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COURSE MATERIAL

SCHOOL OF MECHANICAL

DEPARTMENT OF AERONAUTICAL ENGINEERING

SAE1304 - AIRCRAFT STABILITY AND CONTROL

UNIT – I - BASIC CONCEPTS

Degrees of freedom

Degrees of freedom (mechanics), independent displacements and/or rotations that specify the orientation of the body or system

Degrees of freedom (statistics), the number of values in the final calculation of a statistic that is free to vary

Six degrees of freedom

Refers to motion of a rigid body in three-dimensional space, namely the ability to move forward/backward, up/down, left/right combined with rotation about three perpendicular axes (pitch, yaw, roll). As the movement along each of the three axes is independent of each other and independent of the rotation about any of these axes, the motion indeed has six degrees of freedom. Notice that the initial conditions for a rigid body include also the derivatives of these variables (velocity and angular velocity), being therefore a 12-DOF system

Figure 1. Motions of aircraft

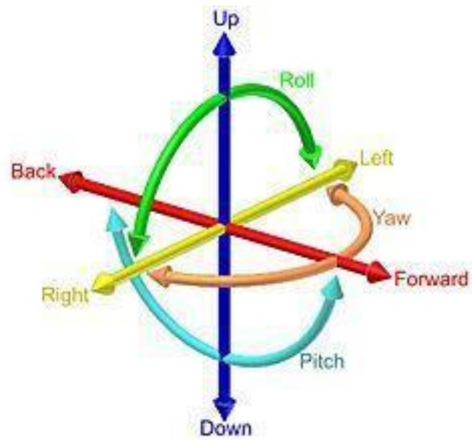


Figure 2. Directional axis

Static stability

As any vehicle moves it will be subjected to minor changes in the forces that act on it, and in its speed.

- If such a change causes further changes that tend to restore the vehicle to its original speed and orientation, without human or machine input, the vehicle is said to be statically stable. The aircraft has positive stability.
- If such a change causes further changes that tend to drive the vehicle away from its original speed and orientation, the vehicle is said to be statically unstable. The aircraft has negative stability.
- If such a change causes no tendency for the vehicle to be restored to its original speed and orientation, and no tendency for the vehicle to be driven away from its original speed and orientation, the vehicle is said to be neutrally stable. The aircraft has zero stability.

For a vehicle to possess positive static stability it is not necessary for its speed and orientation to return to exactly the speed and orientation that existed before the minor change that caused the upset. It is sufficient that the speed and orientation do not continue to diverge but undergo at least a small change back towards the original speed and or

The longitudinal stability of an aircraft refers to the aircraft's stability in the pitching plane - the plane which describes the position of the aircraft's nose in relation to its tail and the horizon. (Other stability modes are directional stability and lateral stability.)

If an aircraft is longitudinally stable, a small increase in angle of attack will cause the pitching moment on the aircraft to change so that the angle of attack decreases. Similarly, a small decrease in angle of attack will cause the pitching moment to change so that the angle of attack increases.

Dynamic Stability

The evaluation of static stability provides some measure of the airplane dynamics, but only a rather crude one. Of greater relevance, especially for lateral motion, is the dynamic response of the aircraft. As seen below, it is possible for an airplane to be statically stable, yet dynamically unstable, resulting in unacceptable characteristics.

Figure 3. Modes of Dynamic stability

Just what constitutes acceptable characteristics is often not obvious, and several attempts have been made to quantify pilot opinion on acceptable handling qualities. Subjective flying qualities evaluations such as Cooper-Harper ratings (The Cooper-Harper rating scale is a set of criteria used by test pilots and flight test engineers to evaluate the handling qualities of aircraft during flight test. The scale ranges from 1 to 10, with 1 indicating the best handling characteristics and 10 the worst) are used to distinguish between "good-flying" and difficult-to-fly aircraft. New aircraft designs can be simulated to determine whether they are acceptable.

Such real-time, pilot-in-the-loop simulations are expensive and require a great deal of information about the aircraft. Earlier in the design process, flying qualities estimate may be made on the basis of various dynamic characteristics. One can correlate pilot ratings to the frequencies and damping ratios of certain types of motion

Flight dynamics

Flight dynamics is the study of dynamics of flight through the air, or beyond planetary bodies' atmospheres. It is chiefly concerned with vehicle attitude, angles and rates of change of angles of the vehicle as well as speed and changes of speed with respect to time.

In another word it is the science of air vehicle orientation and control in three dimensions. The three critical flight dynamics parameters are the angles of rotation in three dimensions about the vehicle's center of mass, known as pitch, roll and yaw.

Aircraft engineers develop control systems for a vehicle's orientation (attitude) about its center of mass. The control systems include actuators, which exert forces in various directions, and generate rotational forces or moments about the center of gravity of the aircraft, and thus rotate the aircraft in pitch, roll, or yaw. For example, a pitching moment is a vertical force applied at a distance forward or aft from the center of gravity of the aircraft, causing the aircraft to pitch up or down.

Roll, pitch and yaw refer, in this context, to rotations about the respective axes starting from a defined equilibrium state. The equilibrium roll angle is known as wings level or zero bank angle, equivalent to a level heeling angle on a ship. Yaw is known as "heading". The equilibrium pitch angle in submarine and airship parlance is known as "trim", but in aircraft, this usually refers to angle of attack, rather than orientation.

A fixed-wing aircraft increases or decreases the lift generated by the wings when it pitches nose up or down by increasing or decreasing the angle of attack (AOA). The roll angle is also known as bank angle on a fixed wing aircraft, which usually "banks" to change the horizontal direction of flight. An aircraft is usually streamlined from nose to tail to reduce drag making it typically advantageous to keep the sideslip angle near zero, though there are instances when an aircraft may be deliberately "sideslipped" for example a slip in a fixed wing aircraft

Flight dynamics (spacecraft)

Spacecraft flight dynamics is the science of space vehicle orientation and control in three dimensions. Three critical flight dynamics parameters are the angles of rotation in three dimensions about the vehicle's center of mass, known as pitch, roll and yaw. For spacecraft, an additional dynamic parameter is translation in space that shifts a vehicle or satellite from one defined orbit to another.

The attitude of a vehicle is its orientation with respect to a defined frame of reference. Satellite Attitude control or flight dynamics refers to the techniques employed to keep the attitude inside a predefined range of values.

Attitude dynamics is the modeling of the changing position and orientation of a vehicle, due to external forces acting on the body. Attitude control is the purposeful manipulation of controllable external forces (using vehicle actuators) to establish a desired attitude, whereas attitude determination is the utilization of vehicle sensors to ascertain the current vehicle attitude.

Inherent stability vs inherent instability

One of the Wright Brothers' breakthroughs (important discovery) was the realization that an inherently stable airplane is difficult to control in that it wants to keep following its original path. Therefore they designed their planes to be inherently (Existing as an essential constituent or characteristic) Unstable. The resulting aircraft were easier to turn but when banked for a turn lift was lost, the aircraft stalled, and at the altitudes they flew at there was no room to recover. This, along with spins and the aircraft breaking up in mid air became a leading cause of death for early flyers.

Rigid Body Dynamics

In physics, rigid body dynamics is the study of the motion of rigid bodies. Unlike particles, which move only in three degrees of freedom (translation in three

directions), rigid bodies occupy space and have geometrical properties, such as a center of mass, moments of inertia, etc., that characterize motion in six degrees of freedom(translation in three directions plus rotation in three directions). Rigid bodies are also characterized as being non-deformable, as opposed to deformable bodies.

As such, rigid body dynamics is used heavily in analyses and computer simulations of physical systems and machinery where rotational motion is important, but material deformation does not have a significant effect on the motion of the system. A rigid body is an idealization of a solid body of finite size in which deformation is neglected. In other words, the distance between any two given points of a rigid body remains constant in time regardless of external forces exerted on it. Even though such an object cannot physically exist due to relativity, objects can normally be assumed to be perfectly rigid if they are not moving near the speed of light.

In classical mechanics a rigid body is usually considered as a continuous mass distribution, while in quantum mechanics a rigid body is usually thought of as a

collection of point masses. For instance, in quantum mechanics molecules (consisting of the point masses: electrons and nuclei) are often seen as rigid bodies

Kinematics

Linear and angular position

The position of a rigid body is the position of all the particles of which it is composed. To simplify the description of this position, we exploit the property that the body is rigid, namely that all its particles maintain the same distance relative to each other. If the body is rigid, it is sufficient to describe the position of at least three non-collinear particles. This makes it possible to reconstruct the position of all the other particles, provided that their time-invariant position relative to the three selected particles is known. However, typically a different, mathematically more convenient, but equivalent approach is used. The position of the whole body is represented by:

1. The linear position or position of the body, namely the position of one of the particles of the body, specifically chosen as a reference point (typically coinciding with the center of mass or centroid of the body), together with

2. The angular position (also known as orientation, or attitude) of the body.

Thus, the position of a rigid body has two components: linear and angular, respectively. The same is true for other kinematic and kinetic quantities describing

the motion of a rigid body, such as linear and angular velocity, acceleration, momentum, impulse, and kinetic energy.

For instance, a basis set with fixed orientation relative to an airplane can be defined as a set of three orthogonal unit vectors b_1, b_2, b_3, such that b_1 is parallel to the chord line of the wing and directed forward, b_2 is normal to the plane of symmetry and directed rightward, and b_3 is given by the cross product $b_3 = b_1 \times b_2$.

In general, when a rigid body moves, both its position and orientation vary with time. In the kinematic sense, these changes are referred to as translation and rotation, respectively. Indeed, the position of a rigid body can be viewed as a hypothetic translation and rotation of the body starting from a hypothetic reference position (not necessarily coinciding with a position actually taken by the body during its motion).

Linear and angular velocity

Velocity (also called linear velocity) and angular velocity are measured with respect to a frame of reference.

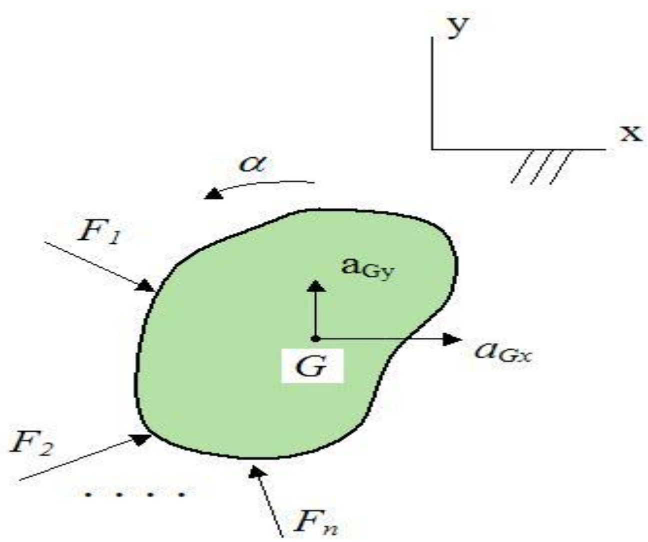
The linear velocity of a rigid body is a vector quantity, equal to the time rate of change of its linear position. Thus, it is the velocity of a reference point fixed to the body. During purely translational motion (motion with no rotation) all points on a rigid body move with the same velocity. However, when motion involves rotation, the instantaneous velocity of any two points on the body will generally not be the same. Two points of a rotating body will have the same instantaneous velocity only if they happen to lay on an axis parallel to the instantaneous axis of rotation.

Angular velocity is a vector quantity that describes the angular speed at which the orientation of the rigid body is changing and the instantaneous axis about which it is rotating (the existence of this instantaneous axis is guaranteed by the Euler's rotation theorem). All points on a rigid body experience the same angular velocity at all times. During purely rotational motion, all points on the body change position except for those lying on the instantaneous axis of rotation. The relationship between orientation and angular velocity is not directly analogous to the relationship between position and velocity. Angular velocity is not the time rate of change of orientation, because there is no such concept as an orientation vector that can be differentiated to obtain the angular velocity.

Two-Dimensional Rigid Body Dynamics

For two-dimensional rigid body dynamics problems, the body experiences motion in one plane, due to forces acting in that plane.

A general rigid body subjected to arbitrary forces in two dimensions is shown below.



The full set of scalar equations describing the motion of the body are:

$$\sum F_x = ma_{Gx}$$

$$\sum F_y = ma_{Gy}$$

$$\sum M_G = I_G \alpha$$

$$\sum M_G \equiv \sum \tau_G$$

Where:

m is the mass of the body

ΣF_x is the sum of the forces in the x-direction

ΣF_y is the sum of the forces in the y-direction

a_{Gx} is the acceleration of the center of mass G in the x-direction, with respect to an inertial reference frame xyz, which is the ground in this case

a_{Gy} is the acceleration of the center of mass G in the y-direction, with respect to ground

α is the angular acceleration of the rigid body with respect to ground

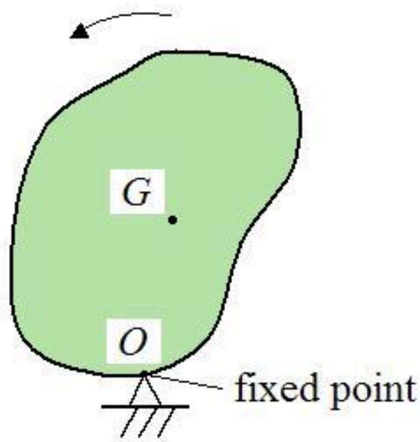
ΣM_G is the sum of the moments about an axis passing through the center of mass G (in the z-direction, pointing out of the page). This is defined as the sum of the torque $\Sigma\tau$ due to the forces acting on the body (about an axis passing through the center of mass G, and pointing in the z-direction). Using ΣM_G is simply a different naming convention.

I_G is the rotational inertia of the rigid body about an axis passing through the center of mass G, and pointing in the z-direction (out of the page)

Note that, if the rigid body were rotating about a fixed point O, the final moment equation would retain the same form if we were to choose point O instead of point G. So, the equation would become:

$$\sum M_o = I_o \alpha$$

The figure below illustrates this situation.

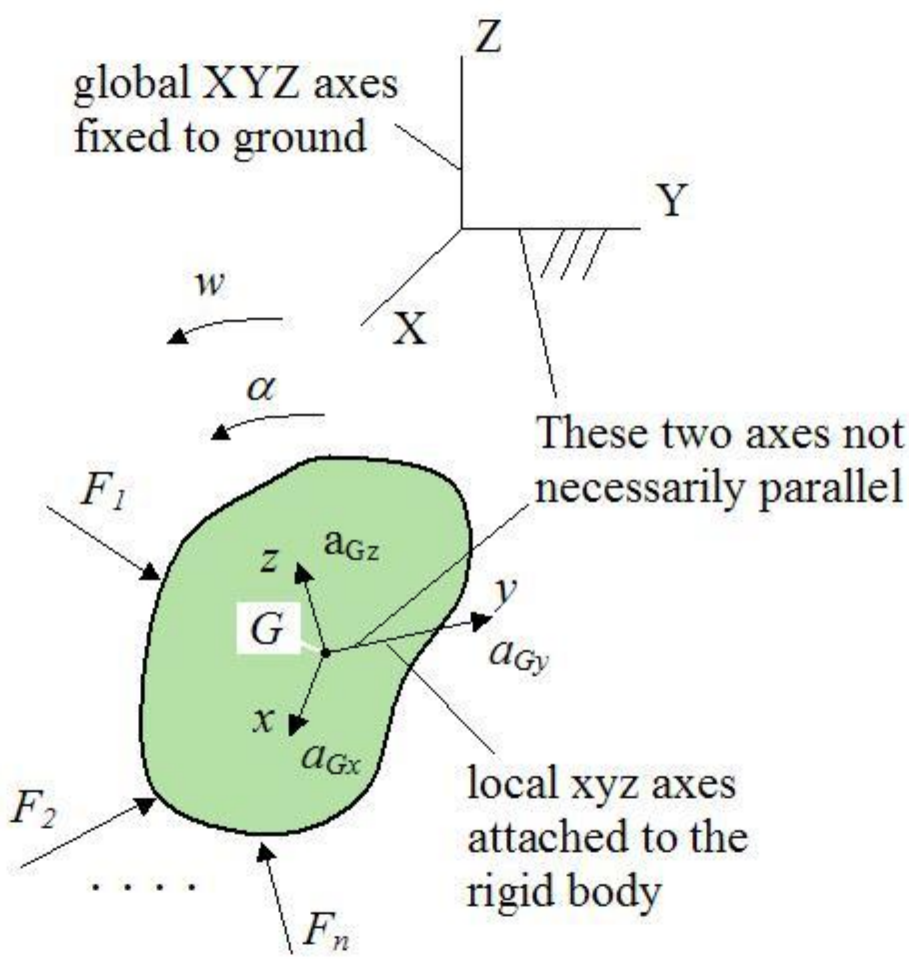


Where the point O is a fixed point attached to ground. A specific example of this would be a pendulum swinging about a fixed point.

Three-Dimensional Rigid Body Dynamics

For three-dimensional rigid body dynamics problems, the body experiences motion in all three dimensions, due to forces acting in all three dimensions. This is the most general case for a rigid body.

A general rigid body subjected to arbitrary forces in three dimensions is shown below.



The first three of the six scalar equations describing the motion of the body are force equations. They are:

$$\sum F_x = ma_{Gx}$$

$$\sum F_y = ma_{Gy}$$

$$\sum F_z = ma_{Gz}$$

Where:

a_{Gx} is the acceleration of the center of mass G in the x-direction, with respect to ground (an inertial reference frame)

a_{Gy} is the acceleration of the center of mass G in the y-direction, with respect to ground

a_{Gz} is the acceleration of the center of mass G in the z-direction, with respect to ground

Note that the subscripts x,y,z indicate that the quantities are resolved along the xyz axes. For example, a force acting along the Z-axis is resolved into its components along the xyz axes in the above three equations. This can generally be done using trigonometry.

However, it is not necessary to resolve the quantities along the xyz axis. For the above three force equations, one can resolve the quantities along the XYZ axes instead.

To solve three-dimensional rigid body dynamics problems it is necessary to calculate six inertia terms for the rigid body, corresponding to the extra complexity of the three dimensional system. To do this, it is necessary to define a local xyz axes which lies within the rigid body and is attached to it (as shown in the figure above), so that it moves with the body. The six inertia terms are

calculated with respect to xyz and depend on the orientation of xyz relative to the rigid body. So, a different orientation of xyz (relative to the rigid body) will result in different inertia terms. The reason that xyz is said to "move with the body" is because the inertia terms will not change with time as the body moves. So you only need to calculate the inertia terms once, at the initial position of the rigid body, and you are done. This has the advantage of keeping the mathematics as simple as possible. An added benefit of having xyz move with the rigid body is when simulating the motion of the body, over time. We can track the orientation of the body by tracking the orientation of xyz (since they move together).

For two-dimensional rigid body dynamics problems there is only one inertia term to consider and it is I_G, as given above. For these problems I_G can be calculated with respect to any orientation of the rigid body, and it will always be the same, since the problem is planar. Therefore, we don't need to define an axes xyz that is attached to the rigid body, and has a certain orientation relative to it (like we do in three-dimensional problems). This is because, for planar problems (where motion is in one plane), I_G would be independent of the orientation of xyz (relative to the rigid body).

For the general case (where we have an arbitrary orientation of xyz within the rigid body), the last three equations describing the motion of the rigid body are moment (torque) equations. They are:

$$\sum M_{Gx} = I_{Gx}\alpha_x - (I_{Gy} - I_{Gz})w_y w_z$$
$$- I_{Gxy}(\alpha_y - w_z w_x) - I_{Gyz}(w_y^2 - w_z^2)$$
$$- I_{Gzx}(\alpha_z + w_x w_y)$$

$$\sum M_{Gy} = I_{Gy}\alpha_y - (I_{Gz} - I_{Gx})w_z w_x$$
$$- I_{Gyz}(\alpha_z - w_x w_y) - I_{Gzx}(w_z^2 - w_x^2)$$
$$- I_{Gxy}(\alpha_x + w_y w_z)$$

$$\sum M_{Gz} = I_{Gz}\alpha_z - (I_{Gx} - I_{Gy})w_x w_y$$
$$- I_{Gzx}(\alpha_x - w_y w_z) - I_{Gxy}(w_x^2 - w_y^2)$$
$$- I_{Gyz}(\alpha_y + w_z w_x)$$

Where:

ΣM_{Gx} is the sum of the moments about the x-axis, passing through the center of mass G

ΣM_{Gy} is the sum of the moments about the y-axis, passing through the center of mass G

ΣM_{Gz} is the sum of the moments about the z-axis, passing through the center of mass G

w_x, w_y, w_z are the components of the angular velocity of the rigid body with respect to ground, and resolved along the local xyz axes. To calculate these components, one must first determine the angular velocity vector of the rigid .

with respect to the global XYZ axes, and then resolve this vector along the x, y, z directions to find the components w_x,w_y, w_z. This is often done using trigonometry.

α_x, α_y, α_z are the components of the angular acceleration of the rigid body with respect to ground, and resolved along the local xyz axes. To calculate these components, one must first determine the angular acceleration vector of the rigid body with respect to the global XYZ axes, and then resolve this vector along the x, y, z directions to find the components α_x, α_y, α_z. This is often done using trigonometry.

I_{Gx} is the rotational inertia of the rigid body about the x-axis, passing through the center of mass G

I_{Gy} is the rotational inertia of the rigid body about the y-axis, passing through the center of mass G

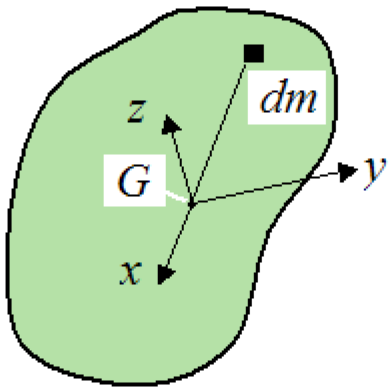
I_{Gz} is the rotational inertia of the rigid body about the z-axis, passing through the center of mass G

I_{Gxy} is the product of inertia (xy) of the rigid body, relative to xyz

I_{Gyz} is the product of inertia (yz) of the rigid body, relative to xyz

I_{Gzx} is the product of inertia (zx) of the rigid body, relative to xyz

The six inertia terms are evaluated as follows, using integration:



$$I_{Gx} = \int (y^2 + z^2) dm$$

$$I_{Gy} = \int (x^2 + z^2) dm$$

$$I_{Gz} = \int (x^2 + y^2) dm$$

$$I_{Gxy} = \int (xy) dm$$

$$I_{Gyz} = \int (yz) dm$$

$$I_{Gzx} = \int (zx) dm$$

The orientation of xyz relative to the rigid body can be chosen such that

$$I_{Gxy} = I_{Gyz} = I_{Gzx} = 0$$

This orientation is defined as the principal direction of xyz.

