Aircraft Design
AE 405

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Team - Term Project

Grading

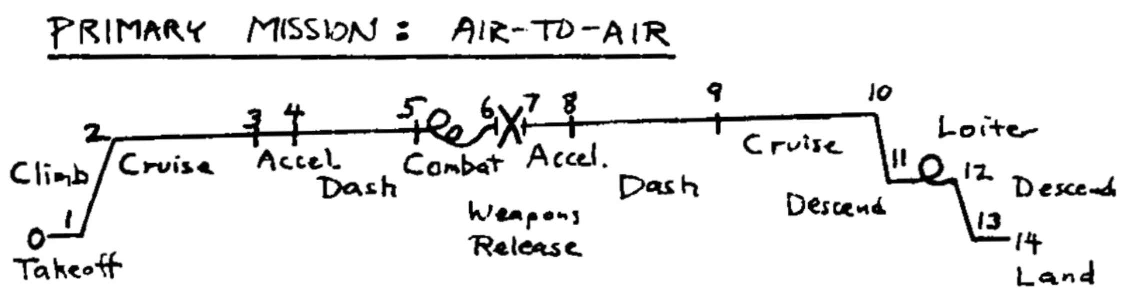
▪ Conceptual design of fighter aircraft,	▪ 40 points
▪ Solid modeling of aircraft Shell,	▪ 20 points
▪ Production of aircraft (wood or foam),	▪ 20 points
▪ Flight test of aircraft (r/c radio controlled)	▪ 20 points
If it flies successfully +20 points	
Due date:	A week ahead of final exam.

Term Project

Design Requirements

- Unmanned,
- Double-engine (new, 'rubber'),
- Stealth, Supercruise,
- Primary mission: air to air combat.

Mission profile



- 3 & 10 (cruise): 250 nm at best cruise Mach and altitude,
- 5 & 9 (dash): 75 nm at 1.5 M at 30,000 ft.
- 6 (Combat): 4 min at max. thrust, M 0.9 at 30,000 ft,
- 12 (Loiter): 30 min at sea level, best loiter speed,
- 7 (weapons release): 400 lb (missiles only)

Term Project

- Payload:
  - 1 laser gun,
  - 2 advanced missiles (200 lb, 5 in x 90 in)
  - Advanced gun (400 lb), 750 rounds ammo (450 lb)
  - Pilot (220 lb)

- Performance requirements;
  - Take off and landing (1000 ft ground roll),
  - Approach speed $\leq$ 120 kts,
  - Maximum Mach $\geq$1.8 (A/B); M 1.5 (Dry),
  - Accelerate M 0.9 to M 1.5 in 30 sec at 30,000 ft,
  - Ps =0 at 5g at 25,000 ft at M 0.9 and M 1.5,
  - $\dot{\varphi} \geq 20\ ^{\circ}/_{sec}$ at 350 kts at 25,000 ft.

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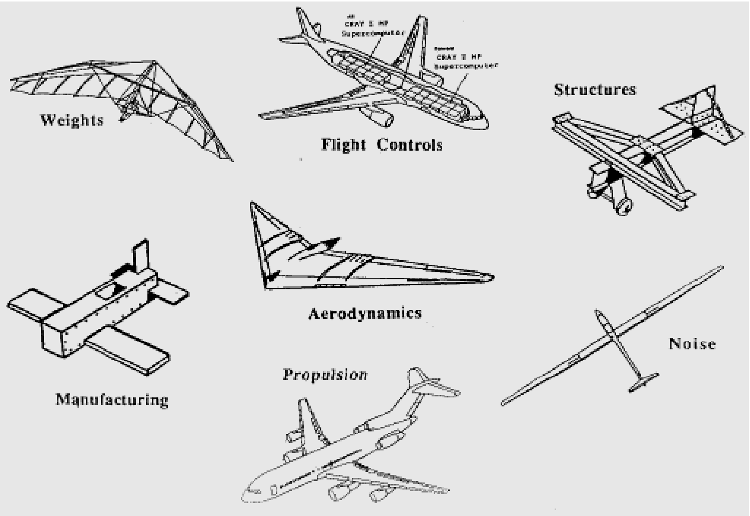
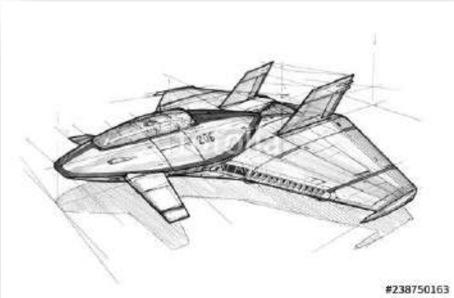
Contents



Chapter 1 - 2 - 3

- Design – a separate discipline
- Overview of the design process
- Sizing from a conceptual sketch

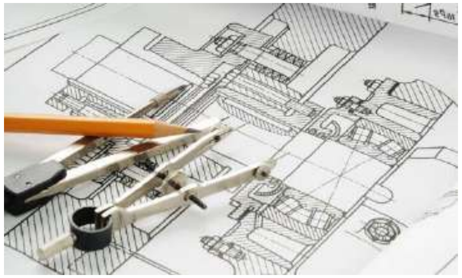


▪ Design – a separate discipline

- Aircraft design is a separate discipline of aeronautical engineering different from the analytical disciplines such as aerodynamics, structures, controls, and propulsion.

- An aircraft designer spends time doing something called "design," creating the geometric description of a thing to be built.

- "Design" looks a lot like "drafting", or in the modern world, "computer-aided drafting".



▪ Design – a separate discipline

A good aircraft design looks like…

F -22

Su -34

- Somehow, the landing gear fits,
- The fuel tanks are near the center of gravity,
- The structural members are simple and lightweight,
- The overall arrangement provides good aerodynamics,
- The engines install in a simple and clean fashion,
- And a host of similar detail seems to fall into place…

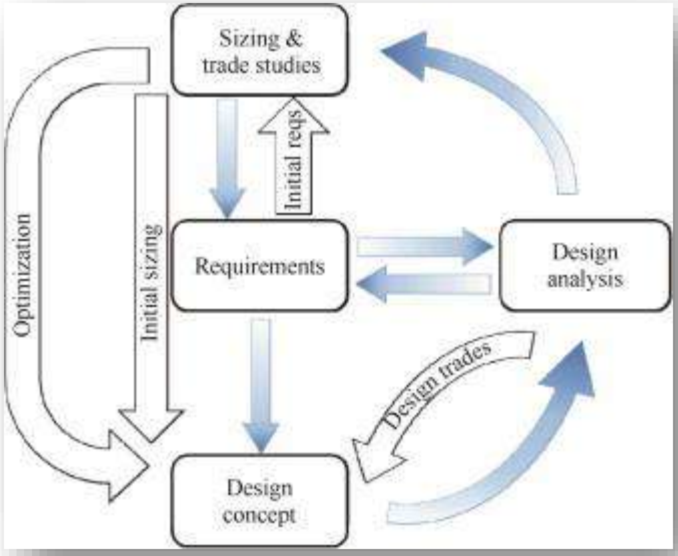
Overview of the design process

- Design is an iterative effort, as shown in the "Design Wheel"

- 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.



Overview of the design process

- Aircraft design can be broken into three major phases;
  - Conceptual design
  - Preliminary design
  - Detail Design

- Conceptual design is the primary focus of this course.

- 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.

Overview of the design process

- Preliminary design can be said to begin when the major changes are over.

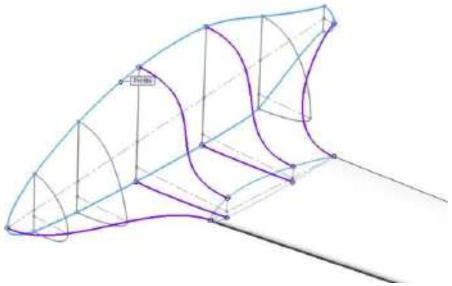
- The big questions such as whether to use a canard or an aft tail have been resolved.

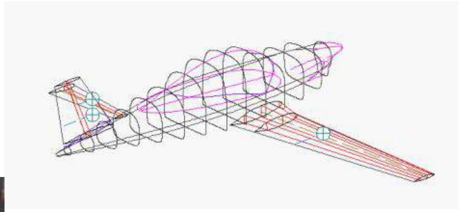
- Testing is initiated in areas such as aerodynamics, propulsion, structures, and stability and control.

- The ultimate objective during preliminary design is to ready the company for the detail design stage, also called full-scale development.

Overview of the design process

- A key activity during preliminary design is "lofting."
- Lofting is the mathematical modeling of the outside skin of the aircraft with sufficient accuracy.
- A mockup may be constructed at this point.

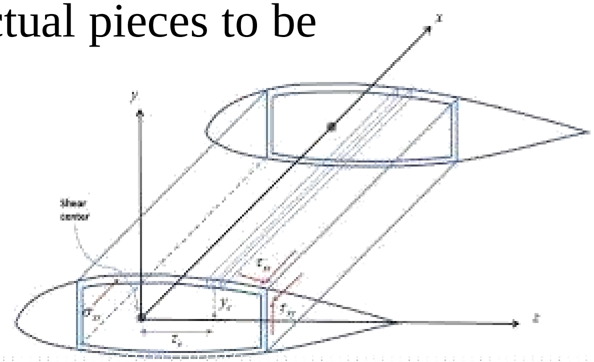




Lockheed Martin unveiled a mockup of F-35 Lighting II in Japan International Aerospace Exhibition 2016.

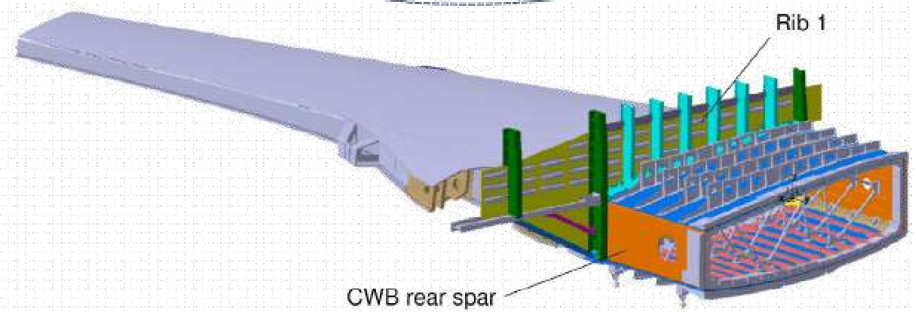
Overview of the design process

- 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.



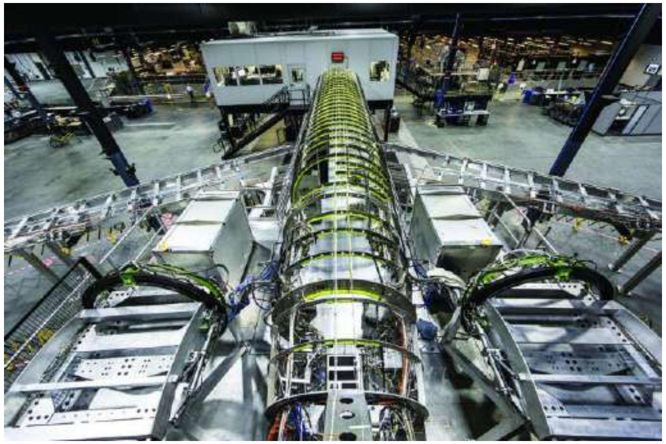
- Another important part of detail design is called production design.

- Specialists determine how the airplane will be fabricated, starting with the smallest and simplest subassemblies and building up to the final assembly process.

Overview of the design process

- 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.

Gulfstream's virtual test plane, or "iron bird."



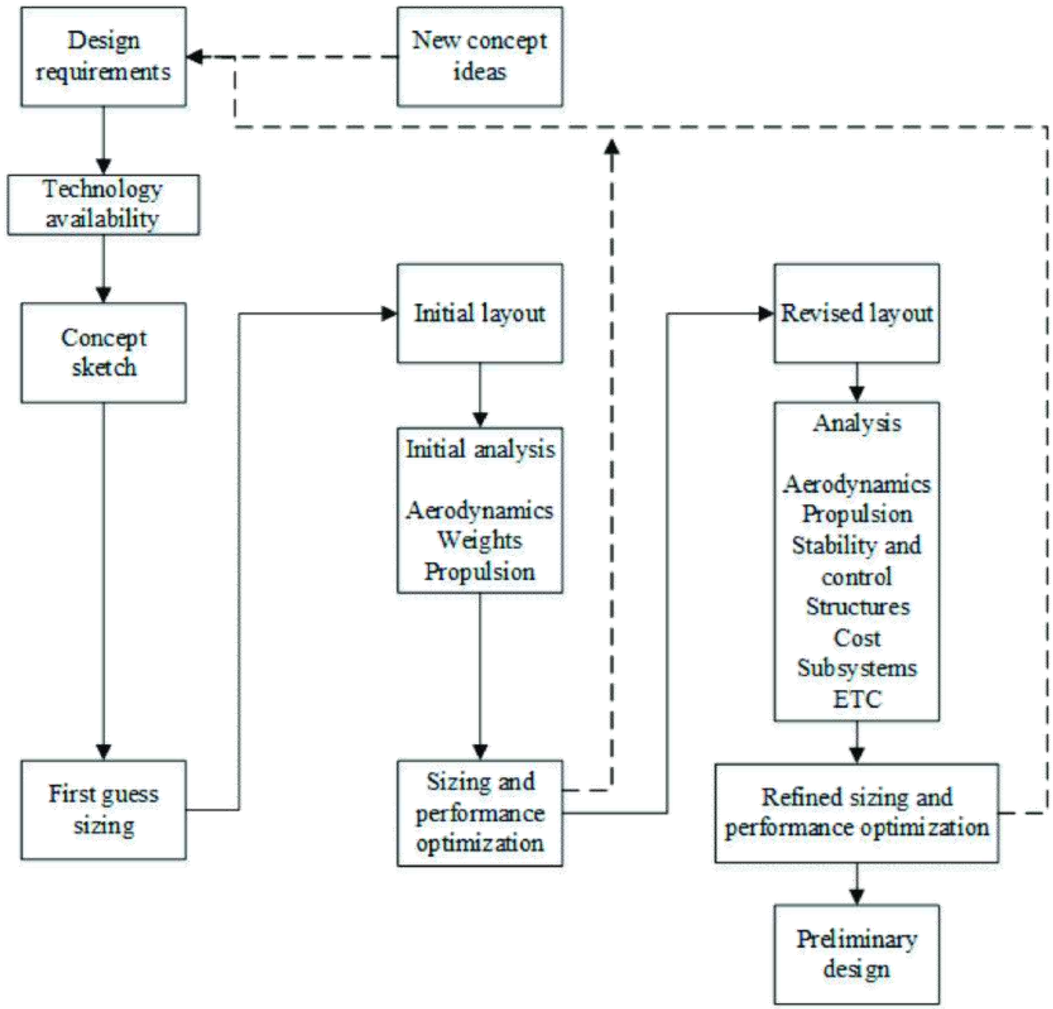
Overview of the design process

- Figure depicts the conceptual design process in greater detail.

- Conceptual design will usually begin with a 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,
  - Speed requirements,
  - structural design limits,
  - pilots' outside vision angles,
  - reserve fuel, and many others



Overview of the design process

- 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.

- Future Technologies may result in a higher development risk.
  - Active flow control by blowing/suction pumps,
  - Laser weapons,
  - All composite fighter,
  - Blended wing body…

- The actual design effort usually begins with a conceptual sketch.

- This is the "back of a napkin" drawing of aerospace legend, and gives a rough indication of what the design may look like.

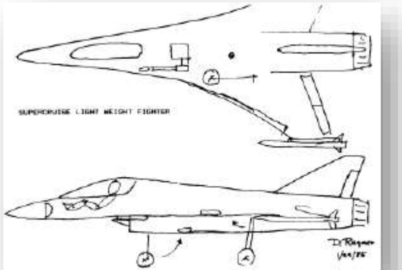


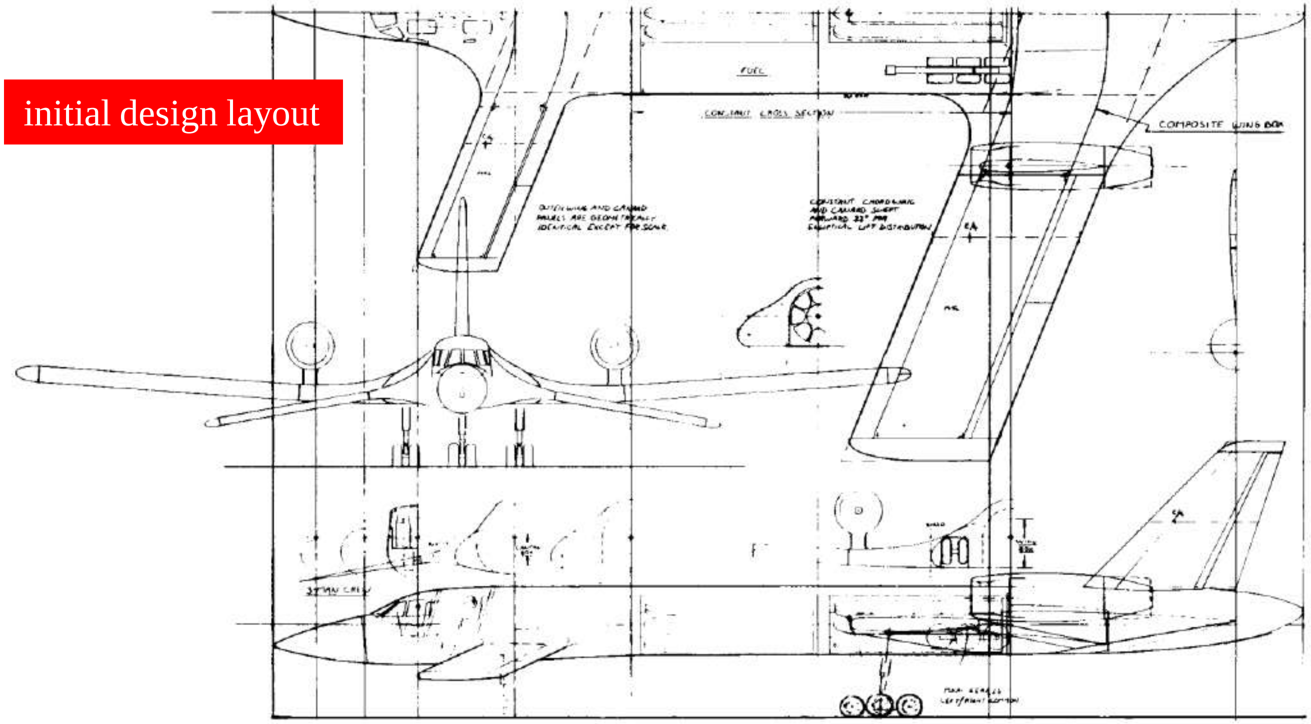
Fig. 2.4 Initial sketch.

Overview of the design process

- 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 "first-order" sizing provides the information needed to develop an initial design layout.

Overview of the design process

initial design layout



- On a computer-aided design system, the design work is usually done in full scale (numerically).

Overview of the design process

- The revised drawing, after some number of iterations, is then examined in detail by an ever-expanding group of specialists.

- For example, controls experts will perform a six-degree-of-freedom analysis to ensure that the designer's estimate for the size of the control surfaces is adequate for control.

- If not, they will instruct the designer as to how much each control surface must be expanded.

- The end product of all this will be an aircraft design that can be confidently passed to the preliminary design phase,

Sizing from a conceptual sketch

- There are many levels of design procedure. The simplest level just adopts past history - **statistics**.

- To get the "right" answer takes several years, many people, and lots of money.

- The most important data for sizing is the weight, especially "takeoff weight. "

- "Design takeoff gross weight" is the total weight of the aircraft as it begins the mission for which it was designed.

$$"W_0"$$

Sizing from a conceptual sketch

takeoff weight buildup

- 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 (3.1) summarizes the takeoff-weight buildup.

$$W_0 = W_{crew} + W_{payload} + W_{fuel} + W_{empty} \tag{3.1}$$

- 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, they are both dependent on the total aircraft weight. Thus an iterative process must be used for aircraft sizing.

Sizing from a conceptual sketch

- To simplify the calculation, both fuel and empty weights can be expressed as fractions of the total takeoff weight,

$$W_0 = W_{crew} + W_{payload} + \left(\frac{W_f}{W_0}\right) W_0 + \left(\frac{W_e}{W_0}\right) W_0 \qquad (3.2)$$

and

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

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

- Now *Wo* can be determined if $+\left(\frac{W_f}{W_0}\right)$ and $+\left(\frac{W_e}{W_0}\right)$ can be estimated.

Sizing from a conceptual sketch

The empty-weight fraction

$$+ \left(\frac{W_e}{W_0}\right)$$

- The empty-weight fraction can be estimated statistically from historical trends as shown in Fig. 3.1.
- Empty-weight fractions vary from about 0.3 to 0.7, and diminish with increasing total aircraft weight.
- Table 3.1 presents statistical curve-fit equations for the trends shown in Fig. 3.1.
- Note that these are all exponential equations based upon takeoff gross weight.

Sizing from a conceptual sketch

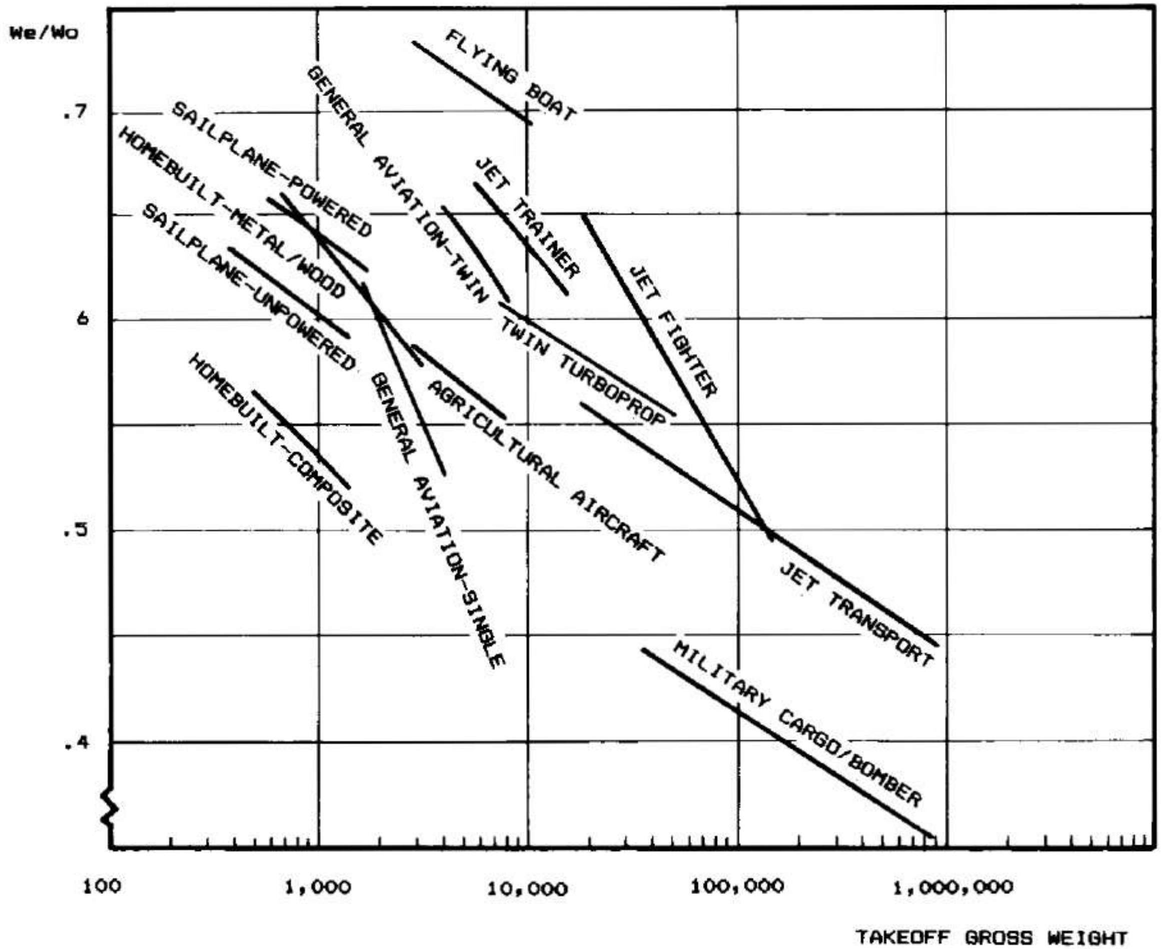


Fig. 3.1 Empty weight fraction trends.

Sizing from a conceptual sketch

Table 3.1 Empty weight fraction vs W_0

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

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

Sizing from a conceptual sketch

- Based on a number of design studies, the empty-weight fraction for composite aircraft can be estimated by multiplying the statistical empty weight fraction by 0.95.
- A variable-sweep wing is heavier than a fixed wing.
- It is accounted for at this initial stage of design by multiplying the empty-weight fraction as determined from the equations in Table 3 .1 by about 1.04.

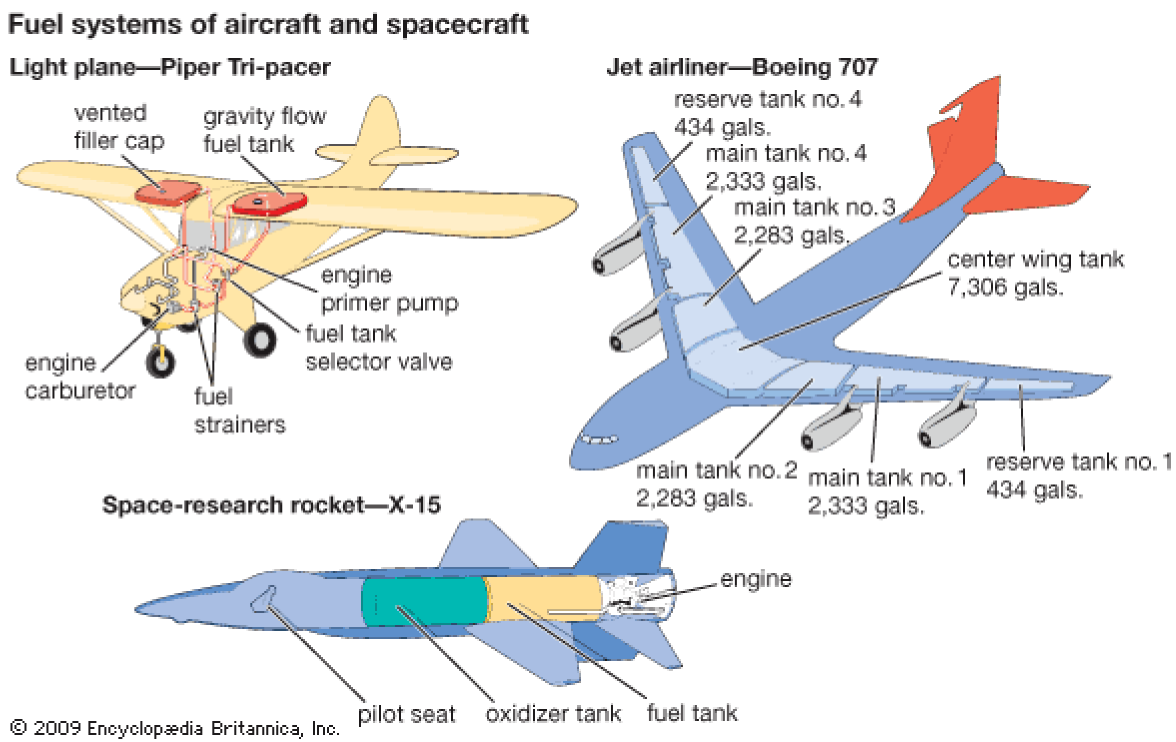
Sizing from a conceptual sketch

The fuel-weight fraction

$$+\left(\frac{W_f}{W_0}\right)$$

- 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 to be flown, the aerodynamics of the aircraft, and the engine's fuel consumption.

Sizing from a conceptual sketch



Fuel tank locations

Sizing from a conceptual sketch

- Fuel fraction can be estimated based on the mission to be flown using approximations of the fuel consumption and aerodynamics.

- Typical mission profiles for various types of aircraft are shown in Fig. 3.2.

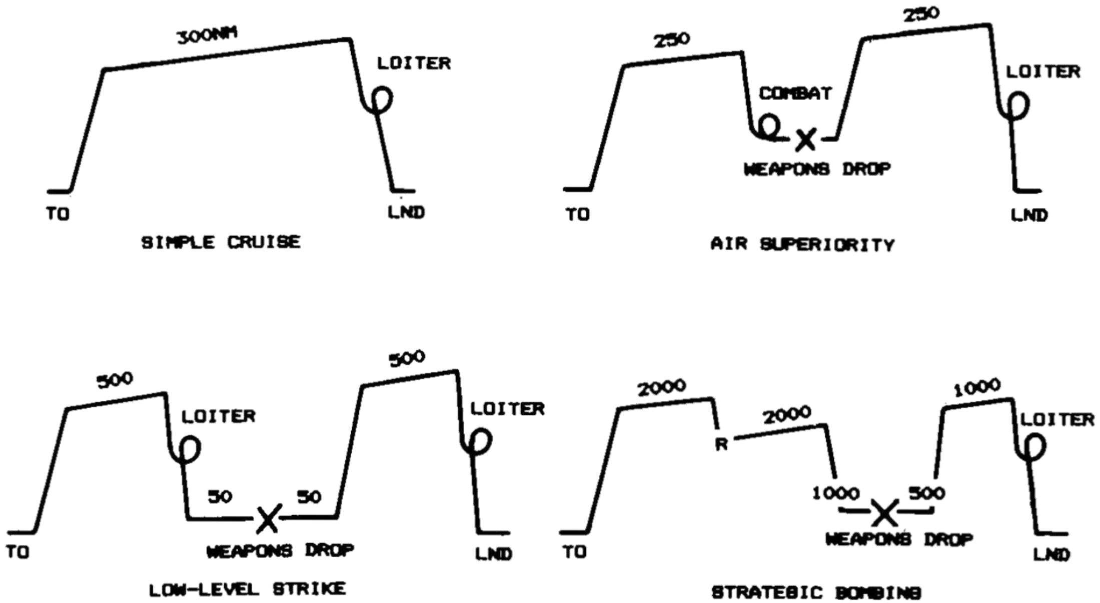


Fig. 3.2 Typical mission profiles for sizing.

Sizing from a conceptual sketch

- The Simple Cruise mission is used for many transport and general aviation designs, including homebuilts.
- The aircraft is sized to provide some required cruise range.
- For safety you would be wise to carry extra fuel in case your intended airport is closed, so a 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.
- FAA requires 30 min of additional cruise fuel for general-aviation aircraft.

Sizing from a conceptual sketch

Mission Segment Weight Fractions

- For analysis, the various mission segments, or "legs," are numbered.

- With zero denoting the start of the mission.

- Mission leg "one" is usually engine warmup and takeoff for first-order sizing estimation.

- The remaining legs are sequentially numbered.

- For example, in the simple cruise mission the legs could be numbered as (1) warmup and takeoff, (2) climb, (3) cruise, (4) loiter, and (5) land.

Sizing from a conceptual sketch

- In a similar fashion, the aircraft weight at each part of the mission can be numbered. Thus, W_0 is the beginning weight.
- For the simple cruise mission, W_1 would be the weight at the end of the first mission segment, which is the warmup and takeoff.
- W_2 would be the aircraft weight at the end of the climb.
- W_3 would be the weight after cruise, and W_4 after loiter.
- Finally, W_5 would be the weight at the end of the landing segment, which is also the end of the total mission.

Sizing from a conceptual sketch

- For any mission segment "*i*," the mission segment weight fraction can be expressed as (W_i/W_{i-1}).
- The warmup, takeoff, and landing weight-fractions can be estimated historically.
- Table 3.2 gives typical historical values for initial sizing.

Table 3.2 Historical mission segment weight fractions

	(W_i/W_{i-1})
Warmup and takeoff	0.970
Climb	0.985
Landing	0.995

Sizing from a conceptual sketch

- Cruise-segment mission weight fractions can be found using the *Breguet* range equation

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

where
R = range
C = specific fuel consumption
V = velocity
L/D = lift-to-drag ratio

- Loiter weight fractions are found from the endurance equation

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

where E = endurance or loiter time.

- Range and endurance are given data (requirement), we need to find out specific fuel consumption and L/D ratio.

Sizing from a conceptual sketch

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 1/hr].

- Figure 3.3 shows SFC vs Mach number.

- Propeller engine SFC is normally given as C_{bhp}, the pounds of fuel per hour to produce one horsepower at the propeller shaft (or one "brake horsepower": bhp = 550 ft-Ib/s).

- A propeller thrust SFC equivalent to the jet-engine SFC can be calculated.

Sizing from a conceptual sketch

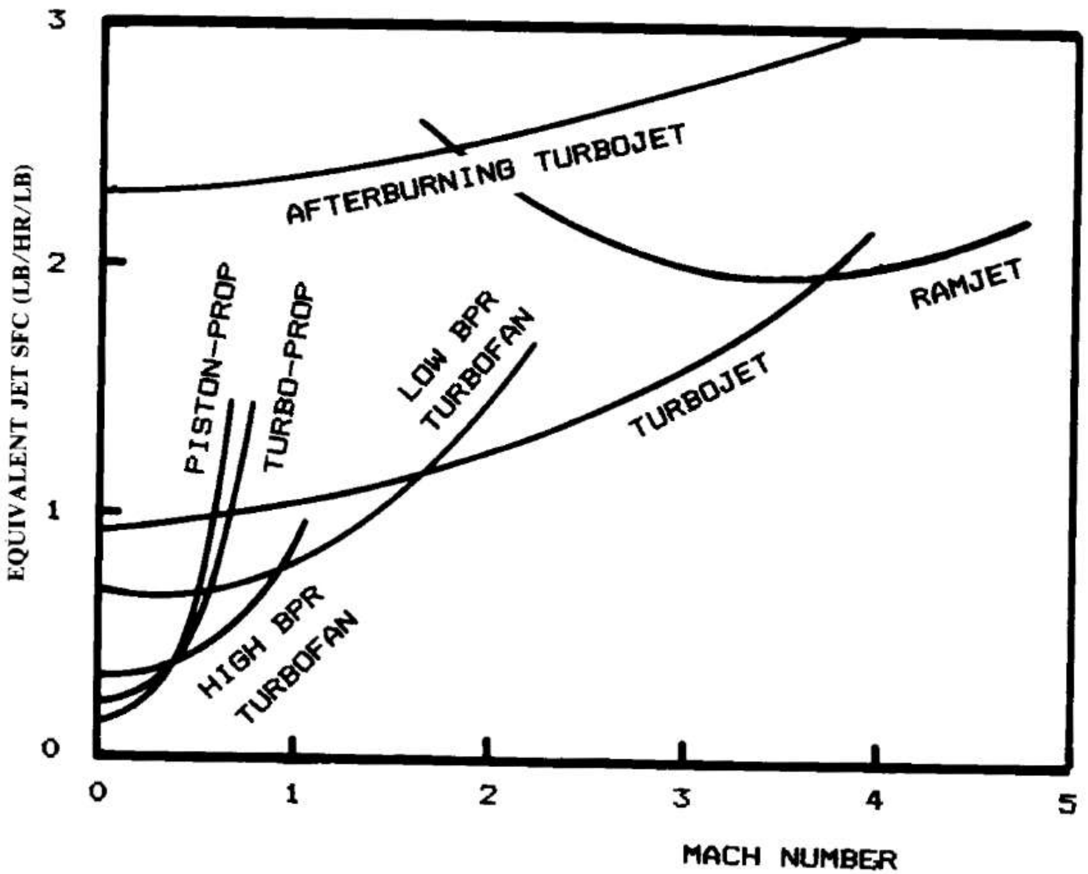


Fig. 3.3 Specific fuel consumption trends.

Sizing from a conceptual sketch

- The engine produces thrust via the propeller, which has an efficiency

$$\eta_p = \frac{TV}{550\ \text{hp}} \quad V \text{ is in feet per second.}$$

- Equation (3.10) shows the derivation of the equivalent-thrust SFC for a propeller-driven aircraft.

$$C = \frac{W_f/\text{time}}{\text{thrust}} = C_{\text{bhp}} \frac{V}{550\ \eta_p}$$

- Table 3.3 _provides typical SFC values for jet engines, while Table 3.4 provides typical C_{bhp} and propeller efficiency values for propeller engines.

- These can be used for rough initial sizing

Sizing from a conceptual sketch

Table 3.3 Specific fuel consumption (C)

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 3.4 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

Sizing from a conceptual sketch

L/D Estimation

- *L/D*, or lift-to-drag ratio, is a measure of the design's overall aerodynamic efficiency.

- At subsonic speeds *L/D* is most directly affected by two aspects of the design: wing span and wetted area.

The wetted area is the area which is in contact with the external airflow.

- This suggests a new parameter, the "Wetted Aspect Ratio," which is defined as the wing span squared divided by the total aircraft wetted area.

- This is very similar to the aspect ratio except that it considers total wetted area instead of wing reference area.

Sizing from a conceptual sketch

- Figure 3.6 plots maximum *L/D for* a number of aircraft vs the wetted aspect ratio, and shows clear trend lines for jet, prop, and fixed-gear prop aircraft.

wetted-area ratio: $\frac{S_{wet}}{S_{ref}}$

- Note that the wetted aspect ratio can be shown to equal the wing geometric aspect ratio divided by the wetted-area ratio,

- *L*/D can now be estimated from a conceptual sketch.

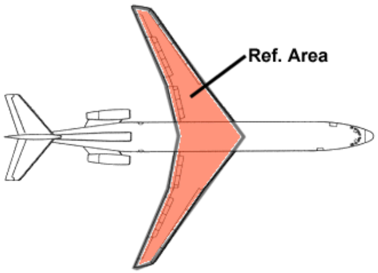
- The designer arranges the major components of the aircraft, from the sketch the wetted-area ratio can be "eyeball-estimated" using Fig. 3.5 for guidance.

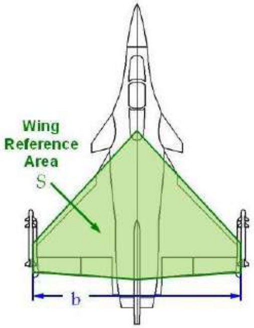
- The wetted aspect ratio can then be calculated as the wing aspect ratio divided by the wetted-area ratio.

