Aircraft Performance

This workbook provides an understanding of how a heavier than air vehicle is able to generate lift and sustain flight.

This understanding is generated by understanding the atmosphere in which an aircraft flies (Section 1 The Atmosphere). Next, the parameters used to represent the 3-dimensional state of the aircraft are defined. This is followed by a discussion of Aerodynamics, Airfoils, Lift and Drag.

1) THE ATMOSPHERE

2) MEASURING SPEEDS AND ALTITUDES

3) AERODYNAMICS, AIRFOILS, LIFT, DRAG and THRUST

4) AIRCRAFT PERFORMANCE

5) ASSESSMENT (UNIT EXAM)

Learning Objectives:

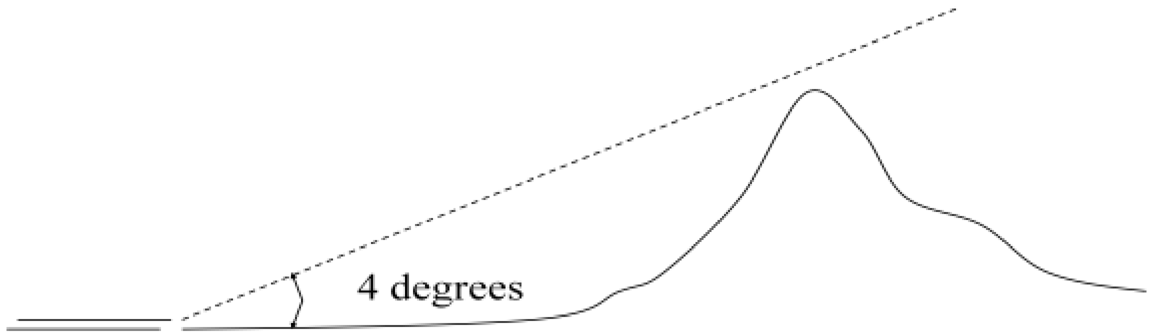
(1) Facts: The student will know and understand all the following facts:
    a. Atmospheric properties (Ambient and Static Pressure, Temperature Density, …)
    b. Altitude and Airspeeds
    c. Wings and Airfoils
    d. Forces: Lift, Drag, Thrust
    e. Aircraft Performance Model for Climb, Cruise and Descent Operations (Vertical Plane)
    f. Aircraft Performance Model for Runway Operations
    g. Aircraft Performance Model for Climb, Cruise and Descent Operations (Lateral Plane)

(2) Skills: The student will be able to build, run, and analyze aerodynamic models to solve real world problems
    a. Obstacle clearance for departure (vertical)
    b. Obstacle clearance for departure (lateral)\

Problem #1:

You are a Flight Operations Engineer at Eventually Airways (motto: We will get you there …eventually).

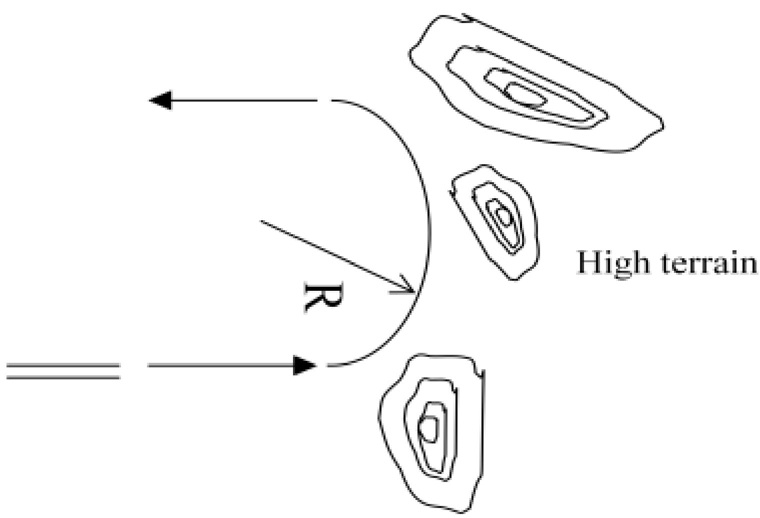
Your company marketing department has decided to provide service in the winter ski season to a popular ski village located in the valley between two steep peaks. You need to determine the type of aircraft that can be used in all weather conditions to climb-out in excess of 4 degrees to clear the peak located in the published departure procedure.



Problem #2:

You are a Flight Operations Engineer at Eventually Airways (motto: We will get you there …eventually).

Your company marketing department has decided to provide service in the winter ski season to a popular ski village located in the valley between two steep peaks. The aircraft selected for the route is not able to clear the mountain in the departure. You need to determine whether the type of aircraft selected can make a 180º turn of <u>no</u> more than 4nm turn radius to avoid high terrain under specific conditions. What bank angle is sufficient?



SECTION 1 THE ATMOSPHERE

For the purposed of studying the performance and dynamics of transport aircraft, the portion of the Earth's atmosphere that will be considered is between 100ft below Sea Level to 60,000 ft above Sea Level.

The atmosphere in which air transports operate is characterized by the following parameters:

1. Altitude
2. Ambient Temperature (T_{amb}) and Temperature Ratio
3. Ambient Pressure
4. Ambient Density (ρ_{amb}) and Density Ratio (σ)

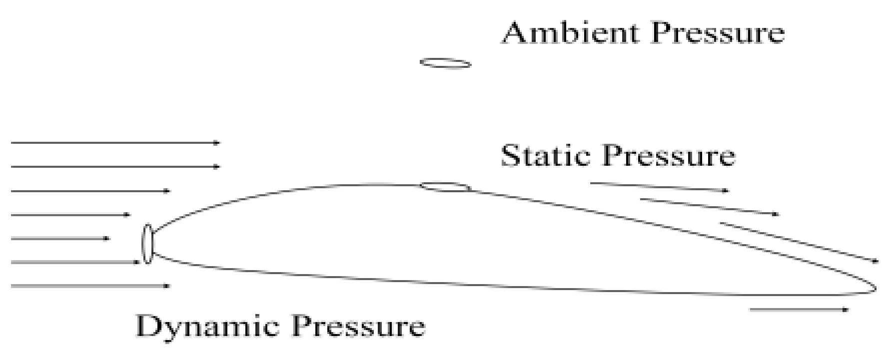
Terminology and Fundamental Concepts

Ambient, Static, Dynamic Measurements

An ambient measurement is a measurement taken at a distance from an object (i.e. airfoil) in the air flow.

A static measurement is a measurement taken on the surface of the object parallel to the air flow. This measurement is affected by the friction effects.

A dynamic measurement is a measurement taken on the surface of the object perpendicular to the air flow. This measurement is affected by the compressibility effects.



International Standard Atmosphere (ISA)

An air-mass (i.e. a large pocket of air) in the atmosphere can be defined by the following parameters:

- Pressure (lbs/ft^2)
- Density (slugs/ft^3)
- Temperature (degrees F or degrees C or degrees K or degrees R)
- Gravity (ft/sec^2)
- Gas Constant (ft/degrees R)

To simplify the mathematics, these parameters are represented as a ratio relative to a reference point at Sea Level at N45°32’ 40”

- Pressure (2,116.22 lbs/ft^2)
- Density (0.002377 slugs/ft^3)
- Temperature (59 degrees F or 15 degrees C or 288.16 degrees K or 518.688 degrees R)
- Gravity (32.1741 ft/sec^2)
- Gas Constant (53.35 ft/degrees R)

The **International Standard Atmosphere** (ISA) is an atmospheric model of how the pressure, temperature, density, and viscosity of the Earth's atmosphere change over a wide range of altitudes.

The International Standard Atmosphere (ISA) sets the base standards as follows:

Parameter	Symbol	Value	Units
Pressure	P_o	2,116.22	Lb/ft^2
Density	ρ_o	0.002377	Slugs/ft^3
Temperature	To	59^{o}F = 15^{o}C	Farenheit or Celsius
Gravity	Go	32.1741	ft/sec2
Gas Constant	R	53.35	Ft/^{o}R

Ratios:

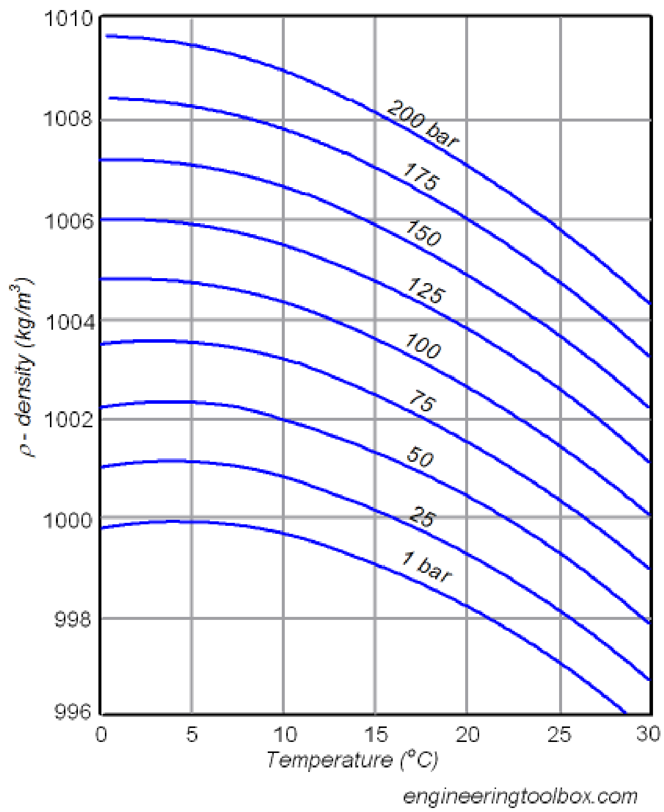
Temperature Ratio = $T_{ambient}$ / $T_{sea\ level}$ = θ = 1 – (6.8753 * 10-6) Pressure Altitude (pressure Altitude < 36,089 ft)

Temperature Ratio = $T_{ambient}$ / $T_{sea\ level}$ = θ = 0.7519 (pressure Altitude $\geq$ 36,089 ft)

Pressure Ratio = $P_{ambient}$ / $P_{sea\ level}$ = δ = 1 – (6.88 * 10-6 * Pressure Altitude)$^{5.26}$ (pressure Altitude < 36,089 ft)

Pressure Ratio =$P_{ambient}$ / $P_{sea\ level}$ = δ = 0.22336 EXP((36,089 – Pressure Alt)/20,805.7) (pressure Altitude $\geq$ 36,089 ft)

Density Ratio = $\rho_{ambient}$ / $\rho_{sea\ level}$ = σ = δ/ θ (from thermodynamics)



Altitude

The height above a reference point is known as Altitude.

There are four measures of altitude useful in aviation.

1) Geometric Altitude – the physical distance of the aircraft to a reference point (e.g. mountain top). This measure is independent of atmospheric effects and can be measured by radar (or another sensor that is not affected by the atmosphere).

RADAR ALTIMETER:

As the name implies, radar (radio detection and ranging) is the underpinning principle of the system. Radiowaves are transmitted towards the ground and the time it takes them to be reflected back and return to the aircraft is timed. Because speed, distance and time are all related to each other, the distance from the surface providing the reflection can be calculated as the speed of the radiowave and the time it takes to travel are known quantities.

Radar Altimeter was invented by Lloyd Espenschied in 1924. It took 14 years before Bell Labs was able to put Espenschied's device in a form that was adaptable for aircraft use.

Radio altimeter RT unit.

Radio altimeter with antenna on underside of fuselage.

2) Pressure Altitude – is a measure of altitude computed by sensing the Static Pressure. The actual pressure altitude is computed by comparing the Static Pressure measured with a reference pressure known as the Barometric Pressure. The Barometric Pressure at a hypothetical reference point known as 'mean sea level' that has an ambient temperature of 15 C and is an ambient pressure of 29.92 inches of mercury (inHg).

