Universidad Pontificia Bolivariana

Fundada en 1936

AIRCRAFT DESIGN I

Aeronautical Engineering

School of Engineering

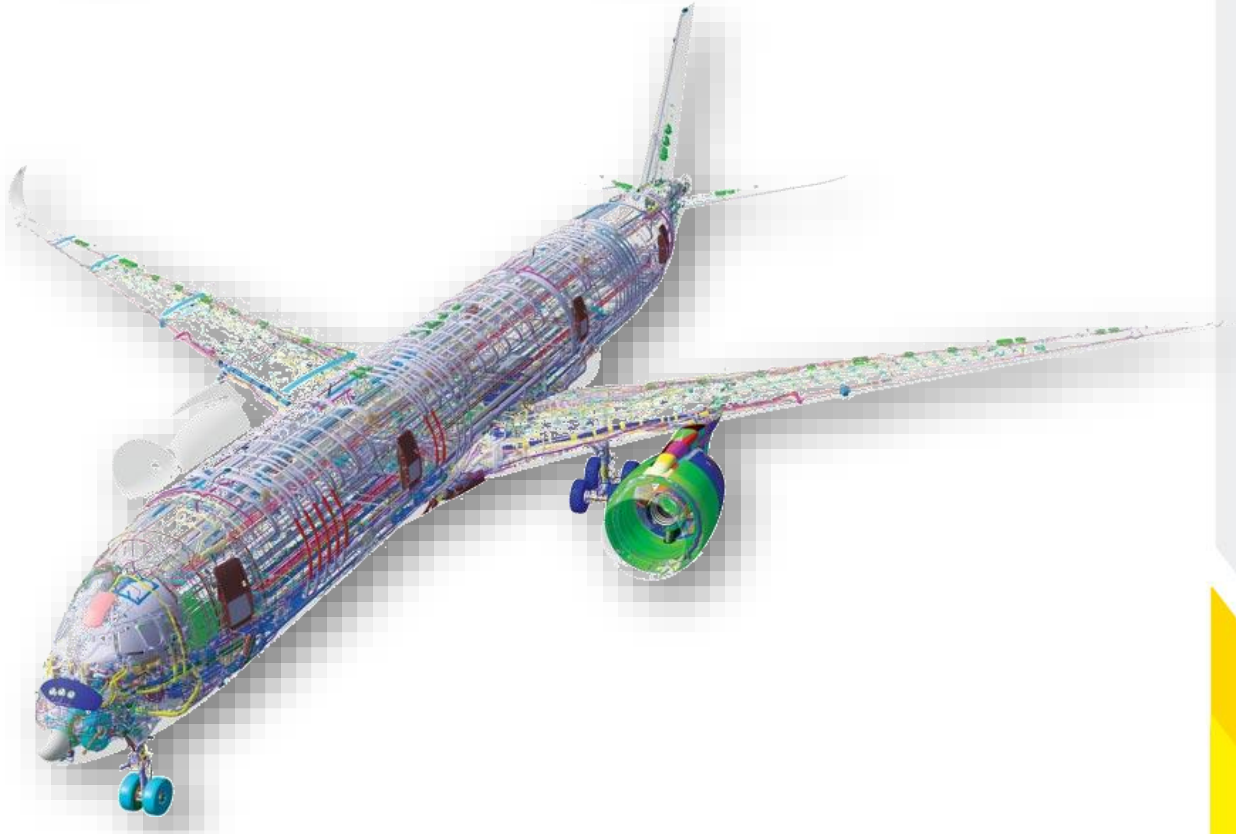
UPB

ENGINE SELECTION

Propulsion system selection.

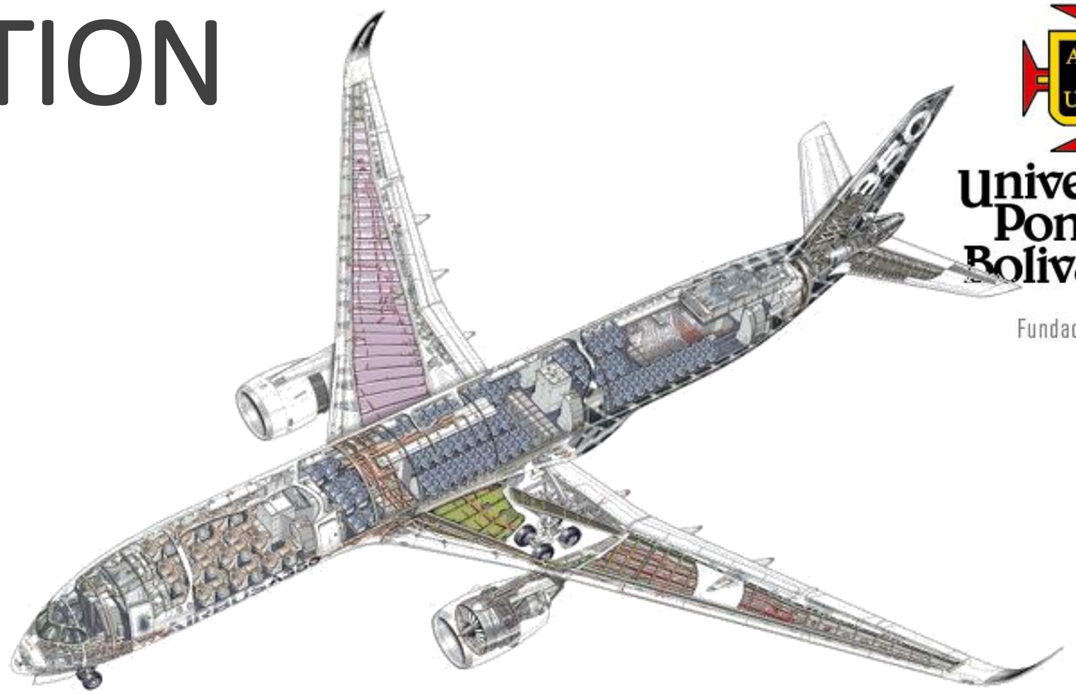
Table 8.1 Propulsion system functions

No.	Category of function	Function
1	Primary function	Generate propulsive force
2	Secondary function	Generate power/energy for various aircraft subsystems such as hydraulic and electric systems
3	Contributing function	Either stabilizing or destabilizing Reduces the comfort of the passengers, crew, and flight attendants due to engine noise Reduces the comfort of the passengers, crew, and flight attendants due to heat exchange to cabin/cockpit Safety contributions in case of one engine inoperative Operating cost due to fuel consumption Structural impact due to engine vibrations



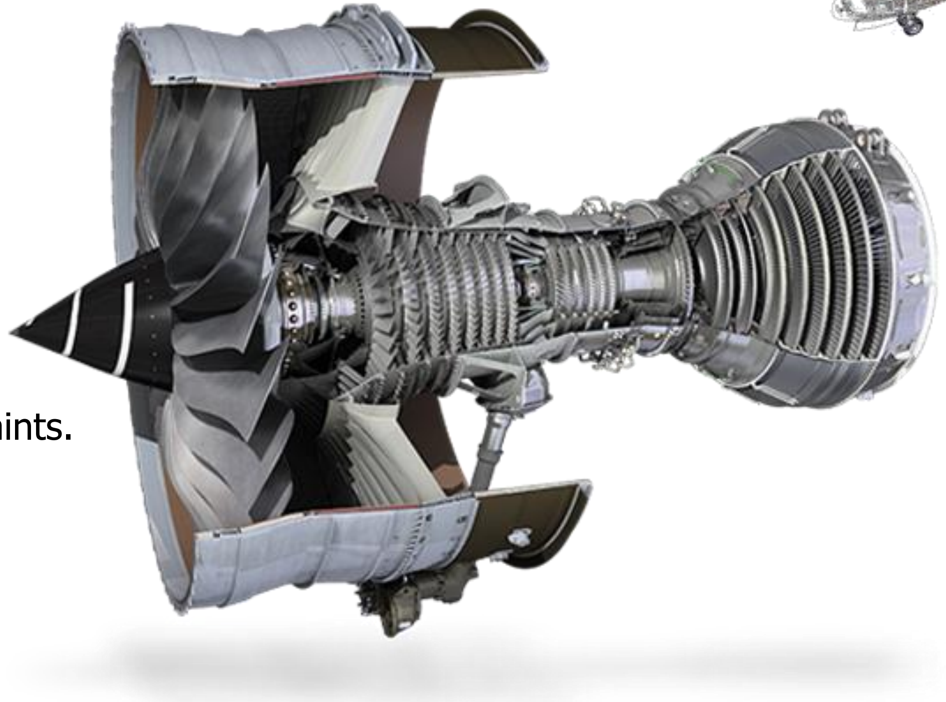
Universidad Pontificia Bolivariana
Fundada en 1936

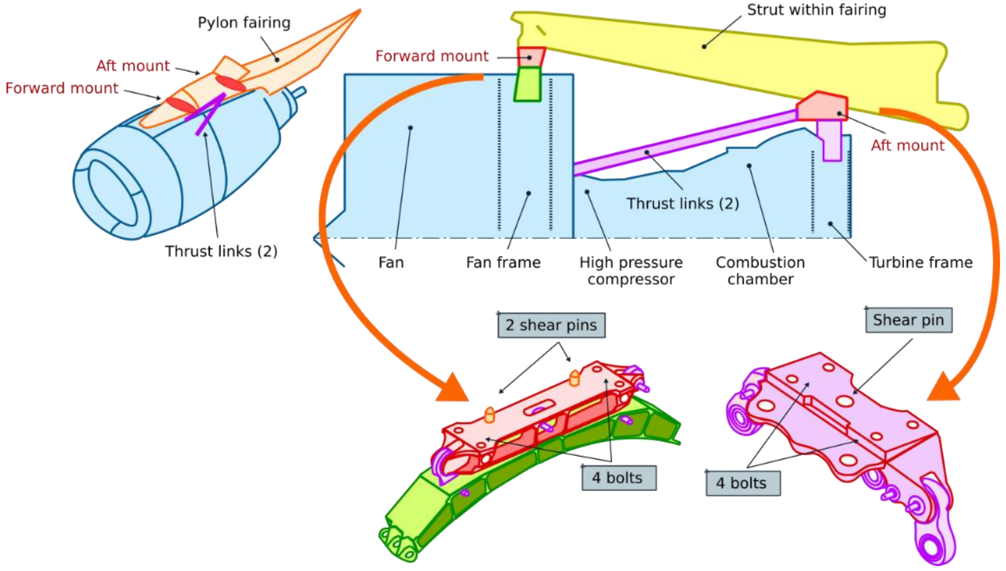
ENGINE SELECTION



In general, the following items are considered as the propulsion system design requirements and constraints:

- Aircraft performance.
- Engine cost.
- Operating cost.
- Engine weight constraints.
- Size constraints.
- Flight safety.
- Engine efficiency.
- Aircraft stability.
- Heat exchange.
- Structural requirements.
- Installation constraints.
- Integration.
- Noise constraints.
- Passenger comfort.
- Passenger appeal.
- Stealth constraints.
- Engine frontal area constraints.
- Maintainability.
- Manufacturability.
- Disposability.





ENGINE SELECTION

Engine types.

ENGINE SELECTION

Engine types.

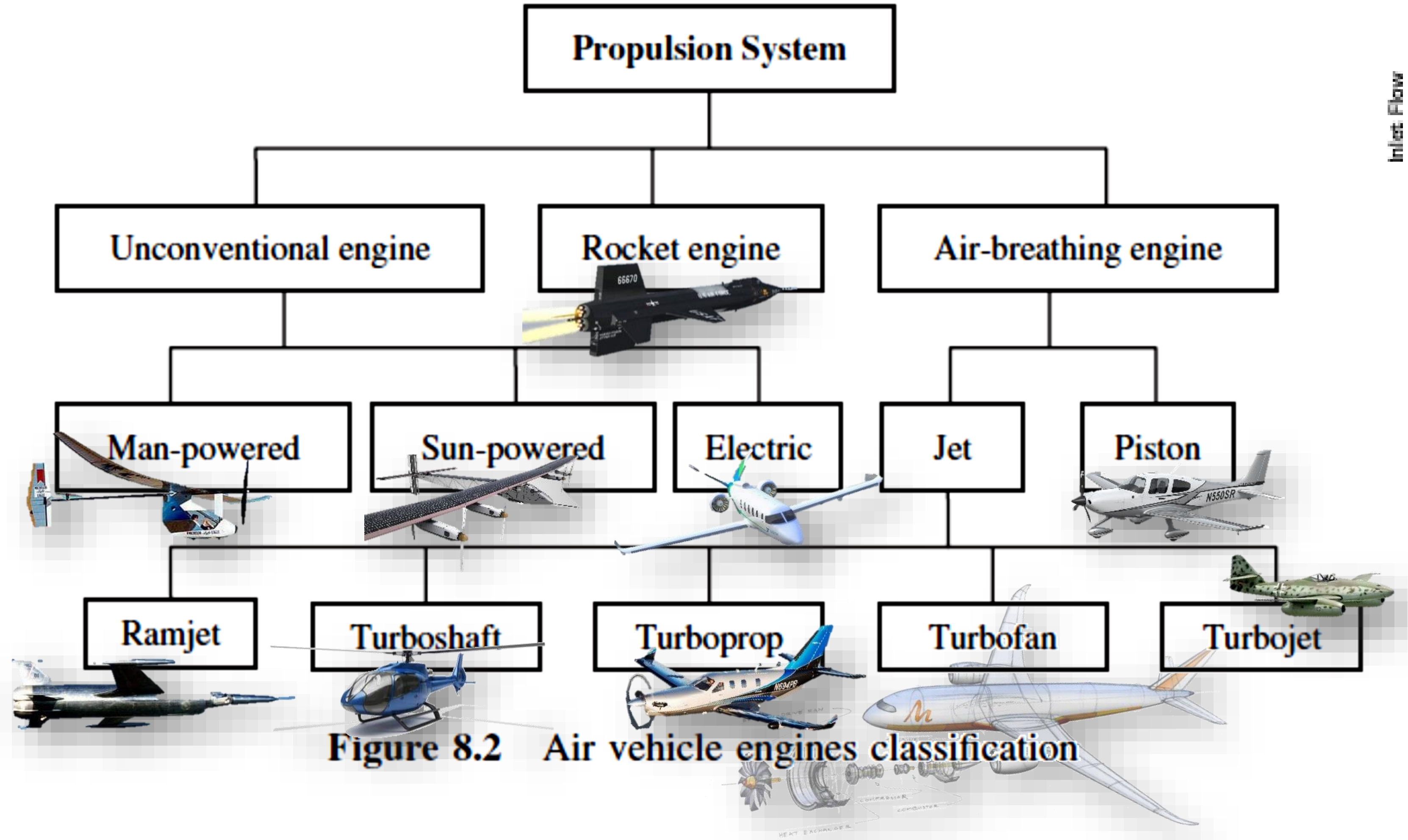
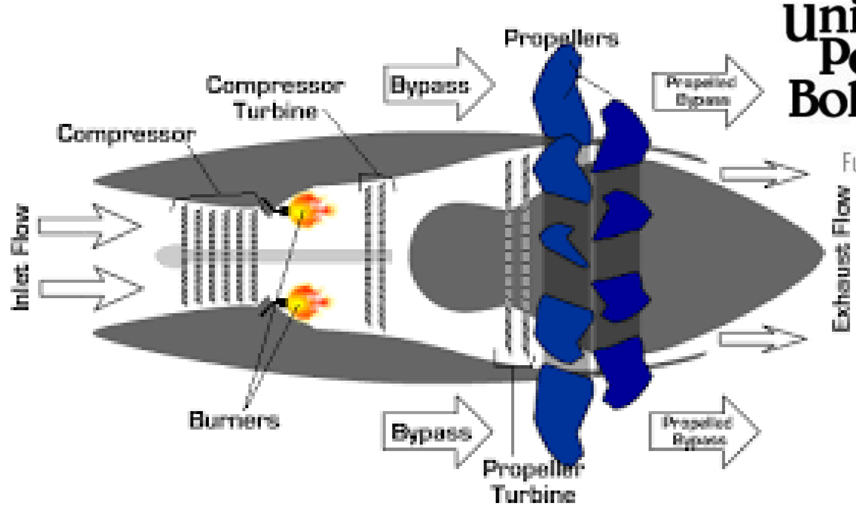
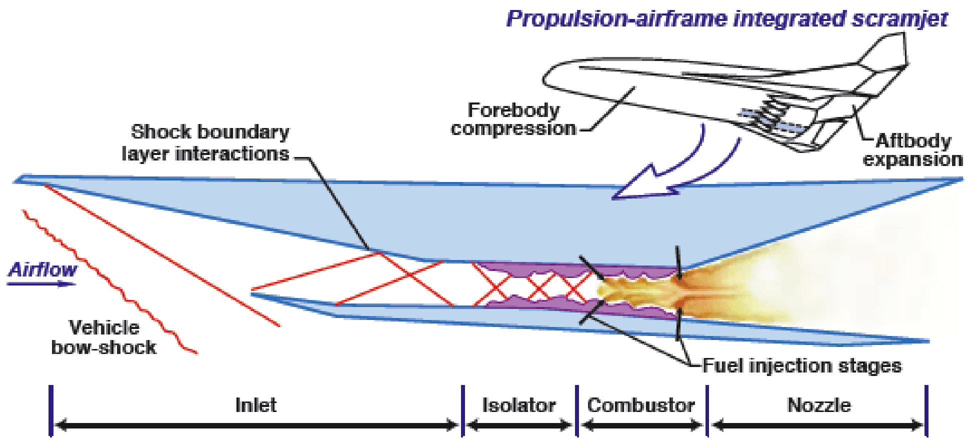


Figure 8.2 Air vehicle engines classification



Propfan or unducted fan



Scramjet

ENGINE SELECTION

Universidad Pontificia Bolivariana

Fundada en 1936

Engine types.

Table 8.3 General comparison of various parameters for 10 different engines

No.	Engine	SFC	Engine cost	Noise	Specific weight	Propulsive efficiency	Maintainability	Ceiling	Aircraft speed
1	Human-powered	0	1[a]	1	1	8	10	1	1
2	Electric	1	2	3	2[b]	10	9	8	2
3	Solar-powered	0	2[c]	2	2	9	8	9	3
4	Piston-prop	2	3	4	5	3	5	2	5
5	Turbojet	6	6	5	8	6	1	6	8
6	Turbofan	5	9	6	9	7	2	5	7
7	Turboprop	4	7	7	6	4	4	4	6
8	Turboshaft	3	8	8	7	5	3	3	4
9	Ramjet	8	4	6	4	2	6	7	9
10	Rocket	10	5–8	10	3[d]	1	7	10	10

[a] This does not imply that human is cheap, but it means that the pilot does not need to purchase an engine.
[b] Without battery.
[c] Excluding solar panels.
[d] Excluding internal fuel.
1: lowest, 10: highest

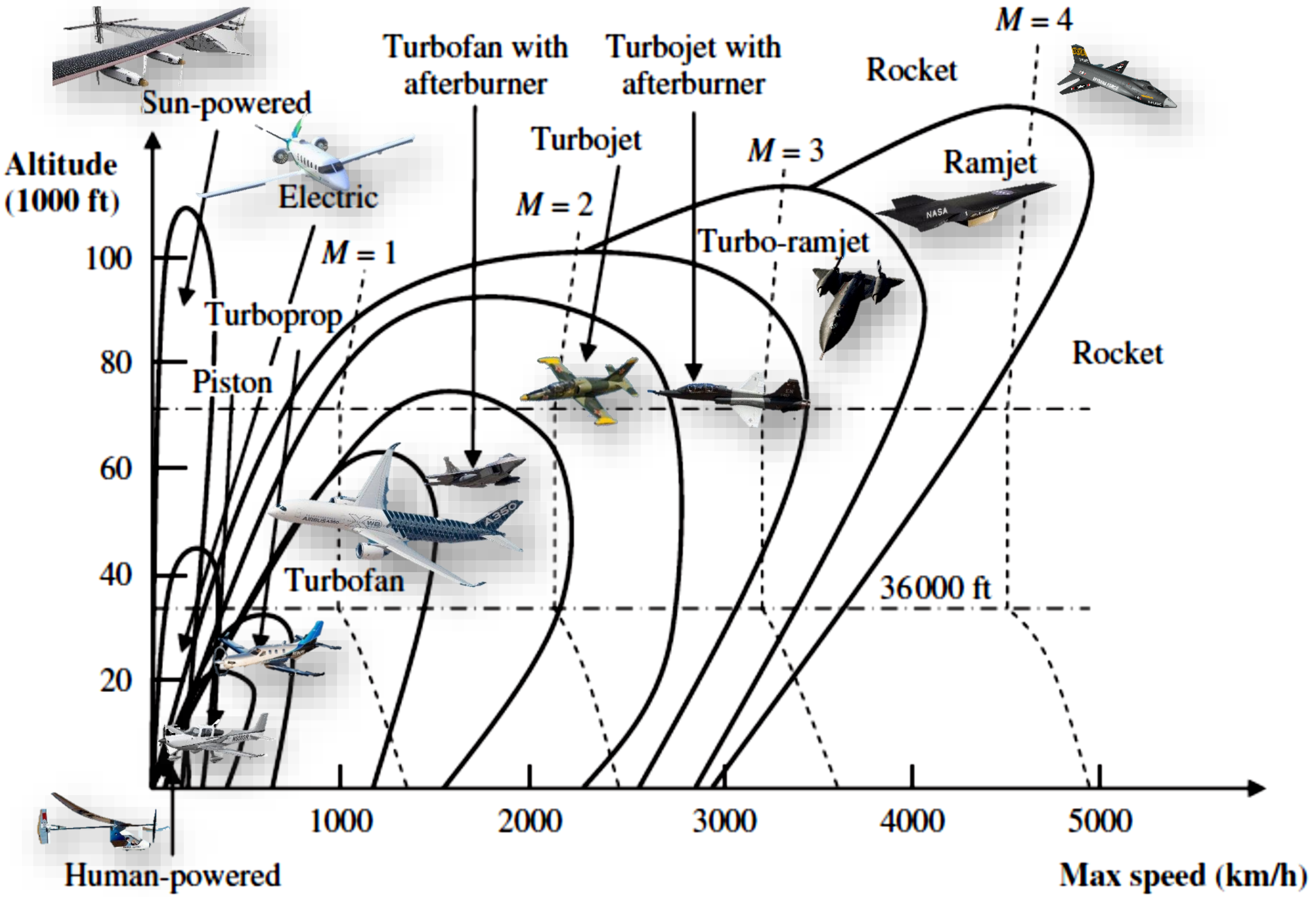
ENGINE SELECTION

Factors that play a role in selecting the type of propulsion system:

- Required cruise speed and/or maximum speed.
- Required maximum operating altitude.
- Required range and range economy.
- FAR 36 noise regulations (only civil aviation).
- Installed weight.
- Reliability and maintainability.
- Fuel amount needed.
- Fuel cost.
- Fuel availability.
- Specific customer or market demands.
- Timely certification.
- Environmental issues (requirements).

Universidad Pontificia Bolivariana
Fundada en 1936

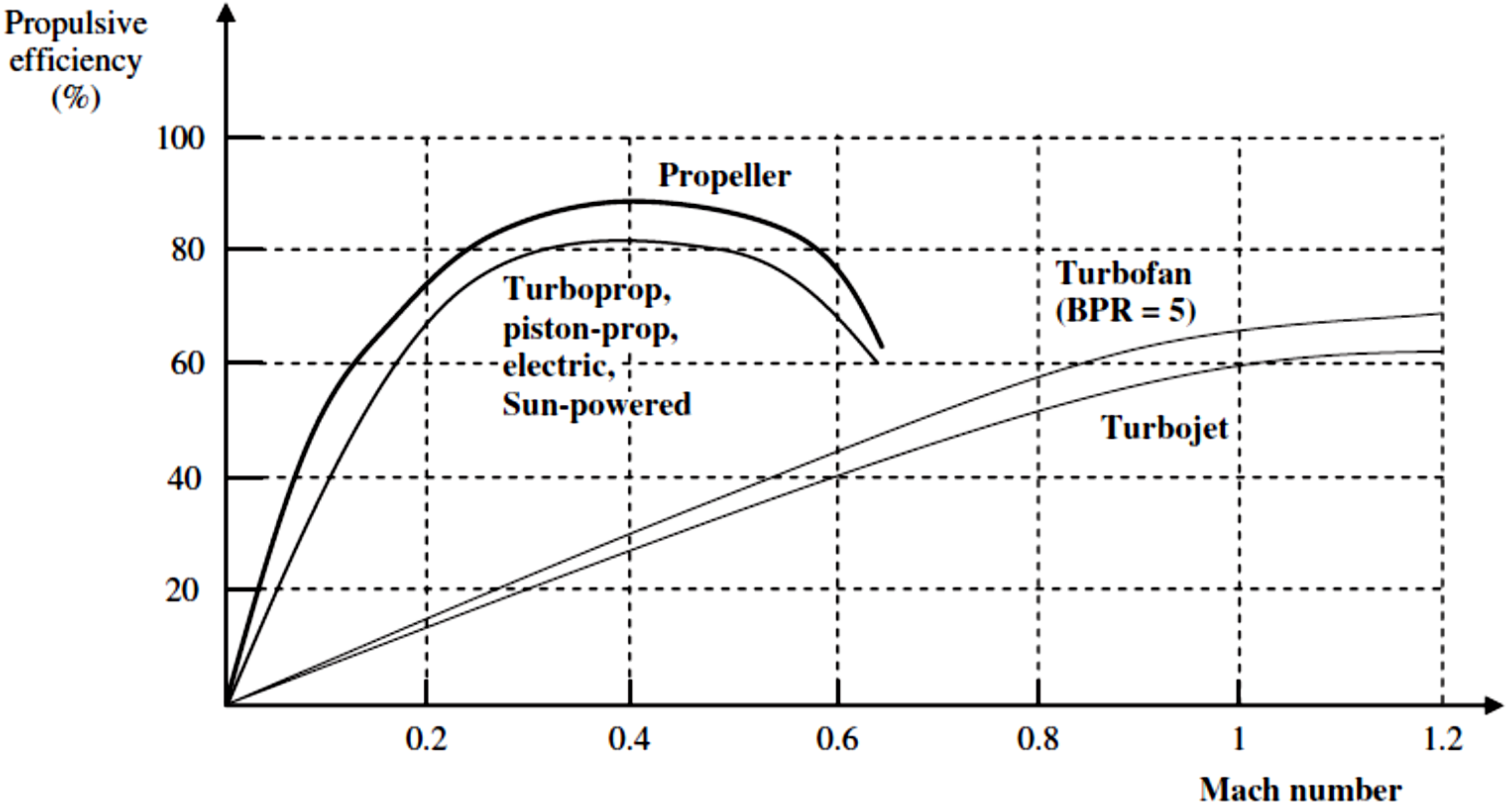
ENGINE SELECTION



Absolute ceiling and Mach number

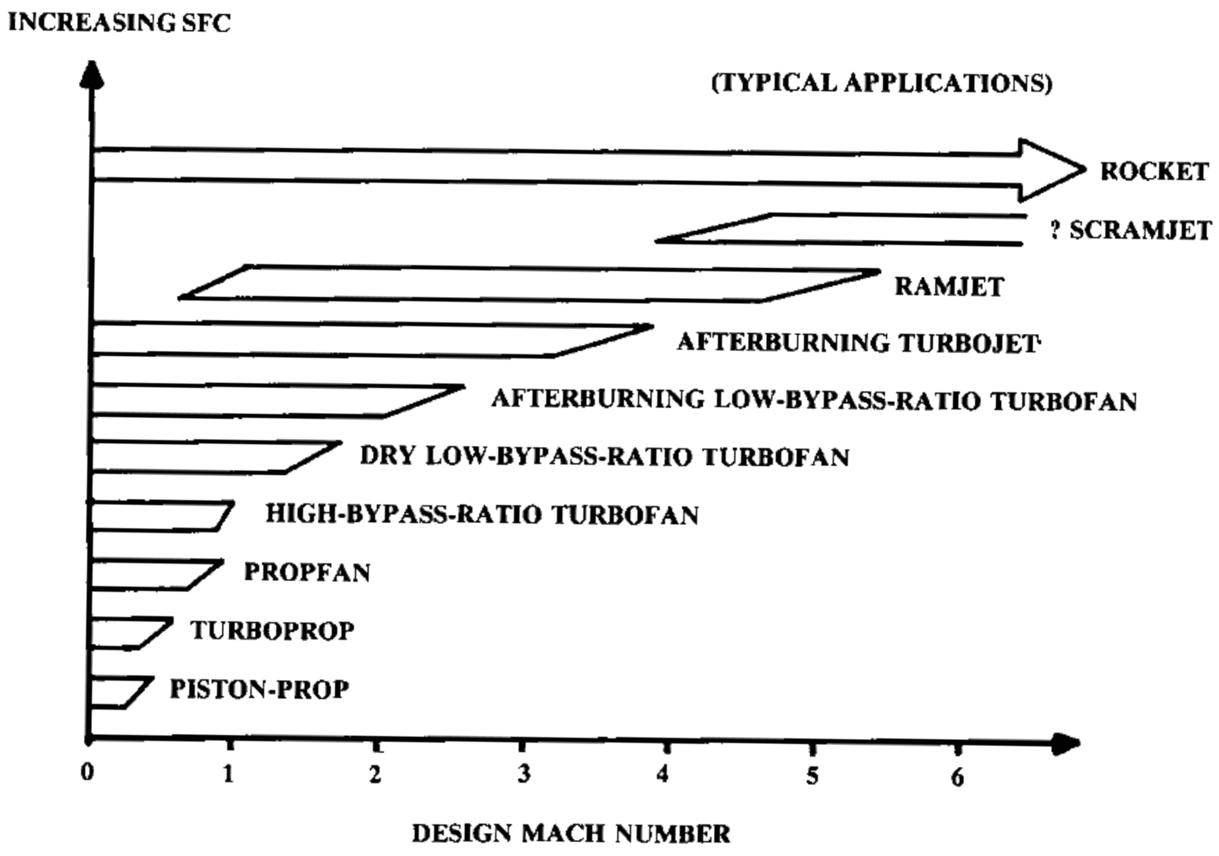
ENGINE SELECTION

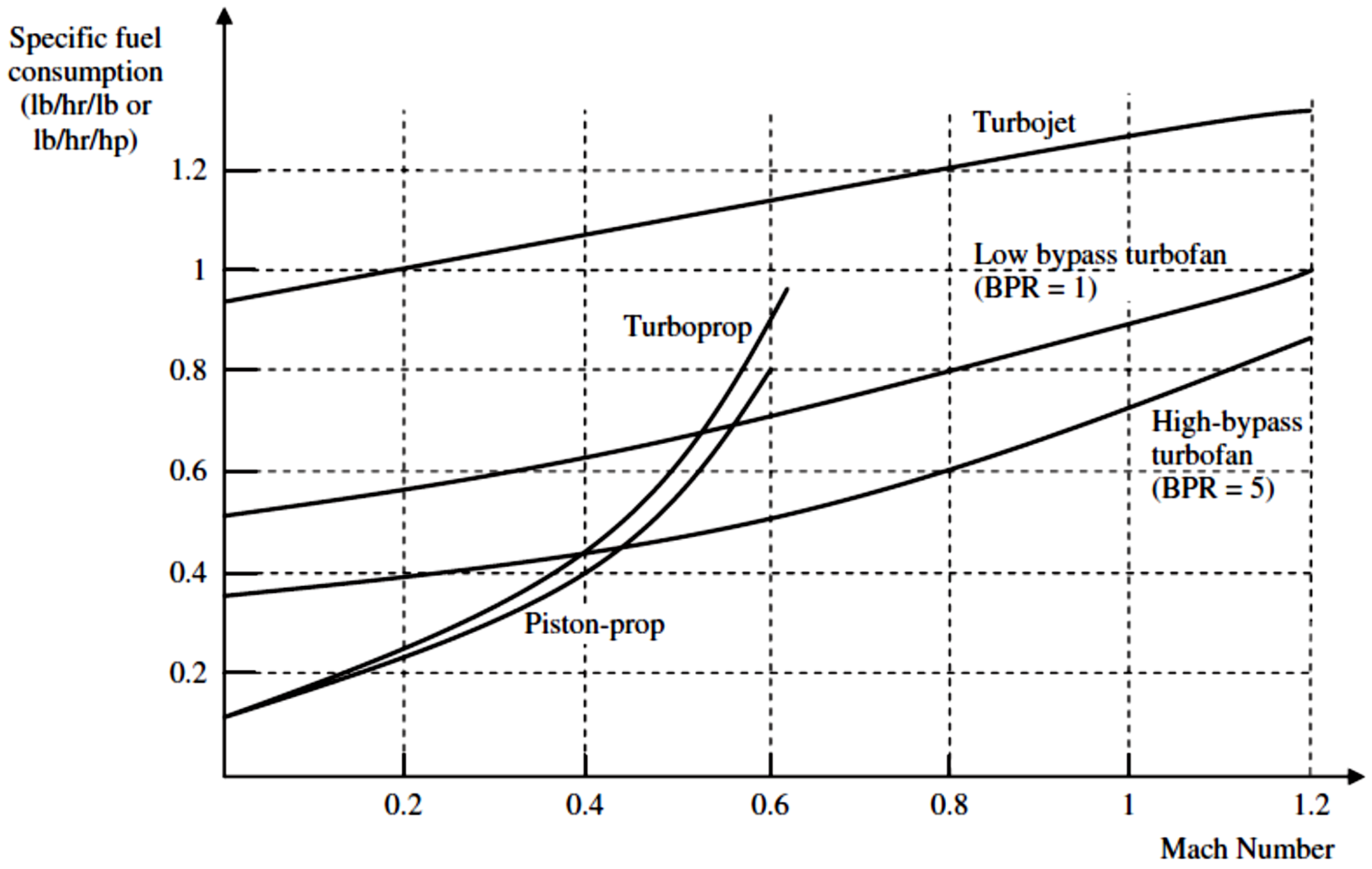
Propulsive efficiency.



ENGINE SELECTION

SFC.





ENGINE SELECTION

Integration of the propulsion system.

Having decided on the type and the number of engines to be employed, the next decision is the location of the engines.

The number of possible arrangements is very large.

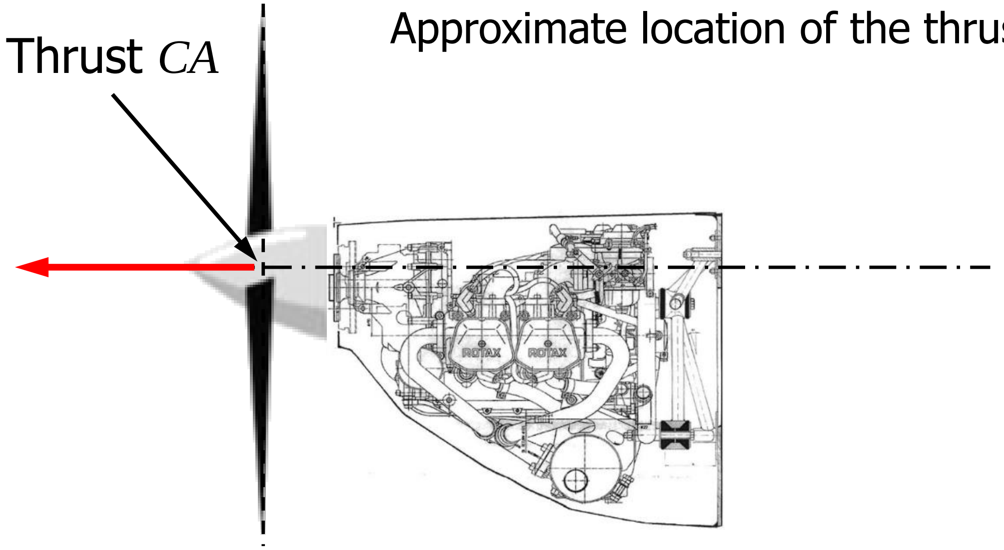
The following factors play a role in deciding on the engine disposition:

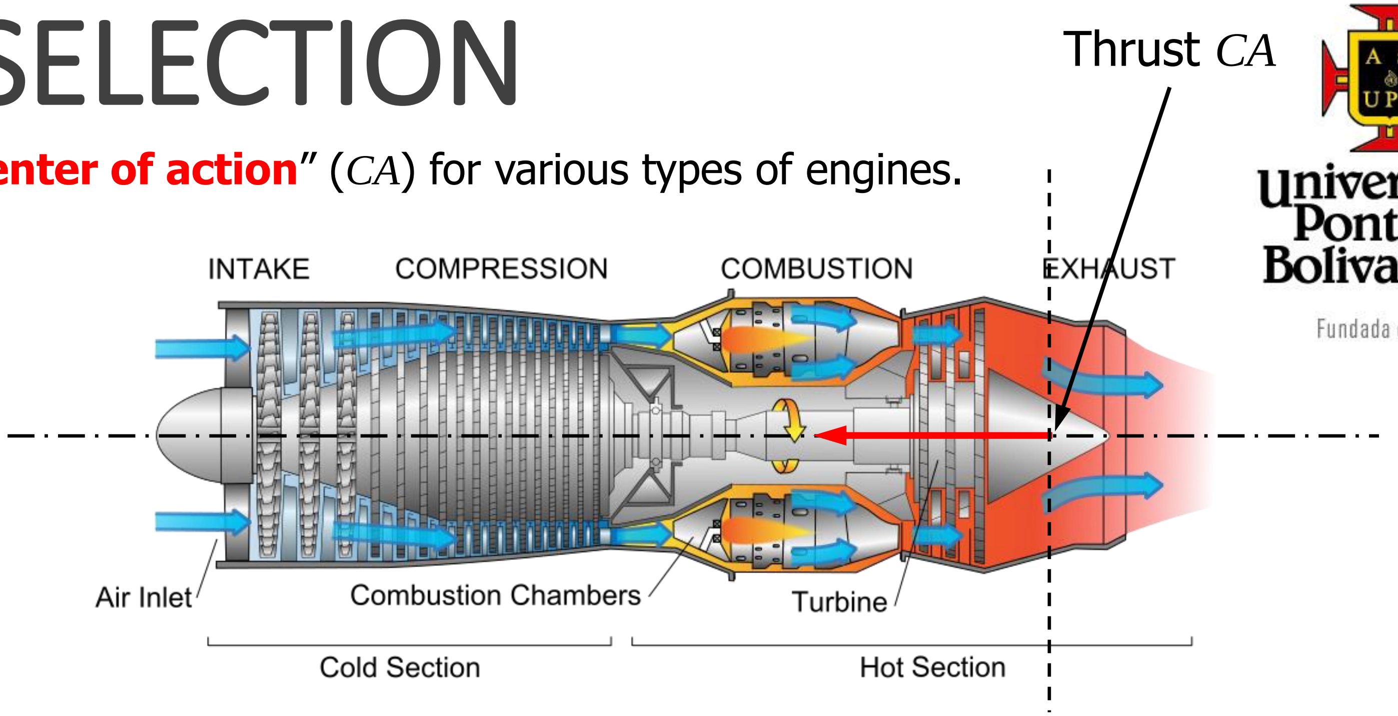
1. Effect of power changes or power failures on stability and control: longitudinal, lateral and directional. The vertical and/or lateral location of the thrust line(s) are critically important in this respect.
2. Drag of the proposed installation.
3. Inlet requirements and resulting effect on "installed" power and efficiency.
4. Accessibility and maintainability.

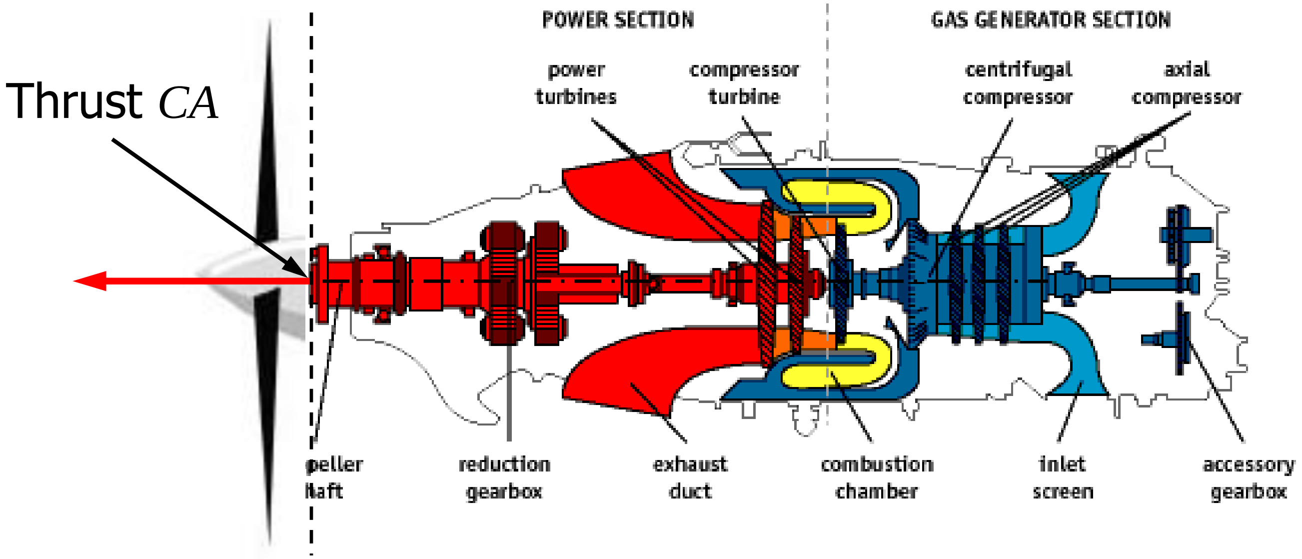
Universidad Pontificia Bolivariana
Fundada en 1936

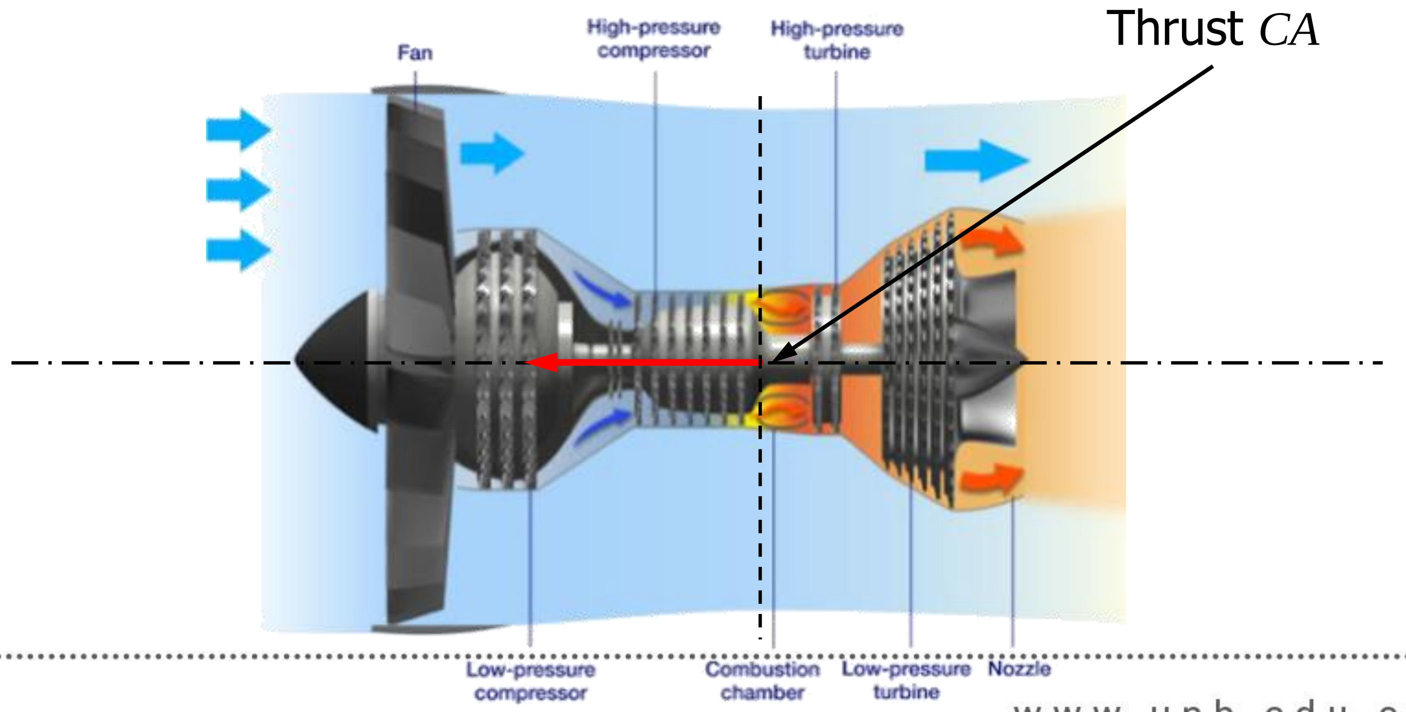
ENGINE SELECTION

Approximate location of the thrust “**center of action**” (CA) for various types of engines.



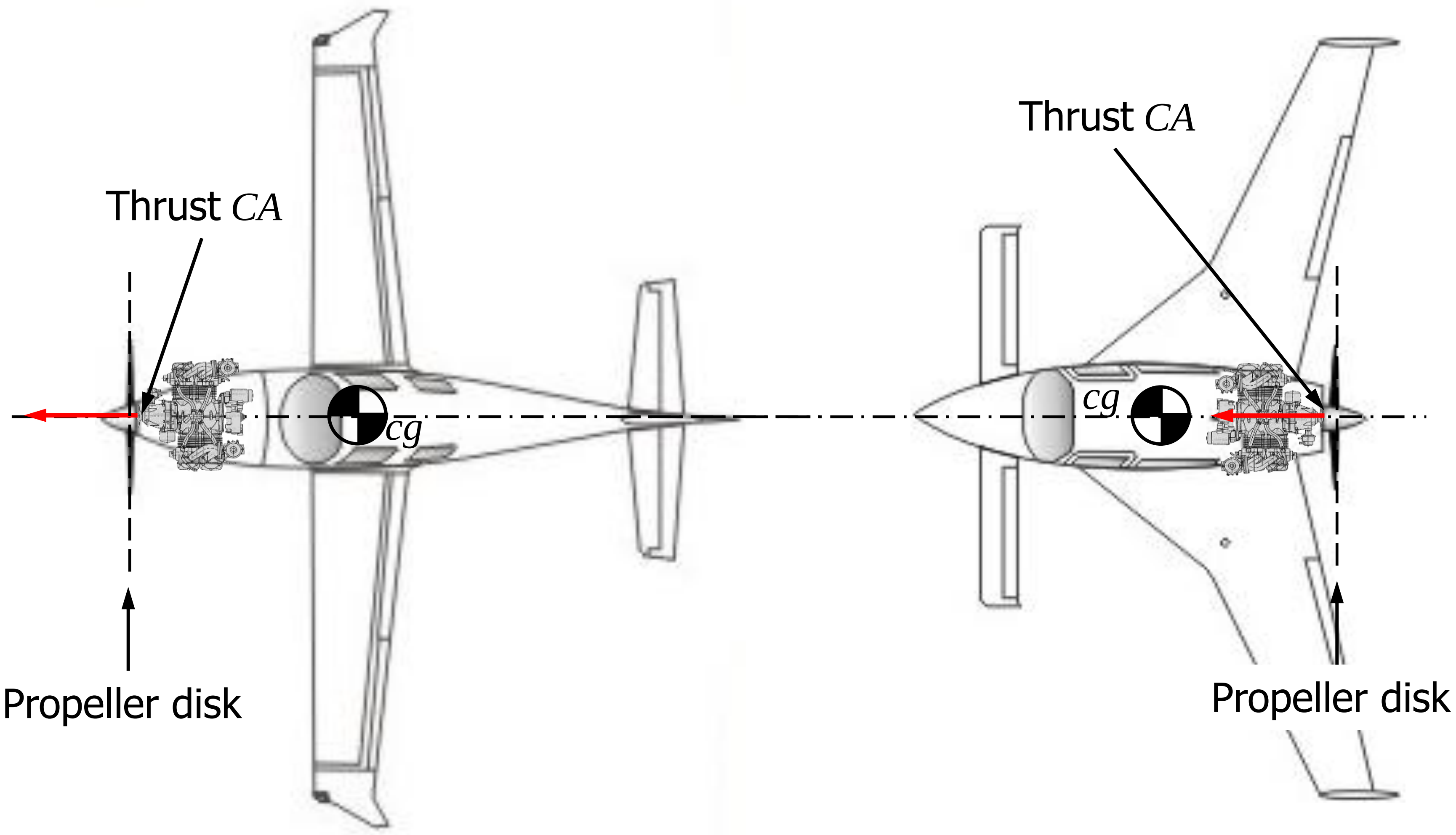






ENGINE SELECTION

Pusher versus tractor.



Tractor – where the engine is located ahead the aircraft center of gravity.

Pusher – where the engine is located behind the aircraft center of gravity.

1. The pusher engine moves the aircraft cg rearward.
2. The tractor engine moves the aircraft cg forward.
3. The net thrust of a prop-driven tractor engine is slightly greater than that of a prop-driven pusher engine.
4. In a take-off operation, a pusher aircraft usually requires less elevator deflection than a tractor aircraft.
5. Due to the relationship between aircraft cg and engine thrust, a tractor configuration provides better directional stability than a pusher configuration.
6. Both pusher and tractor configurations impose some limits on other component configurations.

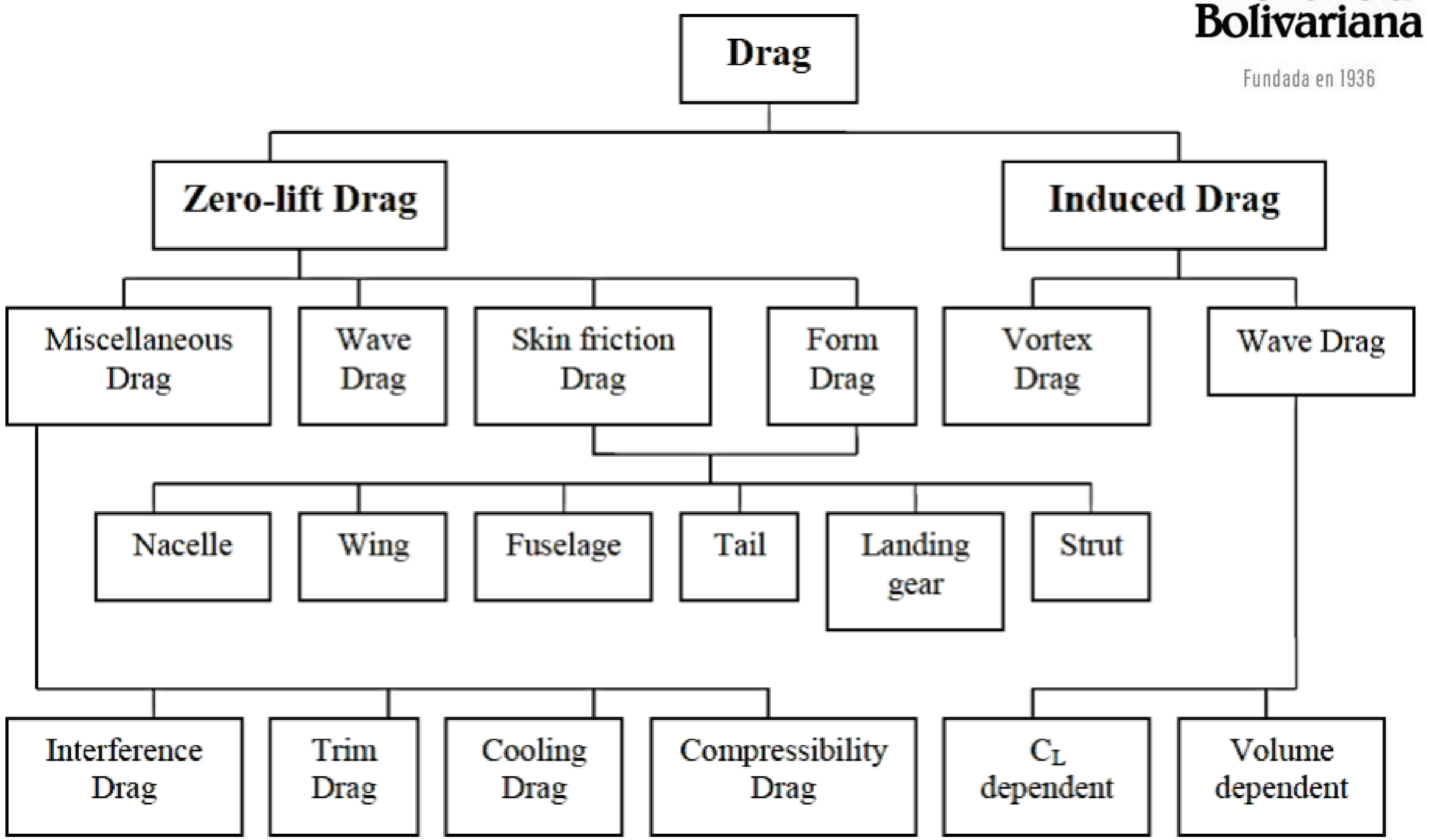
ENGINE SELECTION

Aircraft drag calculation.

$$D_{a/ctotal} = D_0 + D_i \Rightarrow C_D = C_{D_0} + \frac{C_L^2}{\pi \cdot e \cdot AR}$$

$$e = 1.78(1 - 0.045AR^{0.68}) - 0.64$$

$$e = 4.61(1 - 0.045AR^{0.68})[\cos(\Lambda_{LE})]^{0.15} - 3.1$$

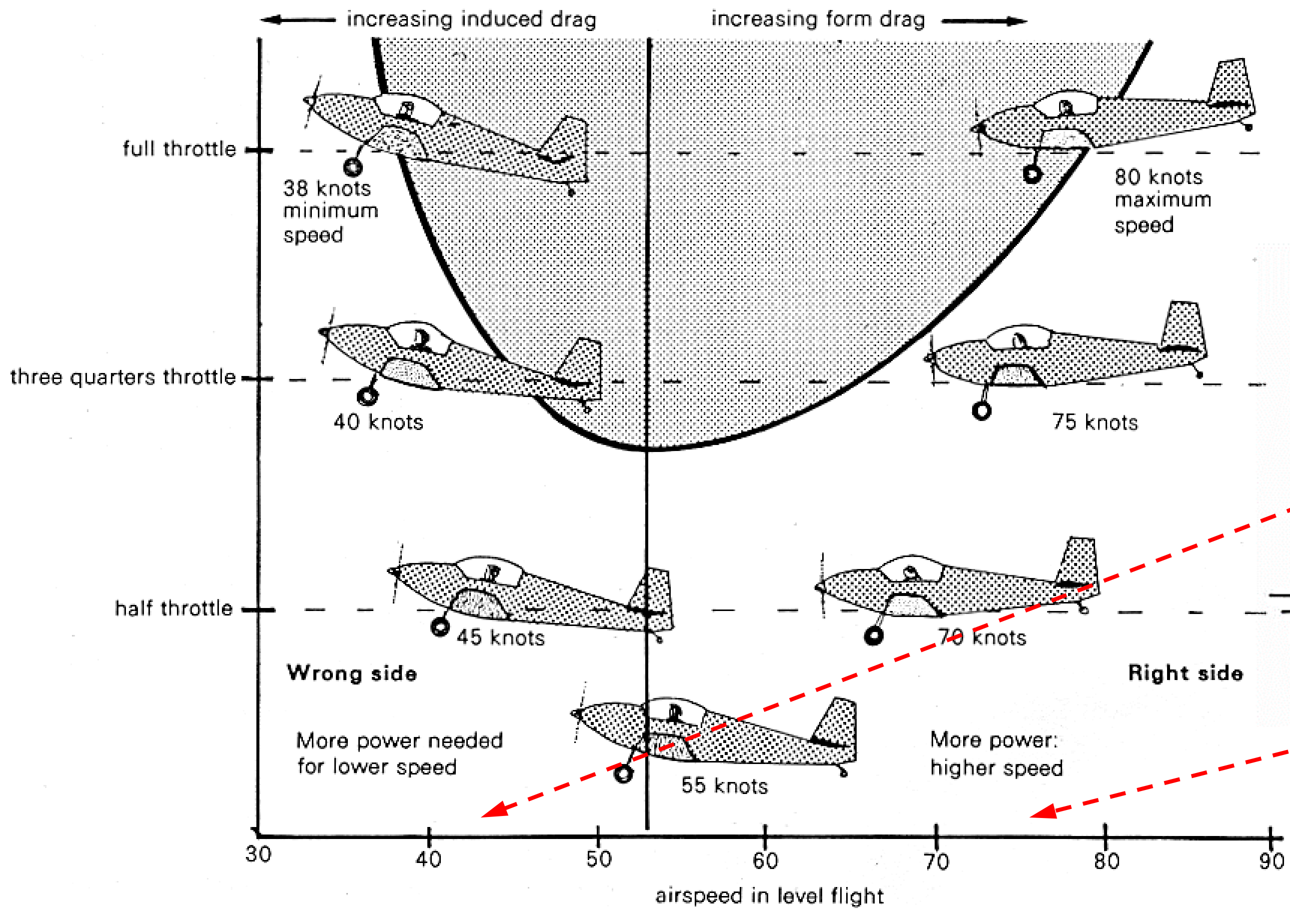


ENGINE SELECTION

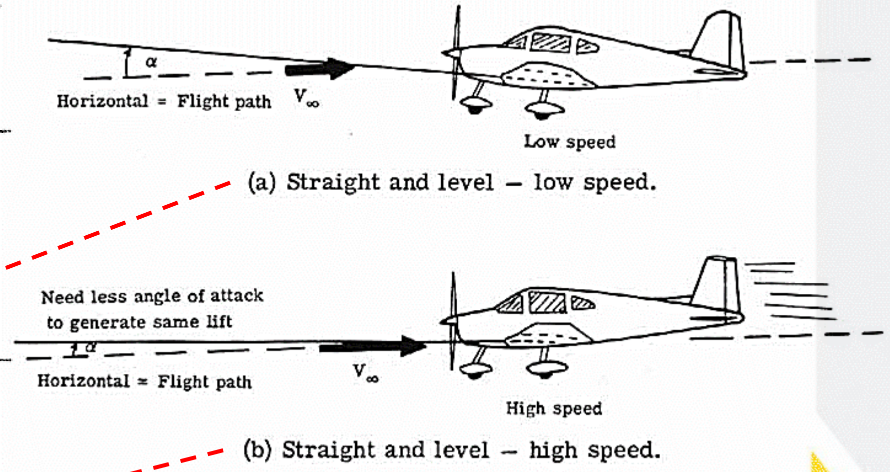
Aircraft drag calculation.

The lift-to-drag (L/D) ratio is a good measure of aerodynamic efficiency.

Minimizing the drag has been one of the strongest drivers in the historical development of applied aerodynamics.



$$C_D = C_{D_0} + \frac{C_L^2}{\pi \cdot e \cdot AR}$$



ENGINE SELECTION

Aircraft drag calculation – method with empirical equations (document in *moodle*).

$$C_{D_0} = C_{D_{0_f}} + C_{D_{0_w}} + C_{D_{0_{ht}}} + C_{D_{0_{vt}}} + C_{D_{0_{LG}}} + C_{D_{0_N}} + C_{D_{0_S}} + C_{D_{0_{HLD}}} + \ldots$$

$C_{D_{0\_f}}$ - fuselage contribution.
$C_{D_{0\_w}}$ - wing contribution.
$C_{D_{0\_ht}}$ - horizontal tail contribution.
$C_{D_{0\_vt}}$ - vertical tail contribution.
$C_{D_{0\_LG}}$ - landing gear contribution.
$C_{D_{0\_N}}$ - nacelle contribution.
$C_{D_{0\_S}}$ - strut contribution.
$C_{D_{0\_HLD}}$ - high lift devices contribution.

Other non-significant components such as: antenna(s), pitot tube, stall horn, wires, interference, and wiper.

In most conventional aircraft, wing and fuselage are each contributing about 30%-40% (totally 60%-80%) to the aircraft C_{D_0}. All other components are contributing around 20%-40% to C_{D_0} of an aircraft.

ENGINE SELECTION

Universidad Pontificia Bolivariana

Fundada en 1936

Aircraft drag calculation.

Examples of typical ranges values of "C_{D_0}" and "e" for several aircraft.

No	Aircraft type	C_{D_o}	e
1	Twin-engine piton prop	0.022-0.028	0.75-0.8
2	Large turboprop	0.018-0.024	0.8-0.85
3	Small GA with retractable landing gear	0.02-0.03	0.75-0.8
4	Small GA with fixed landing gear[1]	0.025-0.04	0.65-0.8
5	Agricultural aircraft with crop duster	0.07-0.08	0.65-0.7
6	Agricultural aircraft without crop duster	0.06-0.065	0.65-0.75
7	Subsonic jet	0.014-0.02	0.75-0.85
8	Supersonic jet	0.02-0.04	0.6-0.8
9	Glider	0.012-0.015	0.8-0.9
10	Remote controlled model aircraft	0.025-0.045	0.75-0.85

ENGINE SELECTION

Aircraft drag calculation.

$C_{D_{0\_corrected}}$ – Corrected coefficient

The calculation of C_{D_0} contributions due to factors is most of the times optimistically made, this because these factors are responsible for an increase up to about 50%, due to this fact a correction factor should be included.

$$C_{D_{0\_corrected}} = K_c C_{D_0} = K_c\left[C_{D_{0_f}} + C_{D_{0_w}} + C_{D_{0_{ht}}} + C_{D_{0_{vt}}} + C_{D_{0_{LG}}} + C_{D_{0_N}} + C_{D_{0_S}} + C_{D_{0_{HLD}}} + \dots\right]$$

K_c is a correction factor and depends on several factors such as the type, year of fabrication, degree of streamline-ness of fuselage, configuration of the aircraft, and number of miscellaneous items.

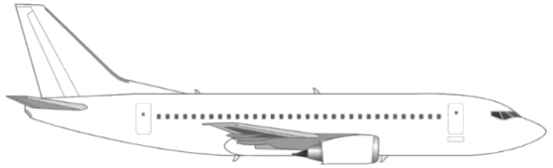
No	Aircraft type	K_c
1	Jet transport	1.1
2	Agriculture	1.5
3	Prop Driven Cargo	1.2
4	Single engine piston	1.3
5	General Aviation	1.2
6	Fighter	1.1
7	Glider	1.05
8	Remote Controlled	1.2

ENGINE SELECTION

Aircraft drag calculation

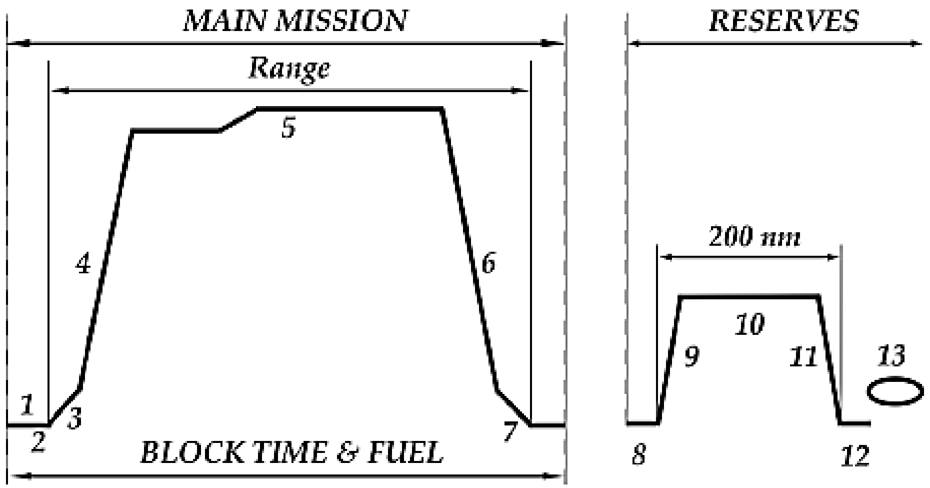
When an aircraft retracts its landing gear, deflects its flap, rotates its control surfaces, exposes any external component (such as gun), releases its store (e.g., missile), or opens its cargo door; it has changed its configuration.

In general, there are three configuration groups that an aircraft may adopt:

1. clean configuration. 

2. take-off configuration. 

3. landing configuration. 



1. Warm-up
2. Taxi
3. Take-off & Climb out
4. Climb
5. Stepped Cruise
6. Descent
7. Land & Taxi
8. Landing overshoot
9. Economy Climb
10. Diversion Cruise
11. Descent
12. Land & Taxi
13. Hold @ 5.000 ft

ENGINE SELECTION

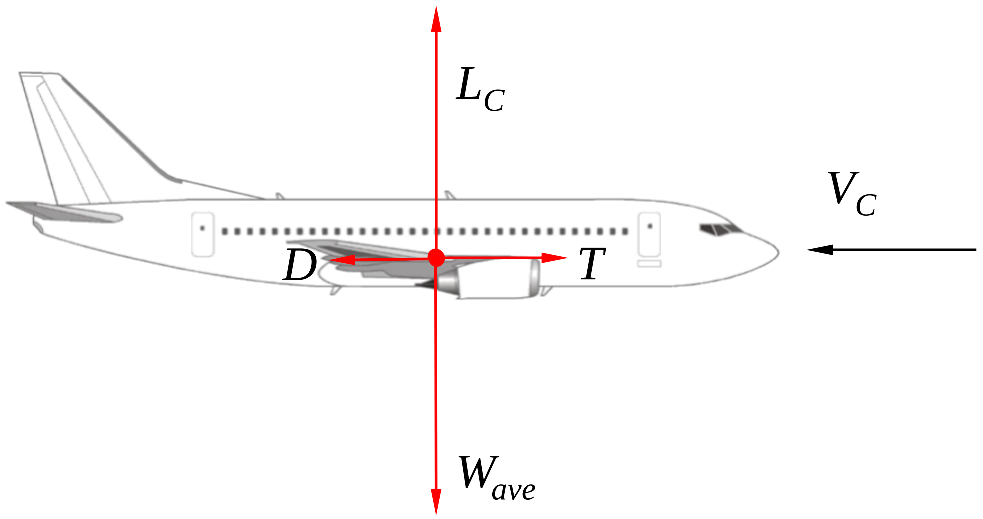
Aircraft drag calculation.

