4-45. *Angle of bank changes load factor.*

It should be noted how rapidly the line denoting load factor rises as it approaches the 90° bank line, which it never quite reaches because a 90° banked, constant altitude turn is not mathematically possible. An aircraft may be banked to 90°, but not in a coordinated turn. An aircraft which can be held in a 90° banked slipping turn is capable of straight knife-edged flight. At slightly more than 80°, the load factor exceeds the limit of 6 Gs, the limit load factor of an acrobatic aircraft.

For a coordinated, constant altitude turn, the approximate maximum bank for the average general aviation aircraft is 60°. This bank and its resultant necessary power setting reach the limit of this type of aircraft. An additional 10° bank increases the load factor by approximately 1 G, bringing it close to the yield point established for these aircraft. *[Figure 4-46]*

Load Factors and Stalling Speeds

Any aircraft, within the limits of its structure, may be stalled at any airspeed. When a sufficiently high AOA is imposed, the smooth flow of air over an airfoil breaks up and separates, producing an abrupt change of flight characteristics and a sudden loss of lift, which results in a stall.

A study of this effect has revealed that the aircraft's stalling speed increases in proportion to the square root of the load factor. This means that an aircraft with a normal unaccelerated stalling speed of 50 knots can be stalled at 100 knots by inducing a load factor of 4 Gs. If it were possible for this aircraft to withstand a load factor of nine, it could be stalled at a speed of 150 knots. A pilot should be aware:

- Of the danger of inadvertently stalling the aircraft by increasing the load factor, as in a steep turn or spiral;
- When intentionally stalling an aircraft above its design maneuvering speed, a tremendous load factor is imposed.

Figures 4-45 and *4-46* show that banking an aircraft greater than 72° in a steep turn produces a load factor of 3, and the stalling speed is increased significantly. If this turn is made in an aircraft with a normal unaccelerated stalling speed of 45 knots, the airspeed must be kept greater than 75 knots to prevent inducing a stall. A similar effect is experienced in a quick pull up, or any maneuver producing load factors above 1 G. This sudden, unexpected loss of control, particularly in a steep turn or abrupt application of the back elevator control near the ground, has caused many accidents.

Since the load factor is squared as the stalling speed doubles, tremendous loads may be imposed on structures by stalling an aircraft at relatively high airspeeds.

The maximum speed at which an aircraft may be stalled safely is now determined for all new designs. This speed is called the "design maneuvering speed" (V_A) and must be entered in the FAA-approved Airplane Flight Manual/Pilot's Operating Handbook (AFM/POH) of all recently designed aircraft. For older general aviation aircraft, this speed is approximately 1.7 times the normal stalling speed. Thus, an older aircraft which normally stalls at 60 knots must never be stalled at above 102 knots (60 knots x 1.7 = 102 knots). An aircraft with a normal stalling speed of 60 knots stalled at 102 knots undergoes a load factor equal to the square of the increase in speed, or 2.89 Gs (1.7 x 1.7 = 2.89 Gs). (The above figures are approximations to be considered as a guide, and are not the exact answers to any set of problems. The design maneuvering speed should be determined from the particular aircraft's operating limitations provided by the manufacturer.)

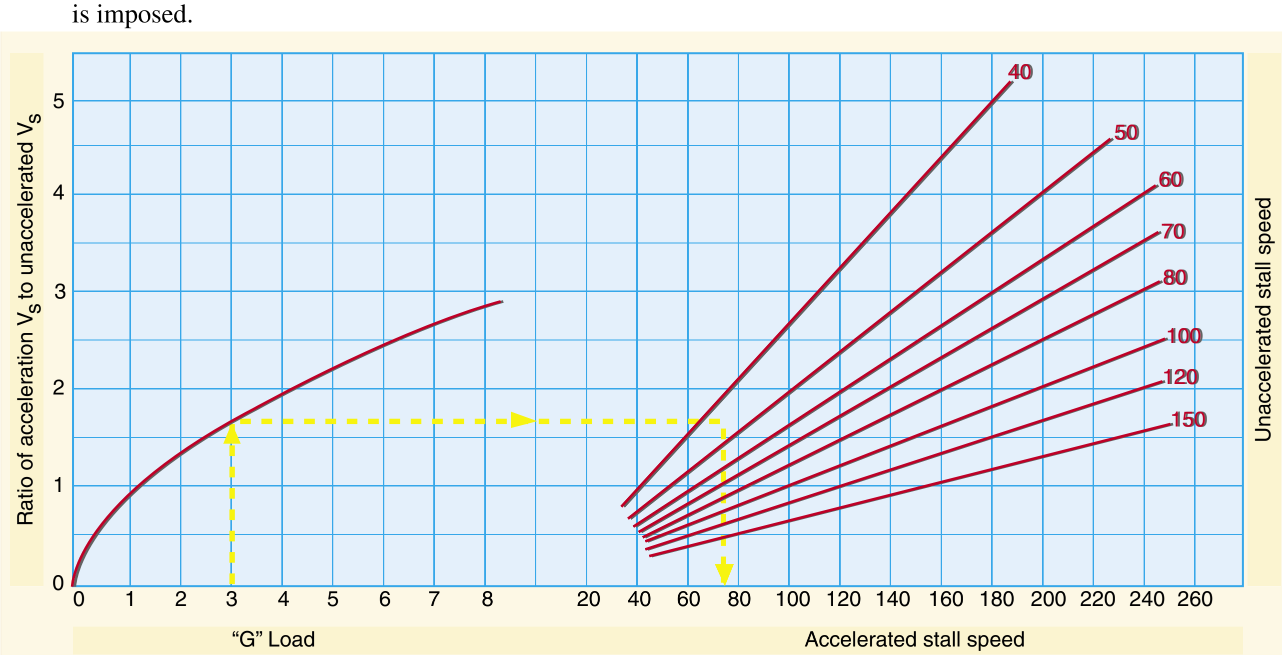


Figure 4-46. *Load factor changes stall speed.*

Since the leverage in the control system varies with different aircraft (some types employ "balanced" control surfaces while others do not), the pressure exerted by the pilot on the controls cannot be accepted as an index of the load factors produced in different aircraft. In most cases, load factors can be judged by the experienced pilot from the feel of seat pressure. Load factors can also be measured by an instrument called an "accelerometer," but this instrument is not common in general aviation training aircraft. The development of the ability to judge load factors from the feel of their effect on the body is important. A knowledge of these principles is essential to the development of the ability to estimate load factors.

A thorough knowledge of load factors induced by varying degrees of bank and the V_A aids in the prevention of two of the most serious types of accidents:

1. Stalls from steep turns or excessive maneuvering near the ground
2. Structural failures during acrobatics or other violent maneuvers resulting from loss of control

Load Factors and Flight Maneuvers

Critical load factors apply to all flight maneuvers except unaccelerated straight flight where a load factor of 1 G is always present. Certain maneuvers considered in this section are known to involve relatively high load factors.

Turns

Increased load factors are a characteristic of all banked turns. As noted in the section on load factors in steep turns, load factors become significant to both flight performance and load on wing structure as the bank increases beyond approximately 45°.

The yield factor of the average light plane is reached at a bank of approximately 70° to 75°, and the stalling speed is increased by approximately one-half at a bank of approximately 63°.

Stalls

The normal stall entered from straight-and-level flight, or an unaccelerated straight climb, does not produce added load factors beyond the 1 G of straight-and-level flight. As the stall occurs, however, this load factor may be reduced toward zero, the factor at which nothing seems to have weight. The pilot experiences a sensation of "floating free in space." If recovery is effected by snapping the elevator control forward, negative load factors (or those that impose a down load on the wings and raise the pilot from the seat) may be produced.

During the pull up following stall recovery, significant load factors are sometimes induced. These may be further increased inadvertently during excessive diving (and consequently high airspeed) and abrupt pull ups to level flight. One usually leads to the other, thus increasing the load factor. Abrupt pull ups at high diving speeds may impose critical loads on aircraft structures and may produce recurrent or secondary stalls by increasing the AOA to that of stalling.

As a generalization, a recovery from a stall made by diving only to cruising or design maneuvering airspeed, with a gradual pull up as soon as the airspeed is safely above stalling, can be effected with a load factor not to exceed 2 or 2.5 Gs. A higher load factor should never be necessary unless recovery has been effected with the aircraft's nose near or beyond the vertical attitude, or at extremely low altitudes to avoid diving into the ground.

Spins

A stabilized spin is not different from a stall in any element other than rotation and the same load factor considerations apply to spin recovery as apply to stall recovery. Since spin recoveries are usually effected with the nose much lower than is common in stall recoveries, higher airspeeds and consequently higher load factors are to be expected. The load factor in a proper spin recovery usually is found to be about 2.5 Gs.

The load factor during a spin varies with the spin characteristics of each aircraft, but is usually found to be slightly above the 1 G of level flight. There are two reasons for this:

1. Airspeed in a spin is very low, usually within 2 knots of the unaccelerated stalling speeds.
2. Aircraft pivots, rather than turns, while it is in a spin.

High Speed Stalls

The average light plane is not built to withstand the repeated application of load factors common to high speed stalls. The load factor necessary for these maneuvers produces a stress on the wings and tail structure, which does not leave a reasonable margin of safety in most light aircraft.

The only way this stall can be induced at an airspeed above normal stalling involves the imposition of an added load factor, which may be accomplished by a severe pull on the elevator control. A speed of 1.7 times stalling speed (about 102 knots in a light aircraft with a stalling speed of 60 knots) produces a load factor of 3 Gs. Only a very narrow margin for error can be allowed for acrobatics in light aircraft. To illustrate how rapidly the load factor increases with airspeed, a high-speed stall at 112 knots in the same aircraft would produce a load factor of 4 Gs.

Chandelles and Lazy Eights
A chandelle is a maximum performance climbing turn beginning from approximately straight-and-level flight, and ending at the completion of a precise 180° of turn in a wings-level, nose-high attitude at the minimum controllable airspeed. In this flight maneuver, the aircraft is in a steep climbing turn and almost stalls to gain altitude while changing direction. A lazy eight derives its name from the manner in which the extended longitudinal axis of the aircraft is made to trace a flight pattern in the form of a figure "8" lying on its side. It would be difficult to make a definite statement concerning load factors in these maneuvers as both involve smooth, shallow dives and pull ups. The load factors incurred depend directly on the speed of the dives and the abruptness of the pull ups during these maneuvers.

Generally, the better the maneuver is performed, the less extreme the load factor induced. A chandelle or lazy eight in which the pull-up produces a load factor greater than 2 Gs will not result in as great a gain in altitude, and in low-powered aircraft it may result in a net loss of altitude.

The smoothest pull up possible, with a moderate load factor, delivers the greatest gain in altitude in a chandelle and results in a better overall performance in both chandelles and lazy eights. The recommended entry speed for these maneuvers is generally near the manufacturer's design maneuvering speed which allows maximum development of load factors without exceeding the load limits.

Rough Air
All standard certificated aircraft are designed to withstand loads imposed by gusts of considerable intensity. Gust load factors increase with increasing airspeed, and the strength used for design purposes usually corresponds to the highest level flight speed. In extremely rough air, as in thunderstorms or frontal conditions, it is wise to reduce the speed to the design maneuvering speed. Regardless of the speed held, there may be gusts that can produce loads which exceed the load limits.

Each specific aircraft is designed with a specific G loading that can be imposed on the aircraft without causing structural damage. There are two types of load factors factored into aircraft design, limit load and ultimate load. The limit load is a force applied to an aircraft that causes a bending of the aircraft structure that does not return to the original shape. The ultimate load is the load factor applied to the aircraft beyond the limit load and at which point the aircraft material experiences structural failure (breakage). Load factors lower than the limit load can be sustained without compromising the integrity of the aircraft structure.

Speeds up to but not exceeding the maneuvering speed allows an aircraft to stall prior to experiencing an increase in load factor that would exceed the limit load of the aircraft.

Most AFM/POH now include turbulent air penetration information, which help today's pilots safely fly aircraft capable of a wide range of speeds and altitudes. It is important for the pilot to remember that the maximum "never-exceed" placard dive speeds are determined for smooth air only. High speed dives or acrobatics involving speed above the known maneuvering speed should never be practiced in rough or turbulent air.

Vg Diagram
The flight operating strength of an aircraft is presented on a graph whose vertical scale is based on load factor. *[Figure 4-47]* The diagram is called a Vg diagram—velocity versus G loads or load factor. Each aircraft has its own Vg diagram which is valid at a certain weight and altitude.

The lines of maximum lift capability (curved lines) are the first items of importance on the Vg diagram. The aircraft in the *Figure 4-47* is capable of developing no more than +1 G at 62 mph, the wing level stall speed of the aircraft. Since the maximum load factor varies with the square of the airspeed, the maximum positive lift capability of this aircraft is 2 G at 92 mph, 3 G at 112 mph, 4.4 G at 137 mph, and so forth. Any load factor above this line is unavailable aerodynamically (i.e., the aircraft cannot fly above the line of maximum lift capability because it stalls). The same situation exists for negative lift flight with the exception that the speed necessary to produce a given negative load factor is higher than that to produce the same positive load factor.

If the aircraft is flown at a positive load factor greater than the positive limit load factor of 4.4, structural damage is possible. When the aircraft is operated in this region, objectionable permanent deformation of the primary structure may take place and a high rate of fatigue damage is incurred. Operation above the limit load factor must be avoided in normal operation.

There are two other points of importance on the Vg diagram. One point is the intersection of the positive limit load factor and the line of maximum positive lift capability. The airspeed at this point is the minimum airspeed at which the limit load can be developed aerodynamically. Any airspeed greater than this provides a positive lift capability sufficient to damage the aircraft. Conversely, any airspeed less than this does not provide positive lift capability sufficient to cause damage from excessive flight loads. The usual term given to this speed is "maneuvering speed," since consideration of subsonic

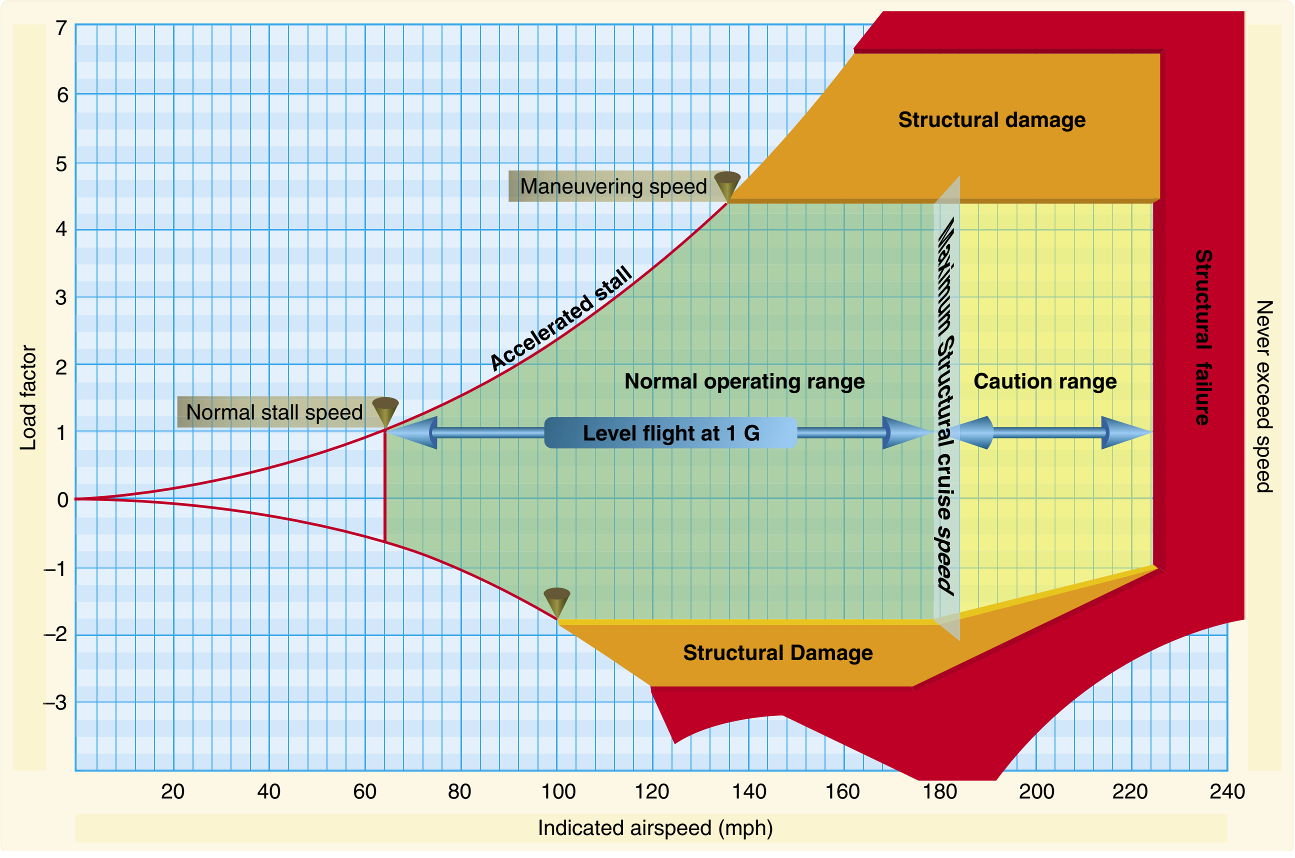


Figure 4-47. *Typical Vg diagram.*

aerodynamics would predict minimum usable turn radius or maneuverability to occur at this condition. The maneuver speed is a valuable reference point, since an aircraft operating below this point cannot produce a damaging positive flight load. Any combination of maneuver and gust cannot create damage due to excess airload when the aircraft is below the maneuver speed.

The other point of importance on the Vg diagram is the intersection of the negative limit load factor and line of maximum negative lift capability. Any airspeed greater than this provides a negative lift capability sufficient to damage the aircraft; any airspeed less than this does not provide negative lift capability sufficient to damage the aircraft from excessive flight loads.

The limit airspeed (or redline speed) is a design reference point for the aircraft—this aircraft is limited to 225 mph. If flight is attempted beyond the limit airspeed, structural damage or structural failure may result from a variety of phenomena.

The aircraft in flight is limited to a regime of airspeeds and Gs which do not exceed the limit (or redline) speed, do not exceed the limit load factor, and cannot exceed the maximum lift capability. The aircraft must be operated within this "envelope" to prevent structural damage and ensure the anticipated service lift of the aircraft is obtained. The pilot must appreciate the Vg diagram as describing the allowable combination of airspeeds and load factors for safe operation. Any maneuver, gust, or gust plus maneuver outside the structural envelope can cause structural damage and effectively shorten the service life of the aircraft.

Rate of Turn

The rate of turn (ROT) is the number of degrees (expressed in degrees per second) of heading change that an aircraft makes. The ROT can be determined by taking the constant of 1,091, multiplying it by the tangent of any bank angle and dividing that product by a given airspeed in knots as illustrated in *Figure 4-48.* If the airspeed is increased and the ROT desired is to be constant, the angle of bank must be increased, otherwise, the ROT decreases. Likewise, if the airspeed is held constant, an aircraft's ROT increases if the bank angle is increased. The formula in *Figures 4-48* through *4-50* depicts the relationship between bank angle and airspeed as they affect the ROT.

NOTE: All airspeed discussed in this section is true airspeed (TAS).

$$\text{ROT} = \frac{1{,}091 \times \text{tangent of the bank angle}}{\text{airspeed (in knots)}}$$

Example The rate of turn for an aircraft in a coordinated turn of 30° and traveling at 120 knots would have a ROT as follows.

$$\text{ROT} = \frac{1{,}091 \times \text{tangent of } 30°}{120 \text{ knots}}$$

$$\text{ROT} = \frac{1{,}091 \times 0.5773 \text{ (tangent of } 30°)}{120 \text{ knots}}$$

ROT = 5.25 degrees per second

Figure 4-48. *Rate of turn for a given airspeed (knots, TAS) and bank angle.*

Example Suppose we were to increase the speed to 240 knots, what is the rate of turn? Using the same formula from above we see that:

$$\text{ROT} = \frac{1{,}091 \times \text{tangent of } 30°}{240 \text{ knots}}$$

ROT = 2.62 degrees per second

An increase in speed causes a decrease in the rate of turn when using the same bank angle.

Figure 4-49. *Rate of turn when increasing speed.*

Example Suppose we wanted to know what bank angle would give us a rate of turn of 5.25° per second at 240 knots. A slight rearrangement of the formula would indicate it will take a 49° angle of bank to achieve the same ROT used at the lower airspeed of 120 knots.

$$\text{ROT } (5.25) = \frac{1{,}091 \times \text{tangent of X}}{240 \text{ knots}}$$

$$240 \times 5.25 = 1{,}091 \times \text{tangent of X}$$

$$\frac{240 \times 5.25}{1{,}091} = \text{tangent of X}$$

$$1.1549 = \text{tangent of X}$$

$$49° = \text{X}$$

Figure 4-50. *To achieve the same rate of turn of an aircraft traveling at 120 knots, an increase of bank angle is required.*

Airspeed significantly effects an aircraft's ROT. If airspeed is increased, the ROT is reduced if using the same angle of bank used at the lower speed. Therefore, if airspeed is increased as illustrated in *Figure 4-49,* it can be inferred that the angle of bank must be increased in order to achieve the same ROT achieved in *Figure 4-50.*

What does this mean on a practicable side? If a given airspeed and bank angle produces a specific ROT, additional conclusions can be made. Knowing the ROT is a given number of degrees of change per second, the number of seconds it takes to travel 360° (a circle) can be determined by simple division. For example, if moving at 120 knots with a 30° bank angle, the ROT is 5.25° per second and it takes 68.6 seconds (360° divided by 5.25 = 68.6 seconds) to make a complete circle. Likewise, if flying at 240 knots TAS and using a 30° angle of bank, the ROT is only about 2.63° per second and it takes about 137 seconds to complete a 360° circle. Looking at the formula, any increase in airspeed is directly proportional to the time the aircraft takes to travel an arc.

So why is this important to understand? Once the ROT is understood, a pilot can determine the distance required to make that particular turn which is explained in radius of turn.

Radius of Turn

The radius of turn is directly linked to the ROT, which explained earlier is a function of both bank angle and airspeed. If the bank angle is held constant and the airspeed is increased, the radius of the turn changes (increases). A higher airspeed causes the aircraft to travel through a longer arc due to a greater speed. An aircraft traveling at 120 knots is able to turn a 360° circle in a tighter radius than an aircraft traveling at 240 knots. In order to compensate for the increase in airspeed, the bank angle would need to be increased.

The radius of turn (R) can be computed using a simple formula. The radius of turn is equal to the velocity squared (V^2) divided by 11.26 times the tangent of the bank angle.

$$R = \frac{V^2}{11.26 \times \text{tangent of bank angle}}$$

Using the examples provided in *Figures 4-48* through *4-50,* the turn radius for each of the two speeds can be computed. Note that if the speed is doubled, the radius is squared. *[Figures 4-51* and *4-52]*

120 knots

$$R = \frac{V^2}{11.26 \times \text{tangent of bank angle}}$$

$$R = \frac{120^2}{11.26 \times \text{tangent of } 30°}$$

$$R = \frac{14{,}400}{11.26 \times 0.5773}$$

R = 2,215 feet

The radius of a turn required by an aircraft traveling at 120 knots and using a bank angle of 30° is 2,215 feet.

Figure 4-51. *Radius at 120 knots with bank angle of 30°.*

240 knots

$$R = \frac{V^2}{11.26 \times \text{tangent of bank angle}}$$

$$R = \frac{240^2}{11.26 \times \text{tangent of } 30^\circ}$$

$$R = \frac{57{,}600}{11.26 \times 0.57735}$$

$$R = 8{,}861 \text{ feet}$$

(four times the radius at 120 knots)

The radius of a turn required by an aircraft traveling at 240 knots using the same bank angle in *Figure 4-51* is 8,861 feet. Speed is a major factor in a turn.

Figure 4-52. *Radius at 240 knots.*

Another way to determine the radius of turn is speed in using feet per second (fps), π (3.1415) and the ROT. Using the example on page 4-34 in the upper right column, it was determined that an aircraft with a ROT of 5.25 degrees per second required 68.6 seconds to make a complete circle. An aircraft's speed (in knots) can be converted to fps by multiplying it by a constant of 1.69. Therefore, an aircraft traveling at 120 knots (TAS) travels at 202.8 fps. Knowing the speed in fps (202.8) multiplied by the time an aircraft takes to complete a circle (68.6 seconds) can determine the size of the circle; 202.8 times 68.6 equals 13,912 feet. Dividing by π yields a diameter of 4,428 feet, which when divided by 2 equals a radius of 2,214 feet *[Figure 4-53],* a foot within that determined through use of the formula in *Figure 4-51.*

$$r = \frac{\dfrac{\text{speed (fps)} \times \dfrac{360}{\text{ROT}}}{\text{Pi } (\pi)}}{2}$$

$$r = \frac{\dfrac{202.8 \times 68.6}{\pi}}{2}$$

$$r = \frac{\dfrac{13{,}912}{\pi}}{2}$$

$$r = \frac{4{,}428}{2} = 2{,}214 \text{ feet}$$

Figure 4-53. *Another formula that can be used for radius.*

In *Figure 4-54,* the pilot enters a canyon and decides to turn 180° to exit. The pilot uses a 30° bank angle in his turn.

Weight and Balance

The aircraft's weight and balance data is important information for a pilot that must be frequently reevaluated. Although the aircraft was weighed during the certification process, this data is not valid indefinitely. Equipment changes or modifications affect the weight and balance data. Too often pilots reduce the aircraft weight and balance into a "rule of thumb" such as: "If I have three passengers, I can load only 100 gallons of fuel; four passengers, 70 gallons."

Weight and balance computations should be part of every preflight briefing. Never assume three passengers are always of equal weight. Instead, do a full computation of all items to be loaded on the aircraft, including baggage, as well as the pilot and passenger. It is recommended that all bags be weighed to make a precise computation of how the aircraft CG is positioned.

The importance of the CG was stressed in the discussion of stability, controllability, and performance. Unequal load distribution causes accidents. A competent pilot understands and respects the effects of CG on an aircraft.

Weight and balance are critical components in the utilization of an aircraft to its fullest potential. The pilot must know how much fuel can be loaded onto the aircraft without violating CG limits, as well as weight limits to conduct long or short flights with or without a full complement of allowable passengers. For example, an aircraft has four seats and can carry 60 gallons of fuel. How many passengers can the aircraft safely carry? Can all those seats be occupied at all times with the varying fuel loads? Four people who each weigh 150 pounds leads to a different weight and balance computation than four people who each weigh 200 pounds. The second scenario loads an additional 200 pounds onto the aircraft and is equal to about 30 gallons of fuel.

The additional weight may or may not place the CG outside of the CG envelope, but the maximum gross weight could be exceeded. The excess weight can overstress the aircraft and degrade the performance.

Aircraft are certificated for weight and balance for two principal reasons:

1. The effect of the weight on the aircraft's primary structure and its performance characteristics
2. The effect of the location of this weight on flight characteristics, particularly in stall and spin recovery and stability

Aircraft, such as balloons and weight-shift control, do not require weight and balance computations because the load is suspended below the lifting mechanism. The CG range in these types of aircraft is such that it is difficult to exceed loading limits. For example, the rear seat position and fuel of a weight-shift control aircraft are as close as possible to the hang point with the aircraft in a suspended attitude. Thus, load variations have little effect on the CG. This also holds true for the balloon basket or gondola. While it is difficult

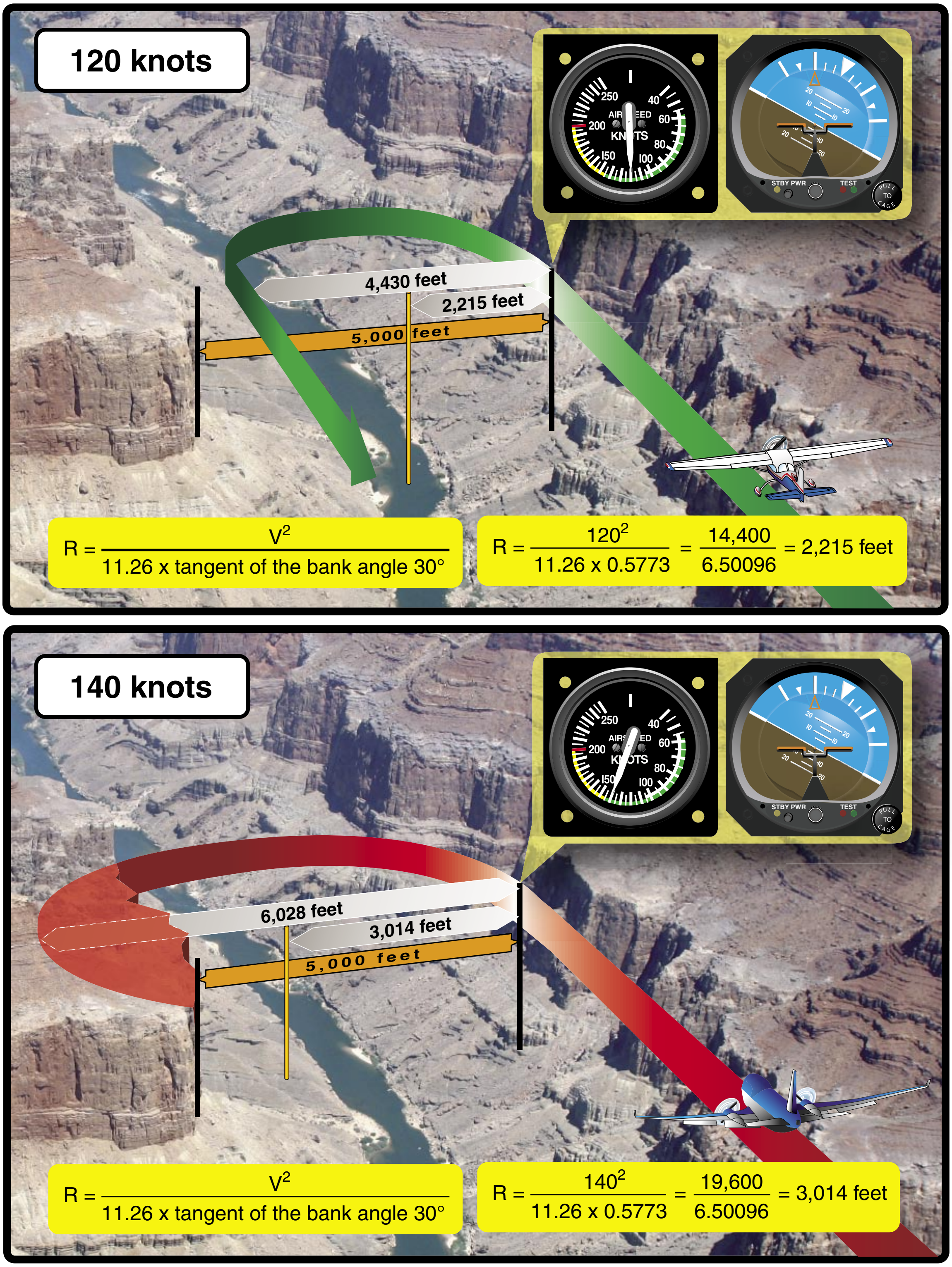


Figure 4-54. *Two aircraft have flown into a canyon by error. The canyon is 5,000 feet across and has sheer cliffs on both sides. The pilot in the top image is flying at 120 knots. After realizing the error, the pilot banks hard and uses a 30° bank angle to reverse course. This aircraft requires about 4,000 feet to turn 180°, and makes it out of the canyon safely. The pilot in the bottom image is flying at 140 knots and also uses a 30° angle of bank in an attempt to reverse course. The aircraft, although flying just 20 knots faster than the aircraft in the top image, requires over 6,000 feet to reverse course to safety. Unfortunately, the canyon is only 5,000 feet across and the aircraft will hit the canyon wall. The point is that airspeed is the most influential factor in determining how much distance is required to turn. Many pilots have made the error of increasing the steepness of their bank angle when a simple reduction of speed would have been more appropriate.*

to exceed CG limits in these aircraft, pilots should never overload an aircraft because overloading causes structural damage and failures. Weight and balance computations are not required, but pilots should calculate weight and remain within the manufacturer's established limit.