Pressure altimeter

A pressure altimeter (also called barometric altimeter) is the altimeter found in most aircraft. In it, an aneroid barometer measures the atmospheric pressure from a static port outside the aircraft. Air pressure decreases with an increase of altitude—approximately 100 hectopascals per 800 meters or one inch of mercury per 1000 feet near sea level.

The altimeter is calibrated to show the pressure directly as an altitude above mean sea level, in accordance with a mathematical model defined by the International Standard Atmosphere (ISA). Modern aircraft use a "sensitive altimeter" which has a primary needle that makes multiple revolutions, and one or more secondary needles that show the number of revolutions, similar to a clock face. In other words, each needle points to a different digit of the current altitude measurement.

3) Density Altitude – is a measure of altitude computed by sensing the Density. Density Altitude can be computed by sensing the density and comparing it to the standard altitude-density table made available by the International Standard Atmosphere (ISA). Density Altitude is not used for flying the aircraft, but is an important parameter in assessing the aircraft performance (e.g. climb rate). Density Altitude is used on missiles (since sensors are cheap and accuracy due to non-standard atmosphere is not important).

4) Geopotentail Altitude – is a measure of altitude relative to the Center of the Earth. This altitude represents the altitude with the same gravitational potential.

EXERCISE: Draw a diagram showing the 4 types of altitude measurement. The diagram should include an aircraft, terrain/ocean, mean-sea-level defined by 29.92 in Hg, center of Earth, and ISA density altitude scale.

Temperature

Ambient Temperature is a measure of the temperature of the air in the region surrounding the aircraft. This pocket of air is not touching the skin of the aircraft or affected by the flow of air around the skin of the aircraft.

The Ambient Temperature varies with altitude as follows:

Below 36,089 feet, the Ambient Temperature (^oR) = -3.566^o * (Altitude/1000)

Above 36,089 feet, the Ambient Temperature (^oR) = 389.988^o

A simplified rule-of-thumb is that ambient temperature drops 1^oC for every 1000 ft increase in altitude

Temperature Ratio is useful parameter that represents the Ambient Temperature at an altitude relative to the Temperature at Sea Level

Below 36,089 feet, Tamb / TSeaLevel = $\theta = 1 - (6.8753 \times 10^{-6})$ * Pressure Altitude

Above 36,089 feet, Tamb / TSeaLevel = $\theta = 0.7519$

Delta ISA is used to account for differences between the standard ambient temperature and the actual air temperature. For example, you are scheduled to fly through an airmass that is $+40^o$ than standard temperature.

$$\Theta = (T_{amb} + \Delta ISA) / (T_{SeaLevel} + \Delta ISA)$$

Static Temperature is the measure of the temperature at the surface of the aircraft. This pocket of air is affected by the flow of air around the skin of the aircraft.

EXERCISE: Plot Standard Ambient Temperature from -1000ft to 43,000ft. Show Altitude on the y-axis. Show Temperature on the x-axis. Convert temperature into degrees Celsius.

Pressure

Ambient Pressure is a measure of the pressure of the air in the region surrounding the aircraft. This pocket of air is not touching the skin of the aircraft or affected by the flow of air around the skin of the aircraft.

Pressure Ratio is useful parameter that represents the Ambient Pressure at an altitude relative to the Pressure at Sea Level

$$\delta = P_{amb} / P_{SeaLevel}$$

Below 36,089 feet, the Ambient Pressure (lb/ft^2) = δ = $(1 - {}^{o} * 6.88 X 10^{-6} * \text{Pressure Altitude})^{5.26}$

Above 36,089 feet, the Ambient Pressure (lb/ft^2) = δ = 0.223360 $e^{((36.089 - \text{Pressure Altitude})/20.8057)}$

Static Pressure is the measure of the pressure at the surface of the aircraft. This pocket of air is affected by the flow of air around the skin of the aircraft.

Due to practical considerations of mounting sensors on an aircraft, aircraft speed and altitude are measured through ports on the surface of the vehicle. To compute the ambient pressure, corrections to the static pressure readings must be made. These corrections include the *Compressibility Effect*.

EXERCISE: Plot Standard Ambient Pressure from -1000ft to 43,000ft. Show Altitude on the y-axis. Pressure on the x-axis.

Density

Ambient Density(ρ) is a measure of the density of the air in the region surrounding the aircraft. This pocket of air is not touching the skin of the aircraft or affected by the flow of air around the skin of the aircraft.

Density Ratio (σ) is useful parameter that represents the Ambient Density at an altitude relative to the Density at Sea Level

σ ($Slugs/ft^3$) = $\rho_{amb} / \rho_{SeaLevel}$

From thermodynamics, Density is Pressure divided by Temperature. So

$\sigma = \delta/\theta$

SECTION 2 MEASURING SPEED AND ALTITUDE

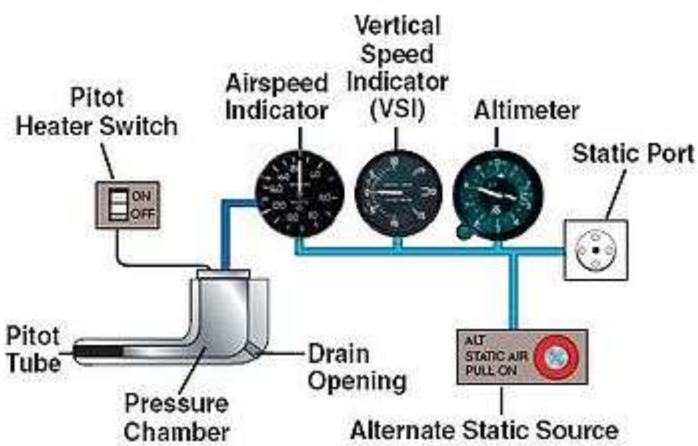
Aircraft speed and altitude are the key parameters in the assessment of the energy and dynamic state of the aircraft.

Measurement of altitude and speed is straightforward when the medium in which the vehicle operates (i.e. airmass) is fixed relative to the reference point (i.e. earth). This situation becomes more complex when the airmass moves with respect to the earth (i.e. wind) and deforms (i.e. compresses).

A complete system is illustrated in the Figure below. The Pitot tube collects dynamic pressure and the Static port collects static pressure.

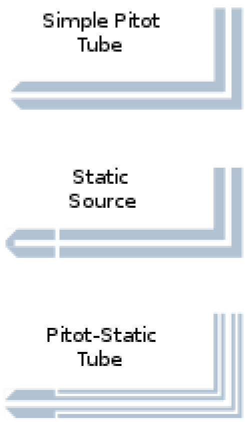
Static pressure is used by the Altimeter to compute altitude, and the Vertical Speed Indicator (VSI) to compute vertical speed.

Dynamic Pressure and Static Pressure is used to compute Airspeed.



Pitot-static systems

A **pitot-static system** is a system of pressure-sensitive instruments that is most often used in aviation to determine an aircraft's airspeed, Mach number, altitude, and altitude trend (i.e. verstical speed). The static port measures static pressure. The pitot tube measures dynamic pressure. Some designs combine both the static and dynamic ports into a single unit.



Pitot pressure

The pitot pressure is obtained from the pitot tube. The pitot pressure is a measure of ram air pressure (the air pressure created by vehicle motion or the air ramming into the tube), which, under ideal conditions, is equal to stagnation pressure, also called total pressure.

The pitot tube is most often located on the wing or front section of an aircraft, facing forward, where its opening is exposed to the relative wind. By situating the pitot tube in such a location, the ram air pressure is more accurately measured since it will be less distorted by the aircraft's structure. When airspeed increases, the ram air pressure is increased, which can be translated by the airspeed indicator.

Static pressure

The static pressure is obtained through a static port. The static port is most often a flush-mounted hole on the fuselage of an aircraft, and is located where it can access the air flow in a relatively undisturbed area.

Some aircraft may have a single static port, while others may have more than one. In situations where an aircraft has more than one static port, there is usually one located on each side of the fuselage. With this positioning, an average pressure can be taken, which allows for more accurate readings in specific flight situations. An alternative static port may be located inside the cabin of the aircraft as a backup for when the external static port(s) are blocked.

A pitot-static tube effectively integrates the static ports into the pitot probe. It incorporates a second coaxial tube (or tubes) with pressure sampling holes on the sides of the probe, outside the direct airflow, to measure the static pressure.

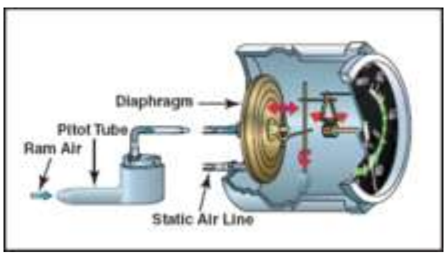
Pitot-static instruments

The pitot-static system obtains pressures for interpretations by the pitot-static instruments. The explanations below explain traditional, mechanical instruments.

Airspeed indicator

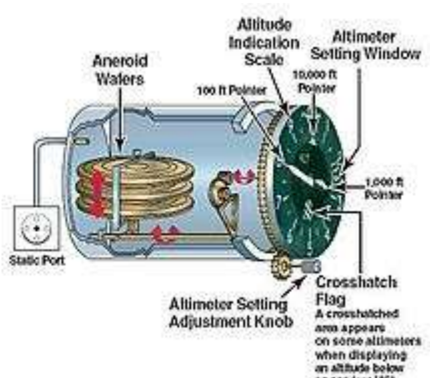
The airspeed indicator is connected to both the pitot and static pressure sources. The difference between the pitot pressure and the static pressure is called "impact pressure". The greater the impact pressure, the higher the airspeed reported.

A traditional mechanical airspeed indicator (see Figure below) contains a pressure diaphragm that is connected to the pitot tube. The case around the diaphragm is airtight and is vented to the static port. The higher the speed, the higher the ram pressure, the more pressure exerted on the diaphragm, and the larger the needle movement through the mechanical linkage.



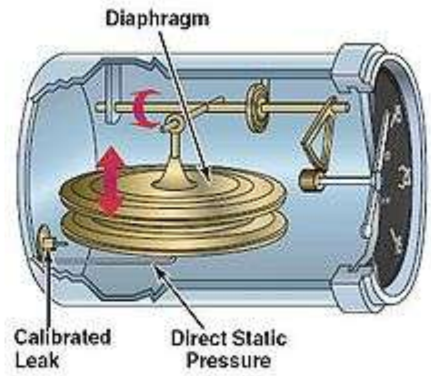
Altimeter

The pressure altimeter, also known as the barometric altimeter, is used to determine changes in air pressure that occur as the aircraft's altitude changes. Pressure altimeters must be calibrated prior to flight to register the pressure as an altitude above sea level. The instrument case of the altimeter is airtight and has a vent to the static port. Inside the instrument, there is a sealed aneroid barometer. As pressure in the case decreases, the internal barometer expands, which is mechanically translated into a determination of altitude. The reverse is true when descending from higher to lower altitudes.



Machmeter

Aircraft designed to operate at transonic or supersonic speeds will incorporate a machmeter. The machmeter is used to show the ratio of true airspeed in relation to the speed of sound. Most supersonic aircraft are limited as to the maximum Mach number they can fly, which is known as the "Mach limit". The Mach number is displayed on a machmeter as a decimal fraction.



Vertical airspeed indicator

The vertical speed indicator (VSI) or the vertical velocity indicator (VVI), is the pitot-static instrument used to determine whether or not an aircraft is flying in level flight. The vertical airspeed specifically shows the rate of climb or the rate of descent, which is measured in feet per minute or meters per second.

The vertical airspeed is measured through a mechanical linkage to a diaphragm located within the instrument. The area surrounding the diaphragm is vented to the static port through a calibrated leak (which also may be known as a "restricted diffuser"). When the aircraft begins to increase altitude, the diaphragm will begin to contract at a rate faster than that of the calibrated leak, causing the needle to show a positive vertical speed. The reverse of this situation is true when an aircraft is descending The calibrated leak varies from model to model, but the average time for the diaphragm to equalize pressure is between 6 and 9 seconds.

