MODULE III

ENGINE SELECTION AND FLIGHT VEHICLE PERFORMANCE

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INITIAL SIZING

3.1- INITIAL SIZING- INTRODUCTION

- Aircraft sizing is the process of determining the takeoff gross weight and fuel weight required for an aircraft concept to perform its design mission. That sizing method was limited to fairly simple design missions.

- This chapter presents a more refined method capable of dealing with most types of aircraft- sizing problems.

- An aircraft can be sized using some existing engine or a new design engine. The existing engine is fixed in size and thrust, and is referred to as a "fixed engine"("fixed" refers to engine size).

- The new design engine can be built in any size and thrust required, and is called a "rubber engine" because it can be "stretched" during the sizing process to provide any required amount of thrust.

- Rubber-engine sizing is used during the early stages of an aircraft development program that is sufficiently important to warrant the development of an all-new engine. This is generally the case for a major military fighter or bomber program, and is sometimes the case for a transport- aircraft project.

- In these cases, the designer will use a rubber engine in the early stages of design, and then updates, when the customer tells the engine company what characteristics the new engine should have. When the engine company finalizes the design for the new engine, it becomes fixed in size and thrust. The aircraft concept will then be finalized around this now-fixed engine

- Developing a new jet engine costs on the order of a billion dollars. Developing and certifying a new piston engine is also very expensive. Most aircraft projects do not rate development of a new engine, and so must rely on selecting the best of the existing engines. However, even projects which must use an existing engine may begin with a rubber-engine design study to determine what characteristics to look for in the selection of an existing engine

- The rubber engine can be scaled to any thrust so the thrust-to-weight ratio can be held to some desired value even as the aircraft weight is varied. The rubber-engine sizing approach allows the designer to size the aircraft to meet both performance and range goals, by solving for takeoff gross weight while holding the thrust-to-weight ratio required to meet the performance objectives. As the weight varies, the rubber-engine is scaled up or down as required.

- This is not possible for fixed-engine aircraft sizing. When a fixed size engine is used, either the mission range or the performance of the aircraft must become a fallout parameter.
- For example, if a certain rate of climb must be attained, then the thrust to- weight ratio cannot be allowed to fall to an extremely low value. If the calculation of the takeoff gross weight required for the desired range indicates that the weight is much higher than expected, then either the range must be reduced or the rate of climb must be relaxed.

3.2-RUBBER ENGINE SIZING

- Sizing an aircraft can be estimated using a configuration sketch and the selected aspect ratio. From this information a crude estimate of the maximum *LID* was obtained. Using approximations of the specific fuel consumption, the changes in weight due to the fuel burned during cruise and loiter mission segments were estimated, expressed as the mission-segment weight fraction *(Wi*+ 1/ Wi). Using these fractions and the approximate fractions for takeoff, climb, and landing which were provided in the data, the total mission weight fraction *(Wx/W0)* was estimated.

- For different classes of aircraft, statistical equations for the aircraft empty-weight fraction were provided in Table – (2.a). Then, the takeoff weight was calculated using Eq. (2.1).

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right)$$

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - (W_f/W_0) - (W_e/W_0)}$$

$W_e/W_0 = A W_0^C K_{vs}$	A	C
Sailplane—unpowered	0.86	−0.05
Sailplane—powered	0.91	−0.05
Homebuilt—metal/wood	1.19	−0.09
Homebuilt—composite	0.99	−0.09
General aviation—single engine	2.36	−0.18
General aviation—twin engine	1.51	−0.10
Agricultural aircraft	0.74	−0.03
Twin turboprop	0.96	−0.05
Flying boat	1.09	−0.05
Jet trainer	1.59	−0.10
Jet fighter	2.34	−0.13
Military cargo/bomber	0.93	−0.07
Jet transport	1.02	−0.06

K_{vs} = variable sweep constant = 1.04 if variable sweep
= 1.00 if fixed sweep

Table.2.1. a – empty weight fraction vs W_0

- Since the empty weight was calculated using a guess of the takeoff weight, it was necessary to iterate towards a solution. This was done by calculating the empty-weight fraction from an initial guess of the takeoff weight and using Eq. (2.1) to calculate the resulting takeoff weight. If the calculated takeoff weight did not equal the initial guess, a new guess was made somewhere Between the two.

Equation (2.1) is limited in use to missions which do not have a sudden weight change, such as a payload drop. Also, in many cases Eq. (2.1) cannot be used for fixed-engine sizing.

3.3- FIXED ENGINE SIZING

- The sizing procedure for the fixed-size engine is similar to the rubber engine sizing, with several exceptions. These result from the fact that either the mission range or the performance must be considered a fallout parameter, and allowed to vary as the aircraft is sized.

- If the range is allowed to vary, the sizing problem is very simple. The required thrust-to-weight ratio *(T/W)* is determined as in the last section to provide all required performance capabilities, using the known characteristics of the selected engine. Then the takeoff gross weight is determined as the total engine takeoff thrust divided by the required takeoff thrust-to weight ratio.

$$W_0 = \frac{NT_{\text{per engine}}}{(T/W)} \quad \text{\_\_(2.3)} \qquad \text{where } N = \text{number of engines.}$$

- With the takeoff weight known, the range capability can be determined from Eq. (2.4) using a modified iteration technique. The known takeoff weight is repeatedly used as the "guess" *W0*, and the range for one or more cruise legs is varied until the calculated Wo equals the known Wo.
- This technique can also be used to vary mission parameters other than range. For example, a research aircraft may be sized for a certain radius (range out and back) with the number of minutes of test time as the variable parameter.

$$W_0 = W_{\text{crew}} + W_{\substack{\text{fixed}\\ \text{payload}}} + W_{\substack{\text{dropped}\\ \text{payload}}} + W_{\text{fuel}} + \left(\frac{W_e}{W_0}\right) W_0 \quad \text{\_\_(2.4)}$$

- If some range requirement must be satisfied, then performance must be the fallout. The takeoff gross weight will be set by fuel requirements, and the fixed-size engine may not necessarily provide the thrust-to-weight ratio desired for performance considerations.

- In this case the takeoff gross weight can be solved by iteration of Eq. (2.4) as for the rubber- engine case, with one major exception. The thrust-to - weight ratio is now permitted to vary during the sizing iterations. Equation (2.5) cannot be used for determining a weight fraction for combat mission legs as it assumes a known *TI W*.

$$W_i / W_{i-1} = 1 - C(T/W)(d) \quad \text{\_\_\_\_\_\_\_\_\_\_\_\_\_\_\_\_(2.5)}$$

- Instead, the fuel burned during combat by a fixed-size engine is treated as a weight drop. For a given engine, the fuel burned during a combat leg of duration d is simply the thrust times the specific fuel consumption times the duration:

$$W_f = CTd$$

3.4 ENGINE SELECTION

Figure illustrates the major options for aircraft propulsion. All aircraft engines operate by compressing outside air. Mixing it with fuel, burning the mixture and extracting energy from the resulting high-pressure hot gases. In a piston-prop, these steps are done intermittently in the cylinders through the reciprocating pistons. In a turbine engine, these steps are done continuously but in three distinct parts of the engine.

The piston prop was the first form of aircraft propulsion which are mainly limited to light airplanes and some agricultural airplanes now. Piston prop engines have two advantages. They are cheap and they have the lowest fuel consumption. However, they are heavy and produce a lot of noise and vibration and also the propeller losses efficiency as the velocity increases.

The turbine engine consists of a compressor, a burner and a turbine. These separately perform the three functions of the reciprocating piston in a piston engine. The compressor takes the air delivered by the inlet system and compresses it to many times atmospheric pressure. This compressed air passes to the burner, where fuel is injected and mixed with the air and the resulting mixture ignited.

The hot gases could be immediately expelled out the rear to provide thrust but they are first passed through a turbine to extract enough mechanical power to drive the compressor.

There are two types of compressors. The centrifugal compressor relies upon centrifugal force to fling air into an increasingly narrow channel, which raises the pressure. In contrast, an axial compressor relies upon blade aerodynamics to force the air into an increasingly narrow channel. An axial compressor typically ahs about six to ten stages, each of which consists of a rotor disk of blades and s stator disk of blades.

The turbo prop and turbofan engines both use a turbine to extract mechanical power from the exhaust gases. This mechanical power is used to accelerate a larger mass of outside air, which increases efficiency at lower speeds.

For the turboprop engine, the outside air is accelerated by a conventional propeller. The prop-fan or unducted fan is essentially a turboprop with an advanced aerodynamics propeller capable of near sonic speeds.

For the turbofan engine, the air is accelerated with a ducted fan of one or several stages, this accelerated air is then split, with part remaining in the engine for further compression and burning and the remainder being bypassed around the engine to exit unburned. The bypass ratio is the mass flow ratio of the bypassed air to the air that goes into the core of the engine. Bypass ratio ranges from as high as 6 to as low as 0.25 (the so-called leaky turbojet).

To lower the temperature seen by the turbine blades, excess air is used. Currently engines are limited to a turbine temperature of about 2000 – 2500 ^{0} F, which required an air/fuel mixture of about 60 to 1. Thus, only about a quarter of the captured and compressed air is actually used for combustion. The exhaust is 75% unused hot air.

If the fuel is injected into this largely-uncombusted hot air, it will ix and burn which will raise the thrust as much as a factor of two and is known as after-burning. Unfortunately, afterburning is inefficient in terms of fuel usage. The fuel flow required to produce kN of thrust in afterburner is approximately double that used to produce a kN of thrust during normal engine operations. Due to the high temperatures produced, afterburning must be done downstream of the turbine. Also, it is usually necessary to divert part of the compressor air to cool the walls of the afterburner and nozzles. Addition of an afterburner will approximately double the length of a turbojet or turbofan engine.

If an aircraft is travelling fast enough, the inlet duct alone will compress the air enough to burn if fuel is added. This is the principle of a “ramjet”. Ramjets must be travelling at above Mach 3 to become competitive with a turbojet in terms of efficiency.

A “Scramjet” is a ramjet that can operate with supersonic internal flow and combustion. Scramjets are largely unproven and are suitable only for operation above Mach 5 or 6, Ramjets and Scramjets require some other form of propulsion for takeoff and acceleration to the high Mach numbers they require for operation.