- Figure 3.6 can then be used to estimate the maximum LID.

Sizing from a conceptual sketch

The reference wing area is defined as the plan area of an aircraft's wing (usually including the area of the wing through the fuselage)





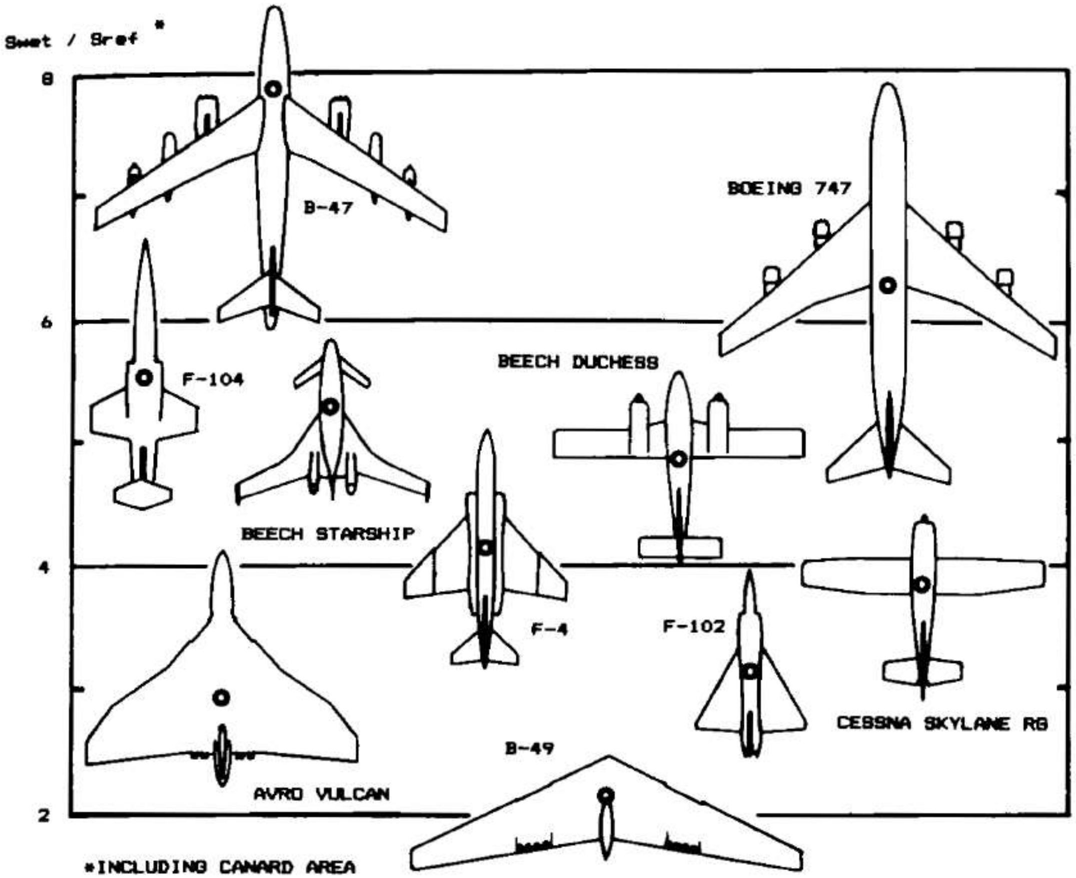


Fig. 3.5 Wetted area ratios.

Sizing from a conceptual sketch

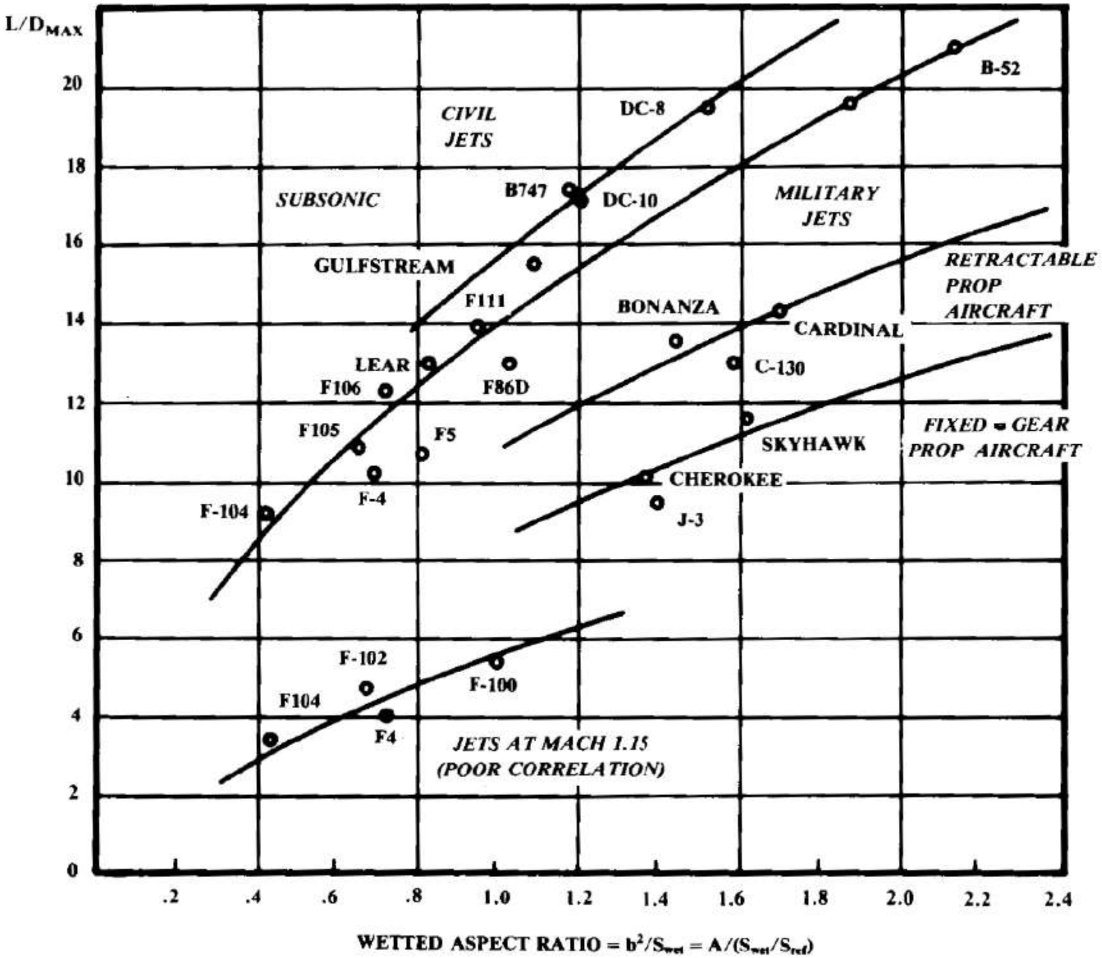


Fig. 3.6 Maximum lift to drag ratio trends.

Sizing from a conceptual sketch

- We use the following vales for *L/D* ratio

	Cruise	Loiter
Jet	$0.866\ L/D_{max}$	L/D_{max}
Prop	L/D_{max}	$0.866\ L/D_{max}$

- Using historical values from Table 3.2 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, W_x/W_o, can be calculated.

$$\frac{W_1}{W_0} \times \frac{W_2}{W_1} \times \frac{W_3}{W_2} \ldots \times \frac{W_x}{W_{x-1}}$$

- All weight lost during the mission must be due to fuel usage. The mission fuel fraction must therefore be equal to

$$(1 - W_x/W_0)$$

Sizing from a conceptual sketch

- If you assume, typically, a 6% allowance for reserve and trapped fuel, the total fuel fraction can be estimated as in Eq. (3 .11).

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

Sizing from a conceptual sketch

T/O weight calculation

- Using the fuel fraction (Eq. (3.11)) and the statistical empty weight equation selected from Table 3 .1, the takeoff gross weight can be found iteratively from Eq. (3.4).

- 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 first-order sizing process is diagrammed in Fig. 3.7.

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right) \quad (3.11)$$

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - (W_f/W_0) - (W_e/W_0)} \quad (3.4)$$

Sizing from a conceptual sketch

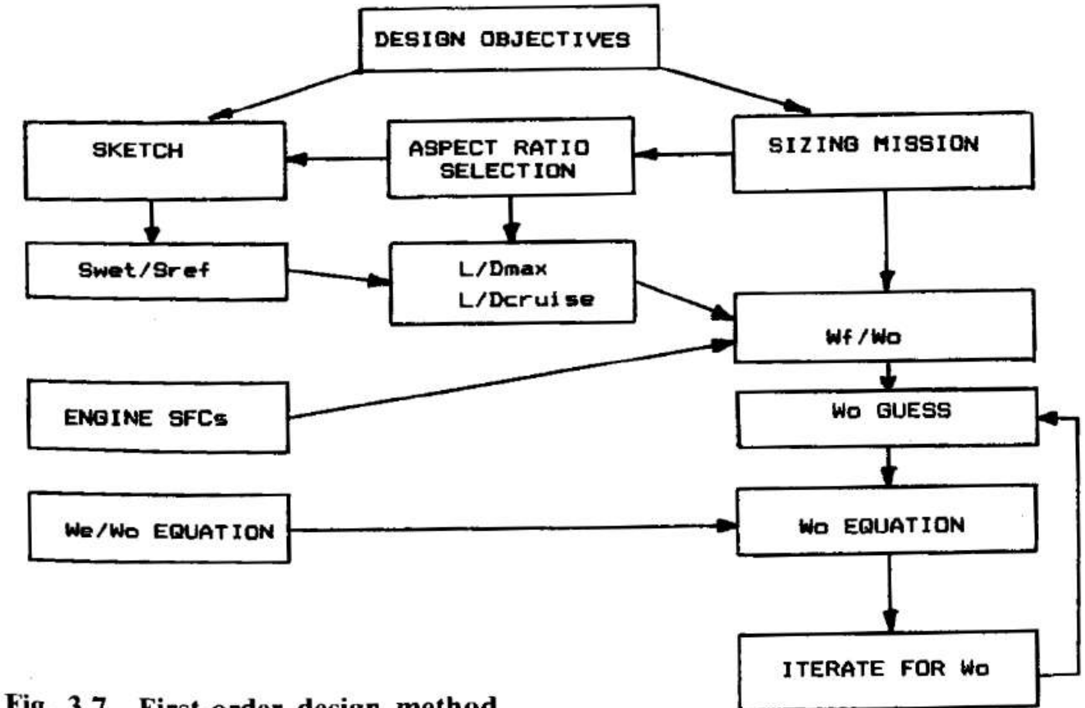


Fig. 3.7 First-order design method.

Sizing from a conceptual sketch

Design Example

- Fig. 3.8 illustrates the mission requirement for anti-submarine warfare (ASW) aircraft.
- The key requirement is the ability to loiter for 3 hours at a distance of 1500 nm from takeoff point.
- It uses sophisticated electronic equipment, is assumed to weigh 10,000 lb.
- 4-man crew is required, 800 lb.
- Aircraft must cruise at 0.6 Mach number.

Sizing from a conceptual sketch

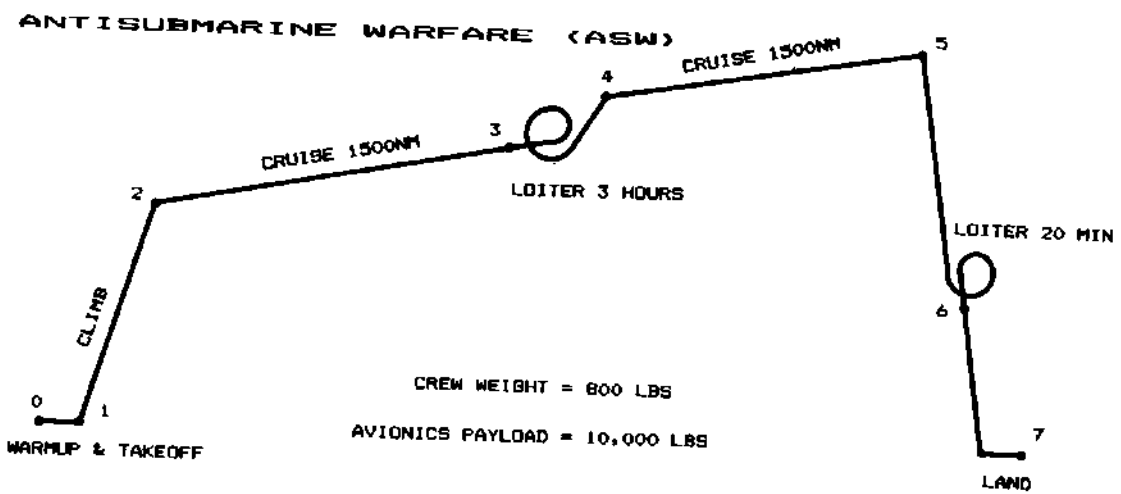
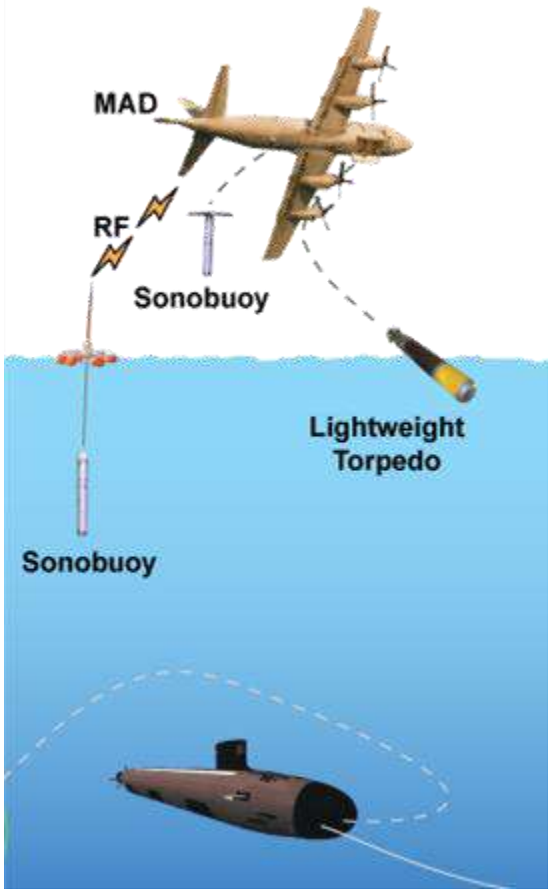


Fig. 3.8 Sample mission profile.



- Anti-submarine warfare (ASW, or in older form A/S) is a branch of underwater warfare that uses surface warships, aircraft, or other submarines to find, track, and deter, damage, or destroy enemy submarines.

Sizing from a conceptual sketch

- Based on previous designs and experience, we may imagine four conceptual approaches as shown in Fig. 3.9;

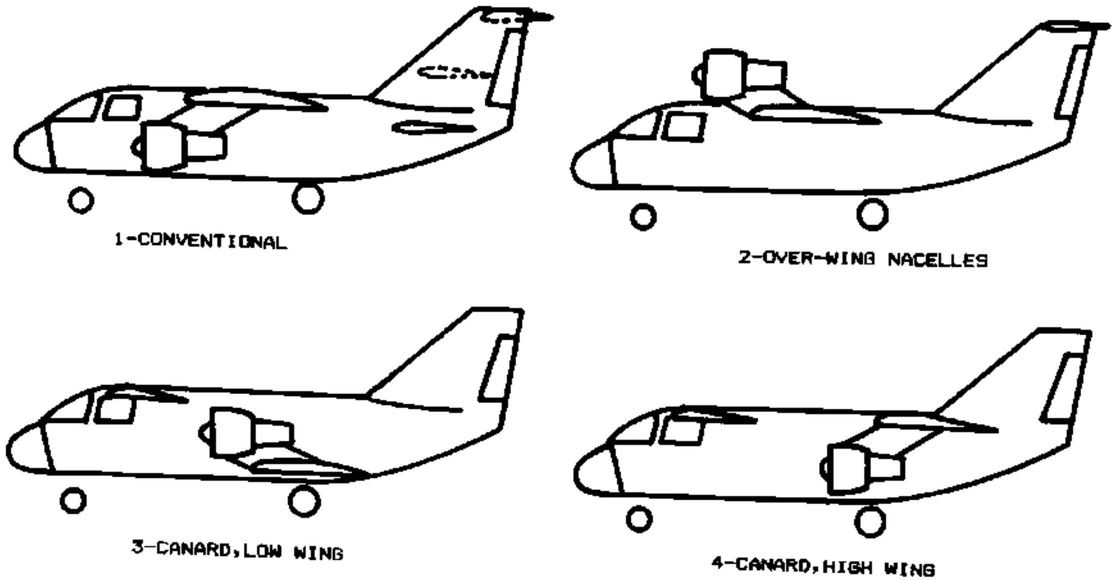
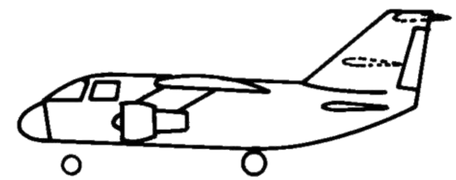


Fig. 3.9 ASW concept sketches.

Sizing from a conceptual sketch

- 1st option;

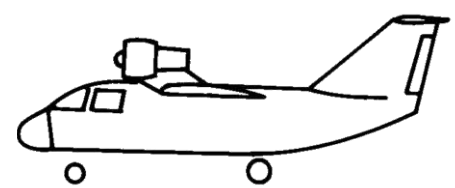
  - Conventional type,
  - Low horizontal tail position means lightest structure,
  - But the tail in the exhaust stream of the jet engines, other positions are available.



1-CONVENTIONAL

- 2nd option;

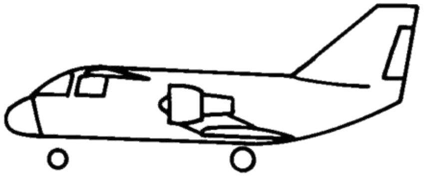
  - It is much like the first except for the engine location,
  - Mounted over the wing provides extra lift,
  - provides greater ground clearance,
  - reduces the tendency of the jet engines to suck up debris, but, reaching the engines for maintenance work is difficult.



Sizing from a conceptual sketch

- 3rd option;

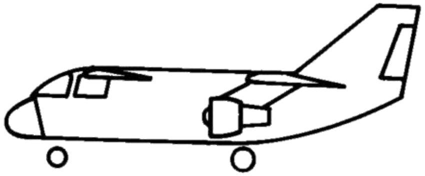
  - Concept has canarded approach,
  - It offers reduced drag,
  - May provide a wider allowable range for the center of gravity,
  - The wing is low, allow the main landing gear to be stowed in the wing root.



3-CANARD, LOW WING

- 4th option;

  - Concept has canarded approach,
  - The wing is high, offers better access to the engines.

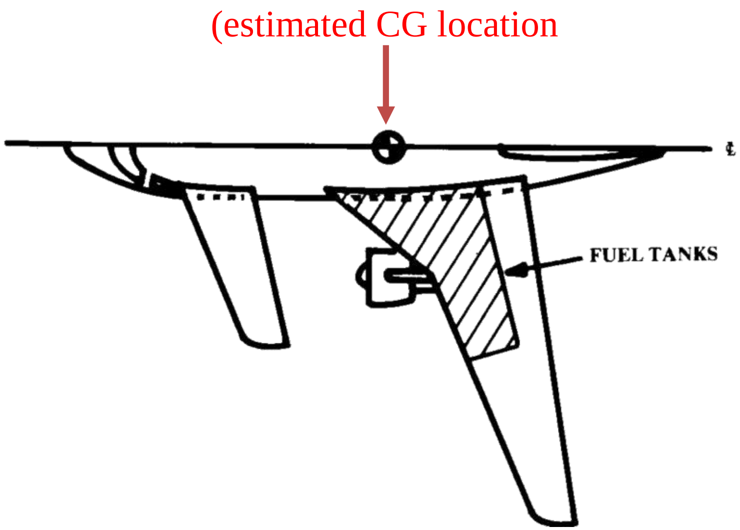


4-CANARD, HIGH WING

4th option was selected for further development.

Sizing from a conceptual sketch

- Figure 3 .10 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.



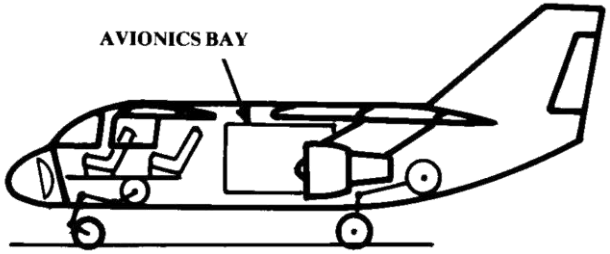


Fig. 3.10 Completed ASW sketch.

- CG position is important, it should be within margins when fuel is consumed.

Sizing from a conceptual sketch

L/D Estimation

- For initial sizing, a wing aspect ratio of about 8 was selected.
- From Fig. 3.5, it would appear that the wetted area ratio is about *5.5;* a wetted aspect ratio of 1.45 (i.e., 8/5.5).
- For a wetted aspect ratio of 1.45, Fig. 3.6 indicates that a maximum lift-to-drag ratio of about 17 (jet engine).
- 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.

Sizing from a conceptual sketch

Takeoff Weight Sizing

- From Table 3.3, initial values for SFC are obtained.

Table 3.3 Specific fuel consumption (C)

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

- 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.

- Table 3.1 may provide an equation for statistically estimating the empty weight fraction of an antisubmarine aircraft.

Table 3.1 Empty weight fraction vs W_0

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

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

- 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.

Sizing from a conceptual sketch

- Lets follow the calculations for sizing this example;

Cruise velocity: 0.6 Mach
Typical cruise altitude: 30,000 ft; speed of sound: 994.8 ft/s
Range : 1500 nm

- Mission Segment Weight Fractions

1) Warmup and takeoff
2) Climb
3) Cruise
4) Loiter
5) Cruise (same as 3)
6) Loiter
7) Land

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

Sizing from a conceptual sketch

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

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

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

$$W_e/W_0 = 0.93\,W_0^{-0.07}$$

(Table 1)

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

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

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

Sizing from a conceptual sketch

Trade Studies

- An important part of conceptual design is the evaluation and refinement with the customer, of the design requirements; range, payload, structure options…

- Range options:
  - 1000 nm,
  - 1500 nm,
  - 2000 nm

<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

Sizing from a conceptual sketch

2000 nm Range

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

$$W_7/W_0 = 0.5713$$

$$W_f/W_0 = 0.4544$$

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

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

Sizing from a conceptual sketch

- Payload weights options;
  - 5000 lb,
  - 15,000 lb,
  - 20,000 lb.

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

Payload = 5000 lb $\quad W_0 = \dfrac{5800}{0.613 - 0.93 W_0^{-0.07}}$

W_0 Guess	W_e/W_0	W_0 Calculated
50,000	0.4361	32,787
35,000	0.4471	34,960
34,970	0.4471	34,965
34,966	0.4471	34,966

Payload = 15,000 lb $\quad W_0 = \dfrac{15,800}{0.613 - 0.93 W_0^{-0.07}}$

W_0 Guess	W_e/W_0	W_0 Calculated
50,000	0.4362	89,366
85,000	0.4202	81,937
82,000	0.4212	82,389
82,350	0.4211	82,335

Payload = 20,000 lb $\quad W_0 = \dfrac{20,800}{0.613 - 0.93 W_0^{-0.07}}$

W_0 Guess	W_e/W_0	W_0 Calculated
90,000	0.4185	106,941
100,000	0.4154	105,272
105,000	0.4140	104,522
104,600	0.4141	104,581

Sizing from a conceptual sketch

- The statistical empty-weight equation used here for sizing was based upon existing military cargo and bomber aircraft, which are all of aluminum construction.

- For composite materials, 95% of the empty-weight fraction obtained for a metal aircraft may be option.

Range : 1500 nm
Payload : 10,000 lb

$$W_e/W_0 = (0.95)(0.93\,W_0^{-0.07}) = 0.8835\,W_0^{-0.07}$$

$$W_0 = \frac{10{,}800}{1 - 0.387 - \dfrac{W_e}{W_0}} = \frac{10{,}800}{0.613 - 0.8835\ W_0^{-0.07}}$$

W_0 Guess	W_e/W_0	W_0 Calculated
50,000	0.4143	54,344
54,000	0.4120	53,742
53,800	0.4122	53,771

- The use of composite materials reduces the takeoff gross weight from 59,310 lb to only 53,771 lb.

Sizing from a conceptual sketch

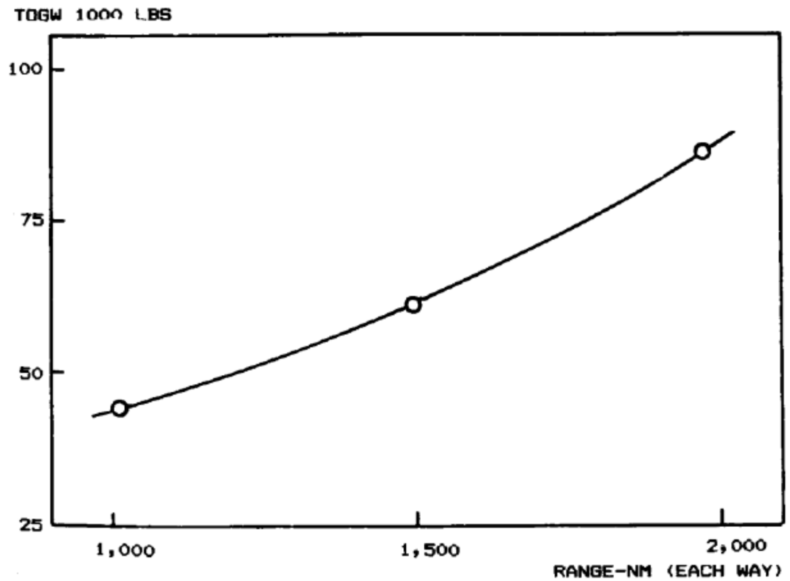


Fig. 3.11 Range trade.

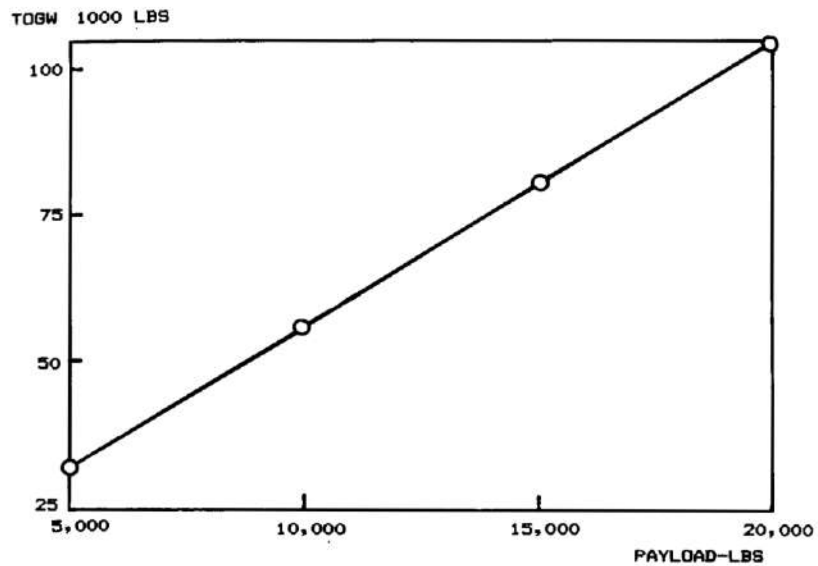


Fig. 3.12 Payload trade.

- There are many trade studies which could be conducted other than range, payload, and material.

Aircraft Design
AE 405

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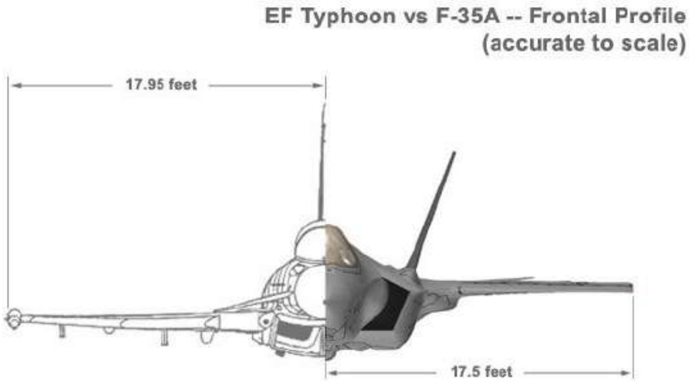
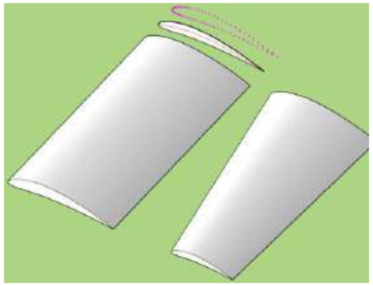
Contents



Chapter 4 – 5

- Airfoil and Geometry Selection
- Thrust-to-Weight Ratio and Wing Loading



Airfoil and Geometry Selection

- Before the design layout can be started, values for a number of parameters must be chosen;

  - Airfoil(s), the wing and tail geometries,
  - Wing loading, thrust-to-weight or horsepower-to-weight ratio,
  - Estimated takeoff gross weight and fuel weight,
  - Estimated wing, tail, and engine sizes, and the required fuselage size.

- This chapter covers selecting the airfoil and the wing and tail geometry, estimation of the required wing loading and thrust-to weight ratio (horsepower-to-weight ratio for a propeller aircraft).

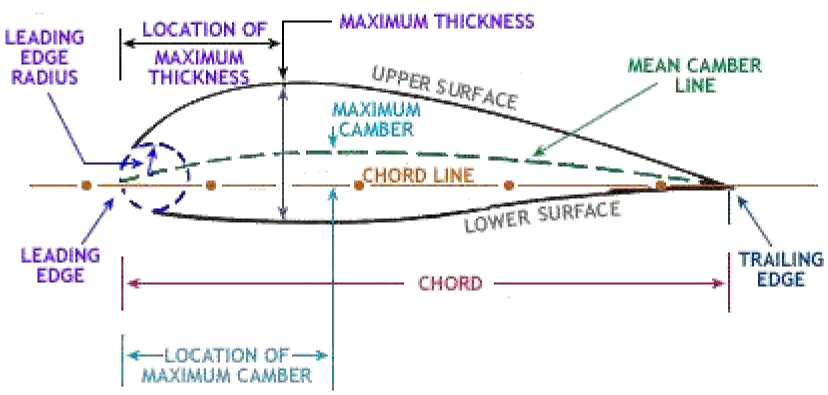
Airfoil and Geometry Selection

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.

- Figure illustrates the key geometric parameters of an airfoil.



Airfoil and Geometry Selection

- 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" *(t/c)* refers to the maximum thickness of the airfoil divided by its chord.
- "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.

Airfoil and Geometry Selection

- The airfoil section lift, drag, and pitching moment are defined in nondimensional form in Eqs. (4.1), (4.2), and (4.3).

- 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.

- Note that 2-D airfoil characteristics are denoted by lowercase subscripts

$$C_\ell = \frac{\text{section lift}}{qc} \tag{4.1}$$

$$C_d = \frac{\text{section drag}}{qc} \tag{4.2}$$

$$C_m = \frac{\text{section moment}}{qc^2} \tag{4.3}$$

c = chord length
q = dynamic pressure = $\rho V^2/2$
α = angle of attack

Airfoil and Geometry Selection

- The point about which the pitching moment remains constant for any angle of attack is called the "aerodynamic center."
- The center of pressure is usually behind the aerodynamic center.
- Location of the center of pressure varies with angle of attack for most airfoils.
- Airfoil characteristics are strongly affected by the "Reynolds number" at which they are operating.

Airfoil and Geometry Selection

- The Reynolds number influences whether the flow will be laminar or turbulent, and whether flow separation will occur.

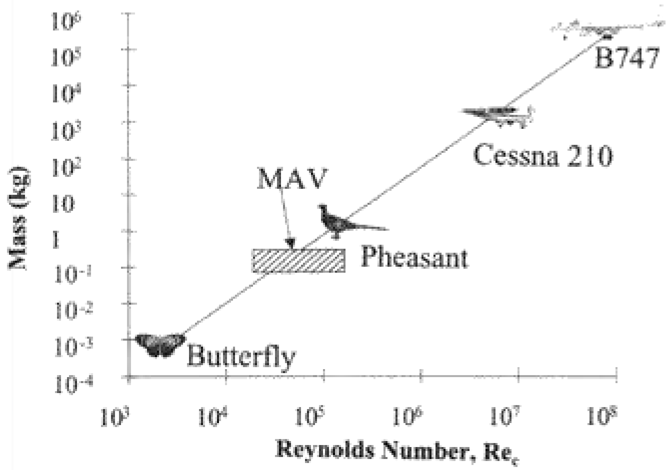
- A typical aircraft wing operates at a Reynolds number of about 10^6.

The Reynolds number is defined as[3]

$$\mathrm{Re} = \frac{\rho u L}{\mu} = \frac{u L}{\nu}$$

where:

- ρ is the density of the fluid (SI units: kg/m^3)
- u is the velocity of the fluid with respect to the object (m/s)
- L is a characteristic linear dimension (m)
- μ is the dynamic viscosity of the fluid (Pa·s or N·s/m^2 or kg/m·s)
- ν is the kinematic viscosity of the fluid (m^2/s).



Airfoil and Geometry Selection

- In several canarded homebuilt designs with laminar airfoils, entering a light rainfall will cause the canard's airflow to become turbulent,
- Reducing Canard’s lift and causing the aircraft to pitch downward.
- Earlier, nonlaminar airfoils were designed assuming turbulent flow at all times and do not experience this effect.

Airfoil and Geometry Selection

- A variety of airfoils is shown in Fig. 4.6.

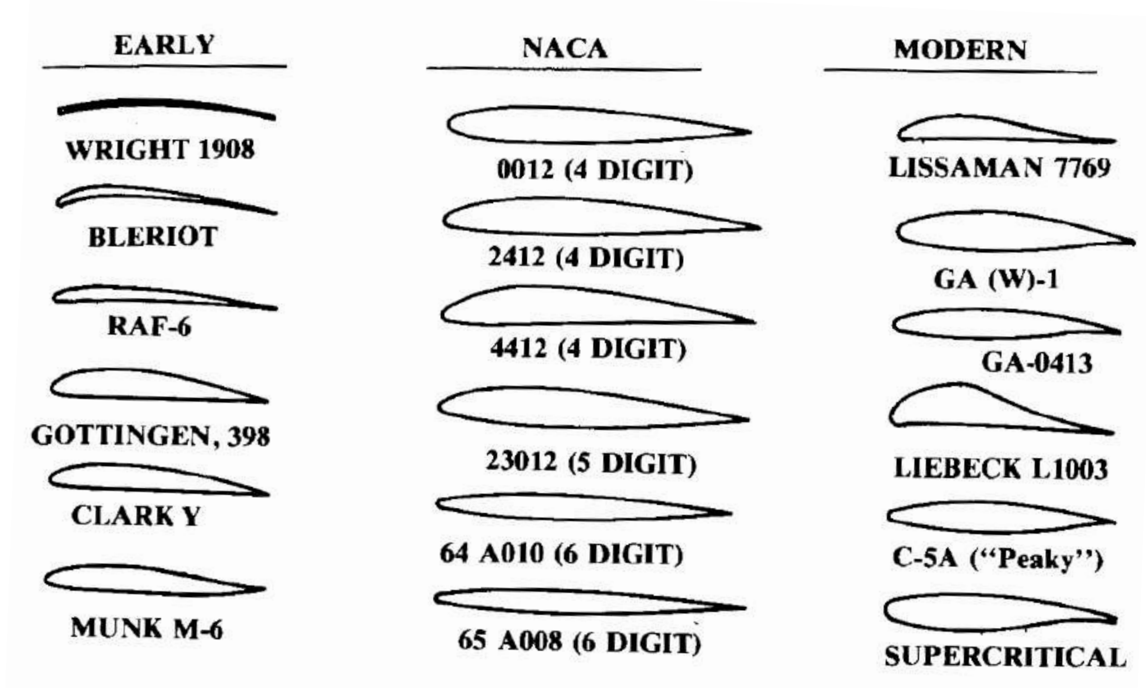
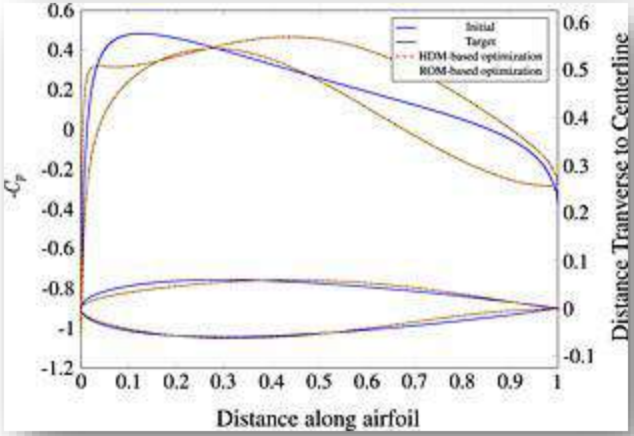


Fig. 4.6 Typical airfoils.

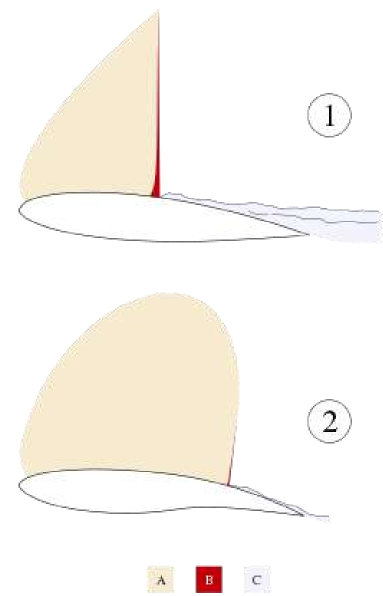
Airfoil and Geometry Selection

- 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.
- 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.



Airfoil and Geometry Selection

- Another consideration in modern airfoil design is the desire to maintain laminar flow over the greatest possible part of the airfoil.
- Modern computational methods allow design of airfoils in which the upper surface shock is minimized or even eliminated.
- A "supercritical" airfoil is one designed to minimize shock wave effects.
- This increases the Critical Mach Number.