Pitot-static errors

The pitot tube is susceptible to becoming clogged by ice, water, insects or some other obstruction. For this reason, aviation regulatory agencies such as the U.S. Federal Aviation Administration (FAA) recommend that the pitot tube be checked for obstructions prior to any flight.

To prevent icing, many pitot tubes are equipped with a heating element. A heated pitot tube is required in all aircraft certificated for instrument flight.

A blocked pitot tube is a pitot-static problem that will only affect airspeed indicators.

A blocked pitot tube will cause the airspeed indicator to register an increase in airspeed when the aircraft climbs, even though indicated airspeed is constant. This is caused by the pressure in the pitot system remaining constant when the atmospheric pressure (and static pressure) are decreasing. In reverse, the airspeed indicator will show a decrease in airspeed when the aircraft descends.

A blocked static port is a more serious situation because it affects all pitot-static instruments. One of the most common causes of a blocked static port is airframe icing. A blocked static port will cause the altimeter to freeze at a constant value, the altitude at which the static port became blocked. The vertical speed indicator will become frozen at zero and will not change at all, even if vertical airspeed increases or decreases. The airspeed indicator will reverse the error that occurs with a clogged pitot tube and cause the airspeed be read less than it actually is as the aircraft climbs. When the aircraft is descending, the airspeed will be over-reported. In most aircraft with unpressurized cabins, an alternative static source is available and can be toggled from within the cockpit of the airplane.

Density errors affect instruments reporting airspeed and altitude. This type of error is caused by variations of pressure and temperature in the atmosphere.

A *compressibility error* can arise because the impact pressure will cause the air to compress in the pitot tube. At standard sea level pressure altitude the calibration equation (see calibrated airspeed) correctly accounts for the compression so there is no compressibility error at sea level. At higher altitudes the compression is not correctly accounted for and will cause the instrument to read greater than equivalent airspeed. A correction may be obtained from a chart. Compressibility error becomes significant at altitudes above 10,000 feet (3,000 m) and at airspeeds greater than 200 knots (370 km/h).

Hysteresis is an error that is caused by mechanical properties of the aneroid capsules located within the instruments. These capsules, used to determine pressure differences, have physical properties that resist change by retaining a given shape, even though the external forces may have changed.

Reversal errors are caused by a false static pressure reading. This false reading may be caused by abnormally large changes in an aircraft's pitch. A large change in pitch will cause a momentary showing of movement in the opposite direction. Reversal errors primarily affect altimeters and vertical speed indicators.

A *position error* is produced by the aircraft's static pressure being different from the air pressure remote from the aircraft. This error is caused by the air flowing past the static port at a speed different from the aircraft's true airspeed. Position errors may provide positive or negative errors, depending on one of several factors. These factors include airspeed, angle of attack, aircraft weight, acceleration, aircraft configuration, and in the case of helicopters, rotor downwash. There are two categories of position errors, which are "fixed errors" and "variable errors". Fixed errors are defined as errors which are specific to a particular make of aircraft. Variable errors are caused by external factors such as deformed panels obstructing the flow of air, or particular situations which may overstress the aircraft.

Final Note:

Modern aircraft use air data computers (ADC) to calculate airspeed, rate of climb, altitude and mach number. The ADC's replace the diaphragms with electrnic sensors. Two (or three) ADCs receive total and static pressure from independent pitot tubes and static ports, and the aircraft's flight data computer compares the information from each computers and checks one against the other. There are also "standby instruments", which are back-up pneumatic instruments employed in the case of problems with the primary instruments.

Air France Flight 447

NOVA video:
http://www.pbs.org/wgbh/nova/space/crash-flight-447.html

Pitot 25 minutes
Lift 35 minutes
Power indication 43 minutes
Stall recovery 45 minutes

Official Accident Report:
http://www.bea.aero/docspa/2009/f-cp090601e3.en/pdf/f-cp090601e3.en.pdf

On 31 May 2009, flight AF447 took off from Rio de Janeiro Galeão airport bound for Paris Charles de Gaulle. The airplane was in contact with the Brazilian ATLANTICO ATC on the INTOL – SALPU – ORARO - TASIL route at FL350. At around 2 h 02, the Captain left the cockpit. At around 2 h 08, the crew made a course change of about ten degrees to the left, probably to avoid echoes detected by the weather radar.

At 2 h 10 min 05, likely following the **obstruction of the Pitot probes in an ice crystal environment**, the speed indications became erroneous and the automatic systems disconnected. The airplane's flight path was not brought under control by the two copilots, who were rejoined shortly after by the Captain. The airplane went into a stall that lasted until the impact with the sea at 2 h 14 min 28.

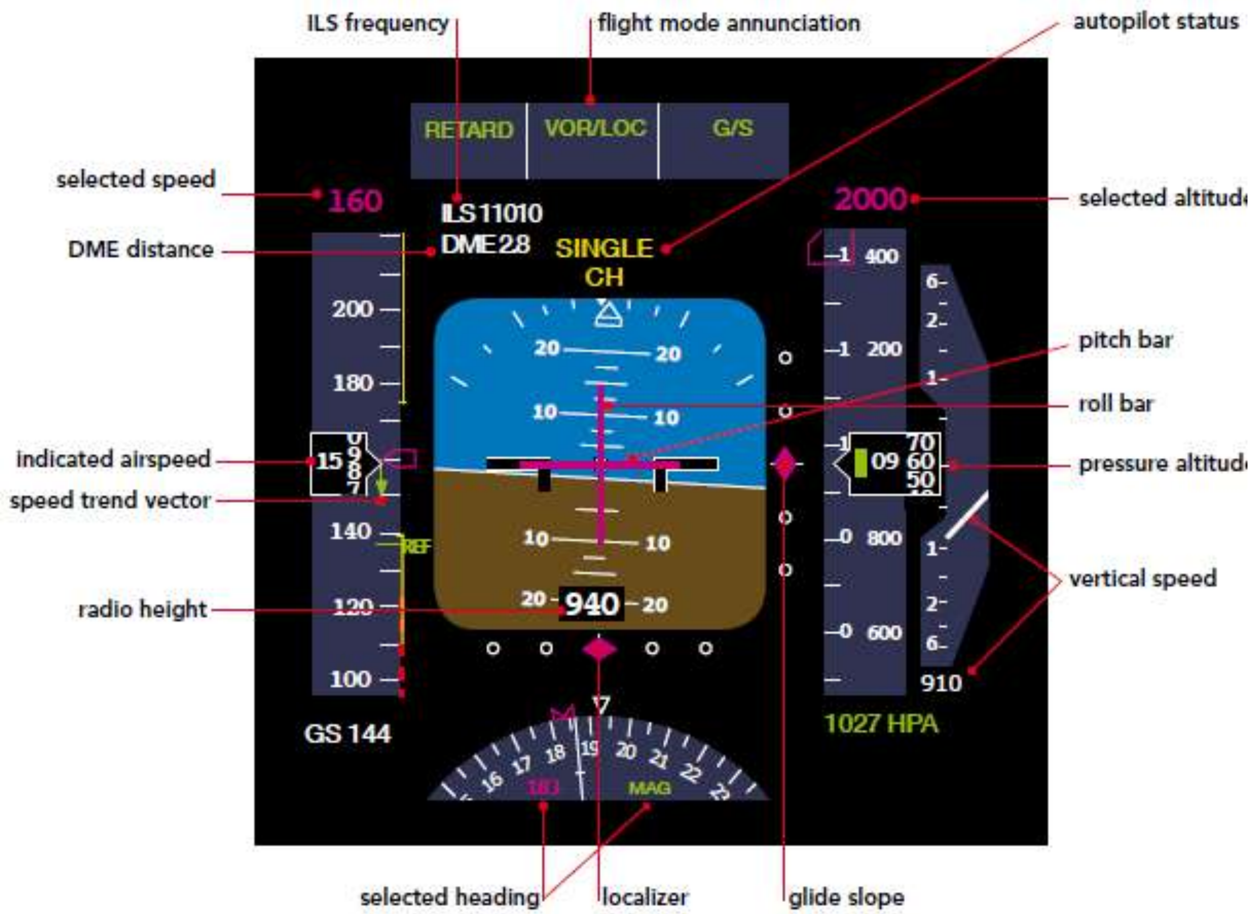
Accident Factors: (49 minutes)

- Big storm hidden by smaller storm on weather radar
- Super-cooled water
- Pitot tubes in atmosphere outside of design requirements
- Narrow speed envelope
- Procedures – abnormal (85% thrust, 5 degrees)
- Stall recovery techniques
- Overreliance on automation
- Limits of simulation capabilities

YOU ARE THE PILOT

How do pilots know the aircraft altitude and speed?

PRIMARY FLIGHT DISPLAY



Vertical Speed Indicator (VSI)

Right handside **pointer**= Vertical Speed in feet per minute. Below middle mark(0 fpm = level flight), rate of descent. Above middle mark (0 fpm = level flight), rate of climb. Image shows rate-of-descent at -900 fpm.

Descent on 3 degree glidepath, approximately -1000 to -1500 fpm. Range of display -6000 fpm to +6000 fpm.

Vertical Speed Limits: There are NO vertical speed limits annunciated. A rate of descent in excess of 6000fpm would exceed the speed envelope. Very difficult to maintain a sustained rate-of-climb in excess of 4000 fpm.

Vertical Speed Targets: Some PFD's display desired vertical speed and a bug on the gauge.

Altitude Tape

Right handside **sliding tape** = Pressure Altitude in feet. Background tape with altitude values slides past box in middle with actual Pressure Altitude. Pressure Altitude in image 960 feet. Range of display -1000 ft to 45,000 ft.

Altitude Limits displayed. Ground proximity warning.

Altitude Targets: Air Traffic Control (ATC) ceiling/floor/ desired altitude is displayed in magenta above the altitude tape. Image shows selected altitude 2000' (Note: this image is of approach in which the aircraft has descended below the target altitude. When the aircraft is cleared to land, the pilots will set the target altitude to the missed approach clearance altitude in preparation for a missed approach (if it is required). This is the only time, the aircraft is allowed to violate the target altitude.

Indicated Airspeed Tape

Left handside **sliding tape** = Indicated Airspeed in knots. Background tape with airspeed values slides past box in middle with actual Indicated Airspeed. Indicated Airspeed in image 159 knots. Range of display 0 knots to 340 knots..

Airspeed Limits displayed.(1) 1.3 VStall – minimum safe speed plus 30% buffer, (2) Vstall – minimum speed at which lift is lost, (3) Maximum speed for buffet and compressibility.

Airspeed Targets: displayed above the airspeed tape (160 knots) and by a magenta bug on the tape.

There is also an airspeed trend indicator. This is small green arrow that points either up or down on the speed tape. The tip of the arrow identifies the speed in 10 seconds. The average change in velocity over the past 10 seconds is used to forecast the velocity in the next 10 seconds. In this example, the aircraft is decelerating and will be at 150 knots in 10 seconds.

Radio Height

Geometric height above the terrain is displayed as a number in the bottom section of the Horizontal Situation Indicator (HSI). This value is derived from a radio altimeter and is only available within approx 5000 feet of the terrain. Used for takeoff and landing.

Groundspeed

Groundspeed (knots) is displayed as a number under the Airspeed tape. The Groundspeed is equal to the Airspeed minus the headwind (i.e. wind component parallel to aircraft flightpath opposite to the direction of flight).

Historic Note:

Before the “glass cockpit’ (i.e. digital avionics) era there was the “steam gauge” era. The basic 6 instruments were arranged in a "basic-T" to provide an easy scan for pilots. From left top row: airspeed indicator, attitude indicator, altimeter. From left bottom row: turn coordinator, heading indicator, and vertical speed indicator.