With this simplification, the moment equations become:

$$\sum M_{Gx} = I_{Gx}\alpha_x - (I_{Gy} - I_{Gz})w_y w_z$$

$$\sum M_{Gy} = I_{Gy}\alpha_y - (I_{Gz} - I_{Gx})w_z w_x$$

$$\sum M_{Gz} = I_{Gz}\alpha_z - (I_{Gx} - I_{Gy})w_x w_y$$

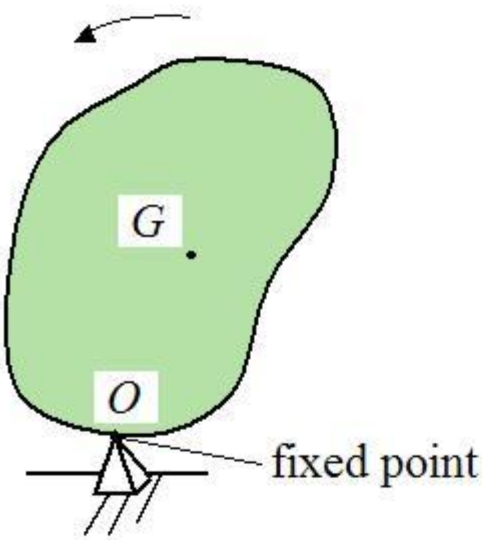
These are known as the Euler equations of motion. Clearly, it is a good idea to choose the orientation of xyz so that it lies in the principal direction. For every rigid body a principal direction exists. If a body has two or three planes of symmetry, the principal directions will be aligned with these planes. For the case where there are no symmetry planes in the body, the principal direction can still be found, but it involves solving a rather complicated cubic equation.

Note that, for the three moment equations and six inertia terms, their quantities must be with respect to the xyz axes (this is unlike the first three force equations, where this is optional). For the inertia terms, the reason for this is obvious since this is how they are defined. But for the moment equations, the reason is rather complicated, but basically it comes down to how they are derived, which is discussed here.

For example, moment acting about the Y-axis must be resolved into its components along the xyz axes in order to use the above moment equations. This can generally be done using trigonometry.

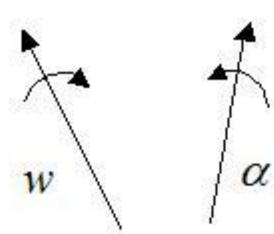
Note that, if the rigid body were rotating about a fixed point O, the above moment equations and six inertia terms would retain the same form if we were to choose point O instead of point G. You just replace the subscript G with the subscript O, and everything else stays the same. Note that the xyz axes would have origin at point O instead of point G.

The figure below illustrates this situation.



Where the point O is a fixed point attached to ground. A specific example of this would be a spinning top precession around a fixed point.

For two-dimensional rigid body dynamics problems the angular acceleration vector is always pointing in the same direction as the angular velocity vector. However, for three-dimensional rigid body dynamics problems these vectors might be pointing in different directions, as shown below.

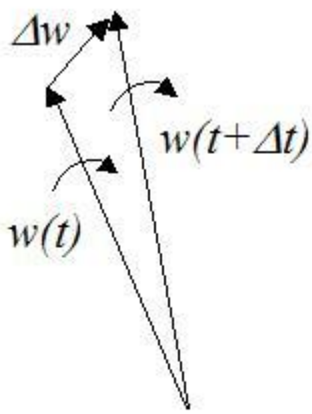


These vectors can be expressed as:

$$\vec{w} = (w_x, w_y, w_z)$$

$$\vec{\alpha} = (\alpha_x, \alpha_y, \alpha_z)$$

In two-dimensions, to find the angular acceleration you simply differentiate the magnitude of the angular velocity, with respect to time. In three-dimensions you have to account for the change in magnitude and direction of the angular velocity vector (since both might change with time), so this does complicate matters a bit. This is done by calculating the difference in the angular velocity vector over a very small time step Δt, where Δt→0. To illustrate, see the figure below.



Using calculus, the angular acceleration is calculated as follows (taking the limit as Δt→0):

$$\vec{\alpha} = \frac{\Delta \vec{w}}{\Delta t} \to \frac{d\vec{w}}{dt}$$

Inertia

Inertia is the resistance of any physical object to a change in its state of motion or rest, or the tendency of an object to resist any change in its motion. It is proportional to an object's mass. The principle of inertia is one of the fundamental principles of classical physics which are used to describe the motion of matter and how it is affected by applied forces. Inertia comes from the Latin word, iners, meaning idle, or lazy. Isaac Newton defined inertia as his first law in his Philosophiæ Naturalis Principia Mathematica, which states

The vis insita, or innate force of matter, is a power of resisting by which everybody, as much as in it lies, Endeavour to preserve its present state, whether it be of rest or of moving uniformly forward in a straight line.

In common usage the term "inertia" may refer to an object's "amount of resistance to change in velocity" (which is quantified by its mass), or sometimes to its momentum, depending on the context. The term "inertia" is more properly understood as shorthand for "the principle of inertia" as described by Newton in his First Law of Motion; that an object not subject to any net external force moves at a constant velocity. Thus an object will continue moving at its current velocity until some force causes its speed or direction to change.

On the surface of the Earth inertia is often masked by the effects of friction and gravity, both of which tend to decrease the speed of moving objects (commonly to the point of rest). This misled classical theorists such as Aristotle, who believed that objects would move only as long as force was applied to them

INERTIA FORCES

- Inertia
  - Tendency for an object at rest to remain at rest, or
  - Tendency of an object in motion to remain in motion.
- Force
  - The energy required to move or accelerate the object.
- Inertia forces
  - Forces that move or accelerate an object
  - They are proportional to the object's weight.

- Seismic forces on buildings are inertia forces and are weight driven.

Stability derivative vs. Control derivative

Stability derivatives and Control derivatives are related because they both are measures of forces and moments on a vehicle as other parameters change. Often the words are used together and abbreviated in the term "S&C derivatives". They differ in that stability derivatives measure the effects of changes in flight conditions while control derivatives measure effects of changes in the control surface positions:

- A stability derivative measures how much change occurs in a force or moment acting on the vehicle when there is a small change in a flight condition parameter such as angle of attack, airspeed, altitude, etc. (Such parameters are called "states".)

- A control derivative measures how much change occurs in a force or moment acting on the vehicle when there is a small change in the deflection of a control surface such as the ailerons, elevator, and rudder.

Names for the axes of vehicles

Air vehicles use a coordinate system of axes to help name important parameters used in the analysis of stability. All the axes run through the center of gravity (called the "CG"):

"X" or "x" axis runs from back to front along the body, called the Roll Axis.

"Y" or "y" axis runs left to right along the wing, called the Pitch Axis.

"Z" or "z" runs from top to bottom, called the Yaw Axis.

Two slightly different alignments of these axes are used depending on the situation: "Body-fixed Axes", and "Stability Axes".

Body-fixed Axes

Body-fixed axes, or "Body Axes", are defined and fixed relative to the body of the vehicle.

X body axis is aligned along the vehicle body and is usually positive toward the normal direction of motion.

Y body axis is at a right angle to the x body axis and is oriented along the wings of the vehicle. If there are no wings (as with a missile), a "horizontal" direction is defined in a way that is useful. The Y body axis is usually taken to be positive to right side of the vehicle.

Z body axis is perpendicular to wing-body (XY) plane and usually points downward.

Stability Axes

Aircraft (usually not missiles) operate at a nominally constant "trim" angle of attack. The angle of the nose (the X Axis) does not align with the direction of the oncoming air. The difference in these directions is the angle of attack. So, for many purposes, parameters are defined in terms of a slightly modified axis system called "stability axes". The stability axis system is used to get the X axis aligned with the oncoming flow direction. Essentially, the body axis system is rotated about the Y body axis by the trim angle of attack and then "re-fixed" to the body of the aircraft

X stability axis is aligned into the direction of the oncoming air in steady flight. (It is projected into the plane made by the X and Z body axes if there is sideslip).

Y stability axis is the same as the Y body-fixed axis.

Z stability axis is perpendicular to the plane made by the X stability axis and the Y body axis.

Forces and velocities along each of the axes

Forces on the vehicle along the body axes are called "Body-axis Forces":

- X, or F_X, is used to indicate forces on the vehicle along the X axis
- Y, or F_Y, is used to indicate forces on the vehicle along the Y axis

- Z, or F_Z, is used to indicate forces on the vehicle along the Z axis

- u (lower case) is used for speed of the oncoming flow along the X body axis
- v (lower case) is used for speed of the oncoming flow along the Y body axis
- w (lower case) is used for speed of the oncoming flow along the Z body axis

  It is helpful to think of these speeds as projections of the relative wind vector on to the three body axes, rather than in terms of the translational motion of the vehicle relative to the fluid. As the body rotates relative to direction of the relative wind, these components change, even when there is no net change in speed.

Moments and angular rates around each of the axes

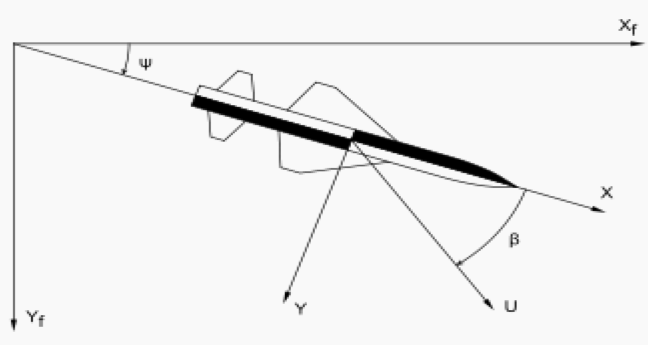
- L is used to indicate the "rolling moment", which is around the X axis. Whether it is around the X body axis or the X stability axis depends on context (such as a subscript).
- M is used to indicate the name of the "pitching moment", which is around the Y axis.

- N is used to indicate the name of the "yawing moment", which is around the Z axis. Whether it is around the Z body axis or the Z stability axis depends on context (such as a subscript).

- "P" or "p" is used for angular rate about the X axis ("Roll rate about the roll axis"). Whether it is around the X body axis or the X stability axis depends on context (such as a subscript).
- "Q" or "q" is used for angular rate about the Y axis ("Pitch rate about the pitch axis").
- "R" or "r" is used for angular rate about the Z axis ("Yaw rate about the yaw axis"). Whether it is around the Z body axis or the Z stability axis depends on context (such as a subscript).

Equations of Motion

The use of stability derivatives is most conveniently demonstrated with missile or rocket configurations, because these exhibit greater symmetry than aero planes, and the equations of motion are correspondingly simpler. If it is assumed that the vehicle is roll-controlled, the pitch and yaw motions may be treated in isolation. It is common practice to consider the yaw plane, so that only 2D motion need be considered. Furthermore, it is assumed that thrust equals drag, and the longitudinal equation of motion may be ignored.



Derivation of Equations of Motion

The body is oriented at angle ψ (psi) with respect to inertial axes. The body is oriented at an angle β (beta) with respect to the velocity vector, so that the components of velocity in body axes are:

$u = U\cos\beta$

$v = U\sin\beta$

where U is the speed.

The aerodynamic forces are generated with respect to body axes, which is not an inertial frame. In order to calculate the motion, the forces must be referred to inertial axes. This requires the body components of velocity to be resolved through the heading angle (β) into inertial axes.

Resolving into fixed (inertial) axes:

$u_f = U\cos(\beta)\cos(\psi) - U\sin(\beta)\sin(\psi) = U\cos(\beta + \psi)$

$v_f = U\sin(\beta)\cos(\psi) + U\cos(\beta)\sin(\psi) = U\sin(\beta + \psi)$

The acceleration with respect to inertial axes is found by differentiating these components of velocity with respect to time:

$$\frac{du_f}{dt} = \frac{dU}{dt}\cos(\beta+\psi) - U\frac{d(\beta+\psi)}{dt}\sin(\beta+\psi)$$
$$\frac{dv_f}{dt} = \frac{dU}{dt}\sin(\beta+\psi) + U\frac{d(\beta+\psi)}{dt}\cos(\beta+\psi)$$

From Newton's Second Law, this is equal to the force acting divided by the mass. Now forces arise from the pressure distribution over the body, and hence are generated in body axes, and not in inertial axes, so the body forces must be resolved to inertial axes, as Newton's Second Law does not apply in its simplest form to an accelerating frame of reference.

Resolving the body forces:

$X_f = X\cos(\psi) - Y\sin(\psi)$

$Y_f = Y\cos(\psi) + X\sin(\psi)$

Newton's Second Law, assuming constant mass:

$$X_f = m\frac{du_f}{dt}$$
$$Y_f = m\frac{dv_f}{dt}$$

Where m is the mass, Equating the inertial values of acceleration and force, and resolving back into body axes, yields the equations of motion:

$$X = m\frac{dU}{dt}\cos(\beta) - mU\frac{d(\beta+\psi)}{dt}\sin(\beta)$$
$$Y = m\frac{dU}{dt}\sin(\beta) + mU\frac{d(\beta+\psi)}{dt}\cos(\beta)$$

The sideslip, β, is a small quantity, so the small perturbation equations of motion become:

$$X = m\frac{dU}{dt}$$
$$Y = mU\frac{d(\beta+\psi)}{dt}$$

The first resembles the usual expression of Newton's Second Law, whilst the second is essentially the centrifugal acceleration. The equation of motion governing the rotation of the body is derived from the time derivative of angular momentum:

$$N = C\frac{d^2\psi}{dt^2}$$

Where C is the moment of inertia about the yaw axis, assuming constant speed, there are only two state variables; β and $\frac{d\psi}{dt}$, which will be written more compactly as the yaw rate r. There is one force and one moment, which for a given flight condition will each be functions of β, r and their time derivatives. For typical missile configurations the forces and moments depend, in the short term, on β and r. The forces may be expressed in the form:

$$Y = Y_0 + \frac{\partial Y}{\partial \beta}\beta + \frac{\partial Y}{\partial r}r$$

Where Y_0 is the force corresponding to the equilibrium condition (usually called the trim) whose stability is being investigated. It is common practice to employ shorthand:

$$\frac{\partial Y}{\partial \beta} = Y_\beta$$

The partial derivative $\frac{\partial Y}{\partial \beta}$ and all similar terms characterizing the increments in forces and moments due to increments in the state variables are called stability derivatives. Typically, $\frac{\partial Y}{\partial r}$ is insignificant for missile configurations, so the equations of motion reduce to:

$$\frac{d\beta}{dt} = \frac{Y_\beta}{mU}\beta - r$$
$$\frac{dr}{dt} = \frac{N_\beta}{C}\beta + \frac{N_r}{C}r$$

Compressibility

In thermodynamics and fluid mechanics, **compressibility** is a measure of the relative volume change of a fluid or solid as a response to a pressure (or mean stress) change.

.

$$\beta = -\frac{1}{V}\frac{\partial V}{\partial p}$$

Where V is volume and p is pressure

Note: most textbooks use the notation κ for this quantity

The above statement is incomplete, because for any object or system the magnitude of the compressibility depends strongly on whether the process is adiabatic or isothermal. Accordingly **isothermal** compressibility is defined:

$$\beta_T = -\frac{1}{V}\left(\frac{\partial V}{\partial p}\right)_T$$

Where the subscript T indicates that the partial differential is to be taken at constant temperature

Adiabatic compressibility is defined:

$$\beta_S = -\frac{1}{V}\left(\frac{\partial V}{\partial p}\right)_S$$

Where S is entropy, for a solid, the distinction between the two is usually negligible.

The inverse of the compressibility is called the bulk modulus, often denoted K (sometimes B). That page also contains some examples for different materials.

The compressibility equation relates the isothermal compressibility (and indirectly the pressure) to the structure of the liquid.

In the present study, a computational investigation was carried out to understand the influence of flexibility on the aerodynamic performance of a hovering wing. A flexible, two-dimensional, two-link model moving within a viscous fluid was considered. The Navier–Stokes equations governing the fluid dynamics were solved together with the equations governing the structural dynamics by using a strongly coupled fluid–structure interaction scheme. Harmonic kinematics was used to prescribe the motions of one of the links, thus effectively reducing the wing to a single degree-of-freedom oscillator. The wing's flexibility was characterized by the ratio of the flapping frequency to the natural frequency of the structure. Apart from the rigid case, different values of this frequency ratio (only in the range of 1/2 to 1/6) were considered at the Reynolds numbers of 75, 250 and 1000. It was found that flexibility can enhance aerodynamic performance and that the best performance is realized when the wing is excited by a non-linear resonance at 1/3 of the natural frequency. Specifically, at Reynolds numbers of 75, 250 and 1000, the aerodynamic performance that is characterized by the ratio of lift coefficient to drag coefficient is respectively increased by 28%, 23% and 21% when compared with the corresponding ratios of a rigid wing driven with the same kinematics. For all Reynolds numbers, the lift generated per unit driving power is also enhanced in a similar manner. The wake capture mechanism is enhanced, due to a stronger flow around the wing at stroke reversal, resulting from a stronger end of stroke vortex at the trailing edge. The present study provides some clues about how flexibility affects the aerodynamic performance in low Reynolds number flapping flight. In addition, it points to the importance of considering non-linear resonances for enhancing aerodynamic performance

UNIT – IV - MANEUVERABILITY

Adverse yaw

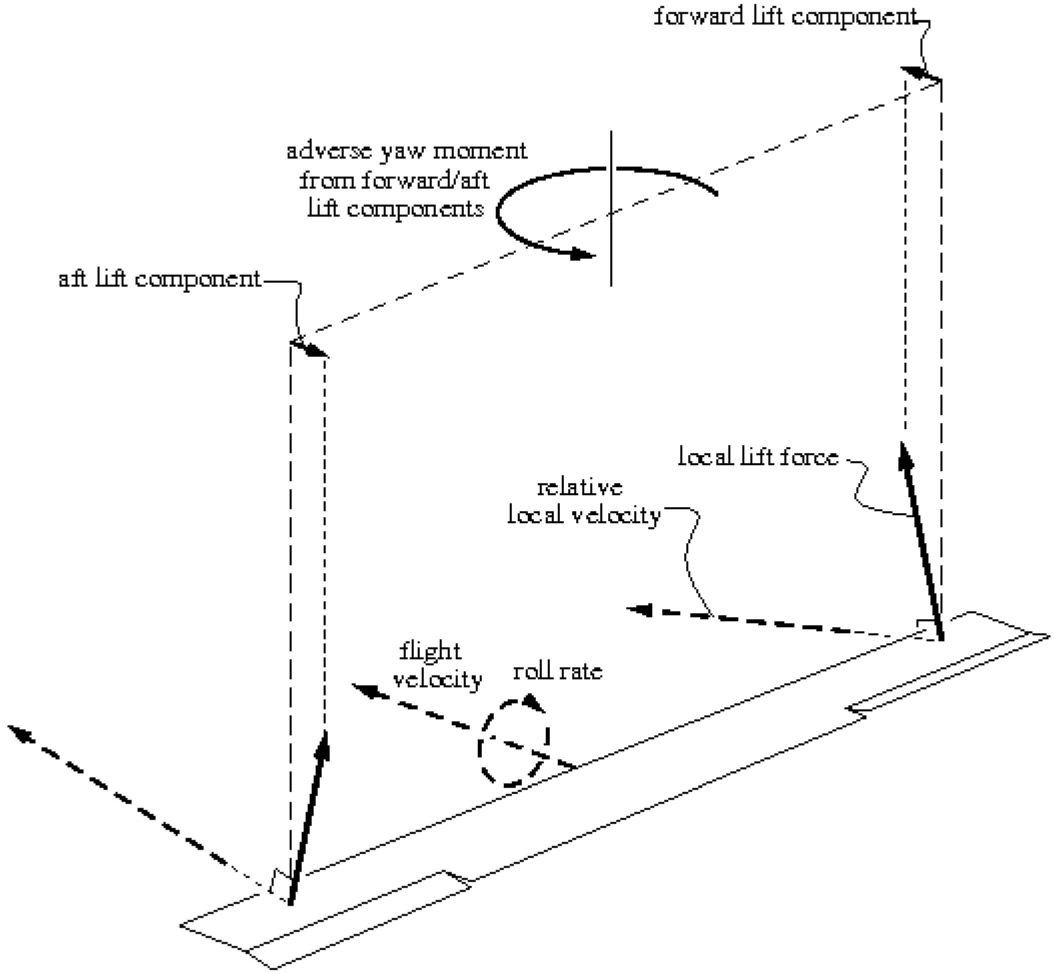
Adverse yaw is a yaw moment on an aircraft which results from an aileron deflection and a roll rate, such as when entering or exiting a turn. It is called "adverse" because it acts opposite to the yaw moment needed to execute the desired turn. Adverse yaw has three mechanisms, listed below in decreasing order of importance. Assuming a roll rate to the right, as in the diagram below, these three mechanisms are explained as follows:

1) By definition, lift is perpendicular to the oncoming flow. Hence, as the left wing moves up, its lift vector tilts back; as the right wing descends, its lift vector tilts forward. The result is an adverse yaw moment to the left, opposite to the intended right turn.

2) The downward aileron deflection on the left increases the airfoil camber, which will typically increase the profile drag. Conversely, the upward aileron deflection on the right will decrease the camber and profile drag. The profile drag imbalance adds to the adverse yaw. The exception is on a Frise aileron, described father below.

3) Initiating the right roll rate requires a briefly greater lift on the left than the right. This also causes a greater induced drag on the left than the right, which further adds to the adverse yaw.

Adverse Yaw Moment due to Roll Rate



Adverse yaw is countered by using the aircraft's rudder to perform a coordinated turn, however an aircraft designer can reduce the amount of correction required by careful design of the aircraft. Some methods are common:

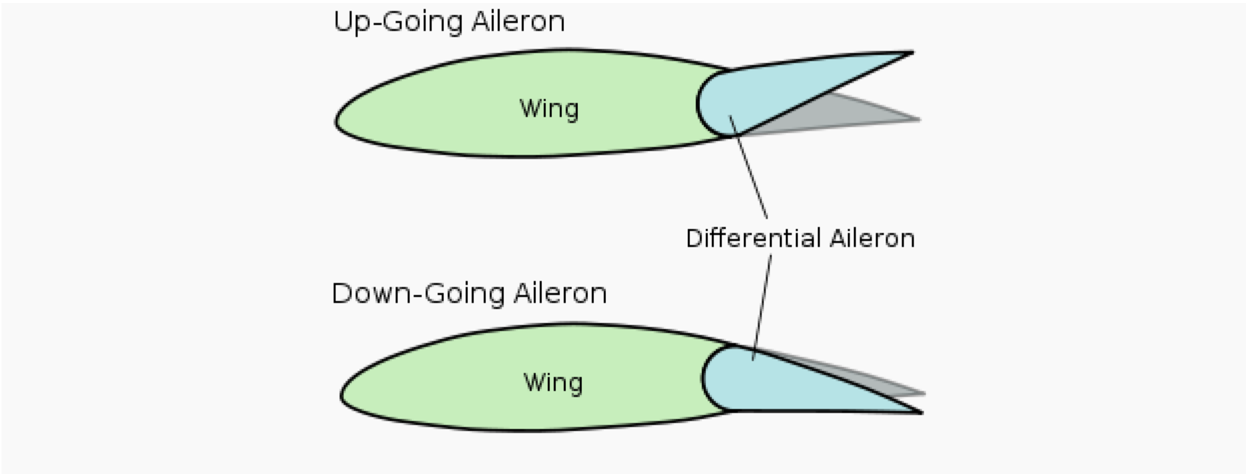
General characteristics

As the tilting of the left/right lift vectors is the major cause to adverse yaw, an important parameter is the magnitude of these lift vectors, or the aircraft's lift coefficient to be more specific. Flight at low lift coefficient (or high speed compared to minimum speed) produces less adverse yaw.

