LECTURE NOTES

ON

FLIGHT VEHICLE DESIGN

B.Tech VII Sem (IARE-R16)

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UNIT-I
OVERVIEW OF THE DESIGN PROCESS

1.1. INTRODUCTION

Those involved in design can never quite agree as to just where the design process begins. The designer thinks it starts with a new airplane concept. The sizing specialist knows that nothing can begin until an initial estimate of the weight is made. The customer, civilian or military, feels that the design begins with requirements.

They are all correct. Actually, design is an iterative effort, as shown in the "Design Wheel" of Fig 1.1. Requirements are set by prior design trade studies. Concepts are developed to meet requirements. Design analysis frequently points toward new concepts and technologies, which can initiate a whole new design effort. However a particular design is begun, all of these activities are equally important in producing a good aircraft concept

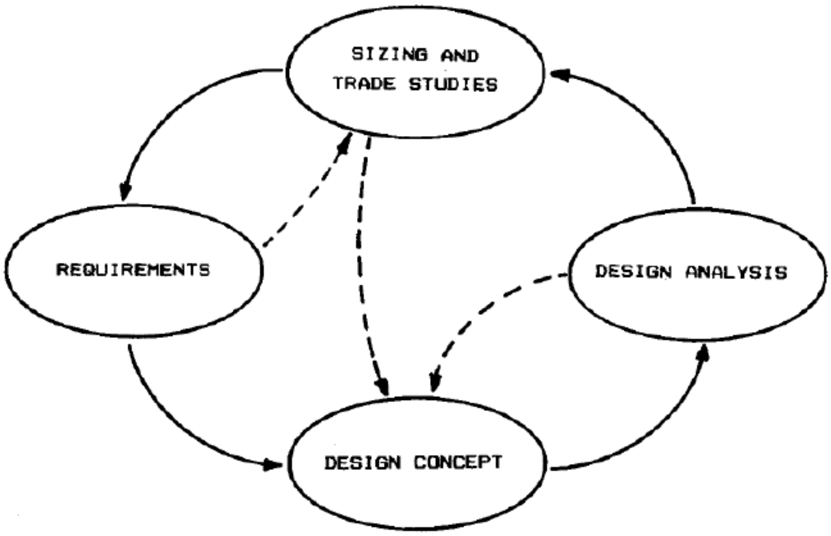

Figure 1.1: The design wheel

1.2. PHASES OF AIRCRAFT DESIGN

1.2.1. Conceptual Design

Aircraft design can be broken into three major phases, as depicted. Conceptual design is the primary focus of this chapter. It is in conceptual design that the basic questions of configuration arrangement size and weight, and performance are answered. The first question is, "Can an affordable aircraft be built that meets the requirements?" If not, the customer may wish to relax the requirements. Conceptual design is a very fluid process.

New ideas and problems emerge as a design is investigated in ever-increasing detail. Each time the latest design is analyzed and sized, it must be redrawn to reflect the new gross weight, fuel weight, wing size, engine size, and other changes. Early 1nner tests often reveal problems requiring some changes to the configuration. The steps of conceptual design are described later in more detail.

1.2.2. Preliminary Design

Preliminary design can be said to begin when the major changes are over. The big questions such as whether to use a canard or an aft tail have been resolved. The configuration arrangement can be expected to remain about as shown on current drawings, although minor revisions may occur. At some Point late in preliminary design, even minor changes are stopped when a decision is made to freeze the configuration.

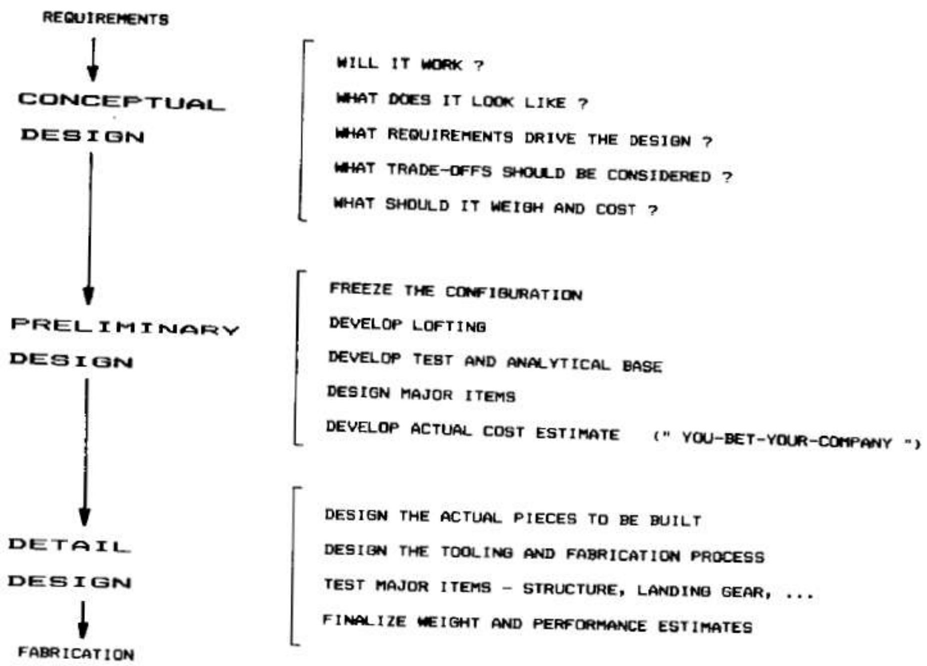

Figure 1.2: Three phases of aircraft design

During preliminary design the specialists in areas such as structures, landing gear, and control systems will design and analyze their portion of the aircraft. Testing is initiated in areas such as aerodynamics, propulsion, structures, and stability and control. A mockup may be constructed at this point.

A key activity during preliminary design is "lofting." Lofting is the mathematical modeling of the outside skin of the aircraft with sufficient accuracy to insure proper fit between its different parts, even if they are designed by different designers and possibly fabricated in different locations. Lofting originated in shipyards and was originally done with long flexible rulers called "splines." This work was done in a loft over the shipyard; hence the name.

The ultimate objective during preliminary design is to ready the company for the detail design stage, also called full-scale development. Thus, the end of preliminary design usually involves a full-scale development proposal. In today's environment, this can result in a situation jokingly referred to as "you-bet-your-company." The possible loss on an overrun contract or from lack of sales can exceed the net worth of the company! Preliminary design must establish confidence that the airplane can be built on time and at the estimated cost.

Assuming a favorable decision for entering full-scale development, the detail design phase begins in which the actual pieces to be fabricated are designed. For example, during conceptual and preliminary design the wing box will be designed and analyzed as a whole. During detail design, that whole will be broken down into individual ribs, spars, and skins, each of which must be separately designed and analyzed.

Another important part of detail design is called production design. Specialists determine how the airplane will be fabricated, starting with the smallest and simplest subassemblies and building up to the final assembly process. Production designers frequently wish to modify the design for ease of manufacture; that can have a major impact on performance or weight. Compromises are inevitable, but the design must still meet the original requirements.

It is interesting to note that in the Soviet Union, the production design is done by a completely different design bureau than the conceptual and preliminary design, resulting in superior producibility at some expense in performance and weight.

During detail design, the testing effort intensifies. Actual structure of the aircraft is fabricated and tested. Control laws for the flight control system are tested on an "iron-bird" simulator, a detailed working model of the actuators and flight control surfaces. Flight simulators are developed and flown by both company and customer test-pilots.

Detail design ends with fabrication of the aircraft. Frequently the fabrication begins on part of the aircraft before the entire detail-design effort is completed. Hopefully, changes to already-fabricated pieces can be avoided.

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1.2.3. Detail Design

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Detail design ends with fabrication of the aircraft. Frequently the fabrication begins on part of the aircraft before the entire detail-design effort is completed. Hopefully, changes to already-fabricated pieces can be avoided. The further along a design progresses, the more people are involved. In fact, most of the engineers who go to work for a major aerospace company will work in preliminary or detail design

1.3. AIRCRAFT CONCEPTUAL DESIGN PROCESS

Figure 1.3 depicts the conceptual design process in greater detail. Conceptual design will usually begin with either a specific set of design requirements established by the prospective customer or a company-generated guess as to what future customers may need. Design requirements include parameters such as the aircraft range and payload, takeoff and landing distances, and maneuverability and speed requirements.

The design requirements also include a vast set of civil or military design specifications which must be met. These include landing sink-speed, stall speed, structural design limits, pilots' outside vision angles, reserve fuel, and many others. Sometimes a design will begin as an innovative idea rather than as a response to a given requirement. The flying wings pioneered by John Northrop were not conceived in response to a specific Army Air Corps requirement at that time, but instead were the product of one man's idea of the "better airplane." Northrop pursued this idea for years before building a flying wing to suit a particular military requirement.

Before a design can be started, a decision must be made as to what technologies will be incorporated. If a design is to be built in the near future, it must use only currently-available technologies as well as existing engines and avionics. If it is being designed to be built in the more distant future, then an estimate of the technological state of the art must be made to determine which emerging technologies will be ready for use at that time.

For example, an all-composite fighter has yet to enter high-rate promotion as of this date (1989), but can confidently be predicted by 1999. On the other hand, active laminar flow control by suction pumps shows great payoff analytically, but would be considered by many to be too risky to incorporate into a new transport jet in the near future. An optimistic estimate of the technology availability will yield a lighter, cheaper aircraft to perform a given mission, but will also result in a higher development risk. The actual design effort usually begins with a conceptual sketch (Fig.1.3). This is the "back of a napkin" drawing of aerospace legend, and gives a rough indication of what the design may look like. A good conceptual sketch will include the approximate wing and tail geometries, the fuselage shape, and the internal locations of the major components such as the engine, cockpit, payload/passenger compartment, landing gear, and perhaps the fuel tanks. The conceptual sketch can be used to estimate aerodynamics and weight fractions by comparison to previous designs.

These estimates are used to make a first estimate of the required total weight and fuel weight to perform the design mission, by a process called "sizing." The conceptual sketch may not be needed for initial sizing if the design resembles previous ones. The "first-order" sizing provides the information needed to develop an initial design layout (Fig1.3). This is a three-view drawing complete with

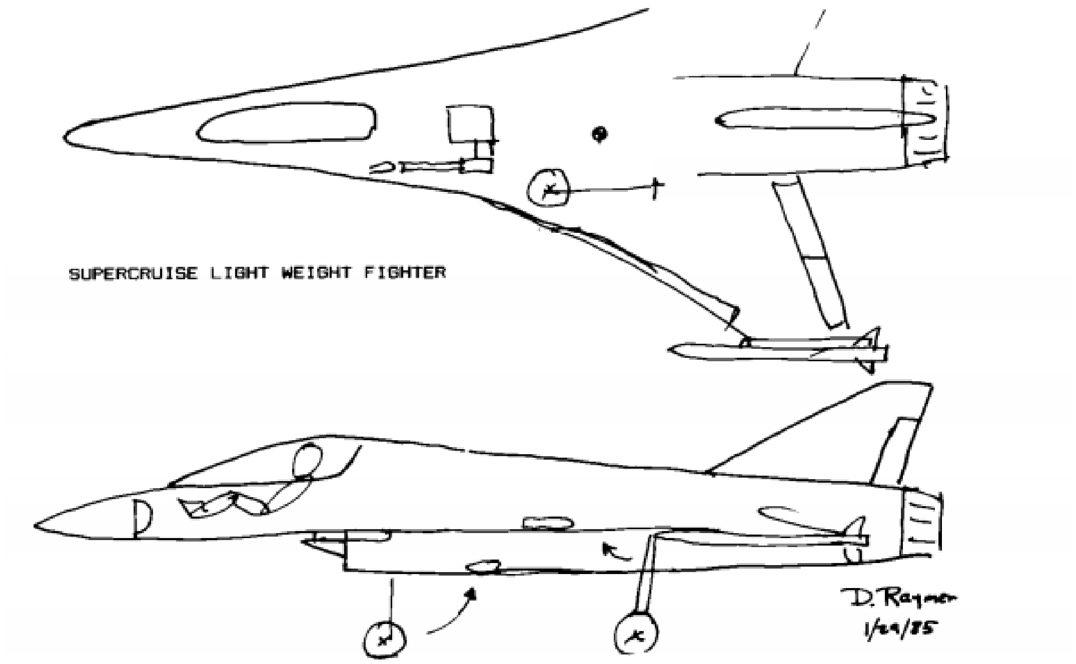


Figure 1.3 Initial Sketche

The more important internal arrangement details, including typically the landing gear, payload or passenger compartment, engines and inlet ducts, fuel tanks, cockpit, major avionics, and any other internal components which are large enough to affect the overall shaping of the aircraft. Enough cross-sections are shown to verify that everything fits. On a drafting table, the three-view layout is done in some convenient scale such as 1/10, 1/20, 1/40, or 1/100 (depending upon the size of the airplane and the available paper). On a computer-aided design system, the design work is usually done in full scale (numerically).

This initial layout is analyzed to determine if it really will perform the mission as indicated by the first-order sizing. Actual aerodynamics, weights, and installed propulsion characteristics are analyzed and subsequently used to do a detailed sizing calculation. Furthermore, the performance capabilities of the design are calculated and compared to the requirements mentioned above. Optimization techniques are used to find the lightest or lowest-cost aircraft that will both perform the design mission and meet all performance requirements.

The results of this optimization include a better estimate of the required total weight and fuel weight to meet the mission. The results also include required revisions to the engine and wing sizes.

This frequently requires a new or revised design layout, in which the designer incorporates these changes and any others suggested by the effort to date. The revised drawing, after some number of iterations, is then examined in detail by an ever-expanding group of specialists, each of whom insures that the design meets the requirements of that specialty.

1.4. SIZING FROM A CONCEPTUAL SKETCH

There are many levels of design procedure. The simplest level just adopts past history. For example, if you need an immediate estimate of the takeoff weight of an airplane to replace the Air Force F-15 fighter, use 44,500 lb. That is what the F-15 weighs, and is probably a good number to start with. To get the "right" answer takes several years, many people, and lots of money. Actual design requirements must be evaluated against a number of candidate designs, each of which must be designed, analyzed, sized, optimized, and redesigned any number of times. Analysis techniques include all manner of computer code as well as correlations to wind-tunnel and other tests. Even with this extreme level of design sophistication, the actual airplane when flown will never exactly match predictions. In between these extremes of design and analysis procedures lie the methods used for most conceptual design activities. As an introduction to the design process, this chapter presents a quick method of estimating takeoff weight from a conceptual sketch. The simplified sizing method presented can only be used for missions who do not include any combat or payload drops. While admittedly crude, this method introduces all of the essential features of the most sophisticated design by the major aerospace manufacturers. In a later chapter, the concepts introduced here will be expanded to a sizing method capable of handling all types of mission

1.4.1 Takeoff-Weight Buildup

Design takeoff gross weight" is the total weight of the aircraft as it begins the mission for which it was designed. This is not necessarily the same as the "maximum takeoff weight." Many military aircraft can be overloaded beyond design weight but will suffer a reduced maneuverability. Unless specifically mentioned, takeoff gross weight, or "W_0 ," is assumed to be the design weight. Design takeoff gross weight can be broken into crew weight, payload (or passenger~ weight, fuel weight, and the remaining (or "empty") weight. The empty weight includes the structure, engines, landing gear, fixed equipment, avionics, and anything else not considered a part of crew, payload, or fuel. Equation summarizes the takeoff-weight buildup

$$W_o = W_{Crew} + W_{payload} + W_{fuel} + W_{empt}$$

The crew and payload weights are both known since they are given in the design requirements. The only unknowns are the fuel weight and empty weight. However, they are both dependent on the total aircraft weight. Thus an iterative process must be used for aircraft sizing. To simplify the calculation, both fuel and empty weights can be expressed as fractions of the total takeoff weight, i.e., (W_F/W_0) and (W_e/W_0).

$$W_0 = W_{\text{crew}} + W_{\text{payload}} + W_{\text{fuel}} + W_{\text{empty}}$$

This can be solved for W0 as follows:

$$W_0 = W_{\text{crew}} + W_{\text{payload}} + \left(\frac{W_f}{W_0}\right)W_0 + \left(\frac{W_e}{W_0}\right)W_0$$

$$W_0 - \left(\frac{W_f}{W_0}\right)W_0 - \left(\frac{W_e}{W_0}\right)W_0 = W_{\text{crew}} + W_{\text{payload}}$$

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

Now W_0 can be determined if (W_F/W_o) and $(W_e!W_o)$ can be estimated

1.4.2. Empty-Weight Estimation

The empty-weight fraction (W_e/W_0) can be estimated statistically from historical trends as shown in Fig. 1.4, developed by the author from data taken and other sources. Empty-weight fractions vary from about 0.3 to 0.7, and diminish with increasing total aircraft weight. As can be seen, the type of aircraft also has a strong effect, with flying boats having the highest empty-weight fractions and long-range military aircraft having the lowest. Flying boats are heavy because they need to carry extra weight for what amounts to a boat hull.

Notice also that different types of aircraft exhibit different slopes to the trend lines of empty-weight fraction vs. takeoff weight Table presents statistical curve-fit equations for the trends shown in Fig.1.4. Note that these are all exponential equations based upon takeoff gross weight. The exponents are small negative numbers, which indicates that the empty weight fractions decrease with increasing takeoff weight, as shown by the trend lines in Fig 1.3.

The differences in exponents for different types of aircraft reflect the different slopes to the trend lines, and imply that some types of aircraft are more sensitive in sizing than others .A variable-sweep wing is heavier than a fixed wing, and is accounted for at this initial stage of design by multiplying the empty-weight fraction as determined from the equations.

Table 1.1 Empty weight fraction vs W_e

$W_e/W_0 = A W_0^C K_{vs}$	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

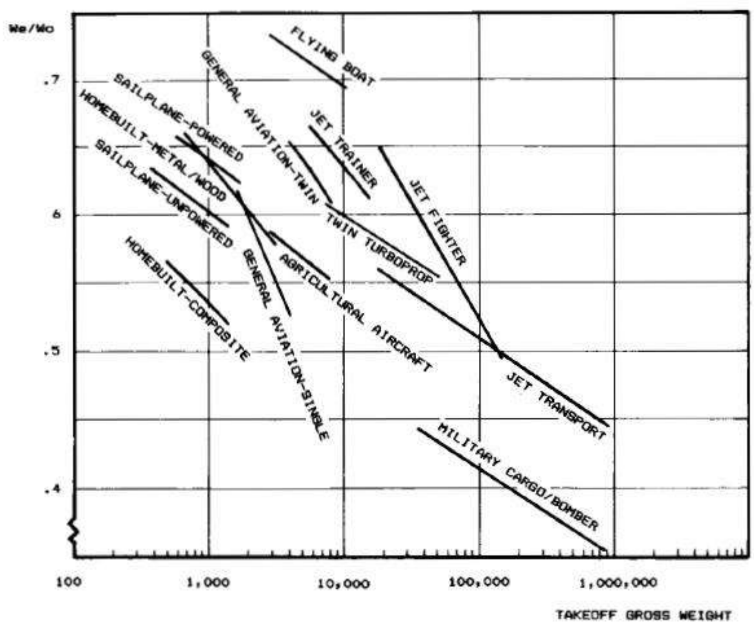


Figure 1.4: Empty weight fraction trends

Advanced composite materials such as graphite-epoxy are replacing aluminum in a number of new designs. There have not yet been enough composite aircraft flown to develop statistical equations. Based on a number? Design studies, the empty-weight fraction for other types of composite aircraft can be estimated at this stage by multiplying the statistical empty weight fraction by 0.95."Composite" homebuilt aircraft are typically of fiberglass-epoxy construction rather than an advanced composite material.

The statistically estimated empty weight fraction for fiberglass-epoxy composite home built is approximately0.85 times the metal homebuilt empty-weight fraction (0.99/1.19=0.85). However, this is not due to the material used for construction as much as the different design philosophies concerning ease of manufacture, passenger comfort, maintenance accessibility, and similar factors

1.5. FUEL -FRACTION ESTIMATION

Only part of the aircraft's fuel supply is available for performing the mission ("mission fuel"). The other fuel includes reserve fuel as required by civil or military design specifications, and also includes "trapped fuel," which is the fuel which cannot be pumped out of the tanks.

The required amount of mission fuel depends upon the mission the flown, the aerodynamics of the aircraft, and the engine's fuel consumption the aircraft weight during the mission affects the drag, so the fuel used is a function of the aircraft weight.

As a first approximation, the fuel used can be considered to be proportional to the aircraft weight, so the fuel fraction (Wt/Wo) is approximately independent of aircraft weight. Fuel fraction can be estimated base on the mission to be flown using approximations of the fuel consumption and aerodynamics

1.5.1. Mission Profiles

Typical mission profiles for various types of aircraft are shown in Fig. 1.5. The Simple Cruise mission is used for many transport and general aviation designs, including home built. The aircraft is sized to provide some required cruise range. For safety you would be wise to carry extra fuel m case your intended airport is closed, loiter of typically 20-30 min is added.

Alternatively, additional range could be included, representing the distance to the nearest other airport or some fixed number of minutes of flight at cruise speed (the FAA requires 30 min of additional cruise fuel for general-aviation aircraft). Other missions are more complex.

The typical Air Superiority mission includes a cruise out, a combat consisting of either a certain number of turns or a certain number of minutes at maximum power, a weapons drop, a cruise back, and a loiter. The weapons drop refers to the firing of gun and missiles, and is often left out of the sizing analysis to insure that the aircraft has enough fuel to return safely if the weapons aren't used. Note that the second cruise segment is identical to the first, indicating that the aircraft must return to its base at the end of the mission.

The Low-Level Strike mission includes "dash" segments that must be flown at just a few hundred feet off the ground. This is to improve the survivability of the aircraft as it approaches its target. Unfortunately, the aerodynamic efficiency of an aircraft, expressed as "lift-to-drag ratio" (LID), is greatly reduced during low-level, high-speed flight, as is the engine efficiency. The aircraft may burn almost as much fuel during the low-level dash segment as it burns in the much-longer cruise segment. The Strategic Bombing mission introduces another twist.

After the initial cruise, a refueling segment occurs, as indicated by an "R." Here the aircraft meets up with a tanker aircraft such as an Air Force KC-135 and receives some quantity of fuel.

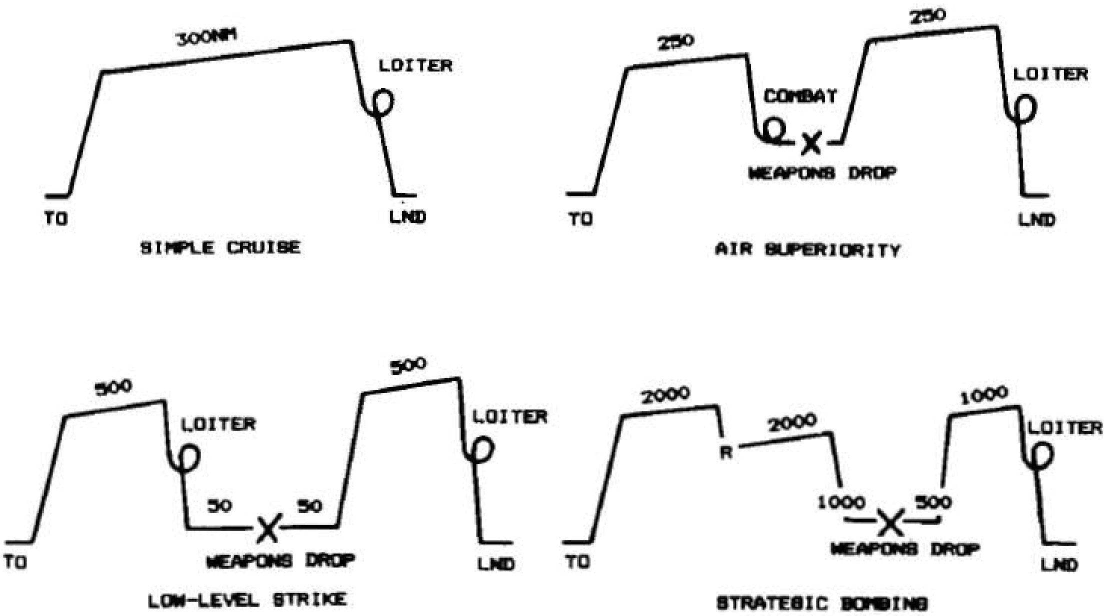


Figure 1.5: Typical mission profiles for sizing

This enables the bomber to achieve far more range, but adds to the overall operating cost because a fleet of tanker aircraft must be dedicated to up porting the. Bombers. Also note that the bomber in this typical strategic mission will fly at low level as it nears the target area to improve its chances of survival. Earlier bombers such as the B-52 were originally designed to cruise at high altitudes throughout the mission. Another difference in this strategic mission is the fact that the return cruise range is far shorter than the outbound range.

This is necessary because of the extreme range required. If the aircraft were sized to return to its original base, it would probably weigh several million pounds. Instead, it is assumed that strategic bombers will land on bases in friendly countries for refueling after completion of their mission. These are merely typical missions, and the ranges shown are just examples. When an aircraft is designed, the actual mission profile and ranges will be provided by the customer or determined by operational analysis methods beyond the scope of this book. In addition to the mission profile, requirements will be established for a number of performance parameters such as takeoff distance values can vary somewhat depending on aircraft type, but the averaged values given in the table are reasonable for initial sizing.

In our simple sizing method we ignore descent, assuming that the cruise ends with a descent and that the distance traveled during descent is part of the cruise range. Cruise-segment mission weight fractions can be found using the Breguet range equation

$$R = \frac{V}{C}\frac{L}{D} \ln \frac{W_{i-1}}{W_i}$$

$$\frac{W_i}{W_{i-1}} = \exp \frac{-RC}{V(L/D)}$$

Where

R =range
C = specific fuel consumption
V =velocity
LID = lift-to-drag ratio

Loiter weight fractions are found from the endurance equation

$$E = \frac{L/D}{C} \ln \frac{W_{i-1}}{W_i}$$

$$\frac{W_i}{W_{i-1}} = \exp \frac{-EC}{L/D}$$

Where E = endurance or loiter time. (Note: It is very important to use consistent units! Also note that C and L/D vary with speed and altitude. Furthermore, C varies with throttle setting, and L/D varies with aircraft weight. These will be discussed in detail in later chapters.)

1.5.2. Specific Fuel Consumption

Specific fuel consumption ("SFC" or simply "C") is the rate of fuel consumption divided by the resulting thrust. For jet engines, specific fuel consumption is usually measured in pounds of fuel per hour per pound of thrust [(lb/hr)/lb, or 11hr]. Figure shows SFC vs. Mach number. Propeller engine SFC is normally given, the pounds of fuel per hour to produce one horsepower at the propeller shaft (or one "brake horsepower": bhp = 550 ft.-Ibis).A propeller thrust SFC equivalent to the jet-engine SFC can be calculated. The engine produces thrust via the propeller, which has efficiency

Table 1.2: Specific fuel consumption

Typical jet SFC's	Cruise	Loiter
Pure turbojet	0.9	0.8
Low-bypass turbofan	0.8	0.7
High-bypass turbofan	0.5	0.4

Table 1.3. Propeller specific fuel consumption(C_{bhp})

Propeller: $C = C_{bhp} V/(550\eta_p)$ Typical C_{bhp} and η_p	Cruise	Loiter
Piston-prop (fixed pitch)	0.4/0.8	0.5/0.7
Piston-prop (variable pitch)	0.4/0.8	0.5/0.8
Turboprop	0.5/0.8	0.6/0.8

1.5.3. L/D Estimation

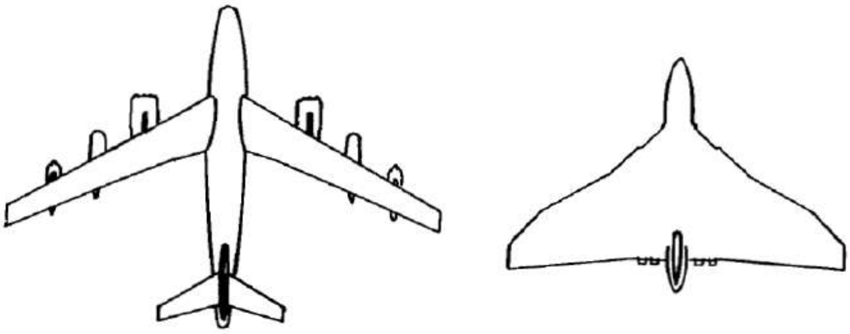
The remaining unknown in both range and loiter equations is the L/D, or l ift-to-drag ratio, which is a measure of the design's overall aerodynamic efficiency. Unlike the parameters estimated above, the LID is highly dependent upon the configuration arrangement. At subsonic speeds LID is most directly affected by two aspects of the design: wing span and wetted area.

In level flight, the lift is known. It must equal the aircraft weight. Thus, L/D is solely dependent upon drag.The drag at subsonic speeds is composed of two parts. "Induced" drag is the drag caused by the generation of lift. This is primarily a function of the wing span. "Zero-lift," or "parasite" drag is the drag which is not related to lift. This is primarily skin-friction drag, and as such is directly proportional to the total surface area of the aircraft exposed ("wetted") to the air.

The "aspect ratio" of the wing has historically been used as the primary indicator of wing efficiency. Aspect ratio is defined as the square of the wing span divided by the wing reference area. For a rectangular wing the aspect ratio is simply the span divided by chord. Aspect ratios range from under one for re-entry lifting bodies to over thirty for sailplanes. Typical values range between three and eight. For initial design purposes, aspect ratio can be selected from historical data.

For final determination of the best aspect ratio, a trade study as discussed in should be conducted. Aspect ratio could be used to estimate subsonic LID, but for one major problem. The parasite drag is not a function of just the wing area, as expressed by aspect ratio, but also of the aircraft's total wetted area. Figure 1.4 shows two widely different aircraft concepts, both designed to perform the same mission of strategic bombing

The Boeing B-47 features a conventional approach. With its aspect ratio of over 9, it is not surprising



	B-47	AVRO VULCAN
S reference	1430	3446
S wetted	11300	9600
SPAN	116	90
Swet/Sref	7.9	2.8
ASPECT RATIO	9.4	3.0
WETTED ASPECT RATIO	1.2	1.1
L/D max	17.2	17.0

Figure 1.6 Does aspect ratio predict drag?

Figure 1.6 plots maximum *L/D* number of aircraft vs. the wetted aspect ratio, and shows clear trend Jet, prop, and fixed-gear pop aircraft. Note that the wetted aspect ratio can be shown to equal the wing geometric aspect ratio divided by the wetted-area ratio, Swet/ Sref.

It should be clear at this point that the designer has control over the *L/D*. The designer picks the aspect ratio and determines the configuration arrangement, which in turn determines the wetted area rat10.

However, the designer must strike a compromise between a high *L/D* and the conflicting desire for low weight. The statistical equations provided above for estimating the empty-weight fraction are based on "normal" If the aspect ratio selected is much higher than that of other aircraft in its class, the empty-weight fraction would be higher than estimated by these simple statistical equations. *L/D* can now be estimated from a conceptual sketch.

This is the crude, "back of a napkin" drawing mentioned earlier*?* the conceptua1 sketch the designer arranges the major components of the aircraft, including wings, tails, fuselage, engines, payload or passenger compartment, landing gear, fuel tanks, and others as needed From the sketch the wetted-area ratio can be "eyeball-estimated" using Fig. 1.5 for guidance.

The wetted aspect ratio can then be calculated as the wing aspect ratio divided by the wetted-area ratio. Figure 1.6 can then be used to estimate the maximum L/D. For initial sizing, these percent's can be multiplied times the maximum LID as estimated using Fig. 1.6 to determine the L/D for

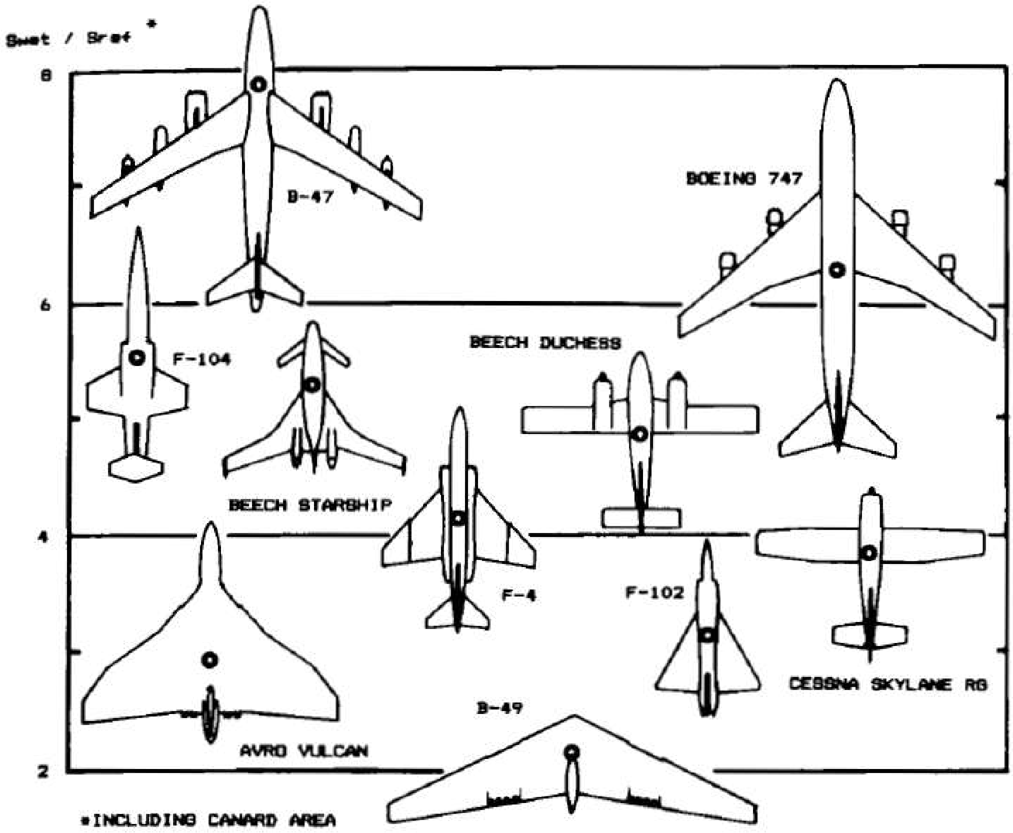


Figure 1.7: Wetted area ratios

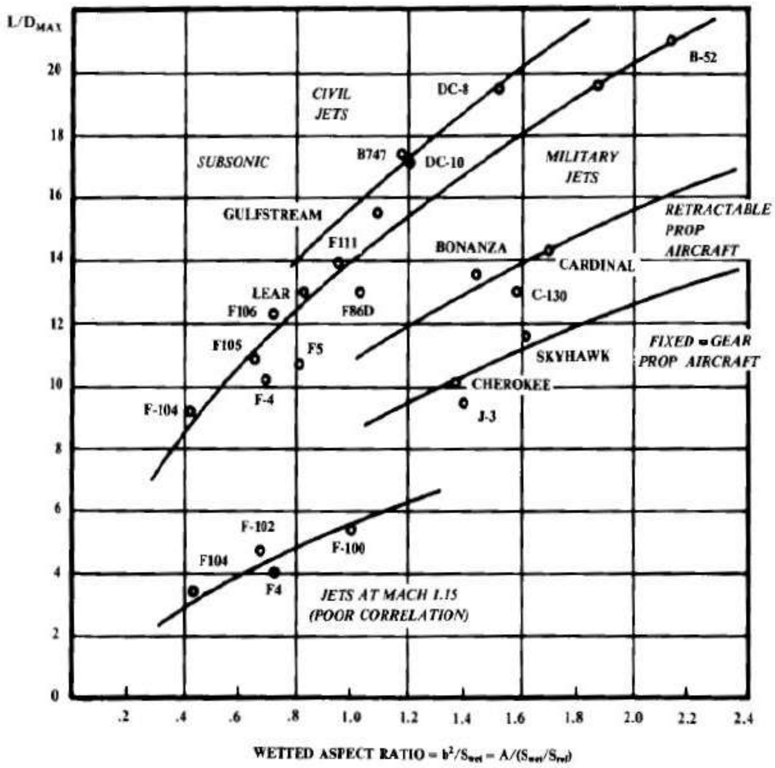


Figure 1.8: maximum lift to drag ratio trends

1.6. FUEL-FRACTION ESTIMATION

Using historical values from Table 1.2 and the equations for cruise and loiter segments, the mission-segment weight fractions can now be estimated. By multiplying them together, the total mission weight fraction, Wx/Wo, can be calculated. Since this simplified sizing method does not allow mission segments involving payload drops, all weight lost during the mission must be due to fuelusage. The mission fuel fraction must therefore be equal to (1 – Wx/Wo). If you assume, typically, a 6% allowance for reserve and trapped fuel, the total fuel fraction can be estimated as in Eq.

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

1.6.1. Takeoff-Weight Calculation

Using the fuel fraction found with above Eq .and the statistical empty weight equation selected from the takeoff gross weight can be found iteratively from Eq This is done by guessing the takeoff gross weight, calculating the statistical empty-weight fraction, and then calculating the takeoff gross weight. If the result doesn't match the guess value, a value between the two is used as the next guess. This wills usually converging just a few iterations. This first-order sizing process is diagrammed in fig

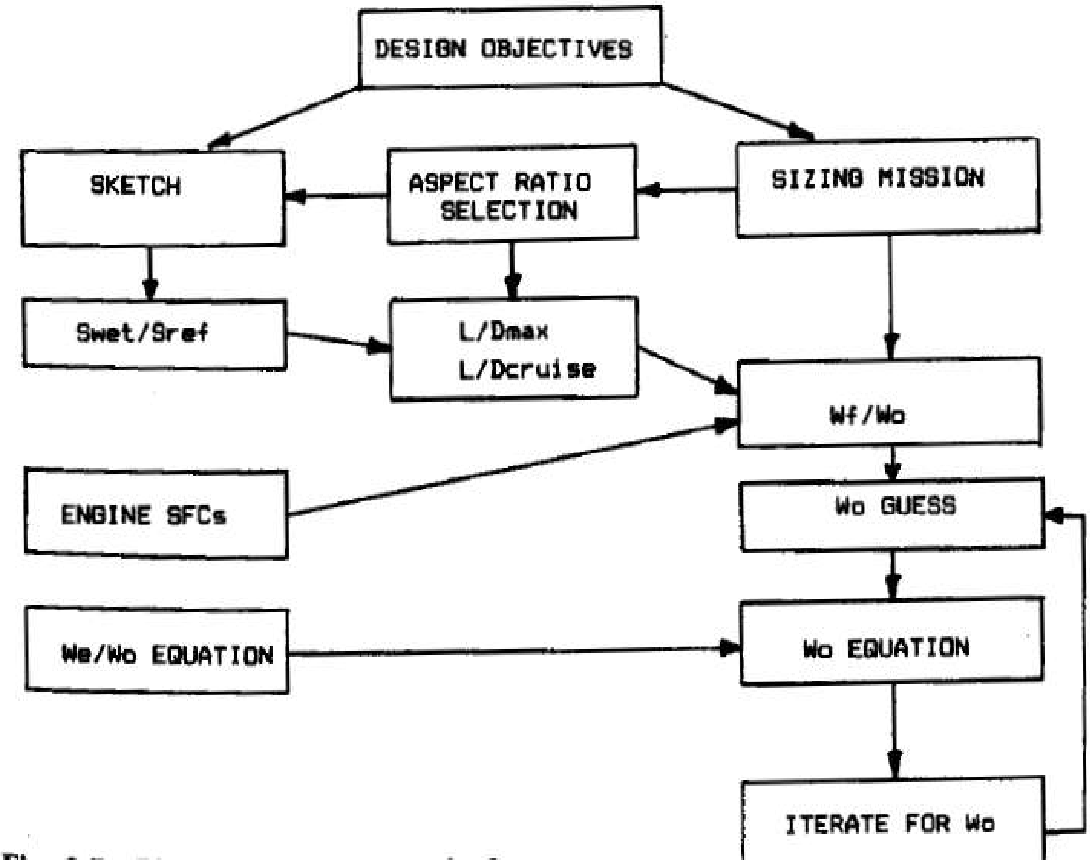


Figure 1.9: First order design method

1.6.2. Design Example: ASW Aircraft

ASW Requirements

Figure illustrates e mission requirement for a hypothetical antisubmarine warfare (ASW) aircraft the key. Requirement is the ability to loiter for three hours loitering distance o 1500 n.m1. From the takeoff point. While loitering on station

This type of aircraft uses sophisticated electronic equipment to detect and track submarines. For the sizmg example is assumed to weigh 10000 lb. Also a four-man crew is required, totaling 800 lb. The aircraft must cruise at 0.6 Mach number

Conceptual Sketches
Figure shows four conceptual approaches considered by the designer these mission requirements. Concept one is the conventional.

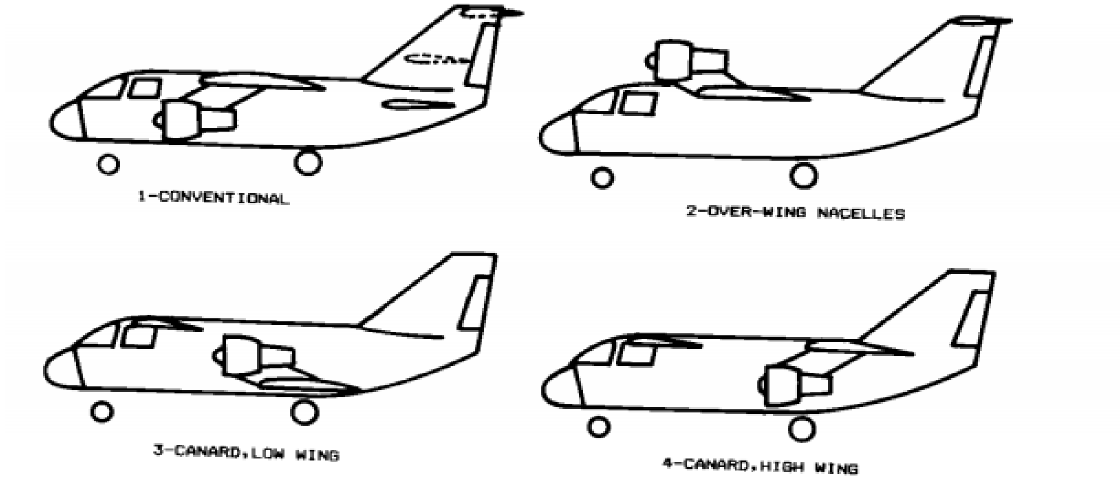
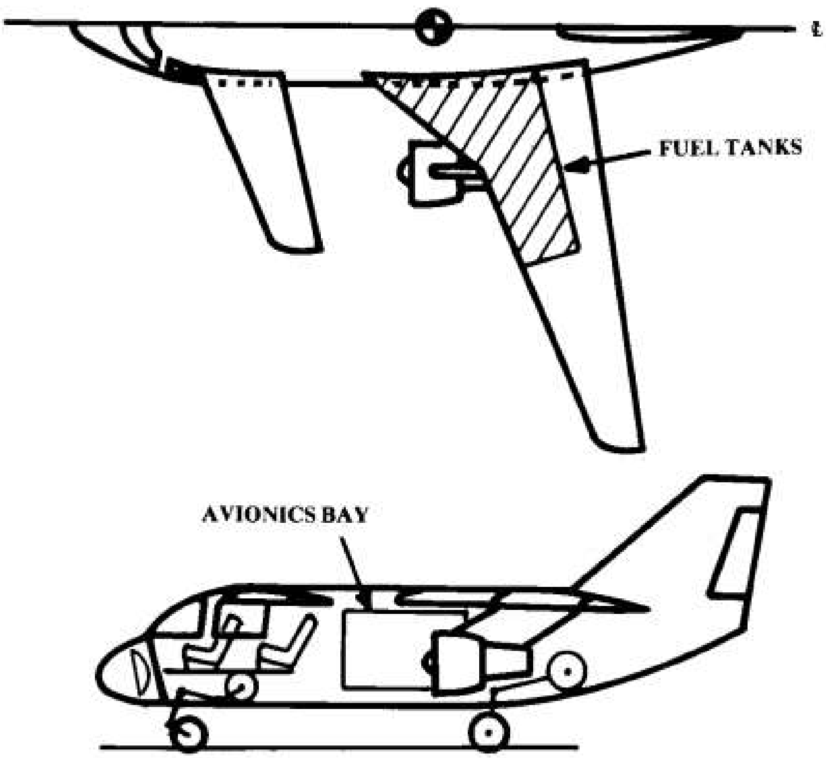


Figure 1.10: ASW concept sketches

Approach, looking much like the Lockheed S-3A that currently performs a similar mission. The low horizontal tail position shown in solid line would offer the lightest structure, but may place the tail in the exhaust stream of the engines, so other positions for the horizontal tail are shown in dotted lines the second concept is much like the first except for the engine location.

Here the engines are shown mounted over the wing. This provides extra lift due to the exhaust over the wings, and also provides greater ground clearance for the engines, which reduces the tendency of the jet engines to suck up debris. However, the disadvantage of this concept is the difficulty in reaching the engines for maintenance work. Concepts three and four explore the canard approach. Canards offer the potential for reduced trim drag and may provide a wider allowable range for the center of gravity. In concept three, the wing is low and the engines are mounted over the wing as in concept two. This would allow the main landing gear to be stowed in the wing root. In concept four, the wing is high with the engines mounted below.

This last approach offers better access to the engines and for this reason was selected for further development. Figure 1 .11 is a conceptual sketch prepared, in more detail, for the selected concept. Note the locations indicated for the landing-gear stowage, crew station, and fuel tanks. The fuel tanks should be placed so that the fuel is evenly distributed about the aircraft center of gravity (estimated location shown by the circle with two quarters



shaded).

Figure: 1.11 completed ASW sketch

This is necessary so that the aircraft when loaded has nearly the same center of gravity as when its fuel is almost gone. However, the wing is located aft of the center of gravity whenever a canard is used, so the fuel located in the wing is also aft of the center of gravity. One solution to this problem would be to add fuel tanks in the fuselage, forward of the center of gravity.

This would increase the risk of fire in the fuselage during an accident, and is forbidden in commercial aircraft. Although this example is a military aircraft, fire safety should always be considered. Another solution, shown on the sketch, is to add a wing strake full of fuel. This solution is seen on the Beech Starship among others. The strakes do add to the aircraft wetted area, which reduces cruise aerodynamic efficiency.

This example serves to illustrate an extremely important principle of aircraft design; namely, that there is no such thing as a free lunch! All aircraft design entails a series of tradeoffs. The canard offers lower trim drag, but in this case may require a higher wetted area. The only true way to determine whether a canard is a good idea for this or any aircraft is to design several aircraft, one with and one without a canard. This type of trade study comprises the majority of the design effort during the conceptual design process.

1.6.3. L/D Estimation

For initial sizing, a wing aspect ratio of about 11 was selected. With the area of the wing and canard both included, this is equivalent to a combined aspect ratio of about 8. Comparing the sketch to the examples of Fig. 3.5, it would appear that the wetted area ratio (S_{wet}/S_{ref}) is about 5.5. This yields a wetted aspect ratio of 1.45 (i.e., 8/5.5). For a wetted aspect ratio of 1.45, Fig. 1.6 indicates that a maximum lift-to-drag ratio of about 17 would be expected. This value, obtained from an initial sketch and the selected aspect ratio, can now be used for initial sizing. Since this is a jet aircraft, the maximum L/D is used for loiter calculations. For cruise, a value of 0.866 times the maximum L/D, or about 15, is used

1.6.4. Takeoff-Weight Sizing

From, initial values for SFC are obtained. For a subsonic aircraft the best SFC values are obtained with high-bypass turbofans, which have typical values of about 0.5 for cruise and 0.4 for loiter.
Does not provide an equation for statistically estimating the empty weight fraction of an antisubmarine aircraft. However, such an aircraft is basically designed for subsonic cruise efficiency so the equation for military cargo/bomber can be used. The extensive ASW avionics would not be included in that equation, so it is treated as a separate payload weight.

Gives the calculations for sizing this example. Note the effort _to insure consistent dimensions, including the convers10n of cruise velocity (Mach 0.6) to ft/s by assuming a typical cruise altitude of 30,000 ft. At t is altitude the speed of sound is 994.8 ft/s.

The calculations in box indicate a takeoff gross weight of 59,310 lb. Although these calculations are based upon crude estimates of aerodynamics, weights, and propulsion parameters, it is interesting to note that the actual takeoff gross weight. While strict accuracy should not be expected, this simple size method will usually yield an answer in the "right ballpark."

ASW sizing calculations

Mission Segment Weight Fractions

1) Warmup and takeoff	$W_1/W_0 = 0.97$	(Table 2)
2) Climb	$W_2/W_1 = 0.985$	(Table 2)
3) Cruise	$R = 1500\text{nm} = 9{,}114{,}000 \text{ ft}$	
	$C = 0.5\ 1/\text{hr} = 0.0001389\ 1/\text{s}$	
	$V = 0.6\text{M} \times (994.8 \text{ ft/s}) = 569.9 \text{ ft/s}$	
	$L/D = 16 \times 0.866 = 13.9$	
	$W_3/W_2 = e^{[-RC/VL/D]} = e^{-0.16} = 0.852$	
4) Loiter	$E = 3 \text{ hours} = 10{,}800 \text{ s}$	
	$C = 0.4\ 1/\text{hr} = 0.0001111\ 1/\text{s}$	
	$L/D = 16$	
	$W_4/W_3 = e^{[-EC/L/D]} = e^{-0.075} = 0.9277$	
5) Cruise (same as 3)	$W_5/W_4 = 0.852$	
6) Loiter	$E = ⅓ \text{ hours} = 1200 \text{ s}$	
	$C = 0.0001111\ 1/\text{s}$	
	$L/D = 16$	
	$W_6/W_5 = e^{-0.0083} = 0.9917$	
7) Land	$W_7/W_6 = 0.995$	(Table 2)

$$W_7/W_0 = (0.97)(0.985)(0.852)(0.9277)(0.852)(0.9917)(0.995) = 0.635$$

$$W_f/W_0 = 1.06(1 - 0.6505) = 0.387$$

$$W_e/W_0 = 0.93 W_0^{-0.07} \qquad \text{(Table 1)}$$

$$W_0 = \frac{10{,}800}{1 - 0.387 - \dfrac{W_e}{W_0}} = \frac{10{,}800}{0.613 - 0.93\ W_0^{-0.07}}$$

W_0 Guess	W_e/W_0	W_0 Calculated
50,000	0.4361	61,057
60,000	0.4305	59,191
59,200	0.4309	59,328
59,300	0.4309	59,311
59,310	0.4309	59,309.6

1.6.5. Trade studies

An important part of conceptual design is the evaluation and refinement with the customer, of the design requirements In the ASW design the required f 1500 n.mi. 1gn example e range 1500n.mi. (Each way) is probably less than the customer would really like. A "range trade" can be calculated to determine the increase m design takeoff gross weight if the required range is increased.

<u>1000 nm Range</u>

$$W_3/W_2 = W_5/W_4 = e^{-0.1065} = 0.899$$

$$W_7/W_0 = 0.7069$$

$$W_f/W_0 = 1.06\,(1 - 0.7069) = 0.3107$$

$$W_0 = \frac{10{,}800}{1 - 0.3107 - \dfrac{W_e}{W_0}} = \frac{10{,}800}{0.6893 - 0.93\,W_0^{-0.07}}$$

W_0 Guess	W_e/W_0	W_0 Calculated
50,000	0.4361	42,657
45,000	0.4393	43,203
43,500	0.4403	43,384
43,400	0.4404	43,396
43,398	0.4404	43,397

<u>2000 nm Range</u>

$$W_3/W_2 = W_5/W_4 = \;-0.213 = 0.8082$$

$$W_7/W_0 = 0.5713$$

$$W_f/W_0 = 0.4544$$

$$W_0 = \frac{10{,}800}{1 - 0.4544 - \dfrac{W_e}{W_0}} = \frac{10{,}800}{0.5456 - 0.93\,W_0^{-0.07}}$$

W_0 Guess	W_e/W_0	W_0 Calculated
50,000	0.4361	98,660
80,000	0.4219	87,331
86,000	0.4198	85,889
85,900	0.4199	85,913
85,910	0.4199	85,911

This is done by recalculating the weight fractions for the cruise mission segments, using arbitrarily selected ranges. For example, instead of the required 1500 n.mi., we will calculate the cruise weight fractions using 1000 and 2000 n.mi., and will size the aircraft separately for each of those ranges. These calculations are shown in Box , and the results are plotted

In a similar fashion, a "payload trade" can be made. The mission-segment weight fractions and fuel fraction are unchanged but the numerator of the sizing equation, Eq, is parametrically varied by assuming different payload weights. The given payload requirement is 10,000 lb of avionics equipment. Box 1.3 shows the sizing calculations assuming payload weights of 5000, 15,000, and 20,000 lb. The results are plotted in Fig 1.12

The statistical empty-weight equation used here for sizing was based upon existing military cargo and bomber aircraft, which are all of aluminum construction. The takeoff gross weight calculations above have thus implicitly assumed that the new aircraft would also be built of aluminum.
To determine the effect of building the aircraft out of composite materials, the designer must adjust the empty-weight equation. As previously mentioned, this can be approximated in the early stages of design by taking 95% of the empty-weight fraction obtained for a metal aircraft. The calculations for resizing the aircraft using composite materials are shown in box above

The use of composite materials reduces the takeoff gross weight from 59,310 lb to only 53,771 lb yet the aircraft can still perform

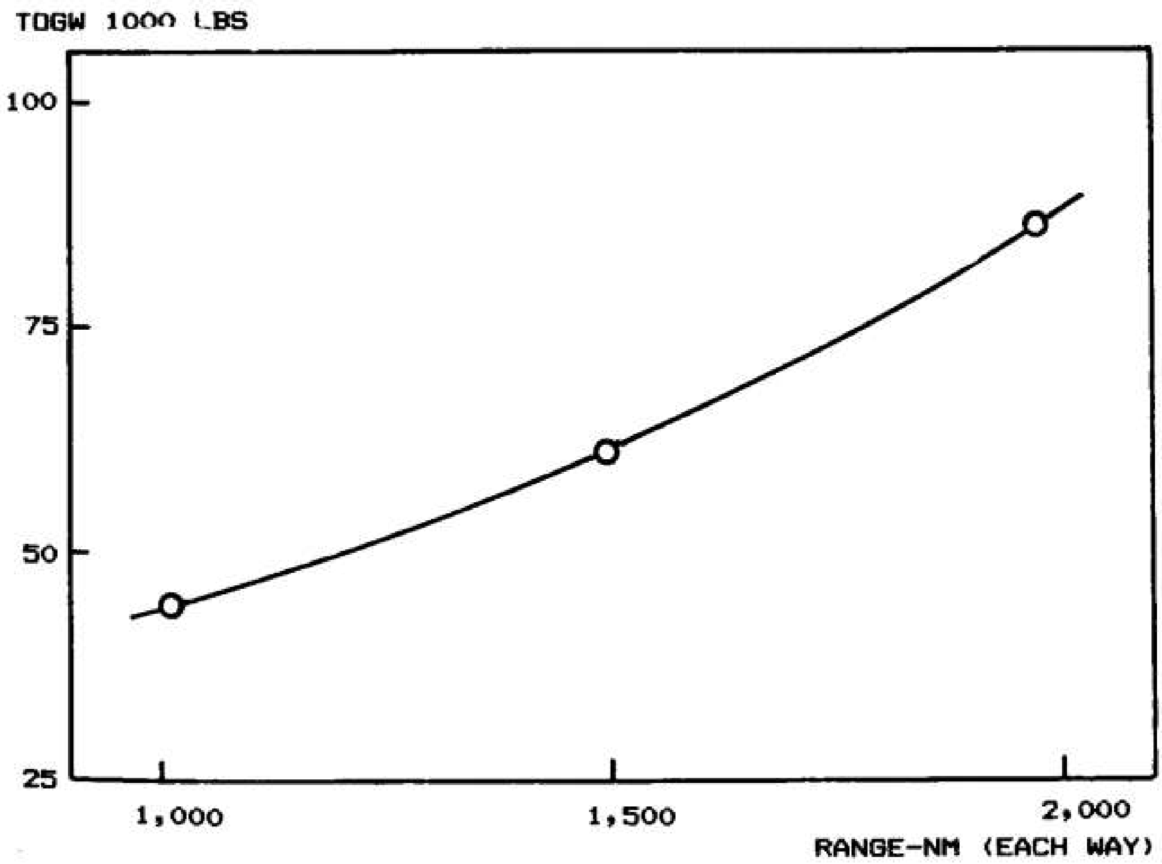


Figure 1.12: Range trade

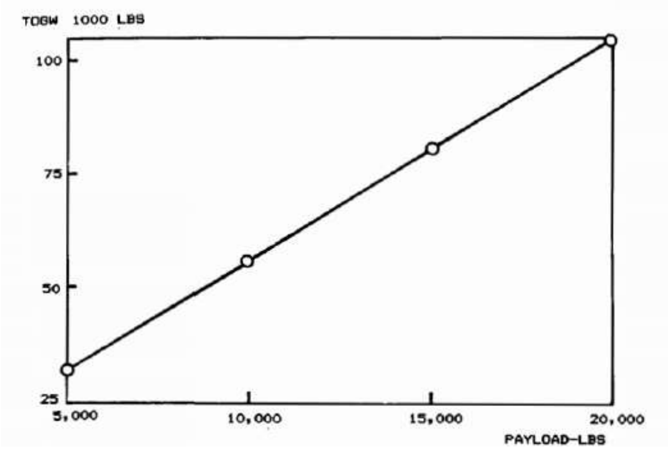


Figure 1.13: payload trade

Mission. This is a 9.30Jo takeoff-weight savings, resulting from only a 5 empty-weight saving. This result sounds erroneous, but is actually typical of the "leverage" effect of the sizing equation. Unfortunately, this works both ways. If the empty weight creeps up during the detail-design process, it will require a more-than proportional increase in takeoff gross weight to maintain the capability to perform the sizing mission. Thus it is crucial that realistic estimates of empty weight be used during early conceptual design, and that the weight be strictly controlled during later stages of design.

There are many trade studies which could be conducted other than range, payload, and material. Methods for trade studies are discussed in detail in. The remainder of the book presents better methods for design, analysis, sizing, and trade studies, building on the concepts just given. In this chapter a conceptual sketch was made, but no guidance was provided as to how to make the sketch or why different features may be good or bad. Following chapters address these issues and illustrate how to develop a complete three view drawing for analysis. Then more-sophisticated methods of analysis, sizing, and trade studies will be provided

1.6.6. Airfoil Selection

The airfoil, in many respects, is the heart of the airplane. The airfoil affects the cruise speed, takeoff and landing distances, stall speed, handling qualities (especially near the stall), and overall aerodynamic efficiency during all phases of flight.

Much of the Wright Brothers' success can be traced to their development of airfoils using a wind tunnel of their own design, and the in-flight validation of those airfoils in their glider experiments of 1901-1902. The P-51 was regarded as the finest fighter of World War II in part because of its radical laminar-flow airfoil. Recently, the low-speed airfoils developed by Peter Lissaman contributed to the success of the man-powered Gossamer Condor.

1.6.7. Airfoil Geometry

illustrates the key geometric parameters of an airfoil. The front of the airfoil is defined by a leading-edge radius which is tangent to the upper and lower surfaces. An airfoil designed to operate in supersonic flow will have a sharp or nearly-sharp leading edge to prevent a drag-producing bow shock. (As discussed later, wing sweep may be used instead of a sharp leading edge to reduce the supersonic drag.) The chord of the airfoil is the straight line from the leading edge to the trailing edge. It is very difficult to build a perfectly sharp trailing edge, so most airfoils have a blunt trailing edge with some small finite thickness.

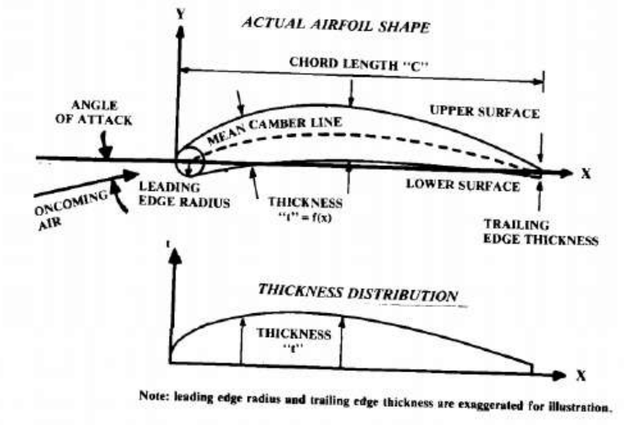


Figure: 1.14: Airfoil geometry

"Camber" refers to the curvature characteristic of most airfoils. The "mean camber line" is the line equidistant from the upper and lower surfaces. Total airfoil camber is defined as the maximum distance of the mean camber line from the chord line, expressed as a percent of the chord. In earlier days, most airfoils had flat bottoms, and it was common to refer to the upper surface shape as the "camber." Later, as airfoils with curved bottoms came into usage, they were known as "double-cambered" airfoils. Also, an airfoil with a concave lower surface was known as an "under-cambered" airfoil. These terms are technically obsolete but are still in common usage.

The thickness distribution of the airfoil is the distance from the upper surface to the lower surface, measured perpendicular to the mean camber line, and is a function of the distance from the leading edge. The "airfoil thickness ratio" (tic) refers to the maximum thickness of the airfoil divided by its chord. For many aerodynamic calculations, it has been traditional to separate the airfoil into its thickness distribution and a zero-thickness camber line.