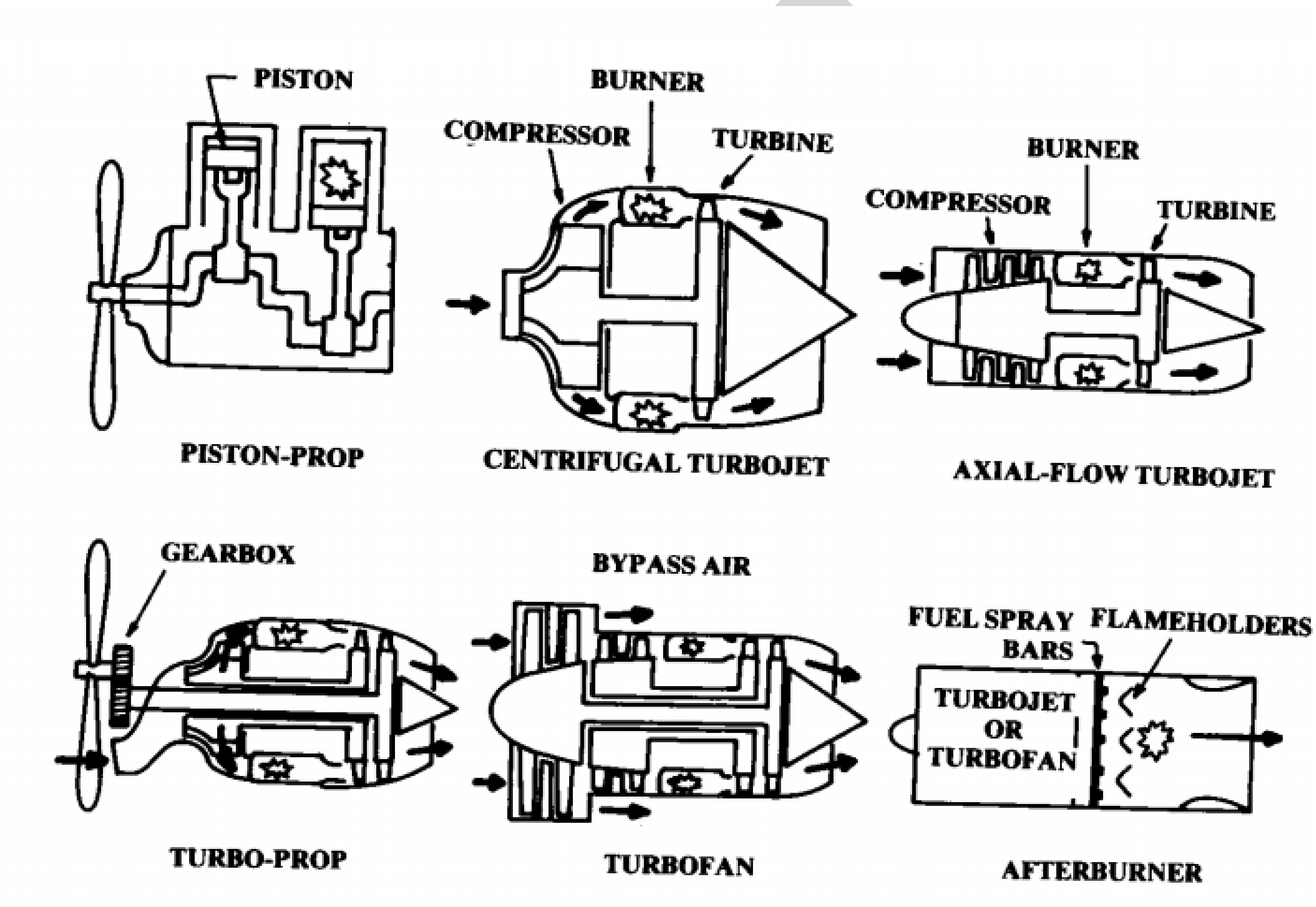


Figure 3.1: Propulsion systems

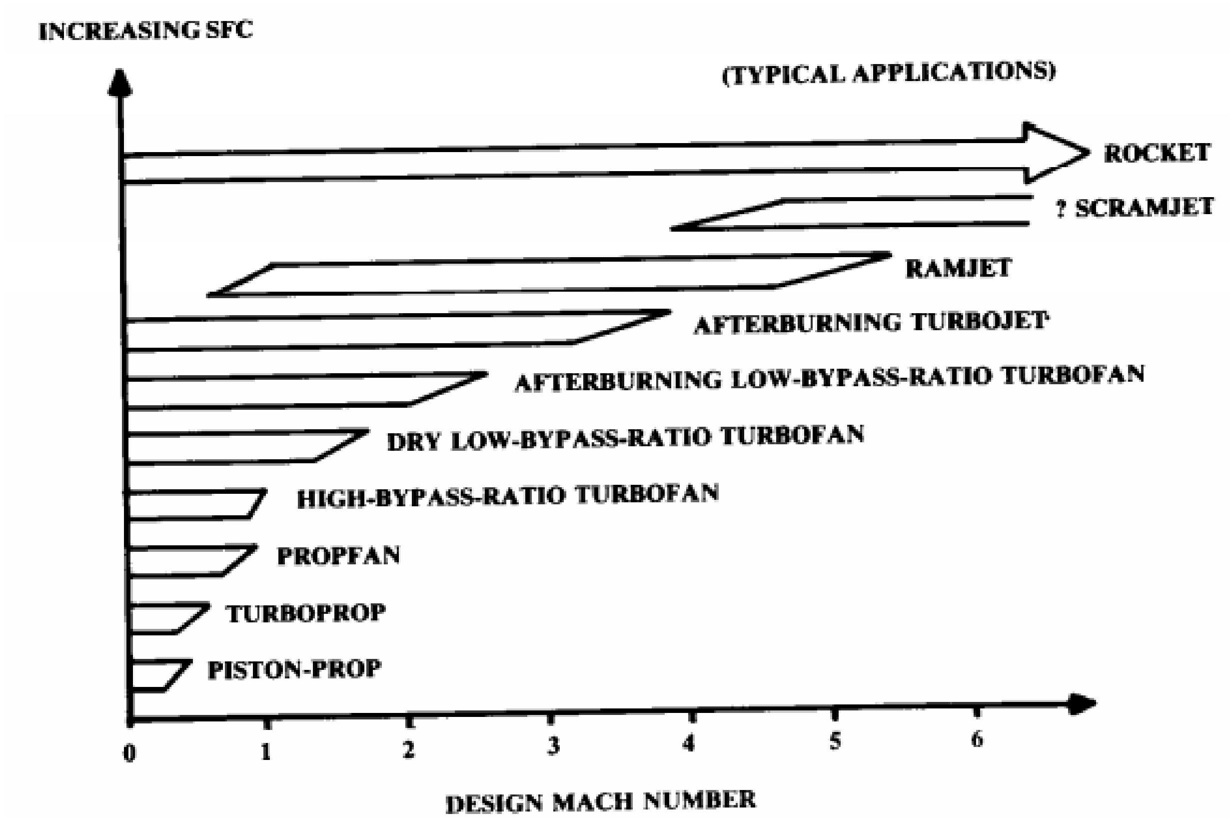


Figure 3.2: Propulsion system limits

The selection of the type of propulsion system – piston-prop, turbo prop, turbofan, turbojet, ramjet – will usually be from the design requirements. Aircraft maximum speed limits the choices as shown in the above figure. In most cases there is no reason to select a propulsion system other than the lowest on the chart for the design Mach number.

The choice between a piston prop and turboprop can depend upon several additional factors. The turboprop uses more fuel than a piston prop of the same horsepower, but is substantially lighter and more reliable. Also, turboprops are usually quieter. For these reasons gas turbine engines have largely replaced piston engines for most business twins and short range commuter airplanes regardless of design speed. However, piston-props are substantially cheaper and will likely remain the only choice for light aircraft for a long time.

3.5 JET-ENGINE INTEGRATION

If the aircraft is designed using an existing, off-the -shelf engine, the dimensions are obtained from the manufacturer. If a “rubber” engine is being used, the dimensions for the engine must be obtained by scaling from some nominal engine size by whatever scale factor is required to provide the desired thrust. The nominal engine can be obtained by several methods.

On the major aircraft companies, designers can obtain estimated data for hypothetical "rubber" engines from the engine companies. This data is presented for a nominal engine size, and precise scaling laws are provided.

Another method for defining a nominal engine assumes that the new engine will be scaled version of an existing one, perhaps with some performance improvement due to the use of newer technologies.

Figure illustrates the dimensions that must be scaled from the nominal engine. The scale factor "SF" is the ratio between the required thrust and the actual thrust of the nominal engine.

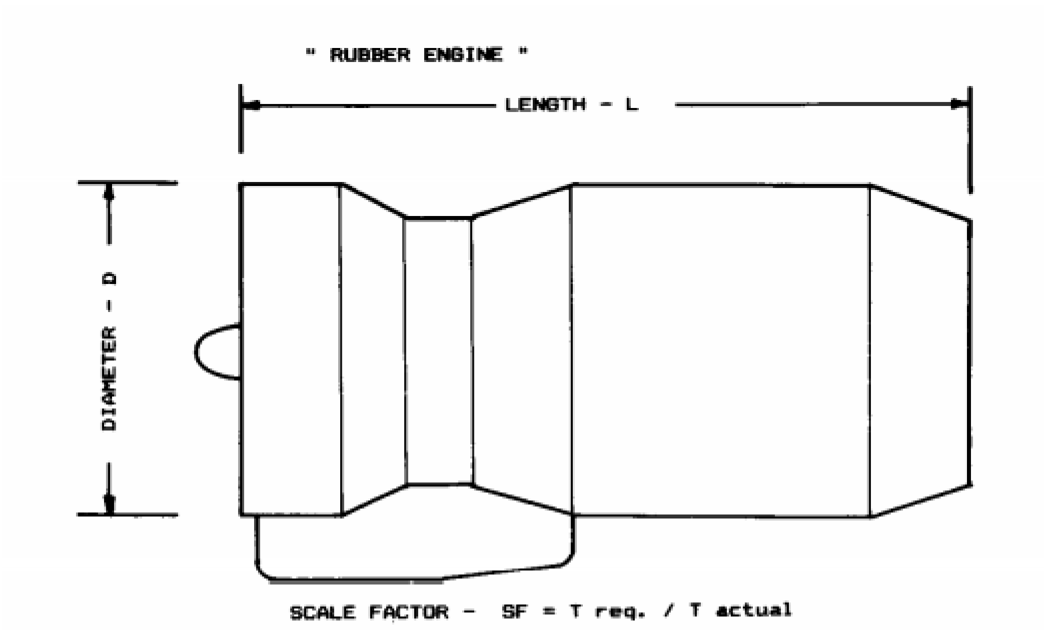


Figure 3.3: Engine Scaling

Equations below show how length, diameter and weight vary with the scale factor for the typical jet engine.

$$L = L_{\text{actual}}(\text{SF})^{0.4}$$

$$D = D_{\text{actual}}(\text{SF})^{0.5}$$

$$W = W_{\text{actual}}(\text{SF})^{1.1}$$

The engine accessories package beneath the engine includes fuel pumps, oil pumps, power takeoff gear boxes and engine control boxes. The location and size of the accessory package varies widely for different types of engines, in the absence of a drawing, the accessory package can be assumed to extend below the engine to a radius of about 20-40 % greater than the engine radius. On some engines these accessories have been located in the compressor spinner or other places.

If a parametric deck is unavailable, and no existing engines come closer enough to the desired characteristics to be rubberized and updated as described above, than a parametric statistical approach can be used to define the nominal engine.

Equations below define two first-order statistical jet-engine models based upon the existing data. One model is for subsonic non-afterburning engines such as found on commercial transports and covers a bypass-ratio range from zero to 6. The other model is for afterburning engines foe supersonic fighters and bombers (M<2.5) and includes bypass ratios from zero to just under 1.

Non-afterburning engines:

$$W = 0.084T^{1.1}e^{(-0.045\ \mathrm{BPR})}$$

$$L = 2.22T^{0.4}M^{0.2}$$

$$D = 0.393T^{0.5}e^{(0.04\ \mathrm{BPR})}$$

$$\mathrm{SFC}_{\max T} = 0.67e^{(-0.12\ \mathrm{BPR})}$$

$$T_{\mathrm{cruise}} = 0.60T^{0.9}e^{(0.02\ \mathrm{BPR})}$$

$$\mathrm{SFC}_{\mathrm{cruise}} = 0.88e^{(-0.05\ \mathrm{BPR})}$$

Afterburning engines:

$$W = 0.063T^{1.1}M^{0.25}e^{(-0.81\ \mathrm{BPR})}$$

$$L = 3.06T^{0.4}M^{0.2}$$

$$D = 0.288T^{0.5}e^{(0.04\ \mathrm{BPR})}$$

$$\mathrm{SFC}_{\max T} = 2.1e^{(-0.12\ \mathrm{BPR})}$$

$$T_{\mathrm{cruise}} = 1.6T^{0.74}e^{(0.023\ \mathrm{BPR})}$$

$$\mathrm{SFC}_{\mathrm{cruise}} = 1.04e^{(-0.186\ \mathrm{BPR})}$$

where

W = weight
T = takeoff thrust
BPR = bypass ratio
M = max Mach number
Cruise is at 36,000 ft and $0.9M$.

These equations represent a very unsophisticated model for initial estimation of engine dimensions. They should not be applied beyond the given bypass-ratio and speed ranges.

3.4 JET-ENGINE THRUST CONSIDERATIONS

A jet engine develops thrust by taking in air, compressing it, mixing in fuel, burning the mixture and expanding and accelerating the resulting high-pressure, high temperature gasses out the rear through a nozzle.

To provide power to drive the compressor, a turbine is placed in the exhaust stream which extracts mechanical power from the high-pressure gases. If greater thrust is required for a short period of time, an afterburner can be placed downstream of the turbine permitting the unburned air in the turbine exhaust to combust with additional fuel and thereby increase the exhaust velocity.

'Gross thrust" is produced as a result of the total momentum in the high-velocity exhaust stream. "Net thrust" is calculated as the gross thrust minus the "ram drag", which is the total momentum in the inlet stream. The ram drag which results from the deceleration of the air taken into the inlet, is included in the engine cycle analysis performed by the engine manufacturer to determine net uninstalled thrust.

An increase in air density such as at low altitude or low outside air temperature would therefore increase thrust by increasing mass flow. Hot day takeoffs from a high-elevation airport pose problems because the reduction in air density causes a reduction in mass flow and hence thrust.

Similarly, an increase in aircraft velocity also increases thrust due to ram effect increasing the mass flow. However, for a typical subsonic jet, the exhaust comes out the nozzle at a choked condition and so the exit velocity equals the speed of sound regardless of aircraft velocity. As air craft velocity approaches the speed of sound, the thrust is therefore reduced for the choked exit nozzle. When combined with the favorable ram effect, this results in a relatively constant thrust as velocity increases for the typical subsonic jet, dropping off as transonic speeds are reached.

For supersonic jet engines, a variable area converging diverging nozzle is typically employed which permits supersonic exhaust velocities. Therefore, the ram effect does cause the thrust tend to increase with increasing velocity until at high Mach numbers where excessive total pressure losses occur in the inlet, resulting in thrust degradation. The Mach number at which inlet losses become excessive is determined by the number of shocks and the extent of variable geometry employed.

Thrust and propulsive efficiency are strongly affected by the engine’s overall pressure ratio (OPR). OPR is the ratio of the pressure at the engine exhaust plane and inlet front face. This pressure ratio is a measure of the engine’s ability to accelerate the exhaust, which produces thrust. OPRs usually range from about 15 to 1 toa bout 30 to 1.

Another key parameter which currently limits turbine engine performance is the turbine inlet temperature (TIT). This results in less thrust and thermal efficiency and so a key objective in propulsion technology development has always been the increase in allowable TIT.

To increase propulsive efficiency, the turbofan engine uses an oversized fan with some of the accelerated fan air “bypassed” around the engine, not being used for combustion. This has the effect of allowing the engine to accelerate a larger cross-sectional area of air by a smaller change in velocity, which increases the efficiency. The bypass ratio is defined as the ratio of the mass flows of the bypasses air and the air that goes through the core of the engine to be used for combustion.

A higher bypass ratio, which enables the engine to accelerate a larger cross section of air, produces higher efficiency and hence greater thrust for a given expenditure of fuel. However, the fan alone cannot efficiently accelerate the air to transonic or supersonic exit speeds and so this favorable effect works only at lower speeds. Thus, high bypass turbofan is best at subsonic speeds and low bypass ratio turbofan at the low supersonic speeds. At higher supersonic speeds, ie., about Mach 2.2, the pure turbojet is superior.

3.5 TURBOJET INSTALLED THRUST:

The statistical methods for estimating installed thrust and specific fuel consumption for jet engines are suitable for initial sizing and performance estimation. For a more sophisticated analysis it is necessary to estimate analytically the installed performance of an existing engine or a proposed new engine from the uninstalled engine data.