Airfoil and Geometry Selection

Design Lift Coefficient

- The first consideration in initial airfoil selection is the design lift coefficient.

- As a first approximation, it can be assumed that the wing lift coefficient, C_L, equals the airfoil lift coefficient, C_l.

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

$$W = L = qSC_L \cong qSC_\ell \tag{4.4}$$

$$C_\ell = \frac{1}{q}\left(\frac{W}{S}\right) \tag{4.5}$$

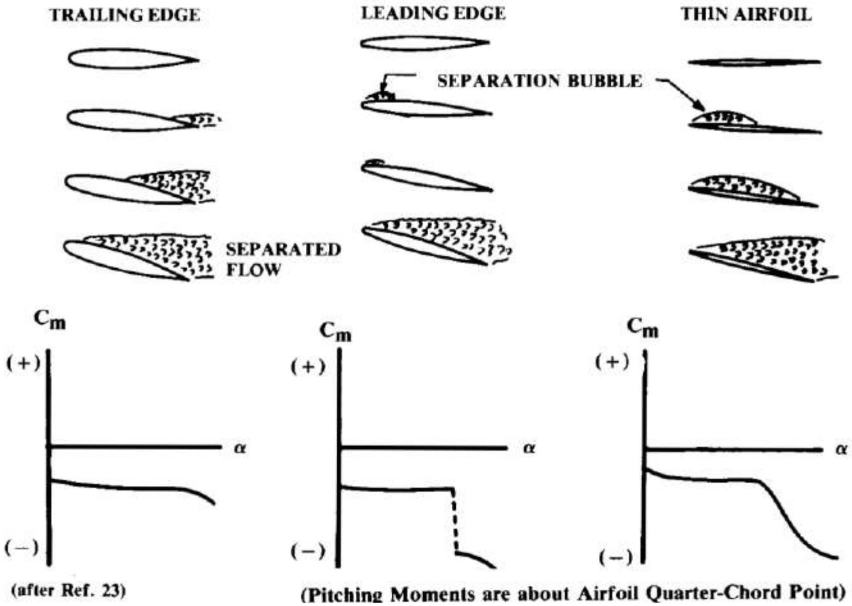
- Dynamic pressure *(q)* is a function of velocity and altitude.

- Assuming a wing loading (*W/S)* as described later, the design lift coefficient can be calculated for the velocity and altitude of the design mission.

Airfoil and Geometry Selection

Stall

- We have three types of stall characteristics; depending on fat, thinner, and thin airfoils



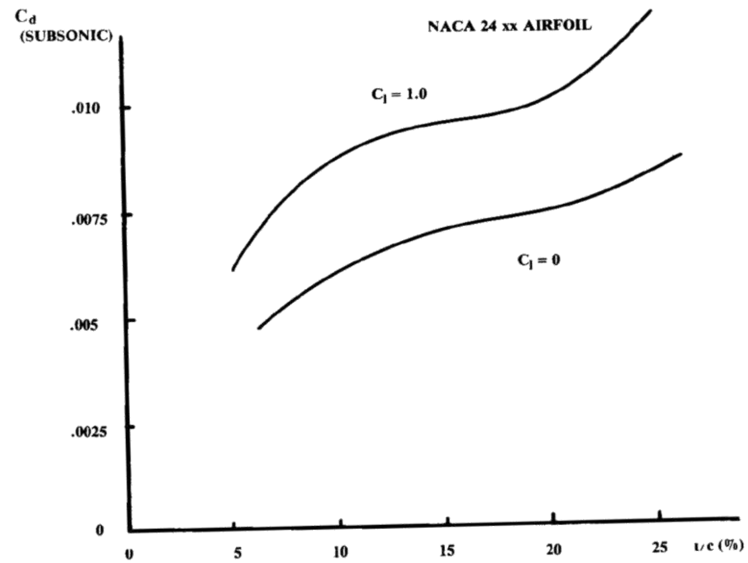
- However, wing stall is directly related to airfoil stall only for high-aspect-ratio, unswept wings.

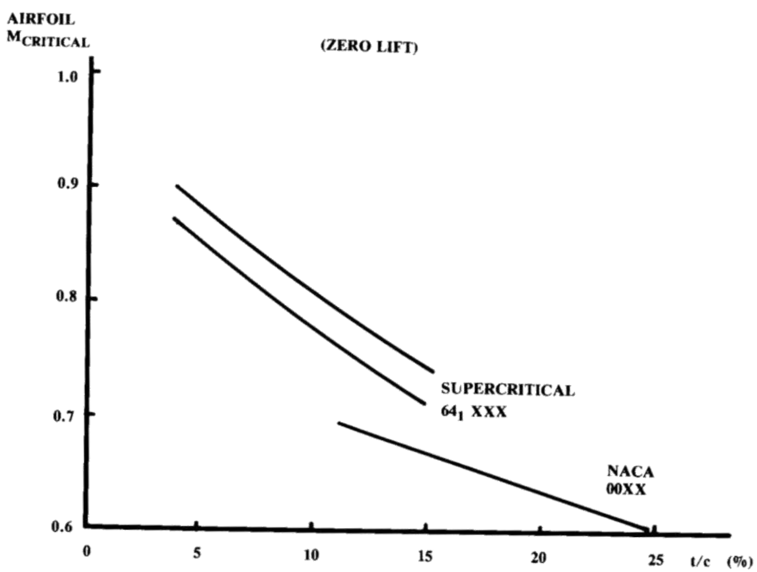
- For lower aspect ratio or highly swept wings the 3-D effects dominate stall characteristics, and airfoil stall characteristics can be essentially ignored in airfoil selection.

Airfoil and Geometry Selection

Airfoil Thickness Ratio

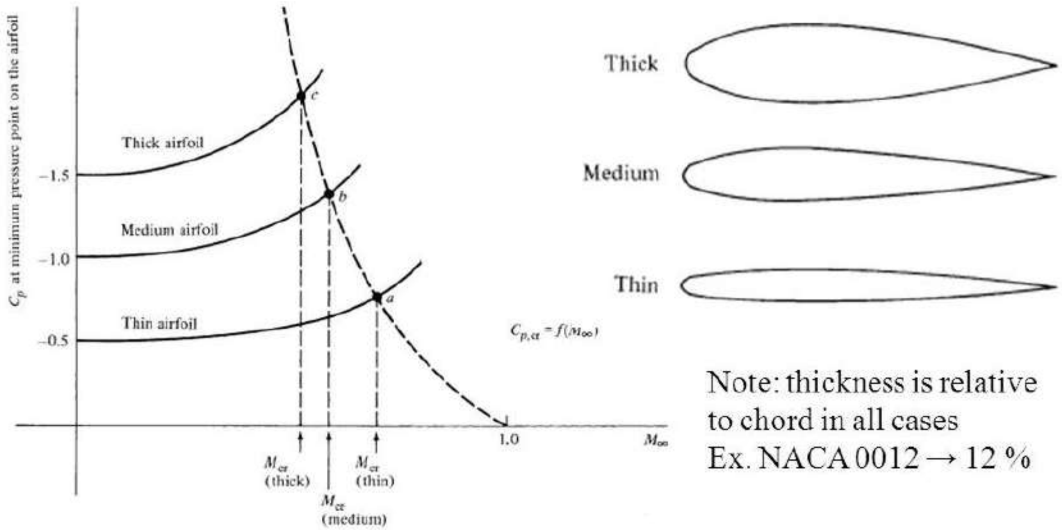
- Airfoil thickness ratio has a direct effect on drag, maximum lift, stall characteristics, and structural weight.
- Figure 4.11 illustrates the effect of thickness ratio on subsonic drag.





- Figure 4.12 shows the impact of thickness ratio on Critical Mach Number.
- the Mach number at which supersonic flow first appears over the wing.

Airfoil and Geometry Selection



Note: thickness is relative to chord in all cases
Ex. NACA 0012 → 12 %

- Thick airfoils have a lower critical Mach number than thin airfoils
- Desirable to have MCR as high as possible
- Implication for design → high speed wings usually design with thin airfoils
  - Supercritical airfoil is somewhat thicker

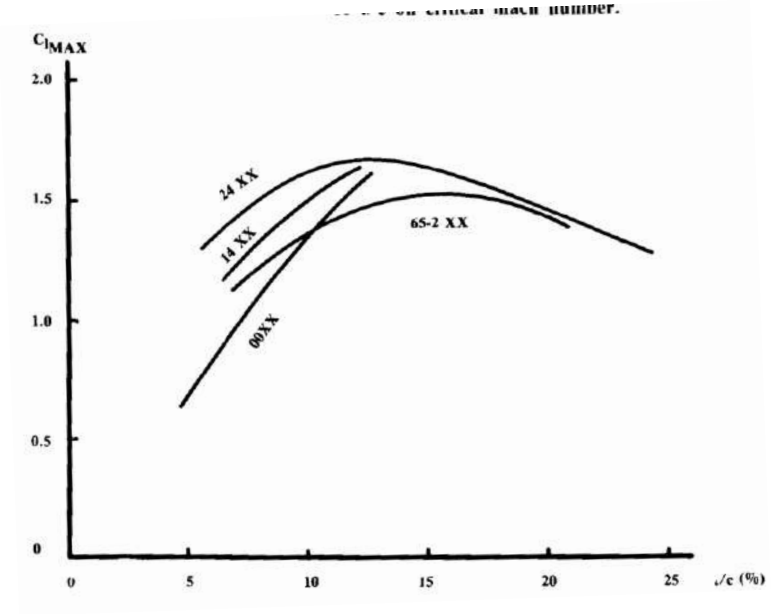
Airfoil and Geometry Selection

- 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, as shown in Fig. 4.13.

- The reverse is true for low-aspect-ratio, swept wings, such as a delta wing.

- Here, a sharper leading edge provides greater maximum lift due to the formation of vortices just behind the leading edge.



Airfoil and Geometry Selection

- Thickness also affects the structural weight of the wing.
- Statistical equations for wing weight show that the wing structural weight varies approximately inversely with the square root of the thickness ratio.
- The wing is typically about 15% of the total empty weight, so halving the thickness ratio would increase empty weight by about 6%.
- When applied to the sizing equation, this can have a major impact.

Airfoil and Geometry Selection

- For initial selection of the thickness ratio, the historical trend shown in Fig. 4.14 can be used.

- Note that a supercritical airfoil would tend to be about 10% thicker (i.e., conventional airfoil thickness ratio times 1.1) than the historical trend.

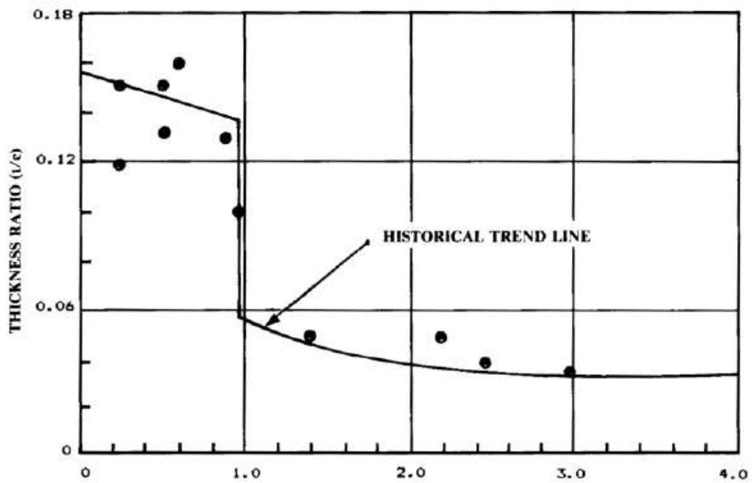
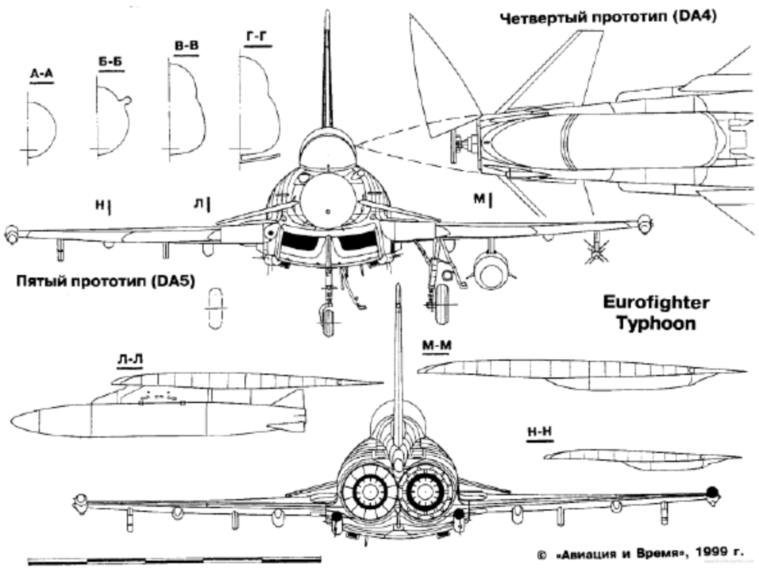


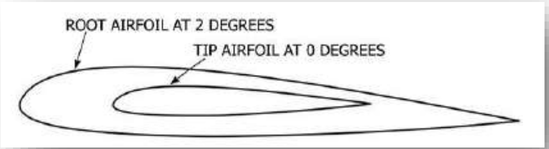
Fig. 4.14 Thickness ratio historical trend.

Airfoil and Geometry Selection

- Frequently the thickness is varied from root to tip.
- Due to fuselage effects the root airfoil of a subsonic aircraft can be as much as 20-60% thicker than the tip airfoil without greatly affecting the drag.



- This is very beneficial, resulting in a structural weight reduction as well as more volume for fuel and landing gear.
- This thicker root airfoil should extend to no more than about 30% of the span.



Airfoil and Geometry Selection

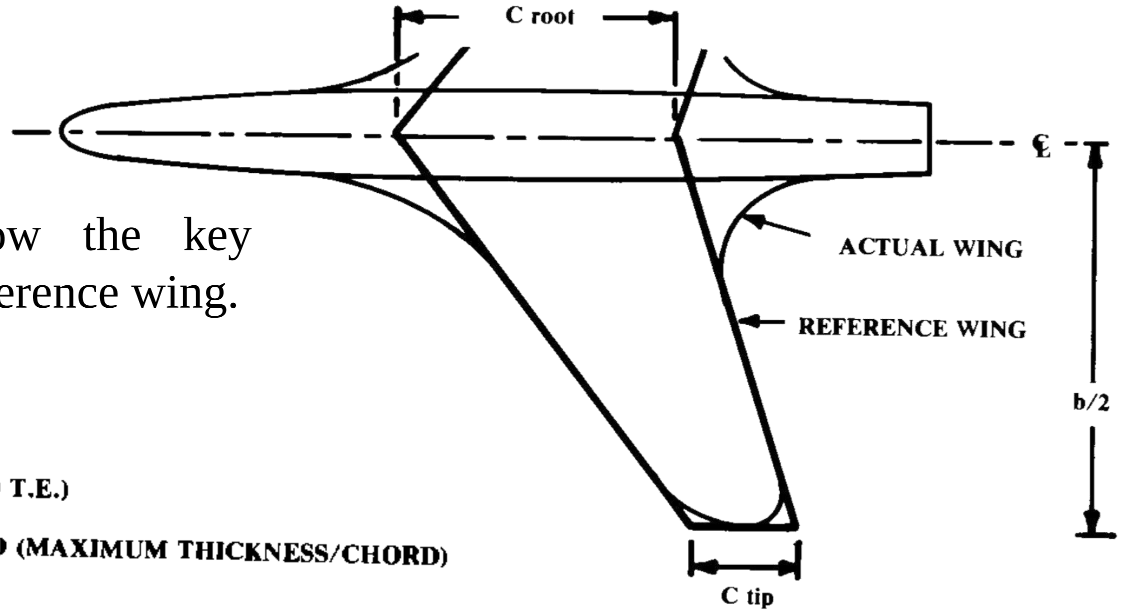
- Another important aspect of airfoil selection is the intended Reynolds number. Each airfoil is designed for a certain Reynolds number.
- Use of an airfoil at a greatly different Reynolds number (half an order of magnitude or so) can produce section characteristics much different from those expected.
- This is especially true for the laminar-flow airfoils, and is most crucial when an airfoil is operated at a lower than design Reynolds 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.

Airfoil and Geometry Selection

Wing Geometry

- The "reference" ("trapezoidal") wing is the basic wing geometry used to begin the layout.

- Figures 4.15 and 4.16 show the key geometric parameters of the reference wing.



S	= REFERENCE WING AREA
C	= CHORD (DISTANCE L.E. TO T.E.)
A	= ASPECT RATIO = b^2/S
t/c	= AIRFOIL THICKNESS RATIO (MAXIMUM THICKNESS/CHORD)
λ	= TAPER RATIO = C tip/C root
b	= SPAN

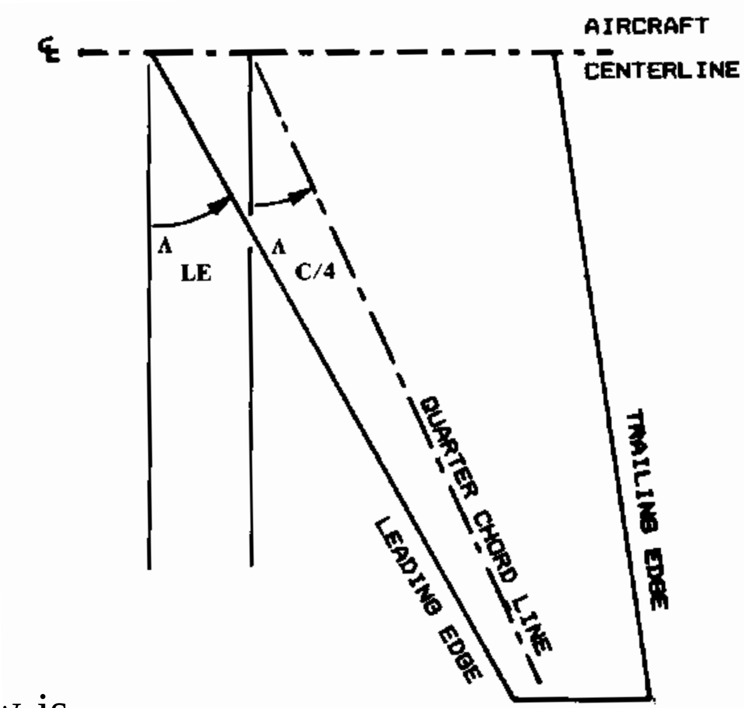
GIVEN: W/S, A, λ

$S = W/(W/S)$ $b = \sqrt{A \cdot S}$ $C_{root} = 2 \cdot S/[b(1+\lambda)]$ $C_{tip} = \lambda \cdot C_{root}$

Fig. 4.15 Wing geometry.

Airfoil and Geometry Selection

- There are two key sweep angles, as shown in Fig. 4.16.
- 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.
- 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").



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

Fig. 4.16 Wing sweep Λ.

Airfoil and Geometry Selection

- In a similar fashion, such a point is defined for the complete trapezoidal wing and is based on the concept of the "mean aerodynamic chord."

- The mean aerodynamic chord (Fig. 4.17) is the chord $\bar{c}$ of an airfoil, located at some distance $\bar{Y}$ from the centerline.

- The designer uses the mean aerodynamic chord and the resulting aerodynamic center point to position the wing properly.

- Figure 4.17 illustrates a graphical method for finding the mean aerodynamic chord of a trapezoidal-wing planform.

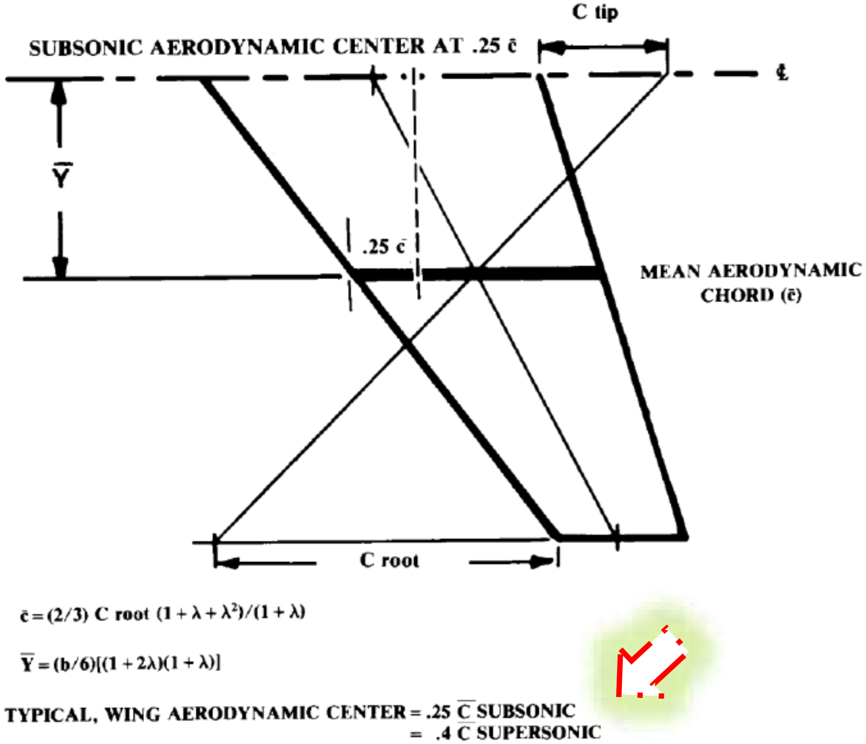


Fig. 4.17 Mean aerodynamic chord.

Airfoil and Geometry Selection

- The required reference wing area ("S") can be determined only after the takeoff gross weight is determined.

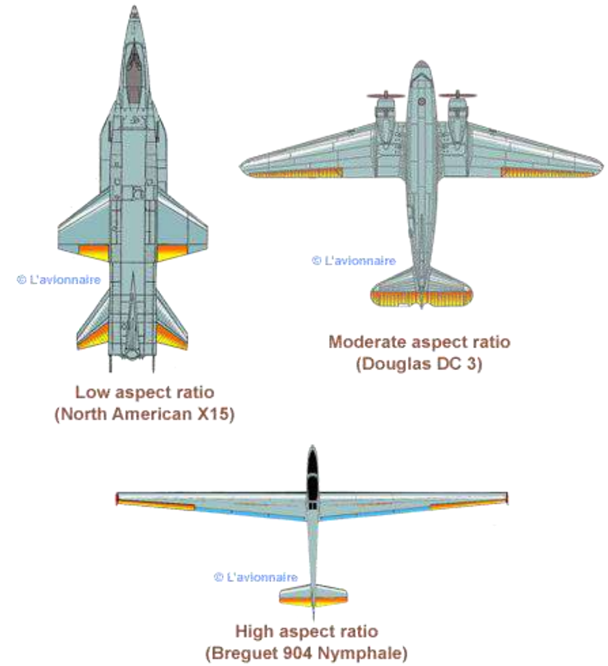
- The shape of the reference wing is determined by its aspect ratio, taper ratio, and sweep.

Aspect Ratio $$AR = \frac{b^2}{S}$$

- 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. However, for a real, 3-D wing, the air can escape around the wing tip.



Airfoil and Geometry Selection

- Air escaping around the wing tip lowers the pressure difference between the upper and the lower surfaces.

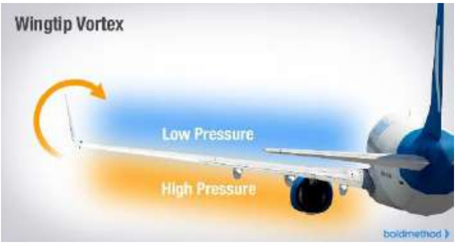
- This reduces lift near the tip.

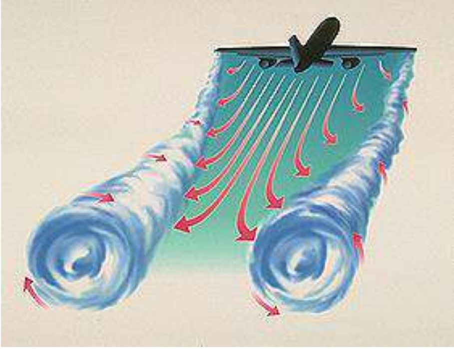
- Also, the air flowing around the tip flows in a circular path when seen from the front.

- 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.

- 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.



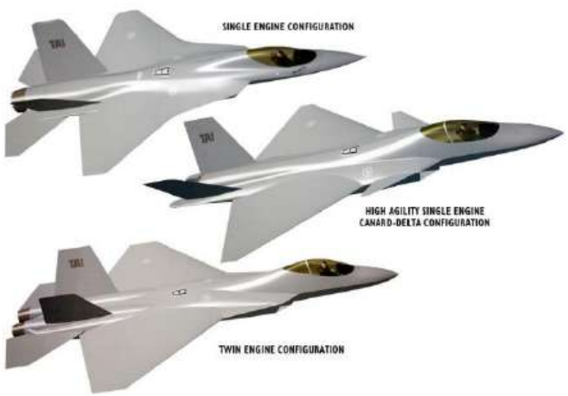


Airfoil and Geometry Selection

- 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. 4.18).

- 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.



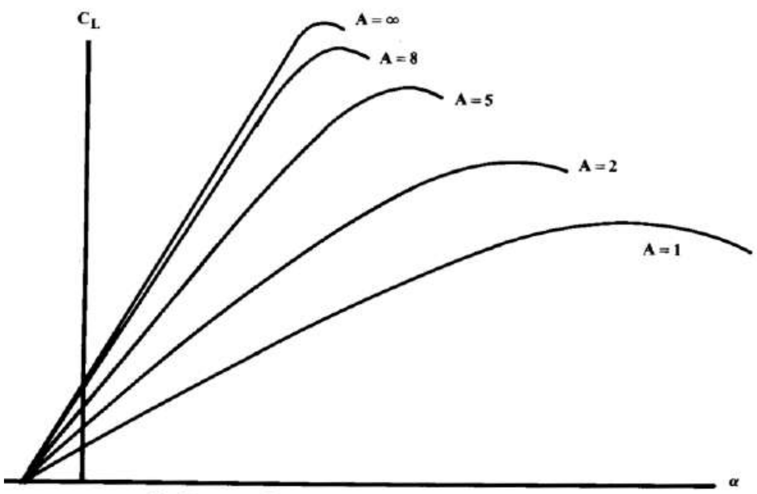


Fig. 4.18 Effect of aspect ratio on lift.

Airfoil and Geometry Selection

- Later in the design process, the aspect ratio will be determined by a trade study in which the aerodynamic advantages of a higher aspect ratio are balanced against the increased weight.

- For initial wing layout the values and equations. provided in Table 4.1 can be used.

- For the canard design, typically, the canard will have about 10-25% of the total lifting area.

- So the wing aspect ratio becomes the statistically determined aspect ratio divided by 0.9-0.75.

Airfoil and Geometry Selection

Table 4.1 Aspect ratio

Sailplane equivalent* aspect ratio $= 4.464$ (best $L/D)^{.69}$

Propeller aircraft	Equivalent aspect ratio	
Homebuilt	6.0	
General aviation—single engine	7.6	
General aviation—twin engine	7.8	
Agricultural aircraft	7.5	
Twin turboprop	9.2	
Flying boat	8.0	
	Equivalent aspect Ratio $= aM_{max}^{C}$	
Jet aircraft	a	C
Jet trainer	4.737	−0.979
Jet fighter (dogfighter)	5.416	−0.622
Jet fighter (other)	4.110	−0.622
Military cargo/bomber	5.570	−1.075
Jet transport	7.50	0

*Equivalent aspect ratio = wing span squared/(wing and canard areas)

Airfoil and Geometry Selection

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.

- At supersonic speeds the loss of lift associated with supersonic flow can be reduced by sweeping the wing.

Airfoil and Geometry Selection

- Figure 4.19 shows a historical trend line for wing leading-edge sweep vs Mach number.

- Note that sweep is defined aft of a line perpendicular to the flight direction, while the Mach angle is defined with respect to the flight direction.

- Thus, the line labeled "90-arcsin(l/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.

- In the high-speed range, it becomes structurally impractical to sweep the wing past the Mach cone.

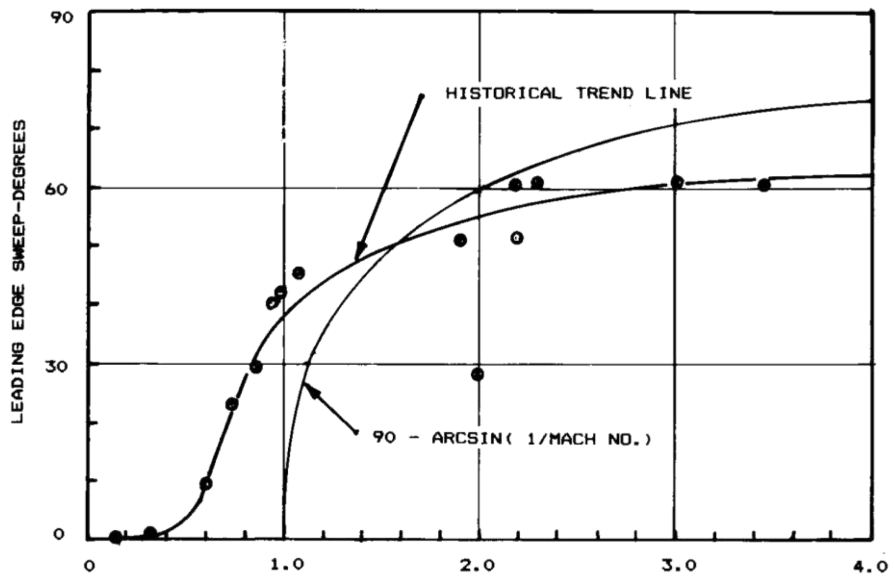


Fig. 4.19 Wing sweep historical trend.

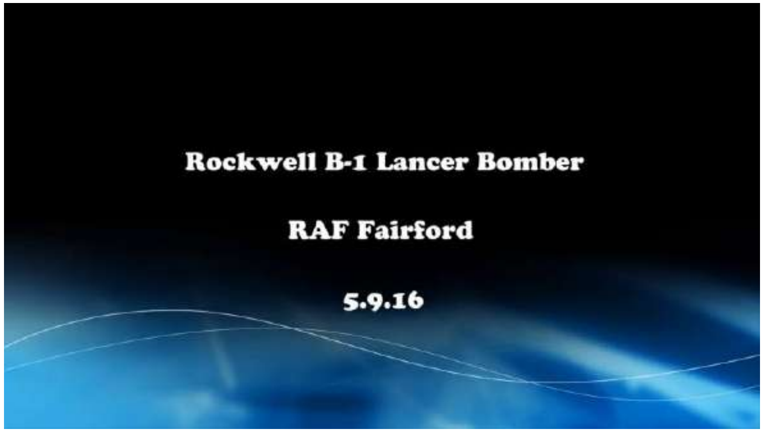
Airfoil and Geometry Selection

- There is no theoretical difference between sweeping a wing aft and sweeping it forward.
- There are other reasons for sweeping a wing; weight balance, better stability…
- The wing sweep and aspect ratio together have a strong effect on the wing-alone pitchup characteristics.
- "Pitchup" 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-16 fighter requires a computerized angle-of-attack limiter to prevent a severe pitchup problem at about 25-deg angle of attack.

Airfoil and Geometry Selection

- Figure 4.20 describes boundaries for pitchup avoidance for combinations of wing quarter chord sweep angle and aspect ratio.

- Pitchup avoidance should be considered for military fighters, aerobatic aircraft, general-aviation aircraft, and trainers.



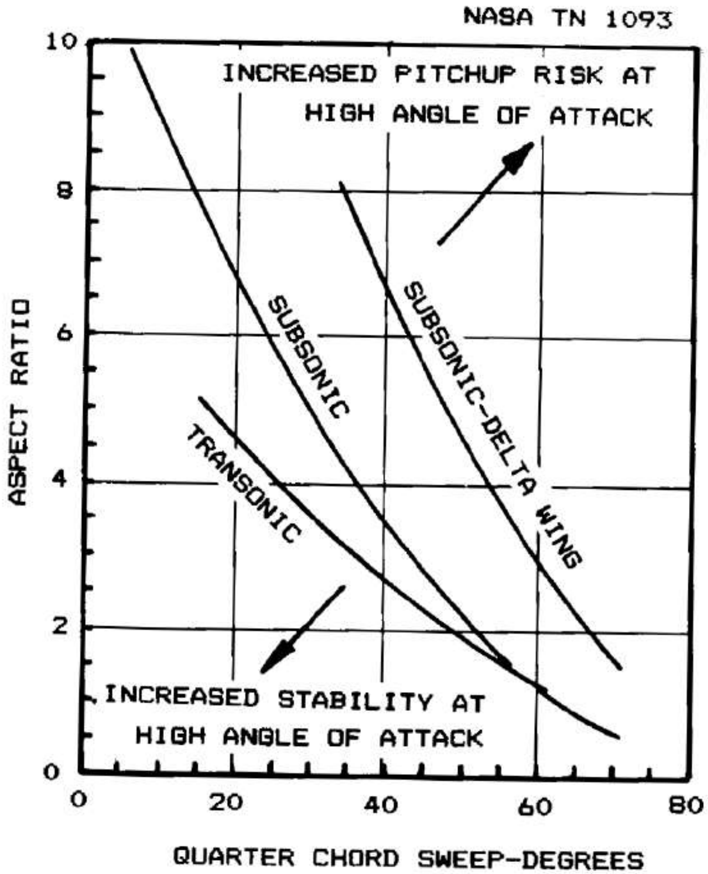


Fig. 4.20 Tail-off pitchup boundaries.

Airfoil and Geometry Selection

- For high-speed flight, a swept wing is desirable. For cruise as well as takeoff and landing, an unswept wing is desirable.
- A wing of variable sweep would offer the best of both worlds.
- Controlling the balance of a variable sweep aircraft is a major design problem.
- To balance the aircraft, either fuel must be pumped to move the center of gravity, or the tail must provide a tremendous down-load (or both).
- Another problem with the variable-sweep wing is the weight penalty associated with the pivot mechanism: a 19% increase in the weight of the wing itself if it has variable sweep.

F-111

B-1B

Tu-160

Airfoil and Geometry Selection

Taper Ratio

- Wing taper ratio, λ, 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.
- The most efficient form, an elliptical wing planform is difficult and expensive to build.
- The easiest wing to build is the untapered (λ=1.0) rectangular wing.
- However, an untwisted rectangular wing has about 7% more drag due to lift than an elliptical wing of the same aspect ratio.

Airfoil and Geometry Selection

- A taper ratio of 0.45 for an unswept wing produces a lift distribution very close to the elliptical ideal (Fig. 4.22).
- This results in a drag due to lift less than I% higher than the ideal, elliptical wing.

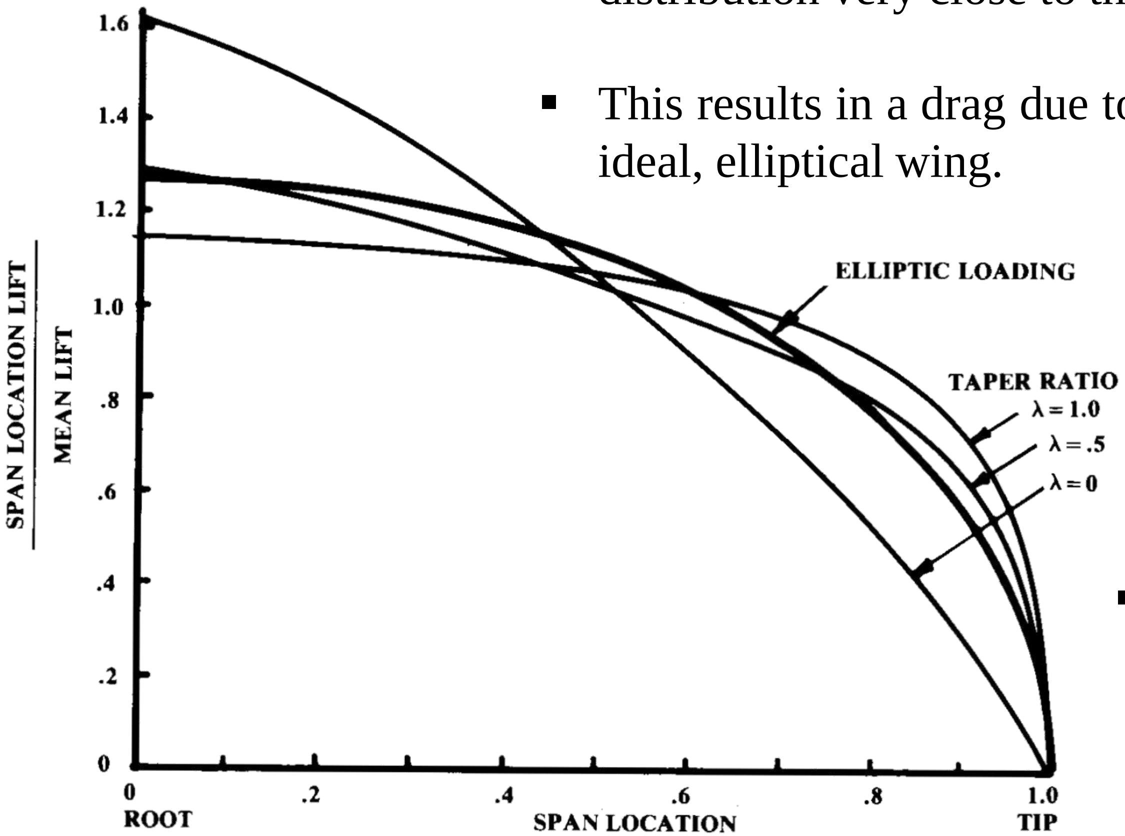


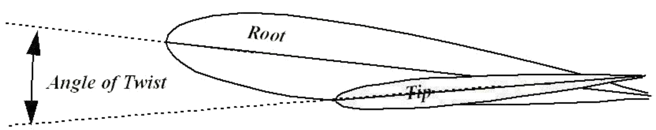
Fig. 4.22 Effect of taper on lift distribution.

- For a wing swept, it is necessary to increase the amount of taper (reduce the taper ratio).

Airfoil and Geometry Selection

- 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.