1. Clean configuration.

Aircraft is at a cruise flight condition.

At this configuration, no flap is deflected, and landing gear is retracted (if it is retractable).

$C_{D_{0\_clean}}$ includes every component (such as wing, tail, and fuselage), and excludes flap and landing gear (if retractable).



$$C_{D_{clean}} = C_{D_{0_{clean}}} + K C_{L_C}^2$$

$$C_{L_C} = \frac{2W_{ave}}{\rho_\infty V_C^2 S} \qquad K = \frac{1}{\pi e AR}$$

$$W_{ave} = \frac{W_i + W_f}{2}$$

ENGINE SELECTION

Aircraft drag calculation.

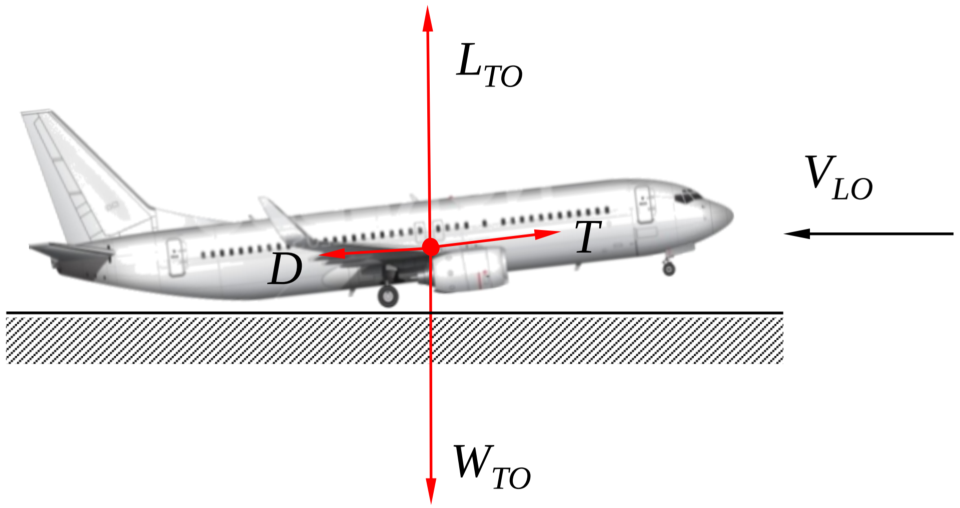
2. Take-off configuration..

The aircraft has high angle of attack, flap is deflected for take-off, and landing gear is not retracted (even if it is retractable).

$C_{D_{0\_TO}}$ depends on the flap type and the deflection angle.

$\delta_f \approx$ 10-30 deg

$K_{LO} = 1.1 \rightarrow 1.3$



$$C_{D_{TO}} = C_{D_{0_{TO}}} + KC_{L_{TO}}^2$$

$$C_{D_{0_{TO}}} = C_{D_{0_{clean}}} + C_{D_{0_{HLD\_TO}}} + C_{D_{0_{LG}}}$$

$$C_{L_{TO}} \cong 0.9\frac{2W_{TO}}{\rho_\infty V_{LO}^2 S}$$

$$V_{LO} = K_{LO}V_s$$

ENGINE SELECTION

Aircraft drag calculation.

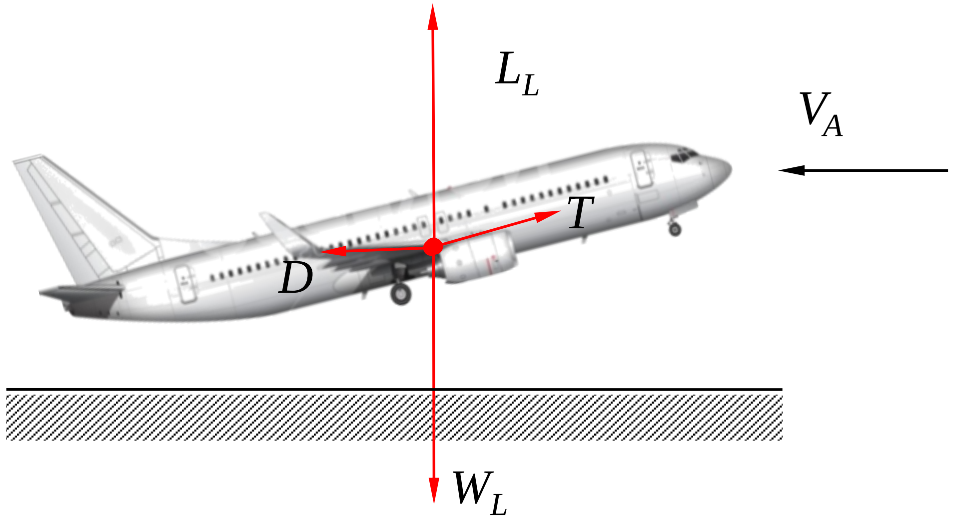
3. Landing configuration.

Aircraft has high angle of attack (even more than take-off condition), flap is deflected (even more than take-off condition), and landing gear is not retracted (even if it is retractable).

$C_{D_{0\_L}}$ depends on the flap type and the deflection angle.

$\delta_f \approx 30\text{-}60\ deg$

$K_A = 1.1 \rightarrow 1.3$



$$C_{D_L} = C_{D_{0_L}} + K C_{L_L}^2$$

$$C_{D_{0_L}} = C_{D_{0_{clean}}} + C_{D_{0_{HLD\_L}}} + C_{D_{0_{LG}}}$$

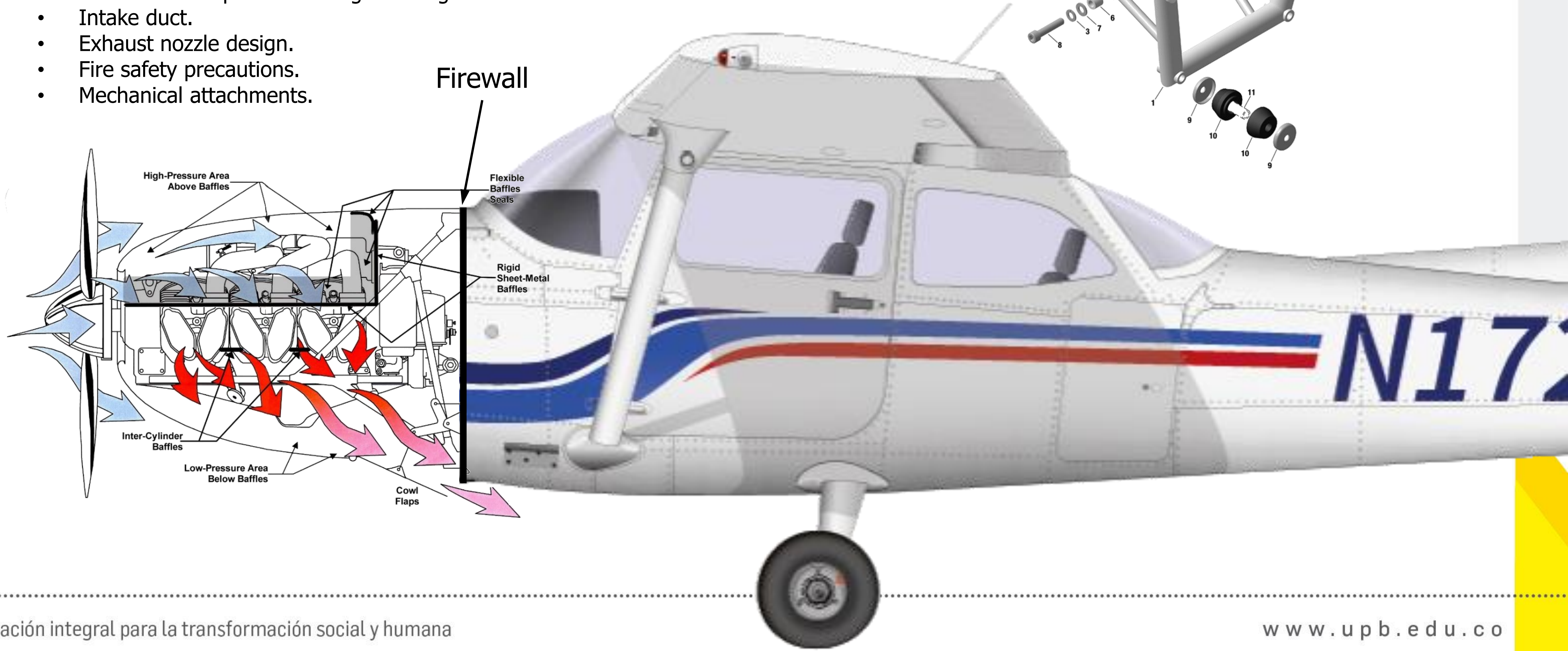
$$C_{L_L} \cong \frac{2W_{TO}}{\rho_\infty V_A^2 S}$$

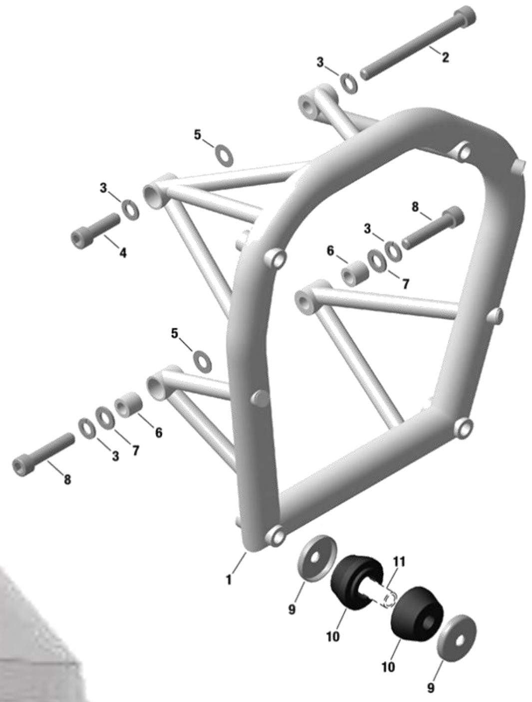
$$V_A = K_A V_s$$

ENGINE SELECTION

Engine installation challenges primarily include the engine:

- Cooling provision.
- Cabin and cockpit isolation against engine heat.
- Intake duct.
- Exhaust nozzle design.
- Fire safety precautions.
- Mechanical attachments.



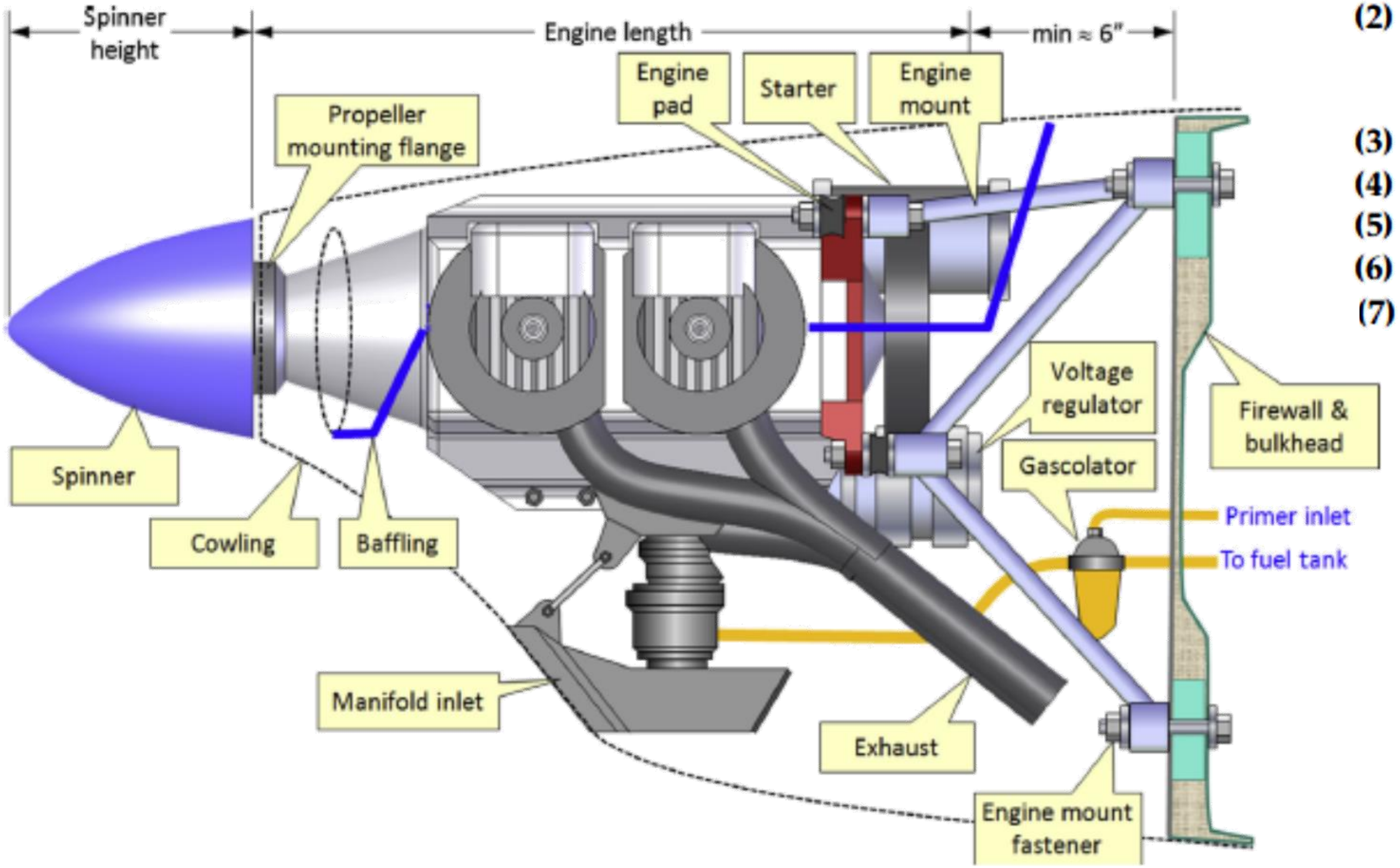


Universidad Pontificia Bolivariana

Fundada en 1936

PISTON ENGINE

Systems integration:



(1) Starter and ignition switch wiring.
(2) Battery, which is often inside the engine compartment, unless it serves a secondary purpose as ballast.
(3) Voltage regulator.
(4) Primer inlet and fuel lines.
(5) Mixture control.
(6) Throttle control.
(7) Carburetor heat control, unless the engine features fuel-injection technology.

Instruments (gages):

(1) Oil pressure gage.
(2) Oil temperature gage.
(3) Tachometer (RPM indicator).
(4) Manifold pressure gage (MAP – often omitted for low-powered engines).
(5) Fuel tank quantity gages.
(6) Fuel flow indicator (omitted for low-performance aircraft).
(7) Hobbs indicator (shows the number of hours on the engine).

PISTON ENGINE

Universidad Pontificia Bolivariana

Fundada en 1936

Engine/Motor installation.

(a) For the purpose of this part, the airplane powerplant installation indudes each component that—
  (1) Is necessary for propulsion; and
  (2) Affects the safety of the major propulsive units.

(b) Each powerplant installation must be constructed and arranged to—
  (1) Ensure safe operation to the maximum altitude for which approval is requested.
  (2) Be accessible for necessary inspections and maintenance.

(c) Engine cowls and nacelles must be easily removable or openable by the pilot to provide adequate access to and exposure of the engine compartment for preflight checks.

(d) Each turbine engine installation must be constructed and arranged to—
  (1) Result in carcass vibration characteristics that do not exceed those established during the type certification of the engine.
  (2) Ensure that the capability of the installed engine to withstand the ingestion of rain, hail, ice, and birds into the engine inlet is not less than the capability established for the engine itself under §23.903(a)(2).

(e) The installation must comply with—
  (1) The instructions provided under the engine type certificate and the propeller type certificate.
  (2) The applicable provisions of this subpart.

(f) Each auxiliary power unit installation must meet the applicable portions of this part.

PISTON ENGINE

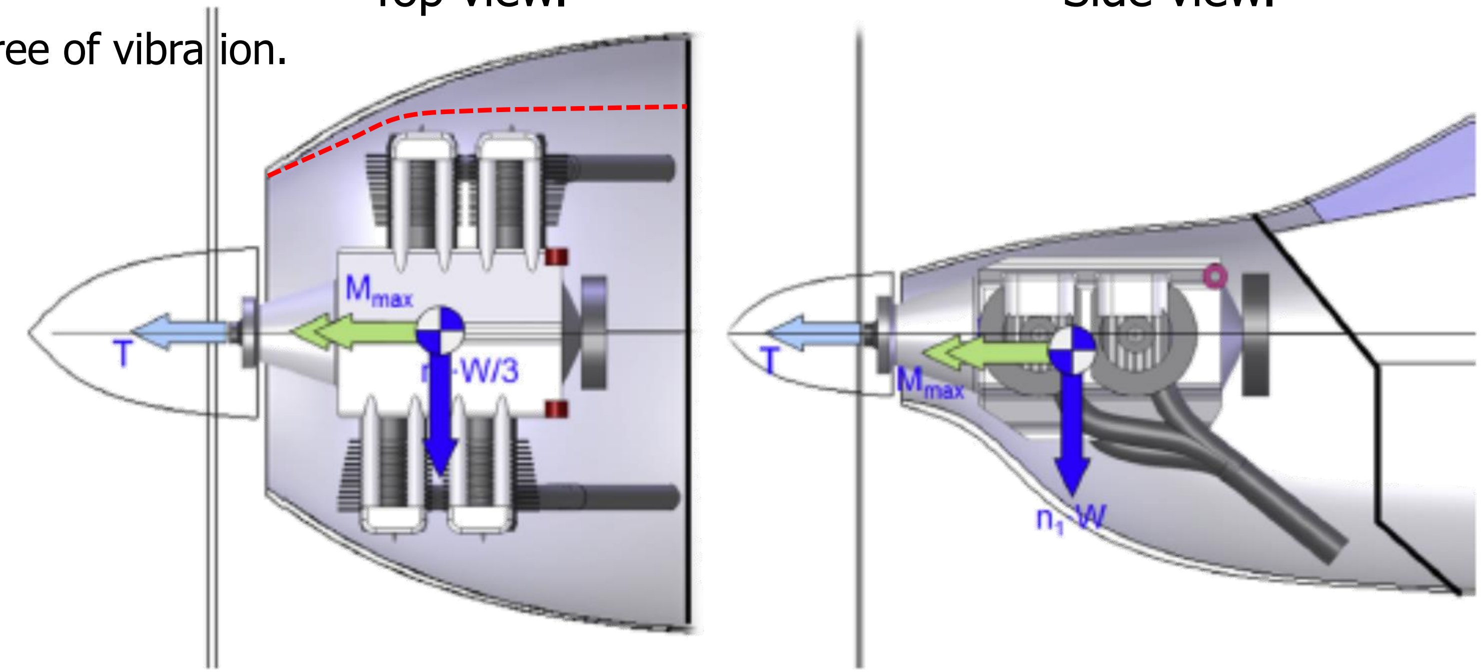
Engine/Motor installation must meet at least these five requirements:

1. Be **structurally sound enough** to react to all loads generated by the engine.
2. Allow for **easy access to maintenance**.
3. Allow for **engine controls to be easily routed** to and from the engine (electrical system, fuel plumbing, and engine controls like throttle, mixture, and pitch control).
4. Must be **fire-resistant**.
5. The **propeller must have a type certificate** and be free of vibration.

The loads generated by the engine installation are primarily **inertia loads** due to gravitation and **maneuvering loads** and loads generated by the engine itself, such as **thrust and gyroscopic moments**.

Top view.

Side view.



Engine loads are applied at the *cg* of the engine, except propeller thrust, which is applied at the hub or the center of action *CA*.

PISTON ENGINE

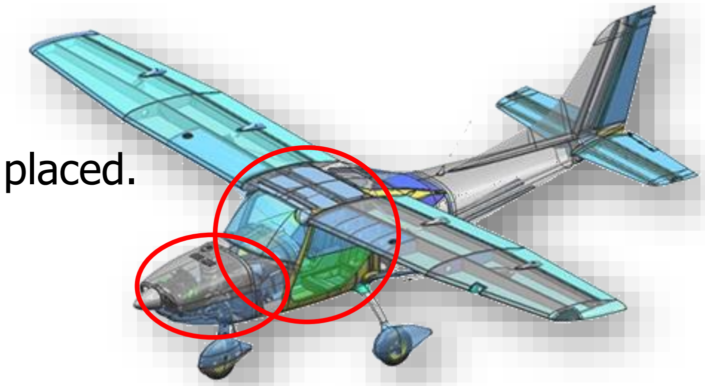
Universidad Pontificia Bolivariana

Fundada en 1936

Fireproofing – is the addition of fire-resisting or retarding material and the installation of fire-suppression chemical dispensers to aircraft intended to make it more fire-resistant and, thus, safer in the case of such an emergency.

There are three areas in an aircraft that are more susceptible to fire than others:

1. the engine compartment,
2. the cabin area, and
3. anywhere electrical wiring is placed.



Designing the fireproofing **<u>does not require much mathematics</u>**, but rather should demonstrate compliance with the applicable federal regulations. GA aircraft must comply with the regulation of *14 CFR Part 23*:

23.865 – Fire protection of flight controls, engine mounts, and other flight structure,
23.1181 – Designated fire zones; regions included,
23.1182 – Nacelle areas behind firewalls,
23.1183 – Lines, fittings, and components,
23.1189 – Shutoff means,
23.1191 – Firewalls,
23.1193 – Cowling and nacelle,
23.1195 – Fire extinguishing systems,
23.1197 – Fire extinguishing agents,
23.1199 – Extinguishing agent containers,
23.1201 – Fire extinguishing systems materials, and
23.1203 – Fire detector system.

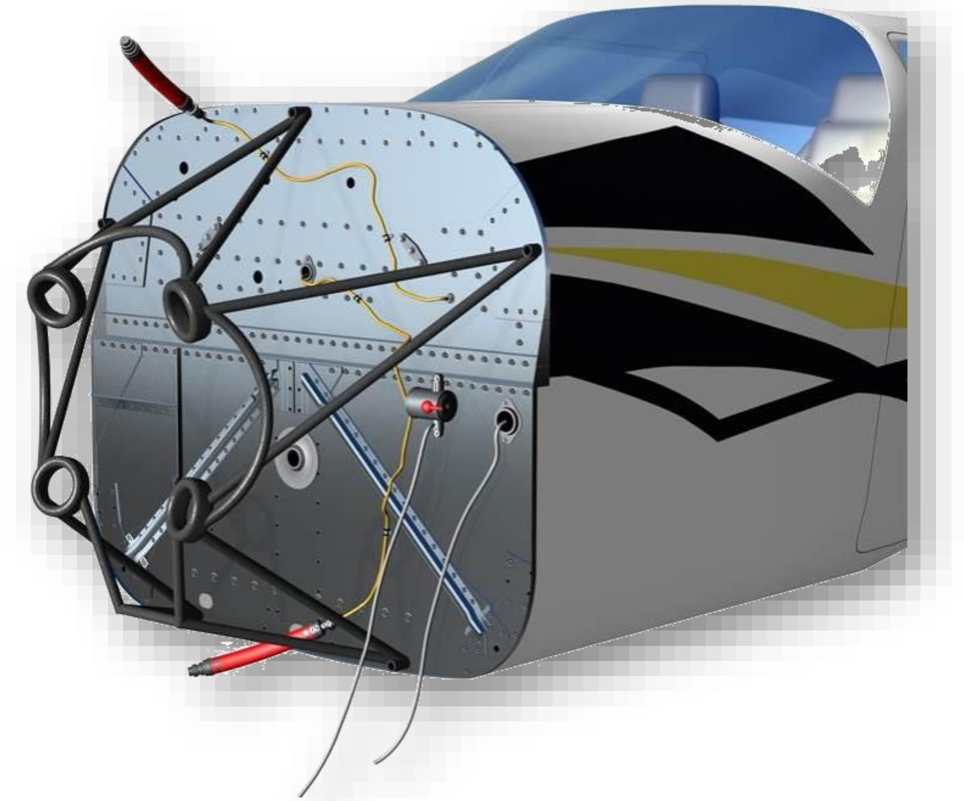
PISTON ENGINE

Firewall – engines are typically mounted on a truss structure made from welded **chrome-molybdenum steel** (SAE 4130), which is then mounted to the airframe and separated from it via the ***firewall***.

Its purpose is to prevent fire from spreading beyond the engine compartment. The firewall is usually made **from stainless steel or other heat-resistant material**.

Some materials are exempt from fire retardation testing:

0.015 inch thick stainless steel sheet,
0.018 inch thick mild steel sheet (coated with aluminum or otherwise protected against corrosion)
0.018 inch thick terne plate
0.018 inch thick Monel metal
0.016 inch thick titanium sheet



Additionally, steel or copper base alloy firewall fittings are exempt as well.

Other materials must be demonstrated to provide fire resistance – for instance, they must be subjected to a flame that is $2000 \pm 150\ °F$ for at least $15\ minutes$ without penetration. Furthermore, the material must be protected against corrosion.

PISTON ENGINE

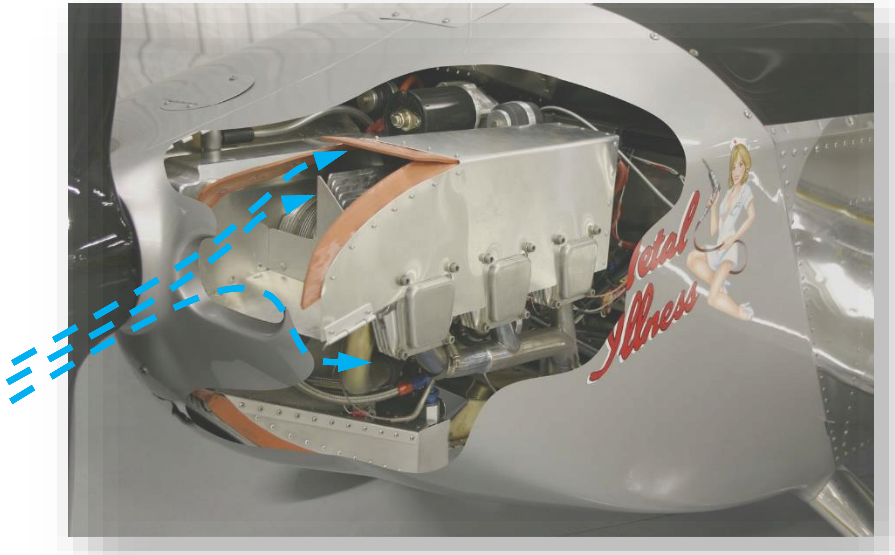
Engine baffles

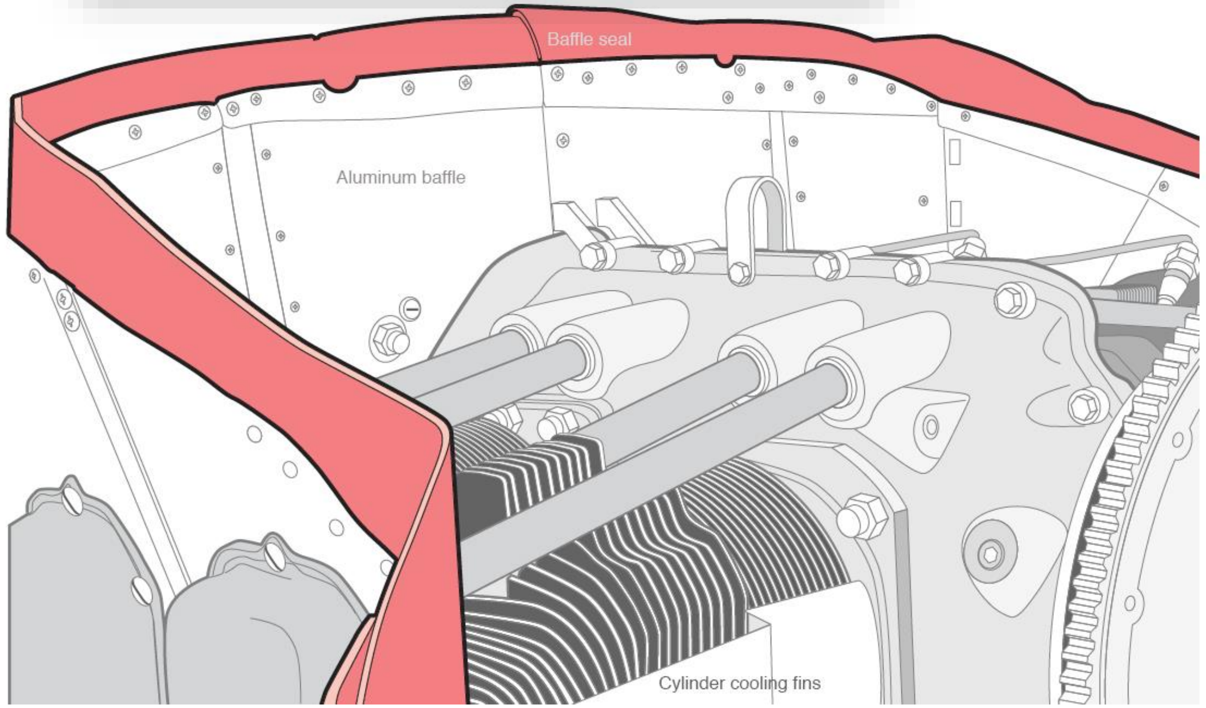
A system of **rigid aluminum baffles** and **flexible baffle seals** creates a chamber of high pressure above the cylinders and another chamber of low pressure below the cylinders and behind the engine. They seal the gap on top of the engine to pressurize the air.

The baffles and baffle seals help to direct air over the cylinders from the high-pressure area to the low-pressure chamber, creating an airflow that travels from top to bottom.

The flexible seals close the gaps between the baffles and the cowling. If they age and become loose or brittle, they won't keep air from leaking past the seal, which results in poor cooling performance, such as an abnormally high cylinder-head or oil temperature.

Baffles and baffle seals aren't visible unless the cowling is off. Older seals are thin and black, whereas newer, silicone rubber seals are usually reddish orange.





Universidad Pontificia Bolivariana

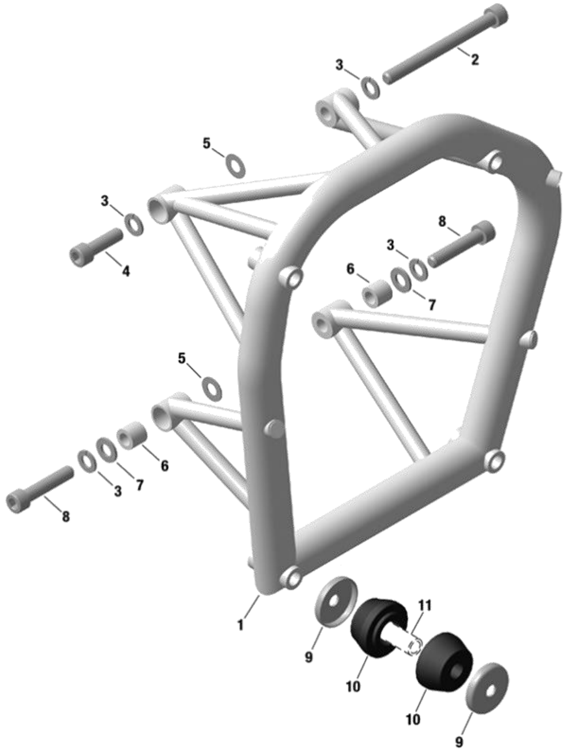
Fundada en 1936

ENGINE SELECTION

The motor mount usually fabricated from welded steel tubing-transfer the engine loads to the corners of the fuselage or the longerons.

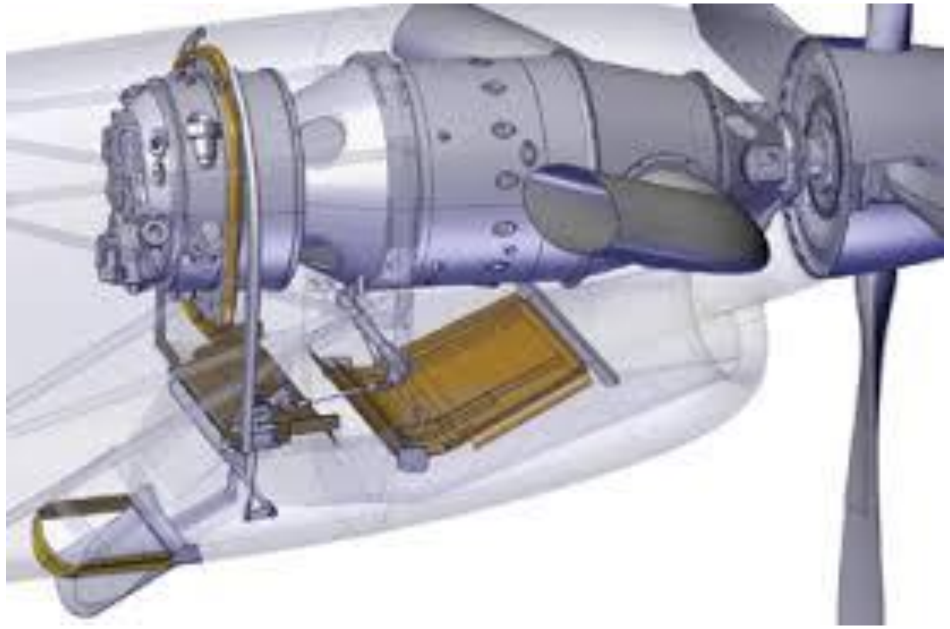
Typically, the engine mount extends the engine forward of the firewall by about half the length of the engine.

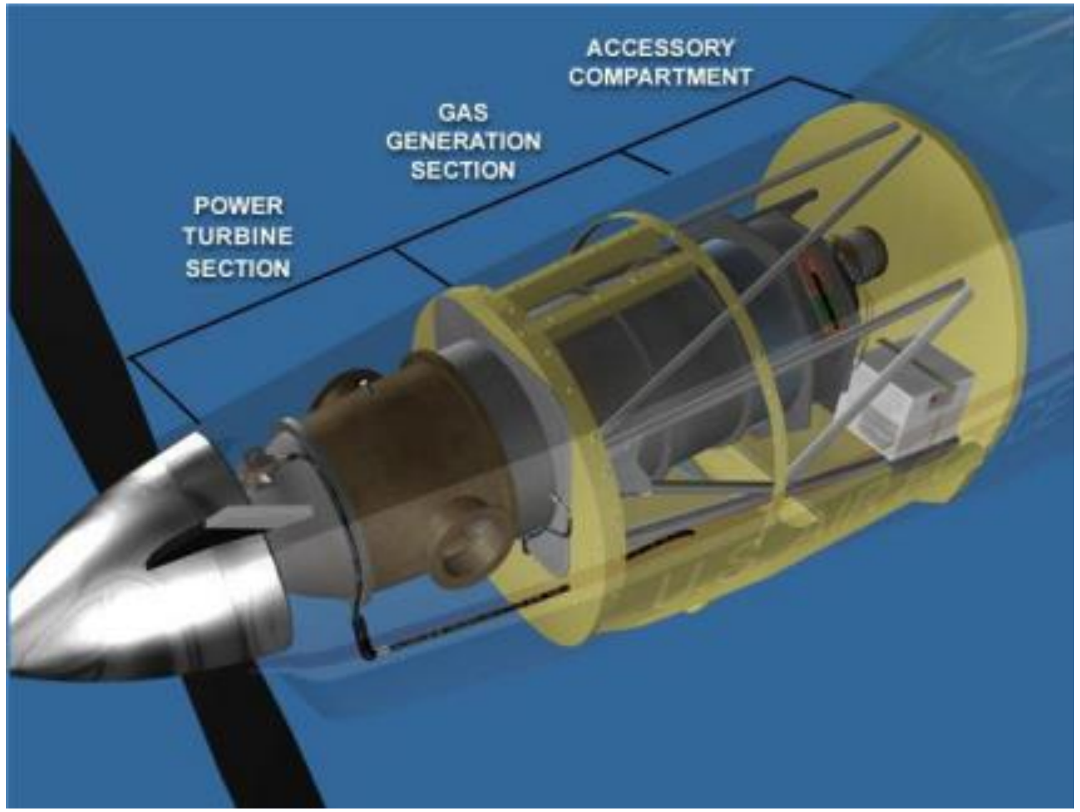
This extra space is used for location of the battery and nose-wheel steering linkages.



Universidad Pontificia Bolivariana
Fundada en 1936

ENGINE SELECTION

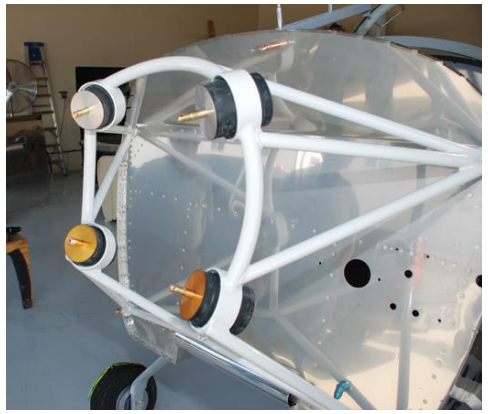




PISTON ENGINE

There are three common means of motor mounting a piston engine to the airplane:

1. **Dynafocal.**
2. **Conical.**
3. **Bed.**



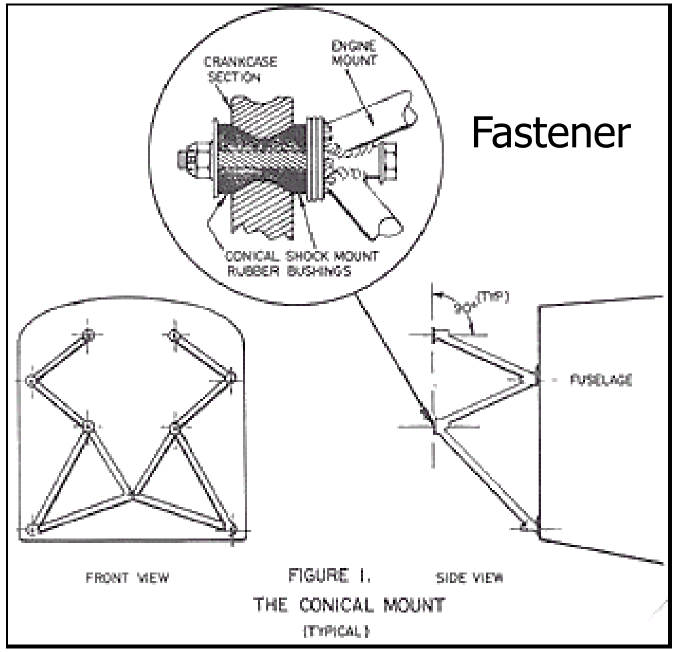
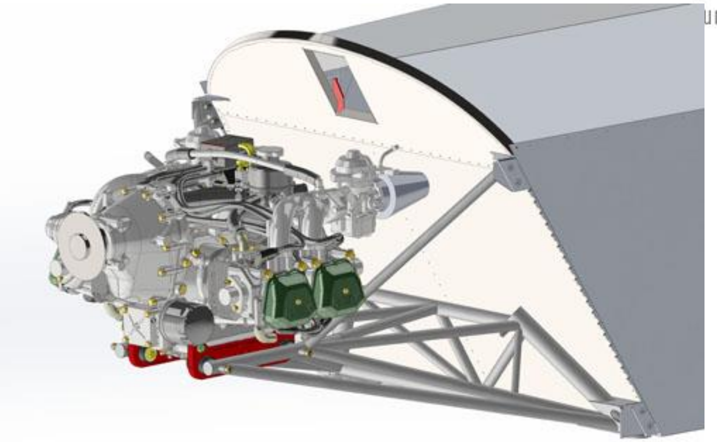


FIGURE 1.
THE CONICAL MOUNT
(TYPICAL)



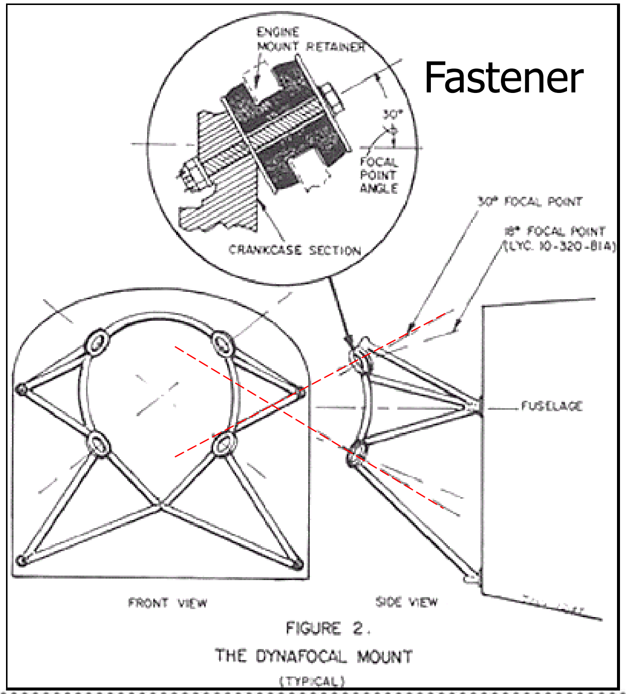
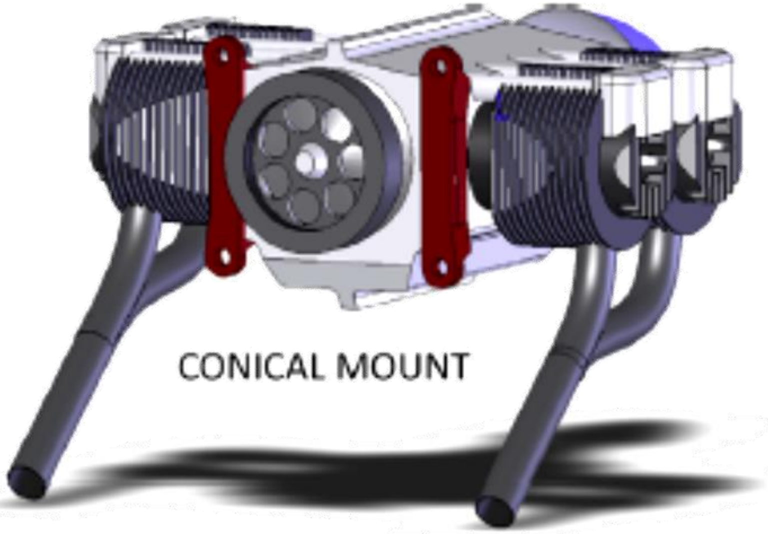
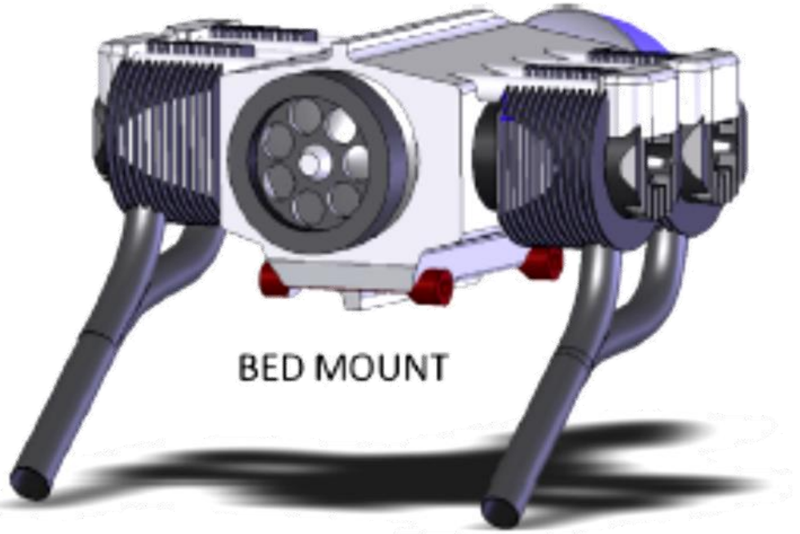


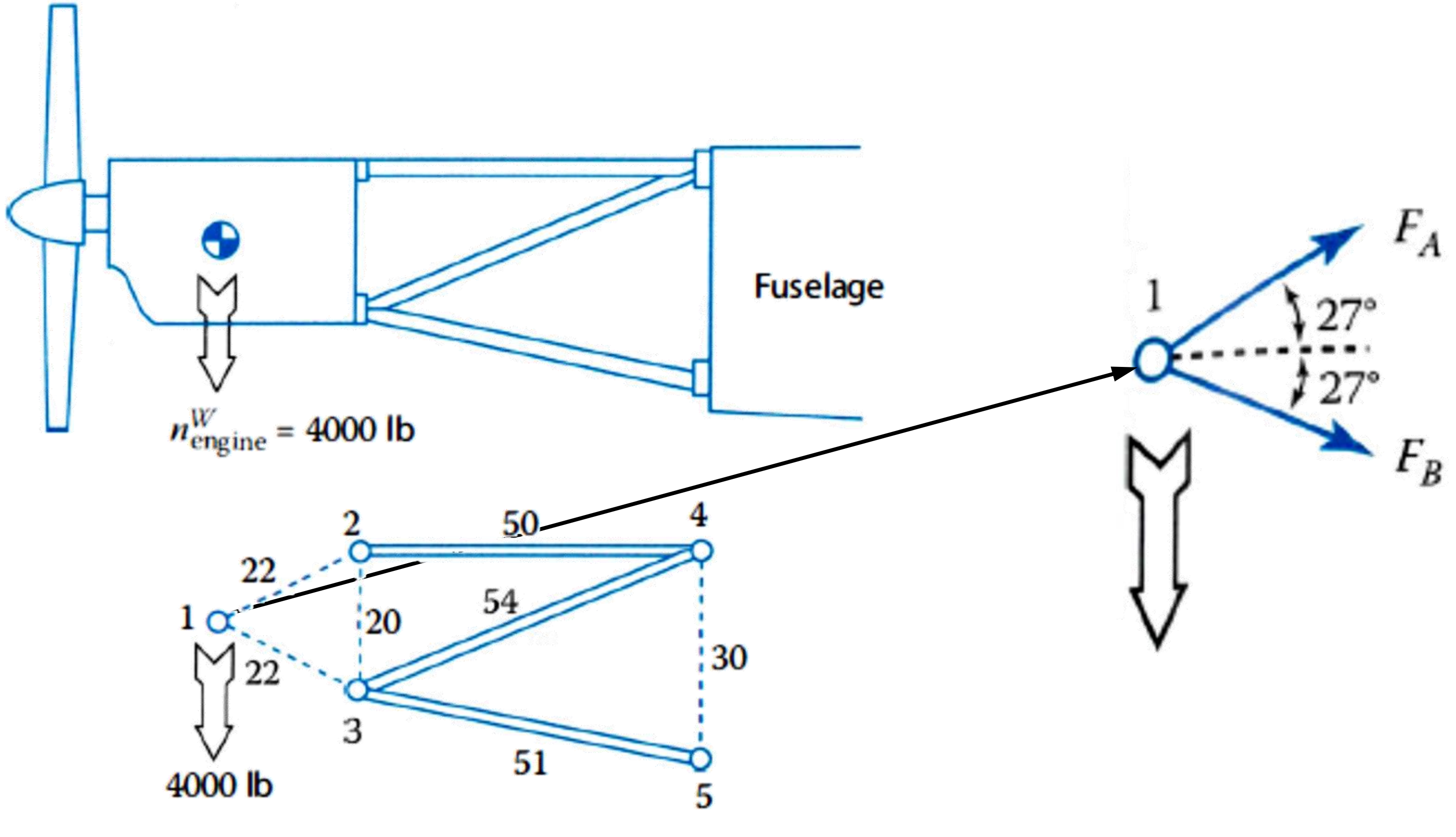
FIGURE 2.
THE DYNAFOCAL MOUNT
(TYPICAL)





PISTON ENGINE

Engine mount – Truss type structure calculation: method of joints.



Joint 1

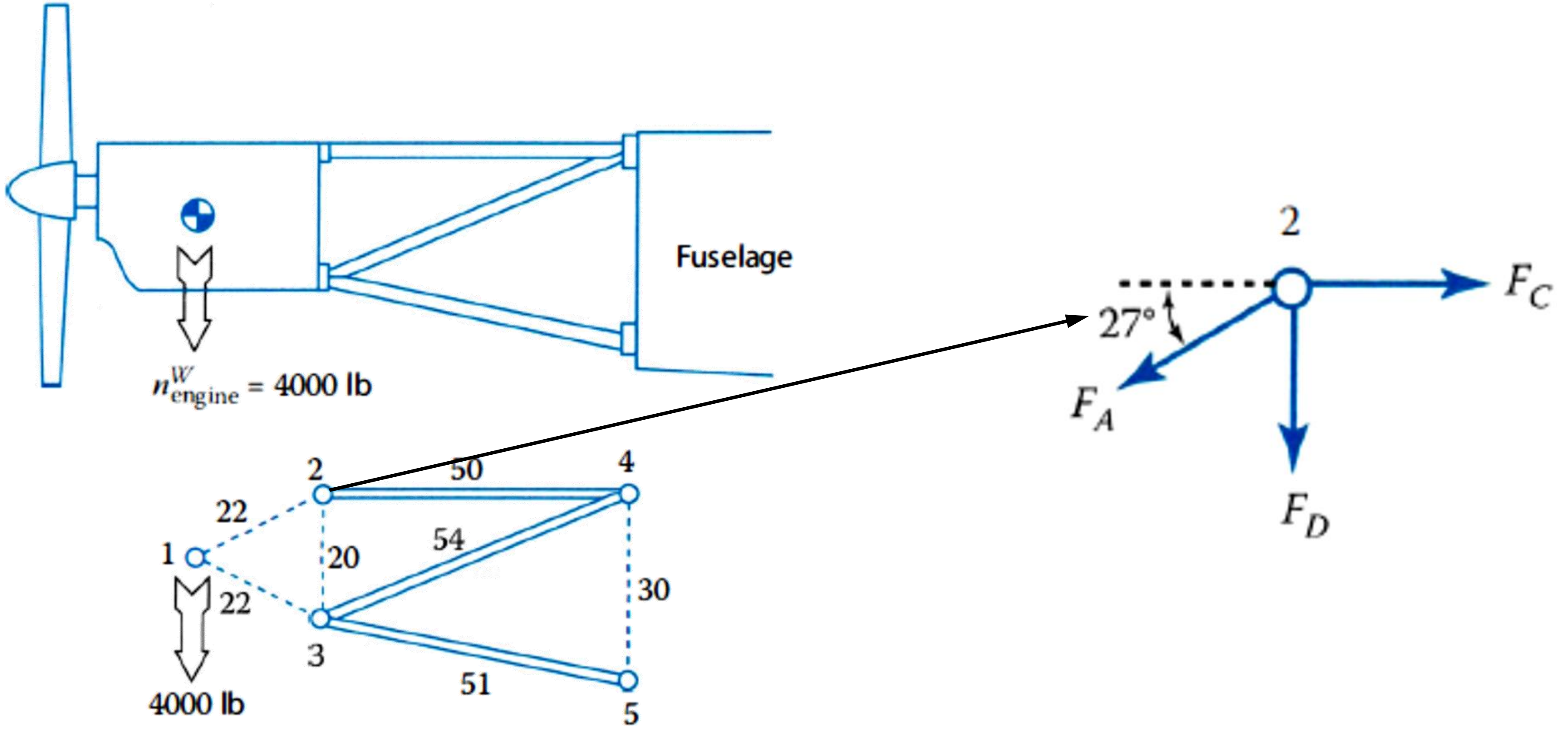
$\Sigma F_H = O = F_A \cos 27 + F_B \cos 27$

$\Sigma F_V = O = F_A \sin 27 - F_B \sin 27 - 4000$

$\boxed{F_A = 4400\ (T) \quad F_B = -4400\ (C)}$

PISTON ENGINE

Engine mount – Truss type structure calculation: method of joints.

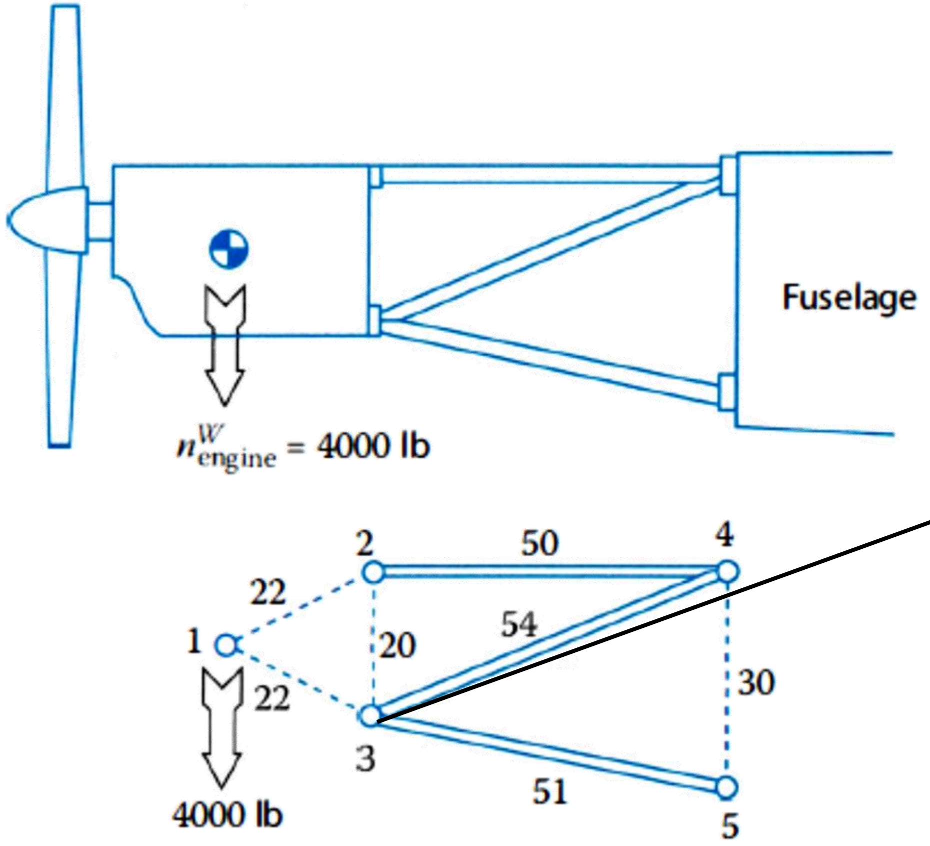


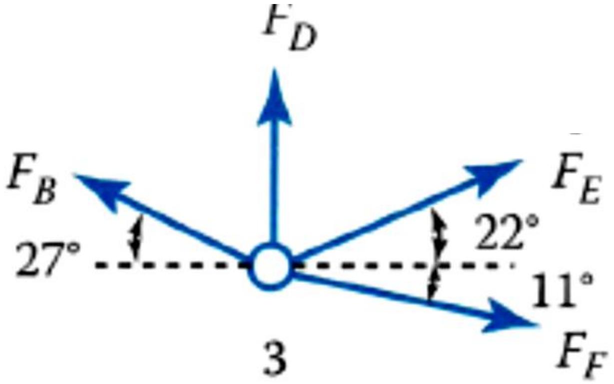
Joint 2

$$\Sigma F_H = O = F_C - F_A \cos 27$$
$$\Sigma F_V = O = -F_D - F_A \sin 27$$
$$F_C = -3919\ (T)$$
$$F_D = -2000\ (C)$$

PISTON ENGINE

Engine mount – Truss type structure calculation: method of joints.





Joint 3

$\Sigma F_H = 0 = F_E \cos 22 + F_F \cos 11 - F_B \cos 27$

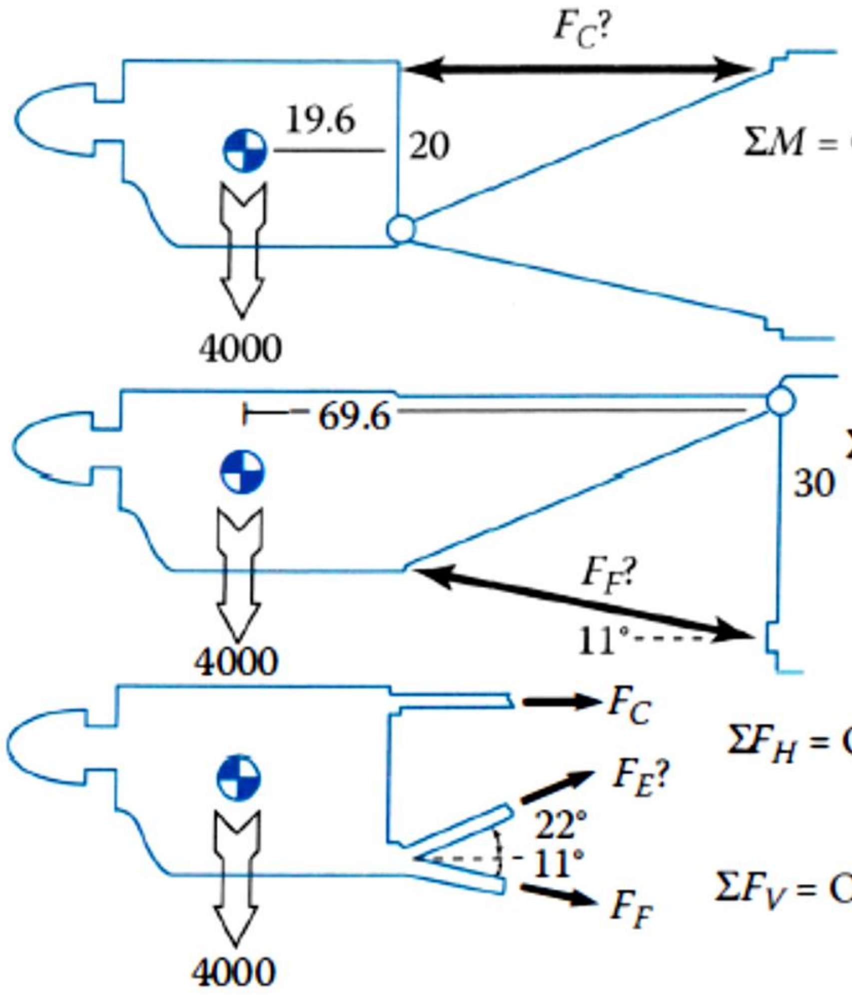
$\Sigma F_V = 0 = F_D - F_B \sin 27 + F_E \sin 22 - F_F \sin 11$

$F_E = 5775$ (T)

$F_F = -9463$ (C)

PISTON ENGINE

Engine mount – Truss type structure calculation: method of moments/shears.



$$\Sigma M = O = -19.6\,(4000) + 20\,F_C$$
$$\boxed{F_C = 3919.2}$$

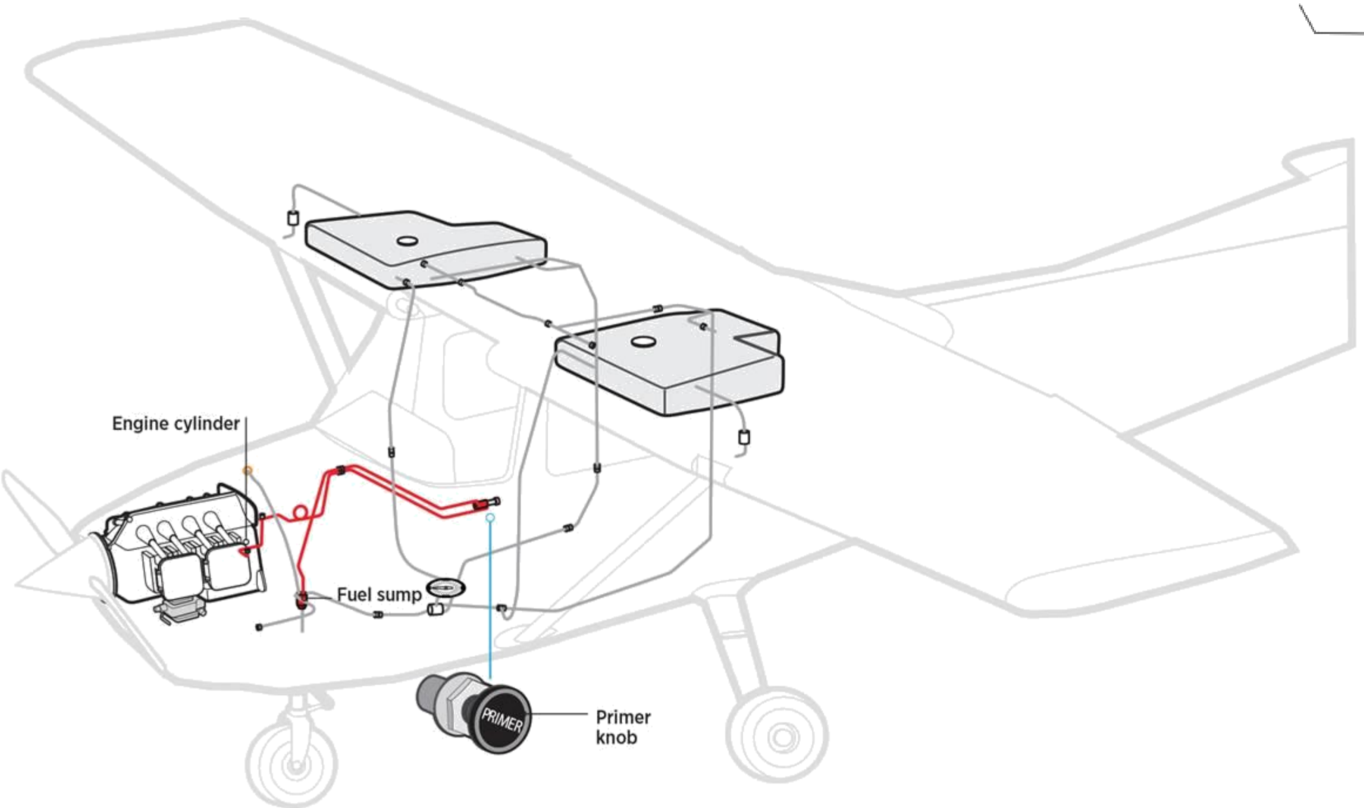
$$\Sigma M = O = -69.9\,(4000) - 30\,F_F \cos 11$$
$$\boxed{F_F = -9463}$$

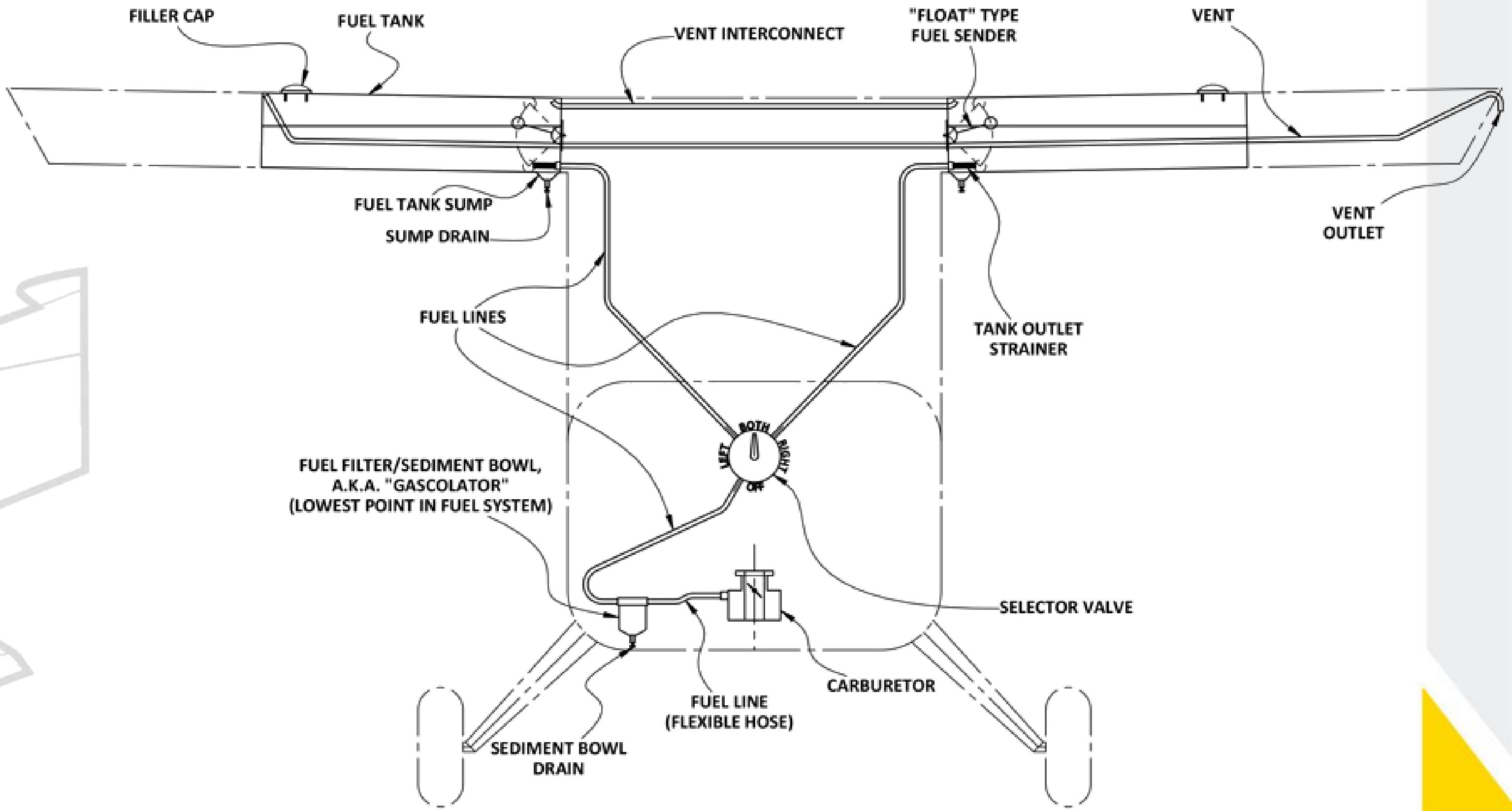
$$\Sigma F_H = O = 3919.2 + F_E \cos 22 + (-9463)\cos 11$$
$$\boxed{F_E = 5775}$$

$$\Sigma F_V = O = -4000 + F_E \sin 22 - (-9463)\sin 11$$
$$\boxed{F_E = 5775}$$

PISTON ENGINE

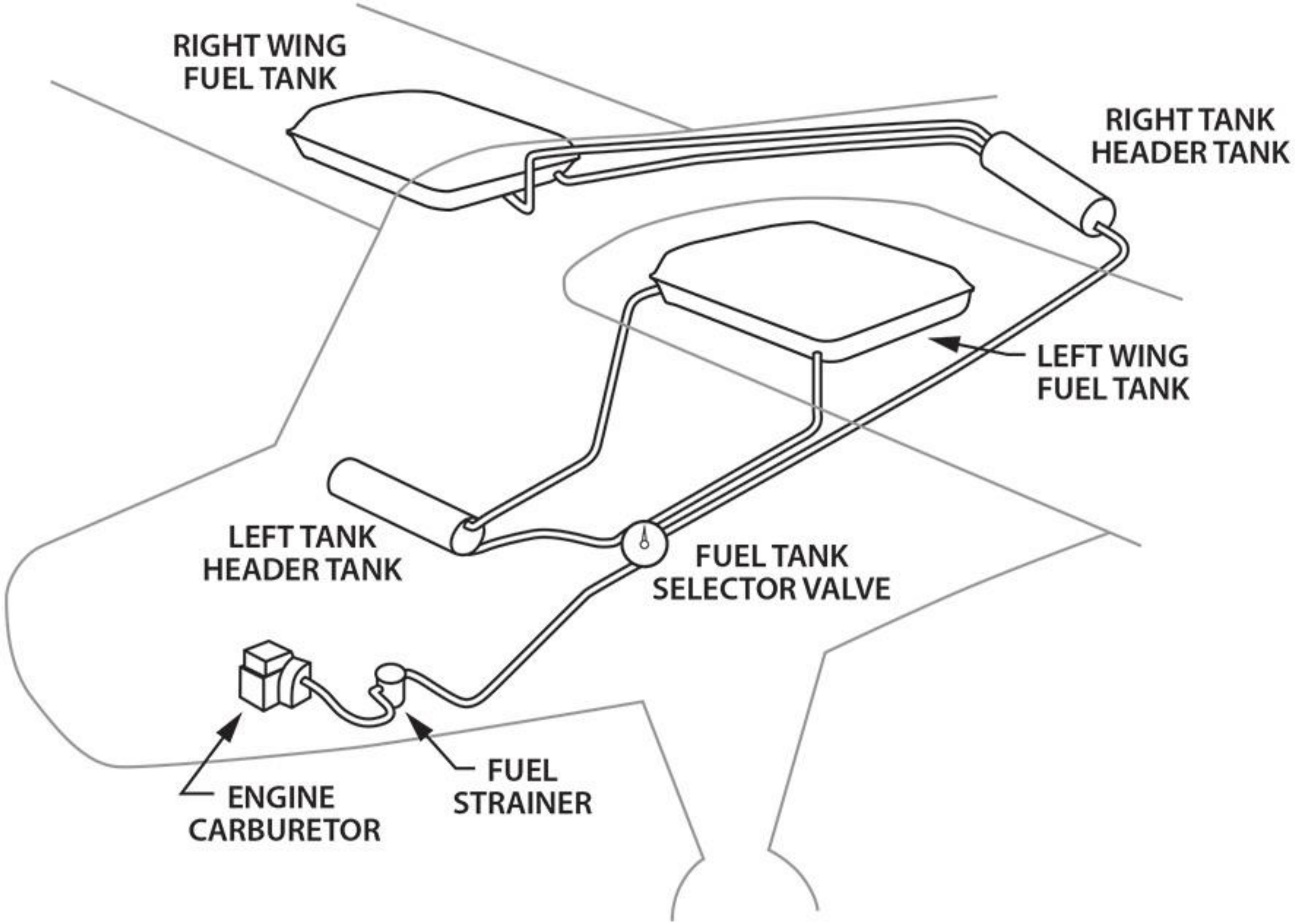
Fuel system.