It is assumed that uninstalled-engine data is available, either from an engine manufacturer, a preliminary cycle analysis or a fudge factor approach based upon some given engine.

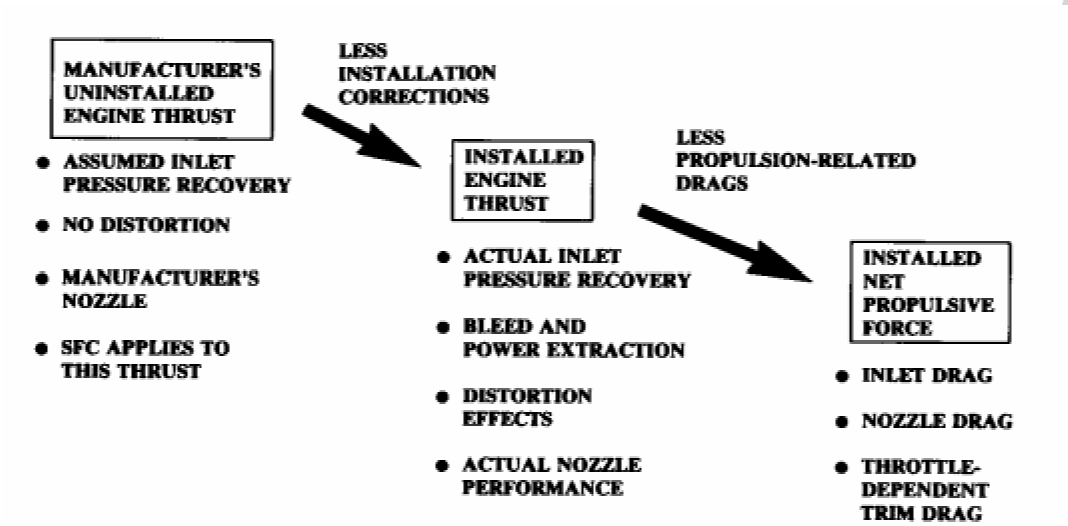


Figure 3.4 : Installed Thrust methodology

Usually, designer would start with existing engine with characteristics that are close to the required and then scale it up or down based scaling laws. It can be scaled based on conservation of mass and momentum for fluids.

$$T = \dot{m}(V_e - V_a) + A_e(P_e - P_a).$$

Turbojet engines are designed to increase the momentum of incoming air, but not the exit pressure, therefore

$$P_e \simeq P_a.$$

If it is considered to be static thrust,

$$T = \dot{m} V_e.$$

$$T = T_{\text{ref}} \frac{(\dot{m} V_e)}{(\dot{m} V_e)_{\text{ref}}}.$$

Assuming that $V_e = V_{ref}$

$$T = T_{\text{ref}} \frac{\dot{m}}{\dot{m}_{\text{ref}}}.$$

Mass flow rate of the engine is

$$\dot{m} = (\rho A V)_e = \left[\rho V \frac{\pi d^2}{4}\right]_e,$$

With respect to the reference engine

$$\dot{m} = \dot{m}_{\text{ref}} \left[\frac{d}{d_{\text{ref}}}\right]_e^2.$$

$$\left[\frac{d}{d_{\text{ref}}}\right]_e = \left[\frac{\dot{m}}{\dot{m}_{\text{ref}}}\right]^{1/2}$$

$$\left[\frac{d}{d_{\text{ref}}}\right]_e = \left[\frac{T}{T_{\text{ref}}}\right]^{1/2}.$$

Altitude and velocity effects:

Always, thrust value mentioned by the engine manufacturer corresponds to sea level thrust, if in case thrust is to be calculated at an cruise altitude, sea level thrust should be corrected for that altitude As per ideal gas relation

$$\rho = \frac{P}{R\theta},$$

$$T_H = T_{\text{SL}} \frac{P_H}{P_{\text{SL}}} \frac{\theta_{\text{SL}}}{\theta_H}.$$

Ve – Va decreases as the speed of the Aircraft increases, therefore thrust will be decreased with increase in airspeed, so this will be compensated by "ram effect" in turn increases the thrust

$$\mathrm{TSFC} = \frac{w_f}{T},$$

INSTALLED ENGINE THRUST CORRECTIONS:

The manufacturer's uninstalled engine thrust is based upon an assumed inlet pressure-recovery. For a subsonic engine, it is typically assumed that the inlet has perfect recovery ie., 1.0. Supersonic military aircraft engines are usually defined using an inlet pressure-recovery of 1.0 at subsonic speeds and the inlet recovery at supersonic speeds. The pressure recovery loss due to internal flow in the duct adds to 2-3% to the losses.

Figure 3.4 shows this reference inlet pressure-recovery plotted versus Mach number, compared to the recovery available for a normal shock inlet and external compression inlets with one, two and three ramps.

$$\left(\frac{P_1}{P_0}\right)_{\mathrm{ref}} = 1 - 0.075(M_\infty - 1)^{1.35}$$

The external compression inlets of figure are of movable ramp design with a perfectly optimized schedule of ramp angles as a function of Mach number. To determine the pressure recovery of a fixed or less-than perfect inlet, the shock tables should be used.

The pressure losses inside the inlet duct are determined by the length and diameter of the duct, the presence of bends in the duct and internal Mach number.

For initial evaluation of a typical inlet duct, an internal pressure recovery of 0.96 for a straight duct and 0.94 for an S duct may be used. The short duct of a subsonic podded nacelle will have a pressure recovery of 0.98 or better. Most detailed estimation of inlet internal-pressure loss is based upon experimental data and requires a separate evaluation at each Mach number.

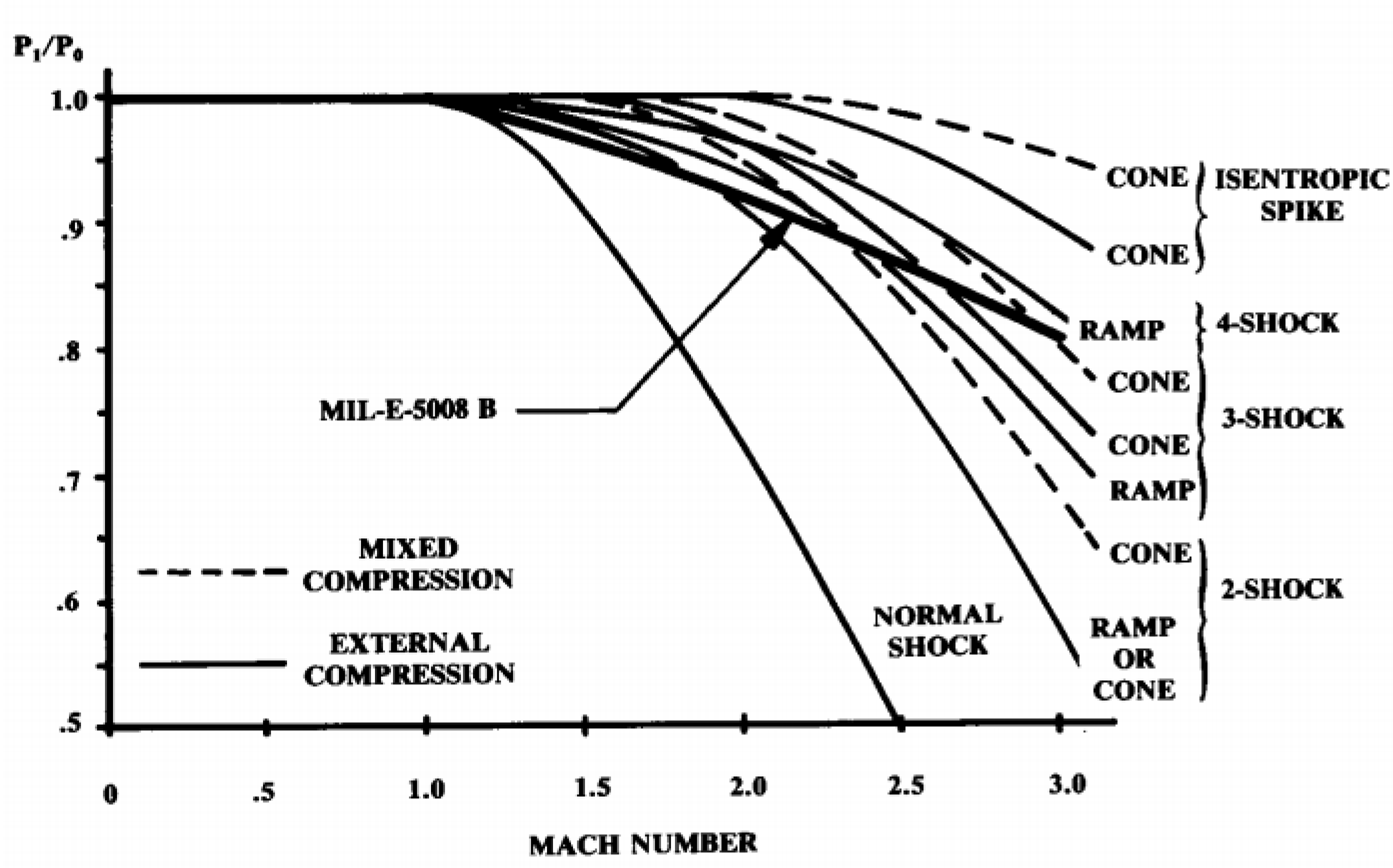


Figure 3.5: Reference and available inlet pressure recovery rate

Figure 3.5 provides the actual inlet pressure recoveries of some existing designs. This may be used for pressure-recovery estimation during early design studies. Reducing inlet pressure recovery has a greater-than-proportional effect upon the engine thrust. The inlet "ram recover correction factor (C_{ram})" is provided by the manufacturer for various altitudes, Mach numbers, air temperatures and thrusts settings. Typically C_{ram} ranges from 1.2-1.5. If the manufacturer's data is not available C_{ram} may be approximated as 1.35 for subsonic flight and by the below equation for supersonic flight.

$$\text{Percent thrust loss} = C_{ram}\left[\left(\frac{P_1}{P_0}\right)_{ref} - \left(\frac{P_1}{P_0}\right)_{actual}\right] \times [100]$$

$$\text{Supersonic:} \quad C_{ram} \cong 1.35 - 0.15(M_\infty - 1)$$

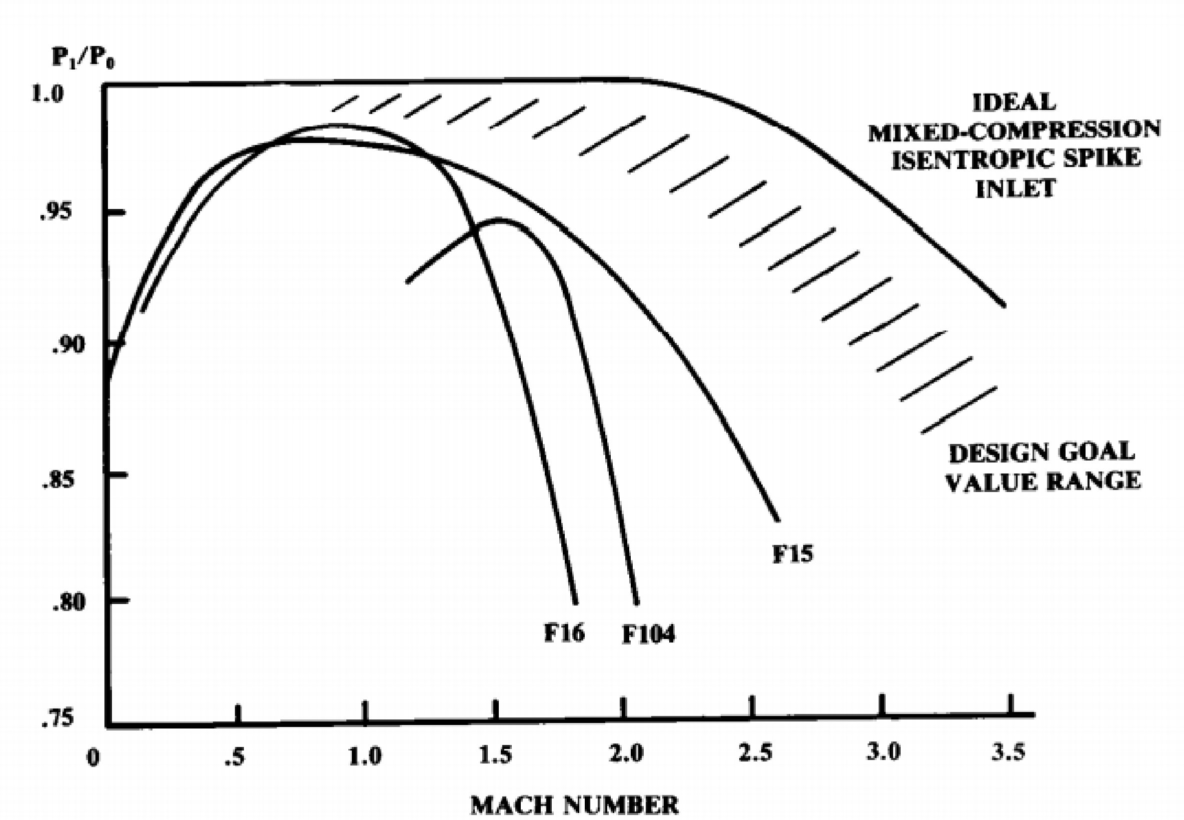


Figure 3.6: Actual inlet pressure recoveries

High-pressure air bled from the engine compressor for cabin air, anti-icing and other used. This engine bleed air exacts a thrust penalty that is also more-than-proportional to the percent of the total engine mass flow extracted as bleed air.