Wing Twists

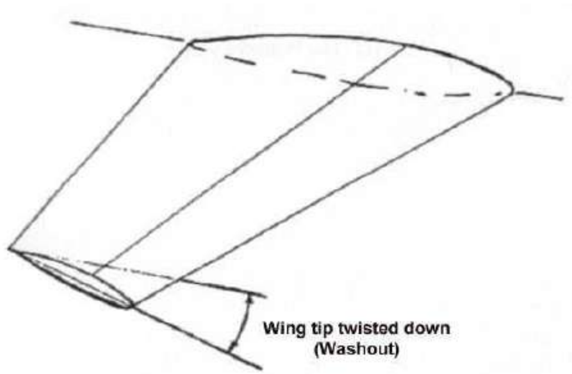


Geometric Twist: Angles of attack at wing root and wing tip are different.

Washout: Angle of attack at tip lower than that at root. Used to reduce tip vortices.

Washin: Angle of attack at tip larger than that at root.

Aerodynamic Twist: Airfoil shape varies from wing root to wing tip.



- "Aerodynamic twist" is the angle between the zero-lift angle of an airfoil and the zero-lift angle of the root airfoil.
- For initial design purposes, historical data should be used. Typically, 3 deg of twist provides adequate stall characteristics.

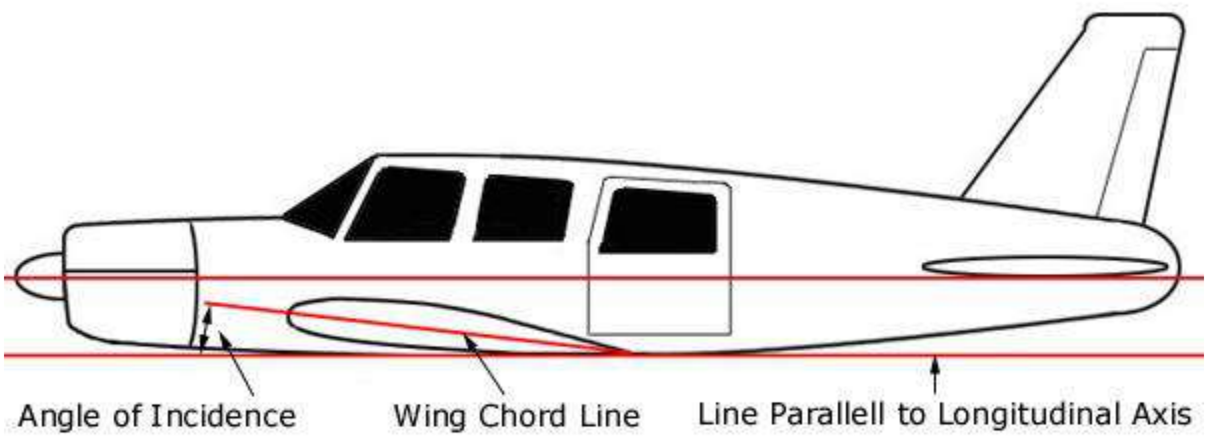
Airfoil and Geometry Selection

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 chordlines.
- If the wing is twisted, the incidence is defined with respect to some arbitrarily chosen spanwise 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.

Airfoil and Geometry Selection

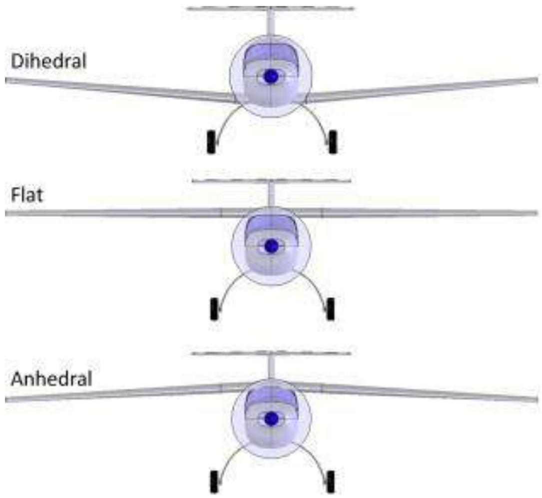
- It can be assumed that general aviation and homebuilt aircraft will have an incidence of about 2 deg, transport aircraft about 1 deg, and military aircraft approximately zero.



Airfoil and Geometry Selection

Dihedral

- 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.
- Table 4.2 provides initial estimates of dihedral.



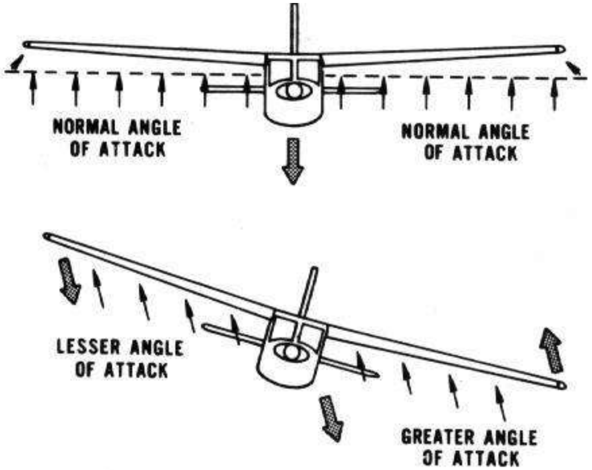


Figure 17–29 Dihedral for Lateral Stability

Airfoil and Geometry Selection

Table 4.2 Dihedral guidelines

	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

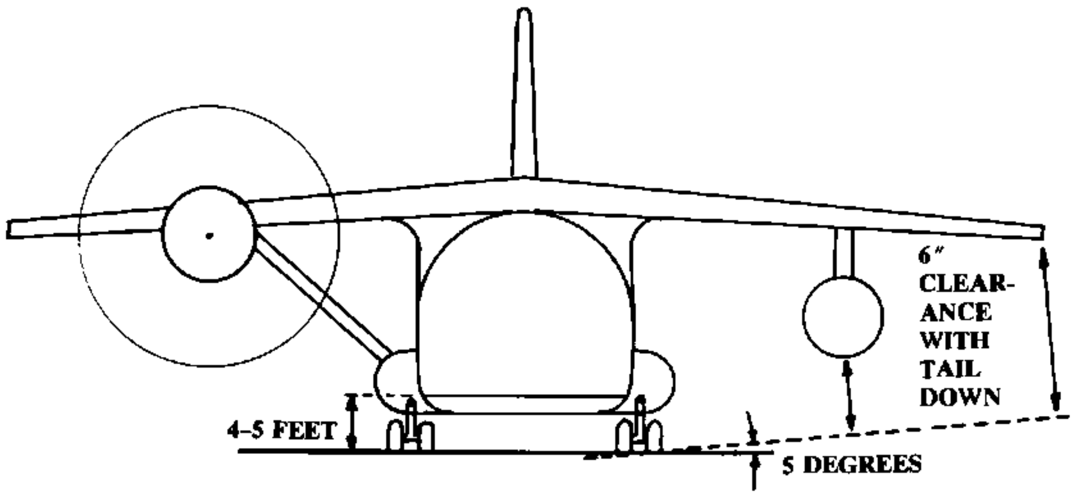
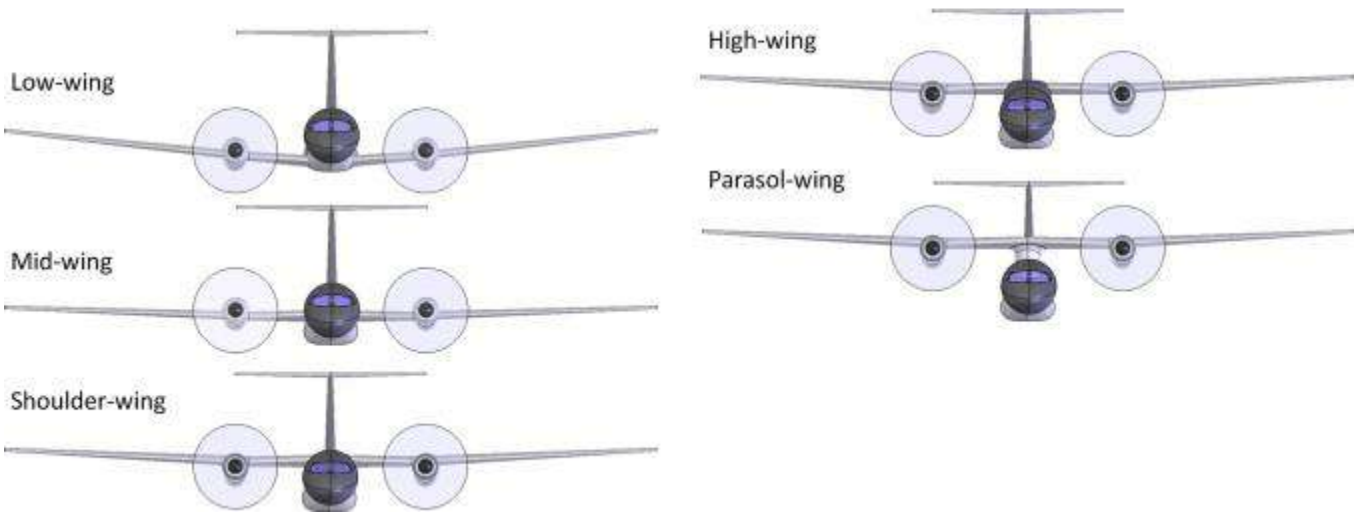


Fig. 4.24 High wing.

Airfoil and Geometry Selection

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.



Airfoil and Geometry Selection

- For example, virtually all high-speed commercial transport aircraft are of low-wing design.

- Military transport aircraft designed to similar mission profiles and payload weights are all of high-wing design.

- The major benefit of a high wing is that it allows placing the fuselage closer to the ground (Fig. 4.24).

- For military transport aircraft such as the C-5 and C-141, this allows loading and unloading the cargo without special ground handling gear.

Airfoil and Geometry Selection

- With a high wing, jet engines or propellers will have sufficient ground clearance without excessive landing-gear length.

- Another structural benefit occurs if the wing box is carried over the top of the fuselage rather than passing through it.

- When the wing box passes through the fuselage, the fuselage must be stiffened around the cut-out area.

- This adds weight to the fuselage. However, passing the wing box over the fuselage will increase drag due to the increase in frontal area.

Airfoil and Geometry Selection

- For an aircraft designed with short takeoff and landing (STOL) requirements, a high wing offers several advantages.

- The high position allows room for the very large wing flaps needed for a high lift coefficient.

Airfoil and Geometry Selection

- There are several disadvantages to the high-wing arrangement.
- While landing-gear weight tends to be lower than other arrangements, the fuselage weight is usually increased because it must be strengthened to support the landing-gear loads.
- The fuselage is also usually flattened at the bottom to provide the desired cargo-floor height above ground.
- This flattened bottom is heavier than the optimal circular fuselage.
- For small aircraft, the high wing arrangement can block the pilot's visibility

Airfoil and Geometry Selection

- If the fuselage is roughly circular and fairings are not used, the mid-wing arrangement (Fig. 4.25) provides the lowest drag.

- High- and low-wing arrangements must use fairings to attain acceptable interference drag with a circular fus

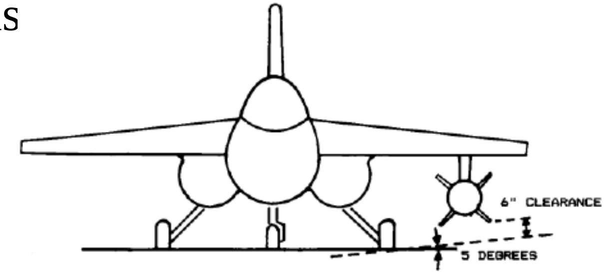


Fig. 4.25 Mid wing.

- Structural carrythrough presents the major problem with the mid wing.

- The major advantage of the low-wing approach (Fig. 4.26) 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

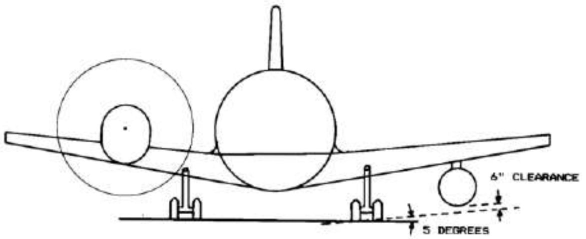


Fig. 4.26 Low wing.

Airfoil and Geometry Selection

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.
- A far more important effect is the influence the tip shape has upon the lateral spacing of the tip vortices.
- An obvious way to prevent induced drag would be to mount a vertical plate at the wing tip.
- One problem with winglets is that they add weight behind the elastic axis of the wing, which can aggravate flutter tendencies.

Airfoil and Geometry Selection

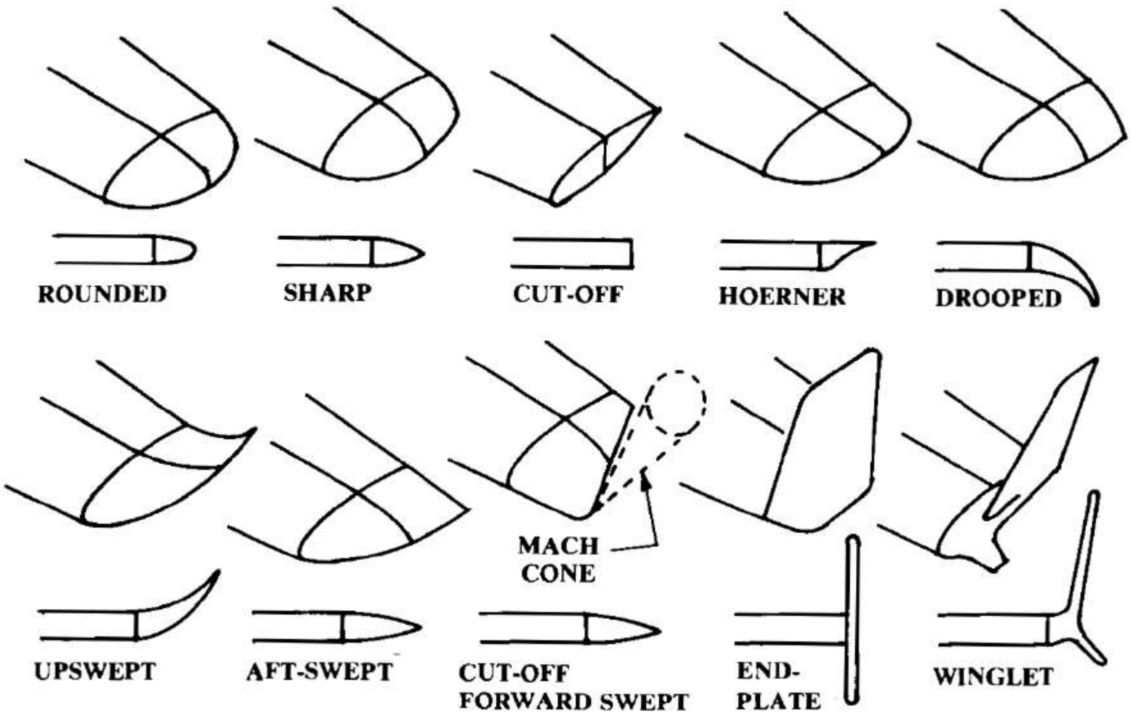


Fig. 4.27 Wing tips.

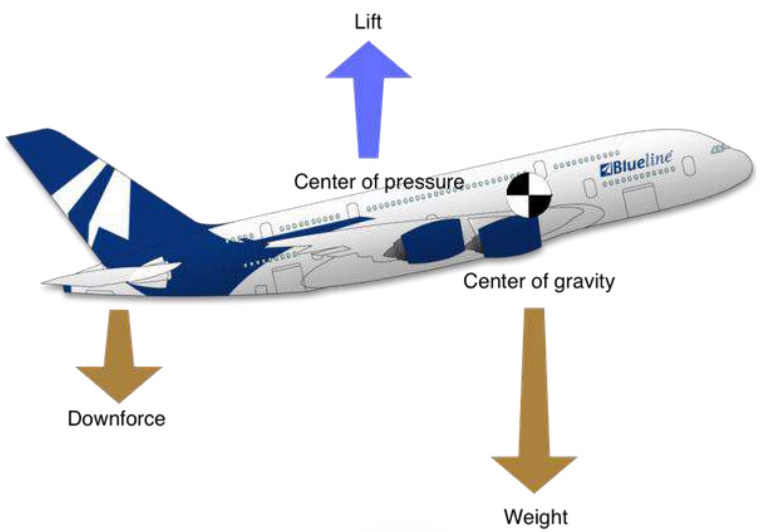
Boeing 777 Folding Wingtips

Boeing 737 Max

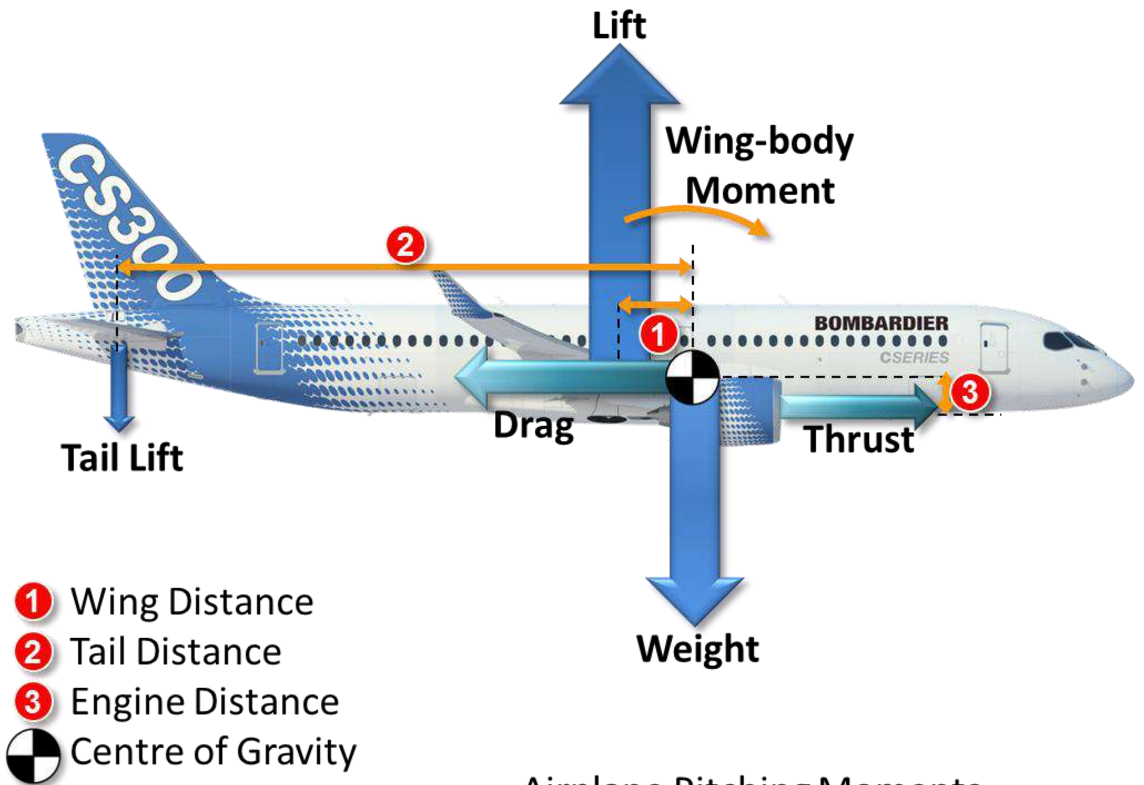
Airfoil and Geometry Selection

Tail Geometry & Arrangement

- Tails are little wings. Much of the previous discussion concerning wings can also be applied to tail surfaces.
- 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 other moment produced by the aircraft.



Airfoil and Geometry Selection



Airplane Pitching Moments

Airfoil and Geometry Selection

- 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-engined aircraft must be capable of providing a sufficient trim force in the event of an engine failure.

Airfoil and Geometry Selection

- A single-engine propeller tends to "drag" the air into a rotational motion in the same direction that the propeller spins.

- Since the vertical tail is above the fuselage, it will be pushed on by the rotating propwash, causing a nose-left motion for the normal direction of engine rotation.

- To counter this, some single-engine propeller airplanes have the vertical tail offset several degrees.

Airfoil and Geometry Selection

- The other major function of the tail is control.
- The tail must be sized to provide adequate control power at all critical conditions.
- These critical conditions for the horizontal tail or canard typically include nosewheel liftoff, low-speed flight with flaps down, and transonic maneuvering.

- For the vertical tail, critical conditions typically include engine-out flight at low speeds, maximum roll rate, and spin recovery.
- Note that control power depends upon the size and type of the movable surface as well as the overall size of the tail itself.

Airfoil and Geometry Selection

Tail Arrangement

- Figure 4.28 illustrates some of the possible variations in aft-tail arrangement.

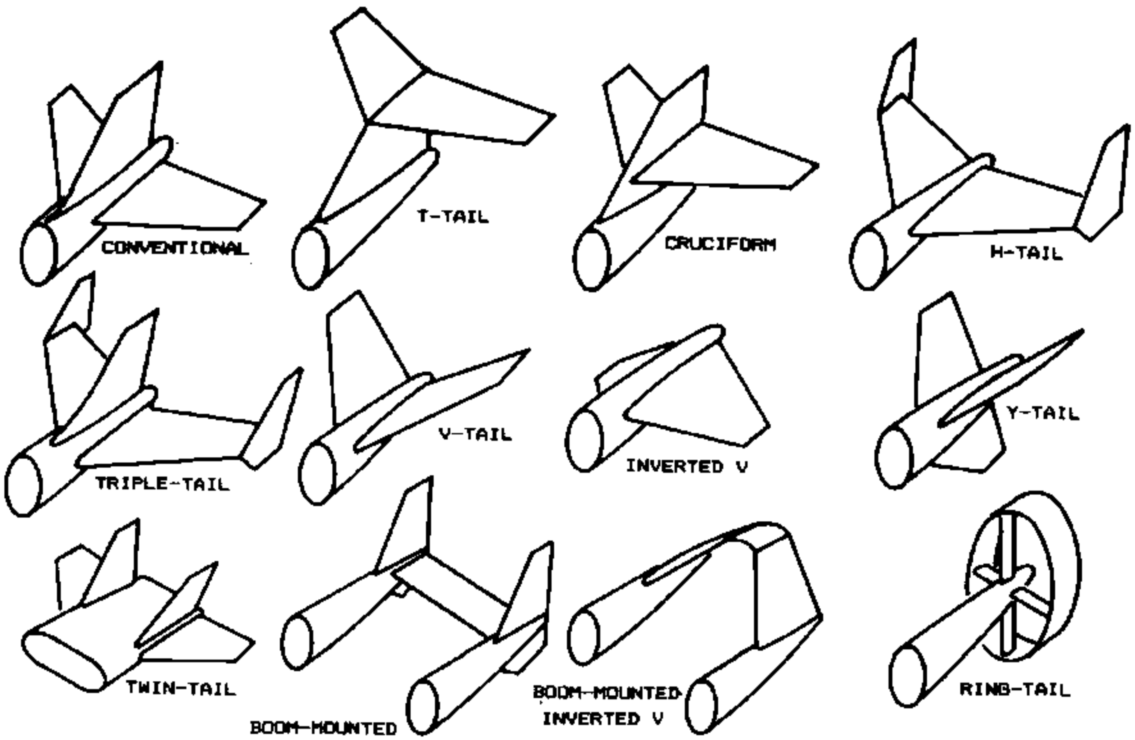


Fig. 4.28 Aft tail variations.

Airfoil and Geometry Selection

- The first shown has become "conventional" for the simple reason that it works.
- For most aircraft designs, the conventional tail will usually provide adequate stability and control at the lightest weight.
- Probably 70% or more of the aircraft in service have such a tail arrangement.
- The "T-tail" is also widely used. A T-tail is inherently heavier than a conventional tail because the vertical tail must be strengthened to support the horizontal tail.
- The T-tail lifts the horizontal tail clear of the wing wake and propwash, which makes it more efficient and hence allows reducing its size.

Airfoil and Geometry Selection

- This also reduces buffet on the horizontal tail, which reduces fatigue for both the structure and the pilot.

- The "H-tail" is used primarily to position the vertical tails in undisturbed air or to position the rudders in the propwash on a multiengine aircraft to enhance engine-out control.

AN-225

- The H-tail is heavier than the conventional tail, but its endplate effect allows a smaller horizontal tail.

- On the A-10, the H-tail serves to hide the hot engine nozzles from heatseeking missiles when viewed from an angle off the rear of the aircraft.

A-10

- H-tails and the related triple-tails have also been used to lower the tail height to allow an aircraft such as the Lockheed Constellation to fit into existing hangars.

Airfoil and Geometry Selection

- The "V-tail" is intended to reduce wetted area.
- However, to obtain satisfactory stability and control, the V surfaces must be upsized to about the same total area as would be required for separate horizontal and vertical surfaces.

Predator Avenger

the proper movement of the V-tail "ruddervators

Airfoil and Geometry Selection

- The inverted V-tail avoids stability problems, but this tail arrangement can cause difficulties in providing adequate ground clearance.

MQ-1 Predator

- 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.

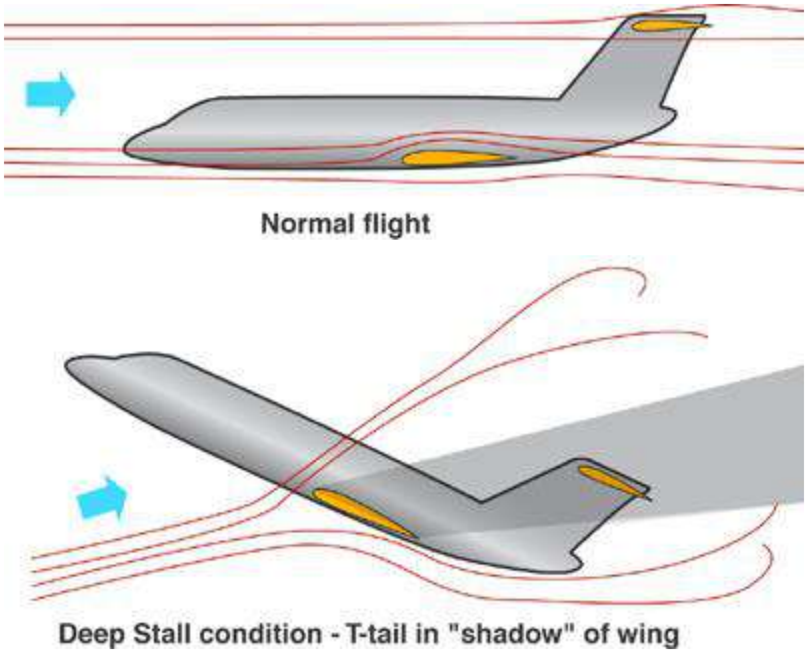
J-20

- Twin tails are usually heavier than an equal-area centerline-mounted single tail, but are often more effective.
- Twin tails are seen on most large modern fighters such as the F-14, F-15, F-18, and MiG-25.

Airfoil and Geometry Selection

- The location of an aft horizontal tail with respect to the wing is critical to the stall characteristics of the aircraft.

- If the tail enters the wing wake during the stall, control will be lost and pitchup may be encountered.



- Several T-tailed aircraft encountered "deep stall" from which they could not be Extricated/freed.

- Figure 4.29 illustrates the boundaries of the acceptable locations for a horizontal tail to avoid this problem. Note that low tails are best for stall recovery.

Airfoil and Geometry Selection

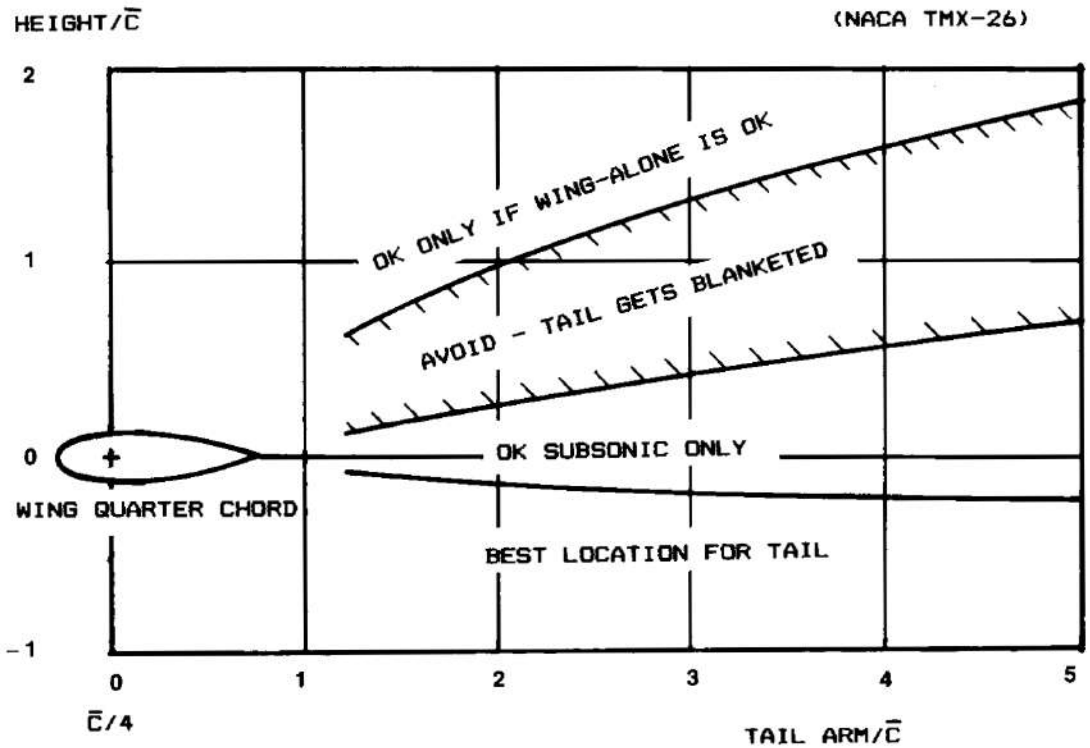


Fig. 4.29 Aft tail positioning.

Airfoil and Geometry Selection

- Other possible tail arrangements are depicted in Fig. 4.30.
- Canards were used by the Wright brothers, but fell out of favor due to the difficulty of prov1dmg sufficient stab1hty.

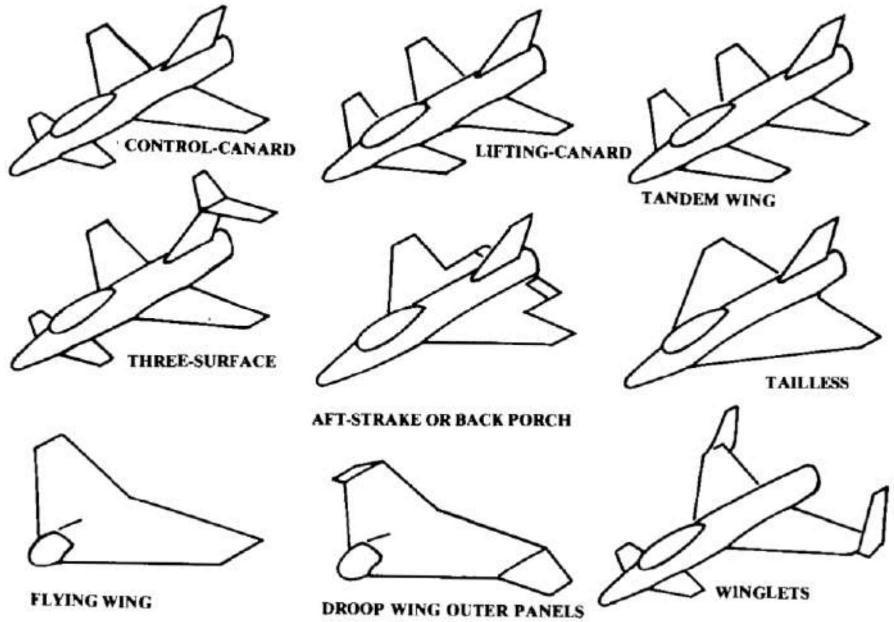


Fig. 4.30 Other tail configurations.

Airfoil and Geometry Selection

- There are actually two distinct classes of canard: the control-canard and the lifting-canard.
- In the control-canard, the wing carries most of the lift, and the canard is used primarily for control (as is an aft tail).
- Both the Wright Flyer and the Grumman X-29 are of this type.
  - In contrast, a lifting-canard aircraft uses both the wing and the canard to provide lift under normal flight conditions.
  - This requires that the aircraft center of gravity be well forward of the normal location with respect to the wing when compared to an aft-tailed aircraft.
  - A lifting-canard will usually have a higher aspect ratio and greater airfoil camber than a control-canard to reduce the canard's drag-due-to-lift.

Airfoil and Geometry Selection

- The lifting-canard arrangement is theoretically more efficient than an aft-tailed aircraft.
- Because the canard's lift reduces the lift that must be produced by the wing, which permits a smaller wing and also reduces total drag-due-to-lift.
- An aft-tail design frequently flies with a download on the tail to produce stability, which actually increases the amount of lift that the wing must produce.
- However, the lifting-canard suffers from several drawbacks that reduce the net benefit: stability and control (stall recovery) problems…

Airfoil and Geometry Selection

- On a Cessna 172, if the wing stalls before the tail, you'll still have elevator controllability to pitch down.

- If the tail stalls before the wing, the aircraft will naturally pitch down. In either case, stall recovery is natural.

- However, if your aircraft has a canard instead of a tail-mounted horizontal stabilizer, you're in real trouble if the wing stalls first.

- In this case, the center of gravity would drop the wing and tail, pitching the nose up. The aircraft now enters a deeper stall and becomes unrecoverable

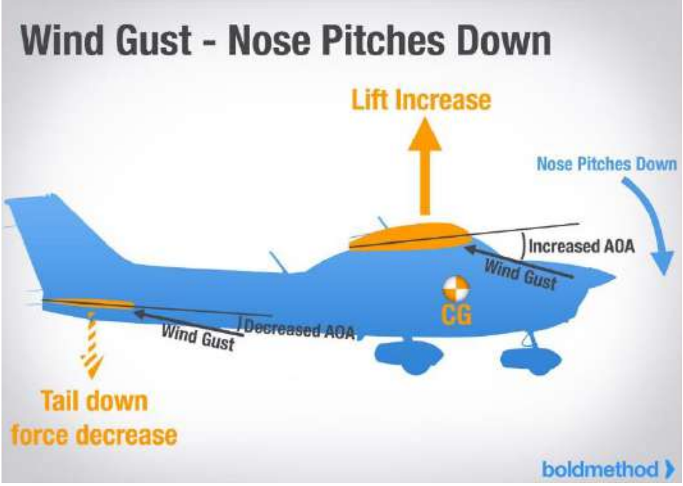
Airfoil and Geometry Selection

- Canards can also make an airplane unstable.

- If a wind gust briefly increases the angle of attack on a Cessna 172, the aircraft tends to pitch nose down and return to it's original attitude.

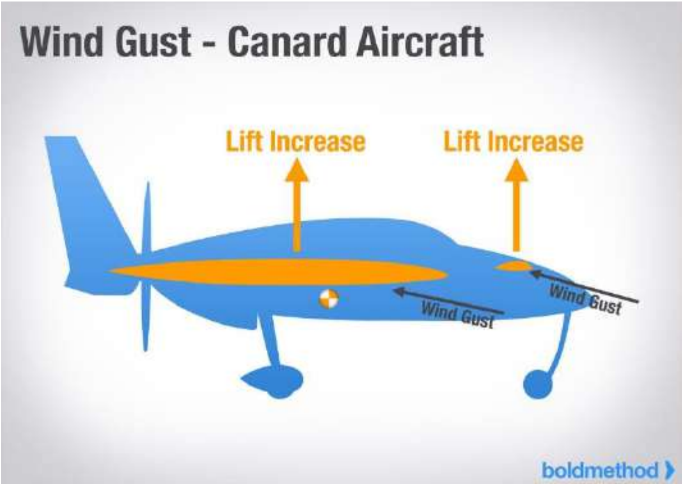
- In the Cessna's case, the increased angle of attack increases the wing's lift.

- However, it actually decreases the tail down force, because it decreases the horizontal stabilizer's angle of attack.



Airfoil and Geometry Selection

- However, the canard can actually make your aircraft pitch up further.
- The increase in angle of attack causes both the canard and the wing to generate more lift.
- If the canard's increase in lift is greater than the wing's, the nose will pitch further up.



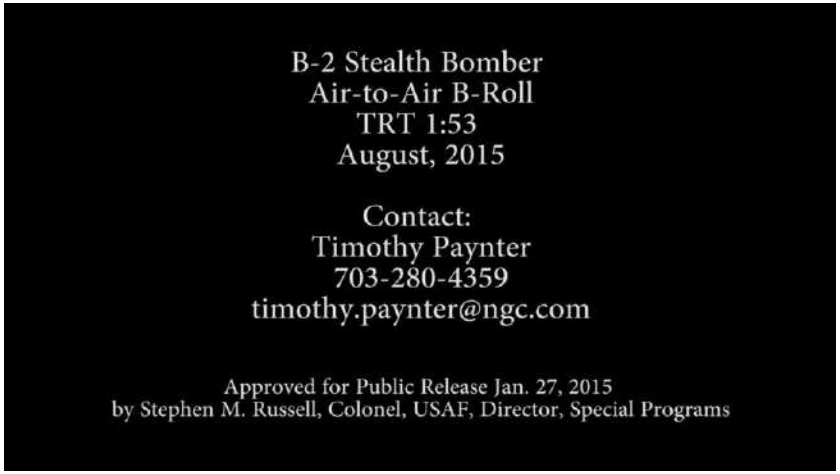
Airfoil and Geometry Selection

- The tailless configuration offers the lowest weight and drag of any tail configuration, if it can be made to work.

- However, the fully tailless (flying-wing) design is probably the most difficult configuration to stabilize, either naturally or by computer.

- Fully-tailless designs must rely exclusively upon wing control surfaces for control, unless vectored thrust is provided.

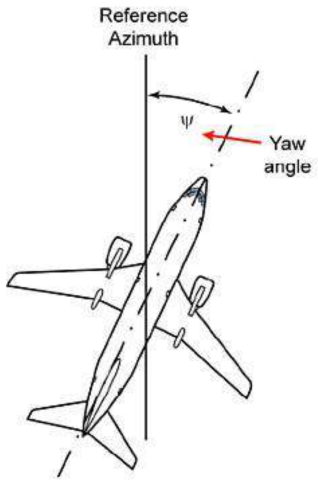
- Rudder control is usually provided by wingtip-mounted drag devices.