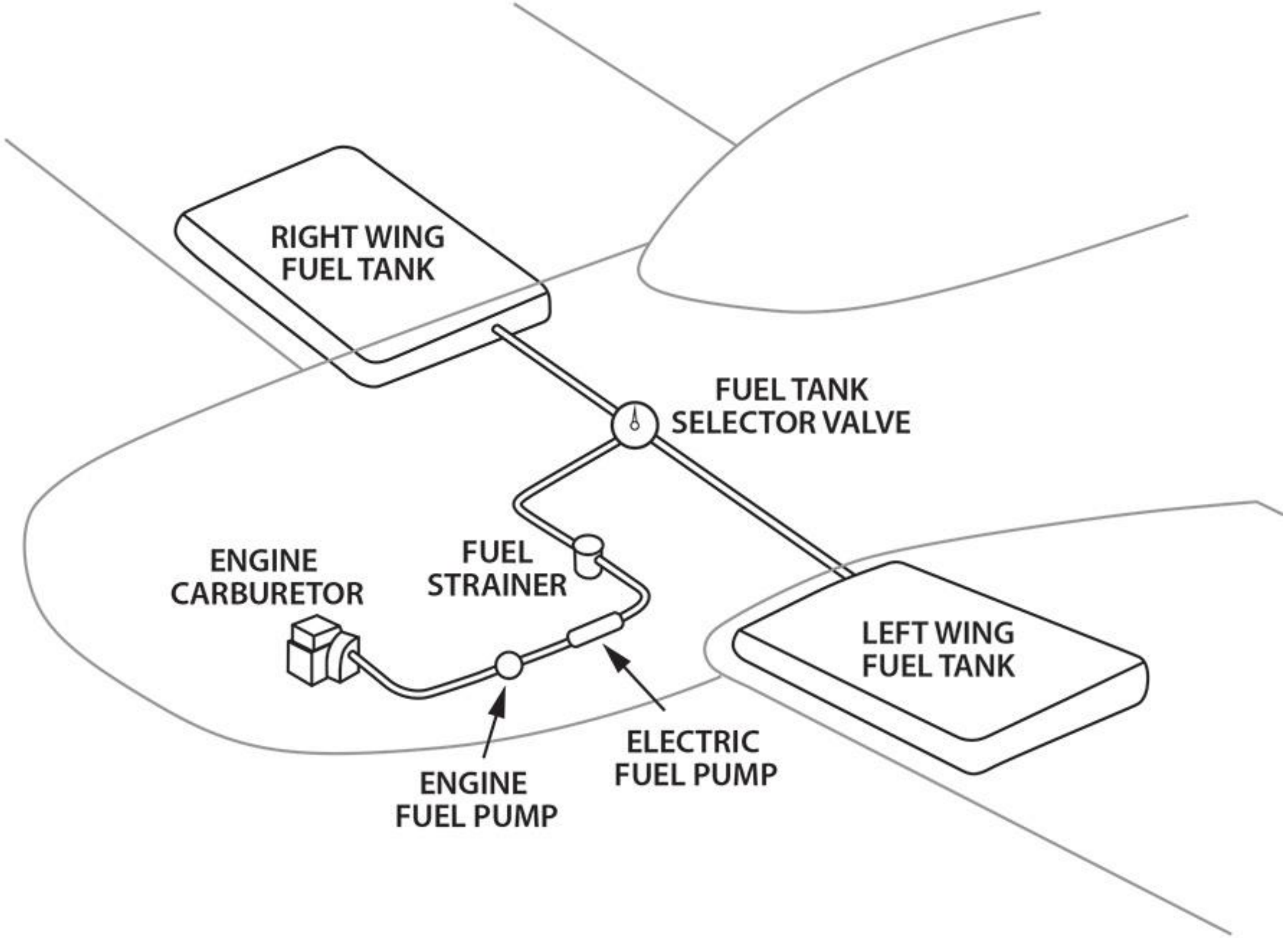
PISTON ENGINE

Fuel system.



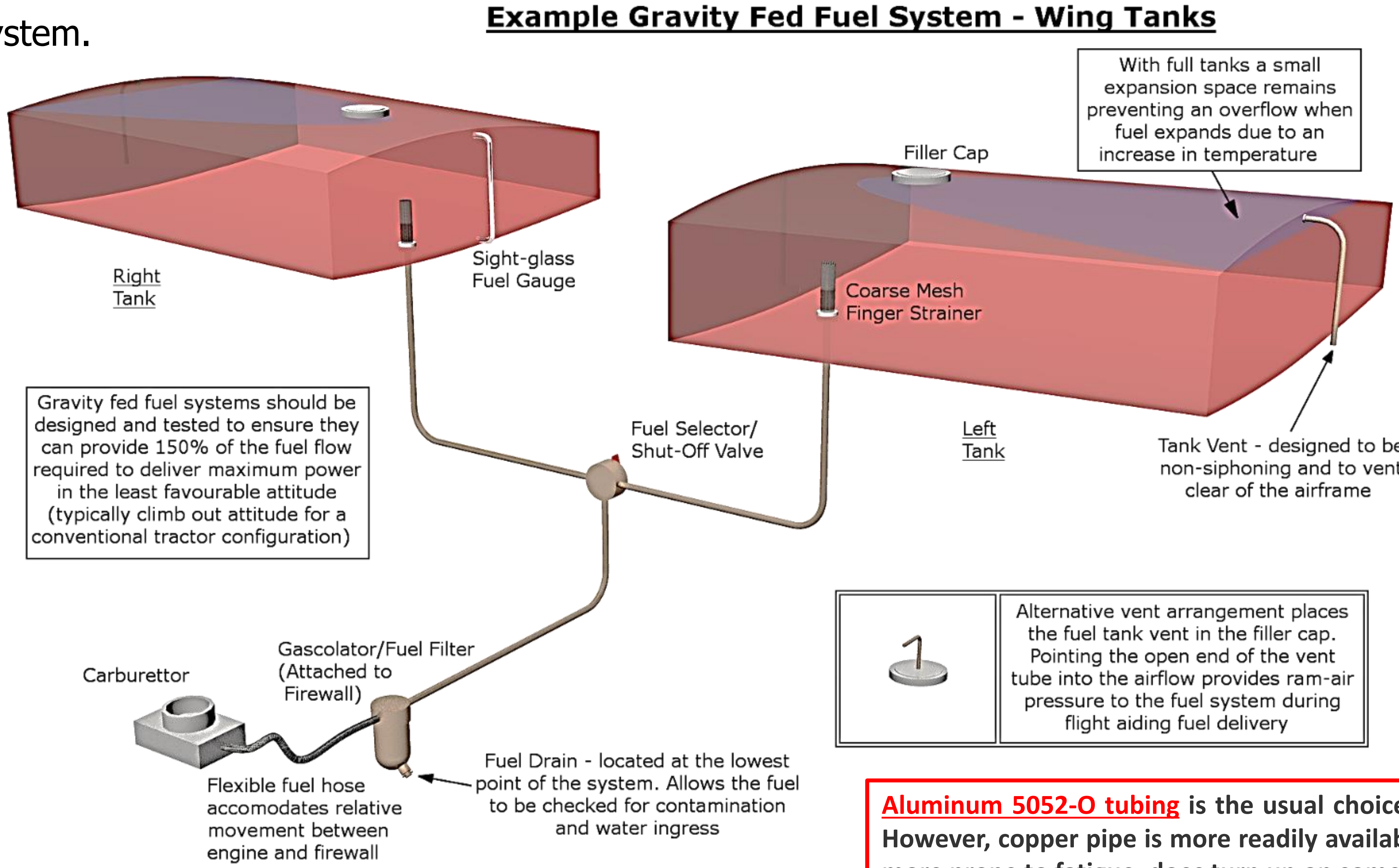
PISTON ENGINE

Fuel system.



PISTON ENGINE

Fuel system.

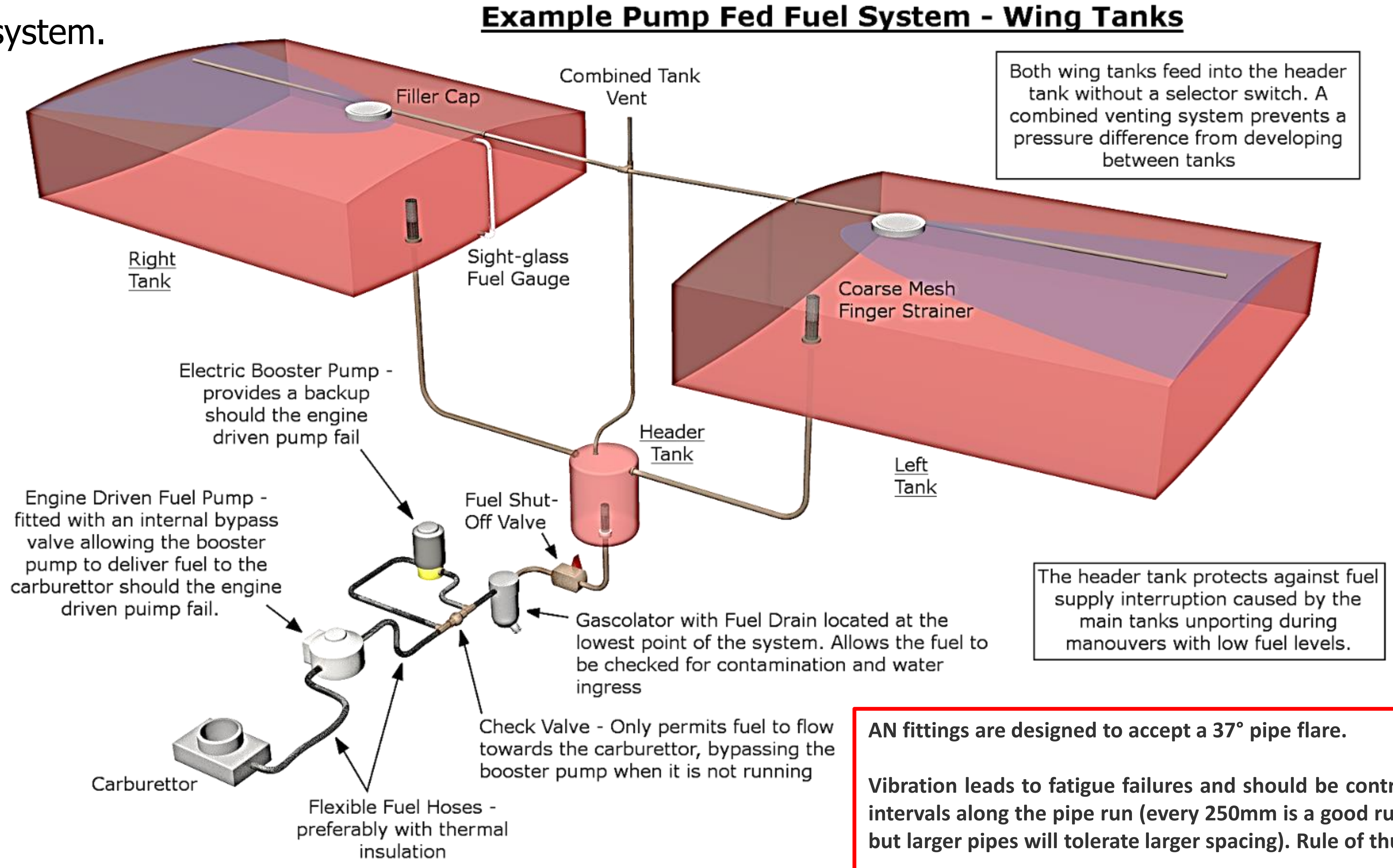


Aluminum 5052-O tubing is the usual choice for fuel lines in GA applications. However, copper pipe is more readily available and, despite being heavier and more prone to fatigue, does turn up on some homebuilt airplanes.

PISTON ENGINE

Universidad Pontificia Bolivariana
Fundada en 1936

Fuel system.



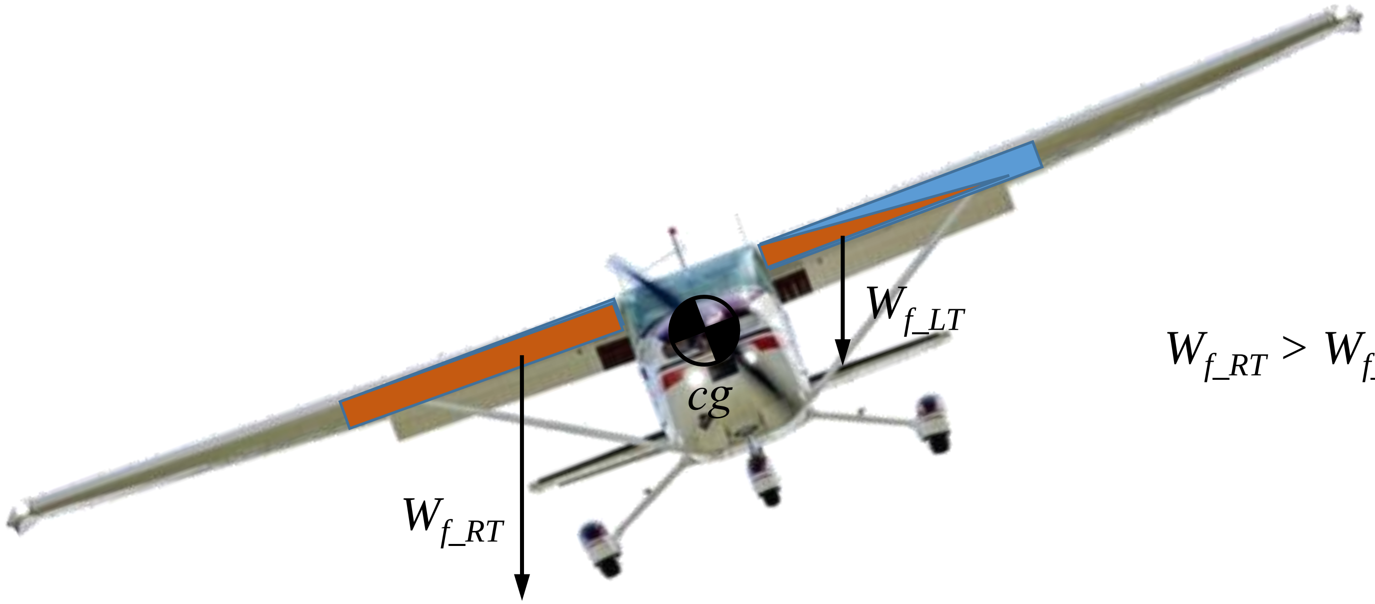
AN fittings are designed to accept a 37° pipe flare.

Vibration leads to fatigue failures and should be controlled by providing support at regular intervals along the pipe run (every 250mm is a good rule-of-thumb for small diameter pipes, but larger pipes will tolerate larger spacing). Rule of thumb for small-diameter

PISTON ENGINE

Fuel system.

$$W_{f\_RT} = W_{f\_LT}$$



$$W_{f\_RT} > W_{f\_LT}$$

PISTON ENGINE

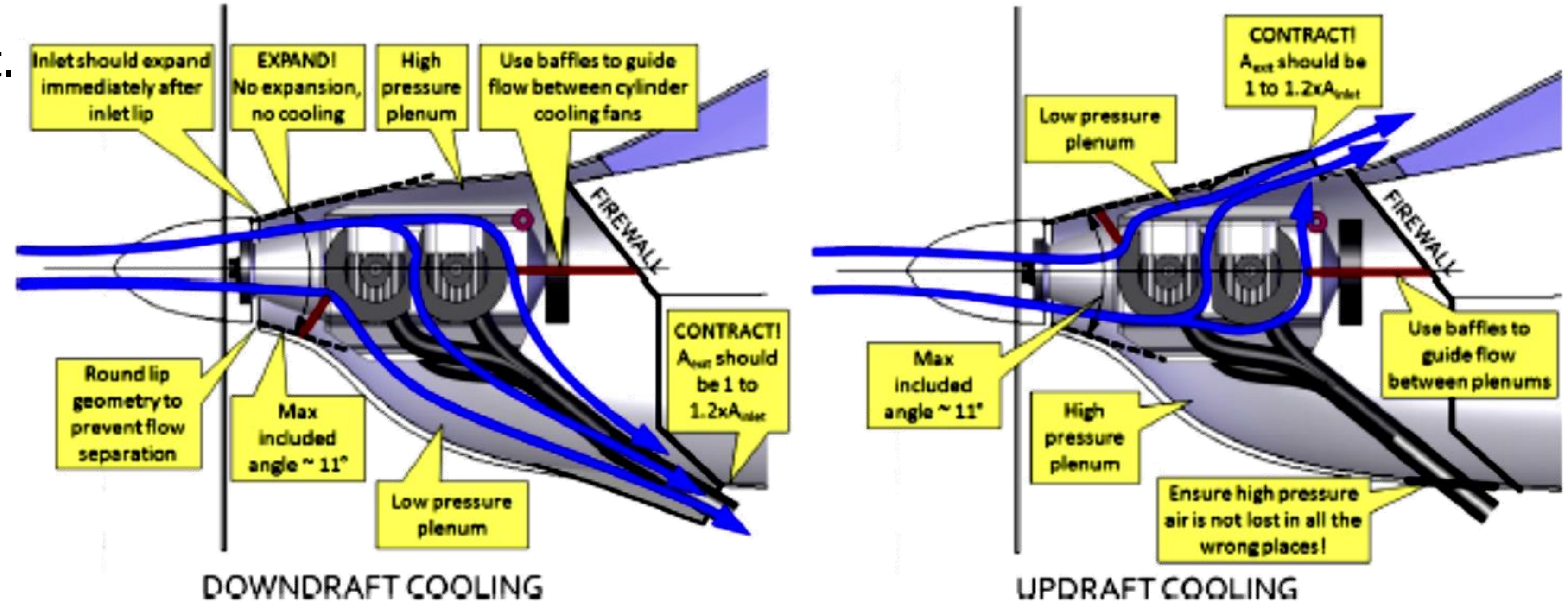
Airflow inlet and outlet.

PISTON ENGINE

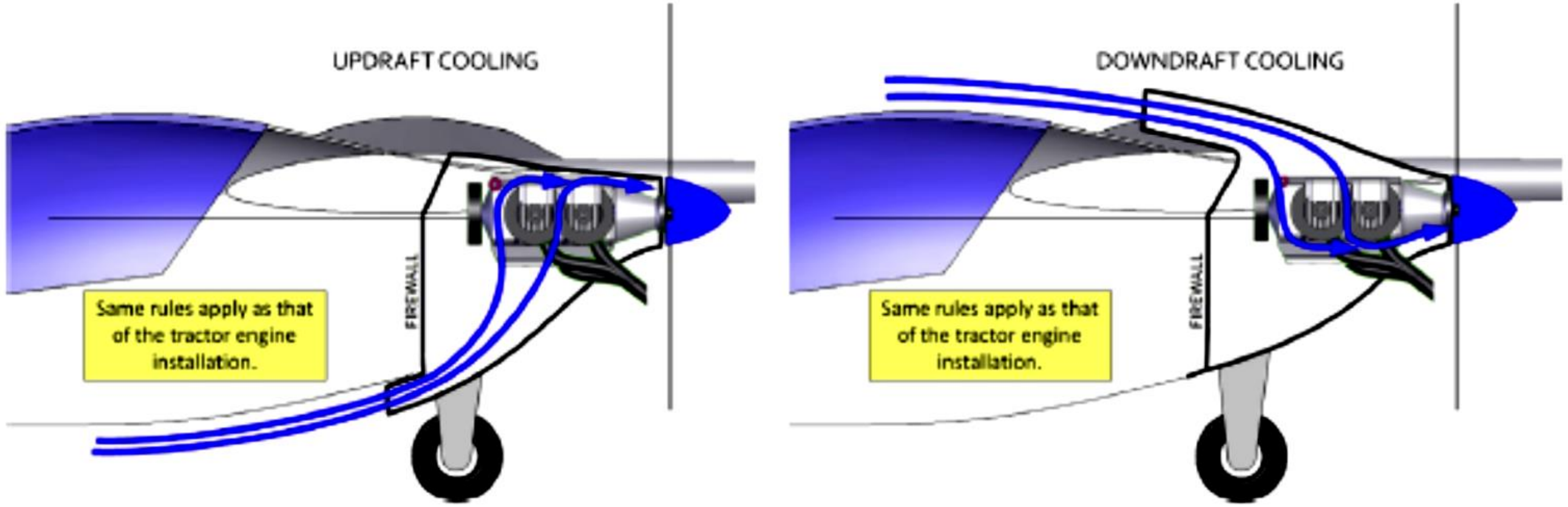
Universidad Pontificia Bolivariana
Fundada en 1936

Airflow inlet and outlet.

Tractor.

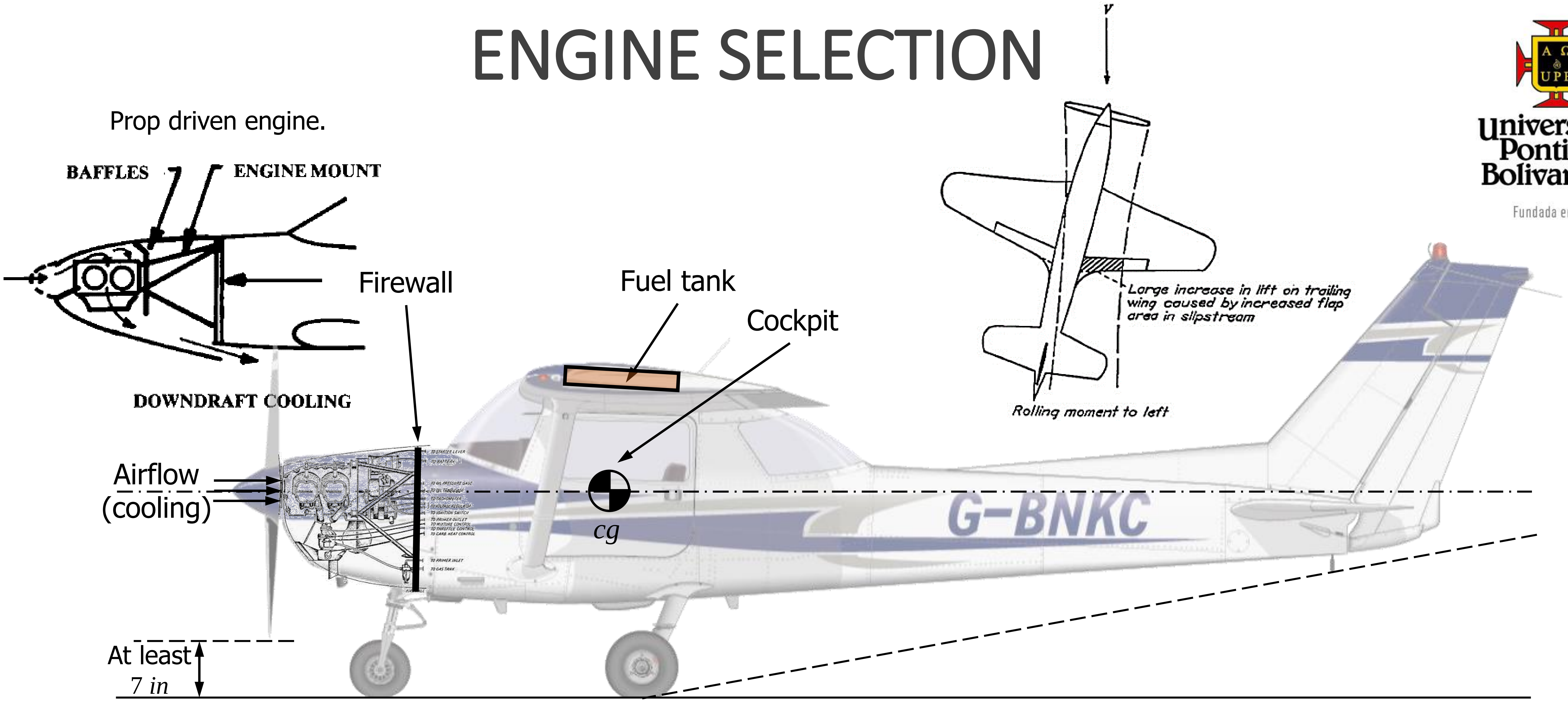


Pusher.



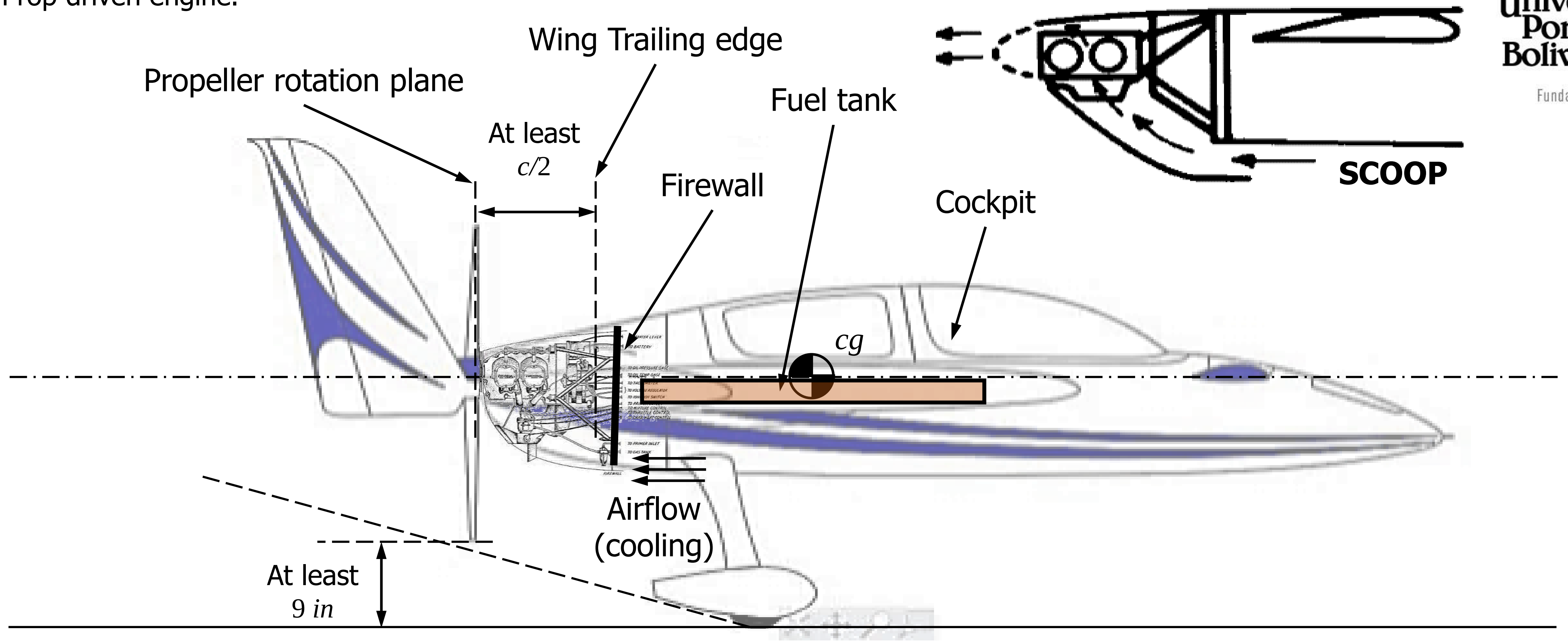
ENGINE SELECTION

Prop driven engine.



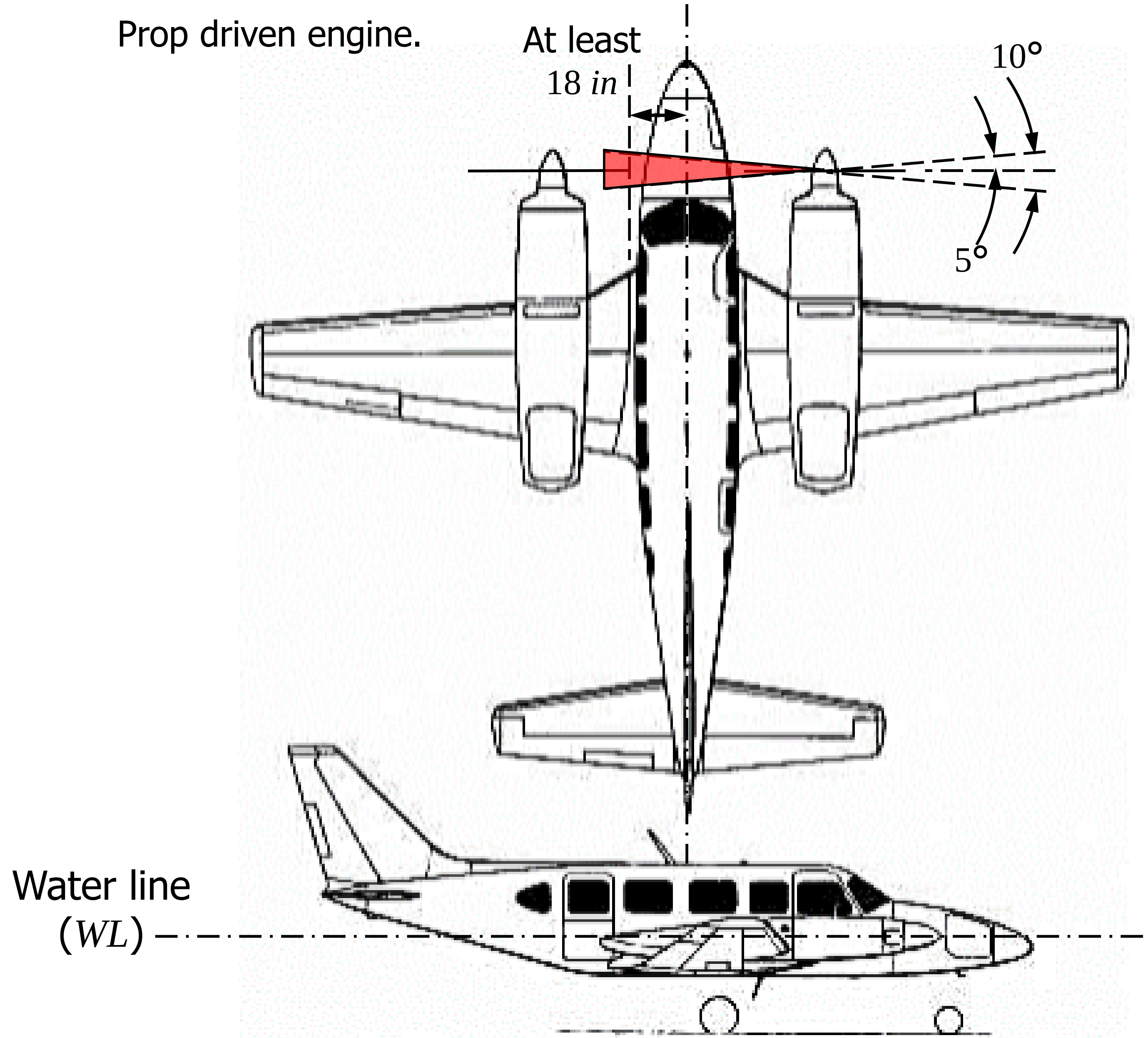
ENGINE SELECTION

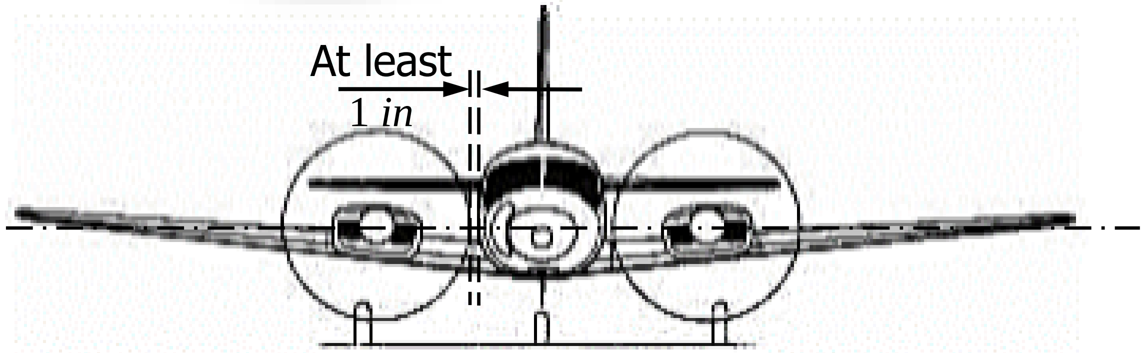
Prop driven engine.



ENGINE SELECTION

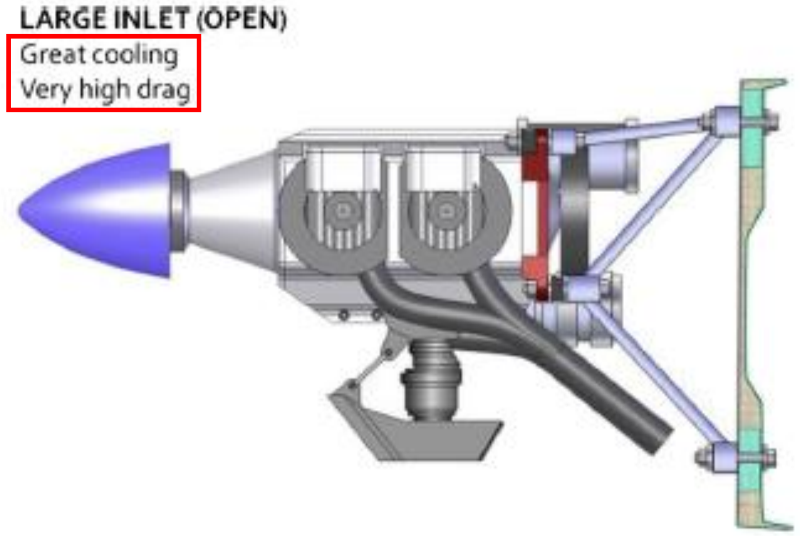
Prop driven engine.

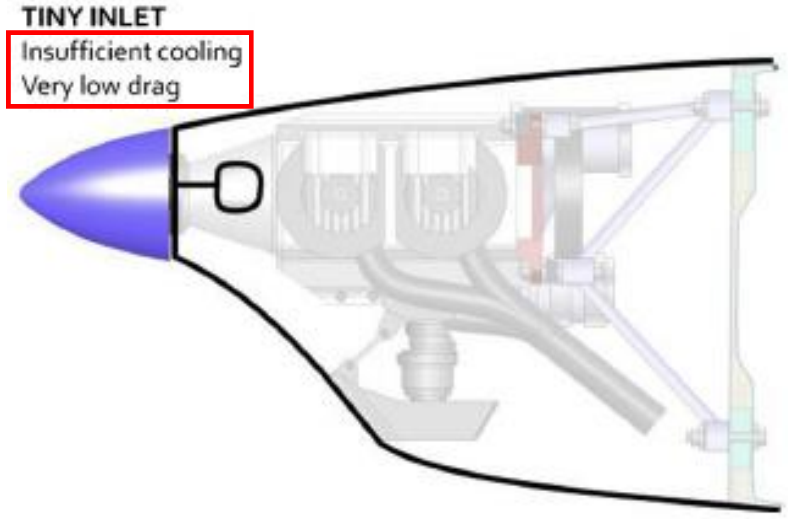


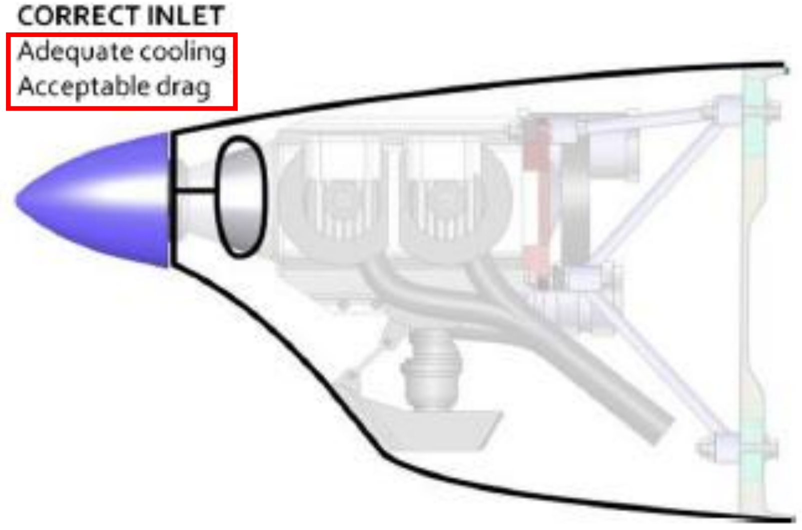


PISTON ENGINE

Airflow inlet and outlet.







PISTON ENGINE

Method 1: Inlet-exit-dependent Heat Transfer.

Assumes that the inlet area must equal the exit area required to carry the engine heat away. The heat carried away by the air flowing through the engine compartment can be found in the **equation of mass flow rate**:

$$\dot{Q} = \dot{m} \cdot C_p \cdot \Delta T = \rho V_E A_E \cdot C_p \cdot \Delta T$$

$\dot{m}$ = mass flow of air flowing through the engine compartment

C_p = specific heat of pressure for air and

ΔT = rise in temperature as it flows through the radiator or the cylinder fins

For air, the value of C_p is as follows: $C_p = 1000 \frac{J}{kg \cdot K}$

V_E = airspeed at the exit of the engine compartment

A_E = exit area

This can be solved for A_E, which then can be used as the inlet area, A_I:

$$A_E = A_I = \frac{\dot{Q}}{\rho V_E \cdot C_p \cdot \Delta T}$$

The engineer should evaluate area requirements during climb (low speed, full power), approach (low speed, low power), and cruise (high speed, moderate power) on hot days (whichever is the most critical condition) and use these to evaluate the area.

The fact that it takes the engine some time to overheat is sometimes used to get by with smaller inlets and outlets.

Universidad Pontificia Bolivariana

Fundada en 1936

PISTON ENGINE

Method 1: Inlet-exit-dependent Heat Transfer – Example.

The installation manual for a Rotax 912 four-stroke aircraft piston engine recommends a radiator capable of transferring thermal energy that amounts to 28 kW. If air warms up by 50 K as it flows through the radiator, size the exit area as a function of the airspeed through it in m/s. How large must the exit area be for a cruising speed of 50 m/s and during climb at 30 m/s?

Solution

The heat carried away by air can be found from:

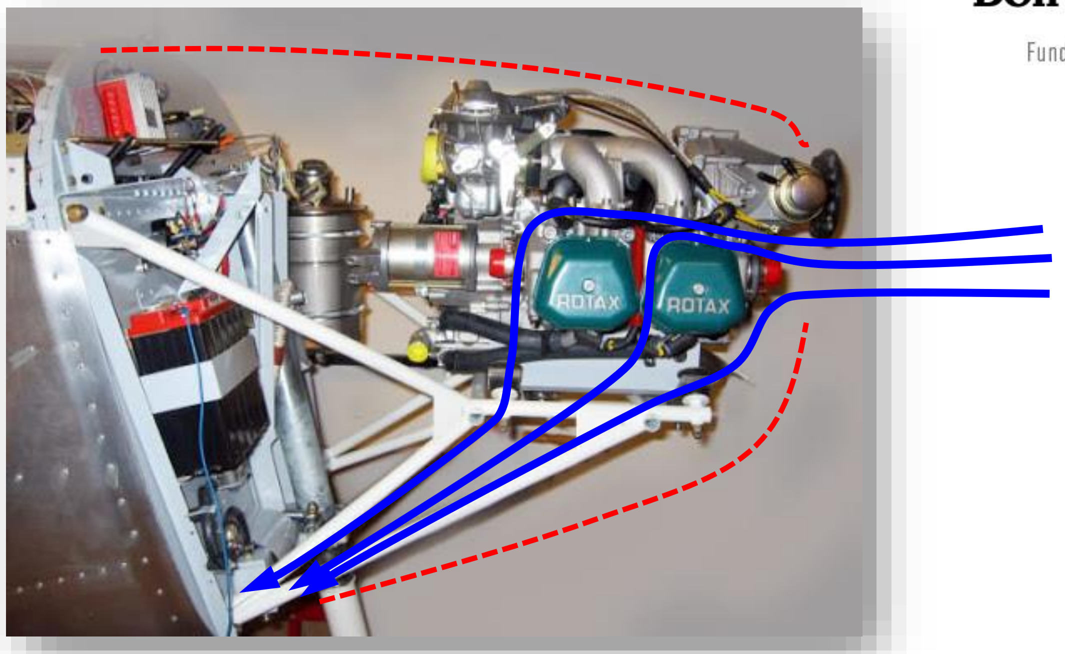
$$\dot{Q} = \dot{m} \cdot C_p \cdot \Delta T = 28{,}000 \ W$$

Therefore we can write:

$$\dot{Q} = \dot{m} \cdot C_p \cdot \Delta T = \rho V_E A_E \cdot C_p \cdot \Delta T$$

Solve for A_E:

$$A_E = \frac{\dot{Q}}{\rho V_E \cdot C_p \cdot \Delta T} = \frac{28{,}000}{(1.223) V_E (1000)(50)} = \frac{0.4579}{V_E} \ \text{m}^2$$



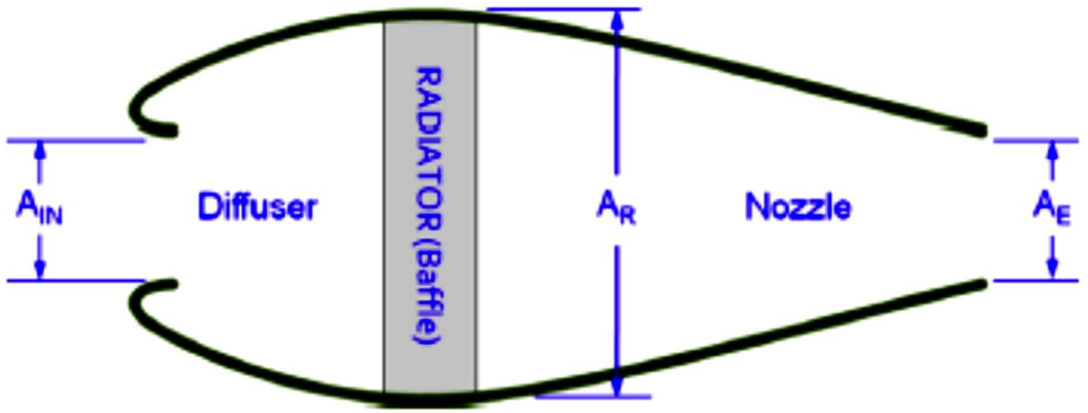
At airspeed of 50 m/s the exit area must be 0.00916 m^2 (0.0986 ft^2 or 14.2 in^2)

At airspeed of 30 m/s the exit area must be 0.0153 m^2 (0.164 ft^2 or 23.7 in^2)

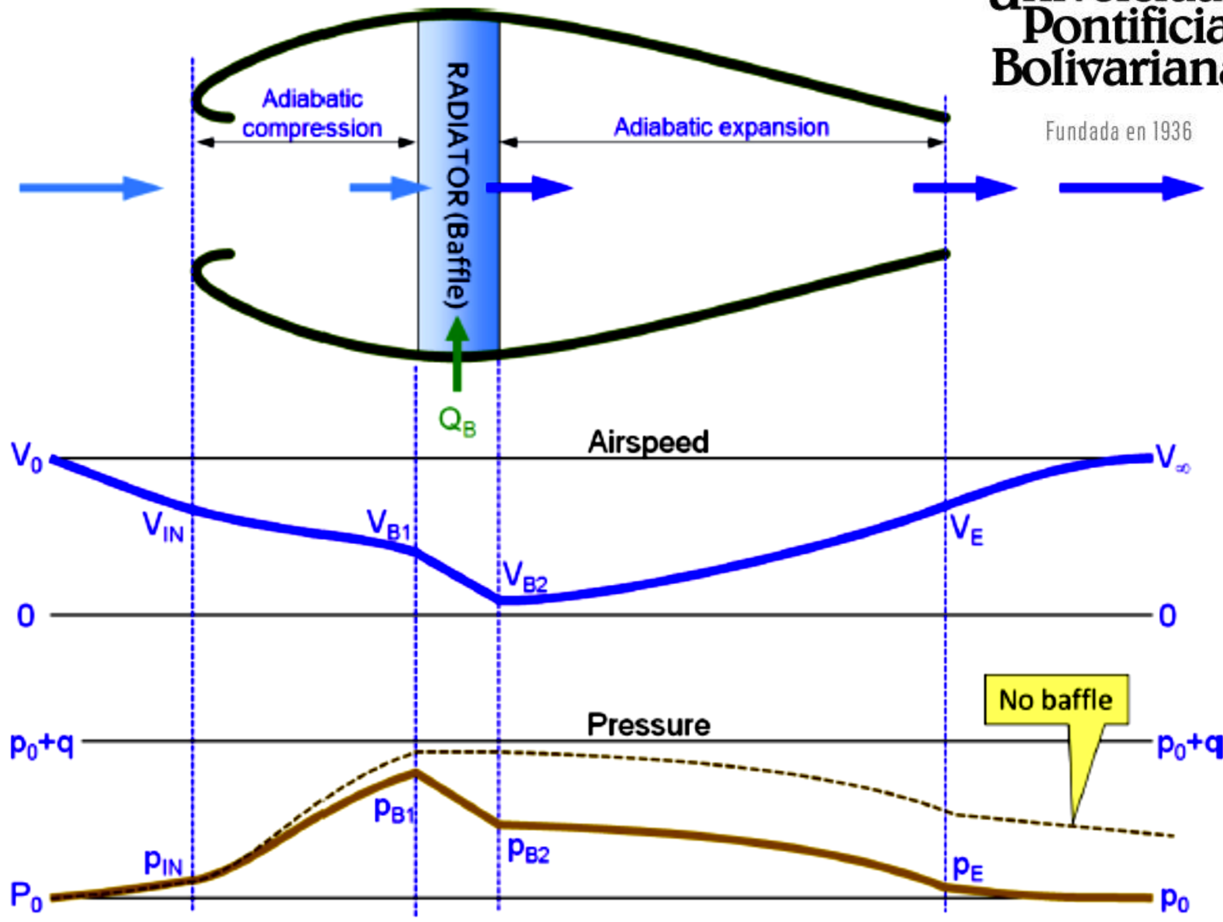
PISTON ENGINE

Method 2: Inlet-radiator-exit Method.

It requires data from the engine manufacturer, which, unfortunately are not always available. The piston engine (or a heat exchanger, such as a radiator) can be analyzed based on the idealized configuration:



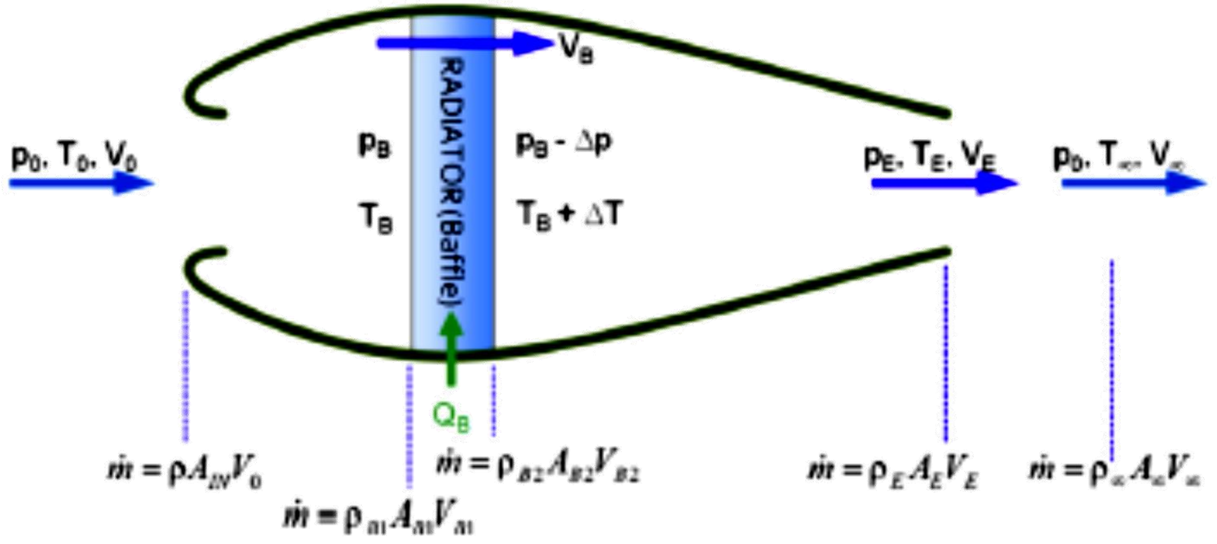
The method assumes that air entering the diffuser will slow down gradually to a minimum value at the forward face of the radiator. The effectiveness of the cooling requires a pressure differential to exist across the radiator.



A common pressure recovery in airplane piston engine cowling inlets is of the order of 60-80%, based on airspeed. For carefully designed scoops, it can be as large as 80-90% depending on airspeed.

PISTON ENGINE

Method 2: Inlet-radiator-exit Method.



$$\frac{p}{p_0} = \left(\frac{T}{T_0}\right)^{\frac{\gamma}{\gamma-1}} \Rightarrow \frac{p}{p_0} = \frac{p_0 + kq}{p_0} = \left(\frac{T}{T_0}\right)^{\frac{\gamma}{\gamma-1}}$$

p and p_0 = pressure at condition and reference, respectively
T and T_0 = temperature at condition and reference, respectively
k = pressure recovery coefficient (1 = complete recovery, 0.5 = 50% recovery, etc.)

A_{IN} = inlet area

A_B = reference area of the baffle (radiator)

A_E = exit area

$\dot{m}$ = mass (or weight) flow rate

ρ = density of air

T_0 = far-field temperature

T_{B1} = temperature at the baffle forward face

T_{B2} = temperature at the baffle aft face

T_E = temperature at the exit

ΔT = temperature increase through the baffle

T_∞ = temperature in the streamtube behind the nozzle

Q_B = heat flow into heat exchanger

V_0 = far-field airspeed

V_{B1} = airspeed in front of the baffle

V_{B2} = airspeed aft of the baffle

V_E = airspeed at the exit

V_∞ = airspeed in the streamtube behind the nozzle

p_0 = far-field pressure

p_{B1} = pressure at the baffle (radiator) forward face

p_{B2} = pressure at the baffle aft face

p_E = pressure at the exit

Δp = pressure drop through the baffle

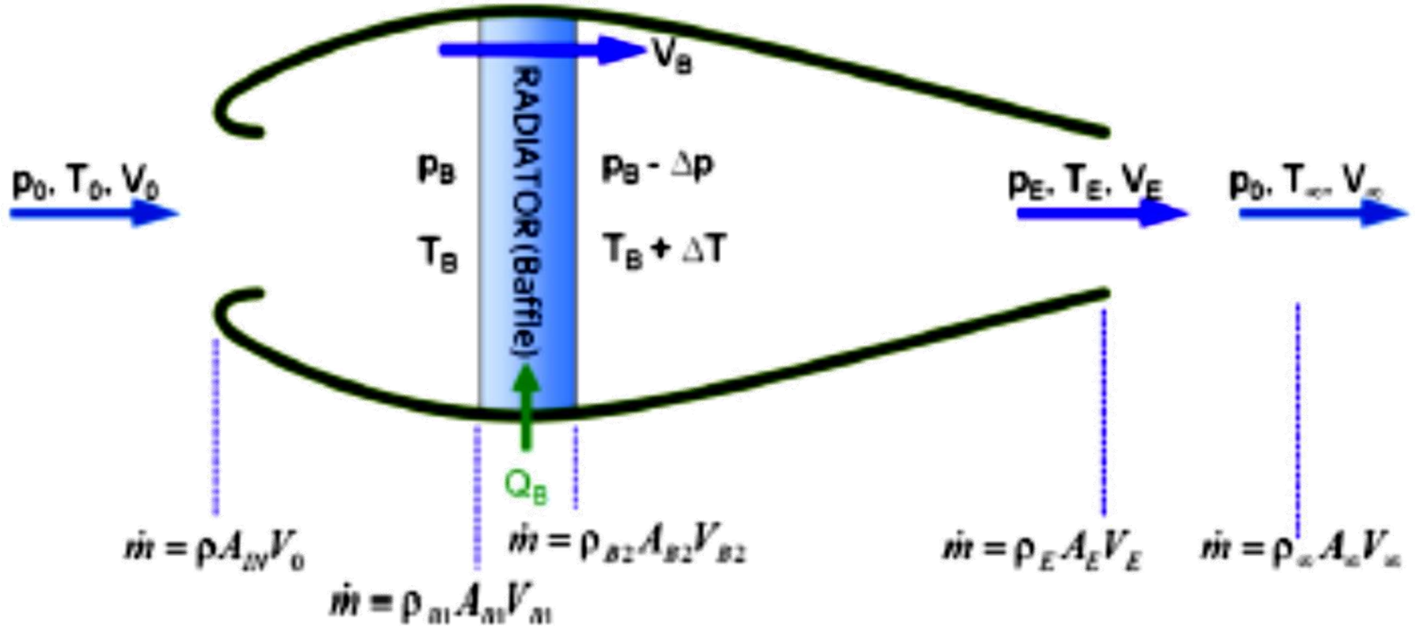
p_0 = far-field pressure

The factor k indicates how much of the dynamic pressure is preserved as the speed of the airflow is slowed down and is an indicator of the efficiency of the diffuser. If $k = 0$, there is no recovery, and the total pressure remains that of the ambient pressure in the far-field. If $k = 1$, there is 100% recovery, and all the dynamic pressure is converted into total air pressure without any losses. <u>This is generally highly desirable.</u>

Universidad Pontificia Bolivariana
Fundada en 1936

PISTON ENGINE

Method 2: Inlet-radiator-exit Method.



With the pressure known, the airspeed at condition can be determined using the compressible Bernoulli equation:

$$V = \sqrt{V_0^2 + \frac{2\gamma}{\gamma - 1}\left(\frac{p_0}{\rho_0} - \frac{p}{\rho}\right)}$$

Determining the pressure drop through the radiator is not a simple task. Usually this is done by empirical methods (read as trial and error). However, the pressure drop across baffle depends on:

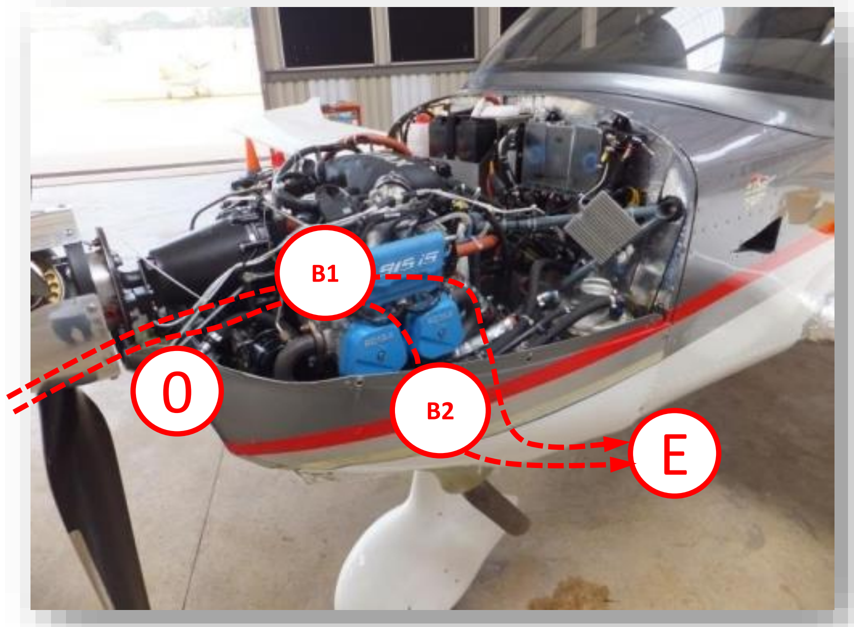
$$\Delta p \propto \frac{1}{2}\rho V_{B1}^2$$

where

V_{B1} = airspeed at the forward face of the radiator.

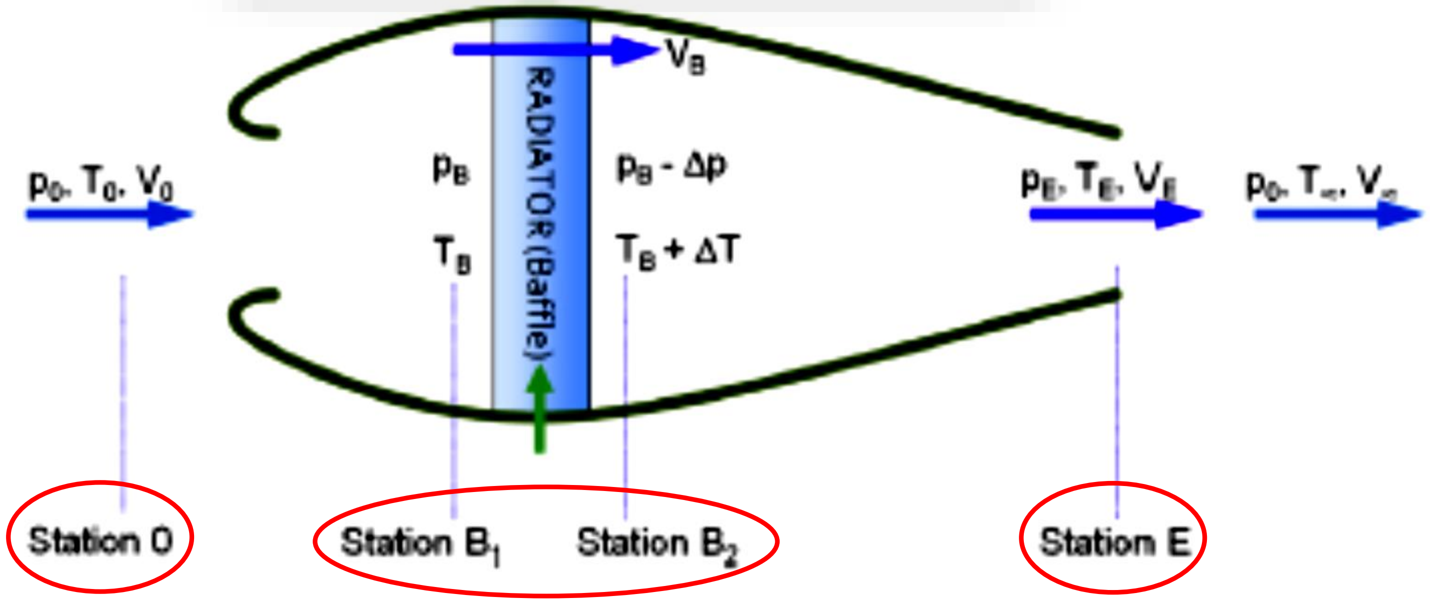
The mass flow through the radiator is given by: $\dot{m} = \rho A_B V_B$

PISTON ENGINE



Method 2: Inlet-radiator-exit Method – Example.

A piston engine is being operated at 10,000 ft and airspeed of 185 KTAS where it delivers 230 BHP. OAT is 30 °F above standard temperature. The manufacturer recommends a constant cylinder-head temperature (CHT) of 450 °F for maximum engine life. Size the inlet and exit area assuming 75% pressure recovery at the radiator and that air temperature rises by 150°F across the cylinders. Estimate how much engine power is lost to cooling.



PISTON ENGINE

Method 2: Inlet-radiator-exit Method – Example.