The below equation illustrates this, where the "bleed correction factor (C_{bleed})" is provided by the manufacturer for various flight conditions. For initial analysis, C_{bleed} can be approximated as 2.0. the bleed mass flow typically ranges from 1-5% of the engine mass flow.

$$\text{Percent thrust loss} = C_{\text{bleed}}\left(\frac{\text{bleed mass flow}}{\text{engine mass flow}}\right) \times [100]$$

Installed engine thrust is also affected by horsepower extraction. Jet engines are equipped with rotating mechanical shafts turned by the turbines. The electrical generators, hydraulic pumps and other such components connect to these shafts.

This extraction is typically less than 200 hp for a 30,000 lb thrust engine, and usually has only a small effect upon installed thrust, but can be ignored for initial analysis.

Thrust reported by the manufacturer is uninstalled thrust, thrust that is relevant to the Aircraft performance is said to be installed thrust, uninstalled thrust is corrected for installed thrust and called as installed thrust corrections

Installation effect includes:

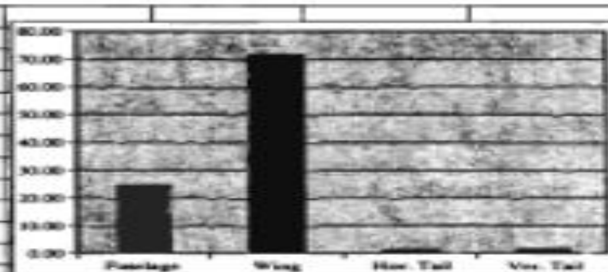
- Inlet pressure recovery
- Bleed air and power extraction
- Inlet airflow distortion
- Exit nozzle performance

Added propulsion drag originated from

- Inlet drag caused by air spillage
- Exit nozzle drag
- Trim drag results from added control trim

SPREAD SHEET FOR TURBO ENGINE SIZING

Drag:	Drag (lbf)	% Total
Fuselage	2705.0	25.17
Wing	7739.0	72.00
Hor. Tail	140.8	1.31
Ver. Tail	163.8	1.52
Total	**10748.6**	100.00



Uninstalled Thrust:		
Altitude	55000	ft
p/p_{SL}	1.05E-01	
θ/θ_{SL}	0.752	
Treq	10749	lbf
No. Engines	4	
Treq/Eng	2687.15	lbf
% Max T	0.8	
Mach Factor	2.39	
Tref	2412.94	lbf
Treq/Tref	1.11	
Reference Engine Data:		
Wref	2,240	lbf
Lref	158.8	in
Dref	34.8	in
mref	146	lbm/s
Scaled Engine Data:		
Weight	2494.6	lbf
Length	167.6	in
Diameter	36.7	in
Total Thrust	**12250.1**	lbf
m	162.6	lbm/s
Take-off		
Total Thrust	49000.23	
T-O Weight	90523	
(T/W) TO	0.54	

Reference Engine:
GE-F404-100D

H (ft)	p/p_{SL}	θ/θ_{SL}	T (lbs)
0	1.000	1.000	**11000.00**
16000	0.542	0.890	6704.20
20000	0.460	0.863	5864.87
24000	0.388	0.835	5111.02
28000	0.326	0.808	4433.88
32000	0.272	0.780	3827.61
36000	0.225	0.753	3286.95
40000	0.186	0.752	2717.68
44000	0.153	0.752	2244.11
48000	0.127	0.752	1853.50
52000	0.105	0.752	1530.64
56000	0.086	0.752	**1264.47**

PISTON ENGINE PERFORMANCE:

The aircraft piston engine operates on the four stroke Otto cycle used by the automobiles. For design purposes the most important thing to know that the piston engine is that the horse power produced is directly proportional to the mass flow of the air into the intake manifold. Here, horsepower is approximately 620 times the air mass flow (kg/s)

Mass flow into the engine is affected by the outside air density (altitude, temperature and humidity) and intake manifold pressure. The below equations accounts for the air-density affect upon horsepower and is attributed to Gagg and Ferrar of the Wright Aeronautical Company (0934). This equation indicates that at an altitude of 20,000 ft a piston engine has less than half of its sea level horse power.

$$\mathbf{bhp} = \mathbf{bhp_{SL}} \left(\frac{\rho}{\rho_0} - \frac{1 - \rho/\rho_0}{7.55} \right)$$

The intake is usually at atmospheric pressure. A forward-facing air-intake scoop can provide some small increase in manifold pressure at higher speeds. Large increases in manifold pressure require mechanical pumping via a “supercharger” of “Turbo supercharger”

The supercharger is a centrifugal air compressor mechanically driven by a shaft from the engine. The amount of air compression available is proportional to engine RPM. The turbo supercharger or “turbocharger” is driven by a turbine placed in the exhaust pipe. This recovers energy which would otherwise be wasted and decouples the available amount of compression form the engine RPM.

Supercharging or turbo supercharging is usually used to maintain seal level pressure in the intake manifold as the aircraft climbs. Typically, the sea level pressure can be maintained up to an altitude of about 15,000 – 20,000 ft. Above this altitude the manifold pressure and hence the horsepower drops. Figure 3.6 show typical engine performance for non-supercharged, supercharged and turbocharged engines.

Piston engine performance charts are provided by the manufacturer as a function of manifold pressure, altitude and RPM.

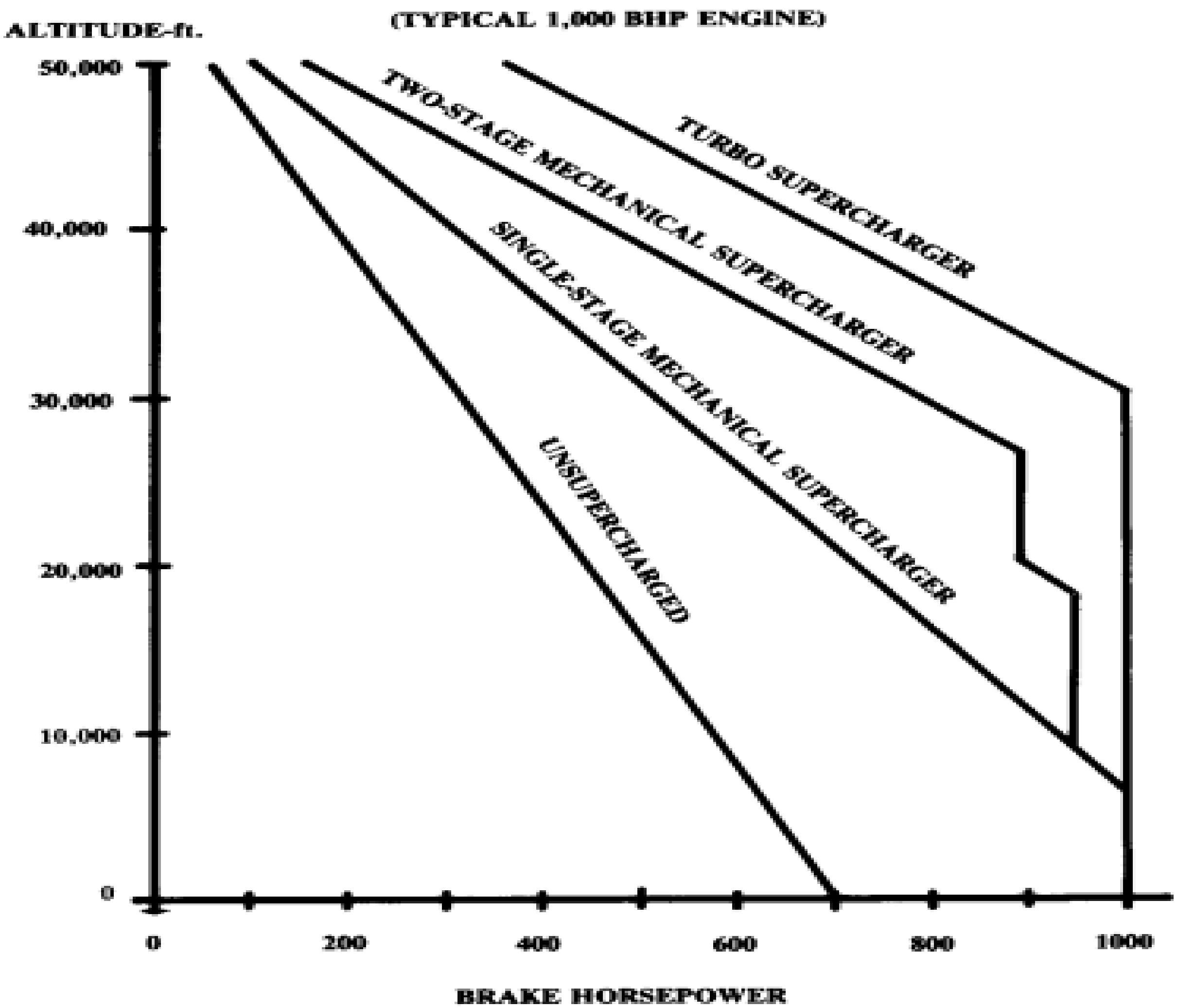


Figure 3.7: Effects of supercharging

PROPELLER PERFORMANCE:

A propeller is a rotating airfoil that generates thrust much as a wing generates lkft. Like a wing, the propeller is designed to a particular flight condition the propeller has a selected design lift coefficient (usually around 0.5) and the twist of the airfoil is selected to give the optimal airfoil angle of attack at the design condition.

Since the tangential velocities of the propeller airfoil sections increase with distance from the hub, the airfoils must be set at progressively reduced pitch-angles going from root to tip. The overall “pitch” of a propeller refers to the blade angle at 75% of the radius.

While propeller theories are useful for propeller designers, the aircraft designers usually work with experimental propeller data provided by the propeller companies.

As for a wing, the properties of a propeller are expressed in coefficient form. Experimental data for design purposes are expressed using a variety of parameters and coefficients, as described below.

Advance Ratio: $J = V/nD$

Power Coefficient: $c_P = \dfrac{P}{\rho n^3 D^5} = \dfrac{550 \text{ bhp}}{\rho n^3 D^5}$

Thrust Coefficient: $c_T = T/\rho n^2 D^4$

Speed-Power Coefficient: $c_S = V \sqrt[5]{\rho/Pn^2}$

Activity Factor:

$$\mathrm{AF}_{\text{per blade}} = \frac{10^5}{D^5}\int_{0.15R}^{R} cr^3\,\mathrm{d}r = \frac{10^5 c_{\text{root}}}{16D}\,[0.25 - (1-\lambda)0.2]$$

Propeller Efficiency: $\eta_p = \dfrac{TV}{P} = \dfrac{TV}{550 \text{ bhp}} = J\dfrac{c_T}{c_P}$

Thrust: $T = \dfrac{550 \text{ bhp}\ \eta_p}{V} = \dfrac{c_T}{c_P}\dfrac{550 \text{ bhp}}{nD}$

where

T	= thrust (lb)
V	= velocity (ft/s)
P	= power (ft-lb/s)
bhp	= brake horsepower
n	= rotation speed (rev/s)
D	= propeller diameter (ft)
c	= propeller airfoil chord (ft)

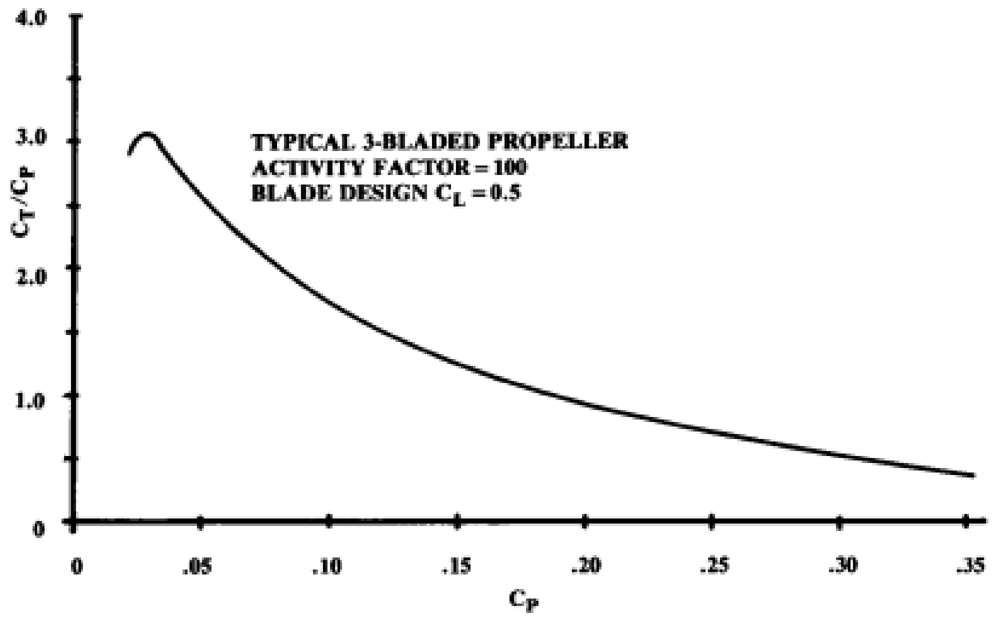


Figure 3.8: Static propeller thrust

Propeller data is available from the manufacturers as well as various NASA/NACA reports. This data is provided in many different formats using different combinations of the above parameters and coefficients. The thrust equation is used to determine the propeller thrust at a given flight condition.