The former provides the major influence on the profile drag, while the latter provides the major influence upon the lift and the drag due to lift. When an airfoil is scaled in thickness, the camber line must remain unchanged, so the scaled thickness distribution is added to the original camber line to produce the new, scaled airfoil. In a similar fashion, an airfoil which is to have its camber changed is broken into its camber line and thickness distribution. The camber line is scaled to produce the desired maximum camber; then the original thickness distribution is added to obtain the new airfoil. In this fashion, the airfoil can be reshaped to change either the profile drag or lift characteristics, without greatly affect mg the other.

1.6.8. Airfoil Lift and Drag

An airfoil generates lift by changing the velocity of the air passing and under itself. The airfoil angle of attack and/or camber causes the a1r over the top of the wing to travel faster than the air beneath the wing. Bernoulli's equation shows that higher velocities produce lower pressures, so the upper surface of the airfoil tends to be pulled upward by lower-than-ambient pressures while the lower surface of the a1rfoil tends to be pushed upward by higher-than-ambient pressures. The integrated differences in pressure between the top and bottom of the a1rfoil generate the net lifting force. Figure 1.15 shows typical pressure distributions for the upper and lower surfaces of a lifting airfoil at subsonic speeds. Note that the upper surface of the wing contributes about two-thirds of the total lift. Figure 1.15 illustrates the flow field around a typical airfoil as a number of airflow velocity vectors, with the vector length represent mg local velocity magnitude. In Fig. 1.15, the free stream velocity vector is subtracted from each local velocity vector, leaving only the change velocity vector caused by the presence of the airfoil. It can be seen that _the effect of the airfoil is to introduce a change in airflow, which seems to c1rculate around the airfoil in a clockwise fashion if the airfoil nose is to the left of zero lift."

This negative angle is approximately equal (in degrees) to the percent camber of the airfoil.
Odd as it sounds, an airfoil in two-dimensional (2-D) flow does not experience any drag due to the creation of lift. The pressure forces produced in the generation of lift are at right angles to the oncoming air. All two-dimensional airfoil drag is produced by skin friction and pressure

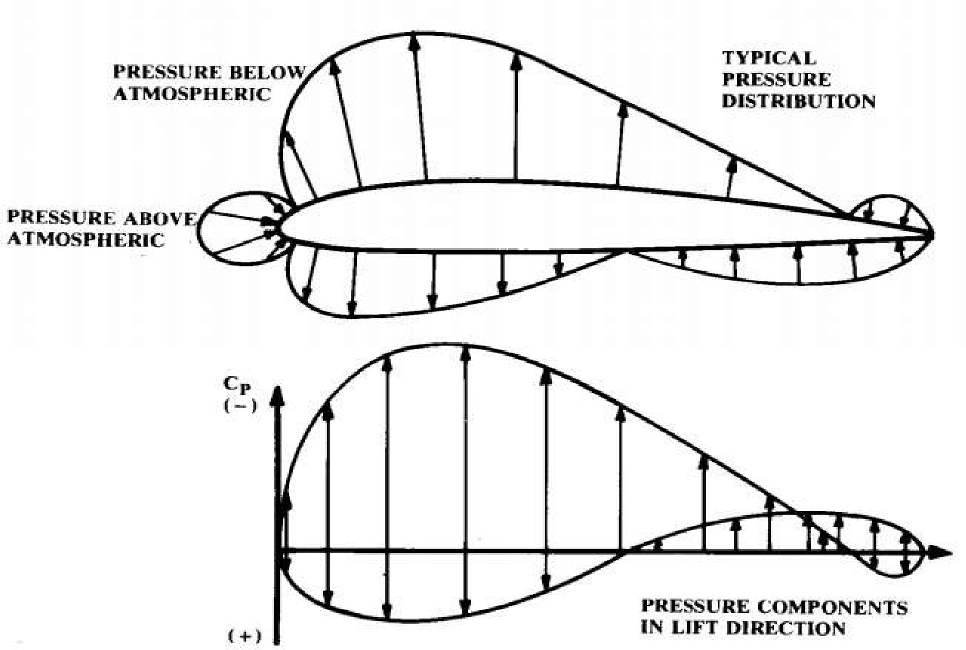


Figure 1.15: Typical airfoil pressure distribution

Flow separation and shocks. It is only in three-dimensional (3-D) flow that drags due to lift is produced. The airfoil section lift, drag, and pitching moment are defined in non-dimensional form. By definition, the lift force is perpendicular to the flight direction while the drag force is parallel to the flight direction. The pitching moment is usually negative when measured about the aerodynamic Center, implying a nose-down moment

$$C_\ell = \frac{\text{section lift}}{qc}$$

$$C_d = \frac{\text{section drag}}{qc}$$

$$C_m = \frac{\text{section moment}}{qc^2}$$

.

c = chord length

q = dynamic pressure = pV_2 /2

a = angle of attack

C_l = slope of the lift curve = 2Π (typically)

The point about which the pitching moment remains constant for any angle of attack is called the "aerodynamic center." The aerodynamic center is not the same as the airfoil's center of pressure (or lift). The center of pressure is usually behind the aerodynamic center. The location of the center of pressure varies with angle of attack for most airfoils.

1.6.9. Airfoil Families

A variety of airfoils is shown in Figure. The early airfoils were developed mostly by trial and error. In the 1930's, the NACA developed a widely-used family of mathematically defined airfoils called the "four-digit" airfoils. In these, the first digit defined the percent camber, the second define the location

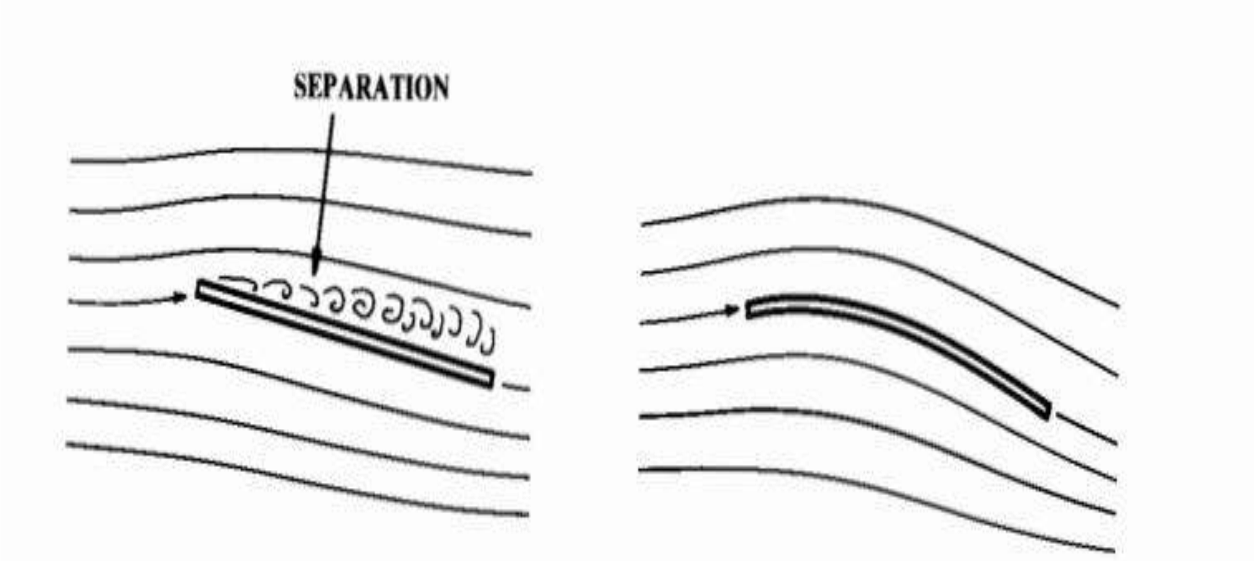


Figure1.16: Effect of camber on separation

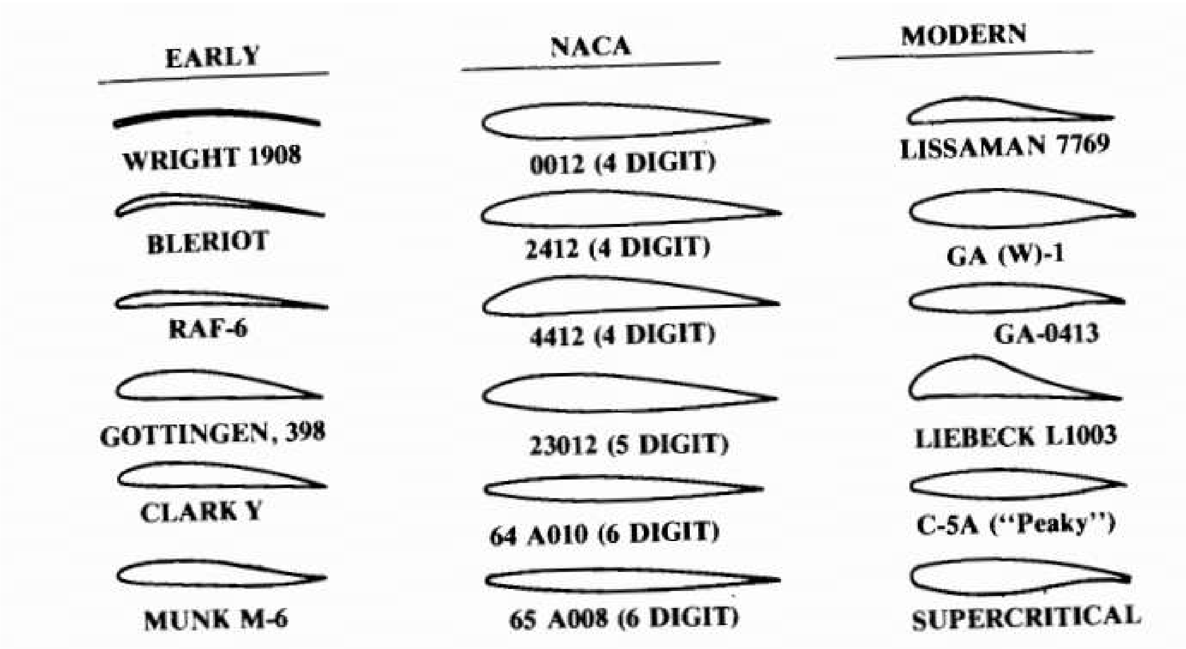


Figure1.17: Typical airfoils

1.7. AIRFOIL DESIGN

In the past, the designer would select an airfoil (or airfoils) from such a catalog. This selection would consider factors such as the airfoil drag during cruise, stall and pitching-moment characteristics, the thickness available for structure and fuel and the ease of manufacture. With today's computational airfoil design capabilities, it is becoming common for the airfoil shapes for a wing to be custom-designed.

Modern airfoil design is based upon inverse computational solutions for desired pressure (or velocity) distributions on the airfoil. Methods have been Comes from the tendency of the rapid pressure rise across the shock to thicken or even separate the boundary layer.

A "supercritical" airfoil is one designed to_ minimize these effects. Modern computational methods allow design of airfoils m which the uppers face shock is minimized or even eliminated by spreading the hft m. t e chord wise direction, thus reducing the upper surface velocity for a required total lift. This increases the Critical Mach Number.

1.7.1. Design Lift Coefficient

For early conceptual design work, the designer must frequently rely upon existing airfoils. From existing airfoils, the. One should be selected a comes closest to having the desired characteristics h aircraft should be designed so that it flies the design milestone at or near the design lift coefficient to maximize the aerodynamic efficient equals the airfoil lift Coefficient, C_l.

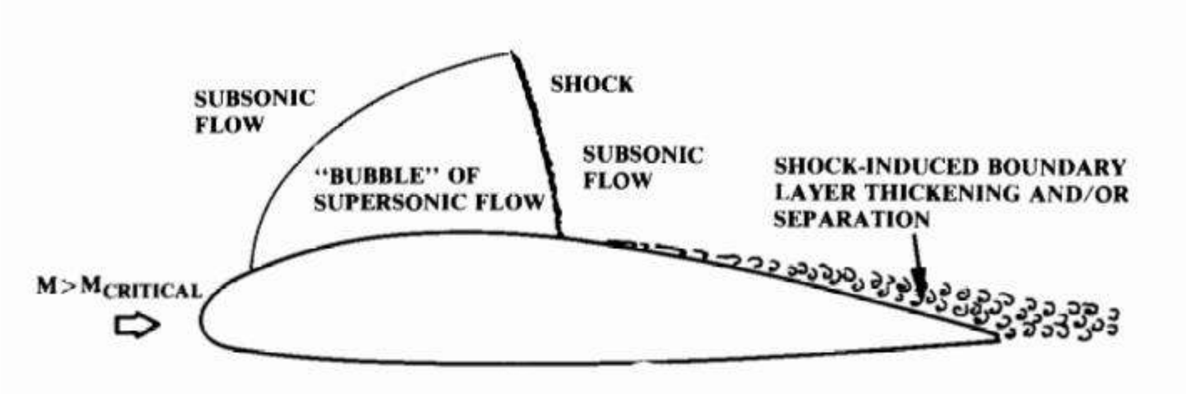


Figure1.18: transonic effects

In level flight the lift must quail the weight, so the required design lift coefficient can be found as follows.

$$W = L = qSC_L \cong qSC_\ell$$

$$C_\ell = \frac{1}{q}\left(\frac{W}{S}\right)$$

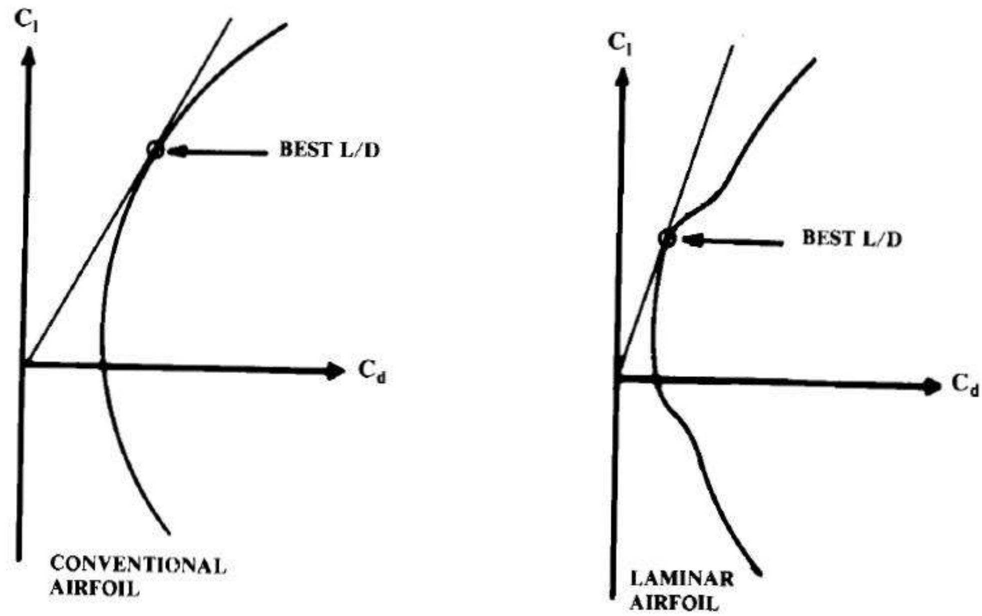


Figure1.19: design lift coefficient

1.7.2. Stall

Stall characteristics play an important role in airfoil selection. Some Airfoils exhibit a gradual reduction in lift during a stall, while others show loss of lift, accompanied by a rapid change in pitching moment. This difference reflects the existence of three entirely different types of airfoil stall.

"Fat" airfoils (round leading edge and t / c greater than about 14%) stall from the tra1hng edge. The turbulent boundary layer increases with angle of attack. at around 10 deg the boundary layer begins to separate, starting at the trailing edge and moving forward as the angle of attack is further Increased. The loss of lift is gradual.

The pitching moment changes only a Small amount. Thinner airfoils stall from the leading edge. If the airfoil is of moderate thickness (about 6-1_4%), the flow separates near the nose at a very small Angle of attack, but 1mmed1ately reattaches itself so that little effect is felt.

At some higher angle of attack the flow fails to reattach, which almost immediately stalls the entire airfoil.

This causes an abrupt change in lift and pitching moment. Very thin airfoils exhibit another form of stall. As before, the flow Separates from the nose at a small angle of attack and reattaches almost immediately.

However, for a very thin airfoil this "bubble" continues to stretch towards the trailing edge as the angle of attack is increased.

At the Angle of attack where the bubble stretches all the way to the trailing edge, the airfoil reaches its maximum lift. Beyond that angle of attack, the flow is separated over the whole airfoil, so the stall occurs.

The loss of lift is smooth, but large changes in pitching moment are experienced. The three types of stall characteristics are depicted in Fig. 1.20.

Twisting the wing such that the tip airfoils have a reduced angle of attack compared to the root ("washout") can cause the wing to stall first at the root.

This provides a gradual stall even for a wing with a poorly stalling airfoil. Horizontal tail, notifying the pilot that a stall is imminent.

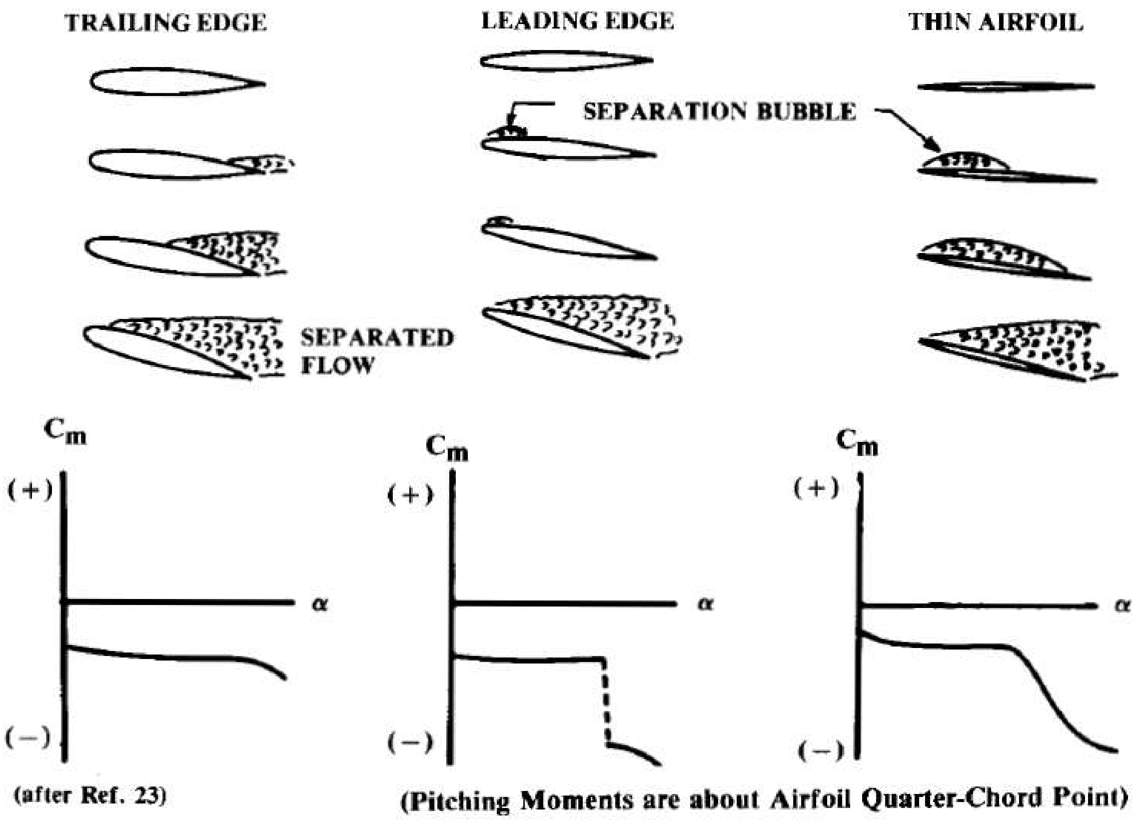


Figure1.20: Types of stall

Root and tip, with a tip airfoil selected which stalls at a higher angle of Attack than the root airfoil. This provides good flow over the ailerons for roll control at an angle of attack where the root is stalled. If different airfoils are used at the root and tip, the designer must develop the intermediate airfoils by interpolation (discussed later).

These intermediate airfoils will have section characteristics somewhere between those of the root and tip airfoils, and can also be estimated by interpolation.

This interpolation of section characteristics does not work or modern supercritical or laminar-flow airfoils. Estimation of the section characteristics those cases must be done computationally. Stall characteristics for thinner airfoils

Dynamic pressure *(q)* is a function of velocity and altitude. By assuming a wing loading (*W/S)* as described later, the design lifts. Coefficient can be calculated for the velocity and altitude of the design mission. Note that the actual wing loading will decrease during the mission as fuel is burned. Thus, to stay at the design lift coefficient, the dynamic pressure must be steadily reduced during the mission by either slowing down, which

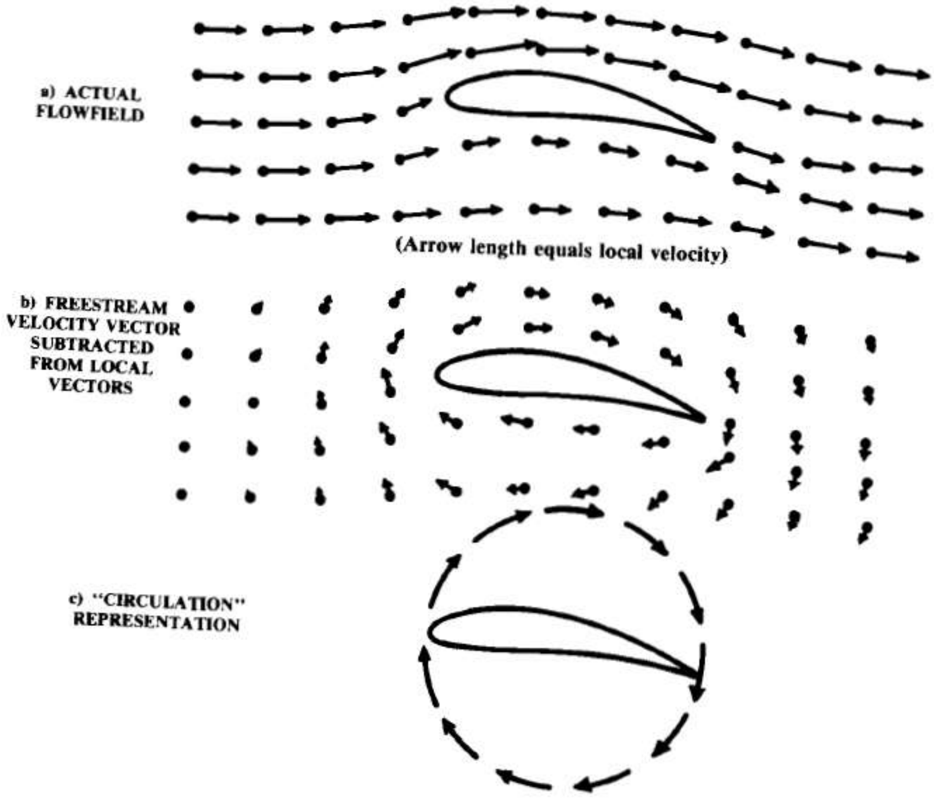


Figure 1.21: airfoil flow field and circulations

1.7.3. Airfoil Thickness Ratio

Airfoil thickness ratio has a direct effect on drag, maximum lift, stall characteristics, and structural weight. Figure illustrates the effect of thickness ratio on subsonic drag. The drag increases with increasing thickness due to increased separation

The thickness ratio affects the maximum lift and stall characteristics primarily by its effect on the nose shape. For a wing of fairly high aspect ratio and moderate sweep, a larger nose radius provides a higher stall angle and a greater maximum lift coefficient.

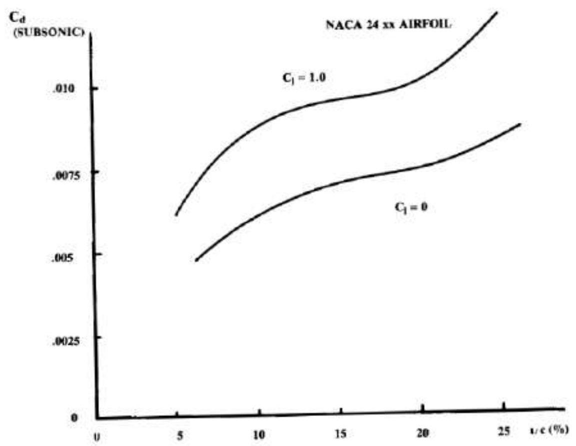


Figure1.22: Effects of t/c on drag

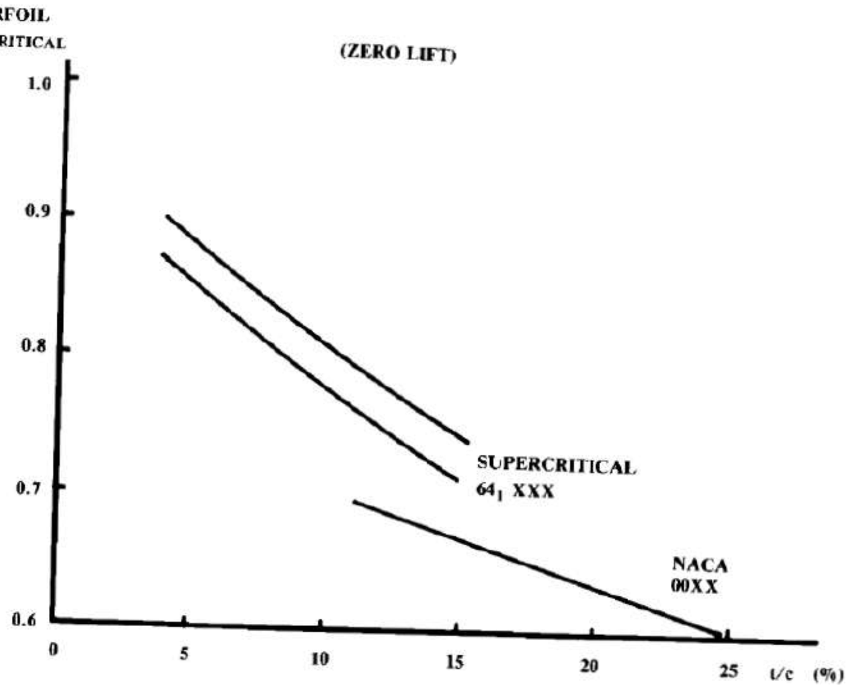


Figure1.23: effect of t/c on critical Mach number

The laminar airfoils require extremely smooth skins as well as exact control over the actual, as-manufactured shape. These can drive the cost up significantly. Also, the camouflage paints used on military aircraft are rough compared to bare metal or composite skins. This must be considered before selecting certain airfoils.

While an understanding of the factors important to airfoil selection is important, an aircraft designer should not spend too much time trying to pick exactly the "right" airfoil in early conceptual design. Later trade studies and analytical design tools will determine the desired airfoil characteristics and geometry. For early conceptual layout, the selected airfoil is
Important mostly for determining the thickness available for structure, landing gear, and fuel

1.8. WING GEOMETRY

The "reference" ("trapezoidal") wing is the basic wing geometry used to beg n the layout. Following Figures .how the key geometric parameters of the reference wing.

Note that the reference wing is fictitious, and extends through the fuselage to the aircraft centerline. Thus the reference wing area includes the part of the reference wing which sticks into the fuselage. For the reference wing, the root airfoil is the airfoil of the trapezoidal reference wing at the centerline of the aircraft, not where the actual wing connects to the fuselage.
There are two key sweep angles, as shown in Fig. 1.16.

The leading-edge sweep is the angle of concern in supersonic flight. To reduce drag it is common to sweep the leading edge behind the Mach cone. The sweep of the quarter-chord line is the sweep most related to subsonic flight. It is important to avoid confusing these two sweep angles.

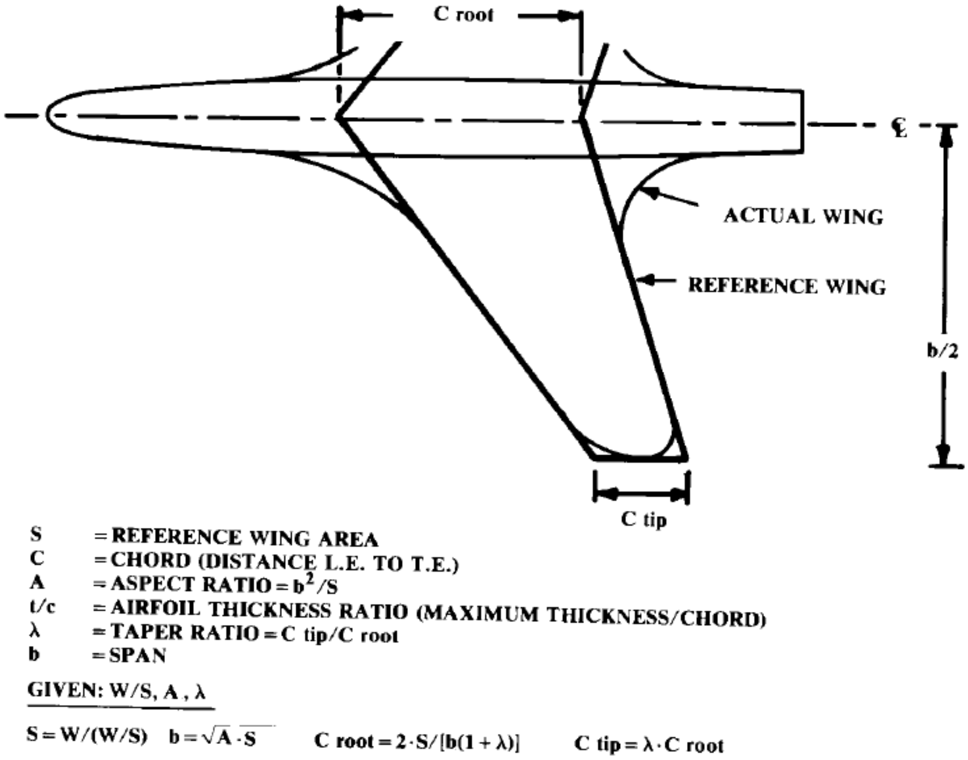


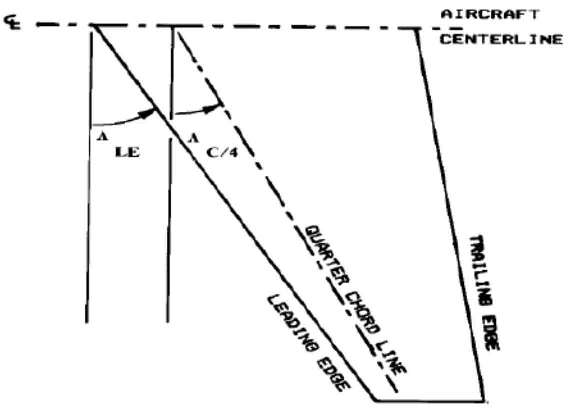
Figure1.24: Wing Geometry

The equation at the bottom of Fig allows converting from one sweep angle to the other. Airfoil pitching moment data in subsonic flow is generally provided about the quarter-chord point, where the airfoil pitching moment is essentially constant with changing angle of attack (i.e., the "aerodynamic center". The mean aerodynamic chord is the chord *"c"* of an airfoil, located at some distance *"Y"* from the centerline.

Ina similar fashion, such a point is defined for the complete trapezoidal wing and is based on the concept of the "mean aerodynamic chord alone. In subsonic flow, this is at the quarter-chord point on the mean aerodynamic chord. In supersonic flow, the aerodynamic center moves back to about 40% of the mean aerodynamic chord.

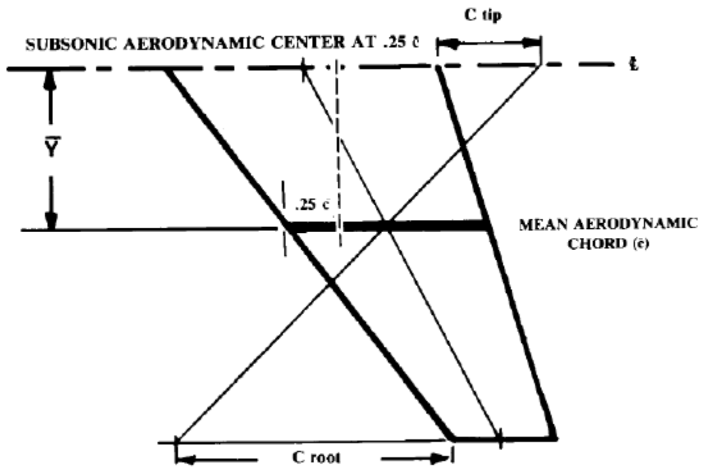
The designer uses the mean aerodynamic chord and the resulting aerodynamic center point to position the wing properly. Also, the mean aerodynamic chord will be important to stability calculations.

Figure 1.24 illustrates a graphical method for finding the mean aerodynamic chord of a trapezoidal-wing plan form. The required reference wing area ("S") can be determined only after the takeoff gross weight is determined. The shape of the reference wing is determined by its aspect ratio, taper ratio, and sweep.



$\tan\Lambda_{LE} = \tan\Lambda_{C/4} + [(1-\lambda)/A(1+\lambda)]$

Figure1.25: wing sweep Λ



$\bar{c} = (2/3)\ C\ root\ (1+\lambda+\lambda^2)/(1+\lambda)$

$\bar{Y} = (b/6)[(1+2\lambda)/(1+\lambda)]$

TYPICAL, WING AERODYNAMIC CENTER = .25 $\bar{C}$ SUBSONIC
= .4 $\bar{C}$ SUPERSONIC

Figure1.26: Mean Aerodynamic Chord

1.8.1. Aspect Ratio

The first to investigate aspect ratio in detail were the Wright Brothers, using a wind tunnel they constructed. They found that a long, skinny wing (high aspect ratio) has less drag for a given lift than a short, fat wing (low aspect ratio). This is due to the 3-D effects.

As most early wings were rectangular in shape, the aspect ratio was initially defined as simply the span divided by the chord. For a tapered wing, the aspect ratio is defined as the span squared divided by the area (which defaults to the earlier definition for a wing with no taper).

When a wing is generating lift, it has a reduced pressure on the upper surface and an increased pressure on the lower surface. The air would like to "escape" from the bottom of the wing, moving to the top. This is not possible in 2-D flow unless the airfoil is leaky (a real problem with some fabric wing materials unless properly treated). However, for a real, 3-Dwing, the air can escape around the wing tip.

Air escaping around the wing tip lowers the pressure difference between the upper and the lower surfaces. This reduces lift near the tip. Also, the air flowing around the tip flows in a circular path when seen from the front, and in effect pushes down on the wing. Strongest near the tip, this reduces the effective angle of attack of the wing airfoils. This circular or "vortex" flow pattern continues downstream behind the wing.

A wing with a high aspect ratio has tips farther apart than an equal area wing with a low aspect ratio. Therefore, the amount of the wing affected by the tip vortex is less for a high aspect ratio wing than for a low-aspect-ratio wing, and the strength of the tip vortex is reduced. Thus, the high-aspect ratio wing does not experience as much of a loss of lift and increase of drag due to tip effects as a low-aspect-ratio wing of equal area.

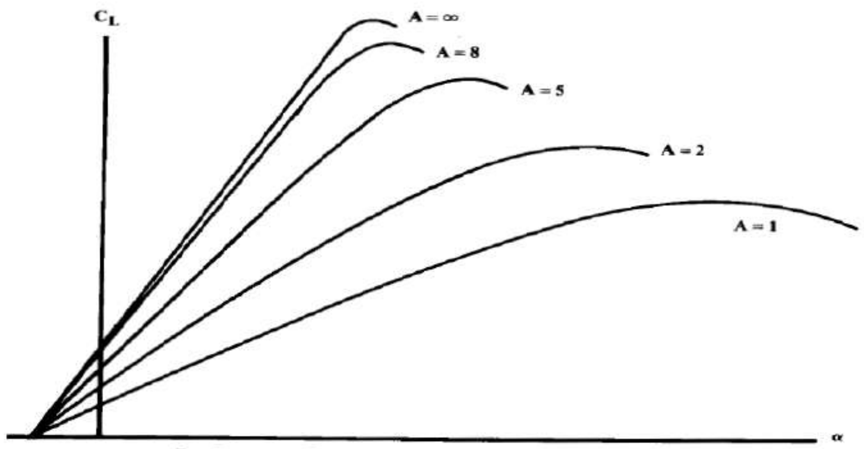


Figure 1.27: effect of aspect ratio on lift

Another effect of changing aspect ratio is a change in stalling angle. Due to the reduced effective angle of attack at the tips, a lower-aspect-ratio wing will stall at a higher angle of attack than a higher-aspect-ratio wing (Fig..27). This is one reason why tails tend to be of lower aspect ratio. Delaying tail stall until well after the wing stalls assures adequate control

1.8.2. Wing Sweep

Wing sweep is used primarily to reduce the adverse effects of transonic and supersonic flow. Theoretically, shock formation on a swept wing is determined not by the actual velocity of the air passing over the wing, but rather by the air velocity in a direction perpendicular to the leading edge of the wing. This result, first applied by the Germans during World War II Allows an increase in Critical Mach Number by the use of sweep

.

At supersonic speeds the loss of lift associated with supersonic flow can be reduced by sweeping the wing leading edge aft of the Mach cone angle [arcsine(1/Mach No.)] .is the wing sweep required to place the wing leading edge exactly on the Mach cone. The historical trend differs from this theoretical result for two reasons. In the high-speed range, it becomes structurally impractical to sweep the wing past the Mach cone. In this speed regime, over about Mach 2.5, it is necessary to use sharp or nearly sharp airfoils.

Selecting the wing sweep to equal the Mach-cone angle would indicate a zero sweep for speeds at or below Mach 1.0. However, in the transonic speed regime (roughly Mach 0.9-1.2) the desire for a high critical Mach number predominates. This requires subsonic airflow velocity over the airflow velocity over (when measured perpendicular to the leading edge), and thus a swept wing

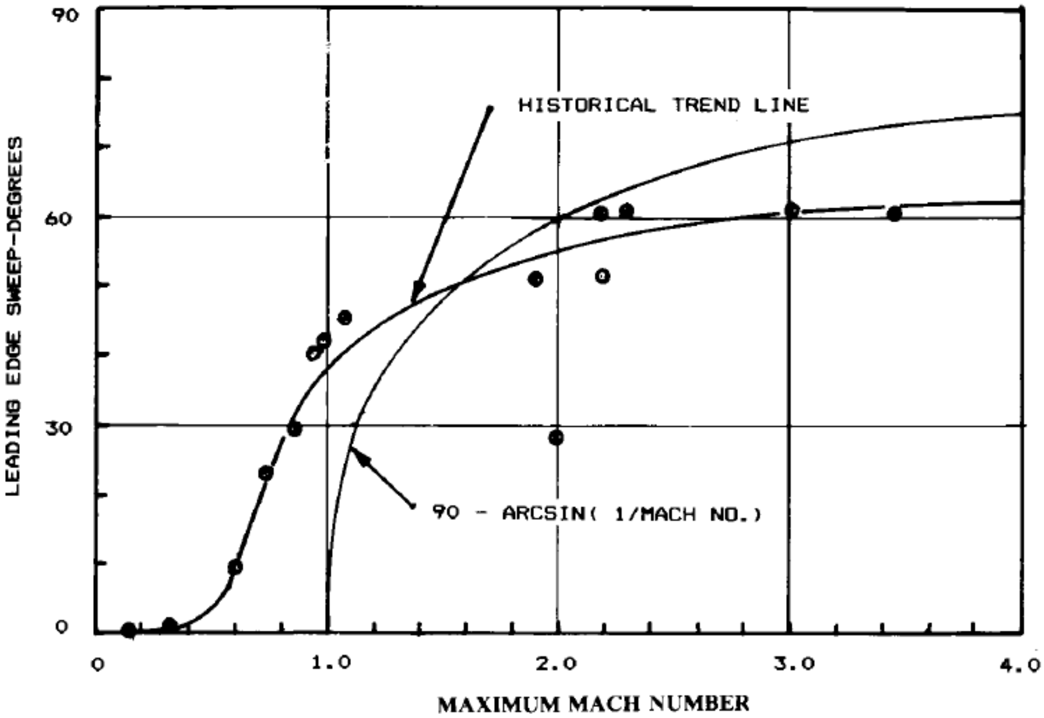


Figure 1.28: Wing sweep historical trend.

The exact wing sweep required to provide the desired Critical Mach Number depends upon the selected airfoil(s), thickness ratio, taper ratio, and other factors. For initial wing layout the trend line of Figure is reasonable. There is no theoretical difference between sweeping a wing aft and sweeping it forward. In the past, wings have been swept aft because of the structural divergence problem associated with forward sweep. With the use of composite materials, this can be avoided for a small weight penalty. Also, there is no reason why one cannot sweep one wing aft and the other wing forward, creating an "oblique wing." This arrangement produces unusual control responses, but a computerized flight control system can easily provide normal handling qualities. The oblique wing also tends to have lower wave drag.

There are other reasons for sweeping a wing. For example, the fuselage layout may not otherwise allow locating the wing carry-through structure at the correct place for balancing the aircraft. Canard aircraft with pusher engines are frequently tail-heavy, requiring wing sweep to move the aerodynamic center back far enough for balance. This is why most canard pushers have swept wings. Wing sweep improves stability.

A swept wing has a natural dihedral effect. In fact, it is frequently necessary to use zero or negative dihedral on a swept wing to avoid excessive stability.
If an aircraft has its vertical tails at the wingtips, sweeping the wing will Push the tails aft, increasing their effectiveness.

This is also seen on many canard pusher aircraft. Note on Fig. 1.l8 the data point at Mach 2.0 and leading-edge sweep just under 30 deg. This is the Lockheed F-104, which used a different approach. for reducing drag at supersonic speeds. The F-104 had a razor-sharp leading edge, so sharp that it was covered on the ground for the safety of line personnel. The F-104 also had a very thin wing, only 3.4% thick

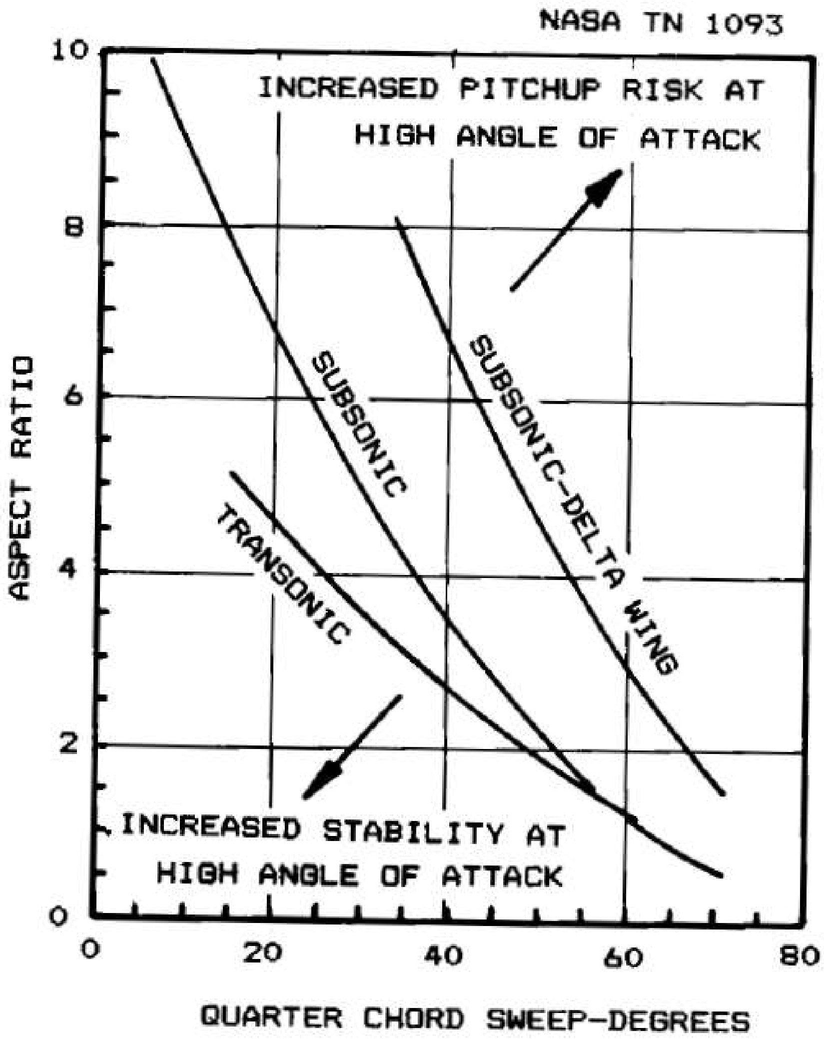


Figure 1.29: Take-off pitch up boundaries

The wing sweep and aspect ratio together have a strong effect on the Wing-alone pitch up characteristics. "Pitch up" is the highly undesirable tendency of some aircraft, upon reaching an angle of attack near stall, to suddenly and uncontrollably increase the angle of attack. The aircraft continues pitching up until it stalls and departs totally out of control.

The F-16fighter requires a computerized angle-of-attack limiter to prevent a severe pitch up problem at about 25-deg angle of attack.

Describes boundaries for pitch up avoidance for combinations of wing quarter-chord sweep angle and aspect ratio. Pitch up avoidance should be considered for military fighters, aerobatic aircraft, and general-aviation aircraft, and trainers.

1.8.3. Taper Ratio

Wing taper ratio is the ratio between the tip chord and the centerline root chord. Most wings of low sweep have a taper ratio of about 0.4-0.5.Most swept wings have a taper ratio of about 0.2-0.3.Taper affects the distribution of lift along the span of the wing. As proven by the Prandtl wing theory early in this century, minimum drag due to lift, or "induced" drag, occurs when the lift is distributed in an elliptical fashion. For an untwisted and unswept wing, this occurs when the wing plan

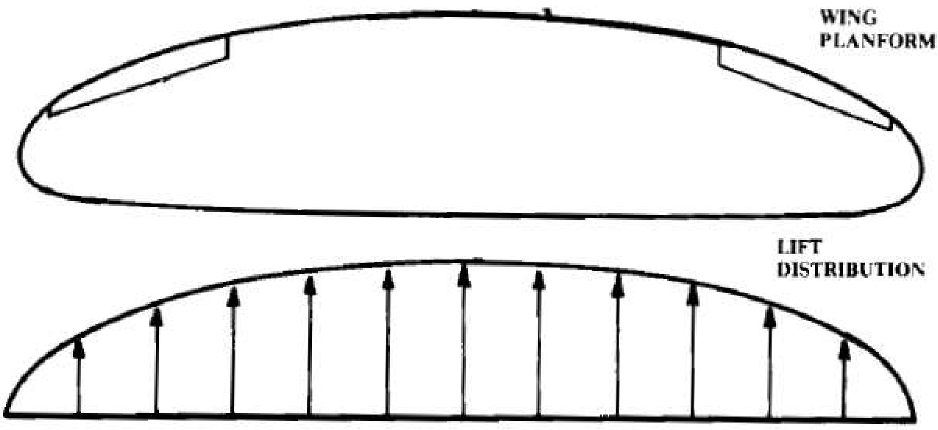


Figure 1.30: Elliptical wing

An elliptical wing plan form is difficult and expensive to build. The easiest wing to build is the untapered (A = 1.0) rectangular wing. However, the untapered wing has constant chord length along the span, and so has excessive chord towards the tip when compared to the ideal elliptical wing. This "loads up" the tip, causing the wing to generate more of its lift toward the tip than is ideal.

The end result is that an untwisted rectangular wing has about 7% more drag due to lift than an elliptical wing of the same aspect ratio. When a rectangular wing is tapered, the tip chords become shorter, alleviating the undesired effects of the constant-chord rectangular wing. In fact a taper ratio of 0.45 almost completely. Eliminates those effects for annual wept wing, and produces a lift distribution very close to the elliptical ideal this results in a drag due to lift less than 1% higher than the ideal, elliptical wing.

A wing swept aft tends to divert the air outboard, towards the tips. This loads up the tips, creating more lift outboard than for an equivalent upswept wing. To return the lift distribution to the desired elliptical lift distribution, it is necessary to increase the amount of taper

1.8.4. Twist

- Wing twist is used to prevent tip stall and to revise the lift distribution to approximate an ellipse. Typically, wings are twisted between zero and five degrees.
- "Geometric twist" is the actual change in airfoil angle of incidence, usually measured with respect to the root airfoil. Awing whose tip airfoil
- "Aerodynamic twist" is the angle between the zero-lift angles of an airfoil and the zero-lift angle of the root airfoil. If the identical airfoil is used from root to tip, the aerodynamic twist is the same as the geometric twist in the other hand, a wing with no geometric twist can have aerodynamic twist if, for example, the root airfoil is symmetric (zero-lift angle is zero) but the tip airfoil is highly cambered (zero-lift angle is nonzero). The total wing aerodynamic twist equals the wing geometric twist plus the root airfoil zero-lift angle, minus the tip airfoil zero-lift angle.

1.8.5. Wing Incidence

The wing incidence angle is the pitch angle of the wing with respect to the fuselage. If the wing is untwisted, the incidence is simply the angle between the fuselage axis and the wing's airfoil chord lines. If the wing is twisted, the incidence is defined with respect to some arbitrarily chosen span wise location of the wing, usually the root of the exposed wing where it intersects the fuselage. Frequently the incidence is given at the root and tip, which then defines the twist as the difference between the two.

Wing incidence angle is chosen to minimize drag at some operating condition, usually cruise. The incidence angle is chosen such that when the wing at the correct angle of attack for the selected designs condition, the fuselage is at the angle of attack for minimum drag.

1.8.6. Dihedral

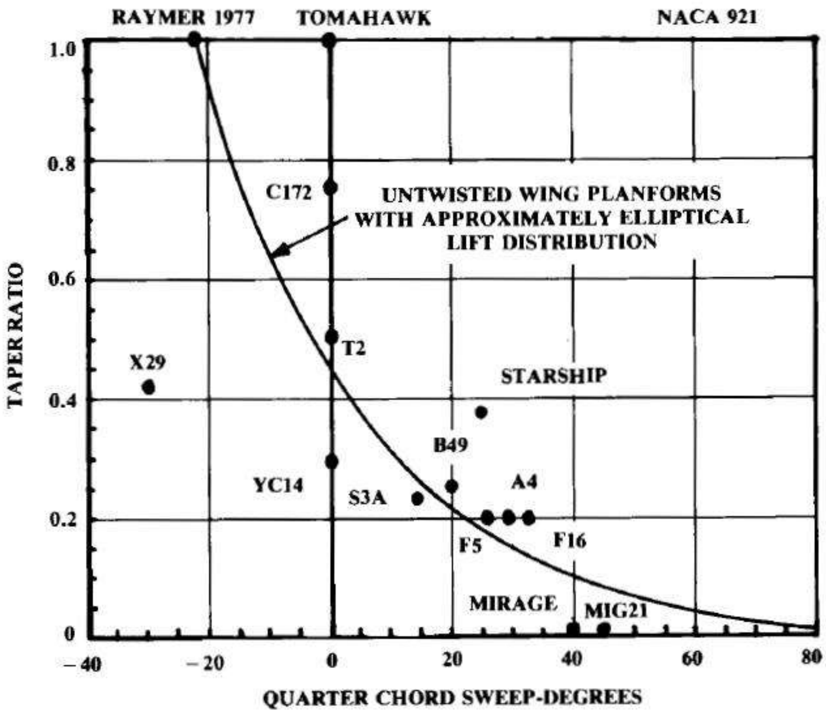


Figure 1.31: effects of sweep on desired taper ratio

Wing dihedral is the angle of the wing with respect to the horizontal when seen from the front. Dihedral tends to roll the aircraft level whenever it is banked. This is frequently, and incorrectly, explained as the result of a greater projected area for the wing that is lowered.

Actually, the rolling moment is caused by a sideslip introduced by the bank angle. The aircraft "slides" toward the lowered wing, which increase sits angle of attack. The resulting rolling moment is approximately proportional to the dihedral angle.

Wing sweep also produces a rolling moment due to sideslip, caused by the change in relative sweep of the left and right wings. For an aft-swept wing, the rolling moment produced is negative and proportional to the sine of twice the sweep angle. This creates an effective dihedral that adds to any actual geometric dihedral.

1.9. 7. Wing Vertical Location

The wing vertical location with respect to the fuselage is generally set by the real-world environment in which the aircraft will operate. For example, virtually all high-speed commercial transport aircraft are of low-wing design, yet military transport aircraft designed to similar mission profiles and payload weights are all of high-wing design.

Table 1.7: dihedral guidence

	Wing position		
	Low	Mid	High
Unswept (civil)	5 to 7	2 to 4	0 to 2
Subsonic swept wing	3 to 7	−2 to 2	−5 to −2
Supersonic swept wing	0 to 5	−5 to 0	−5 to 0

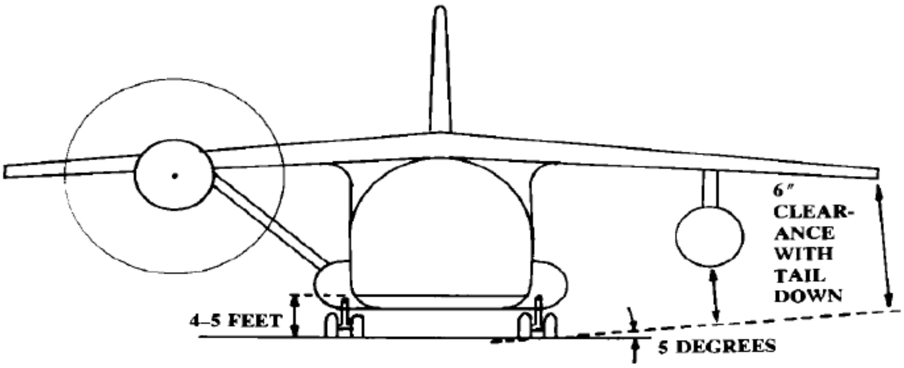


Figure1.32: High wing

For low-speed aircraft, external struts can be used to greatly lower wing weight. However, external struts add substantially to the drag. Since roughly two-thirds of the lift is contributed by the upper surface of the wing, it follows that less drag impact will be seen if the strut disturbs the airflow on the lower surface of the wing than if the strut is above the wing, as would be necessary for a strut-brace, low wing.

For small aircraft, the high wing arrangement can block the pilot's visibility in a turn, obscuring the direction toward which the aircraft is turning. Also, the high wing can block upward visibility in a climb. (A classic mid-air collision features a high-wing aircraft climbing into a low-wing one descend

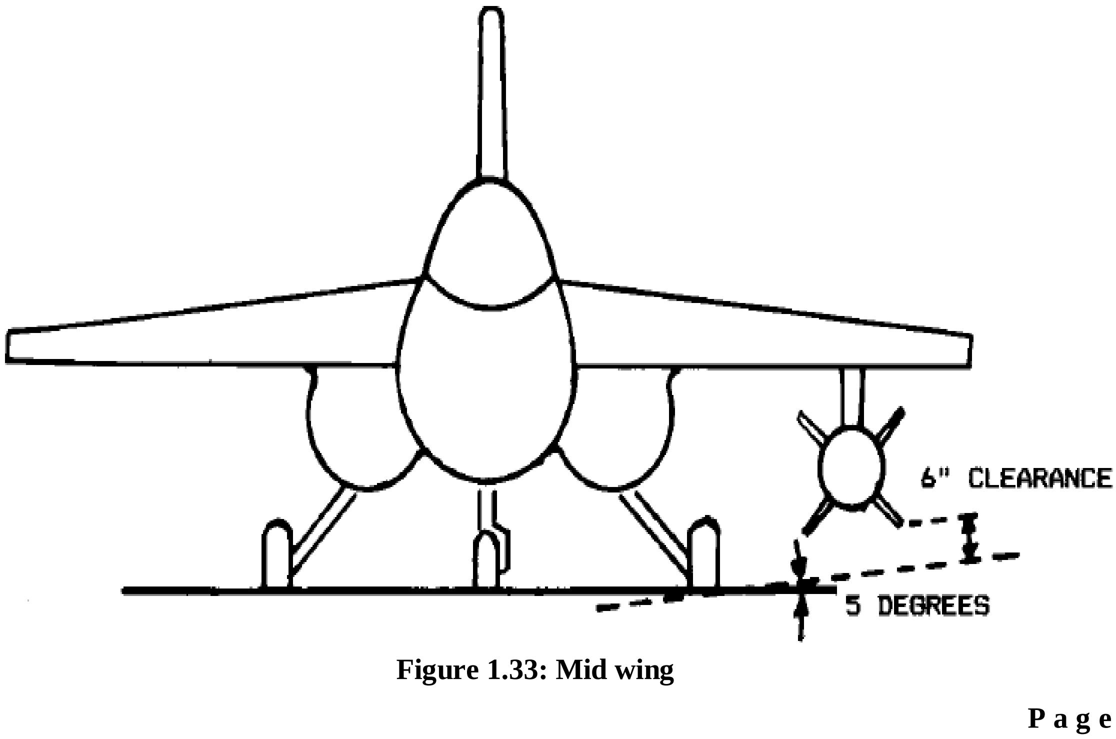


Figure 1.33: Mid wing

The major advantage of the low-wing approach comes in landing-gear stowage. With a low wing, the trunnion about which the gear is retracted can be attached directly to the wing box which, being strong already, will not need much extra strengthening to absorb the gear loads.

When retracted, the gear can be stowed in the wing itself, in the wing-fuselage fairing, or in the nacelle. This eliminates the external blister usually used with the high-wing approach.

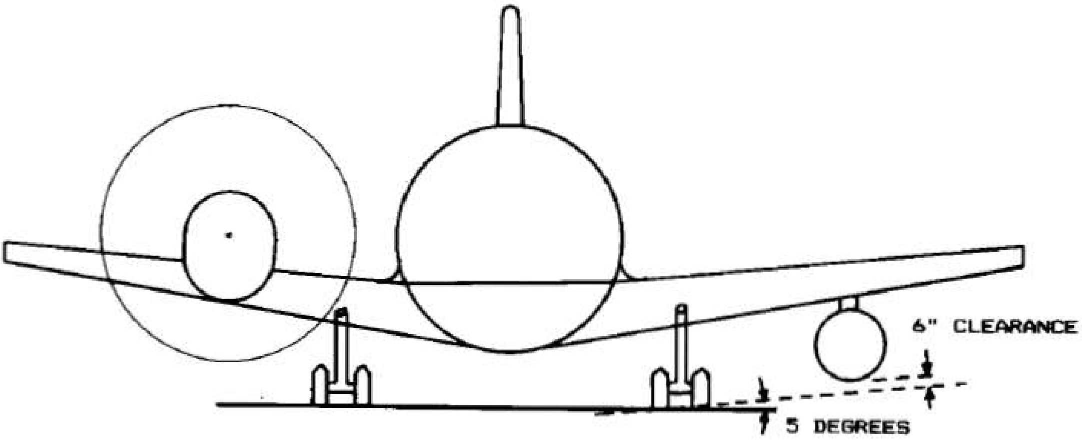


Figure 1.34: low wing

1.9.1. Wing Tips

Wing-tip shape has two effects upon subsonic aerodynamic performance. The tip shape affects the aircraft wetted area, but only to a small extent. Afar more important effect is the influence the tip shape has upon the lateral spacing of the tip vortices.

This is largely determined by the ease with which the higher-pressure air on the bottom of the wing can "escape" around the tip to the top of the wing.

A smoothly-rounded tip (when seen nose-on) easily permits the air to flow around the tip. A tip with a sharp edge (when seen nose-on) makes it more difficult, thus reducing the induced drag. Most of the new low-drag wingtips use some form of sharp edge.

In fact, even a simple cut-off tip offers less drag than a rounded-off tip, due to the sharp edges where the upper and lower surfaces end.

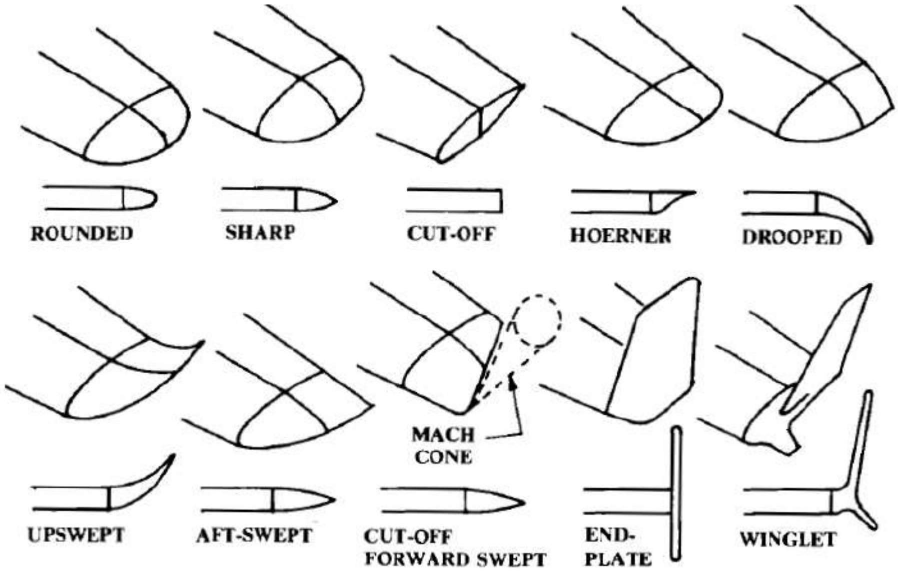


Figure 1.35: wing tips

1.10. BIPLANE WINGS

Biplanes dominated aviation for the first thirty years. The Wright Brothers were influenced by Octave Chanute, a noted architect and civil engineer who applied a structural concept used in bridge building to create light weight biplane gliders. The early airfoils were thin and birdlike, requiring external bracing, and the biplane arrangement provided more structural efficiency than an externally-braced monoplane.