Step 1: Determine Conditions at Station 0

Calculate the far-field temperature using the information given in the problem:

$$T_0 = 518.69(1 - 0.0000068756 \times 10{,}000) + 30 = 513.0°R$$

Calculate pressure at altitude using the hydrostatic gas equation:

$$p_0 = 2116(1 - 0.0000068756 \times 10{,}000)^{5.2561} = 1455 \text{ psf}$$

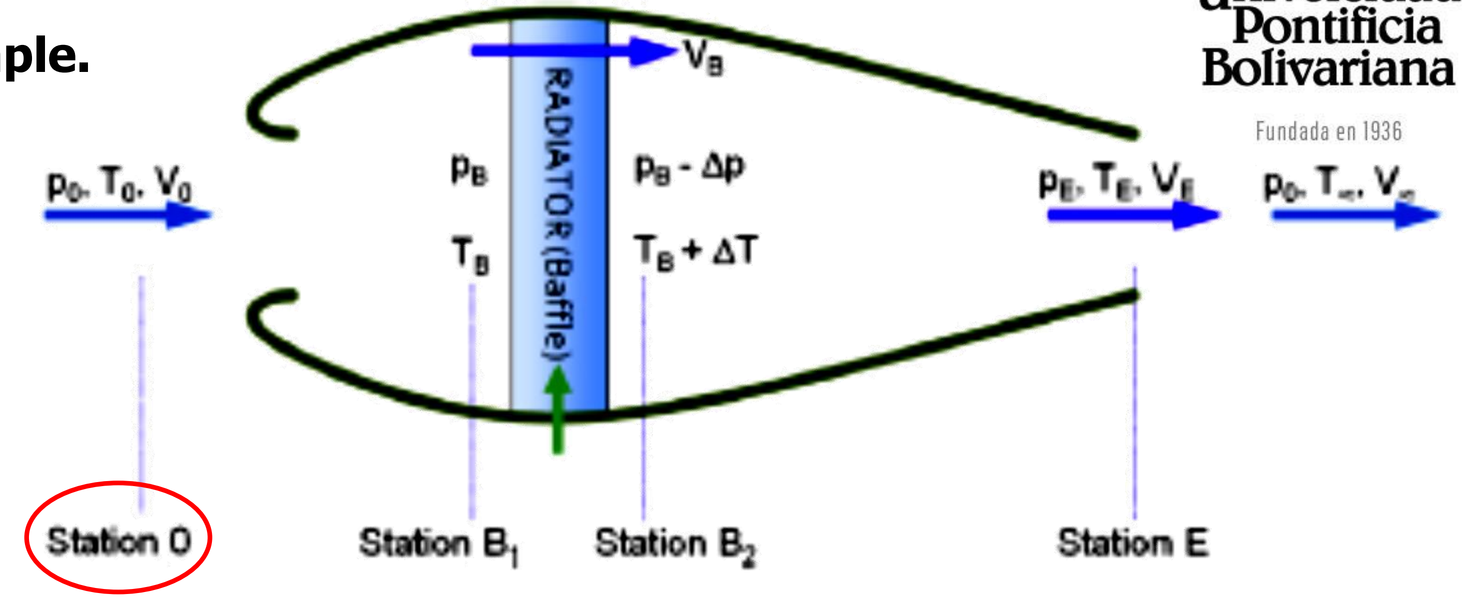
Calculate density using the ideal gas equation:

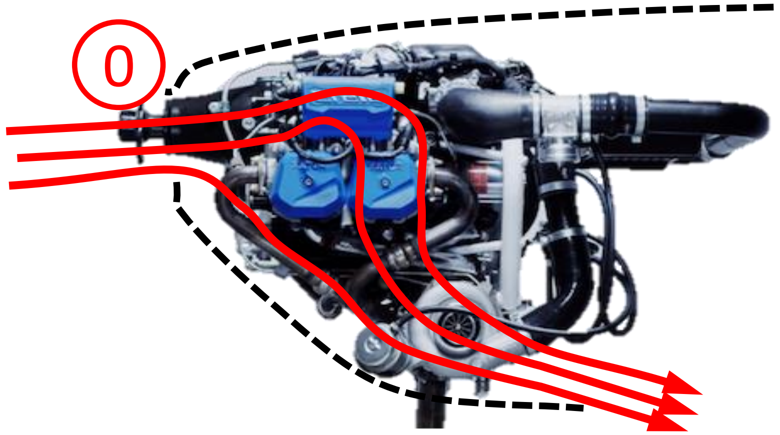
$$\rho = \frac{p_0}{RT_0} = \frac{1455}{(1716)(513.0)} = 0.001653 \text{ slugs/ft}^3$$

Airspeed:

$$V_0 = 185 \times 1.688 = 312.3 \text{ ft/s}$$

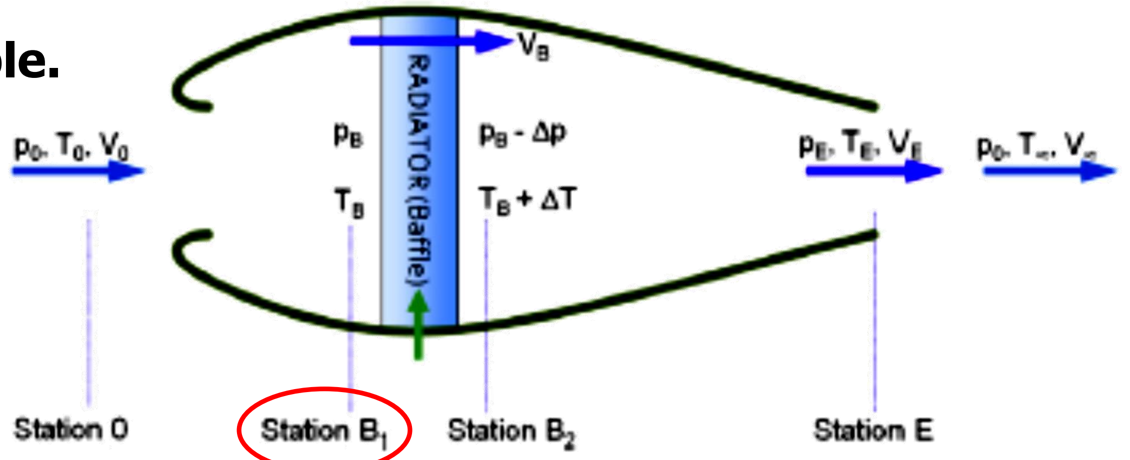
We have thus completely defined p, T, and V in the far-field (station 0).





PISTON ENGINE

Method 2: Inlet-radiator-exit Method – Example.



Step 2: Determine Conditions at Station B_1

Determine the temperature at the radiator front face using the adiabatic gas relation, assuming that adiabatic expansion takes place inside the diffuser.

Therefore, the temperature at the baffle is:

$$\frac{p_0 + kq}{p_0} = \left(\frac{T_{B1}}{T_0}\right)^{\frac{\gamma}{\gamma-1}} \Rightarrow T_{B1} = T_0\left(\frac{p_0 + kq}{p_0}\right)^{\frac{\gamma-1}{\gamma}}$$

Where k is the pressure recovery factor. With that said, let's calculate the dynamic pressure in the far-field to evaluate the impact of the pressure recovery:

$$q = \frac{1}{2}(0.001653)(312.3)^2 = 80.60 \text{ psf}$$

Inserting values (where k = 0.75 for 75% pressure recovery):

$$\boxed{T_{B1} = T_0\left(\frac{p_0 + kq}{p_0}\right)^{\frac{\gamma-1}{\gamma}} = (513.0)\left(\frac{1455 + 0.75(80.60)}{1455}\right)^{\frac{0.4}{1.4}} = 519.0^\circ R}$$

This corresponds to approximately 59.4 °F. Pressure at the baffle assuming 75% pressure recovery:

$$\boxed{p_{B1} = p_0 + kq = 1455 + 0.75 \times 80.60 = 1516 \text{ psf}}$$

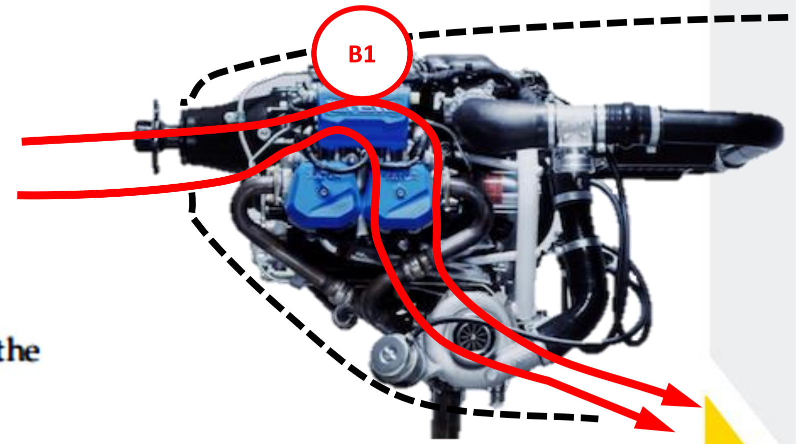
Calculate density using the ideal gas equation:

$$\boxed{\rho = \frac{p_{B1}}{RT_{B1}} = \frac{1516}{(1716)(519.0)} = 0.001702 \text{ slugs/ft}^3}$$

The airspeed at the baffle can now be calculated from the compressible Bernoulli equation:

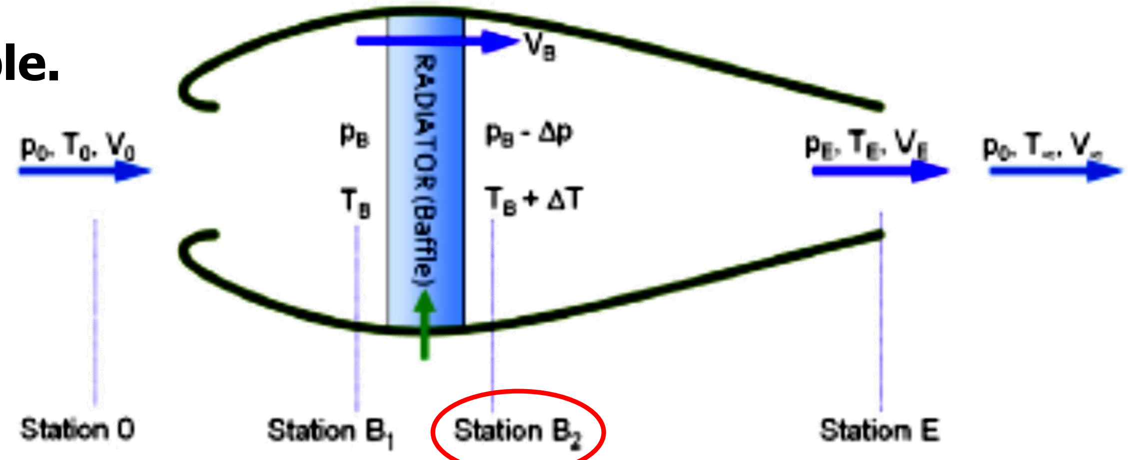
$$\boxed{V_{B1} = \sqrt{V_0^2 + \frac{2\gamma}{\gamma - 1}\left(\frac{p_0}{\rho_0} - \frac{p_{B1}}{\rho_{B1}}\right)} = 159.5 \text{ ft/s}}$$

We have thus completely defined p, T, and V in the far-field (station B1).



PISTON ENGINE

Method 2: Inlet-radiator-exit Method – Example.



Step 3: Determine Conditions at Station B2

$$T_{B1} = 59.4\ °F$$

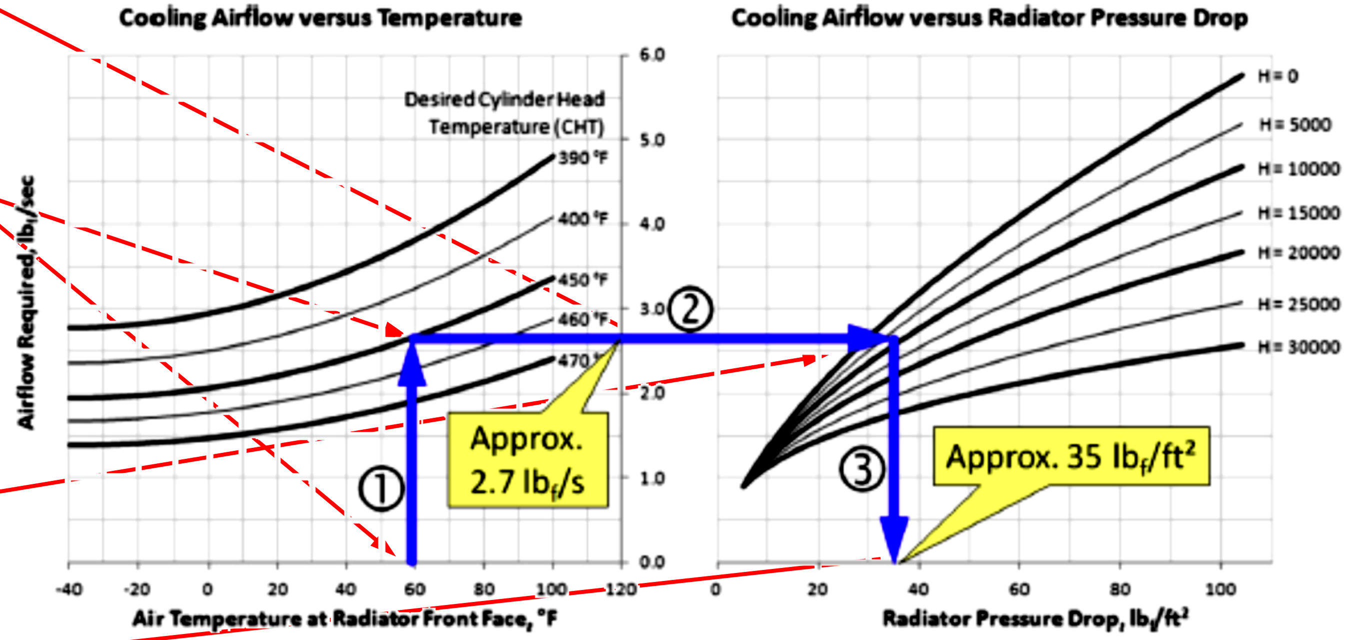
This means that $2.7\ lbf$ of air must flow through the radiator every second to adequately cool the engine

The manufacturer recommends a constant cylinder-head temperature (CHT) of 450 °F for maximum engine life.

The pressure altitude can be estimated using Equation:

$$H_P = 145442\left(1 - \left(\frac{p}{p_0}\right)^{0.19026}\right)$$

$$= 145442\left(1 - \left(\frac{1516}{2116}\right)^{0.19026}\right) = 8947\ \text{ft}$$



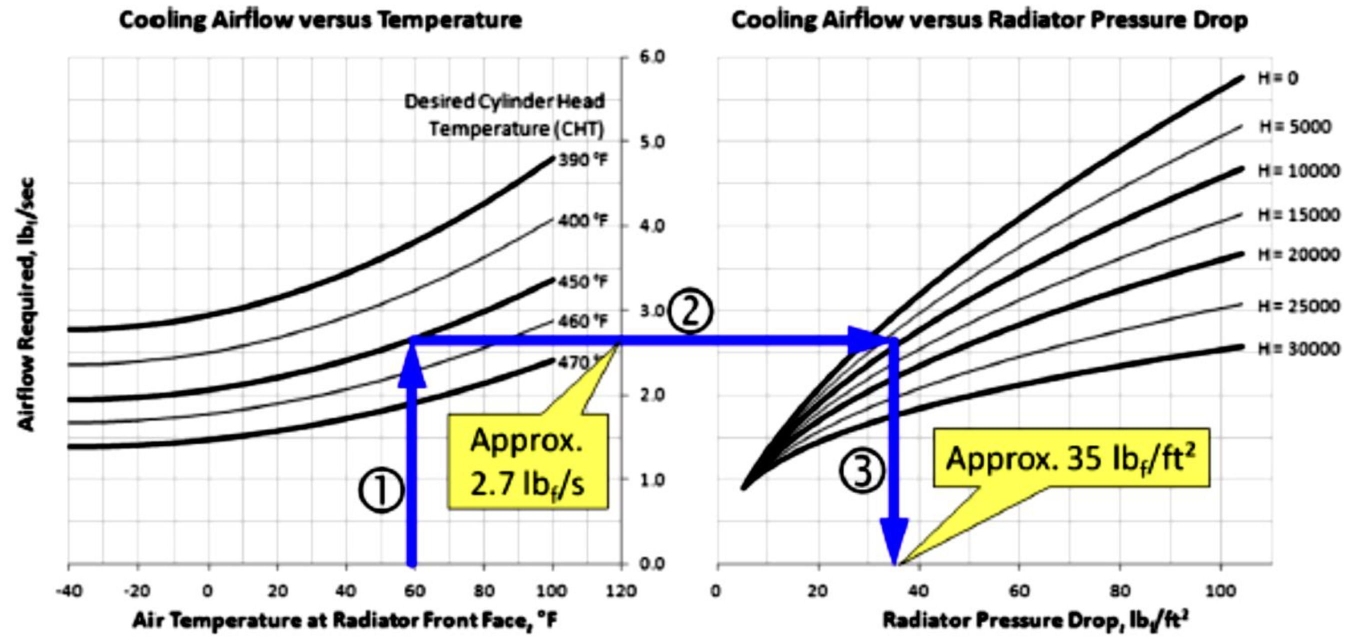
Pressure drop across the radiator

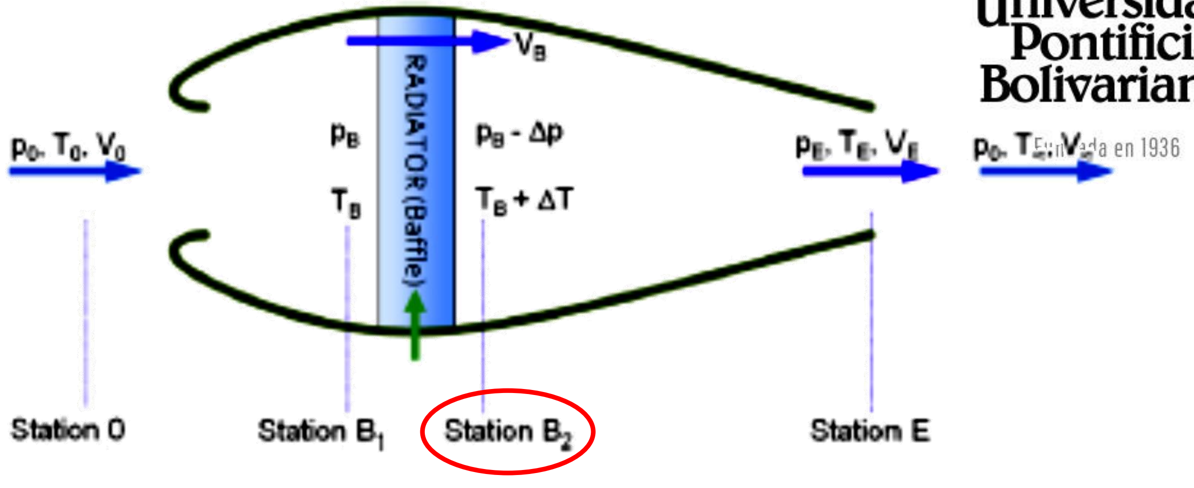
Special graphs supplied by the engine manufacturer are used to extract the required cooling airflow for the engine. The graphs do not represent any particular engine type.

PISTON ENGINE

Universidad Pontificia Bolivariana

Method 2: Inlet-radiator-exit Method – Example.





Required cooling airflow: $\dot{m} = 2.7 \ \text{lb}_f/\text{s}$

Resulting pressure drop: $\Delta p = 35 \ \text{psf}$

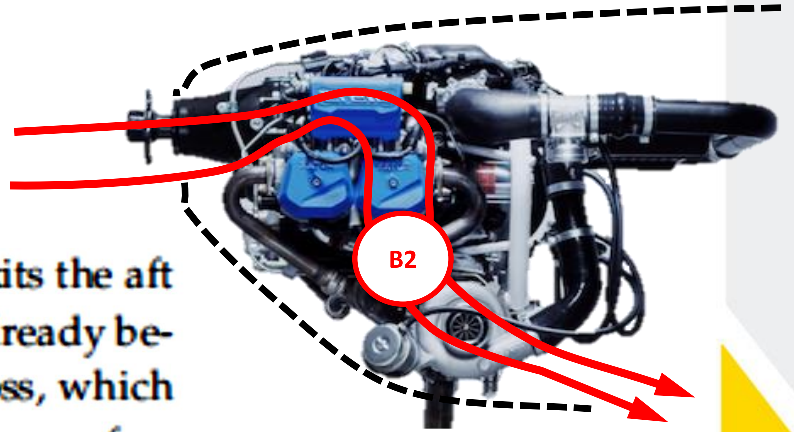
Pressure downstream of the baffle:

$$p_{B2} = p_{B1} - \Delta p = 1516 - 35 = 1481 \ \text{psf}$$

Temperature (+150 °F) rise downstream of the baffle:

$$T_{B2} = T_{B1} + \Delta T = 519.0 + 150.0 = 669.0 \ ^\circ\text{R}$$

It is assumed the speed of air is near zero as it exits the aft face of the radiator. This is based on the speed already being accounted for in the value of the pressure loss, which was empirically determined (by the engine manufacturer). Therefore we say that $V_{B2} = 0$ ft/s and, thus, claim we have completely defined p, T, and V in the far-field (station B1).



PISTON ENGINE

Universidad Pontificia Bolivariana

Fundada en 1936

Method 2: Inlet-radiator-exit Method – Example.

Step 4: Determine Conditions at Station E

The pressure at the exit is assumed to be the atmospheric pressure in the far-field.

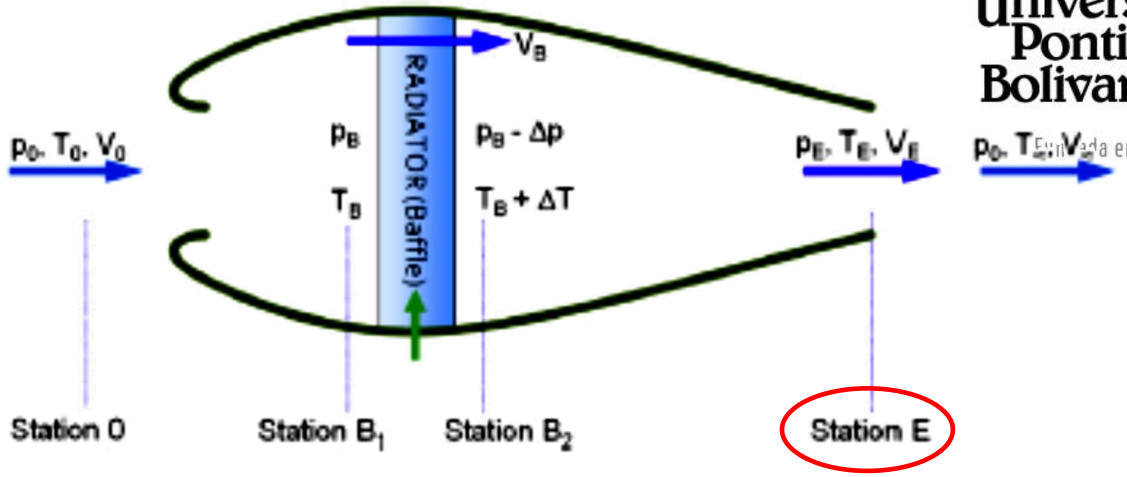
$$\boxed{p_E = p_0}$$

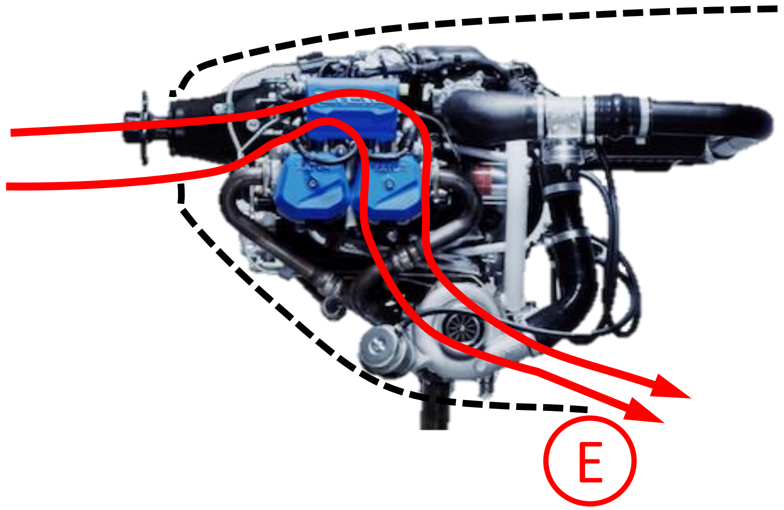
The density at the exit can be found from the adiabatic relation:

$$\frac{p_E}{p_{B2}} = \left(\frac{\rho_E}{\rho_{B2}}\right)^{\gamma} \Rightarrow \rho_E = \rho_{B2}\left(\frac{p_E}{p_{B2}}\right)^{\frac{1}{\gamma}} \Rightarrow \rho_E$$

$$= 0.001290\left(\frac{1455}{1481}\right)^{\frac{1}{\gamma}} = \boxed{0.001274 \text{ slugs/ft}^3}$$

Where: $$\rho_{B2} = \frac{p_{B2}}{RT_{B2}} = \frac{1481}{(1716)(669.0)} = 0.001290 \text{ slugs/ft}^3$$





PISTON ENGINE

Method 2: Inlet-radiator-exit Method – Example.

Step 4: Determine Conditions at Station E

Using Bernoulli's equation to find the airspeed at the exit:

$$\frac{V_{B2}^2}{2} + \frac{\gamma}{\gamma - 1}\frac{p_{B2}}{\rho_{B2}} = \frac{V_E^2}{2} + \frac{\gamma}{\gamma - 1}\frac{p_E}{\rho_E}$$

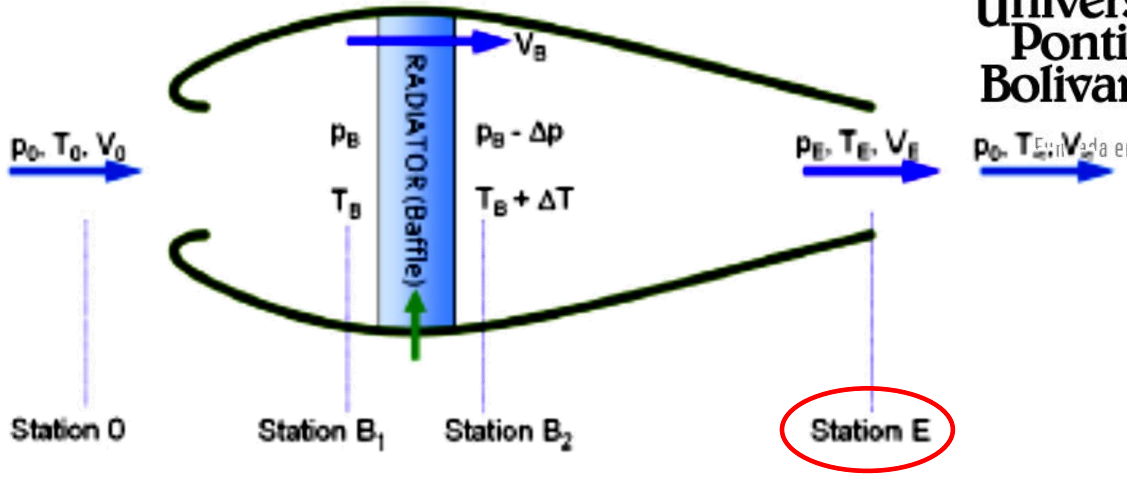
Assuming speed V_{B2} through the baffle to be small:

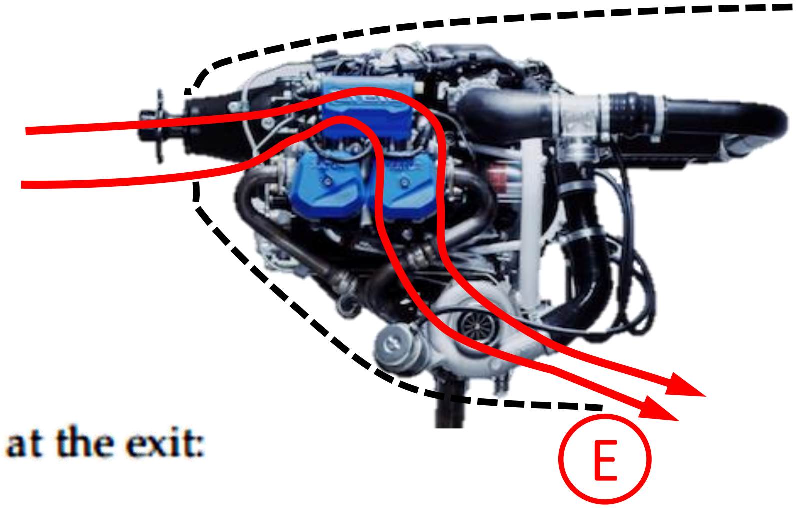
$$0 + \frac{\gamma}{\gamma - 1}\frac{p_{B2}}{\rho_{B2}} = \frac{V_E^2}{2} + \frac{\gamma}{\gamma - 1}\frac{p_E}{\rho_E}$$

$$\Rightarrow \ V_E = \sqrt{\frac{2\gamma}{\gamma - 1}\left(\frac{p_{B2}}{\rho_{B2}} - \frac{p_E}{\rho_E}\right)}$$

The airspeed at the exit comes to:

$$\boxed{\begin{aligned} V_E &= \sqrt{\frac{2\gamma}{\gamma - 1}\left(\frac{p_{B2}}{\rho_{B2}} - \frac{p_E}{\rho_E}\right)} \\ &= \sqrt{\frac{2(1.4)}{1.4 - 1}\left(\frac{1487}{0.001295} - \frac{1455}{0.001275}\right)} = 199.3 \ \text{ft/s} \end{aligned}}$$





Temperature at the exit:

$$\boxed{T_E = T_{B2}\left(\frac{P_E}{P_{B2}}\right)^{\frac{\gamma-1}{\gamma}} = (669.0)\left(\frac{1455}{1481}\right)^{\frac{0.4}{1.4}} = 665.7^\circ R}$$

PISTON ENGINE

Universidad Pontificia Bolivariana

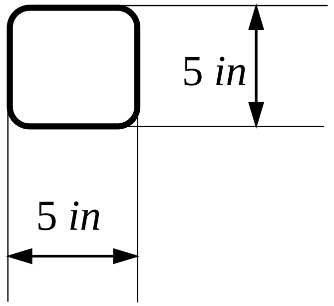
Method 2: Inlet-radiator-exit Method – Example.

Step 4: Determine Conditions at Station E

Now we have enough information to size the inlet area:

$$\dot{m} = \rho_0 V_0 A_{IN} \Rightarrow A_{IN} = \frac{\dot{m}}{\rho_0 V_0} \Rightarrow A_{IN}$$

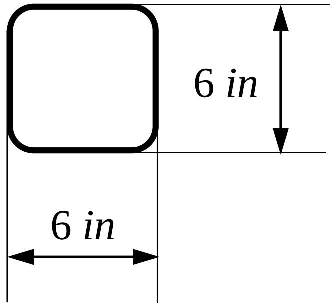
$$= \frac{2.7/32.174}{(0.001653)(312.3)} = \boxed{0.163 \ \text{ft}^2}$$

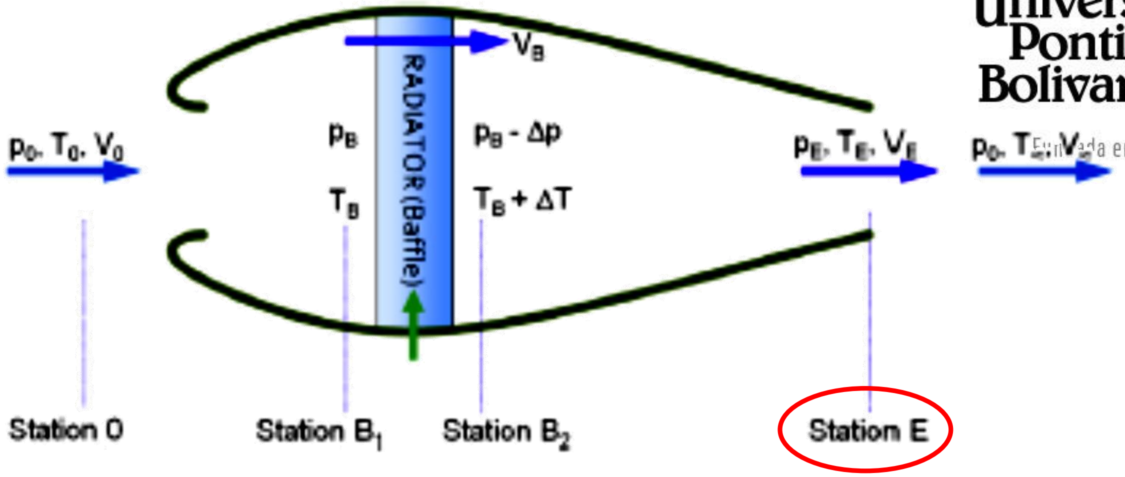


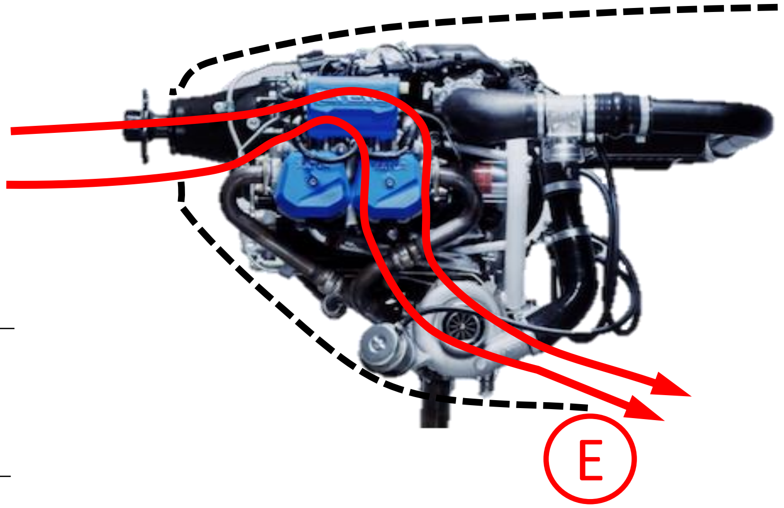
And the outlet area:

$$\dot{m} = \rho_E V_E A_E \Rightarrow A_E = \frac{\dot{m}}{\rho_E V_E}$$

$$\boxed{\Rightarrow A_E = \frac{2.7/32.174}{(0.001274)(199.3)} = 0.331 \ \text{ft}^2}$$



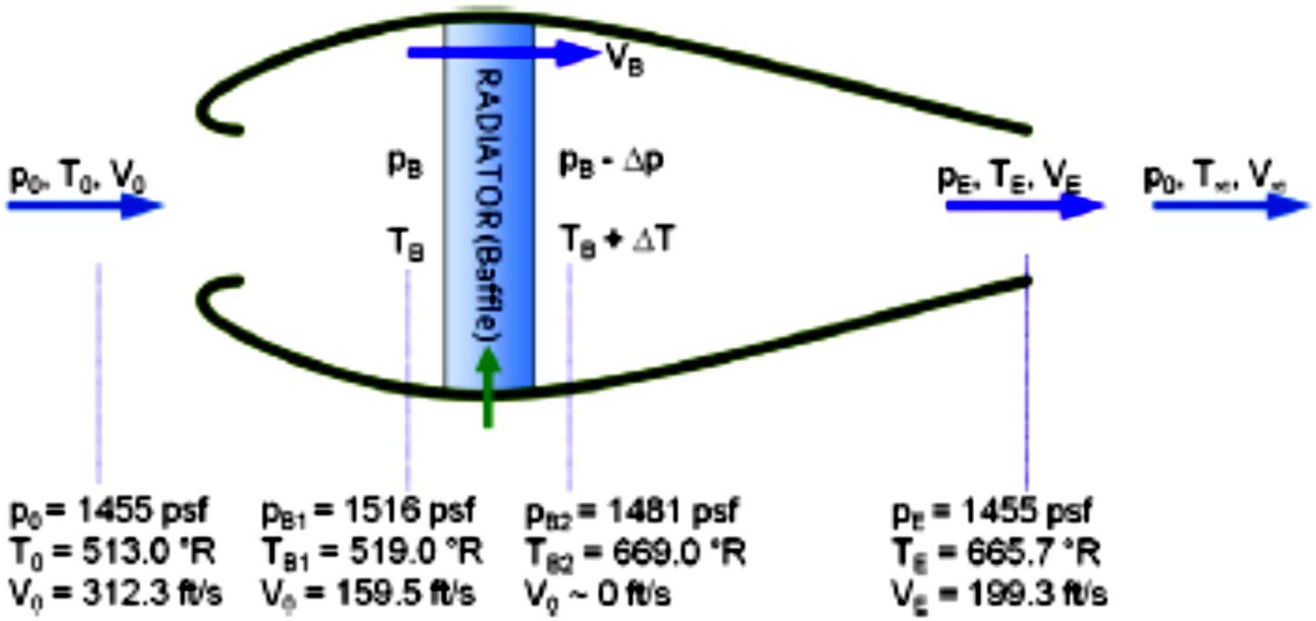


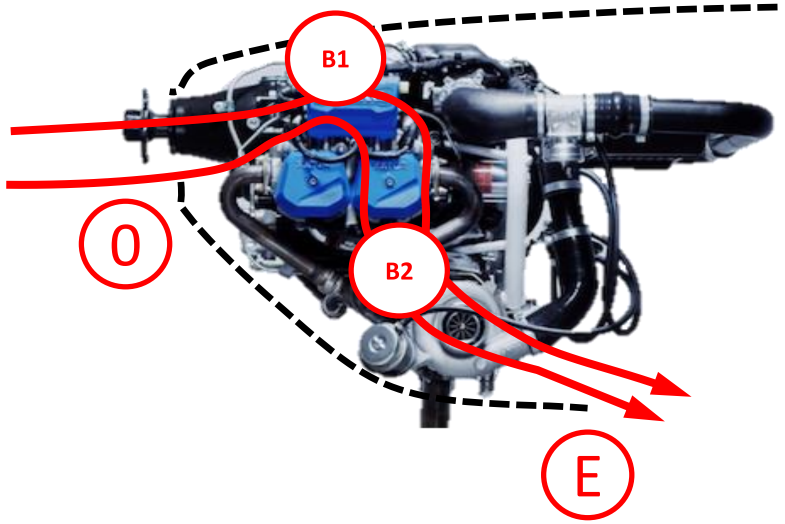


PISTON ENGINE

Method 2: Inlet-radiator-exit Method – Example.

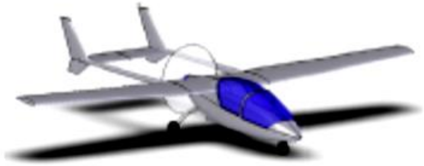
Universidad Pontificia Bolivariana
Fundada en 1936





PROPELLERS

Propeller configuration – methods of mounting.

Configuration	Advantages	Disadvantages
Tractor (A) 	The incoming air is undisturbed. Ground clearance is not an issue during rotation at T-O, or flare before touch-down.[a] Placing the fuselage behind the propeller allows for a reduced streamtube inflow distortion and less asymmetric disc loading that would increase blade stresses. There is less chance that the propeller suffers damage due to FOD, in particular when the aircraft is moving. Propeller is not subject to excessive heat from the exhaust. Propwash can help with T-O rotation during soft- or short-field take-offs, by increasing the dynamic pressure at the horizontal tail.	Obstructed forward view, which usually requires the back side of the prop to be painted black. Increased cabin noise. Normal force forward of CG decreases stability. The turbulent high-speed wake flows over a fuselage or a nacelle and (at least theoretically) increases drag. When placed in the front with nose landing gear, there is a risk of ground-strike in a hard landing. Frequency of propwash pressure pulses may excite structural frequencies at some specific RPMs, resulting is structural vibrations.
Pusher (B) 	Unobstructed forward view. Reduced cabin noise. Normal force aft of the CG increases stability. Turbulent high-speed wake does not flow over a fuselage or a nacelle, and, at least theoretically, will result in less drag. The streamtube will energize the flow in front of propeller and suppress flow separation on the body, even at high *AOA*.	Ground clearance may be an issue during rotation at T-O, or flare before landing. Fuselage ahead of the propeller may distort flow inside the streamtube, causing asymmetric disc loading and increased blade stresses. This distortion may affect the propeller's performance. Propeller may suffer FOD, in terms of both pebbles shot by tires and ingestion by ice shedding off a fuselage. Propeller may be subject to excessive heat from the exhaust. Special regulatory requirements for pushers are stipulated in 14 CFR 23.905. Results in a higher fly-over noise and possible propeller corrosion issues (explained below).

[a]Per 14 CFR Part 23.925 (Propeller clearance), there must be a minimum 7″ ground clearance for a tricycle landing gear and 9″ for taildraggers, at the most adverse combination of CG, weight, the most adverse pitch position of the propeller, and at static ground deflection of the landing gear. Aircraft with leaf spring struts as landing gear must comply with 1.5 times its most adverse weight.

PROPELLERS

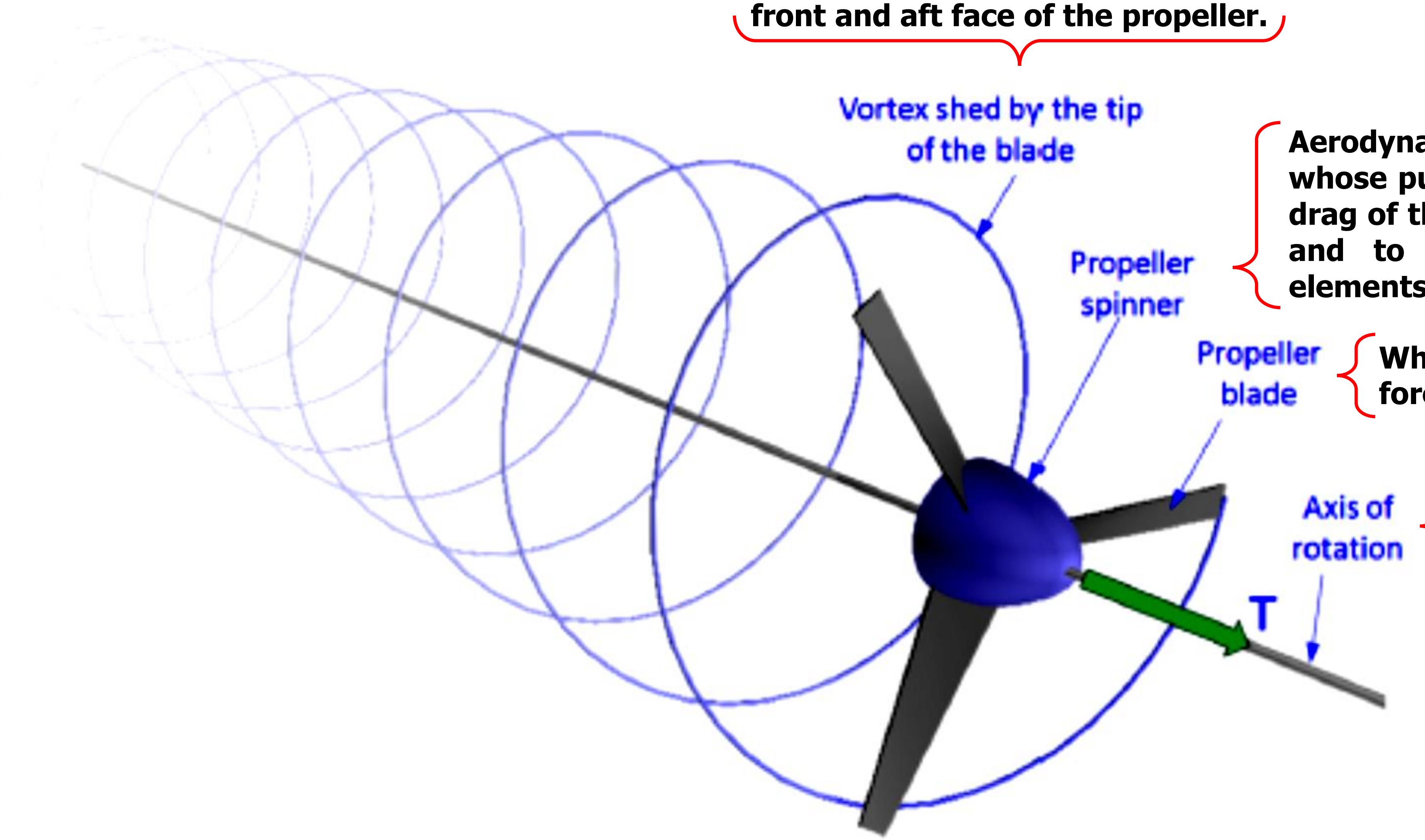
Table 8.10 Features of several propellers

No.	Manufacturer	Designation	Number of blades	Diameter	Features	Aircraft
1	Smiths	R381	6	12 ft 6 in.	Constant speed	Saab 2000
2	Smiths	R391	6	13 ft 6 in.	Constant speed, composite	Lockheed C-130J, Alenia C-27J
3	Smiths	R408	6	13 ft 6 in.	Constant speed, composite	Dash 8 Q400, Y-8F600
4	Sensenich	W72CK-42	2	70 in.	Wood	Aeronca 7AC
5	Hamilton Sundstrand	14SF-5	4	13 ft	Aluminum and composite	ATR 72, Canadair 215T, Dash 8 Q300
6	Hamilton Sundstrand	568F-1	6	13 ft	Constant speed, composite	Casa c-295, Ilyushin Il-114
7	Hartzell	HC-E4A-2/E9612	4	77 in.	Metal, fully feathering	Beechcraft T-6
8	Hartzell	–	5	77 in.	Composite, constant speed, reversible pitch	Embraer EMB-314 Super Tucano
9	Hartzell	HC-D4N-ZA/09512A	4	77 in.	Metal, constant speed, fully feathering	Pilatuc PC-9 M
10	Hartzell	HC-B3TN-3	4	77 in.	Metal, constant speed, fully feathering	Beechcraft King Air 90
11	Hartzell	HC-E5N	5	77 in.	Aluminum, constant speed, fully feathering, reversible pitch	Piaggio P-180 Avanti II
12	Hartzell	HC-E4N-3Q	4	77 in.	Metal	Piper PA-46-500TP Meridian
13	Hartzell	BHC-J2YF-1BF	2	72 in.	Metal	Cirus SR20
14	Hartzell	HC-B5MP-3F	5	72 in.	Metal, constant speed, feathering, reversible pitch	Air tractor AT-802
15	Hartzell	HC-C2YK-1BF	2	74 in.	Metal, constant speed	Aermacchi SF-260EA

PROPELLERS

Propeller geometry.

Pressure differential between the front and aft face of the propeller.



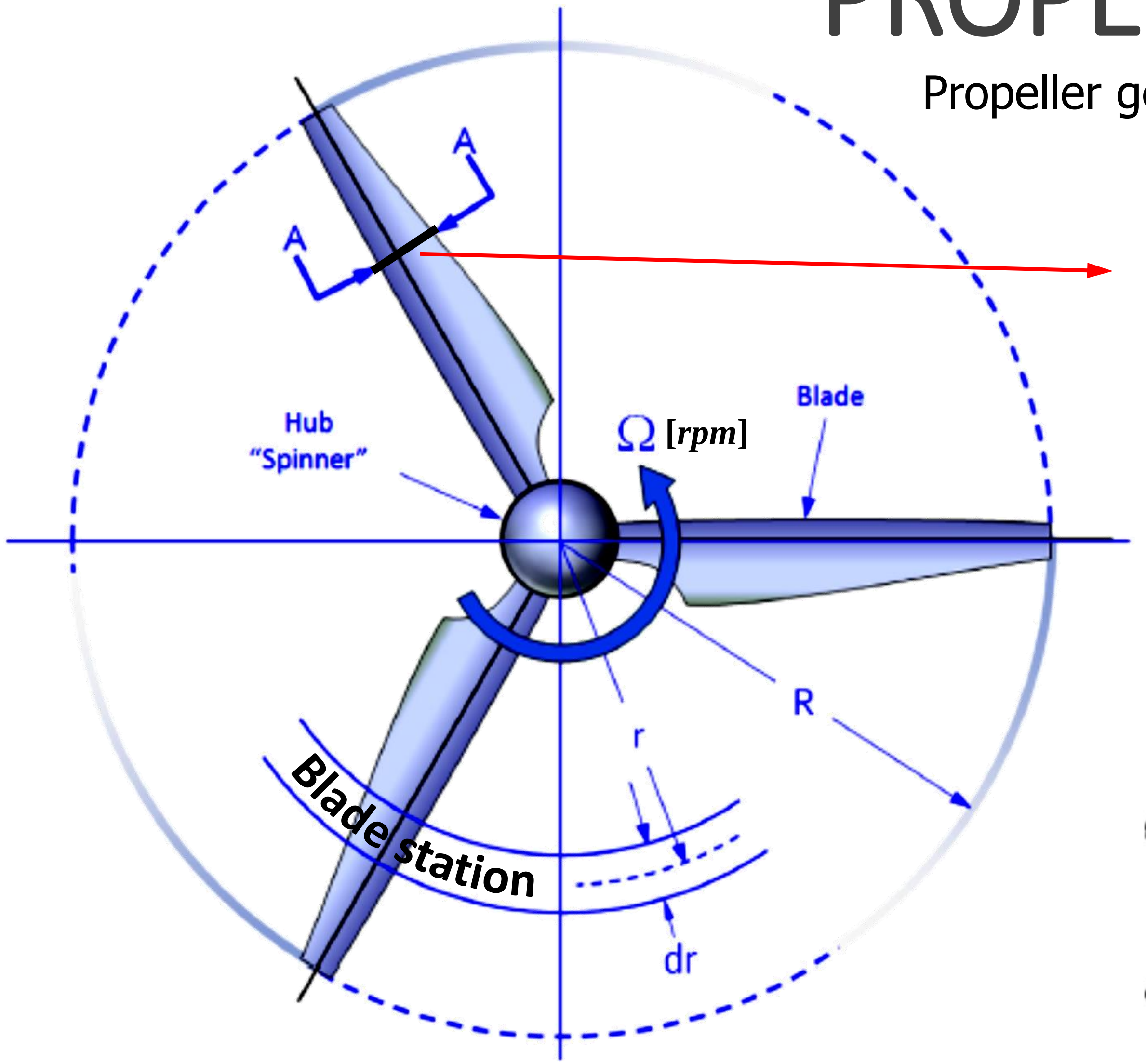
Aerodynamically shaped cover, whose purpose is to reduce the drag of the hub of the propeller and to protect it from the elements.

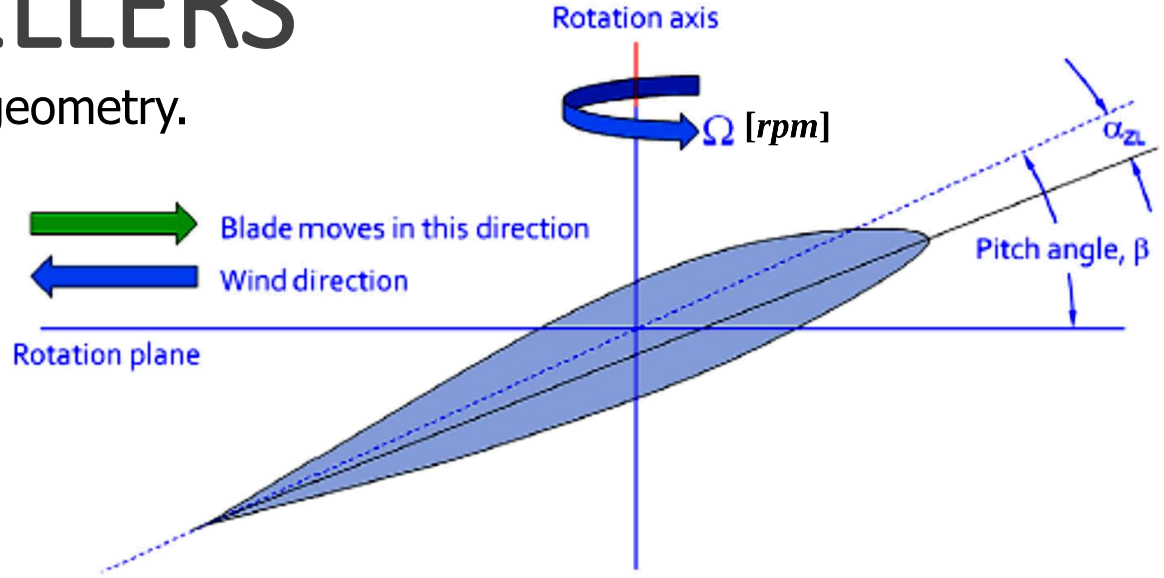
What generates the thrust force (T) of the device.

Propeller rotates about this axis.

PROPELLERS

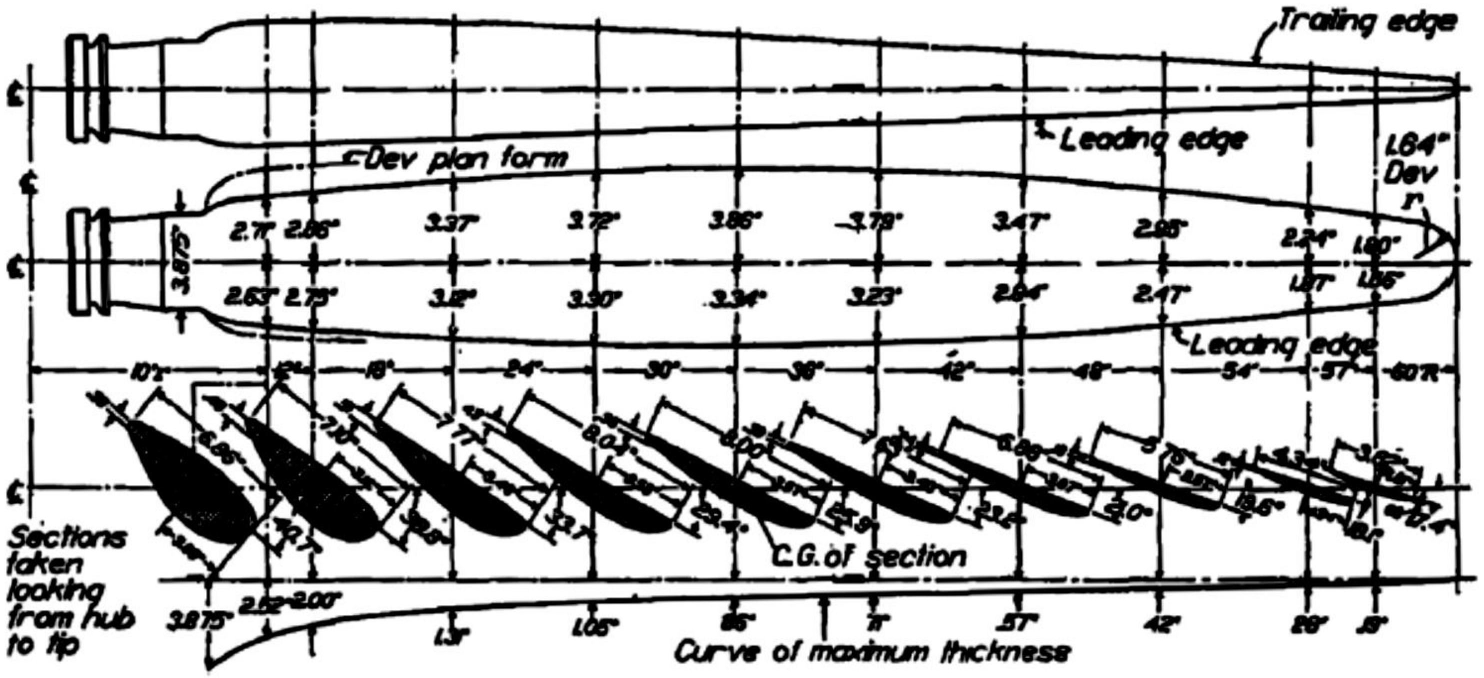
Propeller geometry.





Factors of crucial importance to optimize a propeller.

Twist angle.
Airfoil camber.

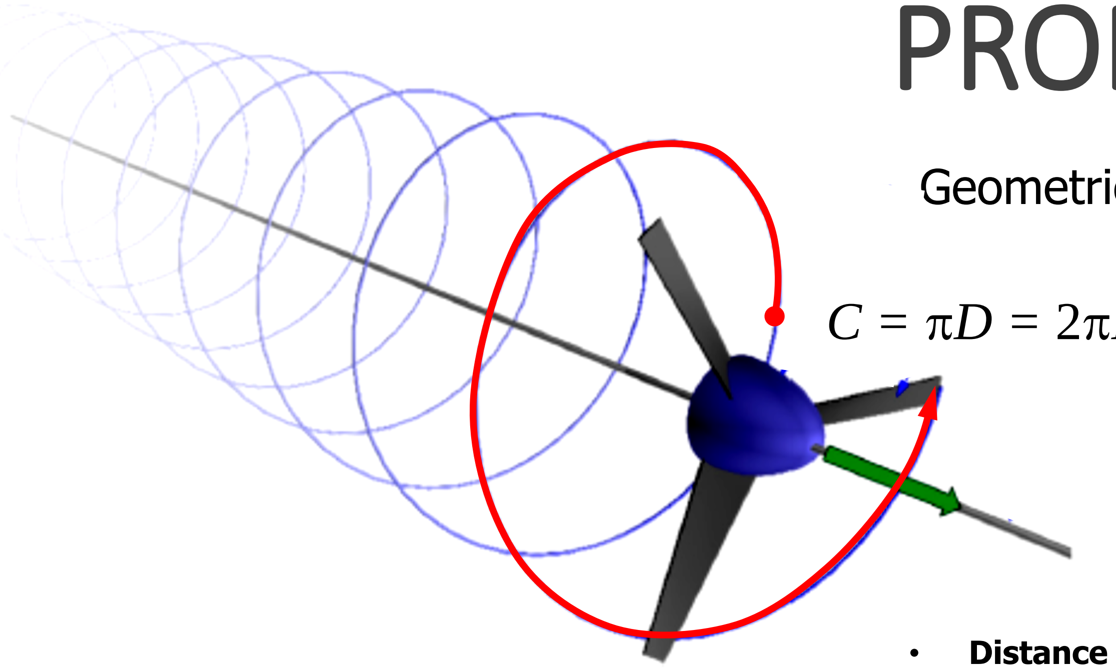


Operational "costs":
- **Power required to rotate the blades.**
- **Noise.**

Universidad Pontificia Bolivariana
Fundada en 1936

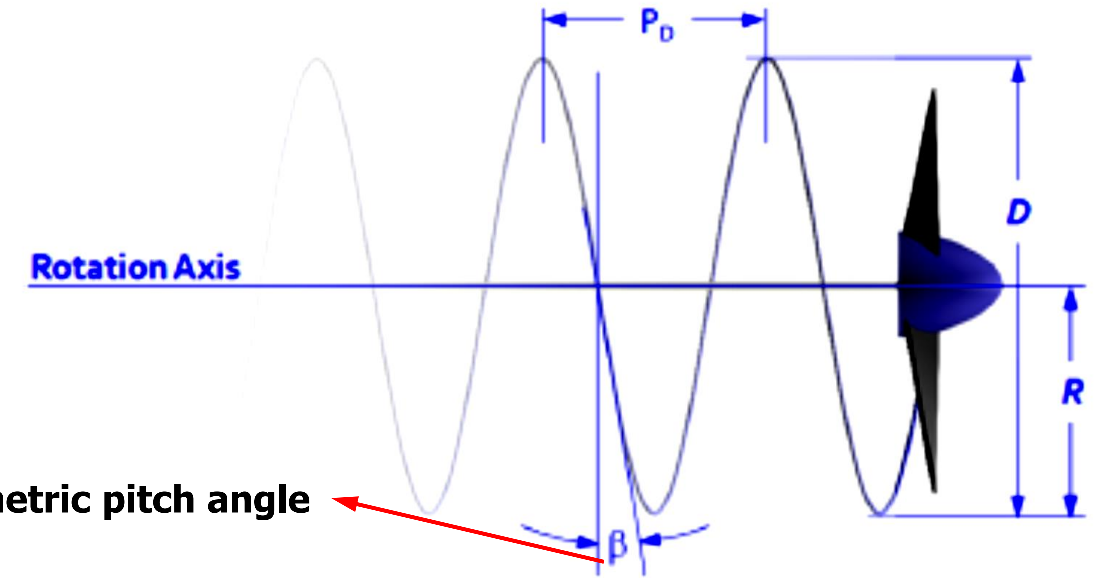
PROPELLERS

Geometric propeller pitch.



$C = \pi D = 2\pi R$

- **Distance cover by the propeller in one revolution.**
- **It ranges from 60% to 85% of D.**



Geometric pitch angle

Pitch angle:

$$\tan\beta = \frac{P_D}{2\pi r_{ref}}$$

where

r_{ref} = reference radius, usually 75% of the propeller radius R

P_D = pitch distance of the propeller

A propeller designated as 42-inch pitch prop. → The propeller will move 42 inches forward in one revolution.

The *pitch-to-diameter ratio* is also used to identify propellers:

$$\frac{P_D}{D} = \frac{\pi \cdot r}{R}\tan\beta$$

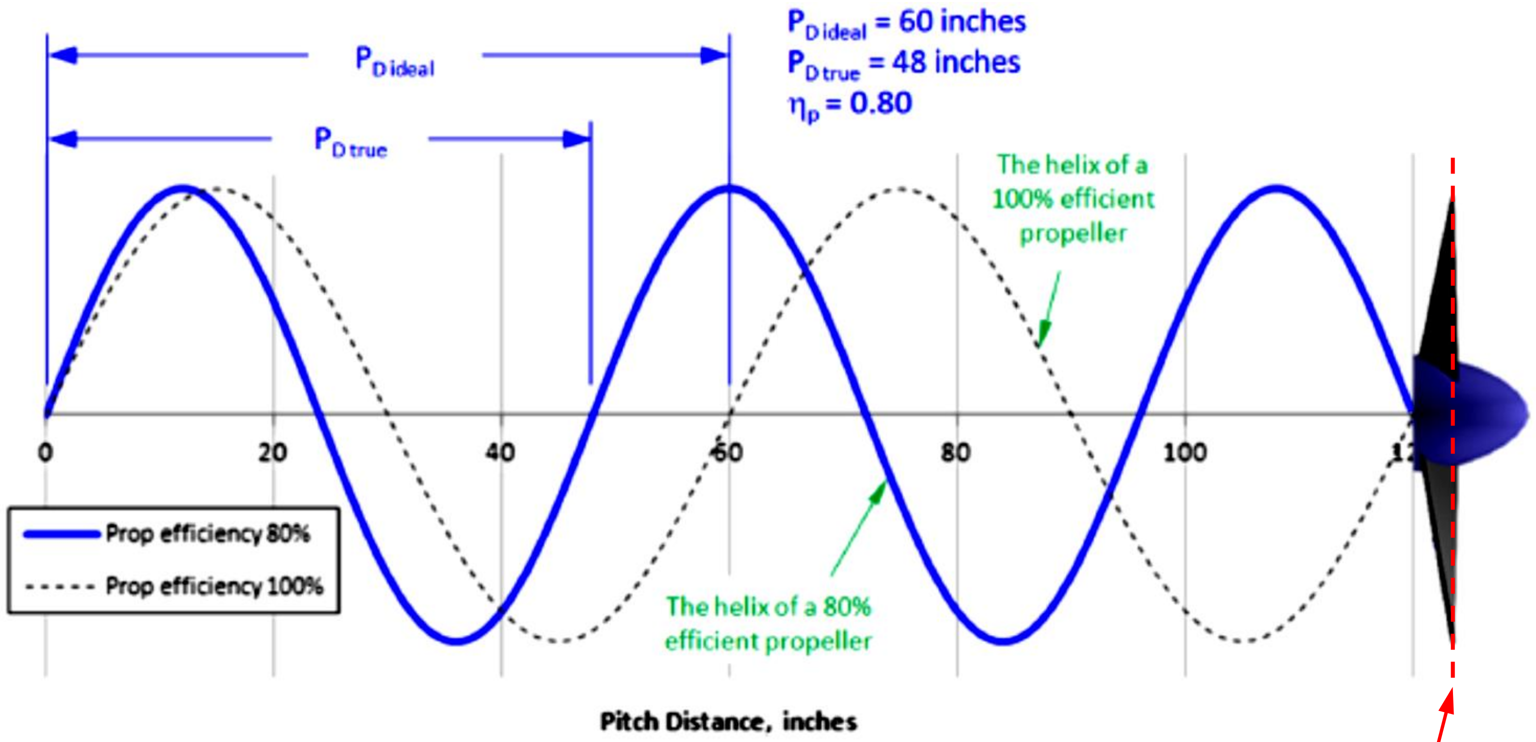
Universidad Pontificia Bolivariana

Fundada en 1936

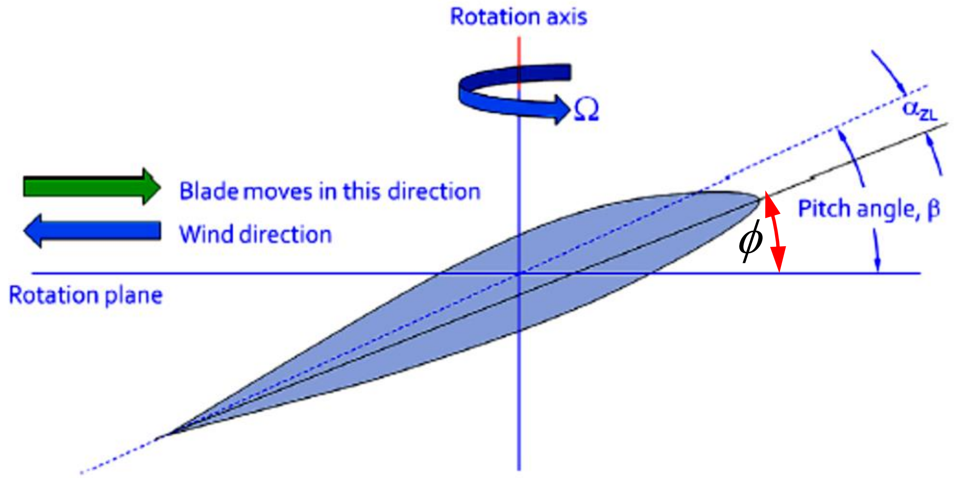
PROPELLERS

Geometric propeller pitch.