AIRPSEED, MACH NUMBER AND ALTITUDE

Mach Number

The **speed of sound** is the distance travelled during a unit of time by a sound wave propagating through an elastic medium. In dry air at 20 °C (68 °F), the speed of sound is 343.2 metres per second (1,126 ft/s). This is 1,236 kilometres per hour (768 mph), or about one kilometer in three seconds, or approximately one mile in five seconds.

In fluid dynamics, the speed of sound in a fluid medium (gas or liquid) is used as a relative measure of speed itself. The speed (in distance per time) divided by the speed of sound in the fluid is called the **Mach number**. Objects moving at speeds greater than *Mach1.0* are traveling at supersonic speeds. Objects moving at speeds less than Mach = 1.0 are traveling at subsonic speeds.

Speed of Sound in Dry air (m/s) = 33.4 + 0.6 * T (m/s) – below 36,000 ft

Speed of Sound in Dry air (knots) = 29.06 SQRT (518.7 – (3.57 * Altitude)) - below 36,000 ft

Temperature stabilizes at -69.7 degrees F at 36,000 ft → speed stabilizes at 573 knots

Where:

T is Temperature in degrees Celsius

Altitude 1000 ft

For example:

at 15º C, Speed of Sound = 340.5 m/s = 762.8 m/hr

As altitude increases, the temperature decreases by 2º per 1000 ft → Speed of Sound decreases with altitude

<u>MACH, TAS and ALTITUDE</u>

TAS for Constant Mach = Speed of Sound * Constant Mach

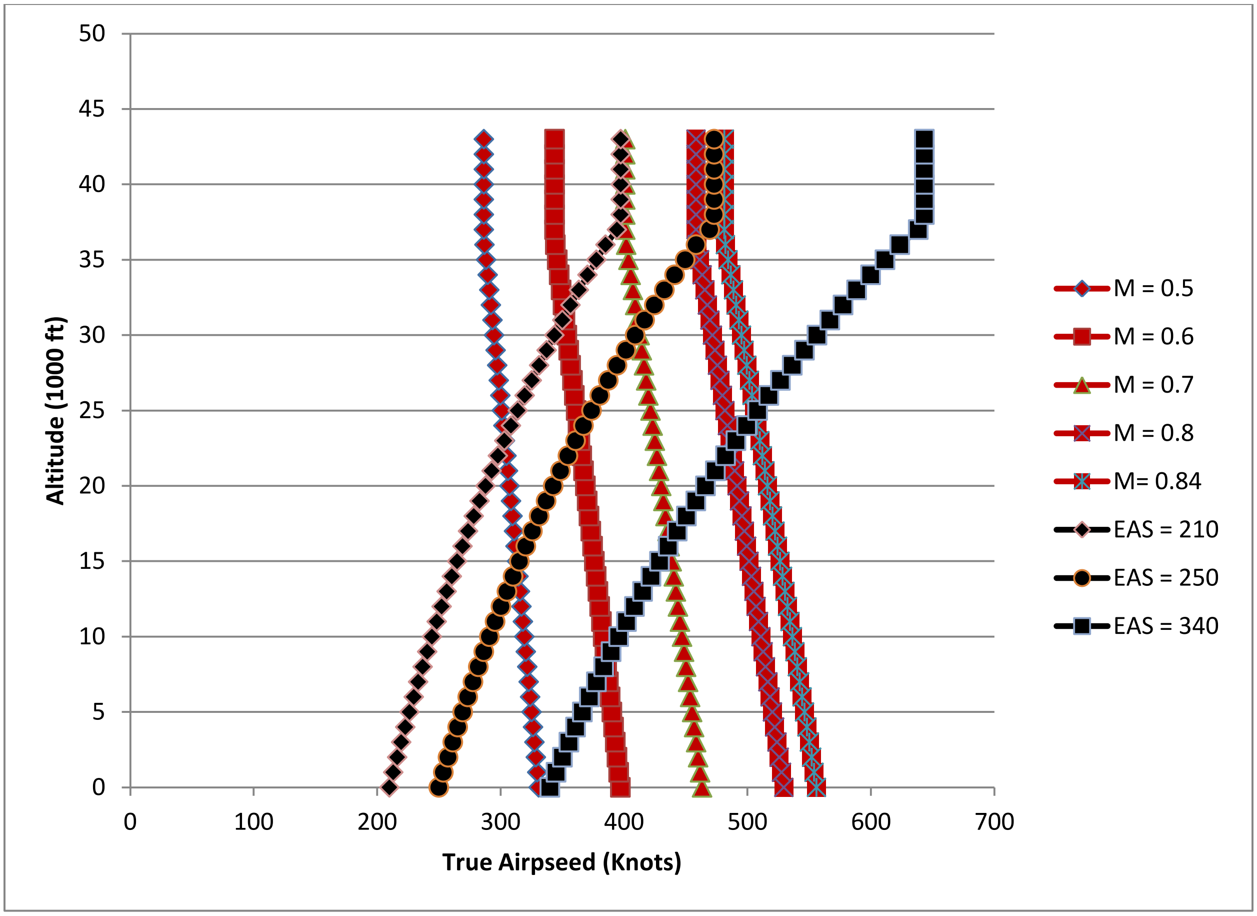
δ =POWER((1-6.88*POWER(10,-6)*Altitude),5.26)

Θ =(1-(6.8753*POWER(10,-6))*(Altitude))

TAS for Constant CAS = Constant CAS /SQRT(δ / Θ)

- CAS/Mach Transition Altitude (e.g. 340 knots and 0.7 Mach)
- "Coffin Corner" where Mach $_{Max}$ meets CAS_{Min}
- 250 knots 10,000'
- Econ Climb CAS/Mach (e.g. 280/0.78
- Econ Cruise Mach (e.g. 0.82)
- Econ Descent Mach/CAS (e.g. 0.77/268)

Altitude (1000 ft)	**Speed of Sound (Knots TAS)**	**Mach (250 knots)**	**TAS for 0.5 M**	**TAS for 0.6 M**	**TAS for 0.7 M**	**TAS for 0.8M**	**TAS for 0.84M**	**Lambda**	**Theta**	**TAS for 210 EAS**	**TAS**
0	=29.06*SQRT(518.7-(3.57*A2))	=340/B2	=B2*0.5	=B2*0.6	=B2*0.7	=B2*0.8	=B2*0.84	=POWER((1-6.88*POWER(10,-6)*A2*1000),5.26)	=(1-(6.8753*POWER(10,-6))*(A2*1000))	=210/SQRT(I2/J2)	=25
=A2+1	=29.06*SQRT(518.7-(3.57*A3))	=340/B3	=B3*0.5	=B3*0.6	=B3*0.7	=B3*0.8	=B3*0.84	=POWER((1-6.88*POWER(10,-6)*A3*1000),5.26)	=(1-(6.8753*POWER(10,-6))*(A3*1000))	=210/SQRT(I3/J3)	=25
=A3+1	=29.06*SQRT(518.7-(3.57*A4))	=340/B4	=B4*0.5	=B4*0.6	=B4*0.7	=B4*0.8	=B4*0.84	=POWER((1-6.88*POWER(10,-6)*A4*1000),5.26)	=(1-(6.8753*POWER(10,-6))*(A4*1000))	=210/SQRT(I4/J4)	=25



50
45
40
35
30
25
20
15
10
5
0
Altitude (1000 ft)
0
100
200
300
400
500
600
700
True Airpseed (Knots)
M = 0.5
M = 0.6
M = 0.7
M = 0.8
M= 0.84
EAS = 210
EAS = 250
EAS = 340

=

SECTION 3) AERODYNAMICS, AIRFOILS, LIFT, DRAG, AND THRUST

Aerodynamics is the specialization of the wider field of fluid dynamics.

Aerodynamics concerned with study of fluids that are susceptible to compressibility (e.g. air at speeds above 200 ft/sec).

Wings

The wing is the main component of the aircraft that generates the upward force, known as lift, that makes heavier than air flight feasible.

Aircraft wings are built in many shapes and sizes for different applications. One the tradeoffs that must be made in wing design is between lift, balance, and stability in flight. Straight wings are more stable but generate drag (that keeps speeds low and increases fuel burn and emissions). Sweptback and delta wings reduce drag but decrease stability.

Typical wing leading and trailing edge shapes

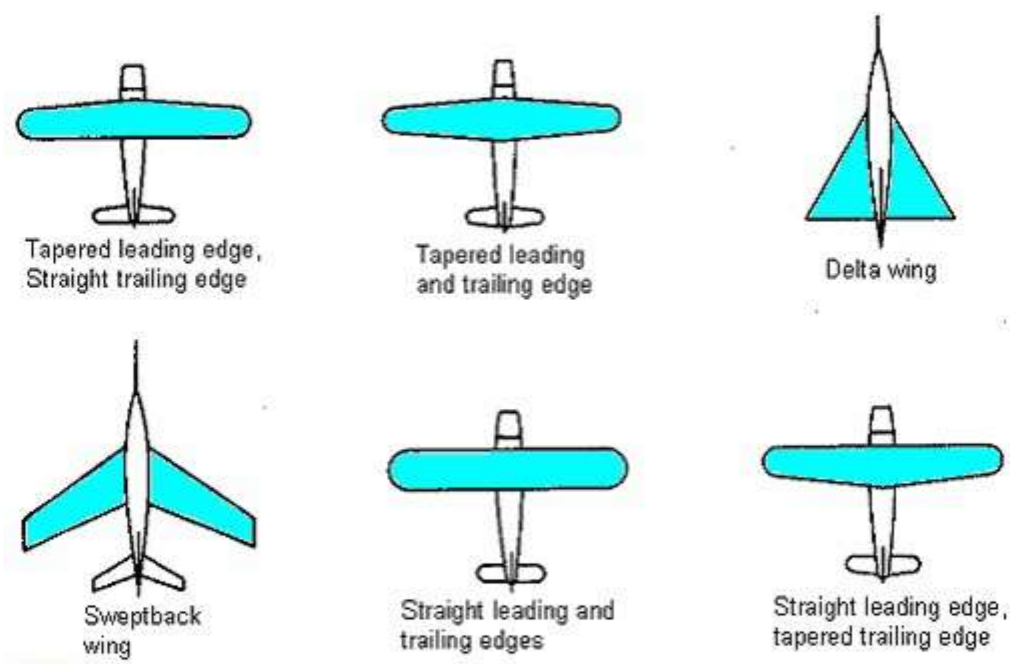
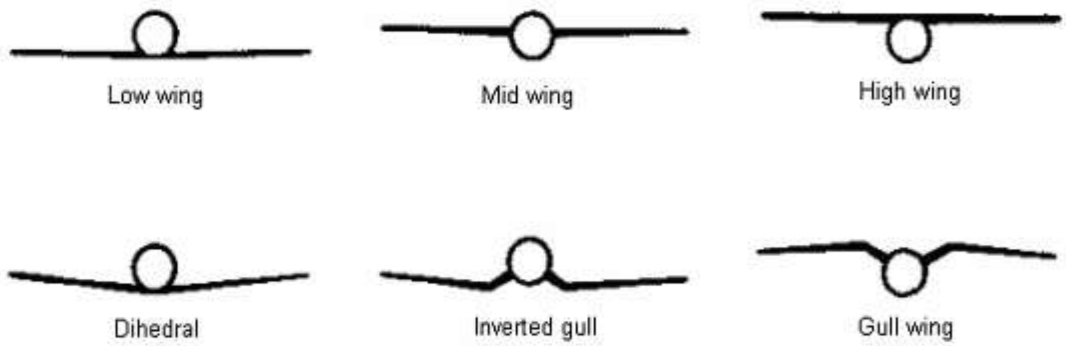


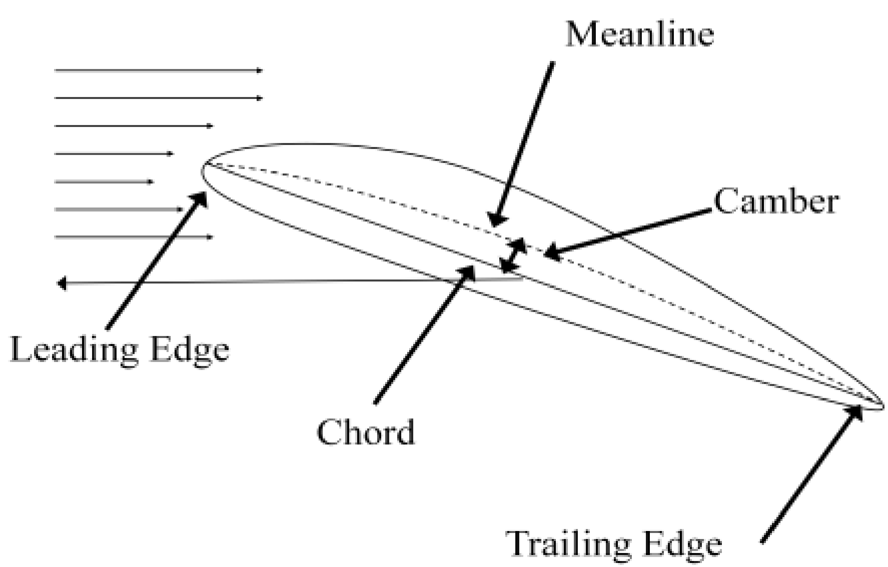
Figure below shows the common wing forms and configuration.