Effect of Weight on Flight Performance

The takeoff/climb and landing performance of an aircraft are determined on the basis of its maximum allowable takeoff and landing weights. A heavier gross weight results in a longer takeoff run and shallower climb, and a faster touchdown speed and longer landing roll. Even a minor overload may make it impossible for the aircraft to clear an obstacle that normally would not be a problem during takeoff under more favorable conditions.

The detrimental effects of overloading on performance are not limited to the immediate hazards involved with takeoffs and landings. Overloading has an adverse effect on all climb and cruise performance which leads to overheating during climbs, added wear on engine parts, increased fuel consumption, slower cruising speeds, and reduced range.

The manufacturers of modern aircraft furnish weight and balance data with each aircraft produced. Generally, this information may be found in the FAA-approved AFM/POH and easy-to-read charts for determining weight and balance data are now provided. Increased performance and load-carrying capability of these aircraft require strict adherence to the operating limitations prescribed by the manufacturer. Deviations from the recommendations can result in structural damage or complete failure of the aircraft's structure. Even if an aircraft is loaded well within the maximum weight limitations, it is imperative that weight distribution be within the limits of CG location. The preceding brief study of aerodynamics and load factors points out the reasons for this precaution. The following discussion is background information into some of the reasons why weight and balance conditions are important to the safe flight of an aircraft.

In some aircraft, it is not possible to fill all seats, baggage compartments, and fuel tanks, and still remain within approved weight or balance limits. For example, in several popular four-place aircraft, the fuel tanks may not be filled to capacity when four occupants and their baggage are carried. In a certain two-place aircraft, no baggage may be carried in the compartment aft of the seats when spins are to be practiced. It is important for a pilot to be aware of the weight and balance limitations of the aircraft being flown and the reasons for these limitations.

Effect of Weight on Aircraft Structure

The effect of additional weight on the wing structure of an aircraft is not readily apparent. Airworthiness requirements prescribe that the structure of an aircraft certificated in the normal category (in which acrobatics are prohibited) must be strong enough to withstand a load factor of 3.8 Gs to take care of dynamic loads caused by maneuvering and gusts. This means that the primary structure of the aircraft can withstand a load of 3.8 times the approved gross weight of the aircraft without structural failure occurring. If this is accepted as indicative of the load factors that may be imposed during operations for which the aircraft is intended, a 100-pound overload imposes a potential structural overload of 380 pounds. The same consideration is even more impressive in the case of utility and acrobatic category aircraft, which have load factor requirements of 4.4 and 6.0, respectively.

Structural failures which result from overloading may be dramatic and catastrophic, but more often they affect structural components progressively in a manner that is difficult to detect and expensive to repair. Habitual overloading tends to cause cumulative stress and damage that may not be detected during preflight inspections and result in structural failure later during completely normal operations. The additional stress placed on structural parts by overloading is believed to accelerate the occurrence of metallic fatigue failures.

A knowledge of load factors imposed by flight maneuvers and gusts emphasizes the consequences of an increase in the gross weight of an aircraft. The structure of an aircraft about to undergo a load factor of 3 Gs, as in recovery from a steep dive, must be prepared to withstand an added load of 300 pounds for each 100-pound increase in weight. It should be noted that this would be imposed by the addition of about 16 gallons of unneeded fuel in a particular aircraft. FAA-certificated civil aircraft have been analyzed structurally and tested for flight at the maximum gross weight authorized and within the speeds posted for the type of flights to be performed. Flights at weights in excess of this amount are quite possible and often are well within the performance capabilities of an aircraft. This fact should not mislead the pilot, as the pilot may not realize that loads for which the aircraft was not designed are being imposed on all or some part of the structure.

In loading an aircraft with either passengers or cargo, the structure must be considered. Seats, baggage compartments, and cabin floors are designed for a certain load or concentration of load and no more. For example, a light plane baggage compartment may be placarded for 20 pounds because of the limited strength of its supporting structure even though the aircraft may not be overloaded or out of CG limits with more weight at that location.

Effect of Weight on Stability and Controllability

Overloading also effects stability. An aircraft that is stable and controllable when loaded normally may have very different flight characteristics when overloaded. Although the distribution of weight has the most direct effect on this, an increase in the aircraft's gross weight may be expected to have an adverse effect on stability, regardless of location of the CG. The stability of many certificated aircraft is completely unsatisfactory if the gross weight is exceeded.

Effect of Load Distribution

The effect of the position of the CG on the load imposed on an aircraft's wing in flight is significant to climb and cruising performance. An aircraft with forward loading is "heavier" and consequently, slower than the same aircraft with the CG further aft.

Figure 4-55 illustrates why this is true. With forward loading, "nose-up" trim is required in most aircraft to maintain level cruising flight. Nose-up trim involves setting the tail surfaces to produce a greater down load on the aft portion of the fuselage, which adds to the wing loading and the total lift required from the wing if altitude is to be maintained. This requires a higher AOA of the wing, which results in more drag and, in turn, produces a higher stalling speed.

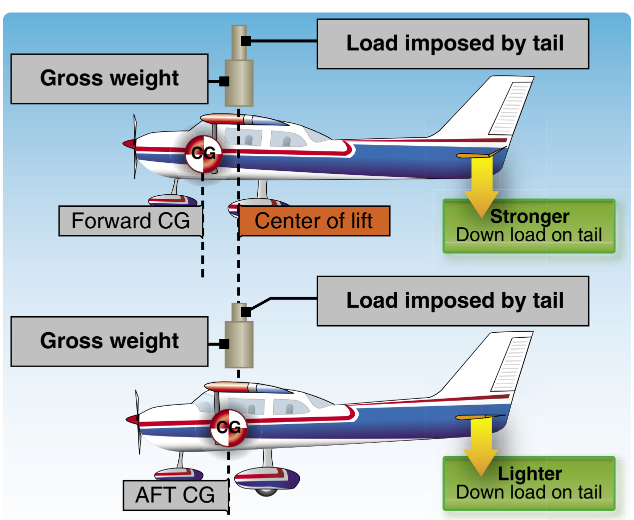


Figure 4-55. *Effect of load distribution on balance.*

With aft loading and "nose-down" trim, the tail surfaces exert less down load, relieving the wing of that much wing loading and lift required to maintain altitude. The required AOA of the wing is less, so the drag is less, allowing for a faster cruise speed. Theoretically, a neutral load on the tail surfaces in cruising flight would produce the most efficient overall performance and fastest cruising speed, but it would also result in instability. Modern aircraft are designed to require a down load on the tail for stability and controllability.

A zero indication on the trim tab control is not necessarily the same as "neutral trim" because of the force exerted by downwash from the wings and the fuselage on the tail surfaces.

The effects of the distribution of the aircraft's useful load have a significant influence on its flight characteristics, even when the load is within the CG limits and the maximum permissible gross weight. Important among these effects are changes in controllability, stability, and the actual load imposed on the wing.

Generally, an aircraft becomes less controllable, especially at slow flight speeds, as the CG is moved further aft. An aircraft which cleanly recovers from a prolonged spin with the CG at one position may fail completely to respond to normal recovery attempts when the CG is moved aft by one or two inches.

It is common practice for aircraft designers to establish an aft CG limit that is within one inch of the maximum which allows normal recovery from a one-turn spin. When certificating an aircraft in the utility category to permit intentional spins, the aft CG limit is usually established at a point several inches forward of that permissible for certification in the normal category.

Another factor affecting controllability, which has become more important in current designs of large aircraft, is the effect of long moment arms to the positions of heavy equipment and cargo. The same aircraft may be loaded to maximum gross weight within its CG limits by concentrating fuel, passengers, and cargo near the design CG, or by dispersing fuel and cargo loads in wingtip tanks and cargo bins forward and aft of the cabin.

With the same total weight and CG, maneuvering the aircraft or maintaining level flight in turbulent air requires the application of greater control forces when the load is dispersed. The longer moment arms to the positions of the heavy fuel and cargo loads must be overcome by the action of the control surfaces. An aircraft with full outboard wing tanks or tip tanks tends to be sluggish in roll when control situations are marginal, while one with full nose and aft cargo bins tends to be less responsive to the elevator controls.

The rearward CG limit of an aircraft is determined largely by considerations of stability. The original airworthiness requirements for a type certificate specify that an aircraft in flight at a certain speed dampens out vertical displacement of the nose within a certain number of oscillations. An aircraft loaded too far rearward may not do this. Instead, when the nose is momentarily pulled up, it may alternately climb and dive becoming steeper with each oscillation. This instability is not only uncomfortable to occupants, but it could even become dangerous by making the aircraft unmanageable under certain conditions.

The recovery from a stall in any aircraft becomes progressively more difficult as its CG moves aft. This is particularly important in spin recovery, as there is a point in rearward loading of any aircraft at which a "flat" spin develops. A flat spin is one in which centrifugal force, acting through a CG located well to the rear, pulls the tail of the aircraft out away from the axis of the spin, making it impossible to get the nose down and recover.

An aircraft loaded to the rear limit of its permissible CG range handles differently in turns and stall maneuvers and has different landing characteristics than when it is loaded near the forward limit.

The forward CG limit is determined by a number of considerations. As a safety measure, it is required that the trimming device, whether tab or adjustable stabilizer, be capable of holding the aircraft in a normal glide with the power off. A conventional aircraft must be capable of a full stall, power-off landing in order to ensure minimum landing speed in emergencies. A tailwheel-type aircraft loaded excessively nose-heavy is difficult to taxi, particularly in high winds. It can be nosed over easily by use of the brakes, and it is difficult to land without bouncing since it tends to pitch down on the wheels as it is slowed down and flared for landing. Steering difficulties on the ground may occur in nosewheel-type aircraft, particularly during the landing roll and takeoff. To summarize the effects of load distribution:

- The CG position influences the lift and AOA of the wing, the amount and direction of force on the tail, and the degree of deflection of the stabilizer needed to supply the proper tail force for equilibrium. The latter is very important because of its relationship to elevator control force.
- The aircraft stalls at a higher speed with a forward CG location. This is because the stalling AOA is reached at a higher speed due to increased wing loading.
- Higher elevator control forces normally exist with a forward CG location due to the increased stabilizer deflection required to balance the aircraft.
- The aircraft cruises faster with an aft CG location because of reduced drag. The drag is reduced because a smaller AOA and less downward deflection of the stabilizer are required to support the aircraft and overcome the nose-down pitching tendency.
- The aircraft becomes less stable as the CG is moved rearward. This is because when the CG is moved rearward it causes an increase in the AOA. Therefore, the wing contribution to the aircraft's stability is now decreased, while the tail contribution is still stabilizing. When the point is reached that the wing and tail contributions balance, then neutral stability exists. Any CG movement further aft results in an unstable aircraft.
- A forward CG location increases the need for greater back elevator pressure. The elevator may no longer be able to oppose any increase in nose-down pitching. Adequate elevator control is needed to control the aircraft throughout the airspeed range down to the stall.

A detailed discussion and additional information relating to weight and balance can be found in Chapter 9, Weight and Balance.

High Speed Flight

Subsonic Versus Supersonic Flow

In subsonic aerodynamics, the theory of lift is based upon the forces generated on a body and a moving gas (air) in which it is immersed. At speeds of approximately 260 knots, air can be considered incompressible in that, at a fixed altitude, its density remains nearly constant while its pressure varies. Under this assumption, air acts the same as water and is classified as a fluid. Subsonic aerodynamic theory also assumes the effects of viscosity (the property of a fluid that tends to prevent motion of one part of the fluid with respect to another) are negligible, and classifies air as an ideal fluid, conforming to the principles of ideal-fluid aerodynamics such as continuity, Bernoulli's principle, and circulation.

In reality, air is compressible and viscous. While the effects of these properties are negligible at low speeds, compressibility effects in particular become increasingly important as speed increases. Compressibility (and to a lesser extent viscosity) is of paramount importance at speeds approaching the speed of sound. In these speed ranges, compressibility causes a change in the density of the air around an aircraft.

During flight, a wing produces lift by accelerating the airflow over the upper surface. This accelerated air can, and does, reach sonic speeds even though the aircraft itself may be flying subsonic. At some extreme AOAs, in some aircraft, the speed of the air over the top surface of the wing may be double the aircraft's speed. It is therefore entirely possible to have both supersonic and subsonic airflow on an aircraft at the same time. When flow velocities reach sonic speeds at some location on an aircraft (such as the area of maximum camber on the wing), further acceleration results in the onset of compressibility effects such as shock wave formation, drag increase, buffeting, stability, and control difficulties. Subsonic flow principles are invalid at all speeds above this point. *[Figure 4-56]*

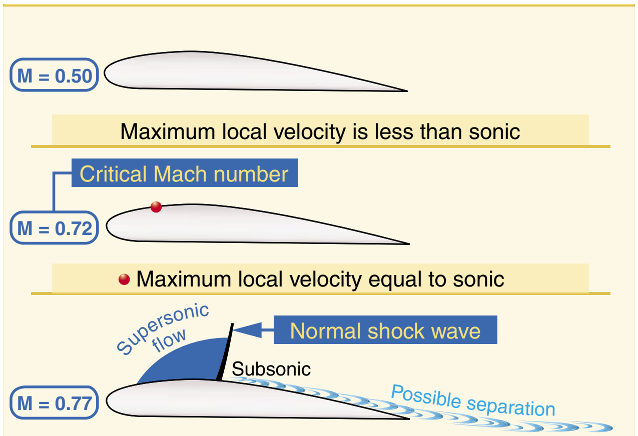


Figure 4-56. *Wing airflow.*

Speed Ranges

The speed of sound varies with temperature. Under standard temperature conditions of 15 °C, the speed of sound at sea level is 661 knots. At 40,000 feet, where the temperature is –55 °C, the speed of sound decreases to 574 knots. In high-speed flight and/or high-altitude flight, the measurement of speed is expressed in terms of a "Mach number"—the ratio of the true airspeed of the aircraft to the speed of sound in the same atmospheric conditions. An aircraft traveling at the speed of sound is traveling at Mach 1.0. Aircraft speed regimes are defined approximately as follows:

Subsonic—Mach numbers below 0.75

Transonic—Mach numbers from 0.75 to 1.20

Supersonic—Mach numbers from 1.20 to 5.00

Hypersonic—Mach numbers above 5.00

While flights in the transonic and supersonic ranges are common occurrences for military aircraft, civilian jet aircraft normally operate in a cruise speed range of Mach 0.7 to Mach 0.90.

The speed of an aircraft in which airflow over any part of the aircraft or structure under consideration first reaches (but does not exceed) Mach 1.0 is termed "critical Mach number" or "Mach Crit." Thus, critical Mach number is the boundary between subsonic and transonic flight and is largely dependent on the wing and airfoil design. Critical Mach number is an important point in transonic flight. When shock waves form on the aircraft, airflow separation followed by buffet and aircraft control difficulties can occur. Shock waves, buffet, and airflow separation take place above critical Mach number. A jet aircraft typically is most efficient when cruising at or near its critical Mach number. At speeds 5–10 percent above the critical Mach number, compressibility effects begin. Drag begins to rise sharply. Associated with the "drag rise" are buffet, trim and stability changes, and a decrease in control surface effectiveness. This is the point of "drag divergence." *[Figure 4-57]*

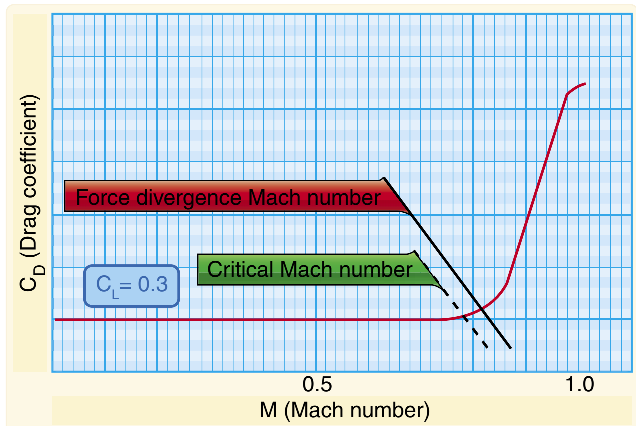


Figure 4-57. *Critical Mach.*

V_{MO}/M_{MO} is defined as the maximum operating limit speed. V_{MO} is expressed in knots calibrated airspeed (KCAS), while M_{MO} is expressed in Mach number. The V_{MO} limit is usually associated with operations at lower altitudes and deals with structural loads and flutter. The M_{MO} limit is associated with operations at higher altitudes and is usually more concerned with compressibility effects and flutter. At lower altitudes, structural loads and flutter are of concern; at higher altitudes, compressibility effects and flutter are of concern.

Adherence to these speeds prevents structural problems due to dynamic pressure or flutter, degradation in aircraft control response due to compressibility effects (e.g., Mach Tuck, aileron reversal, or buzz), and separated airflow due to shock waves resulting in loss of lift or vibration and buffet. Any of these phenomena could prevent the pilot from being able to adequately control the aircraft.

For example, an early civilian jet aircraft had a V_{MO} limit of 306 KCAS up to approximately FL 310 (on a standard day). At this altitude (FL 310), an M_{MO} of 0.82 was approximately equal to 306 KCAS. Above this altitude, an M_{MO} of 0.82 always equaled a KCAS less than 306 KCAS and, thus, became the operating limit as you could not reach the V_{MO} limit without first reaching the M_{MO} limit. For example, at FL 380, an M_{MO} of 0.82 is equal to 261 KCAS.

Mach Number Versus Airspeed

It is important to understand how airspeed varies with Mach number. As an example, consider how the stall speed of a jet transport aircraft varies with an increase in altitude. The increase in altitude results in a corresponding drop in air density and outside temperature. Suppose this jet transport is in the clean configuration (gear and flaps up) and weighs 550,000 pounds. The aircraft might stall at approximately 152 KCAS at sea level. This is equal to (on a standard day) a true velocity of 152 KTAS and a Mach number of 0.23. At FL 380, the aircraft will still stall at approximately 152 KCAS but the true velocity is about 287 KTAS with a Mach number of 0.50.

Although the stalling speed has remained the same for our purposes, both the Mach number and TAS have increased. With increasing altitude, the air density has decreased; this requires a faster true airspeed in order to have the same pressure sensed by the pitot tube for the same KCAS or KIAS (for our purposes, KCAS and KIAS are relatively close to each other). The dynamic pressure the wing experiences at FL 380 at 287 KTAS is the same as at sea level at 152 KTAS. However, it is flying at higher Mach number.

Another factor to consider is the speed of sound. A decrease in temperature in a gas results in a decrease in the speed of sound. Thus, as the aircraft climbs in altitude with outside temperature dropping, the speed of sound is dropping. At sea level, the speed of sound is approximately 661 KCAS, while at FL 380 it is 574 KCAS. Thus, for our jet transport aircraft, the stall speed (in KTAS) has gone from 152 at sea level to 287 at FL 380. Simultaneously, the speed of sound (in KCAS) has decreased from 661 to 574 and the Mach number has increased from 0.23 (152 KTAS divided by 661 KTAS) to 0.50 (287 KTAS divided by 574 KTAS). All the while the KCAS for stall has remained constant at 152. This describes what happens when the aircraft is at a constant KCAS with increasing altitude, but what happens when the pilot keeps Mach constant during the climb? In normal jet flight operations, the climb is at 250 KIAS (or higher (e.g. heavy)) to 10,000 feet and then at a specified en route climb airspeed (such as about 330 if a DC10) until reaching an altitude in the "mid-twenties" where the pilot then climbs at a constant Mach number to cruise altitude.

Assuming for illustration purposes that the pilot climbs at a M_{MO} of 0.82 from sea level up to FL 380. KCAS goes from 543 to 261. The KIAS at each altitude would follow the same behavior and just differ by a few knots. Recall from the earlier discussion that the speed of sound is decreasing with the drop in temperature as the aircraft climbs. The Mach number is simply the ratio of the true airspeed to the speed of sound at flight conditions. The significance of this is that at a constant Mach number climb, the KCAS (and KTAS or KIAS as well) is falling off.

If the aircraft climbed high enough at this constant M_{MO} with decreasing KIAS, KCAS, and KTAS, it would begin to approach its stall speed. At some point the stall speed of the aircraft in Mach number could equal the M_{MO} of the aircraft, and the pilot could neither slow up (without stalling) nor speed up (without exceeding the max operating speed of the aircraft). This has been dubbed the "coffin corner."

Boundary Layer

The viscous nature of airflow reduces the local velocities on a surface and is responsible for skin friction. As discussed earlier in the chapter, the layer of air over the wing's surface that is slowed down or stopped by viscosity, is the boundary layer. There are two different types of boundary layer flow: laminar and turbulent.

Laminar Boundary Layer Flow

The laminar boundary layer is a very smooth flow, while the turbulent boundary layer contains swirls or "eddies." The laminar flow creates less skin friction drag than the turbulent flow, but is less stable. Boundary layer flow over a wing surface begins as a smooth laminar flow. As the flow continues back from the leading edge, the laminar boundary layer increases in thickness.

Turbulent Boundary Layer Flow

At some distance back from the leading edge, the smooth laminar flow breaks down and transitions to a turbulent flow. From a drag standpoint, it is advisable to have the transition from laminar to turbulent flow as far aft on the wing as possible, or have a large amount of the wing surface within the laminar portion of the boundary layer. The low energy laminar flow, however, tends to break down more suddenly than the turbulent layer.

Boundary Layer Separation

Another phenomenon associated with viscous flow is separation. Separation occurs when the airflow breaks away from an airfoil. The natural progression is from laminar boundary layer to turbulent boundary layer and then to airflow separation. Airflow separation produces high drag and ultimately destroys lift. The boundary layer separation point moves forward on the wing as the AOA is increased. *[Figure 4-58]*

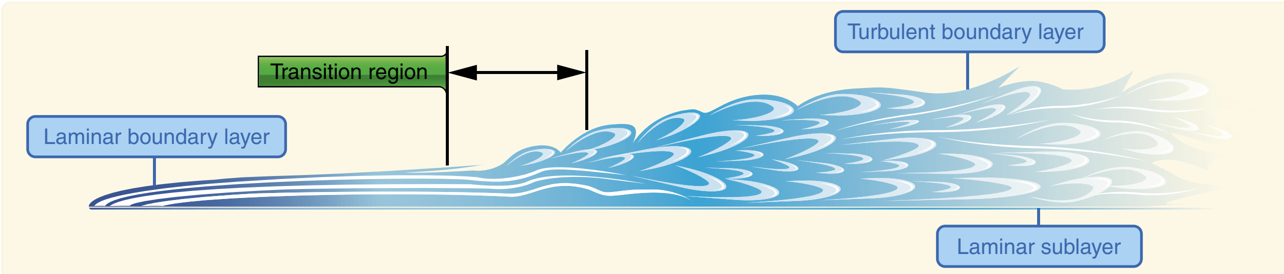


Figure 4-58. *Boundary layer.*

Vortex generators are used to delay or prevent shock wave induced boundary layer separation encountered in transonic flight. They are small low aspect ratio airfoils placed at a 12° to 15° AOA to the airstream. Usually spaced a few inches apart along the wing ahead of the ailerons or other control surfaces, vortex generators create a vortex which mixes the boundary airflow with the high energy airflow just above the surface. This produces higher surface velocities and increases the energy of the boundary layer. Thus, a stronger shock wave is necessary to produce airflow separation.

Shock Waves

When an airplane flies at subsonic speeds, the air ahead is "warned" of the airplane's coming by a pressure change transmitted ahead of the airplane at the speed of sound. Because of this warning, the air begins to move aside before the airplane arrives and is prepared to let it pass easily. When the airplane's speed reaches the speed of sound, the pressure change can no longer warn the air ahead because the airplane is keeping up with its own pressure waves. Rather, the air particles pile up in front of the airplane causing a sharp decrease in the flow velocity directly in front of the airplane with a corresponding increase in air pressure and density.

As the airplane's speed increases beyond the speed of sound, the pressure and density of the compressed air ahead of it increase, the area of compression extending some distance ahead of the airplane. At some point in the airstream, the air particles are completely undisturbed, having had no advanced warning of the airplane's approach, and in the next instant the same air particles are forced to undergo sudden and drastic changes in temperature, pressure, density, and velocity. The boundary between the undisturbed air and the region of compressed air is called a shock or "compression" wave. This same type of wave is formed whenever a supersonic airstream is slowed to subsonic without a change in direction, such as when the airstream is accelerated to sonic speed over the cambered portion of a wing, and then decelerated to subsonic speed as the area of maximum camber is passed. A shock wave forms as a boundary between the supersonic and subsonic ranges.

Whenever a shock wave forms perpendicular to the airflow, it is termed a "normal" shock wave, and the flow immediately behind the wave is subsonic. A supersonic airstream passing through a normal shock wave experiences these changes:

- The airstream is slowed to subsonic.
- The airflow immediately behind the shock wave does not change direction.
- The static pressure and density of the airstream behind the wave is greatly increased.
- The energy of the airstream (indicated by total pressure—dynamic plus static) is greatly reduced.

Shock wave formation causes an increase in drag. One of the principal effects of a shock wave is the formation of a dense high pressure region immediately behind the wave. The instability of the high pressure region, and the fact that part of the velocity energy of the airstream is converted to heat as it flows through the wave is a contributing factor in the drag increase, but the drag resulting from airflow separation is much greater. If the shock wave is strong, the boundary layer may not have sufficient kinetic energy to withstand airflow separation. The drag incurred in the transonic region due to shock wave formation and airflow separation is known as "wave drag." When speed exceeds the critical Mach number by about 10 percent, wave drag increases sharply. A considerable increase in thrust (power) is required to increase flight speed beyond this point into the supersonic range where, depending on the airfoil shape and the angle of attack, the boundary layer may reattach.