With the thicker airfoils now in use, the biplane arrangement is mainly reserved for recreational purposes. However, it should be considered whenever low structural weight is more important to the design than aerodynamic efficiency, or when low speed is required without complicated high lift devices or excessive wing span.

1.10.1. TAIL GEOMETRY AND ARRANGEMENT

1.10.1.1. Tail Functions

Tails are little wings. Much of the previous discussion concerning wing scan also be applied to tail surfaces. The major difference between a wing and a tail is that, while the wing is designed routinely to carry a substantial amount of lift, a tail is designed to operate normally at only a fraction of its lift potential. Any time in flight that a tail comes close to its maximum lift potential, and hence its stall angle, something is very wrong Tails provide for trim, stability, and control.

Trim refers to the generation of a lift force that, by acting through some tail moment arm about the center of gravity, balances some 0th.er moment produced by the aircraft.

For the horizontal tail, trim primarily refers to the balancing of the moment created by the wing. An aft horizontal tail typically has a negative incidence angle of about 2-3 deg to balance the wing pitching moment.

As the wing pitching moment varies under different flight conditions, the horizontal tail incidence is usually adjustable through a range of about 3 deg up and down.

For the vertical tail, the generation of a trim force is normally not required because the aircraft is usually left-right symmetric and does not create any unbalanced yawing moment.

The vertical tail of a multi-engine aircraft must be capable of providing a sufficient trim force in the event of an engine failure.

1.10.2. Tail Arrangement

Figure 1.36 illustrates some of the possible variations in aft-tail arrangement. The first shown has become "conventional" for the simple reason

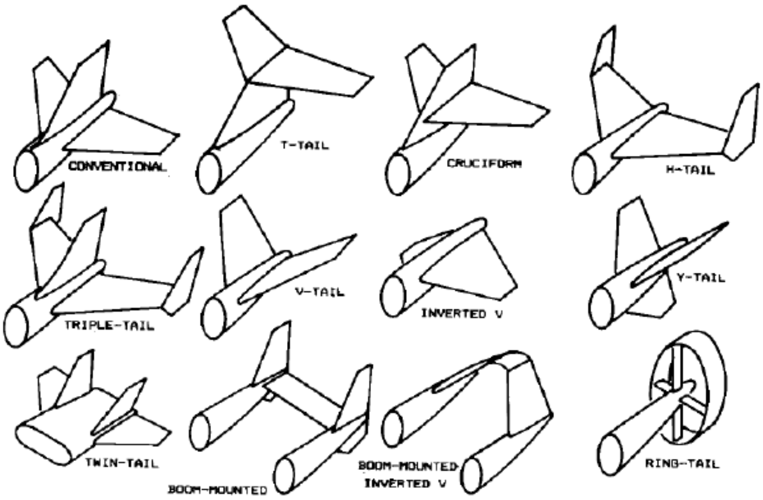


Figure 1.36: aft tail variations

The "H-tail" is used primarily to position the vertical tails in undisturbed air during high angle-of-attack conditions (as on the T-46) or to position the rudders in the prop wash on a multiengine aircraft to enhance engine-out control. The H-tail is heavier than the conventional tail, but its endplate effect allows a smaller horizontal tail.

Twin tails on the fuselage can position the rudders away from the aircraft centerline, which may become blanketed by the wing or forward fuselage at high angles of attack. Also, twin tails have been used simply to reduce the height required with a single tail.

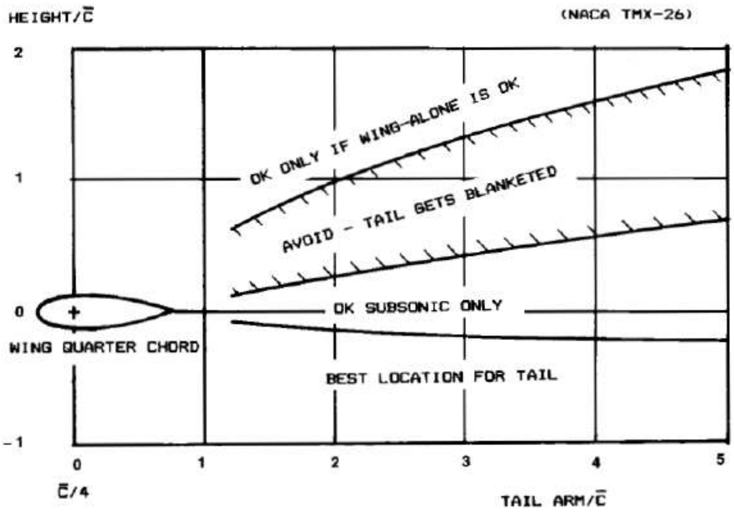


Figure 1.37: aft tail positioning

A T-tail requires a wing designed to avoid pitch up without a horizontal tail, as described by Fig. 1.37. This requires an aircraft stable enough to recover from a stall even when the tail is blanketed by the wing wake. Several general aviation aircraft use this approach, which has the added benefit of a positive warning to the pilot of impending stall caused by buffeting on the tail as it enters the wing wake at high angle of attack. Other possible tail arrangements are depicted in Fig. 1.37. Canards were used by the Wright brothers as a war of ensuring equate

But fell out of favor due to the difficulty of prov1dmg sufficient stability. The early Wright airplanes were, in fact, quite unstable and required a well-trained pilot with quick reflexes. Movie footage taken by passengers shows the Wright canards being continuously manipulated from almost full-up to full-down as the pilot responded to gusts. A three-surface arrangement provides both aft-tail and lifting-canard surfaces. This allows the use of the lifting-canard for reduction of wing drag-due-to-lift without the difficulty of incorporating wing flaps as seen on a canard-only configuration.

The three-surface aircraft theoretically offers minimum trim drag. A canard or aft-tail, when generating lift for trim purposes, will change the aircraft total lift distribution, which increases total induced drag. On a three-surface configuration the canard and aft-tail can act in opposite directions, thus cancelling out each other's effect upon the total lift distribution. (For example, to generate a nose-up trim the canard can generate an upward lift force while the tail generates an equal downward lift force.

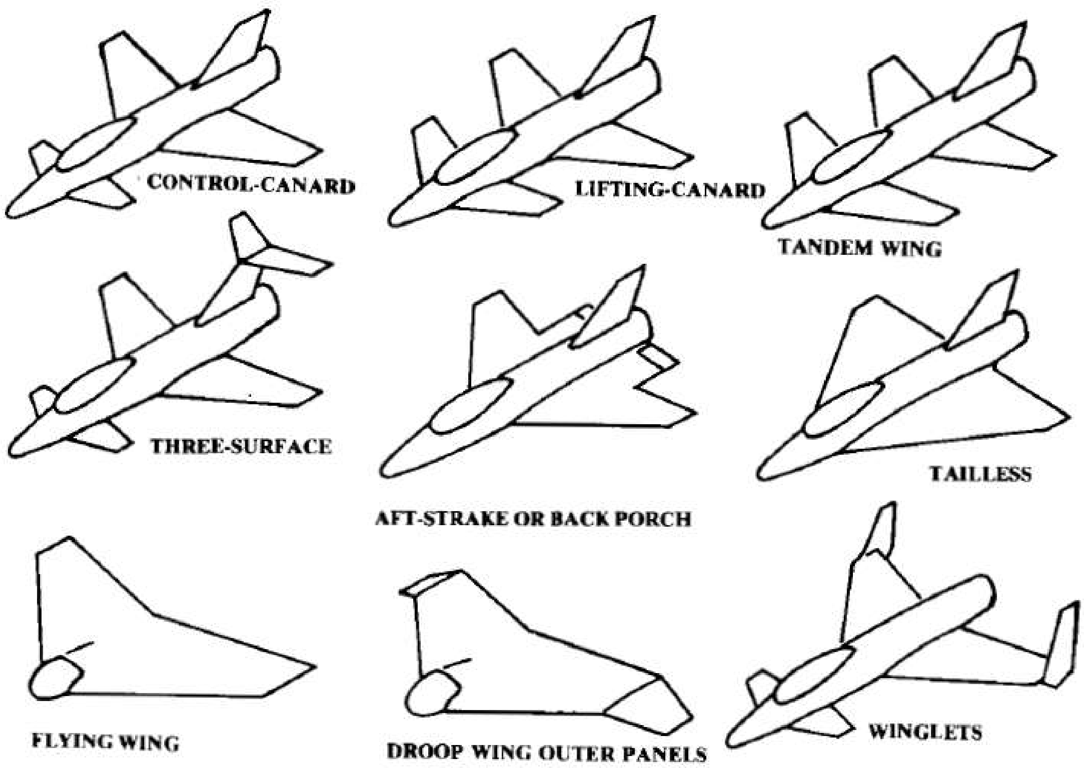


Figure1.38: Other tail configuration

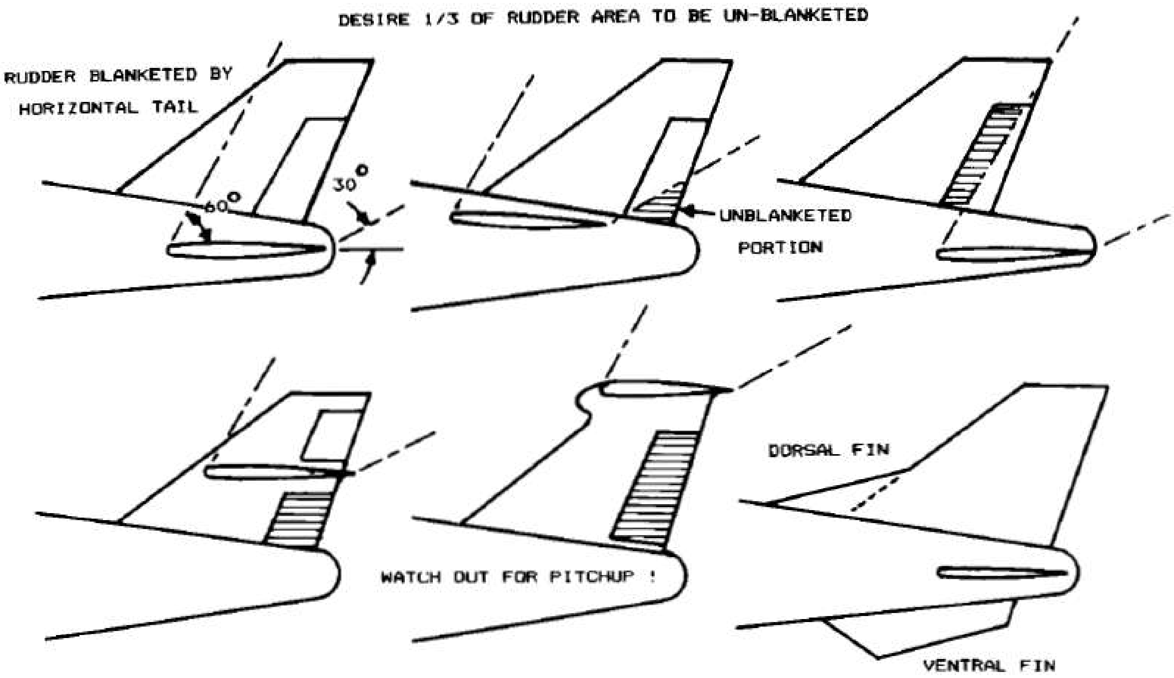


Figure1.39: Tail geometry for spin recovery

1.10.3. Tail Geometry

The surface areas required for all types of tails are directly proportional to the aircraft's wing area, so the tail areas cannot be selected until the initial estimate of aircraft takeoff gross weight has been made. The initial estimation of tail area is made using the "tail volume coefficient" method.Other geometric parameters for the tails can be selected at this time. Tail aspect ratio and taper ratio show little variation over a wide range of aircraft types. Table below provides guidance for selection of tail aspect ratio and taper ratio. Note that T-tail aircraft have lower vertical-tail aspect ratios to reduce the weight impact of the horizontal tail's location on top of the verti1cal-tail. Also, some general-aviation aircraft

Table: Tail aspect ratio and taper ratio

	Horizontal tail		Vertical tail	
	A	λ	A	λ
Fighter	3–4	0.2–0.4	0.6–1.4	0.2–0.4
Sail plane	6–10	0.3–0.5	1.5–2.0	0.4–0.6
Others	3–5	0.3–0.6	1.3–2.0	0.3–0.6
T-Tail	–	–	0.7–1.2	0.6–1.0

Tail thickness ratio is usually similar to the wing thickness ratio, as determined by the historical guidelines provided in the wing-geometry section. For a high-speed aircraft, the horizontal tail is frequently about 10% thinner than the wing to ensure that the tail has a higher Critical Mach Number. Note that a lifting canard or tandem wing should be designed using the guidelines and procedures given for initial wing design, instead of the tail-design guidelines described above.

1.10.4. THRUST-TO-WEIGHT RATIOAND WING LOADING

1.10.4.1. THRUST-TO-WEIGHT

The thrust-to-weight ratio *(T/W)* and the wing loading *(W/S)* are the two most important parameters affecting aircraft performance. Optimization of these parameters forms a major part of the analytical design activities conducted after an initial design layout. However, it is essential that a credible estimate of the wing loading and Thrust-to-weight ratio be made before the initial design layout is begun. Otherwise, the optimized aircraft may be so unlike the as-drawn aircraft that the design must be completely redone.

Due to this interconnection, it is frequently difficult to use historical data to independently select initial values for wing loading and thrust-to-weight ratio. Instead, the designer must guess at one of the parameters and use that guess to calculate the other parameter from the critical design requirements.

1.10.4.2. Thrust-to-Weight Ratio Definitions

T/W directly affects the performance of the aircraft. An aircraft with a higher *T/W* will accelerate more quickly, climb more rapidly, reach a higher maximum speed, and sustain higher turn rates. On the other hand, the larger Engines will consume more fuel throughout the mission, which will drive up the aircraft's takeoff gross weight to perform the design mission.

It is very important to avoid confusing the takeoff *T/ W* with the *T/*W at other conditions in the calculations below. If a required *T/*W is calculated at some other condition, it must be adjusted back to takeoff conditions for use in selecting the number and size of the engines. These *T/W* adjustments will be discussed later.

1.10.4.3. Power Loading and Horsepower-to-Weight

The term "thrust-to-weight" is associated with jet-engine aircraft. For propeller-powered aircraft, the equivalent term has classically been the "power loading," expressed as the weight of the aircraft divided by its horsepower (W/hp).

Power loading has an opposite connotation from *T/W* because a high power loading indicates a smaller engine. Power loadings typically range from 10-15 for most aircraft. An aerobatic aircraft may have a power loading of about six. A few aircraft have been built with power loadings a slow as three or four. One such over-powered airplane was the Pitts Sampson, a one-of-a-kind airshow airplane.

$$\frac{T}{W} = \left(\frac{550\ \eta_p}{V}\right)\left(\frac{\text{hp}}{W}\right)$$

Note that this equation includes the term hp/ *W*, the horsepower-to weight ratio. This is simply the inverse of the classical power loading *(W/*hp). To avoid confusion when discussing requirements affecting both jet and propeller-powered aircraft, this book refers to the horsepower-to weight ratio rather than the classical power loading.

Also, to avoid excessive verbiage in the discussions below, the term "thrust-to-weight ratio" should be understood to include the horsepower-to-weight ratio for propeller aircraft.

1.10.4.4. Statistical Estimation of TIW

Tables provide typical values for *TlW* and hp/W for different classes of aircraft. Table also provides reciprocal values, i.e., power loadings, for propelled aircraft. These values are all at maximum power settings at sea level and zero velocity ("static").

Table 1.4: Thrust to weight ratio (T/W)

Aircraft type	Typical installed T/W
Jet trainer	0.4
Jet fighter (dogfighter)	0.9
Jet fighter (other)	0.6
Military cargo/bomber	0.25
Jet transport	0.25

Table 1.5: Horsepower to weight ratio (T/W)

Aircraft type	Typical installed T/W
Jet trainer	0.4
Jet fighter (dogfighter)	0.9
Jet fighter (other)	0.6
Military cargo/bomber	0.25
Jet transport	0.25

1.10.4.5. Thrust Matching

For aircraft designed primarily for efficiency during cruise, a better initial estimate of the required *TIW* can be obtained by "thrust matching." This refers to the comparison of the selected engine's thrust available during cruise to the estimated aircraft drag. In level accelerating flight, the thrust must equal the drag. Likewise, the weight must equal the lift (assuming that the thrust is aligned with the flight path). Thus, *T/W* must equal the inverse of *L/D*

Table 1.6. : T/W_0 vs M_{max}

$T/W_0 = a\, M_{max}^C$	a	C
Jet trainer	0.488	0.728
Jet fighter (dogfighter)	0.648	0.594
Jet fighter (other)	0.514	0.141
Military cargo/bomber	0.244	0.341
Jet transport	0.267	0.363

Table 1.7.: weight ratio

$hp/W_0 = a\ V_{max}^C$	a	C
Sailplane—powered	0.043	0
Homebuilt—metal/wood	0.005	0.57
Homebuilt—composite	0.004	0.57
General aviation—single engine	0.024	0.22
General aviation—twin engine	0.034	0.32
Agricultural aircraft	0.008	0.50
Twin turboprop	0.012	0.50
Flying boat	0.029	0.23

L/D can be estimated in a variety of ways will discuss the detailed drag-buildup approach. For the first estimation of T/W the method for L/D estimation when the wing loading has been selected, as described later in this chapter, the L/D at the actual cruise conditions should be calculated and used to recheck the initial estimate for T/W.

$$\left(\frac{T}{W}\right)_{takeoff} = \left(\frac{T}{W}\right)_{cruise}\left(\frac{W_{cruise}}{W_{takeoff}}\right)\left(\frac{T_{takeoff}}{T_{cruise}}\right)$$

$$\left(\frac{hp}{W}\right)_{takeoff} = \left(\frac{V_{cruise}}{550\ \eta_p}\right)\left(\frac{1}{(L/D)_{cruise}}\right)\left(\frac{W_{cruise}}{W_{takeoff}}\right)\left(\frac{hp_{takeoff}}{hp_{cruise}}\right)$$

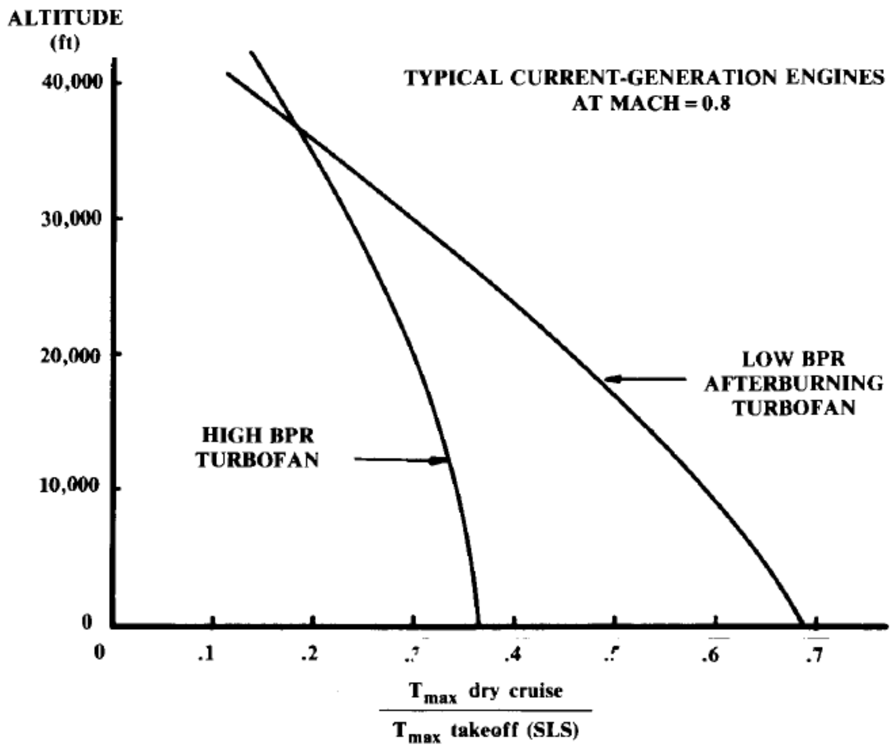


Figure 1.40: Thrust lapse at cruise

1.11. WING LOADING

The wing loading is the weight of the aircraft divided by the area of the reference (not exposed) wing. As with the thrust-to-weight ratio, the term "wing loading" normally refers to the takeoff wing loading, but can also refer to combat and other flight conditions. Wing loading affects stall speed, climb rate, takeoff and landing distances, and turn performance. The wing loading determines the design lift coefficient, and impacts drag through its effect upon wetted area and wingspan.

Table 1.7. : wing loading

Historical trends	Typical takeoff W/S (lb/ft^2)
Sailplane	6
Homebuilt	11
General aviation—single engine	17
General aviation—twin engine	26
Twin turboprop	40
Jet trainer	50
Jet fighter	70
Jet transport/bomber	120

This material generally assumes that an initial estimate of T/W has been made using the methods presented in the last section. However, most of the equations could also be used to solve for T/W if the wing loading is defined by some unique requirement (such as stall speed).

These methods estimate the wing loading required for various performance conditions. To ensure that the wing provides enough lift in all circumstances, the designer should select the lowest of the estimated wing loadings.

However, if an unreasonably low wing loading value is driven by only one of these performance conditions, the designer should consider another way to meet that condition For example, if the wing loading required to meet a stall speed requirement is well below all other requirements, it may be better to equip the aircraft with a high-lift flap system. If takeoff distance or rate of climb requires a very low wing loading, perhaps the thrust-to-weight ratio should be increased.

1.11.1. Stall Speed

The stall speed of an aircraft is directly determined by the wing loading and the maximum lift coefficient. Stall speed is a major contributor to flying safety, with a substantial number of fatal accidents each year due to "failure to maintain flying speed."

Also, the approach speed, which is the most important factor in landing distance and also contributes to post-touch down accidents, is defined by the stall speed.

$$W = L = q_{\text{stall}} S C_{L\,\text{max}} = \tfrac{1}{2} \rho V_{\text{stall}}^2 S C_{L\,\text{max}}$$

$$W/S = \tfrac{1}{2} \rho V_{\text{stall}}^2 C_{L\,\text{max}}$$

The maximum lift coefficient for an aircraft designed for short takeoff and landing (STOL) applications will typically be about 3.0. For a regular transport aircraft with flaps and slats (leading-edge flaps with slots to improve airflow), the maximum lift coefficient As a crude approximation, the designer can ignore this effect.

Then the maximum lift can be estimated by determining the maximum angle of attack before some part of the wing stalls. Typically, the part of the wing with the flap deflected will stall first. Then, for that angle of attack the lift contributions of the flapped and flapped sections can be summed, weighted by their areas (see Fig. for definitions of flapped and un flapped areas).This crude approximation for wings of a fairly high aspect ratio is given.

$$C_{L\,\max} \cong 0.9\left\{(C_{\ell_{\max}})_{\text{flapped}}\frac{S_{\text{flapped}}}{S_{\text{ref}}} + (C_\ell)_{\text{unflapped}}\frac{S_{\text{unflapped}}}{S_{\text{ref}}}\right\}$$

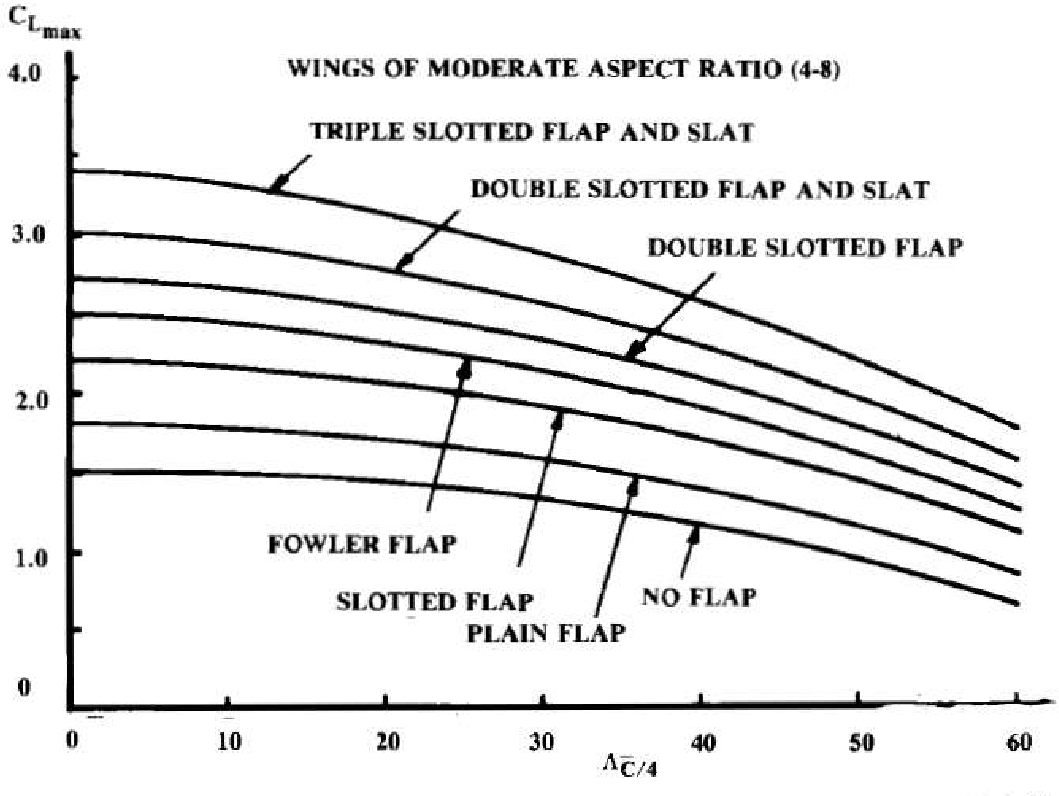


Figure 1.41: Maximum lift coefficient

1.11.2. Takeoff Distance

A number of different values are referred to as "takeoff distance." The "ground roll" is the actual distance traveled before the wheels leave the ground. The liftoff speed for a normal takeoff is 1.1 times the stall speed. The "obstacle clearance distance" is the distance required from brake release until the aircraft has reached some specified altitude. This is usually 50ft for military or small civil aircraft and 35ft for commercial aircraft. The "balanced field length" is the length of the field required for safety the event of an engine failure at the worst possible time in a multiengine aircraft.

If the aircraft is nearly at liftoff speed and an engine fails, the pilot would-be unable to stop safely and instead would continue the takeoff on the remaining engines. The takeoff lift coefficient is the actual lift coefficient at takeoff, not the maximum lift coefficient at takeoff conditions as used for stall calculation. To determine the required wing loading to meet a given takeoff distance requirement, the takeoff parameter is obtained from Figure above. Then the following expressions give the maximum allowable wing loading for the given takeoff distance:

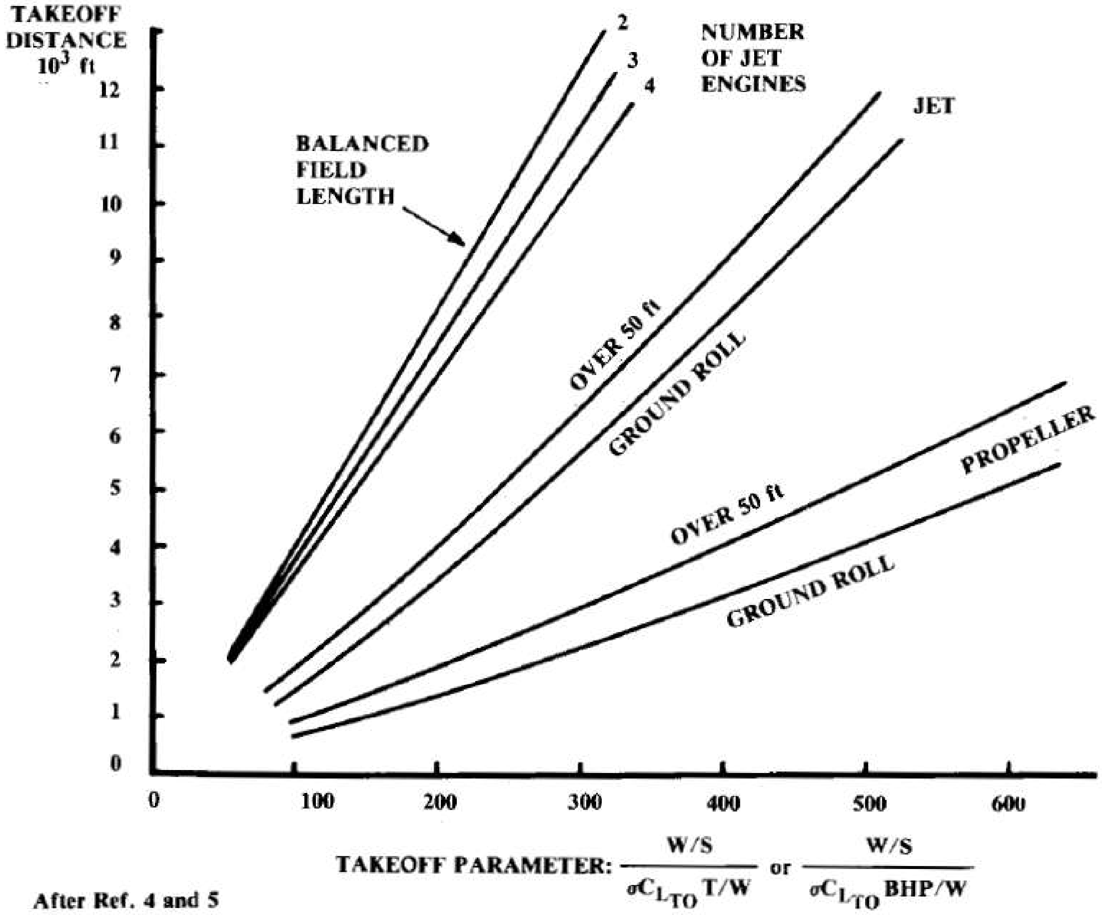


Figure 1.42: Takeoff distance estimation

$$\text{Prop:} \quad (W/S) = (\text{TOP})\sigma C_{L\,\text{TO}}(\text{hp}/W)$$

$$\text{Jet:} \quad (W/S) = (\text{TOP})\sigma C_{L\,\text{TO}}(T/W)$$

$$\left(\frac{W}{S}\right)_{\text{landing}} = \frac{1}{2}\rho(V_{\text{end}} + V_{\text{wod}})^2 \frac{(C_{L\,\max})_{\text{takeoff}}}{1.21}$$

1.11.3. SELECTION OF THRUST-TO-WEIGHT AND WING LOADING

An initial estimate of the thrust-to-weight (or horsepower-to-weight) ratio was previously made. From the wing loadings estimated above, the lowest value should be selected to ensure that the wing is large enough for all flight conditions. Don't forget to convert all wing loadings to takeoff conditions prior to comparisons. A low wing-loading will always increase aircraft weight and cost. If a very low wing-loading is driven by only one of the requirements, change in design assumptions (such as a better high-lift system) may allow a higher wing-loading.

Aerodynamic optimizations for a portion of the mission. If these give wing loadings far lower they may be ignored. When the best compromise for wing loading has been selected, the thrust weight ratio should be rechecked to ensure that all requirements are still met. The equations in the last section which use *T/W* should be recalculated with the selected *W/S* and *T/W*. Only then can the next step of design, initial sizing, be initiated.

UNIT-II
INITIAL SIZING & CONFIGURATION LAYOUT

2.1. 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 such as the SST.

2.2. SIZING WITH FIXED ENGINE AND WITH RUBBER ENGINE

This information a crude estimate of the maximum *L/D* 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 (W_{i+1}/ W_i). Using these fractions and the approximate fractions for takeoff, climb, and landing which were provided in mission weight fraction (W_{i+1}/W_i) was estimated.

For different classes of aircraft, statistical equations for the aircraft empty-weight fraction were provided. Then, the takeoff weight was calculated using repeated below as Eq. 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. 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.

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

Where

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

Is limited in use to missions which do not have a sudden Weight change, such as a payload drop. Also, in many cases Eq. cannot be used for fixed-engine sizing.

2.2.1. Refined Sizing Equation

For missions with a payload drop or other sudden weight change, a slightly different sizing equation must be used. The takeoff weight is calculated by summing the crew weight, payload weight, fuel weight, and empty weight.

The previous equation which resembles equation except that the payload now includes a fixed payload and a dropped payload. The empty weight is again expressed as an empty-weight fraction, but the fuel weight is determined directly.

$$W_0 = W_{\text{crew}} + W_{\substack{\text{fixed} \\ \text{payload}}} + W_{\substack{\text{dropped} \\ \text{payload}}} + W_{\text{fuel}} + W_{\text{empty}}$$

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

As before, an initial guess of the takeoff weight is used to determine a calculated takeoff weight, and the solution is iterated until the two are approximately equal to within a few percent. Refined methods for determining the empty-weight fraction and fuel used are discussed below**.**

2.2.2. Refined Sizing Method

The design and sizing method presented above, as summarized resembles in many respects the first-order method presented

This information a crude estimate of the maximum *L/D* 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 (W_{i+1}/ W_i). Using these fractions and the approximate fractions for takeoff, climb, and landing which were provided in mission weight fraction (W_{i+1}/W_i) was estimated.

For different classes of aircraft, statistical equations for the aircraft empty-weight fraction were provided. Then, the takeoff weight was calculated using repeated below as Eq. Since the empty weight was calculated using a guess of the takeoff weight, it was necessary to iterate towards a solution.

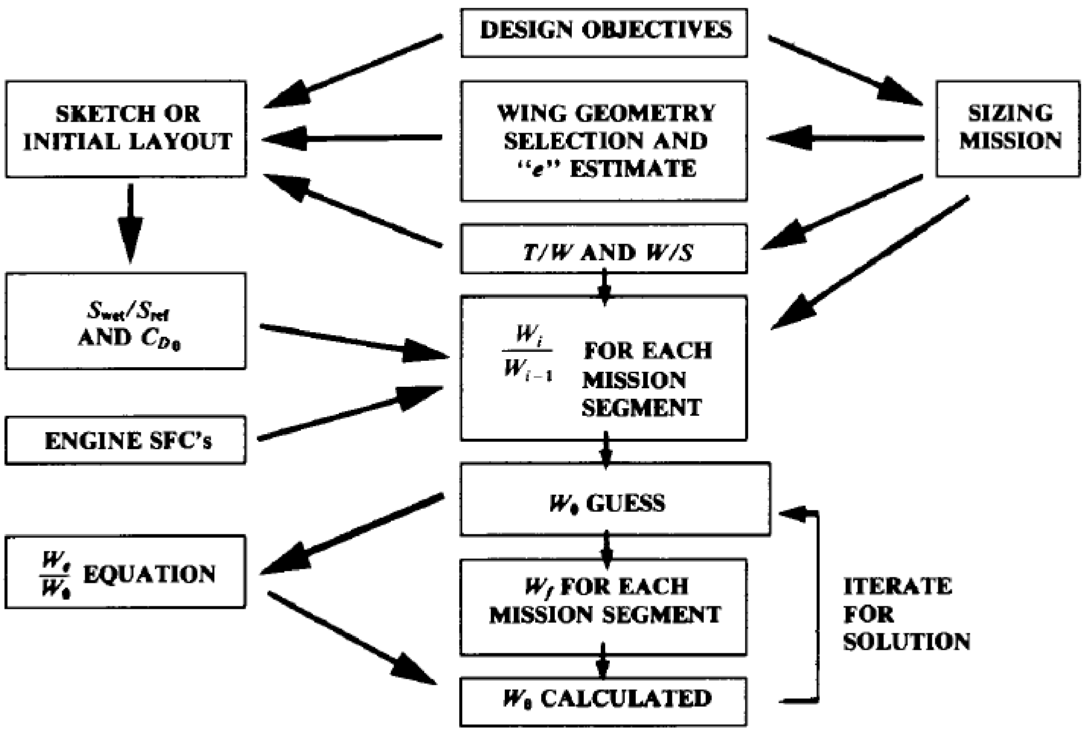


Figure 2.1: Refined Sizing Method

2.3. GEOMETRY SIZING OF FUSELAGE, WING, TAIL, CONTROL SURFACES,
2.3.1. GEOMETRY SIZING

Fuselage once the takeoff gross weight has been estimated, the fuselage, wing, and tails can be sized. Many methods exist to initially estimate the required fuselage size. For certain types of aircraft, the fuselage size is determined strictly by "real-world constraints." For example, a large passenger aircraft devotes most of its length to the passenger compartment.

Once the number of passengers is known and the number of seats across is selected, the fuselage length and diameter are essentially determined. For initial guidance during fuselage layout and tail sizing, this provides statistical equations for fuselage length developed from data provided . These are based solely upon takeoff gross weight, and give remarkably good correlations to most existing aircraft. Fuselage fineness ratio is the ratio between the fuselage length and its maximum diameter. If the fuselage cross section is not a circle, an equivalent diameter is calculated from the cross-sectional area.

2.3.2. Wing

The actual wing size can now be determined simply as the takeoff gross weight divided by the takeoff wing loading. Remember that this is the reference area of the theoretical, trapezoidal wing, and includes the area extending into the aircraft centerline

2.3.3. Tail Volume Coefficient

For the initial layout, a historical approach is used for the estimation of tail size. The effectiveness of a tail in generating a moment about the center of gravity is proportional to the force (i.e., lift) produced by the tail and to the tail moment arm.

The primary purpose of a tail is to counter the moments produced by the wing. Thus, it would be expected that the tail size would be in some way related to the wing size. In fact, there is a directly proportional relationship between the two, as can be determined by examining the moment equations presented before. Therefore, the tail area divided by the wing area should show some consistent relationship for different aircraft, if the effects of tail moment arm could be accounted for the force due to tail lift is proportional to the tail area.

Thus, the tail effectiveness is proportional to the tail area times the tail moment arm. This product has units of volume, which leads to the "tail volume coefficient" method for initial estimation of tail size. Rendering this parameter non dimensional requires dividing by some quantity with units of length. For a vertical tail, the wing yawing moments which must be countered are most directly related to the wing span *bw*.

This leads to the "vertical tail volume coefficient," For a horizontal tail or canard, the pitching moments which must be countered are most directly related to the wing mean chord (Cw).

$$c_{VT} = \frac{L_{VT} S_{VT}}{b_W S_W}$$

$$c_{HT} = \frac{L_{HT} S_{HT}}{\overline{C}_W S_W}$$

For an airplane with a computerized "active" flight control system, the statistically estimated tail areas may be reduced by approximately 10% provided that trim, engine-out, and nose wheel liftoff requirements can be met.

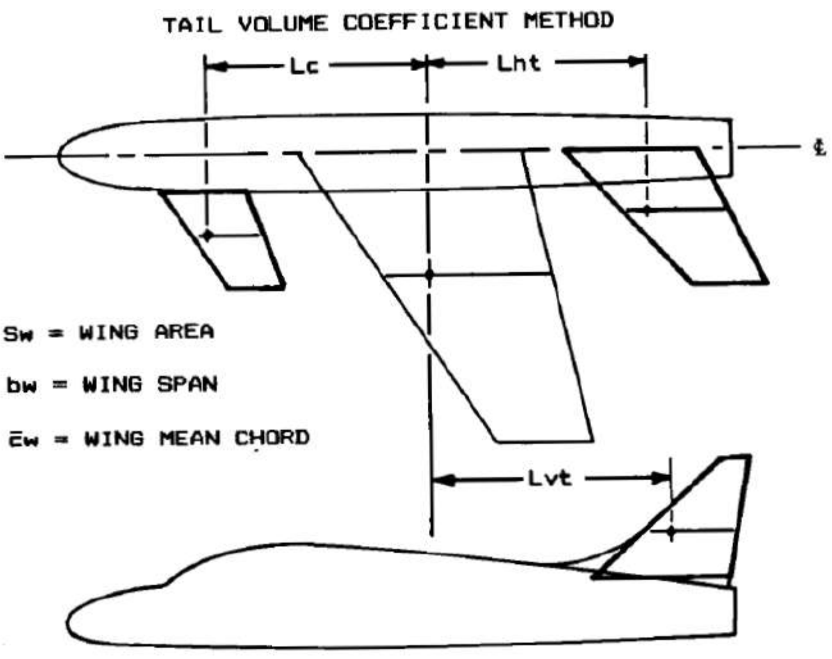


Figure 2.2. Initial Tail Sizing

TABLE: 2.1. Tail Volume Coefficient

	Typical values	
	Horizontal c_{HT}	Vertical c_{VT}
Sailplane	0.50	0.02
Homebuilt	0.50	0.04
General aviation—single engine	0.70	0.04
General aviation—twin engine	0.80	0.07
Agricultural	0.50	0.04
Twin turboprop	0.90	0.08
Flying boat	0.70	0.06
Jet trainer	0.70	0.06
Jet fighter	0.40	0.07
Military cargo/bomber	1.00	0.08
Jet transport	1.00	0.09

2.4. CONTROL SURFACE SIZING

The primary control surfaces are the ailerons (roll), elevator (pitch), and rudder (yaw). Final sizing of these surfaces is based upon dynamic analysis of control effectiveness, including structural bending and control-system effects. For initial design, the following guidelines are offered.

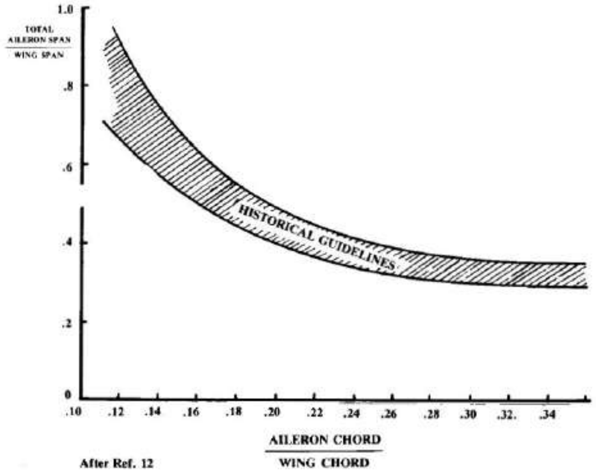
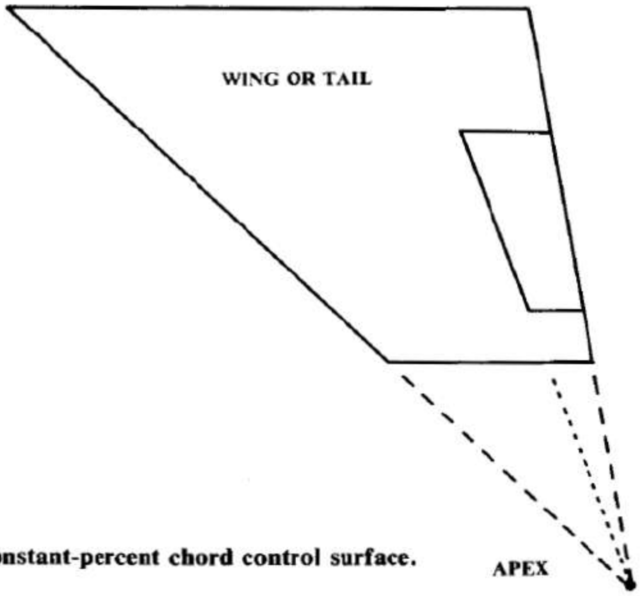


Figure 2.3: Aileron Guidance

Wing flaps occupy the part of the wing span inboard of the ailerons. If a large maximum lift coefficient is required, the flap span should be as large as possible. One way of accomplishing this is through the use of spoilers rather than ailerons. Spoilers are plates located forward of the flaps on the top of the wing, typically aft of the maximum thickness point. Spoilers are deflected upward into the



Constant-percent chord control surface.

slipstream to reduce the wing's lift.

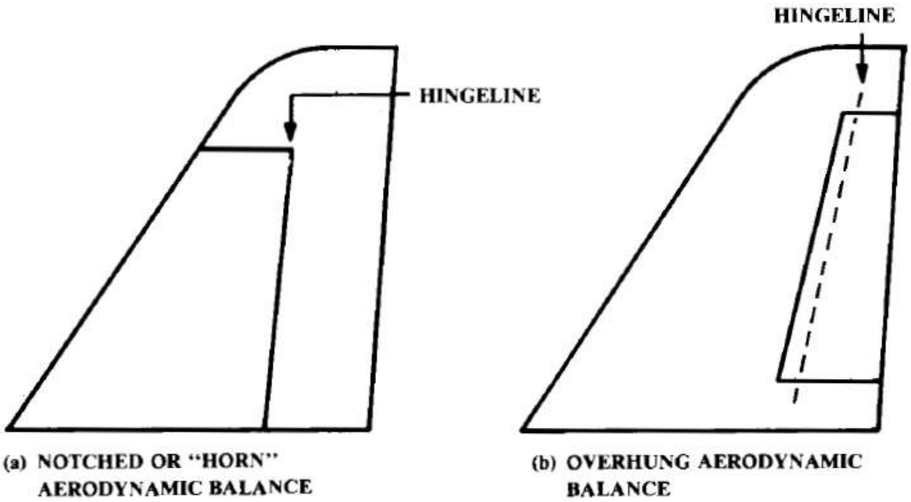


Figure 2.4: Aerodynamic Balance

Control surfaces are usually tapered in chord by the same ratio as the wing or tail surface so that the control surface maintains a constant percent chord (Fig.2.4). This allows spars to be straight-tapered rather than curved. Ailerons and flaps are typically about 15-250Jo of the wing chord. Rudders and elevators are typically about 25-500Jo of the tail chord.

Control-surface "flutter," a rapid oscillation of the surface caused by the air loads, can tear off the control surface or even the whole wing. Flutter tendencies are minimized by using mass balancing and aerodynamic balancing.

2.4.1. Development of Configuration Lay Out From Conceptual Sketch, the Inboard Profile Drawing

The process of aircraft conceptual design includes numerous statistical estimations, analytical predictions, and numerical optimizations. However, the product of aircraft design is a drawing. While the analytical tasks are vitally important, the designer must remember that these tasks serve only to influence the drawing, for it is the drawing alone that ultimately will be used to fabricate the aircraft.

All of the analysis efforts to date were performed to guide the designer in the layout of the initial drawing. Once that is completed, a detailed analysis can be conducted to resize the aircraft and determine its actual performance.

2.4.1.1. End Products of Configuration Layout

The outputs of the configuration layout task will be design drawings of several types as well as the geometric information required for further analysis. The design layout process generally begins with a number of conceptual sketches.

The inboard profile is far more detailed than the initial layout. For example, while the initial layout may merely indicate an avionics bay based upon a statistical estimate of the required avionics volume, the inboard profile drawing will depict the actual location of every piece of avionics (i.e., “black boxes") as well as the required wire bundles and cooling ducts. The inboard profile is generally a team project, and takes many weeks. During the preparation of the inboard profile it is not uncommon to find that the initial layout must be changed to provide enough room for everything. As this can result in weeks of lost effort, it is imperative that the initial layout be as well thought-out as possible

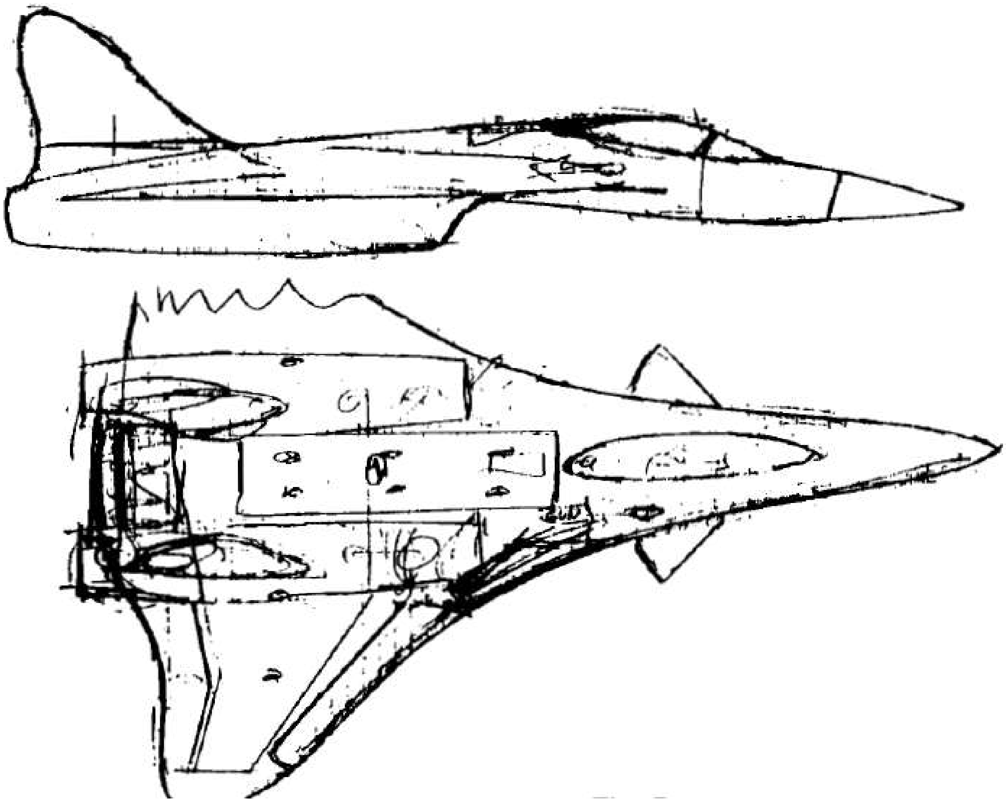


Figure 2.5 Design Sketch

2.5. CONIC LOFTING, FLAT WRAP LOFTING

"Lofting" is the process of defining the external geometry of the aircraft. “Production lofting," the most detailed form of lofting, provides an exact, mathematical definition of the entire aircraft including such minor details as the intake and exhaust ducts for the air conditioning. A production-loft definition is expected to be accurate to within a few hundredths of an inch (or less) over the entire aircraft. This allows the different parts of the aircraft to be designed and fabricated at different plant sites yet fit together perfectly during final assembly.

Most aircraft companies now use computer-aided loft systems that incorporate methods discussed. For an initial layout it is not necessary to go into as much detail. However, the overall lofting of the fuselage, wing, tails, and nacelles must be defined sufficiently to show that these major components will properly enclose the required internal components and fuel tanks while providing a smooth aerodynamic contour.

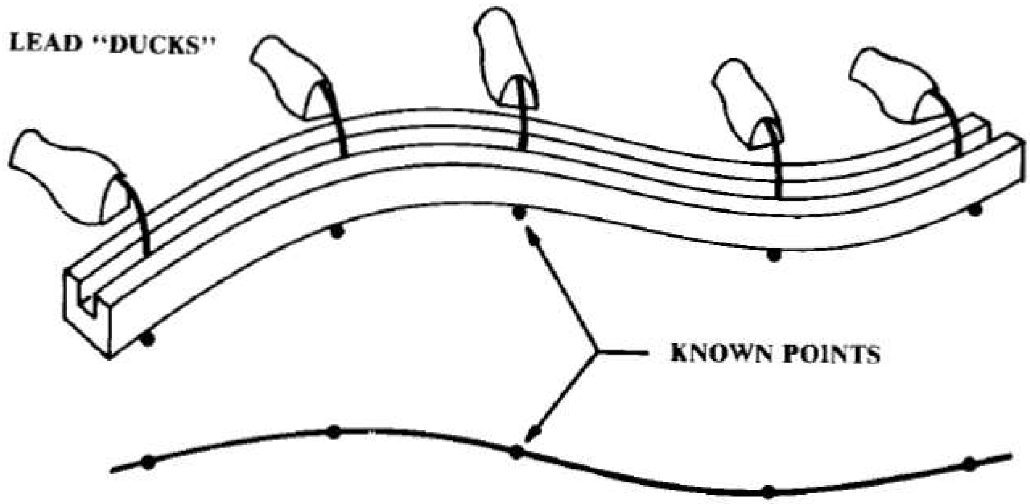


Figure 2.6: Spline Lofting

Lofting gets its name from shipbuilding. The definition of the hull shape was done m the loft over the shipyard, using enormous drawings. To provide a smooth longitudinal contour, points taken from the desired cross-sections were connected longitudinally on the drawing by flexible “splines," long, thin wood or plastic rulers held down at certain points by Lead "ducks".

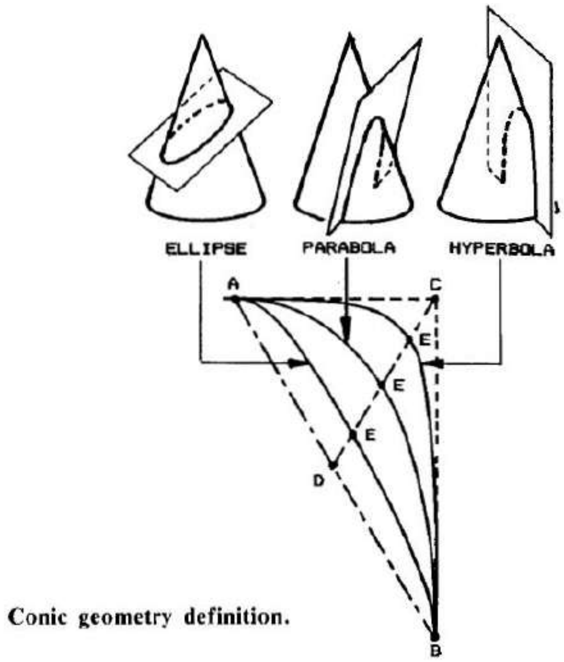


Figure 2.7: Conic Geometry Definition

2.6. WETTED AREA

Aircraft wetted area (S_{wet}), the total exposed surface area, can be visualized as the area of the external parts of the aircraft that would get wet if it were dipped into water. The wetted area must be calculated for drag estimation, as it is the major contributor to friction drag. The wing and tail wetted areas can be approximated from their plan forms, as shown in Fig. 2.7. The wetted area is estimated by multiplying the true-view exposed plan form area ($S_{exposed}$) times a factor based upon the wing or tail thickness ratio.

If a wing or tail were paper-thin, the wetted area would be exactly twice the true plan form area (i.e., top and bottom). The effect of finite thickness is to increase the wetted area, as approximated by Eqs below. Note that the true exposed plan form area is the projected (top-view) area divided by the cosine of the dihedral angle.

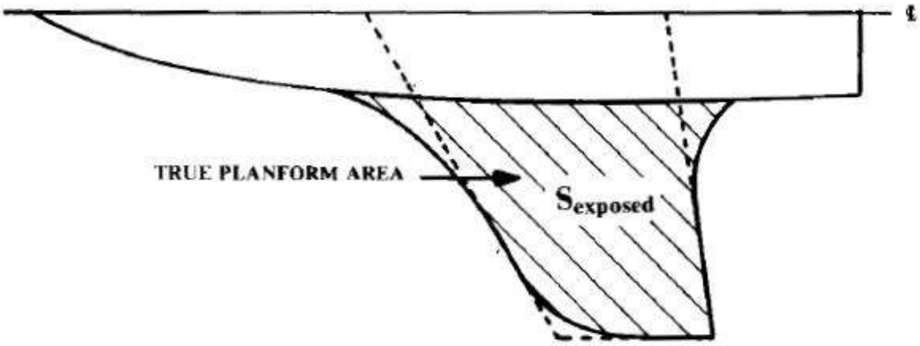
If $t/c < .05$

$$S_{wet} = 2.003\ S_{exposed}$$

If $t/c > .05$

$$S_{wet} = S_{exposed}[1.977 + 0.52(t/c)]$$

For a long, thin body circular in cross section, this average projected area



times

Figure:2.8 : Wing/tail wetted area estimate

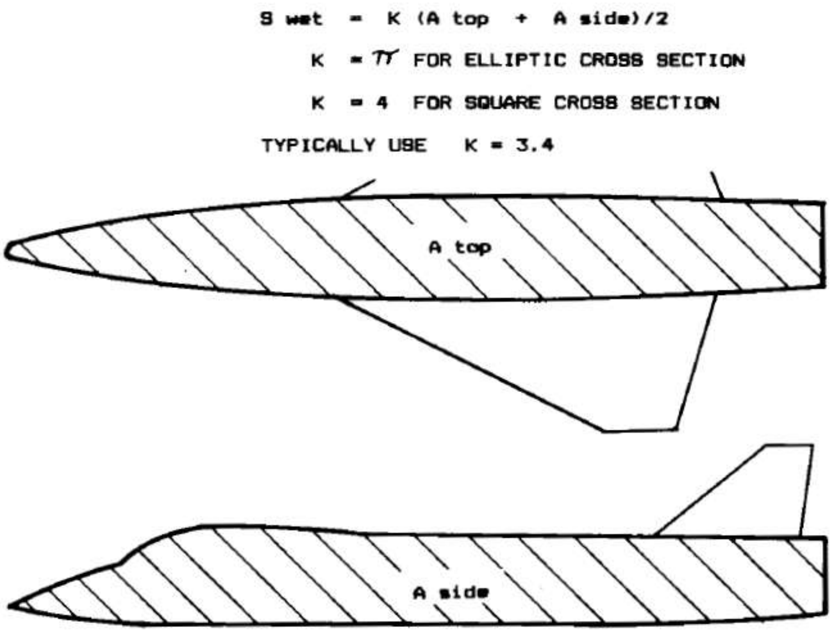


Figure 2.9: Quick fuselage wetted area estimate

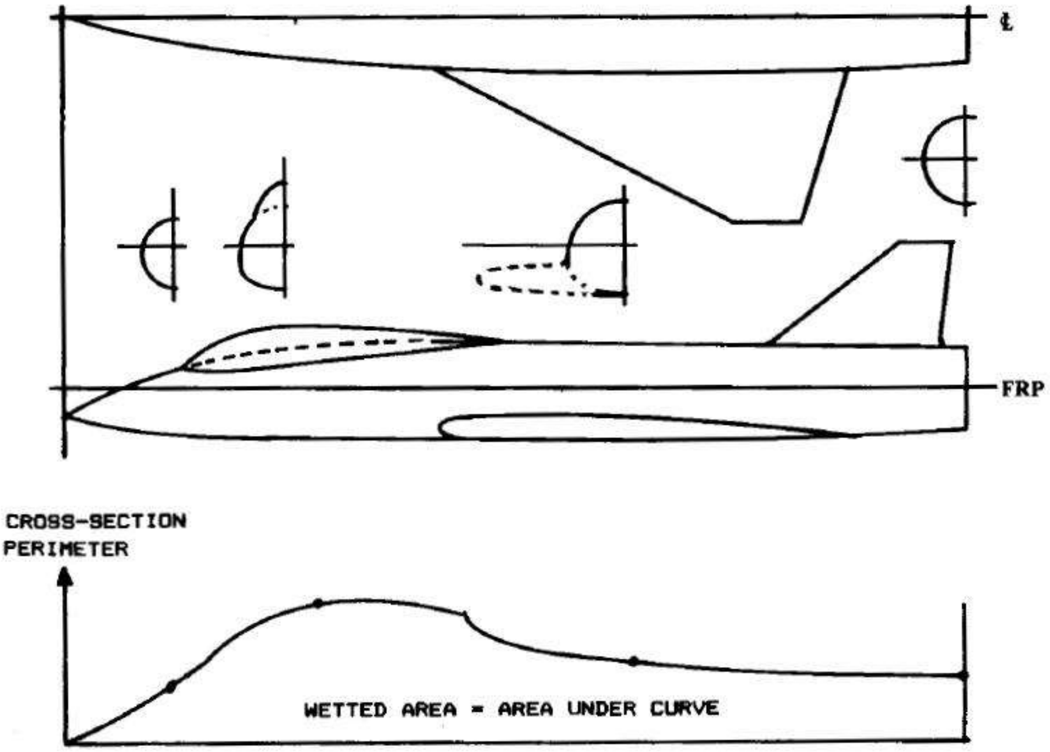


Figure 2.10: fuselage wetted plot

$$S_{\text{wet}} \cong 3.4\left(\frac{A_{\text{top}} + A_{\text{side}}}{2}\right)$$

A more accurate estimation of wetted area can be obtained by graphical integration using a number of fuselage cross sections. If the perimeters of the cross sections are measured and plotted vs. longitudinal location, using the same units on the graph, then the integrated area under the resulting curve gives the wetted area

Aircraft internal volume can be estimated in a similar fashion to the wetted-area estimation. A crude estimate of the fuselage internal volume cane made using which uses the side and top view projected areas as used in *"L"* in is the fuselage length.

$$\text{Vol} \cong 3.4\,\frac{(A_{\text{top}})(A_{\text{side}})}{4L}$$

2.6.1. VOLUME DISTRIBUTION AND FUEL VOLUME PLOTS

The aircraft internal volume can be used as a measure of the reasonableness of a new design, by comparing the volume to existing aircraft of similar weight and type. This is frequently done by customer engineering groups, using statistical data bases which correlate internal volume with takeoff gross weight for different classes of aircraft. An aircraft with a less-than typical internal volume will probably be tightly packed, which makes for poor maintainability.