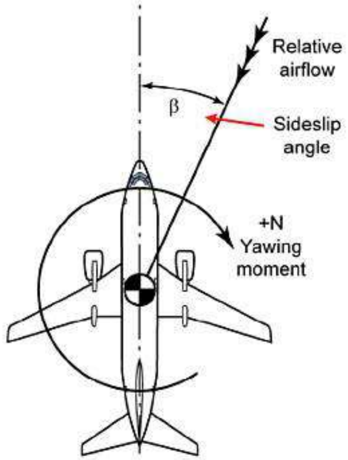


Airfoil and Geometry Selection

Tail Arrangement for Spin Recovery

- The vertical tail plays a key role in spin recovery.
- An aircraft in a spin is essentially falling vertically and rotating about a vertical axis, with the wing fully stalled.
- The aircraft is also typically at a large sideslip angle.





Airfoil and Geometry Selection

- To recover from the spin requires that the wing be unstalled, so the angle of attack must be reduced.

- However, first the rotation must be stopped and the sideslip angle reduced, or the aircraft will immediately enter another spin.

- This requires adequate rudder control even at the high angles of attack seen in the spin.

- Figure 4.31 illustrates the effect of tail arrangement upon rudder control at high angles of attack.

- At high angles of attack the horizontal tail is stalled, producing a turbulent wake extending upward at approximately a 45-deg angle.

Airfoil and Geometry Selection

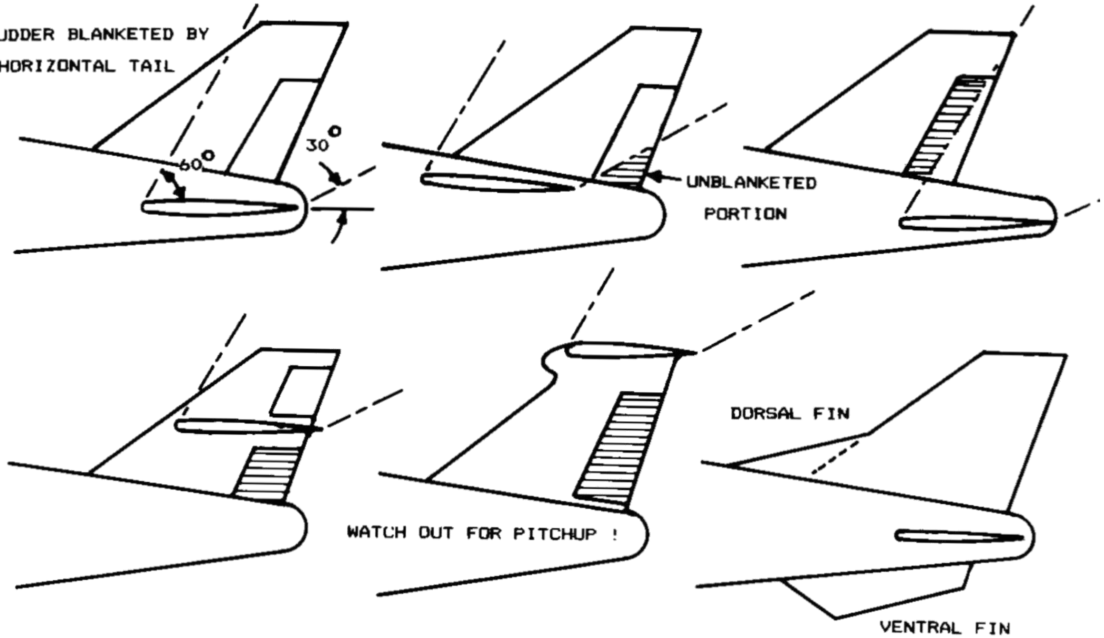


Fig. 4.31 Tail geometry for spin recovery.

- As a rule of thumb, at least a third of the rudder should be out of the wake.
- The next two examples show the effect of moving the horizontal tail upward. The T-tail arrangement completely uncovers the rudder, but can result in pitchup.

Airfoil and Geometry Selection

- The last illustration in Fig. 4.31 shows the use of dorsal and ventral fins.
- The dorsal fin improves tail effectiveness at high angles of sideslip by creating a vortex that attaches to the vertical tail.
- This tends to prevent the high angles of sideslip seen in spins, and augments rudder control in the spin.

- The ventral tail also tends to prevent high sideslip, and has the extra advantage of being where it cannot be blanketed by the wing wake.
- Ventral tails are also used to avoid lateral instability in high-speed flight.

Airfoil and Geometry Selection

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 (discussed later).
- Tail aspect ratio and taper ratio show little variation over a wide range of aircraft types.
- Table 4.3 provides guidance for selection of tail aspect ratio and taper ratio.

Airfoil and Geometry Selection

Table 4.3 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

- Leading-edge sweep of the horizontal tail is usually set to about 5 deg more than the wing sweep.
- This tends to make the tail stall after the wing, and also provides the tail with a higher Critical Mach Number than the wing, which avoids loss of elevator effectiveness due to shock formation.
- Vertical-tail sweep varies between about 35 and 55 deg. For a low-speed aircraft, there is little reason for vertical-tail sweep beyond about 20 deg.
- For a high-speed aircraft, vertical-tail sweep is used primarily to sure that the tail's Critical Mach Number is higher than the wing's.

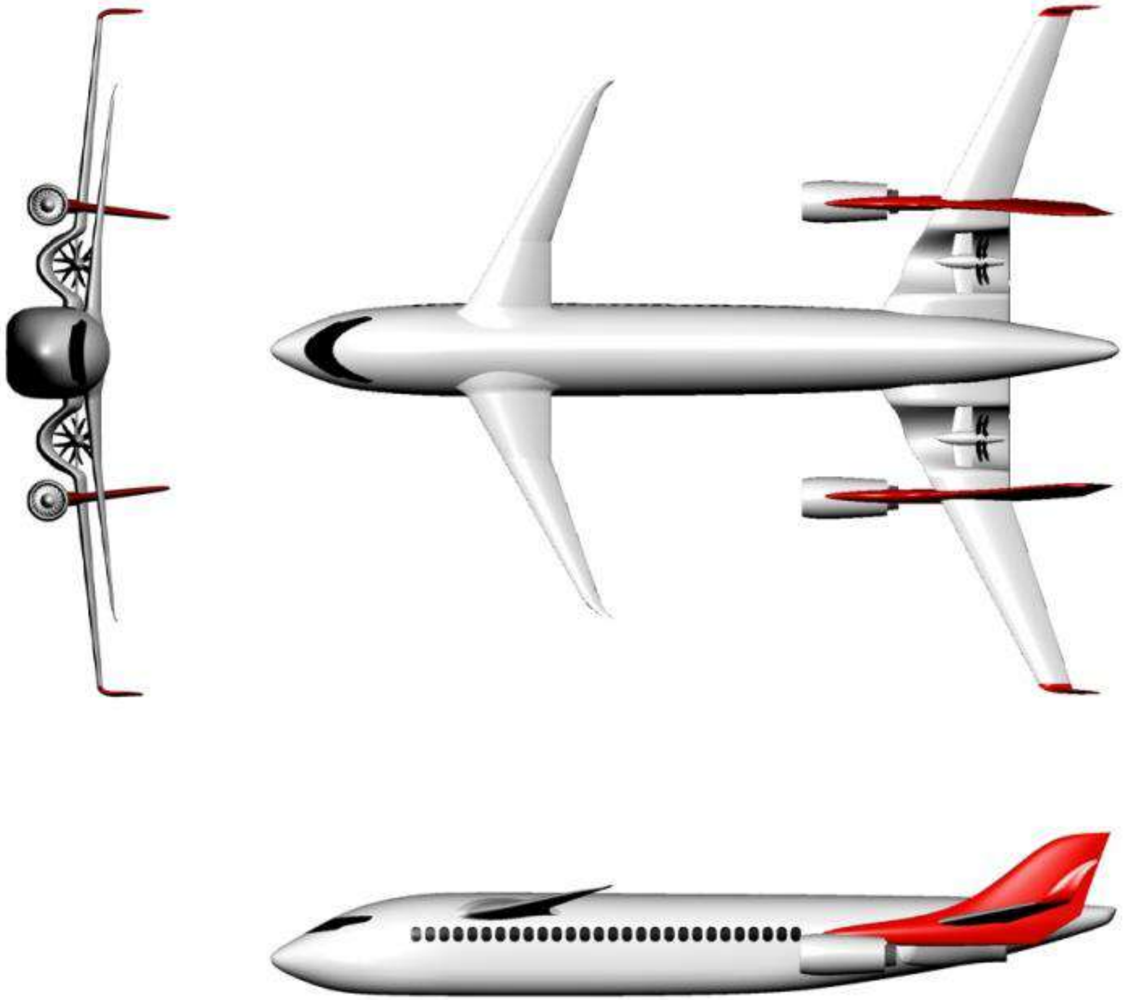
Airfoil and Geometry Selection

- Tail thickness ratio is usually similar to the wing thickness ratio, as determined by the historical guidelines provided in the wing-geometry section.

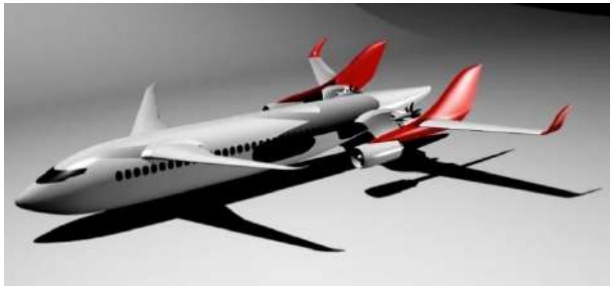
- For a high-speed aircraft, the horizontal tail is frequently about 10% thinner than the wing to ensure that the tail has a higher Critical Mach Number.

- 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.

Airfoil and Geometry Selection

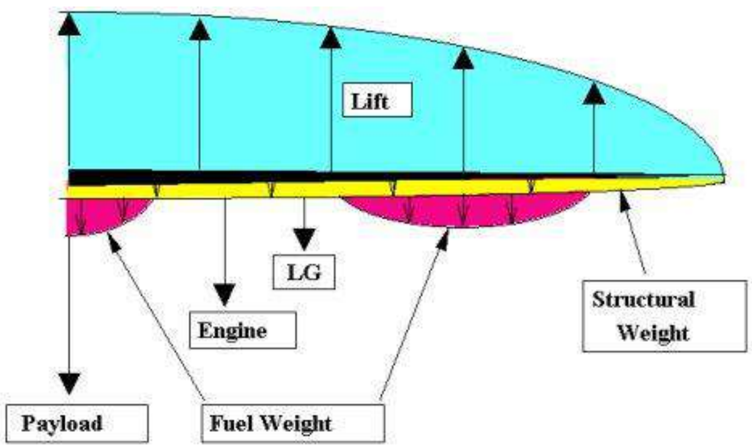


Tandem wing



Thrust-to-Weight Ratio and Wing Loading

- The thrust-to-weight ratio (T/W) and the wing loading (W/S) are the two most important parameters affecting aircraft performance.



- Wing loading and thrust-to-weight ratio are interconnected for a number of performance calculations, such as takeoff distance…

- 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.

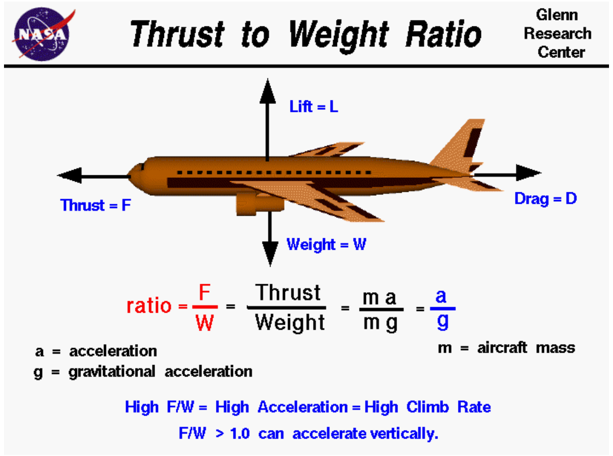
- So, the designer must guess at one of the parameters and use that guess to calculate the other parameter from the critical design requirements.

Thrust-to-Weight Ratio and Wing Loading

- In many cases, the critical requirement for wing loading will be the stall speed during the approach for landing.

- Approach stall speed is independent of engine size, so the wing loading can be estimated based upon stall speed alone.

- The estimated wing loading can then be used to calculate the T/W required to attain other performance drivers such as the single-engine rate of climb.



  - However, in this book, the thrust-to-weight ratio appears as the first guess.

  - Because that parameter better lends itself to a statistical approach, and also because it shows less variation within a given class of aircraft.

Thrust-to-Weight Ratio and Wing Loading

Thrust-to-Weight 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.

- *T/W* is not a constant.

- The weight of the aircraft varies during flight as fuel is burned. Also, the engine's thrust varies with altitude and velocity.

Thrust-to-Weight Ratio and Wing Loading

- When designers speak of an aircraft's thrust-to-weight ratio they generally refer to
  - the *T/W* during sea-level static (zero-velocity),
  - standard-day conditions
  - at design takeoff weight and maximum throttle setting.

- Another commonly referred-to *T/W* concerns combat conditions.

- The term "thrust-to-weight" is associated with jet-engined 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).

Thrust-to-Weight Ratio and Wing Loading

- 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 as low as three or four.
- an equivalent *T/W* for propellered aircraft can be expressed as follows:

$$\frac{T}{W} = \left(\frac{550\,\eta_p}{V}\right)\left(\frac{\text{hp}}{W}\right) \qquad (5.1)$$

propeller efficiency, η_p

- Note that this equation includes the term hp/*W*, the horsepower-to-weight ratio.

Thrust-to-Weight Ratio and Wing Loading

- Tables 5.1 and 5.2 provide typical values for *T/W* and hp/*W* for different classes of aircraft.
- Table 5.2 also provides reciprocal values, i.e., power loadings, for propellered aircraft.
- These values are all at maximum power settings at sea level and zero velocity ("static").

Table 5.1 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

Thrust-to-Weight Ratio and Wing Loading

Table 5.2 Horsepower-to-weight ratio

Aircraft type	Typical hp/W	Typical power loading (W/hp)
Powered sailplane	0.04	25
Homebuilt	0.08	12
General aviation—single engine	0.07	14
General aviation—twin engine	0.17	6
Agricultural	0.09	11
Twin turboprop	0.20	5
Flying boat	0.10	10

- Note that the current generation of dogfighters approaches a *T/W* of 1.0, implying that the thrust is nearly equal to the weight.

- At combat conditions when some fuel has been burned off, these aircraft have T/W values exceeding 1, and are capable of accelerating while going straight up!

- The jet dogfighter T/W values are with afterburning engines, whereas the other jets typically do not have afterburning.

Thrust-to-Weight Ratio and Wing Loading

Table 5.3 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 5.4 hp/W_0 **vs** V_{max} **(mph)**

$hp/W_0 = a\ V_{max}^C$	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

Thrust-to-Weight Ratio and Wing Loading

- For aircraft designed primarily for efficiency during cruise, a better initial estimate of the required *T/W* 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 unaccelerating flight, the thrust must equal the drag.

- Likewise, the weight must equal the lift.

- Thus, *T/W* must equal the inverse of *L/D*;

$$\left(\frac{T}{W}\right)_{\text{cruise}} = \frac{1}{(L/D)_{\text{cruise}}} \qquad (5.2)$$

- For the first estimation of *T/W* the method for *L/D* estimation presented in Chapter 3 is adequate.

Thrust-to-Weight Ratio and Wing Loading

- Note that this method assumes that the aircraft is cruising at approximately the optimum altitude for the as-yet-unknown wing loading.

- The thrust-to-weight ratio estimated using Eq. (5.2) is at cruise conditions, not takeoff.

- The aircraft will have burned off part of its fuel before beginning the cruise, and will burn off more as the cruise progresses.

- A typical aircraft will therefore have a weight at the beginning of cruise of about 0.956 times the takeoff weight.

- This value is used below to adjust the cruise T/W back to takeoff conditions.

Thrust-to-Weight Ratio and Wing Loading

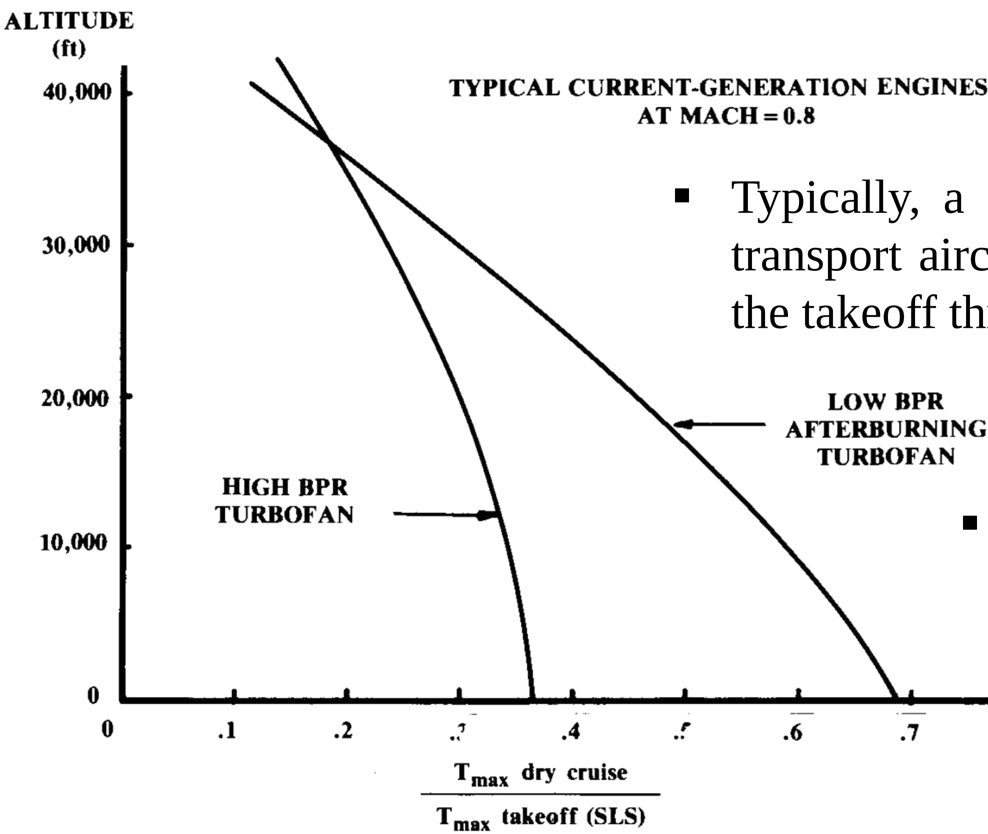


Fig. 5.1 Thrust lapse at cruise.

- Typically, a subsonic, high-bypass-ratio turbofan for a transport aircraft will have a cruise thrust of 30-35% of the takeoff thrust,

- While a low-bypass afterburning turbofan or turbojet will have a cruise thrust of 40-70% of the takeoff maximum value (see Fig. 5.1).

Thrust-to-Weight Ratio and Wing Loading

- For a piston-powered, propeller-driven aircraft, the power available varies with the density of the air provided to the intake manifold.

- If the engine is not supercharged, then the power falls off with increasing altitude according to the density ratio, σ.

- For example, a nonsupercharged engine at 10,000 ft will have about 73% of its sea-level power.

- Piston-powered aircraft typically cruise at about 75% of takeoff power, (see Fig. 5.2).

Thrust-to-Weight Ratio and Wing Loading

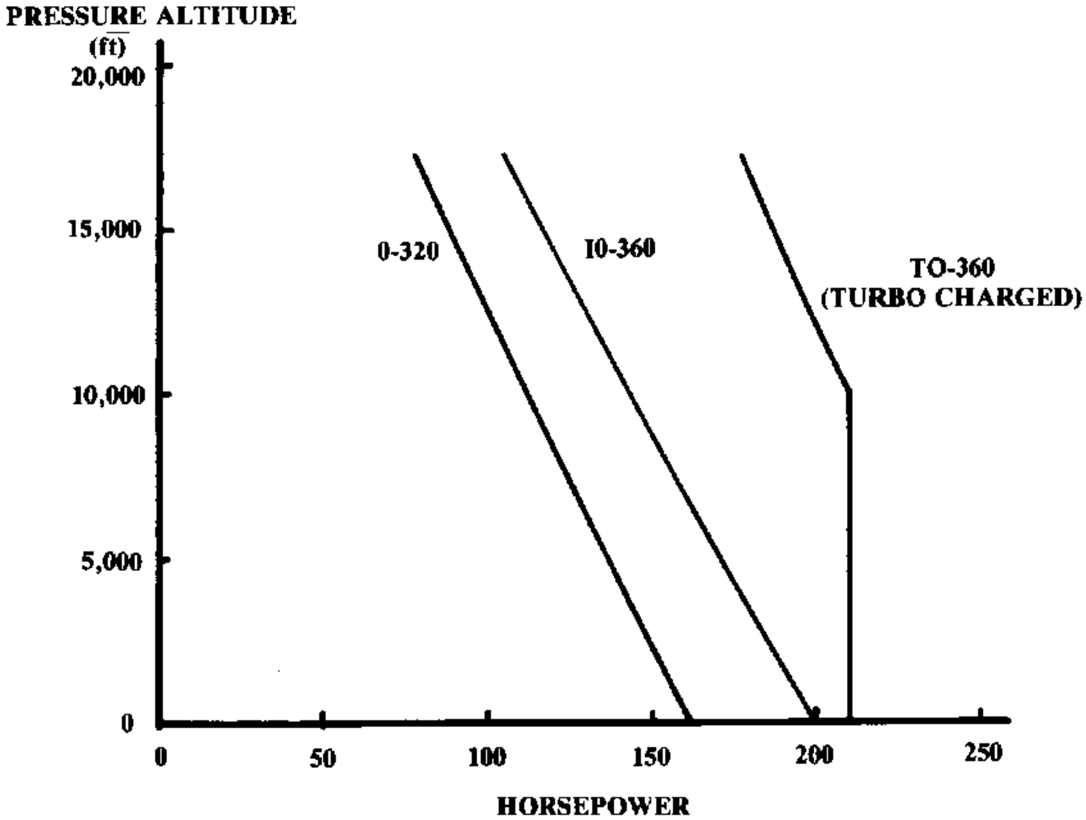
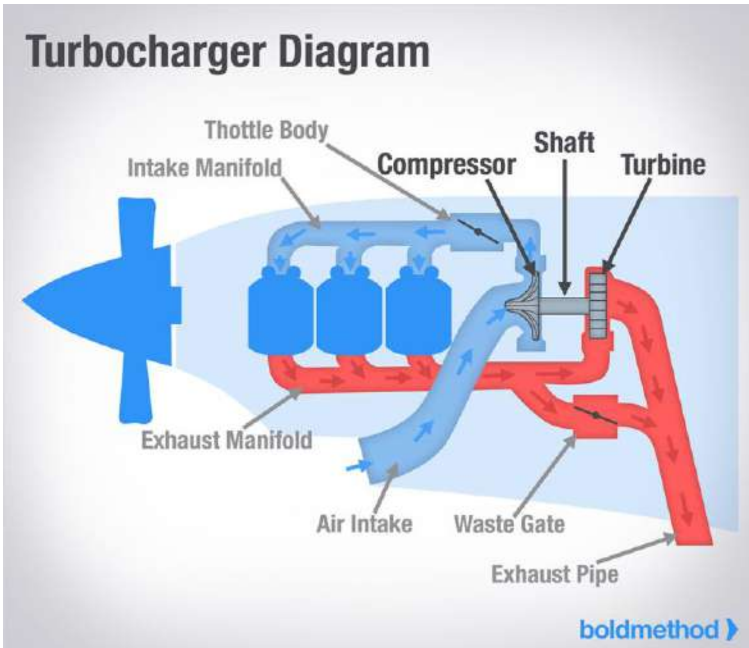
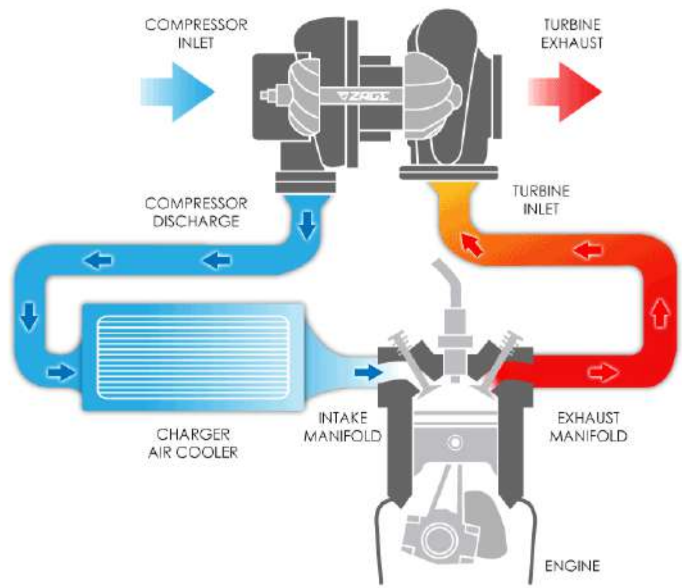


Fig. 5.2 Piston engine power variation with altitude.

Thrust-to-Weight Ratio and Wing Loading





- Many piston engines use a supercharger to maintain the air provided to the manifold at essentially sea-level density up to the compression limit of the supercharger.

Thrust-to-Weight Ratio and Wing Loading

- For a turbine-powered, propeller-driven (turboprop) aircraft, the horsepower available increases somewhat with increasing speed,
- But the thrust drops off anyway due to the velocity effect on the propeller [Eq. (5.1)).
- With a turboprop, there is an additional, residual thrust contribution from the turbine exhaust.
- It is customary to convert this thrust to its horsepower equivalent and add it to the actual horsepower, creating an "equivalent shaft horsepower (eshp)."
- For a typical turboprop engine installation, the cruise eshp is about 60-80% of the takeoff value.

Thrust-to-Weight Ratio and Wing Loading

- The takeoff T/W required for cruise matching can now be approximated using Eq. (5.3).

- The ratio between initial cruise weight and takeoff weight was shown to be about 0.956.

- If a better estimate of this ratio is available, it should be used.

$$\left(\frac{T}{W}\right)_{\text{takeoff}} = \left(\frac{T}{W}\right)_{\text{cruise}} \left(\frac{W_{\text{cruise}}}{W_{\text{takeoff}}}\right) \left(\frac{T_{\text{takeoff}}}{T_{\text{cruise}}}\right) \tag{5.3}$$

- The thrust ratio between takeoff and cruise conditions should be obtained from actual engine data if possible.

Thrust-to-Weight Ratio and Wing Loading

- For a propeller aircraft, the required takeoff hp/W can be found by combining Eqs. (5.1) and (5.2):

$$\left(\frac{hp}{W}\right)_{takeoff} = \left(\frac{V_{cruise}}{550\ \eta_p}\right)\left(\frac{1}{(L/D)_{cruise}}\right)\left(\frac{W_{cruise}}{W_{takeoff}}\right)\left(\frac{hp_{takeoff}}{hp_{cruise}}\right) \qquad (5.4)$$

where typically $\eta_p = 0.8$.

- For the first-pass estimate, the *T/W (or* hp/W) should be selected as the higher of either the statistical value obtained from the appropriate equation in Tables 5.3 and 5.4,

- or the value obtained from the cruise matching as described above.

Thrust-to-Weight Ratio and Wing Loading

Wing Loading

- 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 wing span.
- Table 5.5 provides representative wing loadings.

Thrust-to-Weight Ratio and Wing Loading

Table 5.5 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

- To ensure that the wing provides enough lift in all circumstances, the designer should select the lowest of the estimated wing loadings.

Thrust-to-Weight Ratio and Wing Loading

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-touchdown accidents, is defined by the stall speed.

- For civil applications, the approach speed must be at least 1.3 times the stall speed.

- For military applications, the multiple must be at least 1.2 (1.15 for carrier-based aircraft).

Thrust-to-Weight Ratio and Wing Loading

- Approach speed may be explicitly stated in the design requirements or will be selected based upon prior, similar aircraft.
- Then the required stall speed is found by division by 1.3, 1.2, or 1.15.
- Civil and military design specifications establish maximum allowable stall speeds for various classes of aircraft.
- In some cases the stall speed is explicitly stated.
- FAR 23 certified aircraft (under 12,500-lb TOGW) must stall at no more than 61 knots, unless they are multiengined and meet certain climb requirements.
- While not stated in any design specifications, a stall speed of about 50 knots would be considered the upper limit for a civilian trainer or other aircraft.

Thrust-to-Weight Ratio and Wing Loading

- Equation (5.6) solves for the required wing loading to attain a given stall speed with a certain maximum lift coefficient.

- The air density, ρ, is typically the sea-level standard value (0.00238 slugs/cubic ft) or sometimes the 5000-ft-altitude, hot-day value (0.00189).

$$W = L = q_{\text{stall}} S C_{L_{\max}} = \tfrac{1}{2}\rho V_{\text{stall}}^2 S C_{L_{\max}} \tag{5.5}$$

$$W/S = \tfrac{1}{2}\rho V_{\text{stall}}^2 C_{L_{\max}} \tag{5.6}$$

- The remaining unknown, the maximum lift coefficient, can be very difficult to estimate.

- Values range from about 1.2 to 1.5 for a plain wing with no flaps to as much as 5.0 for a wing with large flaps immersed in the propwash or jetwash.

Thrust-to-Weight Ratio and Wing Loading

- 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 is about 2.4.

- Other aircraft, with flaps on the inner part of the wing, will reach a lift coefficient of about 1.6-2.0.

- Most aircraft use a different flap setting for takeoff and landing.

- Typically, the takeoff maximum lift coefficient is about 80% that of the landing value.

Thrust-to-Weight Ratio and Wing Loading

- For a wing of fairly high aspect ratio (over about 5), the maximum lift coefficient will be approximately 90% of the airfoil maximum lift coefficient at the same Reynolds number, provided that the lift distribution is nearly elliptical.

- Crude approximation for wings of a fairly high aspect ratio is given in Eq. (5.7).

$$C_{L_{\max}} \cong 0.9\left\{(C_{\ell_{\max}})_{\text{flapped}}\frac{S_{\text{flapped}}}{S_{\text{ref}}} + (C_{\ell})_{\text{unflapped}}\frac{S_{\text{unflapped}}}{S_{\text{ref}}}\right\} \tag{5.7}$$

- For a better initial estimate of maximum lift, it is necessary to resort to test results and historical data.

- Figure 5.3 provides maximum-lift trends vs sweep angle for several classes of aircraft.

Thrust-to-Weight Ratio and Wing Loading

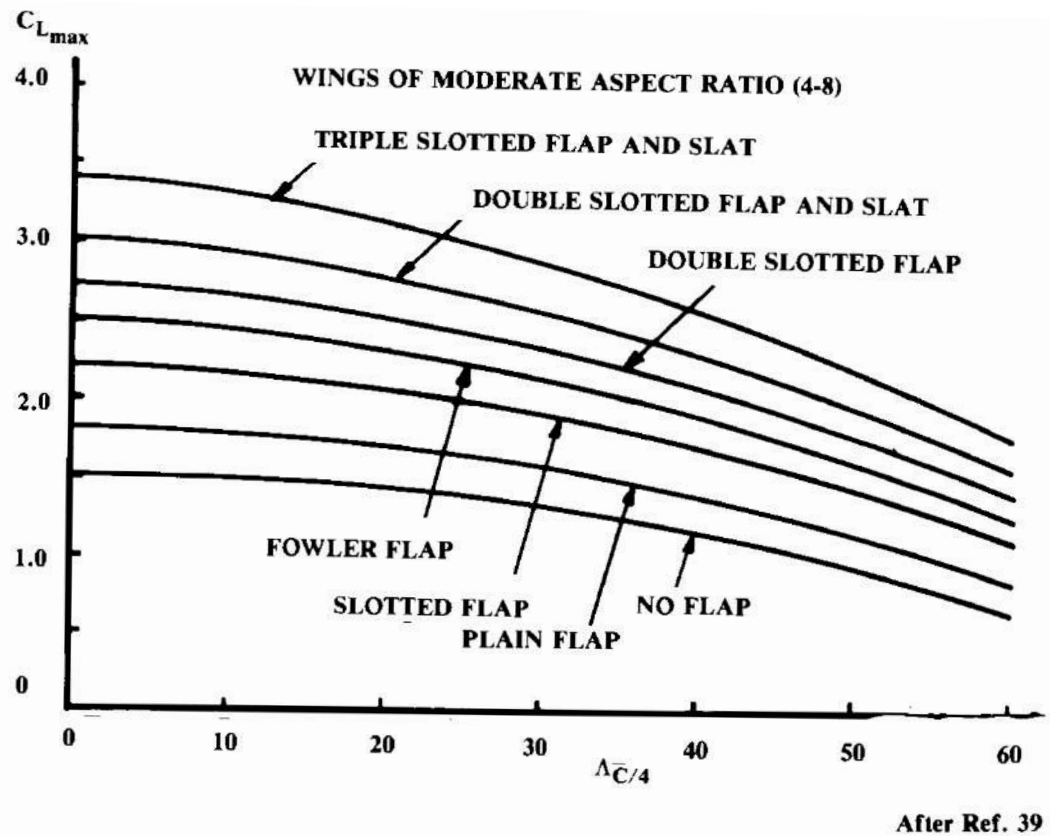


Fig. 5.3 Maximum lift coefficient.

Thrust-to-Weight Ratio and Wing Loading

Takeoff Distance

- Both the wing loading and the thrust-to-weight ratio contribute to the takeoff distance.

- The equations below assume that the thrust-to-weight ratio has been selected and can be used to determine the required wing loading to attain some required takeoff distance.

- However, the equations could be solved for T/W if the wing loading is known.

- For initial estimation of the required wing loading, a statistical approach for estimation of takeoff distance can be used.

- Figure 5.4 permits estimation of the takeoff ground roll, takeoff distance to clear a 50-ft obstacle, and a 35-ft obstacle.

Thrust-to-Weight Ratio and Wing Loading

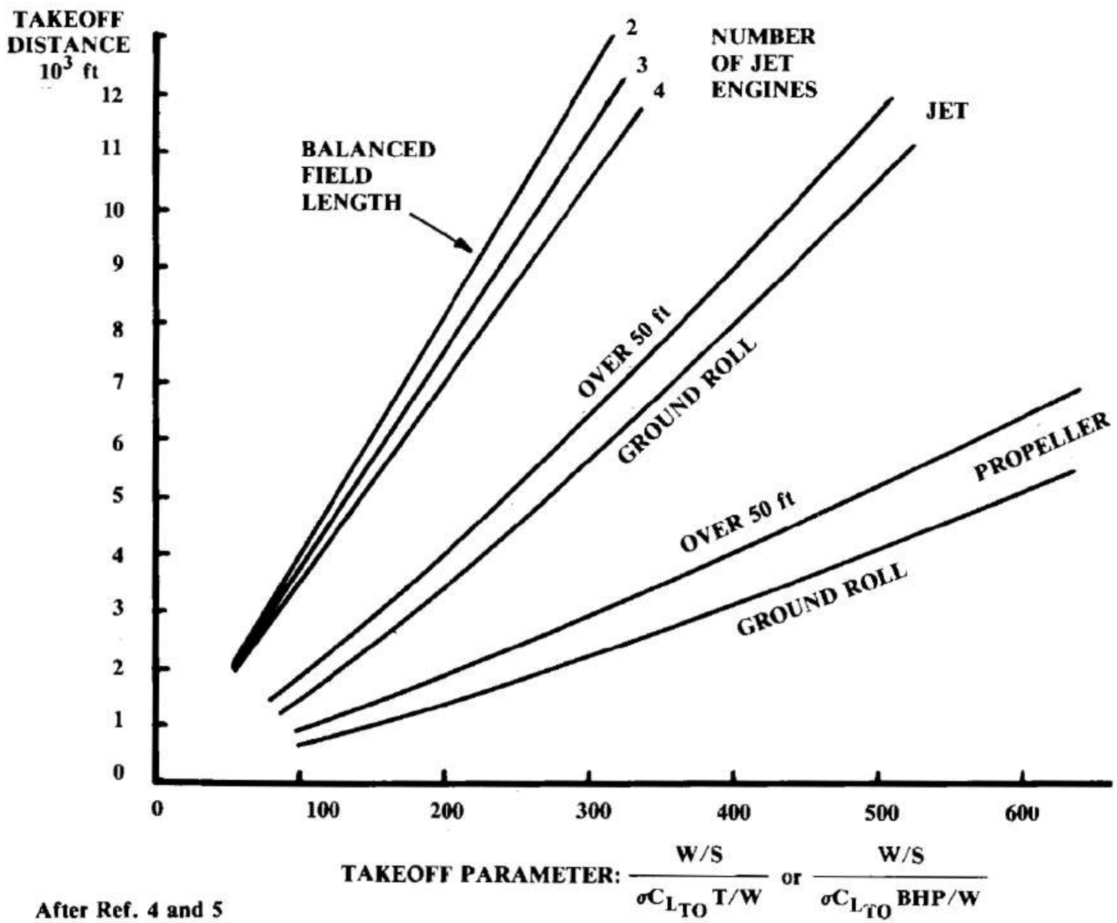


Fig. 5.4 Takeoff distance estimation.

Thrust-to-Weight Ratio and Wing Loading

- To determine the required wing loading to meet a given takeoff distance requirement, the takeoff parameter is obtained from Fig. 5.4.

- Then the following expressions give the maximum allowable wing loading for the given takeoff distance:

$$\text{Prop:} \quad (W/S) = (\mathrm{TOP})\sigma C_{L_{\mathrm{TO}}}(\mathrm{hp}/W) \tag{5.8}$$

$$\text{Jet:} \quad (W/S) = (\mathrm{TOP})\sigma C_{L_{\mathrm{TO}}}(T/W) \tag{5.9}$$

σ The density ratio

- The takeoff lift coefficient is the actual lift coefficient at takeoff, not the maximum lift coefficient at takeoff conditions as used for stall calculation.