Normal shock waves form on the wing's upper surface and form an additional area of supersonic flow and a normal shock wave on the lower surface. As flight speed approaches the speed of sound, the areas of supersonic flow enlarge and the shock waves move nearer the trailing edge. *[Figure 4-59]*

Associated with "drag rise" are buffet (known as Mach buffet), trim and stability changes, and a decrease in control

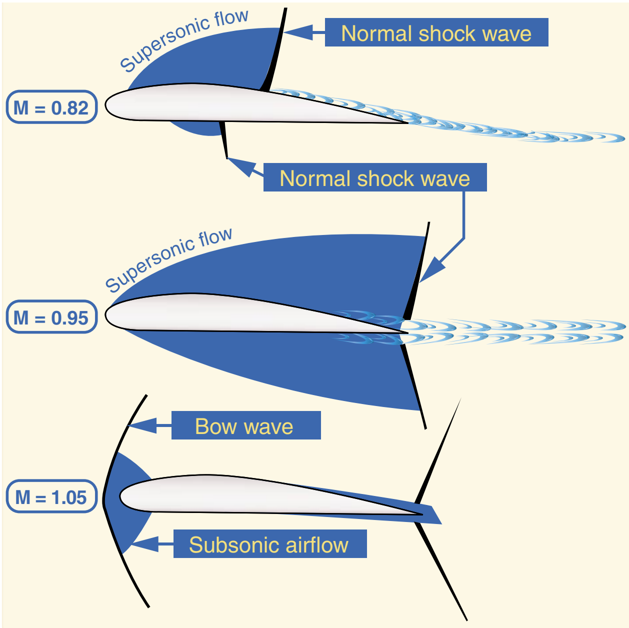


Figure 4-59. *Shock waves.*

force effectiveness. The loss of lift due to airflow separation results in a loss of downwash, and a change in the position of the center pressure on the wing. Airflow separation produces a turbulent wake behind the wing, which causes the tail surfaces to buffet (vibrate). The nose-up and nose-down pitch control provided by the horizontal tail is dependent on the downwash behind the wing. Thus, an increase in downwash decreases the horizontal tail's pitch control effectiveness since it effectively increases the angle of attack that the tail surface is seeing. Movement of the wing CP affects the wing pitching moment. If the CP moves aft, a diving moment referred to as "Mach tuck" or "tuck under" is produced, and if it moves forward, a nose-up moment is produced. This is the primary reason for the development of the T-tail configuration on many turbine-powered aircraft, which places the horizontal stabilizer as far as practical from the turbulence of the wings.

Sweepback

Most of the difficulties of transonic flight are associated with shock wave induced flow separation. Therefore, any means of delaying or alleviating the shock induced separation improves aerodynamic performance. One method is wing sweepback. Sweepback theory is based upon the concept that it is only the component of the airflow perpendicular to the leading edge of the wing that affects pressure distribution and formation of shock waves. *[Figure 4-60]*

On a straight wing aircraft, the airflow strikes the wing leading edge at 90°, and its full impact produces pressure and lift. A wing with sweepback is struck by the same airflow at an angle smaller than 90°. This airflow on the swept wing has

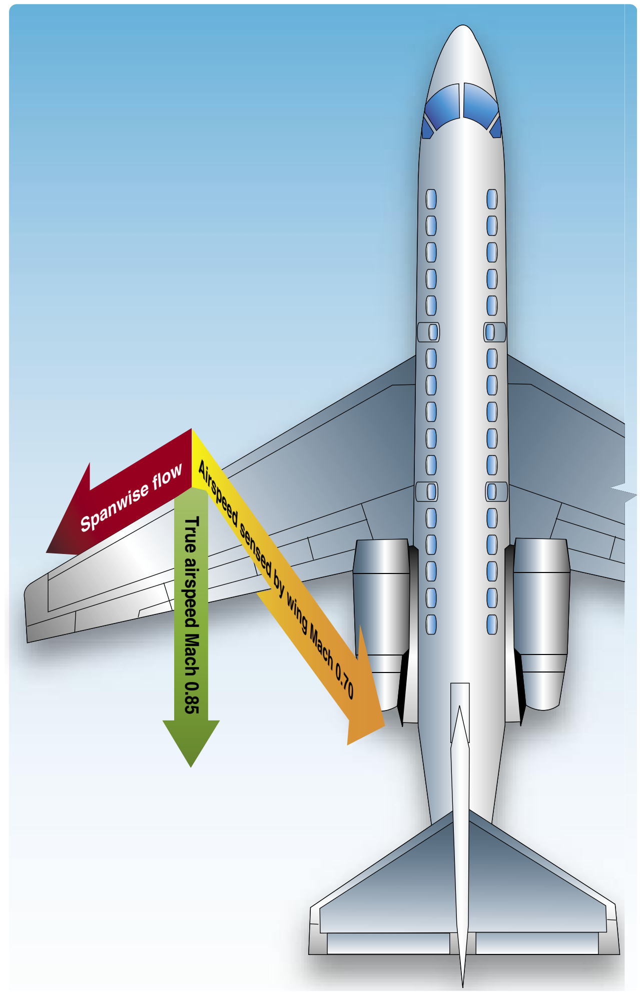


Figure 4-60. *Sweepback effect.*

the effect of persuading the wing into believing that it is flying slower than it really is; thus the formation of shock waves is delayed. Advantages of wing sweep include an increase in critical Mach number, force divergence Mach number, and the Mach number at which drag rises peaks. In other words, sweep delays the onset of compressibility effects.

The Mach number, which produces a sharp change in drag coefficient, is termed the "force divergence" Mach number and, for most airfoils, usually exceeds the critical Mach number by 5 to 10 percent. At this speed, the airflow separation induced by shock wave formation can create significant variations in the drag, lift, or pitching moment coefficients. In addition to the delay of the onset of compressibility effects, sweepback reduces the magnitude in the changes of drag, lift or moment coefficients. In other words, the use of sweepback "softens" the force divergence.

A disadvantage of swept wings is that they tend to stall at the wingtips rather than at the wing roots. *[Figure 4-61]* This is because the boundary layer tends to flow spanwise toward the tips and to separate near the leading edges. Because the tips of a swept wing are on the aft part of the wing (behind the CL), a wingtip stall causes the CL to move forward on the wing, forcing the nose to rise further. The tendency for tip stall is greatest when wing sweep and taper are combined.

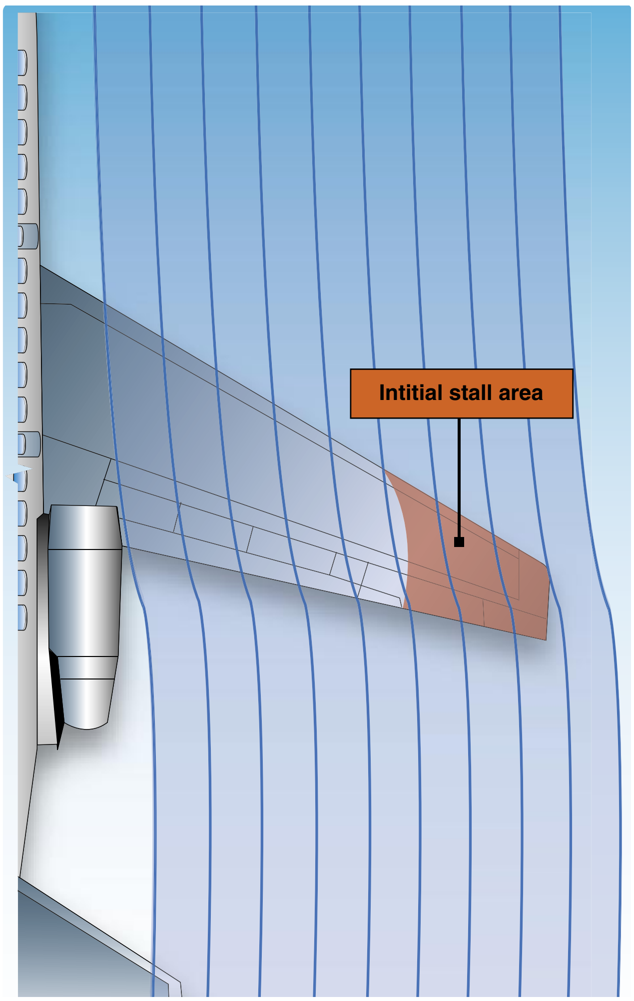


Figure 4-61. *Wingtip stall.*

The stall situation can be aggravated by a T-tail configuration, which affords little or no pre-stall warning in the form of tail control surface buffet. *[Figure 4-62]* The T-tail, being above the wing wake remains effective even after the wing has begun to stall, allowing the pilot to inadvertently drive the wing into a deeper stall at a much greater AOA. If the horizontal tail surfaces then become buried in the wing's wake, the elevator may lose all effectiveness, making it impossible to reduce pitch attitude and break the stall. In the pre-stall and immediate post-stall regimes, the lift/drag qualities of a swept wing aircraft (specifically the enormous increase in drag at low speeds) can cause an increasingly descending flightpath with no change in pitch attitude, further increasing the AOA. In this situation, without reliable AOA information, a nose-down pitch attitude with an increasing airspeed is no guarantee that recovery has been effected, and up-elevator movement at this stage may merely keep the aircraft stalled.

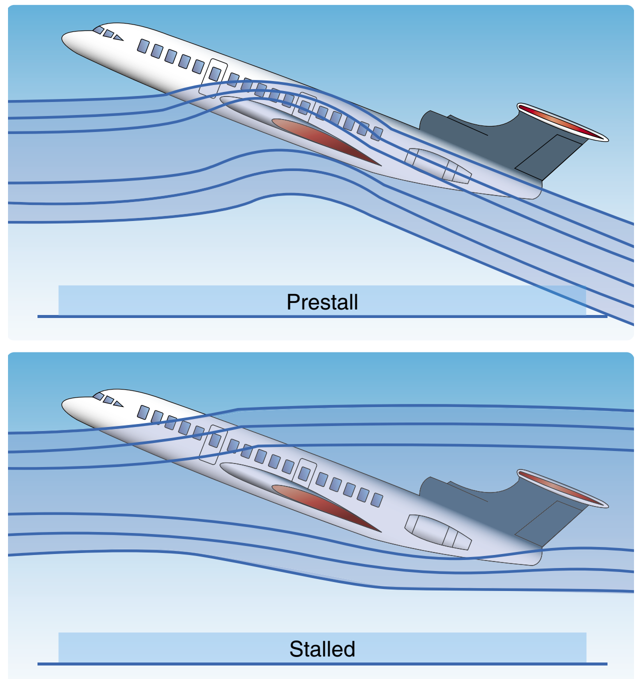


Figure 4-62. *T-tail stall.*

It is a characteristic of T-tail aircraft to pitch up viciously when stalled in extreme nose-high attitudes, making recovery difficult or violent. The stick pusher inhibits this type of stall. At approximately one knot above stall speed, pre-programmed stick forces automatically move the stick forward, preventing the stall from developing. A G-limiter may also be incorporated into the system to prevent the pitch down generated by the stick pusher from imposing excessive loads on the aircraft. A "stick shaker," on the other hand provides stall warning when the airspeed is five to seven percent above stall speed.

Mach Buffet Boundaries

Mach buffet is a function of the speed of the airflow over the wing—not necessarily the speed of the aircraft. Any time that too great a lift demand is made on the wing, whether from too fast an airspeed or from too high an AOA near the MMO, the "high-speed" buffet occurs. There are also occasions when the buffet can be experienced at much lower speeds known as the "low-speed Mach buffet."

An aircraft flown at a speed too slow for its weight and altitude necessitating a high AOA is the most likely situation to cause a low-speed Mach buffet. This very high AOA has the effect of increasing airflow velocity over the upper surface of the wing until the same effects of the shock waves and buffet occur as in the high-speed buffet situation. The AOA of the wing has the greatest effect on inducing the Mach buffet at either the high-speed or low-speed boundaries for the aircraft. The conditions that increase the AOA, the speed of the airflow over the wing, and chances of Mach buffet are:

- High altitudes—the higher an aircraft flies, the thinner the air and the greater the AOA required to produce the lift needed to maintain level flight.
- Heavy weights—the heavier the aircraft, the greater the lift required of the wing, and all other things being equal, the greater the AOA.
- G loading—an increase in the G loading on the aircraft has the same effect as increasing the weight of the aircraft. Whether the increase in G forces is caused by turns, rough control usage, or turbulence, the effect of increasing the wing's AOA is the same.

High Speed Flight Controls

On high-speed aircraft, flight controls are divided into primary flight controls and secondary or auxiliary flight controls. The primary flight controls maneuver the aircraft about the pitch, roll, and yaw axes. They include the ailerons, elevator, and rudder. Secondary or auxiliary flight controls include tabs, leading edge flaps, trailing edge flaps, spoilers, and slats.

Spoilers are used on the upper surface of the wing to spoil or reduce lift. High speed aircraft, due to their clean low drag design use spoilers as speed brakes to slow them down. Spoilers are extended immediately after touchdown to dump lift and thus transfer the weight of the aircraft from the wings onto the wheels for better braking performance. *[Figure 4-63]*

Jet transport aircraft have small ailerons. The space for ailerons is limited because as much of the wing trailing edge as possible is needed for flaps. Also, a conventional size aileron would cause wing twist at high speed. For that reason, spoilers are used in unison with ailerons to provide additional roll control.

Some jet transports have two sets of ailerons, a pair of outboard low-speed ailerons and a pair of high-speed inboard ailerons. When the flaps are fully retracted after takeoff, the outboard ailerons are automatically locked out in the faired position.

When used for roll control, the spoiler on the side of the up-going aileron extends and reduces the lift on that side, causing the wing to drop. If the spoilers are extended as speed brakes, they can still be used for roll control. If they are the differential type, they extend further on one side and retract on the other side. If they are the non-differential type, they extend further on one side but do not retract on the other side. When fully extended as speed brakes, the non-differential spoilers remain extended and do not supplement the ailerons.

To obtain a smooth stall and a higher AOA without airflow separation, the wing's leading edge should have a well-rounded almost blunt shape that the airflow can adhere to at the higher AOA. With this shape, the airflow separation starts at the trailing edge and progresses forward gradually as AOA is increased.

The pointed leading edge necessary for high-speed flight results in an abrupt stall and restricts the use of trailing edge flaps because the airflow cannot follow the sharp curve around the wing leading edge. The airflow tends to tear loose rather suddenly from the upper surface at a moderate AOA. To utilize trailing edge flaps, and thus increase the $C_{L\text{-}MAX}$, the wing must go to a higher AOA without airflow separation. Therefore, leading edge slots, slats, and flaps are used to improve the low-speed characteristics during takeoff, climb, and landing. Although these devices are not as powerful as trailing edge flaps, they are effective when used full span in combination with high-lift trailing edge flaps. With the aid of these sophisticated high-lift devices, airflow separation is delayed and the $C_{L\text{-}MAX}$ is increased considerably. In fact, a 50 knot reduction in stall speed is not uncommon.

The operational requirements of a large jet transport aircraft necessitate large pitch trim changes. Some requirements are:

- A large CG range
- A large speed range
- The ability to perform large trim changes due to wing leading edge and trailing edge high-lift devices without limiting the amount of elevator remaining
- Maintaining trim drag to a minimum

These requirements are met by the use of a variable incidence horizontal stabilizer. Large trim changes on a fixed-tail aircraft require large elevator deflections. At these large deflections, little further elevator movement remains in the same direction. A variable incidence horizontal stabilizer is designed to take out the trim changes. The stabilizer is larger than the elevator, and consequently does not need to be moved through as large an angle. This leaves the elevator streamlining the tail plane with a full range of movement up and down. The variable incidence horizontal stabilizer can

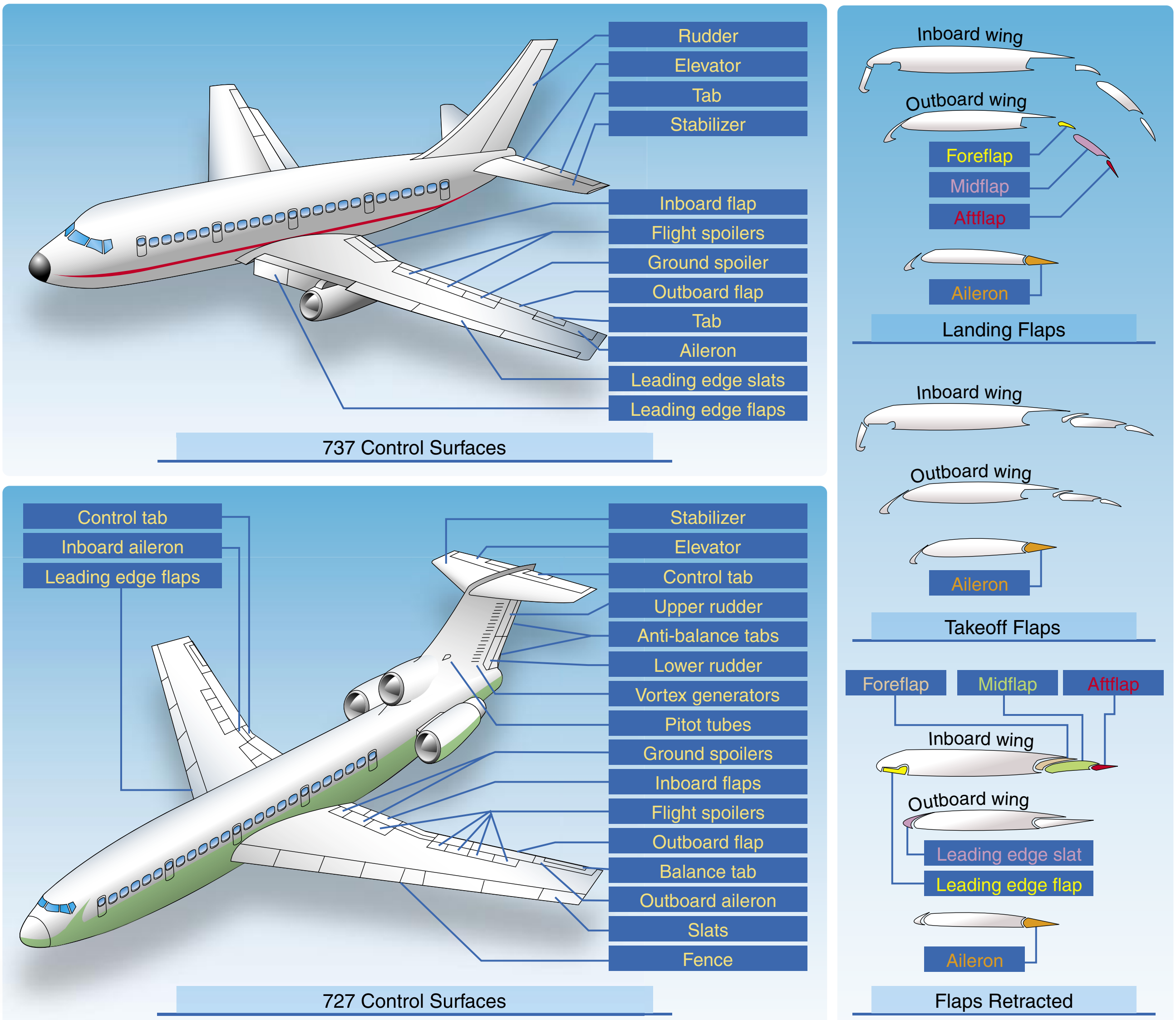


Figure 4-63. *Control surfaces.*

be set to handle the bulk of the pitch control demand, with the elevator handling the rest. On aircraft equipped with a variable incidence horizontal stabilizer, the elevator is smaller and less effective in isolation than it is on a fixed-tail aircraft. In comparison to other flight controls, the variable incidence horizontal stabilizer is enormously powerful in its effect.

Because of the size and high speeds of jet transport aircraft, the forces required to move the control surfaces can be beyond the strength of the pilot. Consequently, the control surfaces are actuated by hydraulic or electrical power units. Moving the controls in the flight deck signals the control angle required, and the power unit positions the actual control surface. In the event of complete power unit failure, movement of the control surface can be effected by manually controlling the control tabs. Moving the control tab upsets the aerodynamic balance which causes the control surface to move.

Chapter Summary

In order to sustain an aircraft in flight, a pilot must understand how thrust, drag, lift, and weight act on the aircraft. By understanding the aerodynamics of flight, how design, weight, load factors, and gravity affect an aircraft during flight maneuvers from stalls to high speed flight, the pilot learns how to control the balance between these forces. For information on stall speeds, load factors, and other important aircraft data, always consult the AFM/POH for specific information pertaining to the aircraft being flown.

Chapter 10

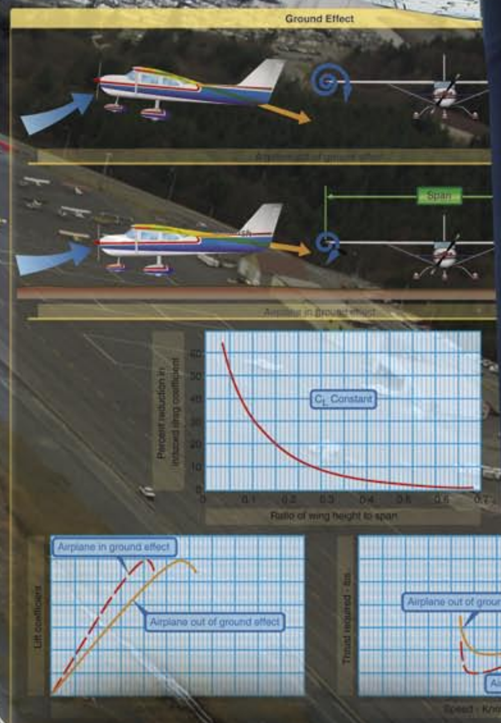
Aircraft Performance

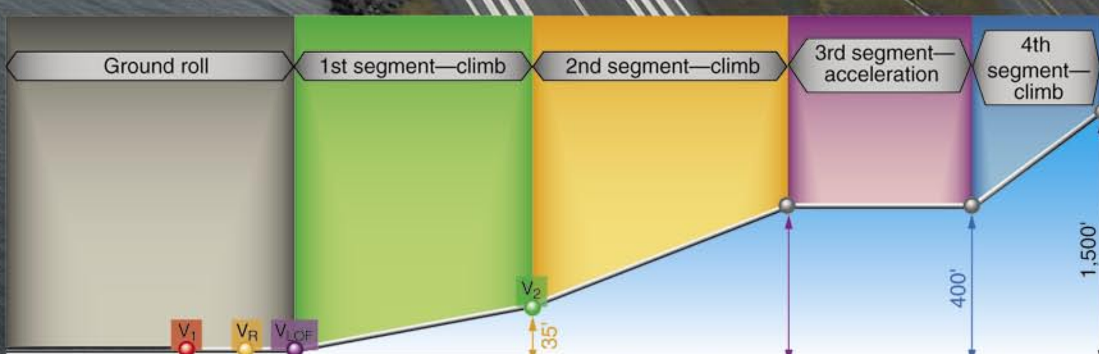
Introduction

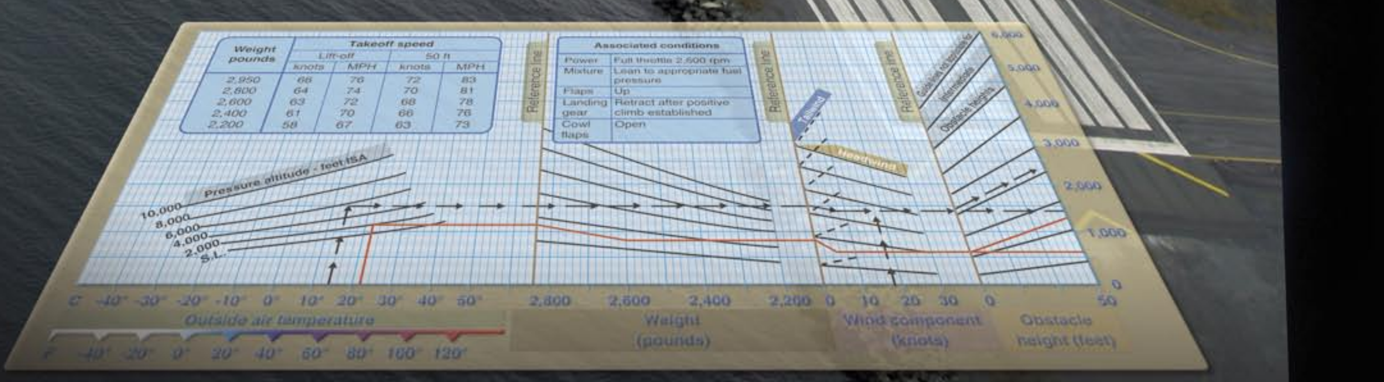
This chapter discusses the factors that affect aircraft performance, which include the aircraft weight, atmospheric conditions, runway environment, and the fundamental physical laws governing the forces acting on an aircraft.

Importance of Performance Data

The performance or operational information section of the Aircraft Flight Manual/Pilot's Operating Handbook (AFM/POH) contains the operating data for the aircraft; that is, the data pertaining to takeoff, climb, range, endurance, descent, and landing. The use of this data in flying operations is mandatory for safe and efficient operation. Considerable knowledge and familiarity of the aircraft can be gained through study of this material.







It must be emphasized that the manufacturers' information and data furnished in the AFM/POH is not standardized. Some provide the data in tabular form, while others use graphs. In addition, the performance data may be presented on the basis of standard atmospheric conditions, pressure altitude, or density altitude. The performance information in the AFM/POH has little or no value unless the user recognizes those variations and makes the necessary adjustments.

To be able to make practical use of the aircraft's capabilities and limitations, it is essential to understand the significance of the operational data. The pilot must be cognizant of the basis for the performance data, as well as the meanings of the various terms used in expressing performance capabilities and limitations.

Since the characteristics of the atmosphere have a major effect on performance, it is necessary to review two dominant factors—pressure and temperature.

Structure of the Atmosphere

The atmosphere is an envelope of air that surrounds the Earth and rests upon its surface. It is as much a part of the Earth as its land and water. However, air differs from land and water inasmuch as it is a mixture of gases. It has mass, weight, and indefinite shape.

Air, like any other fluid, is able to flow and change its shape when subjected to even minute pressures because of the lack of strong molecular cohesion. For example, gas will completely fill any container into which it is placed, expanding or contracting to adjust its shape to the limits of the container.

The atmosphere is composed of 78 percent nitrogen, 21 percent oxygen, and 1 percent other gases, such as argon or helium. Most of the oxygen is contained below 35,000 feet altitude.

Atmospheric Pressure

Though there are various kinds of pressure, pilots are mainly concerned with atmospheric pressure. It is one of the basic factors in weather changes, helps to lift the aircraft, and actuates some of the most important flight instruments in the aircraft. These instruments often include the altimeter, the airspeed indicator (ASI), the vertical speed indicator, and the manifold pressure gauge.

Though air is very light, it has mass and is affected by the attraction of gravity. Therefore, like any other substance, it has weight; because it has weight, it has force. Since it is a fluid substance, this force is exerted equally in all directions, and its effect on bodies within the air is called pressure. Under standard conditions at sea level, the average pressure exerted by the weight of the atmosphere is approximately 14.7 pounds per square inch (psi). The density of air has significant effects on the aircraft's performance. As air becomes less dense, it reduces:

- Power, because the engine takes in less air.
- Thrust, because the propeller is less efficient in thin air.
- Lift, because the thin air exerts less force on the airfoils.

The pressure of the atmosphere varies with time and altitude. Due to the changing atmospheric pressure, a standard reference was developed. The standard atmosphere at sea level is a surface temperature of 59 degrees Fahrenheit (°F) or 15 degrees Celsius (°C) and a surface pressure of 29.92 inches of mercury ("Hg) or 1013.2 millibars (mb). *[Figure 10-1]*

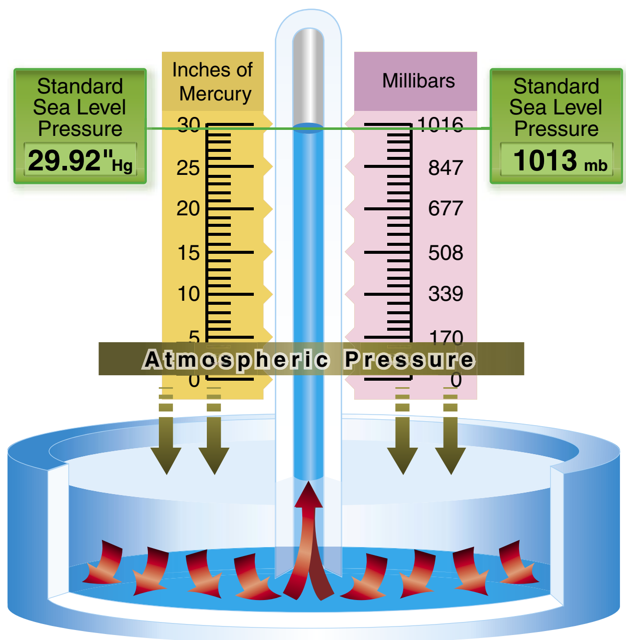


Figure 10-1. *Standard sea level pressure.*

A standard temperature lapse rate is one in which the temperature decreases at the rate of approximately 3.5 °F or 2 °C per thousand feet up to 36,000 feet. Above this point, the temperature is considered constant up to 80,000 feet. A standard pressure lapse rate is one in which pressure decreases at a rate of approximately 1 "Hg per 1,000 feet of altitude gain to 10,000 feet. *[Figure 10-2]* The International Civil Aviation Organization (ICAO) has established this as a worldwide standard, and it is often referred to as International Standard Atmosphere (ISA) or ICAO Standard Atmosphere. Any temperature or pressure that differs from the standard lapse rates is considered nonstandard temperature and pressure. Adjustments for nonstandard temperatures and pressures are provided on the manufacturer's performance charts.

Altitude (ft)	Pressure ("Hg)	Temperature (°C)	Temperature (°F)
0	29.92	15.0	59.0
1,000	28.86	13.0	55.4
2,000	27.82	11.0	51.9
3,000	26.82	9.1	48.3
4,000	25.84	7.1	44.7
5,000	24.89	5.1	41.2
6,000	23.98	3.1	37.6
7,000	23.09	1.1	34.0
8,000	22.22	–0.9	30.5
9,000	21.38	–2.8	26.9
10,000	20.57	–4.8	23.3
11,000	19.79	–6.8	19.8
12,000	19.02	–8.8	16.2
13,000	18.29	–10.8	12.6
14,000	17.57	–12.7	9.1
15,000	16.88	–14.7	5.5
16,000	16.21	–16.7	1.9
17,000	15.56	–18.7	–1.6
18,000	14.94	–20.7	–5.2
19,000	14.33	–22.6	–8.8
20,000	13.74	–24.6	–12.3

Figure 10-2. *Properties of standard atmosphere.*

Since all aircraft performance is compared and evaluated with respect to the standard atmosphere, all aircraft instruments are calibrated for the standard atmosphere. Thus, certain corrections must apply to the instrumentation, as well as the aircraft performance, if the actual operating conditions do not fit the standard atmosphere. In order to account properly for the nonstandard atmosphere, certain related terms must be defined.