A propeller moving through a low-viscosity fluid like air will cover less distance per revolution than the geometric pitch would indicate.



Rotational plane



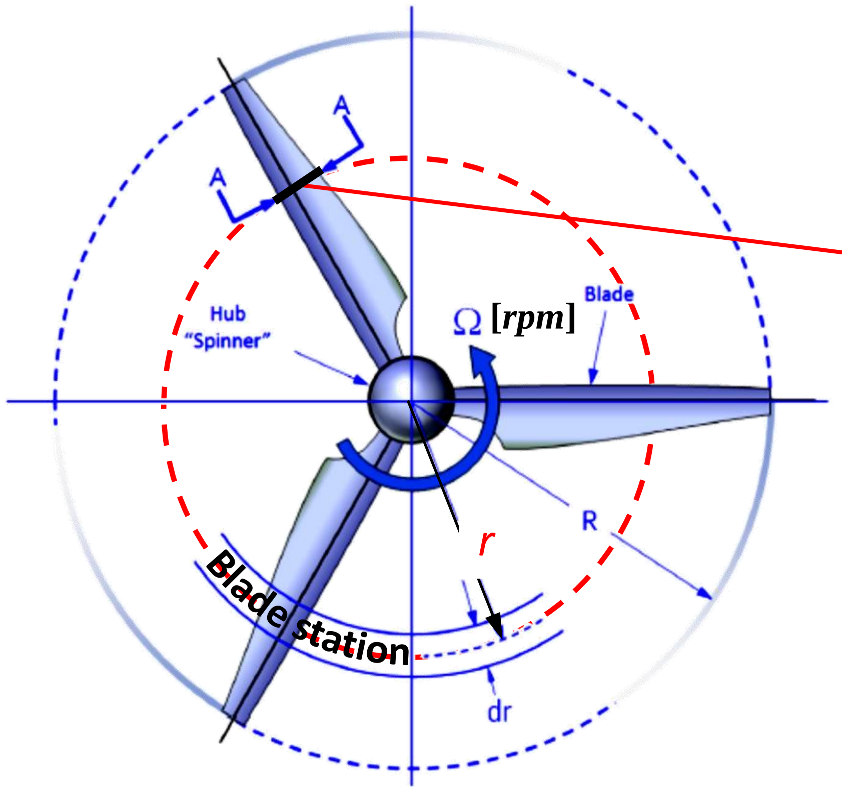
Helix angle:

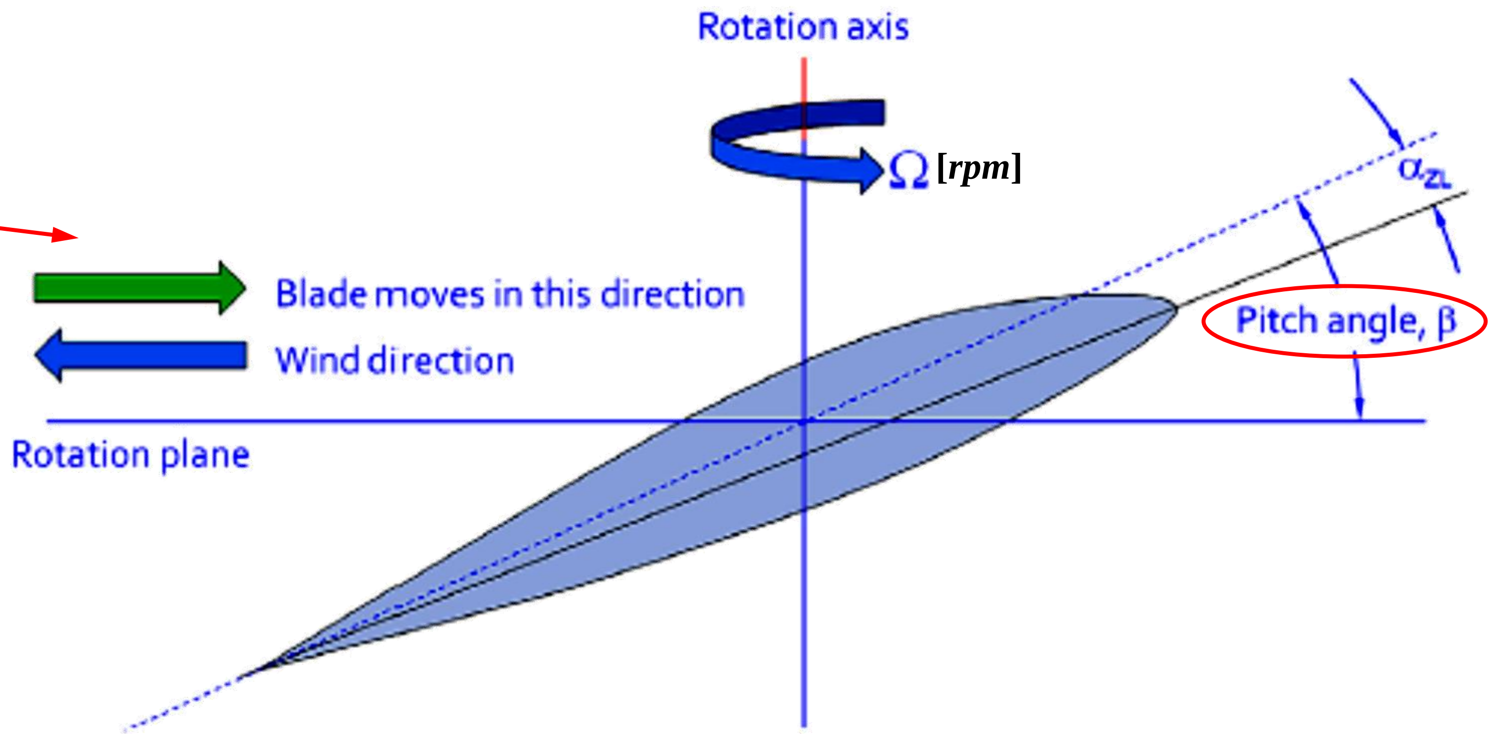
$$\tan\phi = \frac{2\pi\ r\ n}{V_0} = \frac{\pi\ r\ RPM}{30 \cdot V_0}$$

PROPELLERS

Pitch angle or Geometric Pitch (β).

For the propeller to generate thrust in a forward direction its chord line must form a positive angle-of-attack to the relative wind as it moves about its rotation axis.

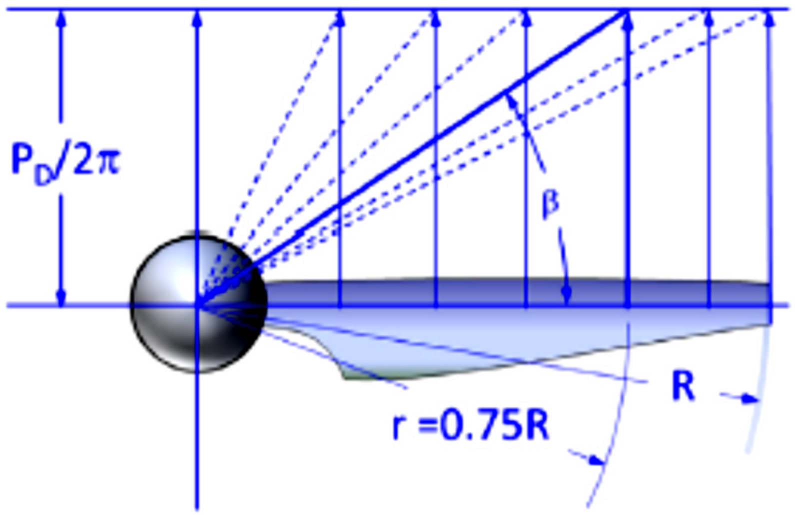




PROPELLERS

Pitch angle or Geometric Pitch (β).

Constant-pitch propeller.



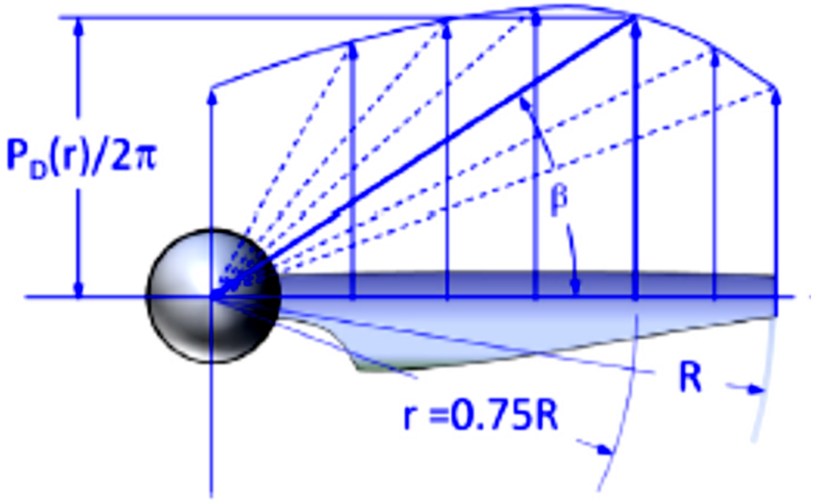
Each radial station moves a constant pitch distance.

Constant-pitch propeller:

$$\tan\beta = \frac{P}{2\pi r}$$

where; β = pitch angle and r = arbitrary blade station along a propeller blade.

Variable-pitch propeller.

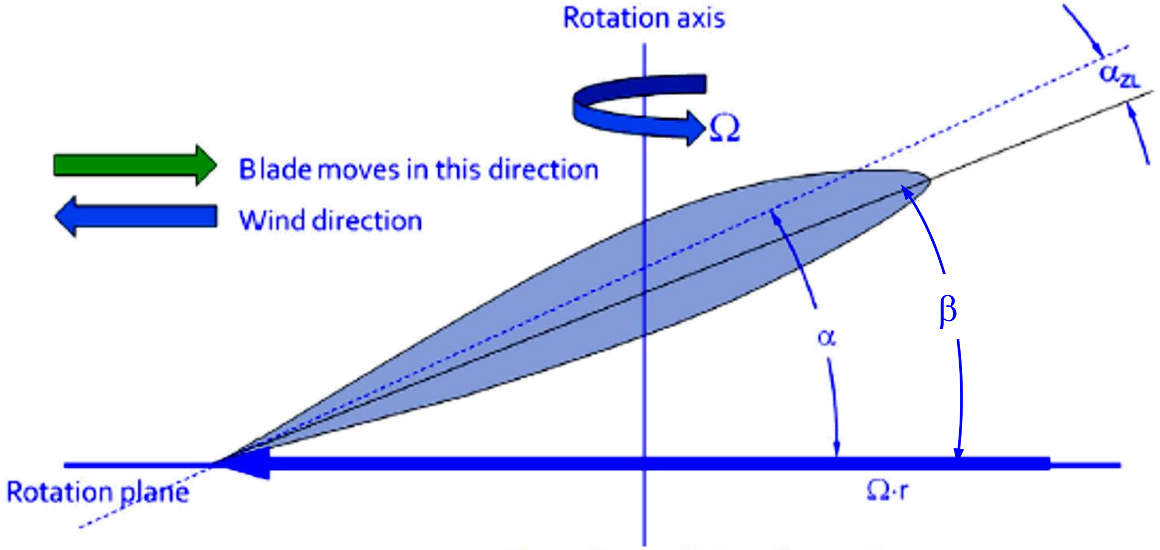


Pitch distance is not fixed but changes gradually along the span of the propeller. The purposes of this are:

- To load the propeller disc plane in some desirable way.
- To prevent the inboard sections from stalling under some specific conditions.
- To prevent the section C_l at the tip from exceeding a maximum value (airplane's operation cond.).
- To modify the distribution of section C_l (improve efficiency).

PROPELLERS

Fundamental Relationships of Propeller Rotations.



$$V_R = \sqrt{V_0^2 + V_{ROT}^2} = \sqrt{V_0^2 + (\Omega \cdot r)^2}$$

Propeller helical tip speed:

$$V_{tip} = \sqrt{V_0^2 + (\Omega R_{prop})^2} = \sqrt{V_0^2 + (\pi n D)^2}$$
$$= \sqrt{V_0^2 + \left(\frac{\pi \cdot RPM \cdot D}{60}\right)^2}$$

Tip Mach number:

$$M_{tip} = \frac{V_{tip}}{a_0} = \frac{V_{tip}}{\sqrt{\gamma R T}}$$

Propeller rotational speed:

$$V_{ROT} = \Omega \cdot r$$

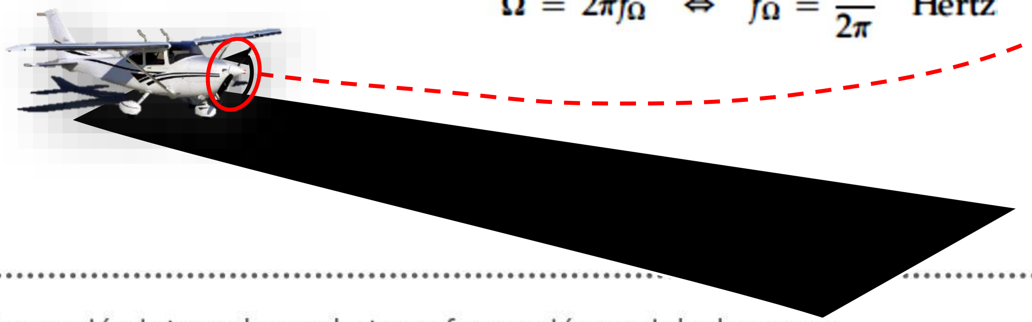
Period of rotation:

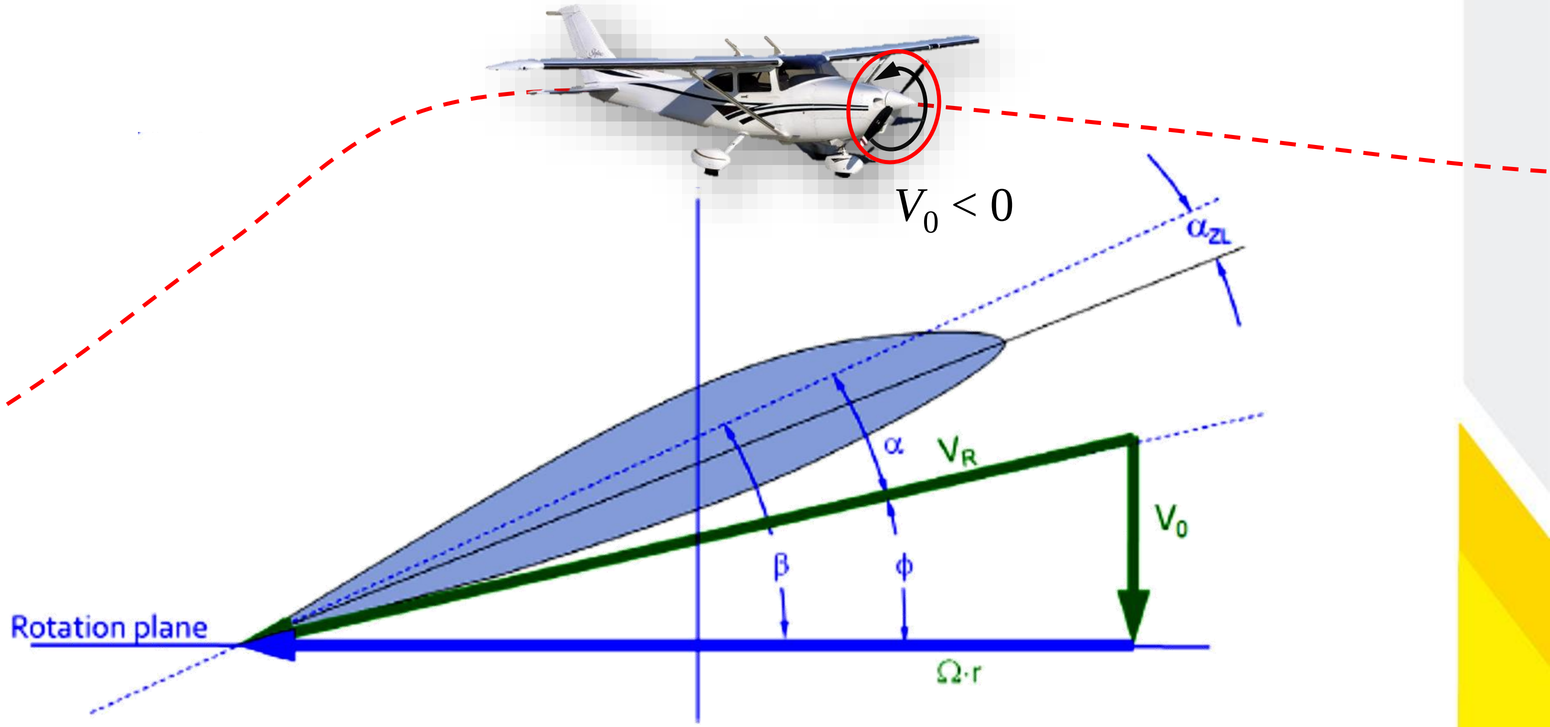
$$\Omega = \frac{2\pi}{P_\Omega} \Leftrightarrow P_\Omega = \frac{2\pi}{\Omega} \quad \text{seconds}$$

Frequency of rotation:

$$\Omega = 2\pi f_\Omega \Leftrightarrow f_\Omega = \frac{\Omega}{2\pi} \quad \text{Hertz}$$

$V_0 = 0$





PROPELLERS

Determination of the Desired Pitch for Fixed-Pitch Propellers.

The selection of a propeller requires its diameter and pitch to be specified. While it is recommended the designer consults with a propeller manufacturer on the appropriate dimensions.

Propeller pitch distance:

Cruising speed [$KTAS$]

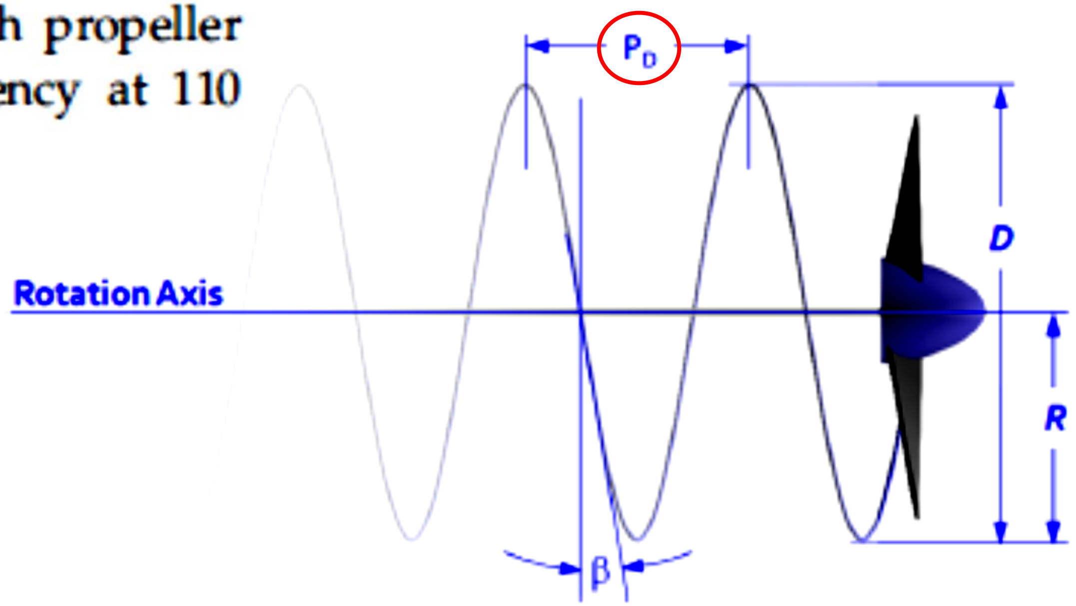
$$P_D \approx 1251 \left(\frac{V_{KTAS}}{RPM}\right)\left(\frac{1}{\eta_P}\right)$$

Propeller efficiency

Example: **Calculate the required pitch of a fixed pitch propeller rotating at 2400 RPM, assuming 75% efficiency at 110 KTAS.**

Solution

$$P_D \approx 1251 \left(\frac{V_{KTAS}}{RPM}\right)\left(\frac{1}{\eta_P}\right) = 1251 \left(\frac{110}{2400}\right)\left(\frac{1}{0.75}\right)$$
$$= \boxed{76.5 \text{ inches}}$$

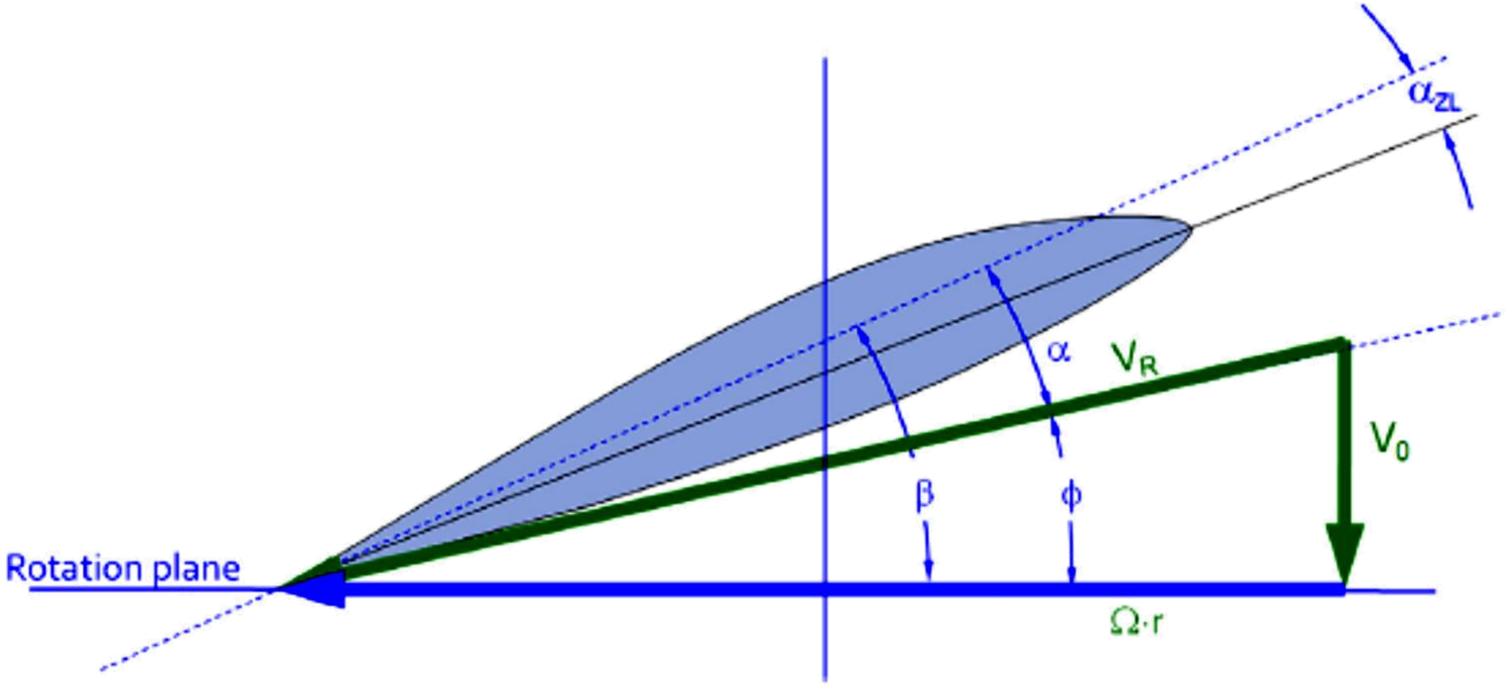


Universidad Pontificia Bolivariana
Fundada en 1936

PROPELLERS

Fixed and Constant-Speed Propellers.

Propeller efficiency (η_p) is an indicator of how much engine power is being converted into propulsive power ($P_R = T \times V_0$). Thus, a particular propeller may be 0.80 efficient at a specific condition. This means that 80% of the engine power (BHP) is being converted into propulsive power.



Variations in airspeed (V_0)

↓

Variations in AoA

↓

Variations in propeller thrust → $\eta_p = f(V_0)$

Aircraft propellers are designed for a particular V_0 or range of airspeeds specified by the airframe manufacturer.

For low-speed operations use a low pitch.
For high-speed operations, use a high pitch.

PROPELLERS

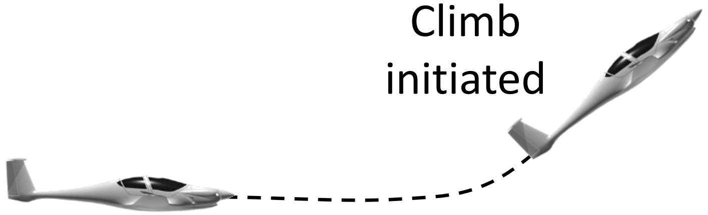
Fixed and Constant-Speed Propellers

A **fixed-pitch propeller** is one in which the blade pitch angles are permanently fixed.

Advantages: Simple, light and inexpensive.

Disadvantages: Their best η_p is achieved at a particular V_0 only (i.e., climb, cruise).

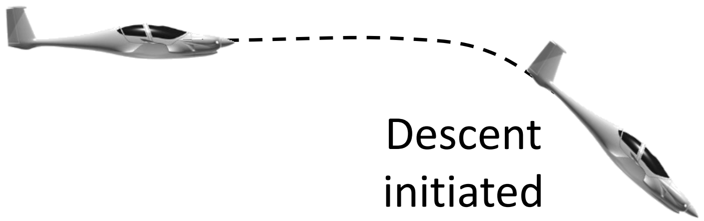
Climb initiated



Cruise

Engine RPM $\downarrow$ $\rightarrow$ $V_0 \downarrow$ $\rightarrow$ Propeller blade AoA $\uparrow$ $\rightarrow$ $D \uparrow$ $\rightarrow$ Higher Q is being generated and this slows down the engine RPM.

If the pilot does not change the power setting, RPM will drop as well, and the propeller will now be operating at a different η_p



Descent initiated

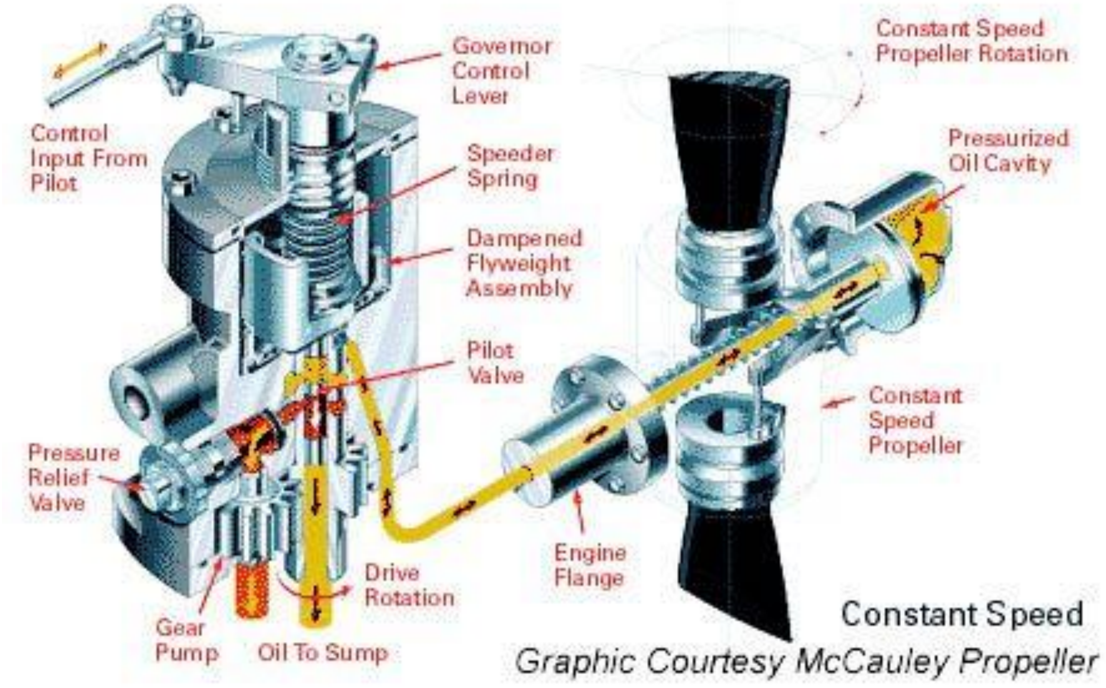
Engine RPM $\uparrow$ $\rightarrow$ $V_0 \uparrow$ $\rightarrow$ Propeller blade AoA $\downarrow$ $\rightarrow$ $D \downarrow$ $\rightarrow$ Lower Q is being generated and this increases the engine RPM.

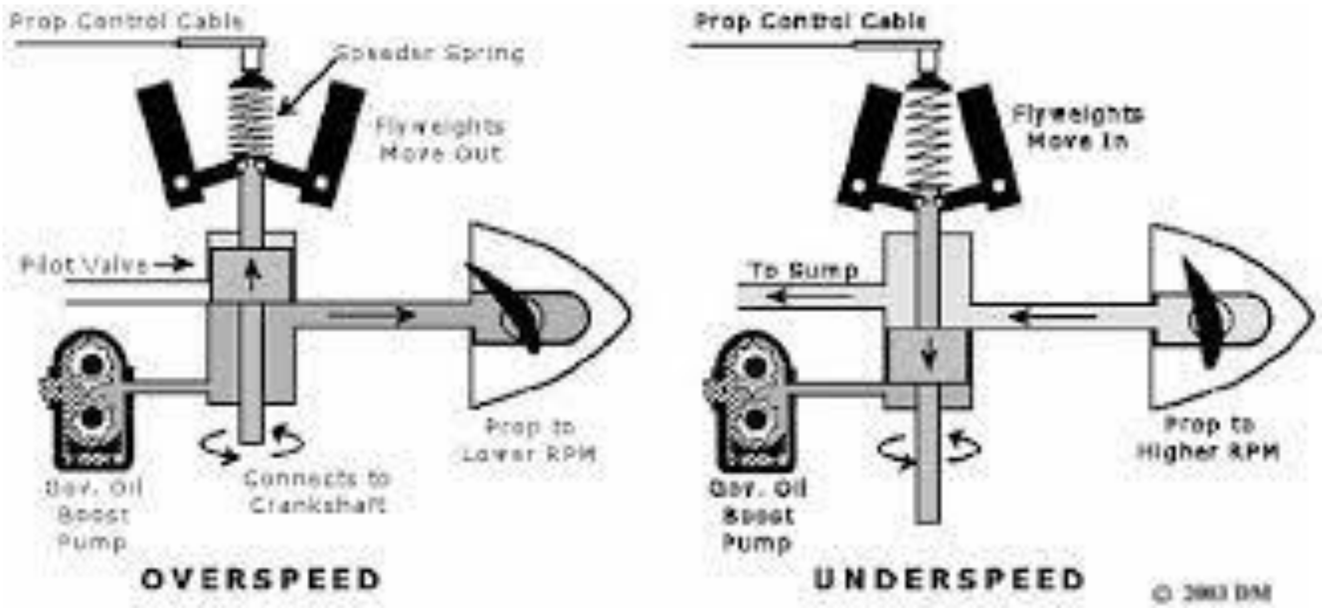
If the pilot does not change the power setting, RPM will increase as well, and the propeller will now be operating at a different η_p

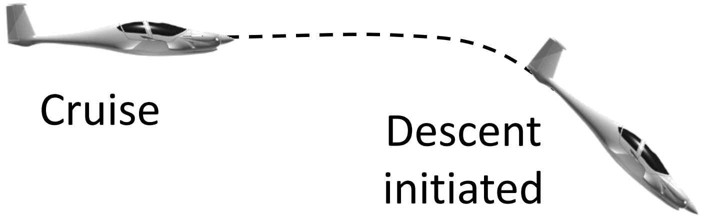
PROPELLERS

Fixed and **Constant-Speed** Propellers.

A **constant-speed propeller** is one in which tends to maintain the RPM and therefore its peak efficiency at different flight conditions.







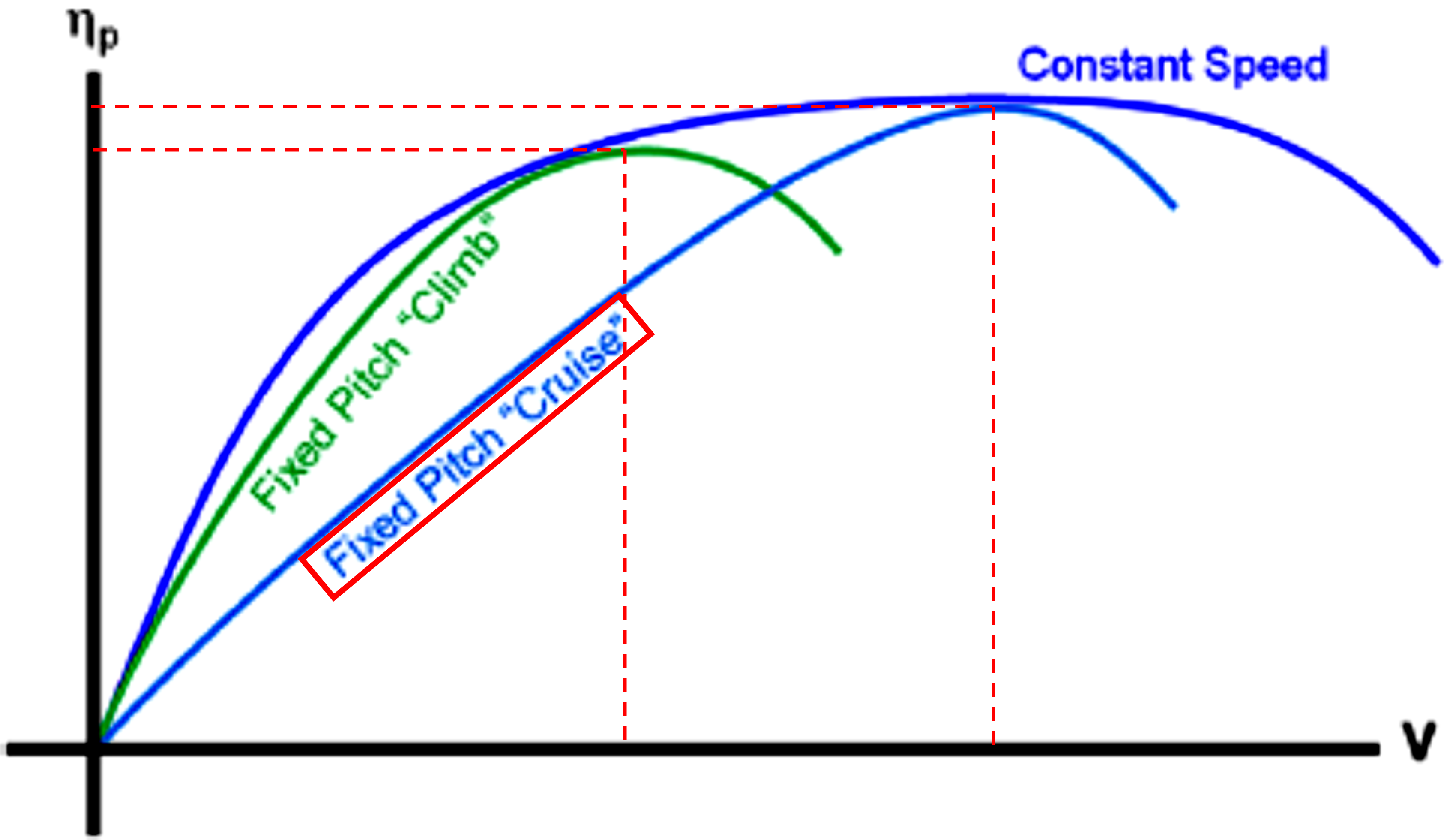
Cruise

Descent initiated

Engine RPM $\uparrow \longrightarrow V_0 \uparrow \longrightarrow$ Governor $\longrightarrow \uparrow$ Propeller pitch $\longrightarrow \downarrow$ RPM

PROPELLERS

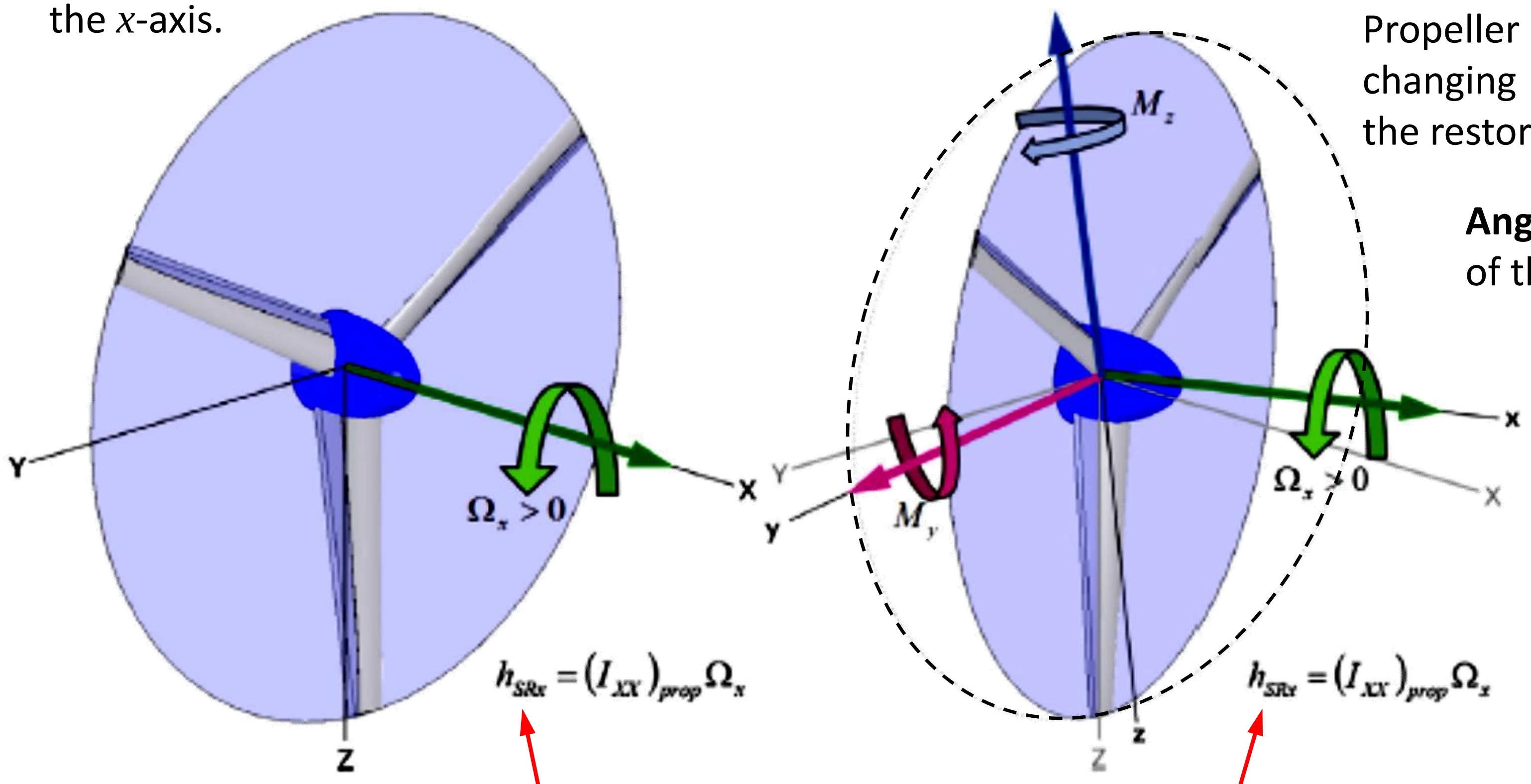
Fixed and **Constant-Speed** Propellers.



PROPELLERS

Propeller effects – Angular Momentum/Gyroscopic Effects.

Rotates only along the x-axis.



Magnitude of the angular momentum.

where

$(I_{XX})_{prop}$ = moment of inertia of the propeller about its axis of rotation

Ω_x = angular velocity of the propeller; the subscript x denotes rotation about the x-axis

Propeller is in the process of changing orientation, which induces the restoring moments M_y and M_z. → The change in orientation is called ***gyroscopic precession.***

Angular velocity of the precession. → $\vec{\omega}_P = p\mathbf{i} + q\mathbf{j} + r\mathbf{k}$

Gyroscopic moments:

$$\mathbf{M}_P = \begin{bmatrix} M_x \\ M_y \\ M_z \end{bmatrix} = \vec{\omega}_P \times \mathbf{h}_{SR} = \begin{bmatrix} i & j & k \\ p & q & r \\ h_{SRx} & h_{SRy} & h_{SRz} \end{bmatrix}$$

The components p, q, and r can be interpreted as the rotation rates of an airplane about its x-, y-, and z-axes, respectively.

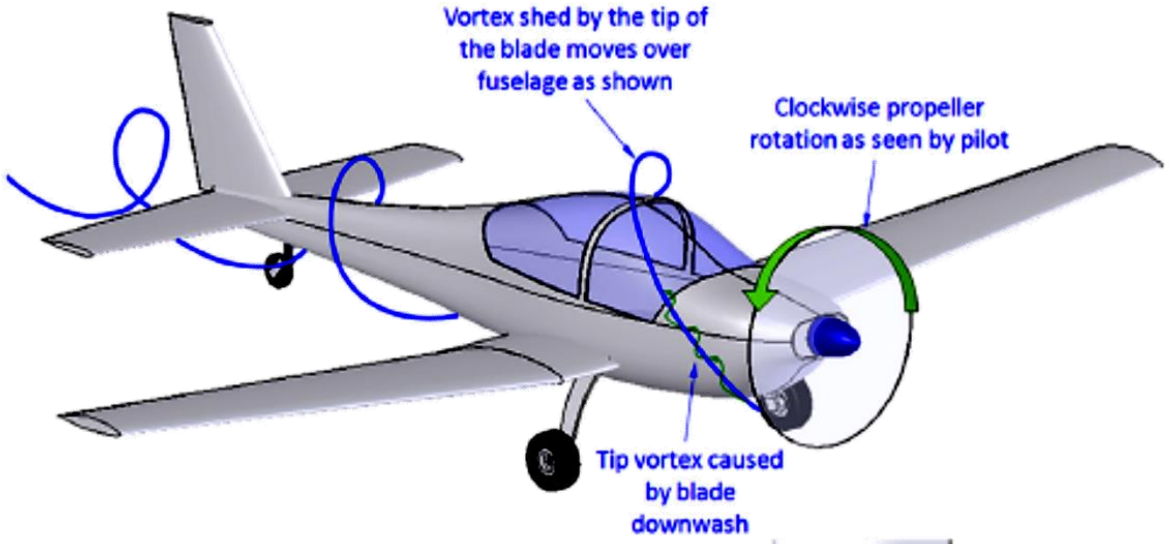
$$\begin{bmatrix} M_x \\ M_y \\ M_z \end{bmatrix} = \begin{bmatrix} \mathbf{i} & \mathbf{j} & \mathbf{k} \\ p & q & r \\ h_{SRx} & 0 & 0 \end{bmatrix}$$

$$= (0)\mathbf{i} - (0 - rh_{SRx})\mathbf{j} + (0 - qh_{SRx})\mathbf{k}$$

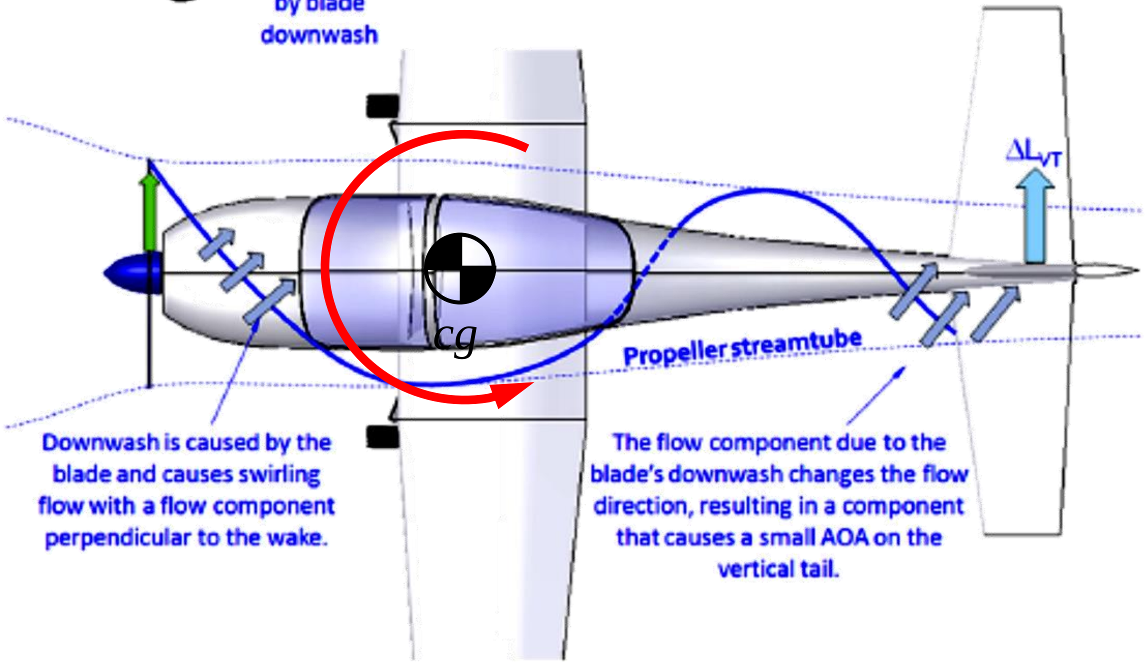
$$= rh_{SRx}\mathbf{j} - qh_{SRx}\mathbf{k}$$

PROPELLERS

Propeller effects – Slipstream Effects.



The swirling motion of air inside the propeller stream-tube will induce a slight change in the AoA of the vertical tail; for a clockwise-turning propeller, this will introduce a left yaw tendency.



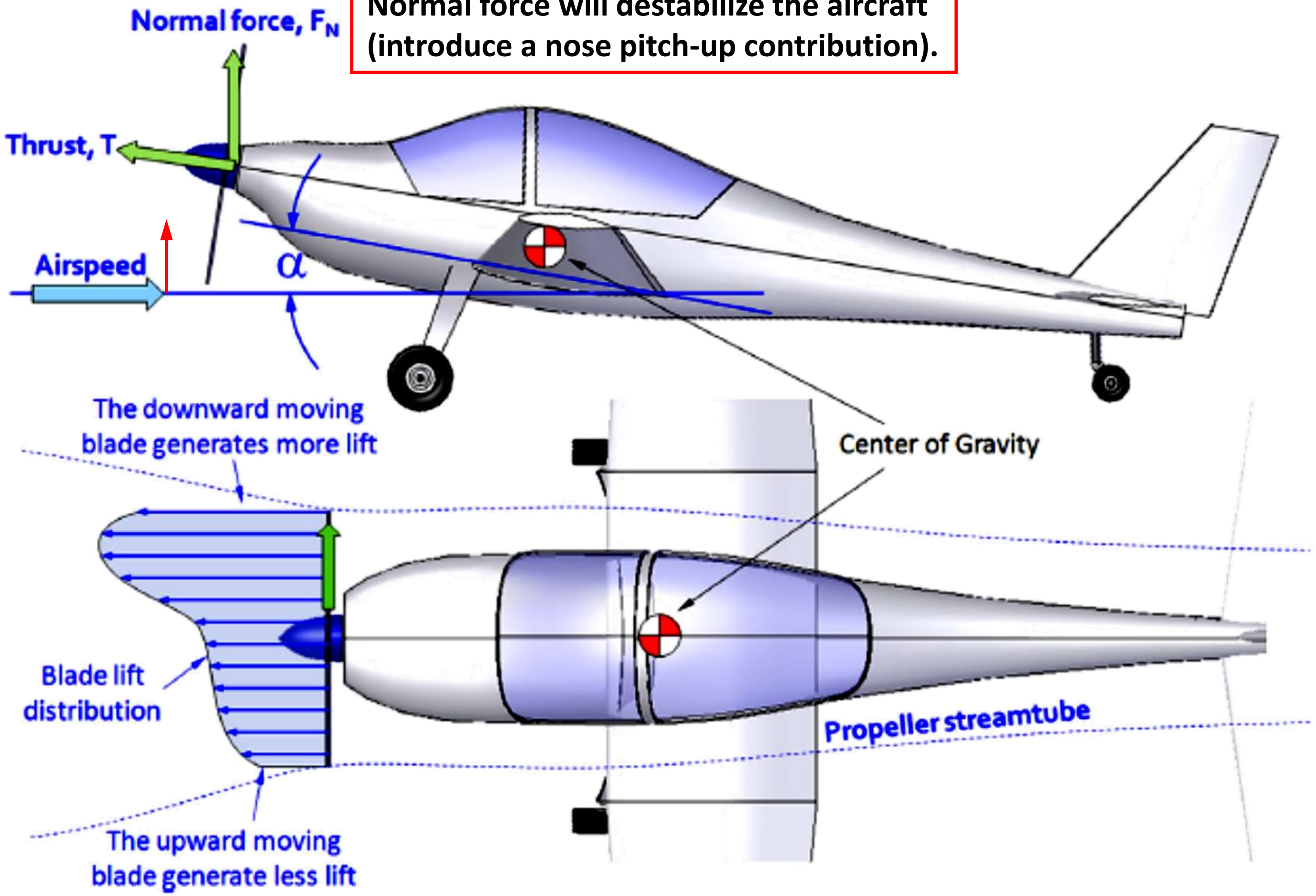
It can be suppressed by a:

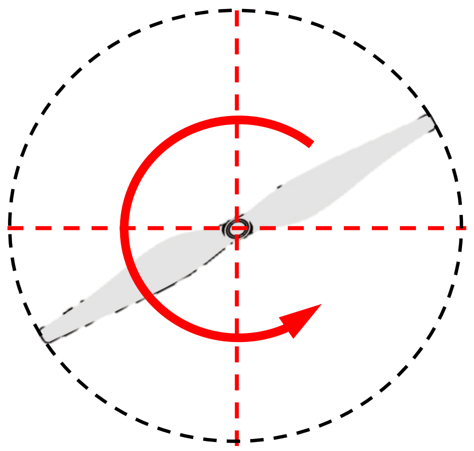
- Rudder deflected trailing edge right (step on the right rudder).
- Cambered airfoil (camber on the left side).
- Small angle-of-incidence adjustment (leading edge left).

PROPELLERS

Propeller effects – **Normal** and Side Force.

Normal force will destabilize the aircraft (introduce a nose pitch-up contribution).

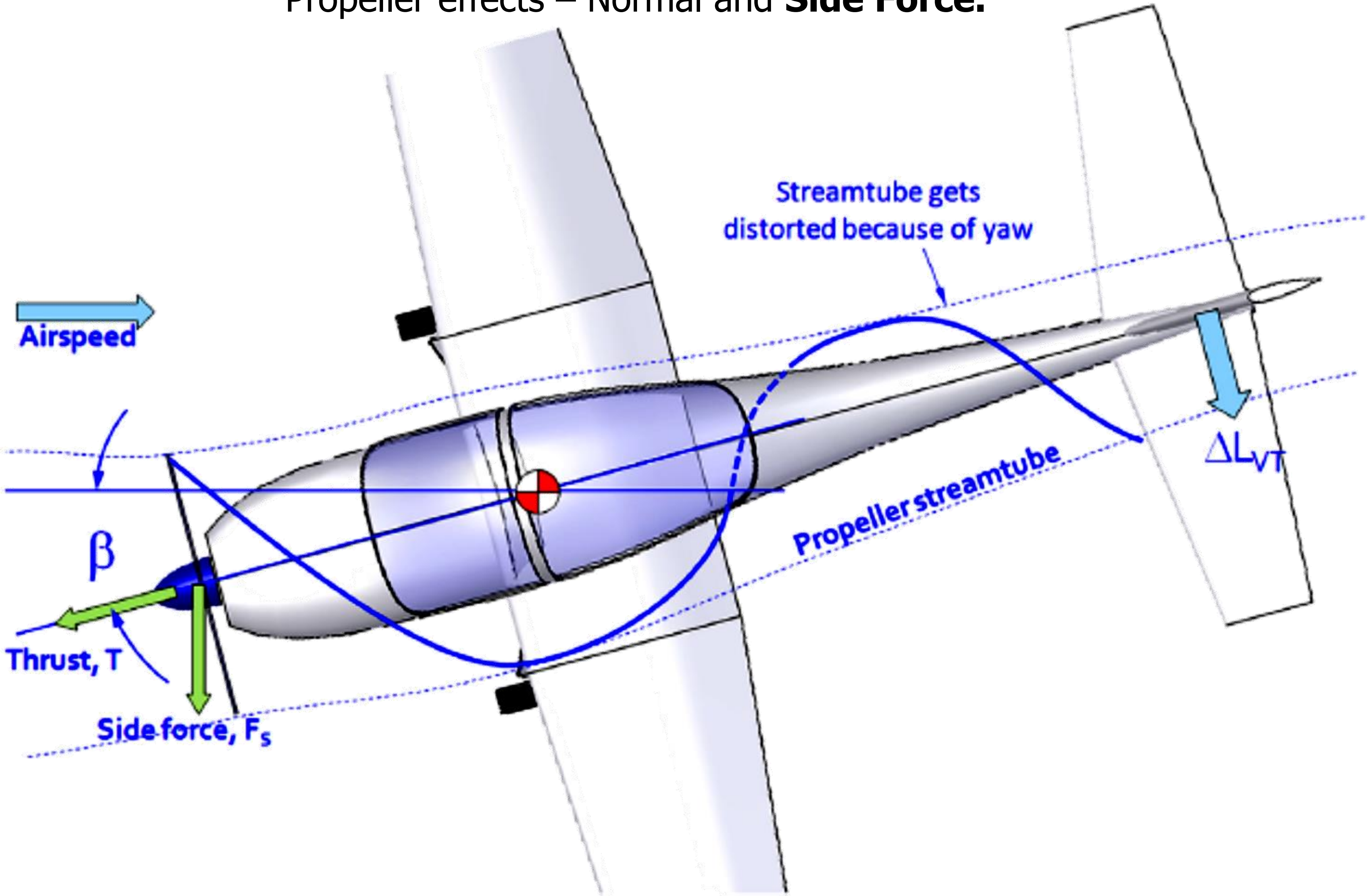




Normal component airspeed (i.e., gust).

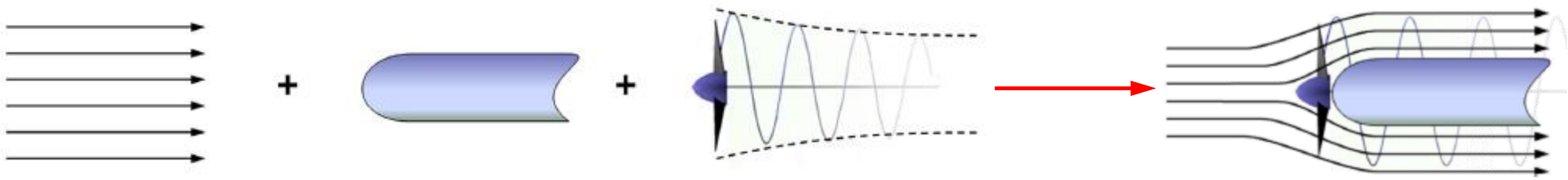
PROPELLERS

Propeller effects – Normal and **Side Force.**



PROPELLERS

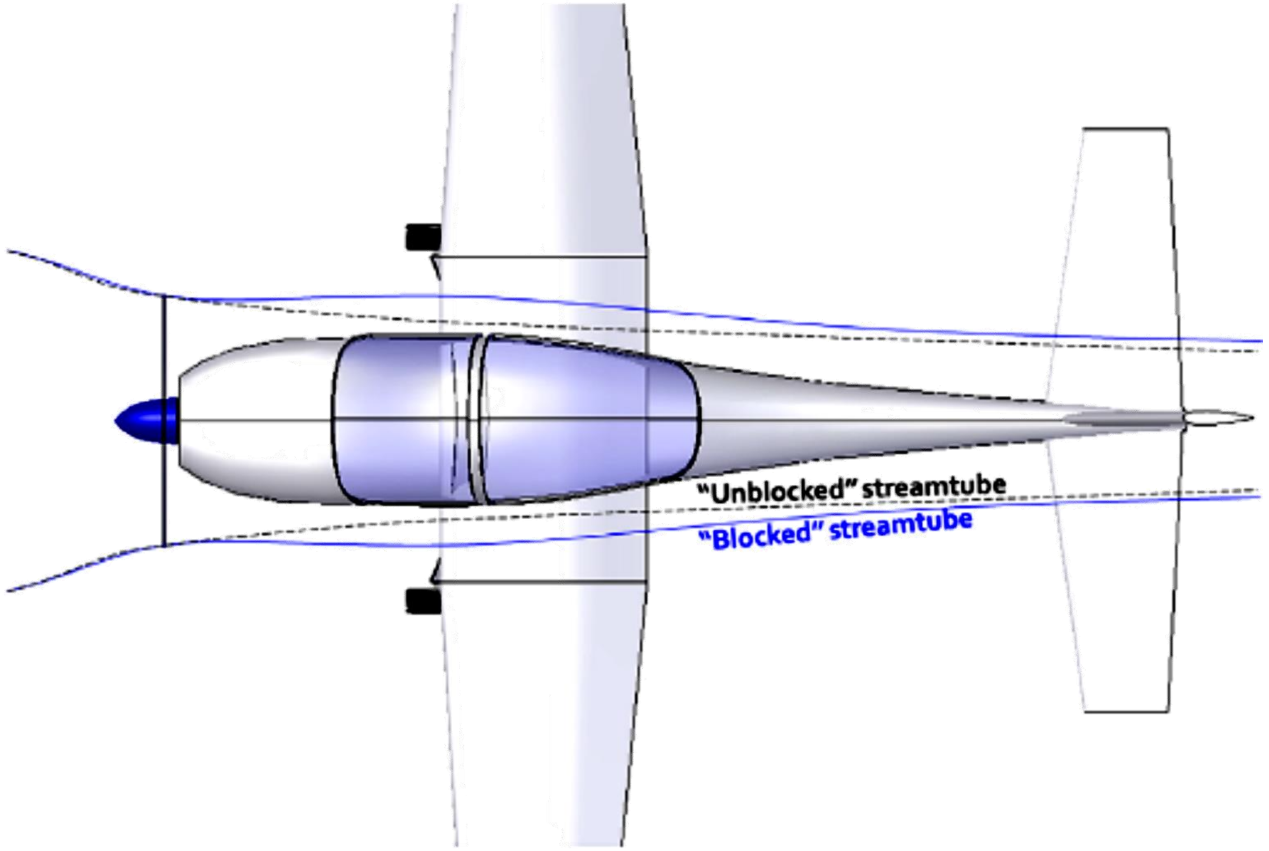
Propeller effects – Blockage Effects.



Placing an obstruction such as a fuselage (or a nacelle) into the stream-tube will distort it and prevent the formation of a proper vena contracta. This will reduce the acceleration of the flow in the stream-tube and, thus, the resulting thrust.

Aircraft designers can help mitigate blockage problems by:

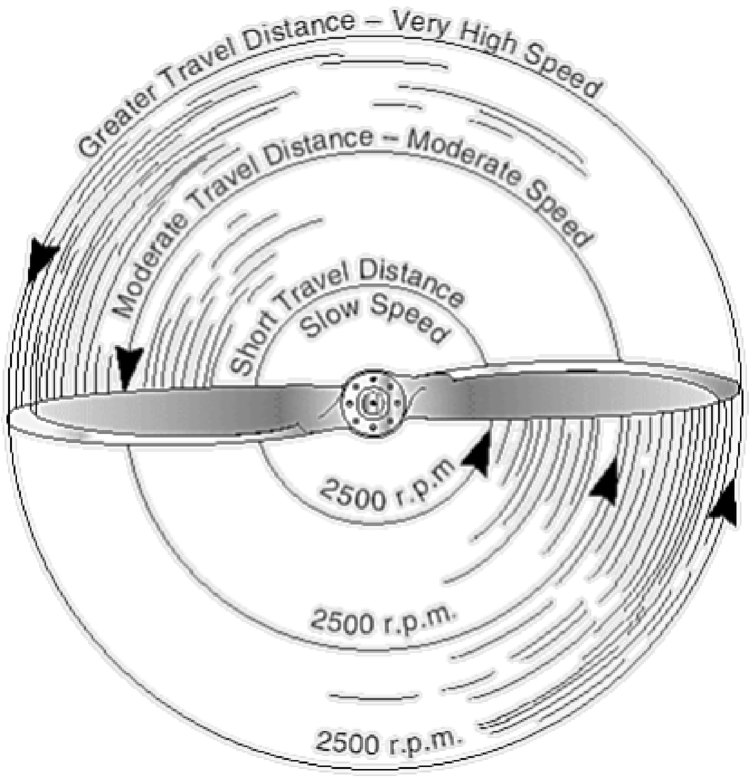
- **Considering a hub extension for piston-engine aircraft and select the largest spinner possible to cover the inboard and least efficient portion of the blade.**
- **The previous allows the cowling or nacelle to be sculpted to provide less obstruction to the flow.**
- **Cowling or nacelle should be designed to be as axisymmetric as possible.**



PROPELLERS

Tips for Selecting a Suitable Propeller

- The tip (which refers to rotational and not helical) speed should be as high as possible, but there is an upper limit. Too high a tip speed increases noise and reduces efficiency (via shock formation).
  - Rotational tip speeds for **<u>wooden</u>** propellers are in the $M_\infty \approx 0.6$ range.
  - Rotational tip speeds for **<u>metal</u>** propellers are in the $M_\infty \approx 0.75–0.8$ range.
  - Rotational tip speeds for **<u>advanced composite</u>** propellers are in the $M_\infty \approx 0.75–0.8$ range.
- If the engine RPM is too high, a gear-reduction drive may be required to slow it down.
- High rotational speed requires fewer blades or a smaller blade area to generate thrust.
- High-powered engines whose rotational rate limits the propeller radius need more blades to help convert engine power into thrust.
- A diameter that is too large may result in high tip speeds and noise, but it can also present some ground proximity problems.
- Weight – a three-bladed prop can be of a smaller diameter, but three blades weigh more than two blades.



ENGINE SELECTION

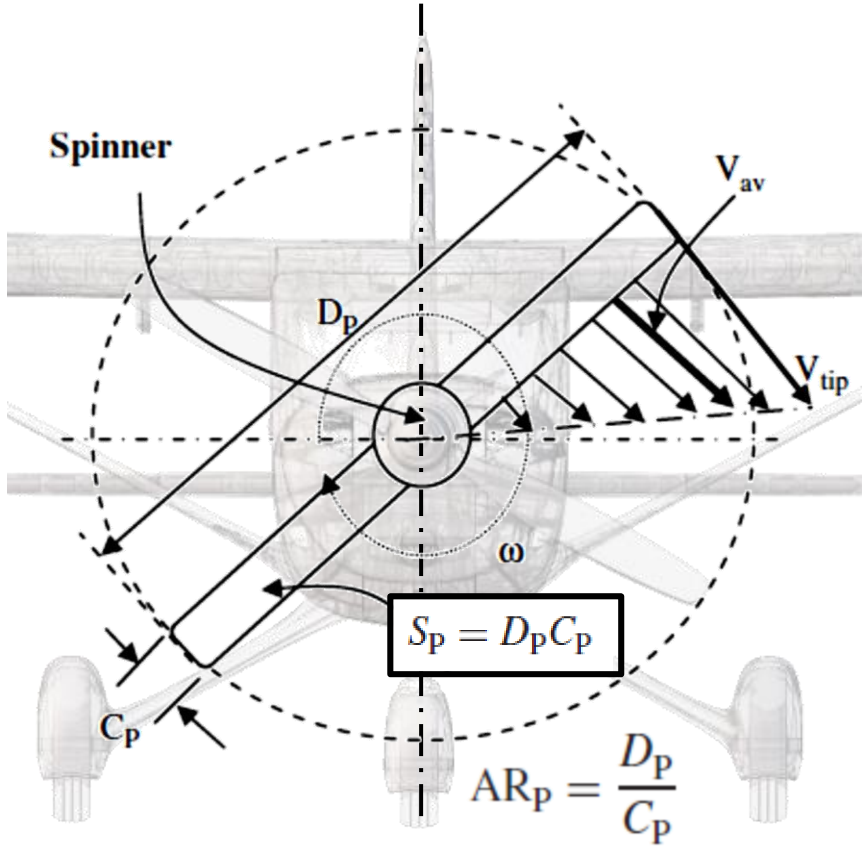
Propeller sizing. propeller efficiency (η_P) → 75–85%

$$T = \frac{P \cdot \eta_P}{V_C}$$

$$L_P = \frac{1}{2}\rho V_{av}^2 S_P C_{L_P}$$

$$L_P = T \Rightarrow \frac{1}{2}\rho V_{av}^2 S_P C_{L_P} = \frac{P\eta_P}{V_C}$$

$$V_{av} = 0.7 V_{tip_{cruise}}$$



The required propeller planform area to generate such a lift (i.e., thrust), when the engine power P is supplied.