Yaw stability

A strong directional stability is the first way to reduce adverse yaw.[1] That means important vertical tail moment (area and lever arm about gravity center).

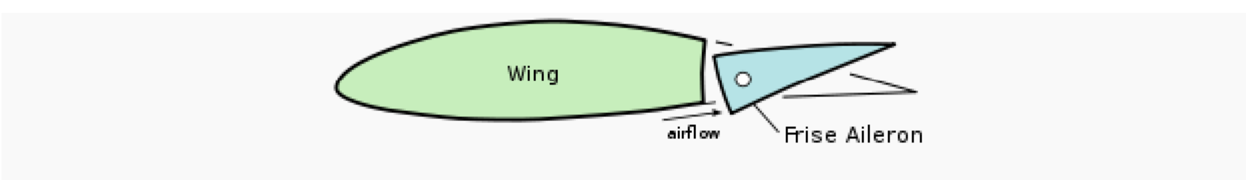
Differential deflection ailerons



Because downwards deflection of an aileron typically causes more profile drag than an upwards deflection, a simple way of mitigating adverse yaw would be to rely solely on the upward deflection of the opposite aileron to cause the aircraft to roll. However, this would lead to a slow roll rate - and therefore a better solution is to make a compromise between adverse yaw and roll rate. This is what occurs in Differential ailerons.

As can be seen from the diagram, the down-going aileron moves through a smaller angle than the up-going aileron, reducing the amount of aileron drag, and thus reducing the effect of adverse yaw. The De Havilland Tiger Moth biplane uses this method of roll control to avoid adverse yaw problems.

Frise ailerons



Frise ailerons are designed so that when up aileron is applied, some of the forward edge of the aileron will protrude downward into the airflow, causing increased drag on this (down-going) wing. This will counter the drag produced by the other aileron, thus reducing adverse yaw.

Unfortunately, as well as reducing adverse yaw, Frise ailerons will increase the overall drag of the aircraft much more than applying rudder correction. Therefore

they are less popular in aircraft where minimizing drag is important (e.g. in a glider). Frise ailerons are primarily designed to reduce roll control forces. Contrary to the illustration, the aileron leading edge has to be rounded to prevent flow separation and flutter at negative deflections. That prevents important differential drag forces.

Roll spoilers

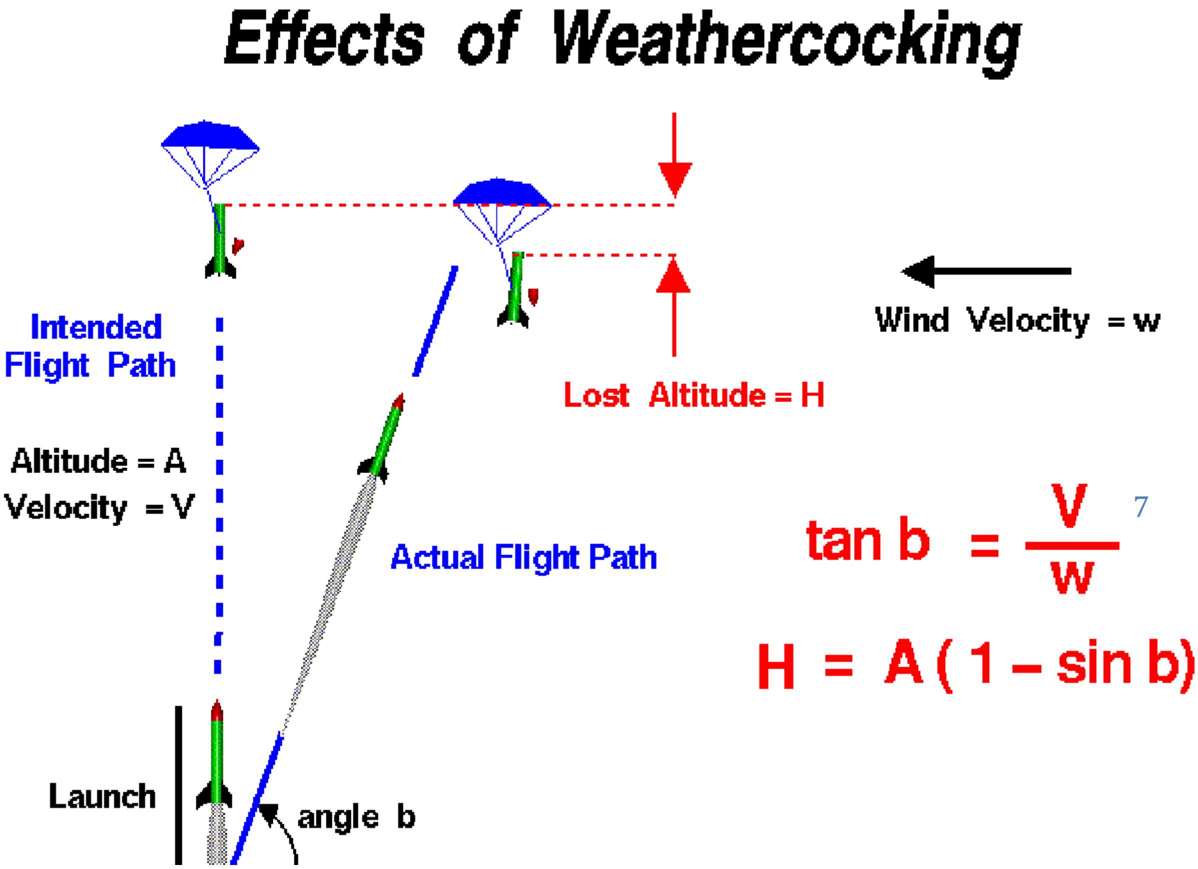
On large aircraft where rudder use is inappropriate at high speeds or ailerons are too small at low speeds, roll spoilers can be used to minimize adverse yaw or increase roll moment. To function as a lateral control, the spoiler is raised on the down-going wing (up aileron) and remains retracted on the other wing. The raised spoiler increases the drag, and so the yaw is in the same direction as the roll.

Control reversal

Control reversal is an adverse effect on the controllability of aircraft. The flight controls reverse themselves in a way that is not intuitive, so pilots may not be aware of the situation and therefore provide the wrong inputs; in order to roll to the left, for instance, they have to push the control stick to the right, the opposite of the normal direction.

Causes

There are several causes for this problem: pilot error, effects of high-speed flight, incorrectly connected controls, and various coupling forces on the aircraft.



[7]

Following the liftoff of a model rocket, it often turns into the wind. This maneuver is called weather cocking and it is caused by aerodynamic forces on the rocket. Wind striking the side of the rocket generates a side force which acts through the center of pressure. For stability reasons, the center of pressure is located below the center of gravity of the rocket. The effect of the side force is to rotate the rocket about the center of gravity until the nose is inclined at the angle b to the horizontal. Angle b is the effective flow direction. If the wind velocity is w and the flight velocity is V, then:

tan b = V / w

where "tan" is the trigonometric tangent function.

The rotation of the rocket produces a new flight path into the wind, as shown at the left of the figure. When the new flight path is aligned with the effective flow direction, there is no longer any lift force and the rocket will continue to fly in the new flight direction. The flight path is inclined to the horizontal at angle b. The chief effect of weather cocking is that the maximum altitude of the flight is reduced. We can estimate the amount of lost altitude H by using some trigonometry. If the maximum, vertical altitude is denoted by A, the lost altitude is given by:

$$H = A * (1 - \sin b)$$

where "sin" is the trigonometric sine function. As a check, if the wind velocity is zero, the angle b is 90 degrees, and the lost altitude is zero.

Directional stability

Directional stability is stability of a moving body or vehicle about an axis which is perpendicular to its direction of motion. Stability of a vehicle concerns itself with the tendency of a vehicle to return to its original direction in relation to the oncoming medium (water, air, road surface, etc.) when disturbed (rotated) away from that original direction. If a vehicle is directionally stable, a restoring moment is produced which is in a direction opposite to the rotational disturbance. This "pushes" the vehicle (in rotation) so as to return it to the original orientation, thus tending to keep the vehicle oriented in the original direction.

Directional stability is frequently called "weather vaning" because a directionally stable vehicle free to rotate about its center of mass is similar to a weather vane rotating about its (vertical) pivot.

With the exception of spacecraft, vehicles generally have a recognizable front and rear and are designed so that the front points more or less in the direction of motion. Without this stability, they may tumble end over end, spin or orient themselves at a high angle of attack, even broadside on to the direction of motion. At high angles of attack, drag forces may become excessive, the vehicle may be impossible to control, or may even experience structural failure. In general, land, sea, air and underwater vehicles are designed to have a natural tendency to point in the direction of motion.

Slip (aerodynamic)

A slip is an aerodynamic state where an aircraft is moving somewhat sideways as well as forward relative to the oncoming airflow. In other words, for a conventional aircraft, the nose will not be pointing directly into the relative wind (in the side-to-side sense).

A slip is also a piloting maneuver where the pilot deliberately puts the aircraft into a slip.

Forward-slip vs. Sideslip

Two forms are employed, the forward-slip and the sideslip. Aerodynamically these are identical once established, but they are entered in different manners and will create different ground tracks and headings relative to those prior to entry. Slips are particularly useful in performing a short field landing over an obstacle (such as trees, or power lines), or to avoid an obstacle (such as a single tree on the extended centerline of the runway), and may be practiced as part of emergency landing procedures. These methods are also commonly employed in flying into farmstead or rough country airstrips where approach hazards are present.

Forward-slip

The forward slip will change the heading of the aircraft away from the down wing, while retaining the original track (flight path over the ground) of the aircraft.

A forward-slip is useful when a pilot has set up for a landing approach with excessive height or must descend steeply beyond a tree line to touchdown near the start of a short runway. Assuming that the runway is properly lined up, the forward slip will allow the aircraft track to be maintained while steepening the descent without adding excessive airspeed. Since the heading is not aligned with the runway, the slip must be removed before touchdown to avoid excessive side loading on the landing gear, and if a cross wind is present an appropriate side slip may be necessary at touchdown as described below.

Sideslip

The sideslip also uses opposite aileron and rudder. In this case it is entered by lowering a wing and exactly enough opposite rudder so the airplane does not turn (maintaining the same heading), while adding airspeed as required.

In the sideslip condition, the airplane's longitudinal axis remains parallel to the original flight path, but the airplane no longer flies straight along its original track. Now, the horizontal component of lift forces the airplane to move sideways toward the low wing.

A sideslip is also one of the methods used by pilots to execute a crosswind landing. In order to land crosswind using the sideslip method, the pilot puts the airplane into a sideslip toward the wind to maintain runway centerline position while maintaining heading on the centerline with the rudder, touching one main landing gear, followed by the second main gear, and finally the nose gear (or tail gear if employed). This allows the wheels to be constantly aligned with the track, thus avoiding any side load at touchdown.

The sideslip method for cross-wind landings is not suitable for long winged and low sitting aircraft such as sailplanes, where instead a crab angle (heading into the wind) is maintained until a moment before touchdown.

Uses of the slip

Other uses

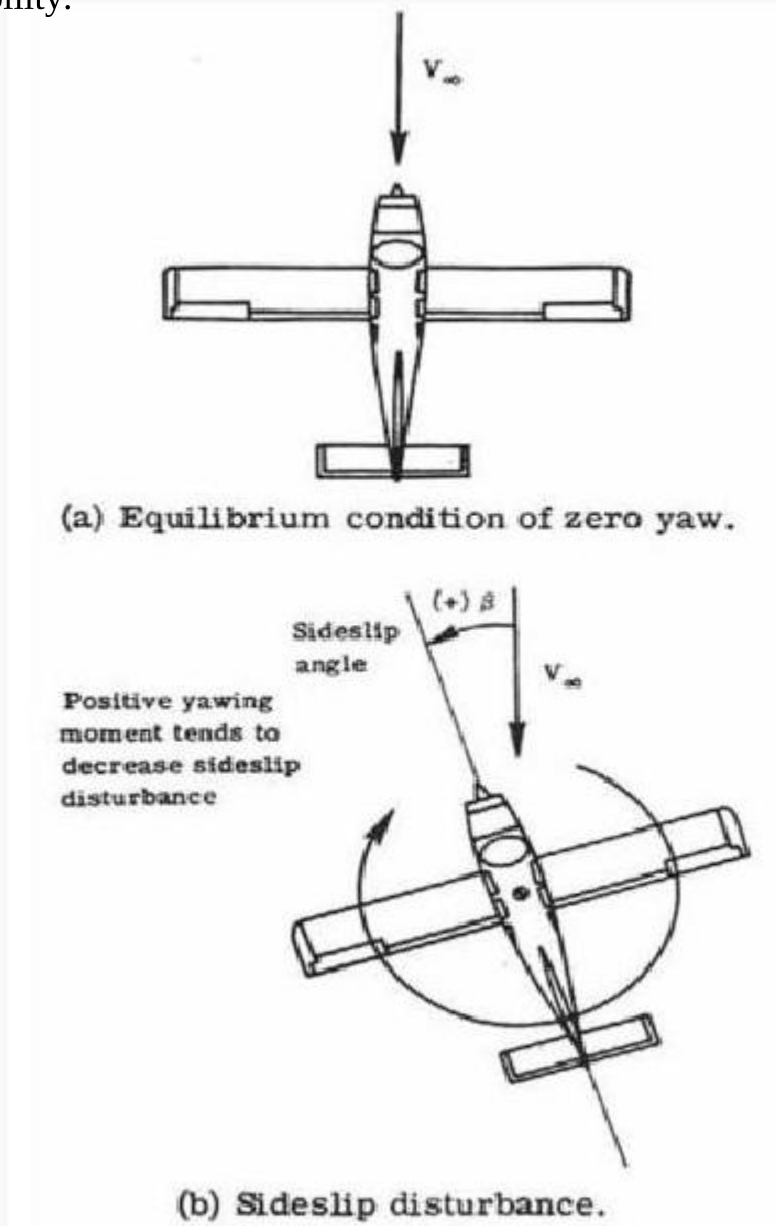
There are other, specialized circumstances where slips can be useful in aviation. For example, during aerial photography, a slip can lower one side of the aircraft to allow ground photos to be taken through a side window. Pilots will also use a slip to land in icing conditions if the front windshield has been entirely iced over — by landing slightly sideways, the pilot is able to see the runway through the aircraft's side window. Slips also play a role in aerobatics and aerial combat.

How a Slip affects flight

When an aircraft is put into a side slip with no other changes to the throttle or elevator, the pilot will notice an increased rate of descent (or reduced rate of ascent). This is usually mostly due to increased drag on the fuselage. The airflow over the fuselage is at a sideways angle, increasing the relative frontal area, which increases drag.

Sideslip angle

In figure (a) an airplane flies so that the sideslip angle is zero, in figure (b) it yaws to have a positive sideslip angle. The circular arrow shows the moment needed for directional stability.



(a) Equilibrium condition of zero yaw.

(b) Sideslip disturbance.

Sideslip angle, also called angle of sideslip (AOS, AoS, β, Greek letter Beta), is a term used in fluid dynamics and aerodynamics and aviation. It relates to the rotation of the aircraft centerline from the relative wind. In flight dynamics it is given the shorthand notation β and is usually assigned to be "positive" when the relative wind is coming from the right of the nose of the airplane. The sideslip angle β is essentially the directional angle of attack of the airplane. It is the primary parameter in directional stability considerations.

UNIT – V - MODERN CONTROL THEORY

Equations of motion

Equations of motion are equations that describe the behavior of a system in terms of its motion as a function of time (e.g., the motion of a particle under the influence of a force). Sometimes the term refers to the differential equations that the system satisfies (e.g., Newton's second law or Euler–Lagrange equations),

Aircraft dynamic modes

The dynamic stability of an aircraft is how the motion of an aircraft behaves after it has been disturbed from steady non-oscillating flight

Longitudinal modes

Oscillating motions can be described by two parameters, the period of time required for one complete oscillation, and the time required to damp to half-amplitude, or the time to double the amplitude for a dynamically unstable motion. The longitudinal motion consists of two distinct oscillations, a long-period oscillation called a phugoid mode and a short-period oscillation referred to as the short-period mode.

Phugoid (longer period) oscillations

The longer period mode, called the "phugoid mode" is the one in which there is a large-amplitude variation of air-speed, pitch angle, and altitude, but almost no angle-of-attack variation. The phugoid oscillation is really a slow interchange of kinetic energy (velocity) and potential energy (height) about some equilibrium energy level as the aircraft attempts to re-establish the equilibrium level-flight condition from which it had been disturbed. The motion is so slow that the effects of inertia forces and damping forces are very low. Although the damping is very weak, the period is so long that the pilot usually corrects for this motion without being aware that the oscillation even exists. Typically the period is 20–60 seconds.

Short period oscillations

With no special name, the shorter period mode is called simply the "short-period mode". The short-period mode is a usually heavily damped oscillation with a period of only a few seconds. The motion is a rapid pitching of the aircraft about the center of gravity. The period is so short that the speed does not have time to change, so the oscillation is essentially an angle-of-attack variation. The time to damp the amplitude to one-half of its value is usually on the order of 1 second. Ability to quickly self damp when the stick is briefly displaced is one of the many criteria for general aircraft certification.

Lateral-directional modes

"Lateral-directional" modes involve rolling motions and yawing motions. Motions in one of these axes almost always couples into the other so the modes are generally discussed as the "Lateral-Directional modes".[2]

There are three types of possible lateral-directional dynamic motion: roll subsidence mode, Dutch roll mode, and spiral mode.

Roll subsidence mode

Roll subsidence mode is simply the damping of rolling motion. There is no direct aerodynamic moment created tending to directly restore wings-level, i.e. there is no returning "spring force/moment" proportional to roll angle. However, there is a damping moment (proportional to roll rate) created by the slewing-about of long wings. This prevents large roll rates from building up when roll-control inputs are made or it damps the roll rate (not the angle) to zero when there are no roll-control inputs.

Roll mode can be improved by adding dihedral effects to the aircraft design, such as high wings, dihedral angles or sweep angles.

Spiral mode

If a spirally unstable aircraft, through the action of a gust or other disturbance, gets a small initial roll angle to the right, for example, a gentle sideslip to the right is produced. The sideslip causes a yawing moment to the right. If the dihedral stability is low, and yaw damping is small, the directional stability keeps turning the aircraft while the continuing bank angle maintains the sideslip and the yaw angle. This spiral gets continuously steeper and tighter until finally, if the motion is not checked, a steep, high-speed spiral dive results. The motion develops so gradually, however that it is usually corrected unconsciously by the pilot, who may not be aware that spiral instability exists. If the pilot cannot see the horizon, for instance because of clouds, he might not notice that he is slowly going into the spiral dive, which can lead into the graveyard spiral.

To be spirally stable, an aircraft must have some combination of a sufficiently large dihedral, which increases roll stability, and a sufficiently long vertical tail arm, which increases yaw damping. Increasing the vertical tail area then magnifies the degree of stability or instability.

The spiral dive should not be confused with a spin.

Detection

While descending turns are commonly performed by pilots as a standard flight maneuvers, the spiral dive is differentiated from a descending turn owing to its feature of accelerating speed. It is therefore an unstable flight condition, and pilots are trained to recognise its onset and to implement recovery procedures safely and immediately. Without intervention by the pilot, acceleration of the aircraft will lead to structural failure of the airframe, either as a result of excess aerodynamic loading or flight into terrain. Spiral dive training therefore revolves around pilot recognition and recovery.

Recovery

Spiral dive accidents are typically associated with visual flight (non-instrument flight) in conditions of poor visibility, where the pilot's reference to the visual natural horizon is effectively reduced, or prevented entirely, by such factors as cloud or darkness. The inherent danger of the spiral dive is that the condition, especially at onset, cannot be easily detected by the sensory mechanisms of the human body. The physical forces exerted on an airplane during a spiral dive are effectively balanced and the pilot cannot detect the banked attitude of the spiral descent. If the pilot detects acceleration, but fails to detect the banked attitude associated with the spiral descent, a mistaken attempt may be to recovery with mere back pressure (pitch-up inputs) on the control wheel. However, with the lift vector of the aircraft now directed to the centre of the spiral turn, this erred nose-up input simply tightens the spiral condition and increases the rate of acceleration and increases dangerous airframe loading. To successfully recover from a spiral dive, the lift vector must first be redirected upward (relative to the natural horizon) before backpressure is applied to the control column. Since the acceleration can be very rapid, recovery is dependent on the pilot's ability to quickly close the throttle (which is contributing to the acceleration), position the lift vector upward, relative to the Earth's surface before the dive recovery is implemented; any factor that would impede the pilot's external reference to the Earth's surface could delay or prevent recovery. The quick and efficient completion of these tasks is crucial as the aircraft can accelerate through maximum speed limits within only a few seconds, where the structural integrity of the airframe will be compromised.

For the purpose of flight training, instructors typically establish the aircraft in a descending turn with initially slow but steadily accelerating airspeed – the initial slow speed facilitates the potentially slow and sometimes erred response of student pilots. The cockpit controls are released by the instructor and the student is instructed to recover. It is not uncommon for a spiral dive to result from an unsuccessful attempt to enter a spin, but the extreme nose-down attitude of the aircraft during the spin-spiral transition makes this method of entry ineffective for training purposes as there is little room to permit student error or delay.

All spiral dive recoveries entail the same recovery sequence: first, the throttle must be immediately closed; second, the aircraft is rolled level with co-ordinate use of ailerons and rudder; and third, backpressure is exerted smoothly on the control wheel to recover from the dive.

Dutch roll

The second lateral motion is an oscillatory combined roll and yaw motion called Dutch roll, perhaps because of its similarity to an ice-skating motion of the same name made by Dutch skaters; the origin of the name is unclear. The Dutch roll may be described as a yaw and roll to the right, followed by a recovery towards the equilibrium condition, then an overshooting of this condition and a yaw and roll to the left, then back past the equilibrium attitude, and so on. The period is usually on the order of 3–15 seconds, but it can vary from a few seconds for light aircraft to a minute or more for airliners. Damping is increased by large directional stability and small dihedral and decreased by small directional stability and large dihedral. Although usually stable in a normal aircraft, the motion may be so slightly damped that the effect is very unpleasant and undesirable. In swept-back wing aircraft, the Dutch roll is solved by installing a yaw damper, in effect a special-purpose automatic pilot that damps out any yawing oscillation by applying rudder corrections. Some swept-wing aircraft have an unstable Dutch roll. If the Dutch roll is very lightly damped or unstable, the yaw damper becomes a safety requirement, rather than a pilot and passenger convenience. Dual yaw dampers are required and a failed yaw damper is cause for limiting flight to low altitudes, and possibly lower mach numbers, where the Dutch roll stability is improved.