Pressure Altitude

Pressure altitude is the height above the standard datum plane (SDP). The aircraft altimeter is essentially a sensitive barometer calibrated to indicate altitude in the standard atmosphere. If the altimeter is set for 29.92 "Hg SDP, the altitude indicated is the pressure altitude—the altitude in the standard atmosphere corresponding to the sensed pressure.

The SDP is a theoretical level where the pressure of the atmosphere is 29.92 "Hg and the weight of air is 14.7 psi. As atmospheric pressure changes, the SDP may be below, at, or above sea level. Pressure altitude is important as a basis for determining aircraft performance, as well as for assigning flight levels to aircraft operating at above 18,000 feet.

The pressure altitude can be determined by either of two methods:

1. By setting the barometric scale of the altimeter to 29.92 "Hg and reading the indicated altitude, or
2. By applying a correction factor to the indicated altitude according to the reported "altimeter setting."

Density Altitude

The more appropriate term for correlating aerodynamic performance in the nonstandard atmosphere is density altitude—the altitude in the standard atmosphere corresponding to a particular value of air density.

Density altitude is pressure altitude corrected for nonstandard temperature. As the density of the air increases (lower density altitude), aircraft performance increases. Conversely, as air density decreases (higher density altitude), aircraft performance decreases. A decrease in air density means a high density altitude; an increase in air density means a lower density altitude. Density altitude is used in calculating aircraft performance. Under standard atmospheric condition, air at each level in the atmosphere has a specific density; under standard conditions, pressure altitude and density altitude identify the same level. Density altitude, then, is the vertical distance above sea level in the standard atmosphere at which a given density is to be found.

The computation of density altitude must involve consideration of pressure (pressure altitude) and temperature. Since aircraft performance data at any level is based upon air density under standard day conditions, such performance data apply to air density levels that may not be identical to altimeter indications. Under conditions higher or lower than standard, these levels cannot be determined directly from the altimeter.

Density altitude is determined by first finding pressure altitude, and then correcting this altitude for nonstandard temperature variations. Since density varies directly with pressure, and inversely with temperature, a given pressure altitude may exist for a wide range of temperature by allowing the density to vary. However, a known density occurs for any one temperature and pressure altitude. The density of the air, of course, has a pronounced effect on aircraft and engine performance. Regardless of the actual altitude at which the aircraft is operating, it will perform as though it were operating at an altitude equal to the existing density altitude.

For example, when set at 29.92 "Hg, the altimeter may indicate a pressure altitude of 5,000 feet. According to the AFM/POH, the ground run on takeoff may require a distance of 790 feet under standard temperature conditions.

However, if the temperature is 20 °C above standard, the expansion of air raises the density level. Using temperature correction data from tables or graphs, or by deriving the density altitude with a computer, it may be found that the density level is above 7,000 feet, and the ground run may be closer to 1,000 feet.

Air density is affected by changes in altitude, temperature, and humidity. High density altitude refers to thin air while low density altitude refers to dense air. The conditions that result in a high density altitude are high elevations, low atmospheric pressures, high temperatures, high humidity, or some combination of these factors. Lower elevations, high atmospheric pressure, low temperatures, and low humidity are more indicative of low density altitude.

Using a flight computer, density altitude can be computed by inputting the pressure altitude and outside air temperature at flight level. Density altitude can also be determined by referring to the table and chart in *Figures 10-3 and 10-4*.

Method for Determining Pressure Altitude	
Altimeter setting	Altitude correction
28.0	1,825
28.1	1,725
28.2	1,630
28.3	1,535
28.4	1,435
28.5	1,340
28.6	1,245
28.7	1,150
28.8	1,050
28.9	955
29.0	865
29.1	770
29.2	675
29.3	580
29.4	485
29.5	390
29.6	300
29.7	205
29.8	110
29.9	20
29.92	0
30.0	–75
30.1	–165
30.2	–255
30.3	–350
30.4	–440
30.5	–530
30.6	–620
30.7	–710
30.8	–805
30.9	–895
31.0	–965

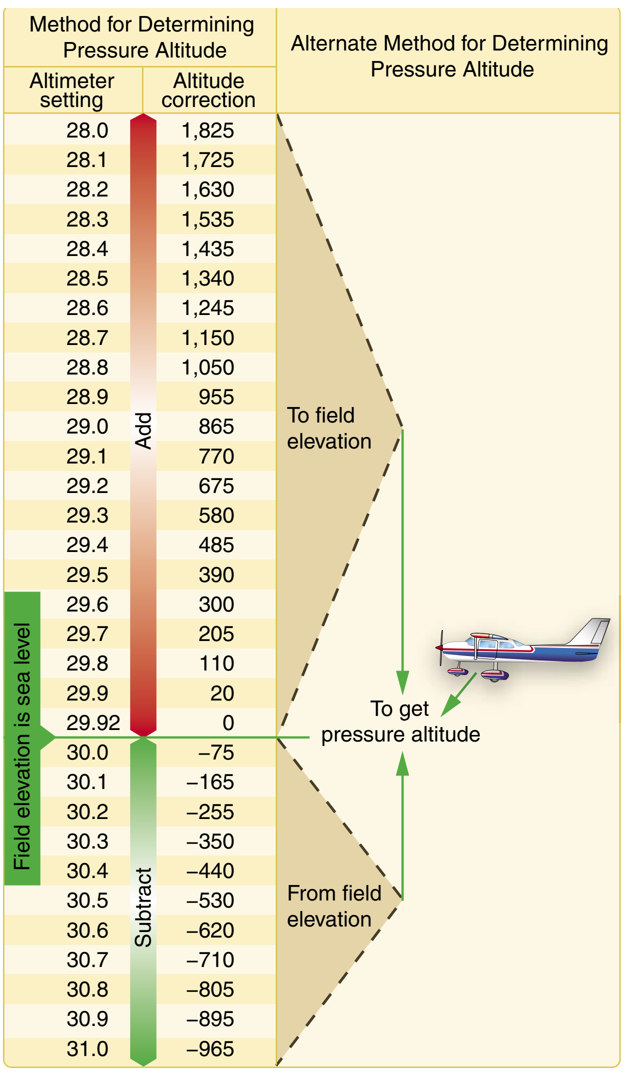


Figure 10-3. *Field elevation versus pressure. The aircraft is located on a field which happens to be at sea level. Set the altimeter to the current altimeter setting (29.7). The difference of 205 feet is added to the elevation or a PA of 205 feet.*

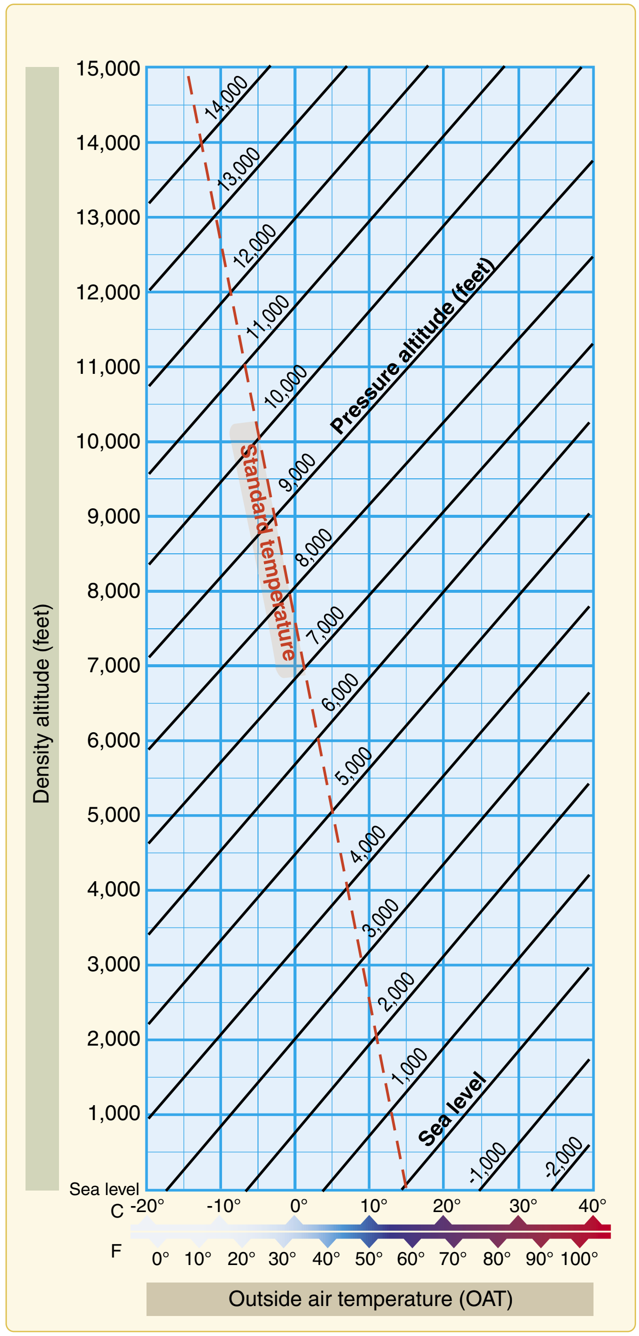


Figure 10-4. *Density altitude chart.*

Effects of Pressure on Density

Since air is a gas, it can be compressed or expanded. When air is compressed, a greater amount of air can occupy a given volume. Conversely, when pressure on a given volume of air is decreased, the air expands and occupies a greater space. That is, the original column of air at a lower pressure contains a smaller mass of air. In other words, the density is decreased. In fact, density is directly proportional to pressure. If the pressure is doubled, the density is doubled, and if the pressure is lowered, so is the density. This statement is true only at a constant temperature.

Effects of Temperature on Density

Increasing the temperature of a substance decreases its density. Conversely, decreasing the temperature increases the density. Thus, the density of air varies inversely with temperature. This statement is true only at a constant pressure.

In the atmosphere, both temperature and pressure decrease with altitude, and have conflicting effects upon density. However, the fairly rapid drop in pressure as altitude is increased usually has the dominant effect. Hence, pilots can expect the density to decrease with altitude.

Effects of Humidity (Moisture) on Density

The preceding paragraphs are based on the presupposition of perfectly dry air. In reality, it is never completely dry. The small amount of water vapor suspended in the atmosphere may be negligible under certain conditions, but in other conditions humidity may become an important factor in the performance of an aircraft. Water vapor is lighter than air; consequently, moist air is lighter than dry air. Therefore, as the water content of the air increases, the air becomes less dense, increasing density altitude and decreasing performance. It is lightest or least dense when, in a given set of conditions, it contains the maximum amount of water vapor.

Humidity, also called relative humidity, refers to the amount of water vapor contained in the atmosphere, and is expressed as a percentage of the maximum amount of water vapor the air can hold. This amount varies with the temperature; warm air can hold more water vapor, while colder air can hold less. Perfectly dry air that contains no water vapor has a relative humidity of zero percent, while saturated air that cannot hold any more water vapor has a relative humidity of 100 percent. Humidity alone is usually not considered an essential factor in calculating density altitude and aircraft performance; however, it does contribute.

The higher the temperature, the greater amount of water vapor that the air can hold. When comparing two separate air masses, the first warm and moist (both qualities making air lighter) and the second cold and dry (both qualities making it heavier), the first must be less dense than the second. Pressure, temperature, and humidity have a great influence on aircraft performance because of their effect upon density. There is no rule-of-thumb or chart used to compute the effects of humidity on density altitude, but it must be taken into consideration. Expect a decrease in overall performance in high humidity conditions.

Performance

Performance is a term used to describe the ability of an aircraft to accomplish certain things that make it useful for certain purposes. For example, the ability of an aircraft to land and take off in a very short distance is an important factor to the pilot who operates in and out of short, unimproved airfields. The ability to carry heavy loads, fly at high altitudes at fast speeds, or travel long distances is essential performance for operators of airline and executive type aircraft.

The primary factors most affected by performance are the takeoff and landing distance, rate of climb, ceiling, payload, range, speed, maneuverability, stability, and fuel economy. Some of these factors are often directly opposed: for example, high speed versus short landing distance, long range versus great payload, and high rate of climb versus fuel economy. It is the preeminence of one or more of these factors that dictates differences between aircraft and explains the high degree of specialization found in modern aircraft.

The various items of aircraft performance result from the combination of aircraft and powerplant characteristics. The aerodynamic characteristics of the aircraft generally define the power and thrust requirements at various conditions of flight, while powerplant characteristics generally define the power and thrust available at various conditions of flight. The matching of the aerodynamic configuration with the powerplant is accomplished by the manufacturer to provide maximum performance at the specific design condition (e.g., range, endurance, and climb).

Straight-and-Level Flight

All of the principal components of flight performance involve steady-state flight conditions and equilibrium of the aircraft. For the aircraft to remain in steady, level flight, equilibrium must be obtained by a lift equal to the aircraft weight and a powerplant thrust equal to the aircraft drag. Thus, the aircraft drag defines the thrust required to maintain steady, level flight. As presented in Chapter 4, Aerodynamics of Flight, all parts of an aircraft contribute to the drag, either induced (from lifting surfaces) or parasite drag.

While the parasite drag predominates at high speed, induced drag predominates at low speed. *[Figure 10-5]* For example,

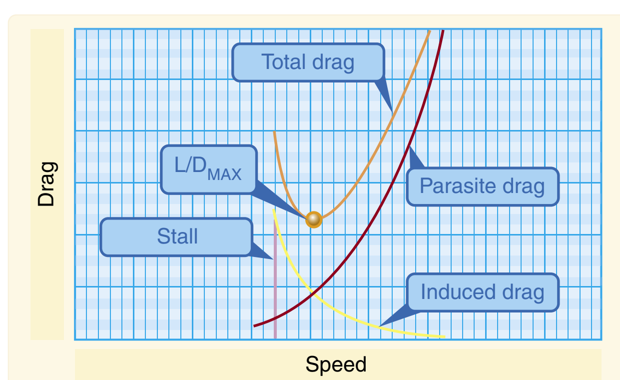


Figure 10-5. *Drag versus speed.*

if an aircraft in a steady flight condition at 100 knots is then accelerated to 200 knots, the parasite drag becomes four times as great, but the power required to overcome that drag is eight times the original value. Conversely, when the aircraft is operated in steady, level flight at twice as great a speed, the induced drag is one-fourth the original value, and the power required to overcome that drag is only one-half the original value.

When an aircraft is in steady, level flight, the condition of equilibrium must prevail. The unaccelerated condition of flight is achieved with the aircraft trimmed for lift equal to weight and the powerplant set for a thrust to equal the aircraft drag.

The maximum level flight speed for the aircraft will be obtained when the power or thrust required equals the maximum power or thrust available from the powerplant. *[Figure 10-6]* The minimum level flight airspeed is not usually defined by thrust or power requirement since conditions of stall or stability and control problems generally predominate.

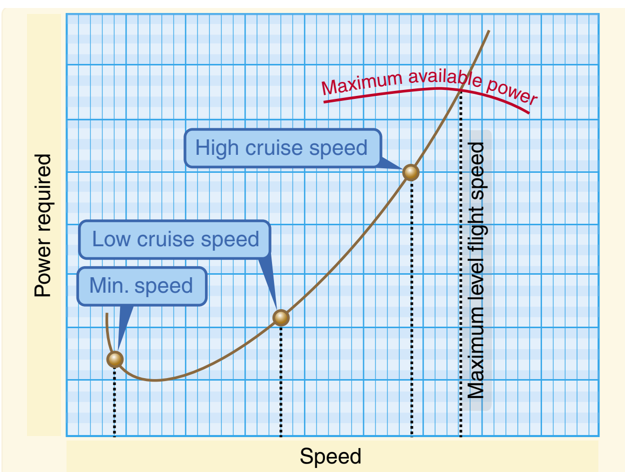


Figure 10-6. *Power versus speed.*

Climb Performance

Climb performance is a result of using the aircrafts potential energy provided by one, or a combination of two factors. The first is the use of excess power above that required for level flight. An aircraft equipped with an engine capable of 200 horsepower (at a given altitude) but using 130 horsepower to sustain level flight (at a given airspeed) has 70 excess horsepower available for climbing. A second factor is that the aircraft can tradeoff its kinetic energy and increase its potential energy by reducing its airspeed. The reduction in airspeed will increase the aircraft's potential energy thereby also making the aircraft climb. Both terms, power and thrust are often used in aircraft performance however, they should not be confused.

Although the terms "power" and "thrust" are sometimes used interchangeably, erroneously implying that they are synonymous, it is important to distinguish between the two when discussing climb performance. Work is the product of a force moving through a distance and is usually independent of time. Work is measured by several standards; the most common unit is called a foot-pound. If a one pound mass is raised one foot, a work unit of one foot-pound has been performed. The common unit of mechanical power is horsepower; one horsepower is work equivalent to lifting 33,000 pounds a vertical distance of one foot in one minute. The term power implies work rate or units of work per unit of time, and as such is a function of the speed at which the force is developed. Thrust, also a function of work, means the force that imparts a change in the velocity of a mass. This force is measured in pounds but has no element of time or rate. It can be said then, that during a steady climb, the rate of climb is a function of excess thrust.

This relationship means that, for a given weight of an aircraft, the angle of climb depends on the difference between thrust and drag, or the excess power. *[Figure 10-7]* Of course, when the excess thrust is zero, the inclination of the flightpath is zero, and the aircraft will be in steady, level flight. When the thrust is greater than the drag, the excess thrust will allow a climb angle depending on the value of excess thrust. On the other hand, when the thrust is less than the drag, the deficiency of thrust will allow an angle of descent.

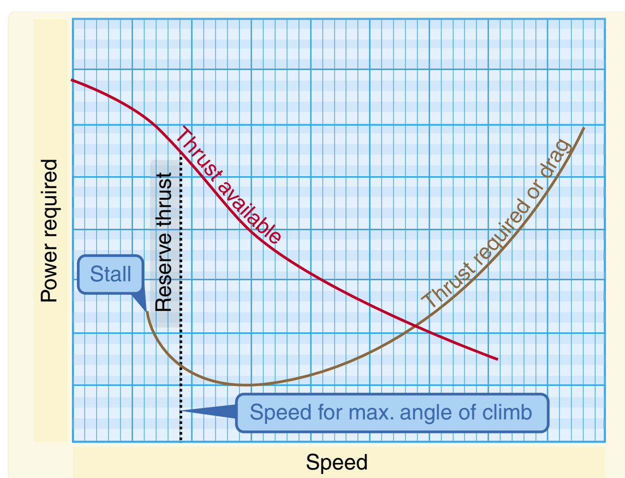


Figure 10-7. *Thrust versus climb angle.*

The most immediate interest in the climb angle performance involves obstacle clearance. The most obvious purpose for which it might be used is to clear obstacles when climbing out of short or confined airports.

The maximum angle of climb would occur where there exists the greatest difference between thrust available and thrust required; i.e., for the propeller-powered airplane, the maximum excess thrust and angle of climb will occur at some speed just above the stall speed. Thus, if it is necessary to clear an obstacle after takeoff, the propeller-powered airplane will attain maximum angle of climb at an airspeed close to—if not at—the takeoff speed.

Of greater interest in climb performance are the factors that affect the rate of climb. The vertical velocity of an aircraft depends on the flight speed and the inclination of the flightpath. In fact, the rate of climb is the vertical component of the flightpath velocity.

For rate of climb, the maximum rate would occur where there exists the greatest difference between power available and power required. *[Figure 10-8]* The above relationship means that, for a given weight of an aircraft, the rate of climb depends on the difference between the power available and the power required, or the excess power. Of course, when the excess power is zero, the rate of climb is zero and the aircraft is in steady, level flight. When power available is greater than the power required, the excess power will allow a rate of climb specific to the magnitude of excess power.

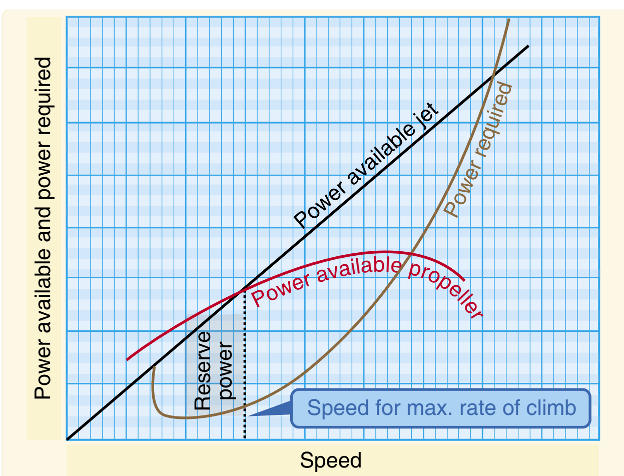


Figure 10-8. *Power versus climb rate.*

During a steady climb, the rate of climb will depend on excess power while the angle of climb is a function of excess thrust.

The climb performance of an aircraft is affected by certain variables. The conditions of the aircraft's maximum climb angle or maximum climb rate occur at specific speeds, and variations in speed will produce variations in climb performance. There is sufficient latitude in most aircraft that small variations in speed from the optimum do not produce large changes in climb performance, and certain operational considerations may require speeds slightly different from the optimum. Of course, climb performance would be most critical with high gross weight, at high altitude, in obstructed takeoff areas, or during malfunction of a powerplant. Then, optimum climb speeds are necessary.

Weight has a very pronounced effect on aircraft performance. If weight is added to an aircraft, it must fly at a higher angle of attack (AOA) to maintain a given altitude and speed. This increases the induced drag of the wings, as well as the parasite drag of the aircraft. Increased drag means that additional thrust is needed to overcome it, which in turn means that less reserve thrust is available for climbing. Aircraft designers go to great effort to minimize the weight since it has such a marked effect on the factors pertaining to performance.

A change in an aircraft's weight produces a twofold effect on climb performance. First, a change in weight will change the drag and the power required. This alters the reserve power available, which in turn, affects both the climb angle and the climb rate. Secondly, an increase in weight will reduce the maximum rate of climb, but the aircraft must be operated at a higher climb speed to achieve the smaller peak climb rate.

An increase in altitude also will increase the power required and decrease the power available. Therefore, the climb performance of an aircraft diminishes with altitude. The speeds for maximum rate of climb, maximum angle of climb, and maximum and minimum level flight airspeeds vary with altitude. As altitude is increased, these various speeds finally converge at the absolute ceiling of the aircraft. At the absolute ceiling, there is no excess of power and only one speed will allow steady, level flight. Consequently, the absolute ceiling of an aircraft produces zero rate of climb. The service ceiling is the altitude at which the aircraft is unable to climb at a rate greater than 100 feet per minute (fpm). Usually, these specific performance reference points are provided for the aircraft at a specific design configuration. *[Figure 10-9]*

In discussing performance, it frequently is convenient to use the terms power loading, wing loading, blade loading, and disk loading. Power loading is expressed in pounds per horsepower and is obtained by dividing the total weight of the aircraft by the rated horsepower of the engine. It is a significant factor in an aircraft's takeoff and climb capabilities. Wing loading is expressed in pounds per square foot and is obtained by dividing the total weight of an airplane in pounds by the wing area (including ailerons) in square feet. It is the airplane's wing loading that determines the landing speed. Blade loading is expressed in pounds per square foot and is obtained by dividing the total weight of a helicopter by the area of the rotor blades. Blade loading is not to be confused with disk loading, which is the total weight of a helicopter divided by the area of the disk swept by the rotor blades.

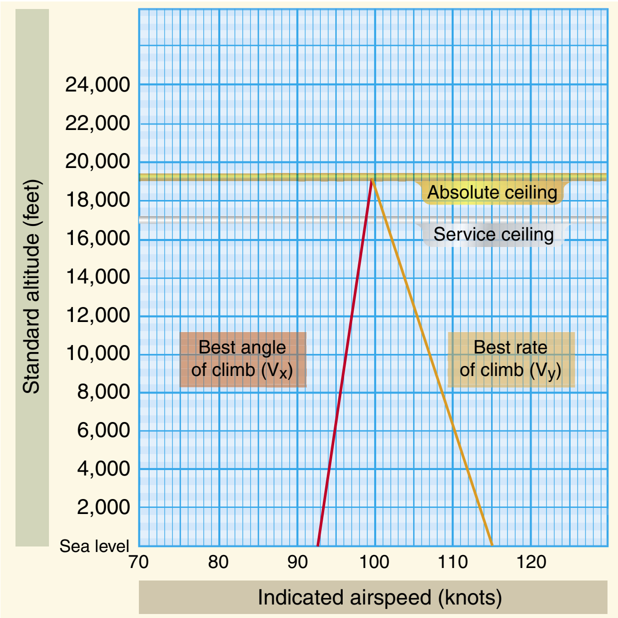


Figure 10-9. *Absolute and service ceiling.*

Range Performance

The ability of an aircraft to convert fuel energy into flying distance is one of the most important items of aircraft performance. In flying operations, the problem of efficient range operation of an aircraft appears in two general forms:

1. To extract the maximum flying distance from a given fuel load
2. To fly a specified distance with a minimum expenditure of fuel

A common element for each of these operating problems is the specific range; that is, nautical miles (NM) of flying distance versus the amount of fuel consumed. Range must be clearly distinguished from the item of endurance. Range involves consideration of flying distance, while endurance involves consideration of flying time. Thus, it is appropriate to define a separate term, specific endurance.

$$\text{specific endurance} = \frac{\text{flight hours}}{\text{pounds of fuel}}$$

or

$$\text{specific endurance} = \frac{\text{flight hours/hour}}{\text{pounds of fuel/hour}}$$

or

$$\text{specific endurance} = \frac{1}{\text{fuel flow}}$$

Fuel flow can be defined in either pounds or gallons. If maximum endurance is desired, the flight condition must provide a minimum fuel flow. In *Figure 10-10* at point A the airspeed is low and fuel flow is high. This would occur during ground operations or when taking off and climbing. As airspeed is increased, power requirements decrease due to aerodynamic factors and fuel flow decreases to point B. This is the point of maximum endurance. Beyond this point increases in airspeed come at a cost. Airspeed increases require additional power and fuel flow increases with additional power.

Cruise flight operations for maximum range should be conducted so that the aircraft obtains maximum specific range throughout the flight. The specific range can be defined by the following relationship.

$$\text{specific range} = \frac{\text{NM}}{\text{pounds of fuel}}$$

or

$$\text{specific range} = \frac{\text{NM/hour}}{\text{pounds of fuel/hour}}$$

or

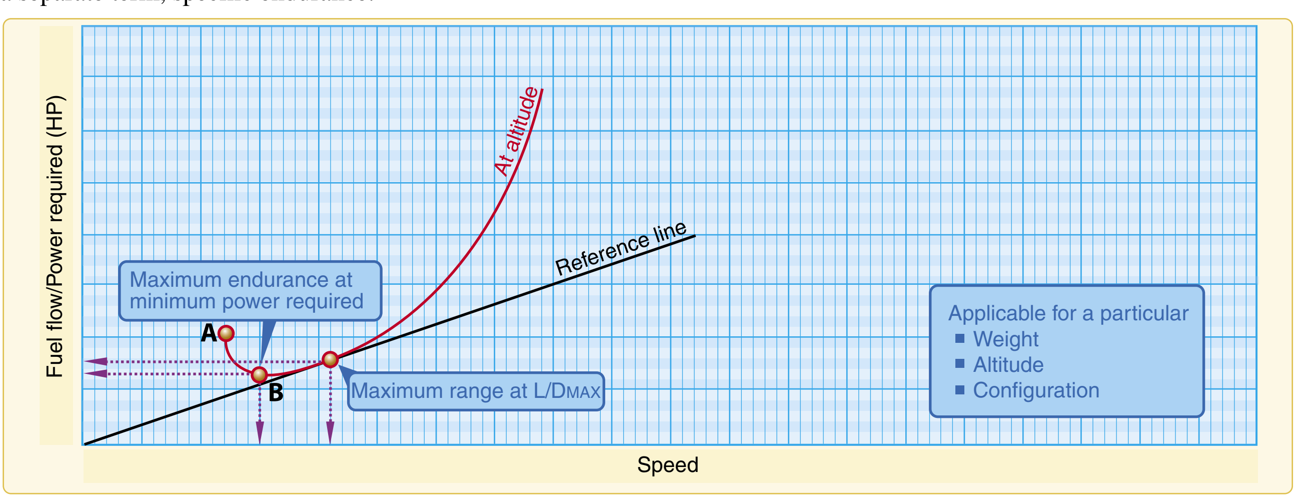


Figure 10-10. *Airspeed for maximum endurance.*

$$\text{specific range} = \frac{\text{knots}}{\text{fuel flow}}$$

If maximum specific range is desired, the flight condition must provide a maximum of speed per fuel flow. While the peak value of specific range would provide maximum range operation, long-range cruise operation is generally recommended at some slightly higher airspeed. Most long-range cruise operations are conducted at the flight condition that provides 99 percent of the absolute maximum specific range. The advantage of such operation is that one percent of range is traded for three to five percent higher cruise speed. Since the higher cruise speed has a great number of advantages, the small sacrifice of range is a fair bargain. The values of specific range versus speed are affected by three principal variables:

1. Aircraft gross weight
2. Altitude
3. The external aerodynamic configuration of the aircraft.

These are the source of range and endurance operating data included in the performance section of the AFM/POH.