- The aircraft takes off at about 1.1 times the stall speed so the takeoff lift coefficient is the maximum takeoff lift coefficient divided by 1.21

Thrust-to-Weight Ratio and Wing Loading

Catapult Takeoff

- Most Naval aircraft must be capable of operation from an aircraft carrier.
- For takeoff from a carrier, a catapult accelerates the aircraft to flying speed in a very short distance.
- A light aircraft can be accelerated to a higher speed by the catapult than a heavy one.

Thrust-to-Weight Ratio and Wing Loading

- Figure 5.5 depicts the velocities attainable as a function of aircraft weight for three catapults in use by the U.S. Navy. Note that a rough guess of takeoff weight is required.

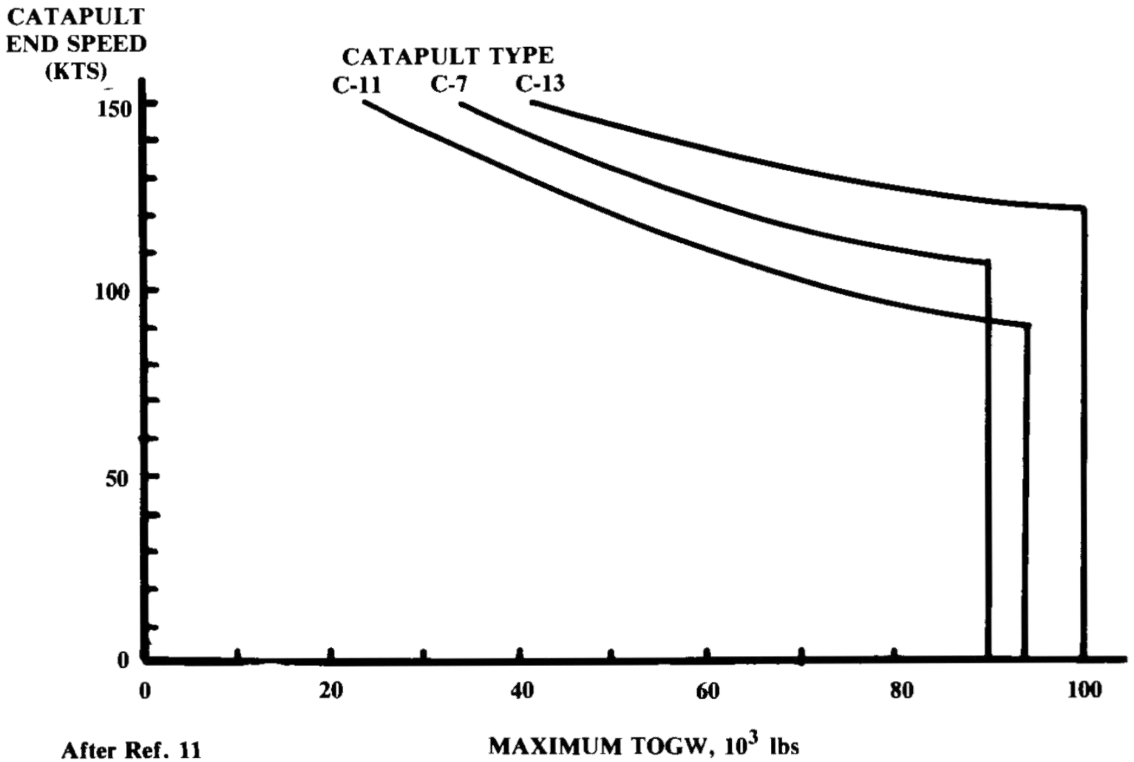


Fig. 5.5 Catapult end speeds.

Thrust-to-Weight Ratio and Wing Loading

- For a catapult takeoff, the airspeed as the aircraft leaves the catapult must exceed the stall speed by 10%.

- Airspeed is the sum of the catapult end speed and the wind-over-deck of the carrier.

- Once the end speed is known, the maximum wing loading is defined by:

$$\left(\frac{W}{S}\right)_{\text{landing}} = \frac{1}{2}\rho(V_{\text{end}} + V_{\text{wod}})^2 \frac{(C_{L\,\max})_{\text{takeoff}}}{1.21} \qquad (5.10)$$

Thrust-to-Weight Ratio and Wing Loading

Thrust-to-Weight Ratio and Wing Loading

Landing Distance

- Landing distance is largely determined by wing loading.

- Wing loading affects the approach speed, which must be a certain multiple of stall speed (1.3 for civil aircraft, 1.2 for military aircraft).

- A reasonable first-guess of the total landing distance in feet, including obstacle clearance, is approximately 0.3 times the square of the approach speed in knots.

- Equation (5.11) provides a better approximation of the landing distance, which can be used to estimate the maximum landing wing loading.

- The first term represents the ground roll to absorb the kinetic energy at touchdown speed. The constant term, S_a, represents the obstacle-clearance distance.

Thrust-to-Weight Ratio and Wing Loading

$$S_{\text{landing}} = 80\left(\frac{W}{S}\right)\left(\frac{1}{\sigma C_{L_{\max}}}\right) + S_a \tag{5.11}$$

where

S_a = 1000 (airliner-type, 3-deg glideslope)
= 600 (general aviation-type power-off approach)
= 450 (STOL, 7-deg glideslope)

- For aircraft equipped with thrust reversers or reversible-pitch propellers, multiply the ground portion of the landing [first term in Eq. (5.11)) by 0.66.
- For commercial (FAR 25) aircraft, multiply the total landing distance calculated with Eq. (5 .11) by 1.67 to provide the required safety margin.

Thrust-to-Weight Ratio and Wing Loading

- Part 21 – Certification Procedures for Products and Parts
- Part 23 – Airworthiness Standards: Normal, Utility, Acrobatic and Commuter Airplanes
- Part 25 – Airworthiness Standards: Transport Category Airplanes
- Part 27 – Airworthiness Standards: Normal Category Rotorcraft
- Part 29 – Airworthiness Standards: Transport Category Rotorcraft
- Part 33 – Airworthiness Standards: Aircraft Engines

- The landing wing loading must be converted to takeoff conditions by dividing by the ratio of landing weight to takeoff weight.

- For most propeller-powered aircraft and jet trainers, the aircraft must meet its landing requirement at or near the design takeoff weight, so the ratio is about 1.0.

- For most jet aircraft, the landing is typically calculated at a weight of about 0.85 times the takeoff weight.

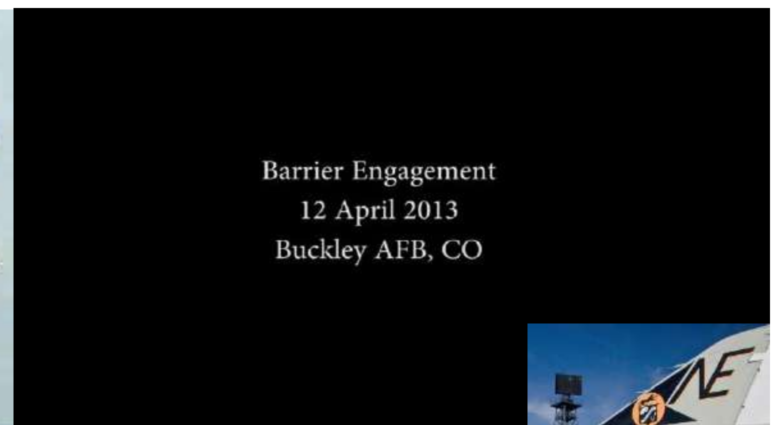
- Military design requirements will frequently specify full payload and some percent of fuel remaining (usually 50%) for the landing.

Thrust-to-Weight Ratio and Wing Loading

Arrested Landing

- Aircraft which land on Navy aircraft carriers are stopped by a cable-and brake arrangement called "arresting gear."

- One of several cables strung across the flight deck is caught by a hook attached to the rear of the aircraft.



Thrust-to-Weight Ratio and Wing Loading

- For carrier-based aircraft, the approach speed (1.15 times the stall speed) is the same as the touchdown speed.

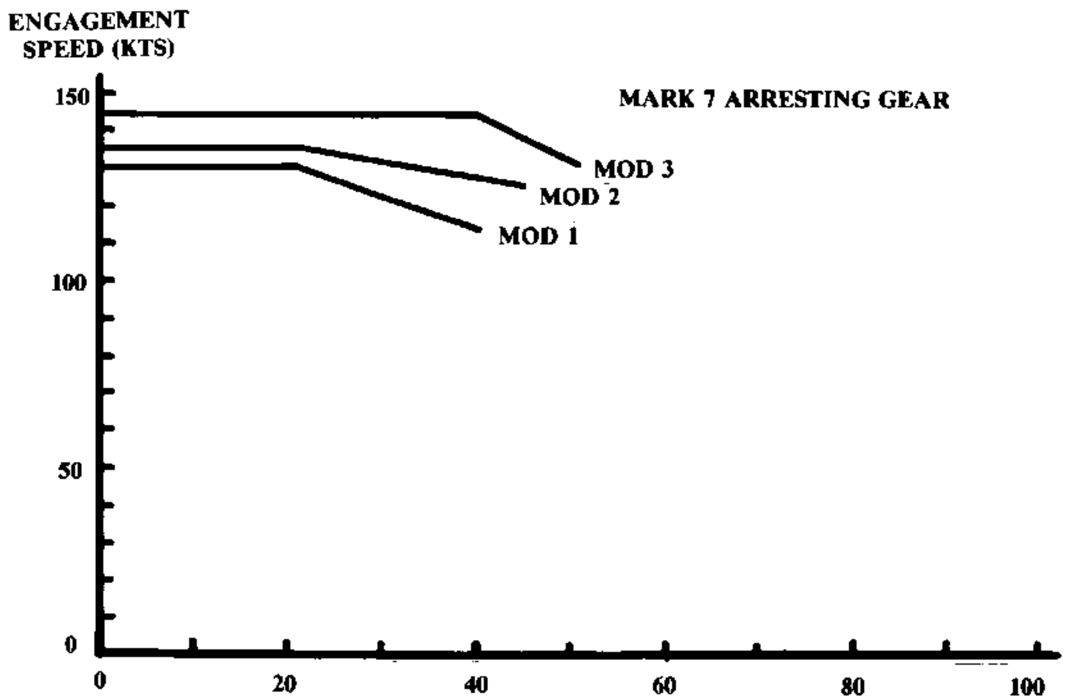
- Carrier pilots do not flare and slow down for landing.

- The landing weight limits for three standard arresting gears are depicted in Fig. 5.6.

- This figure can be used to determine the allowable approach speed based upon a first-guess of the landing weight.

- The approach speed divided by 1.15 defines the stall speed, which can then be used to estimate the wing loading.

Thrust-to-Weight Ratio and Wing Loading



After Ref. 11

Fig. 5.6 Arresting gear weight limits.

Thrust-to-Weight Ratio and Wing Loading

Wing Loading for Cruise

- At this point we must bring in the use of two aerodynamic coefficients, C_{D0} and *"e."*

- C_{D0} is the zero lift drag coefficient, and equals approximately 0.015 for a jet aircraft, 0.02 for a clean propeller aircraft, and 0.03 for a dirty, fixed-gear propeller aircraft.

- The Oswald efficiency factor *e* is a measure of drag due to lift efficiency, and equals approximately 0.6 for a fighter and 0.8 for other aircraft.

- To maximize range a propeller aircraft should fly such that:

$$qSC_{D_0} = qS\frac{C_L^2}{\pi Ae} \tag{5.12}$$

Remember: The speed for best *LID* can be shown to result in parasite drag equaling the induced drag

Thrust-to-Weight Ratio and Wing Loading

- Wing loading for maximum range for a propeller aircraft, (L=W).

$$\textbf{Maximum Prop Range:}\quad W/S = q\sqrt{\pi A e C_{D_0}} \qquad \textbf{(5.13)}$$

- A jet aircraft flying a cruise-climb will obtain maximize range by flying at a wing loading such that the parasite drag is three times the induced drag (remember flight mechanics).

- This yields the following formula for wing-loading selection for range optimization of jet aircraft.

$$\textbf{Maximum Jet Range:}\quad W/S = q\sqrt{\pi A e C_{D_0}/3} \qquad \textbf{(5.14)}$$

Wing Loading for Loiter Endurance

- Most aircraft will have some loiter requirement during the mission, typically 20 min of loiter before landing.
- Unless the loiter requirement is a substantial fraction of the total mission duration, it is better to optimize the wing loading for cruise.
- Aircraft which may be concerned with loiter endurance are patrol aircraft, airborne command posts, and intelligence-gathering aircraft.
- For an aircraft which must be optimized for loiter, the wing loading should be selected to provide a high *L/D*.
- For jet aircraft, the best loiter occurs at maximum *LID* so Eq. (5.12) should be used.

Thrust-to-Weight Ratio and Wing Loading

- For a propeller aircraft, loiter is optimized when the induced drag is three times the parasite drag, which yields Eq. (5.16).

- This also provides the wing loading for minimum power required.

$$\text{Maximum Jet Loiter: } W/S = q\sqrt{\pi A e C_{D_0}} \tag{5.15}$$

$$\text{Maximum Prop Loiter: } W/S = q\sqrt{3\pi A e C_{D_0}} \tag{5.16}$$

- These equations assume that the loiter velocity and altitude are known.

- For initial design purposes, it can be assumed that the best loiter velocity will be about 150-200 knots for turboprops and jets, and about 80-120 knots for piston-props.

- If altitude is not specified, the altitude for best fuel consumption should be selected.

Thrust-to-Weight Ratio and Wing Loading

- The wing loading estimated from Eqs. (5 .15) or (5 .16) is the average during the loiter.
- This must be converted to takeoff conditions by dividing the loiter wing loading by the ratio of the average loiter weight to the takeoff weight.
- In the absence of better information, this ratio can be assumed to be about 0.85.
- Remember that Eqs. (5.15) and (5.16) are to be used for designing an aircraft optimized solely for loiter.
- Optimizing for loiter alone is very rare in aircraft design. For most aircraft, the wing loading will be selected for best cruise and the loiter capabilities will be a "fallout."

Thrust-to-Weight Ratio and Wing Loading

Instantaneous Turn

- An aircraft designed for air-to-air dogfighting must be capable of high turn rate.
- A turn rate superiority of 2 deg/s is considered significant.
- There are two important turn rates.
- The "sustained" turn rate for some flight condition is the turn rate at which the thrust of the aircraft is just sufficient to maintain velocity and altitude in the turn.
- If the aircraft turns at a quicker rate, the drag becomes greater than the available thrust, so the aircraft begins to slow down or lose altitude.
- The "instantaneous" turn rate is the highest turn rate possible, ignoring the fact that the aircraft will slow down or lose altitude.

Thrust-to-Weight Ratio and Wing Loading

- Turn rate is equal to the radial acceleration divided by the velocity. For a level turn, this results in Eq. (5.17).

$$\dot{\psi} = \frac{g\sqrt{n^2 - 1}}{V} \tag{5.17}$$

$$n = \frac{qC_L}{W/S} \tag{5.18}$$

- Instantaneous turn rate is limited only by the usable maximum lift, up to the speed at which the maximum lift exceeds the load-carrying capability of the wing structure.

- Typically, a fighter aircraft will be designed to an operational maximum load factor of 7.33g, although newer fighters are being designed to 8- or 9-g.

- This g limit must be met at some specified combat weight.

Thrust-to-Weight Ratio and Wing Loading

- Design specifications will usually require some maximum turn rate at some flight condition.

- Equation (5.17) can be solved for the load factor at the specified turn rate as follows:

$$n = \sqrt{\left(\frac{\dot{\psi} V}{g}\right)^2 + 1} \qquad (5.19)$$

- Typically, a modern fighter has a corner speed of about 300-350 knots indicated airspeed (i.e., dynamic pressure) regardless of altitude.

- Required wing loading can be solved for in Eq. (5.18) as follows:

$$\frac{W}{S} = \frac{qC_{L_{max}}}{n} \qquad (5.20)$$

Thrust-to-Weight Ratio and Wing Loading

- The only unknown is the maximum lift coefficient at combat conditions.
- This is not the same as the maximum lift coefficient for landing.
- During combat, use of full flap settings is not usually possible.
- Also, there is a Mach-number effect which reduces maximum lift at higher speeds.
- For initial design purposes, a combat maximum lift coefficient of about 0.6-0.8 should be assumed for a fighter with only a simple trailing-edge flap for combat.
- For a fighter with a complex system of leading- and trailing edge flaps which can be deployed during combat, a maximum usable lift coefficient of about 1.0-1.5 is attainable.

Thrust-to-Weight Ratio and Wing Loading

- Again, the resulting wing loading must be divided by the ratio of combat weight to takeoff weight to obtain the required takeoff wing loading.

- Usually the combat weight is specified as the aircraft design takeoff weight with any external fuel tanks dropped and 50% of the internal fuel gone.

- This is approximately 0.85 times the takeoff weight for most fighters.

- The resulting wing loading is the maximum which will allow the required instantaneous turn.

Thrust-to-Weight Ratio and Wing Loading

Sustained Turn

- Sustained turn rate is usually expressed in terms of the maximum load factor at some flight condition that the aircraft can sustain without slowing or losing altitude.

- For example, the ability for sustaining 4- or 5-g at 0.9 Mach number at 30,000 ft is frequently specified.

- The wing loading to exactly attain a required sustained load factor *n* using all of the available thrust can be determined by equating the thrust and drag,

$$T = qSC_{D_0} + qS\left(\frac{C_L^2}{\pi Ae}\right) = qSC_{D_0} + \frac{n^2W^2}{qS\pi Ae} \tag{5.23}$$

$$\frac{T}{W} = \frac{qC_{D0}}{W/S} + \frac{W}{S}\left(\frac{n^2}{q\pi Ae}\right) \tag{5.24}$$

$$\frac{W}{S} = \frac{(T/W) \pm \sqrt{(T/W)^2 - (4n^2C_{D0}/\pi Ae)}}{2n^2/q\pi Ae} \tag{5.25}$$

Thrust-to-Weight Ratio and Wing Loading

- The thrust-to-weight ratio for this calculation is at combat conditions.

- If the term within the square root in Eq. (5.25) becomes negative, there is no solution.

- This implies that, at a given load factor, the following must be satisfied regardless of the wing loading:

$$\frac{T}{W} \geq 2n\sqrt{\frac{C_{D_0}}{\pi A e}} \tag{5.26}$$

- Unfortunately, the above equations for turning flight are very sensitive to thee value.

- At high angles of attack the e value may be reduced by 30% or more.

Thrust-to-Weight Ratio and Wing Loading

Climb and Glide

- Numerous climb requirements are available for FAR or military aircraft.

- These specify rate of climb for various combinations of factors such as engine-out, landing-gear position, and flap settings.

- While the details may vary, the method for selecting a wing loading to satisfy such requirements is the same.

- Rate of climb is a vertical velocity, typically expressed in feet-per-minute.

- Climb gradient, "G," is the ratio between vertical and horizontal distance traveled.

- At normal climb angles the climb gradient equals the excess thrust divided by the weight,

Thrust-to-Weight Ratio and Wing Loading

$$G = (T - D)/W \qquad \frac{D}{W} = \frac{T}{W} - G \tag{5.28}$$

$$\frac{D}{W} = \frac{qSC_{D_0} + qS(C_L^2/\pi Ae)}{W} = \frac{qC_{D_0}}{W/S} + \frac{W}{S}\frac{1}{q\pi Ae} \tag{5.29}$$

- Equating Eqs. (5.28) with (5.29) and solving for wing loading yields:

$$\frac{W}{S} = \frac{[(T/W) - G] \pm \sqrt{[(T/W) - G]^2 - (4C_{D_0}/\pi Ae)}}{2/q\pi Ae} \tag{5.30}$$

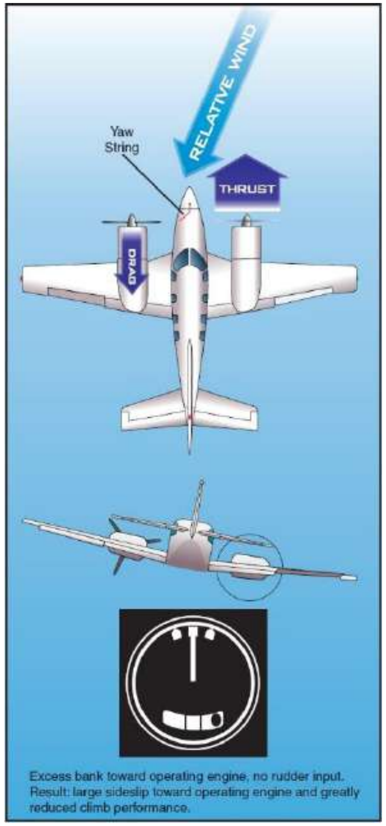
- The resulting *W/S* must then be rationed to a takeoff-weight value.

- The term within the square root symbol in Eq. (5.30) cannot go below zero, so the following must be true regardless of the wing loading:

$$\frac{T}{W} \geq G + 2\sqrt{\frac{C_{D_0}}{\pi Ae}} \tag{5.31}$$

Thrust-to-Weight Ratio and Wing Loading

- C_{D0} and e values for some of the climb conditions must include the effects of flaps and landing gear.
- Sometimes the rate of climb must also be calculated with one engine windmilling or stopped.
- The thrust loss due to a "dead" engine can be accounted for in the T/W.
- For example, if a three-engined aircraft loses one engine, the T/W becomes two-thirds of the original T/W.



Windmilling is a typical process to start a jet engine in case it stops working mid-air. When the aircraft is cruising and the engine suddenly stops working , this method is used to start the engine (if APU power is unavailable).

Thrust-to-Weight Ratio and Wing Loading

- Equation (5.30) can also be used to establish the wing loading required to attain some specified glide angle, by setting T/W to zero and using a negative value of.

- If a particular sink rate must be attained, the value of G to use is the sink rate divided by the forward velocity.

- Make sure that both are in the same units.

Maximum Ceiling

- Equation (5.30) can be used to calculate the wing loading to attain some maximum ceiling, given the T/W at those conditions.
- The climb gradient G can be set to zero to represent level flight at the desired altitude.
- Frequently a small residual climb capability, such as 100 ft/min, is required at maximum ceiling.
- This can be included in Eq. (5.30) by first solving for the climb gradient G (climb rate divided by forward velocity).

$$\frac{W}{S} = \frac{[(T/W) - G] \pm \sqrt{[(T/W) - G]^2 - (4C_{D_0}/\pi Ae)}}{2/q\pi Ae} \tag{5.30}$$

Thrust-to-Weight Ratio 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.

- When the best compromise for wing loading has been selected, the thrust-to-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.

Aircraft Design
AE 405

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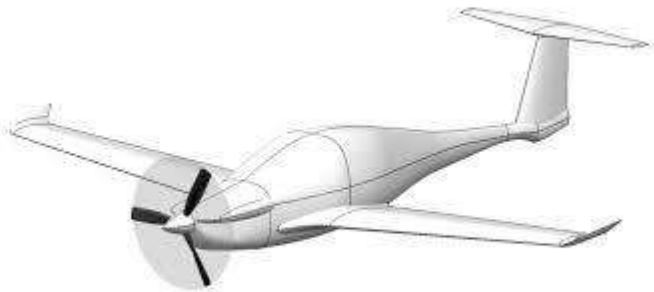
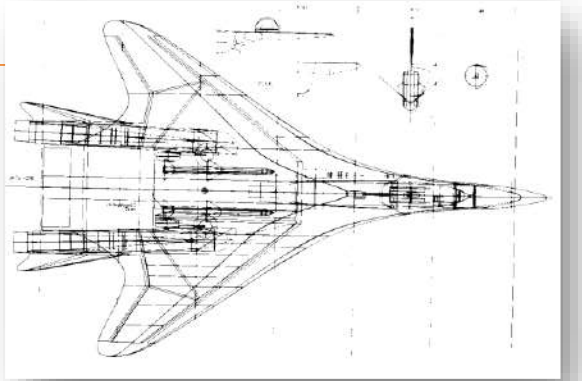
Contents



Chapter 6 – 7 – 8

- Initial Sizing
- Configuration Layout and Loft
- Special Considerations in Configuration Layout



Initial Sizing

- Aircraft sizing is the process of determining the takeoff gross weight and fuel weight required for an aircraft concept to perform its design mission.

- We see basic approach for A/C sizing in Chapter 3, now we will see more details.

- 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"

- 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.

- 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 (Super Sonic Transport).

Initial Sizing

RUBBER-ENGINE SIZING

- Chapter 3 presented a quick method of sizing an aircraft based on statistics and proper equations;

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - (W_f/W_0) - (W_e/W_0)} \qquad (6.1)$$

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right) \qquad (6.2)$$

- Equation (6.1) is limited in use to missions which do not have a sudden weight change, such as a payload drop.

- For missions with a payload drop or other sudden weight change, a slightly different sizing equation must be used.

Initial Sizing

$$W_0 = W_{crew} + W_{\substack{\text{fixed}\\\text{payload}}} + W_{\substack{\text{dropped}\\\text{payload}}} + W_{fuel} + W_{empty} \tag{6.3}$$

$$W_0 = W_{crew} + W_{\substack{\text{fixed}\\\text{payload}}} + W_{\substack{\text{dropped}\\\text{payload}}} + W_{fuel} + \left(\frac{W_e}{W_0}\right) W_0 \tag{6.4}$$

- Refined methods for determining the empty-weight fraction and fuel used are discussed below.
- The empty-weight fraction is estimated using improved statistical equations via Tables 6.1 and 6.2,
- to provide empty-weight equations which better reflect the weight impact of the major design variables.
- These are takeoff weight (W_0), the aspect ratio (A), thrust-to-weight (or horsepower-to-weight) ratio (T/W_0), wing loading (W_0/S), and maximum Mach and speed (M_{max}, V_{max}).

Initial Sizing

Table 6.1 Empty weight fraction vs W_0, A, T/W_0, W_0/S, and M_{max}

$$W_e/W_0 = (a + b W_0^{C1} A^{C2} (T/W_0)^{C3} (W_0/S)^{C4} M_{max}{}^{C5}) K_{vs}$$

	a	b	$C1$	$C2$	$C3$	$C4$	$C5$
Jet trainer	0	4.28	−0.10	0.10	0.20	−0.24	0.11
Jet fighter	−0.02	2.16	−0.10	0.20	0.04	−0.10	0.08
Military cargo/bomber	0.07	1.71	−0.10	0.10	0.06	−0.10	0.05
Jet transport	0.32	0.66	−0.13	0.30	0.06	−0.05	0.05

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

Initial Sizing

Table 6.2 Empty weight fraction vs W_0, A, hp/W_0, W_0/S, and V_{max} (mph)

$$W_e/W_0 = a + b W_0^{C1} A^{C2} (\mathrm{hp}/W_0)^{C3} (W_0/S)^{C4} V_{max}^{C5}$$

	a	b	$C1$	$C2$	$C3$	$C4$	$C5$
Sailplane—unpowered	0	0.75	−0.05	0.14	0	−0.30	0.06
Sailplane—powered	0	1.20	−0.04	0.14	0.19	−0.20	0.05
Homebuilt—metal/wood	0	0.69	−0.10	0.05	0.10	−0.05	0.17
Homebuilt—composite	0	0.59	−0.10	0.05	0.10	−0.05	0.17
General aviation—single engine	−0.25	1.14	−0.20	0.08	0.05	−0.05	0.27
General aviation—twin engine	−0.90	1.32	−0.10	0.08	0.05	−0.05	0.20
Agricultural aircraft	0	1.64	−0.14	0.07	0.10	−0.10	0.11
Twin turboprop	0.37	0.08	−0.06	0.08	0.08	−0.05	0.30
Flying boat	0	0.41	−0.01	0.10	0.05	−0.12	0.18

Initial Sizing

Fuel Weight

- The remaining unknown in Eq. (6.4) is the fuel weight.

$$(1 - W_x/W_0).$$

- The previous weight fractions based equation cannot be assumed if the mission includes a weight drop.

- The mission segment weight fractions (W_i/W_{i-1}) are calculated as before for all mission segments other than those which are weight drops.

- For each mission segment, the fuel burned is then equal to:

$$W_{f_i} = \left(1 - \frac{W_i}{W_{i-1}}\right) W_{i-1} \qquad (6.5)$$

- The total mission fuel, W_{fm}, then is equal to:

$$W_{f_m} = \sum_{1}^{x} W_{f_i} \qquad (6.6)$$

Initial Sizing

- The total aircraft fuel includes the mission fuel as well as an allowance for reserve and trapped fuel.

- This reserve fuel allowance is usually 5%, and accounts for an engine with poorer-than-normal fuel consumption.

- An additional allowance of 1% for trapped (i.e., unusable) fuel is typical.

- Thus, the total aircraft fuel is:

$$W_f = 1.06\left(\sum_{1}^{x} W_{f_i}\right) \qquad (6.7)$$

- The methods used for estimating the mission segment weight fractions are presented below.

- These are a combination of analytical and statistical methods, similar to the methods used in Chapter 3.

Initial Sizing

Engine Start, Taxi, and Takeoff

- As before, the mission segment weight fraction for engine start, taxi, and takeoff is estimated historically. A reasonable estimate is:

$$W_1/W_0 = 0.97\text{–}0.99 \tag{6.8}$$

Climb and Accelerate

- The weight fraction for an aircraft climbing and accelerating to cruise altitude and Mach number *"M,"* (starting at Mach 0.1), will be approximately as follows:

$$\text{Subsonic:} \quad W_i/W_{i-1} = 1.0065 - 0.0325M \tag{6.9}$$

$$\text{Supersonic:} \quad W_i/W_{i-1} = 0.991 - 0.007M - 0.01M^2 \tag{6.10}$$

- For an acceleration beginning at other than Mach 0.1, the weight fraction calculated by Eqs. (6.9) or (6.10) for the given ending Mach number should be divided by the weight fraction calculated for the beginning Mach number using Eqs. (6.9) or (6.10).

Mach 0.8–2.0 (0.937/0.9805), or 0.956.

Initial Sizing

Cruise

Jet: $$\frac{W_i}{W_{i-1}} = \exp \frac{-RC}{V(L/D)} \quad (6.11)$$

where

R = range
C = specific fuel consumption
V = velocity
L/D = lift-to-drag ratio

Prop: $$\frac{W_i}{W_{i-1}} = \exp \left[\frac{-RC_{\text{bhp}}}{550\,\eta_p (L/D)} \right] \quad (6.12)$$

$$C = \frac{W_f/\text{time}}{\text{thrust}} = C_{\text{bhp}} \frac{V}{550\,\eta_p}$$

where η_p = propeller efficiency.

$$\frac{L}{D} = \frac{1}{\dfrac{qC_{D0}}{W/S} + \dfrac{W}{S}\dfrac{1}{q\pi Ae}} \quad (6.13)$$

- Note that the wing loading used in Eq. (6.13) and subsequent weight fraction equations is the actual wing loading at the condition being evaluated, not the takeoff wing loading.

Initial Sizing

Loiter

- The weight fraction for a loiter mission segment is:

$$\text{Jet:} \quad \frac{W_i}{W_{i-1}} = \exp \frac{-EC}{L/D} \tag{6.14}$$

where E = endurance or loiter time.

$$\text{Prop:} \quad \frac{W_i}{W_{i-1}} = \exp \frac{-EVC_{\text{bhp}}}{550\, \eta_p (L/D)} \tag{6.15}$$

Combat

- The combat mission leg is normally specified as either a time duration *(d)* at maximum power (typically *d* = 3 min),
- or as a certain number of combat turns at maximum power at some altitude and Mach number.

$$W_i/W_{i-1} = 1 - C(T/W)(d) \tag{6.16}$$

Initial Sizing

- If the combat is defined by some number of turns (x), the duration of combat (d) must be calculated.

$$d = \frac{2\pi x}{\dot{\psi}} = \frac{2\pi V x}{g\sqrt{n^2 - 1}} \tag{6.17}$$

$$n = (T/W)(L/D) \tag{6.18}$$

maximum structural load factor

$$n \le n_{\max} \tag{6.19}$$

maximum available lift

$$n \le \frac{q C_{L\,\max}}{W/S} \tag{6.20}$$

$$\frac{L}{D} = \frac{1}{q\dfrac{C_{D0}}{n(W/S)} + \dfrac{n(W/S)}{q\pi A e}} \tag{6.21}$$

Initial Sizing

Descent for Landing

- Descent is estimated historically:

$$W_i/W_{i-1} = 0.990 \text{ to } 0.995 \qquad (6.22)$$

Landing and Taxi Back

- Again, a historical approximation is used.

$$W_i/W_{i-1} = 0.992 \text{ to } 0.997 \qquad (6.23)$$

Initial Sizing

- The design and sizing method presented above, as summarized in Fig. 6.1;

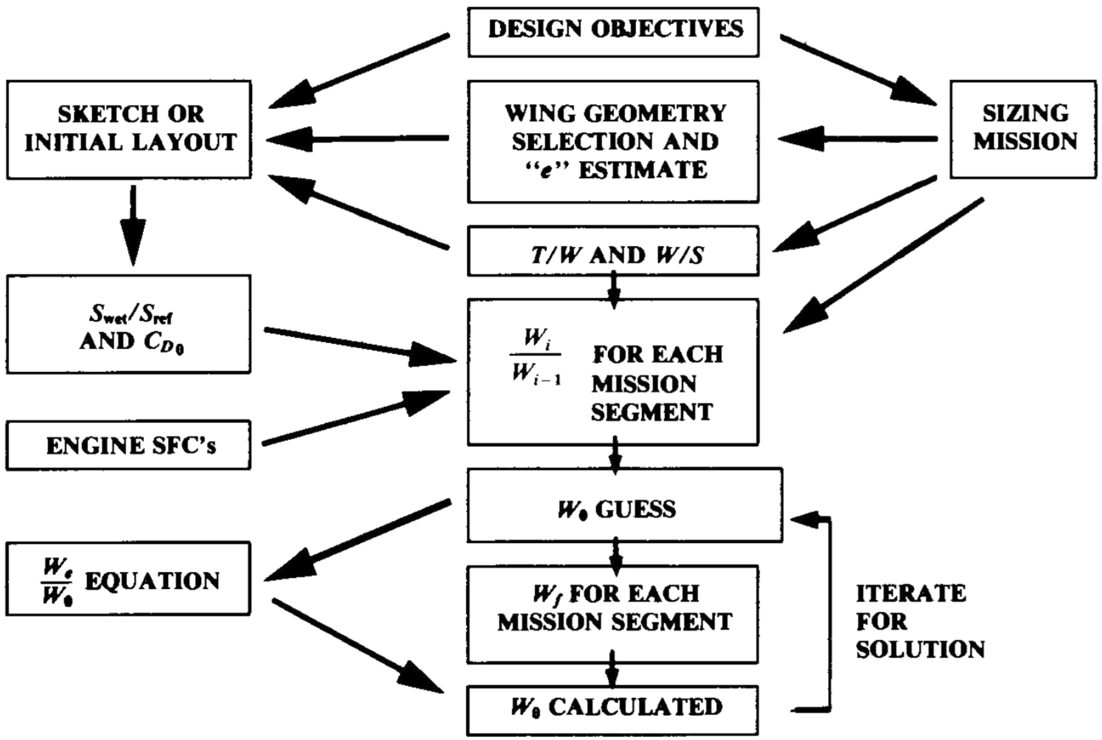


Fig. 6.1 Refined sizing method.

Initial Sizing

Fixed-engine Sizing

- The sizing procedure for the fixed-size engine is similar to the rubber engine sizing, with several exceptions.
- If the range is allowed to vary, the sizing problem is very simple.
- The required thrust-to-weight ratio (T/W) is determined using the known characteristics of the selected engine.
- Then the takeoff gross weight is determined as

$$W_0 = \frac{NT_{\text{per engine}}}{(T/W)} \tag{6.24}$$

where N = number of engines.

Geometry Sizing - Fuselage

Initial Sizing

- Once the takeoff gross weight has been estimated, the fuselage, wing, and tails can be sized.
- 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, Table 6.3 provides statistical equations for fuselage length

Initial Sizing

Table 6.3 Fuselage length vs W_0

Length $= aW_0^C$	a	C
Sailplane—unpowered	0.86	0.48
Sailplane—powered	0.71	0.48
Homebuilt—metal/wood	3.68	0.23
Homebuilt—composite	3.50	0.23
General aviation—single engine	4.37	0.23
General aviation—twin engine	0.86	0.42
Agricultural aircraft	4.04	0.23
Twin turboprop	0.37	0.51
Flying boat	1.05	0.40
Jet trainer	0.79	0.41
Jet fighter	0.93	0.39
Military cargo/bomber	0.23	0.50
Jet transport	0.67	0.43

Initial Sizing

- 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.

- Theoretically, for a fixed internal volume the subsonic drag is minimized by a fineness ratio of about 3.0,

- while supersonic drag is minimized by a fineness ratio of about 14.

- Most aircraft fall between these values.

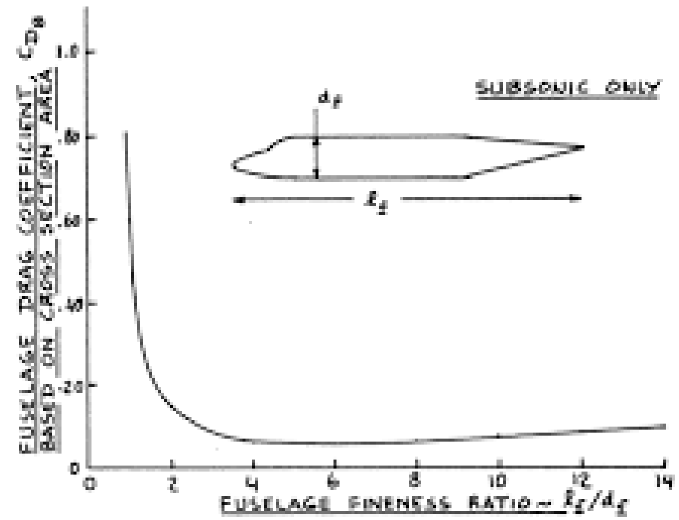


Figure 3.1 Effect of Fineness Ratio on Fuselage Drag

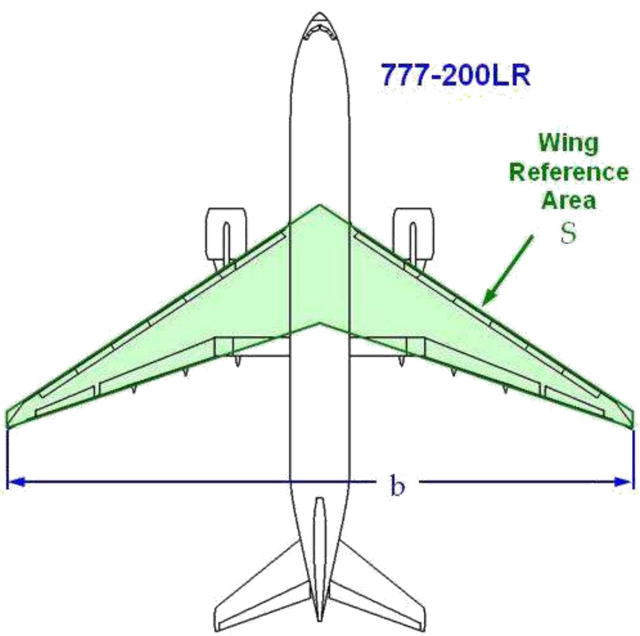
Initial Sizing

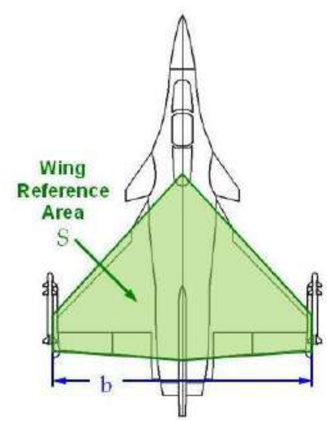
Geometry Sizing

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.