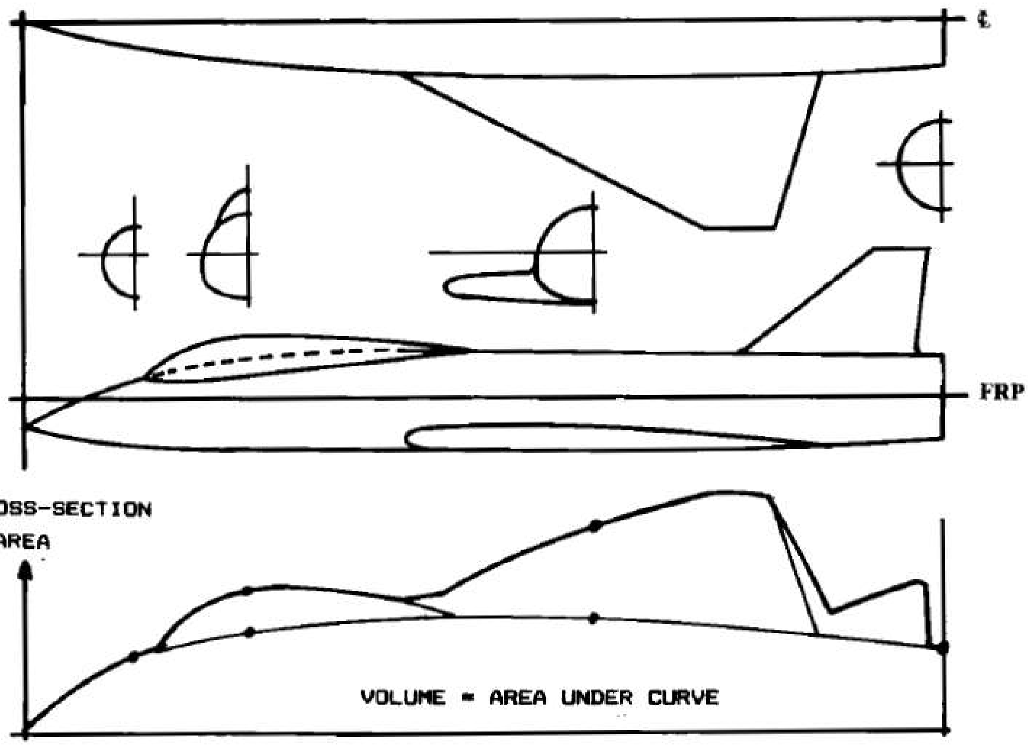


Figure 2.11: aircraft volume plot

2.7. SPECIAL CONSIDERATION IN CONFIGURATION LAY OUT:

This will focus on the required provisions for specific internal components, such as the crew station and landing gear. All of these are numerically analyzed in later stages of the design process. During configuration layout the designer must consider their impact in a qualitative sense.

2.7.1. AERODYNAMIC CONSIDERATIONS

The overall arrangement and smoothness of the fuselage can have a major effect upon aerodynamic efficiency. A poorly designed aircraft can have excessive flow separation, transonic drag rise, and supersonic wave drag. Also, a poor wing-fuselage arrangement can cause lift losses or disruption of the desired elliptical lift distribution.

Aerodynamic analysis will be discussed and variety of first-order estimation methods will be presented. During concept layout, the designer must consider the requirements for aerodynamics based upon experience and a "good eye."

Minimization of wetted area is the most powerful aerodynamic consideration for virtually all aircraft. Wetted area directly affects the friction drag. Fuselage wetted area is minimized by tight internal packaging and a low fineness ratio (i.e., a short, fat fuselage). However, excessively tight packaging should be avoided for maintainability considerations. Also, a short, fat fuselage will have a short tail moment arm which increases the required tail areas. The short, fat fuselage will also have high supersonic wave drag.

Another major driver for good aerodynamic designs during fuselage layouts the maintenance of smooth longitudinal contours. These can be provided by the use of smooth longitudinal control lines. Generally, longitudinal breaks in contour should follow a radius at least equal to the fuselage diameter at that point.

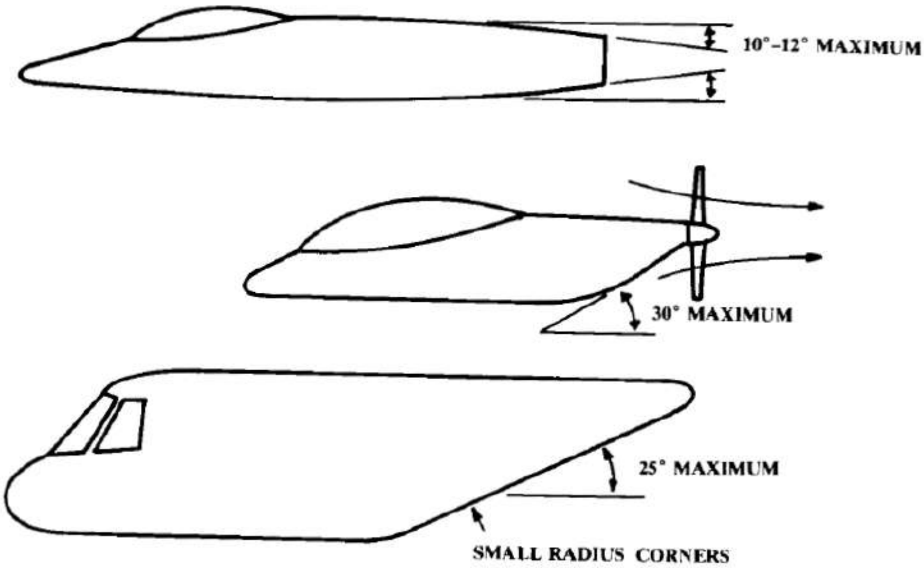


Figure 2.12: longitudinal contour guidelines

A lower-surface upsweep of about 25 deg can be tolerated for a rear-loading transport aircraft provided that the fuselage lower corners are fairly sharp. This causes a vortex-flow pattern that reduces the drag penalty. In general, aft-fuselage upsweep should be minimized as much as possible, especially for high-speed aircraft. The importance of well-designed wing fillets has already been discussed. Fillets are especially important for low-wing, high-speed aircraft such as jet transports.

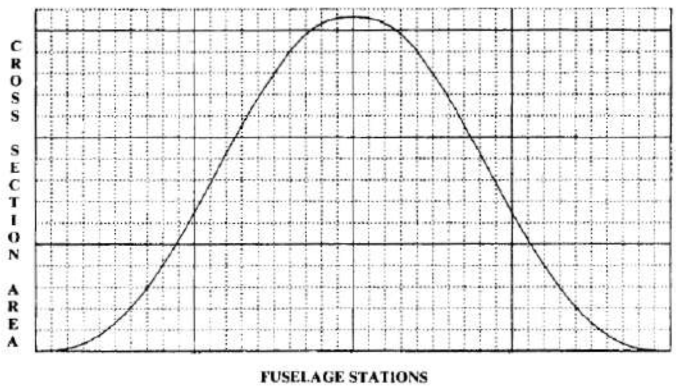


Figure2.13: Sears hack volume distribution

However, it is usually impossible to exactly or even approximately match the Sears-Haack shape for a real aircraft. Fortunately, major drag reductions can be obtained simply by smoothing the volume distribution shape. As shown in Figure, the main contributors to the cross-sectional area are the wing and the fuselage.

A typical fuselage with a trapezoidal wing will have an irregularly-shaped volume distribution with the maximum cross-sectional area located near the center of the wing. By "squeezing" the fuselage at that point, the volume-distribution shape can be smoothed and the maximum cross- sectional area reduced. While area-ruling was developed for minimization of supersonic drag.

There is reason to believe that even low-speed aircraft can benefit from it to some extent. The airflow over the wing tends to separate toward the trailing edge.

If an aircraft is designed such that the fuselage is increasing in cross-sectional area towards the wing trailing edge, this may "push" air onto the wing, thus reducing the tendency to separate. The Wittman Tailwind, which is remarkably efficient, uses this approach.

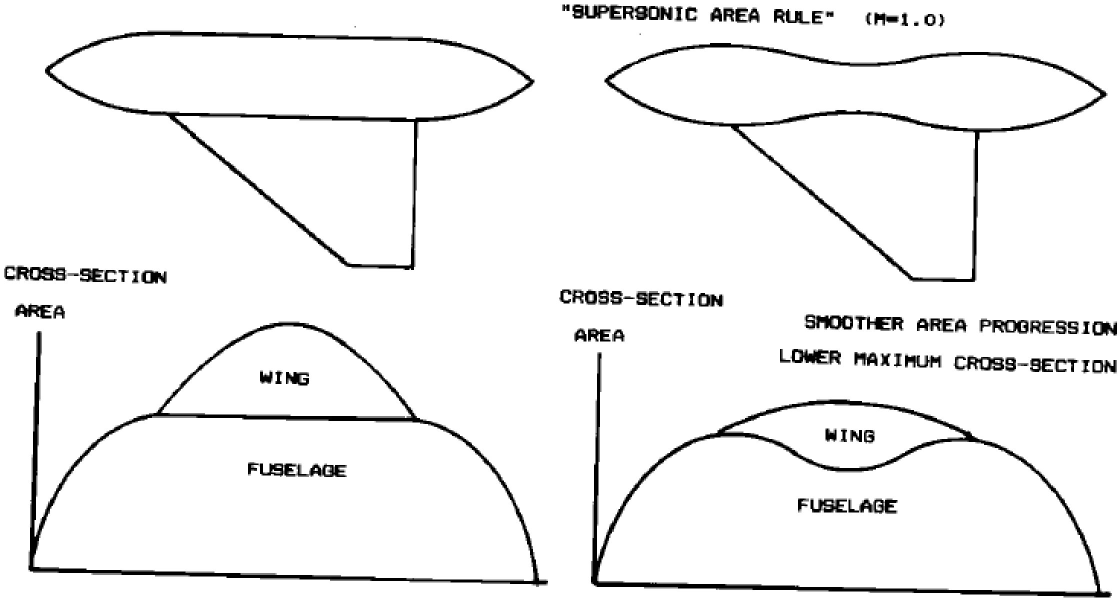


Figure 2.14: design for low wave drag

2.8. STRUCTURAL CONSIDERATIONS, STRUCTURAL LOAD PATHS

In larger companies, the configuration designer is not ultimately responsible for the structural arrangement of the aircraft. That is the responsibility of the structural design group. However, a good configuration designer will consider the structural impacts of the general arrangement of the aircraft, and will in fact have at least an initial idea as to a workable structural arrangement.

The primary concern in the development of a good structural arrangements the provision of efficient "load paths"-the structural elements by which opposing forces are connected.

The primary forces to be resolved are the lift of the wing and the opposing weight of the major parts of the aircraft, such as the engines and payload. The size and weight of the structural members will be minimized by locating these opposing forces near to each other

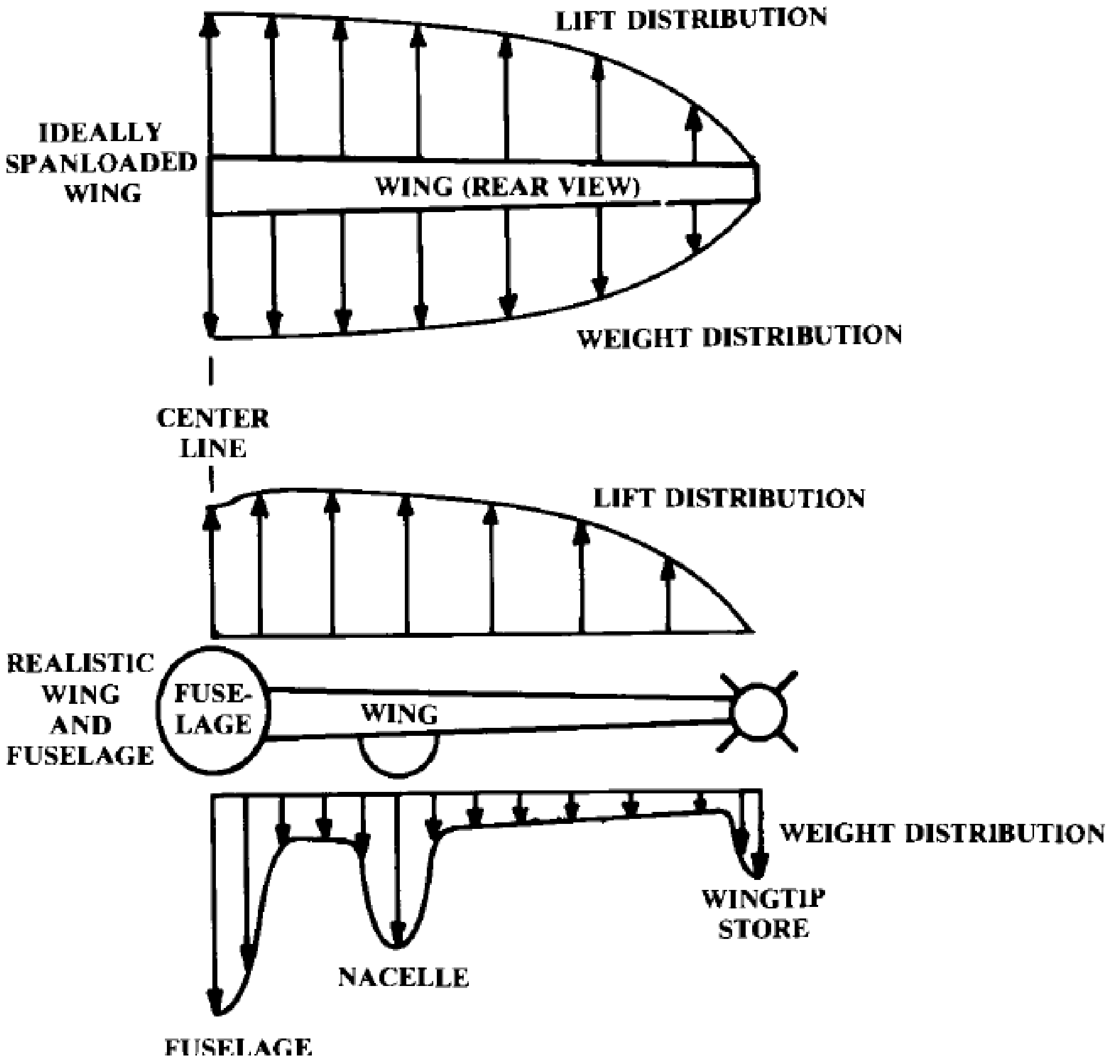


Figure1.15: Span loading for weight reduction

An especially poor arrangement (seen on some older fighter aircraft) has the main landing gear retracting into the wing-box area, which requires a large cutout where the loads are the greatest. When possible, structural cut outs should be avoided altogether. For example, a jet engine that is buried in the fuselage requires a cutout for the inlet, a cutout for the exhaust, and in most cases another cutout for removal

Aircraft with the landing gear in the wing will usually have the gear located aft of the wing box, with a single trailing-edge spar behind the gear to carry the flap loads, as shown Ribs carry the loads from the control surfaces, store stations, and landing gear to the spars and skins.

A multi spar wing box will have comparatively few ribs, located only where major loads occur. Another form of wing structure, the "multi rib" or "stringer panel" box, has only two spars, plus a large number of span wise stringers attached to the wing skins. Numerous ribs are used to maintain the shape of the box under bending. Variable sweep and folding capability add considerably to the wing structural weight. On the other hand, use of a delta wing will reduce the structural weight.

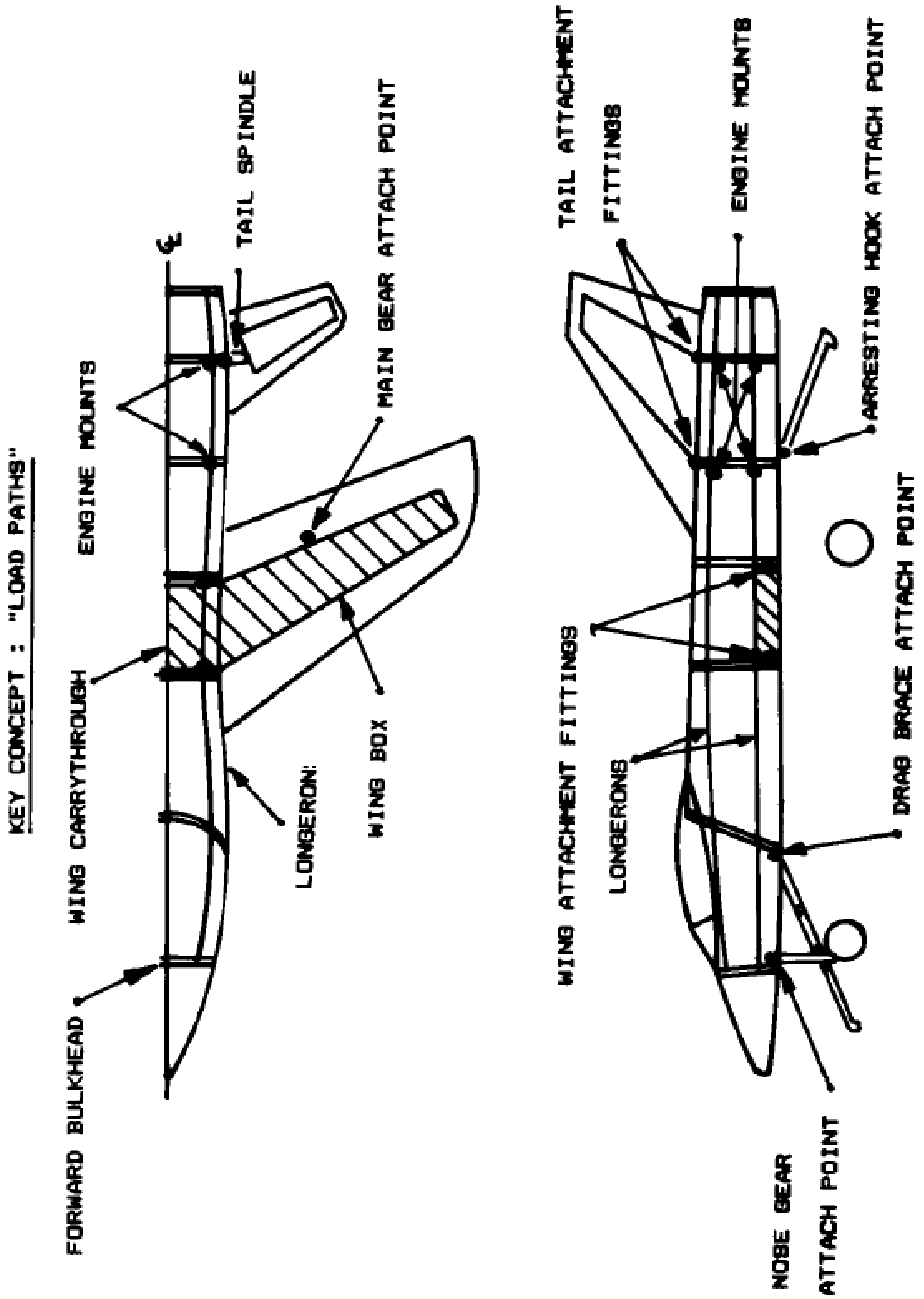


Figure 2.16 structural arrangement

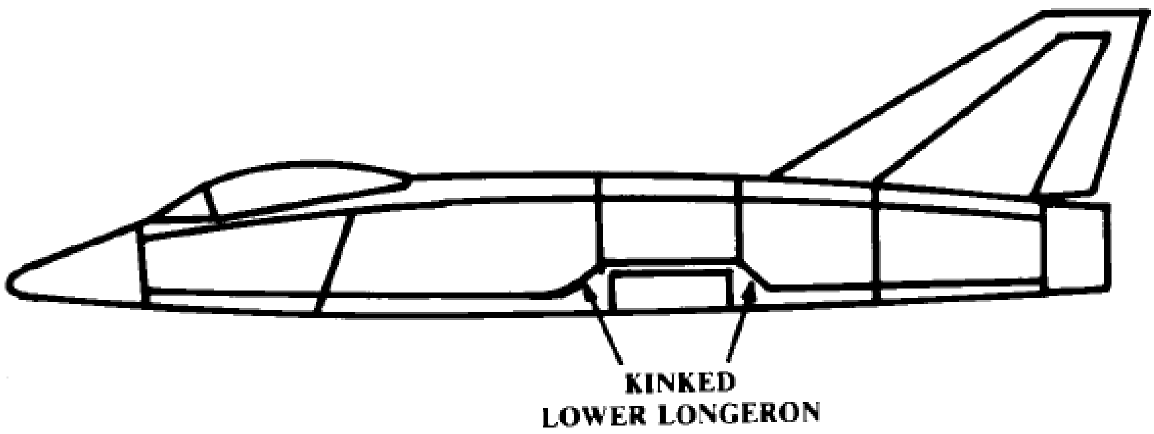


Figure 2.17: Kinked lower longeron

These are further discussed. First-order structural sizing will be discussed. For initial layout purposes the designer must guess at the amount of clearance required for structure around the internal components.

A good designer with a "calibrated eyeball" can prevent a lot of lost effort, for the aircraft may require substantial redesign if later structural analysis determines that more room is required for the structural members.

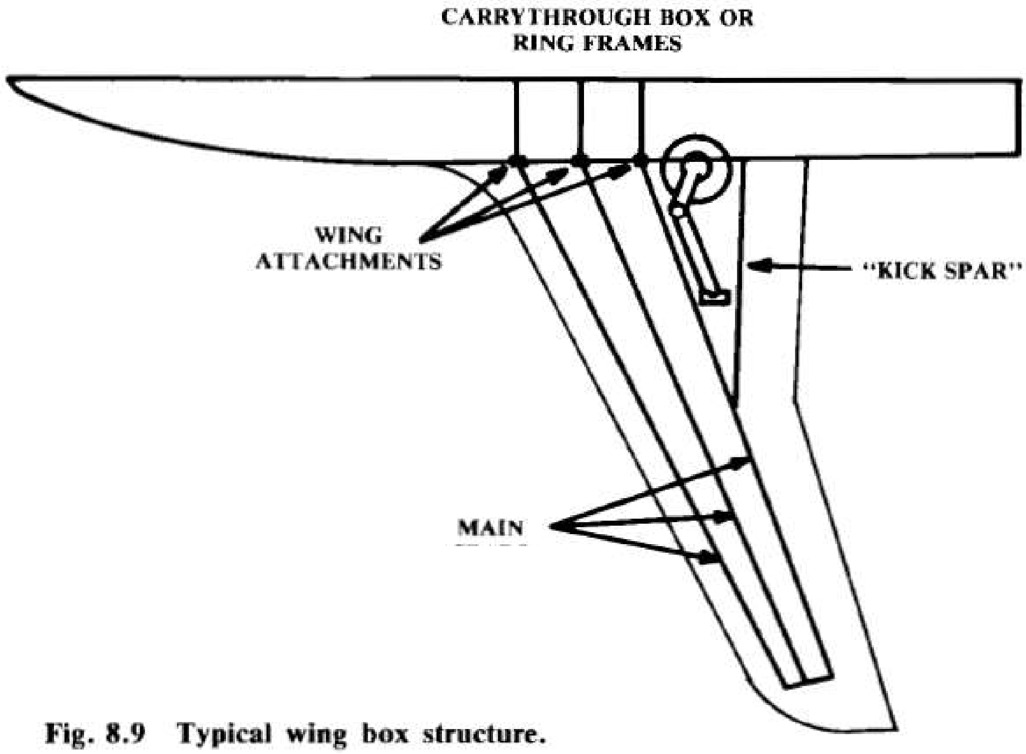


Figure 2.18: Typical wing box structure

2.9. SEARS-HAACK VOLUME DISTRIBUTION, RADAR, IR, VISUAL DETECT-ABILITY

Ever since the dawn of military aviation attempts have been made to reduce the detectability of aircraft. During World War I, the only "sensor" in use was the human eyeball. Camouflage paint in mottled patterns was used on both sides to reduce the chance of detection. Radar (acronym for Radio Detection And Ranging), the primary sensor used against aircraft today, consists of a transmitter antenna that broadcasts a directed beam of electromagnetic radio waves and a receiver antenna which picks up the faint radio waves that bounce off objects "illuminated" by the radio beam. Usually the transmitter and receiver antennas are collocated("monostatic radar"), although some systems have them m different locations ("bistatic radar").

Detectability to radar has been a concern since radar was first used in World War II. "Chaff" was the first radar "stealth" technology. Chaff,also called "window," consists of bits of metal foil or metallized fibers dropped by an aircraft to create many radar echos that hide its actual echo return. Chaff is still useful against less-sophisticated radars**.**

One of the largest contributions to airframe RCS occurs any time a relatively flat surface of the aircraft is perpendicular to the incoming radar beam. Imagine shining a flashlight at a shiny aircraft in a dark hanger. Any spots where the beam is reflected directly back at you will have an enormous RCS contribution.

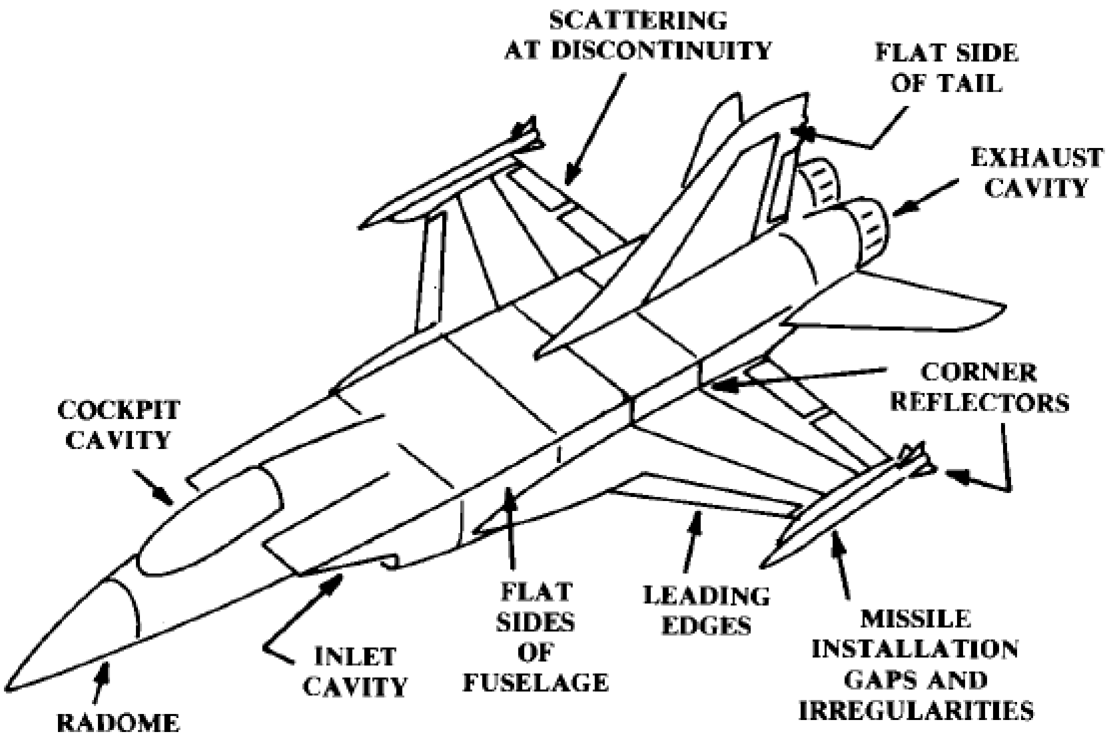


Figure 2.19: Major RCS Contributors

Another area of the aircraft which can present a perpendicular bounce for the radar is the round leading edge of the wing and tail surfaces. If the aircraft is primarily designed for low detectability by a nose-on threat radar, the wings and tails can be highly swept to reduce their contribution to RCS. Note that this and many other approaches to reducing the RCS will produce a penalty in aerodynamic efficiency. It is also important to avoid any "corner reflectors," i.e., intersecting surfaces that form approximately a right angle, as shown in Figure at the wing-fuselage junction.

One of the largest contributions to airframe RCS occurs any time a relatively flat surface of the aircraft is perpendicular to the incoming radar beam. Imagine shining a flashlight at a shiny aircraft in a dark hanger. Any spots where the beam is reflected directly back at you will have an enormous RCS contribution.

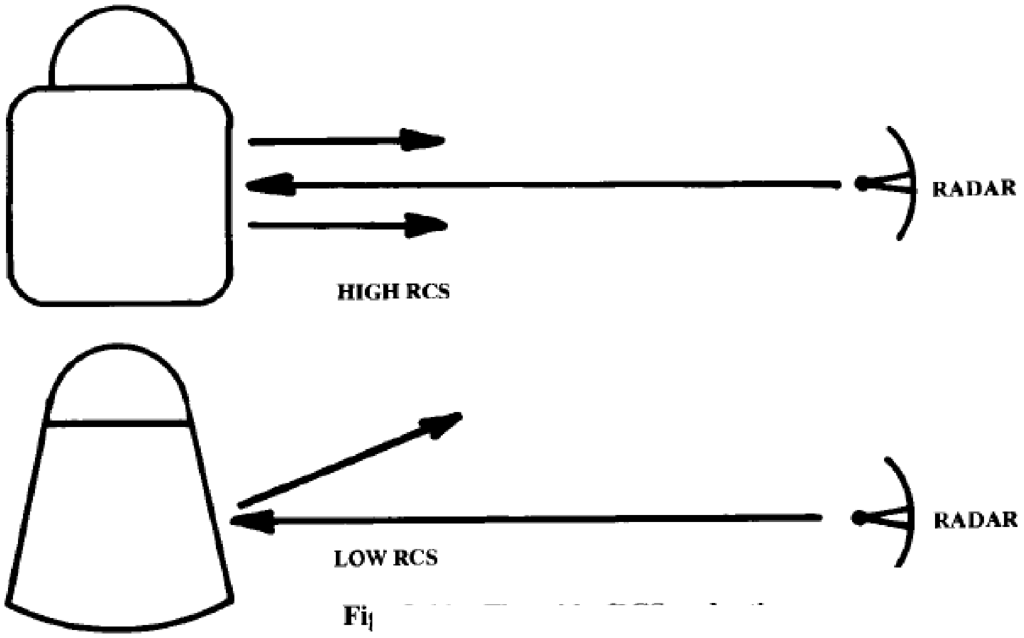


Figure 2.20: flat side RCS reduction

This effect is much lower in intensity than the specular return, but is still sufficient for detection. The effect is strongest when the discontinuity is straight and perpendicular to the radar beam.

Thus, the discontinuities such as at the wing and tail trailing edges can be swept to minimize the detectability from the front. Carried to the extreme, this leads to diamond- or saw tooth-shaped edges on every door, access plate, and other discontinuity on the aircraft,

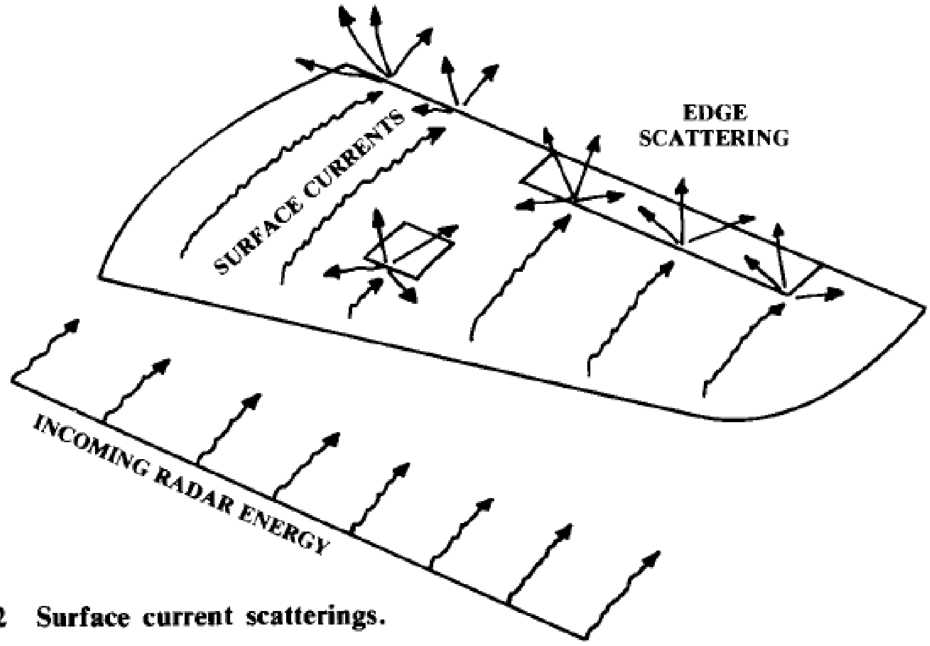


Fig. 8.12 Surface current scatterings.

Figure 2.21: Surface current scatterings

2.9.1. Aural Signature

Aural signature (noise) is important for civilian as well as military aircraft. Commercial airports frequently have antm01se ordinances that restrict some aircraft. Aircraft noise is largely caused by airflow shear layers, primarily due to the engine exhaust**.**

A small-diameter, high-velocity jet exhaust produces the greatest noise, while a large-diameter propeller with a low tip-speed produces the least noise. A turbofan falls somewhere in between. Blade shaping and maternal duct shaping can somewhat reduce noise. Piston exhaust stacks are also a source of noise. This noise can be controlled with mufflers, and by aiming the exhaust stacks away from the ground and possibly over the wings. Within the aircraft, noise is primarily caused by the engines. Well-designed engine mounts, mufflers, and insulation materials can be used to reduce the noise. Internal noise will be created if the exhaust from a piston engine impinges upon any part of the aircraft, especially the cabin.

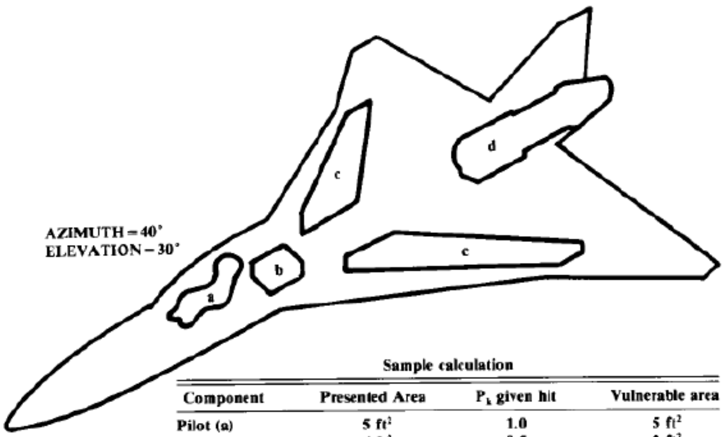
Wing-mounted propellers can have a tremendous effect on maternal nose. All propellers should have a minimum clearance to the fuselage of about 1ft, and should preferably have a minimum clearance of about one-half of the propeller radius. However, the greater the propeller clearance, the larger the vertical tail must be to counter the engine-out yaw.

2.9.2. Considerations of Vulnerability

Vulnerability concerns the ability of the aircraft to sustain battle damage, continue flying, and return to base. An aircraft can be "killed" in many ways. A single bullet through a non-redundant elevator actuator is as bad as a big missile up the tailpipe "Vulnerable area" is a key concept. This refers to the product of the projected area (square feet or meters) of the aircraft components, times the probability that each component will, if struck, cause the aircraft to be lost. Vulnerable area is different for each threat direction.

Typical components with a high aircraft kill probability (near 1.0) are the crew compartment, engine (if single-engine), fuel tanks (unless self-sealing), and weapons. Figure shows a typical vulnerable area calculation. When assessing the vulnerability of an aircraft; the first step is to determine the ways in which it can be "killed." Referred to as a "failure modes and effects analysis (FMEA)," this step will typically be performed during the later stages of conceptual design. The FMEA considers both the ways in which battle damage can affect individual aircraft components, and the ways in which damage to each component will affect the other components.

Redundancy of critical components can be used to allow the survival of the aircraft when a critical component is hit. Typical components that could be redundant include the hydraulic system, electrical system, flight control system, and fuel system. Note that while redundancy improves the survivability and reliability, it worsens the maintenance requirements because there are more components to fail



Sample calculation

Component	Presented Area	P_k given hit	Vulnerable area
Pilot (a)	5 ft²	1.0	5 ft²
Computer (b)	4 ft²	0.5	2 ft²
Fuel (c)	80 ft²	0.3	24 ft²
Engine (d)	50 ft²	0.4	20 ft²
		Total vulnerable area	51 ft²

Figure 2.22: vulnerable area calculation

2.10. CRASHWORTHINESS, PRODUCE

Airplanes crash careful design can reduce the probability of injury in a moderate crash. Several suggestions have been mentioned above, including positioning the propellers so that the blades will not strike anyone if they fly off during a crash. Also mentioned was the desire to avoid placing fuel Tanks in the fuselage of a passenger airplane (although fuel in the wing box carry through structure is usually acceptable).

Figure 2.22 shows several other design suggestions which were learned the hard way. A normal, vertical firewall in a propeller aircraft has a sharp lower corner which tends to dig into the ground, stopping the aircraft dangerously fast. Sloping the lower part of the firewall back as shown will prevent digging in therefore reducing the deceleration.

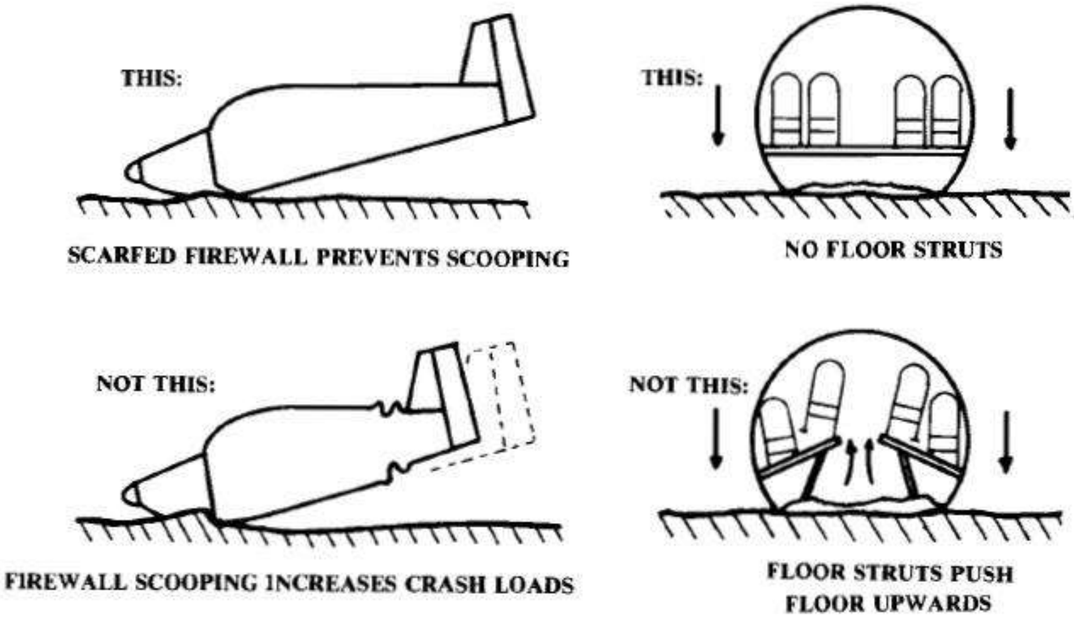


Figure 2.23: crashworthiness design

2.11. ABILITY, MAINTAINABILITY:

Maintainability means simply the ease with which the aircraft can be fixed. "Reliability and Maintainability" (R&M) are frequently bundled together and measured in "Maintenance ManhoursPerFlighthour" (MMH/FH).

MMH/FH's range from less than one for a small private aircraft to well over a hundred for a sophisticated supersonic bomber or interceptor. Reliability is usually out of the hands of the conceptual designer. Reliability depends largely upon the detail design of the avionics, engines, and other subsystems.

The configuration designer can only negatively impact reliability by placing delicate components, such as avionics, too near to vibration and heat sources such as the engines. Anybody who has attempted to repair a car will already know what the major driver is for maintainability. Getting at the internal components frequently takes longer than fixing them.

Accessibility depends upon the packaging density, number and location of doors, and number of components that must be removed to get at the broken component. Packaging density has already been discussed.

The number and location of doors on modern fighters have greatly improved over prior-generation designs. Frequently the ratio between the total area of the access doors and the total wetted area of the aircraft's fuselage is used as a measure of merit, with modern fighters approaching a value of one-half. A structural weight penalty must be paid for such access.

This leads to the temptation to use "structural doors" that carry skin loads via heavy hinges and latches. These are always more difficult to open than nonload-bearing doors because the airframe's deflection from its own weight will bind the latches and hinges. In extreme cases, the aircraft must be supported on jacks or a cradle to open these structural doors.

2.11.1. Fuselage Design- Crew Station

The crew station will affect the Conceptual design primarily in the vision requirements. Requirements for unobstructed outside vision for the pilot can determine both the location of the cockpit and the fuselage shape in the vicinity of the cockpit. For example, the pilot must be able to see the runway while on final approach, so the nose of the aircraft must slope away from the pilot's eye at some specified angle.

While this may produce greater drag than a more streamlined nose, the need for safety overrides drag considerations. Similarly, the need for over-side vision may prevent locating the cockpit directly above the wing. When laying out an aircraft's cockpit, it is first necessary to decide what range of pilot sizes to accommodate. For most military aircraft, the design requirements include accommodation of the 5th to the 95th percentile of male pilots.

Due to the expense of designing aircraft that will accommodate smaller or larger pilots, the services exclude such people from pilot training. Women are only now entering the military flying profession in substantial numbers, and a standard percentile range for the accommodation of female pilots had not yet been established as this was written. Future military aircraft might require the accommodation of approximately the 20th percentile female and larger. This may affect the detailed layout of cockpit controls and displays, but should have little impact upon conceptual cockpit layout.

2.11.2. Passenger Compartment:

General-aviation cockpits are designed to whatever range of pilot sizes the marketing department feels is needed for isomer appeal, but typically are comfortable only for those under about 72 in. Commercial-airliner cockpits are designed to accommodate pilot sizes similar to those of military Aircraft. This cockpit layout uses a typical 13-deg seatback angle, but seatback angles of 30 deg are in use (F-16), and angles of up to 70 deg have been considered for advanced fighter studies. This entails a substantial penalty

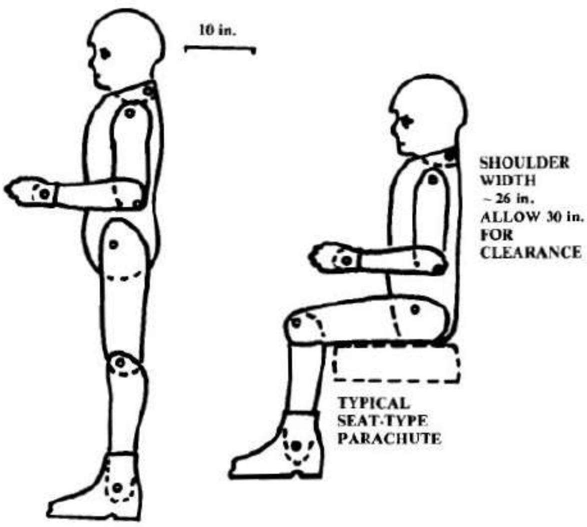


Figure 2.24: average 95th percentile pilot

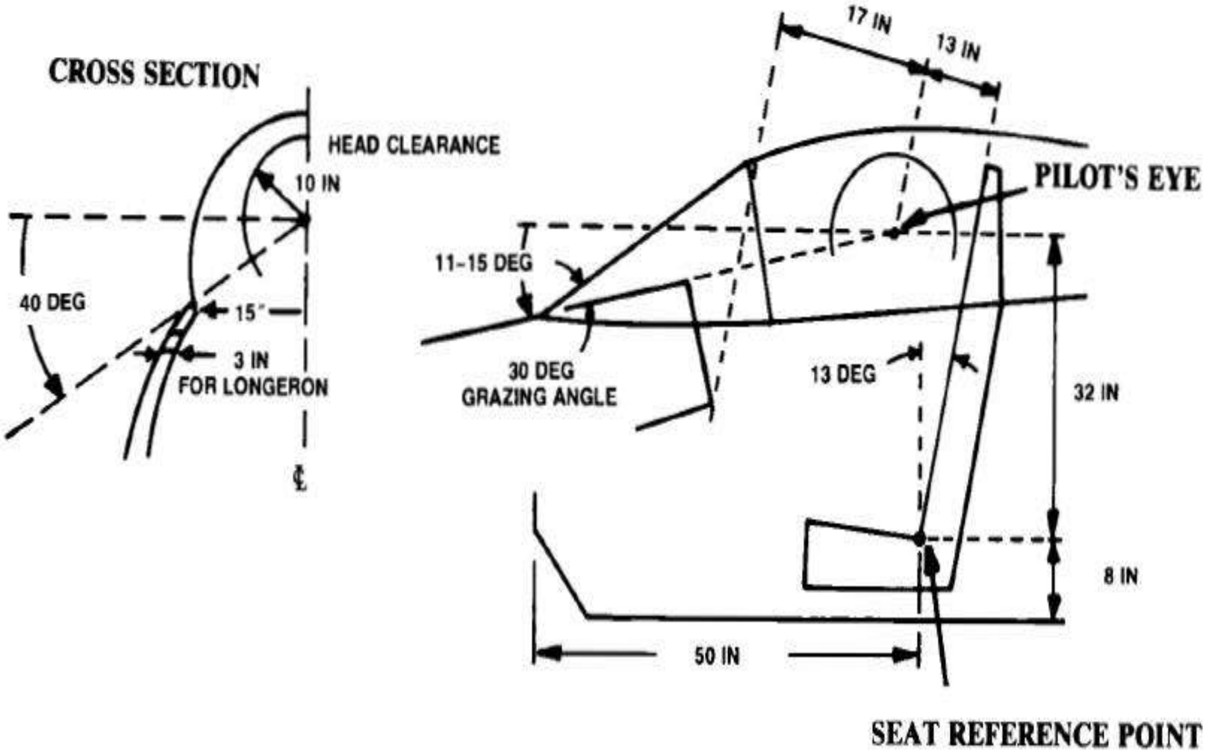


Figure 2.25: Typical fighter cockpit

2.11.3. PASSENGER COMPARTMENT

The actual cabin arrangement for a commercial aircraft is determined more by marketing than by regulations. Defines the dimensions of interest. "Pitch" of the seats is defined as the distance from the back of one seat to the back of the next. Pitch includes fore and aft seat length as well as leg room. "Headroom" is the height from the floor to the roof over the seats. For many smaller aircraft the sidewall of the fuselage cuts off a portion of the outer seat's headroom, as shown. In such a case it is important to assure that the outer passenger has a 10-in. clearance radius about the eye position.

Table 2.2: Typical passenger compartment data

	First class	Economy	High density/ small aircraft
Seat pitch (in.)	38–40	34–36	30–32
Seat width (in.)	20–28	17–22	16–18
Headroom (in.)	>65	>65	–
Aisle width (in.)	20–28	18–20	≥ 12
Aisle height (in.)	>76	>76	>60
Passengers per cabin staff (international-domestic)	16–20	31–36	≤ 50
Passengers per lavatory ($40'' \times 40''$)	10–20	40–60	40–60
Galley volume per passenger (ft^3/pass)	5–8	1–2	0–1

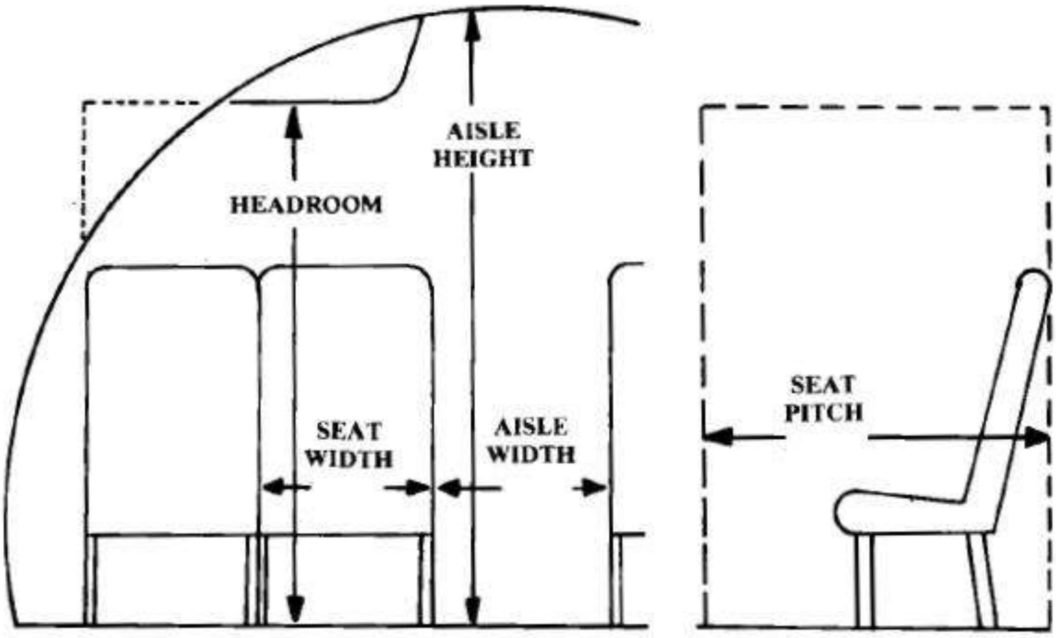


Figure 2.26: commercial passenger allowance

2.11.4. CARGO PROVISIONS:

Cargo must be carried in a secure fashion to prevent shifting while in flight. Large civilian transports use standard cargo containers that are pre-loaded with cargo and luggage and then placed into the belly of the aircraft. During conceptual design it is best to attempt to use an existing container rather than requiring purchase of a large inventory of new containers.

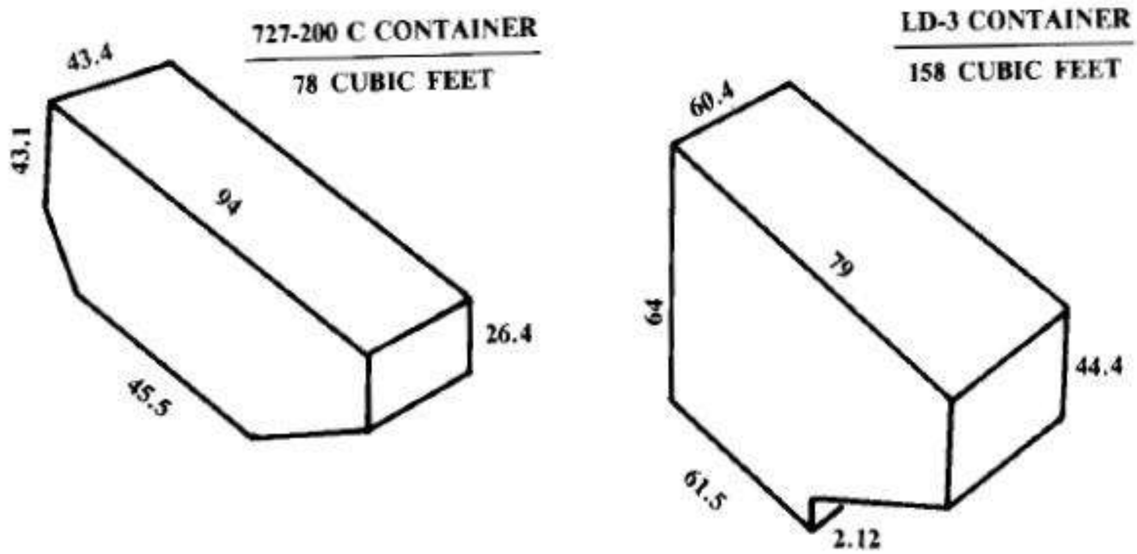


Figure 2.27: cargo containers

Cargo must be carried in a secure fashion to prevent shifting while in flight. Large civilian transports use standard cargo containers that are pre-loaded with cargo and luggage and then placed into the belly of the aircraft. During conceptual design it is best to attempt to use an existing container rather than requiring purchase of a large inventory of new containers.

2.12. WEAPONS CARRIAGE

Carriage of weapons is the purpose of most military aircraft. Traditional weapons include guns, bombs, and missiles. Lasers and other exotic technologies may someday become feasible as airborne weapons but will not be discussed here the weapons are a substantial portion of the aircraft's total weight. This requires that the weapons be located near the aircraft's center of gravity. Otherwise the aircraft would pitch up or down when the weapons are released.

Missiles different room bombs primarily in that missiles are powered. Today, virtually all missiles are also guided in some fashion. Most bombs are "dumb," or unguided, and are placed upon a target by some bombsight mechanism or computer which releases them at the proper position and velocity so that they free-fall to the desired target. However, "smart bombs," which have some guidance mechanism, are also in use. Missiles are launched from the aircraft in one of two ways.

Most of the smaller missiles such as the AIM-9 are rail-launched. A rail-launcher is mounted to the aircraft, usually at the wingtip or on a pylon under the wing. Attached to the missile are several mounting lugs, which slide onto the rail as shown

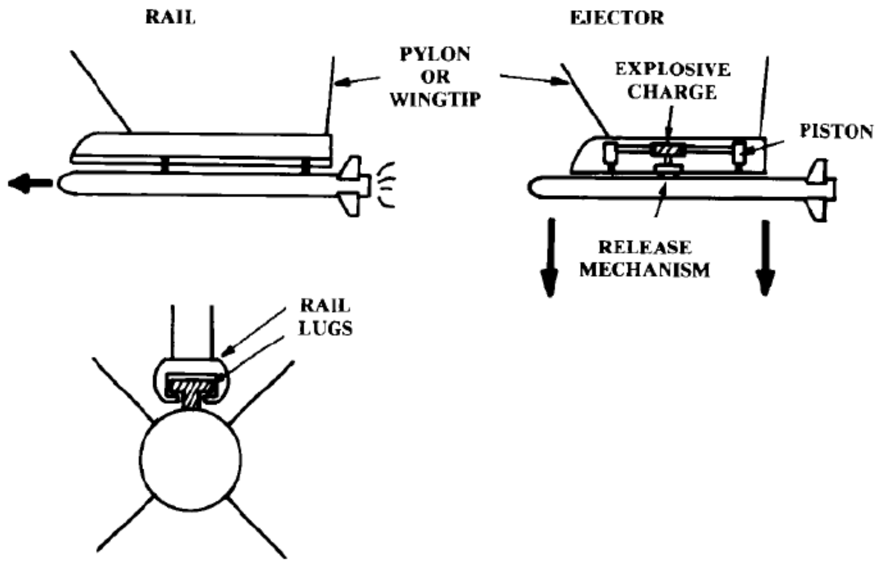
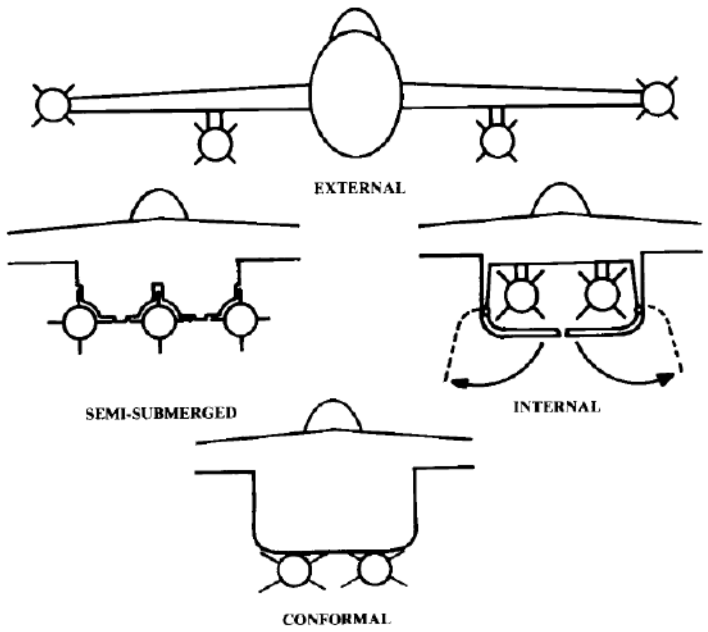


Figure 2.27 : Missile carriage/launch

While most fighter aircraft are designed to an air-to-au role, the ability to perform an additional air-to-ground role is often imposed. To avoid analyzing the aircraft's performance when "clean" (i.e., set up for dog fighting), most fighter aircraft have "hard points" under the wing and fuselage to which weapon pylons can be attached, as shown in Fig.2.27.

There are used to carry additional external weapons, and are removed for maximum dogfighting performance



2.12.1. Weapon carriage options

Clearance around the missiles and bombs is also important for safety. To insure that the weapons never strike the ground, the designer should provide

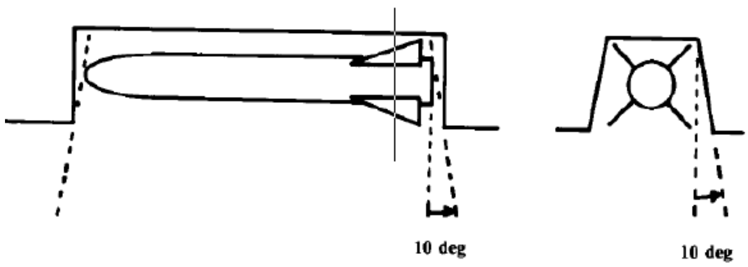


Figure: 2.28 weapon release clearance

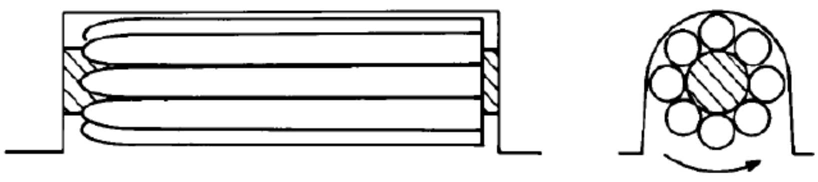


Figure 2.29: Rotary weapons bay

2.13. GUN INSTALLATION

The gun has been the primary weapon of the air-to-air fighter since the First World War I scout pilot took a shot at an opposing scout pilot with a Handgun. For a time during the 1950's it was felt that the then-new air-to air Missiles would replace the gun, and in fact several fighters such as the F-4 and F-104 were originally designed without guns.

History proved that missiles cannot be solely relied upon, and all new fighters are being designed with guns. The standard U.S. air-to-air gun today is the M61Al "Vulcan" six-barrel Gatling gun, shown in Figure. This is used in the F-15, F-16, F-18, and others. Note the ammunition container.

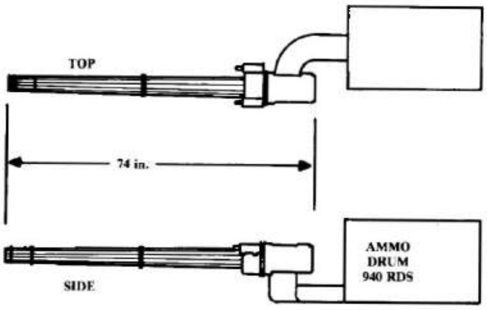


Figure 2.30 : M61 “VULCAN” gun

This must be located near the attend of the gun. Rounds of ammo are fed out of the container ("drum") through feed chutes and into the gun. Ammo is loaded into the drum by attaching an ammo loading cart to the feed chute shown. The door to this loading chute must be accessible from the ground.

UNIT-III
PROPULSION AND FUEL SYSTEM INTEGRATION, LANDING GEAR AND SUBSYSTEMS

3.1. PROPULSION SELECTION

Figure 3.1 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 via 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. By the dawn of the jet era, a 5500-hp piston-prop engine was in development. Today piston props are mainly limited to light airplanes and some agricultural aircraft. 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. Also, the propeller loses 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 are first passed through a turbine to extract enough mechanical power to drive the compressor. It is interesting to note that one early jet engine used a separate piston engine to drive the compressor.

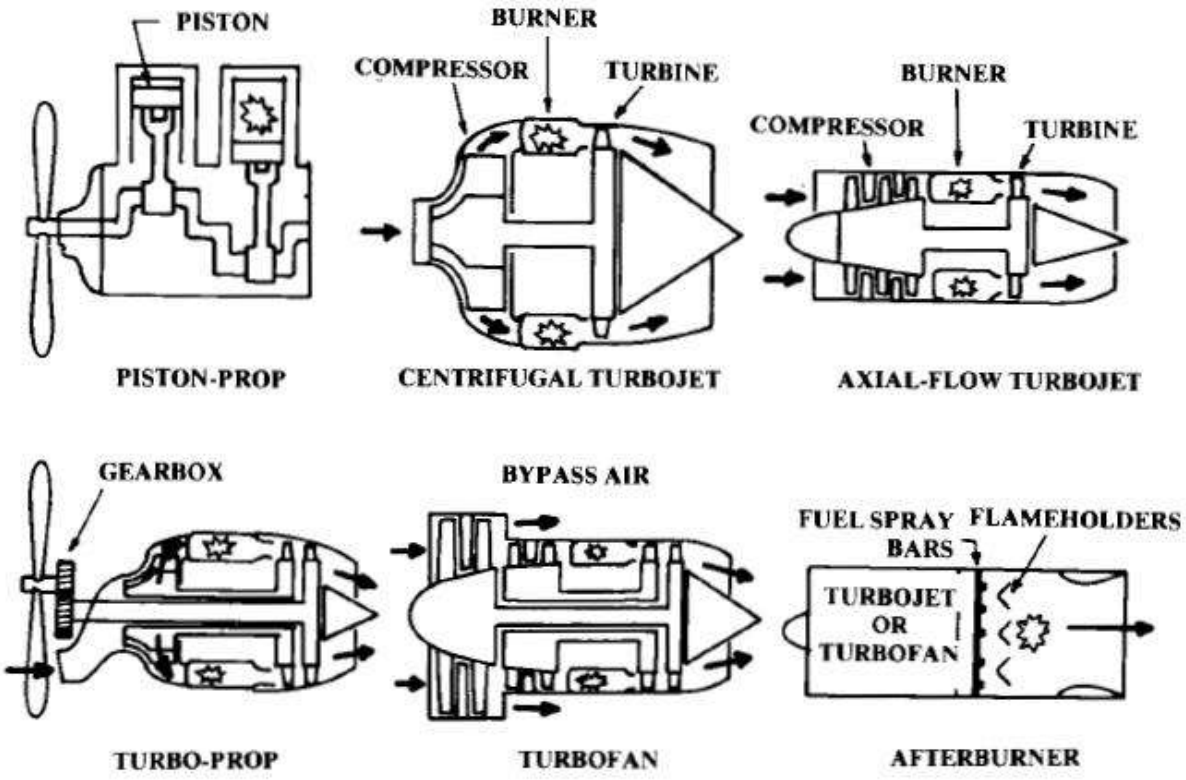


Figure 3.1: Propulsion system options

The axial compressor, relying upon blade aerodynamics, is intolerant to distortions in the incoming air such as swirl or pressure variations. These distortions can stall the blades, causing a loss of compression and a possible engine flame-out.