Autorotation

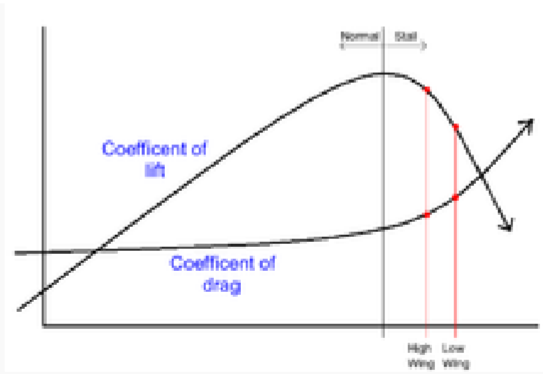
In aviation, **autorotation** refers to processes in both fixed-wing and rotary-wing aircraft. The term means significantly different things in each context.

For fixed-wing aircraft, autorotation refers to the tendency of an aircraft in or near a stall to roll spontaneously to the right or left, leading to a spin (a state of continuous autorotation).

In helicopters and auto gyros, autorotation refers to generation of lift by the main rotor when it is not being driven by an engine. Should an engine fail, a helicopter is able to use autorotation lift to slow its descent and land in a controlled manner.

Auto gyros' main rotors are unpowered, so they rely continuously on autorotation for lift.

Autorotation in fixed-wing aircraft



A typical graph of lift coefficient and drag coefficient versus angle of attack, At any angle of attack greater than the stalling angle an increase in angle of attack causes a reduction in lift coefficient, and a decrease in angle of attack causes an increase in lift coefficient.

When the angle of attack is less than the stalling angle any increase in angle of attack causes an increase in lift coefficient that causes the wing to rise. As the wing rises the angle of attack decreases, which tends to restore the wing to its original angle of attack. Conversely any decrease in angle of attack causes a decrease in lift coefficient which causes the wing to descend. As the wing descends, the angle of attack increases, which tends to restore the wing to its original angle of attack. For this reason the angle of attack is stable when it is less than the stalling angle. The aircraft displays damping in roll.

When the wing is stalled and the angle of attack is greater than the stalling angle any increase in angle of attack causes a decrease in lift coefficient that causes the wing to descend. As the wing descends the angle of attack increases, which causes the lift coefficient to decrease and the angle of attack to increase, Conversely any

decrease in angle of attack causes an increase in lift coefficient that causes the wing to rise. As the wing rises the angle of attack decreases and causes the lift coefficient to increase further towards the maximum lift coefficient. For this reason the angle of attack is unstable when it is greater than the stalling angle. Any disturbance of the angle of attack on one wing will cause the whole wing to roll spontaneously and continuously.

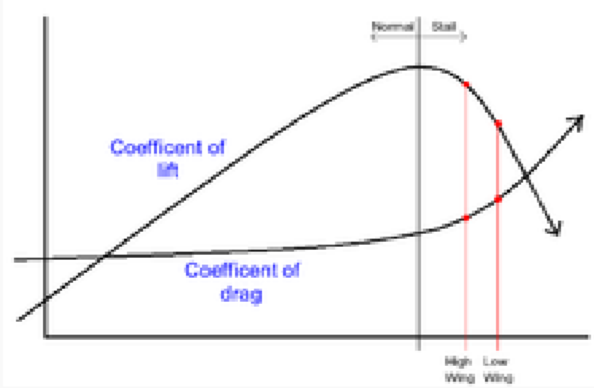
When the angle of attack on the wing of an aircraft reaches the stalling angle the aircraft is at risk of autorotation. This will eventually develop into a spin if the pilot does not take corrective action.

Spin (flight)

In aviation, a spin is an aggravated stall resulting in autorotation about the spin axis wherein the aircraft follows a corkscrew path. Spins can be entered unintentionally or intentionally, from any flight attitude and from practically any airspeed—all that is required is sufficient yaw rate while an aircraft is stalled. In either case, however, a specific and often counterintuitive set of actions may be needed for an effective recovery to be made. If the aircraft exceeds published limitations regarding spins, or is loaded improperly, or if the pilot uses incorrect technique to recover, the spin can lead to a crash. In a spin, one or both wings are in a stalled condition, if both are stalled one wing will be in a deeper stall condition than the other.[1] This causes the aircraft to auto rotate (yaw) towards the deeper-stalled wing due to its higher drag. Spins are also characterized by high angle of attack, low airspeed, and high rate of descent.

Spins differ from spiral dives which are characterized by low angle of attack and high airspeed. A spiral dive is not a type of spin because neither wing is stalled. In a spiral dive the airplane will respond conventionally to the pilot's inputs to the flight controls.

How a spin occ



Certificated, light, single-engine airplanes must meet specific criteria regarding stall and spin behavior. Many types of airplane will only spin if the pilot simultaneously yaws and stalls the airplane (intentionally or unintentionally). Under these circumstances, one wing tends to stall more deeply than the other. The wing that stalls first will drop, increasing its angle of attack and deepening the stall. Both wings must be stalled for a spin to occur. The other wing will rise, decreasing its angle of attack, and the aircraft will yaw towards the more deeply-stalled wing. The difference in lift between the two wings causes the aircraft to roll, and the difference in drag causes the aircraft to yaw.

One common scenario that can lead to an unintentional spin is an uncoordinated turn towards the runway during the landing sequence. A pilot who is overshooting the turn to final approach may be tempted to apply rudder to increase the rate of turn. The result is twofold: the nose of the airplane drops below the horizon and the bank angle increases. Reacting to these unintended changes, the pilot may then begin to pull the elevator control aft (thus increasing the angle of attack) while applying opposite aileron to decrease bank angle. Taken to its extreme, this can result in an uncoordinated turn with sufficient angle of attack to cause the aircraft to stall. This is called across-control stall, and is very dangerous if it happens at low altitude where the pilot has little time to recover.

Autopilot

An autopilot is a system used to control the trajectory of a vehicle without constant 'hands-on' control by a human operator being required. Autopilots do not replace a human operator, but assist them in controlling the vehicle, allowing them to focus on broader aspects of operation, such as monitoring the trajectory, weather and systems.Autopilots are used in aircraft, boats (known as self-steering gear), spacecraft, missiles, and others. Autopilots have evolved significantly over time, from early autopilots that merely held an attitude to modern autopilots capable of performing automated landings under the supervision of a pilot.

The autopilot system on airplanes is sometimes colloquially referred to as "George"

First autopilots

Gyroscopic autopilot

In the early days of aviation, aircraft required the continuous attention of a pilot in order to fly safely. As aircraft range increased allowing flights of many hours, the constant attention led to serious fatigue. An autopilot is designed to perform some of the tasks of the pilot.

The first aircraft autopilot was developed by Sperry Corporation in 1912. The autopilot connected a gyroscopic heading indicator and attitude indicator to hydraulically operated elevators and rudder. (Ailerons were not connected as wing dihedral was counted upon to produce the necessary roll stability.) It permitted the aircraft to fly straight and level on a compass course without a pilot's attention, greatly reducing the pilot's workload. Lawrence Sperry (the son of famous inventor Elmer Sperry) demonstrated it in 1914 at an aviation safety contest held in Paris. At the contest, Sperry demonstrated the credibility of the invention by flying the aircraft with his hands away from the controls and visible to onlookers of the contest. Elmer Sperry Jr., the son of Lawrence Sperry, and Capt Shiras continued work after the war on the same autopilot, and in 1930 they tested a more compact and reliable autopilot which kept a US Army Air Corps aircraft on a true heading and altitude for three hours.

In 1930, the Royal Aircraft Establishment in England developed an autopilot called a pilots' assister that used a pneumatically-spun gyroscope to move the flight controls.

Further development of the autopilot was performed, such as improved control algorithms and hydraulic servomechanisms. Also, inclusion of additional instrumentation such as the radio-navigation aids made it possible to fly during night and in bad weather. In 1947 a US Air Force C-54 made a transatlantic flight, including takeoff and landing, completely under the control of an autopilot.

In the early 1920s, the Standard Oil tanker J.A. Moffet became the first ship to use an autopilot.

Modern autopilots

The modern flight control unit of an Airbus A340

Not all of the passenger aircraft flying today have an autopilot system. Older and smaller general aviation aircraft especially are still hand-flown, and even small airliners with fewer than twenty seats may also be without an autopilot as they are used on short-duration flights with two pilots. The installation of autopilots in aircraft with more than twenty seats is generally made mandatory by international aviation regulations. There are three levels of control in autopilots for smaller aircraft. A single-axis autopilot controls an aircraft in the roll axis only; such autopilots are also known colloquially as "wing levellers," reflecting their limitations. A two-axis autopilot controls an aircraft in the pitch axis as well as roll, and may be little more than a "wing leveller" with limited pitch oscillation-correcting ability; or it may receive inputs from on-board radio navigation systems to provide true automatic flight guidance once the aircraft has taken off until shortly before landing; or its capabilities may lie somewhere between these two extremes. A three-axis autopilot adds control in the yaw axis and is not required in many small aircraft.

Autopilots in modern complex aircraft are three-axis and generally divide a flight into taxi, takeoff, climb, cruise (level flight), descent, approach, and landing phases. Autopilots exist that automate all of these flight phases except taxi and takeoff. An autopilot-controlled landing on a runway and controlling the aircraft on rollout (i.e. keeping it on the centre of the runway) is known as a CAT IIIb landing or Autoland, available on many major airports' runways today, especially at airports subject to adverse weather phenomena such as fog. Landing, rollout, and taxi control to the aircraft parking position is known as CAT IIIc. This is not used to date, but may be used in the future. An autopilot is often an integral component of a Flight Management System.

Modern autopilots use computer software to control the aircraft. The software reads the aircraft's current position, and then controls a Flight Control System to guide the aircraft. In such a system, besides classic flight controls, many autopilots incorporate thrust control capabilities that can control throttles to optimize the

airspeed, and move fuel to different tanks to balance the aircraft in an optimal attitude in the air. Although autopilots handle new or dangerous situations inflexibly, they generally fly an aircraft with lower fuel consumption than a human pilot.

The autopilot in a modern large aircraft typically reads its position and the aircraft's attitude from an inertial guidance system. Inertial guidance systems accumulate errors over time. They will incorporate error reduction systems such as the carousel system that rotates once a minute so that any errors are dissipated in different directions and have an overall nulling effect. Error in gyroscopes is known as drift. This is due to physical properties within the system, be it mechanical or laser guided, that corrupt positional data. The disagreements between the two are resolved with digital signal processing, most often a six-dimensional Kalman filter. The six dimensions are usually roll, pitch, yaw, altitude, latitude, and longitude. Aircraft may fly routes that have a required performance factor, therefore the amount of error or actual performance factor must be monitored in order to fly those particular routes. The longer the flight, the more error accumulates within the system. Radio aids such as DME, DME updates, and GPS may be used to correct the aircraft position.

A midway between fully automated flight and manual flying is Control Wheel Steering (CWS). Although going out of fashion in modern airliners as a stand-alone option, CWS is still a function on many aircraft today. Generally, an autopilot that is CWS equipped, has three positions being off, CWS and CMD. In CMD (Command) mode the autopilot has full control of the aircraft, and receives its input from either the heading /altitude setting, radio and navaids or the FMS (Flight Management System). In CWS mode, the pilot controls the autopilot through inputs on the yoke or the stick. These inputs are translated to a specific heading and attitude, which the autopilot will then hold until instructed to do otherwise. This provides stability in pitch and roll. Some aircraft employ a form of CWS even in manual mode, such as the MD-11 which uses a constant CWS in roll. In many ways, a modern Airbus fly-by-wire aircraft in Normal Law is always in CWS mode. The major difference is that in this system the limitations of the aircraft are guarded by the Flight Computer, and the pilot can not steer the aircraft past these limits.

Computer system details

The hardware of an autopilot varies from implementation to implementation, but is generally designed with redundancy and reliability as foremost considerations. For example, the Rockwell Collins AFDS-770 Autopilot Flight Director System used on the Boeing 777 uses triplicated FCP-2002 microprocessors which have been formally verified and are fabricated in a radiation resistant process.

Software and hardware in an autopilot is tightly controlled, and extensive test procedures are put in place.

Some autopilots also use design diversity. In this safety feature, critical software processes will not only run on separate computers and possibly even using different architectures, but each computer will run software created by different engineering teams, often being programmed in different programming languages. It is generally considered unlikely that different engineering teams will make the same mistakes. As the software becomes more expensive and complex, design diversity is becoming less common because fewer engineering companies can afford it. The flight control computers on the Space Shuttle used this design: there were five computers, four of which redundantly ran identical software, and a fifth backup running software that was developed independently. The software on the fifth system provided only the basic functions needed to fly the Shuttle, further reducing any possible commonality with the software running on the four primary systems.

Stability augmentation systems

A stability augmentation system (SAS) is another type of automatic flight control system; however, instead of maintaining the aircraft on a predetermined attitude or flight path, the SAS will actuate the aircraft flight controls to dampen out aircraft buffeting regardless of the attitude or flight path. SAS can automatically stabilize the aircraft in one or more axes. The most common type of SAS is the yaw damper which is used to eliminate the Dutch roll tendency of swept-wing aircraft. Some yaw dampers are integral to the autopilot system while others are stand-alone systems.

Yaw dampers usually consist of a yaw rate sensor (either a gyroscope or angular accelerometer), a computer/amplifier and a servo actuator. The yaw damper uses yaw rate sensor to sense when the aircraft begins a Dutch Roll. A computer processes the signals from the yaw rate sensor to determine the amount of rudder movement that is required to dampen out the Dutch roll. The computer then commands the servo actuator to move the rudder that amount. The Dutch roll is dampened out and the aircraft becomes stable about the yaw axis. Because Dutch roll is an instability that is inherent to all swept-wing aircraft, most swept-wing aircraft have some sort of yaw damper system installed.

There are two types of yaw dampers: series yaw dampers and parallel yaw dampers. The servo actuator of a series yaw damper will actuate the rudder independently of the rudder pedals while the servo actuator of a parallel yaw damper is clutched to the rudder control quadrant and will result in pedal movement when the system commands the rudder to move.

Some aircraft have stability augmentation systems that will stabilize the aircraft in more than a single axis. B-52s, for example, require both pitch and yaw SAS in order to provide a stable bombing platform. Many helicopters have pitch, roll and yaw SAS systems. Pitch and roll SAS systems operate much the same way as the yaw damper described above; however, instead of dampening out Dutch roll, they will dampen pitch and roll oscillations or buffeting to improve the overall stability of the aircraft.

Autopilot for ILS landings

Instrument-aided landings are defined in categories by the International Civil Aviation Organization, or ICAO. These are dependent upon the required visibility level and the degree to which the landing can be conducted automatically without input by the pilot.

CAT I - This category permits pilots to land with a decision height of 200 ft (61 m) and a forward visibility or Runway Visual Range (RVR) of 550 m. Autopilots are not required.

CAT II - This category permits pilots to land with a decision height between 200 ft and 100 ft (≈ 30 m) and a RVR of 300 m. Autopilots have a fail passive requirement.

CAT IIIa -This category permits pilots to land with a decision height as low as 50 ft (15 m) and a RVR of 200 m. It needs a fail-passive autopilot. There must be only a 10−6 probability of landing outside the prescribed area.

CAT IIIb - As IIIa but with the addition of automatic roll out after touchdown incorporated with the pilot taking control some distance along the runway. This category permits pilots to land with a decision height less than 50 feet or no decision height and a forward visibility of 250 ft (76 m, compare this to aircraft size, some of which are now over 70 m long) or 300 ft (91 m) in the United States. For a landing-without-decision aid, a fail-operational autopilot is needed. For this category some form of runway guidance system is needed: at least fail-passive but it needs to be fail-operational for landing without decision height or for RVR below 100 m.

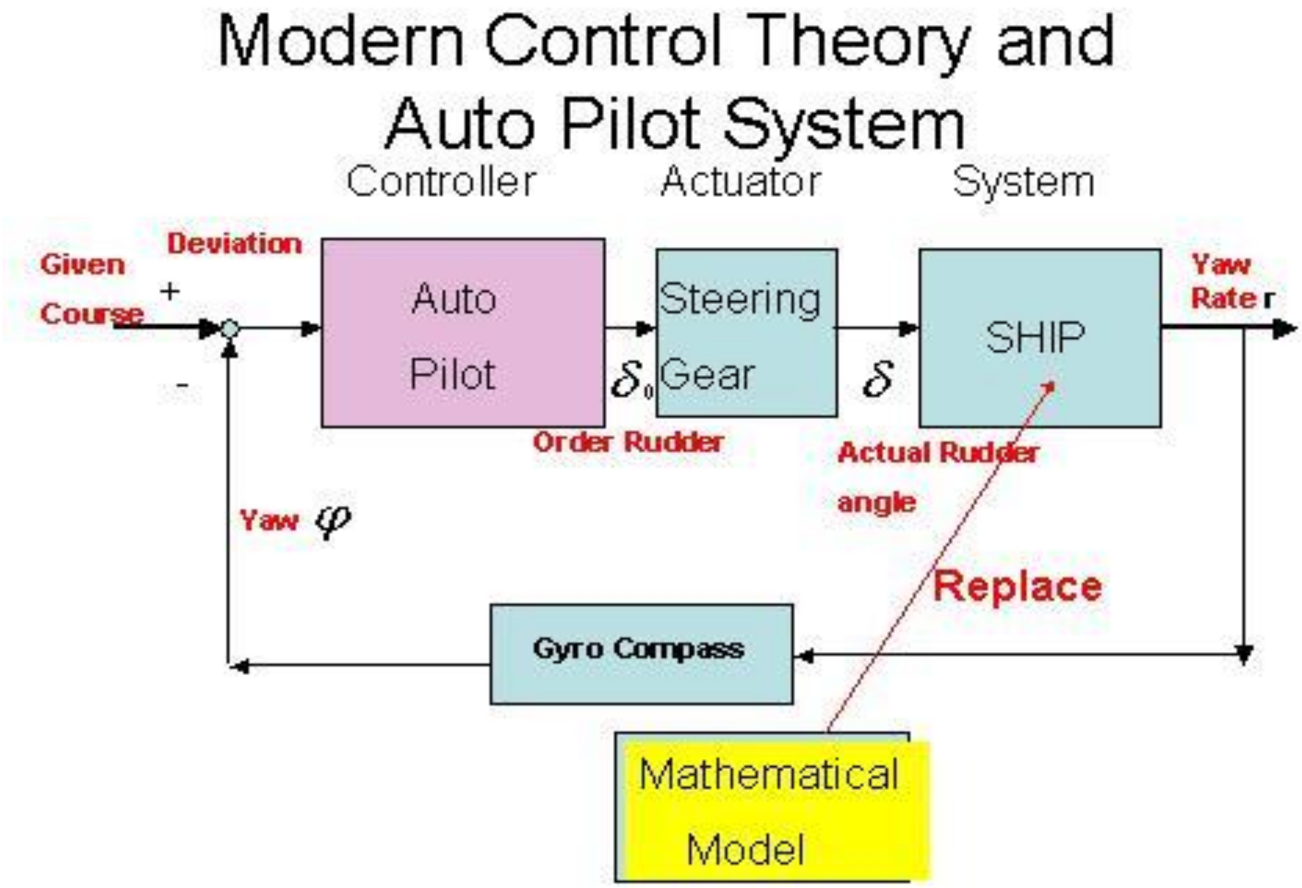
CAT IIIc - As IIIb but without decision height or visibility minimums, also known as "zero-zero"

Fail-passive autopilot: in case of failure, the aircraft stays in a controllable position and the pilot can take control of it to go around or finish landing. It is usually a dual-channel system.

Fail-operational autopilot: in case of a failure below alert height, the approach, flare and landing can still be completed automatically. It is usually a triple-channel system or dual-dual system.

Radio-controlled models

In radio-controlled modelling, and especially RC aircraft and helicopters, an autopilot is usually a set of extra hardware and software that deals with pre-programming the model's flight.



UNIT – II - LONGITUDINAL DYNAMIC STABILITY AND CONTROL

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Static stability

As any vehicle moves it will be subjected to minor changes in the forces that act on it, and in its speed.

- If such a change causes further changes that tend to restore the vehicle to its original speed and orientation, without human or machine input, the vehicle is said to be statically stable. The aircraft has positive stability.
- If such a change causes further changes that tend to drive the vehicle away from its original speed and orientation, the vehicle is said to be statically unstable. The aircraft has negative stability.
- If such a change causes no tendency for the vehicle to be restored to its original speed and orientation, and no tendency for the vehicle to be driven away from its original speed and orientation, the vehicle is said to be neutrally stable. The aircraft has zero stability.

For a vehicle to possess positive static stability it is not necessary for its speed and orientation to return to exactly the speed and orientation that existed before the minor change that caused the upset. It is sufficient that the speed and orientation do not continue to diverge but undergo at least a small change back towards the original speed and orientation

Longitudinal stability

The longitudinal stability of an aircraft refers to the aircraft's stability in the pitching plane - the plane which describes the position of the aircraft's nose in relation to its tail and the horizon. (Other stability modes are directional stability and lateral stability.

If an aircraft is longitudinally stable, a small increase in angle of attack will cause the pitching moment on the aircraft to change so that the angle of attack decreases. Similarly, a small decrease in angle of attack will cause the pitching moment to change so that the angle of attack increases.

The pilot's task

The pilot of an aircraft with positive longitudinal stability, whether it is a human pilot or an autopilot, has an easy task to fly the aircraft and maintain the desired pitch attitude which, in turn, makes it easy to control the speed, angle of attack and fuselage angle relative to the horizon. The pilot of an aircraft with negative longitudinal stability has a more difficult task to fly the aircraft. It will be necessary for the pilot devote more effort, make more frequent inputs to the elevator control, and make larger inputs, in an attempt to maintain the desired pitch attitude.

Most successful aircraft have positive longitudinal stability, providing the aircraft's center of gravity lies within the approved range. Some acrobatic and combat aircraft have low-positive or neutral stability to provide high maneuverability. Some advanced aircraft have a form of low-negative stability called relaxed stability to provide extra-high maneuverability.