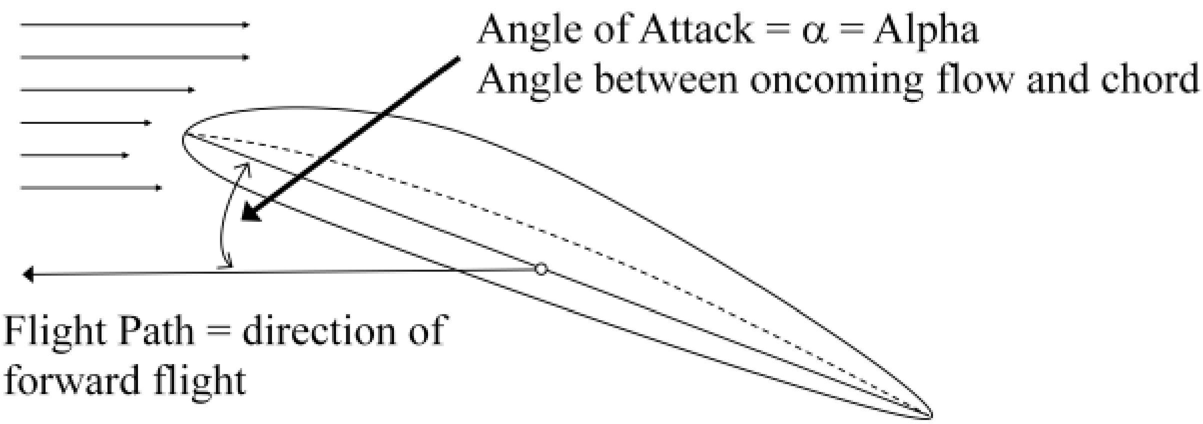
Airfoils

The airfoil section, a cross-section of the wing determines the amount of lift generated. Airfoil section properties are as follows:

1. Leading Edge faces oncoming flow
2. Trailing edge - opposite oncoming flow
3. Chord - straight line from leading edge to trailing edge
4. Meanline - Line midway between upper and lower surface
5. Camber- Maximum difference between meanline and chord (*Symmetrical airfoil, camber = zero)*



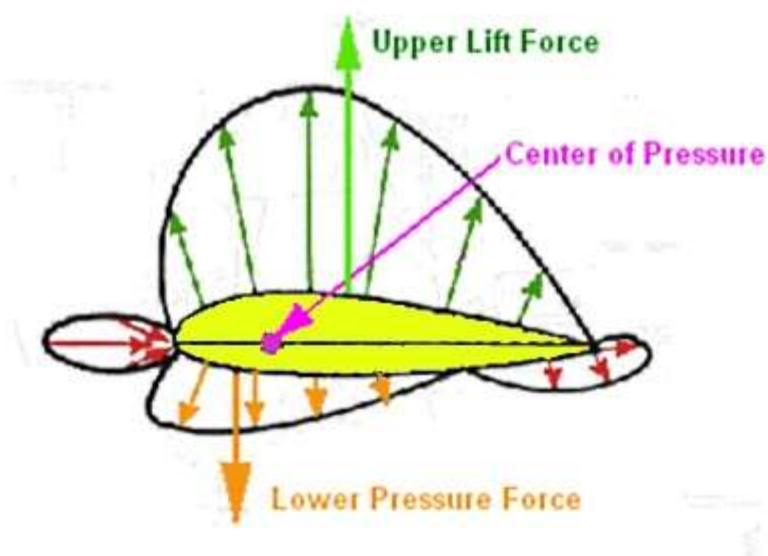
Angle-of-Attack (α = alpha) is the angle between on-coming flow and chord.



Lift

Lift is the upward force that enables flight.

Lift is generated by the pressure differential that exists between the upper and lower surfaces of the wing. The differential pressure is the result of the higher velocity flow that exists on the upper surface of the wing and the relatively lower velocity flow along the lower surface of the wing.



Three factors determine the magnitude of the Lift force:

1) Airfoil chord length. The longer the chord the more surface area over which a differential pressure can act. However, the longer the chord, the more friction drag will generated.

2) Airfoil camber. Camber has the effect of inducing higher pressure differentials for a given surface area. Excessive camber produces drag an dynamic instability.
3) Angle of Attack (α, apha) The higher the alpha the higher pressure differential and the greater the lift. When alpha gets too high, the flow over the upper surface of the airfoil transitions from laminar flow (smooth over the surface) to turbulent flow that separates from the airfoil surface.

Stall

A **stall** occurs beyond a certain angle-of-attack such that the lift begins to decrease. The angle at which this occurs is called the *critical angle of attack*. This critical angle is dependent upon the profile of the wing, itsplanform, its aspect ratio, and other factors, but is typically in the range of 8 to 20 degrees relative to the incoming wind for most subsonic airfoils. The critical angle of attack is the angle of attack on the lift coefficient versus angle-of-attack curve at which the maximum lift coefficient occurs.

Flow separation *begins* to occur at small angles of attack while *attached* flow over the wing is still dominant. As angle of attack increases, the separated regions on the top of the wing increase in size and hinder the wing's ability to create lift. At the critical angle of attack, separated flow is so dominant that further increases in angle of attack produce *less* lift and vastly more drag. (Note, airflow doesn't really separate from the wing, a vacuum does not magically emerge there. Rather, clean laminar flow gets pulled away by messy turbulent flow—the green area in the diagram. "Flow separation" is a useful abstraction though.)

A fixed-wing aircraft during a stall may experience buffeting or a change in attitude (normally nose down). Most aircraft are designed to have a gradual stall with characteristics that will warn the pilot and give the pilot time to react. For example an aircraft that does not buffet before the stall may have an audible alarm or a stick shaker installed to simulate the feel of a buffet by vibrating the stick fore and aft. The "buffet margin" is, for a given set of conditions, the amount of ‘g’, which can be imposed for a given level of buffet. The critical angle of attack in steady straight and level flight can only be attained at low airspeed. Attempts to increase the angle of attack at higher airspeeds can cause a high speed stall or may merely cause the aircraft to climb.

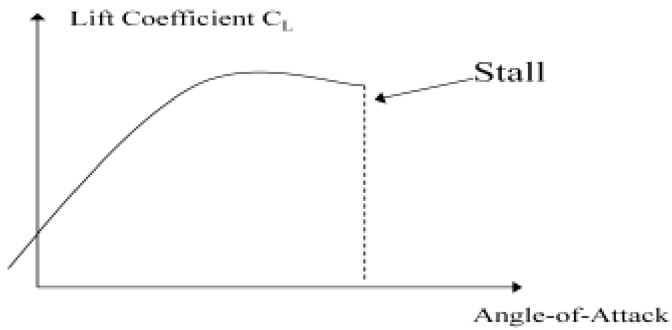
Watch King Training Video on Angle of Attack and Stall

http://www.youtube.com/watch?v=5wIq75_BzOQ&feature=player_embedded#!

Watch King Training Video on impact of flaps during a Stall

http://www.youtube.com/watch?v=sKzbeWwe0wM&feature=related

Stall (Continued)



The graph shows that the greatest amount of lift is produced as the critical angle of attack is reached(quaintly known as the "burble point" in the early days of aviation).

This graph shows the stall angle, yet in practice most pilot operating handbooks (POH) or generic flight manuals describe stalling in terms of airspeed. This is because all aircraft are equipped with an airspeed indicator, but fewer aircraft have an angle of attack indicator. An aircraft's stalling speeds is published by the manufacturer (and is required for certification by flight testing) for a range of weights and flap positions, but the stalling angle of attack is not published.

As speed reduces, angle of attack has to increase to keep lift constant until the critical angle is reached. The airspeed at which this angle is reached is the (1g, unaccelerated) stalling speed of the aircraft in that particular configuration. Deploying flaps/slats decreases the stall speed to allow the aircraft to take off and land at a lower speed.

One symptom of an approaching stall is slow and sloppy controls . As the speed of the aircraft decreases approaching the stall, there is less air moving over the wing and therefore less air will be deflected by the control surfaces (ailerons, elevator and rudder) at this slower speed. Some buffeting may also be felt from the turbulent flow above the wings as the stall is reached. However during a turn this buffeting will not be felt and immediate action must be taken to recover from the stall. The stall warning will sound, if fitted, in most aircraft 5 to 10 knots above the stall speed.

The magnitude of the Lift force can be computed from the following equation:

Lift = ½ * C_L * ρ * Velocity^2 * Wing Area
Where:
Lift = the magnitude of the lift force (lbs)
C_L = the Lift Coefficient
ρ = air density (slugs/ft^3)
Velocity = True Airspeed (TAS) along the Flight Path axis (feet/sec)
Wing Area = area of the wing (ft^2)

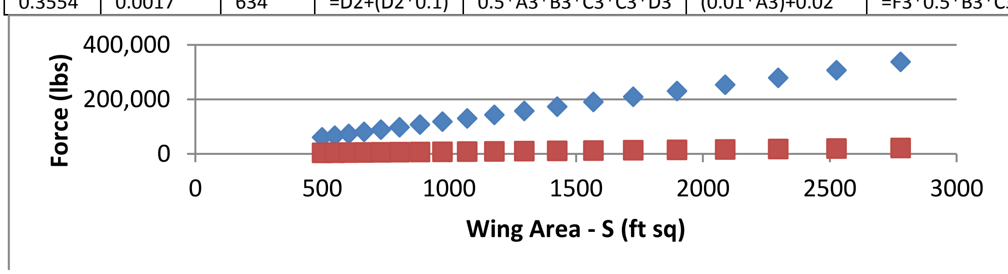
The Lift Coefficient (C_L) is a non-dimensional parameter used to capture the complexities in the generation of Lift. [Note: A non-dimensional parameter enables the use of the scale models in a wind tunnel to estimate the lift forces in full size aircraft]. The Lift Coefficient is estimated in wind tunnel tests by measuring the lift force, L.