The centrifugal compressor is much more forgiving of inlet distortion, but causes the engine to have a substantially higher frontal area, which increases aircraft drag. Also, a centrifugal compressor cannot provide as great a pressure increase (pressure ratio) as an axial compressor. Several Smaller turbine engines use a centrifugal compressor behind an axial compressor to attempt to get the best of both types.

The turboprop 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 "inducted fan" is essentially a turboprop with an advanced aerodynamics propeller capable of near-sonic speeds.

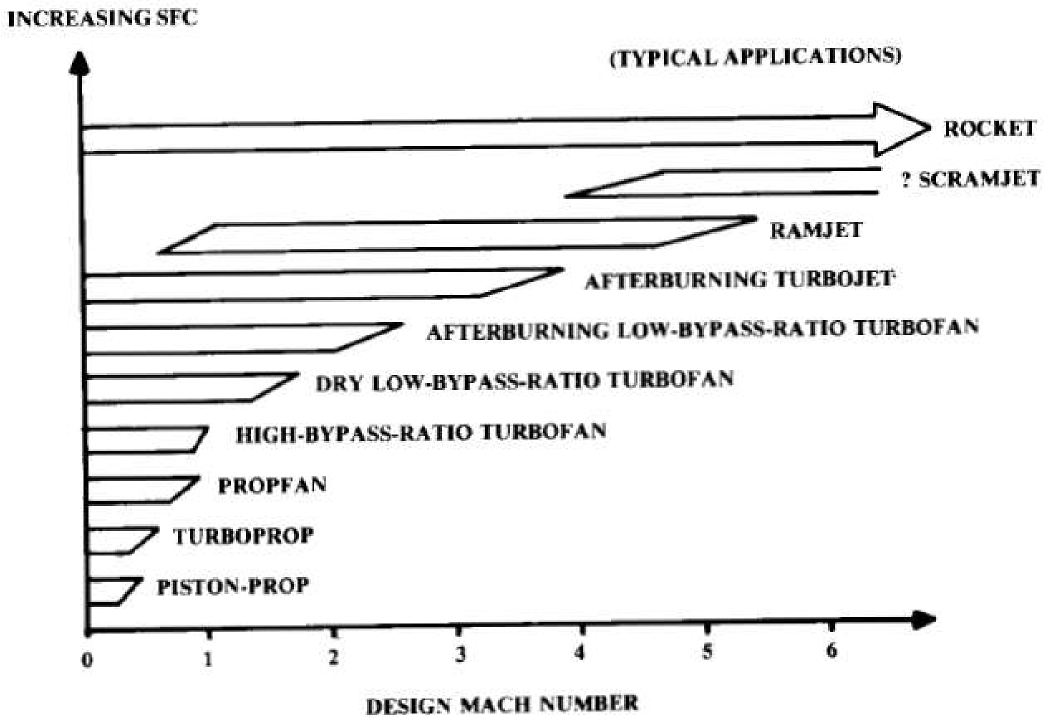


Figure 3.2: propulsion system speed limits

3.2. JET ENGINE INTEGRATION ENGINE DIMENSIONS

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. In 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. Appendix provides data for several hypothetical advanced engines. Better yet, engine companies sometimes provide a "parametric deck," a computer program that will provide performance and dimensional data for an arbitrary advanced-technology engine based upon inputs such as bypass ratio, overall pressure ratio, and turbine-inlet temperature. This kind of program, which provides great flexibility for early trade studies, goes beyond the scope of this book.

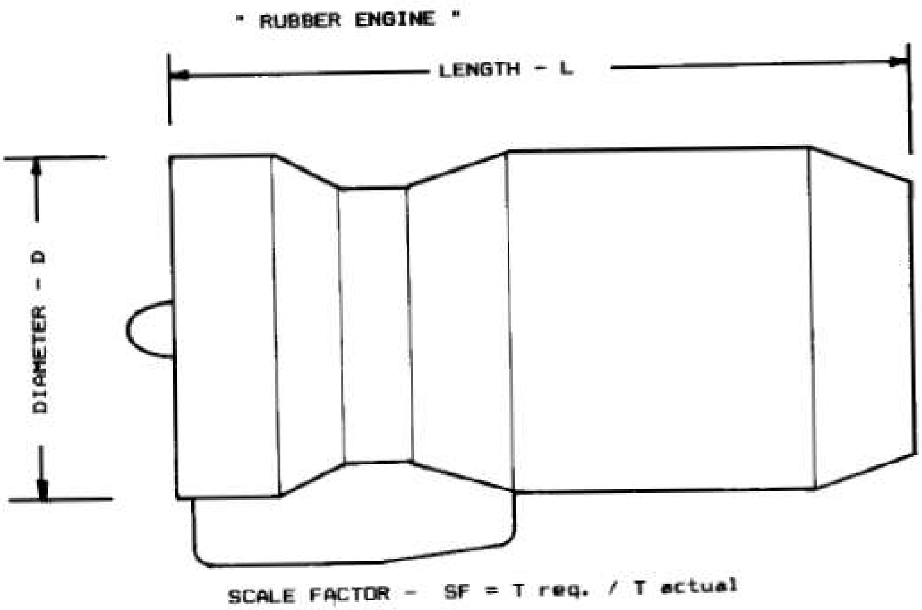


Figure 3.3: Engine scalling

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

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

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

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

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

3.2.1. Inlet Geometry:

Turbojet and turbofan engines are incapable of efficient operation unless the air entering them is slowed to a speed of about Mach 0.4-0.5. This is to keep the tip speed of the compressor blades below sonic speed relative to the incoming air. Slowing down the incoming air is the primary purpose of an inlet system.

The installed performance of a jet engine greatly depends upon the air inlet system. The type and geometry of the inlet and inlet duct will determine the pressure loss and distortion of the air supplied to the engine, which will affect the installed thrust and fuel consumption. Roughly speaking, a1 % reduction in inlet pressure recovery (total pressure delivered to the engine divided by free stream total pressure) will reduce thrust by about 1.3%.

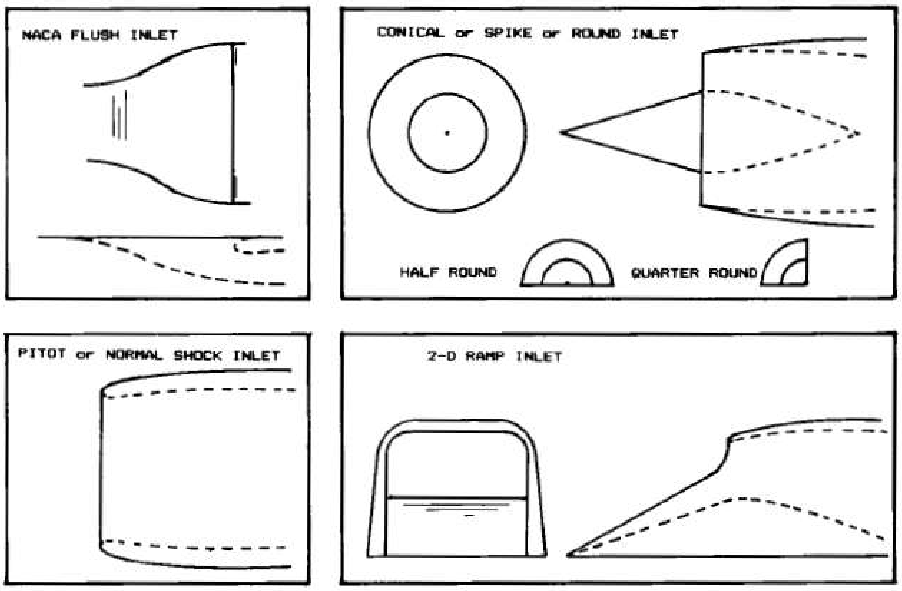


Figure 3.4: Inlet types

Aircraft wetted area and weight if the engine is in the fuselage. The NACA inlet is regularly used for applications in which pressure recovery is less important, such as the intakes for cooling air or for turbine powered aux1hary power units. The *BD-5J,* a jet version of the BD-5 homebuilt, used the NACA inlet, probably to minimize the redesign effort.

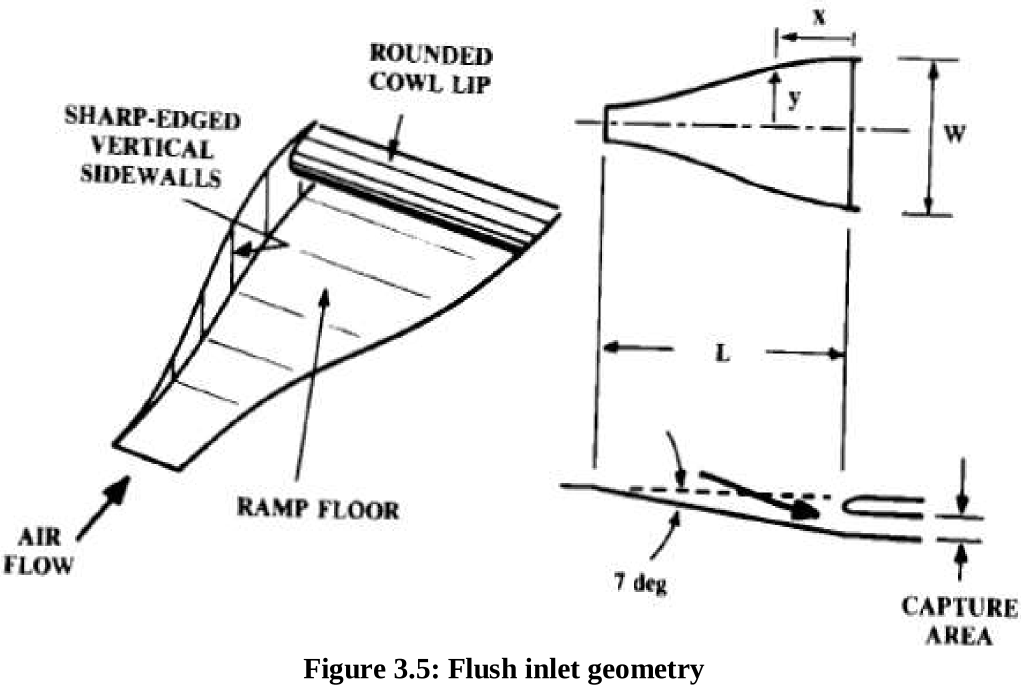


Figure 3.5: Flush inlet geometry

The pilot inlet is simply a forward-facing hole. It works very well sub-sonically and fairly well at low supersonic speeds. It is also called a "normal shock inlet" when used for supersonic flight ("normal" meaning perpendicular in this case). Figure gives design guidance for pilot inlets.

The cowl lip radius has a major influence upon engine performance and aircraft drag. A large lip radius tends to minimize distortion, especially at high angles of attack and sideslip. Also, a large lip radius will readily accommodate the additional air required for takeoff thrust, when the ram air effect is small.

However, a large lip radius will produce shock-separated flow on the outside of the inlet as the speed of sound is approached, and that greatly increases the drag. For supersonic jets, the cowl lip should be nearly sharp. Typically the lip radius will be about 3-50Jo of the inlet front face radius. For subsonic jets, the lip radius ranges from 6-lOOJo of the inlet radius.

To minimize distortion the lip radius on a subsonic inlet is frequently Greater on the inside than the outside, with perhaps an 80Jo inner radius and a 40Jo outer radius. Also, a number of aircraft have a lip radius on the lower part of the inlet up to 500Jo greater than that on the upper lip.

This reduces the effects of angle of attack during takeoff and landing. Note that the inlet front face may not be perpendicular to the engine axis. The desired front-face orientation depends upon the location of the inlet and the aircraft's angle-of-attack range.

Normally the inlet should be about perpendicular to the local flow direction during cruise. If the aircraft is to operate at large angles of attack, it may be desirable to compromise between these angles and the angle at cruise.

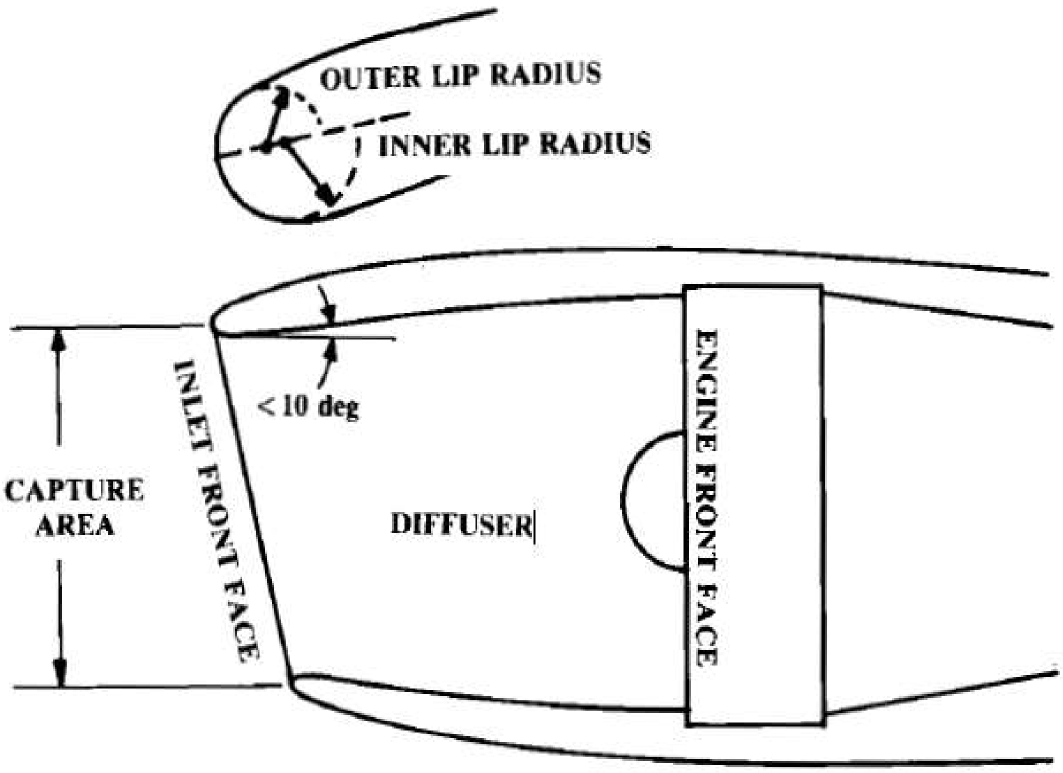


Figure 3.6 Pilot (normal shock) inlet layout

Some form of boundary-layer bleed is required on the ramp to prevent shock-induced separation on the ramp. The bled air is usually dumped overboard out a rearward-facing hole above the inlet duct.

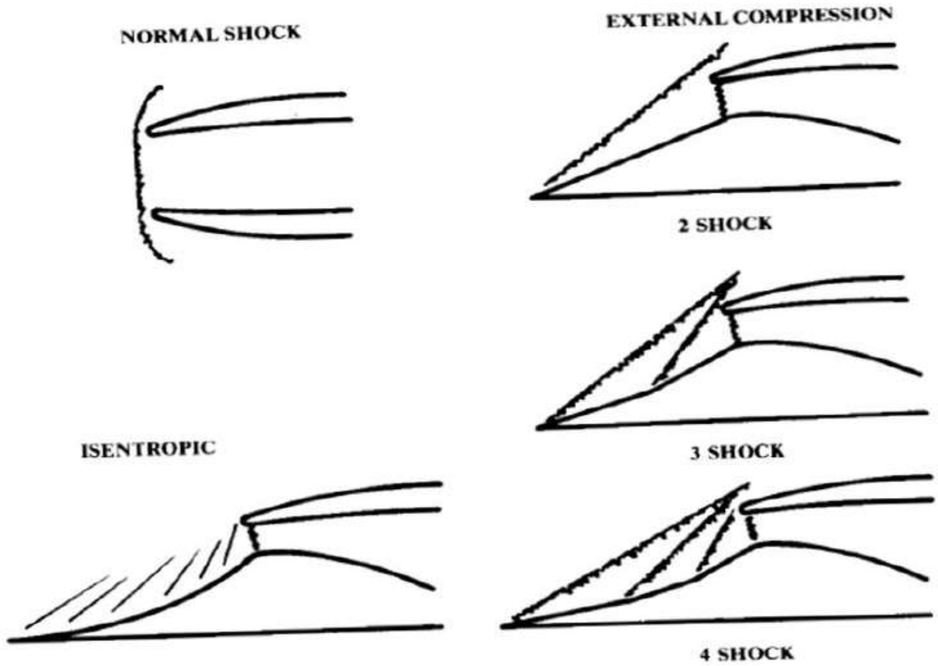


Figure 3.7: supersonic inlets-external shocks

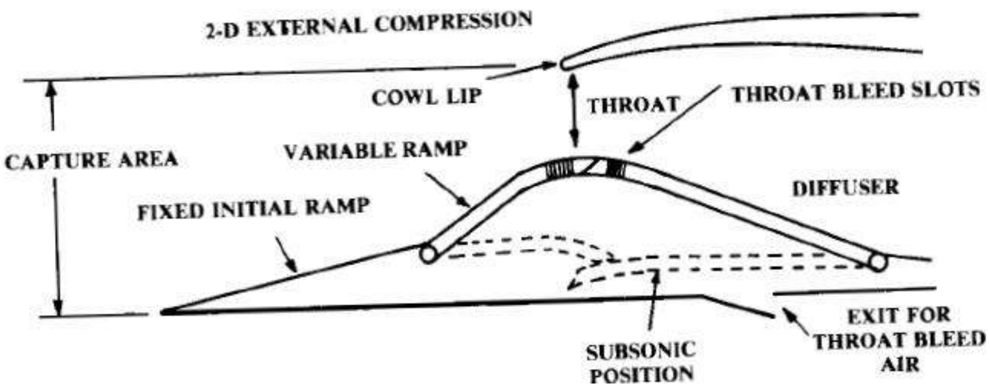


Figure 3.8: variable inlet geometry

For a supersonic application, the theoretical diffuser length for maximum efficiency is about eight times the diameter. Lengths longer than eight times the diameter are permissible but have internal friction losses as well as an additional weight penalty. A supersonic diffuser shorter tt1an about tour times the diameter may produce some internal flow separation, but the weight savings can exceed the engine performance penalty.

Diffusers as short as two times the diameter have been used with axisymmetric spike inlets. For a long diffuser it is important to verify that the cross-sectional area of the flow path is smoothly increasing from the inlet front face back to the engine. This verification is done with a volume-distribution plot of the inlet duct, constructed in the same fashion as the aircraft volume plot shown .To reduce distortion, some aircraft use a diffuser oversized about 5%that "pinches" the flow down to the engine front-face diameter in a very short distance just before the engine

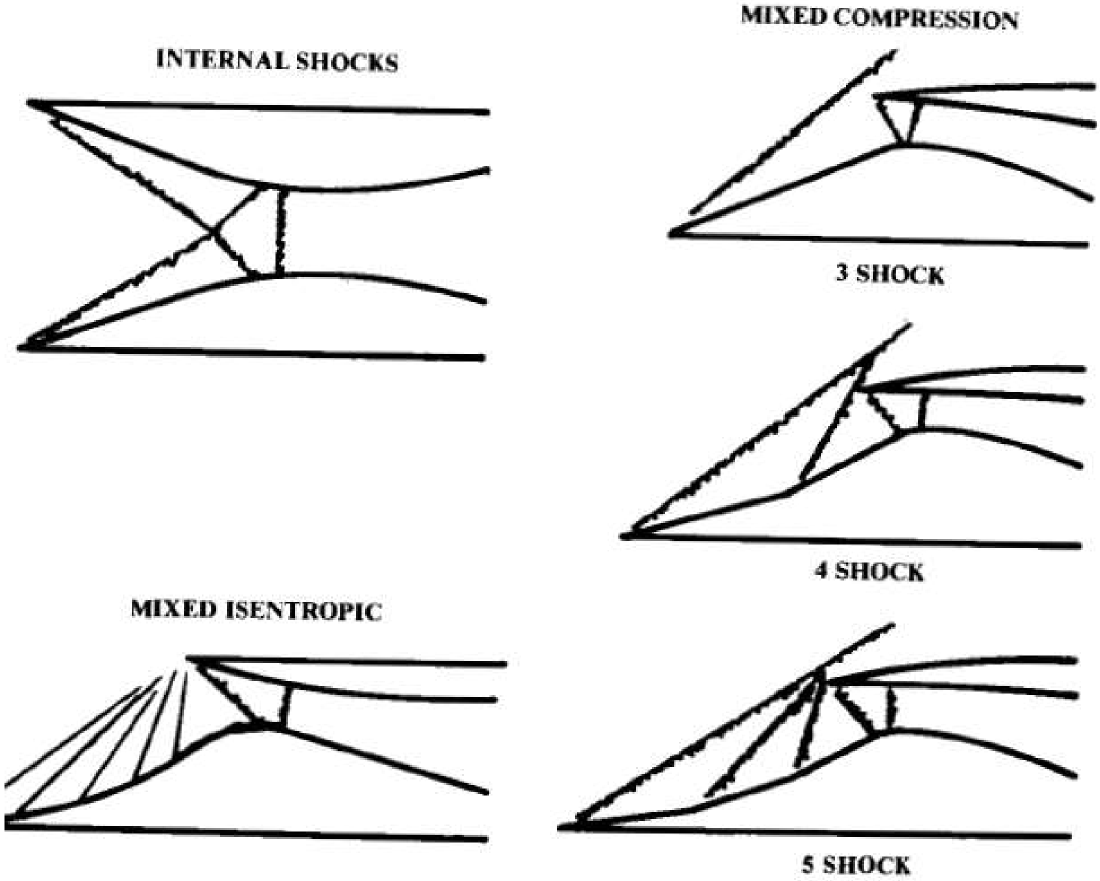


Figure 3.9: supersonic inlets-internal and mixed

Figure 3.10 summarizes the selection criteria for different inlets, based upon design Mach number. Note that these are approximate criteria, and may be overruled by special cons1aerations. Estimated pressure recoveries of these inlets is provided.

3.2.2. INLET LOCATION

The inlet location can have almost as great an effect on engine performance as the inlet geometry. If the inlet is located where it can ingest vortex off the fuselage or a separated wake from a wing, the resulting

An over-fuselage inlet is much Tike an inverted chin inlet, and has a short duct length but without the problems of nose-wheel location. This was used on the unusual F-107. The upper-fuselage inlet is poor at high angle of attack because the fore body blanks the airflow.

Also, many pilots fear that they may be sucked down the inlet if forced to bail out manually. Placed over the wing and near the fuselage, an inlet encounters problems similar to those of an inverted-armpit inlet. It also suffers at angle of attack. An inlet above the aft fuselage for a buried engine is used on the L-1011 and B-727, with the inlet located at the root of the vertical tail.

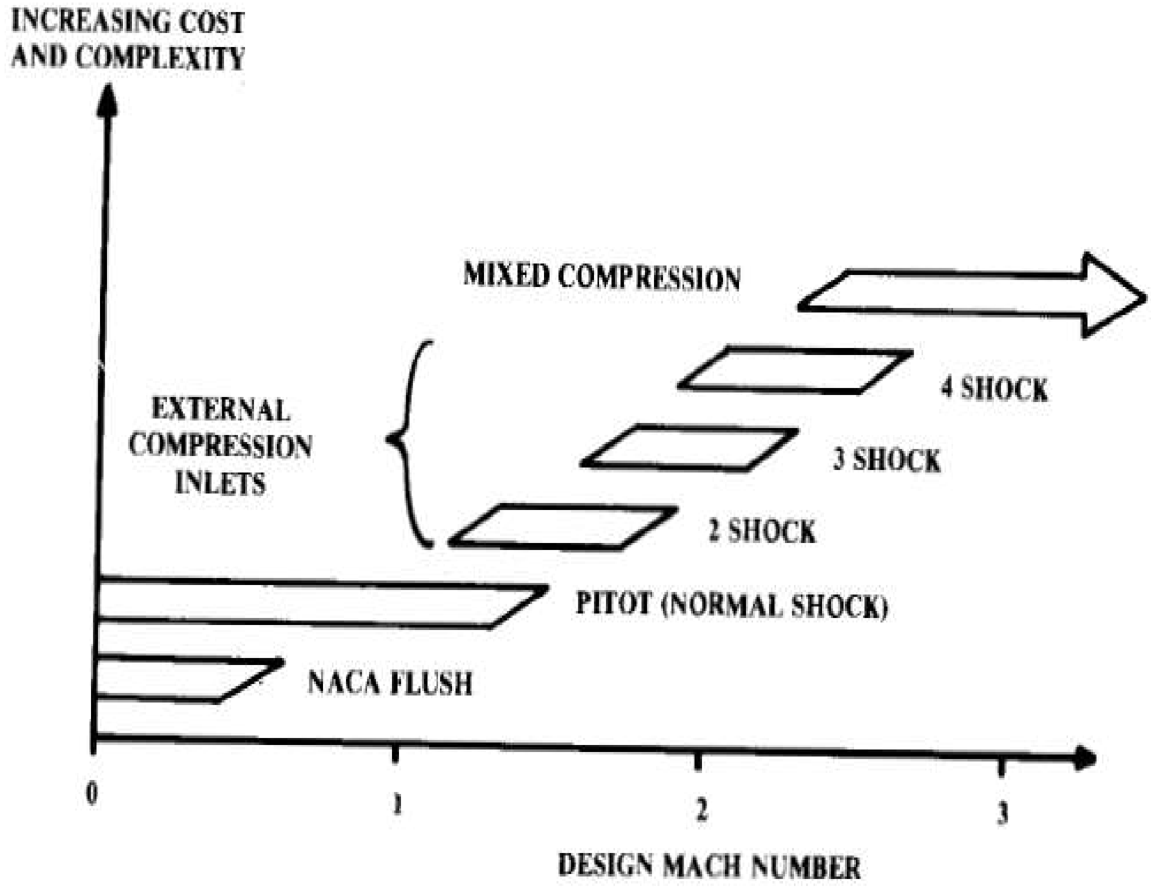


Figure 3.10: Inlet applicability

This arrangement allows the engine exhaust to be placed at the rear of the fuselage, which tends to reduce fuselage separation and drag. The buried engine with a tail inlet must use an "S duct." This requires careful design to avoid internal separation.

Also, the inlet should be well above the fuselage to avoid ingesting the thick boundary layer On the basis of years of wind-tunnel study, design charts for pylon mounted engines have been prepared that minimize the interference effects of the nacelle pod on the wing.

As a classical rule-of-thumb, the inlet for a wing-mounted podded engine should be located approximately two inlet diameters forward and one inlet diameter below the wing leading edge

The over-fuselage podded engine has been used only rarely. Most recently it was used to add a jet engine to the turboprop Rockwell OV-10. Access and cabin noise are undesirable for this installation.

The wing tip-mounted engine has obvious engine-out controllability Problem. It was used on the Soviet supersonic Myasishchev M-52 ("Bounder"), which also had under-wing engine pods.

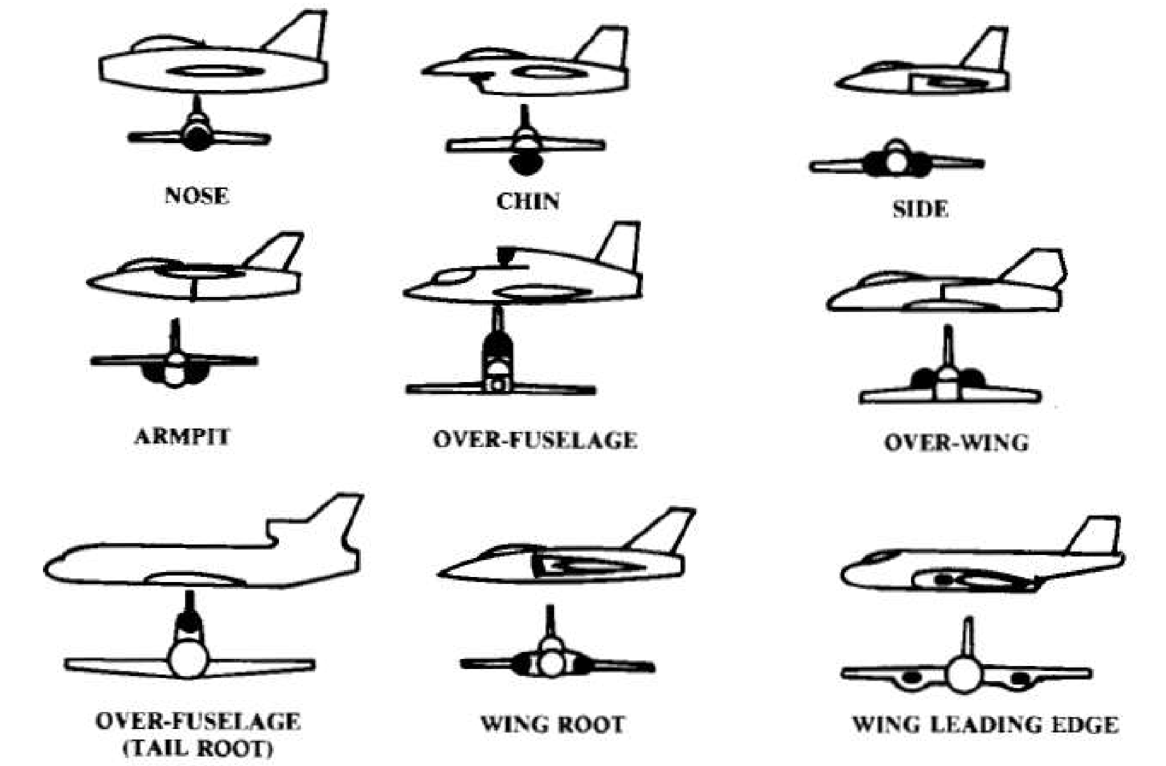


Figure 3.11: inlet locations-buried engines

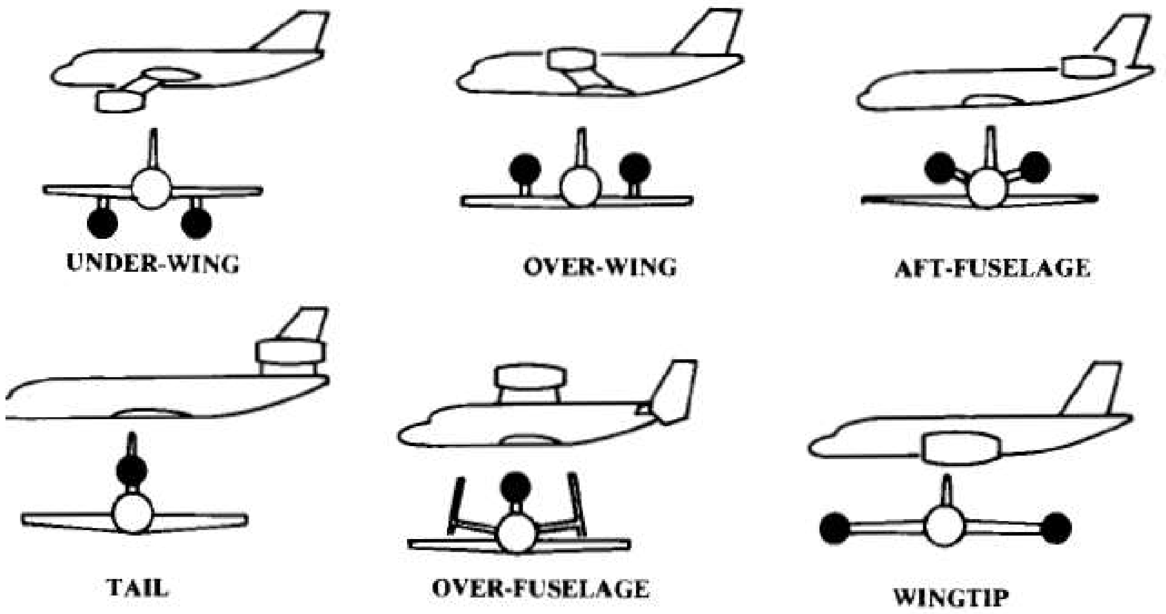


Figure 3.12: inlet location-podded engines

3.2.3. CAPTURE AREA CALCULATION:

Figure below provides a quick method of estimating the required inlet capture area. This statistical method is based upon the design Mach number and the engine mass flow in pounds per second.

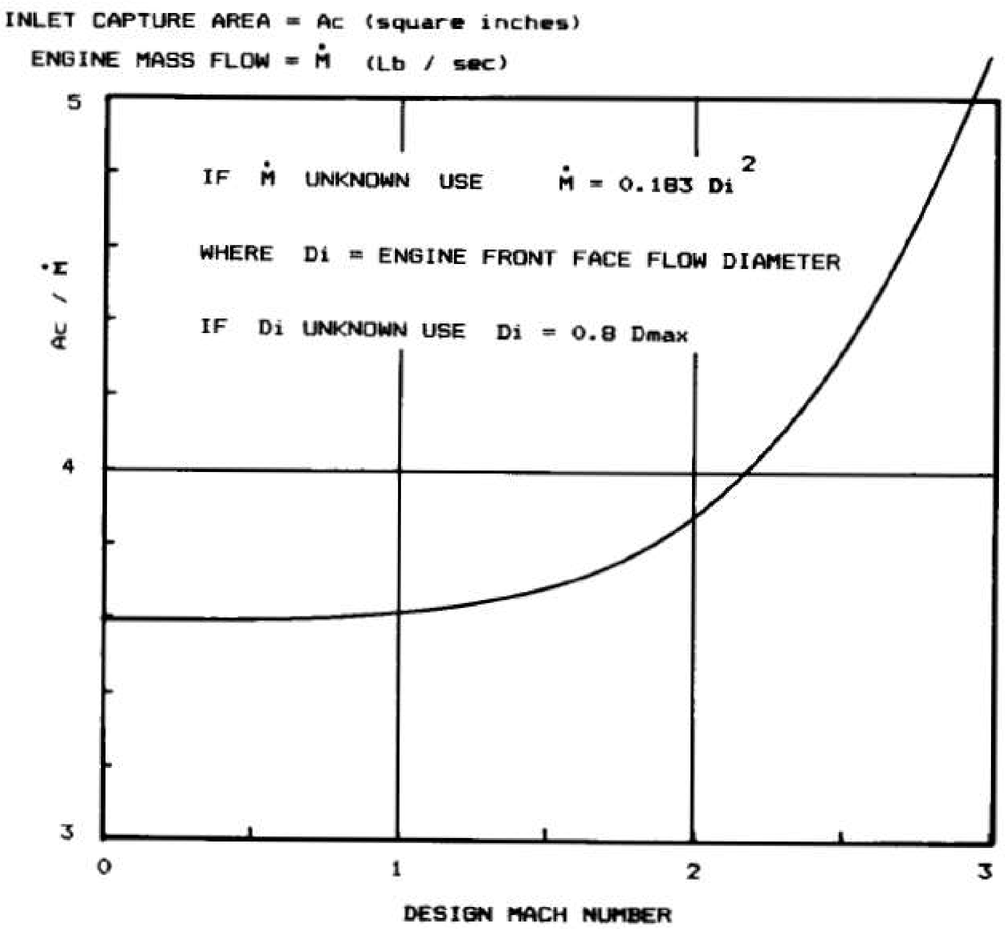


Figure 3.13: Preliminary capture area sizing

Extra air required. The inlet capture area must be sized to provide sufficient air to the engine at all aircraft speeds. For many aircraft the capture area must also provide "secondary air" for cooling and environmental control, and also provide for the air bled off the inlet ramps to prevent boundary-layer buildup. The area at the inlet front face is both the capture area and the throat area. It can be calculated from the following isentropic compressible flow relationship:

$$\frac{A_{\text{throat}}}{A_{\text{engine}}} = \frac{(A/A^*)_{\text{throat}}}{(A/A^*)_{\text{engine}}}$$

$$\frac{A}{A^*} = \frac{1}{M}\left(\frac{1 + 0.2M^2}{1.2}\right)^3$$

Where A^* is the area of the same flow at sonic speed.

For a typical inlet designed to a cruise speed of Mach 0.8, the inlet must slow the air from about Mach 0.6 down to Mach 0.4. The air is slowed from Mach 0.8-0.6 outside the inlet. In sizing a supersonic inlet, a variety of flight conditions must be considered to find the largest required capture area. Typically this will be at the aircraft maximum Mach number, but may also occur during takeoff or subsonic cruise. If the maximum required capture-area occurs during take

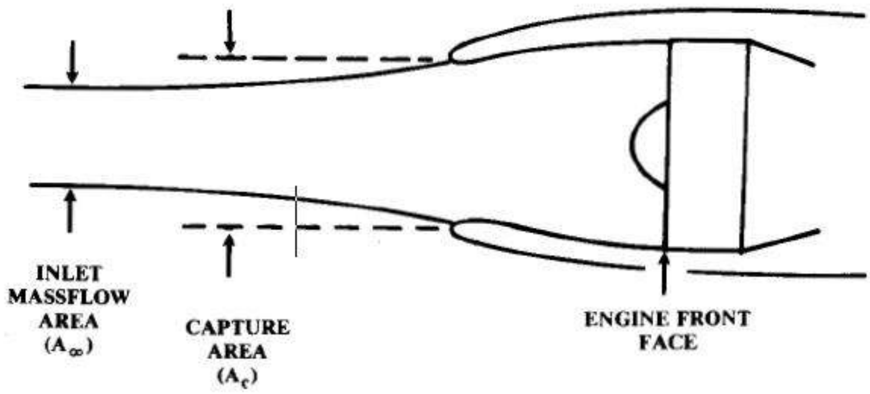


Figure: 3.14: subsonic inlet capture area

If the total mass flow required by the engine, bleed, and secondary flow is known, then Eq. can be solved for the required cross-sectional area upstream of the inlet (at "infinity") using the free stream values for density and velocity. The required engine mass flow is provided by the engine manufacturer, and is a function of the Mach number, altitude, and throttle setting (percent power). Usually the manufacturer's data should be increased by 3% to allow for manufacturing tolerances.

The secondary airflow requirements are accurately determined by an evaluation of the aircraft's subsystems such as environmental control. For initial capture-area estimation Inlet boundary layer bleed should also be determined analytically, but can be approximated using Fig. below. This estimates the required extra capture area for bleed as a percent of the capture area required for the engine and secondary

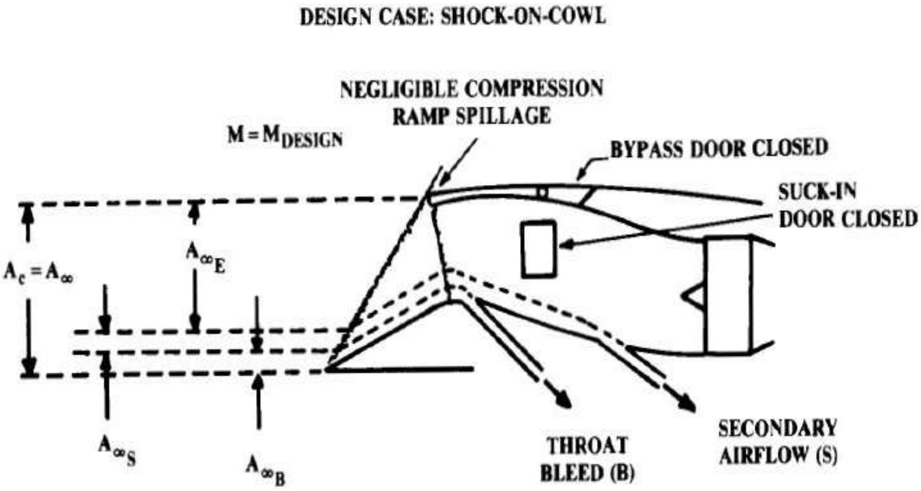


Figure 3.15: supersonic inlet capture area-on design

Conditions in which the inlet bypass doors are closed (no bypass mass flow).

$$\frac{A_\infty}{A_c} = \frac{\dot{m}_E + \dot{m}_S + \dot{m}_{\mathrm{BL}} + \dot{m}_{\mathrm{bypass}}}{g\rho_\infty V_\infty A_c}$$

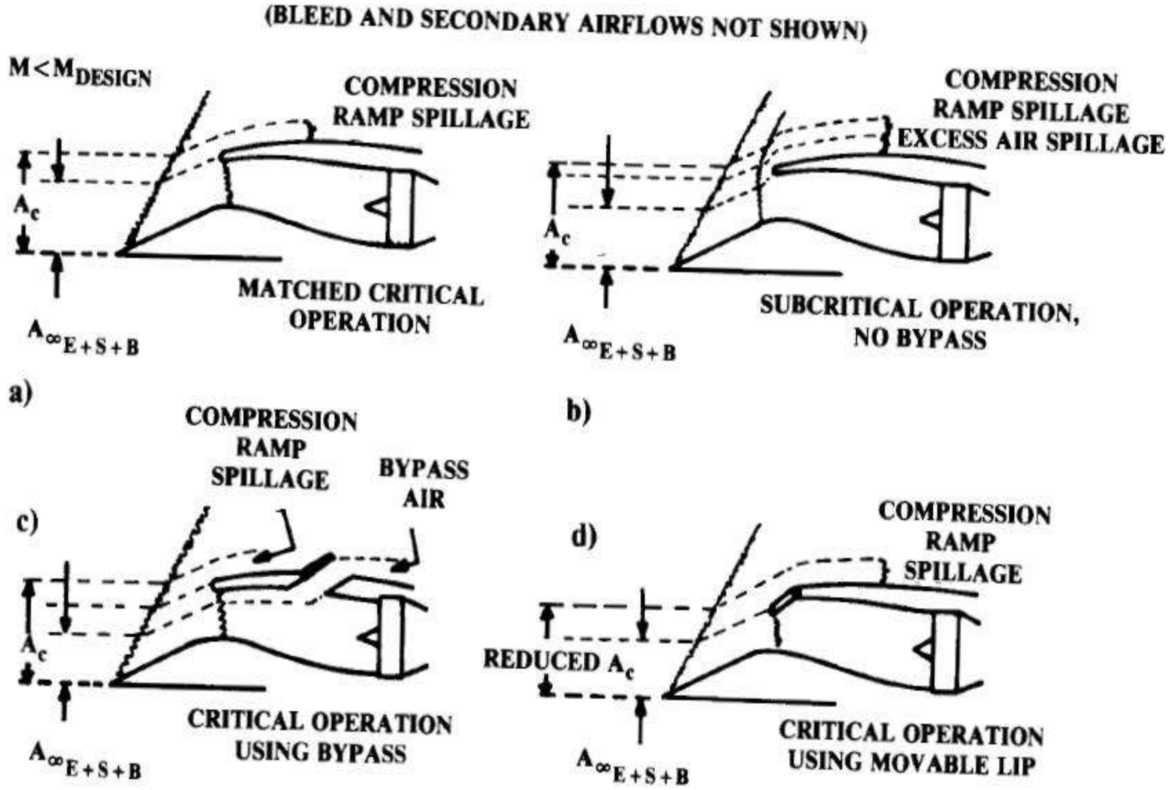


Figure 3.17 off-design inlet operations

Capture-area ratio is calculated by determining the required mass flow and dividing by the mass flow through the capture area far upstream. Note that capture-area ratio is generally critical for

3.4. BOUNDARY LAYER DIVERTERS

Any object moving through air will build up a boundary layer on its surface. In the last section, boundary-layer bleed was included in the capture area calculation. This boundary-layer bleed was used to remove the low-energy boundary layer air from the compression ramps, to prevent shock-induced separation.

The aircraft's fore body builds up its own boundary layer. If this low-energy, turbulent air is allowed to enter the engine, it can reduce engine performance sub sonically and prevent proper inlet operation supersonically. Unless the aircraft's inlets are very near the nose (within two to four inlet diameters), Some form of boundary-layer removal should be used just in front of the inlet.

The boundary-layer air is caught between the splitter plate and the fuselage, and pushed out of the resulting channel by the diverter ramps. The diverter ramps should have an angle of no more than about 30 deg. A very good rule of thumb for the required thickness of a boundary-layer diverter is that it should be between 1 and 3 % of the fuselage length in front of the inlet, with the larger number for fighters which go to high angle of attack . The drag of a boundary-layer diverter depends upon its frontal area. During conceptual layout, the fuselage and inlet should be designed to minimize this area, shown shaded in Fig3.18

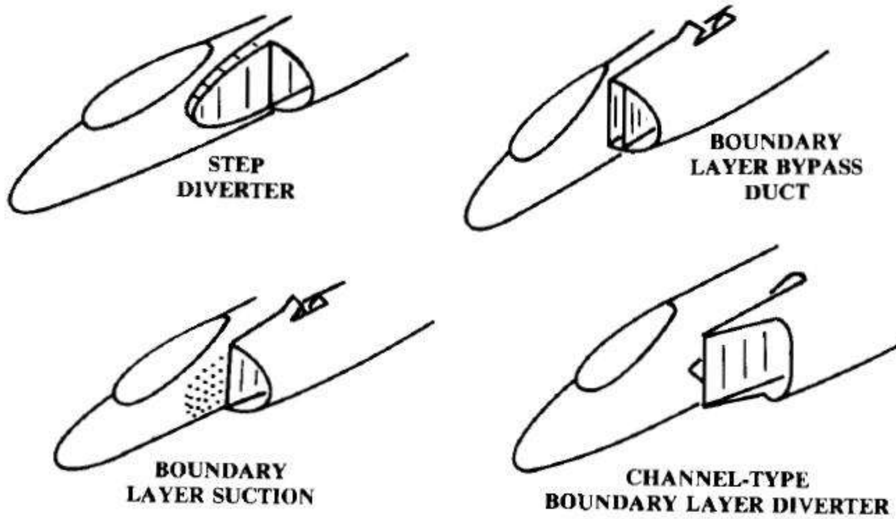


Figure 3.18: Boundary layer removal

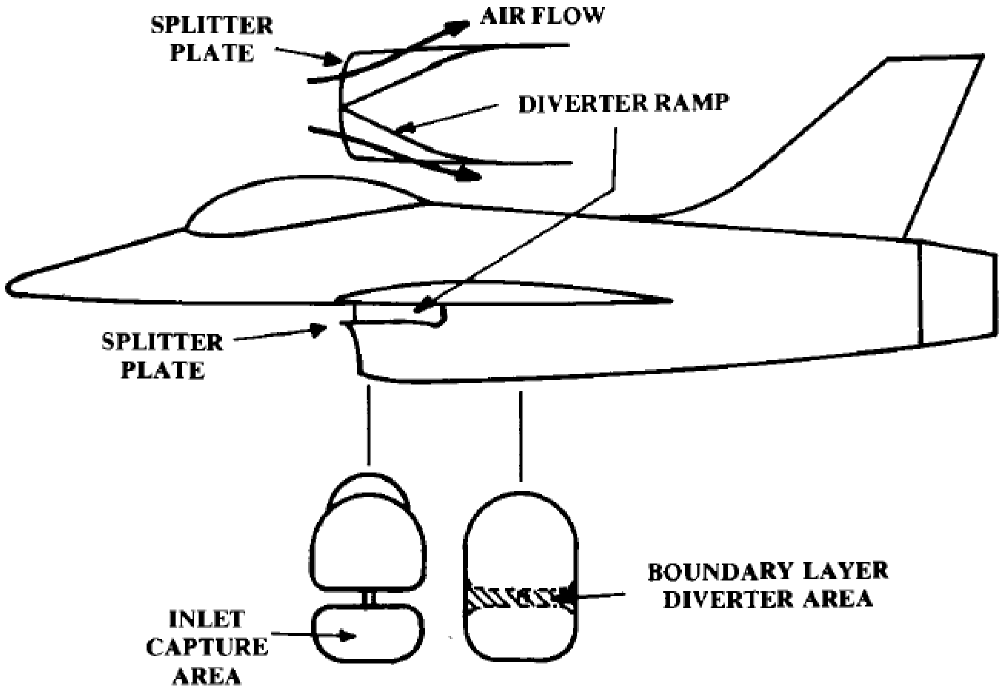


Figure 3.19: Boundary layer diverter

3.5. NOZZLE INTEGRATION

The fundamental problem in jet engine nozzle design is the mismatch in desired exit areas at different speeds, altitudes, and thrust settings. The engine can be viewed as a producer of high-pressure subsonic gases. The nozzle accelerates those gases to the desired exit speed, which is controlled by the exit area. The nozzle must converge to accelerate the exhaust gases to a high subsonic exit speed. If the desired exit speed is supersonic, a converging-diverging nozzle is required. Another means to vary the exit area of a convergent nozzle is the translating plug. This was used on the engine for the Me-262, the first jet to be employed in combat in substantial numbers. The plug slides aft to decrease exit area.

The ejector nozzle takes engine bypass air that has been used to cool the afterburner and ejects it into the exhaust air, thus cooling the nozzle as well. The variable-geometry convergent-divergent ejector nozzle is most commonly applied to supersonic jet aircraft. It allows varying the throat

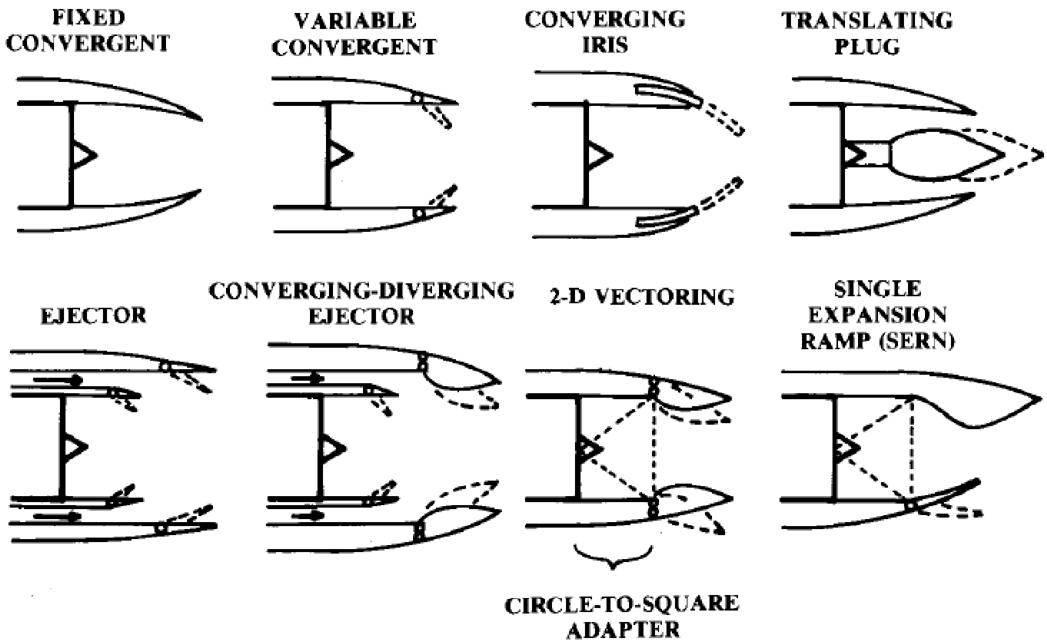


Figure 3.20: Types of nozzles

3.6. ENGINE COOLING PROVISIONS

3.6.1. Engine Size Estimation

The required horsepower has previously been calculated. The dimensions of an engine producing this power must now be determined. In propeller aircraft design it is far more common to size the aircraft to a known, fixed-size engine as opposed to the rubber-engine aircraft sizing more common in jet-aircraft design.

In fact, most propeller-aircraft designs are based around some production engine, probably because very few new piston or turboprop engines have been designed and certified. Most piston engines in production were designed three decades ago. The high cost of developing and certifying a new engine, and the relatively small market, prevent new engines from appearing.

However, rubber-engine trade studies can point to the optimal existing engine. Also, the use of rubber-engine trade studies for comparison of alternate technologies (such as composite vs aluminum structure) can prevent a bias in the results due to the use of a fixed engine size.

3.6.2. Fuel System

An aircraft fuel system includes the fuel tanks, fuel lines, fuel pumps, vents, and fuel-management controls. Usually the tanks themselves are the only components that impact the overall aircraft layout, although the winglets on the round-the-world Rutan Voyager were added solely to raise the fuel vents above the wing tanks when the wing tips bent down to the runway on takeoff.

There are three types of fuel tank: discrete, bladder, and integral. Discrete tanks are fuel containers which are separately fabricated and mounted in the aircraft by bolts or straps. Discrete tanks are normally used only formal general aviation and homebuilt aircraft. Discrete tanks are usually shaped like the front of an airfoil and placed at the inboard wing leading-edge, or are placed in the fuselage directly behind the engine and above the pilot's feet. Bladder tanks are made by stuffing a shaped rubber bag into a cavity in the structure. The rubber bag is thick, causing the loss of about 10% of the available fuel volume. However, bladders are widely used because they can be made "self-sealing." If a bullet passes through a self-sealing tank, the rubber will fill in the hole preventing a large fuel loss and fire hazard. This offers a major improvement in aircraft survivability as approximately a third of combat losses are attributed to hits in the fuel tanks.

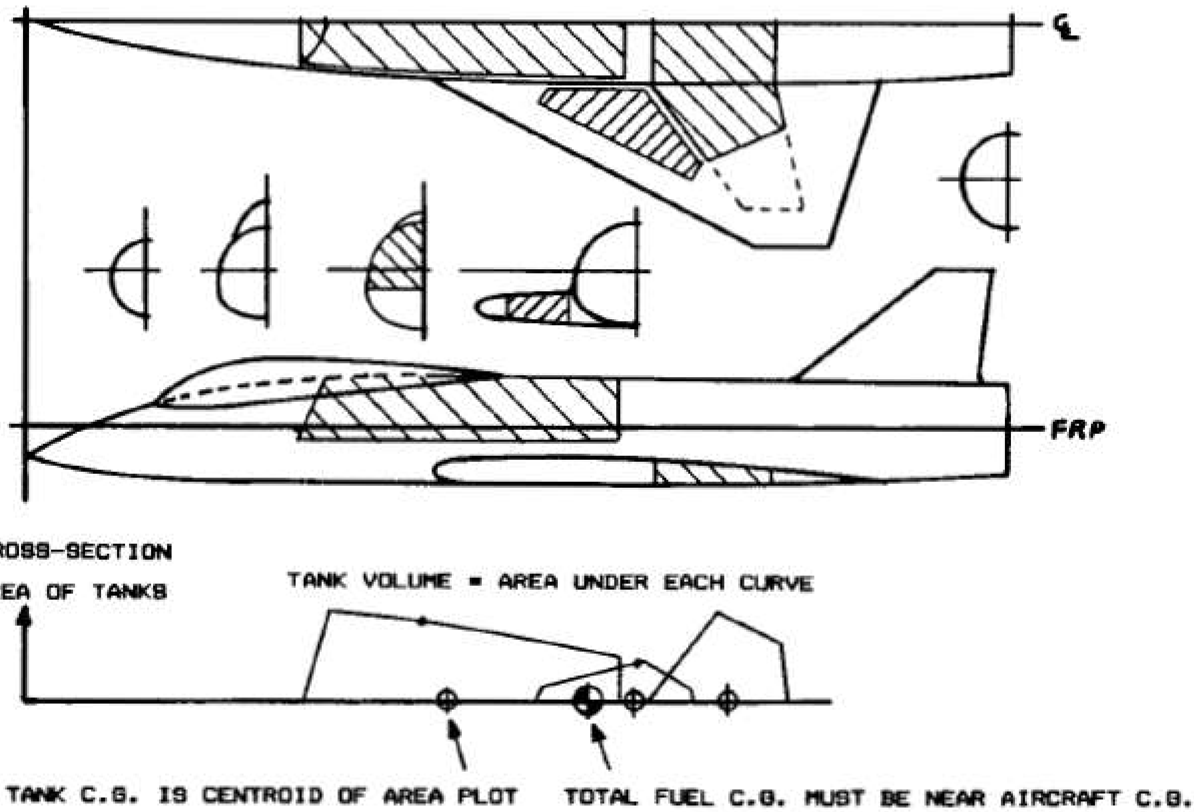


Figure 3.23: fuel tank volume plotting

Table : Fuel densities (lb/gal)

	Average actual density		Mil-spec density
	0°F	100°F	
Aviation gasoline	6.1	5.7	6.0
JP-4	6.7	6.4	6.5
JP-5	7.2	6.8	6.8
JP-8	–	–	6.7

3.6.3. Landing Gear Arrangements

The common options for landing-gear arrangement are shown. The single main gear is used for many sailplanes because of its simplicity. The wheel can be forward of the center of gravity (e.g.), as shown here, or can be aft of the e.g. with a skid under the cockpit. . "Bicycle" gear has two main wheels, fore and aft of the e.g.: with stall "outrigger" wheels. The bicycle landing gear has the aft wheel that the aircraft must take off and land in a flat attitude, which limits this type of gear to aircraft with high lift at low angles of attack (i.e., high-aspect-ratio Wings with large camber and/or flaps). Bicycle gear has been used mainly on aircraft with narrow fuselage and wide wing span such as the B-47.