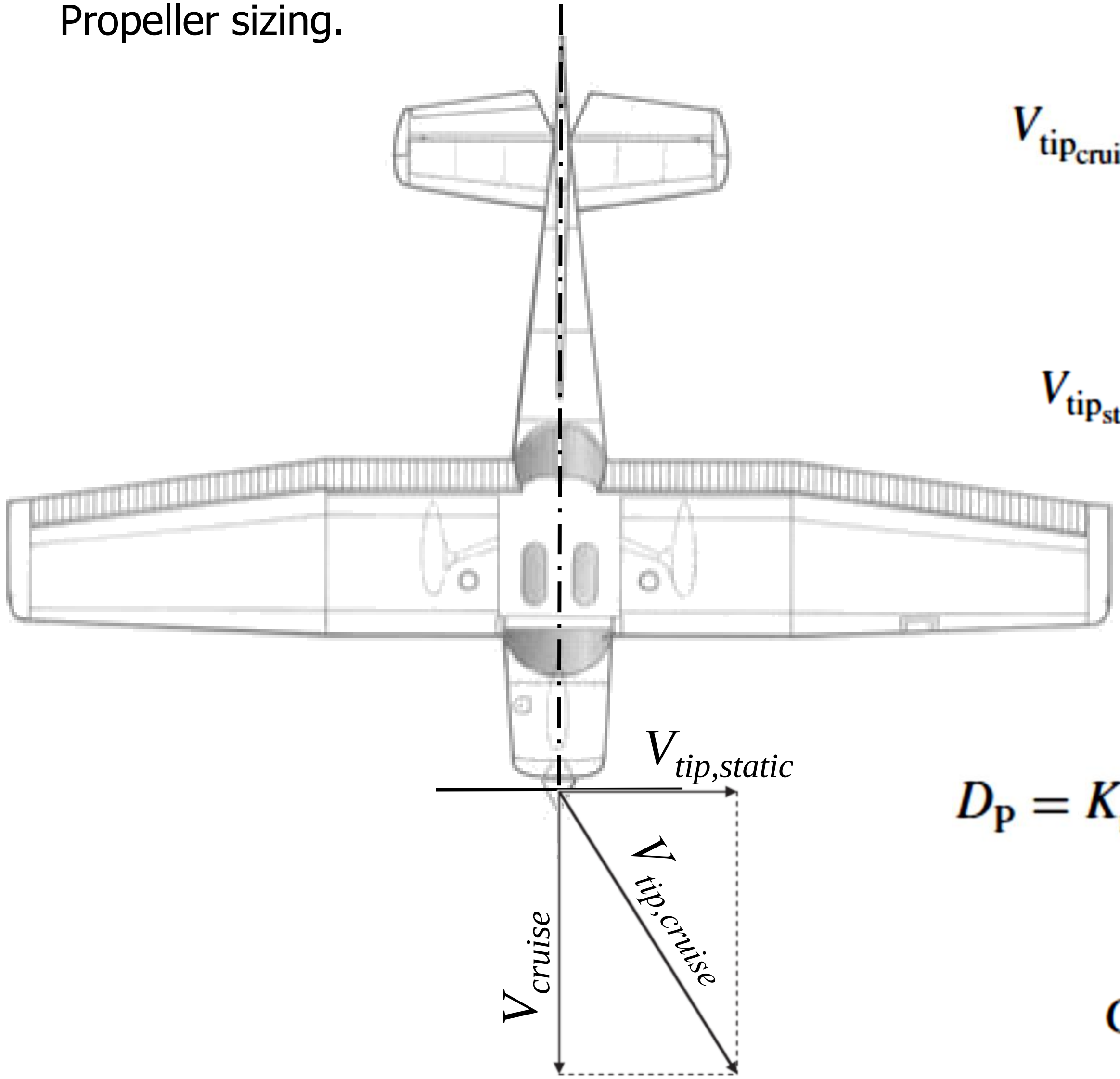
$$S_P = \frac{2P\eta_P}{\rho V_{av}^2 C_{L_P} V_C}$$

Propeller diameter $$D_P = \sqrt{\frac{2P\eta_P AR_P}{\rho V_{av}^2 C_{L_P} V_C}}$$

Typical values:
AR_P = 7 and 15
$C_{L,P}$ = 0.2 and 0.4

ENGINE SELECTION

Propeller sizing.



$$V_{\mathrm{tip}_{\mathrm{cruise}}} = \sqrt{V_{\mathrm{tip}_{\mathrm{static}}}^2 + V_{\mathrm{C}}^2}$$

$$V_{\mathrm{tip}_{\mathrm{static}}} = \frac{D_{\mathrm{P}}}{2}\omega$$

$$\omega = \frac{2\pi \cdot n}{60}(\mathrm{rad/s})$$

Table 8.9 Suggested propeller cruise tip speed limit

No.	Tip speed limit (m/s)	Propeller type
1	310	Metal high-performance prop
2	270	Metal regular prop
3	250	Composite prop
4	210	Wooden prop
5	150	Plastic prop for RC model aircraft

$$D_{\mathrm{P}} = K_{\mathrm{np}}\sqrt{\frac{2P\eta_{\mathrm{P}}\mathrm{AR}_{\mathrm{P}}}{\rho V_{\mathrm{av}}^2 C_{L_{\mathrm{P}}} V_{\mathrm{C}}}}$$

K_{np} – correction factor, is 1 for a two-blade propeller, and 0.72 for a six-bladed and beyond. For any other number of blades, you can use linear interpolation to find the appropriate correction factor.

$$\mathrm{GR} = \frac{n_{\mathrm{P}}}{n_{\mathrm{S}}}$$

PROPELLERS

Rapid Estimation of Required Propeller Diameter (***Gudmundsson***).

Two-bladed wooden propeller:

$$D = 10000 \cdot \sqrt[4]{\frac{P_{BHP}}{53.5 \times RPM^2 \times V_{TAS}}} \text{ (in inches)}$$

Three-bladed wooden propeller:

$$D = 10000 \cdot \sqrt[4]{\frac{P_{BHP}}{75.8 \times RPM^2 \times V_{TAS}}} \text{ (in inches)}$$

Four-bladed wooden propeller:

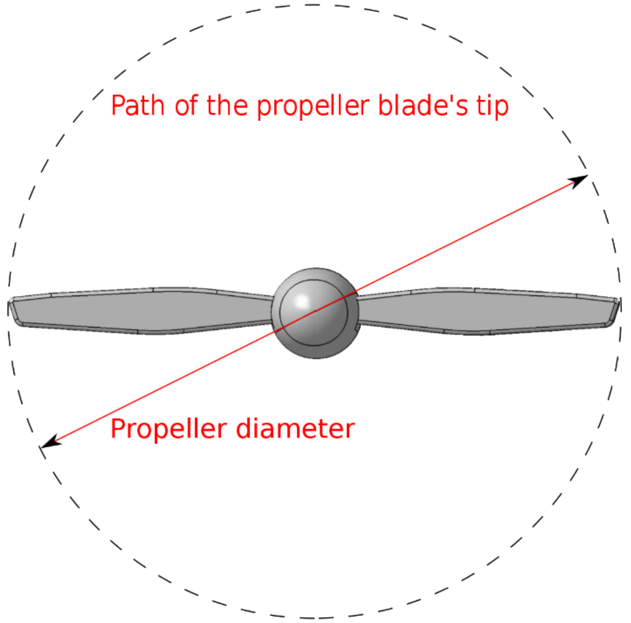
$$D = 10000 \cdot \sqrt[4]{\frac{P_{BHP}}{111 \times RPM^2 \times V_{TAS}}} \text{ (in inches)}$$

Two-bladed metal propellers:

$$D = 22 \sqrt[4]{P_{BHP}} \text{ (in inches)}$$

Three-bladed metal propellers:

$$D = 18 \sqrt[4]{P_{BHP}} \text{ (in inches)}$$



Propeller diameter:

$$D = K_p \sqrt[4]{P_{BHP}}$$

where the factor K_p is obtained from

Type of Propeller	K_p for P in BHP and D in Inches	K_p for P in kW and D in m
Two-bladed	20.4	0.56
Three-bladed	19.2	0.52
Four or more blades	18.0	0.49

PROPELLERS

Example – Rapid Estimation of Required Propeller Diameter (D).

The Aerotrek A240 Light-Sport Aircraft is equipped with a 62.2 inch diameter three-bladed composite propeller. When powered by an 80 BHP Rotax 912UL engine it is capable of flying at 104 KTAS at 75% power. Calculate the recommended diameter of two-, three-, and four-bladed wooden propellers for this airplane, intended to rotate at 2300 RPM, and compare to the actual propeller diameter. What are the corresponding tip speeds?

PROPELLERS

Example – Rapid Estimation of Required Propeller Diameter (D).

Solution

Two-bladed propeller:

$$D = 10000 \cdot \sqrt[4]{\frac{P_{BHP}}{53.5 \times RPM^2 \times V_{TAS}}}$$

$$= 10000 \cdot \sqrt[4]{\frac{0.75 \times 80}{53.5 \times 2300^2 \times 104}} = \boxed{67.2 \text{ inches}}$$

Rotation speed in terms of radians per second:

$$\Omega = 2\pi \left(\frac{RPM}{60}\right) = 2\pi \left(\frac{2300}{60}\right) = 240.9 \quad \text{rad/s}$$

Tip speed:

$$V_{tip} = \Omega \times R = 240.9 \times \frac{67.2/12}{2} = \boxed{675 \text{ ft/s}}$$

Three-bladed propeller:

$$D = 10000 \cdot \sqrt[4]{\frac{P_{BHP}}{75.8 \times RPM^2 \times V_{TAS}}}$$

$$= 10000 \cdot \sqrt[4]{\frac{0.75 \times 80}{75.8 \times 2300^2 \times 104}} = \boxed{61.6 \text{ inches}}$$

Tip speed:

$$V_{tip} = \Omega \times R = 240.9 \times \frac{61.6/12}{2} = \boxed{618 \text{ ft/s}}$$

Four-bladed propeller:

$$D = 10000 \cdot \sqrt[4]{\frac{P_{BHP}}{111 \times RPM^2 \times V_{TAS}}}$$

$$= 10000 \cdot \sqrt[4]{\frac{0.75 \times 80}{111 \times 2300^2 \times 104}} = \boxed{56.0 \text{ inches}}$$

Tip speed:

$$V_{tip} = \Omega \times R = 240.9 \times \frac{56.0/12}{2} = \boxed{562 \text{ ft/s}}$$

PROPELLERS

Example – Rapid Estimation of Required Propeller Diameter (D).

Universidad Pontificia Bolivariana
Fundada en 1936

Estimate the required propeller diameter for a light airplane designed to operate around 120 KTAS, assuming:

(a) A two-bladed wooden prop, if the maximum expected engine power is 80 BHP and max engine RPM is 5800 RPM with a 2.27:1 gear reduction ratio.
(b) A three-bladed metal prop.
(c) Assume a max tip speed of $M = 0.75$ at S-L.

PROPELLERS

Example – Rapid Estimation of Required Propeller Diameter (D).

Solution

(a) First determine RPM delivered to propeller:

$$RPM = \frac{RPM_{engine}}{GR} = \frac{5800}{2.27} = 2555 \text{ RPM}$$

Then, calculate the diameter of a two-bladed wooden propeller:

$$D = 10000 \cdot \sqrt[4]{\frac{P_{BHP}}{53.5 \times RPM^2 \times V_{TAS}}}$$

$$= 10000 \cdot \sqrt[4]{\frac{80}{53.5 \times 2555^2 \times 120}} = \boxed{66 \text{ inches}}$$

(b) A three-bladed metal propeller:

$$D = 18 \sqrt[4]{P_{BHP}} = 18 \sqrt[4]{80} = \boxed{54 \text{ inches}}$$

(c) Maintaining max tip speed of $M = 0.75$:

$$M_{tip} = \frac{V_{tip}}{1116\sqrt{1 - 0.0000068753H}} < 0.75$$

$$\Rightarrow \frac{V_{tip}}{1116} < 0.75$$

$$\Rightarrow V_{tip} < 0.75 \times 1116 = \boxed{837 \text{ ft/s}}$$

PROPELLERS

Example – Rapid Estimation of Required Propeller Diameter (D).

Solution

Therefore, the maximum diameter can be determined as follows:

$$V_{tip} = \sqrt{V^2 + \left(\frac{\pi \cdot RPM \cdot D}{60}\right)^2} < 837 \text{ ft/s}$$

$$\Rightarrow \quad D = \frac{60\sqrt{V_{tip}^2 - V^2}}{\pi \cdot RPM}$$

$$= \frac{60\sqrt{(837)^2 - (120 \times 1.688)^2}}{\pi \cdot 2555}$$

$$= \boxed{6.07 \text{ ft (73 inches)}}$$

PROPELLERS

Universidad Pontificia Bolivariana

Fundada en 1936

Estimation of the Required Propeller Pitch (P_D).

The tables are intended to give the designer an idea of the propeller pitch required. $P_D \approx 1251\left(\frac{V_{KTAS}}{RPM}\right)\left(\frac{1}{\eta_P}\right)$

RPM	INTENDED CRUISING AIRSPEED in KTAS ($\eta_P = 0.90$)														
	50	**60**	**70**	**80**	**90**	**100**	**110**	**120**	**130**	**140**	**150**	**160**	**170**	**180**	**190**
2000	35	42	49	56	63	70	76	83	90	97	104	111	118	125	119
2100	33	40	46	53	60	66	73	79	86	93	99	106	113	119	113
2200	32	38	44	51	57	63	70	76	82	88	95	101	107	114	108
2300	30	36	42	48	54	60	66	73	79	85	91	97	103	109	103
2400	29	35	41	46	52	58	64	70	75	81	87	93	98	104	99
2500	28	33	39	44	50	56	61	67	72	78	83	89	95	100	95
2600	27	32	37	43	48	53	59	64	70	75	80	86	91	96	91
2700	26	31	36	41	46	51	57	**62**	67	72	77	82	88	93	88
2800	25	30	35	40	45	50	55	60	65	70	74	79	84	89	85
2900	24	29	34	38	43	48	53	58	62	67	72	77	82	86	82
3000	23	28	32	37	42	46	51	56	60	65	70	74	79	83	79
3100	22	27	31	36	40	45	49	54	58	63	67	72	76	81	77
3200	22	26	30	35	39	43	48	52	56	61	65	70	74	78	74
3300	21	25	29	34	38	42	46	51	55	59	63	67	72	76	72
3400	20	25	29	33	37	41	45	49	53	57	61	65	70	74	70
3500	20	24	28	32	36	40	44	48	52	56	60	64	68	72	68

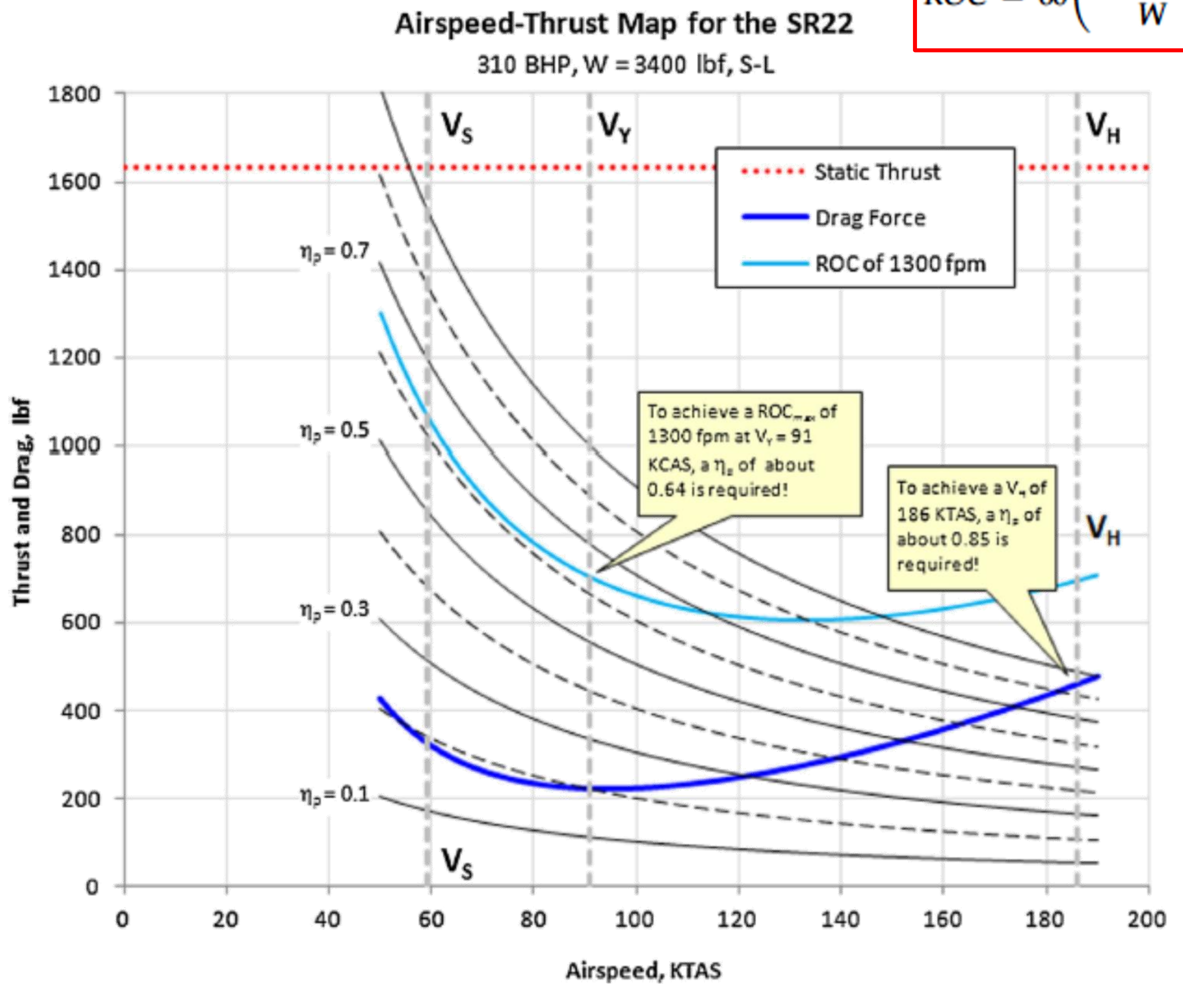
RPM	INTENDED CRUISING AIRSPEED in KTAS ($\eta_P = 0.80$)														
	50	**60**	**70**	**80**	**90**	**100**	**110**	**120**	**130**	**140**	**150**	**160**	**170**	**180**	**190**
2000	39	47	55	63	70	78	86	94	102	109	117	125	133	141	149
2100	37	45	52	60	67	74	82	89	97	104	112	119	127	134	142
2200	36	43	50	57	64	71	78	85	92	100	107	114	121	128	135
2300	34	41	48	54	61	68	75	82	88	95	102	109	116	122	129
2400	33	39	46	52	59	65	72	78	85	91	98	104	111	117	124
2500	31	38	44	50	56	63	69	75	81	88	94	100	106	113	119
2600	30	36	42	48	54	60	66	72	78	84	90	96	102	108	114
2700	29	35	41	46	52	58	64	**70**	75	81	87	93	98	104	110
2800	28	34	39	45	50	56	61	67	73	78	84	89	95	101	106
2900	27	32	38	43	49	54	59	65	70	76	81	86	92	97	102
3000	26	31	36	42	47	52	57	63	68	73	78	83	89	94	99
3100	25	30	35	40	45	50	56	61	66	71	76	81	86	91	96
3200	24	29	34	39	44	49	54	59	64	68	73	78	83	88	93
3300	24	28	33	38	43	47	52	57	62	66	71	76	81	85	90
3400	23	28	32	37	41	46	51	55	60	64	69	74	78	83	87
3500	22	27	31	36	40	45	49	54	58	63	67	72	76	80	85

PROPELLERS

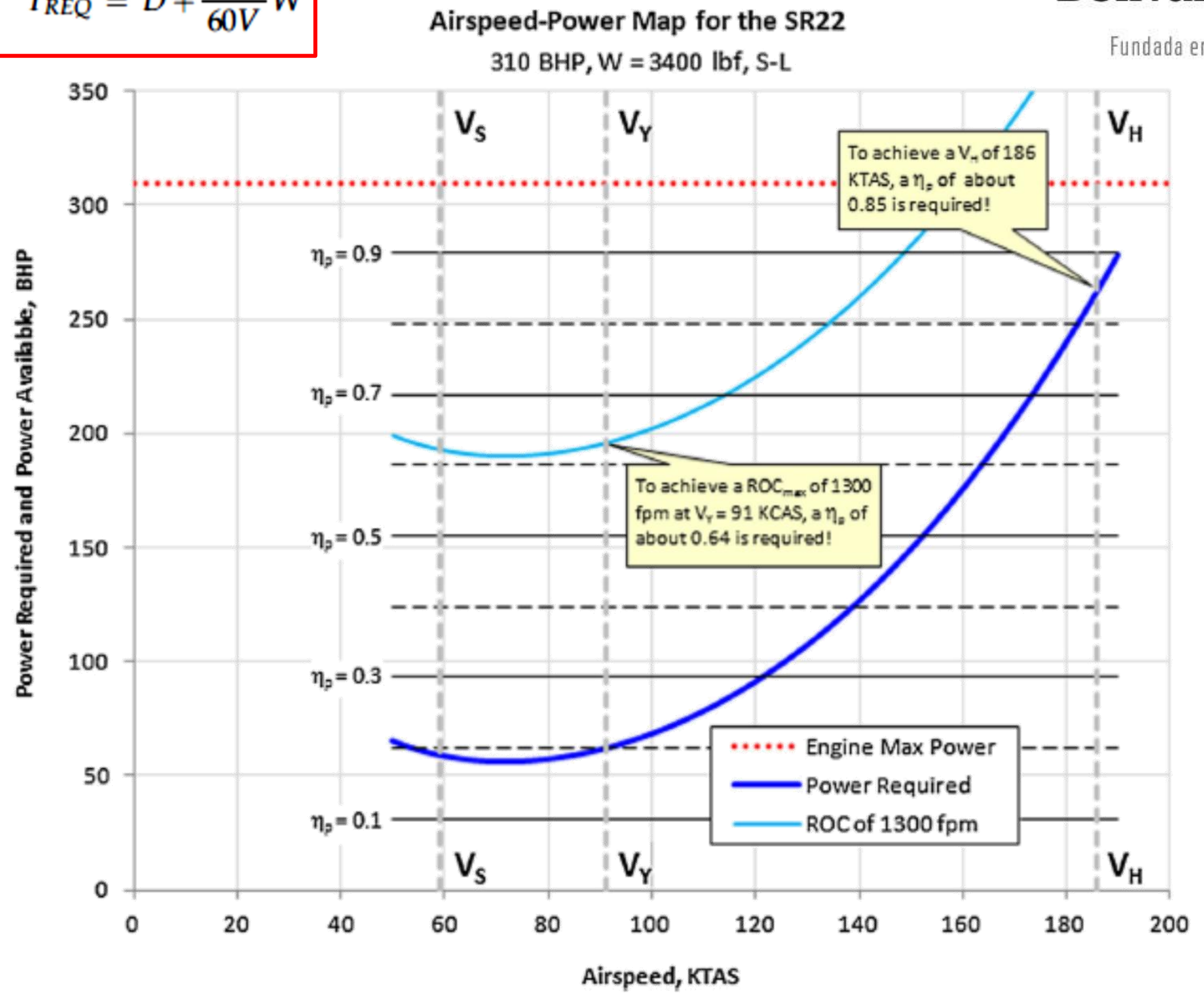
Cirrus Aircraft SR22

Estimation of the Required Propeller Efficiency (η_p).

$$ROC = 60\left(\frac{TV - DV}{W}\right) \Rightarrow T_{REQ} = D + \frac{ROC}{60V}W$$



An airspeed-thrust map shows the propeller efficiency required to meet given performance characteristics in terms of thrust.



An airspeed-thrust map shows the propeller efficiency required to meet given performance characteristics in terms of power.

PROPELLERS

Estimation of the Required Propeller Advance Ratio (J) and Activity Factor (AF).

Universidad Pontificia Bolivariana

Fundada en 1936

J – Is a measure of how far the propeller travels in unit time in terms of its diameter.

For instance, if an advance ratio of 0.5 is measured for a $10\ ft$ diameter propeller rotating at $600\ RPM$, it means the propeller (i.e., the vehicle it is propelling) is moving at a forward speed of $50\ ft/s$.

$$J = \frac{V_0}{nD} = \frac{60 \cdot V_0}{RPM \cdot D}$$

AF – represents the power absorption capability of all blade elements.

Small aircraft have an AF ranging from 80-105, while larger aircraft have an AF of around 130-170.

$$AF = \frac{100\ 000}{16} \int_{0.15}^{1.0} \left(\frac{c}{D}\right)\left(\frac{r}{R}\right)^3 d\left(\frac{r}{R}\right)$$

PROPELLERS

Definition of Power/Torque and Thrust Coefficients.

The following concepts are definitions:

Power coefficient:

$$C_P = \frac{P}{\rho n^3 D^5} = \frac{550 \times P_{BHP}}{\rho \left(\frac{RPM}{60}\right)^3 D^5} = \frac{118\ 800\ 000 \times P_{BHP}}{\rho \cdot RPM^3 \cdot D^5}$$

Thrust coefficient:

$$C_T = \frac{T}{\rho n^2 D^4} = \frac{3600 \cdot T}{\rho \cdot RPM^2 D^4}$$

Torque coefficient:

$$C_Q = \frac{Q}{\rho n^2 D^5} = \frac{3600 \cdot Q}{\rho \cdot RPM^2 \cdot D^5} = \frac{C_P}{2\pi}$$

Power-Torque relation:

$$C_Q = \frac{Q}{\rho n^2 D^5} = \frac{C_P}{2\pi} = \frac{P/\rho n^3 D^5}{2\pi} \quad \Rightarrow \quad P = 2\pi n Q$$

PROPELLERS

Example – Estimation of the Required Propeller.

The SR22 is equipped with a 76 inch diameter propeller. If the airplane is observed to fly at 185 KTAS at 8000 ft with the propeller rotating at 2700 RPM, what are the helix angle, blade rotational and helical tip speeds, Mach number, and advance ratio?

Solution

Tip radius:

$$R = \left(\frac{76/2}{12}\right) = \boxed{3.167 \;\; \text{ft}}$$

Angular velocity:

$$\Omega = 2\pi\left(\frac{RPM}{60}\right) = 2\pi\left(\frac{2700}{60}\right) = \boxed{282.7 \;\; \text{rad/s}}$$

Helix angle:

$$\tan\phi = \frac{\pi R \;\; RPM}{30 \cdot V} = \frac{\pi(3.167) \;\; (2700)}{30 \cdot (185 \times 1.688)} = 2.867$$

$$\Rightarrow \quad \phi = \tan^{-1}(2.867) = \boxed{70.8^\circ}$$

PROPELLERS

Example – Estimation of the Required Propeller.

The SR22 is equipped with a 76 inch diameter propeller. If the airplane is observed to fly at 185 KTAS at 8000 ft with the propeller rotating at 2700 RPM, what are the helix angle, blade rotational and helical tip speeds, Mach number, and advance ratio?

Solution

Propeller rotational tip speed calculated

$$V_{ROT} = \Omega \cdot R = (2827) \cdot (3.167) = \boxed{895.3 \text{ ft/s}}$$

Propeller tip speed:

$$V_{tip} = \sqrt{V_0^2 + \left(\frac{\pi \cdot RPM \cdot D}{60}\right)^2}$$
$$= \sqrt{(185 \times 1.688)^2 + \left(\frac{\pi \cdot 2700 \cdot (76/12)}{60}\right)^2}$$
$$= \boxed{948 \text{ ft/s}}$$

PROPELLERS

Example – Estimation of the Required Propeller.

The SR22 is equipped with a 76 inch diameter propeller. If the airplane is observed to fly at 185 KTAS at 8000 ft with the propeller rotating at 2700 RPM, what are the helix angle, blade rotational and helical tip speeds, Mach number, and advance ratio?

Solution

Tip Mach number:

$$M_{tip} = \frac{948}{1116\sqrt{1 - 0.0000068753 \times 8000}} = \boxed{0.874}$$

Advance ratio:

$$J = \frac{V_0}{nD} = \frac{60 \cdot V_0}{RPM \cdot D} = \frac{60 \cdot (185 \times 1.688)}{2700 \cdot (76/12)} = \boxed{1.096}$$

PROPELLERS

Example – Estimation of the Required Propeller.

The SR22 is equipped with a 76 inch diameter propeller. If the airplane is observed to fly at 185 KTAS at 8000 ft with the propeller rotating at 2700 RPM, what are the helix angle, blade rotational and helical tip speeds, Mach number, and advance ratio?

What is the activity factor of each blade of the airplane whose diameter is 76 inches and chord is a constant 5 inches.

Solution

$$AF = \frac{100\ 000}{16}\int_{0.15}^{1.0}\left(\frac{c}{D}\right)\left(\frac{r}{R}\right)^3 d\left(\frac{r}{R}\right)$$

$$= \frac{100\ 000}{16}\left(\frac{5}{76}\right)\int_{0.15}^{1.0} x^3 dx = 411.2\left[\frac{x^4}{4}\right]_{0.15}^{1.0}$$

$$AF = 411.2\left[\frac{1-0.15^4}{4}\right] = \boxed{102.7}$$

PROPELLERS

Example – Estimation of the Required Propeller.

The SR22 is equipped with a 76 inch diameter propeller. If the airplane is observed to fly at 185 KTAS at 8000 ft with the propeller rotating at 2700 RPM, what are the helix angle, blade rotational and helical tip speeds, Mach number, and advance ratio?

Compute C_P and C_Q for the airplane, if it is known its engine is generating 225 BHP.

Solution

Power coefficient:

$$C_P = \frac{118\ 800\ 000 \times P_{BHP}}{\rho \cdot RPM^3 \cdot D^5}$$
$$= \frac{118\ 800\ 000 \times 225}{(0.002378)(2700)^3(76/12)^5} = \boxed{0.05604}$$

Torque coefficient:

$$C_Q = \frac{Q}{\rho n^2 D^5} = \frac{C_P}{2\pi} = \frac{0.05604}{2\pi} = \boxed{0.008919}$$

PROPELLERS

Propulsive Efficiency (η_p).

Universidad Pontificia Bolivariana

Fundada en 1936

The power generated by an engine is not completely converted to a thrust force. It is inevitable that some losses occur in the process due to an imperfect conversion process.

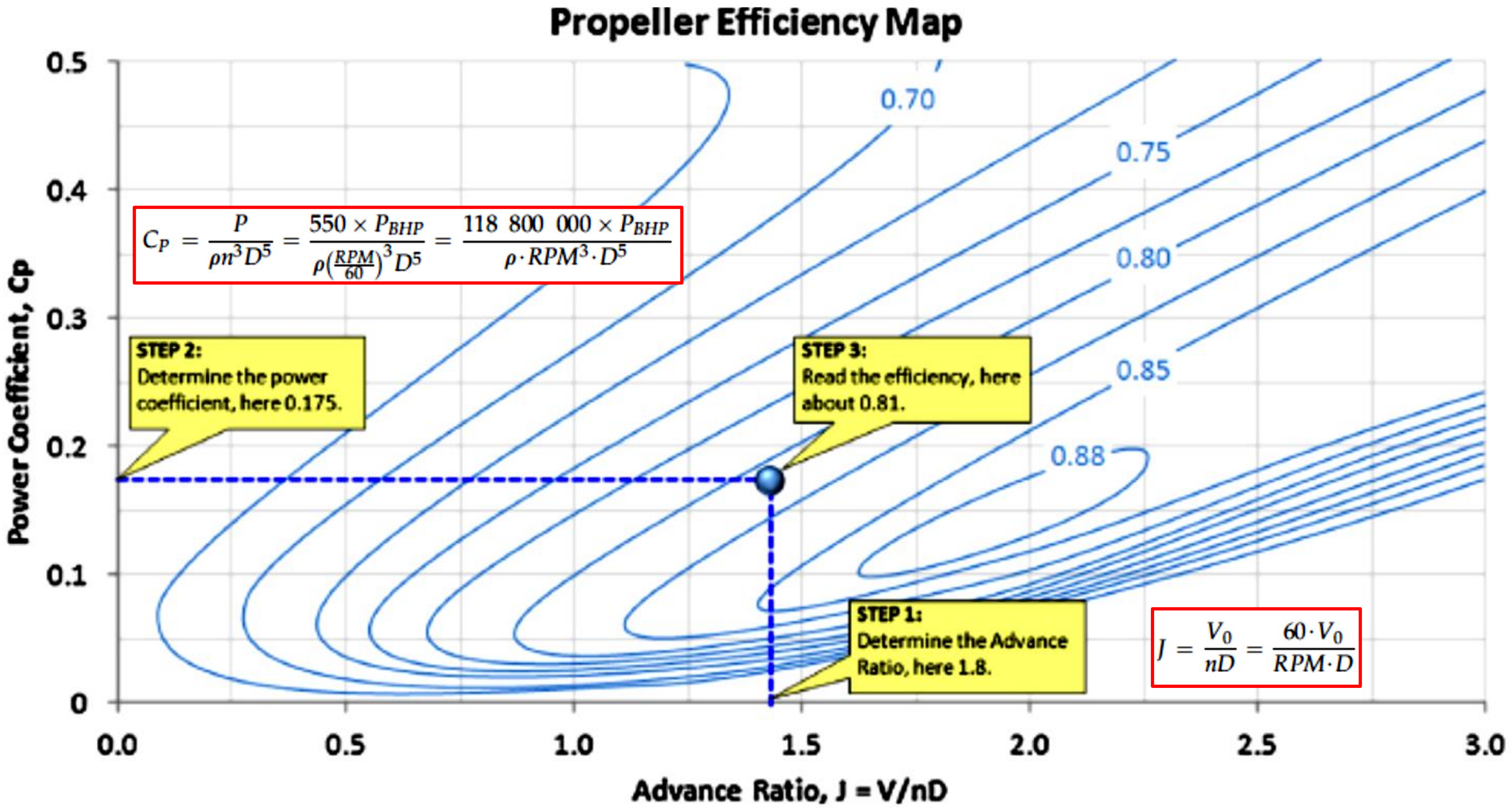
η_p – Can be defined as the fraction of engine power that gets converted into propulsive power (thrust airspeed) to the total engine output power, P. A common expression of this ratio is:

Propeller efficiency:

$$\eta_p = \frac{\mathrm{TV}}{\mathrm{P}} = \frac{TV}{550BHP} = J\frac{C_T}{C_P}$$

PROPELLERS

Propulsive Efficiency (η_p) Map.



PROPELLERS

Determination of propeller thrust – Using a propeller efficiency table.

Sometimes propeller manufacturer provides the designer with a propeller efficiency table:

J/Cp	0.02	0.03	0.04	0.05	0.06	0.07	0.08	0.09	0.1	0.11	0.12	0.13
0.20	0.50	0.46	0.41	0.36	0.31	0.25	0.20	0.17	0.14	0.12	0.10	0.09
0.25	0.57	0.54	0.50	0.45	0.39	0.33	0.26	0.22	0.18	0.15	0.13	0.11
0.30	0.62	0.61	0.57	0.52	0.47	0.40	0.33	0.27	0.22	0.19	0.16	0.14
0.35	0.67	0.66	0.63	0.59	0.54	0.48	0.40	0.32	0.27	0.23	0.20	0.17
0.40	0.70	0.70	0.68	0.64	0.60	0.55	0.48	0.38	0.32	0.27	0.23	0.20
0.45	0.72	0.74	0.71	0.69	0.65	0.61	0.55	0.46	0.37	0.31	0.27	0.23
0.50	0.74	0.76	0.75	0.72	0.69	0.66	0.61	0.53	0.43	0.36	0.30	0.26
0.55	0.75	0.78	0.77	0.75	0.72	0.69	0.66	0.61	0.50	0.40	0.34	0.29
0.60	0.76	0.79	0.79	0.77	0.75	0.72	0.69	0.65	0.59	0.46	0.39	0.33
0.65	0.77	0.81	0.80	0.78	0.76	0.74	0.71	0.68	0.64	0.55	0.44	0.37
0.70	0.77	0.81	0.81	0.80	0.78	0.76	0.73	0.71	0.67	0.64	0.50	0.42
0.75	0.78	0.82	0.82	0.81	0.79	0.77	0.75	0.72	0.69	0.66	0.61	0.49
0.80	0.78	0.82	0.83	0.82	0.80	0.78	0.76	0.74	0.71	0.68	0.65	0.57
0.85	0.78	0.83	0.83	0.82	0.81	0.79	0.77	0.75	0.73	0.70	0.67	0.64
0.90	0.78	0.83	0.83	0.83	0.81	0.80	0.78	0.76	0.74	0.71	0.68	0.65
0.95	0.78	0.83	0.83	0.83	0.82	0.80	0.79	0.77	0.74	0.72	0.69	0.67
1.00	0.77	0.83	0.84	0.83	0.82	0.80	0.79	0.77	0.75	0.73	0.70	0.68
1.05	0.77	0.82	0.83	0.83	0.82	0.81	0.79	0.77	0.76	0.73	0.71	0.68
1.10	0.76	0.82	0.83	0.83	0.82	0.81	0.79	0.78	0.76	0.74	0.71	0.69
1.15	0.76	0.82	0.83	0.83	0.82	0.81	0.79	0.78	0.76	0.74	0.72	0.69
1.20	0.75	0.81	0.83	0.82	0.82	0.81	0.79	0.78	0.76	0.74	0.72	0.70

FIGURE 14-43 A typical propeller efficiency table for a constant-speed propeller (no specific propeller type).

Step 1: Determine Advance Ratio

$$\boxed{J = \frac{V_0}{nD} = \frac{60 \cdot V_0}{RPM \cdot D}}$$

Step 2: Determine Power Coefficient

$$\boxed{C_P = \frac{P}{\rho n^3 D^5} = \frac{550 \times P_{BHP}}{\rho\left(\frac{RPM}{60}\right)^3 D^5} = \frac{118\ 800\ 000 \times P_{BHP}}{\rho \cdot RPM^3 \cdot D^5}}$$

Step 3: Extract Propeller Efficiency

extract the propeller efficiency, using the calculated C_P and J.

Step 4: Calculate Thrust

$$\boxed{T = \frac{\eta_p \times 550 \times P_{BHP}}{V}}$$

PROPELLERS

Universidad Pontificia Bolivariana

Fundada en 1936

Example: Determination of propeller thrust – Using a propeller efficiency table.

An airplane is equipped with a 70″ diameter constant-speed propeller driven by a 150 BHP piston engine. If flying at sea-level at 130 KTAS with the prop swinging at 2700 RPM and max power,
what is the magnitude of the thrust?

Solution

Step 1: Advance ratio

$$J = \frac{60 \cdot V}{RPM \cdot D} = \frac{60 \cdot (1.688 \times 130)}{2700 \cdot (70/12)} = \boxed{0.8360}$$

Step 2: Power Coefficient

$$C_P = \frac{P}{\rho n^3 D^5} = \frac{(550 \times 150)}{(0.002378)(2700/60)^3(70/12)^5} = \boxed{0.05637}$$

PROPELLERS

Example: Determination of propeller thrust – Using a propeller efficiency table.

$$J = \frac{60 \cdot V}{RPM \cdot D} = \frac{60 \cdot (1.688 \times 130)}{2700 \cdot (70/12)} = \boxed{0.8360} \qquad C_P = \frac{P}{\rho n^3 D^5} = \frac{(550 \times 150)}{(0.002378)(2700/60)^3(70/12)^5} = \boxed{0.05637}$$

J/Cp	0.02	0.03	0.04	0.05	0.06	0.07	0.08	0.09	0.1	0.11	0.12	0.13
0.20	0.50	0.46	0.41	0.36	0.31	0.25	0.20	0.17	0.14	0.12	0.10	0.09
0.25	0.57	0.54	0.50	0.45	0.39	0.33	0.26	0.22	0.18	0.15	0.13	0.11
0.30	0.62	0.61	0.57	0.52	0.47	0.40	0.33	0.27	0.22	0.19	0.16	0.14
0.35	0.67	0.66	0.63	0.59	0.54	0.48	0.40	0.32	0.27	0.23	0.20	0.17
0.40	0.70	0.70	0.68	0.64	0.60	0.55	0.48	0.38	0.32	0.27	0.23	0.20
0.45	0.72	0.74	0.71	0.69	0.65	0.61	0.55	0.46	0.37	0.31	0.27	0.23
0.50	0.74	0.76	0.75	0.72	0.69	0.66	0.61	0.53	0.43	0.36	0.30	0.26
0.55	0.75	0.78	0.77	0.75	0.72	0.69	0.66	0.61	0.50	0.40	0.34	0.29
0.60	0.76	0.79	0.79	0.77	0.75	0.72	0.69	0.65	0.59	0.46	0.39	0.33
0.65	0.77	0.81	0.80	0.78	0.76	0.74	0.71	0.68	0.64	0.55	0.44	0.37
0.70	0.77	0.81	0.81	0.80	0.78	0.76	0.73	0.71	0.67	0.64	0.50	0.42
0.75	0.78	0.82	0.82	0.81	0.79	0.77	0.75	0.72	0.69	0.66	0.61	0.49
0.80	0.78	0.82	0.83	0.82	0.80	0.78	0.76	0.74	0.71	0.68	0.65	0.57
0.85	0.78	0.83	0.83	0.82	0.81	0.79	0.77	0.75	0.73	0.70	0.67	0.64
0.90	0.78	0.83	0.83	0.83	0.81	0.80	0.78	0.76	0.74	0.71	0.68	0.65
0.95	0.78	0.83	0.83	0.83	0.82	0.80	0.79	0.77	0.74	0.72	0.69	0.67
1.00	0.77	0.83	0.84	0.83	0.82	0.80	0.79	0.77	0.75	0.73	0.70	0.68
1.05	0.77	0.82	0.83	0.83	0.82	0.81	0.79	0.77	0.76	0.73	0.71	0.68
1.10	0.76	0.82	0.83	0.83	0.82	0.81	0.79	0.78	0.76	0.74	0.71	0.69
1.15	0.76	0.82	0.83	0.83	0.82	0.81	0.79	0.78	0.76	0.74	0.72	0.69
1.20	0.75	0.81	0.83	0.82	0.82	0.81	0.79	0.78	0.76	0.74	0.72	0.70

FIGURE 14-43 A typical propeller efficiency table for a constant-speed propeller (no specific propeller type).

Step 3: Extract prop efficiency – here, by observation $\boxed{\eta \approx 0.81.}$

Step 4: Thrust

$$T = \frac{\eta_p \times 550 \times P_{BHP}}{V} = \frac{0.81 \times 550 \times 150}{1.688 \times 130} = \boxed{304\ \text{lb}_f}$$

PROPELLERS

Determination of propeller thrust – Using manufacturer's data.

It is always wise for the aircraft designer to establish a good relationship with propeller manufacturers by committing to using their products to:

1. Encourage the propeller manufacturer to provide the designer with useful propeller performance information.
2. If the business case is promising enough, even persuade the manufacturer to design a propeller to match the designer's desired characteristics.

J	C_T	C_P	η
0.3142	0.1250	0.0734	0.535
0.4398	0.1077	0.0710	0.667
0.5655	0.0898	0.0674	0.754
0.6912	0.0703	0.0596	0.815
0.8168	0.0497	0.0475	0.854
0.9425	0.0278	0.0306	0.856

Theoretical calculated performance
Includes inflow and spinner/hub drag

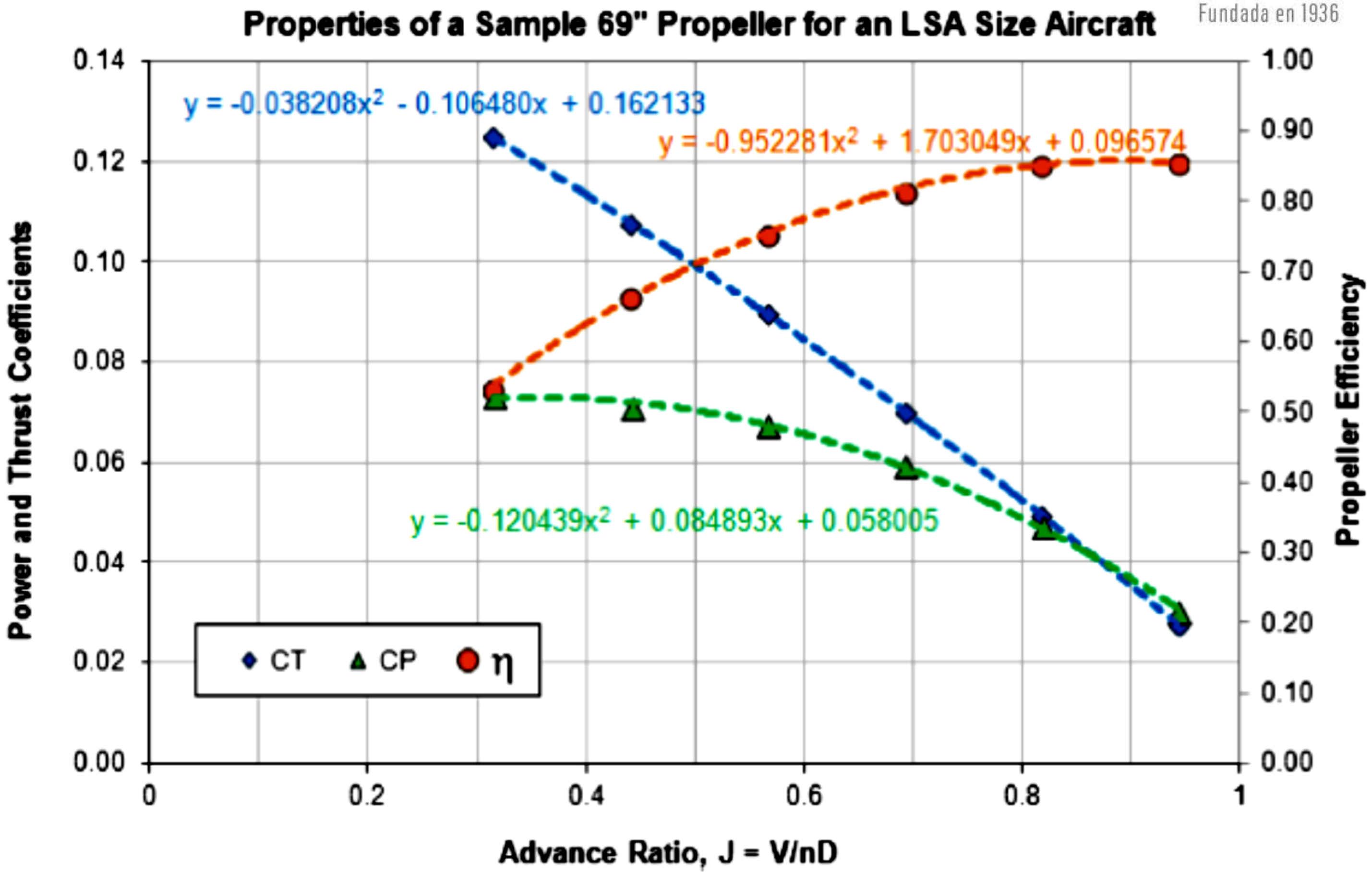


FIGURE 14-45 A typical propeller efficiency graph for a fixed-pitch propeller (no specific propeller type).

PROPELLERS

Determination of propeller thrust – Using manufacturer's data.

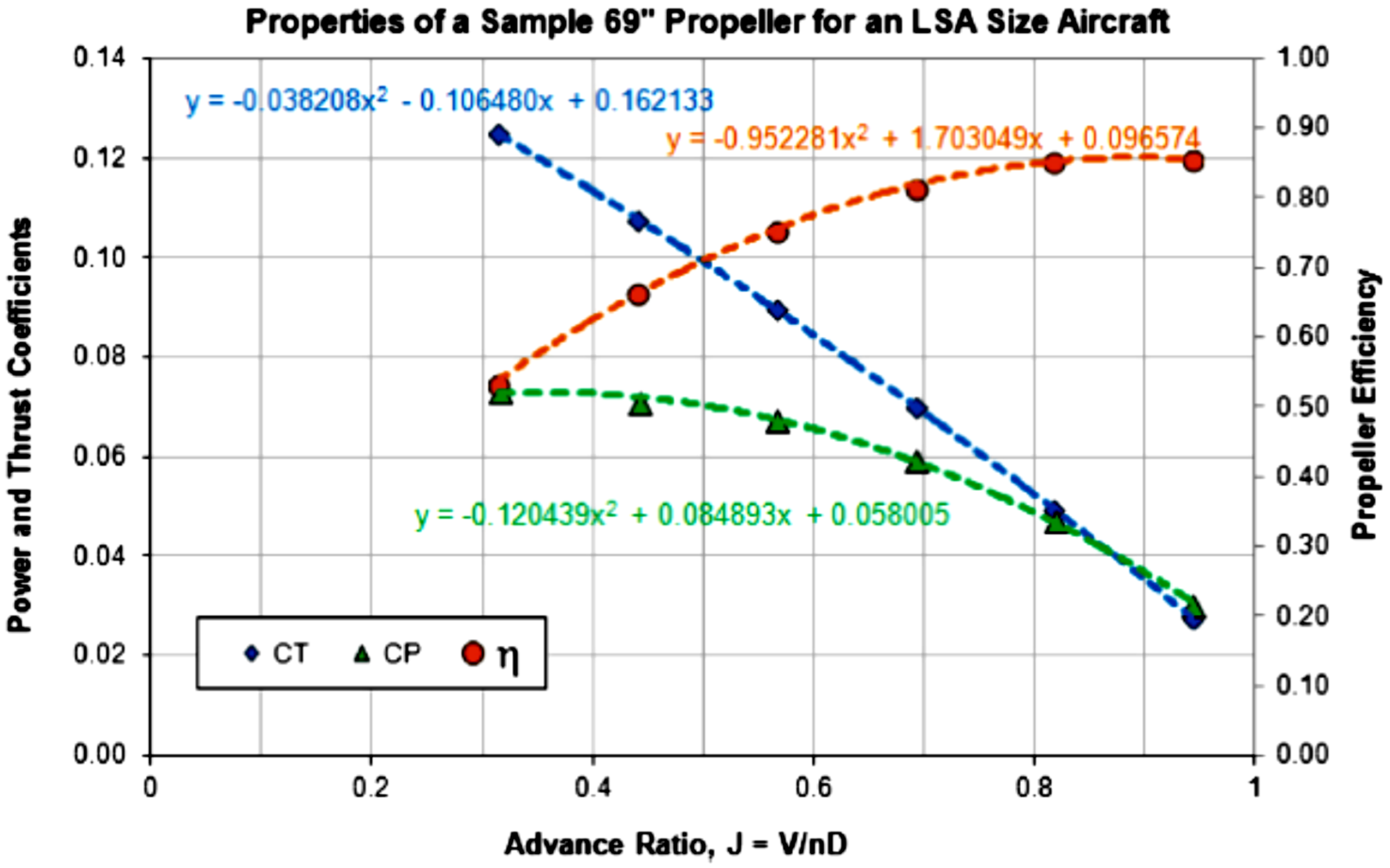


FIGURE 14-45 A typical propeller efficiency graph for a fixed-pitch propeller (no specific propeller type).

J	C_T	C_P	η
0.3142	0.1250	0.0734	0.535
0.4398	0.1077	0.0710	0.667
0.5655	0.0898	0.0674	0.754
0.6912	0.0703	0.0596	0.815
0.8168	0.0497	0.0475	0.854
0.9425	0.0278	0.0306	0.856

Theoretical calculated performance
Includes inflow and spinner/hub drag

$$\eta_P = 0.096574 + 1.703049J - 0.952281J^2$$
$$C_T = 0.162133 - 0.106480J - 0.038208J^2$$
$$C_P = 0.058005 + 0.084893J - 0.120439J^2$$

PROPELLERS

Example: Determination of propeller thrust – Using manufacturer's data.

An airplane is equipped with a 69″ diameter fixed-pitch propeller driven by a 100 BHP piston engine. If flying at sea-level at 110 KCAS with the prop swinging at 2400 RPM and 75% power,
what is the magnitude of the thrust?

Solution

Step 1: Advance ratio

$$J = \frac{60 \cdot V}{RPM \cdot D} = \frac{60 \cdot (1.688 \times 110)}{2400 \cdot (69/12)} = \boxed{0.8073}$$

Step 2: Propeller efficiency

$$\eta_p = 0.096574 + 1.703049J - 0.952281J^2 = \boxed{0.8508}$$

Step 3: Thrust

$$T = \frac{\eta_p \times 550 \times P_{BHP}}{V} = \frac{0.8508 \times 550 \times 75}{1.688 \times 110} = \boxed{189\ \text{lb}_f}$$

ENGINE SELECTION

Table 8.10 Features of several propellers

No.	Manufacturer	Designation	Number of blades	Diameter	Features	Aircraft
1	Smiths	R381	6	12 ft 6 in.	Constant speed	Saab 2000
2	Smiths	R391	6	13 ft 6 in.	Constant speed, composite	Lockheed C-130J, Alenia C-27J
3	Smiths	R408	6	13 ft 6 in.	Constant speed, composite	Dash 8 Q400, Y-8F600
4	Sensenich	W72CK-42	2	70 in.	Wood	Aeronca 7AC
5	Hamilton Sundstrand	14SF-5	4	13 ft	Aluminum and composite	ATR 72, Canadair 215T, Dash 8 Q300
6	Hamilton Sundstrand	568F-1	6	13 ft	Constant speed, composite	Casa c-295, Ilyushin Il-114
7	Hartzell	HC-E4A-2/E9612	4	77 in.	Metal, fully feathering	Beechcraft T-6
8	Hartzell	–	5	77 in.	Composite, constant speed, reversible pitch	Embraer EMB-314 Super Tucano
9	Hartzell	HC-D4N-ZA/09512A	4	77 in.	Metal, constant speed, fully feathering	Pilatuc PC-9 M
10	Hartzell	HC-B3TN-3	4	77 in.	Metal, constant speed, fully feathering	Beechcraft King Air 90
11	Hartzell	HC-E5N	5	77 in.	Aluminum, constant speed, fully feathering, reversible pitch	Piaggio P-180 Avanti II
12	Hartzell	HC-E4N-3Q	4	77 in.	Metal	Piper PA-46-500TP Meridian
13	Hartzell	BHC-J2YF-1BF	2	72 in.	Metal	Cirus SR20
14	Hartzell	HC-B5MP-3F	5	72 in.	Metal, constant speed, feathering, reversible pitch	Air tractor AT-802
15	Hartzell	HC-C2YK-1BF	2	74 in.	Metal, constant speed	Aermacchi SF-260EA

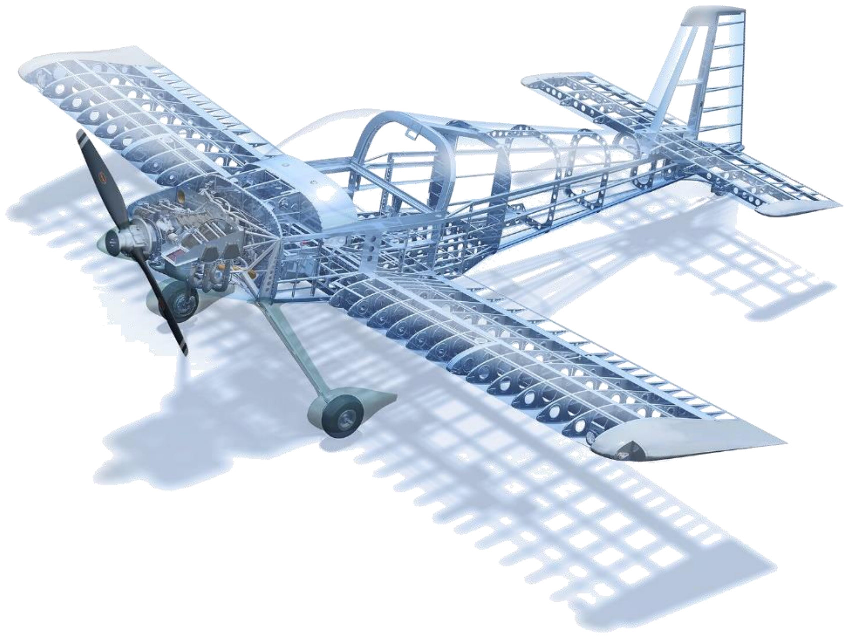
PISTON ENGINE

Universidad Pontificia Bolivariana
Fundada en 1936

TABLE 7-5 Selected Manufacturers of Piston Engines for GA and UAV Aircraft

Maker	Country	Application	Horsepower range (BHP)	Website
Continental Motors	USA	GA, LSA	75–360	www.genuinecontinental.aero
JPX	France	LSA, UAV	14–90	www.jpx.fr
Limbach Engines	Germany	GA, LSA	20–167	www.limflug.de
Hirth Engines	Germany	GA, LSA	14.6–102	www.hirth-motoren.de
Rotax Engines	Austria	GA, LSA	40–115	http://www.rotax-aircraft-engines.com/
SMA Engines	France	GA	227	www.smaengines.com
Textron Lycoming	USA	GA, LSA, UAV	115–400	www.lycoming.com
ULPower Aero Engines	Belgium	GA, LSA, UAV	97–130	www.ulpower.com
Zenoah	Japan	UAV, RC	1.68–5.82	www.zenoah.net
Jabiru Engines	Australia	GA, LSA, UAV	85–120	www.jabiru.net.au

GA = general aviation aircraft, LSA = light sport aircraft, UAV = unmanned aerial vehicles, RC = radio-controlled aircraft.



PISTON ENGINE

TABLE 7-6 Power and Weight of Selected Piston Engines for GA and Experimental Aircraft

Manufacturer	Type	Cylinders	Dsplcmnt in^3	TBO hours	Weight lb$_f$	RPM	Rated power BHP	*SFC* lb$_f$/hr/BHP
Lycoming	O-235	4	235	2400	243–255	2800	115–125	0.6
	O-320	4	320	2000	268–299	2700	150–160	0.6
	O-360	4	360	2000	280–301	2700	168–180	0.6
	IO-390	4	390	2000	308	2700	210	0.6
	IO-580	6	580	–	444	2700	315	0.6
	IO-720	8	720	–	593–607	2650	400	0.6
Continental Motors	IO-360	6	360	–	327–331	2800	200	0.6
	IO-550	6	550	–	467–470	2700	300–310	0.6
Hirth Motoren	3003[a,b]	4	63.6	1000	93	6500	102	0.83–1.80
	3501[a,b]	2	38.1	1000	78	5500	60	
	3701[a,b]	3	57.3	1000	100	6000	100	
Rotax	447UL[a]	2	26.6	300	72	6800	40	–
	503UL[a]	2	30.3	300	85	6800	49	–
	582UL[a]	2	35.4	300	79	6800	65	–
	912UL[a]	4	73.9	1500	122	5800	81	0.47
	912ULS[a]	4	73.9	1500	125	5800	100	0.47
	914UL[a]	4	73.9	1200	154	5800	115	–

[a]*Non-certified.*
[b]*Two-stroke.*
TBO = time between overhauls, RPM = revolutions per minute, *SFC* = specific fuel consumption.

Universidad Pontificia Bolivariana
Fundada en 1936

PISTON ENGINE

Table 8.12 Primary specifications for several piston engines [8]

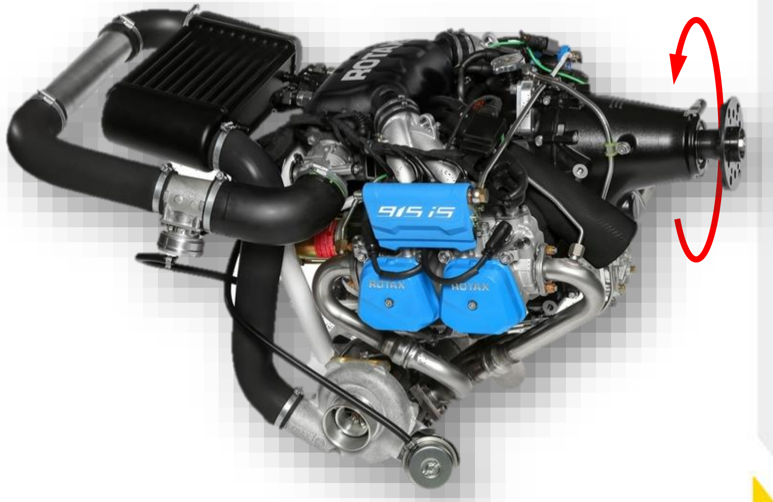
No.	Manufacturer	Designation	Arrangement	Number of cylinders	Cooling	Mass (kg)	Max power (hp)
1	Hirth	F33B	–	1	Air	13	24
2	Rotax	447 UL-1V	In-line	2	Air	26.8	39.6
3	BMW	R115ORS	Opposed	2	Air + oil	76.3	96.6
4	Subaru	EA81-140	Opposed	4	Liquid	100	140
5	Wilksch	WAM 160	In-line	4	Liquid	120	160
6	Textron-Lycoming	O-320-H	Opposed	4	Air	128	160
7	PZL	F 6A6350-C1	Opposed	6	Air	150	205
8	TCM	Tsio-360-RB	Opposed	6	Air	148.6	220
9	Textron-Lycoming	IO-540-C	Opposed	6	Air	170	250
10	TCM	IO-470-D	Opposed	6	Air	193.3	260
11	Bombardier	V300	Vee		Liquid	210	300
12	TCM	TSIOL-550-C	Opposed	6	Liquid	188.4	350
13	Textron-Lycoming	IO-270-A	Opposed	8	Air	258	400
14	VOKBM	M-9F	Radial	9	Air	214	420
15	Orenda	OE600 Turbo	Vee	8	Liquid	–	750
16	PZL	K-9	Radial	9	Air	580	1 170

PISTON ENGINE

Fundamentals – Break Horsepower (BHP), amount of power delivered at the engine output shaft of a piston engine.

TABLE 7-3 Common Fuel Grades for Piston Engine Use

Fuel grades	Color	Comment
80/87	Red	The first number (80) is the octane rating assuming a lean mixture. The second number (87) indicates the rating at a rich mixture. Used for aircraft engines with low compression ratios. No longer produced.
82UL	Purple	UL stands for unleaded. Similar to mogas, but without automotive additives. Intended for low-compression engines such as those common in experimental aircraft and aircraft that have STCs permitting the use of mogas. No longer produced.
91/96 91/96UL	Brown	UL stands for unleaded. Avgas is often intended for military use (e.g. UAVs). Produced today by the Swedish fuel manufacturer Hjelmco in a clear color.
100LL	Blue	LL stands for low lead. The most common avgas in use today. The fuel can be used with engines designed for 80/87.
100/130	Green	Also called avgas 100. Has been superseded by 100LL, although still available in limited quantities.
115/145	Purple	Leaded fuel produced for warbirds and the supercharged radial engines used to power the passenger planes of the 1940s–1960s. Now produced in limited quantities for air races. This fuel is necessary in order to obtain rated power in such engines.



PISTON ENGINE

Fundamentals – Specific fuel consumption (SFC) is one of the most important metrics employed in aviation.

It is important not only in aircraft design but also in the operation of the aircraft. It indicates how efficiently a power plant converts chemical into mechanical energy.

The fuel consumption of piston engines is always measured in terms of mass or weight of fuel flow per unit time, per unit of power. If the power of the engine is known in BHP or kW, the SFC can be computed as shown:

$$SFC = c_{bhp} \equiv \frac{\text{weight of fuel in lbs/hour}}{\text{power in brake horsepower}} = \frac{\text{lb}_f/\text{hr}}{\text{BHP}}$$

$$SFC = c_{ws} \equiv \frac{\text{mass of fuel in grams/sec}}{\text{power in watts}} = \frac{\text{g}}{\text{W} \times \text{sec}} = \frac{\text{g}}{\text{J}}$$

TABLE 7-7 Specific Fuel Consumption of Typical Normally Aspirated Piston Engines for Aircraft

	lb_f/hr/BHP	gr/kW/hr
Two-stroke	0.83–1.80	280–600
Four-stroke	0.42–0.60	140–205

Example

A piston engine is found to consume 10 gallons of fuel per hour while generating 150 BHP. Determine c_{bhp} and c_{ws} (in the UK and SI systems, respectively). Fuel weighs 6 lbs per gallon:

$$c_{bhp} = \frac{10 \text{ gallons/hr}}{150 \text{ BHP}} = \frac{60 \text{ lb}_f/\text{hr}}{150 \text{ BHP}} = \boxed{0.400 \frac{\text{lb}_f/\text{hr}}{\text{BHP}}}$$

$$c_{ws} = \frac{(27.18 \text{ kg/hr})}{150 \text{ BHP}} = \frac{(0.00755 \text{ kg/s})}{111.9 \text{ kW}} = \frac{(7.55 \text{ g/s})}{111.9 \text{ kW}}$$

$$= \frac{(7550 \text{ mg/s})}{111{,}900 \text{ W}} = \boxed{0.06747 \frac{\text{mg}}{\text{W} \times \text{s}}}$$

ENGINE SELECTION

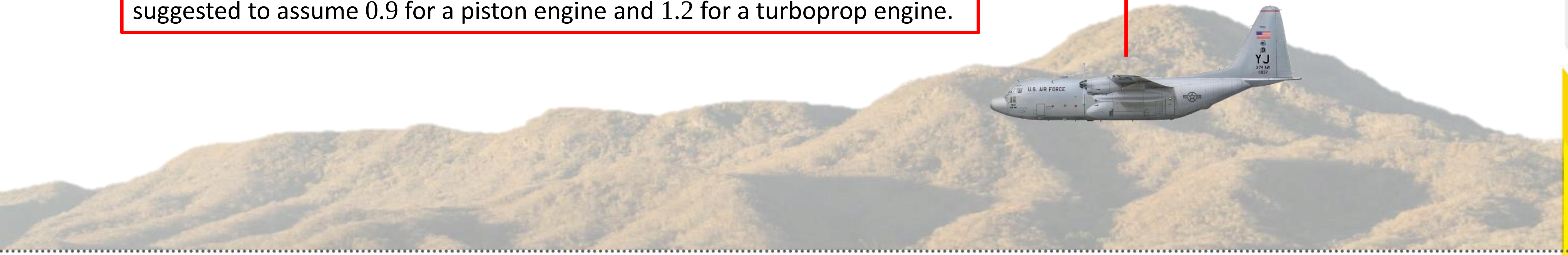
Engine performance.