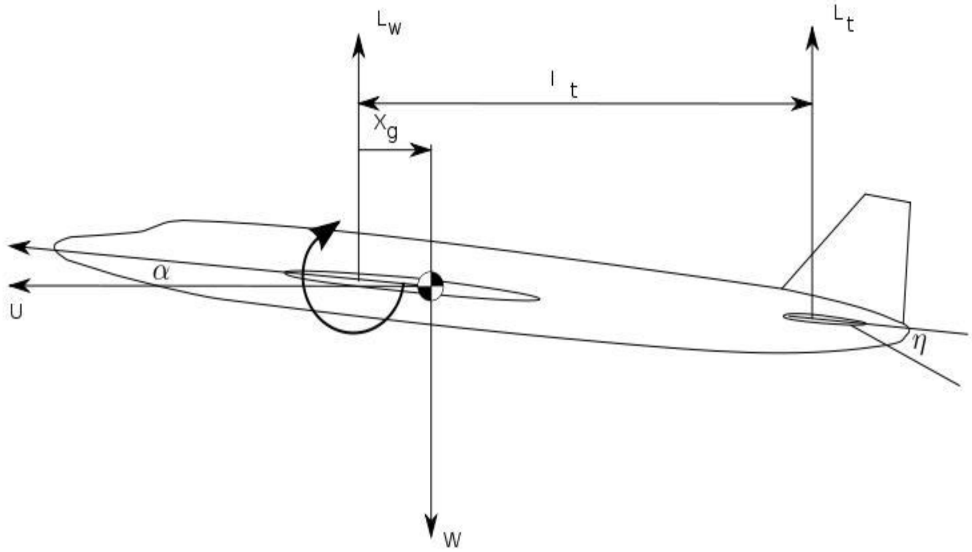
Center of gravity

The longitudinal static stability of an aircraft is significantly influenced by the position of the center of gravity of the aircraft. As the center of gravity moves forward the moment arm between the horizontal stabilizer increases and the longitudinal static stability of the aircraft also increases. As the center of gravity moves aft, the longitudinal static stability of the aircraft decreases.

The limitations specified for an aircraft type and model include limitations on the most forward position, and the most aft position, permitted for the center of gravity. No attempt should be made to fly an aircraft if its center of gravity is outside the approved range, or will move outside the approved range during the flight.

Analysis

Near the cruise condition most of the lift force is generated by the wings, with ideally only a small amount generated by the fuselage and tail. We may analyze the longitudinal static stability by considering the aircraft in equilibrium under wing lift, tail force, and weight. The moment equilibrium condition is called trim, and we are generally interested in the longitudinal stability of the aircraft about this trim cond



Aircraft in Level Flight

Equating forces in the vertical direction:

$W = L_w + L_t$

where W is the weight, L_w is the wing lift and L_t is the tail force.

For a symmetrical airfoil at low angle of attack, the wing lift is proportional to the angle of attack:

$$L_w = qS_w \frac{\partial C_L}{\partial \alpha}(\alpha - \alpha_0)$$

where S_w is the wing area C_L is the (wing) lift coefficient, α is the angle of attack. The term α_0 is included to account for camber, which results in lift at zero angle of attack. Finally q is the dynamic pressure:

$$q = \frac{1}{2}\rho v^2$$

Where ρ is the air density and v is the speed.

Trim

The tail plane is usually a symmetrical airfoil, so its force is proportional to angle of attack, but in general, there will also be an elevator deflection to maintain moment equilibrium (trim). In addition, the tail is located in

For a statically stable aircraft of conventional (tail in rear) configuration, the tail plane force typically acts downward. In canard aircraft, both fore and aft planes are lifting surfaces. The fundamental requirement for static stability is that the coefficient of lift of the fore surface be greater than that of the aft surface; but even this general statement obviously does not apply to tailless aircraft. Violations of this basic principle are exploited in some high performance combat aircraft to enhance agility; artificial stability is supplied by electronic means.

The tail force is, therefore:

$$L_t = qS_t\left(\frac{\partial C_l}{\partial \alpha}\left(\alpha - \frac{\partial \epsilon}{\partial \alpha}\alpha\right) + \frac{\partial C_l}{\partial \eta}\eta\right)$$

Where S_t is the tail area, C_l is the tail force coefficient, η is the elevator deflection, and ϵ is the downwash angle.

Note that for a rear-tail configuration, the aerodynamic loading of the horizontal stabilizer (in $N \cdot m^{-2}$) is less than that of the main wing, so the main wing should stall before the tail, ensuring that the stall is followed immediately by a reduction in angle of attack on the main wing, promoting recovery from the stall. (In contrast, in a canard configuration, the loading of the horizontal stabilizer is greater than that of the main wing, so that the horizontal stabilizer stalls before the main wing, again promoting recovery from the stall.). There are a few classical cases where this favorable response was not achieved, notably some early T-tail jet aircraft. In the event of a very high angle of attack, the horizontal stabilizer became immersed in the downwash from the fuselage, causing excessive download on the stabilizer, increasing the angle of attack still further. The only way an aircraft could recover from this situation was by jettisoning tail ballast or deploying a special tail parachute. The phenomenon became known as 'deep stall'.

Taking moments about the center of gravity, the net nose-up moment is:

$$M = L_w x_g - (l_t - x_g)L_t$$

the angle of attack at the tail plane.

where x_g is the location of the center of gravity behind the aerodynamic center of the main wing, l_t is the tail moment arm. For trim, this moment must be zero. For a given maximum elevator deflection, there is a corresponding limit on center of gravity position at which the aircraft can be kept in equilibrium. When limited by control deflection this is known as a 'trim limit'. In principle trim limits could determine the permissible forwards and rearwards shift of the centre of gravity, but usually it is only the forward cg limit which is determined by the available control, the aft limit is usually dictated by stability.

In a missile context 'trim limit' more usually refers to the maximum angle of attack, and hence lateral acceleration which can be generated.

Static stability

The nature of stability may be examined by considering the increment in pitching moment with change in angle of attack at the trim condition. If this is nose up, the aircraft is longitudinally unstable; if nose down it is stable. Differentiating the moment equation with respect to α:

$$\frac{\partial M}{\partial \alpha} = x_g \frac{\partial L_w}{\partial \alpha} - (l_t - x_g)\frac{\partial L_t}{\partial \alpha}$$

Note: $\frac{\partial M}{\partial \alpha}$ is a stability derivative.

It is convenient to treat total lift as acting at a distance h ahead of the centre of gravity, so that the moment equation may be written:

$$M = h(L_w + L_t)$$

Applying the increment in angle of attack:

$$\frac{\partial M}{\partial \alpha} = h\left(\frac{\partial L_w}{\partial \alpha} + \frac{\partial L_t}{\partial \alpha}\right)$$

Equating the two expressions for moment increment:

$$h = x_g - l_t \frac{\frac{\partial L_t}{\partial \alpha}}{\frac{\partial L_w}{\partial \alpha} + \frac{\partial L_t}{\partial \alpha}}$$

The total lift L is the sum of L_W and L_t so the sum in the denominator can be simplified and written as the derivative of the total lift due to angle of attack, yielding:

$$h = x_g - c\left(1 - \frac{\partial \epsilon}{\partial \alpha}\right) \frac{\frac{\partial C_l}{\partial \alpha}}{\frac{\partial C_L}{\partial \alpha}} \frac{l_t S_t}{c S_w}$$

Where c is the mean aerodynamic chord of the main wing. The term:

$$V_t = \frac{l_t S_t}{c S_w}$$

is known as the tail volume ratio. Its rather complicated coefficient, the ratio of the two lift derivatives, has values in the range of 0.50 to 0.65 for typical configurations, according to Piercy. Hence the expression for h may be written more compactly, though somewhat approximately, as:

$$h = x_g - 0.5 c V_t$$

h is known as the static margin. For stability it must be negative. (However, for consistency of language, the static margin is sometimes taken as − h, so that positive stability is associated with positive static margin.)

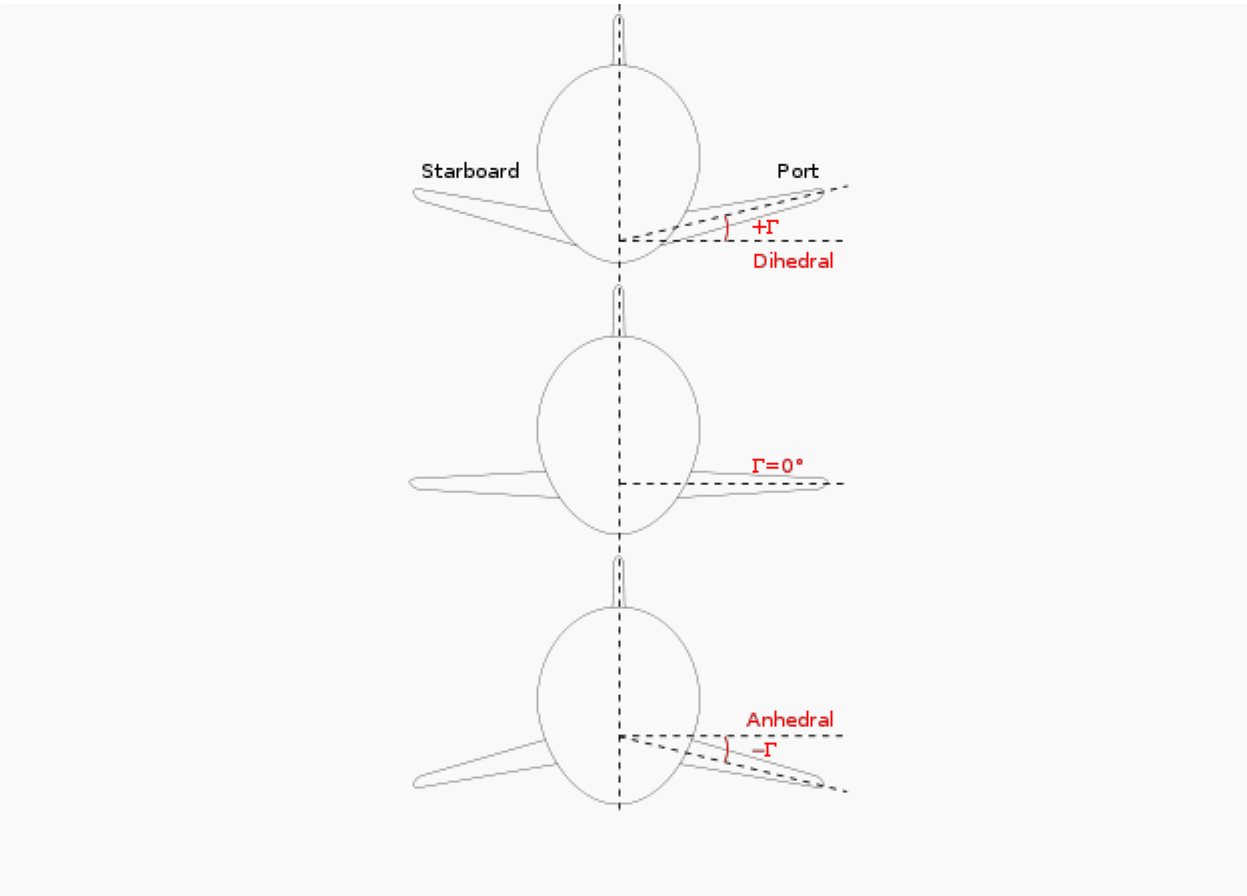
Neutral point

A mathematical analysis of the longitudinal static stability of a complete aircraft (including horizontal stabilizer) yields the position of center of gravity at which stability is neutral. This position is called the neutral point. (The larger the area of the horizontal stabilizer, and the greater the moment arm of the horizontal stabilizer about the aerodynamic center, the further aft is the neutral point.)

The static center of gravity margin (c.g. margin) or static margin is the distance between the center of gravity (or mass) and the neutral point. It is usually quoted as a percentage of the Mean Aerodynamic Chord. The center of gravity must lie ahead of the neutral point for positive stability (positive static margin). If the center of gravity is behind the neutral point, the aircraft is longitudinally unstable (the static margin is negative), and active inputs to the control surfaces are required to maintain stable flight. Some combat aircraft that are controlled by fly-by-wire systems are designed to be longitudinally unstable so they will be highly maneuverable. Ultimately, the position of the center of gravity relative to the neutral point determines the stability, control forces, and controllability of the vehicle.

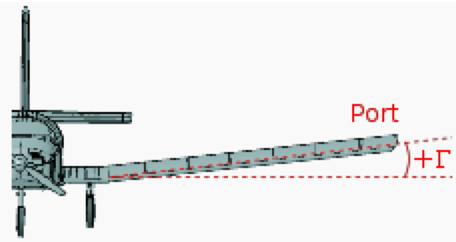
For a tailless aircraft $V_t = 0$, the neutral point coincides with the aerodynamic center, and so for longitudinal static stability the center of gravity must lie ahead of the aerodynamic center

the flow field of the main wing, and consequently experiences a downwash, reducing



Dihedral angle is the upward angle from horizontal of the wings or tailplane of a fixed-wing aircraft. Anhedral angle is the name given to negative dihedral angle, that is, when there is a downward angle from horizontal of the wings or tailplane of a fixed-wing aircraft.

Dihedral angle (or anhedral angle) has a strong influence on dihedral effect, which is named after it. Dihedral effect is the amount of roll moment produced per degree (or radian) of sideslip. Dihedral effect is a critical factor in the stability of an aircraft about the roll axis (the spiral mode). It is also pertinent to the nature of an aircraft's Dutch roll oscillation and to maneuverability about the roll axis.



Measuring the dihedral angle

Longitudinal dihedral is a comparatively obscure term related to the pitch (flight) axis of an airplane. It is the angle between the zero lift axis of the wing and horizontal tail. Longitudinal dihedral can influence the nature of controllability about the pitch axis and the nature of an aircraft's phugoid-mode oscillation.

When the term "dihedral" (of an aircraft) is used by itself it is usually intended to mean "dihedral angle". However, context may otherwise indicate that "dihedral effect" is the intended meaning.

Dihedral angle and dihedral effect

Dihedral angle is the upward angle from horizontal of the wings of a fixed-wing aircraft, or of any paired nominally-horizontal surfaces on any aircraft. The term can also apply to the wings of a bird. Dihedral angle is also used in some types of kites such as box kites. Wings with more than one angle change along the full span are said to be polyhedral.

Dihedral angle has important stabilizing effects on flying bodies because it has a strong influence on the dihedral effect.

Dihedral effect of an aircraft is a rolling moment resulting from the vehicle having a non-zero angle of sideslip. Increasing the dihedral angle of an aircraft increases the dihedral effect on it. However, many other aircraft parameters also have a strong influence on dihedral effect. Some of these important factors are: wing sweep, vertical center of gravity, and the height and size of anything on an aircraft that changes its side wards force as sideslip changes.

Longitudinal dihedral

Dihedral angle on an aircraft almost always implies the angle between two paired surfaces, one on each side of the aircraft. Even then, it is almost always between the left and right wings. However, dihedral in math means the angle between any two planes. So, in aeronautics, in one case, the term "dihedral" is applied to mean the difference in angles between two front-to-back surfaces:

Longitudinal dihedral is the difference between the angle of incidence of the wing and angle of incidence of the horizontal tail.

Longitudinal dihedral can also mean the angle between the zero lift axes of the two surfaces instead of between the root chords of the two surfaces. This is the more meaningful usage because the directions of zero-lift are pertinent to longitudinal trim and stability while the directions of the root chords are not.

Uses of dihedral angle and dihedral effect

Aircraft stability analysis

In analysis of aircraft stability, dihedral effect is also a stability derivative called $\mathbf{C_{l\beta}}$, meaning the change in rolling moment coefficient (the "$\mathbf{C_l}$") per degree (or radian) of change in sideslip (the "β").

Provision of stability

The purpose of dihedral effect is to contribute to stability in the roll axis. It is an important factor in the stability of the spiral mode which is sometimes called "roll stability". It is important to note that dihedral effect does not contribute directly to the restoring of "wings level", but that its action is indirect. It indirectly helps restore "wings level" through its effect on the spiral mode (as described below).

Wing clearance

Aircraft designers may increase dihedral angle to provide increased clearance between wing tips and the runway. The increased dihedral effect caused by this may need to be compensated for by one or more other means, such as decreasing the dihedral angle on the horizontal tail.

Using dihedral angle to adjust dihedral effect

During the design of a fixed-wing aircraft (or any aircraft with horizontal surfaces), changing dihedral angle is usually a relatively simple way to adjust the overall dihedral effect. This is to compensate for other design elements' influence on the dihedral effect. These other elements (such as wing sweep, vertical mount point of the wing, etc.) may be more difficult to change than the dihedral angle. As a result, differing amounts of dihedral angle can be found on different types of fixed-wing aircraft. For example, the dihedral angle is usually greater on low-wing aircraft than on otherwise-similar high-wing aircraft. This is because "highness" of a wing (or "lowness" of vertical center of gravity compared to the wing) naturally creates more dihedral effect itself. This makes it so less dihedral angle is needed to get the amount of dihedral effect nee

How dihedral angle creates dihedral effect and stabilizes the spiral mode

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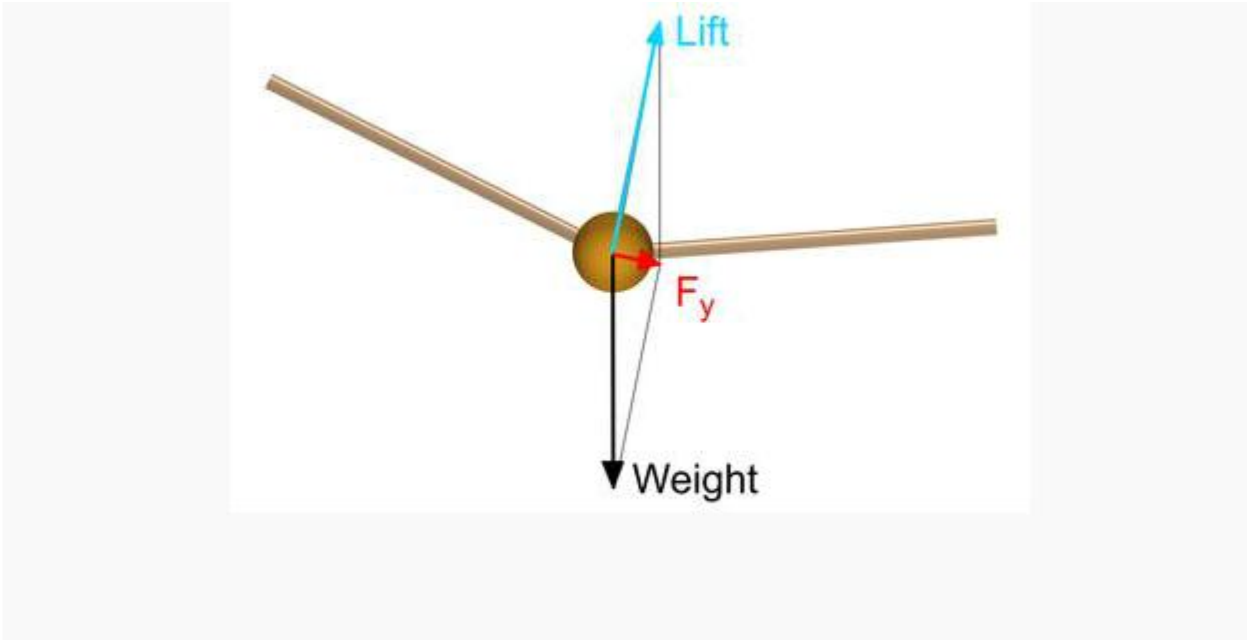
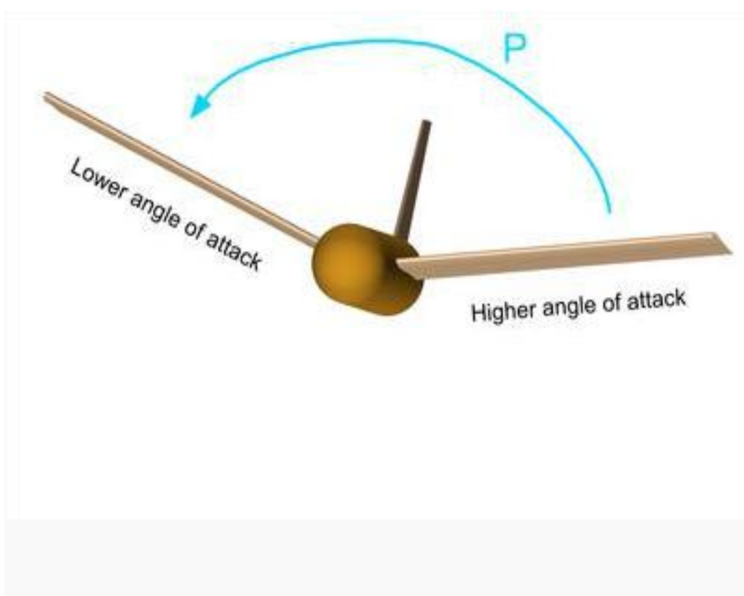
The following discusses how dihedral angle creates dihedral effect and how dihedral effect contributes to stability of the spiral mode. A stable spiral mode will cause the aircraft to eventually return to a nominally "wings level" bank angle when the angle of the wings is disturbed to become off-level.



Uncompensated lift component produces a side force F_y, which causes the aircraft to sideslip.



If a disturbance causes an aircraft to roll away from its normal wings-level position as in Figure 1, the aircraft will begin to move somewhat sideways toward the lower wing. In Figure 2, the airplane's flight path has started to move toward its left while the nose of the airplane is still pointing in the original direction. This means that the oncoming air is arriving somewhat from the left of the nose. Because of this, the airplane now has sideslip angle in addition to the bank angle. Figure 2 shows the airplane as it presents itself to the oncoming air.

How dihedral angle creates rolling moment from sideslip (dihedral effect)

In Figure 2, the sideslip conditions (not the roll angle which is also shown) make the dihedral angle geometrically produce greater angle of attack on the forward-yawed wing and smaller angle of attack on the rearward-yawed wing. This alteration of angle of attack by sideslip is visible to the eye in Figure 2. Since greater angle of attack makes greater lift, the forward wing will have more lift and the rearward wing will have less lift. This difference in lift between the wings is a rolling moment, and since it is caused by sideslip, it is dihedral effect (or more correctly, it is a contribution to the total dihedral effect of the aircraft)

How dihedral effect stabilizes the spiral mode

The rolling moment created by the sideslip (labeled as "P") tends to roll the aircraft back to wings level. More dihedral effect tries to roll the wings in the "leveling" direction more strongly, and less dihedral effect tries to roll the wings in the "leveling" direction less strongly. Dihedral effect helps stabilize the spiral mode by tending to roll the wings toward level in proportion to the amount of sideslip that builds up. It's not the whole picture however. At the same time that angle of sideslip is building up, the vertical fin is trying to turn the nose back into the wind, much like a weathervane, minimizing the amount of sideslip that can be present. If there is no sideslip, there can be no restoring rolling moment. If there is less sideslip, there is less restoring rolling moment. So, yaw stability created by the vertical fin fights the tendency for dihedral effect to roll the wings back level by not letting as much sideslip build up.

The spiral mode is the tendency to slowly diverge from, or the tendency to slowly return to wings level. If the spiral mode is stable, the aircraft will slowly return to wings-level, if it is unstable, the aircraft will slowly diverge from wings-level. Dihedral effect and yaw stability are the two primary factors that affect the stability of the spiral mode, although there are other factors that affect it less strongly.