Figure 3.8 and 3.9, propeller charts for static and forward flight have been chosen as typical of propellers used for modern light and business aircraft. These charts represent a three-bladed propeller with a design lift-coefficient of 0.5 and an activity factor of 100.

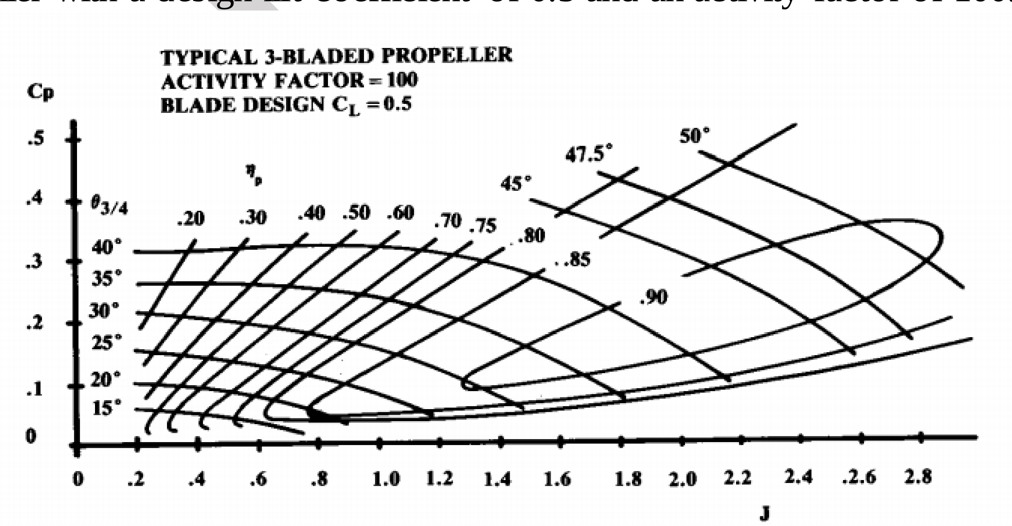


Figure 3.9 : Forward flight thrust and efficiency

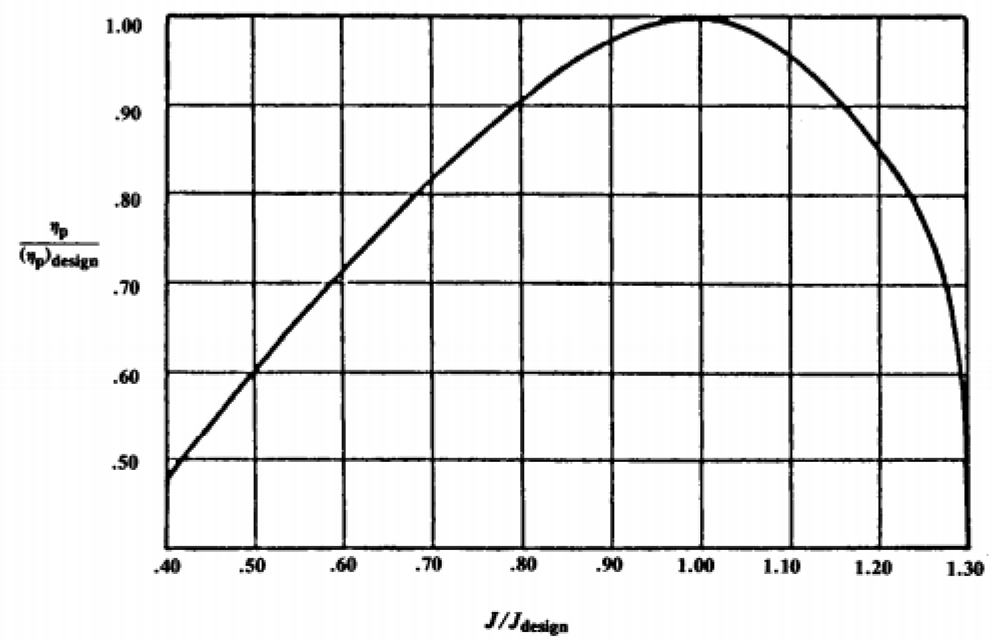


Figure 3.10: Fixed-pitch propeller adjustment

If a fixed-pitch propeller is used, the blade angle cannot be varied in flight to maintain engine RPM at any flight condition. Since the RPM and therefore horsepower will vary with velocity, the efficiency and hence the thrust will be reduced at any speed other than the design speed.

Figure 3.9 could be used to determine the thrust from a fixed pitch propeller by following the appropriate line for selected blade angle. It is simpler to use the approximate method of above figure unless actual propeller data is available.

Figure 3.10 relates the fixed-pitch propeller efficiency at an off-design velocity and RPM to the on-design efficiency, which is attained by the propeller at some selected flight condition. The on-design efficiency is obtained which is also used to get the required blade angle for the design condition.

PISTON-PROP THRUST CORRECTIONS:

As with Jet engine, there are several engine related drag components that must be considered namely, scrubbing drag, cooling drag and engine miscellaneous drag.

Scrubbing drag is the increase in aircraft drag due to the higher velocity and turbulence experienced by the parts of the aircraft within the propwash. This drag could be calculated by determining, for each flight condition, the increased dynamic pressure within the propwash and using that value for the component-drag calculation.

A simpler approach, called the SBAC (Society of British Aircraft Constructors) method, adjusts the propeller efficiency as in the below equation. The subscript "washed" refers to the parts of the aircraft which lie within the propwash. If the parasite-drag coefficient for the propwashed parts of the aircraft cannot be determined, 0.004 is a reasonable estimate.

$$\eta_{p_{\text{effective}}} = \eta_p \left[1 - \frac{1.558}{D^2} \frac{\rho}{\rho_0} \Sigma (C_{f_e} S_{\text{wet}})_{\text{washed}}\right]$$

Where C_{fe} is the equivalent skin-friction (parasite) drag coefficient, referenced to wetted area.

For a pusher-propeller configuration, the scrubbing drag is zero. However, the pusher propeller suffers a loss of efficiency due to the wake of the fuselage and wing. This loss is strongly affected by the actual aircraft configuration and should equal about 2 – 5%.

Colling drag represents the momentum loss of the air passed over the engine for colling. This is highly dependent upon the detail design of the intake, baffles and exit.

Miscellaneous engine drag includes the drag of the oil cooler, air intake, exhaust pipes and other parts. Cooling and miscellaneous drags for a well-designed engine installation can be estimated by the below equations. However, a typical light aircraft engine installation may experience cooling and miscellaneous drag levels 2-3 times the values estimated by these equations.

$$(D/q)_{\text{cooling}} = (4.9 \times 10^{-7}) \frac{\text{bhp} \cdot T^2}{\sigma V}, \ \text{ft}^2$$

$$(D/q)_{\text{misc}} = (2 \times 10^{-4}) \ \text{bhp}, \ \text{ft}^2$$

where

T = air temperature, deg Rankine
V = velocity in ft/s

TURBOPROP PERFORMANCE:

A turboprop is a jet engine that drives a propeller using a turbine in the exhaust. The jet exhaust retains some thrust capability, and can contribute as much as 20%of the total thrust. Thus, the horsepower rating of a turboprop engine includes the horsepower equivalent of this residual thrust. Analysis of the turboprop is a hybrid between the jet and the piston-prop nalysis. The engine is analyzed like a jet, including the inlet effects. The residual thrust is provided by the manufacturer as a horsepower equivalent. The propeller is analyzed including the scrubbing-drag term.

The conventional turboprop, like the piston-prop, is limited by tip Mach number to about Mach 0.7. the turboprop has higher efficiency than the piston-prop at Mach numbers greater than about 0.5 due to the residual jet thrust, but the conventional turboprop in no match for a turbofan engine at the higher subsonic speeds.

Recently, a new type of advanced propeller has been developed that offers good efficiencies up to about Mach 0.85. These are known as “prop fans” or “unducted fans (UDF)”. They are smaller in diameter than the regular propellers and feature numerous wide, thin and swept blades.

FLIGHT VEHICLE PEFORMANCE

TAKEOFF ANALYSIS:

Figure 3.11 illustrates the segments of the takeoff analysis. The ground roll includes two parts, the level ground-roll and ground roll during rotation to the angle of attack for liftoff. After rotation the aircraft follows an approximately circular arc (“transition”) until it reaches the climb angle.

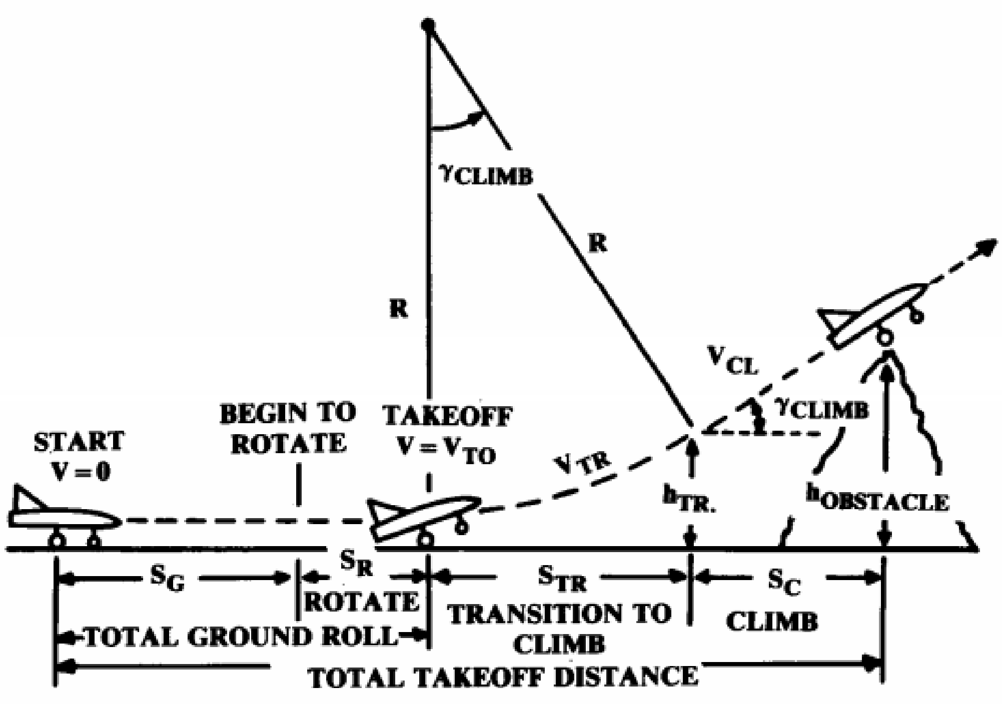


Figure 3.11 : Take off Analysis

Maximum lift coefficient is that parameter to be used in takeoff and landing analysis Take-off flight phase consists of accelerating from rest to a takeoff velocity and climbing to an altitude which should be greater than obstacle height

$$V_{TO} = 1.2V_s = 1.2\left[\left(\frac{W}{S}\right)_{TO}\frac{2}{\rho C_{L_{max}}}\right]^{0.5}$$

Takeoff phase is divided into 4 sub phases

- Ground roll (portion where aircraft accelerates)
- Rotation (Aircraft will pitch up)
- Transition
- Climb

GROUND ROLL:

During the ground roll, the forces on the aircraft are the thrust, drag and rolling friction of the wheels which is expressed as a rolling friction coefficient μ times the weight on the wheels (W-L).a typical value for rolling resistance on a hard runway is 0.03 Values for various runway surfaces are presented in the table below.

Table: Ground rolling_resistance

	μ-typical values	
Surface	Rolling (brakes off)	Brakes on
Dry concrete/asphalt	0.03–0.05	0.3–0.5
Wet concrete/asphalt	0.05	0.15–0.3
Icy concrete/asphalt	0.02	0.06–0.10
Hard turf	0.05	0.4
Firm dirt	0.04	0.3
Soft turf	0.07	0.2
Wet grass	0.08	0.2

The resulting acceleration of the aircraft, as expressed by equation below can be expanded in terms of the aerodynamic coefficients. This requires evaluating the lift and drag of the aircraft in ground effect and with landing gear down and flaps in the takeoff position. The lift coefficient is based on the wing angle of attach on the ground (measured to the zero lift angle) and is typically less than 0.1 unless large takeoff flaps are deployed.

$$a = \frac{g}{W}\left[T - D - \mu(W - L)\right] = g\left[\left(\frac{T}{W} - \mu\right) + \frac{\rho}{2W/S}\left(-C_{D_0} - KC_L^2 + \mu C_L\right)V^2\right]$$

The ground-roll distance is determined by integrating velocity divided by acceleration, as given by the below equation. The takeoff velocity must be no less than 1.1 times the stall speed, which is found by setting maximum lift at stall speed equal to weight and solving for stall speed. The maximum lift coefficient is with the flaps in the takeoff position.

The below equations are used for ground roll distance from $V_{initial}$ to V_{final} where the terms K_T and K_A are also defined which contains the thrust terms and the aerodynamics terms.

$$S_G = \int_{V_i}^{V_f} \frac{V}{a}\,dV = \frac{1}{2}\int_{V_i}^{V_f} \frac{1}{a}\,d(V^2)$$

$$S_G = \frac{1}{2g}\int_{V_i}^{V_f} \frac{d(V^2)}{K_T + K_A V^2} = \left(\frac{1}{2gK_A}\right) \ln\left(\frac{K_T + K_A V_f^2}{K_T + K_A V_i^2}\right)$$

$$K_T = \left(\frac{T}{W}\right) - \mu$$

$$K_A = \frac{\rho}{2(W/S)}\left(\mu C_L - C_{D_0} - K C_L^2\right)$$

Where K – drag-due-to-lift factor
K_A- drag-due-to- aerodynamic factors
K_T- drag-due-to-thrust factors

For greater accuracy the ground roll can be broken into smaller segments and integrated using the averaged thrust for each segment. The averaged thrust is the thrust at 70% of the velocity increase for that segment. Also, K may be reduced due to ground effect.