Aircraft required power

$$\eta_P = \frac{TV}{P_{in}}$$

Aircraft available power

$$P_{max} = P_{max_{SL}} \left(\frac{\rho}{\rho_o} \right)^m$$



P_{max} and ρ represent the maximum shaft power output and air density, respectively, at a given altitude and $P_{max,SL}$ and ρ_0 are the corresponding values at sea level. The value of m changes as the technology advances. It is suggested to assume 0.9 for a piston engine and 1.2 for a turboprop engine.

Universidad Pontificia Bolivariana

Fundada en 1936

PISTON ENGINE

Universidad Pontificia Bolivariana

Fundada en 1936

Effect of altitude on engine power.

Simple altitude-dependency model:

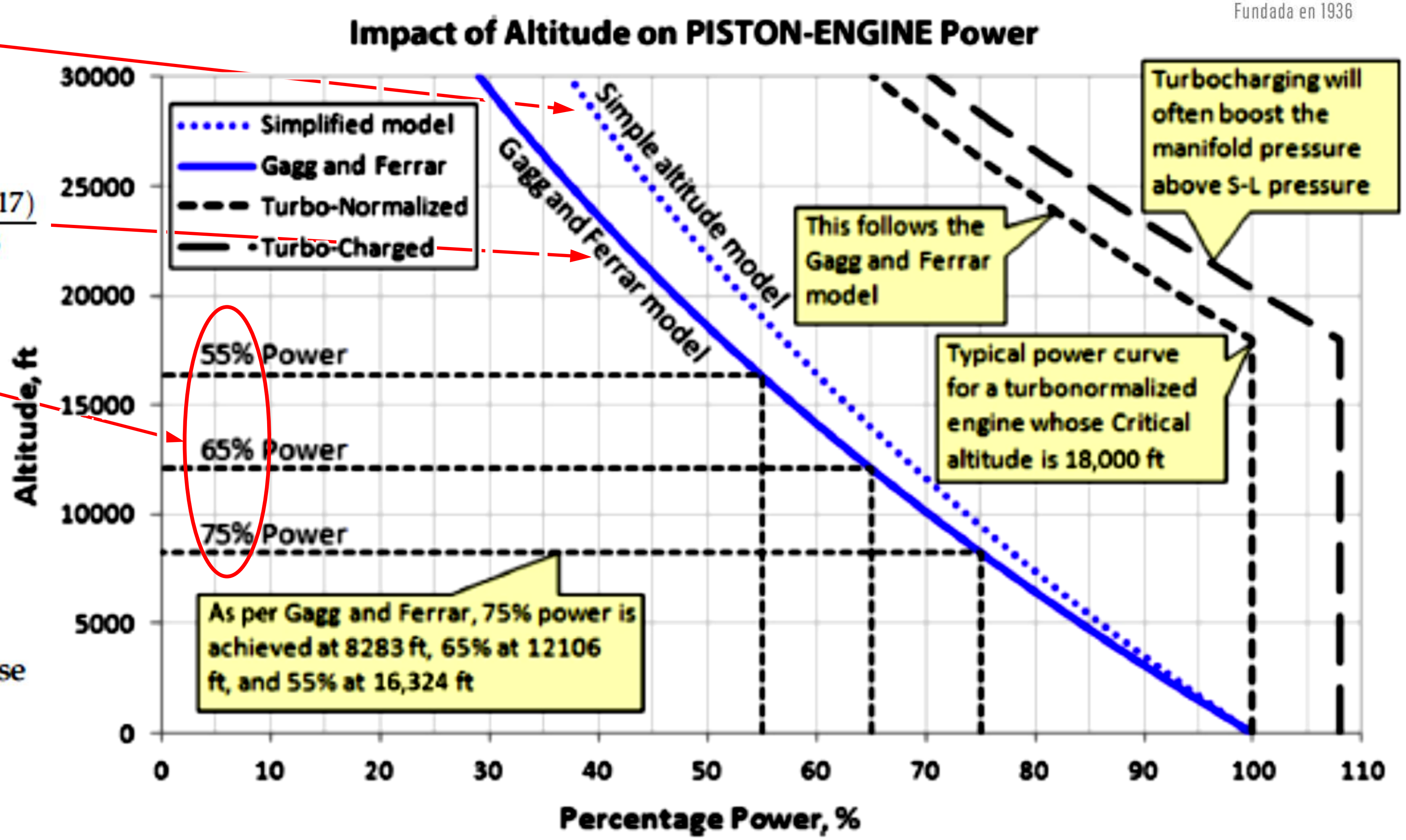
$$P = P_{SL}\left(\frac{\rho}{\rho_{SL}}\right) = P_{SL}\sigma$$

Gagg and Ferrar model: **More accurate model according to *Snorri.***

$$P = P_{SL}\left(\sigma - \frac{(1-\sigma)}{7.55}\right) = P_{SL}(1.132\sigma - 0.132) = P_{SL}\frac{(\sigma - 0.117)}{0.883}$$

Power in *BHP* @ s.l.

Typical power settings reported by manufacturers



Effect of Temperature on engine power.

$$\frac{P}{P_{std}} = \sqrt{\frac{T_{std}}{T_{OAT}}} = \sqrt{\frac{518.67(1-\kappa h)}{T_{OAT}\ {}^\circ R}} = \sqrt{\frac{273.15(1-\kappa h)}{T_{OAT}\ K}}$$

P_{std} = standard power at altitude and ISA κ = lapse rate constant

T_{std} = standard day temperature

T_{OAT} = outside air temperature at condition

h = pressure altitude at condition

PISTON ENGINE

Effect of altitude on engine power – Example.

Universidad Pontificia Bolivariana
Fundada en 1936

Estimate the power of a piston engine rated at 100 BHP while being operated at full power at 10,000 ft on a day on which the OAT is 30 °F (or 30 °R) higher than ISA.

Solution

Method 1: Use of Equation $\frac{P}{P_{std}} = \sqrt{\frac{T_{std}}{T_{OAT}}} = \sqrt{\frac{518.67(1-\kappa h)}{T_{OAT}\ {}^\circ R}} = \sqrt{\frac{273.15(1-\kappa h)}{T_{OAT}\ K}}$

Lapse rate factor:

$$(1-\kappa h) = (1 - 0.0000068756 \times 10{,}000) = 0.9312$$

Standard day temperature at 10,000 ft:

$$T_{std} = 518.67 \times 0.9312 = 483.0\ {}^\circ R$$

Density ratio at 10,000 ft (standard day):

$$\sigma = 0.9312^{4.2561} = 0.7385$$

Maximum power at 10,000 ft per Gagg and Ferrar:

$$P = P_{SL}(1.132\sigma - 0.132)$$
$$= 100(1.132 \times 0.7385 - 0.132) = \boxed{70.4\ \text{BHP}}$$

This is further reduced by the warmer-than-normal day as follows:

$$\boxed{P = P_{std}\sqrt{\frac{T_{std}}{T_{OAT}}} = 70.4\sqrt{\frac{483.0}{483.0+30}} = 68.3\ BHP}$$

Method 2: Ideal gas law with Gagg and Ferrar

The answer can also be estimated using the ideal gas equation as follows:

Pressure at 10,000 ft:

$$p = 2116(1-kh)^{5.2561} = 2116 \times 0.9312^{5.2561} = 1455\ \text{psf}$$

Density at 10,000 ft:

$$\rho = \frac{p}{RT} = \frac{1455}{1716 \times (483+30)} = 0.001653\ \text{slugs/ft}^3$$

Density ratio at 10,000 ft:

$$\sigma = \frac{\rho}{\rho_0} = \frac{0.001653}{0.002378} = 0.6951$$

Gagg and Ferrar:

$$\boxed{P = 100(1.132 \times 0.6951 - 0.132) = 65.5\ \text{BHP}}$$

ENGINE SELECTION

Universidad Pontificia Bolivariana
Fundada en 1936

Table 8.11 Primary specifications for several electric engines [8]

No.	Manufacturer	Designation	Length (mm)	Diameter (mm)	Mass (kg)	Maximum current (A)	Max power (kW)
1	Hacker	A20-26M EVO	28	28	0.042	12 A; 1130 rpm/V	0.150
2	Raiden	T30A	42.7	60	0.271	58	0.400
3	Applied Motion	M1500-232-7-000	190	100	5.7	9.5	1.5
4	Leopard	LBP4074	40	38	0.347	120 A; 2000 rpm/V	2.6
5	Yuneec	Power drive 10	–	160	4.54	180	10
6	Electroavia	GMPE 102 Devoluy	200	210	11.57	250	19.4
7	Electroavia	GMPE 201 Arambre	200	210	12	275	32
8	Yuneec	Power drive 40	–	240	17	285	40

Table 8.12 Primary specifications for several piston engines [8]

No.	Manufacturer	Designation	Arrangement	Number of cylinders	Cooling	Mass (kg)	Max power (hp)
1	Hirth	F33B	–	1	Air	13	24
2	Rotax	447 UL-1V	In-line	2	Air	26.8	39.6
3	BMW	R115ORS	Opposed	2	Air + oil	76.3	96.6
4	Subaru	EA81-140	Opposed	4	Liquid	100	140
5	Wilksch	WAM 160	In-line	4	Liquid	120	160
6	Textron-Lycoming	O-320-H	Opposed	4	Air	128	160
7	PZL	F 6A6350-C1	Opposed	6	Air	150	205
8	TCM	Tsio-360-RB	Opposed	6	Air	148.6	220
9	Textron-Lycoming	IO-540-C	Opposed	6	Air	170	250
10	TCM	IO-470-D	Opposed	6	Air	193.3	260
11	Bombardier	V300	Vee		Liquid	210	300
12	TCM	TSIOL-550-C	Opposed	6	Liquid	188.4	350
13	Textron-Lycoming	IO-270-A	Opposed	8	Air	258	400
14	VOKBM	M-9F	Radial	9	Air	214	420
15	Orenda	OE600 Turbo	Vee	8	Liquid	–	750
16	PZL	K-9	Radial	9	Air	580	1 170

ENGINE SELECTION

Table 8.13 Primary specifications for several turboprop engines [8]

No.	Manufacturer	Designation	Arrangement	Airflow (kg/s)	Length (mm)	Width (mm)	Mass (kg)	Max power (hp)
1	Innodyn	255TE	C	–	762	360	85.3	255
2	Rolls Royce	250-B17	6A + C	1.56	1143	483	88.4	420
3	Turbomeca	Arrius 2F	C	–	945	459	103	504
4	P&WC	PT6A-27	3A + C	3.08	1 575	483	149	680
5	Honeywell	TPE331-3	C + C	3.54	1 092	533	161	840
6	PZL	TWD-10B	6A + C	4.58	2 060	555	230	1 011
7	P&WC	PT6A-65b	4A + C	4.31	1 880	483	225	1 100
8	P&WC	PT6A-69	4A + C	–	1 930	483	259.5	1 600
9	GE	CT7-9	5A + C	5.2	2 438	737	365	1 940
10	P&WC	PW123C	C, C	–	2 143	635	450	2 150
11	Klimov	TV3-113VMA-SB2	12A	9	2 860	880	570	2 500
12	DEMC	WJ5E	10A	14.6	2 381	770	720	2 856
13	Rolls Royce	AE 2100C	14A	16.33	2 743	1151	715.8	3 600
14	Progress	AI-20M	10A	20.7	3 096	842	1 040	3 943
15	P&WC	PW150A	3A + C	–	2 423	767	690	5 071
16	EPI	TP400-D6	5A	26.31	3 500	924.5	1 795	11 000

A: axial stage, C: centrifugal stage, C, C: two stages on different shafts.

Table 8.14 Primary specifications for several turbofan and turbojet engines [8]

No.	Manufacturer	Designation	Arrangement	Airflow (kg/s)	BPR	Length (mm)	Diameter (mm)	Mass (kg)	Max thrust (kN)
1	GE Honda	HF120	1A + 2A + 2A	–	2.9	1 118	538	181	9.12
2	Honeywell	TFE731-20	1F, 4A + C	66.2	3.1	1 547	716	406	15.57
3	Rolls Royce	Viper 680	8A	27.2	0	10 806	740	379	19.39
4	P&W	J-52-408	5A, 7A	64.9	0	3 020	814	1 052	49.8
5	Rolls Royce	Spey 512	5A, 12A	94.3	0.71	2 911	942	1 168	55.8
6	SNECMA	Atar 9K50	9A, a/b	73	0	5 944	1 020	1 582	70.6
7	Volvo/GE	RM12	3F, 7A, a/b	68	0.28	4 100	880	1 050	80.5
8	P&W	JT8D-219	1F + 6A, 7A	221	1.77	3 911	1 250	2 092	93.4
9	SNECMA	M53 P2	3F, 5A, a/b	86	0.35	5 070	1 055	1 500	95
10	CFM	CFM56-2B	1F + 3A, 9A	370	6	2 430	1 735	2 119	97.9
11	P&W	F-100-220P	3F, 10A, a/b	112.5	0.6	5 280	1 181	1 526	120.1
12	Saturn	AL-31FM	4F, 9A, a/b	112	0.57	4 950	1 277	1 488	122.6
13	Soyuz	R-79	5F, 6A, a/b	120	1	5 229	1 100	2 750	152
14	Rolls-Royce	RB211-524B	1F, 7A, 6A	671	4.4	3 106	2 180	4 452	222
15	P&W	JT9D-7R4H	1F + 4A, 11A	769	4.8	3 371	2 463	4 029	249
16	GE	GE90-76B	1F + 3A, 10A	1 361	9	5 182	3 404	7 559	340
17	Rolls-Royce	Trent 895	1F, 8A, 6A	1 217	5.79	4 369	2 794	5 981	425
18	GE	GE90-115B	1F + 4A, 9A	1 641	8.9	7 290	3 442	8 761	511.6

BPR: bypass ratio, a/b: afterburner, A: axial, C: centrifugal, F: fan.

ENGINE SELECTION

Universidad Pontificia Bolivariana
Fundada en 1936

Propulsion system selection steps

1. Identify and list the propulsion system design requirements.
2. Determine the engine type (if not define in *RFP*).
3. Determine the number of engines (if not define in *RFP*).
4. Determine the engine locations.
5. Select an engine from the manufacturers' catalogs (*COTS*) or order an engine design team to design a new engine from scratch.

   Obtain the necessary information on:

   i. Engine geometry and clearance envelope.
   ii. Engine mounting (attachment) points.
   iii. Engine air-ducting requirements.
   iv. Engine thrust reversing requirements.
   v. Engine exhaust system requirements.
   vi. Engine accessory requirements.
   vii. Engine *c.g.* location.
   viii. Engine firewall requirements.

6. Check the propeller diameter (prop-driven engine).

ENGINE SELECTION

Propulsion system selection steps

7. Design the inlet (if a jet engine) – engine inlet requirements can play a major role in the layout of those jet engine installations where long inlets are needed. This is the case in many “buried” installations (For supersonic aircrafts a variable geometry inlet duct is often required).
8. Check propeller power deliver and diameter.
9. Check aircraft’s total drag (thrust or power required) and compare the result with available thrust (or power).
10. Design the engine installation.
11. Check if the propulsion system design satisfies the design requirement.
12. If any design requirement is not met, return to the relevant design step and reselect/recalculate the corresponding parameter.
13. Optimize - Make certain that the proposed engine installations are compatible with such requirements as:
    i. Acceptable FOD characteristics.
    ii. Geometric clearance when static on the ramp: no nacelle or propeller tip may touch the ground with deflated landing gear struts and tires.
    iii. Geometric clearance during Take-Off rotation: no scraping of nacelles or of propeller tips is allowed, with deflated landing gear struts and tires.
    iv. Geometric clearance during a low-speed approach with a 5° bank angle.
    v. No gun exhaust gasses may enter the inlet a jet engine. Such gun gasses are highly corrosive to fan, compressor and turbine blades.

ENGINE SELECTION

Example – Propulsion system selection

Universidad Pontificia Bolivariana
Fundada en 1936

Design a propulsion system for a low-wing, T-tail, transport aircraft to carry eight passengers for a range of 4000 km with the following characteristics:

$$m_{TO} = 7000\,\text{kg},\ S = 29\,\text{m}^2,\ C_{D_o} = 0.028,\ \text{AR} = 8,\ e = 0.92$$

The aircraft must be capable of cruising with a maximum cruising speed of 320 KTAS at 20 000 ft altitude. For this problem, you need to discuss and determine the following:

1. engine thrust and engine power at cruise;
2. engine type;
3. number of engine(s);
4. engine(s) location;
5. engine selection;
6. prop diameter and number of blades (if prop-driven engine).

The propulsion system must be of low manufacturing cost, low operating cost, with high efficiency, and airworthiness requirements must be met. Then, sketch a front view and a top view of the aircraft to show the propulsion system installation. Assume any other parameters as needed.

ENGINE SELECTION

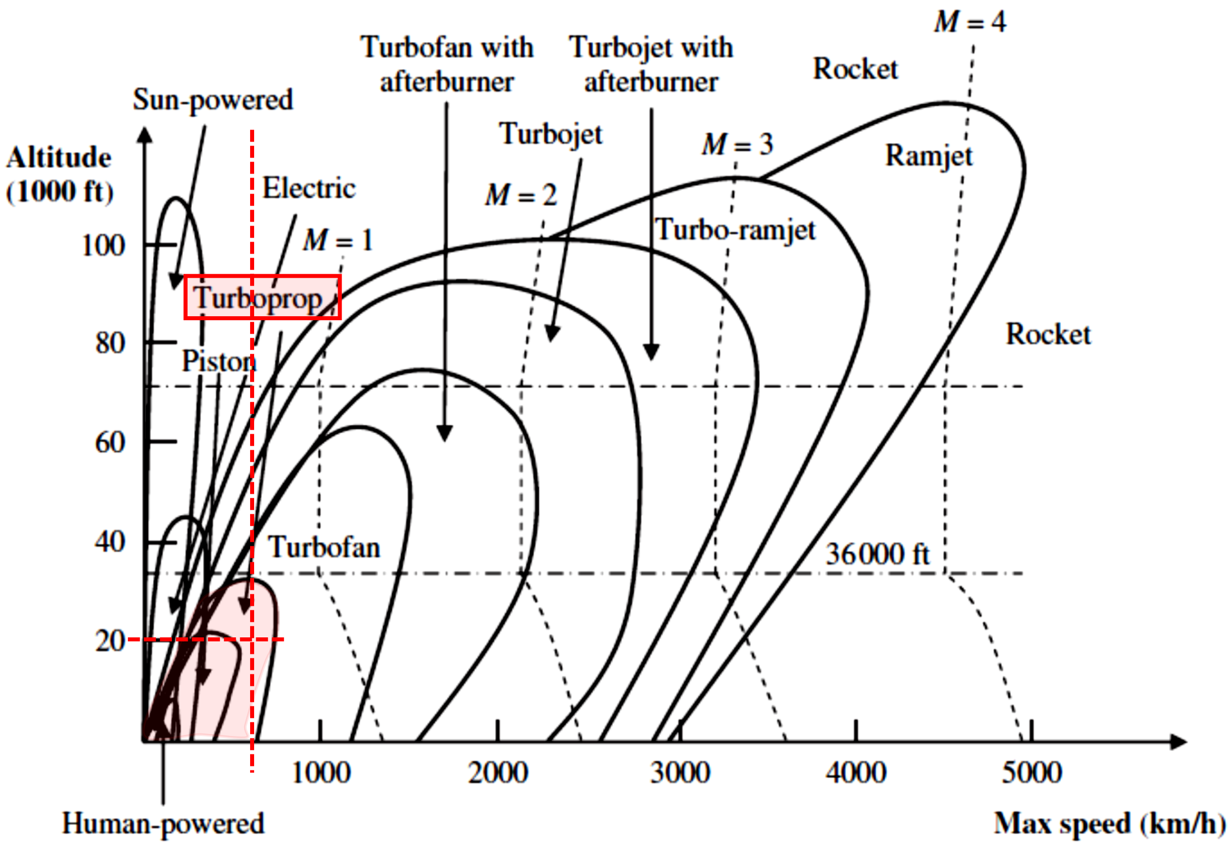
Example – Propulsion system selection

Solution:
The solution is presented in six sections. The optimization is left to the interested reader.

1. **Design Requirements**. The following design requirements are identified and listed in order of importance: aircraft performance (maximum speed), engine manufacturing cost, engine operating cost, flight safety, engine efficiency, maintainability, and manufacturability. Other general requirements (such as structural requirements, installation constraints, and integration) are important, but not considered at this moment. Other performance items such as ceiling, rate of climb, and take-off run are not given by the problem statement, so they are not discussed here.

2. **Engine type**. It is observed that the first three requirements are low manufacturing cost, low operating cost, and high efficiency. Since the vehicle is a transport aircraft carrying human passengers, the noise pollution cabin must be addressed and mitigated. Noise level can be controlled using dynamic vibration absorbers mounted throughout the cabin flight deck plus bagged glass fiber insulation. For these reasons, the prop-driven engine is the most suitable engine for this design problem. Due to the high speed and high altitude, only a turboprop engine will meet these required performance elements. According to Figure 8.7, piston-prop, electric, or solar-powered engines are not capable of meeting a maximum speed of 320 KTAS at 20 000 ft altitude. Although turbofan and turbojet engines are capable of this performance mission, they are costlier (both in manufacturing cost and in operating cost) compared with a turboprop engine.

V_{cruise} = 320 KTAS = 592.64 km/h



ENGINE SELECTION

Example – Propulsion system selection

Universidad Pontificia Bolivariana

Fundada en 1936

Solution:

3. **Number of Engines.** The aircraft is carrying eight passengers. The flight safety of passengers is of prime importance in a civil transport aircraft. To have greater safety, a multi-engine propulsion system is adopted. The more engines, the safer the flight. But as we increase the number of engines, the flight cost plus maintenance cost is increased. We start with two engines; if a suitable rate of safety can be achieved with these we stick with it, otherwise we go for a higher number of engines.

Furthermore, the aircraft range is required to be 4000 km, so it may fly over the ocean such as in a flight from Los Angeles to Hawaii. Statistics clearly indicates that there have been and there will be unfavorable circumstances where an engine may become inoperative during a flight operation. The multi-engine propulsion system configuration is one of the best solutions to the OEI case issue.

First we need to determine the required engine power for this mission. The air density at 20 000 ft is 0.653 kg/m^3. The cruise lift coefficient is:

$$C_{L_C} = \frac{2\,\mathrm{mg}}{\rho S\,(V_C)^2} = \frac{2 \cdot 7000 \cdot 9.81}{0.653 \cdot 29 \cdot (320 \cdot 0.514)^2} = \boxed{0.267}$$

Aircraft drag at cruise:

$$K = \frac{1}{\pi \cdot e \cdot \mathrm{AR}} = \frac{1}{3.14 \cdot 0.92 \cdot 8} \Rightarrow K = \boxed{0.043}$$

$$C_D = C_{D_o} + K{C_L}^2 = 0.028 + 0.043 \cdot 0.267^2 = \boxed{0.031}$$

$$D = \frac{1}{2}\rho V^2 S C_D = \frac{1}{2} \cdot 0.653 \cdot (320 \cdot 0.514)^2 \cdot 29 \cdot 0.031 \Rightarrow \boxed{D = 7980.4\ \mathrm{N}}$$

Required engine thrust at cruise:

$$T = D = 7980.4\ \mathrm{N}$$

Required engine power at 20 000 ft:

$$P_{20\,000} = \frac{TV_C}{\eta_P} = \frac{7980.4 \cdot (320 \cdot 0.514)}{0.75} \Rightarrow P_{20\,000} = 1\,751\,675\ \mathrm{W}$$

$$\boxed{P_{20\,000} = 1751.675\ \mathrm{kW} = 2349\ \mathrm{hp}}$$

ENGINE SELECTION

Example – Propulsion system selection

Universidad Pontificia Bolivariana

Solution:

Required engine power at sea level:

$$P_{max} = P_{max_{SL}} \left(\frac{\rho}{\rho_o}\right)^{1.2} \Rightarrow 1751.6 = P_{max_{SL}} \left(\frac{0.653}{1.225}\right)^{1.2} = P_{max_{SL}}$$

$$\Rightarrow P_{max_{SL}} = \frac{1751.6}{0.47} \Rightarrow P_{max_{SL}} = 3725.73\,\text{kW} = \boxed{4996.3\,\text{hp}}$$

Referring to engine manufacturers' catalogs (e.g., Table 8.12), it is observed that there is no piston engine available in the market that delivers this much power even with three engines. This is another reason that we have chosen the turboprop engine. Moreover, there are very few turboprop engines (e.g., Table 8.13) that generate about 5000 hp; hence, this is another reason for the decision of a multi-engine configuration. There are a number of turboprop engines which deliver about 2500 hp, therefore two turboprop engines are selected for this aircraft.

Table 8.12 Primary specifications for several piston engines [8]

No.	Manufacturer	Designation	Arrangement	Number of cylinders	Cooling	Mass (kg)	Max power (hp)
1	Hirth	F33B	–	1	Air	13	24
2	Rotax	447 UL-1V	In-line	2	Air	26.8	39.6
3	BMW	R115ORS	Opposed	2	Air + oil	76.3	96.6
4	Subaru	EA81-140	Opposed	4	Liquid	100	140
5	Wilksch	WAM 160	In-line	4	Liquid	120	160
6	Textron-Lycoming	O-320-H	Opposed	4	Air	128	160
7	PZL	F 6A6350-C1	Opposed	6	Air	150	205
8	TCM	Tsio-360-RB	Opposed	6	Air	148.6	220
9	Textron-Lycoming	IO-540-C	Opposed	6	Air	170	250
10	TCM	IO-470-D	Opposed	6	Air	193.3	260
11	Bombardier	V300	Vee		Liquid	210	300
12	TCM	TSIOL-550-C	Opposed	6	Liquid	188.4	350
13	Textron-Lycoming	IO-270-A	Opposed	8	Air	258	400
14	VOKBM	M-9F	Radial	9	Air	214	420
15	Orenda	OE600 Turbo	Vee	8	Liquid	–	750
16	PZL	K-9	Radial	9	Air	580	1 170

ENGINE SELECTION

Example – Propulsion system selection

Universidad Pontificia Bolivariana

Solution:

Required engine power at sea level:

$$P_{max} = P_{max_{SL}} \left(\frac{\rho}{\rho_o}\right)^{1.2} \Rightarrow 1751.6 = P_{max_{SL}} \left(\frac{0.653}{1.225}\right)^{1.2} = P_{max_{SL}}$$

$$\Rightarrow P_{max_{SL}} = \frac{1751.6}{0.47} \Rightarrow P_{max_{SL}} = 3725.73\,\text{kW} = \boxed{4996.3\,\text{hp}}$$

Referring to engine manufacturers' catalogs (e.g., Table 8.12), it is observed that there is no piston engine available in the market that delivers this much power even with three engines. This is another reason that we have chosen the turboprop engine. Moreover, there are very few turboprop engines (e.g., Table 8.13) that generate about 5000 hp; hence, this is another reason for the decision of a multi-engine configuration. There are a number of turboprop engines which deliver about 2500 hp, therefore two turboprop engines are selected for this aircraft.

Table 8.13 Primary specifications for several turboprop engines [8]

No.	Manufacturer	Designation	Arrangement	Airflow (kg/s)	Length (mm)	Width (mm)	Mass (kg)	Max power (hp)
1	Innodyn	255TE	C	–	762	360	85.3	255
2	Rolls Royce	250-B17	6A + C	1.56	1143	483	88.4	420
3	Turbomeca	Arrius 2F	C	–	945	459	103	504
4	P&WC	PT6A-27	3A + C	3.08	1 575	483	149	680
5	Honeywell	TPE331-3	C + C	3.54	1 092	533	161	840
6	PZL	TWD-10B	6A + C	4.58	2 060	555	230	1 011
7	P&WC	PT6A-65b	4A + C	4.31	1 880	483	225	1 100
8	P&WC	PT6A-69	4A + C	–	1 930	483	259.5	1 600
9	GE	CT7-9	5A + C	5.2	2 438	737	365	1 940
10	P&WC	PW123C	C, C	–	2 143	635	450	2 150
11	Klimov	TV3-113VMA-SB2	12A	9	2 860	880	570	2 500
12	DEMC	WJ5E	10A	14.6	2 381	770	720	2 856
13	Rolls Royce	AE 2100C	14A	16.33	2 743	1151	715.8	3 600
14	Progress	AI-20M	10A	20.7	3 096	842	1 040	3 943
15	P&WC	PW150A	3A + C	–	2 423	767	690	5 071
16	EPI	TP400-D6	5A	26.31	3 500	924.5	1 795	11 000

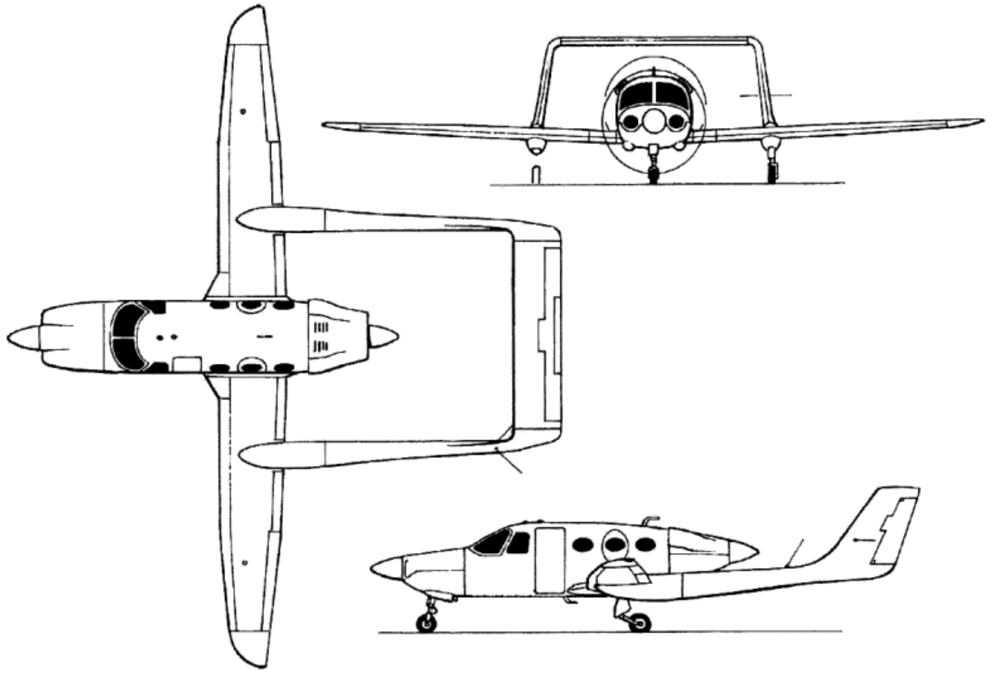
A: axial stage, C: centrifugal stage, C, C: two stages on different shafts.

ENGINE SELECTION

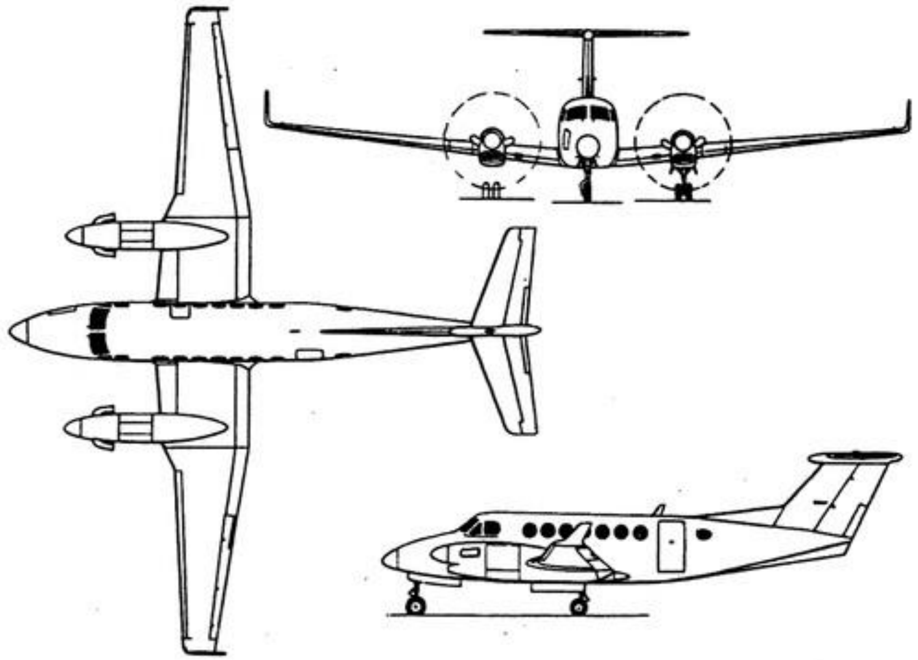
Example – Propulsion system selection

Solution:

4. **Engine locations**. For the case of two engines, both engines must be placed such that the locations satisfy the symmetry requirement. To satisfy this requirement, there are mainly two options. One option is to place an engine in the fuselage nose and another option is to place an engine in the rear fuselage, both along the fuselage center line. The one pusher and one tractor configuration is not a practical and viable alternative for a civil transport aircraft, since one engine obstructs the pilot view and the other engine interferes with the conventional tail.



a twin-engine configuration, positioning the prop-driven engine on the wing (with propeller in front of the wing) often results in the most attractive design from a structural and aerodynamic point of view. Based on these reasons and advantages, both engines are placed on the wing with propellers in front of the wing. Other advantages of locating twin engines on the wing are introduced in Section 8.5. The exact distance between each engine and the fuselage will be determined based on the prop clearance and aerodynamic interference considerations.



ENGINE SELECTION

Example – Propulsion system selection

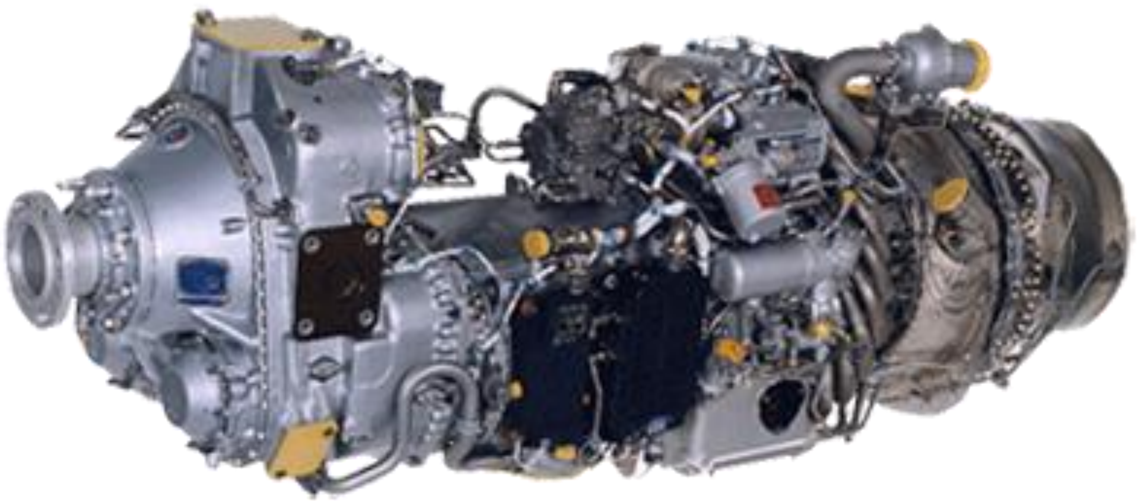
Universidad Pontificia Bolivariana

Fundada en 1936

Solution:

5. **Engine selection from manufacturers' catalogs.** Most turboprop engine manufacturers fabricate engines with a rated power of 2500 hp. Two turboprop engines from Pratt & Whitney are selected with the following features:
designation PW 127; arrangement C, C; prop drive free turbine; length 2134 mm; width 600 mm; dry mass 481 kg; and TO rating 2750 shp

As observed, the engines deliver slightly higher shaft powers. We intentionally selected a more powerful engine for future design precautions and considerations.



Source; Pratt & Whitney PW 127

6. **Propeller design.** The engine power at 20 000 ft is:

$$P_{\max} = P_{\max_{SL}} \left(\frac{\rho}{\rho_o}\right)^{1.2} = 2750 \left(\frac{0.653}{1.225}\right)^{1.2} = 1293\ \text{hp} = 964137\ \text{W}$$

A two-blade propeller with a lift coefficient of 0.3, a prop efficiency of 0.75, and a prop aspect ratio of 9 is adopted. Two regular metal propellers are selected. According to Table 8.9, the suggested tip speed is less than 270 m/s.

Table 8.9 Suggested propeller cruise tip speed limit

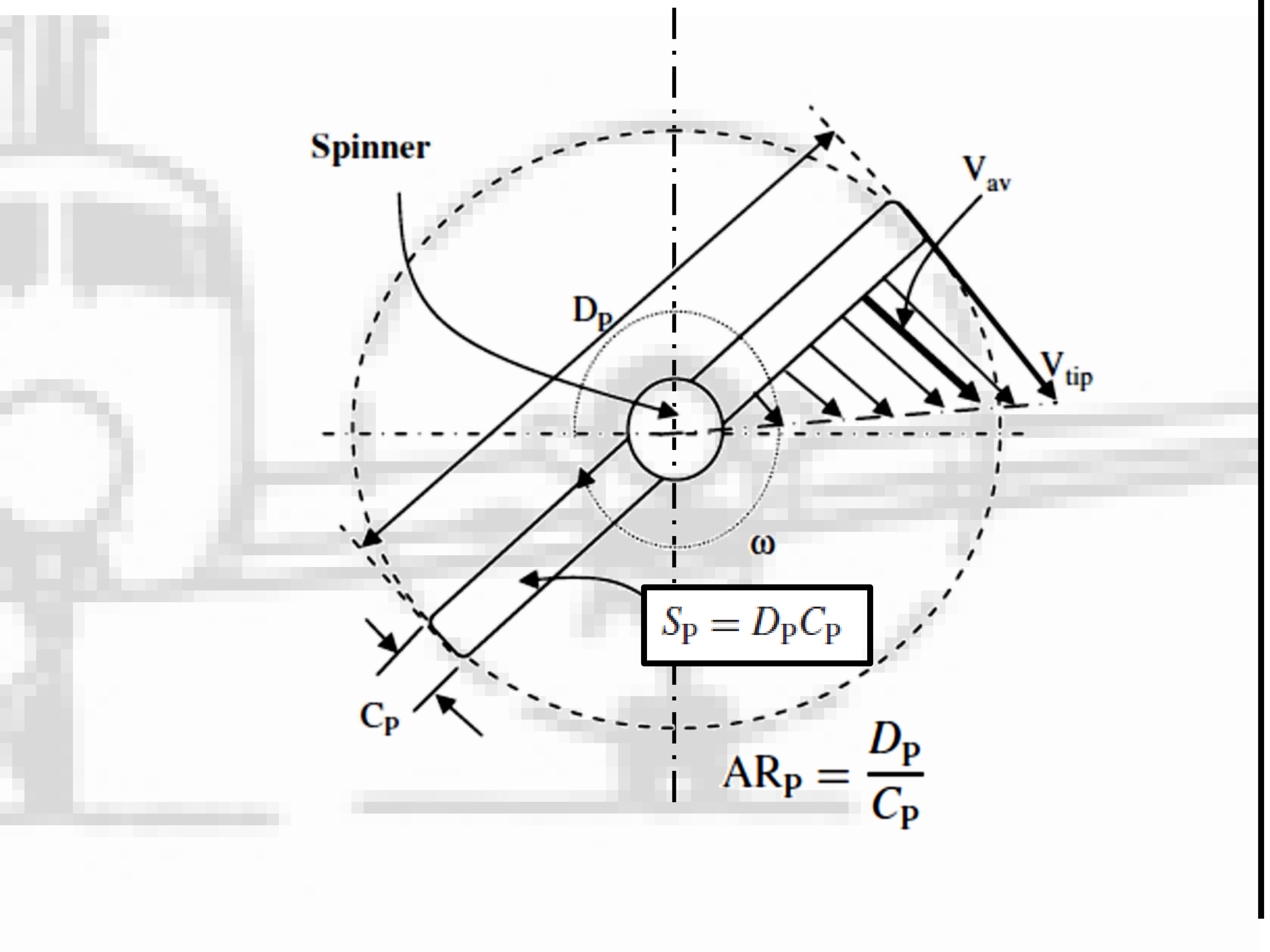
No.	Tip speed limit (m/s)	Propeller type
1	310	Metal high-performance prop
2	270	Metal regular prop
3	250	Composite prop
4	210	Wooden prop
5	150	Plastic prop for RC model aircraft

ENGINE SELECTION

Example – Propulsion system selection

Solution:

6. **Propeller design.**



we can obtain the prop diameter as follows:

$$D_P = K_{np}\sqrt{\frac{2P\,\eta_P AR_P}{\rho\left(0.7V_{tip_{cruise}}\right)^2 C_{L_P} V_C}}$$

$$= 1\cdot\sqrt{\frac{2\cdot 964137\cdot 0.75\cdot 9}{0.653\cdot(0.7\cdot 270)^2\cdot 0.3\cdot(320\cdot 0.514)}} \Rightarrow \boxed{D_P = 3.361\text{ m}}$$

So two propellers with a diameter of 3.361 m are needed.
Propeller rotational speed:

$$V_{tip_{cruise}} = \sqrt{V^2_{tip_{static}} + V_C^2} \Rightarrow V_{tip_{static}} = \sqrt{V^2_{tip_{cruise}} - V_C^2} = \sqrt{270^2 - (320\cdot 0.514)^2}$$

$$\Rightarrow \boxed{V_{tip_{static}} = 214\text{ m/s}}$$

$$V_{tip_{static}} = \frac{D_P}{2}\omega \Rightarrow \omega = \frac{2V_{tip_{static}}}{D_P} = \frac{2\cdot 214}{3.361} = 127.35\text{ rad/s}$$

$$\omega = \frac{2\pi\cdot n}{60} \Rightarrow n = \frac{60\omega}{2\pi} = \frac{60\cdot 127.35}{2\cdot 3.14} = \boxed{1216.1\text{ rpm}}$$

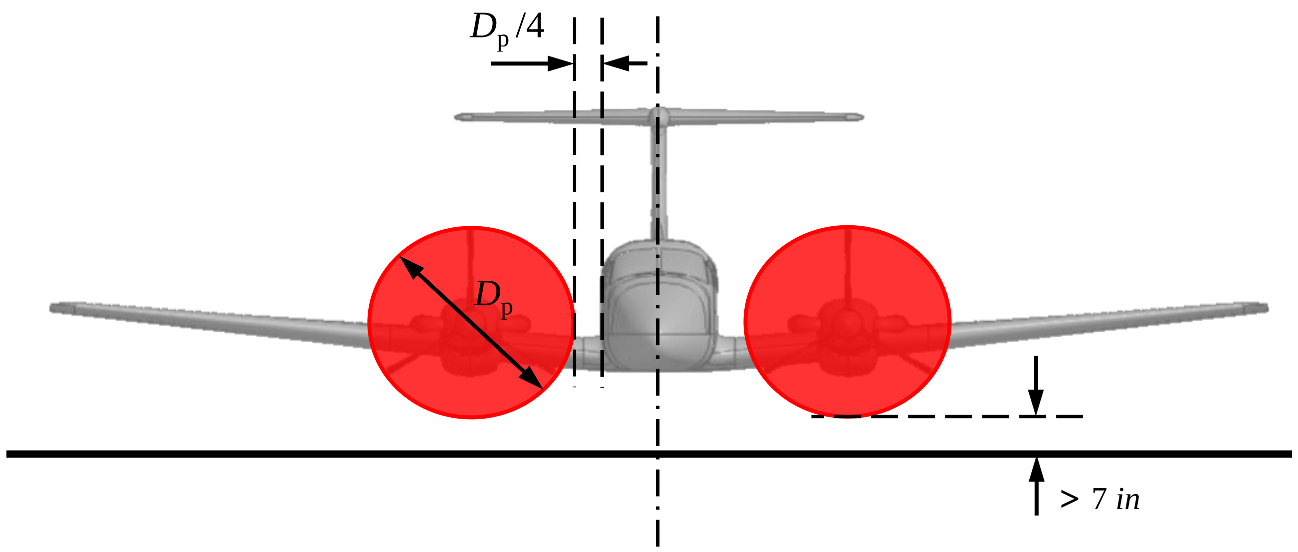
So a gearbox must reduce the engine shaft revolution to 1216.1 rpm

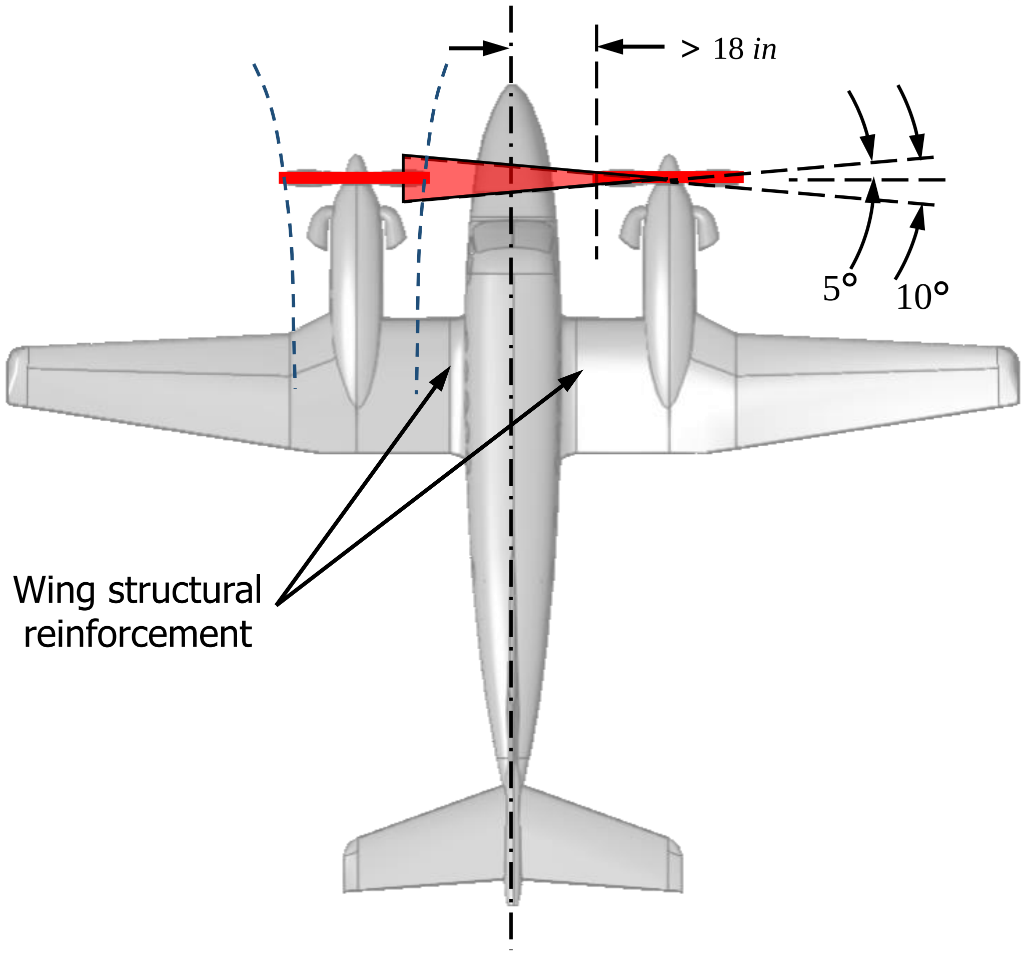
ENGINE SELECTION

Example – Propulsion system selection

Solution:

7. **Engine Installation**. The aircraft weight and type place the aircraft under FAR 23 airworthiness regulations. FAR Part 23 Section 23.771 requires that each pilot compartment must be located with respect to the propellers so that no part of the pilot or the controls lies in the region between the plane of rotation of any inboard propeller and the surface generated by a line passing through the center of the propeller hub making an angle of 5° forward or aft of the plane of rotation of the propeller.





ELECTRIC AVIATION

Electric aviation – key factors.

Energetically efficient airplanes – lighter weight, improvements in aerodynamics.

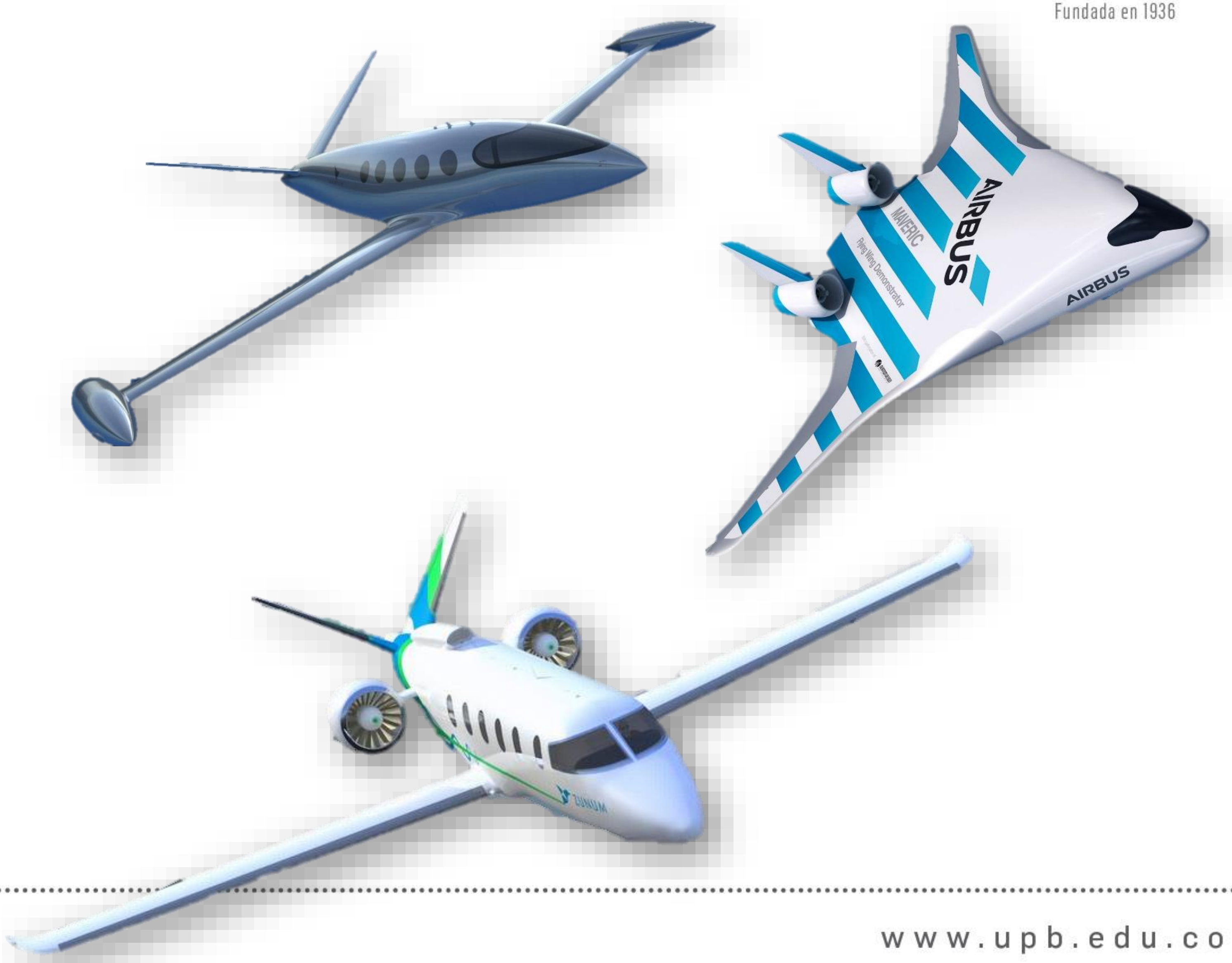
Batteries with a higher energy density.

Powerful and lighter electric motors and controllers.

High fossil fuel prices ($$).

Public consciousness for the reduction of CO_2.

Support of aeronautical authorities and governments.



Universidad Pontificia Bolivariana
Fundada en 1936

ELECTRIC AVIATION

Universidad Pontificia Bolivariana
Fundada en 1936

Electric aviation – Pros and cons.

Pros.

Ecology: No noxious gas emissions to the environment.

Quieter: Less sound emissions.

Easier operation.

Easier maintenance.

Electricity is more economical than gas (petrol).

Less operational and maintenance costs (cheaper).

Cons.

Ecology: electricity must come from renewable sources.

Performance: low endurance and range.

Battery – low life-cycles.

Still a growing technology.

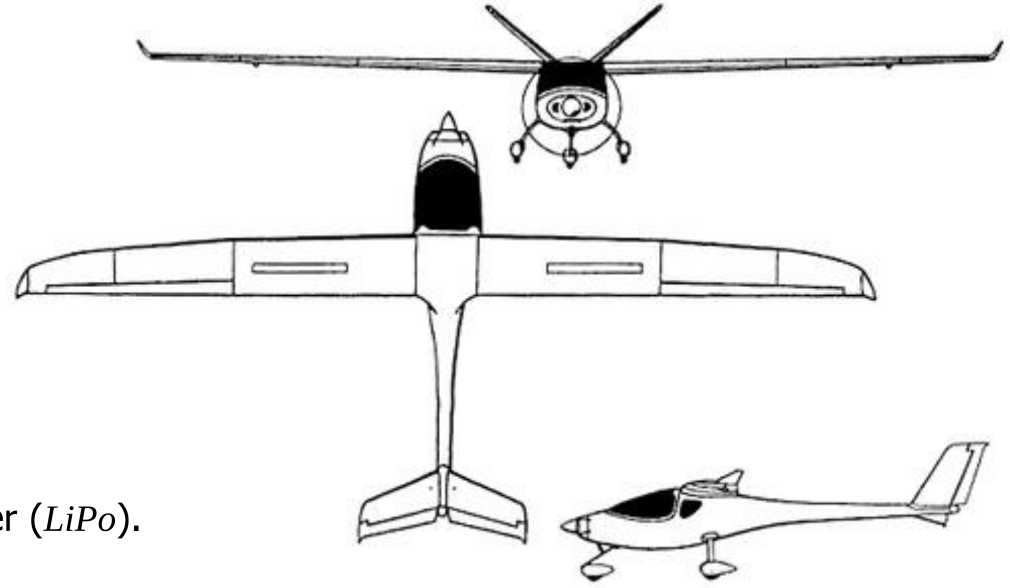
Lack of infrastructure (re-charge stations).

Limitations in bigger airplanes (number of batteries).

ELECTRIC AVIATION

They are becoming an important trend in the light aircraft industry, particularly in light sports aircraft (*LSA*).

Electric motors, capable of delivering power in excess of 80 *HP* (60 *kW*), are now being used to power aircraft that can carry as many as four people.



***Yuneec International E*430 Specs:**

General characteristics:

- Crew: 2 seat.
- Length: 6.98 *m* (22 *ft* 11 *in*).
- Wingspan: 13.8 *m* (45 *ft* 3 *in*).
- Wing area: 11.37 m^2 (122.4 *sq ft*).
- Empty weight: 250 *kg* (551 *lb*) with batteries.
- Gross weight: 470 *kg* (1036 *lb*).
- Powerplant: 1 × *Yuneec Power Drive* 40, powered by *Yuneec OEM* 5-pack Lithium Polymer (*LiPo*). batteries, 13 *kg* (28.6 *lbs*), 66.6*V* (30 *Ah*) each , 40 *kW* (54 *hp*).
- Propellers: 2-bladed fixed pitch.

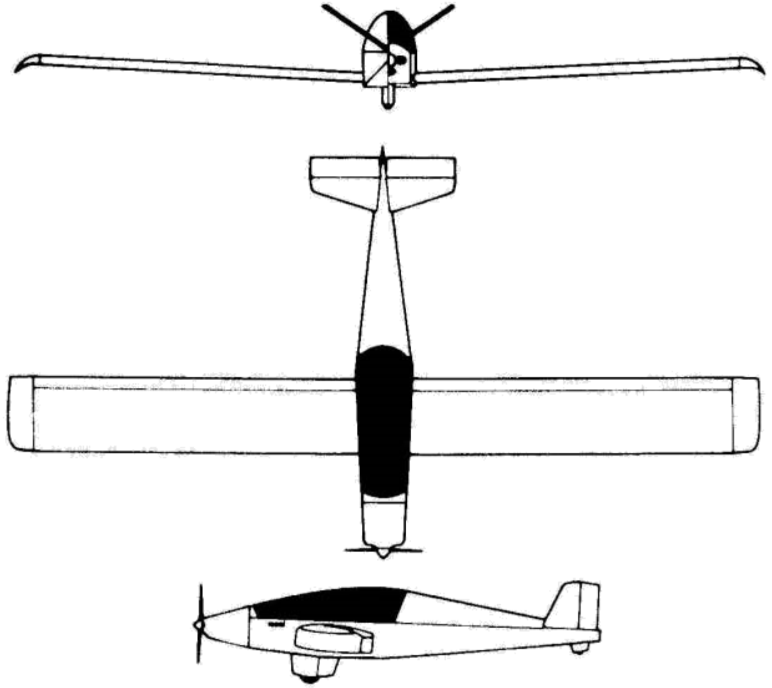
Performance:

- Maximum speed: 150 *km/h* (93 *mph*, 81 *knots*).
- Cruise speed: 90 *km/h* (56 *mph*, 49 *knots*).
- Stall speed: 70 *km/h* (43 *mph*, 38 *knots*).
- Endurance: up to 2 *hours* @ V_{cruise} = 53 *knots*.
- Range: 227 km (141 *mi*, 123 *nmi*).
- Maximum glide ratio: 25:1.
- Rate of climb: 3.5 *m/s* (690 *ft/min*).
- Wing loading: 41.3 kg/m^2 (8.5 *lb/sq ft*).

ELECTRIC AVIATION

They are becoming an important trend in the light aircraft industry, particularly in light sports aircraft (*LSA*).

Electric motors, capable of delivering power in excess of 80 *HP* (60 *kW*), are now being used to power aircraft that can carry as many as four people.



***ElectraFlyer-C* Specs:**

General characteristics:
- Original design: *Monnett Moni Motorglider.*
- Crew: 1 seat.
- Height: 45 *ft* 7 *in* (13.9 *m*).
- Wingspan: 13.9 *m* (45.6 *ft*).
- Wing area: 14.6 m^2 (157 *sq ft*).
- Empty weight: 172 *kg* (380 *lb*) with batteries.
- Gross weight: 283 *kg* (625 *lb*).
- Powerplant: 1 × *ElectraFlyer Direct Drive electric aircraft engine*, powered by a pair of 5.6 *kWh.* Lithium-Ion (*LiIon*) battery packs, 13.0 *kW* (18 *hp*).
- Propellers: 2-bladed fixed pitch.

Performance:
- Maximum speed: 140 *km/h* (90 *mph*, 78 *knots*).
- Cruise speed: 110 *km/h* (70 *mph*, 61 *knots*).
- Stall speed: 65 *km/h* (40 *mph*, 35 *knots*).
- Endurance: up to 1.5 *hours.*

ELECTRIC AVIATION

They are becoming an important trend in the light aircraft industry, particularly in light sports aircraft (*LSA*).

Electric motors, capable of delivering power in excess of 80 *HP* (60 *kW*), are now being used to power aircraft that can carry as many as four people.

Pipistrel Alpha Electro Specs:

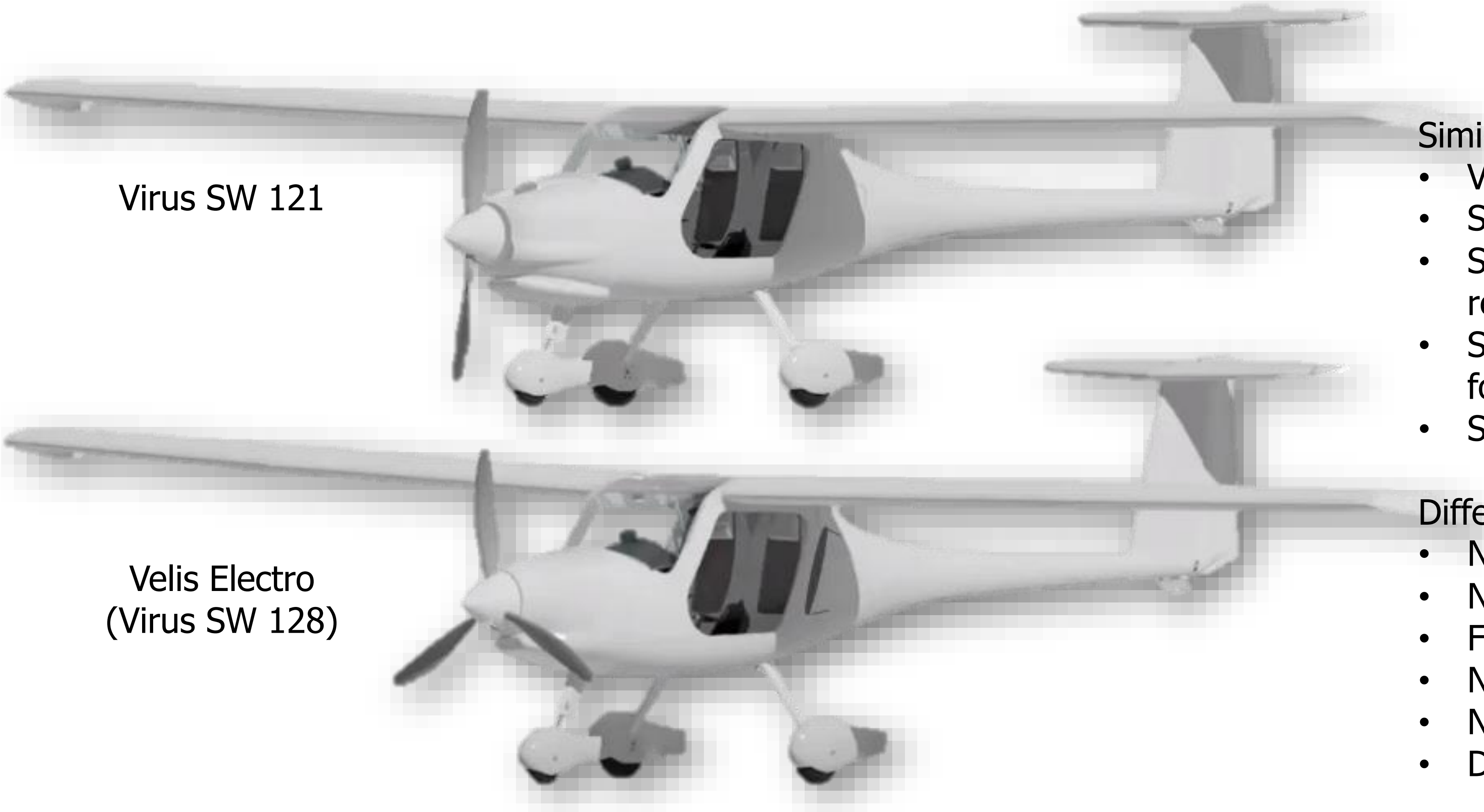
General characteristics:

- Original design: *Pipistrel Alpha Trainer.*
- Crew: 2 seat.
- Length: 6.5 *m* (21 *ft* 4 *in*).
- Wingspan: 10.5 *m* (45 *ft* 3 *in*).
- Wing area: 9.51 m^2 (102.4 *sq ft*).
- Empty weight: 428 *kg* with batteries.
- Gross weight: 600 *kg.*
- Powerplant: 1 × water-cooled *Pipistrel E*-811 58 kW, *Cruise* 50 *kW*@2100-2400 *rpm* powered by 2 x 12 *kWh LiPo* batteries (63 *kg* each).
- Propellers: Fixed pitch *P*-182 3-bladed.