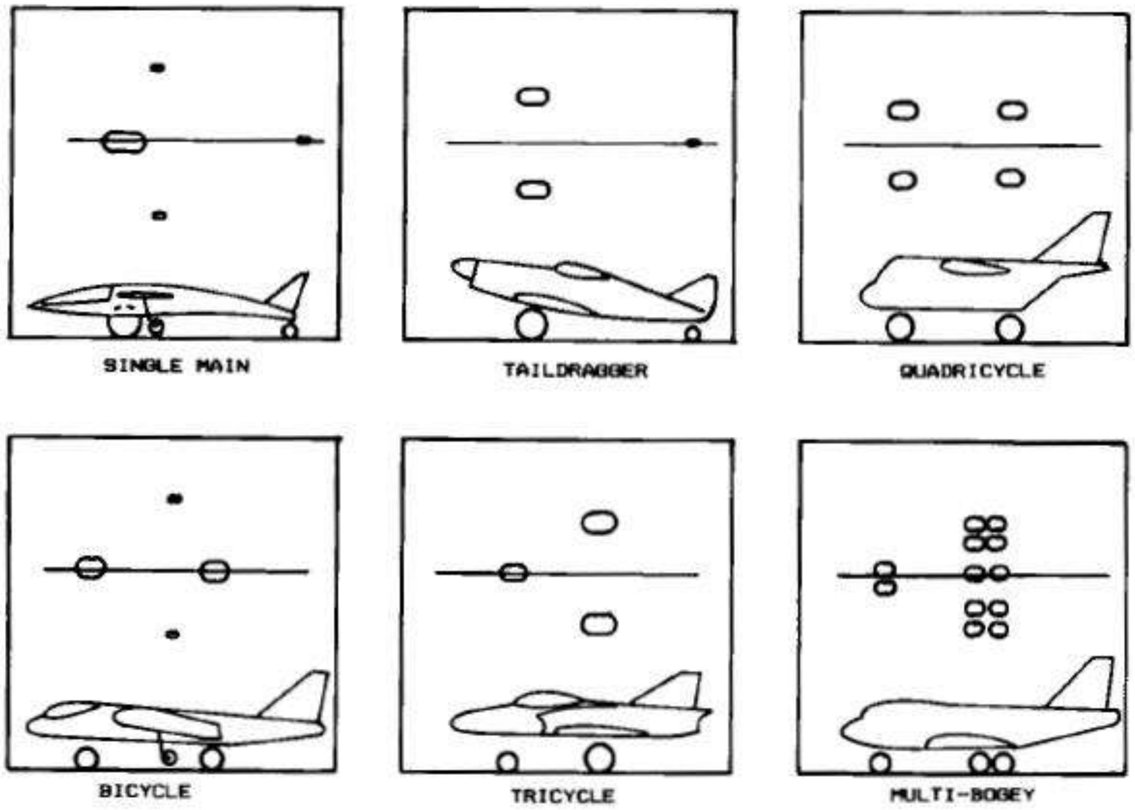


Figure 3.24: landing gear arrangements

The requirements for tail dragger gear are shown in Fig. below. The tail down angle should be about 10-15 deg with the gear in the static position (i.e., tires and shock absorbers compressed the amount seen when the aircraft is stationary on the ground at takeoff gross weight). The e.g. (most forward and most aft) should fall between 16-25 deg back from vertical measured from the main wheel location. If the e.g. is too far forward the aircraft will tend to nose over, and if it is too far back it will tend to ground loop

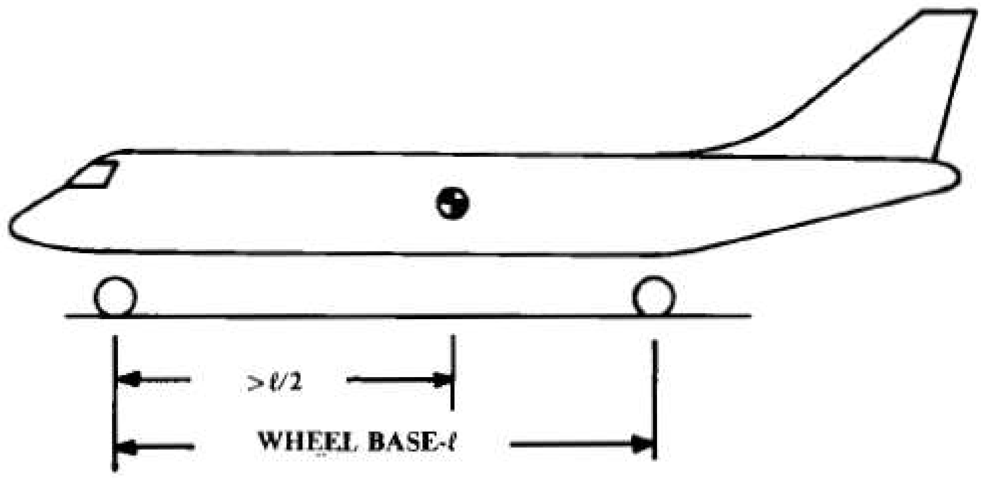


Figure 3.25: Bicycle landing gear

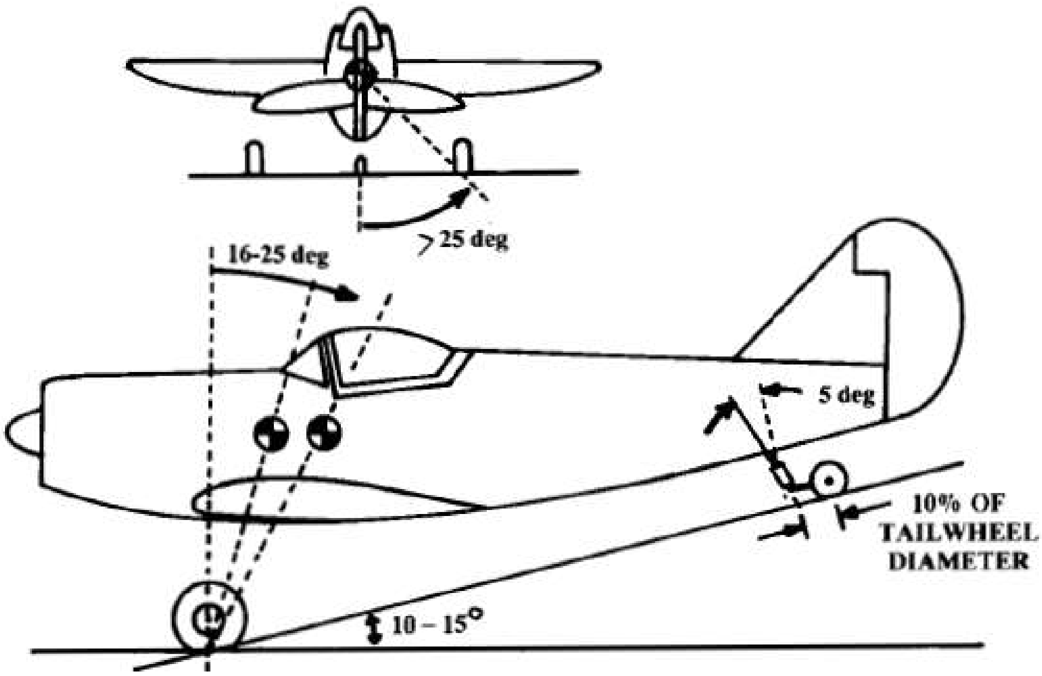


Figure 3.26: Tail dragger landing gear

However, this also makes it difficult to lift the nose for a runway takeoff. If the nose wheel is carrying over 200Jo of the aircraft's weight, the manager is probably too far aft relative to the c.g .On the other hand, if the nose wheel is carrying less than 5% of the aircraft's weight, there will not be enough nose-wheel traction to steer the aircraft. The optimum range for the percent of the aircraft's weight

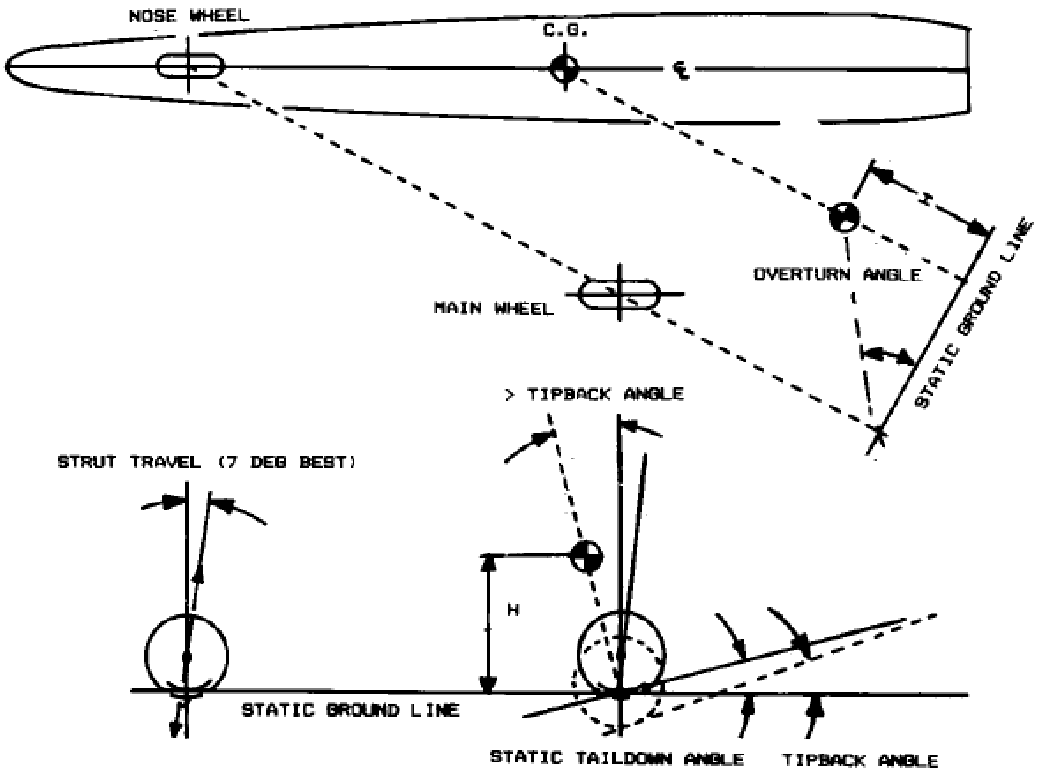


Figure 3.27: Tricycle landing gear geometr

3.6.4. Guidelines for Lay Out

The layout of tricycle landing gear as shown in Fig. 3.27 is even more complex. The length of the landing gear must be set so that the tail doesn't hit the ground on landing. This is measured from the wheel in the static position assuming an aircraft angle of attack for landing which gives 900Jo of the maximum lift. This ranges from about 10-15 deg for most types of aircraft.

The "tip back angle" is the maximum aircraft nose-up attitude with the tail touching the ground and the strut fully extended.

To prevent the aircraft from tipping back on its tail, the angle off the vertical from the main wheel position to the e.g. should be greater than the tip back angle or 15 deg, whichever is larger. For carrier-based aircraft this angle frequently exceeds 25 deg, implying that the e.g. for carrier-based aircraft is well forward of the main wheels.

For carrier-based aircraft this angle frequently exceeds 25 deg, implying that the e.g. for carrier-based aircraft is well forward of the main wheels.

This insures that the rolling of the deck will not cause an aircraft to tip back on its tail. However, this also makes it difficult to lift the nose for a runway takeoff. If the nose wheel is carrying over 200Jo of the aircraft's weight, the main gear is probably too far aft relative.

On the other hand, if the nose wheel is carrying less than 50Jo of the aircraft’s weight, there will not be enough nose-wheel traction to steer the Aircraft.

3.6.5. Shock Absorbers–Types

The landing gear must absorb the shock of a bad landing and smooth out the ride when taxiing. The more common forms of shock absorber are shown in Figure.3.27.

The tires themselves provide some shock-absorbing ability by deflecting when a bump is encountered. Sailplanes and a few homebuilt aircraft have been built with rigid axles, relying solely upon the tires for shock absorbing.

Many World War I fighters used a rigid axle mounted with some vertical movement. The axle was attached to the aircraft with strong rubber chords

The oleo pneumatic *shock* strut, or "oleo," is the most common type of shock-absorbing gear m use today (Fig. 3.28). The oleo concept was patented in 1915 as a recoil device for large cannons

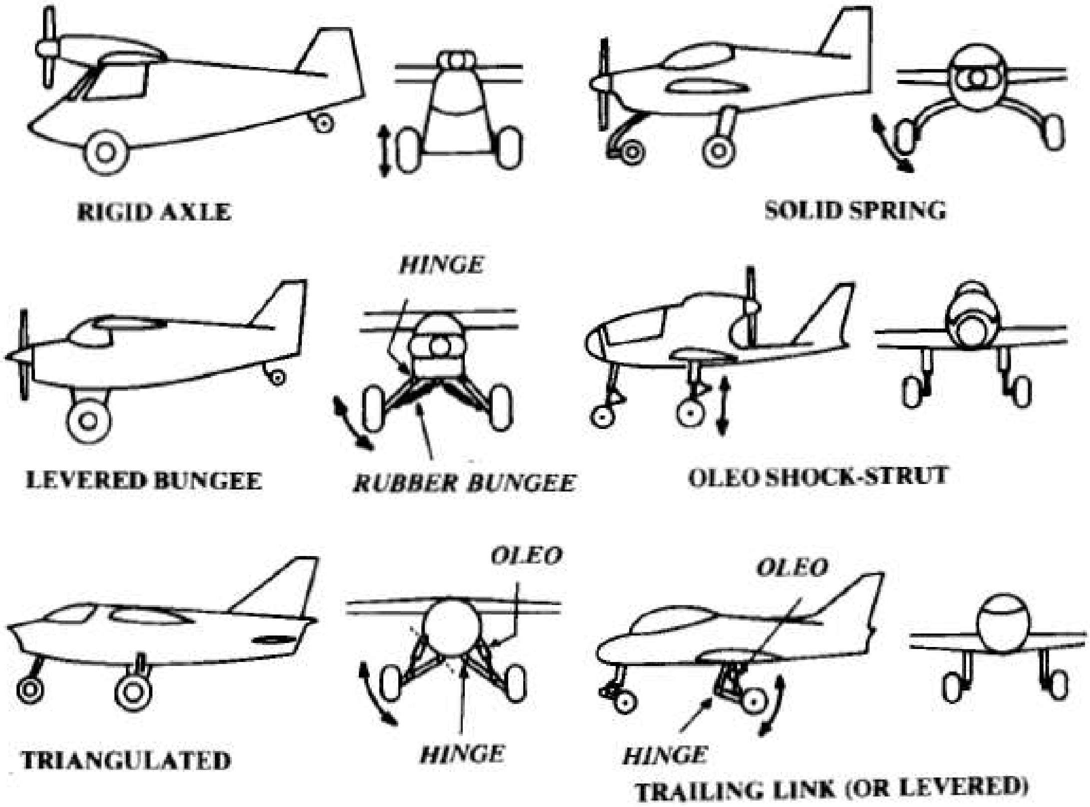


Figure 3.28: Gear/shock arrangements

The oleo combines a spring effect using compressed air with a damping effect using a piston which forces 011 through a small hole (orifice). For maximum efficiency, many oleos have a mechanism for varying the size of the orifice as the oleo compresses the triangulated gear is similar to the levered bungee gear.

When the triangulated gear is deflected, an oleo pneumatic shock absorber is compressed. This provides a leveraged effect in which the oleo cans b~ shorter than the required wheel travel. This is especially useful for earner-based aircraft such as the A-7 that require large amounts of wheel travel to absorb the carrier-landing impact loads. On a triangulated gear, the oleo can be replaced without removing the wheel assembly.

The wheel lateral and braking loads are carried by the solid gear legs, which reduces the oleo weight. However, the complete triangulated gear is usually a little heavier than the oleo shock-strut gear. Also, there is a tire-scrubbing effect that shortens tire life.

The triangulated gear is sometimes seen on smaller aircraft using rubber blocks or springs in compression instead of an oleo pneumatic shock absorber. The rubber blocks or springs can be inside the fuselage which Streamlines the exposed part of the gear but requires the gear leg to support the aircraft's weight in a cantilevered fashion. This increases the gear weight.

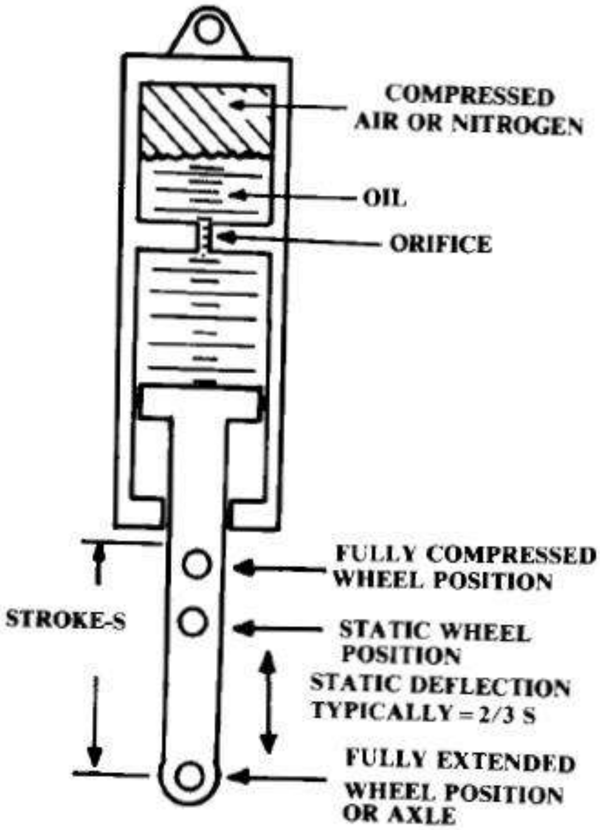


Figure 3.29: Oleo shock absorber (most simple type)

3.6.6. Stroke Determination

The required deflection of the shock-absorbing system (the "stroke") depends upon the vertical velocity at touchdown, the shock-absorbing material and the amount of wing lift still available after touchdown. As rough rule-of-thumb, the stroke in inches approximately equals the vertical velocity at touchdown in (ft/s). This kinetic energy is absorbed by the work of deflecting the shock absorber and tire.

$$KE_{\text{vertical}} = \left(\frac{1}{2}\right)\left(\frac{W_{\text{landing}}}{g}\right) V_{\text{vertical}}^2$$

Where ***W***= total aircraft weight

g= 32.2 ft/s2

If the shock absorber were perfectly efficient, the energy absorbed by deflection would be simply the load times the deflection. Actual efficiencies of shock absorbers range from 0.5-0.9. The actual energy absorbed by deflection is defined in Eq

$$KE_{\text{absorbed}} = \eta L S$$

where

η= shock-absorbing efficiency

L = average total load during deflection (not lift!)

S = stroke

$$\left(\frac{1}{2}\right)\left(\frac{W_{\text{landing}}}{g}\right)V_{\text{vertical}}^2 = (\eta LS)_{\substack{\text{shock}\\\text{absorber}}} + (\eta_T LS_T)_{\text{tire}}$$

$$N_{\text{gear}} = L/W_{\text{landing}}$$

$$S = \frac{V_{\text{vertical}}^2}{2g\,\eta N_{\text{gear}}} - \frac{\eta_T}{\eta}S_T$$

3.7. OLEO SIZING

The actual dimensions of an oleo shock absorber or shock strut can now be estimated. The total oleo stroke is known. For most types of aircraft the static position is approximately 660Jo of the distance from the fully extended to the fully compressed position (see Fig. 11.9). For large transport aircraft the static position is about 840Jo of stroke above the fully extended position'. For a general-aviation aircraft the static position is typically about 600Jo of stroke above the extended position. The total length of the oleo including the stroke distance and the fixed portion of the oleo will be approximately 2.5 times the stroke. For an aircraft with the desired gear attachment point close to the ground, thismm1mum oleo length may

$$D_{\text{oleo}} = 1.3\sqrt{\frac{4L_{\text{oleo}}}{P\pi}} \cong 0.04\sqrt{L_{\text{oleo}}}$$

Require going to a levered gear.Where L_{oleo}= load on the oleo in pounds

3.7.1. SOLID-SPRING GEAR SIZING

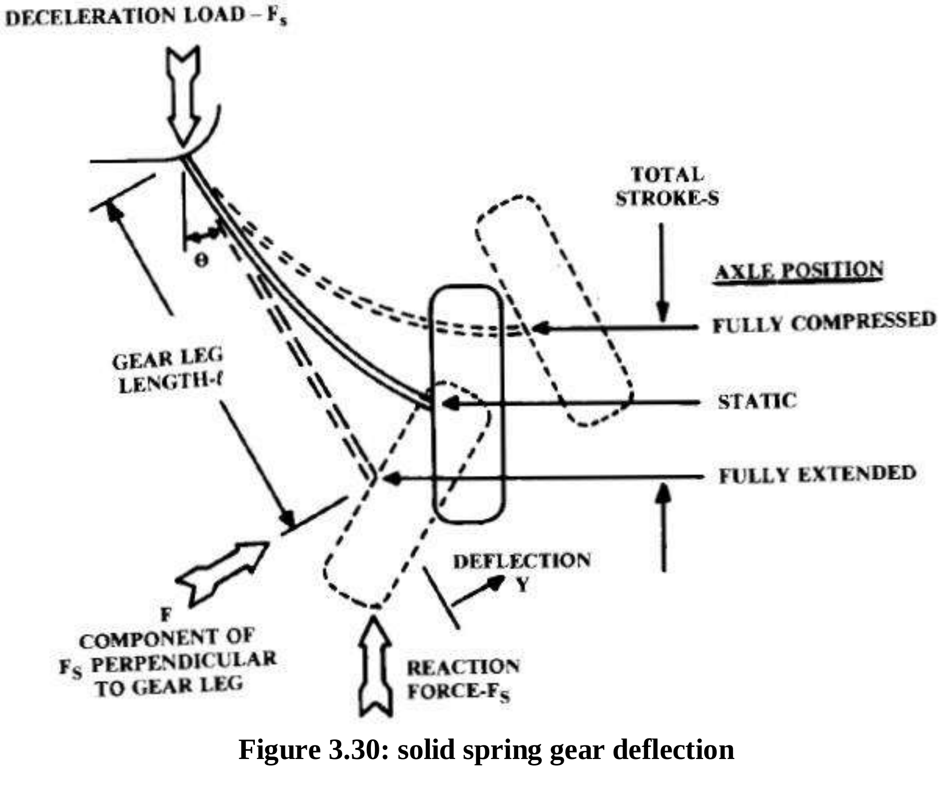


Figure 3.30: solid spring gear deflection

3.7.2. STROKE DETERMINATION, GEAR LOAD FACTORS

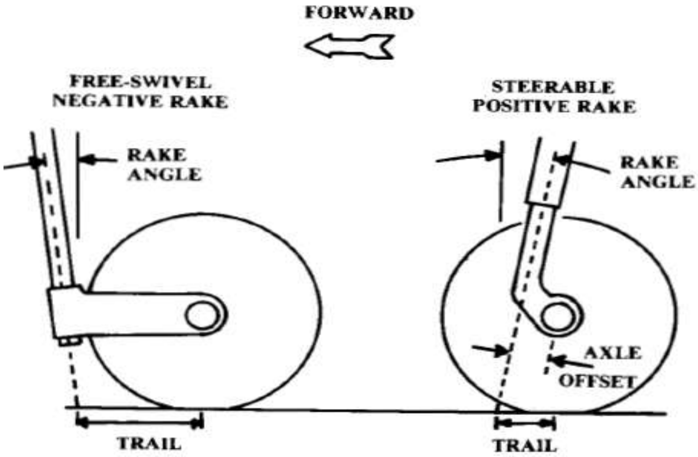


Figure 3.31: casting wheel geometry

3.8. GEAR RETRACTION GEOMETRY

At this point, the required sizes for the wheels, tires, and shock absorbers are known, along with the required down locations of the wheels. The one remaining task is to find a "home for the gear" in the retracted position. A poor location for the retracted gear can ruin an otherwise good design concept! A bad choice for the retracted position can chop up the aircraft structure (increasing weight), reduce the internal fuel volume, or create additional aerodynamic drag.

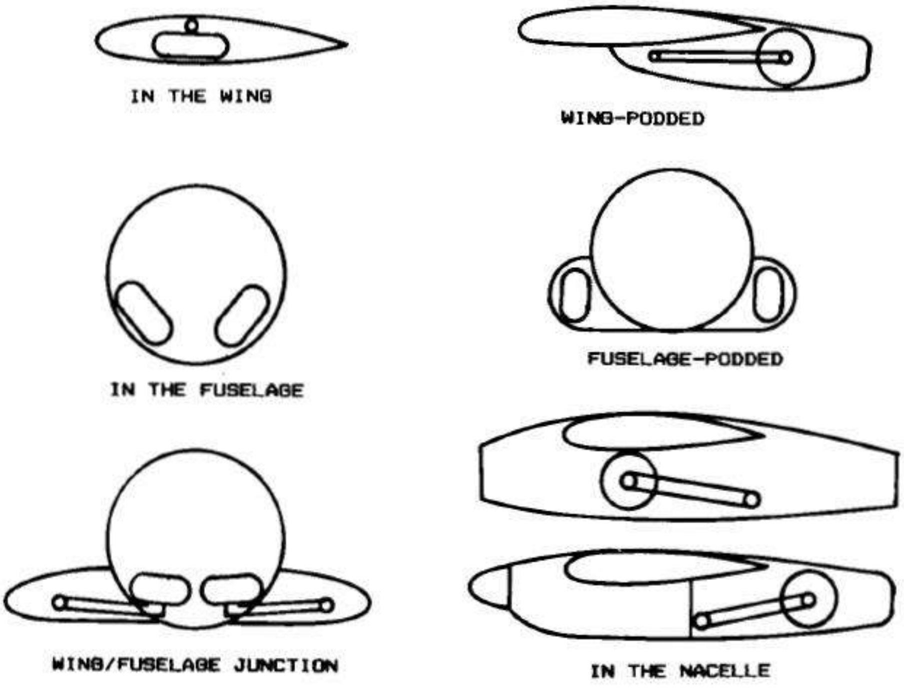


Figure 3.32: A home for the gear

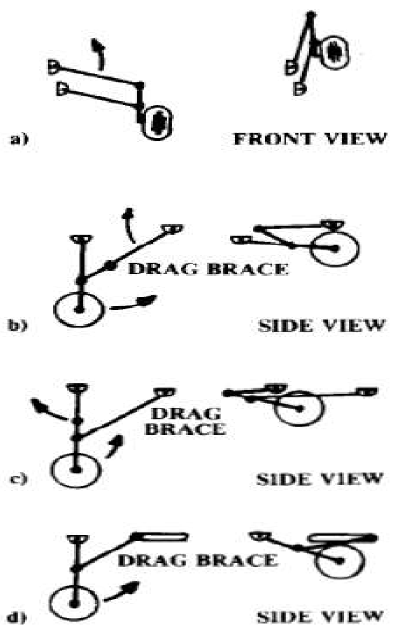


Figure 3.33: Landing Gear Retraction

3.9. AIRCRAFT SUBSYSTEMS

Aircraft subsystems include the hydraulic, electrical, pneumatic, and auxiliary emergency power systems. Also, the avionics can be considered a subsystem (although to the avionics engineers, the airframe is merely the "mobility subsystem" of their avionics package)

In general, the subsystems do not have a major impact on the initial design layout. However, later in the design cycle the configuration designer will have to accommodate the needs of the various subsystems, so a brief introduction is provided below. No attempt is made to provide examples or rules of thumb because the subsystems hardware varies widely between different classes of aircraft. Reference 11 provides additional information on subsystems.

3.9.1. Hydraulics

Hydraulic fluid, a light oil-like liquid, is pumped up to some specified pressure and stored in an "accumulator" (simply a holding tank).

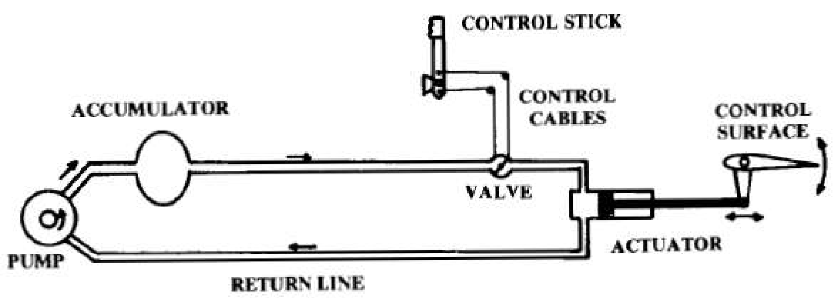


Figure 3.34: simplified hydraulic system

When the valve is opened, the hydraulic fluid flows into the actuator where it presses against the piston, causing it to move and in turn moving the control surface. To move the control surface the other direction, an additional valve (not shown) admits hydraulic fluid to the back side of the piston. The hydraulic fluid returns to the pump by a return line.

To obtain rapid response, the valve must be very close to the actuator. The valve therefore cannot be in or near the cockpit, and instead is usually attached to the actuator. In most current designs the pilot's control inputs are mechanically carried to the actuator by steel cables strung from the control wheel or rudder pedals to the valves on the actuators. In many new aircraft the pilot's inputs are carried electronically to electromechanical valves ("fly-by-wire").

Hydraulics are used for aircraft flight control as well as actuation of the flaps, landing gear, spoilers, speed brakes, and weapon bays. Flight-control hydraulic systems must also include some means of providing the proper control "feel" to the pilot. For example, the controls should become stiffer at higher speeds, and should become heavier in a tight, high-g turn. Such "feel" is provided by a combination of springs, bob weights, and air Bellows. In most cases the hydraulic system will impact the aircraft conceptual design only in the provision of space for the hydraulic pumps, which _are usually attached to the engines. These should be copied from a similar aircraft if better information is not available.

3.9.2. Electrical System

An aircraft electrical system provides electrical power to the avionics, hydraulics, environmental-control, lighting, and other subsystems the electrical system consists of batteries, generators, transformer-rectifiers ("TR's"), electrical controls, circuit breakers,. and cables.
Aircraft generators usually produce alternating current (AC) and are located on or near the engines. TR's are used to convert the alternating current to direct current (DC). Aircraft batteries can be large and heavy if they are used as the only power source for starting.

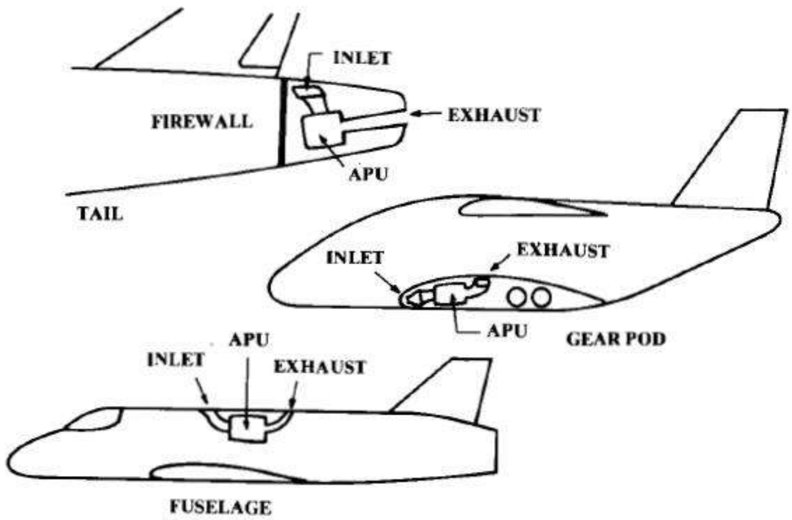


Figure 3.35: APU Installation

3.9.3. Pneumatic System

The pneumatic system provides compressed air for pressurization, environmental control, anti-icing, and in some cases engine starting. Typically the pneumatic system uses pressurized air bled from the engine compressor. This compressed air is cooled through a heat exchanger using outside air. This cooling air is taken from a flush inlet inside the inlet duct (i.e., inlet secondary airflow) or from a separate inlet usually located on the fuselage or at the front of the inlet boundary-layer diverter. The cooled compressor air is then used for cockpit pressurization and avionics cooling. For anti-icing, the compressor bleed air goes uncooled through ducts to the wing leading edge, inlet cowls, and windshield.

Compressed air is sometimes used for starting other engines after one engine has been started by battery. Also, some military aircraft use a ground power cart that provides compressed air through a hose to start the engine

3.9.4. Auxiliary/Emergency Power

Large or high-speed aircraft are completely dependent upon the hydraulic system for flight control. If the hydraulic pumps stop producing pressure for any reason, the aircraft will be uncontrollable. If the pumps are driven off the engines, an engine flame-out will cause an immediate loss of control. For this reason, some form of emergency hydraulic power is required. Also, electrical power must be retained until the engines can be restarted. The three major forms of emergency power are the ram-air turbine (RAT), monopropellant emergency power unit (EPU), and jet-fuel EPU. The ram-air turbine is a windmill extended into the slipstream. Alternatively, a small inlet duct can open to admit air into a turbine The monopropellant EPU uses a monopropellant fuel such as hydrazine to drive a turbine.

The available monopropellants are all toxic and caustic, so monopropellant EPU's are undesirable for operational considerations. However, they have the advantage of not requiring any inlet ducts and can be relied upon to provide immediate power regardless of aircraft altitude, velocity, or attitude. Monopropellant EPU's must be located such that a small fuel leak will not allow the caustic fuel to puddle in the aircraft structure, possibly dissolving it Jet-fuel EPU's are small jet engines that drive a turbine to produce emergency power. These may also be used to start the main engines ("jet-fuel starter"). While they do not require a separate and dangerous fuel, the jet-fuel EPU's require their own inlet duct.

3.9.5. Avionics

Avionics (a contraction of "aviation electronics") includes radios, flight instruments, navigational aids, flight control computers, radar, and other aircraft sensors such as infrared detectors. For initial layout, it is necessary to provide sufficient volume in the avionics bays. Also, the nose of the aircraft should be designed to hold the radar. On the average, avionics has a density of about 30-45 lb/ft3 • The required avionics weight can be estimated from the aircraft empty weight (W_e),which is known at this point

3.10. SIGNIFICANCE TO CONFIGURATION LAY OUT, THE BASELINE DESIGN LAYOUT AND REPORT OF INITIAL SPECIFICATIONS

There are three separate origins of the drag-producing pressure forces. The first, viscous separation, was the source of considerable difficulty during the early theoretical development of aerodynamics. If the theoretical pressure forces in a perfect fluid are integrated over a streamlined body without flow separation, it is found that the pressures around the body which yield a drag force in the flight direction are exactly matched by the pressures around the body which yield a forward force. Thus, if skin friction is ignored the net drag is zero. This was known to be false, and was called d' Alembert's paradox. The paradox was finally resolved by Prandtl who determined that the boundary layer, which is produced by viscosity, causes the flow to separate somewhere on the back half of the body. This prevents the full attainment of the forward-acting force, leaving a net drag force due to viscous separation. Viscous separation drag, also called "form drag," depends upon the locat10n of the separation point on the body. If the flow separates nearer to the front of the body the drag is much higher than if it separates more towards the rear. The location of the separation point depends largely upon the curvature. Also, the separation point is affected by the amount of energy m the flow. Turbulent air has more energy than laminar air, so a turbulent boundary layer actually tends to delay separation. If a body is small and flying at low speed, the Reynolds number will be so low that the flow will remain laminar resulting in separated flow. For this reason a small body may actually have a lower total drag when its skin is rough. This produces turbulent flow, which will remain attached longer than would laminar flow. The dimples on a golf ball are an example of this. For a very long body such as the fuselage of an airliner, the turbulent boundary layer will become so thick that the air near the skin loses most of its energy. This causes separation near the tail of the aircraft resulting in high "boat tail drag." ' . To prevent this, ~mall vanes perpendicular to the skin and angled to the airflow are placed Just upstream of the separation point. These vanes produce vor~1ces off their ends, which mix the boundary layer with higher energy air from outside the boundary layer. This delays separation and reduces boat tail drag. Such "vortex generators" are also used on wing and tail surfaces.

3.10.1. Aerodynamics 261

Viscous separation is largely responsible for the drag of irregular bodies such as landing gear and boundary-layer diverters. It also produces base drag, the pressure drag created by a "cut-off" aft fuselage. The subsonic drag of a streamlined, nonlifting body consists solely of skin friction and viscous separation drag and is frequently called the "profile drag." Profile drag is usually referenced to the maximum cross-sectional area of the body. Note that the terms "profile drag" and "form drag" are frequently intermixed, although strictly speaking the profile drag is the sum of the form drag and the skin-friction drag. Also note that the term "profile drag" is sometimes used for the zero-lift drag of an airfoil. Interference drag is the increase in the drag of the various aircraft components due to the change in the airflow caused by other components. For example, the fuselage generally causes an increase in the wing's drag by encouraging airflow separation at the wing root. Interference drag usually results from an increase in viscous separation, although the skin-friction drag can also be increased if one component causes the airflow over another component to become turbulent or to increase in velocity. "Wave drag" is the drag caused by the formation of shocks at supersonic and high subsonic speeds. At high subsonic speeds, the shocks form first on the upper surface of the wings because the airflow is accelerated as it passes over the wing. Drag forces that are a strong function of lift are known as "induced drag" or "drag-due-to-lift." The induced drag is caused by the circulation about the airfoil that, for a three-dimensional wing, produces vortices in the airflow behind the wing. The energy required to produce these vortices is extracted from the wing as a drag force, and is proportional to the square of the lift.

3.10.2Estimation of lift curve slope:

The slope of the lift curve is essentially linear except near the stall angle, allowing the lift coefficient below stall to be calculated simply as the lift curve Slope times the angle of attack (relative to the zero-lift angle). At the stall, the lift curve has become nonlinear such that the angle for maximum lift is greater than the linear value by an amount shown as .la at CL_{max} . Figure also shows the effect of aspect ratio on lift. For an infinite aspect ratio wing (the 2-D airfoil case) the theoretical low-speed lift-curve slope is two times 71' (per radian).

Actual airfoils have lift-curve slopes between about 90 and l00% of the theoretical value. This percentage of the theoretical value is sometimes called the "airfoil efficiency (17)”.

Reduction of aspect ratio reduces the lift-curve slope, as shown. At very low aspect ratios, the ability of the air to escape around the wing tips tends to prevent stalling even at very high angles of attack. Also note that the lift curve becomes nonlinear for very low aspect ratios.

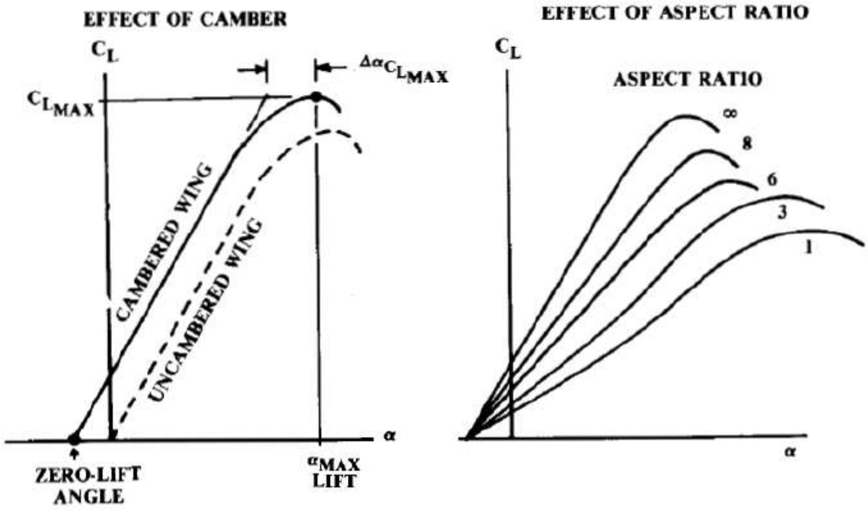


Figure 3.36: wing lift curve

The lift-curve slope is needed during conceptual design for three reasons. First, it is used to properly set the wing incidence angle. This can be especially important for a transport aircraft, in which the floor must be level during cruise. Also, the wing incidence angle influences the required fuselage angle of attack during takeoff and landing, which affects the aft-fuselage upsweep and/or landing-gear length.

3.12. Subsonic Lift-Curve Slope

It is a semi-empirical formula from Ref. 36 for the complete wing lift curve slope (per radian). This is accurate up to the drag-divergent Mach number, and reasonably accurate almost to Mach one for a swept wing.

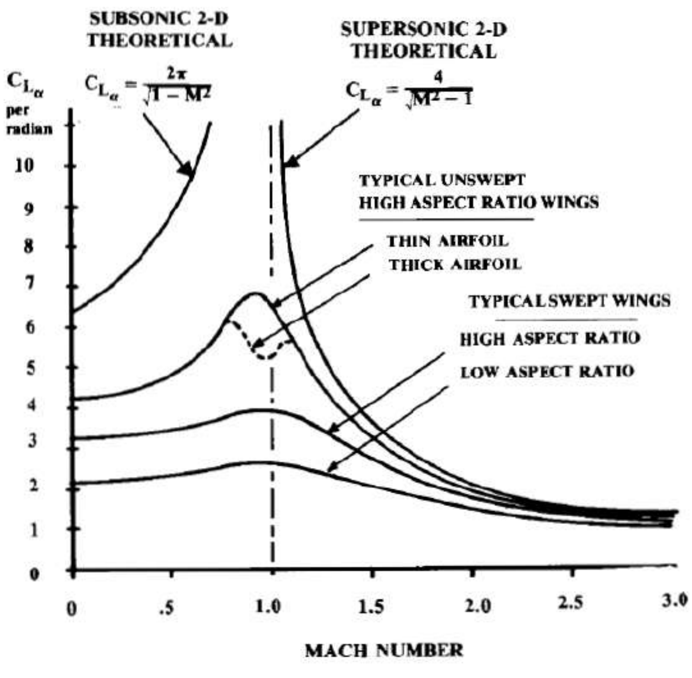


Figure 3.37: Lift curve slope vs Mach number

$$C_{L_\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2\beta^2}{\eta^2}\left(1 + \frac{\tan^2\Lambda_{\max t}}{\beta^2}\right)}}\left(\frac{S_{\text{exposed}}}{S_{\text{ref}}}\right)(F)$$

$$\beta^2 = 1 - M^2$$

$$\eta = \frac{C_{\ell_\alpha}}{2\pi/\beta}$$

The wing aspect ratio *"A "* is the geometric aspect ratio of the complete reference platform. The effective aspect ratio will be increased by wing endplates or winglets.
End plate: $A_{errective} = A\ (1 + 1.9\ h/b)$

3.13. MAXIMUM LIFT COEFFICIENT:

The maximum lift coefficient of the wing will usually determine the wing area. This in turn will have a great influence upon the cruise drag. This strongly affects the aircraft takeoff weight to perform the design mission. Thus, the maximum lift coefficient is critical in determining the aircraft weight; yet the estimation of maximum lift is probably the least reliable of all of the calculations used in aircraft conceptual design. Even refined wind tunnel tests cannot predict maximum lift with great accuracy. Frequently an aircraft must be modified during flight test to achieve the estimated maximum lift.

For high-aspect-ratio wings with moderate sweep and a large airfoil leading
edge radius, the maximum lift depends mostly upon the airfoil characteristics. The maximum lift coefficient of the "clean" wing (i.e., without the use of flaps and other high-lift devices) will usually be about 900Jo of the airfoil's maximum lift as determined from the 2-D airfoil data at a similar Reynolds number.

Sweeping the wing reduces the maximum lift, which can be found by multiplying the unswept maximum lift value by the cosine of the quarter chord sweep

$$CL_{\max} = 0.9C1_{\max}\cos A_{0.25c}$$

If a wing has a low aspect ratio or has substantial sweep and a relatively sharp leading edge, the maximum lift will be increased due to the formation of leading-edge vortices. This vortex formation is strongly affected by the shape of the upper surface of the leading edge.

Leading-edge shape could be defined by the airfoil nose radius. However, the nose radius alone doesn't take into account the effect of airfoil camber on the shape of the upper surface of the airfoil leading edge.

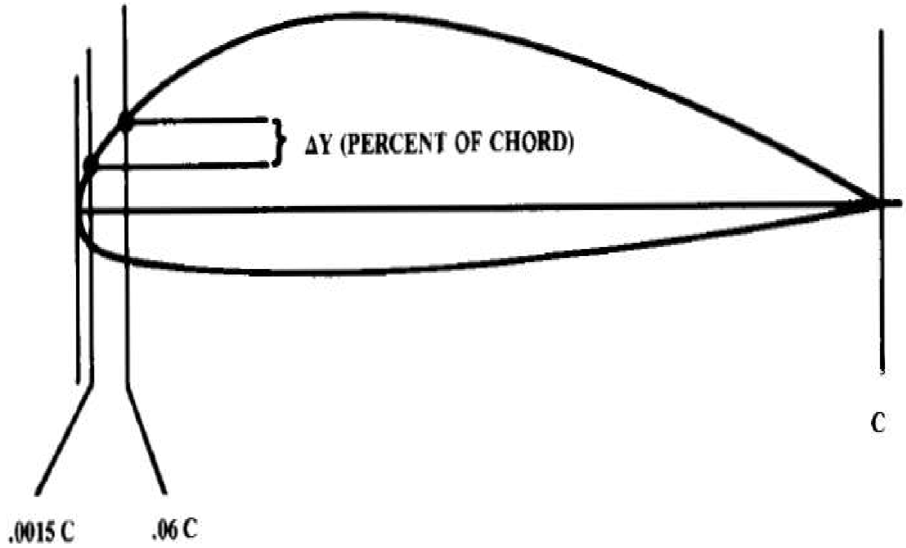


Figure 3.38 Airfoil leading edge sharpness parameter

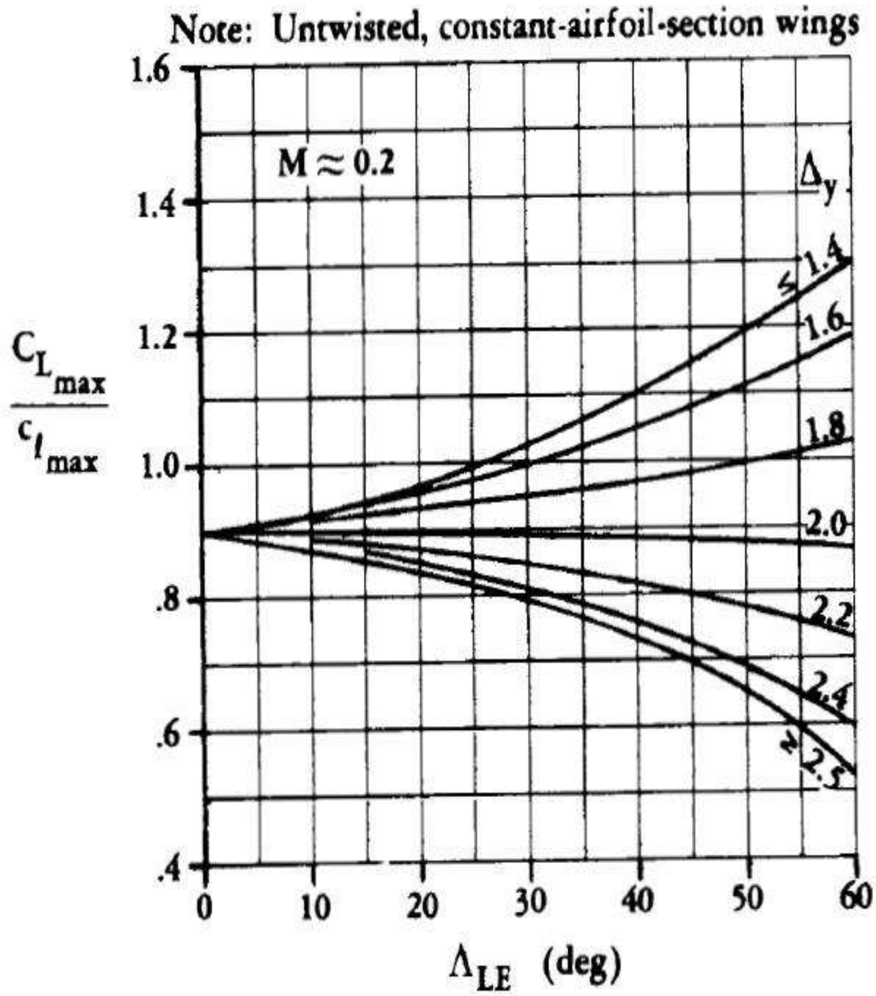


Figure 3.39: Subsonic maximum lift of high aspect ratio wings

$$\text{High Aspect Ratio:} \quad C_{L_{\max}} = C_{l_{\max}} \left(\frac{C_{L_{\max}}}{C_{l_{\max}}} \right) + \Delta C_{L_{\max}}$$

Where *C1max* is the airfoil maximum lift coefficient at *M* = 0.2. The angle of attack for maximum lift is defined in Equation above with the help of Fig. 3.40. Note that the first and second terms represent the angle of attack if the lift curve slope were linear all the way up to stall. The second term may be approximated by the airfoil zero-lift angle, which is negative

$$\text{High Aspect Ratio:} \quad \alpha_{C_{L_{\max}}} = \frac{C_{L_{\max}}}{C_{L_\alpha}} + \alpha_{0L} + \Delta\alpha_{C_{L_{\max}}}$$

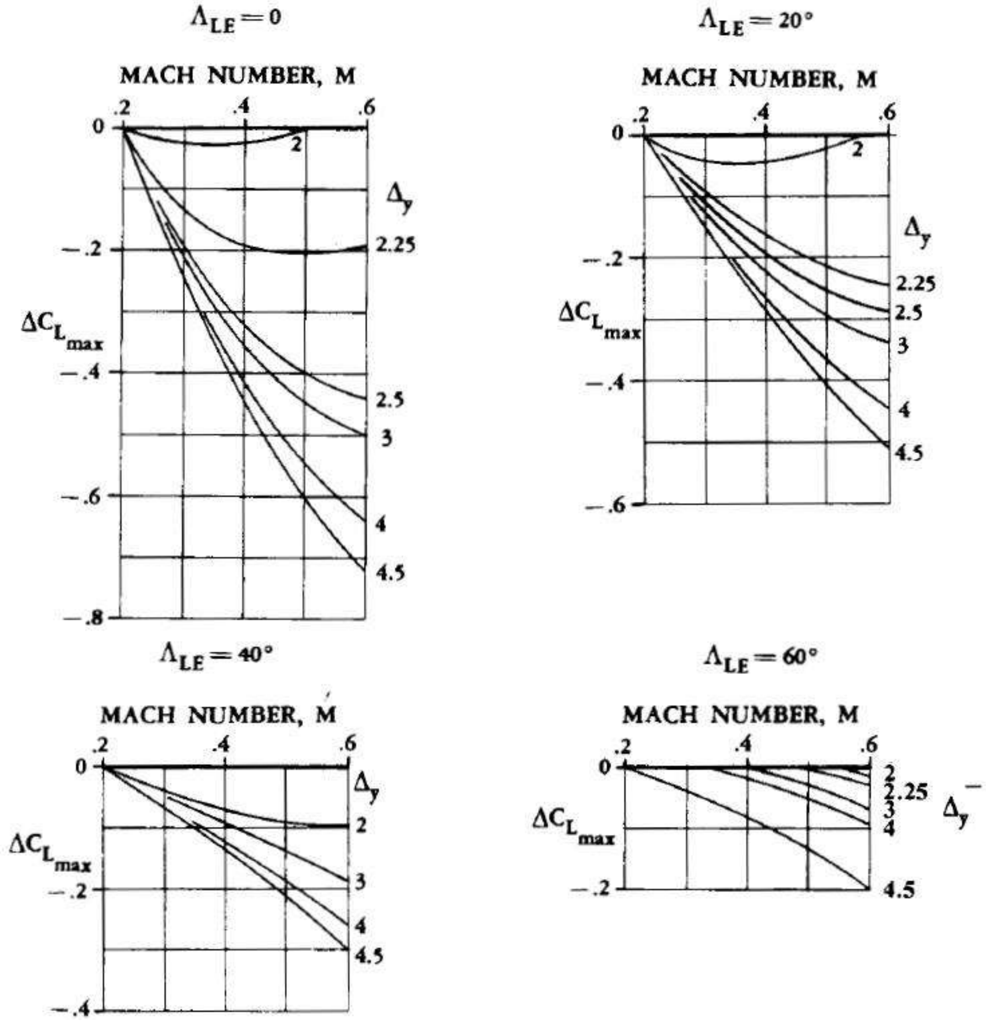


Figure 3.40: Mach-number correction for subsonic maximum lift to high-aspect ratio wings

For a cambered airfoil, if the wing is twisted, the zero-lift angle is approximately the zero lift angles at the mean chord location. The third term in Eq. above is a correction for the nonlinear effects of vortex flow.

3.14. COMPLETE DRAG BUILDS UP:

Two methods for the estimation of the parasite drag *("CD0")* are presented below. The first is based upon the fact that a well-designed aircraft in subsonic cruise will have parasite drag that is mostly skin-friction drag plus a small separation pressure drag. The latter is a fairly consistent percentage of the skin-friction drag for different classes of aircraft. This leads to the concept of an "equivalent skin friction coefficient" *(C_{fe})*, which includes both skin-friction and separation drag. C_{fe} is multiplied by the aircraft's wetted area to obtain an initial estimate of parasite drag. This estimate is suitable for initial subsonic analysis and for checking the results of the more detailed method described in the next section.

$$C_{D_0} = C_{f_e} \frac{S_{wet}}{S_{ref}}$$

Table: equivalent skin friction coefficient

$C_{D_0} = C_{fe} \frac{S_{wet}}{S_{ref}}$	C_{fe}-subsonic
Bomber and civil transport	0.0030
Military cargo (high upsweep fuselage)	0.0035
Air Force fighter	0.0035
Navy fighter	0.0040
Clean supersonic cruise aircraft	0.0025
Light aircraft – single engine	0.0055
Light aircraft – twin engine	0.0045
Prop seaplane	0.0065
Jet seaplane	0.0040

$$(C_{D_0})_{subsonic} = \frac{\Sigma(C_{f_c} FF_c Q_c S_{wet_c})}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}}$$

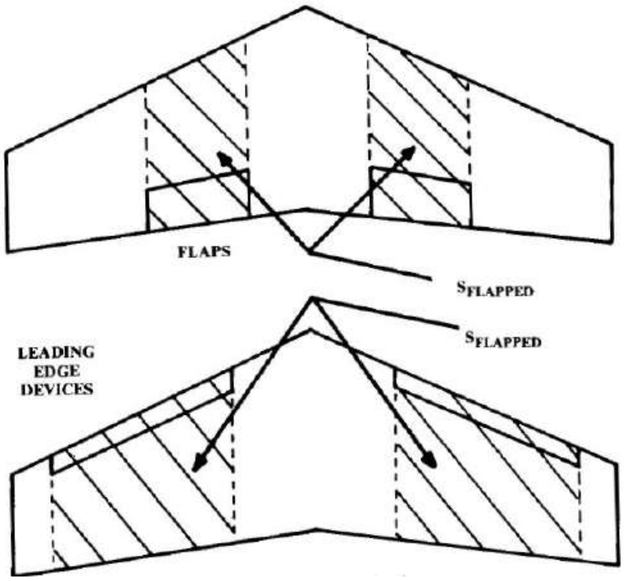


Figure 3.41: flapped wing area

3.15. INSTALLED PERFORMANCE OF AN ENGINE AND INSTALLATION

Aircraft propulsion develop thrust by pushing air (or hot gases) backward. In a simplified case the force obtained can be determined using Newton's equation *(F =ma)* by summing all the accelerations imparted to the air.

The analysis above is too simplistic for actual thrust calculation. It falsely assumes that the fluid velocity is constant throughout the exhaust and that all of the accelerations experienced by the air mass occur at the propeller plane or within the jet engine.

Actually, the exhaust of a jet engine is usually at a higher pressure than the outside air, so the flow expands after leaving the nozzle. In other words the air is still accelerating after the aircraft has passed. For a propeller, the air mass acceleration doesn't even occur at the propeller disk! Roughly half the air mass acceleration occurs before reaching the propeller, and the other half occurs after passing the propeller.

Propulsion force estimation is also complicated by the fact that the propeller flow field or jet intake and exhaust will influence the whole flow field of the aircraft. It has already been mentioned that a pusher propeller will reduce the drag of a stubby aft fuselage by "sucking" air inward and preventing flow separation. Should this reduced drag be considered a part of the propulsive force because it is controlled with the throttle? What about the increased drag due to the propeller wake on a conventional airplane?

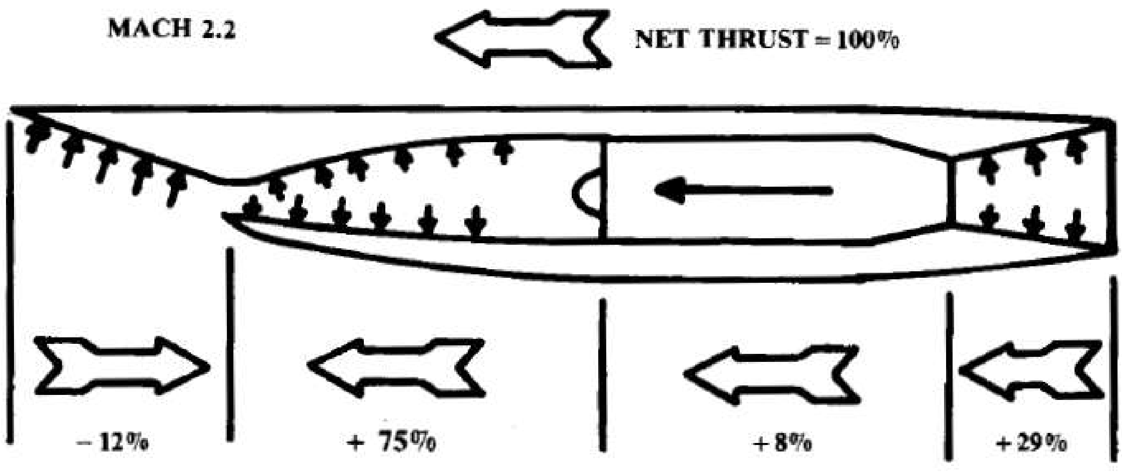


Figure 3.42: Turbojet thrust contributors

For the external inlet system drag. In fact, the subsonic expansion inside the inlet duct is the largest single contributor of thrust! This illustrates the difficulty of calculating thrust by any simple model. This chapter provides methods for estimating the net thrust provided by a propeller or jet engine as a part of the overall vehicle analysis and optimization.

These methods are simplified to permit experiencing the whole design process within the time devoted to the typical college design course. The chapter will also introduce the vastly more complicated process of installed-thrust estimation used at major aerospace firms.(highly recommended) provides a detailed treatment of jet-engine design.

3.16. INSTALLED THRUST METHODOLOGY

The actual available thrust used in performance calculations called the installed net propulsive force" is the uninstalled thrust corrected for installation effects, minus the drag contributions that are assigned to the Propulsion system by the selected thrust-drag book keeping system. This is depicted "manufacturer's uninstalled engine thrust" is obtained by cycle analys1s and/or testing, or can be approximated using fudge-factors as described above. The manufacturer's engine data is based upon some assumed schedule of inlet pressure recovery vs. Mach number. Inlet pressure recovery (P_1/P_0) is the total pressure at the engine front face (1) divided by the total pressure m the free stream (0).

The "installed-engine thrust" is the actual thrust generated by the engme when installed in the aircraft. This is obtained by correcting the thrust for the actual inlet pressure recovery and nozzle performance, and applying thrust losses to account for engine bleed and power extraction. "Inlet distortion" refers to pressure and velocity variations in the air delivered to the engine. It primarily affects the allowable operating envelope of the engine net propulsive force.

The installed net propulsive force is the installed engine thrust minus the inlet, nozzle, and throttle-dependent trim drags. The steps depicted in Fig.3.43

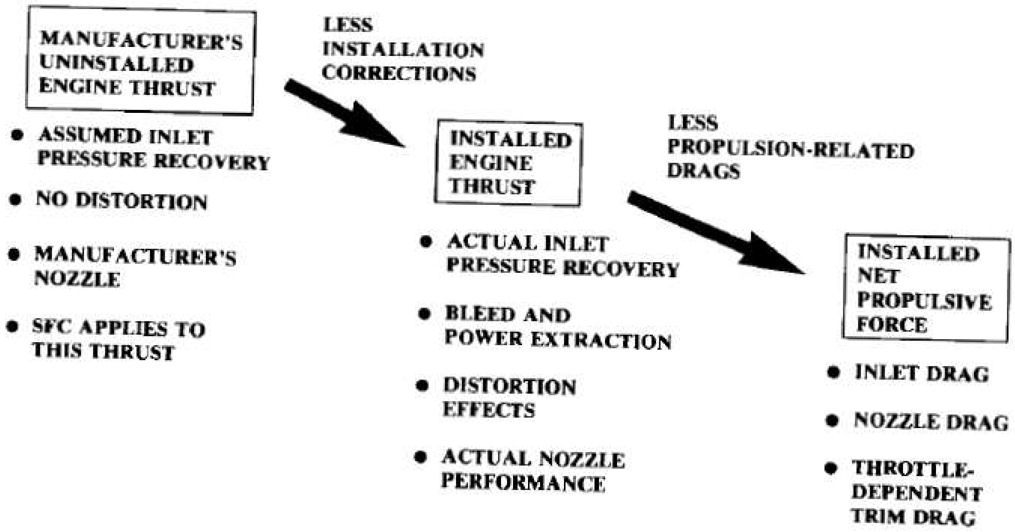


Figure 3.43: installed thrust methodology

3.16.1. Installed Net Propulsive Force Corrections

The "installed engine thrust" is the actual thrust produced by the engine as installed in the aircraft. However, the engine creates three forms of drag that must be subtracted from the engine thrust to determine the thrust force actually available for propelling the aircraft.

This propelling force the "installed net propulsive force," is the thrust value to be used for aircraft performance calculations. Most of the engine-related drag is produced by the inlet as a result of a mismatch between the amount of air demanded by the engine and the amount of air that the inlet can supply at a given flight condition. When the inlet is prov1dmg exactly the amount of air the engine demands (mass flow ratr10 equals 1.0), the inlet drag is negligible.

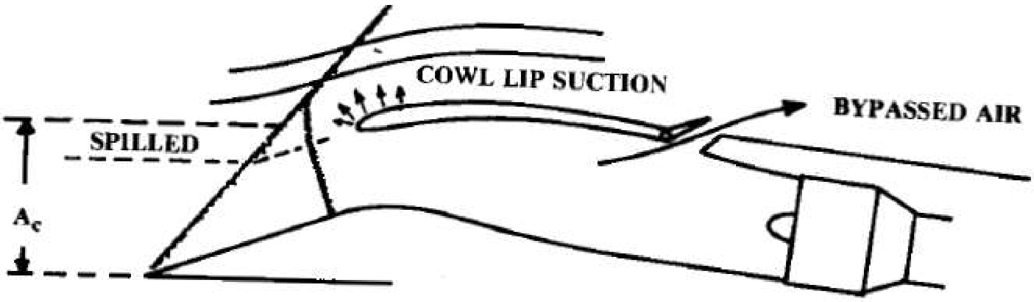


Figure 3.44: additive drag, cowl lip suction and bypass subcritical operation

Allowing the excess air to enter the inlet and be dumped overboard or into an ejector nozzle, will keep the inlet additive drag to a small value. The resulting bypass drag will be substantially less than the additive drag would have been. Bypass drag is calculated by summing the momentum loss experienced by the bypassed air

Another form of inlet drag is the momentum loss associated with the inlet boundary-layer bleed. Air is bled through holes or slots on the inlet ramps and within the inlet to prevent shock-induced separation and to prevent the buildup of a thick turbulent boundary layer within the inlet duct. This air is dumped overboard out an aft-facing discharge exit, which is usually located a few feet behind the inlet.

Calculation of bleed, bypass, and additive drag including cowl-lip suction is a complicated procedure combining analytical and empirical methods.
In a major aircraft company such calculations are made by propulsion specialists using complex computer programs. The results are included in the installed net propulsive force data that are provided to the sizing and performance analyst.

3.17. AIRCRAFT LOADS, CATEGORIES–MANEUVER, GUST, INERTIAL, POWER PLANT, LANDING GEAR LOADS, LIMIT LOADS THE V-N DIAGRAM

When one thinks of aircraft loads the air loads due to high-g maneuvering come immediately to mind while important maneuvering loads are only a part of the total loads that must be withstood by the aircraft structure
Table: aircraft loads the largest load the aircraft is actually expected to encounter is called the "limit," or "applied," load. For the fighter of Fig.3.45, the limit load on the wing occurs during an *8-g* maneuver.

To provide a margin of safety, the aircraft structure is always designed to withstand a higher load than the limit load. The highest load the structure is designed to withstand without breaking is the "design," or "ultimate," load.

The "factor of safety" is the multiplier used on limit load to determine the design load. Since the 1930's the factor of safety has usually been. This was defined in an Air Corps specification based upon the ratio between the ultimate tensile load and yield load of 24ST aluminum alloy, and has proven to be suitable for other aircraft materials in most cases.