Other dihedral-related terminology

Anhedral

Military fighter aircraft often have near zero or even negative dihedral angle. This reduces dihedral effect, reducing the stability of the spiral mode. A too-stable spiral mode decreases maneuverability and is undesirable for fighter-type aircraft.

Anhedral angles are also seen on aircraft with a high mounted wing, such as the An-124 and Lockheed Galaxy. In such designs, the high mounted wing is above the center of gravity which confers extra dihedral effect due to the pendulum effect also called the keel effect, so additional dihedral angle is often not required. In fact, such designs can have excessive dihedral effect and so be excessively stable in the spiral mode, so the anhedral angle is added to cancel out some of the dihedral effect to ensure that the aircraft can be easily maneuvered.

Polyhedral

Most aircraft have been designed with planar wings with simple dihedral (or anhedral). Some older aircraft such as the Vought F4U Corsair and the Beriev Be-12 were designed with gull wings bent near the root. Modern polyhedral wing designs generally cant upwards near the wingtips, increasing dihedral effect without increasing the angle the wings meet at the root, which may be difficult to alter for some other reason.

Polyhedral is seen on gliders and some other aircraft. The McDonnell Douglas F-4 Phantom II is one such example, unique among jet fighters for having dihedral wingtips. This was added after prototype flight testing (the original prototype of the F-4 had a flat wing) showed the need to correct some unanticipated spiral mode instability - angling the wingtips, which were already designed to fold up for carrier operations, was a more practical solution than re-engineering the entire wing.

Elevator (aircraft)

Elevators are flight control surfaces, usually at the rear of an aircraft, which control the aircraft's orientation by changing the pitch of the aircraft, and so also the angle of attack of the wing. In simplified terms, they make the aircraft nose-up or nose-down.[1] (Ascending and descending are more a function of the wing—aircraft typically land nose up.) An increased wing angle of attack will cause a greater lift to be produced by the profile of the wing, and a slowing of the aircraft speed. A decrease in angle of attack will produce an increase in speed. The elevators may be the only pitch control surface present (and are then called a slab elevator or stabilator), or may be hinged to a fixed or adjustable surface called a tail plane or horizontal stabilizer.

The rear wing to which elevators are attached have the opposite effect to a wing.

They usually create a downward pressure which counters the unbalanced moment due to the airplane's center of gravity not being located exactly on the resulting centre of pressure, which in addition to the lift generated by the main wing includes the effects of drag and engine thrust. An elevator decreases or increases the downward force created by the rear wing. An increased downward force, produced by up elevator, forces the tail down and the nose up so the aircraft speed is reduced (i.e. the wing will operate at a higher angle of attack, which produces a greater lift coefficient, so that the required lift is produced by a lower speed). A decreased downward force at the tail, produced by down elevator, allows the tail to rise and the nose to lower. The resulting lower wing angle of attack provides a lower lift coefficient, so the craft must move faster (either by adding power or going into a descent) to produce the required lift. The setting of the elevator thus determines the airplane's trim speed - a given elevator position has only one speed at which the aircraft will maintain a constant (unaccelerated) condition.

In some aircraft pitch-control surfaces are in the front, ahead of the wing; this type of configuration is called a canard, the French word for duck. The Wright Brothers' early aircraft were of this type. The canard type is more efficient, since the forward surface usually is required to produce upward lift (instead of downward force as with the usual empennage) to balance the net pitching moment. The main wing is also less likely tostall, as the forward control surface is configured to stall before the wing, causing a pitch down and reducing the angle of attack of the wing.

Supersonic aircraft have stabilators, because early supersonic flight research revealed that shock waves generated on the trailing edge of tailplanes rendered hinged elevators ineffective. Delta winged aircraft combineailerons and elevators, and their respective control inputs, into one control surface, called an elevon.

Aircraft rudders

On an aircraft, the **rudder** is a directional control surface along with the rudder-like elevator (usually attached to horizontal tail structure, if not a slab elevator) and ailerons(attached to the wings) that control pitch and roll. The rudder is usually attached to the fin (or vertical stabilizer) which allows the pilot to control yaw in the vertical axis, i.e. change the horizontal direction in which the nose is pointing. The rudder's direction in aircraft since the "Golden Age" of flight between the two World Wars into the 21st century has been manipulated with the movement of a pair of foot pedals by the pilot, while during the pre-1919 era rudder control was most often operated with by a center-pivoted, solid "rudder bar" which usually had pedal and/or stirrup-like hardware on its ends to allow the pilot's feet to stay close to the ends of the bar's rear surface.

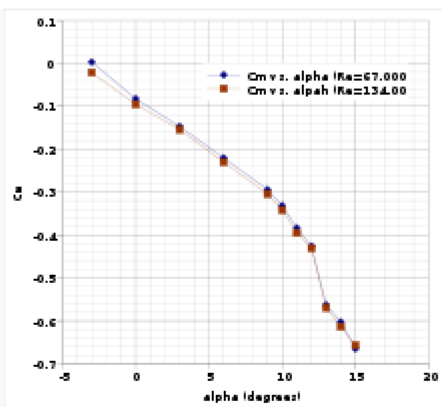
In practice, both aileron and rudder control input are used together to turn an aircraft, the ailerons imparting roll, the rudder imparting yaw, and also compensating for a phenomenon called adverse yaw. Adverse yaw is readily seen if the most simple type of ailerons alone are used for a turn. The downward moving aileron acts like a flap, generating more lift for one wing, and therefore more drag (though since the 1930s, many aircraft have used Frise ailerons or differential ailerons, which compensate for the adverse yaw and require little or no rudder input in regular turns). Initially, this drag yaws the aircraft in the direction opposite to the desired course. A rudder alone will turn a conventional fixed wing aircraft, but much more slowly than if ailerons are also used in conjunction. Use of rudder and ailerons together produces co-ordinate turns, in which the longitudinal axis of the aircraft is in line with the arc of the turn, neither slipping (under-ruddered), nor skidding (over-ruddered). Improperly ruddered turns at low speed can precipitate a spin which can be dangerous at low altitudes.

Sometimes pilots may intentionally operate the rudder and ailerons in opposite directions in a maneuver called a forward slip. This may be done to overcome crosswinds and keep the fuselage in line with the runway, or to more rapidly lose altitude by increasing drag, or both. The pilots of Any aircraft rudder is subject to considerable forces that determine its position via a force or torque balance equation. In extreme cases these forces can lead to loss of rudder control or even destruction of the rudder. (The same principles also apply to water vessels, of course, but it is more important for aircraft because they have lower engineering margins.) The largest achievable angle of a rudder in flight is called its blow down limit; it is achieved when the force from the air or blow down equals the maximum available hydraulic pressure.

Trim tab

Trim tabs are small surfaces connected to the trailing edge of a larger control surface, such as a rudder, on a boat or aircraft, used to control the trim of the controls, i.e. to counteract hydro- or aero-dynamic forces and stabilize the boat or aircraft in a particular desired attitude without the need for the operator to constantly apply a control force. This is done by adjusting the angle of the tab relative to the larger surface.

Changing the setting of a trim tab adjusts the neutral or resting position of a control surface (such as an elevator or rudder). As the desired position of a control surface changes (corresponding mainly to different speeds), an adjustable trim tab will allow the operator to reduce the manual force required to maintain that position—to zero, if used correctly. Thus the trim tab acts as a servo tab. Because the center of pressure of the trim tab is further away from the axis of rotation of the control surface than the center of pressure of the control surface, the movement generated by the tab can match the movement generated by the control surface. The position of the control surface on its axis will change until the movements from the control surface and the trim surface balance each other.



Pitching moment

A graph showing coefficient of pitching moment with respect to attack, the negative slope for positive α indicates stability in pitching In aerodynamics, the pitching moment on an airfoil is the moment (or torque) produced by the aerodynamic force on the airfoil if that aerodynamic force is considered to be applied, not at the center of pressure, but at the aerodynamic center of the airfoil. The pitching moment on the wing of an airplane is part of the total moment that must be balanced using the lift on the horizontal stabilizer.

The lift on an airfoil is a distributed force that can be said to act at a point called the center of pressure. However, as angle of attack changes on a cambered airfoil, there is movement of the center of pressure forward and aft. This makes analysis difficult when attempting to use the concept of the center of pressure. One of the remarkable properties of a cambered airfoil is that, even though the center of pressure moves forward and aft, if the lift is imagined to act at a point called the aerodynamic center the moment of the lift force changes in proportion to the square of the airspeed. If the moment is divided by the dynamic pressure, the area and chord of the airfoil, to compute a pitching moment coefficient, this coefficient changes only a little over the operating range of angle of attack of the airfoil. The combination of the two concepts of aerodynamic center and pitching moment coefficient make it relatively simple to analyze some of the flight characteristics of an aircraft.

Measurement

The aerodynamic center of an airfoil is usually close to 25% of the chord behind the leading edge of the airfoil. When making tests on a model airfoil, such as in a wind-tunnel, if the force sensor is not aligned with the quarter-chord of the airfoil, but offset by a distance **x**, the pitching moment about the quarter-chord point, $M_{c\,/\,4}$ is given by

$$M_{c/4} = M_{\text{indicated}} + \mathbf{x} \times (D_{\text{indicated}}, L_{\text{indicated}})$$

where the indicated values of D and L are the drag and lift on the model, as measured by the force sensor..

Coefficient

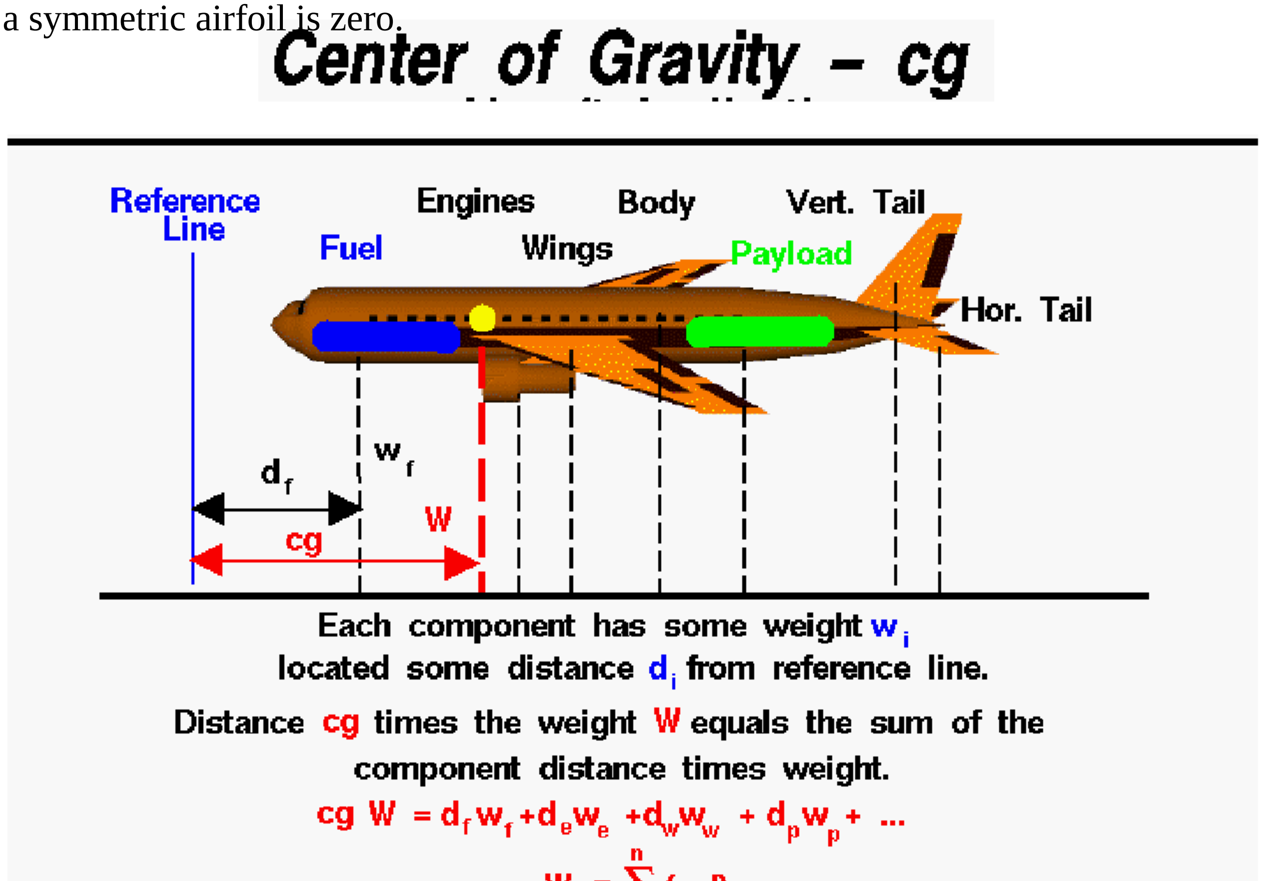
The **pitching moment coefficient** is important in the study of the longitudinal static stability of aircraft and missiles.

The pitching moment coefficient C_m is defined as follows

$$C_m = \frac{M}{qSc}$$

where M is the pitching moment, q is the dynamic pressure, S is the planform area, and c is the length of the chord of the airfoil. C_m is a dimensionless coefficient so consistent units must be used for M, q, S and c.

Pitching moment coefficient is fundamental to the definition of aerodynamic center of an airfoil. The aerodynamic center is defined to be the point on the chord line of the airfoil at which the pitching moment coefficient does not vary with angle of attack, or at least does not vary significantly over the operating range of angle of attack of the airfoil. In the case of a symmetric airfoil, the lift force acts through one point for all angles of attack, and the center of pressure does not move as it does in a cambered airfoil. Consequently the pitching moment coefficient for a symmetric airfoil is zero.



Pitching moment is, by convention, considered to be positive when it acts to pitch the airfoil in the nose-up direction. Conventional cambered airfoils supported at the aerodynamic center pitch nose-down so the pitching moment coefficient of these airfoils is negative.

An airplane in flight can be maneuvered by the pilot using the aerodynamic control surfaces; the elevator, rudder, or ailerons. As the control surfaces change the amount of force that each surface generates, the aircraft rotates about a point called the center of gravity. The center of gravity is the average location of the weight of the aircraft. The weight is actually distributed throughout the airplane, and for some problems it is important to know the distribution. But for total aircraft maneuvering, we need to be concerned with only the total weight and the location of the center of gravity.

How do engineers determine the location of the center of gravity for an airplane which they are designing?

An airplane is a combination of many parts; the wings, engines, fuselage, and tail, plus the payload and the fuel. Each part has a weight associated with it which the engineer can estimate, or calculate, using Newton's weight equation:

$$w = m * g$$

where **w** is the weight, **m** is the mass, and **g** is the gravitational constant which is 32.2 ft/square sec in English units and 9.8 meters/square sec in metric units. To determine the center of gravity cg, we choose a reference location, or reference line. The cg is determined relative to this reference location. The total weight of the aircraft is simply the sum of all the individual weights of the components. Since the center of gravity is an average location of the weight, we can say that the weight of the entire aircraft **W** times the location **cg** of the center of gravity is equal to the sum of the weight **w** of each component times the distance **d** of that component from the reference location:

Center of gravity of an aircraft

The center-of-gravity (CG) is the point at which an aircraft would balance if it were possible to suspend it at that point. It is the mass center of the aircraft, or the theoretical point at which the entire weight of the aircraft is assumed to be concentrated.[1] Its distance from the reference datum is determined by dividing the total moment by the total weight of the aircraft.[2] The center-of-gravity point affects the stability of the aircraft. To ensure the aircraft is safe to fly, the center-of-gravity must fall within specified limits established by the manufacturer.

Terminology

Ballast

Ballast is removable or permanently installed weight in an aircraft used to bring the center of gravity into the allowable range.

Center-of-gravity limits

CG limits are specified longitudinal (forward and aft) and/or lateral (left and right) limits within which the aircraft's center of gravity must be located during flight. The CG limits are indicated in the airplane flight manual. The area between the limits is called the CG range of the aircraft.

Weight and balance

When the weight of the aircraft is at or below the allowable limit(s) for its configuration (parked, ground movement, take-off, landing, etc.) and its center of gravity is within the allowable range, and both will remain so for the duration of the flight, the aircraft is said to be within weight and balance. Different maximum weights may be defined for different situations; for example, large aircraft may have maximum landing weights that are lower than maximum take-off weights (because some weight is expected to be lost as fuel is burned during the flight). The center-of-gravity may change over the duration of the flight as the aircraft's weight changes due to fuel burn.

Reference datum

The reference datum is a reference plane that allows accurate, and uniform, measurements to any point on the aircraft. The location of the reference datum is established by the manufacturer and is defined in the aircraft flight manual. The horizontal reference datum is an imaginary vertical plane or point, arbitrarily fixed somewhere along the longitudinal axis of the aircraft, from which all horizontal distances are measured for weight and balance purposes. There is no fixed rule for its location, and it may be located forward of the nose of the aircraft. For helicopters, it may be located at the rotor mast, the nose of the helicopter, or even at a point in space ahead of the helicopter. While the horizontal reference datum can be anywhere the manufacturer chooses, most small training helicopters have the horizontal reference datum 100 inches forward of the main rotor shaft centerline. This is to keep all the computed values positive. The lateral reference datum, is usually located at the center of the helicopter.[3]

Arm

The arm is the chord wise (fore-and-aft) distance from the datum to any point within the aircraft.

Moment

The moment is a measure of force that results from an object's weight acting through an arc that is centered on the zero point of the reference datum distance. Moment is also referred to as the tendency of an object to rotate or pivot about a point (the zero point of the datum, in this case). The further an object is from this point, the greater the force it exerts. Moment is calculated by multiplying the weight of an object by its arm.

Mean Aerodynamic Chord (MAC)

A specific chord line of a tapered wing, At the mean aerodynamic chord, the center of pressure has the same aerodynamic force, position, and area as it does on the rest of the wing. The MAC represents the width of an equivalent rectangular wing in given conditions. On some aircraft, the center of gravity is expressed as a percentage of the length of the MAC. In order to make such a calculation, the position of the leading edge of the MAC must be known ahead of time. This position is defined as a distance from the reference datum and is found in the aircraft's flight manual and also on the aircraft's type certificate data sheet. If a general MAC is not given but a LeMAC (leading edge mean aerodynamic chord) and a TeMAC (trailing edge mean aerodynamic chord)are given (both of which would be referenced as an arm measured out from the datum line) then your MAC can be found by finding the difference between your LeMAC and your TeMAC.

Calculation

Center of gravity is calculated as follows:

- Determine the weights and arms of all mass within the aircraft.
- Multiply weights by arms for all mass to calculate moments.
- Add the moments of all mass together.
- Divide the total moment by the total weight of the aircraft to give an overall arm.

The arm that results from this calculation must be within the arm limits for the center of gravity that are dictated by the manufacturer. If it is not, weight in the aircraft must be removed, added (rarely), or redistributed until the center of gravity falls within the prescribed limits.

For the sake of simplicity, center of gravity calculations are usually performed along only a single line from the zero point of the reference datum, usually the line that represents the roll axis of the aircraft (to calculate fore-aft balance). In complex situations, more than one line may be separately calculated, e.g., one calculation for fore-aft balance and one calculation for left-right balance.

Weight is calculated simply by adding up all weight in the aircraft. This weight must be within the allowable weight limits for the aircraft.

The weight and moment of fixed portions of the aircraft (engines, wings, etc.) does not change and is provided by the manufacturer. The manufacturer also provides information facilitating the calculation of moments for fuel loads. Other removable weight must be properly accounted for in the calculation by the operator.

In larger aircraft, weight and balance is often expressed as a percentage of mean aerodynamic chord, or MAC. For example, assume that by using the calculation method above, the center of gravity (CG) was found to be 76 inches aft of the aircraft's datum and the leading edge of the MAC is 62 inches aft of the datum. Therefore, the CG lies 14 inches aft of the leading edge of the MAC. If the MAC is 80 inches in length, the percentage of MAC is found by calculating what percentage 14 is of 80. In this case, one could say that the CG is 17.5% of MAC. If the allowable limits were 15% to 35%, the aircraft would be properly loaded.

Example

Given:

	Weight (lb)	**Arm (in)**	**Moment (lb-in)**
Empty weight	1,495.0	101.4	151,593.0

Pilot and passengers	380.0	64.0	24,320.0
Fuel (30 gallons @ 6 lb/gal)	180.0	96.0	17,280.0
Totals	2,055.0	94.01	193,193.0

To find the center of gravity, we divide the total moment of mass by the total mass of the aircraft: 193,193 ÷ 2,055 = **94.01 inches** behind the datum plane.

Incorrect weight and balance in fixed-wing aircraft

When the center of gravity or weight of an aircraft is outside the acceptable range, the aircraft may not be able to sustain flight, or it may be impossible to maintain the aircraft in level flight in some or all circumstances. Placing the CG or weight of an aircraft outside the allowed range can lead to an unavoidable crash of the aircraft.

Center of gravity out of range

When the fore-aft center of gravity is out of range, the aircraft may pitch uncontrollably down or up, and this tendency may exceed the control authority available to the pilot, causing a loss of control. The excessive pitch may be apparent in all phases of flight, or only during certain phases, such as take-off or descent. Because the burning of fuel gradually produces a loss of weight and possibly a shift in the center of gravity, it is possible for an aircraft to take off with the center of gravity in a position that allows full control, and yet later develop an imbalance that exceeds control authority. Calculations of center of gravity must take this into account (often part of this is calculated in advance by the manufacturer and incorporated into CG limits).

Weight out of range

Few aircraft impose a minimum weight for flight (although a minimum pilot weight is often specified), but all impose a maximum weight. If the maximum weight is exceeded, the aircraft may not be able to achieve or sustain controlled, level flight. Excessive take-off weight may make it impossible to take off within available runway lengths, or it may completely prevent take-off. Excessive weight in flight may make climbing beyond a certain altitude difficult or impossible, or it may make it impossible to maintain an altitude.

Incorrect weight and balance in helicopters

The center of gravity is even more critical for helicopters than it is for fixed-wing aircraft (weight issues remain the same). As with fixed-wing aircraft, a helicopter may be properly loaded for takeoff, but near the end of a long flight when the fuel tanks are almost empty, the CG may have shifted enough for the helicopter to be out of balance laterally or longitudinally.[2] For helicopters with a single main rotor, the CG is usually close to the main rotor mast. Improper balance of a helicopter's load can result in serious control problems. In addition to making a helicopter difficult to control, an out-of-balance loading condition also decreases maneuverability since cyclic control is less effective in the direction opposite to the CG location.

The pilot tries to perfectly balance a helicopter so that the fuselage remains horizontal in hovering flight, with no cyclic pitch control needed except for wind correction. Since the fuselage acts as a pendulum suspended from the rotor, changing the center of gravity changes the angle at which the aircraft hangs from the rotor. When the center of gravity is directly under the rotor mast, the helicopter hangs horizontal; if the CG is too far forward of the mast, the helicopter hangs with its nose tilted down; if the CG is too far aft of the mast, the nose tilts up.