Cruise control of an aircraft implies that the aircraft is operated to maintain the recommended long-range cruise condition throughout the flight. Since fuel is consumed during cruise, the gross weight of the aircraft will vary and optimum airspeed, altitude, and power setting can also vary. Cruise control means the control of the optimum airspeed, altitude, and power setting to maintain the 99 percent maximum specific range condition. At the beginning of cruise flight, the relatively high initial weight of the aircraft will require specific values of airspeed, altitude, and power setting to produce the recommended cruise condition. As fuel is consumed and the aircraft's gross weight decreases, the optimum airspeed and power setting may decrease, or, the optimum altitude may increase. In addition, the optimum specific range will increase. Therefore, the pilot must provide the proper cruise control procedure to ensure that optimum conditions are maintained.

Total range is dependent on both fuel available and specific range. When range and economy of operation are the principal goals, the pilot must ensure that the aircraft is operated at the recommended long-range cruise condition. By this procedure, the aircraft will be capable of its maximum design-operating radius, or can achieve flight distances less than the maximum with a maximum of fuel reserve at the destination.

A propeller-driven aircraft combines the propeller with the reciprocating engine for propulsive power. Fuel flow is determined mainly by the shaft power put into the propeller rather than thrust. Thus, the fuel flow can be related directly to the power required to maintain the aircraft in steady, level flight and on performance charts power can be substituted for fuel flow. This fact allows for the determination of range through analysis of power required versus speed.

The maximum endurance condition would be obtained at the point of minimum power required since this would require the lowest fuel flow to keep the airplane in steady, level flight. Maximum range condition would occur where the ratio of speed to power required is greatest. *[Figure 10-10]*

The maximum range condition is obtained at maximum lift/drag ratio (L/D_{MAX}), and it is important to note that for a given aircraft configuration, the L/D_{MAX} occurs at a particular AOA and lift coefficient, and is unaffected by weight or altitude. A variation in weight will alter the values of airspeed and power required to obtain the L/D_{MAX}. *[Figure 10-11]*

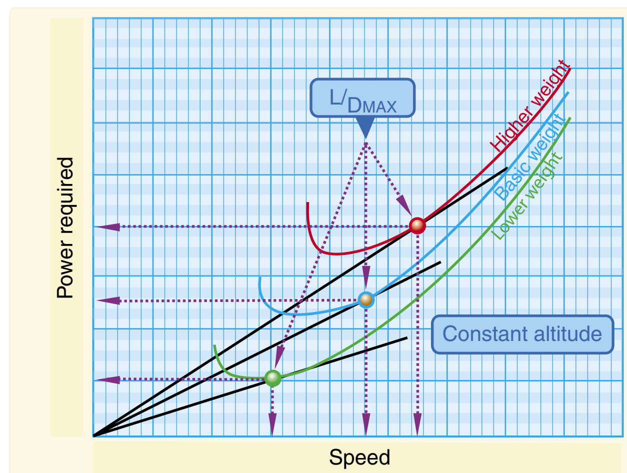


Figure 10-11. *Effect of weight.*

The variations of speed and power required must be monitored by the pilot as part of the cruise control procedure to maintain the L/D_{MAX}. When the aircraft's fuel weight is a small part of the gross weight and the aircraft's range is small, the cruise control procedure can be simplified to essentially maintaining a constant speed and power setting throughout the time of cruise flight. However, a long-range aircraft has a fuel weight that is a considerable part of the gross weight, and cruise control procedures must employ scheduled airspeed and power changes to maintain optimum range conditions.

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The effect of altitude on the range of a propeller-driven aircraft is illustrated in *Figure 10-12.* A flight conducted at high altitude has a greater true airspeed (TAS), and the power required is proportionately greater than when conducted at sea level. The drag of the aircraft at altitude is the same as the drag at sea level, but the higher TAS causes a proportionately greater power required. NOTE: The straight line that is tangent to the sea level power curve is also tangent to the altitude power curve.

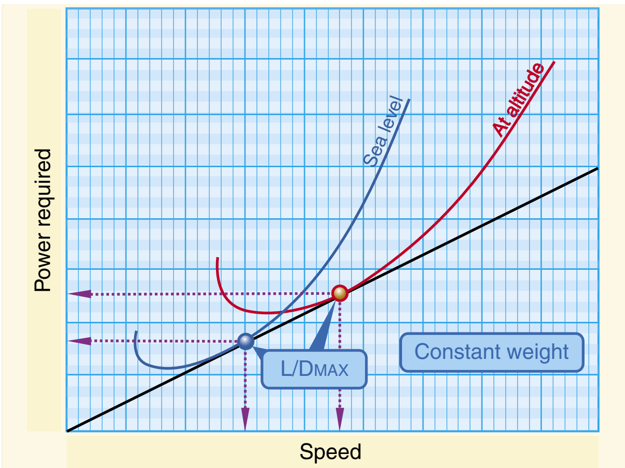


Figure 10-12. *Effect of altitude on range.*

The effect of altitude on specific range also can be appreciated from the previous relationships. If a change in altitude causes identical changes in speed and power required, the proportion of speed to power required would be unchanged. The fact implies that the specific range of a propeller-driven aircraft would be unaffected by altitude. Actually, this is true to the extent that specific fuel consumption and propeller efficiency are the principal factors that could cause a variation of specific range with altitude. If compressibility effects are negligible, any variation of specific range with altitude is strictly a function of engine/propeller performance.

An aircraft equipped with a reciprocating engine will experience very little, if any, variation of specific range up to its absolute altitude. There is negligible variation of brake specific fuel consumption for values of brake horsepower below the maximum cruise power rating of the engine that is the lean range of engine operation. Thus, an increase in altitude will produce a decrease in specific range only when the increased power requirement exceeds the maximum cruise power rating of the engine. One advantage of supercharging is that the cruise power may be maintained at high altitude, and the aircraft may achieve the range at high altitude with the corresponding increase in TAS. The principal differences in the high altitude cruise and low altitude cruise are the TAS and climb fuel requirements.

Region of Reversed Command

The aerodynamic properties of an aircraft generally determine the power requirements at various conditions of flight, while the powerplant capabilities generally determine the power available at various conditions of flight. When an aircraft is in steady, level flight, a condition of equilibrium must prevail. An unaccelerated condition of flight is achieved when lift equals weight, and the powerplant is set for thrust equal to drag. The power required to achieve equilibrium in constant-altitude flight at various airspeeds is depicted on a power required curve. The power required curve illustrates the fact that at low airspeeds near the stall or minimum controllable airspeed, the power setting required for steady, level flight is quite high.

Flight in the region of normal command means that while holding a constant altitude, a higher airspeed requires a higher power setting and a lower airspeed requires a lower power setting. The majority of aircraft flying (climb, cruise, and maneuvers) is conducted in the region of normal command.

Flight in the region of reversed command means flight in which a higher airspeed requires a lower power setting and a lower airspeed requires a higher power setting to hold altitude. It does not imply that a decrease in power will produce lower airspeed. The region of reversed command is encountered in the low speed phases of flight. Flight speeds below the speed for maximum endurance (lowest point on the power curve) require higher power settings with a decrease in airspeed. Since the need to increase the required power setting with decreased speed is contrary to the normal command of flight, the regime of flight speeds between the speed for minimum required power setting and the stall speed (or minimum control speed) is termed the region of reversed command. In the region of reversed command, a decrease in airspeed must be accompanied by an increased power setting in order to maintain steady flight.

Figure 10-13 shows the maximum power available as a curved line. Lower power settings, such as cruise power, would also appear in a similar curve. The lowest point on the power required curve represents the speed at which the lowest brake horsepower will sustain level flight. This is termed the best endurance airspeed.

An airplane performing a low airspeed, high pitch attitude power approach for a short-field landing is an example of operating in the region of reversed command. If an unacceptably high sink rate should develop, it may be possible for the pilot to reduce or stop the descent by applying power. But without further use of power, the airplane would probably stall or be incapable of flaring for the landing.

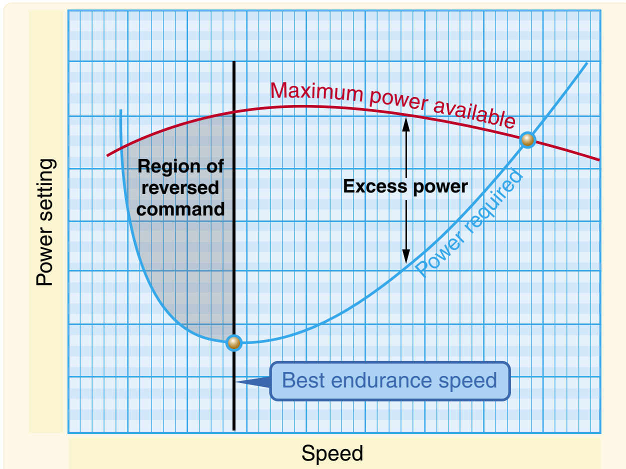


Figure 10-13. *Power required curve.*

Merely lowering the nose of the airplane to regain flying speed in this situation, without the use of power, would result in a rapid sink rate and corresponding loss of altitude.

If during a soft-field takeoff and climb, for example, the pilot attempts to climb out of ground effect without first attaining normal climb pitch attitude and airspeed, the airplane may inadvertently enter the region of reversed command at a dangerously low altitude. Even with full power, the airplane may be incapable of climbing or even maintaining altitude. The pilot's only recourse in this situation is to lower the pitch attitude in order to increase airspeed, which will inevitably result in a loss of altitude.

Airplane pilots must give particular attention to precise control of airspeed when operating in the low flight speeds of the region of reversed command.

Takeoff and Landing Performance

The majority of pilot-caused aircraft accidents occur during the takeoff and landing phase of flight. Because of this fact, the pilot must be familiar with all the variables that influence the takeoff and landing performance of an aircraft and must strive for exacting, professional procedures of operation during these phases of flight.

Takeoff and landing performance is a condition of accelerated and decelerated motion. For instance, during takeoff, an aircraft starts at zero speed and accelerates to the takeoff speed to become airborne. During landing, the aircraft touches down at the landing speed and decelerates to zero speed. The important factors of takeoff or landing performance are:

- The takeoff or landing speed is generally a function of the stall speed or minimum flying speed.
- The rate of acceleration/deceleration during the takeoff or landing roll. The speed (acceleration and deceleration) experienced by any object varies directly with the imbalance of force and inversely with the mass of the object. An airplane on the runway moving at 75 knots has four times the energy it has traveling at 37 knots. Thus, an airplane requires four times as much distance to stop as required at half the speed.
- The takeoff or landing roll distance is a function of both acceleration/deceleration and speed.

Runway Surface and Gradient

Runway conditions affect takeoff and landing performance. Typically, performance chart information assumes paved, level, smooth, and dry runway surfaces. Since no two runways are alike, the runway surface differs from one runway to another, as does the runway gradient or slope. *[Figure 10-14]*

TAKEOFF DISTANCE
MAXIMUM WEIGHT 3800 LBS

SHORT FIELD

CONDITIONS:
Flaps 10°
2850 RPM, Full Throttle and Mixture Set at Placard Fuel Flow Prior to Brake Release
Cowl Flaps Open
Paved, Level, Dry Runway
Zero Wind

MIXTURE SETTING	
PRESS ALT	PPH
S.L.	144
2000	138
4000	132
6000	126
8000	120

NOTES:
1. Short field technique as specified in Section 4.
2. Landing gear extended until takeoff obstacle is cleared.
3. Where distance value has been deleted, climb performance after lift-off is less than 150 fpm. Rate of climb is based on landing gear extended and flaps 10° at takeoff speed.
4. Decrease distances 10% for each 10 knots headwind. For operation with tailwinds up to 10 knots, increase distances by 10% for each 2.5 knots.
5. For operation on a dry, grass runway, increase distances by 15% of the "ground roll" figure.

WEIGHT LBS	TAKEOFF SPEED KIAS		PRESS ALT FT	0°C		10°C		20°C		30°C		40°C	
	LIFT	AT		GRND	TOTAL TO CLEAR	GRND	TOTAL TO CLEAR	GRND	TOTAL TO CLEAR	GRND	TOTAL TO CLEAR	GRND	TOTAL TO CLEAR

Figure 10-14. *Takeoff distance chart.*

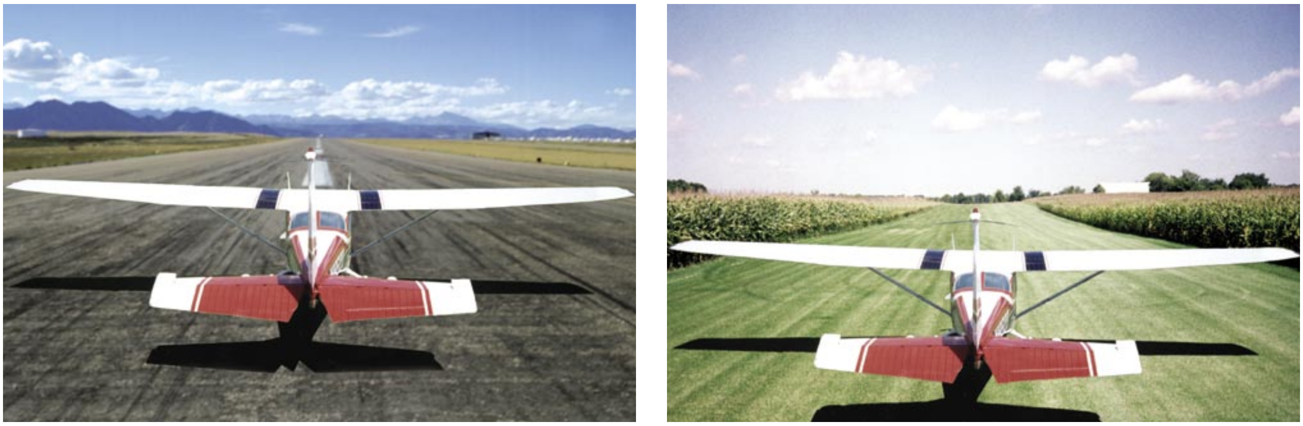


Figure 10-15. *An aircraft's performance depends greatly on the runway surface.*

Runway surfaces vary widely from one airport to another. The runway surface encountered may be concrete, asphalt, gravel, dirt, or grass. The runway surface for a specific airport is noted in the Airport/Facility Directory (A/FD). Any surface that is not hard and smooth will increase the ground roll during takeoff. This is due to the inability of the tires to roll smoothly along the runway. Tires can sink into soft, grassy, or muddy runways. Potholes or other ruts in the pavement can be the cause of poor tire movement along the runway. Obstructions such as mud, snow, or standing water reduce the airplane's acceleration down the runway. Although muddy and wet surface conditions can reduce friction between the runway and the tires, they can also act as obstructions and reduce the landing distance. *[Figure 10-15]* Braking effectiveness is another consideration when dealing with various runway types. The condition of the surface affects the braking ability of the airplane.

The amount of power that is applied to the brakes without skidding the tires is referred to as braking effectiveness. Ensure that runways are adequate in length for takeoff acceleration and landing deceleration when less than ideal surface conditions are being reported.

The gradient or slope of the runway is the amount of change in runway height over the length of the runway. The gradient is expressed as a percentage such as a 3 percent gradient. This means that for every 100 feet of runway length, the runway height changes by 3 feet. A positive gradient indicates the runway height increases, and a negative gradient indicates the runway decreases in height. An upsloping runway impedes acceleration and results in a longer ground run during takeoff. However, landing on an upsloping runway typically reduces the landing roll. A downsloping runway aids in acceleration on takeoff resulting in shorter takeoff distances. The opposite is true when landing, as landing on a downsloping runway increases landing distances. Runway slope information is contained in the A/FD. *[Figure 10-16]*

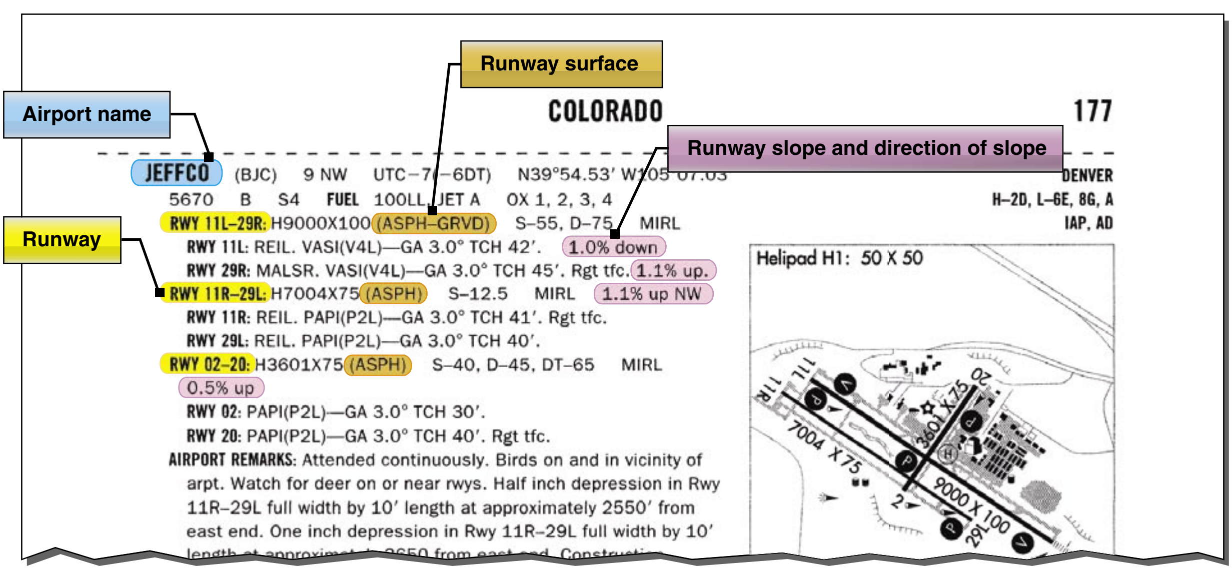


Figure 10-16. *Airport/facility directory (A/FD) information.*

Water on the Runway and Dynamic Hydroplaning

Water on the runways reduces the friction between the tires and the ground, and can reduce braking effectiveness. The ability to brake can be completely lost when the tires are hydroplaning because a layer of water separates the tires from the runway surface. This is also true of braking effectiveness when runways are covered in ice.

When the runway is wet, the pilot may be confronted with dynamic hydroplaning. Dynamic hydroplaning is a condition in which the aircraft tires ride on a thin sheet of water rather than on the runway's surface. Because hydroplaning wheels are not touching the runway, braking and directional control are almost nil. To help minimize dynamic hydroplaning, some runways are grooved to help drain off water; most runways are not.

Tire pressure is a factor in dynamic hydroplaning. Using the simple formula in *Figure 10-17,* a pilot can calculate the minimum speed, in knots, at which hydroplaning will begin. In plain language, the minimum hydroplaning speed is determined by multiplying the square root of the main gear tire pressure in psi by nine. For example, if the main gear tire pressure is at 36 psi, the aircraft would begin hydroplaning at 54 knots.

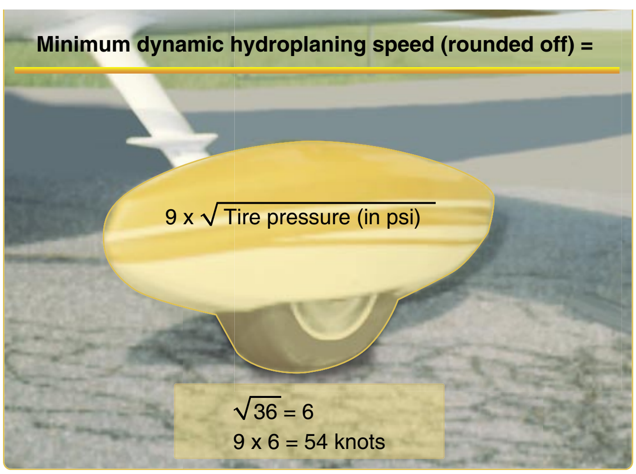


Figure 10-17. *Tire pressure.*

Landing at higher than recommended touchdown speeds will expose the aircraft to a greater potential for hydroplaning. And once hydroplaning starts, it can continue well below the minimum initial hydroplaning speed.

On wet runways, directional control can be maximized by landing into the wind. Abrupt control inputs should be avoided. When the runway is wet, anticipate braking problems well before landing and be prepared for hydroplaning. Opt for a suitable runway most aligned with the wind. Mechanical braking may be ineffective, so aerodynamic braking should be used to its fullest advantage.

Takeoff Performance

The minimum takeoff distance is of primary interest in the operation of any aircraft because it defines the runway requirements. The minimum takeoff distance is obtained by taking off at some minimum safe speed that allows sufficient margin above stall and provides satisfactory control and initial rate of climb. Generally, the lift-off speed is some fixed percentage of the stall speed or minimum control speed for the aircraft in the takeoff configuration. As such, the lift-off will be accomplished at some particular value of lift coefficient and AOA. Depending on the aircraft characteristics, the lift-off speed will be anywhere from 1.05 to 1.25 times the stall speed or minimum control speed.

To obtain minimum takeoff distance at the specific lift-off speed, the forces that act on the aircraft must provide the maximum acceleration during the takeoff roll. The various forces acting on the aircraft may or may not be under the control of the pilot, and various procedures may be necessary in certain aircraft to maintain takeoff acceleration at the highest value.

The powerplant thrust is the principal force to provide the acceleration and, for minimum takeoff distance, the output thrust should be at a maximum. Lift and drag are produced as soon as the aircraft has speed, and the values of lift and drag depend on the AOA and dynamic pressure.

In addition to the important factors of proper procedures, many other variables affect the takeoff performance of an aircraft. Any item that alters the takeoff speed or acceleration rate during the takeoff roll will affect the takeoff distance.

For example, the effect of gross weight on takeoff distance is significant and proper consideration of this item must be made in predicting the aircraft's takeoff distance. Increased gross weight can be considered to produce a threefold effect on takeoff performance:

1. Higher lift-off speed
2. Greater mass to accelerate
3. Increased retarding force (drag and ground friction)

If the gross weight increases, a greater speed is necessary to produce the greater lift necessary to get the aircraft airborne at the takeoff lift coefficient. As an example of the effect of a change in gross weight, a 21 percent increase in takeoff weight will require a 10 percent increase in lift-off speed to support the greater weight.

A change in gross weight will change the net accelerating force and change the mass that is being accelerated. If the aircraft has a relatively high thrust-to-weight ratio, the change in the net accelerating force is slight and the principal effect on acceleration is due to the change in mass.

For example, a 10 percent increase in takeoff gross weight would cause:

- A 5 percent increase in takeoff velocity.
- At least a 9 percent decrease in rate of acceleration.
- At least a 21 percent increase in takeoff distance.

With ISA conditions, increasing the takeoff weight of the average Cessna 182 from 2,400 pounds to 2,700 pounds (11 percent increase) results in an increased takeoff distance from 440 feet to 575 feet (23 percent increase).

For the aircraft with a high thrust-to-weight ratio, the increase in takeoff distance might be approximately 21 to 22 percent, but for the aircraft with a relatively low thrust-to-weight ratio, the increase in takeoff distance would be approximately 25 to 30 percent. Such a powerful effect requires proper consideration of gross weight in predicting takeoff distance.

The effect of wind on takeoff distance is large, and proper consideration also must be provided when predicting takeoff distance. The effect of a headwind is to allow the aircraft to reach the lift-off speed at a lower groundspeed while the effect of a tailwind is to require the aircraft to achieve a greater groundspeed to attain the lift-off speed.

A headwind that is 10 percent of the takeoff airspeed will reduce the takeoff distance approximately 19 percent. However, a tailwind that is 10 percent of the takeoff airspeed will increase the takeoff distance approximately 21 percent. In the case where the headwind speed is 50 percent of the takeoff speed, the takeoff distance would be approximately 25 percent of the zero wind takeoff distance (75 percent reduction).

The effect of wind on landing distance is identical to its effect on takeoff distance. *Figure 10-18* illustrates the general effect of wind by the percent change in takeoff or landing distance as a function of the ratio of wind velocity to takeoff or landing speed.

The effect of proper takeoff speed is especially important when runway lengths and takeoff distances are critical. The takeoff speeds specified in the AFM/POH are generally the minimum safe speeds at which the aircraft can become airborne. Any attempt to take off below the recommended speed means that the aircraft could stall, be difficult to

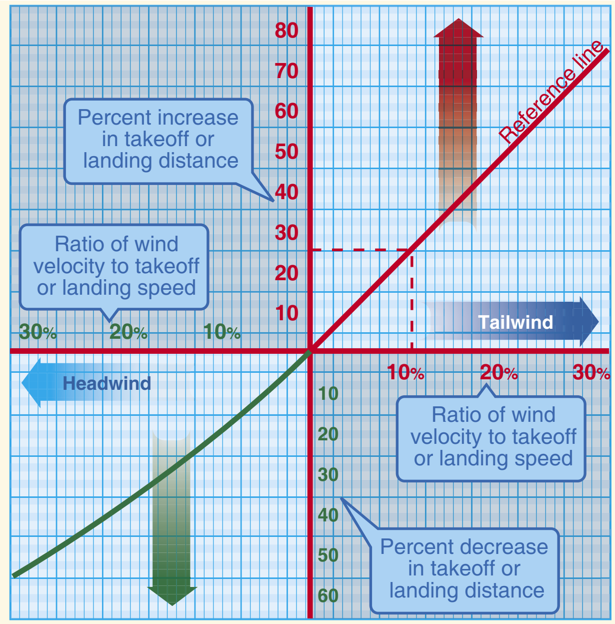


Figure 10-18. *Effect of wind on takeoff and landing.*

control, or have a very low initial rate of climb. In some cases, an excessive AOA may not allow the aircraft to climb out of ground effect. On the other hand, an excessive airspeed at takeoff may improve the initial rate of climb and "feel" of the aircraft, but will produce an undesirable increase in takeoff distance. Assuming that the acceleration is essentially unaffected, the takeoff distance varies with the square of the takeoff velocity.

Thus, ten percent excess airspeed would increase the takeoff distance 21 percent. In most critical takeoff conditions, such an increase in takeoff distance would be prohibitive, and the pilot must adhere to the recommended takeoff speeds.

The effect of pressure altitude and ambient temperature is to define the density altitude and its effect on takeoff performance. While subsequent corrections are appropriate for the effect of temperature on certain items of powerplant performance, density altitude defines specific effects on takeoff performance. An increase in density altitude can produce a twofold effect on takeoff performance:

1. Greater takeoff speed
2. Decreased thrust and reduced net accelerating force

If an aircraft of given weight and configuration is operated at greater heights above standard sea level, the aircraft requires the same dynamic pressure to become airborne at the takeoff lift coefficient. Thus, the aircraft at altitude will take off at the same indicated airspeed (IAS) as at sea level, but because of the reduced air density, the TAS will be greater.

The effect of density altitude on powerplant thrust depends much on the type of powerplant. An increase in altitude above standard sea level will bring an immediate decrease in power output for the unsupercharged reciprocating engine. However, an increase in altitude above standard sea level will not cause a decrease in power output for the supercharged reciprocating engine until the altitude exceeds the critical operating altitude. For those powerplants that experience a decay in thrust with an increase in altitude, the effect on the net accelerating force and acceleration rate can be approximated by assuming a direct variation with density. Actually, this assumed variation would closely approximate the effect on aircraft with high thrust-to-weight ratios.

Proper accounting of pressure altitude and temperature is mandatory for accurate prediction of takeoff roll distance. The most critical conditions of takeoff performance are the result of some combination of high gross weight, altitude, temperature, and unfavorable wind. In all cases, the pilot must make an accurate prediction of takeoff distance from the performance data of the AFM/POH, regardless of the runway available, and strive for a polished, professional takeoff procedure.

In the prediction of takeoff distance from the AFM/POH data, the following primary considerations must be given:

- Pressure altitude and temperature—to define the effect of density altitude on distance
- Gross weight—a large effect on distance
- Wind—a large effect due to the wind or wind component along the runway
- Runway slope and condition—the effect of an incline and retarding effect of factors such as snow or ice

Landing Performance

In many cases, the landing distance of an aircraft will define the runway requirements for flight operations. The minimum landing distance is obtained by landing at some minimum safe speed, which allows sufficient margin above stall and provides satisfactory control and capability for a go-around. Generally, the landing speed is some fixed percentage of the stall speed or minimum control speed for the aircraft in the landing configuration. As such, the landing will be accomplished at some particular value of lift coefficient and AOA. The exact values will depend on the aircraft characteristics but, once defined, the values are independent of weight, altitude, and wind.

To obtain minimum landing distance at the specified landing speed, the forces that act on the aircraft must provide maximum deceleration during the landing roll. The forces acting on the aircraft during the landing roll may require various procedures to maintain landing deceleration at the peak value.

A distinction should be made between the procedures for minimum landing distance and an ordinary landing roll with considerable excess runway available. Minimum landing distance will be obtained by creating a continuous peak deceleration of the aircraft; that is, extensive use of the brakes for maximum deceleration. On the other hand, an ordinary landing roll with considerable excess runway may allow extensive use of aerodynamic drag to minimize wear and tear on the tires and brakes. If aerodynamic drag is sufficient to cause deceleration, it can be used in deference to the brakes in the early stages of the landing roll; i.e., brakes and tires suffer from continuous hard use, but aircraft aerodynamic drag is free and does not wear out with use. The use of aerodynamic drag is applicable only for deceleration to 60 or 70 percent of the touchdown speed. At speeds less than 60 to 70 percent of the touchdown speed, aerodynamic drag is so slight as to be of little use, and braking must be utilized to produce continued deceleration. Since the objective during the landing roll is to decelerate, the powerplant thrust should be the smallest possible positive value (or largest possible negative value in the case of thrust reversers).