The time to rotate to liftoff attitude depends mostly upon the pilot. Maximum elevator deflection is rarely employed. A typical assumption for large aircraft is that is that rotaion takes three seconds. The acceleration is assumed to be negligible over that short time interval, so the rotation ground-roll distance S_R is approximated by three times V_{TO}. For small aircraft the rotational time is on the order of 1s and $S_R = V_{TO}$.

TRANSITION:

During transition, the aircraft accelerates from takeoff speed (1.1 V_{stall}). The average velocity during transition is therefore about 1.15 V_{stall}. The average lift coefficient during transition can be assumed to be about 90% of the maximum lift coefficient with takeoff flaps. The average vertical acceleration in terms of load factor can then be obtained from the below equation.

$$n = \frac{L}{W} = \frac{\frac{1}{2}\rho S(0.9C_{L_{\max}})(1.15V_{\text{stall}})^2}{\frac{1}{2}\rho S C_{L_{\max}} V_{\text{stall}}^2} = 1.2$$

$$n = 1.0 + \frac{V_{TR}^2}{Rg} = 1.2$$

$$R = \frac{V_{TR}^2}{g(n-1)} = \frac{V_{TR}^2}{0.2g} \cong 0.205 V_{\text{stall}}^2$$

The vertical load factor must also equal 1.0 plus the centripetal acceleration required to cause the aircraft to follow the circular transition-arc.

The climb angle γ at the end of the transition is determined from the below equation and the climb angle is equal to the included angle of the transition-arc as shown in figure 3.11, so the horizontal distance travelled during transition can also be determined. The altitude gained during transition id determined from the geometry of figure 3.11 to be as indicated below.

$$\sin\gamma_{\text{climb}} = \frac{T-D}{W} \cong \frac{T}{W} - \frac{1}{L/D}$$

$$S_T = R\sin\gamma_{\text{climb}} = R\left(\frac{T-D}{W}\right) \cong R\left(\frac{T}{W} - \frac{1}{L/D}\right)$$

$$h_{TR} = R(1 - \cos\gamma_{\text{climb}})$$

If the obstacle height is cleared before the end of the transition segment, then the below equation is used to determine the transition distance.

$$S_T = \sqrt{R^2 - (R - h_{TR})^2}$$

CLIMB:

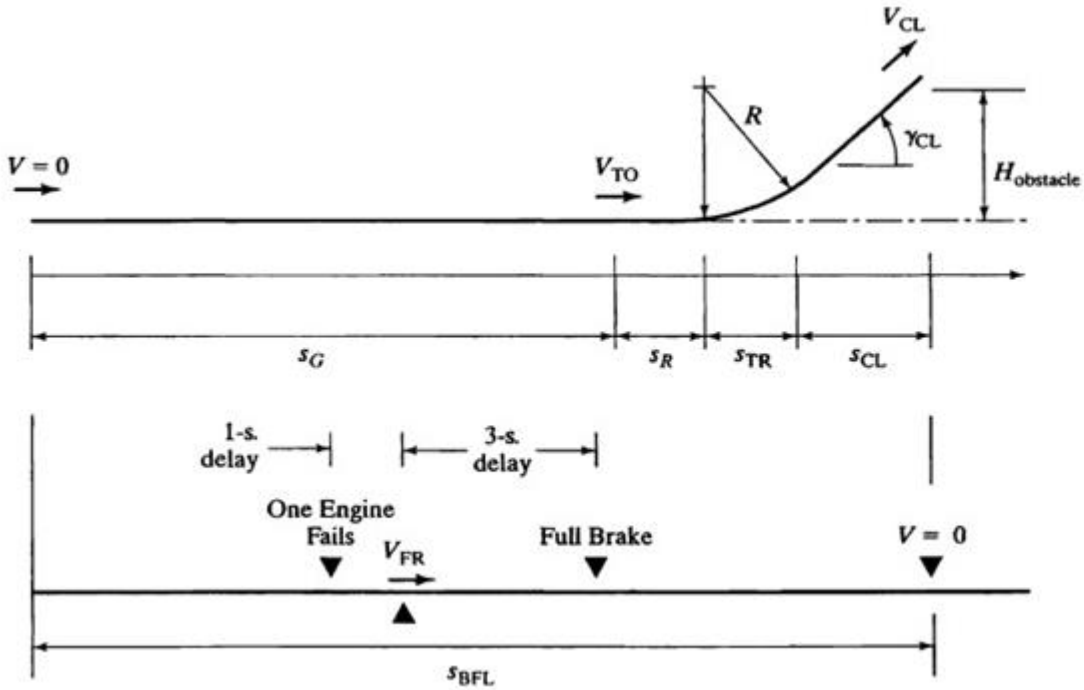
Finally, the horizontal distance travelled during the climb to clear the obstacle height is found from the below equation. The required obstacle clearance is 50 ft for military and small civil aircraft and 35 ft for commercial aircraft.

$$S_c = \frac{h_{\text{obstacle}} - h_{TR}}{\tan\gamma_{\text{climb}}}$$

If the obstacle height was cleared during transition, then Sc is zero.

BALANCE FIELD LENGTH:

The "balance field length" is the total takeoff distance including obstacle clearance when an engine fails at "decision speed" V_1, the speed at which upon an engine failure, the aircraft can either brake to a halt or continue the takeoff in the same total distance. If the engine fails before decision speed, the pilot can easily brake to a halt. If the engine fails after decision speed, the pilot must continue the takeoff.



Balanced Field Length (BFL) is given by the equation,

$$\mathrm{BFL} = \frac{0.863}{1+2.3G}\left(\frac{W/S}{\rho g C_{L_{\text{climb}}}} + h_{\text{obstacle}}\right)\left(\frac{1}{\frac{T_{av}}{W} - U} + 2.7\right) + \left(\frac{655}{\sqrt{\rho/\rho_{SL}}}\right)$$

$$\text{JET: } T_{av} = 0.75\, T_{\substack{\text{takeoff}\\ \text{static}}}\left[\frac{5+\mathrm{BPR}}{4+\mathrm{BPR}}\right]$$

$$\text{PROP: } T_{av} = 5.75\ \mathrm{bhp}\left[\frac{(\rho/\rho_{SL})N_e D_p^2}{\mathrm{bhp}}\right]^{1/3}$$

where

BFL	= balanced field length (ft)
G	$= \gamma_{climb} - \gamma_{min}$
γ_{climb}	= arcsine $[(T\text{-}D)/W]$, 1-engine-out, climb speed
γ_{min}	= 0.024 2-engine; 0.027 3-engine; 0.030 4-engine
$C_{L_{climb}}$	$= C_L$ at climb speed (1.2 V_{stall})
$h_{obstacle}$	= 35 ft commercial, 50 ft military
U	$= 0.01\ C_{L_{max}} + 0.02$ for flaps in takeoff position
BPR	= bypass ratio
bhp	= engine brake horsepower
N_e	= number of engines
D_p	= propeller diameter (ft)

LANDING ANALYSIS:

Landing is much like taking off, only backward. Figure 3.12 illustrates the landing analysis, which contains virtually the same elements as the takeoff.

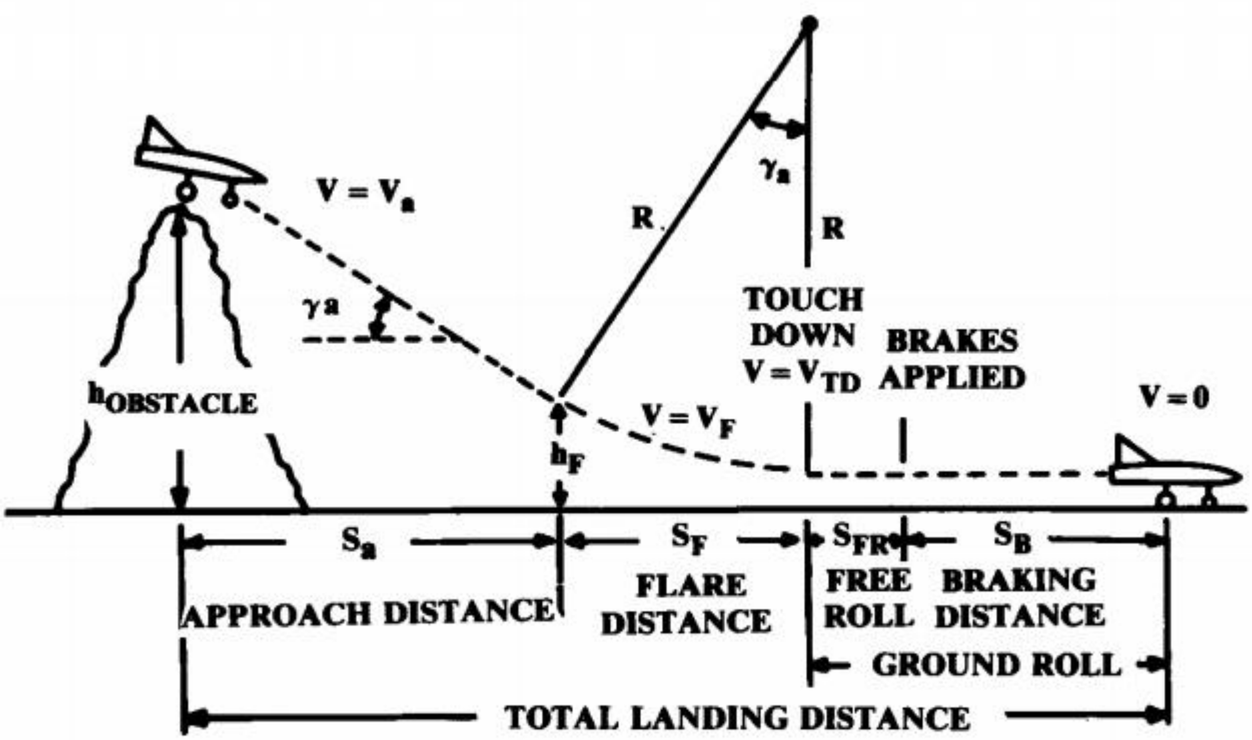


Figure 3.12: Landing Analysis

The aircraft weight for landing analysis is specified in the design requirements and ranges from the takeoff value to about 85% of the takeoff weight. Landing weight is not the end of the mission weight, because this would require dumping large amounts of fuel to land immediately after takeoff in the event of an emergency.

APPROACH:

The approach begins with obstacle clearance over a 50-ft object. Approach speed V_a is 1.3 V_{stall} (1.2 V_{stall} for military). The steepest approach angle cab be calculated from the below equation, with idle thrust and drag with full flaps deflected.

For transport aircraft the approach angle should be no steeper than 3 deg which may require more than idle thrust. Approach distance is determined from the below equation using the flare height, h_f.

$$S_c = \frac{h_{\text{obstacle}} - h_{TR}}{\tan\gamma_{\text{climb}}}$$

FLARE:

Touchdown speed V_{TD} is 1.15 V_{stall} (1.1 V_{stall} for military). The aircraft decelerates from V_a to V_{TD} during the flare. The average velocity during the flare V_f is therefore 1.23V_{stall} (1.15 V_{stall} for military). The radius of the flare circular arc is found by the below equation using V_f and where n = 1.2 for a typical aircraft.

$$\sin\gamma_{\text{climb}} = \frac{T - D}{W} \cong \frac{T}{W} - \frac{1}{L/D}$$

The flare height and the horizontal distance can be determined from the below equations.

$$S_T = R\,\sin\gamma_{\text{climb}} = R\left(\frac{T - D}{W}\right) \cong R\left(\frac{T}{W} - \frac{1}{L/D}\right)$$

$$h_{TR} = R(1 - \cos\gamma_{\text{climb}})$$

GROUND ROLL:

After touchdown, the aircraft rolls free for several seconds before the pilot applies the brakes. The distance is V_{TD} time the assumed delay (1-3 s).

The braking distance is determined by the same equation used for takeoff ground roll. The initial velocity is V_{TD} and the final velocity is zero.

$$S_G = \frac{1}{2g}\int_{V_i}^{V_f}\frac{d(V^2)}{K_T + K_A V^2} = \left(\frac{1}{2gK_A}\right)\ln\left(\frac{K_T + K_A V_f^2}{K_T + K_A V_i^2}\right)$$

The thrust term is the idle thrust. If a jet aircraft is equipped with thrust reversers, the thrust will be negative value approximately equal to 40 or 50% of maximum forward thrust.

Thrust reversers cannot be operated at very slow speeds because of reingestion of the exhaust gases. Thrust reverser "cutoff speed" is determined by the engine manufacturer and is typically about 50 knots (85 ft/s). the ground roll must be broken into two segments and the appropriate values for thrust (negative above cutoff speed, positive idle thrust below cutoff speed).

Reversible propellers produce a reverse thrust of about 40% of static forward thrust (60% for turboprops) and can be used throughout the landing roll.

The drag term may include the additional drag of spoilers, speed brakes or drogue chutes. Drogue chutes have drag coefficient of about 1.4 times the inflated frontal area, divided by the wing reference area.

The rolling resistance will be greatly increased by the application of the brakes. Typical μ values for a hard runway surface are about 0.5 for civil and 0.3 for military aircraft.

The FAA requires that an additional two-thirds be added to the total landing distance of commercial aircraft to allow for pilot technique. Thus "FAR Field Length" is equal to 1.666 times the sum of the approach, flare and total ground roll.