CG forward of forward limit

A forward CG may occur when a heavy pilot and passenger take off without baggage or proper ballast located aft of the rotor mast. This situation becomes worse if the fuel tanks are located aft of the rotor mast because as fuel burns the weight located aft of the rotor mast becomes less.

This condition is recognizable when coming to a hover following a vertical takeoff. The helicopter will have a nose-low attitude, and the pilot will need excessive rearward displacement of the cyclic control to maintain a hover in a no-wind condition. In this condition, the pilot could rapidly run out of rearward cyclic control as the helicopter consumes fuel. The pilot may also find it impossible to decelerate sufficiently to bring the helicopter to a stop.

In the event of engine failure and the resulting autorotation, the pilot may not have enough cyclic control to flare properly for the landing.

A forward CG will not be as obvious when hovering into a strong wind, since less rearward cyclic displacement is required than when hovering with no wind. When determining whether a critical balance condition exists, it is essential to consider the wind velocity and its relation to the rearward displacement of the cyclic control.

CG aft of aft limit

Without proper ballast in the cockpit, exceeding the aft CG may occur when:

- A lightweight pilot takes off solo with a full load of fuel located aft of the rotor mast.
- A lightweight pilot takes off with maximum baggage allowed in a baggage compartment located aft of the rotor mast.
- A lightweight pilot takes off with a combination of baggage and substantial fuel where both are aft of the rotor mast.

An aft CG condition can be recognized by the pilot when coming to a hover following a vertical takeoff. The helicopter will have a tail-low attitude, and the pilot will need excessive forward displacement of cyclic control to maintain a hover in a no-wind condition. If there is a wind, the pilot needs even greater forward cyclic. If flight is continued in this condition, the pilot may find it impossible to fly in the upper allowable airspeed range due to inadequate forward cyclic authority to maintain a nose-low attitude. In addition, with an extreme aft CG, gusty or rough air could accelerate the helicopter to a speed faster than that produced with full forward cyclic control. In this case, asymmetry of lift and blade flapping could cause the rotor disc to tilt aft. With full forward cyclic control already applied, the rotor disc might not be able to be lowered, resulting in possible loss of control, or the rotor blades striking the tail boom.

Lateral balance

In fixed-wing aircraft, lateral balance is often much less critical than fore-aft balance, simply because most mass in the aircraft is located very close to its center. An exception is fuel, which may be loaded into the wings, but since fuel loads are usually symmetrical about the axis of the aircraft, lateral balance is not usually affected. The lateral center of gravity may become important if the fuel is not loaded evenly into tanks on both sides of the aircraft, or (in the case of small aircraft) when passengers are predominantly on one side of the aircraft (such as a pilot flying alone in a small aircraft). Small lateral deviations of CG that are within limits may cause an annoying roll tendency that pilots must compensate for, but they are not dangerous as long as the CG remains within limits for the duration of the flight.

For most helicopters, it is usually not necessary to determine the lateral CG for normal flight instruction and passenger flights. This is because helicopter cabins are relatively narrow and most optional equipment is located near the center line. However, some helicopter manuals specify the seat from which solo flight must be conducted. In addition, if there is an unusual situation, such as a heavy pilot and a full load of fuel on one side of the helicopter, which could affect the lateral CG, its position should be checked against the CG envelope. If carrying external loads in a position that requires large lateral cyclic control displacement to maintain level flight, fore and aft cyclic effectiveness could be dramatically limited.

Fuel dumping and overweight operations

Many large transport-category aircraft are able to take-off at a greater weight than they can land. This is possible because the weight of fuel that the wings can support along their span in flight, or when parked or taxiing on the ground, is greater than they can tolerate during the stress of landing and touchdown, when the support is not distributed along the span of the wing.

Normally the portion of the aircraft's weight that exceeds the maximum landing weight (but falls within the maximum take-off weight) is entirely composed of fuel. As the aircraft flies, the fuel burns off, and by the time the aircraft is ready to land, it is below its maximum landing weight. However, if an aircraft must land early, sometimes the fuel that remains aboard still keeps the aircraft over the maximum landing weight. When this happens, the aircraft must either burn off the fuel (by flying in a holding pattern) or dump it (if the aircraft is equipped to do this) before landing to avoid damage to the aircraft. In an emergency, an aircraft may choose to land overweight, but this may damage it, and at the very least an overweight landing will mandate a thorough inspection to check for any damage.

In some cases, an aircraft may take off overweight deliberately. An example might be an aircraft being ferried over a very long distance with extra fuel aboard. An overweight take-off typically requires an exceptionally long runway. Overweight operations are not permitted with passengers aboard. Many smaller aircraft have a maximum landing weight that is the same as the maximum take-off weight, in which case issues of overweight landing cannot arise.

Static margin

Static margin is a concept used to characterize the static stability and controllability of aircraft and missiles.

- In aircraft analysis, static margin is defined as the distance between the center of gravity and the neutral point of the aircraft

- In missile analysis, static margin is defined as the distance between the center of gravity and the center of pressure.

The differences in the two fields arise largely from the use of cambered wings in aircraft. Missiles are symmetric vehicles and if they have airfoils they too are symmetric. With cambered wings the location of the center of pressure on the wing is a strong function of angle of attack - see movement of the center of pressure.

The response of an aircraft or missile to an angular disturbance such as a pitch disturbance is determined by its static margin.

With the center of gravity forward of the neutral point, an aircraft has positive longitudinal static stability. (For an aircraft this may be described as negative static margin.) With the center of gravity aft of the neutral point, an aircraft is statically unstable, and requires some form of augmentation to be flown with an acceptable workload. (For an aircraft this may be described as positive static margin.)

For missiles, positive static margin implies that the complete vehicle makes a restoring moment for any angle of attack from the trim position. If the center of pressure is behind the center of gravity then the moment will be restoring. For missiles with symmetric airfoils, the neutral point and the center of pressure are coincident and the term neutral point is not used.

If an aircraft in flight suffers a disturbance in pitch that causes an increase (or decrease) in angle of attack, it is desirable that the aerodynamic forces on the aircraft cause a decrease (or increase) in angle of attack so that the disturbance does not cause a continuous increase (or decrease) in angle of attack. This is longitudinal static stability.

Relationship to aircraft and missile stability and control

- If the center of gravity (CG) of an aircraft is forward of the neutral point, or the CG of a missile is forward of the center of pressure, the vehicle will respond to a disturbance by producing an aerodynamic moment that returns the angle of attack of the vehicle towards the angle that existed prior to the disturbance.
- If the CG of an aircraft is behind the neutral point, or the CG of a missile is behind the center of pressure, the vehicle will respond to a disturbance by producing an aerodynamic moment that continues to drive the angle of attack of the vehicle further away from the starting position.

The first condition above is positive static stability. In missile analysis this is described as positive static margin. (In aircraft analysis it may be described as

negative static margin.) The second condition above is negative static stability. In missile analysis this is defined as negative static margin. (In aircraft analysis it may be described as positive static margin.)

Depending on the static margin, humans may not be able to use control inputs to the elevators to control the pitch of the vehicle. Typically, computer based autopilots are required to control the vehicle when it has negative static stability - usually described as negative static margin.

The purpose of the reduced stability (low static margin) is to make an aircraft more responsive to pilot inputs. An aircraft with a large static margin will be very stable and slow to respond to the pilot inputs. The amount of static margin is an important factor in determining the handling qualities of an aircraft. For an unguided rocket, the vehicle must have a large positive static margin so the rocket shows minimum tendency to diverge from the direction of flight given to it at launch. In contrast, guided missiles usually have a negative static margin for increased maneuverability.

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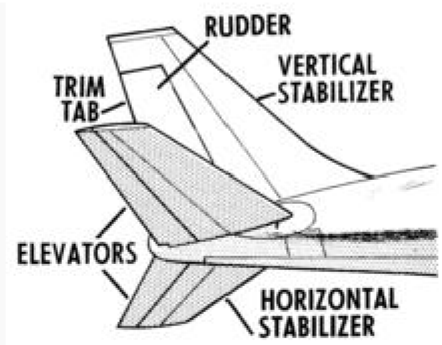
Yaw damper

A yaw damper is a device used on many aircraft (usually jets and turboprops) to damp (reduce) the rolling and yawing oscillations due to Dutch roll mode. It involves yaw rate sensors and a processor that provides a signal to an actuator connected to the rudder. The use of the yaw damper helps to provide a better ride for passengers and on some aircraft is a required piece of equipment to ensure that the aircraft stability remains within certification values.

Trim tabs are small surfaces connected to the trailing edge of a larger control surface on a boat or aircraft, used to control the trim of the controls, i.e. to counteract hydro- or aero-dynamic forces and stabilise the boat or aircraft in a particular desired attitude without the need for the operator to constantly apply a control force. This is done by adjusting the angle of the tab relative to the larger surface.

Changing the setting of a trim tab adjusts the neutral or resting position of a control surface (such as an elevator or rudder). As the desired position of a control surface changes (corresponding mainly to different speeds), an adjustable trim tab will allow the operator to reduce the manual force required to maintain that position—to zero, if used correctly. Thus the trim tab acts as a servo tab. Because the center of pressure of the trim tab is farther away from the axis of rotation of the control surface than the center of pressure of the control surface, the moment generated by the tab can match the moment generated by the control surface. The position of the control surface on its axis will change until the movements from the control surface and the trim surface balance each other.



Many airplanes (including gliders) have trim tabs on their elevators, as a simple method of providing trim in the longitudinal axis.

All aircraft must have a system for ensuring trim in the longitudinal axis, though methods other than trim tabs may be used. Alternatives include:

- a spring attached to the control system that can be adjusted by the pilot
- in the case of the elevator, an all-moving horizontal stabilizer whose position can be adjusted in flight by the pilot.

Elevator trim frees the pilot from exerting constant pressure on the pitch controls. Instead, the pilot adjusts a longitudinal trim control (often in the form of a wheel) to cancel out control forces for a given airspeed / weight distribution. Typically, when this trim control is rotated forward, the nose is held down; conversely, if the trim wheel is moved back, the tail becomes "heavy." Many newer aircraft, especially jet aircraft, have electric trim controls.

Many airplanes also have rudder and/or aileron trim systems. On some of these, the rudder trim tab is rigid but adjustable on the ground by bending: it is angled slightly to the left (when viewed from behind) to lessen the need for the pilot to push the rudder pedal constantly to overcome the left-turning tendencies of some prop-driven aircraft. Other aircraft have hinged rudder trim tabs that the pilot can adjust in flight.

When a trim tab is employed, it is moved into the slipstream opposite to the control surface's desired deflection. For example, in order to trim an elevator to hold the nose down, the elevator's trim tab will actually rise up into the slipstream. The increased pressure on top of the trim tab surface caused by raising it will then deflect the entire elevator slab down slightly, causing the tail to rise and the aircraft's nose to move down.[1] In the case of an aircraft where deployment of

high-lift devices (flaps) would significantly alter the longitudinal trim, a supplementary trim tab is arranged to simultaneously deploy with the flaps so that pitch attitude is not markedly changed.

The use of trim tabs significantly reduces pilots' workload during continuous maneuvers (eg: sustained climb to altitude after takeoff or descent prior to landing), allowing them to focus their attention on other tasks such as traffic avoidance or communication with air traffic control.

Both elevator trim and pitch trim affect the small trimming part of the elevator on jet airliners. The former is supposed to be set in a certain position for a longer time, while the pitch trim (controlled with the landing pilot's thumb on the yoke or joystick, and is thereby easy to maneuver) is used all the time after the flying pilot has disabled the autopilot, especially after each time the flaps are lowered or at every change in the airspeed, at the descent, approach and final. Elevator trim is most used for controlling the attitude at cruising by the autopilot.

Beyond reducing pilot workload, proper trim also increases fuel efficiency by reducing drag. For example, propeller aircraft have a tendency to yaw when operating at high power, for instance when climbing: this increases parasite drag because the craft is not flying straight into the apparent wind. In such circumstances, the use of an adjustable rudder trim tab can reduce yaw.

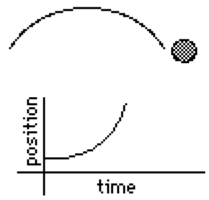
Longitudinal Stability and Trim

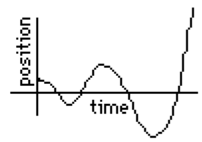
The drag of the system is dependent on the distribution of loads between the surfaces. In order to determine this, and to properly size the tail surface, we must consider the aircraft's stability and trim. Stability is the tendency of a system to return to its equilibrium condition after being disturbed from that point.

Two types

of stability or instability are important.

A static instability: A dynamic instability:

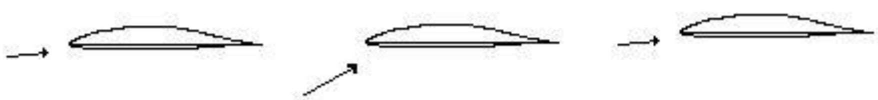




An airplane must be a stable system (well, with some exceptions) with acceptable time constants. To assure this, a careful analysis of the dynamic response and controllability is required, but here we look only at the simplest case: static longitudinal stability and trim. This will tell us something about the aerodynamic design of the surfaces -- the load they must carry, the effect of airfoil properties,

and the drag associated with the surfaces.

If we displace the wing or airplane from its equilibrium flight condition to a higher angle of attack and higher lift coefficient:

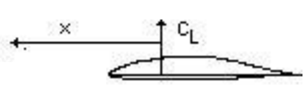


we would like it to return to the lower lift coefficient.

This requires that the pitching moment about the rotation point*, C_m, become negative as we increase C_L:

$$\frac{dC_m}{dC_L} < 0 \quad \text{for static stability.}$$

where x is the distance from the system's center of additional lift to the c.g.



$$\frac{dC_{m_{c.g.}}}{dC_L} = -\frac{x}{c} \equiv -\text{static margin}$$

$$C_{m_{c.g.}} = C_{m_0} - \frac{x}{c} C_L$$

If x were 0, the system would be neutrally stable. x/c represents the margin of static stability and is thus called the static margin. Typical values for stable airplanes range from 5% to 40%. The airplane may therefore be made as stable as desired by moving the c.g. forward (by putting lead in the nose) or moving the wing back. One needs no tail for stability then, only the right position of the c.g..

A very stable airplane.

Although this configuration is stable, it will tend to nose down whenever any lift is produced. In addition to stability we require that the airplane be trimmed (in moment equilibrium) at the desired C_L.

$$C_{m_{c.g.}} = C_{m_0} - \frac{x}{c} C_L = 0$$

With a single wing, generating a sufficient C_m at zero lift to trim with a reasonable static margin and C_L is not so easy. (Most airfoils have negative values of C_{m0}.) Although tailless aircraft can generate sufficiently positive C_{m0} to trim, the more conventional solution is to add an additional lifting surface such as an aft-tail or canard. The following sections deal with some of the considerations in the design of each of these configurations.

Flaperon

A flaperon is a type of aircraft control surface that combines aspects of both flaps and ailerons. In addition to controlling the roll or bank of an aircraft as do conventional ailerons, both flaperons can be lowered together to function similarly to a dedicated set of flaps. Both ailerons could also be raised, which would give spoilerons.

The pilot has separate controls for ailerons and flaps. A mixer is used to combine the separate pilot input into this single set of control surfaces called flaperons. The use of flaperons instead of separate ailerons and flaps can reduce the weight of an aircraft. The complexity is transferred from having a double set of control surfaces (flaps and ailerons) to the mixer.

Stabilator

A stabilator (stabilizer-elevator, also all moving tailplane or all flying tail) is an aircraft control surface that combines the functions of an elevator and a horizontal stabilizer. Most fixed-wing aircraft control pitch using a hinged

horizontal flap — the elevator — attached to the back of the fixed horizontal stabilizer, but some aircraft make the entire stabilizer movable.

Wing warping

Wing warping was an early system for lateral (roll) control of a fixed-wing aircraft. The technique, used and patented by the Wright brothers, consisted of a system of pulleys and cables to twist the trailing edges of the wings in opposite directions. In many respects, this approach is similar to that used to trim the performance of a paper airplane by curling the paper at the back of its wings.

In practice, since most wing warping designs involved flexing of structural members they were difficult to control and liable to cause structural failure. Ailerons had begun to replace wing warping as the most common means of

achieving lateral control as early as 1911, especially in biplane designs. Monoplane wings of the period were much more flexible, and proved more amenable to wing warping - but even for monoplane designs, ailerons became the norm after 1915.

Lateral (roll) control in early aircraft was problematic at best. An overly flexible, involuntarily twisting wing can cause involuntary rolling, but even worse, it can convert attempts at correction, either from wing warping or ailerons, into a counteracting "tab" effect. Once this was fully understood wing structures were made progressively more rigid, precluding wing warping altogether - and aircraft became far more controllable in the lateral plane.

Coordinated flight

When an aircraft is flying with zero sideslip a turn and bank indicator installed on the aircraft’s instrument panel usually shows the ball in the center of the spirit level. There is no lateral acceleration of the aircraft and occupants perceive their weight to be acting straight downwards into their seats.

Particular care to maintain coordinated flight is required by the pilot when entering and leaving turns.

Advantages

Coordinated flight is usually preferred over uncoordinated flight for the following reasons:

- it is more comfortable for the occupants
- it minimizes the drag force on the aircraft
- it causes fuel to be drawn equally from tanks in both wings
- it minimizes the risk of entering a spin

Coordinating the turn

If the pilot were to use only the rudder to initiate a turn in the air, the airplane would tend to "skid" to the outside of the turn.

If the pilot were to use only the ailerons to initiate a turn in the air, the airplane would tend to "slip" towards the lower wing.

If the pilot were to fail to use the elevator to increase the angle of attack throughout the turn, the airplane would also tend to "slip" towards the lower wing.

However, if the pilot makes appropriate use of the rudder, ailerons and elevator to enter and leave the turn such that sideslip and lateral acceleration are zero the airplane will be in coordinated flight.

DETERMINATION OF MANEUVER POINTS IN FLIGHT

Knowledge of the maneuver points of an aircraft important from the view point of safe a maneuvering flight of aircraft. There are two maneuvering points of interest, the stick-fixed maneuver point and the stick-free maneuver point. The stick-fixed maneuver point is defined as the center of gravity position for which no change in the elevator angle (stick travel) is required for a normal acceleration of (ng) and is given by-

$$N = N_0 - \frac{63 \rho C_{m\delta} g l_t}{2 \tau (W/S)} \left(1 + \frac{\mu}{n^2}\right)$$

Where

the stick-fixed maneuver

N_O = **point**

$C_{M\delta}$

τ

l_t = **the elevator power**
= **the elevator effectiveness**
= **the tail length**

μ **0 for pull up ;** **μ = 1 for**

= **turna**

and **ρ, g, W, S** are respectively the air density, acceleration due to gravity, airplane weight an wing area. For any other position X_{cg} of the center of gravity, we have

$$\frac{d\delta_e}{dn} = -\frac{63 g l_t}{\tau V^2} \left(1 + \frac{\mu}{n^2}\right) \frac{H_m}{N_m - N_o}$$

where

Hm = Nm - Xcg maneuver margin stick fixed

we see that the stick travel e per g is proportional to the stick fixed maneuver margin Hm. The stick movement is inversely proportional to the square of the speed neglecting aeroelastic and compressibility effects

$$\frac{d\delta_e}{dn} = \frac{2 (W/S)}{\rho C_{m\delta} V^2} H_m = \frac{(W/S)}{C_{m\delta} q} H_m$$

By combing eqns.(5.B.1) and (5.B.2) from eq. (5.B.3) we see that

$$\frac{d}{dx_{eg}}\left(\frac{d\delta_e}{d_n}\right) = -\frac{(W/S)}{C_{m\delta}\, q}$$

So that the slope of the elevator angle per g versus the c.g. position varies inversely as the dynamic pressure q.

it is used to determine the stick fixed maneuver points in flight by putting the aircraft in a level turn at some speed V with the elevator angle δ_c for trim. If Φ be the bank angle, R the radius nd n the load factor of the turn, we have

$$R = \frac{V^2}{g \tan \phi}$$

; n = sec Φ

The ratio (**d δe / d**$_n$) is measured at a given speed V for a range of c.g. positions and load factors. The stick fixed maneuver point is the obtained as the intersection of the (**d δe / d**$_n$) vs X_{cg} with the X_{cg} axis.

The stick free maneuver point is the center of gravity position for which no change in the hinge moment or stick force is required to maintain a normal acceleration (ng). The stick free maneuver point is given by .

$$N'_m = N'_o + \frac{57.3\ C_{M\delta}}{(W/S)\ Ch_{\delta}}\ \frac{g\rho l_t}{2}\left(C_{h\alpha} - \frac{1.1 C_{h\delta}}{T}\right)\left(1 + \frac{\mu}{n^2}\right)$$

where

$C_{h\alpha}$ =floating tendency

$C_{h\delta}$ =restoring tendency

N'_o =stick free neutral point

For any other position X_{cg} of the center of gravity we have

$$\frac{dF_o}{dn} = -\frac{57.3\ G\eta_L S_\theta C_\theta g \rho l_t}{2}\left(C_{h\alpha} - \frac{1.1C_{h\delta}}{T}\right)\left(1 + \frac{\mu}{n^2}\right)\frac{H'}{N'_m - N'_o}$$

where

H'_m = $N'_m - X_{cg}$ **(stick - free maneuver margin)**

G = **stick force / hinge moment = elevator gearing with a push force assumed positive**

η_t = **tail efficiency**

S_Θ = **elevator area**

C_Θ = **elevator chord**

The variation of stick force with an initially trimmed tab setting is proportional to the maneuver margin stick free.

$$\frac{dF_s}{dn} = -\frac{G\eta_t S_\theta C_\theta (W/S) C_{h\delta}}{C_{m\delta}} H'_m$$

can be used to determine the stick free maneuver point in flight by putting the aircraft in a steady level turn at some speed V with the elevator free. Let Fs be the constant stick force applied to hold the steady turn at load factor n. A measurement of the ratio (dFs/dn) at a given speed for a range of c.g. positions give the maneuver point as the intersection of the graph of (dFs/dn) vs X_{cg} with the X_{cg} axis.

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COURSE MATERIAL

SCHOOL OF MECHANICAL

DEPARTMENT OF AERONAUTICAL ENGINEERING

SAE1304 - AIRCRAFT STABILITY AND CONTROL

UNIT – V - MODERN CONTROL THEORY

Equations of motion

Equations of motion are equations that describe the behavior of a system in terms of its motion as a function of time (e.g., the motion of a particle under the influence of a force). Sometimes the term refers to the differential equations that the system satisfies (e.g., Newton's second law or Euler–Lagrange equations),

Aircraft dynamic modes

The dynamic stability of an aircraft is how the motion of an aircraft behaves after it has been disturbed from steady non-oscillating flight

Longitudinal modes

Oscillating motions can be described by two parameters, the period of time required for one complete oscillation, and the time required to damp to half-amplitude, or the time to double the amplitude for a dynamically unstable motion. The longitudinal motion consists of two distinct oscillations, a long-period oscillation called a phugoid mode and a short-period oscillation referred to as the short-period mode.