In addition to the important factors of proper procedures, many other variables affect the landing performance. Any item that alters the landing speed or deceleration rate during the landing roll will affect the landing distance.

The effect of gross weight on landing distance is one of the principal items determining the landing distance. One effect of an increased gross weight is that a greater speed will be required to support the aircraft at the landing AOA and lift coefficient. For an example of the effect of a change in gross weight, a 21 percent increase in landing weight will require a ten percent increase in landing speed to support the greater weight.

When minimum landing distances are considered, braking friction forces predominate during the landing roll and, for the majority of aircraft configurations, braking friction is the main source of deceleration.

The minimum landing distance will vary in direct proportion to the gross weight. For example, a ten percent increase in gross weight at landing would cause a:

- Five percent increase in landing velocity
- Ten percent increase in landing distance

A contingency of this is the relationship between weight and braking friction force.

The effect of wind on landing distance is large and deserves proper consideration when predicting landing distance. Since the aircraft will land at a particular airspeed independent of the wind, the principal effect of wind on landing distance is the change in the groundspeed at which the aircraft touches down. The effect of wind on deceleration during the landing is identical to the effect on acceleration during the takeoff.

The effect of pressure altitude and ambient temperature is to define density altitude and its effect on landing performance. An increase in density altitude increases the landing speed but does not alter the net retarding force. Thus, the aircraft at altitude lands at the same IAS as at sea level but, because of the reduced density, the TAS is greater. Since the aircraft lands at altitude with the same weight and dynamic pressure, the drag and braking friction throughout the landing roll have the same values as at sea level. As long as the condition is within the capability of the brakes, the net retarding force is unchanged, and the deceleration is the same as with the landing at sea level. Since an increase in altitude does not alter deceleration, the effect of density altitude on landing distance is due to the greater TAS.

The minimum landing distance at 5,000 feet is 16 percent greater than the minimum landing distance at sea level. The approximate increase in landing distance with altitude is approximately three and one-half percent for each 1,000 feet of altitude. Proper accounting of density altitude is necessary to accurately predict landing distance.

The effect of proper landing speed is important when runway lengths and landing distances are critical. The landing speeds specified in the AFM/POH are generally the minimum safe speeds at which the aircraft can be landed. Any attempt to land at below the specified speed may mean that the aircraft may stall, be difficult to control, or develop high rates of descent. On the other hand, an excessive speed at landing may improve the controllability slightly (especially in crosswinds), but causes an undesirable increase in landing distance.

A ten percent excess landing speed causes at least a 21 percent increase in landing distance. The excess speed places a greater working load on the brakes because of the additional kinetic energy to be dissipated. Also, the additional speed causes increased drag and lift in the normal ground attitude, and the increased lift reduces the normal force on the braking surfaces. The deceleration during this range of speed immediately after touchdown may suffer, and it is more probable for a tire to be blown out from braking at this point.

The most critical conditions of landing performance are combinations of high gross weight, high density altitude, and unfavorable wind. These conditions produce the greatest required landing distances and critical levels of energy dissipation required of the brakes. In all cases, it is necessary to make an accurate prediction of minimum landing distance to compare with the available runway. A polished, professional landing procedure is necessary because the landing phase of flight accounts for more pilot-caused aircraft accidents than any other single phase of flight.

In the prediction of minimum landing distance from the AFM/POH data, the following considerations must be given:

- Pressure altitude and temperature—to define the effect of density altitude
- Gross weight—which defines the CAS for landing.
- Wind—a large effect due to wind or wind component along the runway
- Runway slope and condition—relatively small correction for ordinary values of runway slope, but a significant effect of snow, ice, or soft ground

A tail wind of ten knots increases the landing distance by about 21 percent. An increase of landing speed by ten percent increases the landing distance by 20 percent. Hydroplaning makes braking ineffective until a decrease of speed to that determined using *Figure 10-17*.

For instance, a pilot is downwind for runway 18, and the tower asks if runway 27 could be accepted. There is a light rain and the winds are out of the east at ten knots. The pilot accepts because he or she is approaching the extended centerline of runway 27. The turn is tight and the pilot must descend (dive) to get to runway 27. After becoming aligned with the runway and at 50 feet AGL, the pilot is already 1,000 feet down the 3,500 feet runway. The airspeed is still high by about ten percent (should be at 70 knots and is at about 80 knots). The wind of ten knots is blowing from behind.

First, the airspeed being high by about ten percent (80 knots versus 70 knots), as presented in the performance chapter, results in a 20 percent increase in the landing distance. In performance planning, the pilot determined that at 70 knots the distance would be 1,600 feet. However, now it is increased by 20 percent and the required distance is now 1,920 feet.

The newly revised landing distance of 1,920 feet is also affected by the wind. In looking at *Figure 10-18*, the affect of the wind is an additional 20 percent for every ten miles per hour (mph) in wind. This is computed not on the original estimate but on the estimate based upon the increased airspeed. Now the landing distance is increased by another

320 feet for a total requirement of 2,240 feet to land the airplane after reaching 50 feet AGL.

That is the original estimate of 1,600 under planned conditions plus the additional 640 feet for excess speed and the tailwind. Given the pilot overshot the threshhold by 1,000 feet, the total length required is 3, 240 on a 3,500 foot runway; 260 feet to spare. But this is in a perfect environment. Most pilots become fearful as the end of the runway is facing them just ahead. A typical pilot reaction is to brake—and brake hard. Because the aircraft does not have antilock braking features like a car, the brakes lock, and the aircraft hydroplanes on the wet surface of the runway until decreasing to a speed of about 54 knots (the square root of the tire pressure ($\sqrt{36}$) x 9). Braking is ineffective when hydroplaning.

The 260 feet that a pilot might feel is left over has long since evaporated as the aircraft hydroplaned the first 300–500 feet when the brakes locked. This is an example of a true story, but one which only changes from year to year because of new participants and aircraft with different N-numbers.

In this example, the pilot actually made many bad decisions. Bad decisions, when combined, have a synergy greater than the individual errors. Therefore, the corrective actions become larger and larger until correction is almost impossible. Aeronautical decision-making will be discussed more fully in Chapter 17, Aeronautical Decision-Making (ADM).

Performance Speeds

True Airspeed (TAS)—the speed of the aircraft in relation to the air mass in which it is flying.

Indicated Airspeed (IAS)—the speed of the aircraft as observed on the ASI. It is the airspeed without correction for indicator, position (or installation), or compressibility errors.

Calibrated Airspeed (CAS)—the ASI reading corrected for position (or installation), and instrument errors. (CAS is equal to TAS at sea level in standard atmosphere.) The color coding for various design speeds marked on ASIs may be IAS or CAS.

Equivalent Airspeed (EAS)—the ASI reading corrected for position (or installation), or instrument error, and for adiabatic compressible flow for the particular altitude. (EAS is equal to CAS at sea level in standard atmosphere.)

V_{S0}—the calibrated power-off stalling speed or the minimum steady flight speed at which the aircraft is controllable in the landing configuration.

V_{S1}—the calibrated power-off stalling speed or the minimum steady flight speed at which the aircraft is controllable in a specified configuration.

V_Y—the speed at which the aircraft will obtain the maximum increase in altitude per unit of time. This best rate-of-climb speed normally decreases slightly with altitude.

V_X—the speed at which the aircraft will obtain the highest altitude in a given horizontal distance. This best angle-of-climb speed normally increases slightly with altitude.

V_{LE}—the maximum speed at which the aircraft can be safely flown with the landing gear extended. This is a problem involving stability and controllability.

V_{LO}—the maximum speed at which the landing gear can be safely extended or retracted. This is a problem involving the air loads imposed on the operating mechanism during extension or retraction of the gear.

V_{FE}—the highest speed permissible with the wing flaps in a prescribed extended position. This is because of the air loads imposed on the structure of the flaps.

V_A—the calibrated design maneuvering airspeed. This is the maximum speed at which the limit load can be imposed (either by gusts or full deflection of the control surfaces) without causing structural damage. Operating at or below manuevering speed does not provide structural protection against multiple full control inputs in one axis or full control inputs in more than one axis at the same time.

V_{NO}—the maximum speed for normal operation or the maximum structural cruising speed. This is the speed at which exceeding the limit load factor may cause permanent deformation of the aircraft structure.

V_{NE}—the speed which should *never* be exceeded. If flight is attempted above this speed, structural damage or structural failure may result.

Performance Charts

Performance charts allow a pilot to predict the takeoff, climb, cruise, and landing performance of an aircraft. These charts, provided by the manufacturer, are included in the AFM/POH. Information the manufacturer provides on these charts has been gathered from test flights conducted in a new aircraft, under normal operating conditions while using average piloting skills, and with the aircraft and engine in good working order. Engineers record the flight data and create performance charts based on the behavior of the aircraft during the test flights. By using these performance charts, a pilot can determine the runway length needed to take off and land, the amount of fuel to be used during flight, and the time required to arrive at the destination. It is important to remember that the data from the charts will not be accurate if the aircraft is not in good working order or when operating under adverse conditions. Always consider the necessity to compensate for the performance numbers if the aircraft is not in good working order or piloting skills are below average. Each aircraft performs differently and, therefore, has different performance numbers. Compute the performance of the aircraft prior to every flight, as every flight is different. (See appendix for examples of performance charts for a Cessna Model 172R and Challenger 605.)

Every chart is based on certain conditions and contains notes on how to adapt the information for flight conditions. It is important to read every chart and understand how to use it. Read the instructions provided by the manufacturer. For an explanation on how to use the charts, refer to the example provided by the manufacturer for that specific chart. *[Figure 10-19]*

The information manufacturers furnish is not standardized. Information may be contained in a table format, and other information may be contained in a graph format. Sometimes combined graphs incorporate two or more graphs into one chart to compensate for multiple conditions of flight. Combined graphs allow the pilot to predict aircraft performance for variations in density altitude, weight, and winds all on one chart. Because of the vast amount of information that can be extracted from this type of chart, it is important to be very accurate in reading the chart. A small error in the beginning can lead to a large error at the end.

The remainder of this section covers performance information for aircraft in general and discusses what information the charts contain and how to extract information from the charts by direct reading and interpolation methods. Every chart contains a wealth of information that should be used when flight planning. Examples of the table, graph, and combined graph formats for all aspects of flight will be discussed.

Interpolation

Not all of the information on the charts is easily extracted. Some charts require interpolation to find the information for specific flight conditions. Interpolating information means that by taking the known information, a pilot can compute intermediate information. However, pilots sometimes round off values from charts to a more conservative figure.

Using values that reflect slightly more adverse conditions provides a reasonable estimate of performance information and gives a slight margin of safety. The following illustration is an example of interpolating information from a takeoff distance chart. *[Figure 10-20]*

Density Altitude Charts

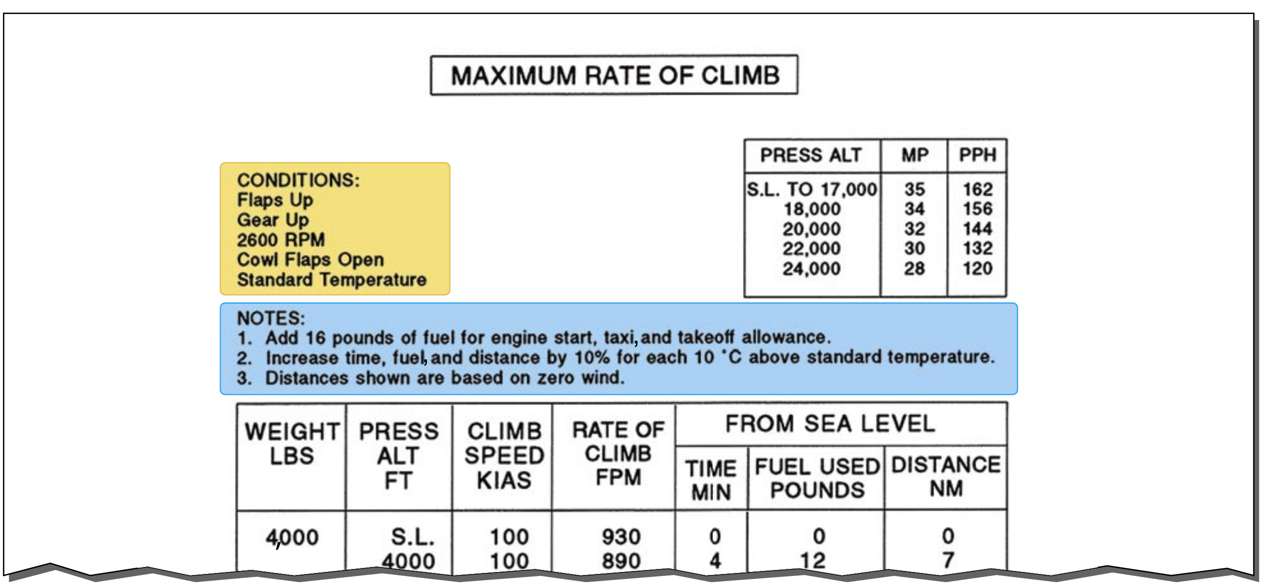
Use a density altitude chart to figure the density altitude at the departing airport. Using *Figure 10-21*, determine the density altitude based on the given information.

Sample Problem 1

Airport Elevation..5,883 feet

OAT..70 °F

Altimeter..30.10" Hg



MAXIMUM RATE OF CLIMB

CONDITIONS:
Flaps Up
Gear Up
2600 RPM
Cowl Flaps Open
Standard Temperature

PRESS ALT	MP	PPH
S.L. TO 17,000	35	162
18,000	34	156
20,000	32	144
22,000	30	132
24,000	28	120

NOTES:
1. Add 16 pounds of fuel for engine start, taxi, and takeoff allowance.
2. Increase time, fuel, and distance by 10% for each 10 °C above standard temperature.
3. Distances shown are based on zero wind.

WEIGHT LBS	PRESS ALT FT	CLIMB SPEED KIAS	RATE OF CLIMB FPM	FROM SEA LEVEL		
				TIME MIN	FUEL USED POUNDS	DISTANCE NM
4,000	S.L.	100	930	0	0	0
	4000	100	890	4	12	7

Figure 10-19. *Conditions notes chart.*

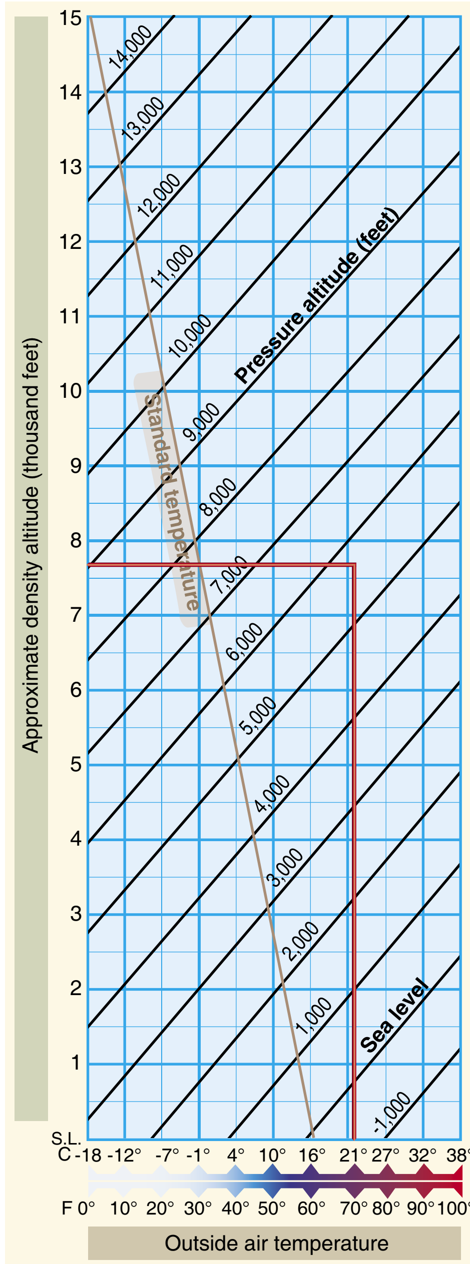
Conditions				TAKEOFF DISTANCE MAXIMUM WEIGHT 2,400 LB										
Flaps 10° Full throttle prior to brake release Paved level runway Zero wind														
Weight (lb)	Takeoff speed KIAS		Press ALT (feet)	0 °C		10 °C		20 °C		30 °C		40 °C		
	Lift off	AT 50 ft		Grnd Roll (feet)	Total feet to clear 50 ft OBS	Grnd Roll (feet)	Total feet to clear 50 ft OBS	Grnd Roll (feet)	Total feet to clear 50 ft OBS	Grnd Roll (feet)	Total feet to clear 50 ft OBS	Grnd Roll (feet)	Total feet to clear 50 ft OBS	
2,400	51	56	S.L.	795	1,460	860	1,570	925	1,685	995	1,810	1,065	1,945	
			1,000	875	1,605	940	1,725	1,015	1,860	1,090	2,000	1,170	2,155	
			2,000	960	1,770	1,035	1,910	1,115	2,060	1,200	2,220	1,290	2,395	
			3,000	1,055	1,960	1,140	2,120	1,230	2,295	1,325	2,480	1,425	2,685	
			4,000	1,165	2,185	1,260	2,365	1,355	2,570	1,465	2,790	1,575	3,030	
			5,000	1,285	2,445	1,390	2,660	1,500	2,895	1,620	3,160	1,745	3,455	
			6,000	1,425	2,755	1,540	3,015	1,665	3,300	1,800	3,620	1,940	3,990	
			7,000	1,580	3,140	1,710	3,450	1,850	3,805	2,000	4,220	- - -	- - -	
			8,000	1,755	3,615	1,905	4,015	2,060	4,480	- - -	- - -	- - -	- - -	

To find the takeoff distance for a pressure altitude of 2,500 feet at 20 °C, average the ground roll for 2,000 feet and 3,000 feet.

$$\frac{1{,}115 + 1{,}230}{2} = 1{,}173 \text{ feet}$$

Figure 10-20. *Interpolating charts.*

First, compute the pressure altitude conversion. Find 30.10 under the altimeter heading. Read across to the second column. It reads "–165." Therefore, it is necessary to subtract 165 from the airport elevation giving a pressure altitude of 5,718 feet. Next, locate the outside air temperature on the scale along the bottom of the graph. From 70°, draw a line up to the 5,718 feet pressure altitude line, which is about two-thirds of the way up between the 5,000 and 6,000 foot lines. Draw a line straight across to the far left side of the graph and read the approximate density altitude. The approximate density altitude in thousands of feet is 7,700 feet.



Altimeter setting ("Hg)	Pressure altitude conversion factor
28.0	1,824
28.1	1,727
28.2	1,630
28.3	1,533
28.4	1,436
28.5	1,340
28.6	1,244
28.7	1,148
28.8	1,053
28.9	957
29.0	863
29.1	768
29.2	673
29.3	579
29.4	485
29.5	392
29.6	298
29.7	205
29.8	112
29.9	20
29.92	0
30.0	–73
30.1	–165
30.2	–257
30.3	–348
30.4	–440
30.5	–531
30.6	–622
30.7	–712
30.8	–803

Figure 10-21. *Density altitude chart.*

Takeoff Charts

Takeoff charts are typically provided in several forms and allow a pilot to compute the takeoff distance of the aircraft with no flaps or with a specific flap configuration. A pilot can also compute distances for a no flap takeoff over a 50 foot obstacle scenario, as well as with flaps over a 50 foot obstacle. The takeoff distance chart provides for various aircraft weights, altitudes, temperatures, winds, and obstacle heights.

Sample Problem 2

Pressure Altitude..2,000 feet

OAT...22 °C

Takeoff Weight...2,600 pounds

Headwind...6 knots

Obstacle Height.....................................50 foot obstacle

Refer to *Figure 10-22*. This chart is an example of a combined takeoff distance graph. It takes into consideration pressure altitude, temperature, weight, wind, and obstacles all on one chart. First, find the correct temperature on the bottom left-

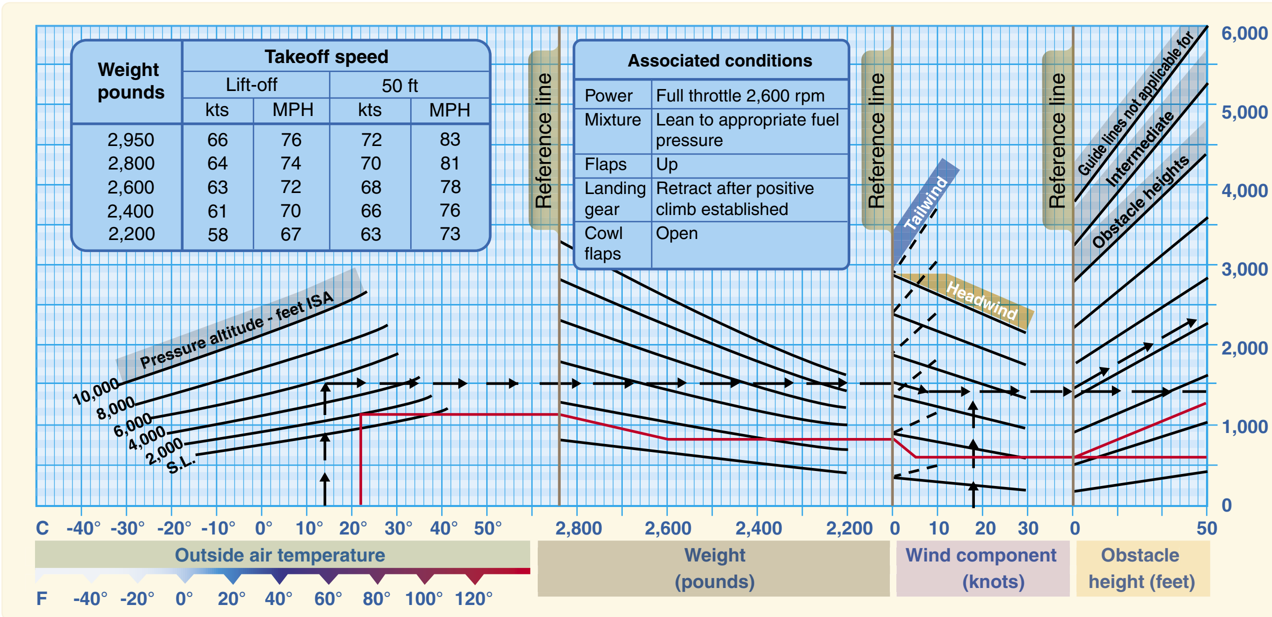


Figure 10-22. *Takeoff distance graph.*

hand side of the graph. Follow the line from 22° C straight up until it intersects the 2,000 foot altitude line. From that point, draw a line straight across to the first dark reference line. Continue to draw the line from the reference point in a diagonal direction following the surrounding lines until it intersects the corresponding weight line. From the intersection of 2,600 pounds, draw a line straight across until it reaches the second reference line. Once again, follow the lines in a diagonal manner until it reaches the six knot headwind mark. Follow straight across to the third reference line and from here, draw a line in two directions. First, draw a line straight across to figure the ground roll distance. Next, follow the diagonal lines again until it reaches the corresponding obstacle height. In this case, it is a 50 foot obstacle. Therefore, draw the diagonal line to the far edge of the chart. This results in a 600 foot ground roll distance and a total distance of 1,200 feet over a 50 foot obstacle. To find the corresponding takeoff speeds at lift-off and over the 50 foot obstacle, refer to the table on the top of the chart. In this case, the lift-off speed at 2,600 pounds would be 63 knots and over the 50 foot obstacle would be 68 knots.

Sample Problem 3

Pressure Altitude...3,000 feet

OAT..30 °C

Takeoff Weight..2,400 pounds

Headwind...18 knots

Refer to *Figure 10-23*. This chart is an example of a takeoff distance table for short-field takeoffs. For this table, first find the takeoff weight. Once at 2,400 pounds, begin reading from left to right across the table. The takeoff speed is in the second column and, in the third column under pressure altitude, find the pressure altitude of 3,000 feet. Carefully follow that line to the right until it is under the correct temperature column of 30 °C. The ground roll total reads 1,325 feet and the total required to clear a 50 foot obstacle is 2,480 feet. At this point, there is an 18 knot headwind. According to the notes section under point number two, decrease the distances by ten percent for each 9 knots of headwind. With an 18 knot headwind, it is necessary to decrease the distance by 20 percent. Multiply 1,325 feet by 20 percent (1,325 x .20 = 265), subtract the product from the total distance (1,325 – 265 = 1,060). Repeat this process for the total distance over a 50 foot obstacle. The ground roll distance is 1,060 feet and the total distance over a 50 foot obstacle is 1,984 feet.

Climb and Cruise Charts

Climb and cruise chart information is based on actual flight tests conducted in an aircraft of the same type. This information is extremely useful when planning a cross-country to predict the performance and fuel consumption of the aircraft. Manufacturers produce several different charts for climb and cruise performance. These charts include everything from fuel, time, and distance to climb, to best power setting during cruise, to cruise range performance.

The first chart to check for climb performance is a fuel, time, and distance-to-climb chart. This chart will give the fuel amount used during the climb, the time it will take to accomplish the climb, and the ground distance that will be covered during the climb. To use this chart, obtain the

Conditions													
Flaps 10° Full throttle prior to brake release Paved level runway Zero wind				**TAKEOFF DISTANCE** MAXIMUM WEIGHT 2,400 LB **SHORT FIELD**									
Notes	1. Prior to takeoff from fields above 3,000 feet elevation, the mixture should be leaned to give maximum rpm in a full throttle, static runup. 2. Decrease distances 10% for each 9 knots headwind. For operation with tailwind up to 10 knots, increase distances by 10% for each 2 knots. 3. For operation on a dry, grass runway, increase distances by 15% of the "ground roll" figure.												
Weight (lb)	Takeoff speed KIAS		Press ALT (FT)	0 °C		10 °C		20 °C		30 °C		40 °C	
	Lift off	AT 50 ft		Grnd Roll (FT)	Total feet to clear 50 ft OBS	Grnd Roll (FT)	Total feet to clear 50 ft OBS	Grnd Roll (FT)	Total feet to clear 50 ft OBS	Grnd Roll (FT)	Total feet to clear 50 ft OBS	Grnd Roll (FT)	Total feet to clear 50 ft OBS
2,400	51	56	S.L.	795	1,460	860	1,570	925	1,685	995	1,810	1,065	1,945
			1,000	875	1,605	940	1,725	1,015	1,860	1,090	2,000	1,170	2,155
			2,000	960	1,770	1,035	1,910	1,115	2,060	1,200	2,220	1,290	2,395
			3,000	1,055	1,960	1,140	2,120	1,230	2,295	1,325	2,480	1,425	2,685
			4,000	1,165	2,185	1,260	2,365	1,355	2,570	1,465	2,790	1,575	3,030
			5,000	1,285	2,445	1,390	2,660	1,500	2,895	1,620	3,160	1,745	3,455
			6,000	1,425	2,755	1,540	3,015	1,665	3,300	1,800	3,620	1,940	3,990
			7,000	1,580	3,140	1,710	3,450	1,850	3,805	2,000	4,220	- - -	- - -
			8,000	1,755	3,615	1,905	4,015	2,060	4,480		- - -	- - -	- - -
2,200	49	54	S.L.	650	1,195	700	1,280	750	1,375	805	1,470	865	1,575
			1,000	710	1,310	765	1,405	825	1,510	885	1,615	950	1,735
			2,000	780	1,440	840	1,545	905	1,660	975	1,785	1,045	1,915
			3,000	855	1,585	925	1,705	995	1,835	1,070	1,975	1,150	2,130
			4,000	945	1,750	1,020	1,890	1,100	2,040	1,180	2,200	1,270	2,375
			5,000	1,040	1,945	1,125	2,105	1,210	2,275	1,305	2,465	1,405	2,665
			6,000	1,150	2,170	1,240	2,355	1,340	2,555	1,445	2,775	1,555	3,020
			7,000	1,270	2,440	1,375	2,655	1,485	2,890	1,605	3,155	1,730	3,450
			8,000	1,410	2,760	1,525	3,015	1,650	3,305	1,785	3,630	1,925	4,005
2,000	46	51	S.L.	525	970	565	1,035	605	1,110	650	1,185	695	1,265
			1,000	570	1,060	615	1,135	665	1,215	710	1,295	765	1,385
			2,000	625	1,160	675	1,240	725	1,330	780	1,425	840	1,525
			3,000	690	1,270	740	1,365	800	1,465	860	1,570	920	1,685
			4,000	755	1,400	815	1,500	880	1,615	945	1,735	1,015	1,865
			5,000	830	1,545	900	1,660	970	1,790	2,145	1,925	1,120	2,070
			6,000	920	1,710	990	1,845	1,070	1,990	2,405	2,145	1,235	2,315
			7,000	1,015	1,900	1,095	2,055	1,180	2,225	2,715	2,405	1,370	2,605
			8,000	1,125	2,125	1,215	2,305	1,310	2,500	1,410	2,715	1,520	2,950

Figure 10-23. *Takeoff distance short field charts.*

information for the departing airport and for the cruise altitude. Using *Figure 10-24,* calculate the fuel, time, and distance to climb based on the information provided.