Performance:

- Maximum speed: 108 *KIAS.*
- Cruise speed (75% power): 100 *KIAS.*
- Stall speed: 45 *KCAS* (no flaps), 38 *KCAS* (with flaps).
- Endurance: up to 50 minutes (plus reserve).
- Range: 75 *nmi.*
- Rate of climb: 1220 *ft/min.*
- Service ceiling: 12000 *ft.*

ELECTRIC AVIATION



Virus SW 121

Velis Electro
(Virus SW 128)

Similarities with the SW 121:
- VFR-Day ops.
- Same MTOW ($600\ kg$)
- Same airframe (less empennage reinforcement), geometry, areas
- Same cg (longitudinal – lower cg loc. better for turn), similar inertia
- Same control system

Differences with the SW 121:
- No negative flaps ($-5\ deg$)
- No air brakes (spoilers)
- Fixed pitch propeller
- No power losses with altitude
- No fuel management
- Different cooling system (glycol-water)

ELECTRIC AVIATION

Universidad Pontificia Bolivariana

Fundada en 1936

Certification issues – problems:

Change in regulations to adapt this type of propulsion system which means that:

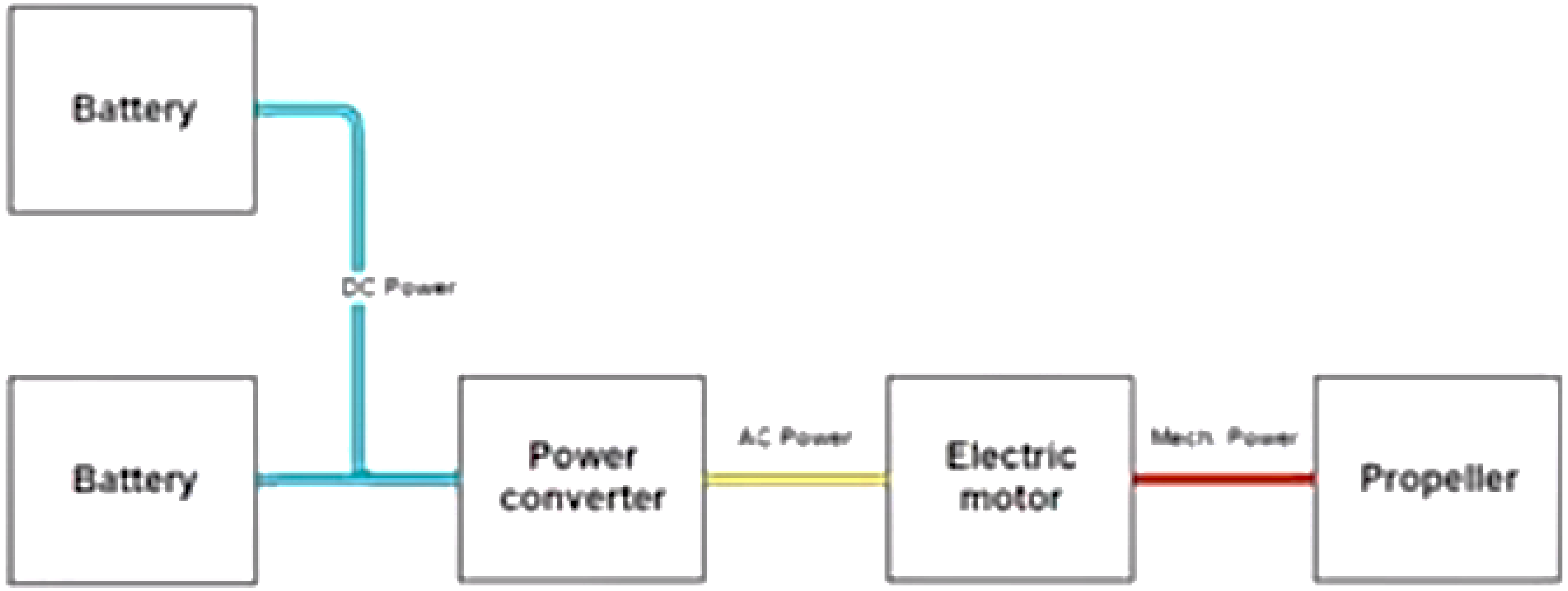
- Operations
- Airfields – adaptation
- Pilot licenses
- Maintenance

Regulations related with electric aviation:

- EASA: SC-VTOL/CS_LSA
- ASTM F44.40: Powerplant (F32.39 – Electric)
- ASTM F39,05: Design, Alteration, and Certification of electric Propulsion Systems
- SAE AE7-D: Aircraft Energy Storage and Charging Committee

ELECTRIC AVIATION

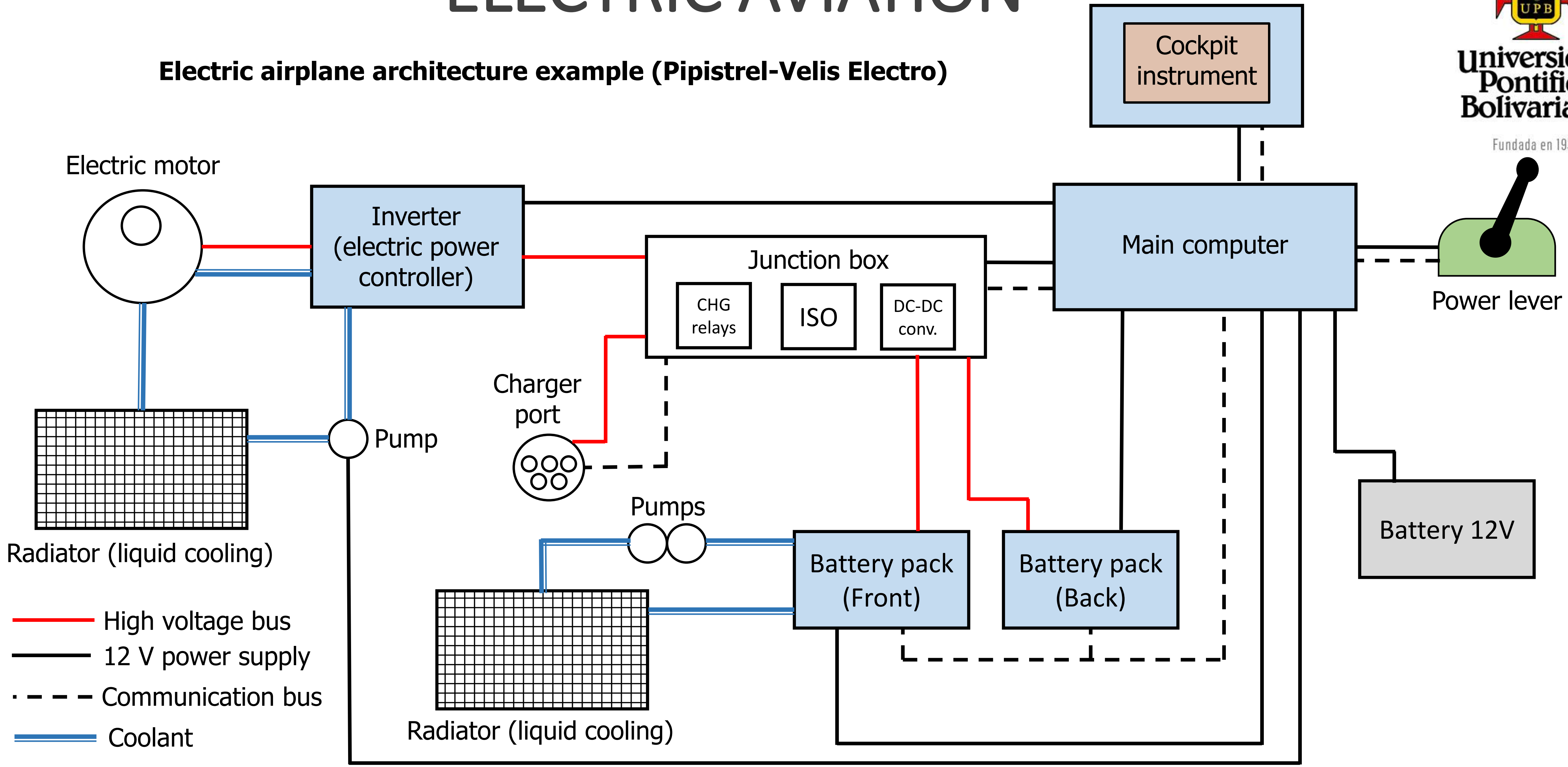
Electric airplane architecture (simplify)



Inverter – converts DC into AC
From the main computer – regulates the electric power given to the motor

ELECTRIC AVIATION

Electric airplane architecture example (Pipistrel-Velis Electro)



ELECTRIC AVIATION

Table 1: Company All-Electric and Hybrid Prototypes

Companies/ EA Models	People/ Passengers	Range (miles)	Autonomous	Vertical Landing and Takeoff	Hybrid
Airbus E-Fan X	1	50	No	No	Yes
Ampaire Electric	6	750	No	No	Yes
Amp. TailWind E	9	350	No	No	No
Amp. TailWind H	9	750	No	No	Yes
Astro ELROY	2	18	No	Yes	No
Bell NeXt	4	150	No	Yes	Yes
Boeing PAV	2	50	Yes	Yes	No
City Airbus 2.2	4	60	No	Yes	No
DeLorean DR7	2	120	No	Yes	No
Dubai Air Taxi	2	31	Yes	Yes	No
Elevation Alice	9	650	No	No	No
Kitty Hawk Cora	2	110	No	Yes	No
Kitty Hawk Flyer	1	20	No	Yes	Yes
Lilium jet	5	186	No	Yes	No
Pipistrel Alpha El.	2	80	No	No	No
SureFly	2	70	No	Yes	Yes
UberAir	5	43.3	No	Yes	No
Wright Electric	150	335	No	No	No
Zunum Aero	12	700	No	No	Yes

ELECTRIC AVIATION

Batteries are devices that convert chemical energy into electrical energy (or the other way around).

There are two basic types of batteries:

- **Primary batteries** are "single-use" and cannot be recharged. Dry cells and (most) alkaline batteries are examples of primary batteries.
- **Secondary batteries** type are the rechargeable ones.

Several issues concern batteries and must be kept in mind:

1. The energy density of current battery technology is very low compared to gasoline. Therefore, a greater battery power requires many batteries (much weight).

2. Energy durability – shelf life. There is more to battery capability than just energy density. For instance, $LiCO_2$ batteries have a higher initial energy density than $LiFePO_4$ batteries. However, after a year of frequent recharge-discharge cycles, the $LiFePO_4$ battery has a similar residual energy density to the $LiCO_2$ battery.

3. Discharge voltage depends on the remaining charge and battery temperature. The initial discharge voltage is usually high but diminishes with the energy used. A fully charged battery yields a reported Take-Off distance for aircraft, but the first touch-n-go requires a much longer runway. This is not acceptable for aircraft transporting people for commercial purposes.

4. The current battery technology poses fire hazards.

ELECTRIC AVIATION

Batteries: The ideal battery for use in airplanes should be light, rechargeable, have a long durability, and with the highest energy density possible.

Currently, two types of batteries are suitable for use in aircraft: $LiFePO_4$ and $LiCoO_2$. Both have their pros and cons.

TABLE 7-12 Common Battery Types

Battery type	Comment
Lead-acid	Best known as the car battery. Low energy option not suited for use in aircraft.
NiCad	Used to be popular for radio-controlled aircraft. Largely obsolete.
NiMH	Popular as rechargeable batteries for robots.
Li-Ion	Lithium-ion battery, best known as battery packs for laptop computers.
LiPo	Current power packages for electric aircraft.

Table 20.1 Battery Specific Energy & Density

	Chemistry	Typical Values (Wh/kg)	Typical Values (Wh/L)	Name	Notes
old	Lead-acid	45	100	Lead acid	automotive
	Alkaline	100	300	Alkaline	flashlights
Nickel	NiFe	25	30	Nickel Iron	locomotives, mining
	NiCd	60	150	Nickel Cadmium	classic "NiCad"
	NiH	75	60	Nickel Hydrogen	space probes
	NiMH	90	300	Nickel Metal Hydride	replaced NiCad
	NiZn	100	280	Nickel Zinc	automobile, electronics
Li-ion[1]	Li-ion	100–265	250–700	Lithium ion	generic term
	Li-ion Polymer	100–265	250–730	Lithium Polymer	polymer electrolyte
	LiCo02	200	—	Lithium Cobalt Oxide	handheld electronics
	LiFePO4	120	170	Lithium Iron Phosphate	tools, vehicles
	LiMn2O4	150	—	Lithium Manganese Oxide	laptops, medical equip
	LiNiMnCoO2	260	500	Lithium Nickel Manganese Cobalt Oxide (NMC)	aircraft, road vehicles
	LiS	400	250	Lithium Sulfur	aircraft, road vehicles
	LiS (2020)	500	1000	Licerion[2] (LiS)	aircraft, road vehicles
	Li titanate	90	170	Lithium Titanate	high power/low energy
	Li-air	600	200	Lithium-Air	experimental
misc	Na-ion	150	50	Sodium Ion	laptops, bikes
	Molten salt	220	290	Molten salt	
	Silver Zinc	200	700	Silver Zinc	laptops, hearing aids
Comparisons	Wood	4500	3600	Wood	it floats
	Coal	8000	10000	Coal	it smells
	Jet Fuel	11000	10000	Jet Fuel	love that smell
	Gasoline	12000	9000	Gasoline	too expensive
	LH2	39406	2790	Liquid Hydrogen	too cold
	Uranium	2.2E + 10	4.3E + 11	Uranium	too scary
	Antimatter (c^2)	9.0E + 10		Antimatter	beam me up

[1] Lithium-ion is a generic term for various batteries in which lithium ions move to the positive electrode during discharge.
[2] Licerion is Scion Power's trade name for its patented rechargeable lithium sulfur battery planned to enter production in late 2018.

ELECTRIC AVIATION

Battery pack: A set of identical batteries or individual battery cells configured in series (S), parallel (P), or a mixture of both to deliver the desired voltage, capacity, or power density.

The components include the individual batteries or cells, and the interconnects that provide electrical conductivity.

Rechargeable packs contain a temperature sensor (used by the charger to detect the end of the charging).

Battery pack

Battery pack

Battery management system

ELECTRIC AVIATION

Universidad Pontificia Bolivariana
Fundada en 1936

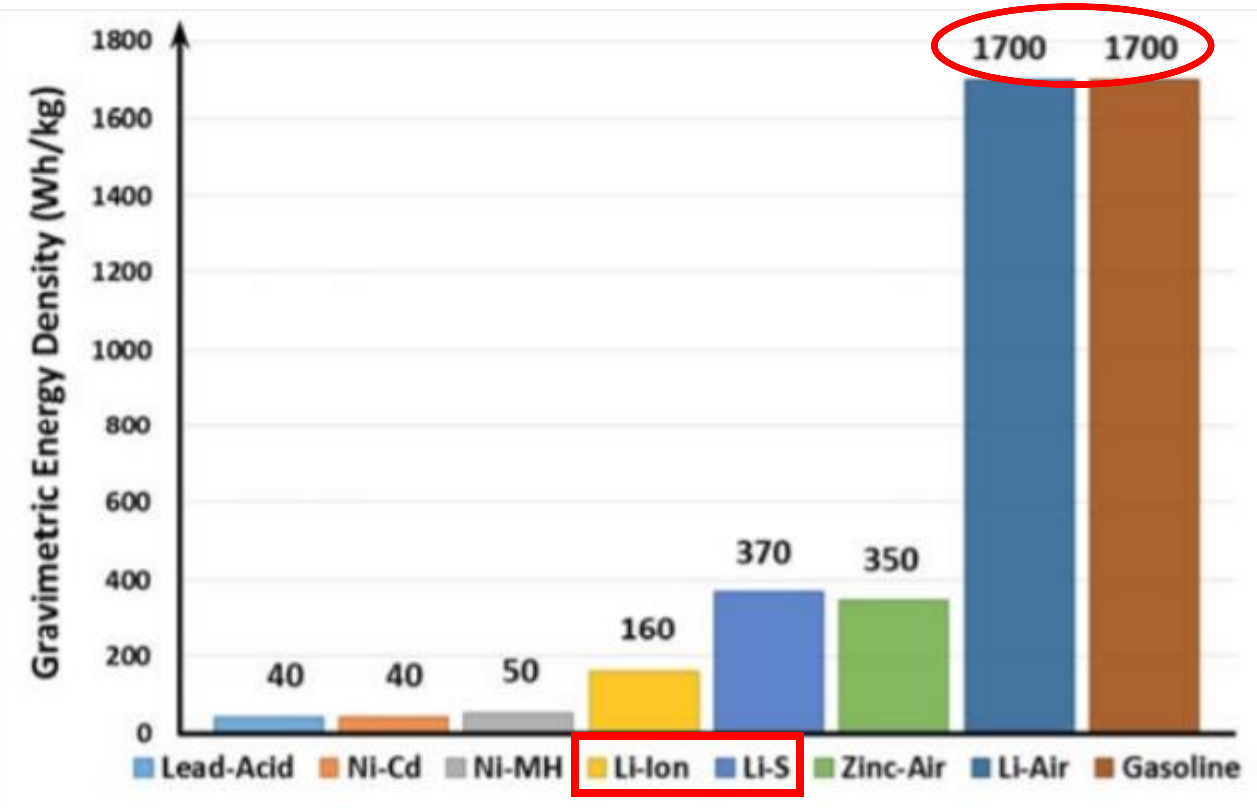
Energy density (Capacity) is the energy stored in a battery's unit weight. It is denoted by E_{BATT} and is typically given in terms of Wh/kg.

The energy density of even the best batteries (LiPo) is substantially lower than that of fossil fuels, about 60 times less!

For example, a $5.6\ kWh$ battery can deliver $5600\ W$ over a period of $1.0\ hour$.

Battery problems:

- Li-Ion batteries have 4 times more capacity than Ni-Cd or Pb.
- Gas (petrol) has around 10 times more energy density than Li-Ion.
- Li-Air – developing technology that will let to reach gasoline ED values (so far not-rechargeable).
- Stability of its chemical components (safety).



ELECTRIC AVIATION

C-rate: Is the measure of the battery charge/discharge rate (how fast).

1C – means that a fully charged battery rated at $1Ah$ should provide $1A$ for one hour (charge/discharge all its capacity in one hour). The same battery discharging at 0.5C should provide $500\ mA$ for two hours, and at 2C, it delivers $2A$ for $30\ minutes$.

Battery problems:

- Great limitation for airplanes.
- Power will change during different flight phases – need for a higher C-rate battery, then, battery degrades faster (need constant replacement).

C-rate	Time
5C	12 min
2C	30 min
1C	**1h**
0.5C or C/2	2h
0.2C or C/5	5h
0.1C or C/10	10h
0.05C or C/20	20h

C-rate and service times when charging and discharging batteries of $1Ah$ ($1000mAh$).

ELECTRIC AVIATION

Universidad Pontificia Bolivariana
Fundada en 1936

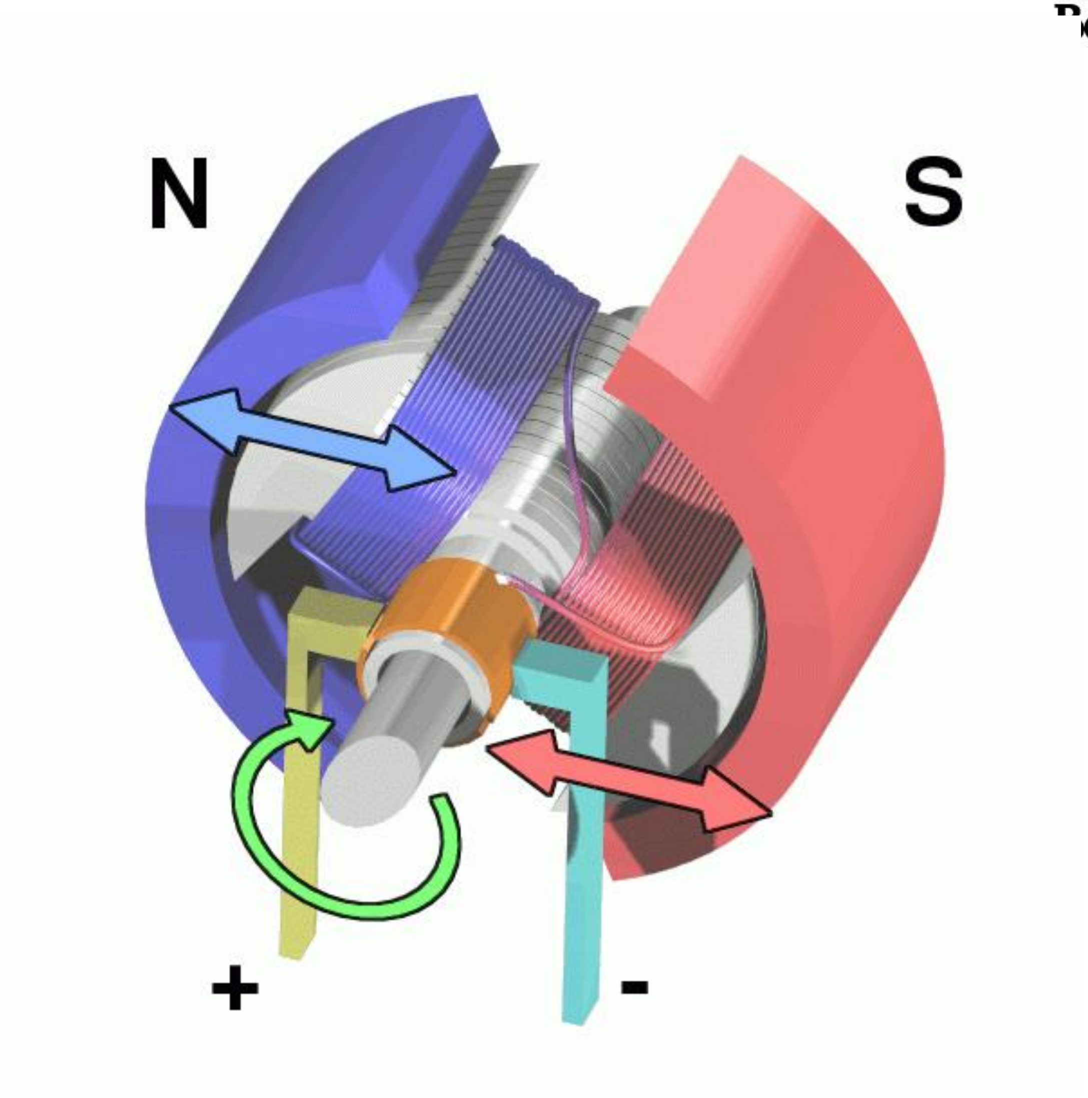
Electric motor (EM): Compared to the internal combustion one, electric motors are quieter, smoother, highly reliable, and low-weight (including the controller).

EM consists of a nonmoving magnetic object (stator) rigged to pull a rotating magnetic thing (rotor) in a circular path.

While EM is classed as DC or AC by the current type, they can accept that inside, they are all AC (current flow must alternate between reversing the magnetic polarity to keep the rotor spinning around).

Modern “brushless” motors use an electronic circuit to sense the rotor position and switch the current flow, improving efficiency and eliminating the brushes’ high potential of sparking.

Electric motors are quieter, smoother, highly reliable, and low-weight.

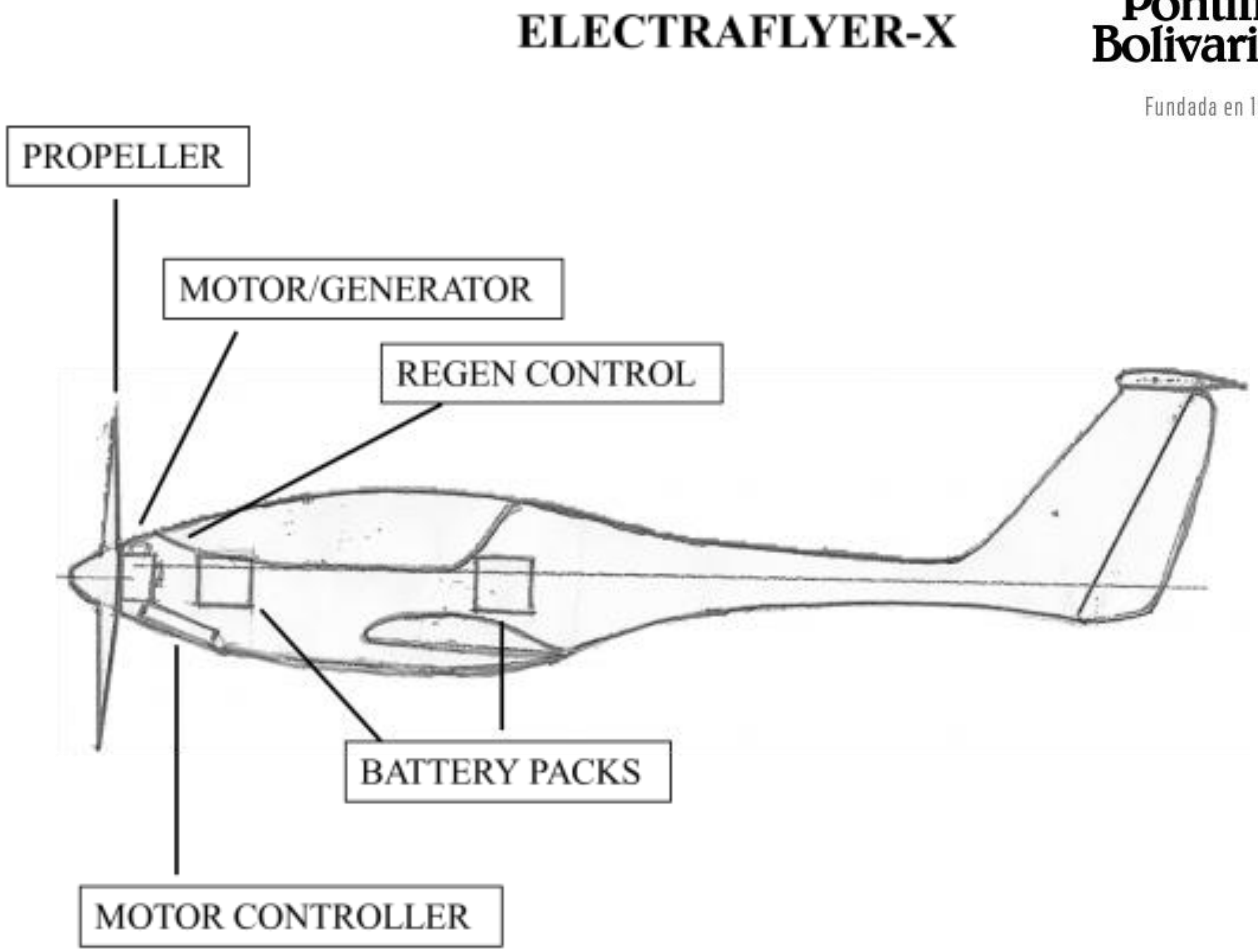


ELECTRIC AVIATION

Electric motor (EM) in aviation: Permanent magnet motors are more efficient than coil (electromagnet) motors.

AC motors require AC power, whereas batteries and solar cells use direct current (DC), so a converter of some sort (motor controller) is required.

Brushless motors are more efficient and reliable, but they require sophisticated controllers to switch the current electronically at the appropriate time, so they cost a bit.



ELECTRIC AVIATION

Table 8.11 Primary specifications for several electric engines [8]

No.	Manufacturer	Designation	Length (mm)	Diameter (mm)	Mass (kg)	Maximum current (A)	Max power (kW)
1	Hacker	A20-26M EVO	28	28	0.042	12 A; 1130 rpm/V	0.150
2	Raiden	T30A	42.7	60	0.271	58	0.400
3	Applied Motion	M1500-232-7-000	190	100	5.7	9.5	1.5
4	Leopard	LBP4074	40	38	0.347	120 A; 2000 rpm/V	2.6
5	Yuneec	Power drive 10	–	160	4.54	180	10
6	Electroavia	GMPE 102 Devoluy	200	210	11.57	250	19.4
7	Electroavia	GMPE 201 Arambre	200	210	12	275	32
8	Yuneec	Power drive 40	–	240	17	285	40

Universidad Pontificia Bolivariana

Fundada en 1936

The following expressions are helpful when solving various problems that involve electrical power.

$$\text{Voltage:}\quad V = \begin{cases} I \times R \\ P/I \\ \sqrt{P \times R} \end{cases} \quad \text{Volts}$$

$$\text{Current:}\quad I = \begin{cases} \sqrt{P/R} \\ P/V \\ V/R \end{cases} \quad \text{Amps}$$

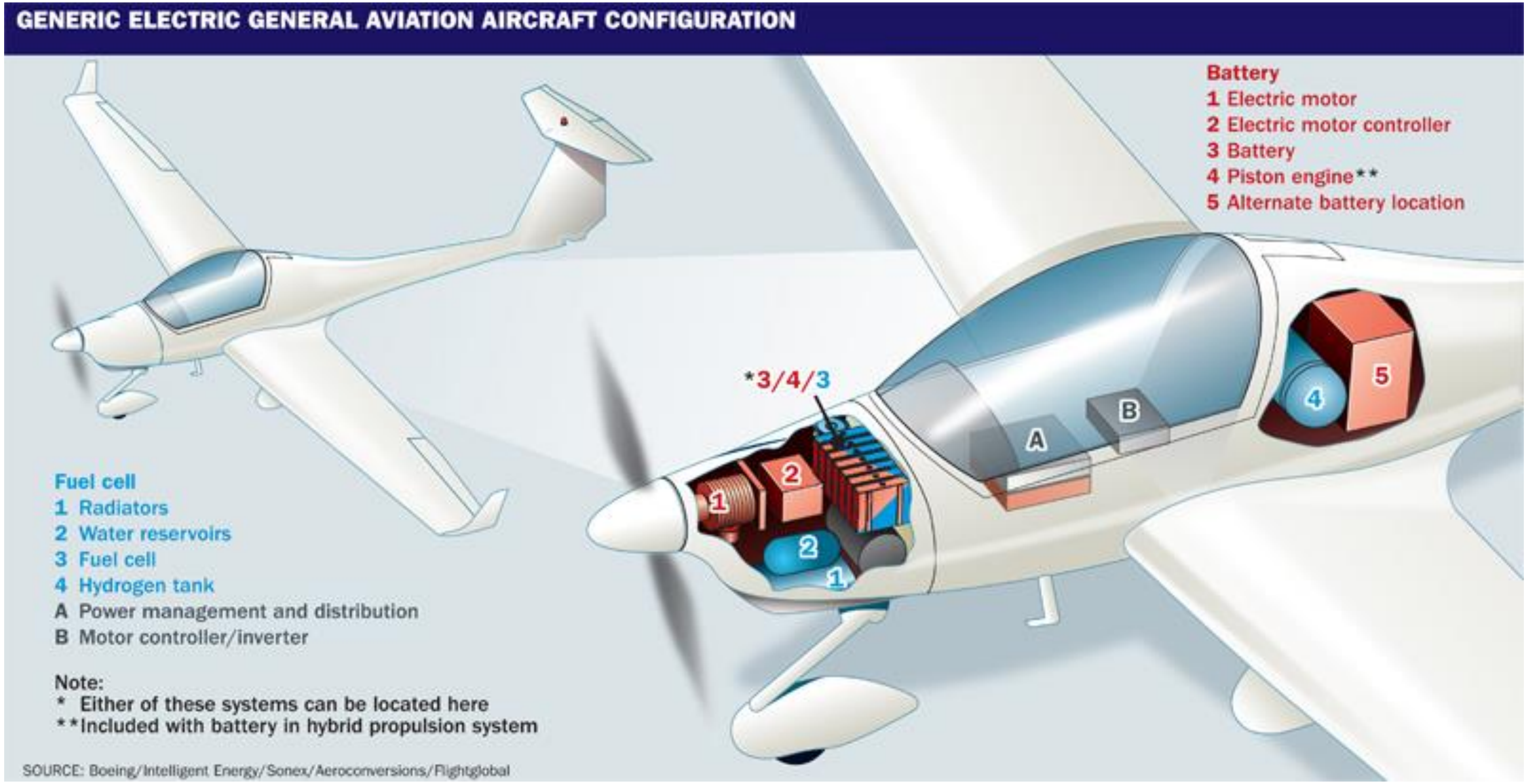
$$\text{Resistance:}\quad R = \begin{cases} V/I \\ V^2/P \\ P/I^2 \end{cases} \quad \text{Ohms}$$

$$\text{Power:}\quad P = \begin{cases} V^2/R \\ R \times I^2 \\ V \times I \end{cases} \quad \text{Watts}$$

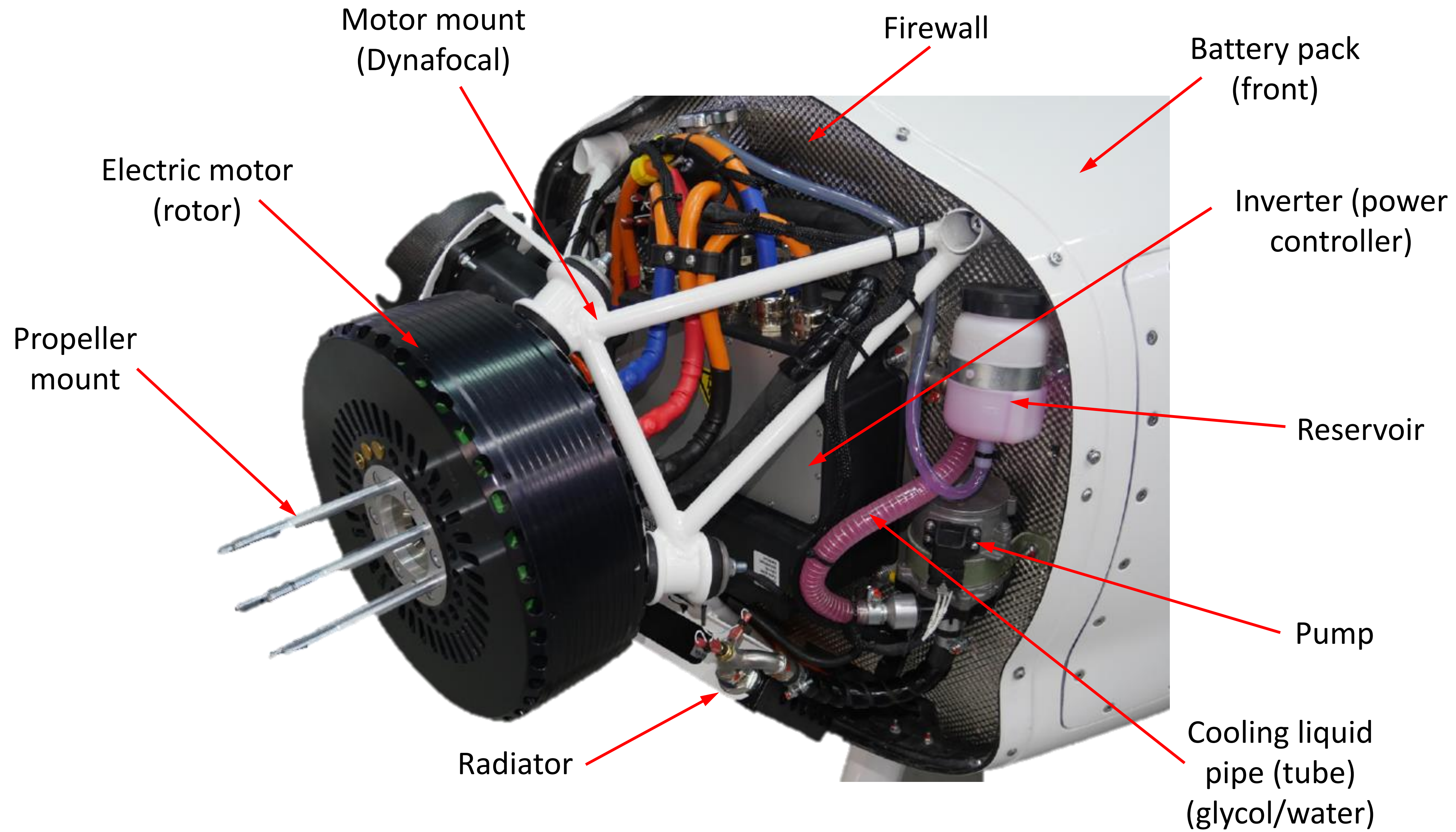
where

I = current (amps)
P = power (watts)
R = resistance (ohms)
V = voltage (volts)

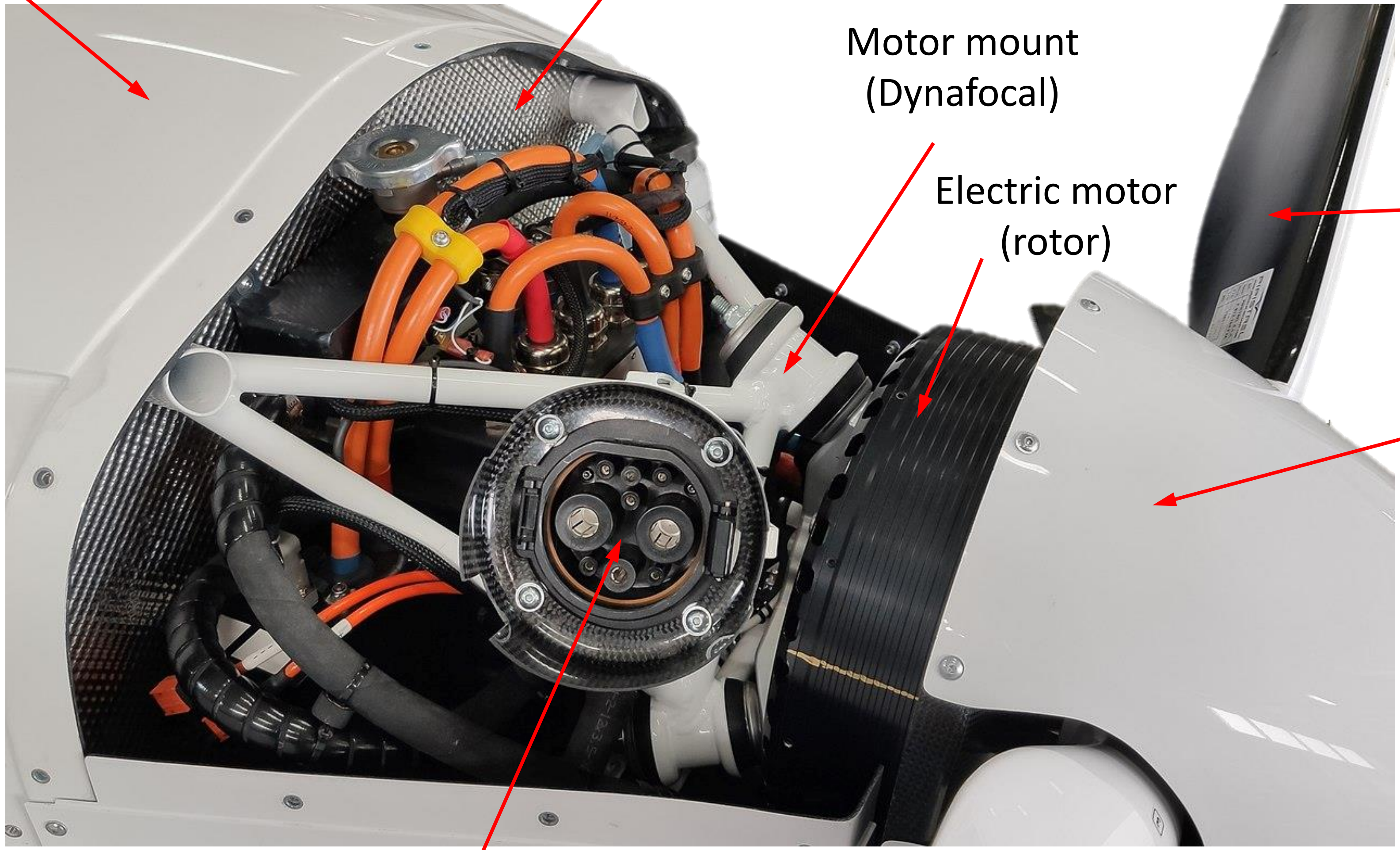
ELECTRIC AVIATION



ELECTRIC AVIATION



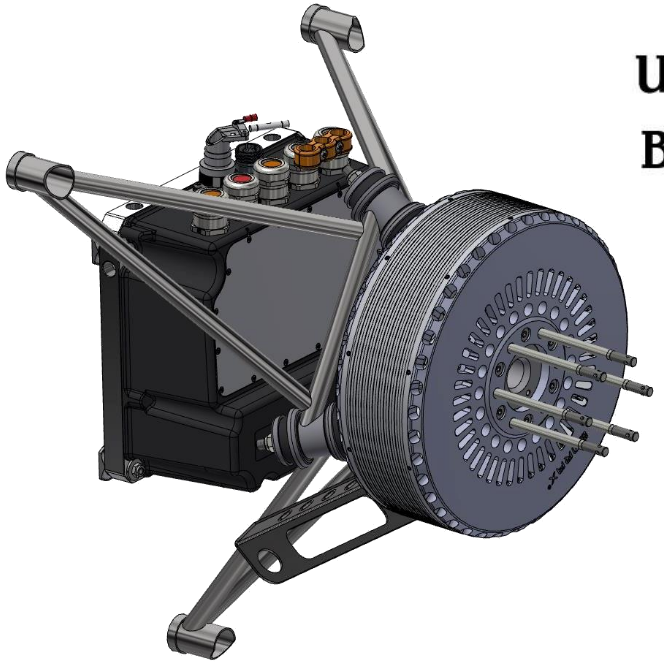
ELECTRIC AVIATION



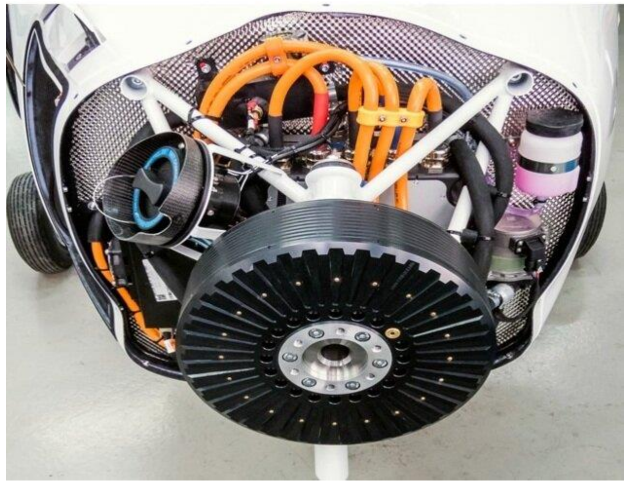
ELECTRIC AVIATION

Example EM – EMRAX 268 (used by Velis Electro/Pipistrel)




- Electric synchronous 3-phase.
- Permanent magnet, outrunner type.
- Rated power $60\ kW$ (up to $75\ kW$).
- Max. temperature $90\ °C$.
- Weight: $20\ kg$.
- Liquid cooled (glycol/water).



ELECTRIC AVIATION

Example Controller – H300A power controller (used by Velis Electro/Pipistrel)

Located at the firewall (high-temperature operation).

Specs:
- $180\ kW$ peak motor power, $120\ kW$ continuous power.
- HV supply ranges from 100 to $450\ Vdc$.
- LV supply range $10\text{-}30\ Vdc$.
- Continuous motor current $300\ Arms$, $1\text{-}minute$ peak $450\ Arms$.
- CAN communication interface.
- Operating temperature (coolant) $-20\text{-}65\ °C$.
- Volume: $230x245x126\ mm$.
- Weight: $7\ kg$.
- IP65 protection.

ELECTRIC AVIATION

Example DC-DC converter and insulation monitor – PCB (used by Velis Electro/Pipistrel)

- Input voltage 220-420 V DC.
- Output voltage 10-15 V DC.
- Adjustable target voltage and current limit.
- Maximum power 500 W.
- Provides power for:
  - 12 V battery charging.
  - BMS (relays, fans, electronics).
  - Avionics (radio, flight instruments).
  - Drive (inverter, cooling pump).
  - Led lights.
  - Main computer.

- Detects insulation failure.
- Output via CAN bus.

ELECTRIC AVIATION

Example Junction box – PCB (used by Velis Electro/Pipistrel)

- It connects the front and rear batteries, charging port, DC-DC converter, and power controller.
- Includes:
  - Charging relays.
  - DC-DC converter and insulation monitor.
  - DC-DC fuse.
  - Charging fuse.

No high voltage devices/power lines/junctions are presented inside the cockpit

ELECTRIC AVIATION

Example charging port (used by Velis Electro/Pipistrel)

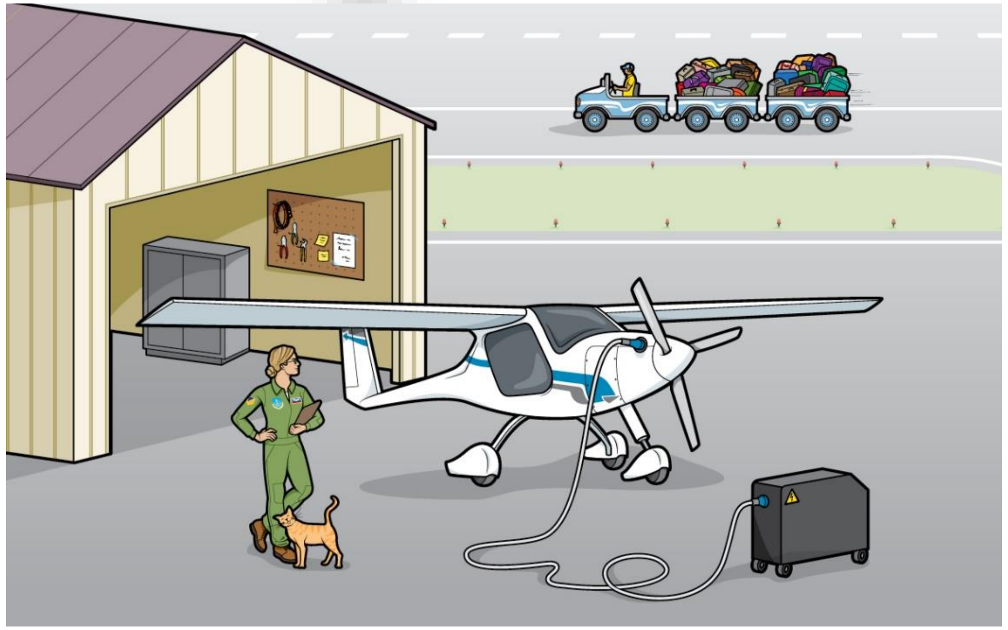
SAE AE-7D (electric aviation standard)

ELECTRIC AVIATION

Example charger (used by Velis Electro/Pipistrel)

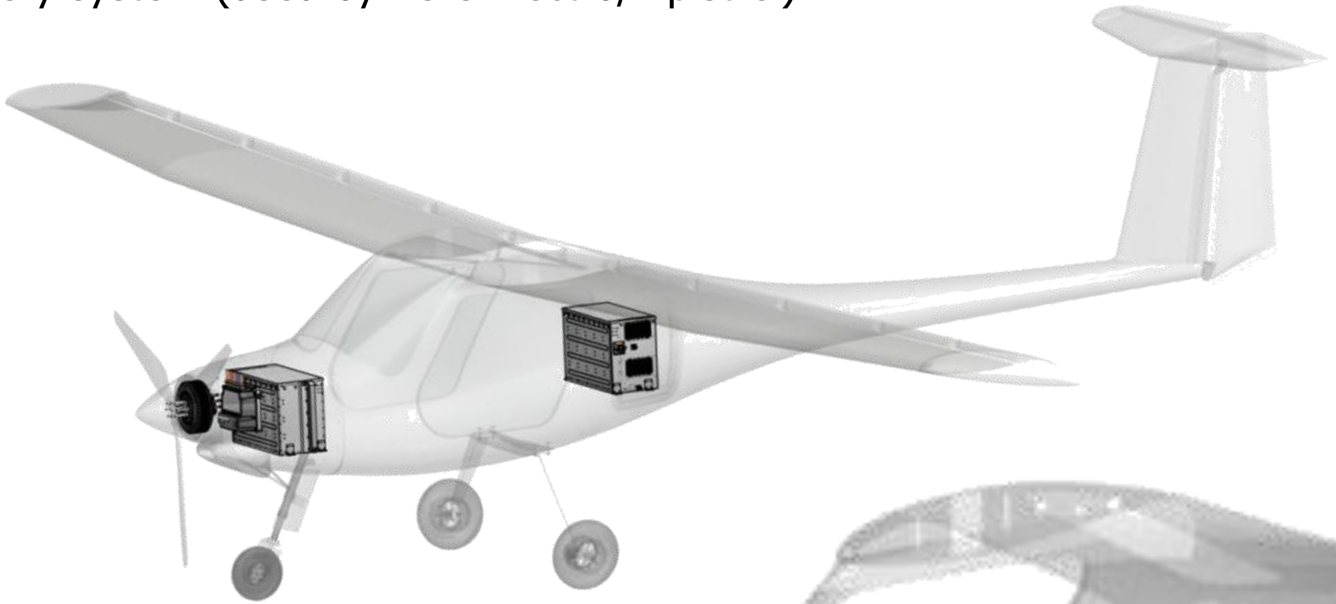
No dedicated onboard charger (like electric cars)
Fabricated by Pipistrel of DC chargers compatible

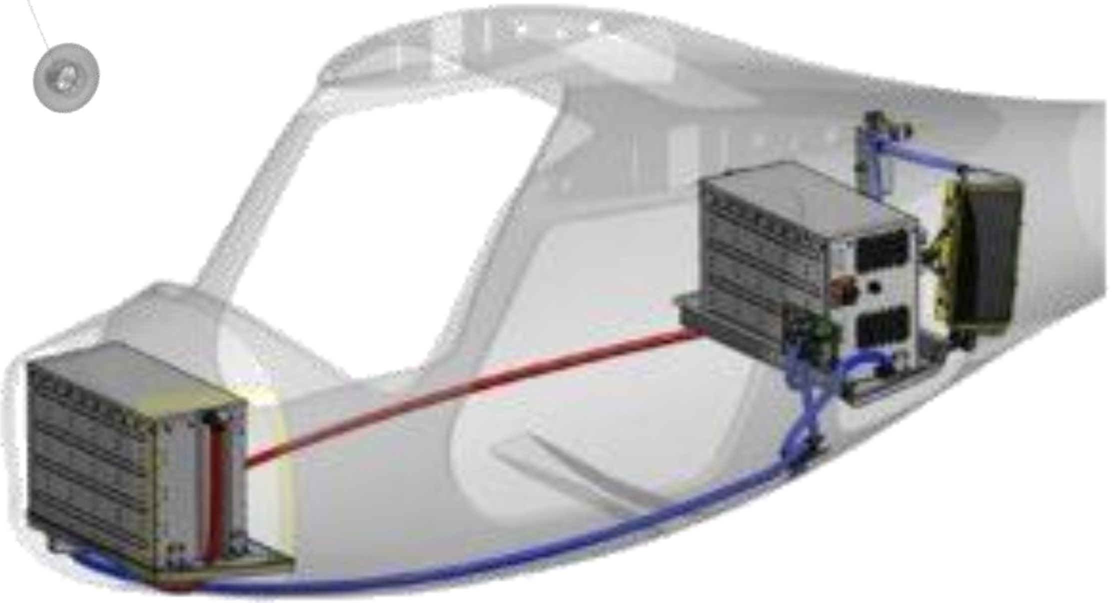
Input voltage 90-265 V (0-50 A; 40-60 Hz)
10 kW version 3 $hours$ to full charge
20 kW version 1.5 $hours$ to full charge
Adjustable charging output



ELECTRIC AVIATION

Example battery system (used by Velis Electro/Pipistrel)

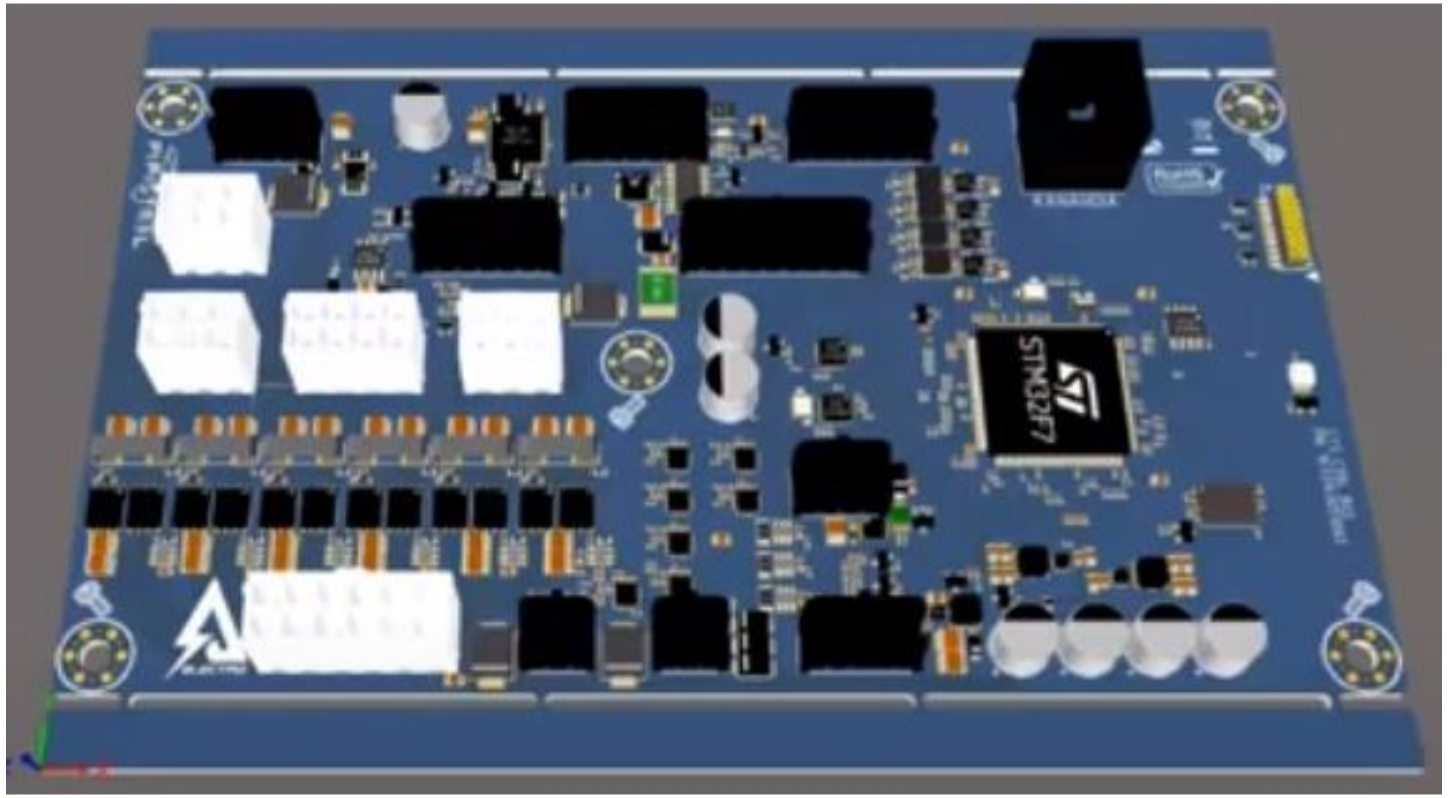




Front and back battery box
Stored energy $10.5\ kWh$ (each box)
Weight: $63\ kg$ each box
Operating temperature: $10\text{-}40\ °C$
Operating voltage: $288V\text{-}400V$
Peak discharge current: $95A$
Passive balancing system
Maximum charging current: $30A$
Battery Management System (BMS)

ELECTRIC AVIATION

Example main computer (used by Velis Electro/Pipistrel)



Power supply management
Power control functions
Battery system control
Error management
Charging control
Data management

ELECTRIC AVIATION

Electric aircraft Run-time, range Loiter, and Climb (*Raymer*)

$$P_{\text{used}}\ \eta_p = TV = DV = \frac{W}{L/D} V = \frac{mg}{L/D} V$$

Run-Time Endurance (hrs):

$$E = \frac{m_b\, E_{sb}\ \eta_{b2s}}{1000\ P_{\text{used}}}$$

where:

m_b = mass of batteries {kg}
E_{sb} = battery specific energy {wh/kg}
η_{b2s} = total system efficiency from battery to motor output shaft
P_{used} = average power used during that period of time {kW}

Level Flight Endurance or Loiter time (hrs):

$$E = 3.6\,\frac{L}{D}\,\frac{E_{sb}\ \eta_{b2s}\ \eta_p}{g\ \ V}\,\frac{m_b}{m}$$

where:

V = velocity {km/h}
E_{sb} = battery specific energy {wh/kg}
η_p = propeller efficiency

ELECTRIC AVIATION

Electric aircraft Run-time, range Loiter, and Climb (*Raymer*)

Level Flight Range (km):

$$R = 3.6 \frac{L}{D} \frac{E_{sb}\ \eta_{b2s}\ \eta_p}{g} \frac{m_b}{m}$$

Rate of Climb (m/s):

$$V_V = \frac{1000\ \eta_p}{g} \frac{P_{\text{used}}}{m} - \frac{V}{3.6\ L/D}$$

where:

V = velocity {km/h}
η_p = propeller efficiency
P_{used}/m = power-to-"weight" ratio {W/g or kW/kg}

ELECTRIC AVIATION

Range and Endurance Estimates for Battery-Powered Aircraft (*Traub, Lance W. – Embry-Riddle Aeronautical University*)

$$\left.\begin{aligned} \text{Preq} &= D \times U \\ C_D &= C_{Do} + kC_L^2 \\ D &= qS(C_{Do} + kC_L^2) \\ L = W &= 0.5\rho U^2 S C_L \end{aligned}\right\} \quad \text{Preq} = \frac{1}{2}\rho U^3 S C_{Do} + \frac{2W^2k}{\rho U S}$$

Peukert's equation

$$t = \frac{C}{i^n} \longrightarrow t = \frac{Rt}{i^n}\left(\frac{C}{Rt}\right)^n$$

$$P_B = Vi \longrightarrow P_B = V\frac{C}{Rt}\left(\frac{Rt}{t}\right)^{1/n}$$

$$P_B = \text{Preq}$$

$$\left(\frac{Rt}{t}\right)^{1/n}\left(\frac{C}{Rt}\right) = \frac{1}{\eta_{\text{tot}} V}\left[\frac{1}{2}\rho U^3 S C_{Do} + \frac{2W^2k}{\rho U S}\right]$$

$$E = t = Rt^{1-n}\left[\frac{\eta_{\text{tot}} V \times C}{\frac{1}{2}\rho U^3 S C_{Do} + (2W^2k/\rho U S)}\right]^n$$

For maximum endurance $\longrightarrow \left\{\begin{aligned} C_{Do} &= \tfrac{1}{3}kC_L^2 \\ U_E &= \sqrt{\frac{2W}{\rho S}\sqrt{\frac{k}{3C_{Do}}}} \end{aligned}\right.$

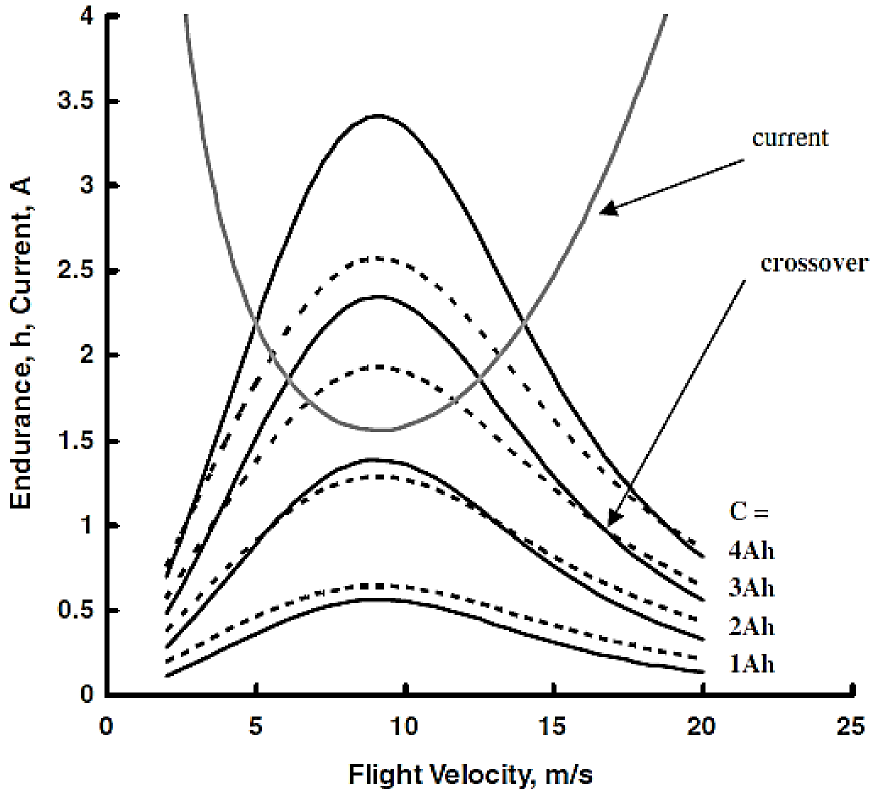
For maximum range $\longrightarrow \left\{\begin{aligned} C_{Do} &= kC_L^2 \\ U_R &= \sqrt{\frac{2W}{\rho S}\sqrt{\frac{k}{C_{Do}}}} \end{aligned}\right.$

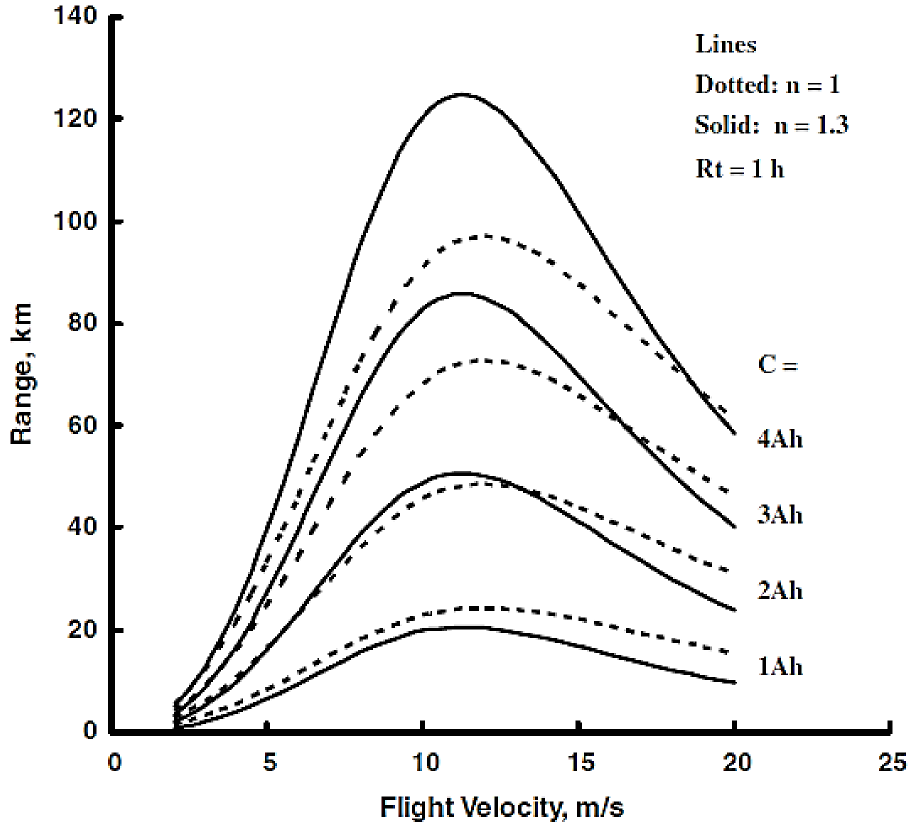
$$R = E \times U_R$$

ELECTRIC AVIATION

Range and Endurance Estimates for Battery-Powered Aircraft (*Traub, Lance W. – Embry-Riddle Aeronautical University*)

assuming constant gross weight.





$$E = t = Rt^{1-n}\left[\frac{\eta_{\text{tot}} V \times C}{\frac{1}{2}\rho U^3 S C_{Do} + (2W^2 k/\rho U S)}\right]^n$$

$$R = E \times U_R$$

ELECTRIC AVIATION

Universidad Pontificia Bolivariana

Fundada en 1936

Range and Endurance Estimates for Battery-Powered Aircraft (*Traub, Lance W. – Embry-Riddle Aeronautical University*)

For a *UAV* with the following characteristics:

- $W = 9.34$ [N]
- $S = 0.32$ [m^2]
- $C_{D,0} = 0.015$
- $k = 0.13$
- $V = 11.1$ [$Volts$]
- $\eta_{tot} = 0.5$

Calculate and graphic the changes in Endurance and Range for different velocity ($U \rightarrow [2.5 - 20.0]$) values for an $Rt = 1$ [h], battery capacities (C) of 1, 2, 3, and 4 [Ah], and including the *Peukert* effect ($n = 1.0$ and $n = 1.3$). Assume $\rho_\infty = 1.2$ [kg/m^3]. With a battery weight per pack of 0.8 [N], determine the number of batteries required for the capacity ranges (1 - 4), using the following approximation:

$$W_{tot} = \frac{j \times W_{batt}}{BR}$$

Where, BR is the battery weight as a fraction of the total weight (typically 0.3 to 0.4), W_{batt} is the weight of each individual battery (pack), W_{tot} is the *UAV*'s total weight and j is a counter expressing the number of batteries (pack).


¡Soy orgullosamente UPB! • Sede central Medellín