C_L = Lift / (½ * ρ * Velocity^2 * Wing Area)

Hear Orville and Wilbur Wright discuss Lift.
http://www.grc.nasa.gov/WWW/Wright/podcast/Podcast_Forces_Lift.m4v

What is the impact of increasing Wing Area on Lift and Drag?
Lift = ½ * C_L * ρ * Velocity^2 * Wing Area
Drag = ½ * C_D * ρ * Velocity^2 * Wing Area
Where:
Lift = the magnitude of the lift force (lbs)
C_L = the Lift Coefficient = 0.3554
C_D = 0.05C_L^2 -0.01C_L +0.02
ρ = air density (slugs/ft^3) = 0.0017
Velocity = True Airspeed (TAS) along the Flight Path axis (feet/sec) = 634 fps
Wing Area = area of the wing (ft^2) = 1000 ft^2

CL	Rho (slugs/ft3)	V (ft/sec)	S (ft sq)	LIFT = ½ * CL * Rho * Velocity2 * Wing Area	CD = 0.05CL2 - 0.01CL +0.02	D=CD*1/2*Rho*V2*S
0.3554	0.0017	634	500	= 0.5*A2*B2*C2*C2*D2	=(0.05*A2*A2)-(0.01*A2)+0.02	=F2*0.5*B2*C2*C2*D2
0.3554	0.0017	634	=D2+(D2*0.1)	= 0.5*A3*B3*C3*C3*D3	=(0.05*A3*A3)-(0.01*A3)+0.02	=F3*0.5*B3*C3*C3*D3



Drag

The penalty of a lift device, such as a wing) is the production of a Drag force, due to the resistance generated by the fluid to the relative motion of the wing.

The Drag force has four measured components:

(1) Form Drag is generated by the energy required to move fluid away from the path of the object.
(2) Viscous Drag is the result of the friction between the object and the fluid
(3) Induced Drag is associated with the production of the Lift force. An object that produces no Lift has no Drag force
(4) Wave Drag, occurs only in supersonic flight, is associated with moving a shock wave through the fluid.

The magnitude of the Drag force can be computed from the following equation:

$$\text{Drag} = ½ * C_D * \rho * \text{Velocity}^2 * \text{Wing Area}$$

Where:
Drag = the magnitude of the Drag force (lbs)
C_D = the Drag Coefficient
ρ = air density (slugs/ft^3)
Velocity = True Airspeed (TAS) along the Flight Path axis (feet/sec)
Wing Area = area of the wing (ft^2)

The Drag Coefficient (C_D) is a non-dimensional parameter used to capture the complexities in the generation of Drag. [Note: A non-dimensional parameter enables the use of the scale models in a wind tunnel to estimate the lift forces in full size aircraft]. The Drag Coefficient is estimated in wind tunnel tests by measuring the drag force, D, and applying the following equation.

$$C_D = \text{Drag} / (½ * \rho * \text{Velocity}^2 * \text{Wing Area})$$

The table below illustrates representative Drag Coefficients various objects. Note the role of the impact surface area (e.g. sphere vs teardrop, cube vs. diamond), as well as the role of the length of object (e.g. longs vs. short cylinder, sphere vs. half sphere)

Object	**CD**

Sphere	0.47
Half-sphere (chop off back half of sphere)	0.42
Cone (point first)	0.5
Cube	1.05
Diamond (cube rotated 45 degrees)	0.8
Long cylinder	0.82
Short cylinder	1.15
Teardrop	0.04

Drag is not all bad. There are several operational scenarios in which an increase in Drag force is useful. In one scenario, Air Traffic Control may request to expedite descent. Increasing drag, while maintain airspeed and thrust setting, will increase the rate of descent. Drag is introduced by extending spoilers (also known as airbrakes), extending the landing gear and flaps/slats. Note landing gear and flaps/slats can only be extended at slower speeds to avoid shearing these devices off their mounts.

THRUST

Thrust Rating

Thrust Rating is the maximum level of thrust that an engine can attain under a given set of environmental conditions. Thrust Rating is used to establish the limits to which an engine can be taken under different flight conditions.

Thrust Rating depends on speed, altitude, and temperature.
As ambient temperature rises beyond a threshold temperature (e.g. 59°F), the thrust generated decreases.
Turbofan rotational speed increases the fan, shaft, and turbines experience aerodynamic and centrifugal forces that create stress levels in the materials that result in structural failure.
Increased altitude resulting in reduced density, will generate less thrust at the same rotor speed.

The following Thrust Ratings commonly used:

- Takeoff (TO) – maximum thrust for the takeoff phase. Usually limited to duration of 5 minutes
- Maximum Climb Thrust (CLB) – maximum thrust extracted during climb phase

- Maximum Continuous Thrust (MCT) – maximum thrust extracted from an engine for unlimited duration. Reserved for operations when an engine has failed.
- Maximum Cruise Thrust (MCT) – maximum thrust that can be extracted from an engine in the cruise regime.

Thrust Available

Thrust Available is the thrust generated at a specific Throttle Setting and the prevailing flight atmospheric conditions.

Thrust Required

Thrust Required is the term used to identify the amount of force that must be generated by the engines to overcome other forces such as drag, inertia and weight. For example, to maintain level flight at a constant speed, will require a specific Thrust Setting (T_1). To accelerate the aircraft to a new speed in level flight will require increased thrust (T_2). T_2 is the Thrust Required.

Fuel Burn Rate

Fuel Burn Rate, also known as Fuel Flow, is a critical parameter. It is used to estimate the required fuel for a flight, as well as efficiency and pollution computations.

Fuel Burn Rate is proportional to thrust. As thrust increases, the fuel burn rate increases. To account for atmospheric conditions, Fuel Burn Rate is corrected as follows:

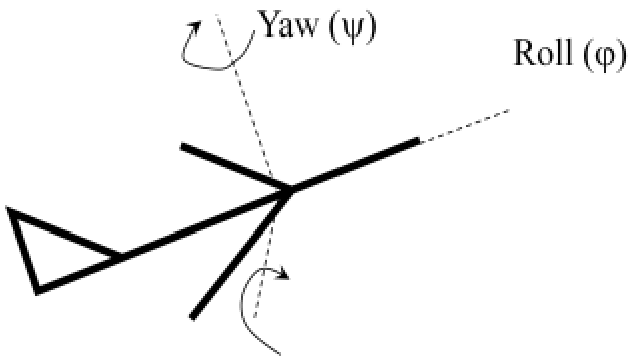
$$W_{fCorrected} = Wf / \delta * \theta$$

An alternate measure of fuel burn rate is Specific Fuel Consumption (SFC). The correction SQRT θ removes the effects of temperature and altitude variation.

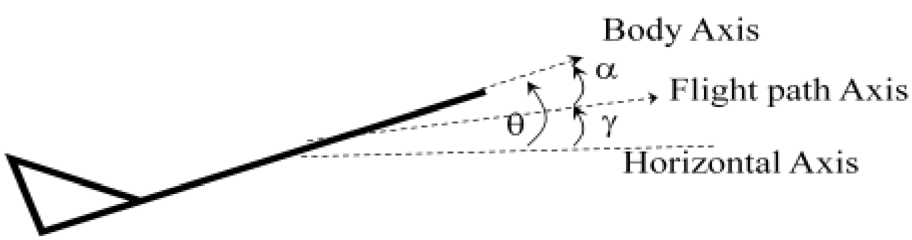
$$SFC = (\text{fuel flow} / \text{thrust}) * \text{SQRT}\ \theta$$

SECTION 4 AIRCRAFT EQUATIONS OF MOTION

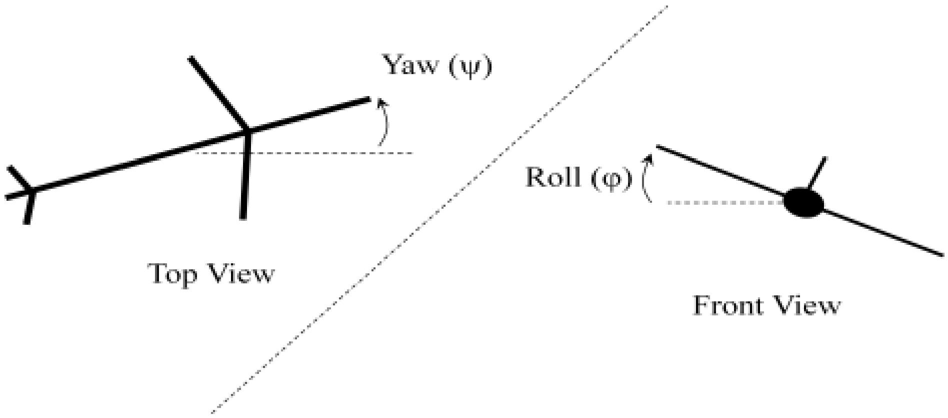
Angles and Axes



Pitch (θ) =
Flight Path Angle (γ) + Angle of Attack (α)



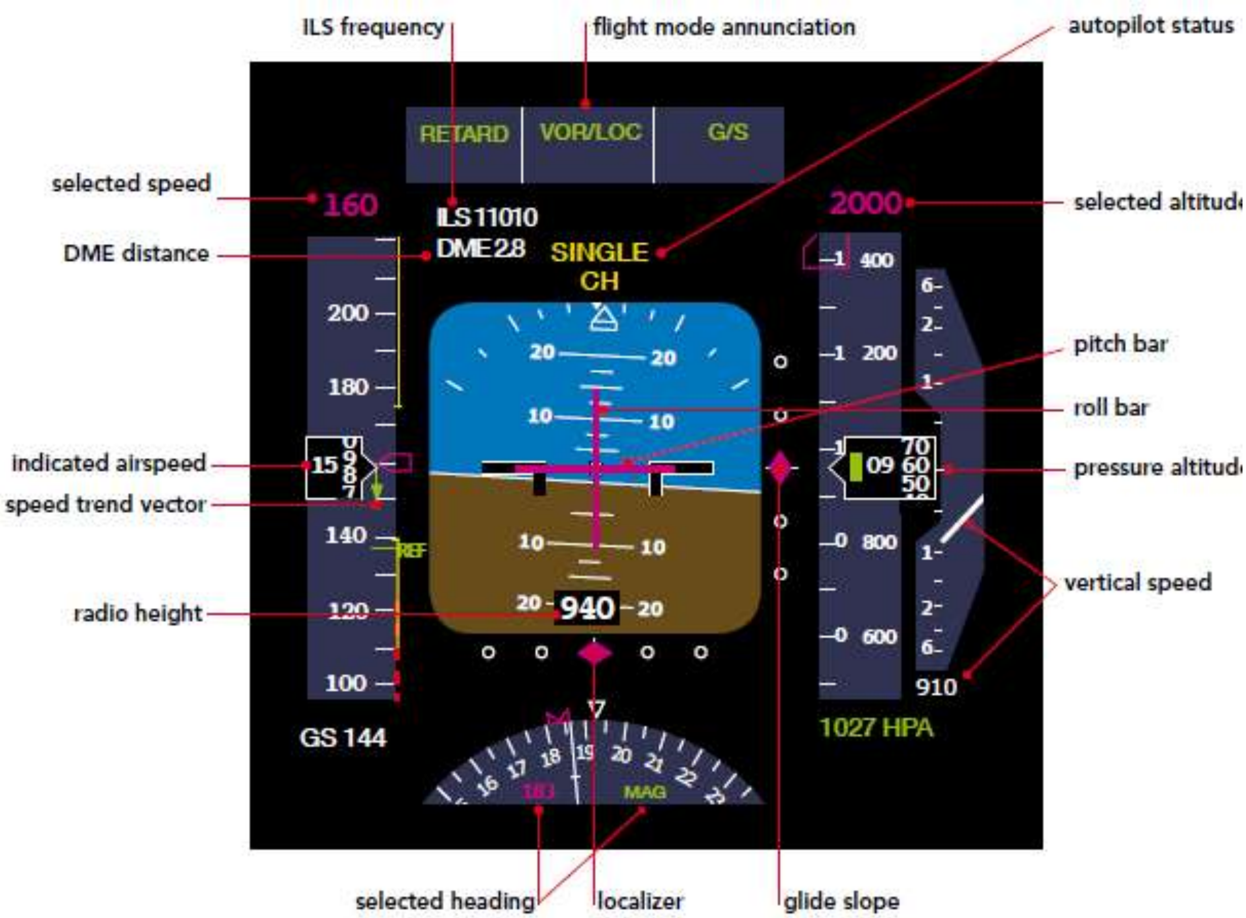
Pitch (θ) =
Flight Path Angle (γ) + Angle of Attack (α)



YOU ARE THE PILOT

How do pilots know the aircraft Pitch, Roll (i.e. Bank Angle), and Yaw?