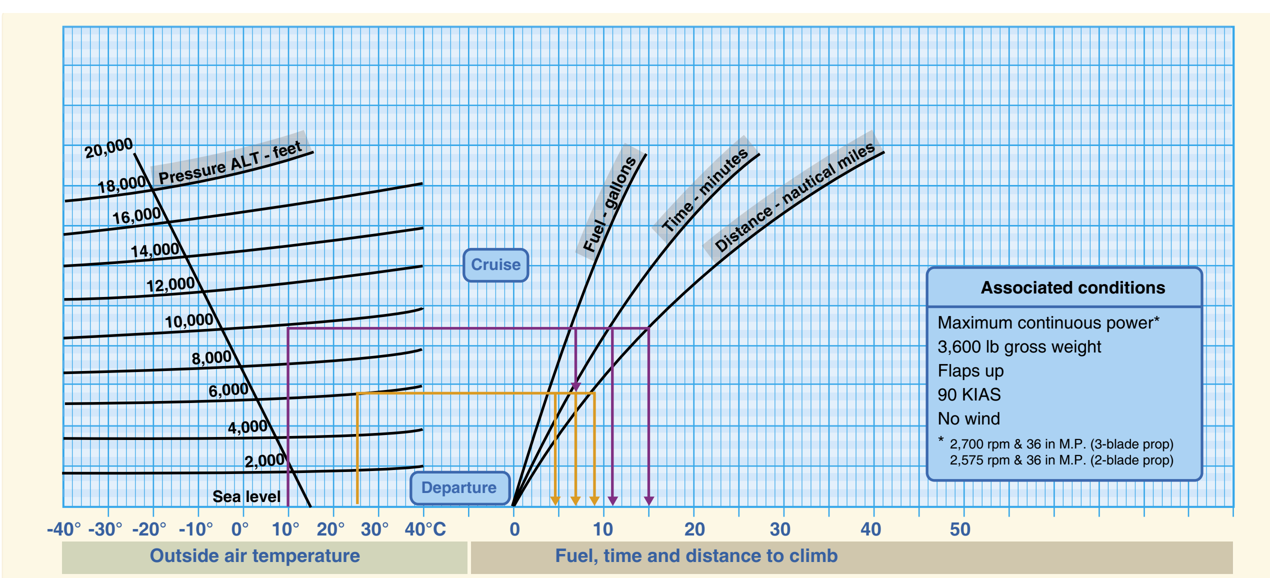


Figure 10-24. *Fuel time distance climb chart.*

Sample Problem 4

Departing Airport Pressure Altitude.................6,000 feet

Departing Airport OAT..25 °C

Cruise Pressure Altitude.................................10,000 feet

Cruise OAT..10 °C

First, find the information for the departing airport. Find the OAT for the departing airport along the bottom, left-hand side of the graph. Follow the line from 25 °C straight up until it intersects the line corresponding to the pressure altitude of 6,000 feet. Continue this line straight across until it intersects all three lines for fuel, time, and distance. Draw a line straight down from the intersection of altitude and fuel, altitude and time, and a third line at altitude and distance. It should read three and one-half gallons of fuel, 6.5 minutes of time, and nine NM. Next, repeat the steps to find the information for the cruise altitude. It should read six and one-half gallons of fuel, 11.5 minutes of time, and 15 NM. Take each set of numbers for fuel, time, and distance and subtract them from one another (6.5 – 3.5 = 3 gallons of fuel). It will take three gallons of fuel and 5 minutes of time to climb to 10,000 feet. During that climb, the distance covered is six NM. Remember, according to the notes at the top of the chart, these numbers do not take into account wind, and it is assumed maximum continuous power is being used.

The next example is a fuel, time, and distance-to-climb table. For this table, use the same basic criteria as for the previous chart. However, it is necessary to figure the information in a different manner. Refer to *Figure 10-25* to work the following sample problem.

Sample Problem 5

Departing Airport Pressure Altitude.................Sea level

Departing Airport OAT..22 °C

Cruise Pressure Altitude..................................8,000 feet

Takeoff Weight..3,400 pounds

To begin, find the given weight of 3,400 in the first column of the chart. Move across to the pressure altitude column to find the sea level altitude numbers. At sea level, the numbers read zero. Next, read the line that corresponds with the cruising altitude of 8,000 feet. Normally, a pilot would subtract these two sets of number from one another, but given the fact that the numbers read zero at sea level, it is known that the time to climb from sea level to 8,000 feet is 10 minutes. It is also known that 21 pounds of fuel will be used and 20 NM will be covered during the climb. However, the temperature is 22 °C, which is 7° above the standard temperature of 15 °C. The notes section of this chart indicate that the findings must be increased by ten percent for each 7° above standard. Multiply the findings by ten percent or .10 (10 x .10 = 1, 1 + 10 = 11 minutes). After accounting for the additional ten percent, the findings should read 11 minutes, 23.1 pounds of fuel, and 22 NM. Notice that the fuel is reported in pounds of fuel, not gallons. Aviation fuel weighs six pounds per gallon, so 23.1 pounds of fuel is equal to 3.85 gallons of fuel (23.1 ÷ 6 = 3.85).

Conditions:
Flaps up
Gear up
2,500 RPM
30" Hg
120 PPH fuel flow
Cowl flaps open
Standard temperature

NORMAL CLIMB
110 KIAS

Notes:
1. Add 16 pounds of fuel for engine start, taxi, and takeoff allowance.
2. Increase time, fuel, and distance by 10% for each 7°C above standard temperature.
3. Distances shown are based on zero wind.

Weight (pounds)	Press ALT (feet)	Rate of climb FPM	From sea level Time (minutes)	Fuel used (pounds)	Distance (nautical miles)
4,000	S.L.	605	0	0	0
	4,000	570	7	14	13
	8,000	530	14	28	27
	12,000	485	22	44	43
	16,000	430	31	62	63
	20,000	365	41	82	87
3,700	S.L.	700	0	0	0
	4,000	665	6	12	11
	8,000	625	12	24	23
	12,000	580	19	37	37
	16,000	525	26	52	53
	20,000	460	34	68	72
	S.L.	810	0	0	0
	4,000	775	5	10	9
3,400	8,000	735	10	21	20
	12,000	690	16	32	31
	16,000	635	22	44	45
	20,000	565	29	57	61

Figure 10-25. *Fuel time distance climb.*

The next example is a cruise and range performance chart. This type of table is designed to give TAS, fuel consumption, endurance in hours, and range in miles at specific cruise configurations. Use *Figure 10-26* to determine the cruise and range performance under the given conditions.

Sample Problem 6

Pressure Altitude...5,000 feet

RPM...2,400 rpm

Fuel Carrying Capacity.................38 gallons, no reserve

Find 5,000 feet pressure altitude in the first column on the left-hand side of the table. Next, find the correct rpm of 2,400 in the second column. Follow that line straight across and read the TAS of 116 mph, and a fuel burn rate of 6.9 gallons per hour. As per the example, the aircraft is equipped with a fuel carrying capacity of 38 gallons. Under this column,

Conditions	Gross weight—2,300 lb. Standard conditions Zero wind Lean mixture							
Notes	Maximum cruise is normally limited to 75% power.							
ALT	RPM	% BHP	TAS MPH	GAL/ Hour	38 gal (no reserve)		48 gal (no reserve)	
					Endr. hours	Range miles	Endr. hours	Range miles
2,500	2,700	86	134	9.7	3.9	525	4.9	660
	2,600	79	129	8.6	4.4	570	5.6	720
	2,500	72	123	7.8	4.9	600	6.2	760
	2,400	65	117	7.2	5.3	620	6.7	780
	2,300	58	111	6.7	5.7	630	7.2	795
	2,200	52	103	6.3	6.1	625	7.7	790
5,000	2,700	82	134	9.0	4.2	565	5.3	710
	2,600	75	128	8.1	4.7	600	5.9	760
	2,500	68	122	7.4	5.1	625	6.4	790
	2,400	61	116	6.9	5.5	635	6.9	805
	2,300	55	108	6.5	5.9	635	7.4	805
	2,200	49	100	6.0	6.3	630	7.9	795
7,500	2,700	78	133	8.4	4.5	600	5.7	755
	2,600	71	127	7.7	4.9	625	6.2	790
	2,500	64	121	7.1	5.3	645	6.7	810
	2,400	58	113	6.7	5.7	645	7.2	820
	2,300	52	105	6.2	6.1	640	7.7	810
10,000	2,650	70	129	7.6	5.0	640	6.3	810
	2,600	67	125	7.3	5.2	650	6.5	820
	2,500	61	118	6.9	5.5	655	7.0	830
	2,400	55	110	6.4	5.9	650	7.5	825
	2,300	49	100	6.0	6.3	635	8.0	800

Figure 10-26. *Cruise and range performance.*

read that the endurance in hours is 5.5 hours and the range in miles is 635 miles.

Cruise power setting tables are useful when planning cross-country flights. The table gives the correct cruise power settings, as well as the fuel flow and airspeed performance numbers at that altitude and airspeed.

Sample Problem 7

Pressure Altitude at Cruise...............................6,000 feet

OAT..36 °F above standard

Refer to *Figure 10-27* for this sample problem. First, locate the pressure altitude of 6,000 feet on the far left side of the table. Follow that line across to the far right side of the table under the 20 °C (or 36 °F) column. At 6,000 feet, the rpm setting of 2,450 will maintain 65 percent continuous power at 21.0 "Hg with a fuel flow rate of 11.5 gallons per hour and airspeed of 161 knots.

Another type of cruise chart is a best power mixture range graph. This graph gives the best range based on power setting and altitude. Using *Figure 10-28,* find the range at 65 percent power with and without a reserve based on the provided conditions.

Sample Problem 8

OAT..Standard

Pressure Altitude..5,000 feet

First, move up the left side of the graph to 5,000 feet and standard temperature. Follow the line straight across the graph until it intersects the 65 percent line under both the reserve and no reserve categories. Draw a line straight down from both intersections to the bottom of the graph. At 65 percent power with a reserve, the range is approximately 522 miles. At 65 percent power with no reserve, the range should be 581 miles.

The last cruise chart referenced is a cruise performance graph. This graph is designed to tell the TAS performance of the airplane depending on the altitude, temperature, and power

CRUISE POWER SETTING
65% MAXIMUM CONTINUOUS POWER (OR FULL THROTTLE)
2,800 POUNDS

Press ALT	ISA -20° (-36 °F)								Standard day (ISA)								ISA +20° (+36 °F)							
	IOAT		Engine speed	Man. press	Fuel flow per engine		TAS		IOAT		Engine speed	Man. press	Fuel flow per engine		TAS		IOAT		Engine speed	Man. press	Fuel flow per engine		TAS	
	°F	°C	RPM	" HG	PSI	GPH	kts	MPH	°F	°C	RPM	" HG	PSI	GPH	kts	MPH	°F	°C	RPM	" HG	PSI	GPH	kts	MPH
S.L.	27	-3	2,450	20.7	6.6	11.5	147	169	63	17	2,450	21.2	6.6	11.5	150	173	99	37	2,450	21.8	6.6	11.5	153	176
2,000	19	-7	2,450	20.4	6.6	11.5	149	171	55	13	2,450	21.0	6.6	11.5	153	176	91	33	2,450	21.5	6.6	11.5	156	180
4,000	12	-11	2,450	20.1	6.6	11.5	152	175	48	9	2,450	20.7	6.6	11.5	156	180	84	29	2,450	21.3	6.6	11.5	159	183
6,000	5	-15	2,450	19.8	6.6	11.5	155	178	41	5	2,450	20.4	6.6	11.5	158	182	79	26	2,450	21.0	6.6	11.5	161	185
8,000	-2	-19	2,450	19.5	6.6	11.5	157	181	36	2	2,450	20.2	6.6	11.5	161	185	72	22	2,450	20.8	6.6	11.5	164	189
10,000	-8	-22	2,450	19.2	6.6	11.5	160	184	28	-2	2,450	19.9	6.6	11.5	163	188	64	18	2,450	20.3	6.5	11.4	166	191
12,000	-15	-26	2,450	18.8	6.4	11.3	162	186	21	-6	2,450	18.8	6.1	10.9	163	188	57	14	2,450	18.8	5.9	10.6	163	188
14,000	-22	-30	2,450	17.4	5.8	10.5	159	183	14	-10	2,450	17.4	5.6	10.1	160	184	50	10	2,450	17.4	5.4	9.8	160	184
16,000	-29	-34	2,450	16.1	5.3	9.7	156	180	7	-14	2,450	16.1	5.1	9.4	156	180	43	6	2,450	16.1	4.9	9.1	155	178

Notes
1. Full throttle manifold pressure settings are approximate.
2. Shaded area represents operation with full throttle.

Figure 10-27. *Cruise power setting.*

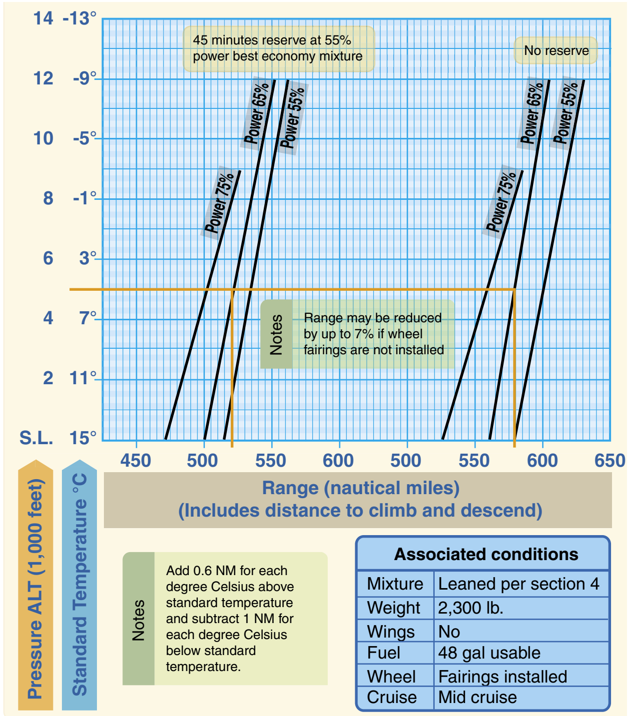


Figure 10-28. *Best power mixture range.*

setting. Using *Figure 10-29,* find the TAS performance based on the given information.

Sample Problem 9

OAT...16 °C

Pressure Altitude...6,000 feet

Power Setting...............................65 percent, best power

Wheel Fairings...Not installed

Begin by finding the correct OAT on the bottom, left side of the graph. Move up that line until it intersects the pressure altitude of 6,000 feet. Draw a line straight across to the 65 percent, best power line. This is the solid line, which represents best economy. Draw a line straight down from this intersection to the bottom of the graph. The TAS at 65 percent best power is 140 knots. However, it is necessary to subtract 8 knots from the speed since there are no wheel fairings. This note is listed under the title and conditions. The TAS will be 132 knots.

Crosswind and Headwind Component Chart

Every aircraft is tested according to Federal Aviation Administration (FAA) regulations prior to certification. The aircraft is tested by a pilot with average piloting skills in 90° crosswinds with a velocity up to 0.2 V_{SO} or two-tenths of the aircraft's stalling speed with power off, gear down, and flaps down. This means that if the stalling speed of the aircraft is 45 knots, it must be capable of landing in a 9-knot, 90° crosswind. The maximum demonstrated crosswind component is published in the AFM/POH. The crosswind and headwind component chart allows for figuring the headwind and crosswind component for any given wind direction and velocity.

Sample Problem 10

Runway..17

Wind...140° at 25 knots

Refer to *Figure 10-30* to solve this problem. First, determine how many degrees difference there is between the runway and the wind direction. It is known that runway 17 means a direction of 170°; from that subtract the wind direction of 140°. This gives a 30° angular difference, or wind angle. Next, locate the 30° mark and draw a line from there until it intersects

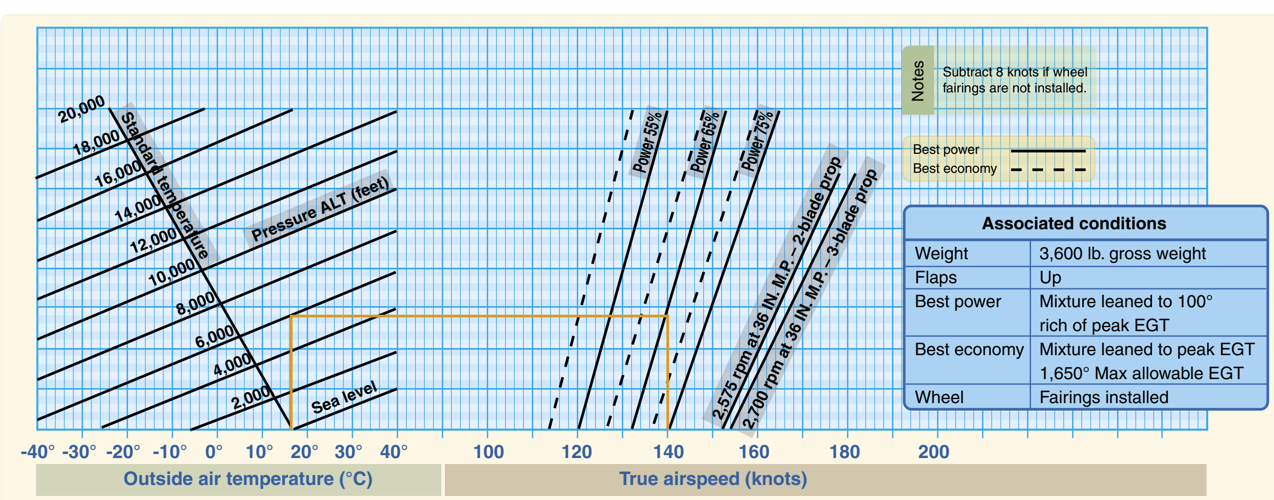


Figure 10-29. *Cruise performance graph.*

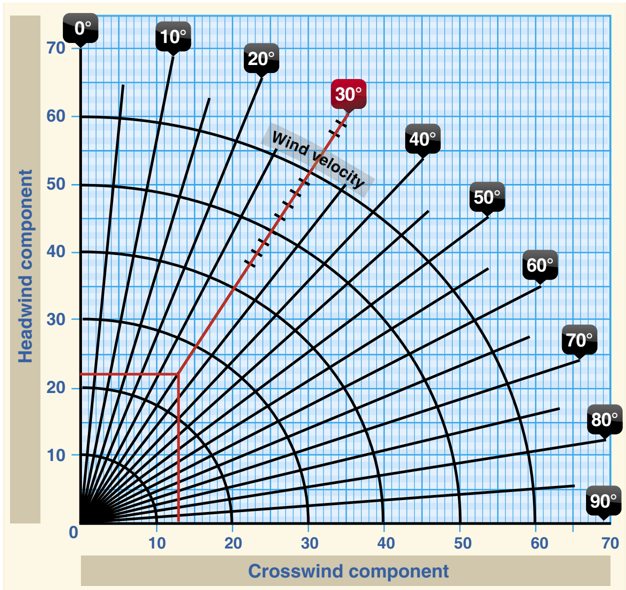


Figure 10-30. *Crosswind component chart.*

the correct wind velocity of 25 knots. From there, draw a line straight down and a line straight across. The headwind component is 22 knots and the crosswind component is 13 knots. This information is important when taking off and landing so that, first of all, the appropriate runway can be picked if more than one exists at a particular airport, but also so that the aircraft is not pushed beyond its tested limits.

Landing Charts

Landing performance is affected by variables similar to those affecting takeoff performance. It is necessary to compensate for differences in density altitude, weight of the airplane, and headwinds. Like takeoff performance charts, landing distance information is available as normal landing information, as well as landing distance over a 50 foot obstacle. As usual, read the associated conditions and notes in order to ascertain the basis of the chart information. Remember, when calculating landing distance that the landing weight will not be the same as the takeoff weight. The weight must be recalculated to compensate for the fuel that was used during the flight.

Sample Problem 11

Pressure Altitude...1,250 feet

Temperature...Standard

Refer to *Figure 10-31*. This example makes use of a landing distance table. Notice that the altitude of 1,250 feet is not on this table. It is, therefore, necessary to interpolate to find the correct landing distance. The pressure altitude of 1,250 is halfway between sea level and 2,500 feet. First, find the column for sea level and the column for 2,500 feet. Take the total distance of 1,075 for sea level and the total distance of 1,135 for 2,500 and add them together. Divide the total by two to obtain the distance for 1,250 feet. The distance is 1,105 feet total landing distance to clear a 50 foot obstacle. Repeat this process to obtain the ground roll distance for the pressure altitude. The ground roll should be 457.5 feet.

Sample Problem 12

OAT.. 57 °F

Pressure Altitude... 4,000 feet

Landing Weight...2,400 pounds

Headwind.. 6 knots

Obstacle Height.. 50 feet

Using the given conditions and *Figure 10-32,* determine the landing distance for the aircraft. This graph is an example of a combined landing distance graph and allows compensation for temperature, weight, headwinds, tailwinds, and varying obstacle height. Begin by finding the correct OAT on the scale on the left side of the chart. Move up in a straight line to the correct pressure altitude of 4,000 feet. From this intersection, move straight across to the first dark reference line. Follow the lines in the same diagonal fashion until the correct landing weight is reached. At 2,400 pounds, continue in a straight line across to the second dark reference line. Once again, draw a line in a diagonal manner to the correct wind component and then straight across to the third dark

LANDING DISTANCE

Conditions:
Flaps lowered to 40°
Power off
Hard surface runway
Zero wind

Gross weight lb	Approach speed IAS, MPH	At sea level & 59 °F		At 2,500 ft & 59 °F		At 5,000 ft & 41 °F		At 7,500 ft & 32 °F	
		Ground roll	Total to clear 50 ft OBS	Ground roll	Total to clear 50 ft OBS	Ground roll	Total to clear 50 ft OBS	Ground roll	Total to clear 50 ft OBS
1,600	60	445	1,075	470	1,135	495	1,195	520	1,255

Note:
1. Decrease the distances shown by 10% for each 4 knots of headwind.
2. Increase the distance by 10% for each 60 °F temperature increase above standard.
3. For operation on a dry, grass runway, increase distances (both “ground roll” and “total to clear 50 ft obstacle”) by 20% of the “total to clear 50 ft obstacle” figure.

Figure 10-31. *Landing distance table.*

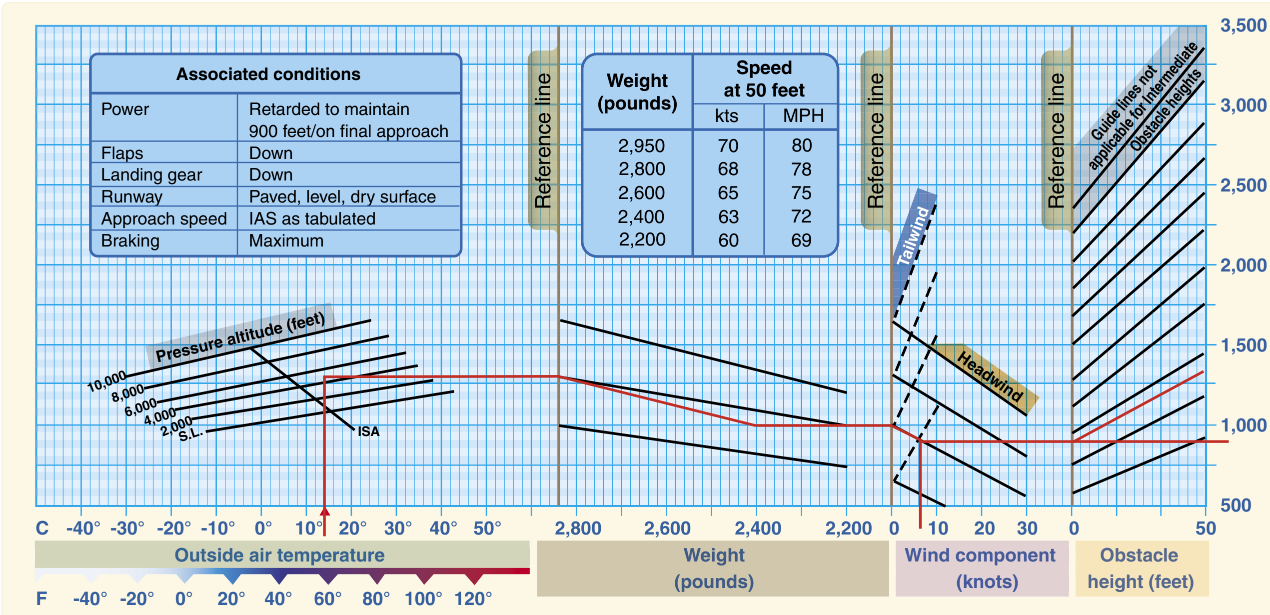


Figure 10-32. *Landing distance graph.*

reference line. From this point, draw a line in two separate directions: one straight across to figure the ground roll and one in a diagonal manner to the correct obstacle height. This should be 900 feet for the total ground roll and 1,300 feet for the total distance over a 50 foot obstacle.

Stall Speed Performance Charts

Stall speed performance charts are designed to give an understanding of the speed at which the aircraft will stall in a given configuration. This type of chart will typically take into account the angle of bank, the position of the gear and flaps, and the throttle position. Use *Figure 10-33* and the accompanying conditions to find the speed at which the airplane will stall.

Sample Problem 13

Power.. OFF

Flaps.. Down

Gear.. Down

Angle of Bank... 45°

First, locate the correct flap and gear configuration. The bottom half of the chart should be used since the gear and flaps are down. Next, choose the row corresponding to a power-off situation. Now, find the correct angle of bank column, which is 45°. The stall speed is 78 mph, and the stall speed in knots would be 68 knots.

Gross weight 2,750 lb			Angle of bank			
			Level	30°	45°	60°
			Gear and flaps up			
Power	On	MPH	62	67	74	88
		knots	54	58	64	76
	Off	MPH	75	81	89	106
		knots	65	70	77	92
			Gear and flaps down			
Power	On	MPH	54	58	64	76
		knots	47	50	56	66
	Off	MPH	66	71	78	93
		knots	57	62	68	81

Figure 10-33. *Stall speed table.*

Performance charts provide valuable information to the pilot. Take advantage of these charts. A pilot can predict the performance of the aircraft under most flying conditions, and this enables a better plan for every flight. The Code of Federal Regulations (CFR) requires that a pilot be familiar with all information available prior to any flight. Pilots should use the information to their advantage as it can only contribute to safety in flight.

Transport Category Airplane Performance

Transport category aircraft are certificated under Title 14 of the CFR (14 CFR) parts 25 and 29. The airworthiness certification standards of part 25 and 29 require proven levels of performance and guarantee safety margins for these aircraft, regardless of the specific operating regulations under which they are employed.

Major Differences in Transport Category Versus Non-Transport Category Performance Requirements

- Full temperature accountability—all of the performance charts for the transport category aircraft require that takeoff and climb performance be computed with the full effects of temperature considered.
- Climb performance expressed as percent gradient of climb—the transport category aircraft's climb performance is expressed as a percent gradient of climb rather than a figure calculated in fpm of climb. This percent gradient of climb is a much more practical expression of performance since it is the aircraft's angle of climb that is critical in an obstacle clearance situation.
- Change in lift-off technique—lift-off technique in transport category aircraft allows the reaching of V_2 (takeoff safety speed) after the aircraft is airborne. This is possible because of the excellent acceleration and reliability characteristics of the engines on these aircraft and due to the larger surplus of power.
- Performance requirements applicable to all segments of aviation—all aircraft certificated by the FAA in the transport category, whatever the size, must be operated in accordance with the same performance criteria. This applies to both commercial and non-commercial operations.

Performance Requirements

The performance requirements that the transport category aircraft must meet are:

Takeoff

- Takeoff speeds
- Takeoff runway required
- Takeoff climb required
- Obstacle clearance requirements

Landing

- Landing speeds
- Landing runway required
- Landing climb required

Takeoff Planning

Listed below are the speeds that affect the transport category aircraft's takeoff performance. The flight crew must be thoroughly familiar with each of these speeds and how they are used in takeoff planning.

- V_S—stalling speed or the minimum steady flight speed at which the aircraft is controllable.
- V_{MCG}—minimum control speed on the ground, with one engine inoperative, (critical engine on two-engine airplanes) takeoff power on other engine(s), using aerodynamic controls only for directional control (must be less than V_1).
- V_{MCA}—minimum control speed in the air, with one engine inoperative, (critical engine on two-engine aircraft) operating engine(s) at takeoff power, maximum of 5° bank into the good engine(s).
- V_1—critical engine failure speed or decision speed. Engine failure below this speed shall result in an aborted takeoff; above this speed the takeoff run should be continued.
- V_R—speed at which the rotation of the aircraft is initiated to takeoff attitude. The speed cannot be less than V_1 or less than 1.05 times V_{MC}. With an engine failure, it must also allow for the acceleration to V_2 at the 35-foot height at the end of the runway.
- V_{LOF}—lift-off speed. The speed at which the aircraft first becomes airborne.
- V_2—the takeoff safety speed which must be attained at the 35-foot height at the end of the required runway distance. This is essentially the best one-engine operative angle of climb speed for the aircraft and should be held until clearing obstacles after takeoff, or until at least 400 feet above the ground.
- V_{FS}—final segment climb speed, which is based upon one-engine inooerative climb, clean configuration, and mximum continuos power setting.

All of the V speeds should be considered during every takeoff. The V_1, V_R, V_2, and V_{FS} speeds should be visibly posted in the flightdeck for reference during the takeoff.

Takeoff speeds vary with aircraft weight. Before takeoff speeds can be computed, the pilot must first determine the maximum allowable takeoff weight. The three items that can limit takeoff weight are runway requirements, takeoff climb requirements, and obstacle clearance requirements.