Initial Sizing

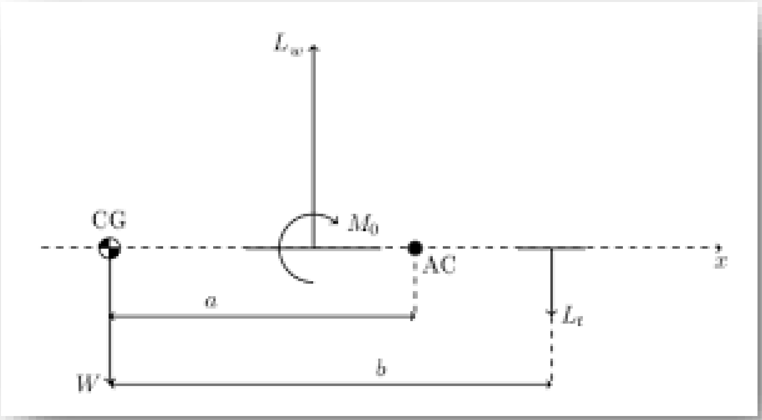
Tail Volume Coefficient

- 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.



Initial Sizing

- 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 nondimensional requires dividing by some quantity with units of length.
- For a horizontal tail or canard, the pitching moments which must be countered are most directly related to the wing mean chord ($\bar{c}_w$).
- This leads to the "horizontal tail volume coefficient,"

Initial Sizing

$$c_{HT} = \frac{L_{HT}S_{HT}}{\overline{C}_W S_W} \qquad (6.27)$$

- For a vertical tail, the wing yawing moments which must be countered are most directly related to the wing span b_w.
- This leads to the "vertical tail volume coefficient.

$$c_{VT} = \frac{L_{VT}S_{VT}}{b_W S_W} \qquad (6.26)$$

- Note that the moment arm (L) is commonly approximated as the distance from the tail quarter-chord to the wing quarter-chord,
- (i.e., 25% of the mean chord length measured back from the leading edge of the mean chord)

Initial Sizing

- The definition of tail moment arm is shown in Fig. 6.2, along with the definitions of tail area.

- Observe that the horizontal tail area is commonly measured to the aircraft centerline, while a canard's area is commonly considered to include only the exposed area.

- If twin vertical tails are used, the vertical tail area is the sum of the two.

Initial Sizing

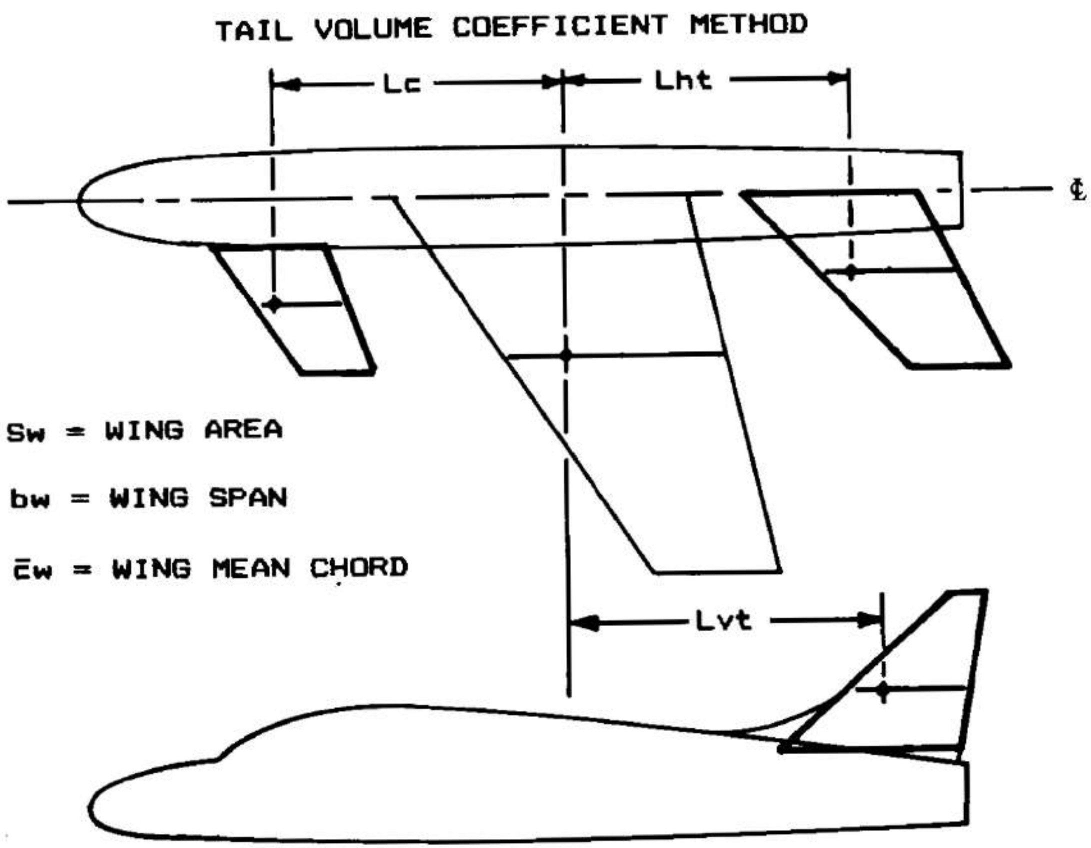


Fig. 6.2 Initial tail sizing.

Initial Sizing

- Table 6.4 provides typical values for volume coefficients for different classes of aircraft.

- These values are used in Eqs. (6.28) or (6.29) to calculate tail area.

$$S_{VT} = c_{VT} b_W S_W / L_{VT} \tag{6.28}$$

$$S_{HT} = c_{HT} \overline{C}_W S_W / L_{HT} \tag{6.29}$$

Table 6.4 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

Initial Sizing

- To calculate tail size, the moment arm must be estimated.
- This can be approximated at this stage of design by a percent of the fuselage length as previously estimated.
- For an aircraft with a front-mounted propeller engine, the tail arm is about 60% of the fuselage length.
- For an aircraft with the engines on the wings, the tail arm is about 50-55% of the fuselage length.
- For aft-mounted engines the tail arm is about 45-50% of the fuselage length.
- A sailplane has a tail moment arm of about 65% of the fuselage length.

Initial Sizing

- For an all-moving tail, the volume coefficient can be reduced by about 10-15%.
- For a "T-tail," the vertical-tail volume coefficient can be reduced by approximately 5% due to the end-plate effect.
- The horizontal tail volume coefficient can be reduced by about 5% due to the clean air seen by the horizontal.
- Similarly, the horizontal tail volume coefficient for an "H tail" can be reduced by about 5%.

Initial Sizing

- For an aircraft which uses a "V-tail," the required horizontal and vertical tail sizes should be estimated as above.
- Then the V surfaces should be sized to provide the same total surface area as required for conventional tails.
- The tail dihedral angle should be set to the arctangent of the square root of the ratio between the required vertical and horizontal tail areas.
- This should be near 45 deg.

Initial Sizing

- The horizontal tail volume coefficient for an aircraft with a control-type canard is approximately 0.1, based upon the relatively few aircraft of this type that have flown.

- For canard aircraft there is a much wider variation in the tail moment arm.

- Typically, the canarded aircraft will have a moment arm of about 30-50% of the fuselage length.

Initial Sizing

- For a lifting-canard aircraft, the volume coefficient method isn't applicable.
- Instead, an area split must be selected by the designer. The required total wing area is then allocated accordingly.
- Typically, the area split allocates about 25% to the canard and 75% to the wing, although there can be wide variation.
- A 50-50 split produces a tandem-wing aircraft.

Initial Sizing

Control-surface Sizing

- The primary control surfaces are the ailerons (roll), elevator (pitch), and rudder (yaw).
- The required aileron area can be estimated from Fig. 6.3.
- In span, the ailerons typically extend from about 50% to about 90% of the span.

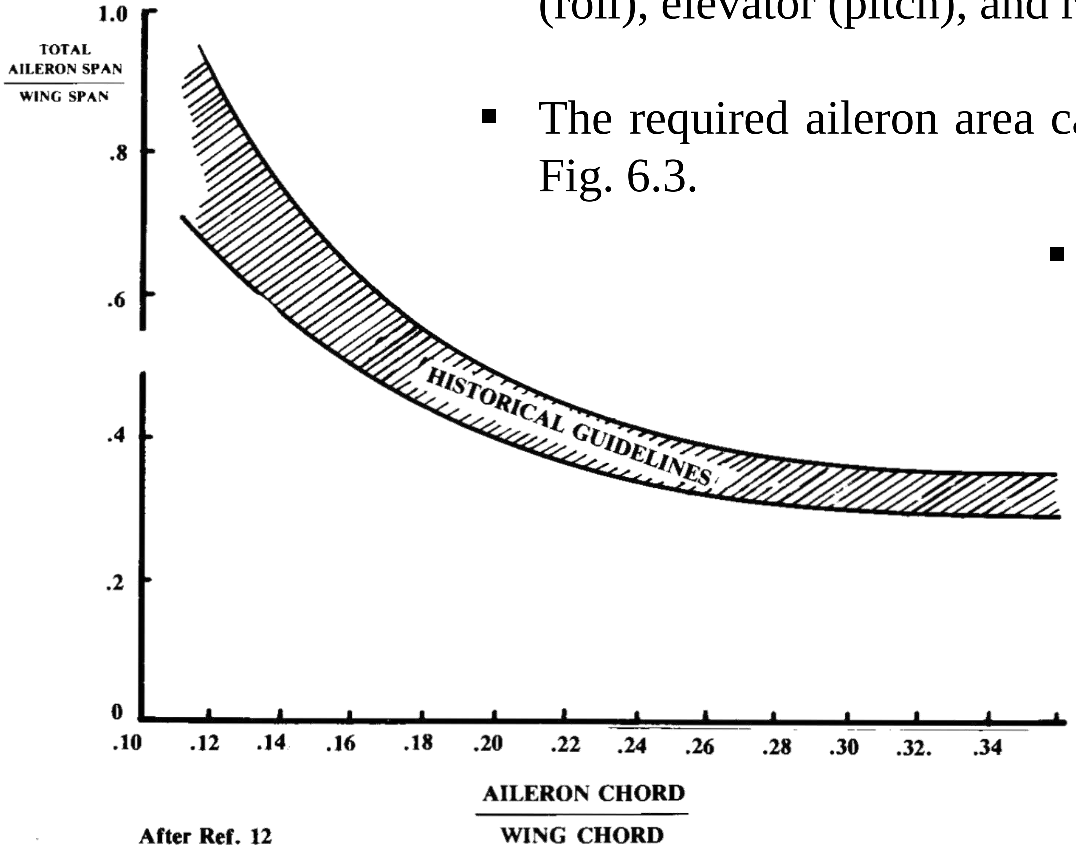
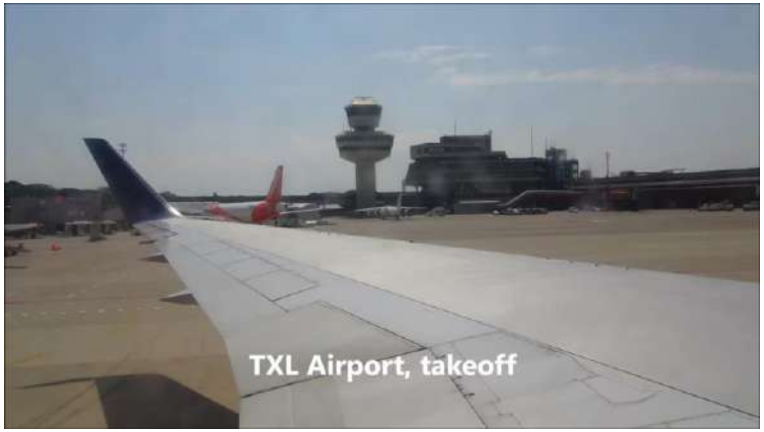


Fig. 6.3 Aileron guidelines.

Initial Sizing

- 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.



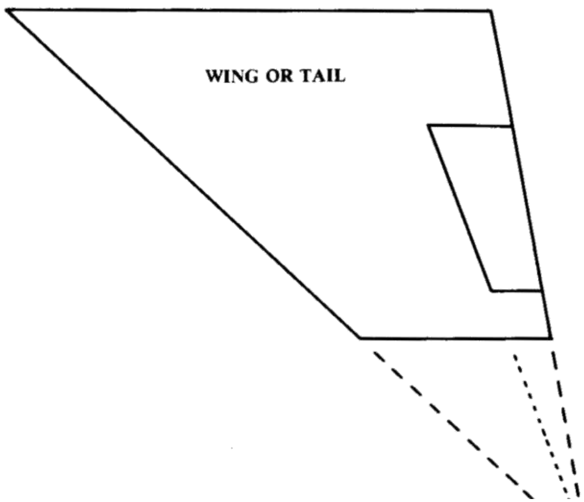
  - Spoilers are plates located forward of the flaps on the top of the wing, typically aft of the maximum thickness point.
  - Spoilers are deflected upward into the slipstream to reduce the wing's lift.
  - Deploying the spoiler on one wing will cause a large rolling moment.

Initial Sizing

- High-speed aircraft can experience a phenomena known as "aileron reversal" in which the air loads placed upon a deflected aileron are so great that the wing itself is twisted.
- To avoid this, many transport jets use an auxiliary, inboard aileron for high-speed roll control.
- Spoilers can also be used for this purpose.
- Several military fighters rely upon "rolling tails" (horizontal tails capable of being deflected nonsymmetrically) to achieve the same result.

Initial Sizing

- 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. 6.4).
- Ailerons and flaps are typically about 15-25% of the wing chord. Rudders and elevators are typically about 25-50% of the tail chord.



Initial Sizing

- Elevators and rudders generally begin at the side of the fuselage and extend to the tip of the tail or to about 90% of the tail span.
- Control-surface "flutter," a rapid oscillation of the surface caused by the airloads, can tear off the control surface or even the whole wing.
- Flutter tendencies are minimized by using mass balancing and aerodynamic balancing.
- Mass balancing refers to the addition of weight forward of the control surface hingeline to counterbalance the weight of the control surface aft of the hingeline.
- This greatly reduces flutter tendencies.

Initial Sizing

- An aerodynamic balance is a portion of the control surface in front of the hinge line.
- This lessens the control force required to deflect the surface, and helps to reduce flutter tendencies.
- The aerodynamic balance can be a notched part of the control surface (Fig. 6.5a), an overhung portion of the control surface (Fig. 6.5b), or a combination of the two.
- The notched balance is not suitable for ailerons or for any surface in high-speed flight.
- The hinge axis should be no farther aft than about 20% of the average chord of the control surface.

Initial Sizing

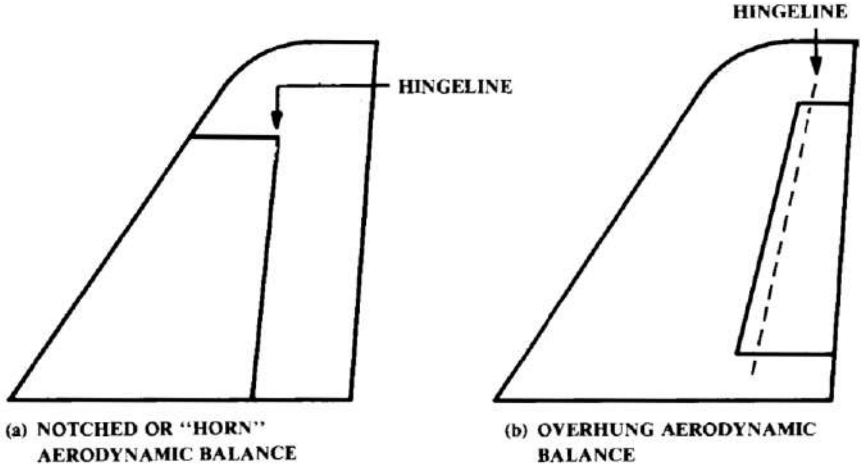


Fig. 6.5 Aerodynamic balance.

Initial Sizing

- Some aircraft have no separate elevator.
- Instead, the entire horizontal tail is mounted on a spindle to provide variable tail incidence.
- This provides outstanding "elevator" effectiveness but is somewhat heavy.

- Some general aviation aircraft use such an all-moving tail.
- But it is most common for supersonic aircraft, where it can be used to trim the rearward shift in aerodynamic center that occurs at supersonic speeds.
- A few aircraft such as the SR-71, J20 have used all-moving vertical tails to increase control authority.

Configuration Layout and Loft

- The process of aircraft conceptual design includes numerous statistical estimations, analytical predictions, and numerical optimizations.
- However, the product of aircraft design is a drawing.
- 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.
- This detailed analysis is time-consuming and costly, so it is essential that the initial drawing be credible.
- Otherwise, substantial effort will be wasted upon analyzing an unrealistic aircraft

Configuration Layout and Loft

- 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.
- A good sketch will show the overall aerodynamic concept and indicate the locations of the major internal components.
- These should include the landing gear, crew station, payload or passenger compartment, propulsion system, fuel tanks, and any unique internal components such as a large radar.

Configuration Layout and Loft

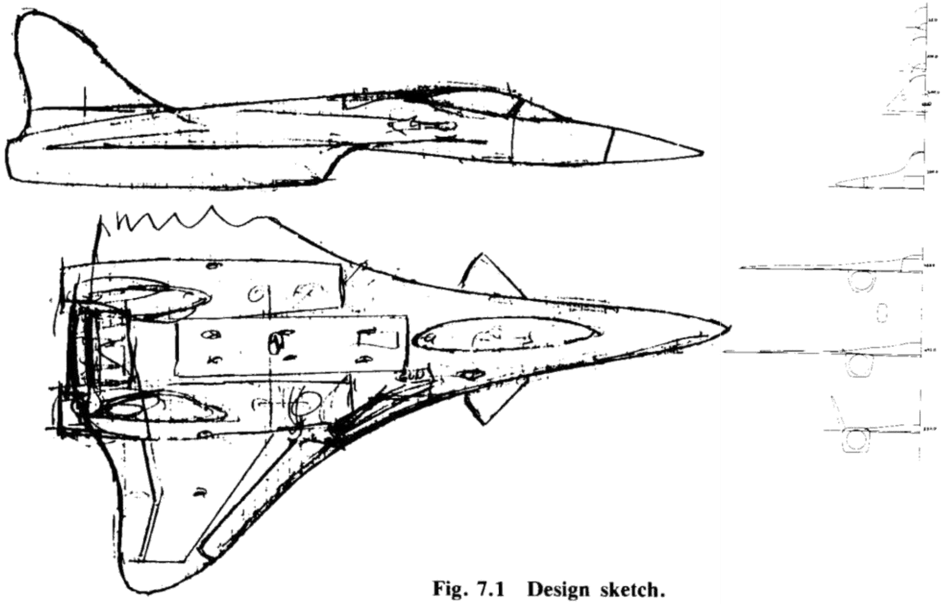


Fig. 7.1 Design sketch.

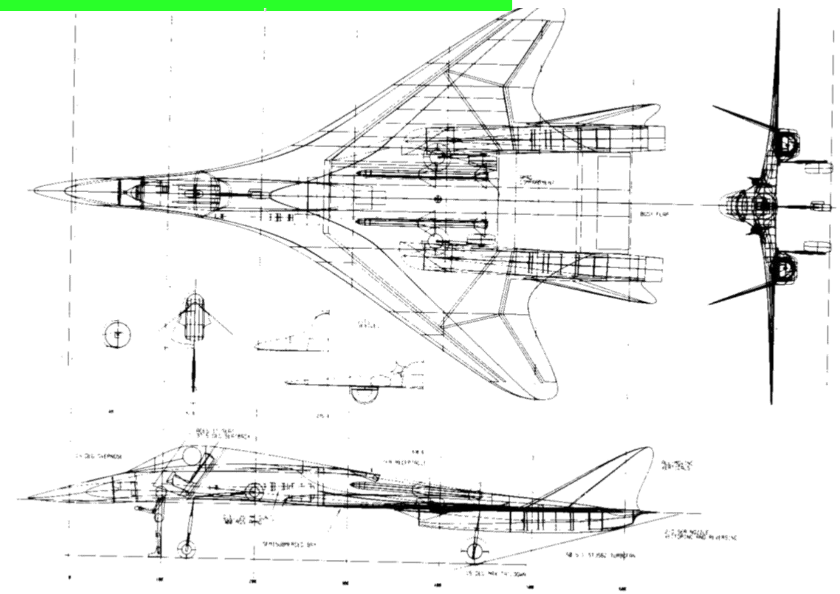


Fig. 7.3 Design layout on a CAD system.

- A design layout represents the primary input into the analysis and optimization tasks.

- Three other inputs must be prepared by the designer:
  - the wetted-area plot,
  - volume distribution plot,
  - and fuel-volume plots for the fuel tanks.

Configuration Layout and Loft

COMPONENT	SURFACE
FUSELAGE	70344.8
VERT TAIL	26165.3
WING	102636.7
CIRCULAR ARC CANOPY	9071.4
NACELLE	25462.9
TOTAL	233681.0

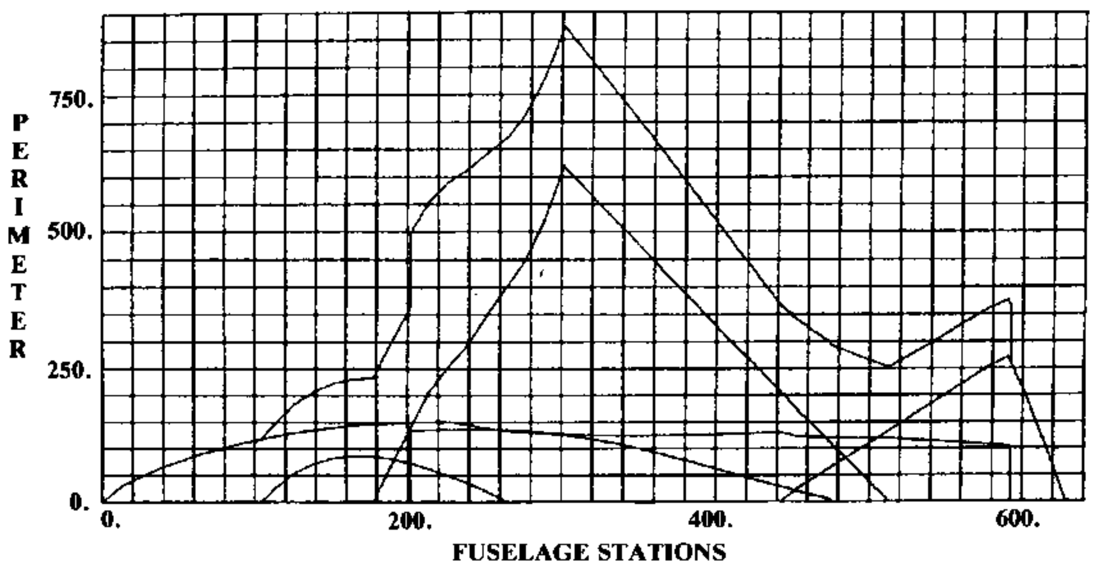


Fig. 7.4 Wetted area plot.

Configuration Layout and Loft

COMPONENT	VOLUME
FUSELAGE	847124.4
VERT TAIL	42903.5
WING	287005.5
CIRCULAR ARC CANOPY	46014.0
NACELLE	95149.8
TOTAL	1318196.8

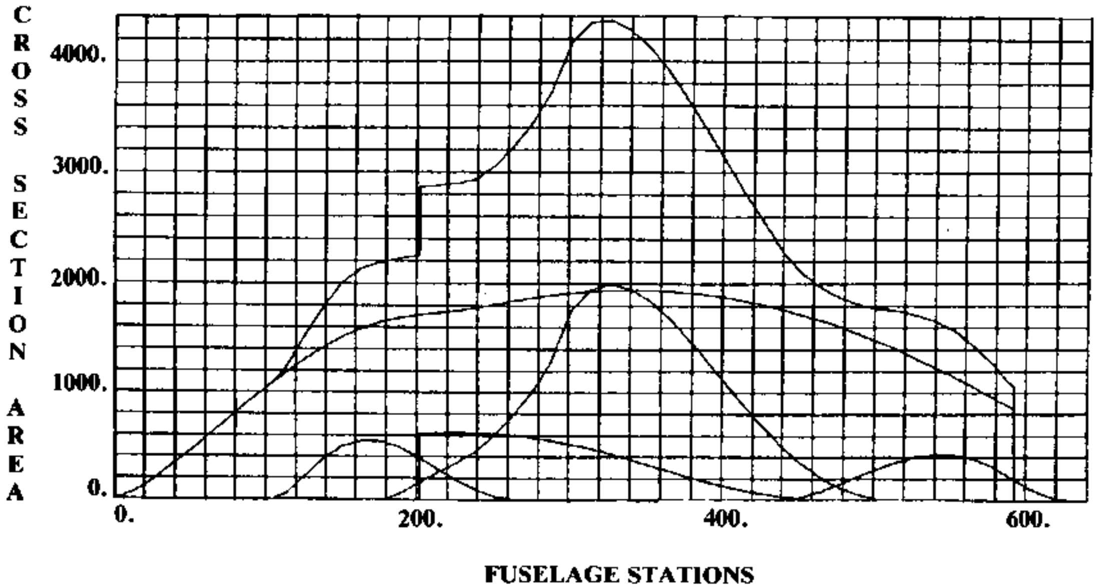


Fig. 7.5 Volume distribution plot.

Configuration Layout and Loft

- Once the design has been analyzed, optimized, and redrawn for a number of iterations of the conceptual design process, a more detailed drawing can be prepared.

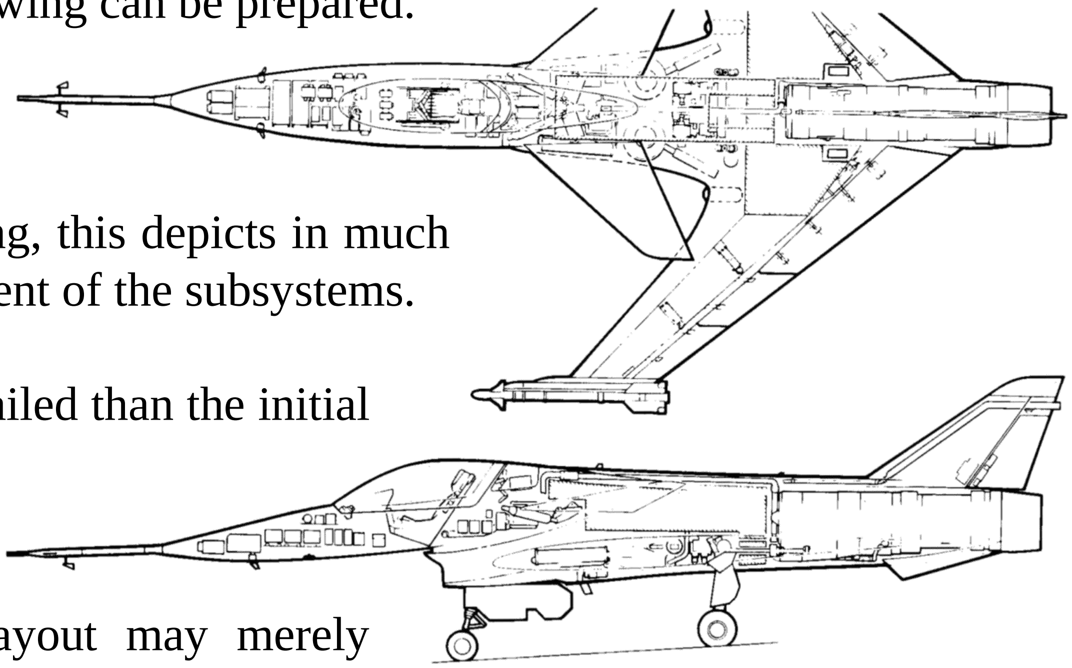


Fig. 7.6 FSW Inboard profile.

- Called the "inboard profile" drawing, this depicts in much greater detail the internal arrangement of the subsystems.
- The inboard profile is far more detailed than the initial layout.
- For example, while the initial layout may merely indicate an avionics, the inboard profile drawing will depict the actual location of every piece of avionics

Configuration Layout and Loft

Conic 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.
- A conic is a second-degree curve whose family includes the circle, ellipse, parabola, and hyperbola.
- The generalized form of the conic is given in Eq. (7.1).

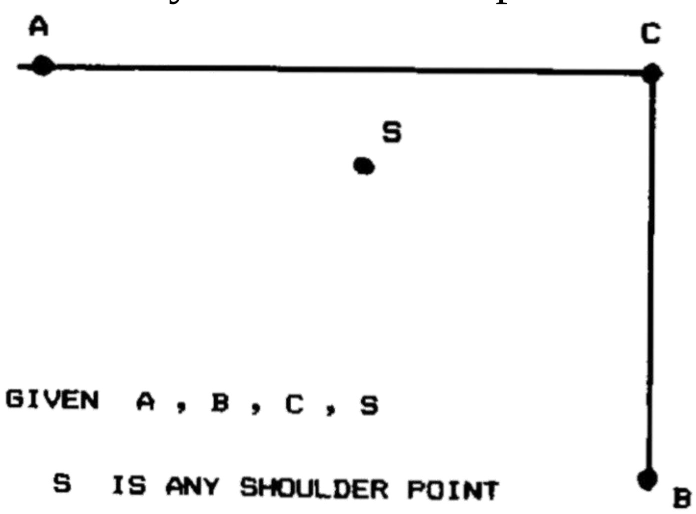
$$C_1X^2 + C_2XY + C_3Y^2 + C_4X + C_5Y + C_6 = 0 \tag{7.1}$$

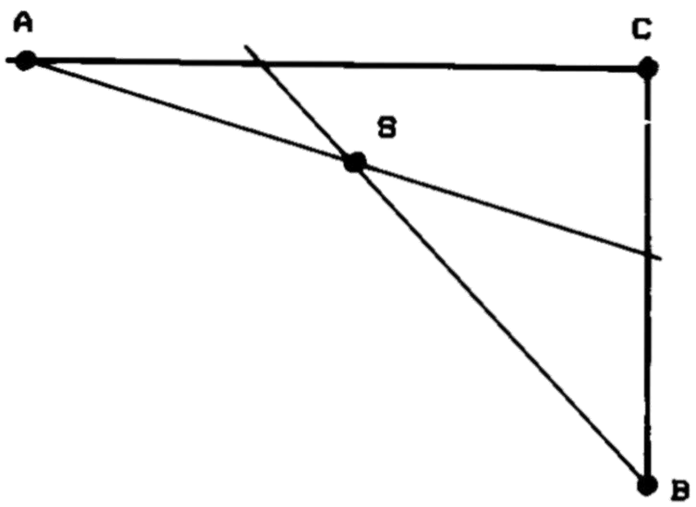
Configuration Layout and Loft

- A conic curve is constructed from the desired start and end points ("A" and "B"), and the desired tangent angles at those points.

- These tangent angles intersect at point "C."

- The shape of the conic between the points A and B is defined by some shoulder point "S."



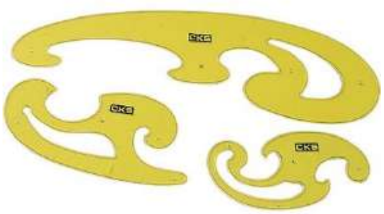


Configuration Layout and Loft

- The first illustration in Fig. 7 .11 shows the given points A, B, C, and S. In the second illustration, lines have been drawn from A and B, passing through S.
- The remaining illustrations show the generation of one point on the conic.
- In the third illustration a line is drawn from point C at an arbitrary angle.
- Note the points where this line intersects the A-Sand B-S lines.
- Lines are now drawn from A and B through the points found in the last step. The intersection of these lines is a point "P" which is on the desired conic curve.

Configuration Layout and Loft

French curve



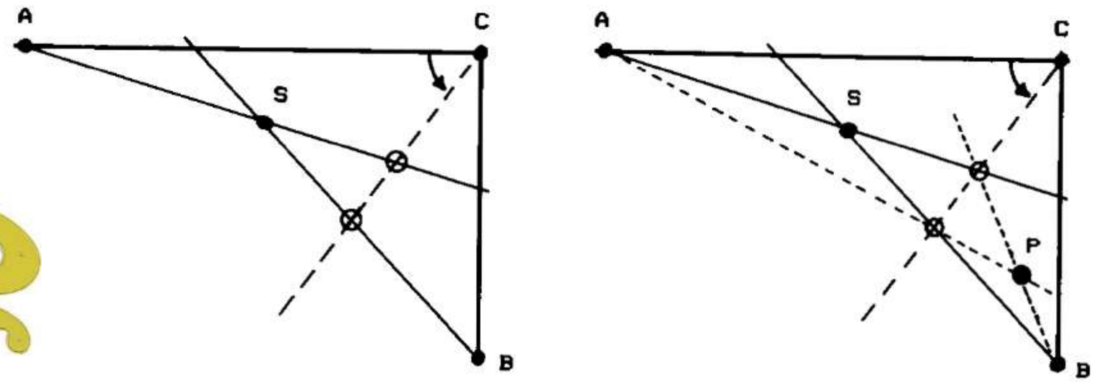


Fig. 7.11 Conic layout.

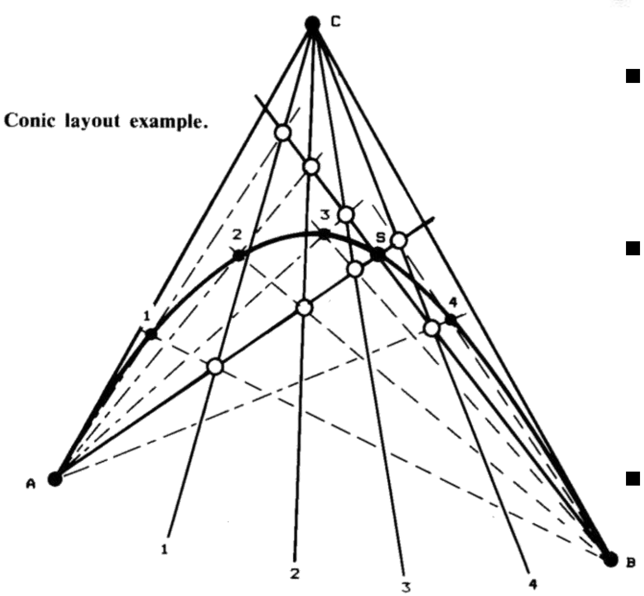


Fig. 7.12 Conic layout example.

- To generate additional points, the last two steps are repeated.
- Another line is drawn from point C at another arbitrary angle, and then the lines from A and B are drawn and their intersection is found.
- When enough points have been generated, a French curve is used to draw the conic.

Configuration Layout and Loft

Longitudinal Control Lines

- To create a smoothly-lofted fuselage using conics, it is necessary only to ensure that the points A, B, C, and S in each of the various cross sections can be connected longitudinally by a smooth line.

- Figure 7 .13 shows the upper half of a simple fuselage, in which the A, B, C, and S points in three cross sections are connected by smooth longitudinal lines.

- These are called "longitudinal control lines" because they control the shapes of the conic cross sections.

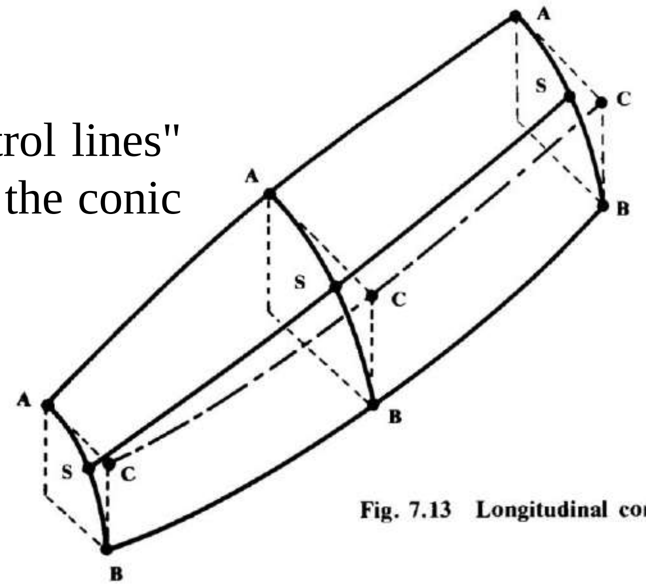


Fig. 7.13 Longitudinal control lines.

Configuration Layout and Loft

- In Fig. 7 .14, the longitudinal control lines are used to create a new cross section, in between the second and third cross sections previously defined.

- This new cross section is created by measuring, from the longitudinal control lines, the positions of the A, B, C, and S points at the desired location of the new cross section.

- As is shown for point A, each point is defined by two measurements, one from side view and one from top view.

- From these points the new cross section can be drawn using the conic layout procedure illustrated in Fig. 7.11.