DESIGN SPREAD SHEET:

Take-Off						
CD_0	0.04		mu_TO	0.05		
A	2		T_max (lb)	49026		
H (f)	1,000		f_LG	30.73		
CL_G	2		A_LG (f^2)	2.5		
W_TO (lb)	90,523		deltCD_0_flap	0.05		
S (f^2)	600		gamma_CL (deg)	3		
			H_obstacle (f)	35		
k	0.2		T/W	0.54		
rho (lbm/f^3)	0.07		f1 (f/s^2)	15.83		
W/S (lb/f^2)	150.87		deltCD_0_LG	0.13		
S (f^2)	600		f2 (f^−1)	−126.1E−6		
V_T-O (f/s)	306.49		R_TR (f)	19447.91		
q_T_O (lb/f^2)	108.63		H_TR (f)	26.65		
			S_G (f)	5470.32		
			S_R (f)	919.46		
			S_TR (f)	1017.82		
			S_CL (f)	159.28		
			S_T-O (f)	**7566.88**		
Landing						
W_L (lb)	23384		D_50 (lb)	19933.81		
W/S (lb/f^2)	38.97		gamma_A (deg)	−58.48		
V_50 (f/s)	168.75		gamma_A _act	−3		
V_TD (f/s)	149.28		R_TR (f)	4166.95		
q_50 (lb/f^2)	32.93		H_TR (f)	5.71		
q_TD (lb/f^2)	25.77		f1 (f/s^2)	−19.32		
mu _L	0.6		f2 (f^−1)	562.8E−6		
T_L (lb)	0					
			S_A (f)	845.1		
			S_TR (f)	218.08		
			S_FR (f)	447.85		
			S_B (f)	930.61		
			S_L (f)	2441.63		
			1.6(S_L) (f)	**3906.61**		

ENHANCED LIFT DESIGN

The value of the maximum lift coefficient affects the take-off and landing distances.

• In aircraft designs, lift enhancing devices are used to achieve the necessary values in order to satisfy the maximum lift requirements that are imposed by such flight phases as take-off and landing, and combat (maximum maneuverability)

• The lift-enhancing devices fall in two categories:
passive and active.

• The passive devices further fall into two sub-categories:

i) trailing-edge devices, which primarily act to increase the camber of the airfoil section, and

ii) leading-edge devices, which primarily act to prevent leading-edge separation.

• Passive lift enhancement is relevant to most of the aircraft designs, except STOL and ultra-STOL. • STOL and ultra-STOL must make use of active lift enhancement

• Active lift enhancement consists of using air streams that are directed over the upper surface of wing in order to energize the boundary layer and prevent flow separation.

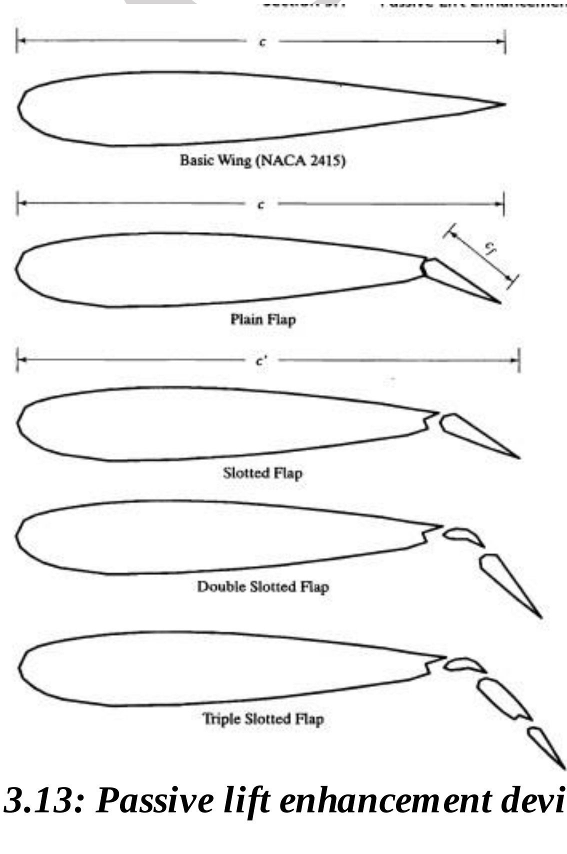


Figure 3.13: Passive lift enhancement devices

PASSIVE LIFT ENHANCEMENT Trailing-Edge (TE) Lift Enhancement Devices

• The most common types of trailing-edge lift enhancement devices are plane flaps, spilt flaps, slotted flaps, and Fowler flaps.

• Further, slotted flaps include single, double, and triple segments. • Schematic illustration of different trailing-edge flap configurations are shown in Fig. 1.

• The effectiveness of any of these devices depends on :

➢ the flap deflection angle, δ_f;

➢ the wing thickness-to-chord ratio (t/c);

➢ the ratio of the flap chord to wing chord, c/c_f ;

➢ the wing sweep angle, and

➢ the aspect ratio.

• In most cases, $c/c_f \approx 0.3$, and the maximum lift occurs at $\delta_f \approx 40^0$

Trailing-Edge (TE) Lift Enhancement Devices

▪ **Plain Flap:** The plain flap is simply a deflection of the trailing edge of the airfoil section. • This type is most widely used on smaller aircraft.

▪ **Split Flap**: • The split flap is similar to the plain flap except that only the bottom half of the airfoil section deflects.

• The lift produced by the split flap is virtually the same as a plain flap, but the drag is larger. • Therefore, the split flaps are rarely used now, but were popular on aircraft built during World War II.

▪ **Slotted Flap:** • A Slotted Flap is essentially a plain flap with the addition of a slot at the hinge point to allow highpressure air from the lower side of the airfoil to pass over the upper surface of the flap.

• This has the effect of adding momentum to the boundary layer on the upper flap surface to allow larger flap deflections before the flow separates.

▪ **Fowler Flap:** • A Fowler flap is a slotted flap that translates rearward, away from the wing.

• This has the benefit of increasing the slot width and increasing the effective wing area.

• A fowler flap is used on the C5-A aircraft.

▪ **Multiple Slotted Flaps**: • These are refined slotted (Fowler) flap designs using two or three flap segments.

• These designs can lead to extremely high lift coefficients (C_{Lmax}>= 3.0) , but are complex and require extra volume inside the wing to be stored during cruise.

▪ Single slotted flaps are most common on mid-size aircraft.

▪ Most larger commercial and transport aircraft use a multiple slotted flap arrangement.

❑ Disadvantages of Trailing Edge Flaps

• Although trailing-edge flaps increase lift at a given AoA, they do not increase the angle of stall, α_s, but actually cause it to decrease.

• This is the result of changes in the location of the stagnation line and local pressure gradient near the leading-edge, which causes a leading-edge flow separation. Sharper LEs are more sensitive to this.

• One solution to leading-edge separation is to increase the leading-edge radius. This is the principal effect of a leading-edge flap.

Leading-Edge (LE) Lift Enhancement Devices

• Increasing the leading-edge radius of an airfoil section prevents flow separation at the leading edge.

• This is the principle of the leading-edge lift enhancement devices.

• These devices consist of a hinged portion of the leading-edge, which deflects downward to effectively increase the leading-edge curvature.

• A variation of this is a Kruger flap, which consists of a hinged flap on the lower side of the wing leading-edge, which extends out into the flow. This approach is lighter in weight and, as a result, popular on large aircraft with large wing spans.

• The most common types of leading-edge lift enhancement devices are a fixed slot, leading edge flap, Kruger flap, and plain slats (slotted leading-edge flap) . These are shown in Fig. 2

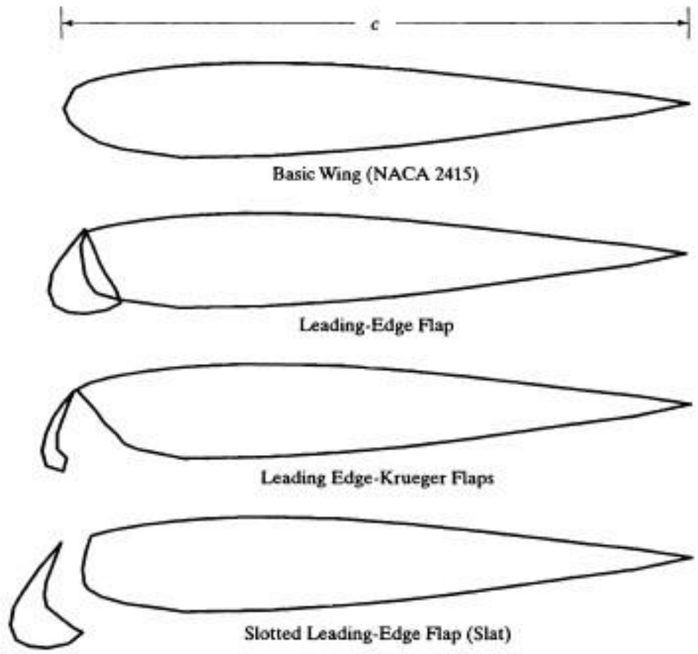


Figure 3.13: Passive lift enhancement devices

- A leading-edge slot works the same way as a slotted flap by allowing air from the high-pressure lower surface to flow to the upper surface to add momentum to the boundary layer and prevent flow separation.
- A slotted leading-edge flap (slat) is the leadingedge equivalent of the trailing-edge slotted flap.
- In this, the leading edge is extended forward and downward to open the slot and simultaneously increase the wing section camber and area.
- As a result of the change in camber, there is also a small change in . This is the arrangement used on C5-A aircraft.
- Of these lift enhancement devices, leading-edge flaps are more effective than slotted flaps on highly swept wings.
- They are also usually located over the outboard half-span of the wing in order to reduce the potential for wing-tip stall.
- The optimum leading-edge flap deflection is approximately 30 – 40 deg.

❑ LIFT DETERMINATION

• Lift determination deals with constructing the 3-D lift coefficient versus AoA for the main wing with flaps and slats.

• Starting with the 3-D lift coefficient Vs AoA for the basic wing, the first step is to find the change in the AoA at zero lift, $\Delta\alpha_{0L}$, produced by the addition of a trailing-edge flap.

• The formula to determine the AoA at zero lift depends on the type of the trailing-edge flap, as shown below:

For *plane flaps*,

$$\Delta_{\alpha_{0_L}} = -\frac{dC_l}{d\delta_f}\frac{K'}{C_{l_\alpha}}\delta_f,$$

Where $C_{l\alpha}$ is the 2D section lift coefficient, which from linear theory should be 2π/rad

K' is a correction for nonlinear effects, and can be found from Fig. 1.

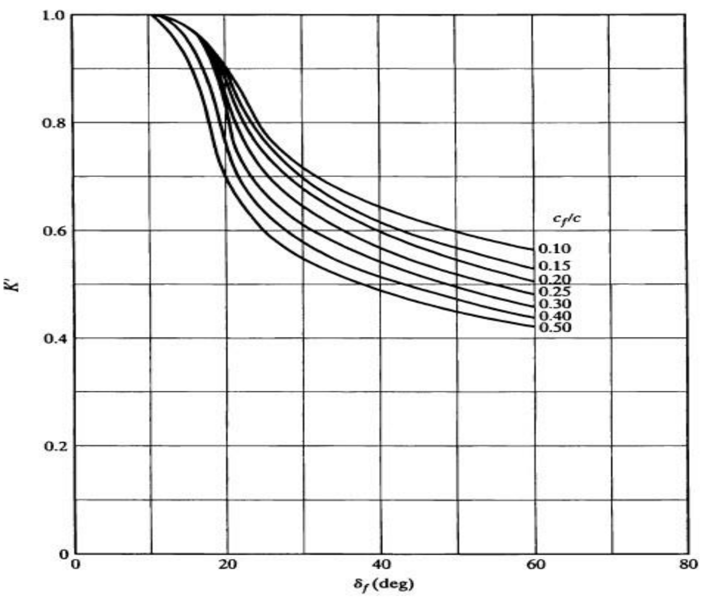
The term $dc_l/d\delta_f$ is the change in the 2-D section lift coefficient with flap deflection. This can be found from Fig. 2., and it is a function of C_f/C and t/c.