Runway Requirements

The runway requirements for takeoff are affected by:

- Pressure altitude
- Temperature
- Headwind component
- Runway gradient or slope
- Aircraft weight

The runway required for takeoff must be based upon the possible loss of an engine at the most critical point, which is at V_1 (decision speed). By regulation, the aircraft's takeoff weight has to accommodate the longest of three distances:

1. Accelerate-go distance—the distance required to accelerate to V_1 with all engines at takeoff power, experience an engine failure at V_1 and continue the takeoff on the remaining engine(s). The runway required includes the distance required to climb to 35 feet by which time V_2 speed must be attained.
2. Accelerate-stop distance—the distance required to accelerate to V_1 with all engines at takeoff power, experience an engine failure at V_1, and abort the takeoff and bring the aircraft to a stop using braking action only (use of thrust reversing is not considered).
3. Takeoff distance—the distance required to complete an all-engines operative takeoff to the 35-foot height. It must be at least 15 percent less than the distance required for a one-engine inoperative engine takeoff. This distance is not normally a limiting factor as it is usually less than the one-engine inoperative takeoff distance.

These three required takeoff runway considerations are shown in *Figure 10-34*.

Balanced Field Length

In most cases, the pilot will be working with a performance chart for takeoff runway required, which will give "balanced field length" information. This means that the distance

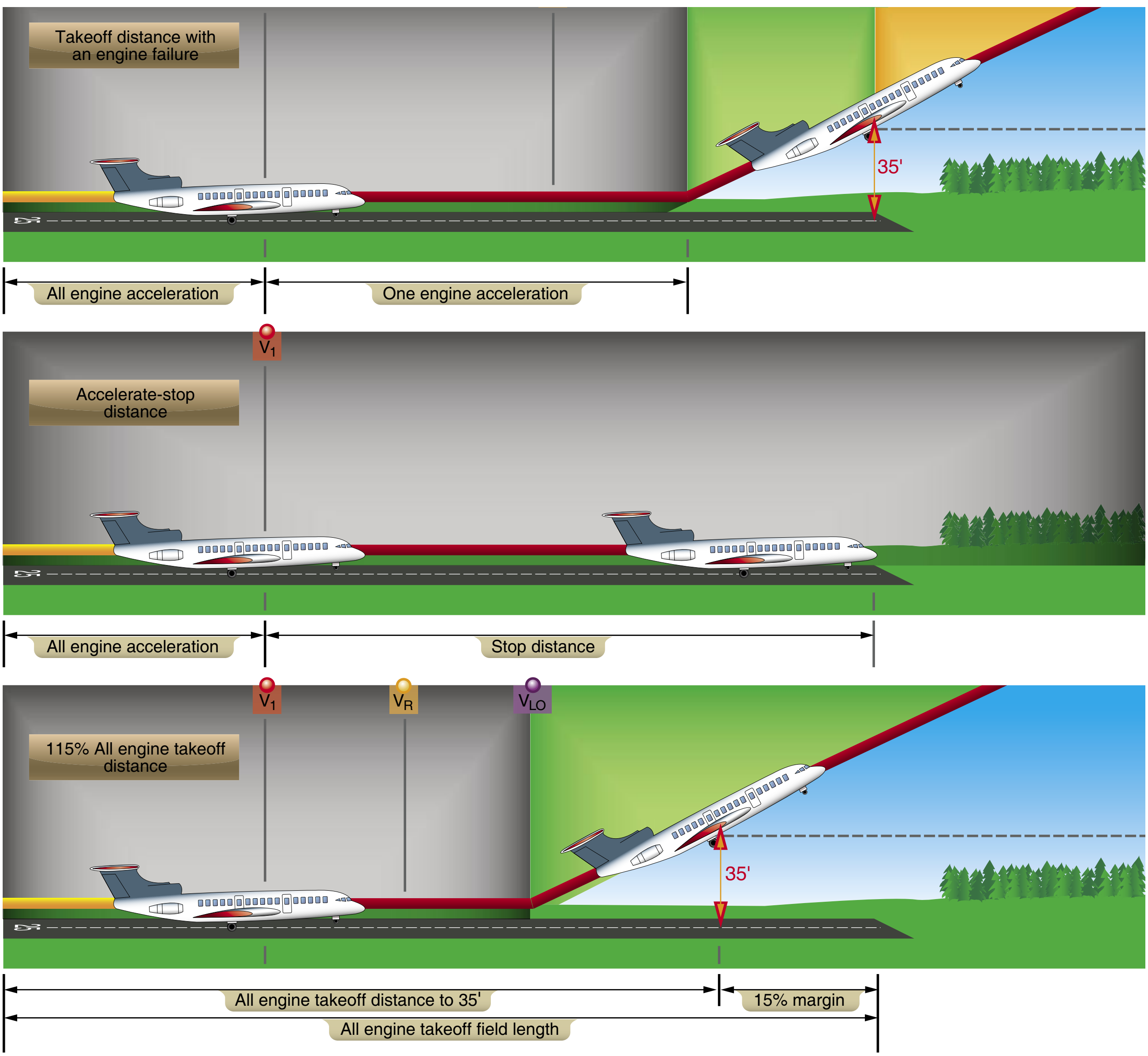


Figure 10-34. *Minimum required takeoff.*

shown for the takeoff will include both the accelerate-go and accelerate-stop distances. One effective means of presenting the normal takeoff data is shown in the tabulated chart in *Figure 10-35*.

The chart in *Figure 10-35* shows the runway distance required under normal conditions and is useful as a quick reference chart for the standard takeoff. The V speeds for the various weights and conditions are also shown.

For other than normal takeoff conditions, such as with engine anti-ice, anti-skid brakes inoperative, or extremes in temperature or runway slope, the pilot should consult the appropriate takeoff performance charts in the performance section of the AFM.

There are other occasions of very high weight and temperature where the runway requirement may be dictated by the maximum brake kinetic energy limits that affect

TAKEOFF RUNWAY REQUIREMENTS
Standard ISA conditions

Conditions:
Cabin pressurization On
Zero slope runway
No flaps—Anti-ice RAM air inlets Off
Anti-skid On
Distances—100 feet (V_1 – KIAS)

Notes:
Shaded area indicates conditions that do not meet second segment climb requirements.
Refer to F.M. for takeoff limitations.

Takeoff gross weight at brake release	Temp. °F	Temp. °C	Pressure altitude (feet): Sea level (V_1)	1,000 (V_1)	2,000 (V_1)	3,000 (V_1)	4,000 (V_1)	5,000 (V_1)	6,000 (V_1)	Headwind (knots)
19,612 V_R = 126 V_2 = 134	30	-1.1	47 (121)	48 (121)	50 (120)	53 (121)	57 (122)	62 (123)	70 (123)	0
	50	10	48 (121)	51 (121)	55 (121)	60 (122)	63 (123)	69 (124)	77 (125)	
	70	21	53 (122)	56 (122)	60 (123)	65 (124)	70 (125)	77 (125)	85 (126)	
	90	32	58 (123)	62 (124)	68 (124)	73 (125)	78 (126)	85 (127)	95 (129)	
	30	-1.1	43 (121)	43 (121)	45 (120)	48 (121)	52 (122)	56 (123)	64 (123)	20
	50	10	43 (121)	46 (121)	50 (122)	55 (122)	57 (123)	63 (124)	70 (125)	
	70	21	48 (122)	51 (122)	55 (123)	59 (124)	63 (125)	70 (125)	77 (126)	
	90	32	53 (123)	57 (124)	62 (124)	66 (125)	71 (126)	77 (127)	85 (129)	
19,000 V_R = 124 V_2 = 131	30	-1.1	45 (118)	45 (118)	47 (117)	50 (118)	54 (119)	59 (120)	66 (120)	0
	50	10	46 (118)	48 (118)	51 (118)	56 (119)	59 (120)	65 (121)	73 (121)	
	70	21	50 (118)	53 (119)	57 (120)	66 (121)	66 (121)	72 (122)	80 (123)	
	90	32	55 (120)	59 (121)	64 (121)	73 (122)	73 (123)	80 (124)	90 (124)	
	30	-1.1	40 (118)	41 (118)	43 (117)	45 (118)	49 (119)	54 (120)	60 (120)	20
	50	10	42 (118)	44 (118)	46 (118)	51 (119)	54 (120)	59 (121)	66 (121)	
	70	21	45 (118)	48 (119)	52 (120)	56 (121)	60 (121)	65 (122)	72 (123)	
	90	32	50 (120)	54 (121)	58 (121)	63 (122)	66 (123)	73 (124)	81 (124)	
18,000 V_R = 119 V_2 = 127	30	-1.1	40 (114)	41 (114)	42 (113)	45 (113)	49 (114)	53 (115)	60 (115)	0
	50	10	41 (115)	43 (114)	46 (114)	50 (115)	53 (115)	59 (116)	66 (117)	
	70	21	45 (114)	48 (115)	51 (115)	56 (116)	59 (116)	65 (116)	72 (117)	
	90	32	50 (115)	53 (116)	58 (116)	62 (117)	66 (118)	73 (118)	80 (119)	
	30	-1.1	36 (114)	37 (114)	38 (113)	41 (113)	45 (114)	48 (115)	54 (115)	20
	50	10	37 (115)	39 (114)	42 (114)	46 (115)	48 (115)	54 (116)	60 (117)	
	70	21	41 (114)	44 (115)	46 (115)	51 (116)	56 (116)	59 (116)	65 (117)	
	90	32	46 (115)	48 (116)	53 (116)	56 (117)	60 (118)	66 (118)	73 (119)	
17,000 V_R = 115 V_2 = 124	30	-1.1	36 (108)	37 (108)	38 (107)	40 (108)	44 (109)	48 (110)	53 (111)	0
	50	10	37 (110)	39 (108)	41 (109)	45 (110)	48 (110)	53 (111)	59 (112)	
	70	21	40 (108)	43 (110)	46 (111)	50 (111)	53 (112)	58 (113)	65 (113)	
	90	32	45 (111)	46 (112)	52 (112)	56 (113)	59 (114)	65 (114)	72 (114)	
	30	-1.1	32 (108)	33 (108)	34 (107)	36 (108)	40 (109)	44 (110)	48 (111)	20
	50	10	34 (110)	35 (108)	37 (109)	41 (110)	44 (110)	48 (111)	54 (112)	
	70	21	36 (108)	39 (110)	42 (111)	45 (111)	48 (112)	53 (111)	59 (113)	
	90	32	41 (111)	44 (112)	47 (112)	51 (113)	54 (114)	59 (114)	65 (114)	
16,000 V_R = 111 V_2 = 120	30	-1.1	32 (104)	33 (103)	34 (103)	36 (103)	39 (105)	43 (106)	48 (106)	0
	50	10	34 (105)	35 (103)	37 (104)	41 (105)	43 (106)	47 (107)	53 (107)	
	70	21	36 (104)	38 (105)	41 (105)	45 (106)	48 (107)	52 (107)	58 (108)	
	90	32	41 (106)	43 (107)	46 (107)	50 (108)	53 (108)	58 (109)	64 (110)	
	30	-1.1	29 (104)	30 (103)	31 (103)	32 (103)	35 (105)	39 (106)	44 (106)	20
	50	10	31 (105)	32 (103)	33 (104)	37 (105)	39 (106)	43 (107)	48 (107)	
	70	21	32 (104)	34 (105)	37 (105)	41 (106)	44 (107)	47 (107)	53 (108)	
	90	32	37 (106)	39 (107)	42 (107)	45 (108)	48 (108)	53 (109)	58 (110)	
15,000 V_R = 106 V_2 = 116	30	-1.1	28 (98)	30 (98)	30 (98)	32 (98)	35 (99)	38 (101)	42 (101)	0
	50	10	30 (100)	31 (98)	33 (99)	36 (100)	38 (101)	42 (102)	46 (102)	
	70	21	32 (99)	34 (100)	37 (101)	40 (102)	42 (102)	46 (102)	51 (103)	
	90	32	36 (101)	38 (102)	41 (102)	44 (103)	47 (104)	51 (104)	56 (105)	
	30	-1.1	25 (98)	27 (98)	27 (98)	29 (98)	32 (99)	34 (101)	38 (101)	20
	50	10	27 (100)	29 (98)	30 (99)	32 (100)	34 (101)	38 (102)	42 (102)	
	70	21	29 (99)	31 (100)	33 (101)	36 (102)	38 (102)	42 (102)	46 (103)	
	90	32	32 (101)	34 (102)	37 (102)	40 (103)	43 (104)	46 (104)	51 (105)	

Figure 10-35. *Normal takeoff runway required.*

the aircraft's ability to stop. Under these conditions, the accelerate-stop distance may be greater than the accelerate-go. The procedure to bring performance back to a balanced field takeoff condition is to limit the V_1 speed so that it does not exceed the maximum brake kinetic energy speed (sometimes called VBE). This procedure also results in a reduction in allowable takeoff weight.

Climb Requirements

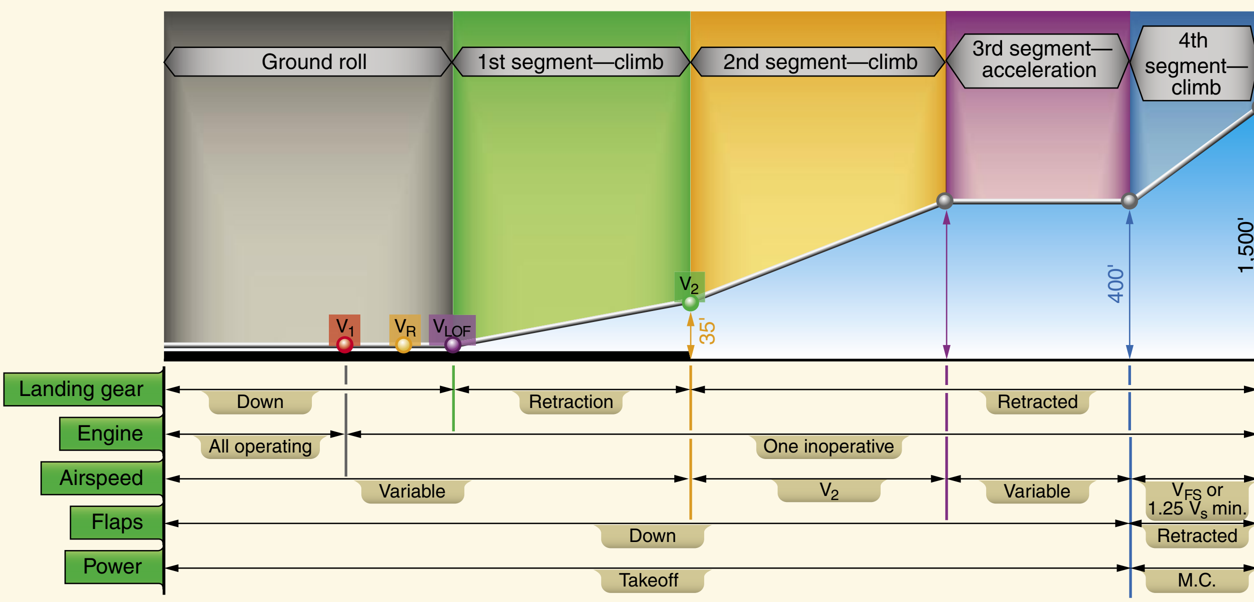
After the aircraft has reached the 35 foot height with one engine inoperative, there is a requirement that it be able to climb at a specified climb gradient. This is known as the takeoff flightpath requirement. The aircraft's performance must be considered based upon a one-engine inoperative climb up to 1,500 feet above the ground. The takeoff flightpath profile with required gradients of climb for the various segments and configurations is shown in *Figure 10-36.*

NOTE: Climb gradient can best be described as being a specific gain of vertical height for a given distance covered horizontally. For instance, a 2.4 percent gradient means that 24 feet of altitude would be gained for each 1,000 feet of distance covered horizontally across the ground.

The following brief explanation of the one-engine inoperative climb profile may be helpful in understanding the chart in *Figure 10-36.*

First Segment

This segment is included in the takeoff runway required charts, and is measured from the point at which the aircraft becomes airborne until it reaches the 35-foot height at the end of the runway distance required. Speed initially is V_{LO} and must be V_2 at the 35 foot height.



Items		1st T/O segment	2nd T/O segment	Transition (acceleration)	Final T/O segment
	2 Engine	Positive	2.4%	Positive	1.2%
*	3 Engine	3.0%	2.7%	Positive	1.5%
	4 Engine	5.0%	3.0%	Positive	1.7%
Wing flaps		T.O.	T.O.	T.O.	Up
Landing gear		Down	Up	Up	Up
Engines		1 Out	1 Out	1 Out	1 Out
Power		T.O.	T.O.	T.O.	M.C.
Air speed		$V_{LOF} \rightarrow V_2$	V_2	$V_2 \rightarrow 1.25\ V_S$(Min)	$1.25\ V_S$(Min)

* Required Absolute Minimum Gradient of Flightpath

M.C. = Maximum continuous
V_1 = Critical-engine-failure speed
V_2 = Takeoff safety speed
V_S = Calibrated stalling speed, or minimum steady flight speed at which the aircraft is controllable
V_R = Speed at which aircraft can start safely raising nose wheel off surface (Rotational Speed)
V_{LOF} = Speed at point where airplane lifts off

Figure 10-36. *One engine inoperative takeoff.*

Second Segment

This is the most critical segment of the profile. The second segment is the climb from the 35 foot height to 400 feet above the ground. The climb is done at full takeoff power on the operating engine(s), at V_2 speed, and with the flaps in the takeoff configuration. The required climb gradient in this segment is 2.4 percent for two-engine aircraft, 2.7 percent for three-engine aircraft, and 3.0 percent for four-engine aircraft.

Third or Acceleration Segment

During this segment, the airplane is considered to be maintaining the 400 feet above the ground and accelerating from the V_2 speed to the V_{FS} speed before the climb profile is continued. The flaps are raised at the beginning of the acceleration segment and power is maintained at the takeoff setting as long as possible (5 minutes maximum).

Fourth or Final Segment

This segment is from the 400 to 1,500 foot AGL altitude with power set at maximum continuous. The required climb in this segment is a gradient of 1.2 percent for two-engine airplanes, 1.55 for three-engine airplanes, and 1.7 percent for four-engine airplanes.

Second Segment Climb Limitations

The second segment climb requirements, from 35 to 400 feet, are the most restrictive (or hardest to meet) of the climb segments. The pilot must determine that the second segment climb is met for each takeoff. In order to achieve this performance at the higher density altitude conditions, it may be necessary to limit the takeoff weight of the aircraft.

It must be realized that, regardless of the actual available length of the takeoff runway, takeoff weight must be adjusted so that the second segment climb requirements can be met. The aircraft may well be capable of lifting off with one engine inoperative, but it must then be able to climb and clear obstacles. Although second segment climb may not present much of a problem at the lower altitudes, at the higher altitude airports and higher temperatures, the second segment climb chart should be consulted to determine the effects on maximum takeoff weights before figuring takeoff runway distance required.

Air Carrier Obstacle Clearance Requirements

Regulations require that large transport category turbine powered aircraft certificated after September 30, 1958, be taken off at a weight that allows a net takeoff flightpath (one engine inoperative) that clears all obstacles either by a height of at least 35 feet vertically, or by at least 200 feet horizontally within the airport boundaries and by at least 300 feet horizontally after passing the boundaries. The takeoff flightpath is considered to begin 35 feet above the takeoff surface at the end of the takeoff distance, and extends to a point in the takeoff at which the aircraft is 1,500 feet above the takeoff surface, or at which the transition from the takeoff to the en route configuration is completed. The net takeoff flightpath is the actual takeoff flightpath reduced at each point by 0.8 percent for two engine aircraft, 0.9 percent for three-engine aircraft, and 1.0 percent for four-engine aircraft.

Air carrier pilots therefore are responsible not only for determining that there is enough runway available for an engine inoperative takeoff (balanced field length), and the ability to meet required climb gradients; but they must also assure that the aircraft will safely be able to clear any obstacles that may be in the takeoff flightpath. The net takeoff flightpath and obstacle clearance required are shown in *Figure 10-37*.

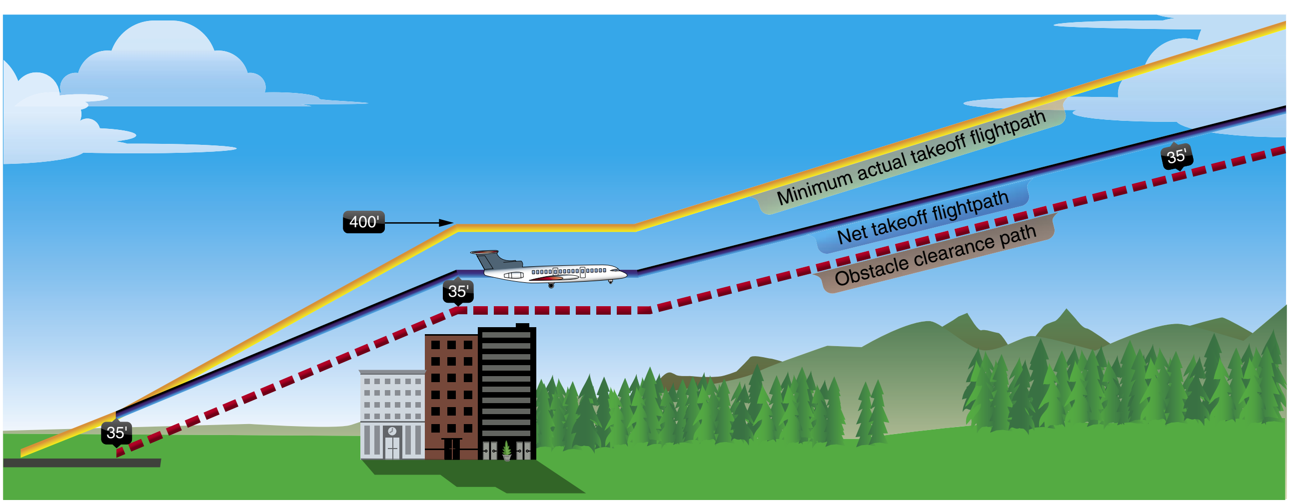


Figure 10-37. *Takeoff obstacle clearance.*

The usual method of computing net takeoff flightpath performance is to add up the total ground distances required for each of the climb segments and/or use obstacle clearance performance charts in the AFM. Although this obstacle clearance requirement is seldom a limitation at the normally used airports, it is quite often an important consideration under critical conditions such as high takeoff weight and/or high density altitude. Consider that at a 2.4 percent climb gradient (2.4 feet up for every 100 feet forward) a 1,500 foot altitude gain would take a horizontal distance of 10.4 NM to achieve.

Summary of Takeoff Requirements

In order to establish the allowable takeoff weight for a transport category aircraft, at any airfield, the following must be considered:

- Airfield pressure altitude
- Temperature
- Headwind component
- Runway length
- Runway gradient or slope
- Obstacles in the flightpath

Once the above details are known and applied to the appropriate performance charts, it is possible to determine the maximum allowable takeoff weight. This weight would be the lower of the maximum weights as allowed by:

- Balanced field length required
- Engine inoperative climb ability (second segment limited)
- Obstacle clearance requirement

In practice, restrictions to takeoff weight at low altitude airports are usually due to runway length limitations; engine inoperative climb limitations are most common at the higher altitude airports. All limitations to weight must be observed. Since the combined weight of fuel and payload in the aircraft may amount to nearly half the maximum takeoff weight, it is usually possible to reduce fuel weight to meet takeoff limitations. If this is done, however, flight planning must be recalculated in light of reduced fuel and range.

Landing Performance

As in the takeoff planning, certain speeds must be considered during landing. These speeds are shown below.

- V_{SO}—stalling speed or the minimum steady flight speed in the landing configuration.
- V_{REF}—1.3 times the stalling speed in the landing configuration. This is the required speed at the 50-foot height above the threshold end of the runway.
- Approach climb—the speed which gives the best climb performance in the approach confguration, with one engine inoperative, and with maximum takeoff power on the operating engine(s). The required gradient of climb in this configuration is 2.1 percent for two-engine aircraft, 2.4 percent for three-emgine aircraft, and 2.7 percent for four-engine aircraft.
- Landing climb—the speed giving the best performance in the full landing configuration with maximum takeoff power on all engines. The gradient of climb required in this configuration is 3.2 percent.

Planning the Landing

As in the takeoff, the landing speeds shown above should be precomputed and visible to both pilots prior to the landing. The V_{REF} speed, or threshold speed, is used as a reference speed throughout the traffic pattern or instrument approach as in the following example:

V_{REF} plus 30K	Downwind or procedure turn
V_{REF} plus 20K	Base leg or final course inbound to final fix
V_{REF} plus 10K	Final or final course inbound from fix (ILS final)
V_{REF}	Speed at the 50 foot height above the threshold

Landing Requirements

The maximum landing weight of an aircraft can be restricted by either the approach climb requirements or by the landing runway available.

Approach Climb Requirements

The approach climb is usually more limiting (or more difficult to meet) than the landing climb, primarily because it is based upon the ability to execute a missed approach with one engine inoperative. The required climb gradient can be affected by pressure altitude and temperature and, as in the second segment climb in the takeoff, aircraft weight must be limited as needed in order to comply with this climb requirement.

Landing Runway Required

The runway distance needed for landing can be affected by the following:

- Pressure altitude
- Temperature
- Headwind component
- Runway gradient or slope
- Aircraft weight

In computing the landing distance required, some manufacturers do not include all of the above items in their charts, since the regulations state that only pressure altitude, wind, and aircraft weight must be considered. Charts are provided for anti-skid on and anti-skid off conditions, but the use of reverse thrust is not used in computing required landing distances.

The landing distance, as required by the regulations, is that distance needed to land and come to a complete stop from a point 50 feet above the threshold end of the runway. It includes the air distance required to travel from the 50 foot height to touchdown (which can consume 1,000 feet of runway distance), plus the stopping distance, with no margin left over. This is all that is required for 14 CFR part 91 operators (non-air carrier), and all that is shown on some landing distance required charts.

For air carriers and other commercial operators subject to 14 CFR part 121, a different set of rules applies stating that the required landing distance from the 50 foot height cannot exceed 60 percent of the actual runway length available. In all cases, the minimum airspeed allowed at the 50 foot height must be no less than 1.3 times the aircraft's stalling speed in the landing configuration. This speed is commonly called the aircraft's V_{REF} speed and varies with landing weight. *Figure 10-38* is a diagram of these landing runway requirements.

Summary of Landing Requirements

In order to establish the allowable landing weight for a transport category aircraft, the following details must be considered:

- Airfield pressure altitude
- Temperature
- Headwind component
- Runway length
- Runway gradient or slope
- Runway surface condition

With these details, it is possible to establish the maximum allowable landing weight, which will be the lower of the weights as dictated by:

- Landing runway requirements
- Approach climb requirements

In practice, the approach climb limitations (ability to climb in approach configuration with one engine inoperative) are seldom encountered because the landing weights upon arrival at the destination airport are usually low. However, as in the second segment climb requirement for takeoff, this approach climb gradient must be met and landing weights must be restricted if necessary. The most likely conditions that would make the approach climb critical would be the landings at high weights and high pressure altitudes and temperatures, which might be encountered if a landing were required shortly after takeoff.

Landing field requirements can more frequently limit an aircraft's allowable landing weight than the approach climb limitations. Again, however, unless the runway is particularly short, this is seldom problematical as the average landing weight at the destination rarely approaches the maximum design landing weight due to fuel burn off.

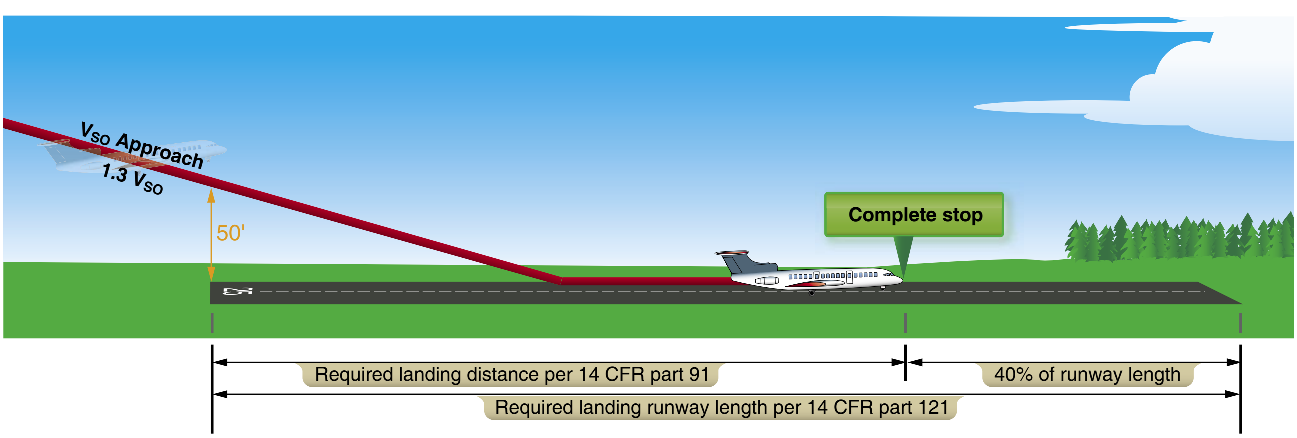


Figure 10-38. *Landing runway requirements.*

Chapter Summary

Performance characteristics and capabilities vary greatly among aircraft. Moreover, aircraft weight, atmospheric conditions, and external environmental factors can significantly affect aircraft performance. It is essential that a pilot become intimately familiar with the performance characteristics and capabilities of the aircraft being flown. The primary source of this information is the AFM/POH.