PRIMARY FLIGHT DISPLAY



Horizontal Situation Indicator (HSI)

The HSI (also known as the Attitude Indicator or *Artificial Horizon*) displays the aircraft's attitude relative to the horizon. This provides information on the bank angle of the wings and the pitch angle (i.e. aircraft nose is pointing above or below the horizon). This is a primary instrument for instrument flight.

The HIS is located in the middle of the PFD. It has a blue (sky) and brown (ground) background with several markings overlaid.

Bank Angle

At the top of the HSI is a half-circle of white hash marks with a white triangular pointer. The hash marks represent 5 degrees of bank angle. The triangle pointer is the current bank angle. In this example the left wing is tipped slightly down, the right wing is tipped slightly up.

There are also two black right angled images on either side of the center line. These images are the wings of the aircraft. The white line separating the blue sky and brown ground represents the horizon. In this example the horizon is higher on the left than on the right indicating a slight bank to the left.

Pitch

The white horizontal hashmarks in the middle of the HSI represent increments of 2.5 degrees of pitch. The pitch of the aircraft is the hash mark that is between two black right angle brackets (wings). In this example the nose of the aircraft is pitched up 2.5 degrees.

The magenta Pitch Bar and Roll bar are part of the Flight Director (Autopilot) and will be discussed in a later chapter.

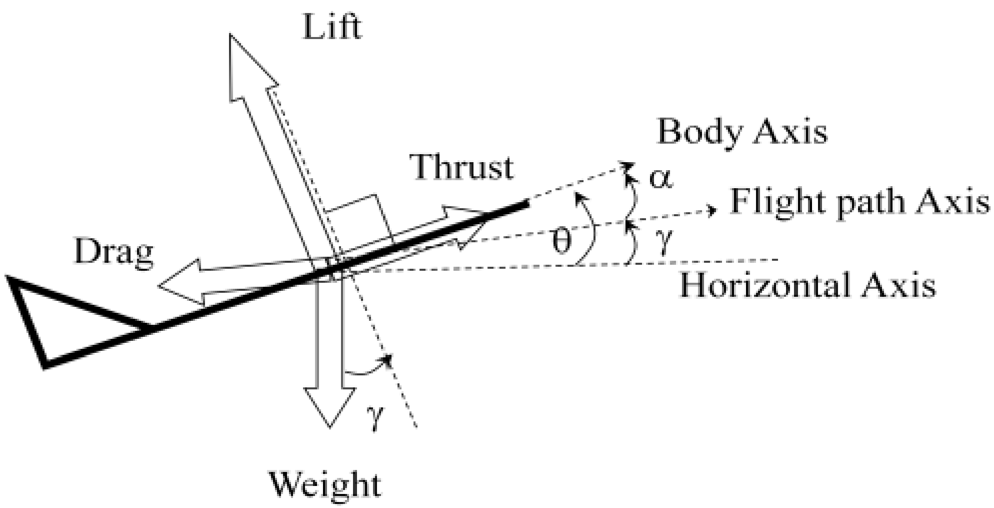
Yaw (Slip and Skid Indicator)

The white rectangle located right below the triangular bank angle indicator. The white rectangle will slide left or right of the base of the triangle indicating the direction and magnitude of the yaw.

Forces

Lift is perpendicular to Flight Path Axis
Thrust is parallel to Body Axis
Drag is parallel to Flight Path Axis
Weight is a result of gravity, perpendicular to Horizontal Axis

Free-body Diagram for Aircraft Performance (Vertical and Longitudinal Axes only)



Equations of Motion (Vertical and Longitudinal Axes only)

Newtons 3rd Law: Sum of the forces equals the Mass times the Acceleartion
Mass * Acceleration = Σ Forces

Mass * Flight Path Acceleration (dV/dt) = Thrust($\cos\alpha$) - Drag – Weight($\sin\gamma$)

Where:
- Acceleration on the Flight-path Axis += dV/dt - ft/sec
- Thrust, Drag, Weight – lbs
- α, γ - radians
- Mass = Weight/g,
- g=32.2 ft/sec^2

Case 1: Level Flight, Constant Speed

Level Flight - $\gamma = 0$
Constant Speed dV/dt = 0

Substitute $\gamma = 0$ and dV/dt = 0 into equation of motion

$M * \ 0 = T(\cos\alpha) - D – 0$
$0 = T(\cos\alpha) – D$
$-T(\cos\alpha) = -D$
$T(\cos\alpha) = D$

Assume $\cos\alpha \sim 1$
Thrust = Drag

To maintain level flight at a constant speed, the Thrust is required to overcome the Drag force.

<u>Case 2: Level Flight, Increasing Speed (Accelerating)</u>

Level Flight - $\gamma = 0$
Constant Speed $dV/dt > 0$

$M * \ dV/dt = T(\cos\alpha) - D – 0$
$(M * dV/dt) + D = T(\cos\alpha)$
$T = D + (M * dV/dt)$

Thrust = Drag + Force Required to Accelerate Mass
To accelerate in level flight, the thrust is required t overcome the Drag force and the force of inertia to accelerate to a new speed.

<u>Case 3: Climbing, Constant Speed</u>

Level Flight - $\gamma > 0$
Constant Speed $dV/dt = 0$

$M * \ 0 = T(\cos\alpha) - D – W(\sin\gamma)$
$W(\sin\gamma) + D = T(\cos\alpha)$

Thrust = Drag + Force Required to Overcome Weight (for selected Flight Path Angle)

Maximum Angle for Climb (γMax) **is determined by Max Thrust, Weight and Drag**

Exercise:

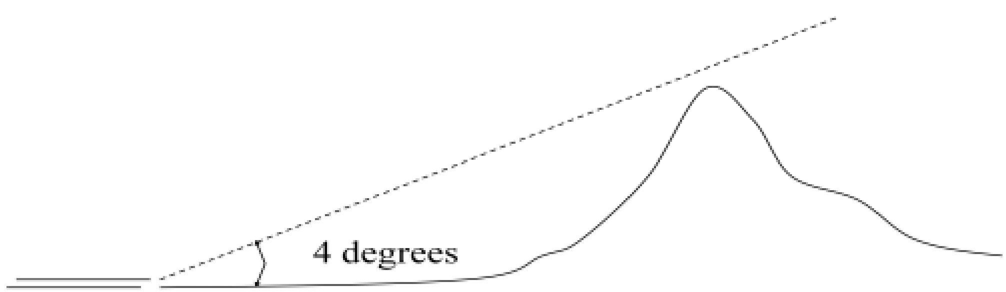
1) Solve the equations of motion for the following scenarios:
   a. Climbing and Accelerating to a new speed
   b. Descending at Constant Speed
   c. Descending and Decelerating to a new speed
2) Draw a state-space diagram with γ on the y-axis and Thrust on the x-axis. Assume constant Drag and Weight. Draw contours for constant speed, accelerating and decelerating

Example Problem:

You are a Flight Operations Engineer at Eventually Airways (motto: We will get you there …eventually).

Your company marketing department has decided to provide service in the winter ski season to a popular ski village located in the valley between two steep peaks. You need to determine the type of aircraft that can be used in all weather conditions to climb-out in excess of 4 degrees to clear the peak located in the published departure procedure.

Compute Flight path angle (γ) for climb with Drag = 6404 lbs, TAS = 634 ft/sec, W=100,000lbs. T=19500. Assume α = 0, no winds.



Solution:

Mass * Flight Path Acceleration (dV/dt) = Thrust($\cos\alpha$) - Drag – Weight($\sin\gamma$)

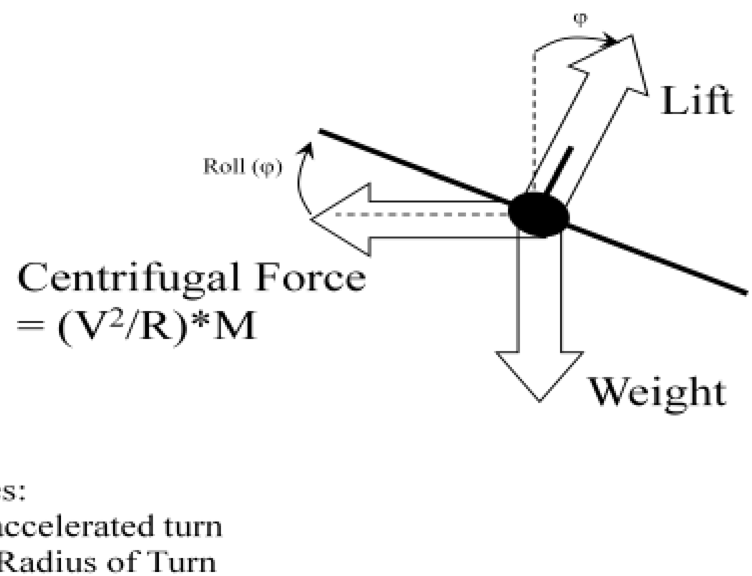
Substitute in values for W, D, T

W($\sin\gamma$) + D = T($\cos\alpha$);

$\sin\gamma$ = (T($\cos\alpha$) – D)/W;

$\cos\alpha = \cos(0) = 1;$
$\sin\gamma = (19500(1) – 6404lbs)/100{,}000lbs$
$\sin\gamma = 0.131$ radians
$\gamma =$ Inverse sin (0.131radians * (360º /2Π)) = 7.5º

Free-body Diagram for Aircraft Performance (Lateral and Vertical Plane)



ΣForces in Lateral Axis:
$(V^2/R)*M = L(\sin\varphi)$
$(V^2/R)*W/g - L(\sin\varphi) = 0$ (1)

To maintain a coordinated turn, Lift component in the horizontal axis equal the Centrifugal force.

ΣForces in Vertical Axis:
$L(\cos\varphi) - W= 0$
$L = W/ (\cos\varphi)$ (2)
To maintain level flight, Lift component in Vertical axis must exceed Weight

To develop the relationship between Bank Angle and Turn Radius

- Substitute equation (2) into equation (1)

$(V^2/R)*W/g - W (\sin\varphi) / (\cos\varphi) = 0$

- Replace W with mg, and $(\sin\varphi) / (\cos\varphi) = (\tan\varphi)$

$(V^2/R)*mg/g - mg (\sin\varphi) / (\cos\varphi) = 0$
$(V^2/R)*m - mg (\tan\varphi) = 0$

- Solve for $\tan\varphi$

$\tan \varphi = (V^2/R)*(1/g)$

Solve for R
$R = V^2/(g \tan \varphi)$

Turn Radius is determined by Speed (V) and Roll Angle (φ)

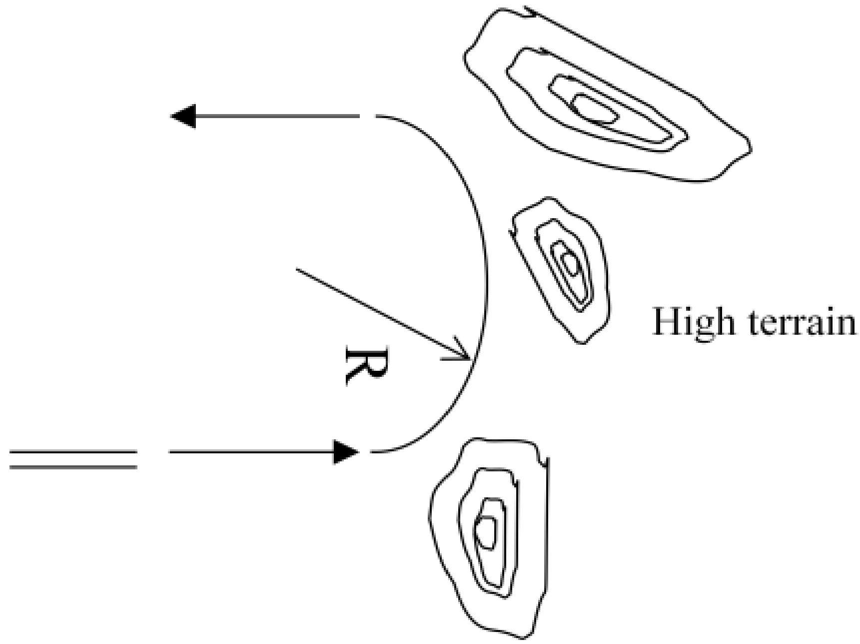
Note: φ is in radians, degrees to radians = 2Pi/360 , g = 32.2 ft/sec^2

Example Problem #2:

You are a Flight Operations Engineer at Eventually Airways (motto: We will get you there …eventually).