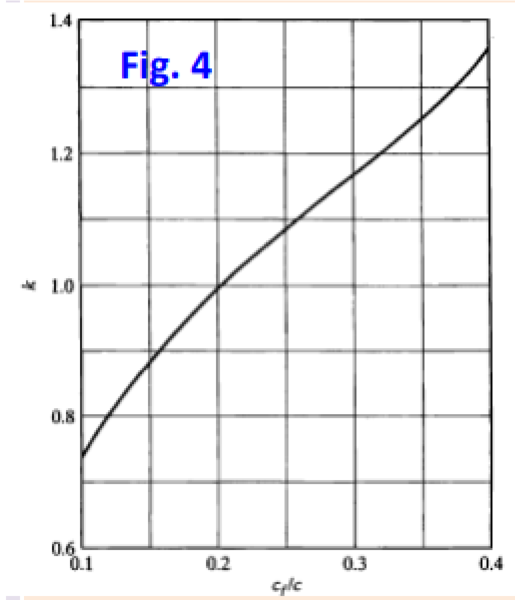
For *single slotted and Fowler flaps*,

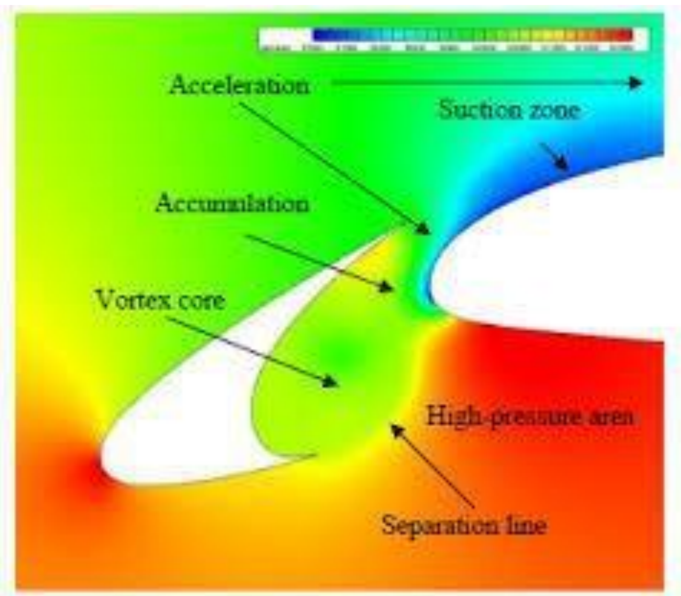
$$\Delta\alpha_{0_L} = -\frac{d\alpha}{d\delta_f}\delta_f,$$

For *split flaps*,

$$\Delta\alpha_{0_L} = -\frac{k}{C_{l_\alpha}}(\Delta C_l)_{\frac{c_f}{c}=0.2},$$







Type	$\Delta C_{l_{max}}$
Fixed Slot	0.2
Leading-Edge Flap	0.3
Kruger Flap	0.3
Slat	0.4

ACTIVE LIFT ENHANCEMENT

• Passive lift enhancement approaches will not provide sufficient C_{Lmax} for the STOL aircraft.

• Assuming sea-level conditions ($\sigma = \rho_{TO}/\rho_{SL} = 1$)

the take-off distance is given by,

$$s_{TO} = 20.9 \frac{W}{S} \frac{W}{T} \frac{1}{C_{L_{max}}} + 87 \sqrt{\frac{W}{S} \frac{1}{C_{L_{max}}}}$$

• For STOL aircraft, $S_{TO} < 1000$ ft

• For medium-size aircraft that would be designed to carry passengers or cargo, efficient cruise would dictate and

• The conditions on for STOL are

$$\frac{4180}{C_{L_{max}}} + \frac{550}{\sqrt{C_{L_{max}}}} < 1000$$

• In order to satisfy the above equation, $C_{Lmax} > 5.47$

• The largest C_{Lmax} that is attainable by passive approaches is approximately 4.0.

• Thus, active approaches are needed for STOL A/C

• Some of the common approaches used for active lift enhancement are shown in Fig. 1.

The active lift enhancement approaches generally fall into 3 categories:

i) Upper Surface Blowing (USB),

ii) Blown Flaps, where air is supplied either externally (EBF) or internally (IBF) ,

and iii) Vectored Thrust.

❑ **USB:** • With USB, a high velocity air stream is directed over upper surface of the main wing. This requires placing the engines above and forward of the wing

❑ **Blown Flaps**: • With blown flaps, high-velocity air is directed specifically at the trailing-edge flaps.

▪ **EBF**: • For externally blown flaps (EBF), the air is supplied by the engine exhaust, and the engine is located below the wing.

• The flaps are slotted in this case so that high momentum air can reach the upper surface and energize the boundary layer over the flaps.

• A portion of the air in this arrangement is also deflected downward.

• The YC-15 aircraft used this arrangement.

▪ **IBF**: • Internally blown flaps (IBF) duct a portion of the engine exhaust air only to the upper side of the trailing-edge flaps.

▪ Generation of Downward Thrust

• In addition to the enhanced aerodynamic lift that these 3 approaches provide, they also generate a component of downward thrust.

• This results because of the "Coanda Effect", which is the ability of an air stream to follow a curved surface.

• When properly designed, the air stream on the upper surface leaves the trailing edge at the angle of the flaps.

❑ **Vectored Thrust:** • Vectored thrust uses an articulated exit nozzle to direct the jet exhaust air downward. • This gives a downward component of thrust, which is independent of any aerodynamic lift enhancement on the wing.

❑ Disadvantages:

• In all the active lift enhancement approaches, there are additional factors that affect the selection of one over another.

• The IBF requires internal ducting that can be heavy and result in internal momentum losses.

• The USB blows hot exhaust air over the wing surface. This generally requires that portion of the wing to be covered with a heat-resistant material (stainless steel), which adds weight. Enhanced Lift Design (contd) CL CD qSw Thrust Cj = C j

• The EBF approach only directs the hot exhaust over the flaps, so that the area of heat-resistant material that has to be covered is less than with the USB. This makes the weight penalty less.

• This, and its relative simplicity, may be the reason that the USB approach appears to be the most popular means for active lift enhancement used by aircraft manufacturers.

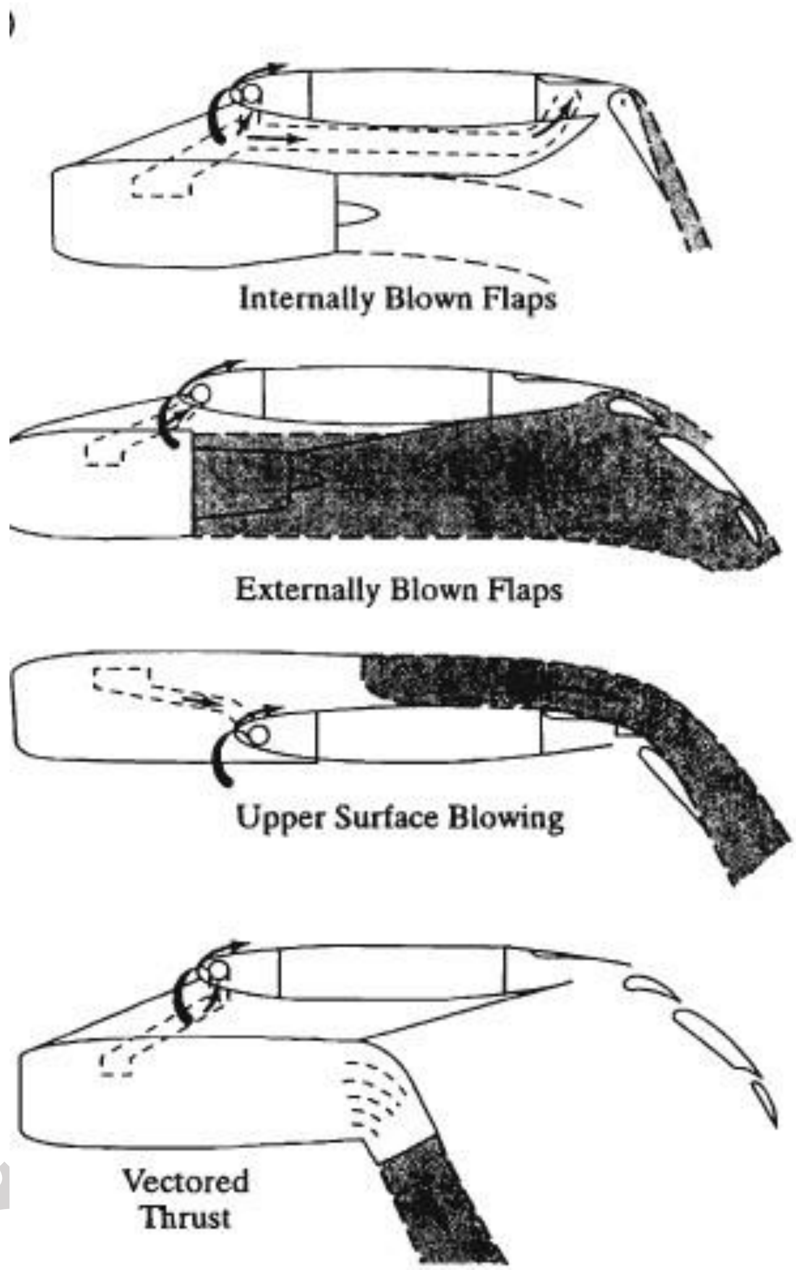


Figure 3.14: Active lift enhancement devices

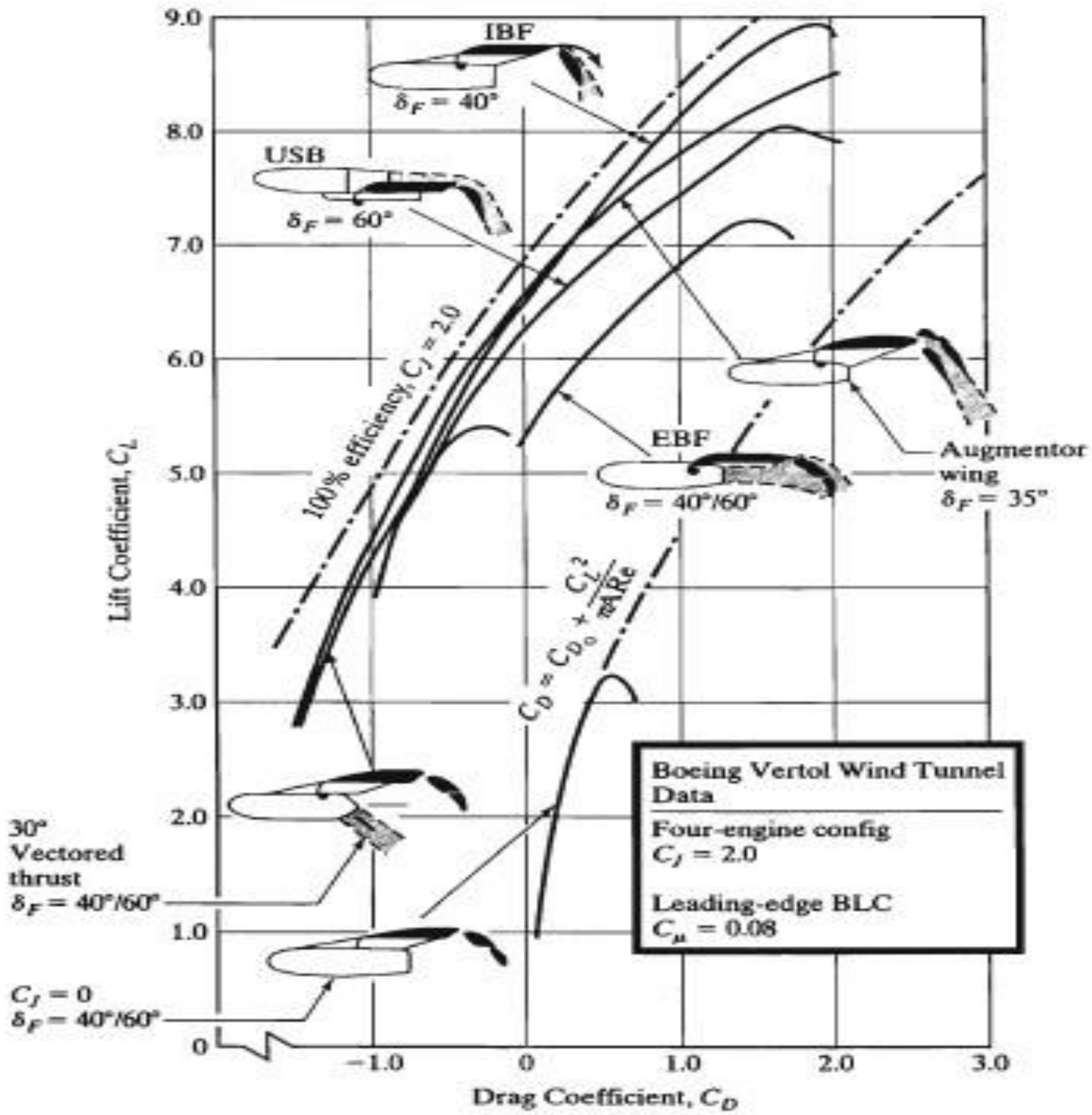


Figure 3.15: Drag polars for Active lift enhancement approaches

Design Spread Sheet for Lift Enhancement

Wing Data:			
Airfoil	NACA	64x	
Λ_{LE}	62	deg	
λ	0.00		
t/c	0.04		
T-O Mach No.	0.1		
β	0.99		
A	2		
$\Lambda_{t/c}$	47.2	deg	
$C_{l\alpha}$(no flap)	0.11	1/deg	
$C_{L\alpha}$ (no flap)	0.04	1/deg	
α_{0L}	−1	deg	
C_{lmax}	1.4		
α_s	12	deg	
Trailing-edge Flap Design:			
Flap type	slot	slot, plane or split	
S_f/S_w	*0.60*		
δ_f	*40*	deg	
c_f/c	*0.25*		
Delta α0L:			
	Plane Flap		
K'	0.58		Fig. 9.3
$dC_l/d\delta_f$	*0.5*		Fig. 9.4
$\Delta\alpha_{0l}$	*−18.06*	deg	
	Single Slotted & Fowler Flap		
$d\alpha/d\delta_f$	−0.4		Fig. 9.5
$\Delta\alpha_{0l}$	−20	deg	
	Split Flaps		
k	1.1		Fig. 9.6
ΔCl	0.8		Fig. 9.7
$\Delta\alpha_{0l}$	−8	deg	
Aspect Ratio Criterion:			
C_1	0		Fig. 9.8
High A criteria	8.52	Low	
Basic Wing-High Aspect Ratio:			
Δy	0.8	%	Fig. 9.10
C_{Lmax}/C_{lmax}	1.3		Fig. 9.9
C_{Lmax}	1.82		
$\Delta\alpha_{CLmax}$	12.5	deg	Fig. 9.11
α_s	52.86	deg	