Phugoid (longer period) oscillations

The longer period mode, called the "phugoid mode" is the one in which there is a large-amplitude variation of air-speed, pitch angle, and altitude, but almost no angle-of-attack variation. The phugoid oscillation is really a slow interchange of kinetic energy (velocity) and potential energy (height) about some equilibrium energy level as the aircraft attempts to re-establish the equilibrium level-flight condition from which it had been disturbed. The motion is so slow that the effects of inertia forces and damping forces are very low. Although the damping is very weak, the period is so long that the pilot usually corrects for this motion without being aware that the oscillation even exists. Typically the period is 20–60 seconds.

Short period oscillations

With no special name, the shorter period mode is called simply the "short-period mode". The short-period mode is a usually heavily damped oscillation with a period of only a few seconds. The motion is a rapid pitching of the aircraft about the center of gravity. The period is so short that the speed does not have time to change, so the oscillation is essentially an angle-of-attack variation. The time to damp the amplitude to one-half of its value is usually on the order of 1 second. Ability to quickly self damp when the stick is briefly displaced is one of the many criteria for general aircraft certification.

Lateral-directional modes

"Lateral-directional" modes involve rolling motions and yawing motions. Motions in one of these axes almost always couples into the other so the modes are generally discussed as the "Lateral-Directional modes".[2]

There are three types of possible lateral-directional dynamic motion: roll subsidence mode, Dutch roll mode, and spiral mode.

Roll subsidence mode

Roll subsidence mode is simply the damping of rolling motion. There is no direct aerodynamic moment created tending to directly restore wings-level, i.e. there is no returning "spring force/moment" proportional to roll angle. However, there is a damping moment (proportional to roll rate) created by the slewing-about of long wings. This prevents large roll rates from building up when roll-control inputs are made or it damps the roll rate (not the angle) to zero when there are no roll-control inputs.

Roll mode can be improved by adding dihedral effects to the aircraft design, such as high wings, dihedral angles or sweep angles.

Spiral mode

If a spirally unstable aircraft, through the action of a gust or other disturbance, gets a small initial roll angle to the right, for example, a gentle sideslip to the right is produced. The sideslip causes a yawing moment to the right. If the dihedral stability is low, and yaw damping is small, the directional stability keeps turning the aircraft while the continuing bank angle maintains the sideslip and the yaw angle. This spiral gets continuously steeper and tighter until finally, if the motion is not checked, a steep, high-speed spiral dive results. The motion develops so gradually, however that it is usually corrected unconsciously by the pilot, who may not be aware that spiral instability exists. If the pilot cannot see the horizon, for instance because of clouds, he might not notice that he is slowly going into the spiral dive, which can lead into the graveyard spiral.

To be spirally stable, an aircraft must have some combination of a sufficiently large dihedral, which increases roll stability, and a sufficiently long vertical tail arm, which increases yaw damping. Increasing the vertical tail area then magnifies the degree of stability or instability.

The spiral dive should not be confused with a spin.

Detection

While descending turns are commonly performed by pilots as a standard flight maneuvers, the spiral dive is differentiated from a descending turn owing to its feature of accelerating speed. It is therefore an unstable flight condition, and pilots are trained to recognise its onset and to implement recovery procedures safely and immediately. Without intervention by the pilot, acceleration of the aircraft will lead to structural failure of the airframe, either as a result of excess aerodynamic loading or flight into terrain. Spiral dive training therefore revolves around pilot recognition and recovery.

Recovery

Spiral dive accidents are typically associated with visual flight (non-instrument flight) in conditions of poor visibility, where the pilot's reference to the visual natural horizon is effectively reduced, or prevented entirely, by such factors as cloud or darkness. The inherent danger of the spiral dive is that the condition, especially at onset, cannot be easily detected by the sensory mechanisms of the human body. The physical forces exerted on an airplane during a spiral dive are effectively balanced and the pilot cannot detect the banked attitude of the spiral descent. If the pilot detects acceleration, but fails to detect the banked attitude associated with the spiral descent, a mistaken attempt may be to recovery with mere back pressure (pitch-up inputs) on the control wheel. However, with the lift vector of the aircraft now directed to the centre of the spiral turn, this erred nose-up input simply tightens the spiral condition and increases the rate of acceleration and increases dangerous airframe loading. To successfully recover from a spiral dive, the lift vector must first be redirected upward (relative to the natural horizon) before backpressure is applied to the control column. Since the acceleration can be very rapid, recovery is dependent on the pilot's ability to quickly close the throttle (which is contributing to the acceleration), position the lift vector upward, relative to the Earth's surface before the dive recovery is implemented; any factor that would impede the pilot's external reference to the Earth's surface could delay or prevent recovery. The quick and efficient completion of these tasks is crucial as the aircraft can accelerate through maximum speed limits within only a few seconds, where the structural integrity of the airframe will be compromised.

For the purpose of flight training, instructors typically establish the aircraft in a descending turn with initially slow but steadily accelerating airspeed – the initial slow speed facilitates the potentially slow and sometimes erred response of student pilots. The cockpit controls are released by the instructor and the student is instructed to recover. It is not uncommon for a spiral dive to result from an unsuccessful attempt to enter a spin, but the extreme nose-down attitude of the aircraft during the spin-spiral transition makes this method of entry ineffective for training purposes as there is little room to permit student error or delay.

All spiral dive recoveries entail the same recovery sequence: first, the throttle must be immediately closed; second, the aircraft is rolled level with co-ordinate use of ailerons and rudder; and third, backpressure is exerted smoothly on the control wheel to recover from the dive.

Dutch roll

The second lateral motion is an oscillatory combined roll and yaw motion called Dutch roll, perhaps because of its similarity to an ice-skating motion of the same name made by Dutch skaters; the origin of the name is unclear. The Dutch roll may be described as a yaw and roll to the right, followed by a recovery towards the equilibrium condition, then an overshooting of this condition and a yaw and roll to the left, then back past the equilibrium attitude, and so on. The period is usually on the order of 3–15 seconds, but it can vary from a few seconds for light aircraft to a minute or more for airliners. Damping is increased by large directional stability and small dihedral and decreased by small directional stability and large dihedral. Although usually stable in a normal aircraft, the motion may be so slightly damped that the effect is very unpleasant and undesirable. In swept-back wing aircraft, the Dutch roll is solved by installing a yaw damper, in effect a special-purpose automatic pilot that damps out any yawing oscillation by applying rudder corrections. Some swept-wing aircraft have an unstable Dutch roll. If the Dutch roll is very lightly damped or unstable, the yaw damper becomes a safety requirement, rather than a pilot and passenger convenience. Dual yaw dampers are required and a failed yaw damper is cause for limiting flight to low altitudes, and possibly lower mach numbers, where the Dutch roll stability is improved.

Autorotation

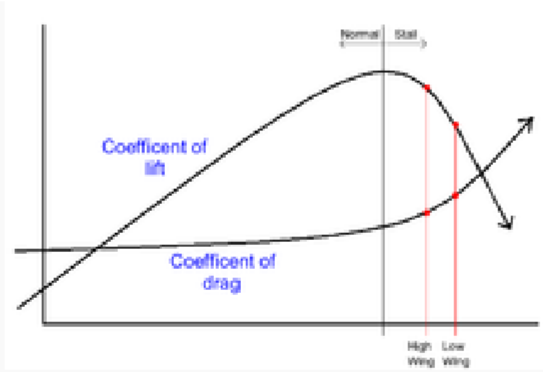
In aviation, **autorotation** refers to processes in both fixed-wing and rotary-wing aircraft. The term means significantly different things in each context.

For fixed-wing aircraft, autorotation refers to the tendency of an aircraft in or near a stall to roll spontaneously to the right or left, leading to a spin (a state of continuous autorotation).

In helicopters and auto gyros, autorotation refers to generation of lift by the main rotor when it is not being driven by an engine. Should an engine fail, a helicopter is able to use autorotation lift to slow its descent and land in a controlled manner.

Auto gyros' main rotors are unpowered, so they rely continuously on autorotation for lift.

Autorotation in fixed-wing aircraft



A typical graph of lift coefficient and drag coefficient versus angle of attack, At any angle of attack greater than the stalling angle an increase in angle of attack causes a reduction in lift coefficient, and a decrease in angle of attack causes an increase in lift coefficient.

When the angle of attack is less than the stalling angle any increase in angle of attack causes an increase in lift coefficient that causes the wing to rise. As the wing rises the angle of attack decreases, which tends to restore the wing to its original angle of attack. Conversely any decrease in angle of attack causes a decrease in lift coefficient which causes the wing to descend. As the wing descends, the angle of attack increases, which tends to restore the wing to its original angle of attack. For this reason the angle of attack is stable when it is less than the stalling angle. The aircraft displays damping in roll.

When the wing is stalled and the angle of attack is greater than the stalling angle any increase in angle of attack causes a decrease in lift coefficient that causes the wing to descend. As the wing descends the angle of attack increases, which causes the lift coefficient to decrease and the angle of attack to increase, Conversely any

decrease in angle of attack causes an increase in lift coefficient that causes the wing to rise. As the wing rises the angle of attack decreases and causes the lift coefficient to increase further towards the maximum lift coefficient. For this reason the angle of attack is unstable when it is greater than the stalling angle. Any disturbance of the angle of attack on one wing will cause the whole wing to roll spontaneously and continuously.

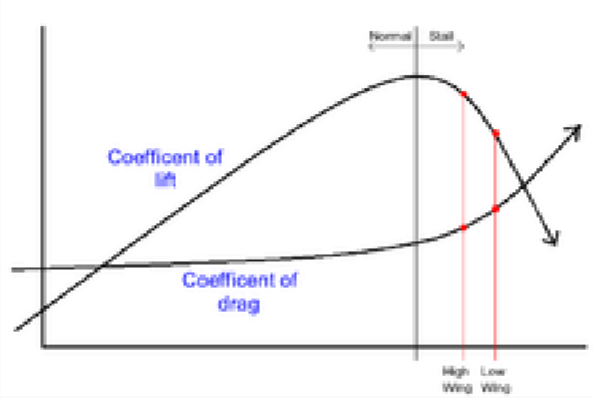
When the angle of attack on the wing of an aircraft reaches the stalling angle the aircraft is at risk of autorotation. This will eventually develop into a spin if the pilot does not take corrective action.

Spin (flight)

In aviation, a spin is an aggravated stall resulting in autorotation about the spin axis wherein the aircraft follows a corkscrew path. Spins can be entered unintentionally or intentionally, from any flight attitude and from practically any airspeed—all that is required is sufficient yaw rate while an aircraft is stalled. In either case, however, a specific and often counterintuitive set of actions may be needed for an effective recovery to be made. If the aircraft exceeds published limitations regarding spins, or is loaded improperly, or if the pilot uses incorrect technique to recover, the spin can lead to a crash. In a spin, one or both wings are in a stalled condition, if both are stalled one wing will be in a deeper stall condition than the other.[1] This causes the aircraft to auto rotate (yaw) towards the deeper-stalled wing due to its higher drag. Spins are also characterized by high angle of attack, low airspeed, and high rate of descent.

Spins differ from spiral dives which are characterized by low angle of attack and high airspeed. A spiral dive is not a type of spin because neither wing is stalled. In a spiral dive the airplane will respond conventionally to the pilot's inputs to the flight controls.

How a spin occ



Certificated, light, single-engine airplanes must meet specific criteria regarding stall and spin behavior. Many types of airplane will only spin if the pilot simultaneously yaws and stalls the airplane (intentionally or unintentionally). Under these circumstances, one wing tends to stall more deeply than the other. The wing that stalls first will drop, increasing its angle of attack and deepening the stall. Both wings must be stalled for a spin to occur. The other wing will rise, decreasing its angle of attack, and the aircraft will yaw towards the more deeply-stalled wing. The difference in lift between the two wings causes the aircraft to roll, and the difference in drag causes the aircraft to yaw.

One common scenario that can lead to an unintentional spin is an uncoordinated turn towards the runway during the landing sequence. A pilot who is overshooting the turn to final approach may be tempted to apply rudder to increase the rate of turn. The result is twofold: the nose of the airplane drops below the horizon and the bank angle increases. Reacting to these unintended changes, the pilot may then begin to pull the elevator control aft (thus increasing the angle of attack) while applying opposite aileron to decrease bank angle. Taken to its extreme, this can result in an uncoordinated turn with sufficient angle of attack to cause the aircraft to stall. This is called across-control stall, and is very dangerous if it happens at low altitude where the pilot has little time to recover.

Autopilot

An autopilot is a system used to control the trajectory of a vehicle without constant 'hands-on' control by a human operator being required. Autopilots do not replace a human operator, but assist them in controlling the vehicle, allowing them to focus on broader aspects of operation, such as monitoring the trajectory, weather and systems.Autopilots are used in aircraft, boats (known as self-steering gear), spacecraft, missiles, and others. Autopilots have evolved significantly over time, from early autopilots that merely held an attitude to modern autopilots capable of performing automated landings under the supervision of a pilot.

The autopilot system on airplanes is sometimes colloquially referred to as "George"

First autopilots

Gyroscopic autopilot

In the early days of aviation, aircraft required the continuous attention of a pilot in order to fly safely. As aircraft range increased allowing flights of many hours, the constant attention led to serious fatigue. An autopilot is designed to perform some of the tasks of the pilot.

The first aircraft autopilot was developed by Sperry Corporation in 1912. The autopilot connected a gyroscopic heading indicator and attitude indicator to hydraulically operated elevators and rudder. (Ailerons were not connected as wing dihedral was counted upon to produce the necessary roll stability.) It permitted the aircraft to fly straight and level on a compass course without a pilot's attention, greatly reducing the pilot's workload. Lawrence Sperry (the son of famous inventor Elmer Sperry) demonstrated it in 1914 at an aviation safety contest held in Paris. At the contest, Sperry demonstrated the credibility of the invention by flying the aircraft with his hands away from the controls and visible to onlookers of the contest. Elmer Sperry Jr., the son of Lawrence Sperry, and Capt Shiras continued work after the war on the same autopilot, and in 1930 they tested a more compact and reliable autopilot which kept a US Army Air Corps aircraft on a true heading and altitude for three hours.

In 1930, the Royal Aircraft Establishment in England developed an autopilot called a pilots' assister that used a pneumatically-spun gyroscope to move the flight controls.

Further development of the autopilot was performed, such as improved control algorithms and hydraulic servomechanisms. Also, inclusion of additional instrumentation such as the radio-navigation aids made it possible to fly during night and in bad weather. In 1947 a US Air Force C-54 made a transatlantic flight, including takeoff and landing, completely under the control of an autopilot.

In the early 1920s, the Standard Oil tanker J.A. Moffet became the first ship to use an autopilot.

Modern autopilots

The modern flight control unit of an Airbus A340

Not all of the passenger aircraft flying today have an autopilot system. Older and smaller general aviation aircraft especially are still hand-flown, and even small airliners with fewer than twenty seats may also be without an autopilot as they are used on short-duration flights with two pilots. The installation of autopilots in aircraft with more than twenty seats is generally made mandatory by international aviation regulations. There are three levels of control in autopilots for smaller aircraft. A single-axis autopilot controls an aircraft in the roll axis only; such autopilots are also known colloquially as "wing levellers," reflecting their limitations. A two-axis autopilot controls an aircraft in the pitch axis as well as roll, and may be little more than a "wing leveller" with limited pitch oscillation-correcting ability; or it may receive inputs from on-board radio navigation systems to provide true automatic flight guidance once the aircraft has taken off until shortly before landing; or its capabilities may lie somewhere between these two extremes. A three-axis autopilot adds control in the yaw axis and is not required in many small aircraft.

Autopilots in modern complex aircraft are three-axis and generally divide a flight into taxi, takeoff, climb, cruise (level flight), descent, approach, and landing phases. Autopilots exist that automate all of these flight phases except taxi and takeoff. An autopilot-controlled landing on a runway and controlling the aircraft on rollout (i.e. keeping it on the centre of the runway) is known as a CAT IIIb landing or Autoland, available on many major airports' runways today, especially at airports subject to adverse weather phenomena such as fog. Landing, rollout, and taxi control to the aircraft parking position is known as CAT IIIc. This is not used to date, but may be used in the future. An autopilot is often an integral component of a Flight Management System.

Modern autopilots use computer software to control the aircraft. The software reads the aircraft's current position, and then controls a Flight Control System to guide the aircraft. In such a system, besides classic flight controls, many autopilots incorporate thrust control capabilities that can control throttles to optimize the airspeed, and move fuel to different tanks to balance the aircraft in an optimal attitude in the air. Although autopilots handle new or dangerous situations inflexibly, they generally fly an aircraft with lower fuel consumption than a human pilot.

The autopilot in a modern large aircraft typically reads its position and the aircraft's attitude from an inertial guidance system. Inertial guidance systems accumulate errors over time. They will incorporate error reduction systems such as the carousel system that rotates once a minute so that any errors are dissipated in different directions and have an overall nulling effect. Error in gyroscopes is known as drift. This is due to physical properties within the system, be it mechanical or laser guided, that corrupt positional data. The disagreements between the two are resolved with digital signal processing, most often a six-dimensional Kalman filter. The six dimensions are usually roll, pitch, yaw, altitude, latitude, and longitude. Aircraft may fly routes that have a required performance factor, therefore the amount of error or actual performance factor must be monitored in order to fly those particular routes. The longer the flight, the more error accumulates within the system. Radio aids such as DME, DME updates, and GPS may be used to correct the aircraft position.

A midway between fully automated flight and manual flying is Control Wheel Steering (CWS). Although going out of fashion in modern airliners as a stand-alone option, CWS is still a function on many aircraft today. Generally, an autopilot that is CWS equipped, has three positions being off, CWS and CMD. In CMD (Command) mode the autopilot has full control of the aircraft, and receives its input from either the heading /altitude setting, radio and navaids or the FMS (Flight Management System). In CWS mode, the pilot controls the autopilot through inputs on the yoke or the stick. These inputs are translated to a specific heading and attitude, which the autopilot will then hold until instructed to do otherwise. This provides stability in pitch and roll. Some aircraft employ a form of CWS even in manual mode, such as the MD-11 which uses a constant CWS in roll. In many ways, a modern Airbus fly-by-wire aircraft in Normal Law is always in CWS mode. The major difference is that in this system the limitations of the aircraft are guarded by the Flight Computer, and the pilot can not steer the aircraft past these limits.

Computer system details

The hardware of an autopilot varies from implementation to implementation, but is generally designed with redundancy and reliability as foremost considerations. For example, the Rockwell Collins AFDS-770 Autopilot Flight Director System used on the Boeing 777 uses triplicated FCP-2002 microprocessors which have been formally verified and are fabricated in a radiation resistant process.

Software and hardware in an autopilot is tightly controlled, and extensive test procedures are put in place.

Some autopilots also use design diversity. In this safety feature, critical software processes will not only run on separate computers and possibly even using different architectures, but each computer will run software created by different engineering teams, often being programmed in different programming languages. It is generally considered unlikely that different engineering teams will make the same mistakes. As the software becomes more expensive and complex, design diversity is becoming less common because fewer engineering companies can afford it. The flight control computers on the Space Shuttle used this design: there were five computers, four of which redundantly ran identical software, and a fifth backup running software that was developed independently. The software on the fifth system provided only the basic functions needed to fly the Shuttle, further reducing any possible commonality with the software running on the four primary systems.

Stability augmentation systems

A stability augmentation system (SAS) is another type of automatic flight control system; however, instead of maintaining the aircraft on a predetermined attitude or flight path, the SAS will actuate the aircraft flight controls to dampen out aircraft buffeting regardless of the attitude or flight path. SAS can automatically stabilize the aircraft in one or more axes. The most common type of SAS is the yaw damper which is used to eliminate the Dutch roll tendency of swept-wing aircraft. Some yaw dampers are integral to the autopilot system while others are stand-alone systems.

Yaw dampers usually consist of a yaw rate sensor (either a gyroscope or angular accelerometer), a computer/amplifier and a servo actuator. The yaw damper uses yaw rate sensor to sense when the aircraft begins a Dutch Roll. A computer processes the signals from the yaw rate sensor to determine the amount of rudder movement that is required to dampen out the Dutch roll. The computer then commands the servo actuator to move the rudder that amount. The Dutch roll is dampened out and the aircraft becomes stable about the yaw axis. Because Dutch roll is an instability that is inherent to all swept-wing aircraft, most swept-wing aircraft have some sort of yaw damper system installed.

There are two types of yaw dampers: series yaw dampers and parallel yaw dampers. The servo actuator of a series yaw damper will actuate the rudder independently of the rudder pedals while the servo actuator of a parallel yaw damper is clutched to the rudder control quadrant and will result in pedal movement when the system commands the rudder to move.

Some aircraft have stability augmentation systems that will stabilize the aircraft in more than a single axis. B-52s, for example, require both pitch and yaw SAS in order to provide a stable bombing platform. Many helicopters have pitch, roll and yaw SAS systems. Pitch and roll SAS systems operate much the same way as the yaw damper described above; however, instead of dampening out Dutch roll, they will dampen pitch and roll oscillations or buffeting to improve the overall stability of the aircraft.

Autopilot for ILS landings

Instrument-aided landings are defined in categories by the International Civil Aviation Organization, or ICAO. These are dependent upon the required visibility level and the degree to which the landing can be conducted automatically without input by the pilot.

CAT I - This category permits pilots to land with a decision height of 200 ft (61 m) and a forward visibility or Runway Visual Range (RVR) of 550 m. Autopilots are not required.

CAT II - This category permits pilots to land with a decision height between 200 ft and 100 ft (≈ 30 m) and a RVR of 300 m. Autopilots have a fail passive requirement.

CAT IIIa -This category permits pilots to land with a decision height as low as 50 ft (15 m) and a RVR of 200 m. It needs a fail-passive autopilot. There must be only a 10−6 probability of landing outside the prescribed area.

CAT IIIb - As IIIa but with the addition of automatic roll out after touchdown incorporated with the pilot taking control some distance along the runway. This category permits pilots to land with a decision height less than 50 feet or no decision height and a forward visibility of 250 ft (76 m, compare this to aircraft size, some of which are now over 70 m long) or 300 ft (91 m) in the United States. For a landing-without-decision aid, a fail-operational autopilot is needed. For this category some form of runway guidance system is needed: at least fail-passive but it needs to be fail-operational for landing without decision height or for RVR below 100 m.

CAT IIIc - As IIIb but without decision height or visibility minimums, also known as "zero-zero"

Fail-passive autopilot: in case of failure, the aircraft stays in a controllable position and the pilot can take control of it to go around or finish landing. It is usually a dual-channel system.

Fail-operational autopilot: in case of a failure below alert height, the approach, flare and landing can still be completed automatically. It is usually a triple-channel system or dual-dual system.

Radio-controlled models

In radio-controlled modelling, and especially RC aircraft and helicopters, an autopilot is usually a set of extra hardware and software that deals with pre-programming the model's flight.