Your company marketing department has decided to provide service in the winter ski season to a popular ski village located in the valley between two steep peaks. The aircraft selected for the route is not able to clear the mountain in the departure. You need to determine whether the type of aircraft selected can make a 180º turn of no more than 4nm turn radius to avoid high terrain under specific conditions. What bank angle is sufficient?

Aircraft speed (V) is 140 knots CAS (= 255 fps TAS).



Solution:

$R = V^2/(g \tan \varphi)$

$R = (255 ft/sec)^2 / (32.2\ ft/sec)\ (\tan (15º * 2Pi/360\ º)$

R = 7548 ft

Convert feet to n.m. (1nm = 6076 ft)

R = 1.24 nm

AIRCRAFT PERFORMANE LIMITS

Stall Speed (VS, 1.3 VS)

Maximum Operating Speed (VMO, MMO)

$$\text{Mass} * \text{Flight Path Acceleration} = \text{Thrust}(\cos\alpha) - \text{Drag} - \text{Weight}(\sin\gamma)$$

$m * dV/dt = T - D - mg(\sin\gamma)$

$dV/dt = (T - D)/m - g(\sin\gamma)$

Integrate over time

$V = -g(\sin\gamma)t + ((T\text{-}D)/m)t$

<u>Case: Starting from level flight and constant speed (T-D=0), increase or decrease Flight Path Angle;</u>

$V = -g(\sin\gamma)t$

Velocity increases as Flight Path Angle decreases.

Velocity decreases as Flight Path Angle increases.

Case: Starting from level flight and constant speed increase or decrease Thrust (while holding Flight Path Angle = 0);

$V = ((T-D)/m)t$

Velocity increases as Thrust increases above Drag.

Velocity decreases as Thrust decreases below Drag.

SECTION 6 ASSESSMENT

UNIT EXAM

TAKEHOME EXAM (Aerodynamic Simulation)

<u>INSTRUCTIONS</u>

1. **Initial: I have adhered to the University Honor Code ______**
   Note: All violations of Honor Code will be reported

2. **OPEN Book**
3. **Write and draw clearly. Provide explanations were required.**
4. **Have Fun!**

<u>Problem #1 State-Space</u>

a) Use Excel (or another tool with computational and graphing capability) to draw a state-space diagram with γ on the y-axis and Thrust on the x-axis. Assume Drag = 6404 lbs, Weight = 100,000. Compute γ for different Thrust (1000 lbs to 17,000 lbs in 2000lb increments) and different rate of acceleration (0, +0.05g, and -0.05g). For each rate of acceleration, plot thrust vs γ.
b) Explain the constant acceleration contours on the state-space diagram
c) Overlay the following trajectory on the state-space diagram.
   a. Aircraft is in level flight ($\gamma = 0$) at constant speed (250 knots). Hint: Compute Thrust for this condition. Don't forget to convert knots to ft/sec.
   b. Aircraft in level flight increases thrust such that the aircraft accelerates at +0.05g and maintains 270 knots.
   c. Aircraft thrust is decreased and the aircraft noses down to a one degree rate of descent while holding constant speed. After descending several thousand feet, the aircraft levels off and maintains 270 knots.

<u>Problem #2 Build a Simulation of Aircraft Performance (using the point-mass model)</u>

Instructions:

1. Use Excel (or any other application or software language)

Example: Excel Spreadsheet

Time	Thrust (lbs)	Drag (lbs)	Weight (lbs	Mass	Flight Path	sin(Flightpa	dV/dt (ft/se	g		VTAS (ft/se	VTAS (Knc	Vground (ft	Vground (K	Distance T	Distance T	Vertical Sp	Vertical Sp
1	6404	6404	100,000	3105.59	0	0	0	0		422	250.0329	422	250.0329	0	0	0	0
2	6404	6404	100,000	3105.59	0	0	0	0		422	250.0329	422	250.0329	422	0.069454	0	0

2. Create a simulation using the following parameters (i.e. columns)
    a. Time (secs) – range 0 to 200. Initial condition is at time step 0.
    b. Thrust (lbs)
    c. Drag (lbs)
    d. Weight (lbs)
    e. Mass = Weight/Gravitational Force
    f. Flight Path Angle (Degrees)
    g. Sin (FlightPath Angle) (radians)
    h. dv/dt = acceleration on Flight Path Axis (ft/sec2)
    i. VTAS (ft/sec) = True Airspeed on Flight Path Axis
    j. VTAS (knots)
    k. VGroundSpeed (ft/sec)
    l. VGroundSpeed (knots)
    m. Distance Travelled (feet)
    n. Distance Travelled (knots)
    o. Vertical Speed (ft/sec)
    p. Vertival Speed (ft/min)
    q. *Altitude (ft)*

    Note1: Parameters shown in *italics* are derived from equations of motion using the parameters in the list. Parameters in plan text are fixed (i.e. constants). Parameters in **bold** are input (see input profile below).

    Note2: All Excel trig functions use radians (not degrees). Convert degrees to radians using RADIANS(angle) function. Convert radians to degrees using DEGREES(radians) function.

3. Initial Conditions
    a. VTAS (ft/sec) = 422 ft/sec = 250 knots
    b. Distance Travelled = 0 ft
    c. Altitude = 0 ft
4. Test Cases. Make sure that your simulation is working correctly. In level flight, when thrust = drag, dV/dt should equal zero. In climb (i.e. Flight Path Angle > 0), at constant speed, it will require more thrust than in level flight. Check all the other cases too.

5. Create the following Charts:
    a. Time (x-axis), Thrust (lbs) (primary y-axis), Flight Path Angle (degrees) (secondary y-axis) and dV/dt (ft/sec2) (secondary y-axis)
    b. Time (x-axis), VTAS (knots) and VGroundspeed (knots) (primary y-axis), Distance Travelled (nm) (secondary y-axis)
    c. Time (x-axis), Flight Path Angle (degrees) (primary y-axis), Altitude (ft) (secondary y-axis), Vertical Speed (fpm) (secondary y-axis)
    d. Distance (nm) (x-axis), Altitude (ft) (primary y-axis), Vertical Speed (fpm) (primary y-axis)
6. The aircraft performs the following maneuvers. Using the charts from #5, complete the table below.

Maneuver	Thrust	Flight	Start	End	Start	End	Start	End	Start	End	Vertic

	Settin g (lbs)	Path Angle (degre es)	Time (secs)	Time (secs)	Distan ce (nm)	Distan ce (nm)	Altitud e (ft)	Altitud e (ft)	VTAS (knots)	VTAS (knots)	al Speed (fpm)
1. Level Flight, constant speed											
2. Level flight accelerat ion											
3. Level Flight, constant speed											
4. Climb Decelera tion											
5. Climb at Constant Speed											
6. Climb with Accelera tion											
7. Climb with Constant Speed											
8. Level Flight Constant Speed											

7. Deliverables:
   a. Table from Question #6
   b. Charts From Question #5
   c. Spreadsheet or Code

8. Input Profile
9.

Time	Thrust (lbs)	Flight Path Angle (degrees)
1	6404	0
2	6404	0
3	6404	0
4	6404	0
5	6404	0
6	6404	0
7	6404	0
8	6404	0
9	6404	0
10	6404	0
11	6404	0
12	6404	0
13	6404	0
14	6404	0
15	6404	0
16	6404	0
17	6404	0
18	6404	0
19	6404	0
20	6404	0
21	19500	0
22	19500	0
23	19500	0
24	19500	0
25	19500	0
26	19500	0
27	19500	0
28	19500	0
29	19500	0

30	19500	0
31	19500	0
32	19500	0
33	19500	0
34	19500	0
35	19500	0
36	19500	0
37	19500	0
38	19500	0
39	19500	0
40	19500	0
41	6404	0
42	6404	0
43	6404	0
44	6404	0
45	6404	0
46	6404	0
47	6404	0
48	6404	0
49	6404	0
50	6404	0
51	6404	0
52	6404	0
53	6404	0
54	6404	0
55	6404	0
56	6404	0
57	6404	0
58	6404	0
59	6404	0
60	6404	3

61	6404	3
62	6404	3
63	6404	3
64	6404	3
65	6404	3
66	6404	3
67	6404	3
68	6404	3
69	6404	3
70	11,750	3
71	11,750	3
72	11,750	3
73	11,750	3
74	11,750	3
75	11,750	3
76	11,750	3
77	11,750	3
78	11,750	3
79	11,750	3
80	11,750	3
81	11,750	3
82	11,750	3
83	11,750	3
84	11,750	3
85	11,750	3
86	11,750	3
87	11,750	3
88	11,750	3
89	11,750	3
90	24,500	3
91	24,500	3

92	24,500	3
93	24,500	3
94	24,500	3
95	24,500	3
96	24,500	3
97	24,500	3
98	24,500	3
99	24,500	3
100	24,500	3
101	24,500	3
102	24,500	3
103	24,500	3
104	24,500	3
105	24,500	3
106	24,500	3
107	24,500	3
108	24,500	3
109	24,500	3
110	24,500	3
111	24,500	3
112	24,500	3
113	24,500	3
114	24,500	3
115	24,500	3
116	24,500	3
117	24,500	3
118	24,500	3
119	24,500	3
120	24,500	3
121	24,500	3
122	24,500	3

123	24,500	3
124	24,500	3
125	24,500	3
126	24,500	3
127	24,500	3
128	24,500	3
129	24,500	3
130	24,500	3
131	24,500	3
132	24,500	3
133	24,500	3
134	24,500	3
135	12,000	3
136	12,000	3
137	12,000	3
138	12,000	3
139	12,000	3
140	12,000	3
141	12,000	3
142	12,000	3
143	12,000	3
144	12,000	3
145	12,000	3
146	12,000	3
147	12,000	3
148	12,000	3
149	12,000	3
150	12,000	3
151	12,000	3
152	12,000	3
153	12,000	3

154	12,000	3
155	12,000	3
156	12,000	3
157	6,404	3
158	6,404	0
159	6,404	0
160	6,404	0
161	6,404	0
162	6,404	0
163	6,404	0
164	6,404	0
165	6,404	0
166	6,404	0
167	6,404	0
168	6,404	0
169	6,404	0
170	6,404	0
171	6,404	0
172	6,404	0
173	6,404	0
174	6,404	0
175	6,404	0
176	6,404	0
177	6,404	0
178	6,404	0
179	6,404	0
180	6,404	0
181	6,404	0
182	6,404	0
183	6,404	0
184	6,404	0

185	6,404	0
186	6,404	0
187	6,404	0
188	6,404	0
189	6,404	0
190	6,404	0
191	6,404	0
192	6,404	0
193	6,404	0
194	6,404	0
195	6,404	0
196	6,404	0
197	6,404	0
198	6,404	0
199	6,404	0
200	6,404	0