Airloads	Landing	Other
– Maneuver	– Vertical load factor	– Towing
– Gust	– Spin-up	– Jacking
– Control deflection	– Spring-back	– Pressurization
– Component interaction	– Crabbed	– Bird strike
– Buffet	– One wheel	– Actuation
	– Arrested	– Crash
	– Braking	
Inertia loads	**Takeoff**	
– Acceleration	– Catapult	
– Rotation	– Aborted	
– Dynamic		
– Vibration		
– Flutter		
Power plant	**Taxi**	
– Thrust	– Bumps	
– Torque	– Turning	
– Gyroscopic		
– Vibration		
– Duct Pressure		

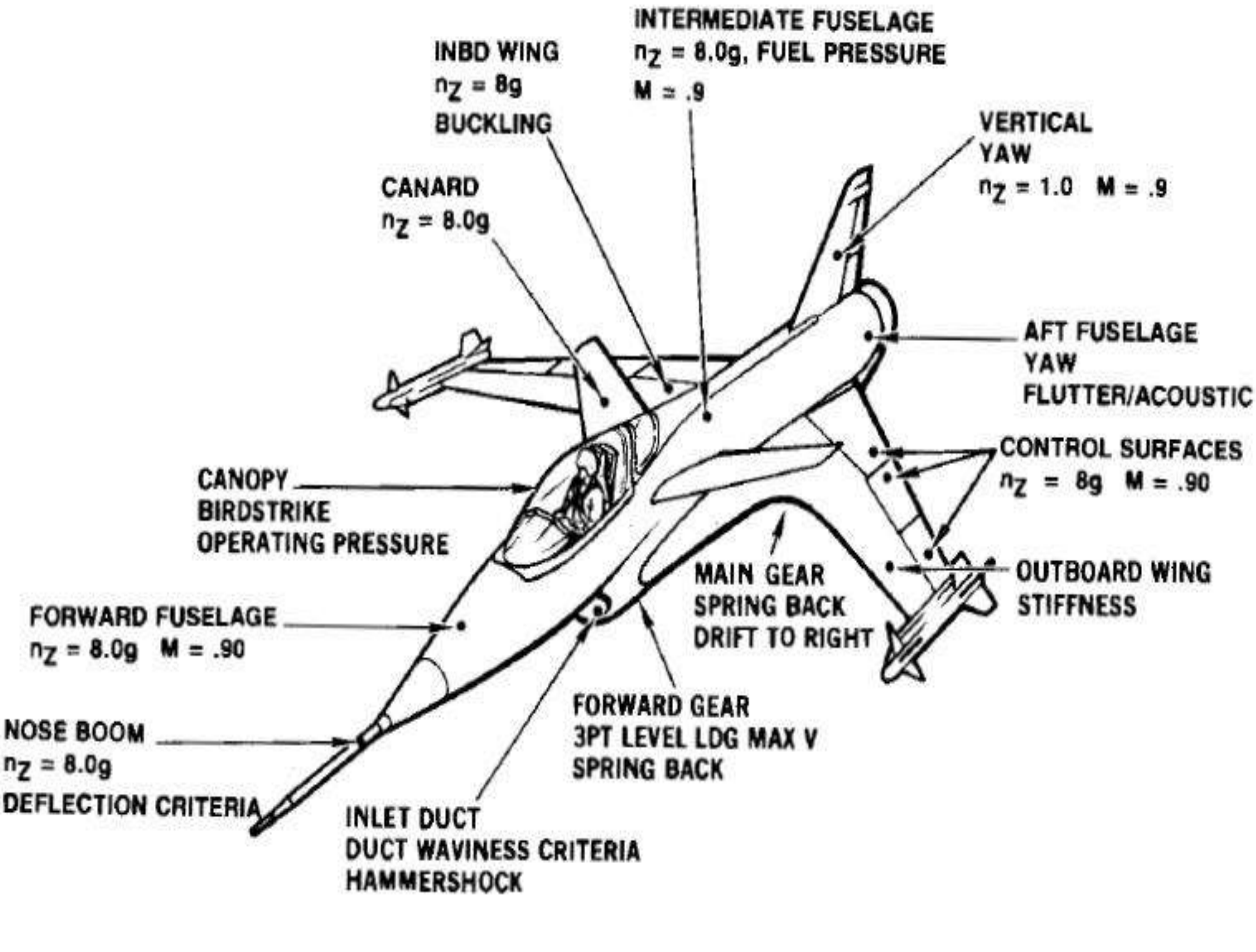


Figure 3.45: typical fighter limit loads

3.17.1. Maneuver Loads

The greatest air loads on an aircraft usually come from the generation of lift during high-g maneuvers. Even the fuselage is almost always structurally sized by the lift of the wing rather than by the air pressures produced directly on the fuselage. Aircraft load factor *(n)* expresses the maneuvering of an aircraft as a multiple of the standard acceleration due to gravity (g = 32.2 ft/s-s).

At lower speeds the highest load factor an aircraft may experience is limited by the maximum lift available. At higher speeds the maximum load factor is limited to some arbitrary value based upon the expected use of the aircraft. The Wright Brothers designed their Flyer to a *5-g* limit load. This remains a reasonable limit load factor for many types of aircraft. Lists typical limit load factors. Note that the required negative load factors are usually much less in magnitude than the positive values.

The *V-n* diagram depicts the aircraft limit load factor as a function of airspeed. The *V-n* diagram of Fig. 14.3 is typical for a general aviation aircraft. Note that the maximum lift load factor equals 1.0 at level-flight stall speed, as would be expected. The aircraft can be stalled at a higher Speed by trying to exceed the available load factor, such as in a steep turn. The point labeled "high A.O.A." (angle of attack) is the slowest speed at which the maximum load factor can be reached without stalling. This part of the flight envelope is important because the load on the wing is approximately perpendicular to the flight direction, not the body-axis vertical direction.

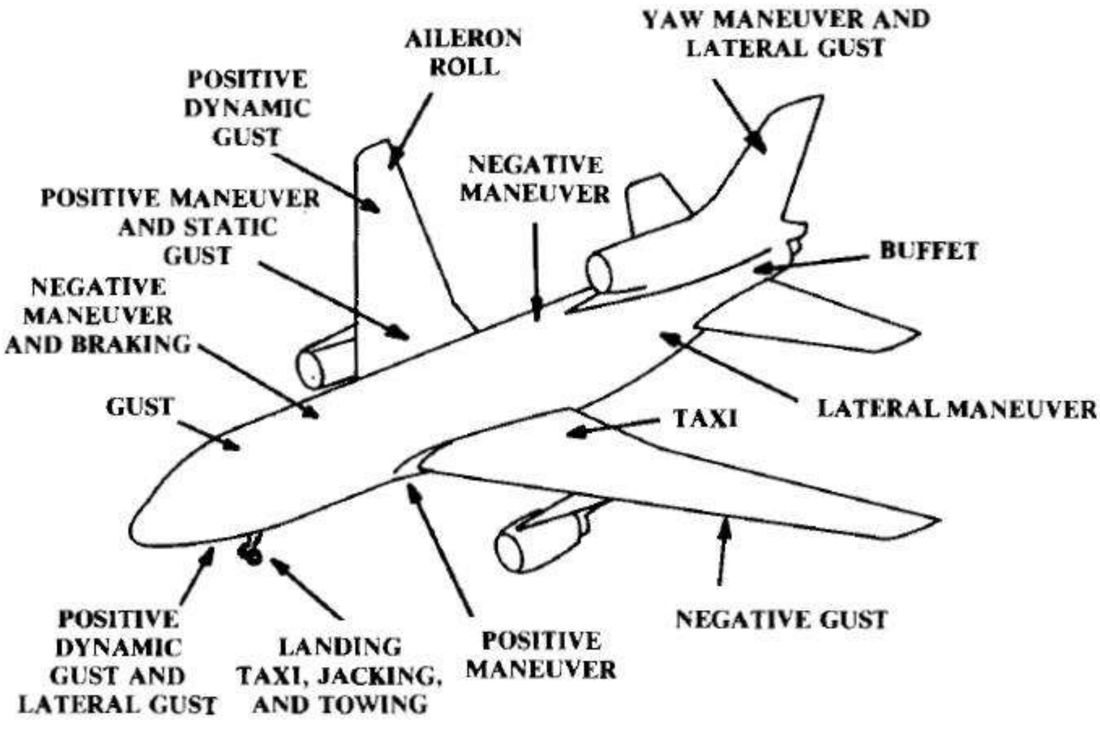
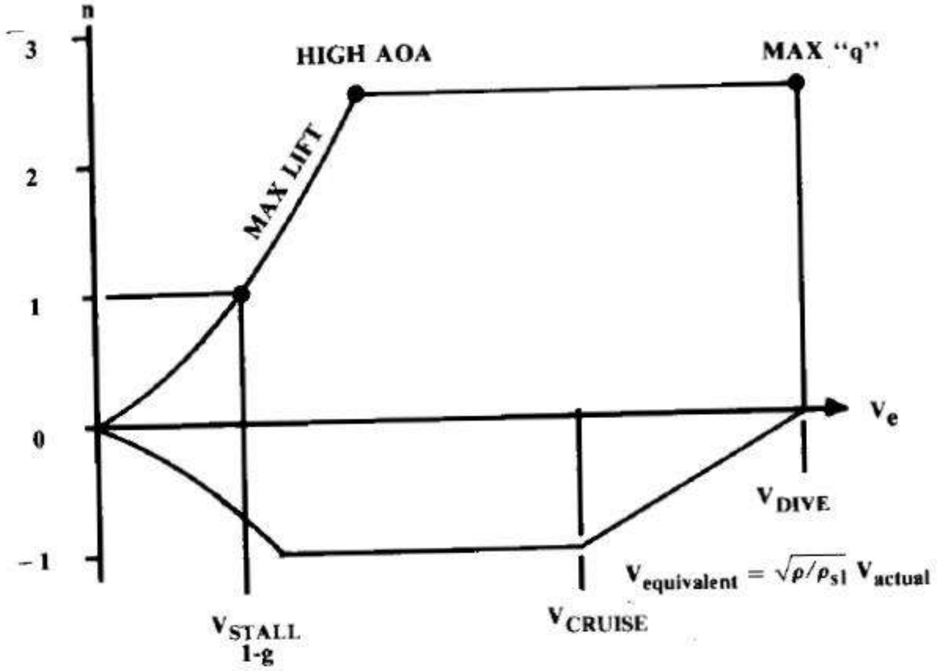


Figure 3.46: L1011 Critical



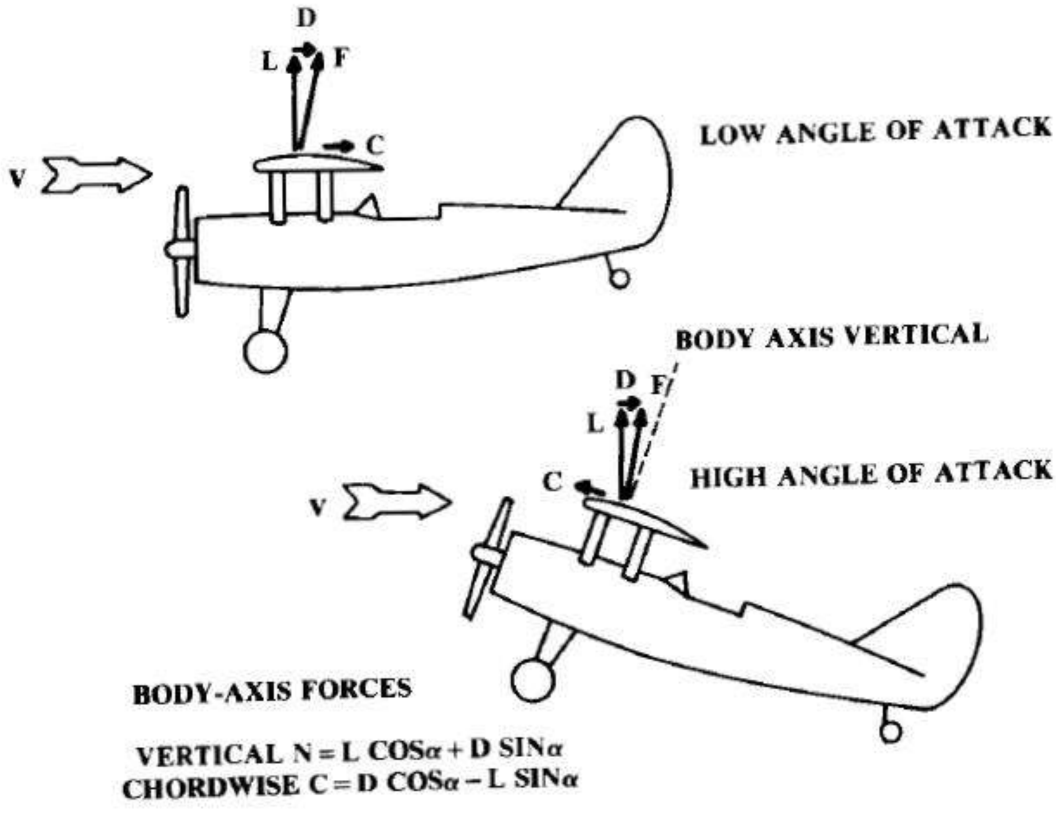


Figure 3.47: V-n diagram(maneuver)

Figure 3.48: wing load direction at angle of attack

3.17.2. GUST LOADS

The loads experienced when the aircraft encounters a strong gust can exceed the maneuver loads in some cases. For a transport aircraft flying near thunderstorms or encountering high-altitude "clear air turbulence," it is not unheard of to experience load factors due to gusts ranging from a negative 1.5 to a positive 3.5 *g* or more. When an aircraft experiences a gust, the effect is an increase (or decrease) in angle of attack. Figure 3.50 illustrates the geometry for an upward gust of velocity *U*. The change in angle of attack, as shown is approximately *U* divided by *V*, the aircraft velocity. The change in aircraft

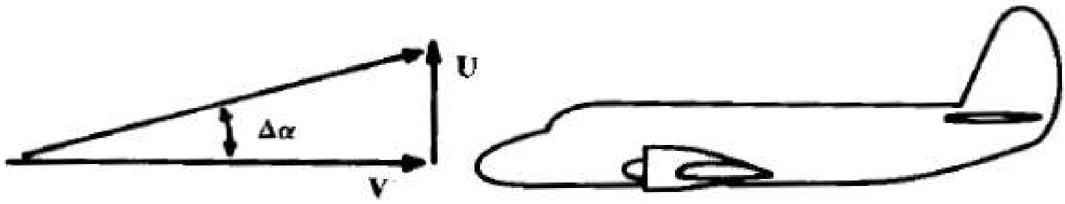


Figure 3.49: Gust Encounter

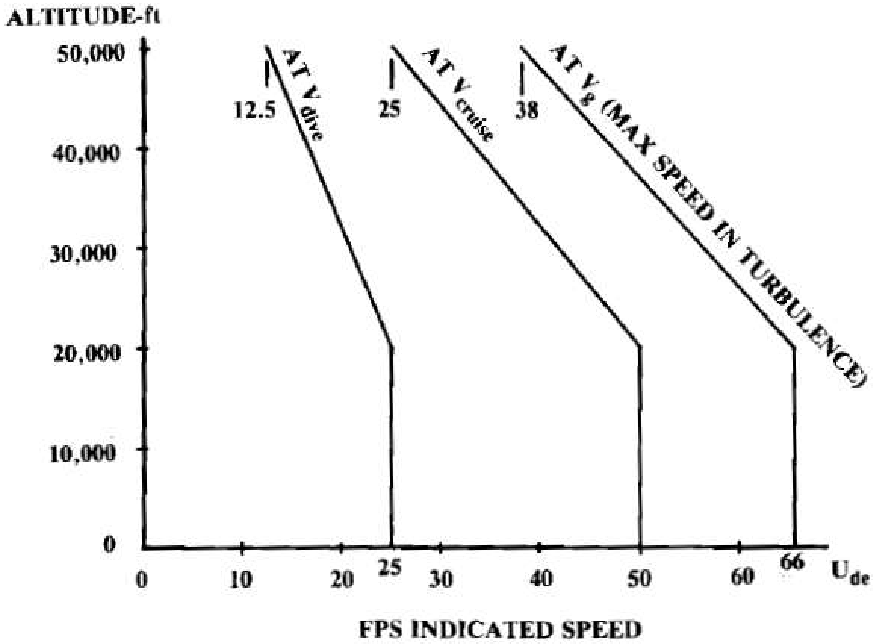


Figure 3.50: Derived equivalent gust velocities (transport)

Aero elastic effects can also influence the load factor due to gusts. An aft-swept wing will bend up under load, which twists the wing and reduces the outboard angle of attack. This reduces total lift and also moves the span wise lift distribution inboard, reducing the wing bending stress. An aft-swept wing will experience roughly 15% lower load factor due to a given gust than an unswept wing.

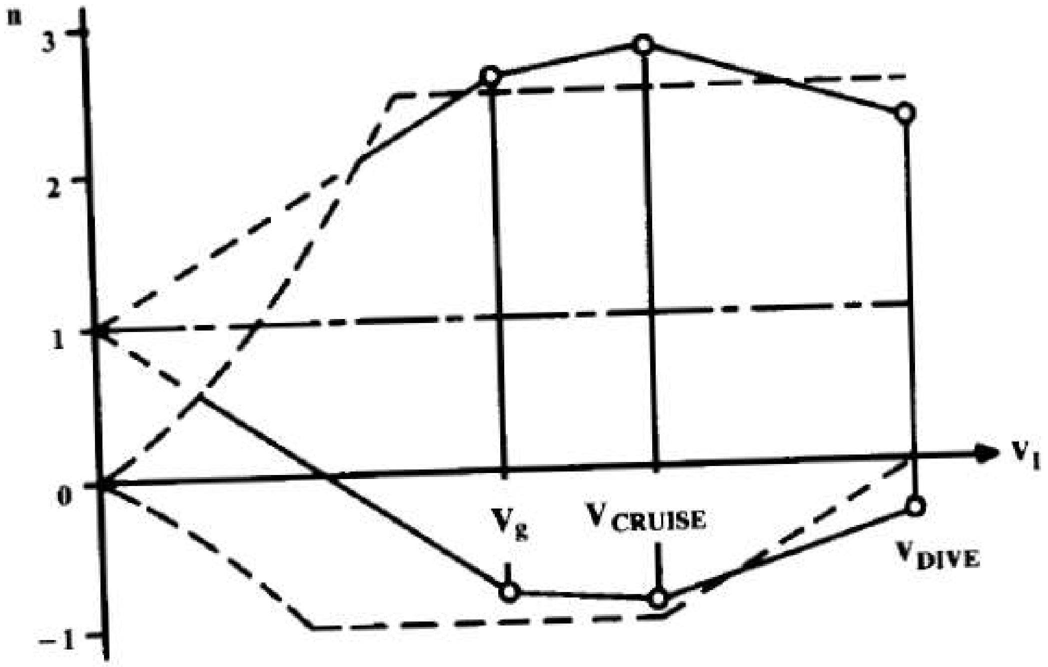


Figure 3.51: V-n diagram (gust)

This method for estimation of gust loads is not as complete or accurate as the methods used at most large aircraft companies. The more accurate methods rely upon a power-spectral-density approach in which the gusts are included in an atmospheric transfer function and the actual aircraft *dynamics* are modeled. However, the methods presented above_ are use ~ or initial analysis and provide an introduction to the more detailed techniques.

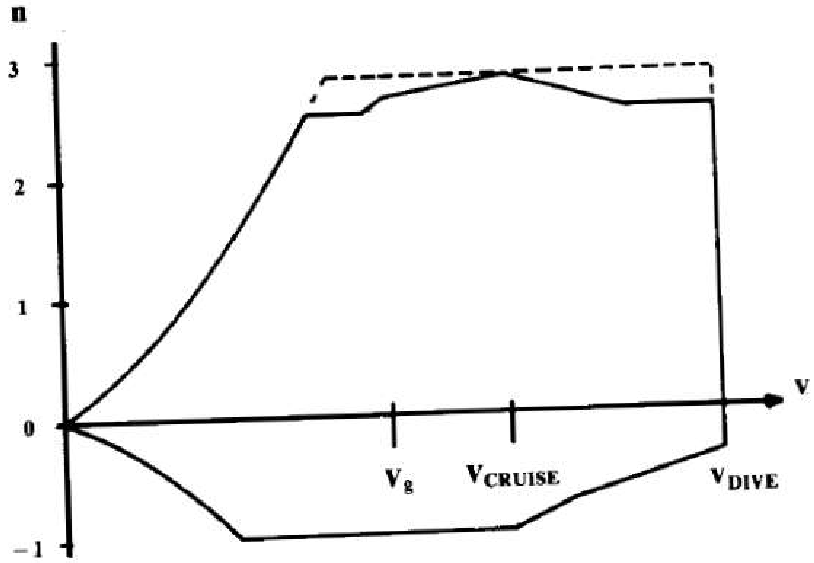


Figure 3.52: combined V-n diagram

3.17.3. POWER-PLANT LOADS

The engine mounts must obviously be able to withstand the thrust of the engine as well as its drag when stopped or wind milling. The mounts must also vertically support the weight of the engine times the design load factor. The engine mounts are usually designed to support a lateral load equal to one-of the vertical design load. The mounts must withstand the gyroscopic loads caused by the rotating machinery (and propeller) at the maximum pitch and yaw rates.

For a propeller-powered aircraft, the engine mounts must withstand the torque of the engine times a safety factor based upon the number of cylinders. This reflects the greater jerkiness of an engine with few cylinders when one cylinder malfunctions. ~or an engine with two cylinders, the safety factor is 4.0; with three cylinders, 3.0; and with four cylinders, 2.0. An engine with five or more Cylinders requires a safety factor of 1.33. These safety factors are multiplied times the maximum torque m normal operation to obtain the design torque for the engine mounts.

For a jet engine, air loads within the inlet duct must be considered as they will frequently bound a part of the flight envelope. A pressure surge known as "hammer shock" is especially severe.

3.17.4. LANDING-GEAR LOADS

The landing gear's main purpose is to reduce the landing loads to a level that can be withstood by the aircraft. Calculations for landing gear stroke to yield an acceptable gear load-factor, as transmitted to the structure of the aircraft.

To analyze fully all the possible gear loads, a number of landing scenarios must be examined. These include a level landing, a tail-down landing, a one-wheel landing, and a crabbed landing. For certification the aircraft may be subjected to drop tests, in which an actual aircraft is dropped from a height of somewhere between 9.2-18. 7 in.

The required drop distance typically will be 3 .6 times the square root of the wing loading. When the tires contact the ground they are not rotating.

During the brief fraction of a second it takes for them to spin up, they exert a large rearward force by friction with the runway. This spin-up force can be as much as half the vertical force due to landing.

When the tire is rotating at the correct speed, the rearward force is relieve and the gear strut "springs back" forward, overshooting the original position and producing a "spring-back" deflection load equal to or greater than the spin-up load.

Another landing-gear load, the braking load, can be estimated by assuming a braking coefficient of 0.8.

The load on the landing gear during retraction is usually based upon the a1rloads plus the assumption that the aircraft is in a *2-g* turn. Other landing gear loads such as taxiing and turning are usually of lesser importance, but must be considered during detail design of the landing gear and supporting structure.

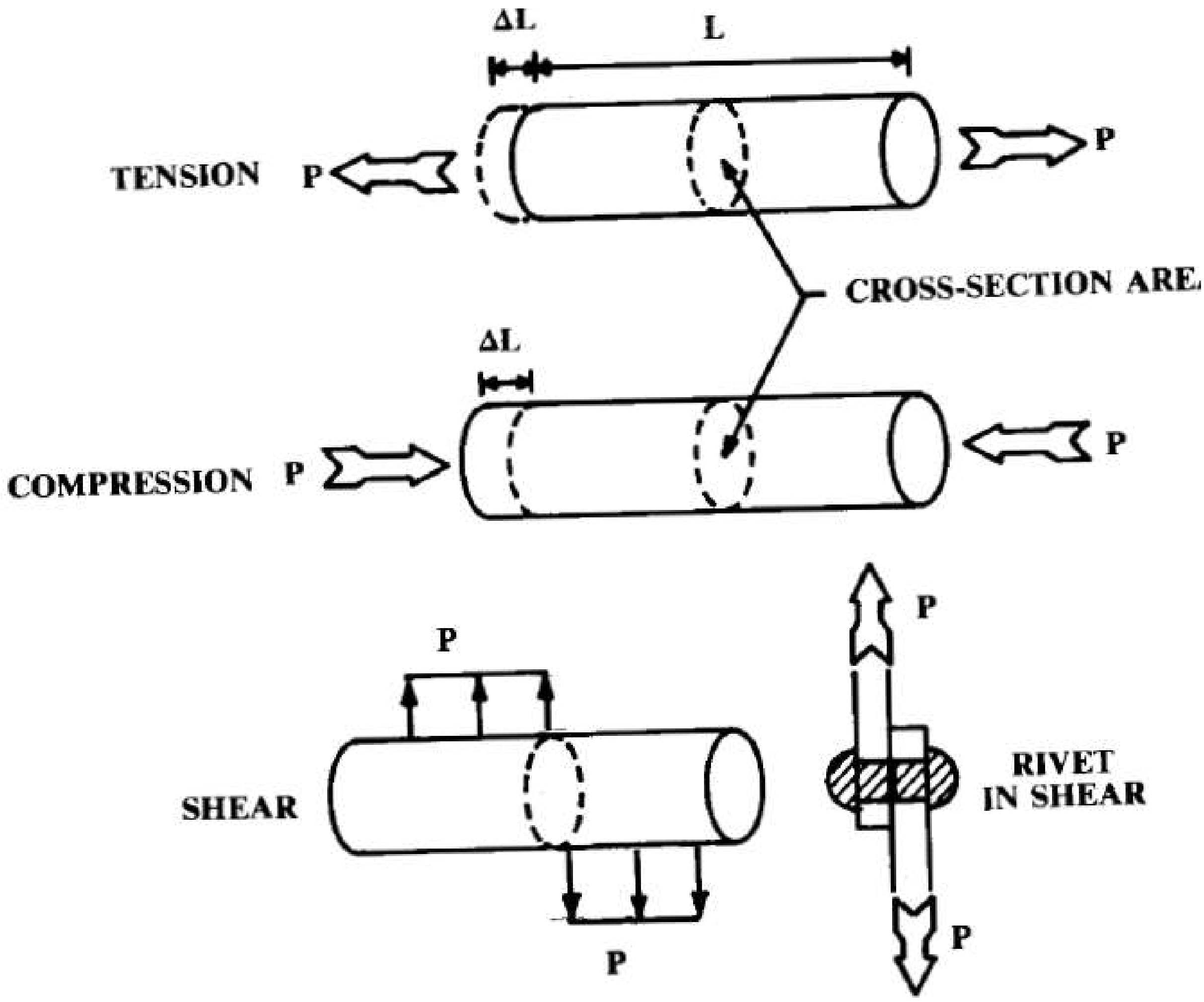


Figure 3.53: Three basic structural loadings

3.18. AIR LOAD DISTRIBUTION ON LIFTING SURFACES

The first step involves a stability-and-control calculation to determine the required lift on the horizontal tail to balance the wing pitching moment at the critical conditions.

Note that the required tail lift will increase or decrease the required wing lift to attain the same load factor. Complicated methods for estimating the lift on the trimmed tail and wing for a given load factor.

These can be initially approximated by a simple summation of wing and tail moments about the aircraft center of gravity, ignoring the effects of downwash, thrust axis, etc. Once the total lift on the wing and tail are known, the span wise and chord wise load distributions can be determined.

Wind-tunnel and aerodynamic panel program data are used if available. For initial design and design of light aircraft, classical approximation methods give reasonably good results.

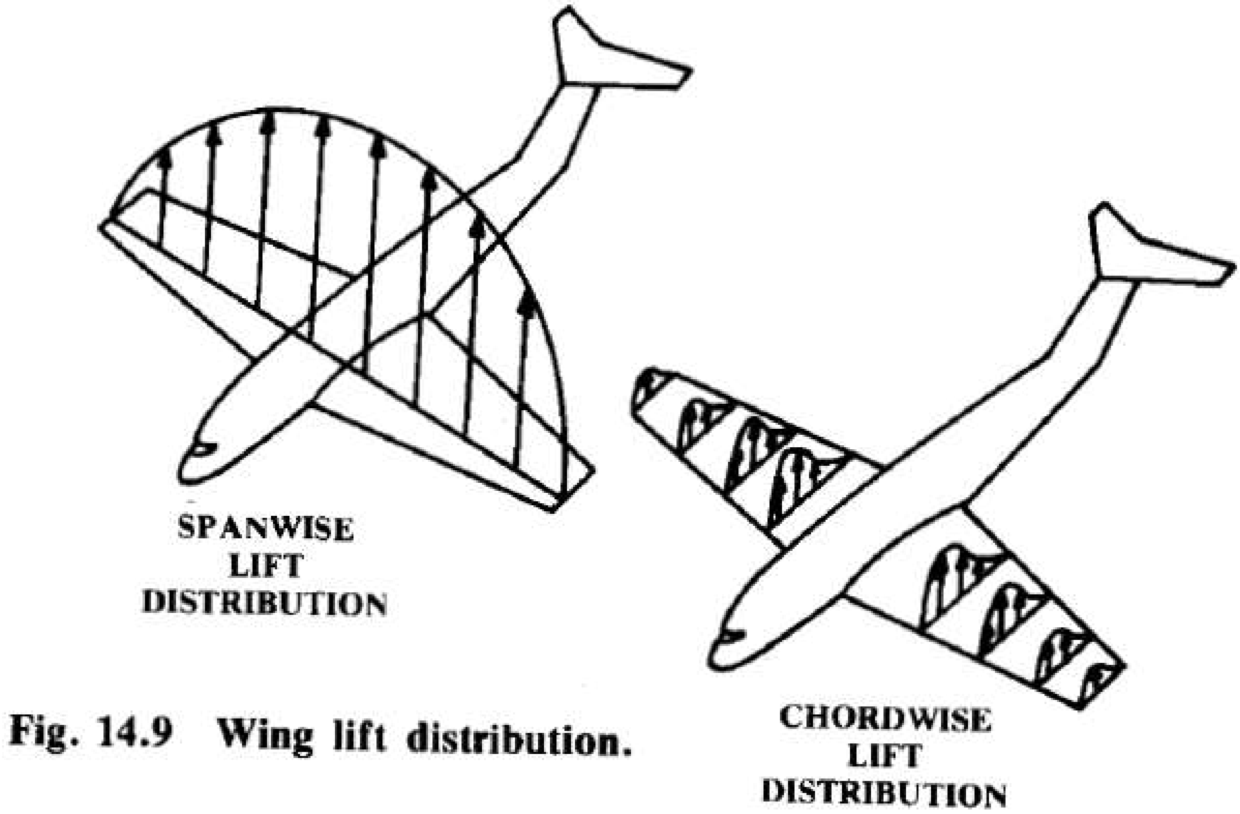


Figure 3.54: Wing lift distribution

Schrenk's Approximation does not apply to to greatly increase the loads at the wing tips . Loads for such a planform must be estimated using computers and wind tunnels.

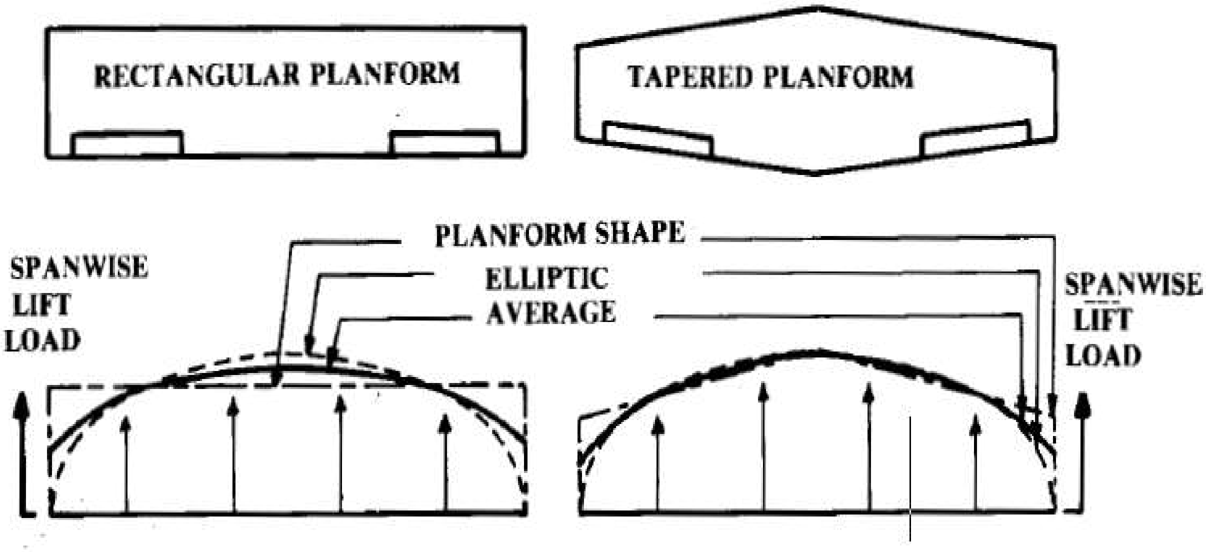


Figure 3.56: schrenk’s approximations

3.19. MATERIAL SELECTION

A number of properties are important to the selection of materials for an aircraft. The selection of the "best" material depends upon the application.

Actors to be considered include yield and ultimate strength, stiffness, density, fracture toughness, fatigue crack resistance, creep, corrosion resistance, temperature limits, producibility, reparability, cost, and availability. Strength, stiffness, and density have been discussed already.

Fracture toughness measures the total energy per unit volume required to deflect the material to the fracture point, and is equivalent to the area under the stress strain curve.

A ductile material with a large amount of inelastic deformation prior io fracture will absorb more work energy in fracturing than a maternal with the same ultimate stress but with little inelastic deformation prior to fracture.

A material subjected to a repeated cyclic loading will eventually experience failure at a much lower stress than the ultimate stress.

This "fatigue" effect 1s largely due to the formation and propagation of cracks and is probably the single most common cause of aircraft material failure.

There are many causes of fatigue; including gust loads, landing impact, and the v1brat10ns of the engine and propeller.

UNIT-IV
BASELINE DESIGN ANALYSIS - II

4.1. Estimation of static pitch stability

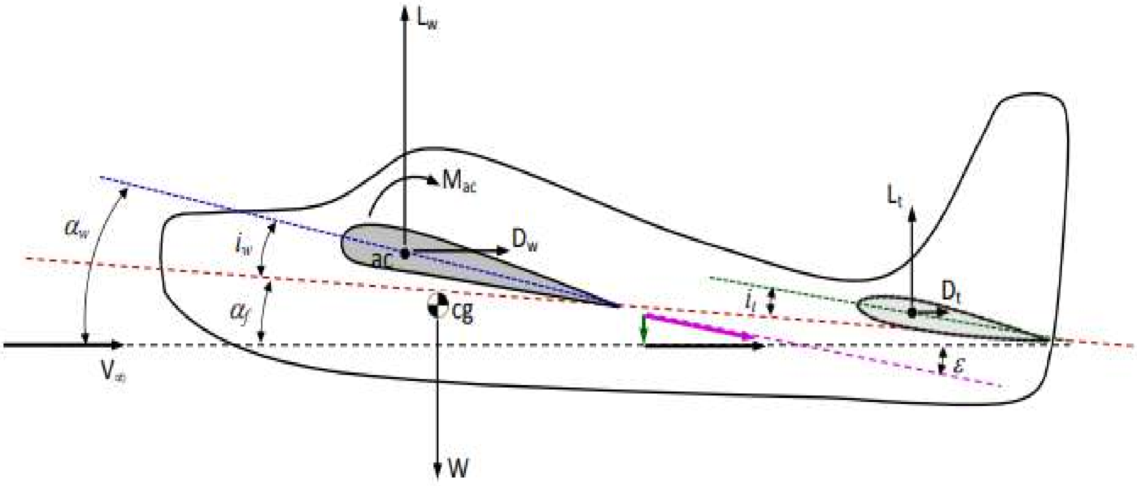


Figure 4.1:Longitudinal (pitch) stability requires a moment

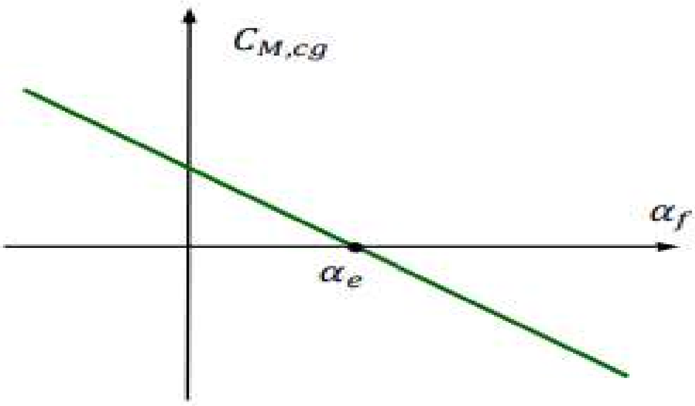


Figure 4.2. cg ($C_{M,cg}$)vs. angle of attack α_f curve as in the following graph:

the cg point. Negative slope of the moment curve provides the static stability such that if the AoA increases (say, due to a momentary gust), a negative pitching moment brings the airplane AoA back to the equilibrium point. There is a certain equilibrium AoA (α_e denoted by solid circle in the graph) at which the pitching moment about cg becomes zero. This corresponds to level flight conditions without any pitching about.

Note that if the airplane is designed for a level cruising flight along the fuselage axis (red dotted line), then the AoA of airplane becomes zero. Therefore, in such a design, equilibrium point has to be achieved at zero AoA in the moment curve(i.e., α_e =0)

4.2. VELOCITY STABILITY AND TRIM

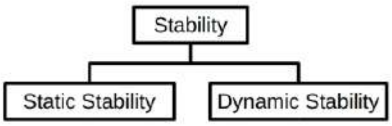


Figure 4.3: Stability

4.2.1. Static Stability

If an airplane disturbed from equilibrium state has "Initial Tendency" to return to its equilibrium state, then the aircraft is assumed to have static stability.

4.2.2. Dynamic Stability

Not only initial tendency, but also the amplitudes of the response due to disturbance decay in finite time to attain the equilibrium state.

In general, when aircraft is being referred to be in stable equilibrium, we mean dynamic stability. However, it so happen that for most of the cases, for conventional aircraft, if it is statically stable, it also automatically satisfies dynamic stability criterion – but not all aircraft! Handling qualities may be different

- Static equilibrium occurs whenever there is no acceleration (linear or angular) of the aircraft. Un-accelerated flight requires that the summations of forces and moments acting on the aircraft are zero.
- Static equilibrium also requires that the side force acting on the airplane is also zero.
- Additionally, the summation of moments about the centre of gravity (CG) in roll, pitch and yaw must all be zero for equilibrium (Trimmed flight).

4.3. STABLE TRIM

4.3.1. Longitudinal (Axial)

Small translational disturbances in axial, normal or side slip velocity must all result in a return to the original trimmed equilibrium condition. This is also referred to as pitch stability.

- An object moving through the air will experience drag that opposes the motion.
- If angle of attack remains fixed, this drag will increase with speed. (Drag opposes increase in speed)
- Thrust developed by engine is either constant with airspeed or decrease with increasing air speed. (Drag increase in speed)
- In static equilibrium with regard to translational in the direction of motion, the forward component of thrust must balance the drag (T = D)
- At constant angle of attack, a small increase in airspeed will result in
  - Increase in Drag
  - Either a decrease in Thrust or No change in Thrust
- Therefore, this force imbalance in the axial direction will result in a deceleration, which will tend (initial tendency) to restore the airspeed to the original value.
- Conversely, if airspeed is decreased by a small disturbance with no change in angle of attack, the drag will become less than the thrust and the aircraft.

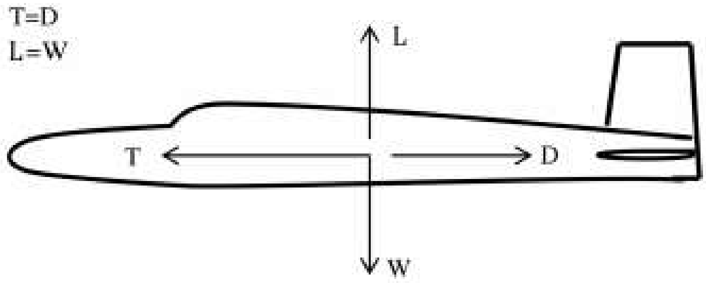


Figure 3: Pitch Stability

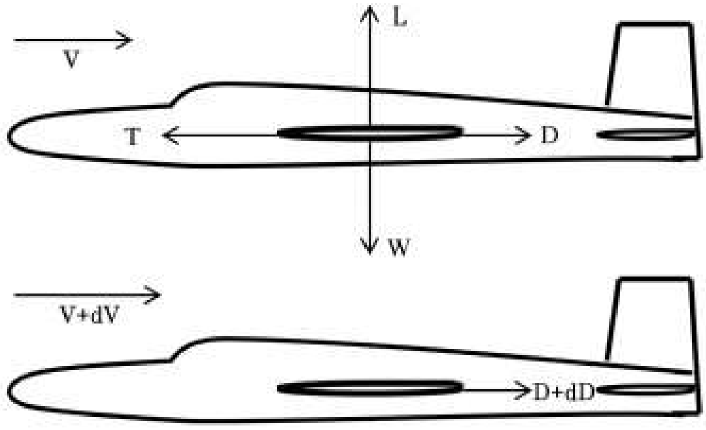


Figure 4.3. : *dD* will oppose *Dv*

If dV is positive; dD will act to reduce/marginalize dV .If dV is negative; dD will tend to increase the speed as in that case T > D.

4.3.1. Estimation of Stability and Control Derivatives

The solution of small perturbation equations for longitudinal motion would be taken up in chapter 8 and for the lateral motion in chapter 9. However, to solve these equations the stability derivatives are required. The following subsections deal with their estimation.

Derivatives due to change of 'Δu'

These derivatives include $\partial X / \partial u$, $\partial Z / \partial u$ and $\partial M / \partial u$

$\partial X / \partial u$

The changes in ΔX are caused by changes in the drag and the thrust i.e

$$\Delta X = -\Delta D + \Delta T$$

Continuing with the linearized treatment of the problem, the variation of ΔX with Δu is expressed as:

$$\Delta X = -\frac{\partial D}{\partial u}\Delta u + \frac{\partial T}{\partial u}\Delta u$$

$$\text{Or } \frac{\partial X}{\partial u} = -\frac{\partial D}{\partial u} + \frac{\partial T}{\partial u} = -\frac{\partial}{\partial u}\left(\frac{1}{2}\rho\, u_0^2\ S\, C_D\right) + \frac{\partial T}{\partial u} = -\frac{1}{2}\rho S\left(u_0^2 \frac{\partial C_D}{\partial u} + 2u_0 C_D\right) + \frac{\partial T}{\partial u}$$

Recall that $X_u = (1 / m)(\partial X / \partial u)$ and let $C_{DU} = \partial CD / \partial (u / u0)$

As regards the term, $\partial T / \partial u$ the following may be noted

a) For gliding flight, T= 0 and hence, $\partial T / \partial u = 0$.
b) For a jet airplane, T is almost constant over small intervals of u and hence, $\partial T / \partial u = 0$
c) For a piston engine airplane with variable pitch propeller, the THP is nearly constant over a small range of u. Hence

$T = THP/u$ and consequently $\partial T/\partial u = - THP/u^2 = - D/u$

As regards C_{DU} the following facts may be noted.

$$C_{Lu} = M_1 \frac{dC_L}{dM_1} = M_1\, \alpha \frac{dC_{L\alpha}}{dM_1}$$

The term $C_{L\alpha}$ as a function of Mach number is given as:

$$C_{L\alpha} = \frac{2\pi A}{2+\sqrt{\frac{A^2\beta_1^2}{K^2}\frac{(1+\tan^2\Lambda_{c/2})}{\beta_1^2}+4}} \text{ in rad}^{-1}$$

a. <u>∂M/∂u</u>

It may be added that:

a) At low subsonic Mach number $C_L u$ can be neglected

$$C_{mu} = M_1 \frac{\partial C_m}{\partial M_1} = M_1\, \alpha \frac{dC_{m\alpha}}{dM_1} \text{ where, } M_1 \text{ is Mach number}$$

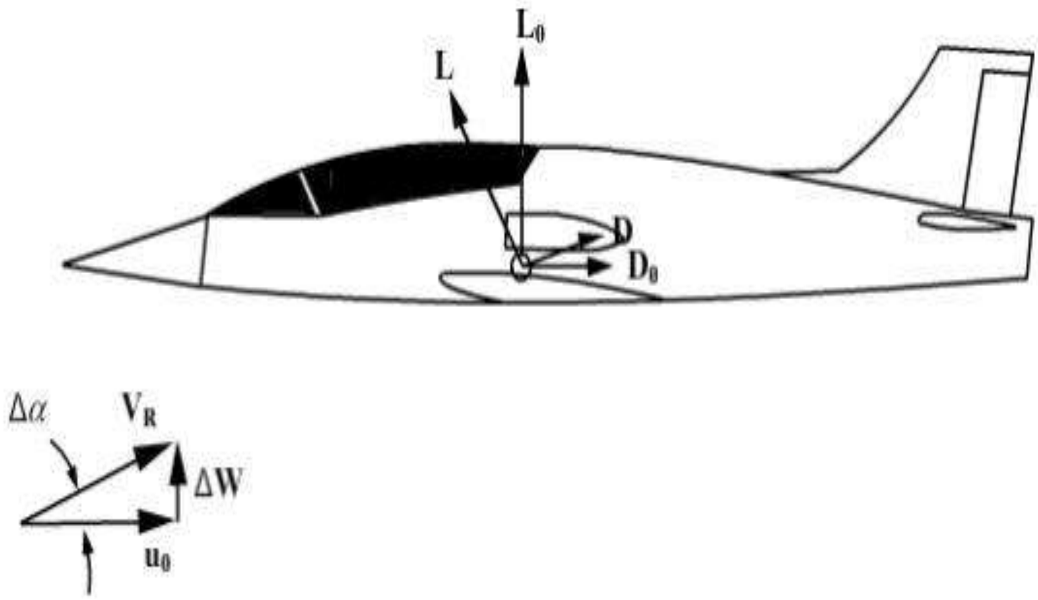


Fig 4.4. : Stability derivatives due changes of w

4.4. STATIC LATERAL, DIRECTIONAL STABILITY AND TRIM

Stability is the tendency of an airplane to fly a prescribed flight course. Dynamic longitudinal stability concerns the motion of a statically stable airplane, one that will return to equilibrium after being disturbed. Basically, there are two primary forms of longitudinal oscillations with regard to an airplane attempting to return to equilibrium after being disturbed.

The first form is the phugoid mode of oscillation, which is a long-period, slow oscillation of the airplane's flight path. The pilot generally can control this oscillation himself. The second oscillation is a short-period variation of the angle of attack.

Usually, this oscillation decreases very quickly with no pilot effort. However, with its natural short period, the oscillation may worsen if a pilot attempts to lessen it by use of a control because of the pilot's slow reaction time where he may get "out of phase" with the oscillation, and thus induce dynamical instability that may eventually lead to destructive forces.

A second type of short-term oscillation occurs if the elevators are left free. This is called the "purposing" mode, and is influenced by the elevator balance. The main effect is vertical accelerations of the airplane that may get out of hand if a coupling between the free elevator and airplane occur.

Proper design is essential here. Insofar as compressibility effects are concerned, the rearward movement of the aerodynamic center of the wing as the airplane goes supersonic is most evident. This condition increases the static stability to such an extent that the airplane may "tuck under" and be extremely stable in a steep dive.

One answer to this problem is to move the center of gravity rearward by a transfer of fuel as the airplane goes supersonic. Other solutions include the double-delta wing configuration or canards placed at the nose of the airplane to develop an additional nose-up moment due to lift in the transonic and supersonic range. (A moment is a measure of the body's tendency to turn about its center of gravity.) This arrangement has an added advantage of contributing to the airplane's lift.

The use of a canard for trim and a rear sailplane for control is beneficial. The canard would trim the rearward shift of the aerodynamic center at supersonic speeds and the strong nose-down moments from high-lift devices (flaps) at low speeds by providing uplift. When not used, the canard can be allowed to trail in the free stream at zero lift and also generate minimum drag.

Many of the basic ideas involving longitudinal stability also apply to directional stability. In the usual equilibrium condition, an airplane flies so that the yaw angle is zero.

To have static directional stability, a positive yawing moment should be generated if the airplane is disturbed to a negative yaw angle or alternatively by convention, a positive sideslip angle ß and a negative yawing moment generated for a negative sideslip angle excursion.

If the airplane holds its disturbed position, it has neutral directional stability. If the tendency is to increase the disturbed position, farther away from equilibrium, the airplane is directionally unstable.

4.4.1. Estimation of Aircraft Dynamical Characteristics

In this section we relate the dimensionless derivatives of the preceding section to the usual aerodynamic derivatives, and provide simple formulas for estimating them. It is natural to express the axial and normal force coefficients in terms of the lift and drag coefficients, but we must take into account the fact that perturbations in angle of attack will rotate the lift and drag vectors with respect to the body axes.

We define the angle of attack as the angle between the instantaneous vehicle velocity vector and the x-axis, and also assume that the propulsive thrust is aligned with the x-axis. we have to within terms linear in angle of attack

$$\mathbf{C}_X = \mathbf{C}_T - \mathbf{C}_D \cos\alpha + \mathbf{C}_L \sin\alpha \approx \mathbf{C}_T - \mathbf{C}_D + \mathbf{C}_L\alpha$$
$$\mathbf{C}_Z = -\mathbf{C}_D \sin\alpha - \mathbf{C}_L \cos\alpha \approx -\mathbf{C}_D\alpha - \mathbf{C}_L$$

4.4.2. Speed Derivatives

We first consider the derivatives with respect to vehicle speed *u*. The derivative Represents the speed damping, and

$$C_{Du} = M\frac{\partial Cd}{\partial M}$$

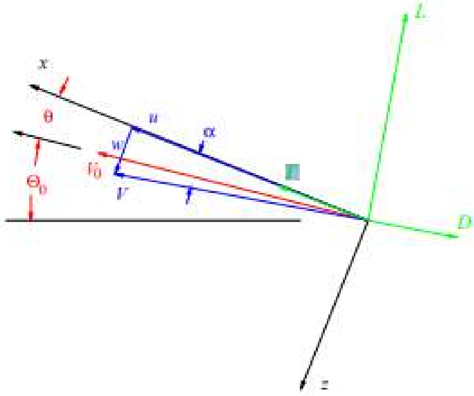


Figure 4.5.: Orientation of body axes with respect to instantaneous and equilibrium vehicle velocity

Orientation of body axes with respect to instantaneous and equilibrium vehicle velocity, illustrating relation between force components in body axes and lifts, drag, and thrust forces. The angle of attack α denotes the angle between the x-axis and the instantaneous velocity vector V, while the perturbation in pitch angle θ denotes the angle between the x-axis and the equilibrium velocity vector V0. Lift and drag act perpendicular to, and anti-parallel to, the instantaneous velocity, while thrust is assumed to act parallel to the x-axis.

4.5. HANDLING QUALITIES

We are continuing our lecture on how to model stick force and try to understand the importance of Hinge moment coefficients on designing a system, reversible control system for aircraft, so the stick force is well within the capability of the pilot, the pilot can fly at ease, right. So, if you recall before I come to stick force modeling let me write few statements.

We first introduce in simple terms what we mean by handling qualities, how we specify them, and how we achieve them. Then we introduce pilot-induced oscillations (PIOs) because they are one of the more serious consequences of failure to design for good handling qualities, and because current

PIO studies offer an interesting example of state-of-the-art research in handling qualities.
Handling qualities have been variously defined. For our purposes they are those characteristics of the dynamic behavior of the aircraft that allow precise control with low pilot workload. A major objective of flight control system design is to bestow good handling (or flying) qualities on the aircraft. Engineers, oblivious to the philosophical fact that measuring a quality transforms it into a quantity, define metrics for handling qualities.

Precision of flight can be quantified in terms of rounds on target for gun tracking, circular error probability for bombing or sink rate for landing, for example. Workload is more difficult to quantify, and for the time being we simply ask the pilot how easy or difficult his job is. Much of the achievement of handling-qualities practitioners has been in acquiring reliable information on pilot workload from pilots. Goodness is generally with reference to the pilot's comments in flight or in the flight simulator, and as summarized by Cooper-Harper ratings

4.6. COOPER HARPER SCALE

The Cooper-Harper Rating Scale is the current standard for evaluating aircraft handling qualities. It makes use of a decision tree that assesses adequacy for task, aircraft characteristics, and demands on the pilot to calculate and rate the handling qualities of an aircraft.

George Cooper's standardized system for rating an aircraft's flying qualities. Cooper developed his rating system over several years as a result of the need to quantify the pilot's judgment of an aircraft's handling in a fashion that could be used in the stability and control design process.

This came about because of his perception of the value that such a system would have, and because of the encouragement of his colleagues in this country and in England who were familiar with his initial attempts.

Cooper's approach forced a specific definition of the pilot's task and of its performance standards. Further, it accounted for the demands the aircraft placed on the pilot in accomplishing a given task to some specified degree of precision.

The Cooper Pilot Opinion Rating Scale was initially published in 1957. After several 27 years of experience gained in its application to many flight and simulator experiments and through its use by the military services and aircraft industry, it was subsequently modified in collaboration with Robert (Bob) Harper of the Cornell Aeronautical Laboratory and became the Cooper-Harper Handling Qualities Rating Scale in 1969.

This rating scale has been one of the enduring contributions of flying qualities research at Ames over the past 40 years; the scale remains as the standard way of measuring flying qualities to this day.

In recognition of his many contributions to aviation safety, Cooper received the Adm. Luis de Florez Flight Safety Award in 1966 and the Richard Hansford Burroughs, Jr., Test Pilot Award in 1971.

After he retired, both he and Bob Harper were selected by the American Institute of Aeronautics and Astronautics to reprise the Cooper-Harper Rating Scale in the 1984 Wright Brothers Lectureship in Aeronautics.

The traditional Cooper Harper (1969) scale has a decision tree format and was used to assess aircraft handling and control by the operator but could also be considered to assess workload.

The Modified Cooper Harper scale (MCH) was developed to be more appropriate in complex and automated systems where operators are not required to actively control systems but are more often monitoring, perceiving, evaluating and problem solving (Wierwille and Cascali 1983).

This also seems more relevant to train driving. Therefore the wording of the scale was replaced to represent activities relevant to such systems and to include task accomplishment, ability, errors, difficulty, performance and mental workload.

The scale also now focuses on assessing perception, cognition and communication. Validation has shown that the MCH is capable of indicating mental workload, and it is therefore appropriate for modern operator-machine systems.

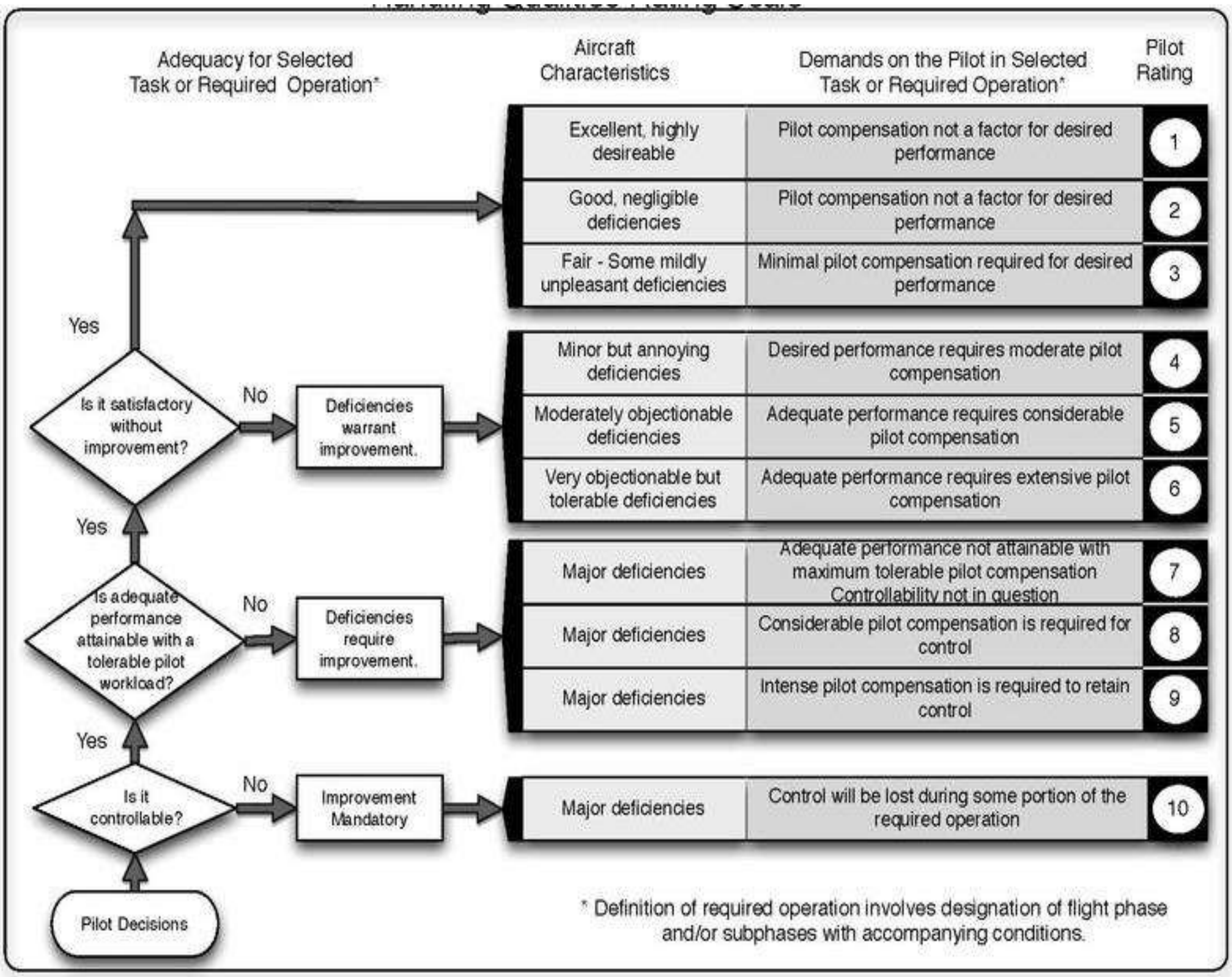


Figure 4.6: Handling qualities Rating scales

4.7. STEADY LEVEL FLIGHT, MINIMUM THRUST REQUIRED FOR LEVEL FLIGHT

During a steady climb the center of gravity of the airplane moves at a constant velocity along a straight line inclined to the horizontal at an angle γ The forces acting on the airplane are shown in Fig4.7.

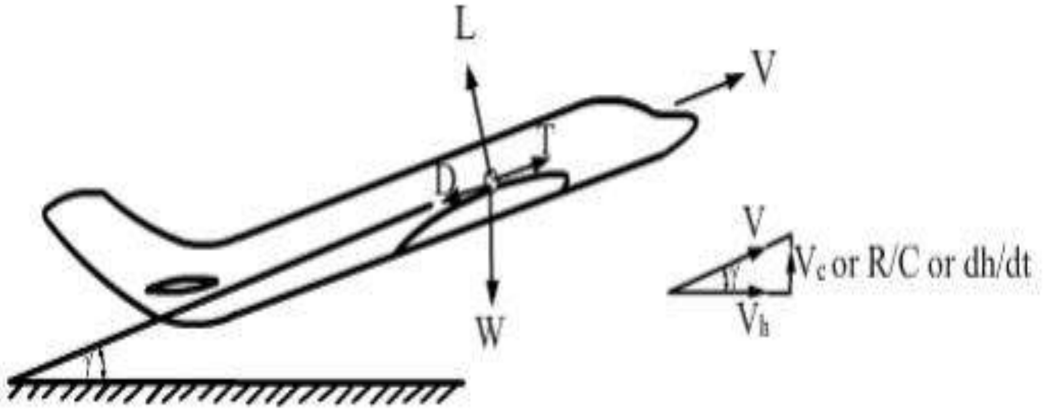


Fig 4.7. Steady climb

Since the flight is steady, the acceleration is zero and the equations of motion in climb can be obtained by resolving the forces along and perpendicular to the flight path and equating their sum to zero i.e.

$$T - D - W \sin \gamma = 0$$
$$L - W \cos \gamma = 0$$

$$\text{Hence, } \sin \gamma = (T\text{- } D / W)$$

From the velocity diagram in Fig. above, the vertical component of the flight velocity (V_c) is given by

$$V_c = V \sin \gamma = (T\text{- } D / W)\, V$$

4.8. THRUST AND POWER REQUIRED FOR A PRESCRIBED RATE OF CLIMB AT A GIVEN FLIGHT SPEED

Here it is assumed that the weight of the airplane (W), the wing area (S) and the drag polar are given. The thrust required and power required for a chosen rate of climb (V_c) at a given altitude (h) and flight speed (V) can be obtained, for a general case, by following the steps given below.

It may be pointed out that the lift and drag in climb are different from those in level flight. Hence, the quantities involved in the analysis of climb performance are, hereafter indicated by the suffix ‘c’ i.e. lift in climb is denoted by L_c

i) Since Vc and V are prescribed, calculate the angle of climb γ from:

$$\gamma = \sin^{-1} (V_c / V)$$

ii) The lift required in climb (L_c) is :

$$L_c = W \cos \gamma$$

iii) Calculate the lift coefficient in climb (CLc) as:

$$C_{Lc} = \frac{L_c}{\frac{1}{2}\rho V^2 S} = \frac{W \cos\gamma}{\frac{1}{2}\rho V^2 S}$$

iv) Obtain the flight Mach number; M = V/a ; a = speed of sound at the chosen altitude.

v) Corresponding to the values of CLc and M, obtain the drag coefficient in climb (CDC) from the drag polar. Hence, drag in climb (Dc) is given by:

$$D_c = (1/2\ \rho V^2\ S\ C_{DC})$$

UNIT-V
COST ESTIMATION, PARAMETRIC ANALYSIS, OPTIMISATION, REFINED SIZING AND TRADE STUDIES

5.1. ELEMENTS OF LIFE CYCLE COST, COST ESTIMATING METHOD

5.1.1. Cost Estimating:

The purpose of cost estimating is to forecast the cost of a project prior to its actual construction. Cost estimating is a method of approximating the probable cost of a project before its construction. The exact cost of a project is known after completion of the project. Cost estimate is prepared at various stages during the life of a project on the basis of the information available during the time of preparation of the estimate. Generally for any construction project, three parties are involved namely owner, design professionals and construction professionals.

In some cases the design professional and construction professional are from the same company or they form a team through a joint venture for providing service to the owner in the project. It is the responsibility of each party involved in the project to estimate the costs during various stages of the project. An early estimate helps the owner to decide whether the project is affordable within the available budget, while satisfying the project's objectives. For cost estimating, work breakdown structure (WBS) serves as an important framework for organized collection project cost data and preparing the cost estimates at different levels.

It is a technique that involves the hierarchical breakdown of the project into different work elements at successive levels and defines the interrelationships between them. For preparation of cost estimates, the estimator performs quantity take-off to quantify each item of work by reviewing the contract drawings and specifications. In cost estimation, quantity take-off is an important task that is carried out before pricing each item of work and quantities should be represented in standard units of measure.

Before bidding for a project, the estimator (along with his or her group) of a construction firm needs to determine the total cost of the project in accordance with contract documents consisting of drawings, specifications and all other technical documents and requirements. The total project cost consists of two components namely direct cost and indirect cost. Direct cost includes cost of materials; equipment and labor associated with each item of work and also includes cost of

5.1.2. Types of Estimates

There are different types of estimates which are prepared at various stages during the life of a project starting from the initial phases to its final phase on the basis of the available information at the time of preparation of the estimates. The range of expected accuracy is more in the estimates which are prepared during initial stages and it narrows down as the project progresses with the availability of more detailed information and increase in the level of project definition.

In addition to the project parameters, the degree accuracy of an estimate also depends on the experience, ability and judgment of the estimator. The construction cost estimate is broadly classified into two types approximate estimates and detailed estimates. The approximate estimates are prepared during initial stages of the project life cycle. These estimates are also known as preliminary, budget or order-of magnitude estimates and are prepared to determine the preliminary cost of the project. From the approximate cost estimates, the owner of the project can be able to know whether the project can be undertaken within the available budget.

There may be more than one design alternative for a project depending on the location, site conditions, type of structure etc. The estimator can determine the approximate cost of various alternatives for the project by taking preliminary design information from designer and can obtain the economical alternative that is affordable within the available budget for the project.

5.2. Estimates during conceptual planning:

This estimate is prepared at the very initial stage i.e. during conceptual planning stage of a project. It is based on little information and on broad parameters namely size of the project, location and job site conditions and the expected construction quality of project as a whole. The size of the project may be expressed in terms of its capacity namely number of rooms for a hostel, number of beds for a hospital, length (km) of a highway etc.

Owner of the project provides adequate input for defining scope of the project and this scope of the project forms the basis on which the conceptual estimate is prepared. This estimate is prepared to establish the preliminary budget of the project and accordingly project funding can be arranged. The degree of accuracy of this estimate is lowest among all the estimates those are prepared during various stages of a project.

5.2.1 Estimates during schematic design

During this phase of the project, the cost estimate is prepared on the basis of preliminary design information along with required schematic documents. The designer may incorporate different design alternatives and the cost estimate is prepared for these design alternatives by the estimators depending on the available information.

The cost estimates of different design alternatives are reviewed keeping in view the project scope and budget and the acceptable alternative(s) selected in this phase is analyzed in a detailed manner in the next phase of the project. This cost estimate is prepared by calculating the cost of major project elements by unit pricing from the available preliminary design information.

Subcontractors or material suppliers may be asked to furnish information while pricing the major project elements. As the cost estimate is prepared using preliminary design information, a contingency may be added in the estimate to accommodate for the unknown design details. With the improved scope of the project, the expected degree of accuracy in this estimate is more as compared to that in conceptual estimate.

5.2.2. Estimates during design development:

During design development phase of the project, the cost estimate is prepared on the basis of more detailed design information and schematic documents. With the improved level of information, the most of the major project items namely volume of earthwork (m^3), volume of concrete (m^3), weight of steel (tons) etc. can be quantified and the cost NPTEL – Civil Engineering – Construction Economics & Finance Joint initiative of IITs and IISc – Funded by MHRD Page 5 of 14 estimate is prepared using the known unit prices.

Detailed information from subcontractors or material suppliers should be obtained and used in pricing the major project items. During this phase, all the identified major systems of the project namely structural systems (reinforced concrete vs. structural steel), masonry (clay brick units vs. concrete masonry units), pile foundation (concrete pile vs. steel pile) etc. are priced and then cost of each system is compared with that obtained from past similar projects.

The project elements costing too high or too low as compared to past data should be reviewed and accordingly adjusted. With the availability of detailed design information and improved system definition, the expected degree of accuracy in this estimate is higher as compared to that in estimate prepared during schematic design phase of the project.

5.2.3. Estimates during procurement (i.e. estimates for construction of the project):

During this phase of the project, the cost estimate is prepared on the basis of complete set of contract documents that defines the project. The contractors bidding for the project prepare the cost estimate in accordance with contract documents by taking into consideration the estimated project duration. As already mentioned in the previous lecture (Lecture-1 of this module), the total cost of project can be divided into two categories namely direct cost and indirect cost.

Direct cost includes cost of materials, equipment and labor associated with each item of work and cost of subcontracted works. Indirect costs are the costs which are not attributed to each item of work and are calculated for the entire project and include overhead cost, contingency and profit. The owner team also prepares the cost estimate to check the accuracy of the bid prices quoted by the contractors and negotiate a reasonable price with the contractor. As this cost estimate is prepared in accordance with complete set of contract documents of the project, the degree of accuracy of this estimate is extremely high.

5.3. RDT&E AND PRODUCTION COSTS

1) RDT & E represent the cost towards research, development, test and evaluation of the airplane. It includes the cost of technology development and research, design engineering, prototype fabrication, flight and ground testing and evaluation of operational suitability. It also includes the cost of certification by the regulating agency. For military airplanes, it includes the cost associated with demonstration of mission capability and compliance with military specifications. RDT & E cost is fixed (non-recurring cost) regardless of how many airplanes are produced. However, the number of airplanes expected to be produced will decide how this cost is apportioned to each airplane.

2) The production cost includes the cost of labor and material to manufacture the airplane including airframe, engine, avionics and other systems. This cost, on one hand includes the tooling cost and on other hand the cost of material and labor for each airplane. The purchase cost of the civil airplane (civil purchase price) is arrived at based on expenses towards

(a) RDT & E,
(b) Production cost and
(c) Fair amount of profit.

5.4. OPERATION AND MAINTENANCE COSTS, COST MEASURES OF MERIT

In the case of military airplane the RDT & E cost may be paid by the government. Generally along with the airplane a certain amount of spares are also purchased which may amount to 10 -15% of the initial cost.

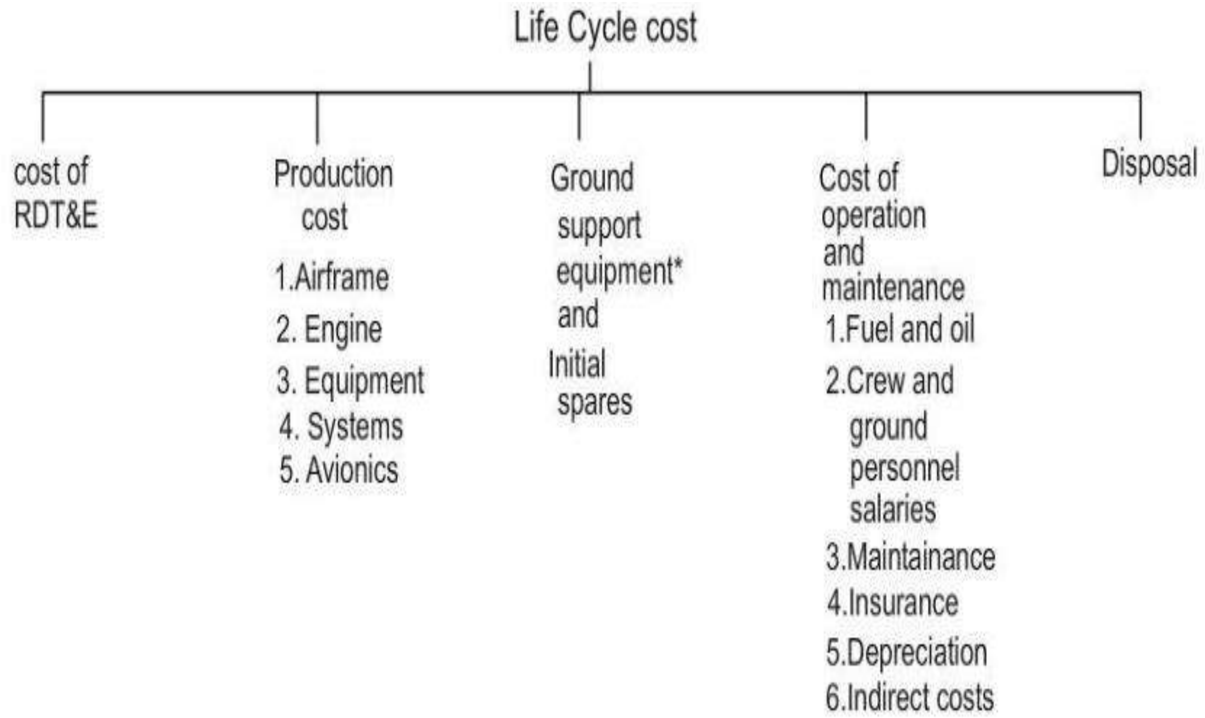


Fig 5.1.: Subdivisions of life cycle cost

5.4.1. Programme cost: The sum of the costs towards

(a) RDT & E,
(b) Production,
(c) Ground support equipment,
(d) Initial spares and
(e) Special construction, constitute the programme cost.

5.4.2. The operations and the maintenance costs include

(a) The costs of fuel and oil,
(b) Salaries of crew and ground personnel,
(c) Cost of maintenance,
(d) Insurance
(e) Depreciation for civil airplane and
(f) Indirect costs. (see remark 8 below)

5.4.3. Disposal cost:

For military aircraft the cost of taking to the disposal location is generally ignored. In the case of civil airplanes, the scrap value of the airplane is typically 10% of the purchase price.

For military airplanes a life time of 20 years is assumed and the cost of operation, maintenance and disposal is added to the program cost of the airplane. This constitutes the life cycle cost. This cost along with the performance of the airplane decides the choice of the airplane. Since the costs are closely related to the weight of the airplane, lighter airplanes are preferred.

The airline operators divide the cost into Direct Operating Cost (DOC) and Indirect Operating Cost (IOC).

The DOC is based on the expenses associated with the flying and maintenance of the airplane. Following these can be divided into

(a) Standing charges,

(b) Maintenance cost and

(c) Flight operational cost.

The maintenance cost includes cost towards airframe and engine maintenance and overheads. The flight operations cost includes

(a) Crew salaries,

(b) Fuel and oil cost and

(c) Airport fees.

5.4.4. Aircraft and airline economics

- Airlines Choose Flights to Maximize Profits
- Flights in a time period are ranked by profit
- Non-profitable flights are eliminated
- When resources are limited, least profitable flights are eliminated

  Low Profitability flights

  Low demand

  O/D linking to small communities

- Elastic markets – O/D linking to small communities
- Unpopular times of day (all links) with passengers with low airfares •
- Unpredictable fluctuations in demand

5.4.5. Model Airline Economics – Assumptions

Airline is rational economic agents

Select aircraft type and schedule to maximize profit

– Profit maximized across network (e.g. connecting

Profit = Revenue – Costs

- Revenue = (Airfare + Fees) * Pax
- Costs = (Non-fuel Costs Rate * Block-hours) + (Fuel Burn Rate * Fuel Price * Block Hours)

– Revenue is driven by (economic health sensitive) demand

– Cost is driven by fuel prices, labor, aircraft performance (i.e. technology)

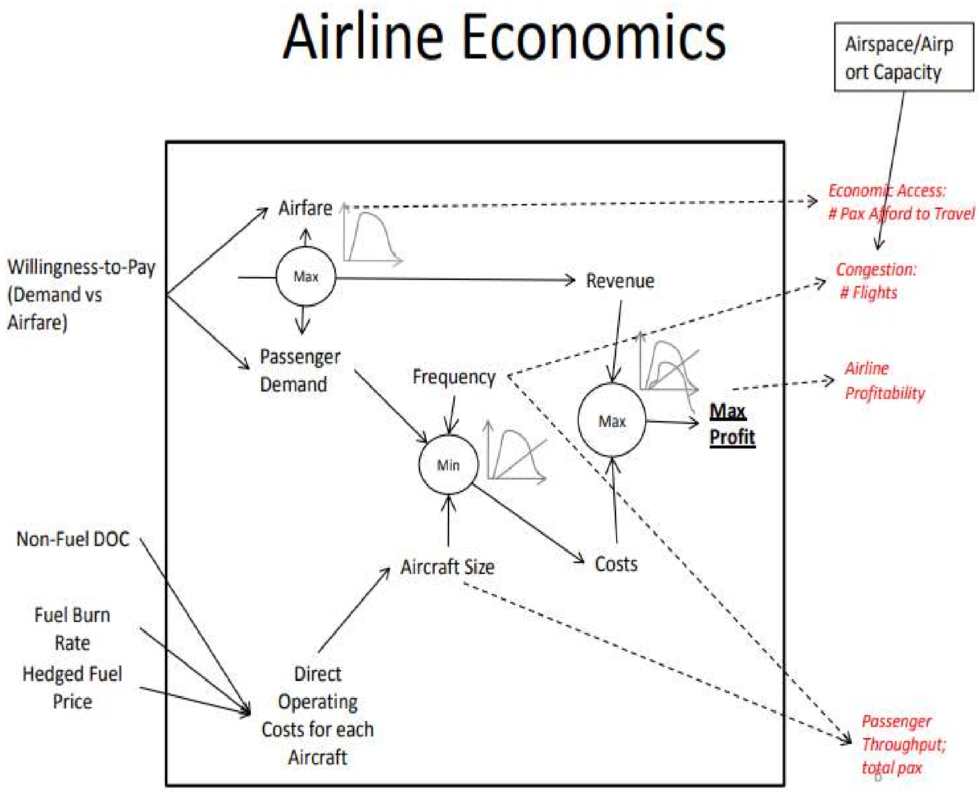


Figure 5.2: Airline Economics

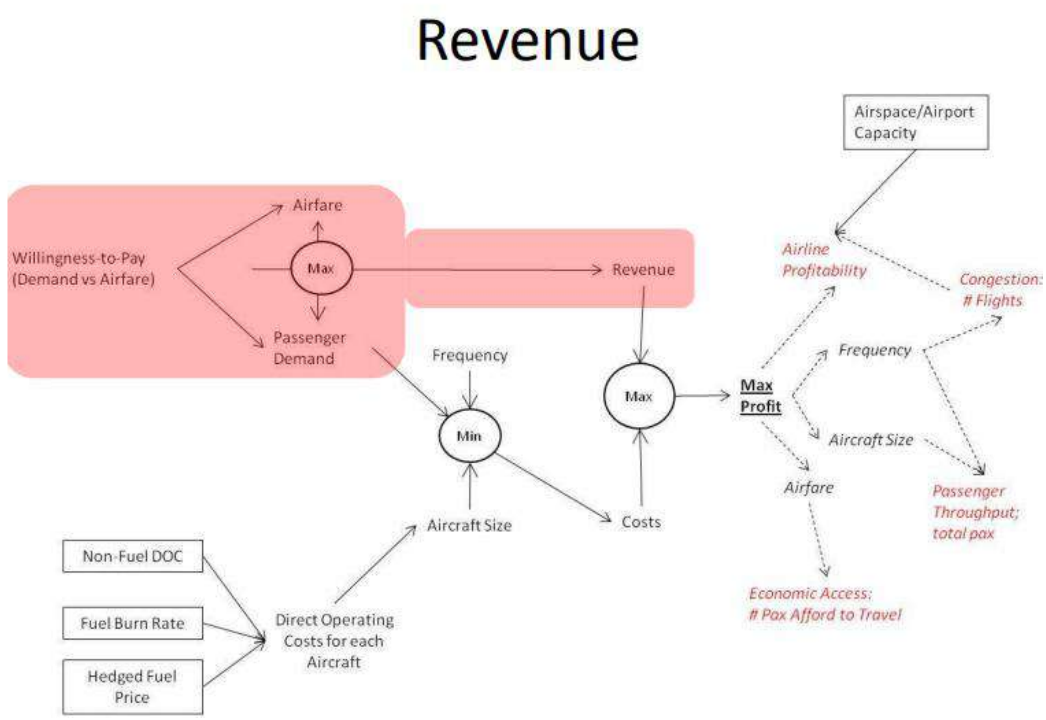


Figure 5.3. : Revenue of airline economics

5.5. DOC AND IOC, AIRLINE REVENUE, BREAKEVEN ANALYSIS, INVESTMENT COST ANALYSIS

- Direct Operating Cost (DOC)
- Indirect Operating Cost (IOC)

5.5.1. Direct Operating Cost (DOC):

The DOC is based on the expenses associated with the flying and maintenance of the airplane. Following these can be divided into (a) standing charges, (b) maintenance cost and (c) flight operational cost. The standing charges include cost of depreciation, insurance and loan repayment. The maintenance cost includes cost towards airframe and engine maintenance and overheads.

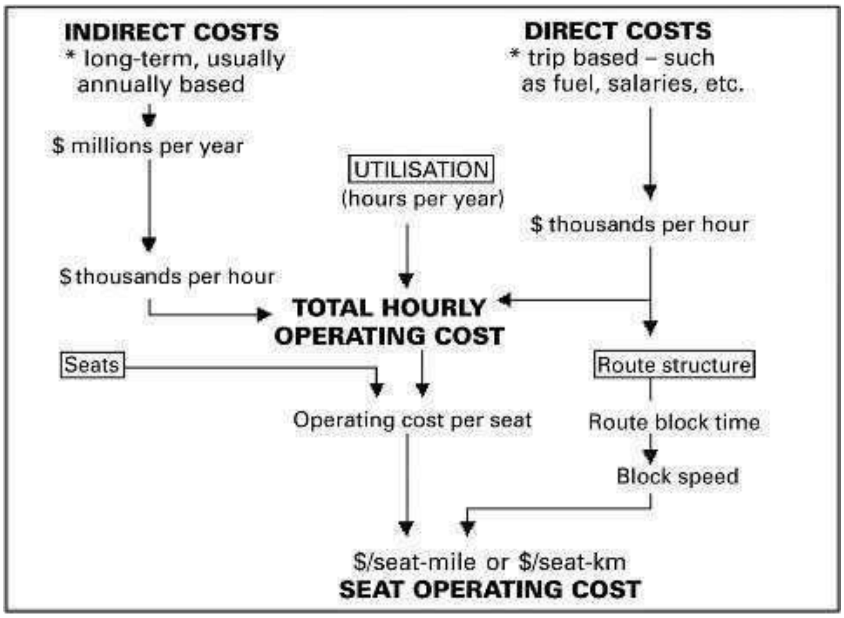


Figure 5.4. : DOC AND IOC

The flight operations cost includes
(a) Crew salaries,
(b) Fuel and oil cost and
(c) Airport fees.

The standing charges are calculated on yearly basis. These are divided by the utilization of the aircraft per year (around 4000 hours). This gives standing cost per flying hour. Based on the experience of the airline the cost of the engine and airframe maintenance (labor and material) are worked out and allocated per hour of flight. The flying cost depends on the flight duration.

The cost is worked out and divided by the flight time in hours. Adding the standing charges, maintenance costs and flight cost per hour gives the total DOC per hour. The stage cost is the cost of flying on a particular route. It is obtained by multiplying DOC per hour by the block time for the flight. The mile cost (or km cost) is obtained by dividing the stage cost by the block distance of the stage.

When the mile cost (km cost) is divided by the maximum number of seats on the airplane, it gives the seat mile (or km) cost. It is expressed as cents per seat-km. This cost is an index of the efficiency of the design.

5.5.2. The Indirect Operating Cost (IOC)

The Indirect Operating Cost is not dependent on the number of hours flown by the airplane. It includes

(a) Cost towards depreciation and maintenance on the ground equipment,
(b) Cost of administrative, technical and customer services,
(c) Advertising, promotion and sales, and
(d) Training IOC is not an insignificant cost. It could be as high as DOC.

Adding IOC and DOC the airline obtains the total operating cost. The revenue from the fraction of the total number of seats which will cover the operating cost is called breakeven load factor. i.e. if on a hundred seated airplane, the revenue from 55 passengers would cover the operating cost then the breakeven load factor is 0.55. The profit of the airline depends on attracting more number of passengers than the breakeven load factor.

5.6. PARAMETRIC ANALYSIS AND OPTIMIZATION

Parametric cost estimate is prepared during early stages of project development. The cost estimate is based on various parameters which define characteristics of the project and includes physical attributes such as size, capacity, weight etc. The parametric cost estimate is based on cost-estimating relationships those use past cost data to obtain the current cost estimate. Cost estimating relationships are statistical models those relate the cost of a product or system to the physical attributes those define its characteristics. One of the most commonly used cost-estimating relationships is power-sizing model.

5.6.1. Power-sizing model is most commonly used for obtaining preliminary cost estimate of industrial plants and equipment. The power sizing model relates the cost of a plant or system to its capacity or size and uses the following relationship for cost estimate

5.6.2. Parametric cost estimate is prepared during early stages of project development. The cost estimate is based on various parameters which define characteristics of the project and includes physical attributes such as size, capacity, weight etc. The parametric cost estimate is based on cost-estimating relationships those use past cost data to obtain the current cost estimate. Cost estimating relationships are statistical models those relate the cost of a product or system to the physical attributes those define its characteristics. One of the most commonly used cost-estimating relationships is power-sizing model.

5.7. IMPROVED CONCEPTUAL SIZING METHODS

Sizing matrix plot and carpet plot, trade studies, At the end of the preliminary design phase, a configuration of the airplane is available. This configuration is called base line configuration. Using this configuration the revised drag polar, fuel required and gross weight can be obtained. Further, parametric studies by varying certain parameters can be carried out to get near optimum shape/parameters. Such studies are called sizing and trade studies.

These studies are carried out by varying the parameter (s) around the values chosen in the baseline configuration. Carrying out such studies here, is outside the scope of the present introductory course material on airplane design.

5.8. DESIGN TRADES

Economic optimization is the process of finding the condition that maximizes financial return or, conversely, minimizes expenses. The factors affecting the economic performance of the design include the types of processing technique and equipment used, arrangement, and sequencing of the processing equipment, and the actual physical parameters for the equipment. The operating conditions are also of prime concern.

Optimization of process design follows the general outline below:

1. Establish optimization criteria: using an objective function that is an economic performance measure.
2. Define optimization problem: establish various mathematical relations and limitations that describe the aspects of the design
3. Design a process model with appropriate cost and economic data

5.8.1. Requirement trades

A part of optimization is assessing trade-offs; usually getting better performance from equipment means higher cost. The objective function must capture this trade-off between cost and benefit. Example: Heat Recovery, total cost captures trade-off between energy savings and capital expense.

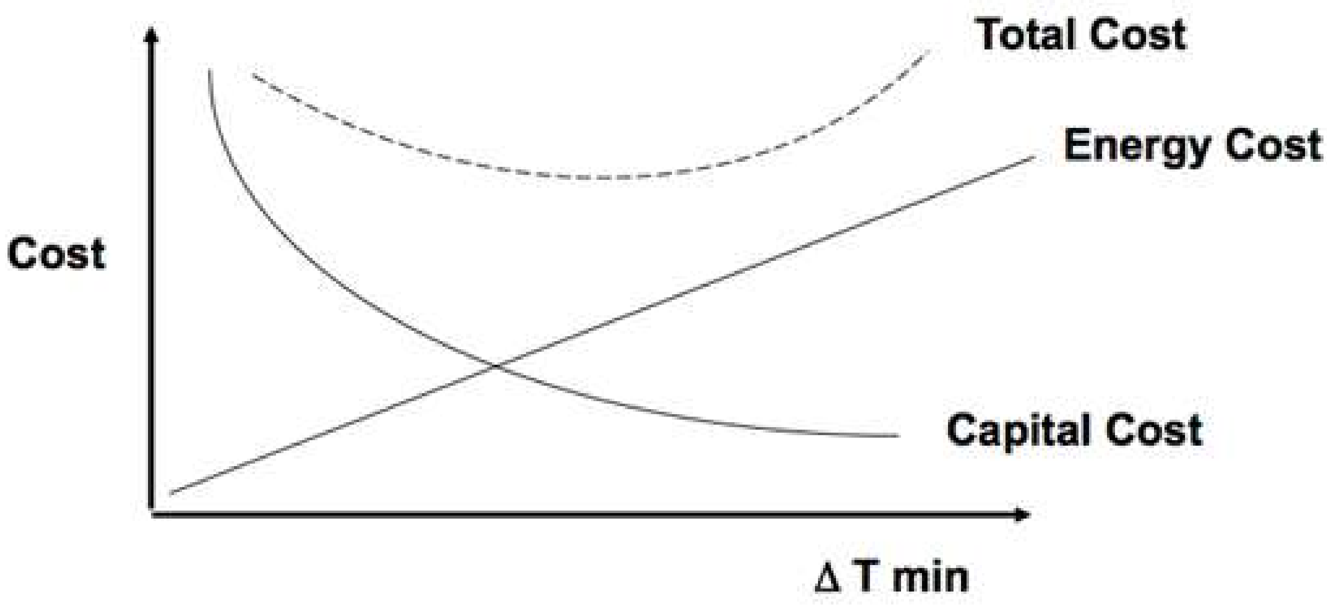


Fig 5.5. : Trade-off example

Some common design trade-offs are more separations equipment versus low product purity, more recycle costs versus increased feed use and increased waste, more heat recovery versus cheaper heat exchange system, and marketable by-product versus more plant expense.

5.8.2. Growth sensitivities

Cases, considering the dropouts in the active group to be failure and the dropouts in the placebo group to be success might be necessary.

5.8.3. Multivariable design optimization methods

If a convex function is to be minimized, the stationary point is the global minimum and analysis is relatively straightforward as discussed earlier. A similar situation exists for maximizing a concave variable function. The necessary and sufficient conditions for the optimization of unconstrained function of several variables are given below.

5.9. CASE STUDIES ON DESIGN OF DC-3

5.9.1. Purpose

The purpose of this exercise is to analyze the unique design of the DC-3 which as the first profit-making passenger aircraft revolutionized commercial aviation. So successful was the DC-3 that it served as the prototype for the design of all commercial propeller-driven aircraft, only to be superseded by the jets. It is still being used on airlines in the United States and abroad.

The DC-3 provides an excellent example of the machine as a solution to a series of problems that were confronting the budding aircraft industry in America during the 1930s. For this reason, the DC-3 should not be presented as the artifact to be studied at the outset of the lesson. It would be more instructive to introduce students to the design features of the DC-3's predecessors: the Fokker and Ford Trimotors and the Boeing 247. Each, though successful in its own way, presented problems to be overcome to make a passenger-carrying aircraft financially successful.

There are at least two ways of studying the DC-3's origins. First, it may be viewed as a logical solution to a series of design problems. It may also be considered as a case study of the inter-relationship between the aircraft industry, the airline industry and the federal government. In fact, both the engineering and entrepreneurial issues were related.

5.9.2. The Problem

The DC-3 can be viewed as a solution to the problems presented by its predecessors. The Ford Trimotor will serve as an excellent artifact for this study. It is important that the student be aware of the design features of this aircraft. While a scale model would be preferable, photographs should more than suffice.

First, the students should be presented with the five essential elements in the design of a successful commercial aircraft. It would be important for the student to be apprised of the market for air travel in the early 1930s as well as the nature of the competition.
The five essential elements are**:**

1. speed
2. reliability
3. comfort
4. safety
5. profit

Needless to say, rather than being presented in class, these five elements could be easily derived by discussion.

The Ford Trimotor, one of the most popular commercial aircraft of the late 20s and early 30s was never profitable as a commercial venture in carrying passengers. Moreover, it failed to meet the requirements of the potential airline passenger; it was slow, noisy, had a limited range, and vibrated mercilessly.

The following are among the most notable of the Trimotor's design features:
•a corrugated aluminum fuselage and wing
•large, non-retractable landing gear
•3 uncovered engines--one mounted at the front of the fuselage, the other two suspended from the wings.

Each of these features should not be presented as "givens," but as solutions to prior problems confronted in aircraft design. The corrugated aluminum wing was a solution to the structural failure of wings built of lacquered fabric stretched over wood. This faulty design was found to be the probable cause of a tragic crash of a Fokker Trimotor in 1931 which took the life of Notre Dame coach Knute Rockne and did much to increase the public's fear of flying.

The cumbersome fixed landing gear was eminently suitable for a plane of short range which often landed on rocky fields and pastures. Furthermore, at the Trimotor's slow speeds the fixed gear presented negligible aerodynamic resistance.

The three-engine design was a response to a concern for reliability, safety and lack of engine power. (The issue of two or three engines came to prominence recently when the FAA granted permission to allow the Boeing 767, a two-engine aircraft, to fly the transatlantic route. The issue was whether the plane could safely make it to an airfield if it lost an engine over water mid-route.

The deciding factor was the record of reliability earned by the latest jet engines.) Fallible as the Trimotor's engines were, they were a solution to a problem which had only been solved a decade earlier: how to get maximum engine power without making the aircraft too heavy to fly to profitably carry freight or passengers. Charles Lindbergh's plane was one of the first to incorporate a "solution"--the development of an air-cooled airplane engine rather than a much bulkier one cooled by water.

5.9.3. The Solution

While the corrugated aluminum fuselage was an improvement over the fabric wing, it too was subject to structural failure. In general, it tended to be very strong in one direction, and weak in another. The solution as developed by the German engineer Adolph Rohrbach was to design a wing of stretched aluminum sheeting over a rib-and-spar framework incorporating a "honeycomb" design. This new design in the DC-3 was subjected to sophisticated stress-testing by driving a steamroller back-and-forth over the wing.

Much of the concern in designing the DC-3 involved an effort to increase the range and speed over its predecessors. The demand was for an aircraft to profitably carry passengers from New York to Chicago.

This was accomplished not only by designing more efficient engines, but by streamlining the plane. In fact, the original plan for the DC-l, the prototype of the DC-3, called for three engines. It was decided that it was the engines themselves which were responsible for much of the aerodynamic drag on the Trimotor.

The solution incorporated into the DC-3 was one that had been devised nearly fifteen years earlier by a man named Northrup in designing the Lockheed Vega--a single-engine plane. For streamlining, Northrup covered the engine with a metal cowling.

In addition, the Douglas engineers decided to incorporate the engines into the wing mounted on nacelles rather than suspend them beneath the wing. Further incorporated into the design was the principle of quickly demountable engines; that is, engines could be quickly removed and replaced, which was seen to be much more efficient in keeping an aircraft serviceable than having to service an engine permanently mounted on the wing, thereby delaying the entire plane. (While virtually all modern-day commercial jet aircraft do not incorporate the engine into the wing, the prototype of the modern passenger jet, the Comet, did.)

Finally, it was realized that it would be essential to eliminate the drag produced by the Trimotor's landing gear. This was accomplished in the DC-3 by devising retractable landing gear. In its initial form the gear was manually retracted, and then only partially--the wheels were left partly protruding from the engine nacelles. This turned out to be fortunate when in an early test flight of the DC-2 the pilot neglected to lower the gear prior to landing. The partially protruding gear protected the belly of the plane and only the props suffered irreparable damage.

Many of the main features of the DC-3 can be seen as solutions to problems encountered in the Ford Trimotor. It can be demonstrated that virtually all of its features were responses to some problem. For example, the unique and characteristic swept wing of the DC-3 was an attempt to keep the wing in its original location as the center of gravity moved toward the rear. The trailing flaps first employed on the wing were an effort to slow down the plane from 200 mph to under 65 mph in order to land on existing airstrips.

An alternative or additional way of studying the DC-3 is to examine it as a product of the interaction between the budding aviation and aircraft industries as well as government intervention. Commercial aviation in America pursued a course of development distinct from that in Europe. In post-World War I Europe, commercial aviation was seen as an essential means of compensating for the massive destruction of the railroads.

In addition, distances between commercial centers were closer than in the United States. The London-Paris route was one of the most traveled in commercial aviation's infancy--a distance easily manageable by post-war aircraft. In America it was the New York to Chicago route which was deemed to be the largest market worth exploiting.

Lindbergh's fame for his historic 1927 transatlantic flight has overshadowed the significant role he played in the development of the DC-3. When Jack Frye organized TWA (then Transcontinental and Western Airlines) he hired Lindbergh as a consultant. As a result, the motto "the Lindbergh line" was emblazoned on the fuselages of Frye's aircraft.

The DC-l was designed in response to Frye's demand for an aircraft that could comfortably carry l2 passengers at l50 miles an hour from New York to Chicago and was capable of landing on existing airstrips. In fact, Frye's initial preference was for a trimotor. But it was Lindbergh who set what the

Douglas engineers found to be the most stringent requirement:
"This ship should be able to take off with a full load from any airport on the TWA route--on one engine!"

The prototype of the DC-3, the DC-1 was designed with this specification in mind, although the engineers were uncertain about the plane's ability to perform such a feat until a successful flight test.

The DC-3 was a solution to a problem presented to Douglas by C.R. Smith, the president of American Airlines. He wanted Douglas to modify the DC-2 to carry passengers overnight in berths. In order to do this profitably the fuselage would have to be lengthened to accommodate more passengers.

This new plane, originally designated the DST (Douglas Sleeper Transport) saw little service as a sleeper when its economic potential as a traditional transport was recognized. Its ability to carry 21 passengers in comfortably configured seating lowered the cost per seat mile to assure the economic success of the DC-3.

5.9.4. A word on modeling

In lieu of a visit to the Air and Space Museum in Washington, viewing or handling scale models of aircraft would no doubt prove useful in acquainting students with the structural characteristics of various aircraft. There does appear to be one shortcoming involved in this methodology: the way in which plastic models are divided in half for building by the hobbyist bears little or no relation to the was the aircraft is actually assembled and can be very misleading.

The innocent student could justifiably assume that the DC-3 was constructed by connecting fuselage halves and inserting the wings to the main fuselage. In fact, the structural integrity of the DC-3 was assured by constructing the fuselage and wing stubs as one unit. The outer sections of the wing were bolted on later and designed to be easily replaced if damaged.

In fact, the fuselage of the aircraft may be viewed as a series of segments connected one in front of or behind the other. Douglas engineers profited from this experience in "stretching" the DC-2 to the DC-3 when in later years they kept modifying the DC-8 jet by "stretching" it. Boeing essentially forfeited this option by designing the 707 with such a low profile that stretching it would have proved prohibitive--the tail scraping the ground on takeoff.

5.9.5. Conclusion

It is the thesis of this abstract that while it is perfectly valid to "read" a machine, it might be exciting for students to take this premise one step further--to "read" machines, define their problems, and be encouraged to "solve" them to produce new machines. Indeed, there appears to be one pitfall inherent in this approach.

It would be too easy with the advantage of hindsight to lead students to the DC-3 as the only possible solution to the problems encountered in the Fokker and Ford Trimotors. We are perhaps fortunate in that most of today's students are ignorant of the DC-3. Alternative solutions should be encouraged. Certainly, such an approach would be no less valid than the laboratory course

As each new generation of wide-body jet aircraft has placed enormous financial pressure on Boeing (including, but not limited to up-and-down earnings results, increased debt burden and a cyclical stock price) thereby significantly increasing the risk of corporate financial failure. Yet despite a few close calls, Boeing has emerged successfully from each cycle, enabling it to maintain industry leadership and generate satisfactory long-term financial returns.

In this, as part of our ongoing research on risk-taking behavior we will focus on the 707 model that formed the template for Boeing's "betting it all" corporate strategy associated with the launch of new generations of wide-body aircraft over the ensuing decades.

5.10.1. Introduction

Now the world's largest aerospace company, Boeing was founded in 1916 by William E. Boeing in Seattle, Washington.

The company is composed of multiple business units: Boeing Commercial Airplanes (BCA); Boeing Defense, Space & Security (BDS); Engineering, Operations & Technology; Boeing Capital; and Boeing Shared Services Group. As top U.S. exporter, the company supports airlines and U.S. and allied government customers in 150 countries.

Boeing's products and tailored services include commercial and military aircraft, satellites, weapons, electronic and defense systems, launch systems, advanced information and communication systems, and performance-based logistics and training.

5.10.2. Boeing Commercial Airplanes

Boeing has been the premier manufacturer of commercial jetliners for over 40 years. Today, their main commercial products are the 737, 747, 767 and 777 families of airplanes and the Boeing Business Jet. New product development efforts are focused on the Boeing 787 Dreamliner, and the 747-8. The company has nearly 12,000 commercial jetliners in service worldwide, which is roughly 75 percent of the world fleet.

Through Boeing Commercial Aviation Services, the company provides round-the-clock technical support to help operators maintain airplanes in peak operating condition.

Commercial Aviation Services offers a full range of world-class engineering, modification, logistics and information services to its global customer base, which includes the world's passenger and cargo airlines, as well as maintenance, repair and overhaul facilities.

5.10.3. Capital Investment Decisions

Capital investment decisions at Boeing are unique and—to some degree—risky. For example, in the mid-1950s, despite failing to profit on civilian planes in two decades, Boeing spent $185 million to develop the first American all-jet transport, the 707.

Despite not having made money in a non-military plane in twenty years, Boeing spent $185 million to develop the 707, the first American all-jet transport.

To put this in context, this capital investment was $36 million or 25% more than Boeing's total net worth of $149 million in 1956.

5.10.4. Conclusions

Boeing's "bet the company strategy" appears to have successively increased earnings power (measured by Net Income) with each generation of new commercial jet aircraft.

Each new revolutionary jet aircraft program eventually is the primary driver in raising total Net Income by several-fold (versus the cycle immediately prior to it).

The 707 led to a 2.5x increase in Net Income (late 1950s/early 1960s versus mid-1950s) and by 1967-68, Net Income was 2x higher than its 1961 level. The 747 helped Boeing surpass the 1967-68 peak by a factor of 7-times by 1980, with the 767 and 777 programs leading to an eventual 7-fold improvement in Net Income by 2011 versus 1980.

Boeing's success in commercial jet aircraft stemmed from its military aircraft business in terms of *risk sharing* (e.g., 707 and its military KC-135 version) and *diversification.*

The strong position in defense-related projects (e.g., Minuteman and Cruise missiles) provided stable, steady cash flow for the entire corporation thereby providing an additional financial cushion to undertake development of new generations of jet aircraft.

The acquisition of the largest US military contractor, McDonnell Douglas, further strengthened Boeing's corporate business portfolio in terms of earnings, cash flow and diversification.

Cyclical, financial and execution risks remain perennially relevant for the commercial jet aircraft business. However, Boeing has a proven performance record of being able to maintain its market leadership and increasing its earning power with each generation of new aircraft.

This includes, but is not limited to, accommodating unique customer demand requirements on a global scale, rationalization in down cycles, improvement of assembly and manufacturing processes and either buying out (e.g., acquiring McDonnell Douglas) or driving out (e.g., Lockheed) its major US commercial jet aircraft competitors.

While Boeing's stock price has been cyclical, investors have learned to be patient every time the company undertakes a bigger bet when launching a new generation of aircraft. In general, as shown in Exhibit II the price of stocks have been more in line with future orders rather than net profit.

5.11. GENERAL DYNAMICS F-16

Figure 5.6. F-16 Fighting Falcon

The General Dynamics F-16 Fighting Falcon is a single-engine supersonic multirole fighter aircraft originally developed by General Dynamics (now Lockheed Martin) for the United States Air Force(USAF).

Designed as an air superiority day fighter, it evolved into a successful all-weather multirole aircraft. Over 4,600 aircraft have been built since production was approved in 1976. Although no longer being purchased by the U.S.

Air Force, improved versions are being built for export customers. In 1993, General Dynamics sold its aircraft manufacturing business to the Lockheed Corporation, which in turn became part of Lockheed Martin after a 1995 merger with Martin Marietta.

The Fighting Falcon's key features include a frameless bubble canopy for better visibility, side-mounted control stick to ease control while maneuvering, an ejection seat reclined 30 degrees from vertical to reduce the effect of g-forces on the pilot, and the first use of a relaxed static stability/fly-by-wire flight control system which helps to make it an agile aircraft.

The F-16 has an internal M61 Vulcan cannon and 11 locations for mounting weapons and other mission equipment.

The F-16's official name is "Fighting Falcon", but "Viper" is commonly used by its pilots and crews, due to a perceived resemblance to a viper snake as well as the Colonial Viper starfighter on Battlestar Galactica which aired at the time the F-16 entered service.

Of the 40 F-16 fighter-bomber aircraft originally acquired by Pakistan, 32 remain in service in 3 squadrons. Pakistan has 71 additional F-16s on order, but delivery has been suspended since 1990 by the United States.

In early 1994 the Clinton Administration initiated consultations with Congress concerning a proposed one-time sale of F-16 fighter aircraft to Pakistan.

Delivery of the planes would be contingent on specific commitments from Pakistan regarding its nuclear program, including a verifiable cap on the production of fissile materials.

When Pakistan declined this proposal, in September 1995 the Clinton Administration proposed revisions to the Pressler Amendment to facilitate cooperation with Pakistan n areas such as combatting terrorism and furthering US commercial interests in Pakistan.

The US would not deliver the controversial F-16 aircraft or resume an official military supply relationship with Pakistan, but the President decided to sell the F-16 aircraft to other countries and return the proceeds to Pakistan.

Pakistan could use the F-16 bombers to drop nuclear weapons on visually acquired targets by improvising the necessary electronic wiring which is omitted from these export models.

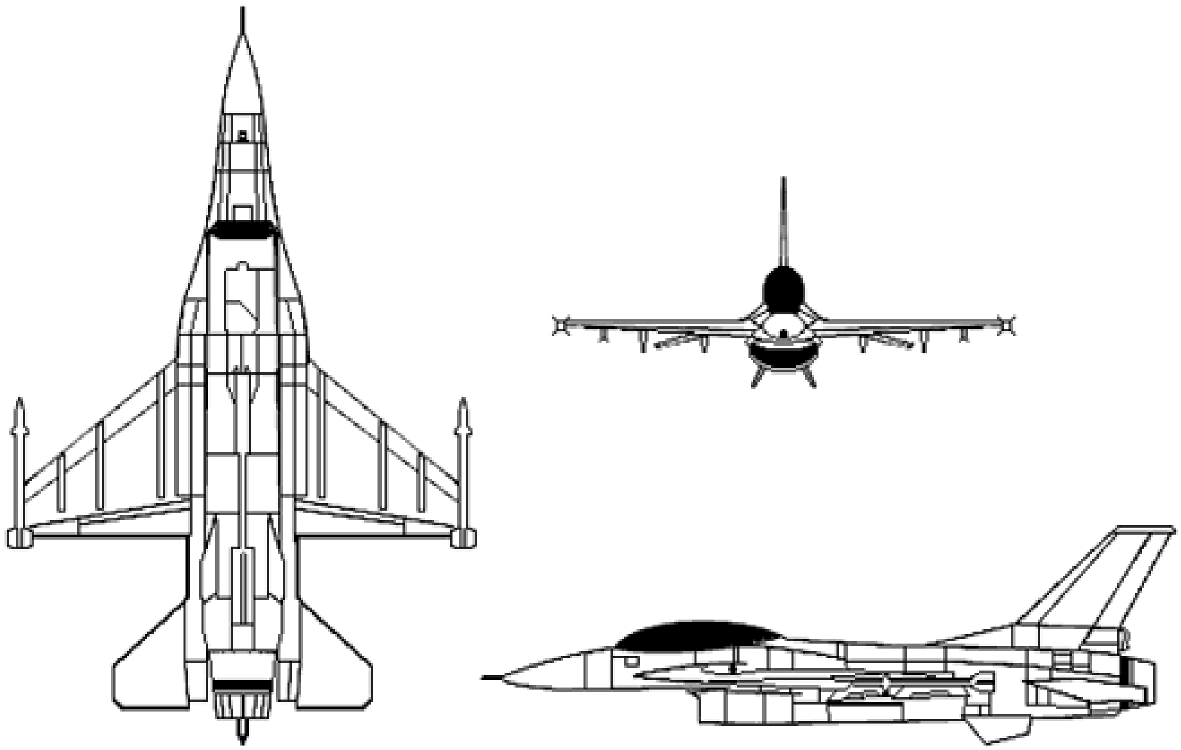


Figure 5.7: views of F-16 Fighting Falcon

5.11.1. General Data

Country of Origin. USA.
Similar Aircraft. F/A-18 Hornet, MiG-29 Fulcrum, Mirage F1.
Crew. One; F-16B--two.
Role. Multirole ground-attack/fighter.
Armament. Cannon, missiles, bombs.
Dimensions. Length: 47 ft, 8 in (14.54 m). Span: 31 ft (9.46 m).

5.11.2. Description

- Wings Mid-mounted, delta-shaped.
- Missiles are normally mounted at the wing tips.
- Engine(s). One in body.
- Oval air intake under the center of the fuselage.
- Single exhaust. Fuselage.
- Long, slender body, widens at air intake.
- Pointed nose.
- Bubble canopy.
- Tail.
- Swept-back, tapered fin with square tip.
- Flats mid-mounted on the fuselage, delta-shaped with square tips, and a slight negative slant.
- Two belly fins.

5.11.3. User Countries

Bahrain, Belgium, Denmark, Egypt, Greece, Indonesia, Israel, Netherlands, Norway, Pakistan, South Korea, Portugal, Singapore, Taiwan, Thailand, Turkey, USA, Venezuela. The fuselage of the F-16 flares out at its juncture with the aluminum-alloy wings, giving the aircraft greater lift and stability at steep angles of attack.

A computerized “fly-by-wire” stabilizing system issues continuous commands to control surfaces in the tail and wings, and a “heads-up-display” instrumentation system projects flying and combat data onto a transparent screen in front of the pilot. In addition, a highly sophisticated bomb-aiming system, using a laser range-finder and high-speed digital data processing, permits ordinary “dumb” bombs to be dropped with precision accuracy from low altitudes. Such structural and electronic innovations made the F-16 a highly capable and versatile aircraft.

It has been built under license in Belgium, the Netherlands, Turkey, and South Korea and is the basis for Japan’s FS-X fighter. It has been sold to U.S. allies in the Middle East, where it proved very effective in air-to-air combat and ground attack in the Israeli-Syrian conflict of 1982 and in the Persian Gulf War of 1990–91.

5.12. SR-71 BLACKBIRD

The Lockheed SR-71 "Blackbird" is a long-range, Mach 3+ strategic reconnaissance aircraft that was operated by the United States Air Force. It was developed as a black project from the Lockheed A-12 reconnaissance aircraft in the 1960s by Lockheed and its Skunk Works division.

American aerospace engineer Clarence "Kelly" Johnson was responsible for many of the design's innovative concepts. During aerial reconnaissance missions, the SR-71 operated at high speeds and altitudes to allow it to outrace threats.

If a surface-to-air missile launch were detected, the standard evasive action was simply to accelerate and outfly the missile. The shape of the SR-71 was based on the A-12 which was one of the first aircraft to be designed with a reduced radar cross-section.

Figure 5.8. SR-71 Blackbird

The SR-71 served with the U.S. Air Force from 1964 to 1998. A total of 32 aircraft were built; 12 were lost in accidents with none lost to enemy action. The SR-71 has been given several nicknames, including "Blackbird" and "Habu". Since 1976, it has held the world record for the fastest air-breathing manned aircraft, a record previously held by the related Lockheed YF-1.

The SR-71 designation is a continuation of the pre-1962 bomber series; the last aircraft built using the series was the XB-70 Valkyrie. However, a bomber variant of the Blackbird was briefly given the B-71 designator, which was retained when the type was changed to SR-71.

During the later stages of its testing, the B-70 was proposed for a reconnaissance/strike role, with an "RS-70" designation.

When the A-12 performance potential clearly was found to be much greater, the Air Force ordered a variant of the A-12 in December 1962. Originally named R-12 by Lockheed, the Air Force version was longer and heavier than the A-12, with a longer fuselage to hold more fuel, two seats in the cockpit, and reshaped chines.

Reconnaissance equipment included signals intelligence sensors, a side-looking airborne radar, and a photo camera.

The CIA's A-12 was a better photo-reconnaissance platform than the Air Force's R-12, since the A-12 flew somewhat higher and faster, and with only one pilot, it had room to carry a superior camera and more instruments.

During the 1964 campaign, Republican presidential nominee Barry Goldwater repeatedly criticized President Lyndon B. Johnson and his administration for falling behind the Soviet Union in developing new weapons.

Johnson decided to counter this criticism by revealing the existence of the YF-12A Air Force interceptor, which also served as cover for the still-secret A-12 and the Air Force reconnaissance model since July 1964.

Air Force Chief of Staff General Curtis LeMay preferred the SR (Strategic Reconnaissance) designation and wanted the RS-71 to be named SR-71. Before the July speech, LeMay lobbied to modify Johnson's speech to read SR-71 instead of RS-71.

The media transcript given to the press at the time still had the earlier RS-71 designation in places, creating the story that the president had misread the aircraft's designation. Johnson only referred to the A-11 to conceal the A-12, while revealing that there was a high speed, high altitude reconnaissance aircraft.

In 1968, Secretary of Defense Robert McNamara cancelled the F-12 interceptor program; the specialized tooling used to manufacture both the YF-12 and the SR-71 was also ordered destroyed.[Production of the SR-71 totaled 32 aircraft with 29 SR-71As, two SR-71Bs, and the single SR-71C

5.13. NORTHROP-GRUMMAN B-2 STEALTH BOMBER.

Figure 5.9: Northrop (later Northrop Grumman) B-2 Spirit

The Northrop (later Northrop Grumman) B-2 Spirit, also known as the Stealth Bomber, is an American heavy strategic bomber, featuring low observable stealth technology designed for penetrating dense anti-aircraft defenses; it is a flying wing design with a crew of two. The bomber can deploy both conventional and thermonuclear weapons, such as up to eighty 500-pound class (230 kg) Mk 82 JDAM Global Positioning System-guided bombs, or sixteen 2,400-pound (1,100 kg) B83 nuclear bombs. The B-2 is the only acknowledged aircraft that can carry large surface standoff in a stealth configuration.

Development started under the "Advanced Technology Bomber" (ATB) project during the Carter administration; its expected performance was one of his reasons for the cancellation of the Mach 2 capable B-1A bomber.

The ATB project continued during the Reagan administration, but worries about delays in its introduction led to the reinstatement of the B-1 program. Program costs rose throughout development. Designed and manufactured by Northrop, later Northrop Grumman, the cost of each aircraft averaged US$737 million (in 1997 dollars). Total procurement costs averaged $929 million per aircraft, which includes spare parts, equipment, retrofitting, and software support. The total program cost, which included development, engineering and testing, averaged $2.1 billion per aircraft in 1997.

Because of its considerable capital and operating costs, the project was controversial in the U.S. Congress. The winding-down of the Cold War in the latter portion of the 1980s dramatically reduced the need for the aircraft, which was designed with the intention of penetrating Soviet airspace and attacking high-value targets. During the late 1980s and 1990s, Congress slashed plans to purchase 132 bombers to 21. In 2008, a B-2 was destroyed in a crash shortly after takeoff, though the crew ejected safely. Twenty B-2s are in service with the United States Air Force, which plans to operate them until 2032.

Further developments

A number of upgrade packages have been applied to the B-2. In July 2008, the B-2's onboard computing architecture was extensively redesigned; it now incorporates a new integrated processing unit (IPU) that communicates with systems throughout the aircraft via a newly installed fiber optic network; a new version of the operational flight program software was also developed, with legacy code converted from the JOVIAL programming language to standard C. Updates were also made to the weapon control systems to enable strikes upon moving targets, such as ground vehicles.

B-2 from below On 29 December 2008, Air Force officials awarded a US$468 million contract to Northrop Grumman to modernize the B-2 fleet's radars. Changing the radar's frequency was required as the United States Department of Commerce had sold that radio spectrum to another operator. In July 2009, it was reported that the B-2 had successfully passed a major USAF audit. In 2010, it was made public that the Air Force Research Laboratory had developed a new material to be used on the part of the wing trailing edge subject to engine exhaust, replacing existing material that quickly degraded.

In July 2010, political analyst Rebecca Grant speculated that when the B-2 becomes unable to reliably penetrate enemy defenses, the Lockheed Martin F-35 Lightning II may take on its strike/interdiction mission, carrying B61 nuclear bombs as a tactical bomber. However, in March 2012, the Pentagon announced that a $2 billion, 10-year-long modernization of the B-2 fleet was to begin. The main area of improvement would be replacement of outdated avionics and equipment.
It was reported in 2011 that the Pentagon was evaluating an unmanned stealth bomber, characterized as a "mini-B-2", as a potential replacement in the near future.

In 2012, Air Force Chief of Staff General Norton Schwartz stated the B-2's 1980s-era stealth technologies would make it less survivable in future contested airspaces, so the USAF is to proceed with the Next-Generation Bomber despite overall budget cuts. In 2012 projections, it was estimated that the Next-Generation Bomber would have an overall cost of $55 billion.

In 2013, the USAF contracted for the Defensive Management System Modernization program to replace the antenna system and other electronics to increase the B-2's frequency awareness.

The Common Very Low Frequency Receiver upgrade will allow the B-2s to use the same very low frequency transmissions as the *Ohio*-class submarines so as to continue in the nuclear mission until the Mobile User Objective System is fielded. In 2014, the USAF outlined a series of upgrades including nuclear war fighting, a new integrated processing unit, the ability to carry cruise missiles, and threat warning improvements.

Although the Air Force previously planned to operate the B-2 to 2058, their FY 2019 budget moved up its retirement to "no later than 2032". It also moved retirement of the B-1 to 2036 while extending the B-52's service life into the 2050s, due to the latter's lower maintenance costs, versatile conventional payload, and ability to carry nuclear cruise missiles (which the B-1 is treaty-prohibited from doing).

The decision to retire the B-2 early was made because the small fleet of 20 is considered too expensive per plane to retain, with its position as a stealth bomber being taken over with the introduction of the B-21 Raider starting in the mid-2020s

5.13.1. Design

Figure 5.10: Northrop-Grumman B-2 Stealth Bomber

5.13.2. Overview

The B-2 Spirit was developed to take over the USAF's vital penetration missions, able to travel deep into enemy territory to deploy ordnance which could include nuclear weapons. The B-2 is a flying wing aircraft, meaning that it has no fuselage or tail. It has significant advantages over previous bombers due to its blend of low-observable technologies with high aerodynamic efficiency and large payload. Low observability provides a greater freedom of action at high altitudes, thus increasing both range and field of view for onboard sensors.

The U.S. Air Force reports its range as approximately 6,000 nautical miles (6,900 mi; 11,000 km). At cruising altitude, the B-2 refuels every six hours, taking on up to 50 short tons (45,000 kg) of fuel at a time.

The development and construction of the B-2 required pioneering use of computer-aided design and manufacturing technologies, due to its complex flight characteristics and design requirements to maintain very low visibility to multiple means of detection. The B-2 bears a resemblance to earlier Northrop aircraft; the YB-35 and YB-49 were both flying wing bombers that had been canceled in development in the early 1950s, allegedly for political reasons. The resemblance goes as far as B-2 and YB-49 having the same wingspan.

The YB-49 also had a small radar cross-section. Approximately 80 pilots fly the B-2. Each aircraft has a crew of two, a pilot in the left seat and mission commander in the right, and has provisions for a third crew member if needed. For comparison, the B-1B has a crew of four and the B-52 has a crew of five. The B-2 is highly automated, and one crew member can sleep in a camp bed, use a toilet, or prepare a hot meal while the other monitors the aircraft, unlike most two-seat aircraft. Extensive sleep cycle and fatigue research was conducted to improve crew performance on long sorties. Advanced training is conducted at the USAF Weapons School

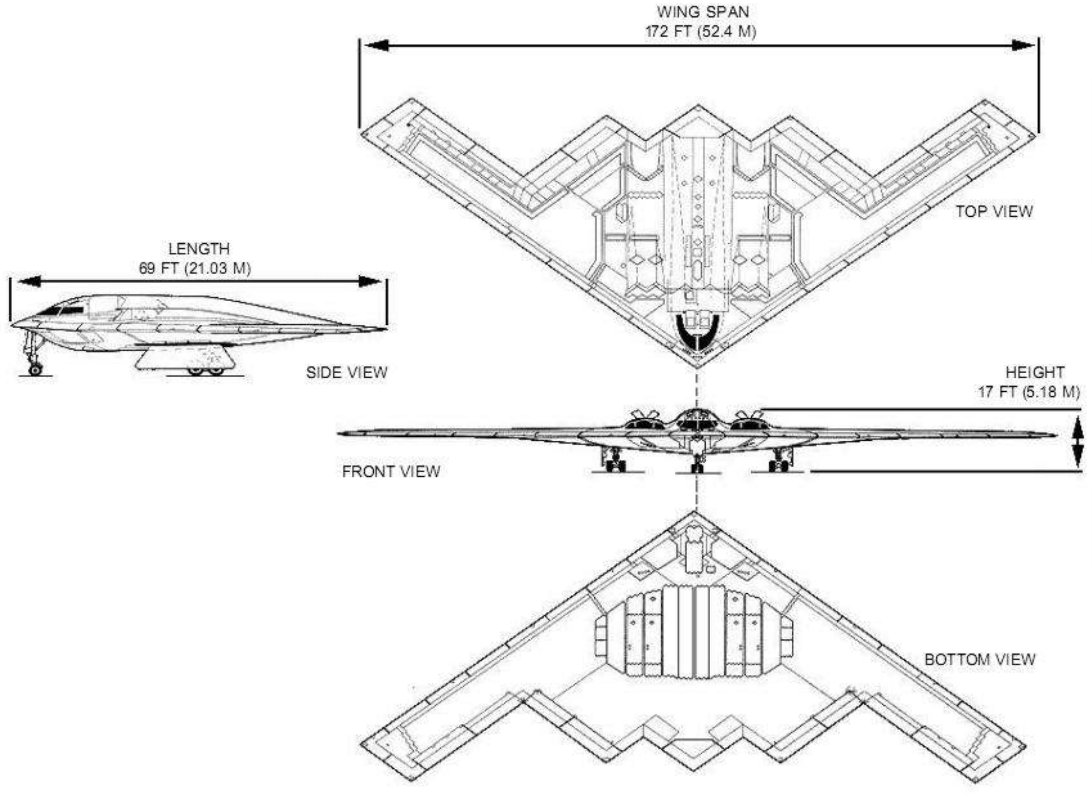


Figure 5.11. Views Northrop-Grumman B-2 Stealth Bomber

5.13.3. General characteristics

- Crew: 2: pilot (left seat) and mission commander (right seat)
- Length: 69 ft (21 m)
- Wingspan: 172 ft (52 m)
- Height: 17 ft (5.2 m)

- Wing area: 5,140 sq ft (478 m^2)
- Empty weight: 158,000 lb (71,668 kg)
- Gross weight: 336,500 lb (152,634 kg)
- Max takeoff weight: 376,000 lb (170,551 kg)
- Fuel capacity: 167,000 pounds (75,750 kg)
- Power plant: 4 × General Electric F118-GE-100 non-afterburning turbofans, 17,300 lbf (77 kN) thrust each

5.13.4. Performance

- Maximum speed: 630 mph (1,014 km/h; 547 kn) at 40,000 ft altitude / Mach 0.95 at sea level[153][*verification needed*]
- Maximum speed: Mach 0.95
- Cruise speed: 560 mph (901 km/h; 487 kn) at 40,000 ft altitude
- Range: 6,905 mi; 11,112 km (6,000 nmi) 11,100 km (6,900 mi)
- Service ceiling: 50,000 ft (15,000 m)
- Wing loading: 67.3 lb/sq ft (329 kg/m^2)

Thrust/weight: 0.205. It has significant advantages over previous bombers due to its blend of low-observable technologies with high aerodynamic efficiency and large payload. Low observability provides a greater freedom of action at high altitudes, thus increasing both range and field of view for onboard sensors.

The Common Very Low Frequency Receiver upgrade will allow the B-2s to use the same very low frequency transmissions

It also moved retirement of the B-1 to 2036 while extending the B-52's service life into the 2050s, due to the latter's lower maintenance costs, versatile conventional payload, and ability to carry nuclear cruise missiles (which the B-1 is treaty-prohibited from doing).

The End