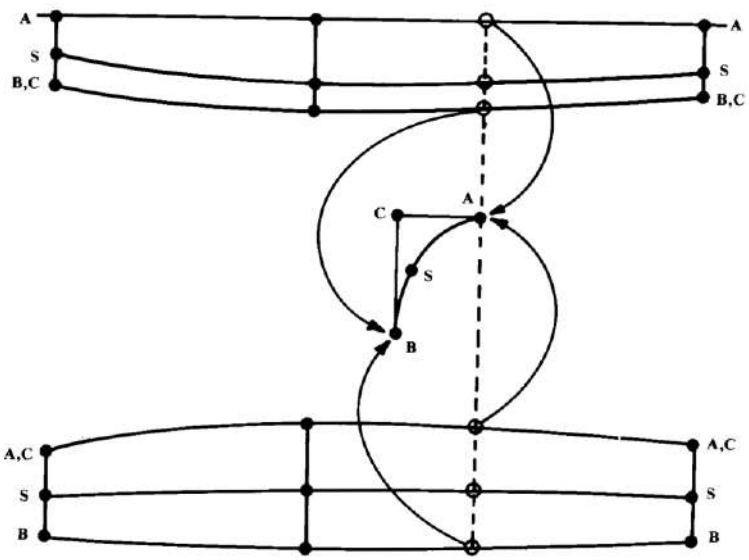


Fig. 7.14 Cross section development from longitudinal control lines.

Configuration Layout and Loft

- The original cross sections that are used to develop the longitudinal control lines are called the "control cross sections," or "control stations. "

- Control stations can also be drawn to match some required shape.

- For example, the last cross section of a single-engined jet fighter with a conventional round nozzle would have to be a circle of the diameter of the nozzle.

- Typically, some five to ten control stations will be required to develop a fuselage that meets all geometric requirements.

- The remaining cross sections of the fuselage can then be drawn from the longitudinal control lines developed from these control stations.

Reference Wing/Tail Layout

- From the parameters;

  - aspect ratio (A),
  - taper ratio (λ),
  - sweep, dihedral,
  - thickness, the selection of an appropriate airfoil,
  - actual sizes for the wing, tails, and fuselage based upon an initial estimate for the takeoff gross weight),

- The geometric dimensions necessary for layout of the reference (trapezoidal) wing or tail can be obtained, as shown in Fig.7 .23 and defined by the following equations:

$$b = \sqrt{AS} \quad (7.5)$$

$$C_{\text{root}} = \frac{2S}{b(1+\lambda)} \quad (7.6)$$

$$C_{\text{tip}} = \lambda C_{\text{root}} \quad (7.7)$$

Configuration Layout and Loft

$$\bar{C} = \left(\frac{2}{3}\right) C_{\text{root}} \frac{1 + \lambda + \lambda^2}{1 + \lambda} \tag{7.8}$$

$$\bar{Y} = \left(\frac{b}{6}\right)\left(\frac{1 + 2\lambda}{1 + \lambda}\right) \tag{7.9}$$

- Figure 7 .23 also shows a quick graphical method of determining the spanwise location of the mean aerodynamic chord (MAC or $\bar{C}$), which is mathematically obtained by Eq. (7.9).

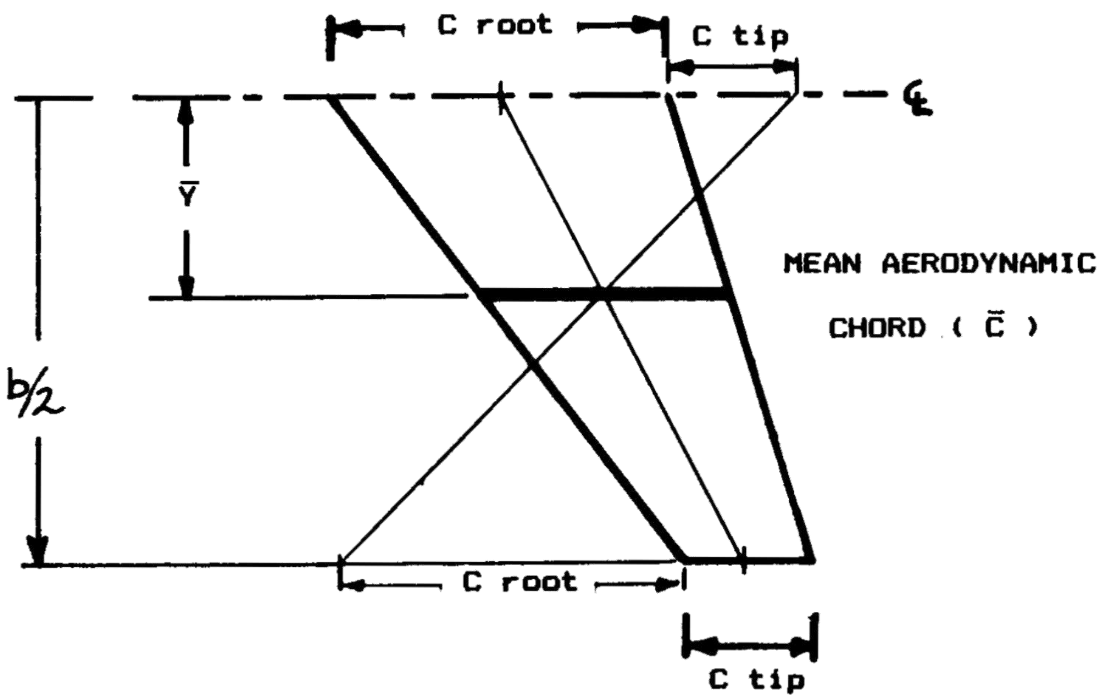


Fig. 7.23 Reference (trapezoidal) wing/tail.

Configuration Layout and Loft

Wing Location with Respect to the Fuselage

- The location and length of the MAC is important.
- Because the wing is located o the aircraft so that some selected percent of the MAC is aligned with the aircraft center of gravity.
- This provides a first estimate of the wing position to attain the required stability characteristics.
- For a stable aircraft with an aft tail, the wing should be initially located such that the aircraft center of gravity is at about 30% of the mean aerodynamic chord.
- When the effects of the fuselage and tail are considered, this will cause the center of gravity to be at about 25% of the total subsonic aerodynamic center of the aircraft.

Configuration Layout and Loft

- For an unstable aircraft with an aft tail, the location of the wing depends upon the selected level of instability,
- but will usually be such that the center of gravity is at about 40% of the mean aerodynamic chord.
- For a control-type canard with a computerized flight control system (i.e., unstable aircraft),
- the wing can be placed such that the aircraft center of gravity is at about 15-25% of the wing’s mean aerodynamic chord.

Configuration Layout and Loft

Wing Fillets

- For improved aerodynamic efficiency, the wing-fuselage connection of most aircraft is smoothly blended using a "wing fillet" (Fig. 7.32).

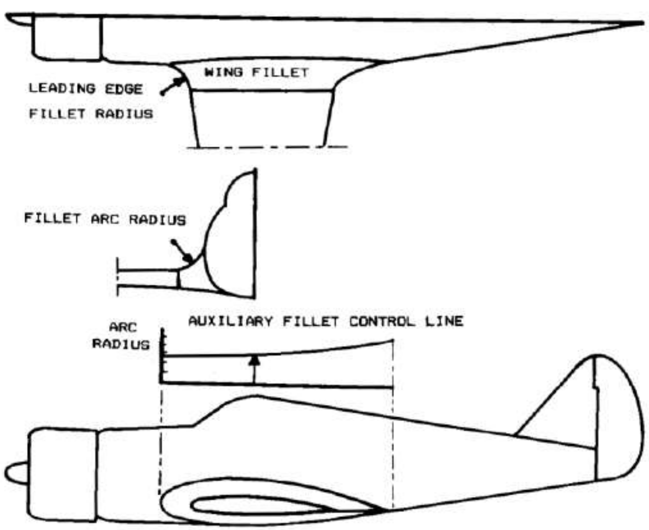


Fig. 7.32 Wing fillet layout.

Configuration Layout and Loft

- A wing fillet is generally defined by a circular arc of varying radius, tangent to both the wing and fuselage.

- Typically a wing fillet has a radius of about 10% of the root-chord length.

- The fillet circular arc is perpendicular to the wing surface, so the arc is in a purely vertical plane only at the maximum thickness point of the wing.

- At the leading edge, the arc is in a horizontal plane.

Configuration Layout and Loft

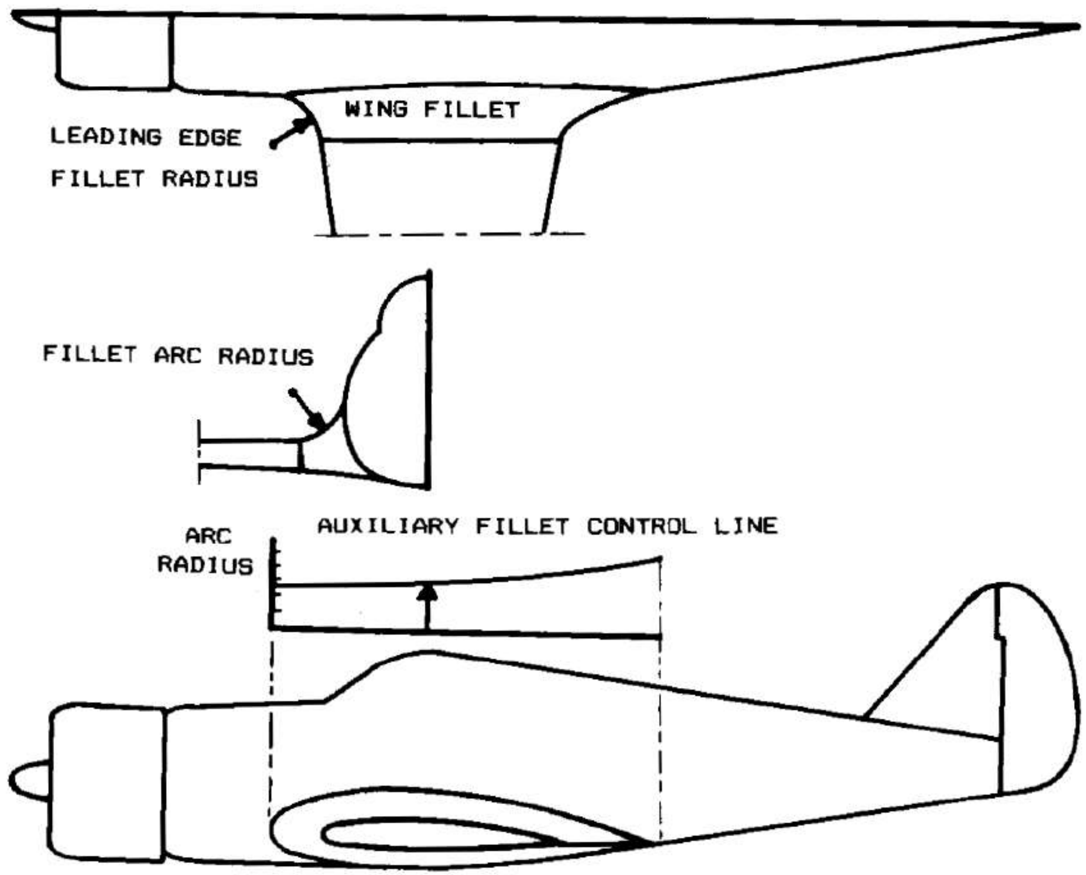


Fig. 7.32 Wing fillet layout.

Configuration Layout and Loft

Aircraft Layout Procedures

- Common drawing scales are 1/10, 1/20, 1/40, and 1/100, depending upon the size of the aircraft.
- Computer-aided design systems usually work in "full scale" on the scope, and any desired scale can be selected if a paper copy is required.
- A minimum number of control stations should be selected,
- to insure that all of the large internal components can be properly enclosed by the aircraft surface.
- Remember that the more control stations selected, the greater the difficulty smooth longitudinal contours.

Configuration Layout and Loft

- The wing and tail trapezoidal geometries should be drawn on separate pieces of paper.

- The mean aerodynamic chord should be shown, including the desired initial location of the wing with respect to the center of gravity.

- The wing drawing can then be slid under the actual drawing and moved to the desired position with respect to the estimated center of gravity location on the fuselage.

- Tails are usually drawn after the fuselage is defined on the drawing.

Configuration Layout and Loft

- The wing and tail geometric parameters should be tabulated somewhere on the drawing,
  - along with the estimated takeoff gross weight,
  - fuel weight and volume,
  - engine type and size (if not 100%),
  - inlet capture area, propeller geometry, etc.

- This information will greatly aid those who later attempt to analyze the drawing.

Wetted Area Determination

- 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 planforms, as shown in Fig. 7.33.

- The wetted area is estimated by multiplying the true-view exposed planform area ($S_{exposed}$) times a factor based upon the wing or tail thickness ratio.

If $t/c < .05$

$$S_{wet} = 2.003\ S_{exposed} \tag{7.10}$$

If $t/c > .05$

$$S_{wet} = S_{exposed}[1.977 + 0.52(t/c)] \tag{7.11}$$

Configuration Layout and Loft

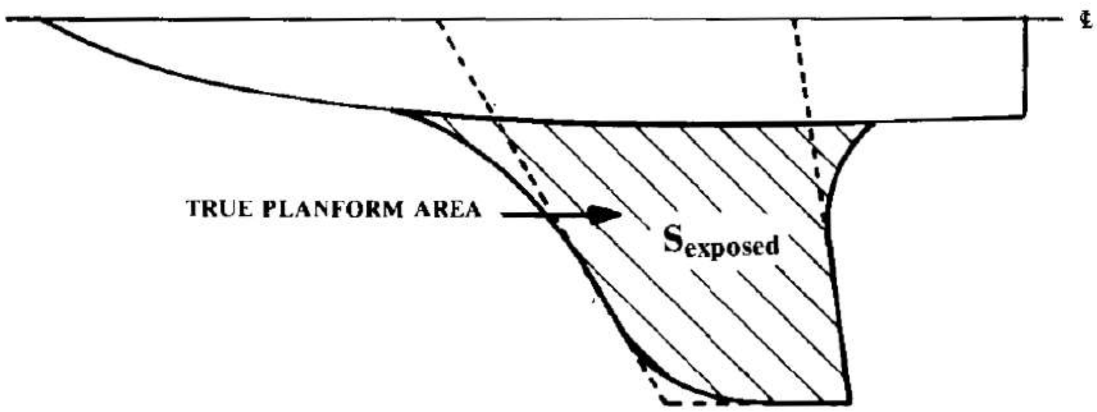
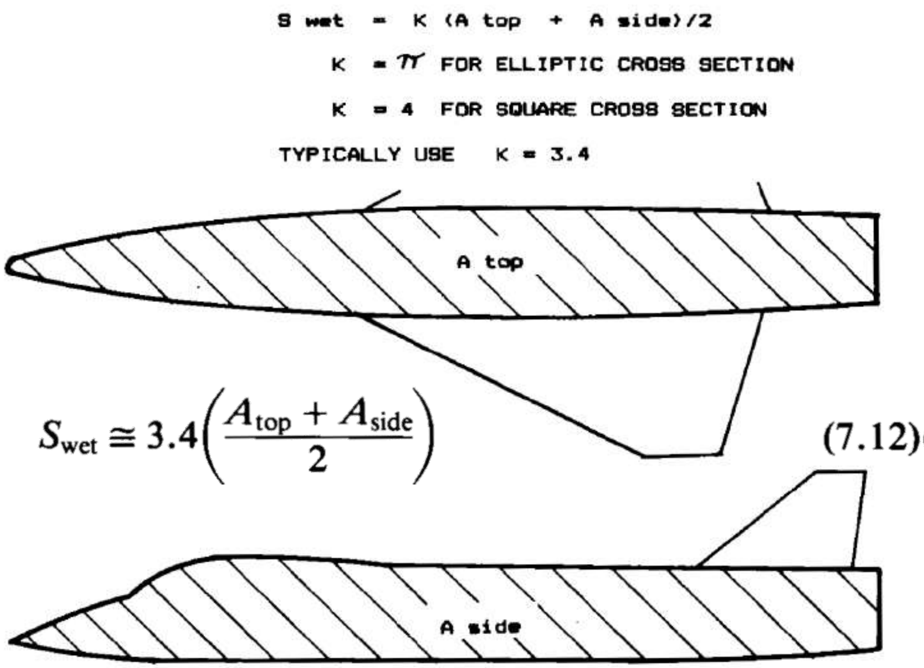


Fig. 7.33 Wing/tail wetted area estimate.

- If a wing or tail were paper-thin, the wetted area would be exactly twice the true planform area (i.e., top and bottom).
- The exposed area shown in Fig. 7.33 can be measured from the drawing in several ways.

Configuration Layout and Loft

- The wetted area of the fuselage can be initially estimated using just the side and top views of the aircraft by the method shown in Fig. 7 .34.



$$S_{wet} \cong 3.4\left(\frac{A_{top} + A_{side}}{2}\right) \tag{7.12}$$

Fig. 7.34 Quick fuselage wetted area estimate.

- For typical aircraft, Eq. (7.12) provides a reasonable approximation.

Configuration Layout and Loft

Volume Determination

- Aircraft internal volume can be estimated in a similar fashion to the wetted-area estimation.

- A crude estimate of the fuselage internal volume can be made using Eq. (7.13), which uses the side and top view projected areas as used in Eq. (7.12).

- "L" in Eq. (7.13) is the fuselage length.

- 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.

- An aircraft with a less than typical internal volume will probably be tightly packed, which makes for poor maintainability.

- Another use of the "volume distribution plot" is to predict and minimize supersonic wave drag and transonic drag rise.

Configuration Layout and Loft

$$\text{Vol} \cong 3.4 \frac{(A_{\text{top}})(A_{\text{side}})}{4L} \tag{7.13}$$

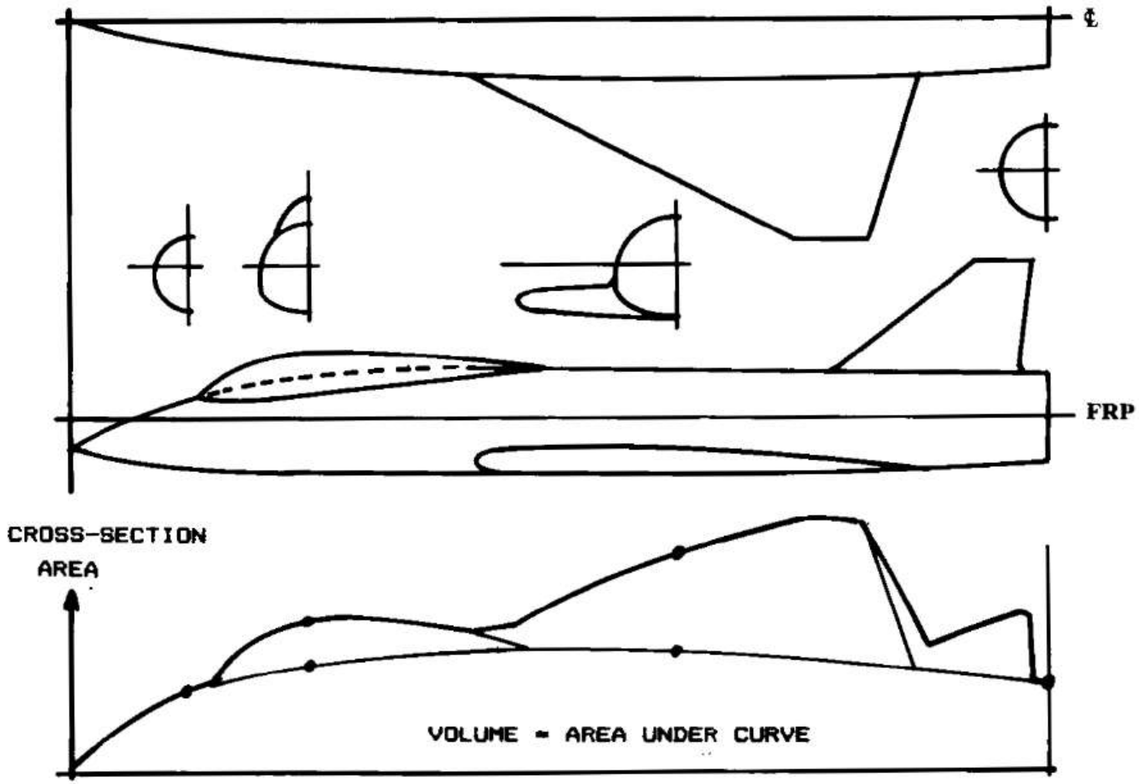


Fig. 7.36 Aircraft volume plot.

Special Considerations in Configuration Layout

- This chapter discusses a number of important intangible/ not mentioned considerations, such as aerodynamics, structures, detectability, vulnerability, producibility, and maintainability.

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.

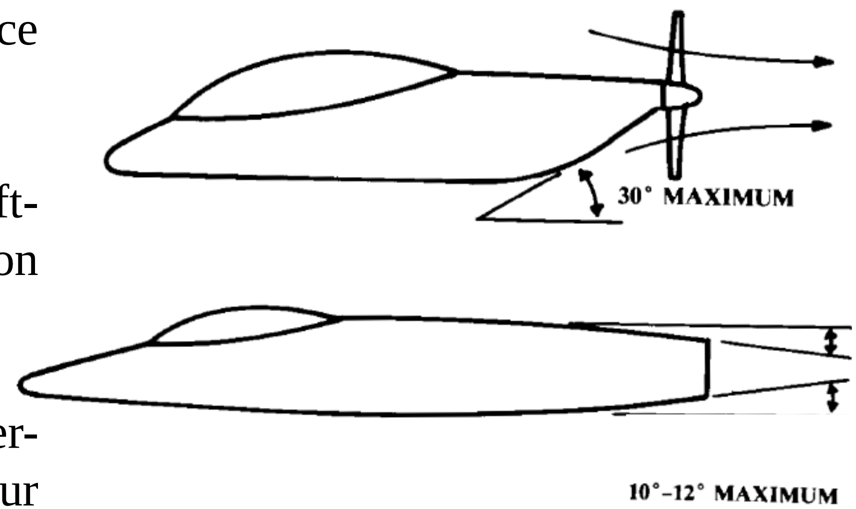
- Also, a poor wing-fuselage arrangement can cause lift losses or disruption of the desired elliptical lift distribution.

Special Considerations in Configuration Layout

- Minimization of wetted area is the most powerful aerodynamic consideration.
- 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.

Special Considerations in Configuration Layout

- Another major driver for good aerodynamic design during fuselage layout is the maintenance of smooth longitudinal contours.
- To prevent separation of the airflow, the aft-fuselage deviation from the freestream direction should not exceed 10-12 deg (Fig. 8.1).
- However, the air inflow induced by a pusher-propeller will prevent separation despite contour angles of up to 30 deg or more.



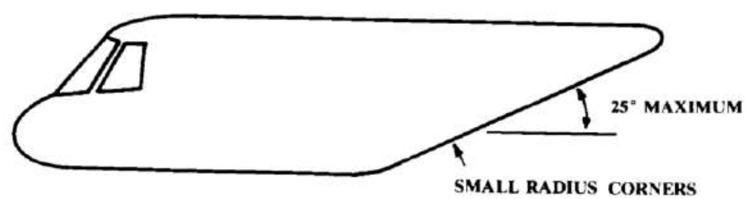


Fig. 8.1 Longitudinal contour guidelines.

Special Considerations in Configuration Layout

- The aerodynamic interaction between different components should be visualized in designing the aircraft.

- For example, a canard should not be located such that its wake might enter the engine inlets at any possible angle of attack.

- Wake ingestion can stall or even destroy a jet engine.

- For supersonic aircraft, the greatest aerodynamic impact upon the configuration layout results from the desire to minimize supersonic wave drag, a pressure drag due to the formation of shocks.

- This is analytically related to the longitudinal change in the aircraft's total cross-sectional area.

Special Considerations in Configuration Layout

- Thus, a "good" volume distribution from a wave-drag viewpoint has the required total internal volume distributed longitudinally in a fashion that minimizes curvature in the volume-distribution plot.

- The "Sears-Haack" body (Fig. 8.2-see Ref. 16) having the lowest wave drag.

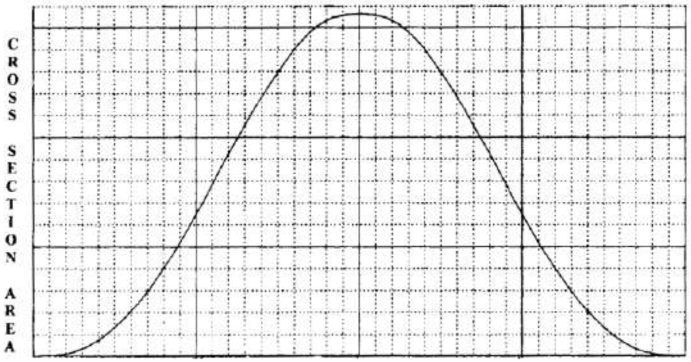


Fig. 8.2 Sears-Haack volume distribution. (the minimum wave drag at Mach 1.0)

Special Considerations in Configuration Layout

- 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.

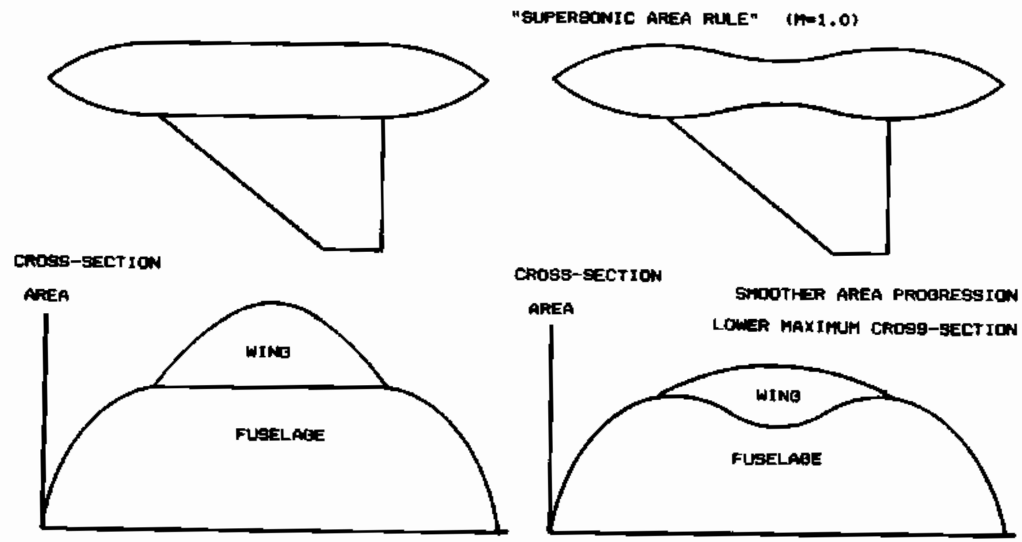


Fig. 8.3 Design for low wave drag.

"area-ruling" or "coke-bottling" and can reduce the wave drag by as much as 500/o.

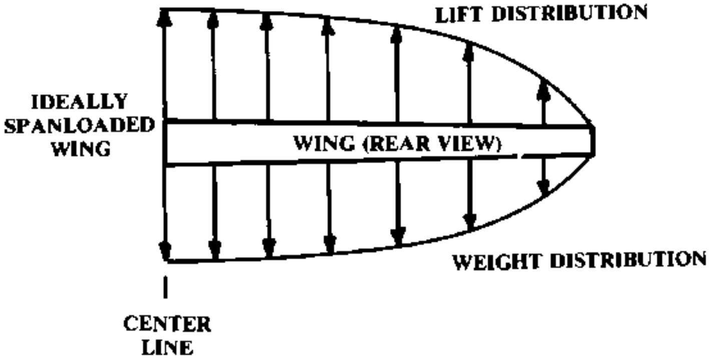
Special Considerations in Configuration Layout

Structural Considerations

- 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 arrangement is 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.

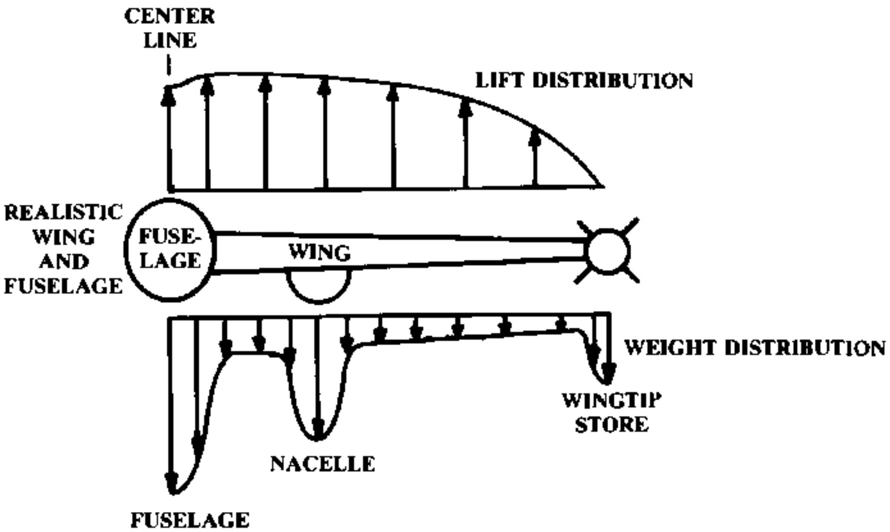
Special Considerations in Configuration Layout

- In a flying wing the lift and weight forces can be located at virtually the same place.
- In the ideal case, the weight of the aircraft would be distributed along the span of the wing exactly as the lift is distributed.
- This is referred to as "spanloading," and eliminates the need for a heavy wing structure to carry the weight of the fuselage to the opposing lift force exerted by the wing.



Special Considerations in Configuration Layout

- While ideal span-loading is rarely possible, the span-loading concept can be applied to more-conventional aircraft by spreading some of the heavy items such as engines out along the wing.
- This will yield noticeable weight savings, but must be balanced against the possible drag increase.



Special Considerations in Configuration Layout

- If the opposing lift and weight forces cannot be located at the same place, then some structural path will be required to carry the load.

- The weight of structural members can be reduced by providing the shortest, straightest load path possible.

- Figure 8.5 illustrates a structural arrangement for a small fighter.

- The major fuselage loads are carried to the wing by "longerons," which are typically I- or H-shaped extrusions running fore and aft and attached to the skin.

- Longerons are heavy, and their weight should be minimized by designing the aircraft so that they are as straight as possible.

Special Considerations in Configuration Layout

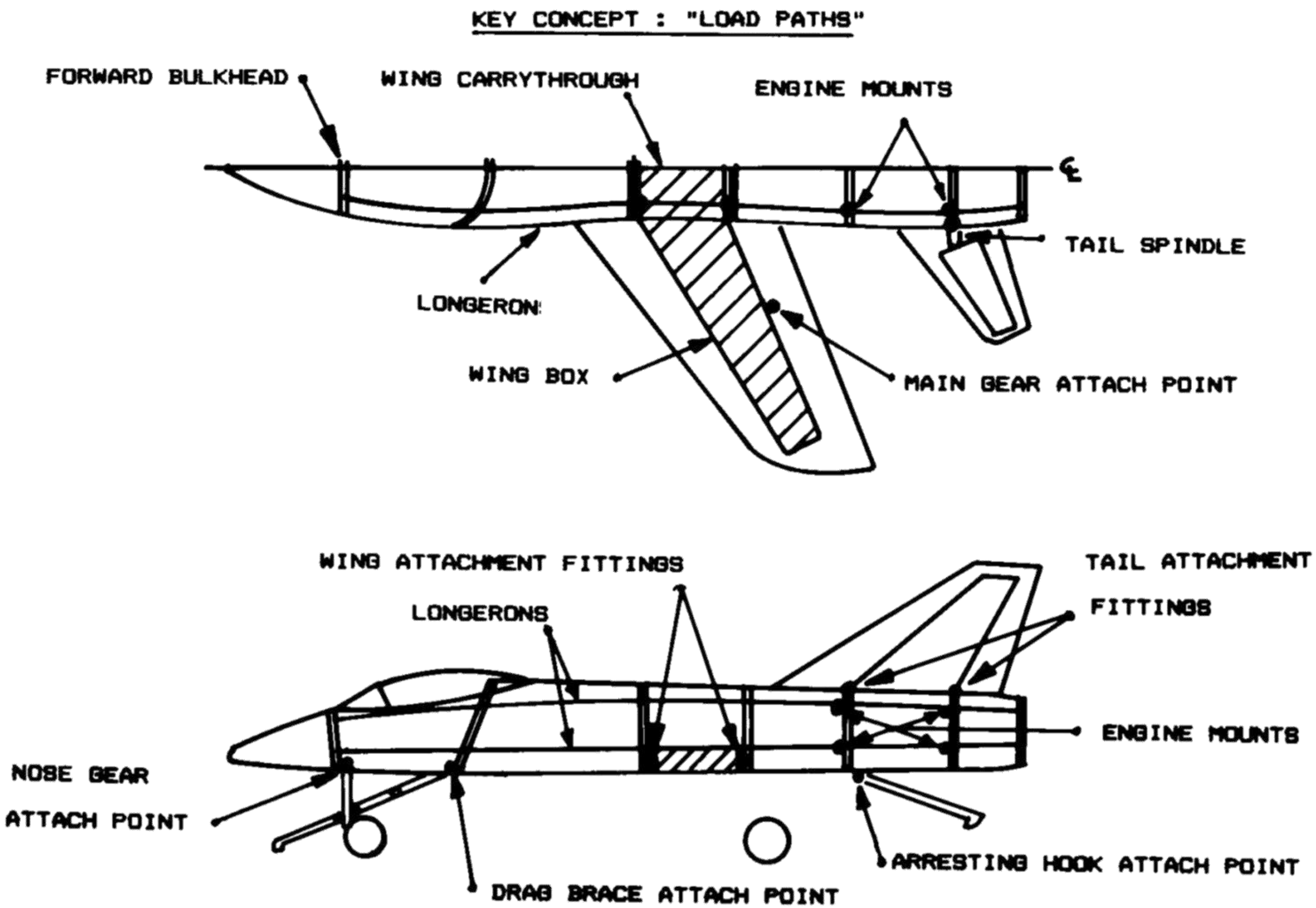


Fig. 8.5 Structural arrangement.

Special Considerations in Configuration Layout

- The purpose of the longeron is to prevent fuselage bending.
- This implies that the lightest longeron structure occurs when the upper and lower longerons are as far apart vertically as possible.
- But this requires a kink to pass over the box.
- For aircraft such as transports, which have fewer cutouts and concentrated loads than a fighter, the fuselage will be constructed with a large number of "stringers",
- which are distributed around the circumference of the fuselage (Fig. 8.7).
- Weight is minimized when the stringers are all straight and uninterrupted.

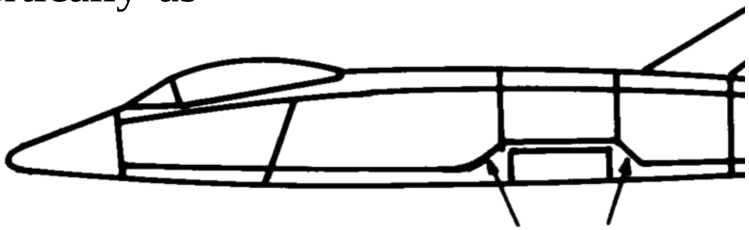
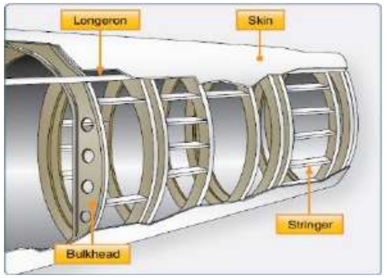


Fig. 8.6 Kinked lower longeron.

Special Considerations in Configuration Layout



Stringer

- If the longitudinal participants are countless (usually 50 to 100), they are named stringers.
- Generally, stringers are of smaller cross-section once contrasted to longerons .
- Stringers frequently take smaller stacks compared to longerons construction.

Longeron

- If the longitudinal participants in a fuselage are limited (usually 4 to 8), they are named longerons.
- Generally, longerons are of greater cross-section once contrasted to stringers.
- Longerons frequently take greater stacks compared to stringers construction.

Special Considerations in Configuration Layout

- Another major structural element used to carry fuselage bending loads is the "keelson."
- This is like the keel on a boat, and is a large beam placed at the bottom of the fuselage as shown in Fig. 8. 7.
- A keelson is frequently used to carry the fuselage bending loads through the portion of the lower fuselage which is cut up by the wheel wells.

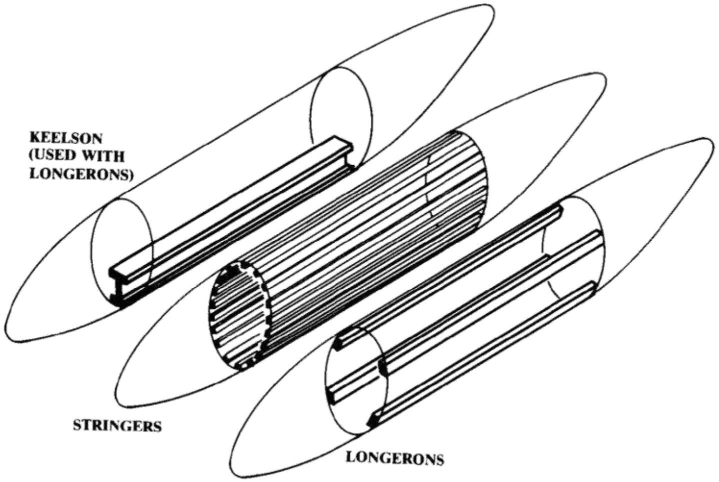


Fig. 8.7 Structural concepts for fuselage loads.

Special Considerations in Configuration Layout

- When possible, structural cutouts should be avoided altogether.
- Required structural cutouts include the cockpit area and a variety of doors (passenger, weapons bay, landing gear, engine access, etc.).
- Large concentrated loads such as the wing and landing gear attachments must be carried by a strong, heavy structural member such as a major fuselage bulkhead.
- The number of such heavy bulkheads can be minimized by arranging the aircraft,
- so that the bulkheads each carry a number of concentrated loads, rather than requiring a separate bulkhead for each concentrated load.

Special Considerations in Configuration Layout

- The lift force on the wing produces a tremendous bending moment where the wing attaches to the fuselage.
- The means by which this bending moment is earned across the fuselage is a key parameter in the structural arrangement,
- and will greatly influence both the structural weight and the aerodynamic drag of the aircraft.
- Figure 8.8 illustrates the four major types of wing carry through structure.

Special Considerations in Configuration Layout

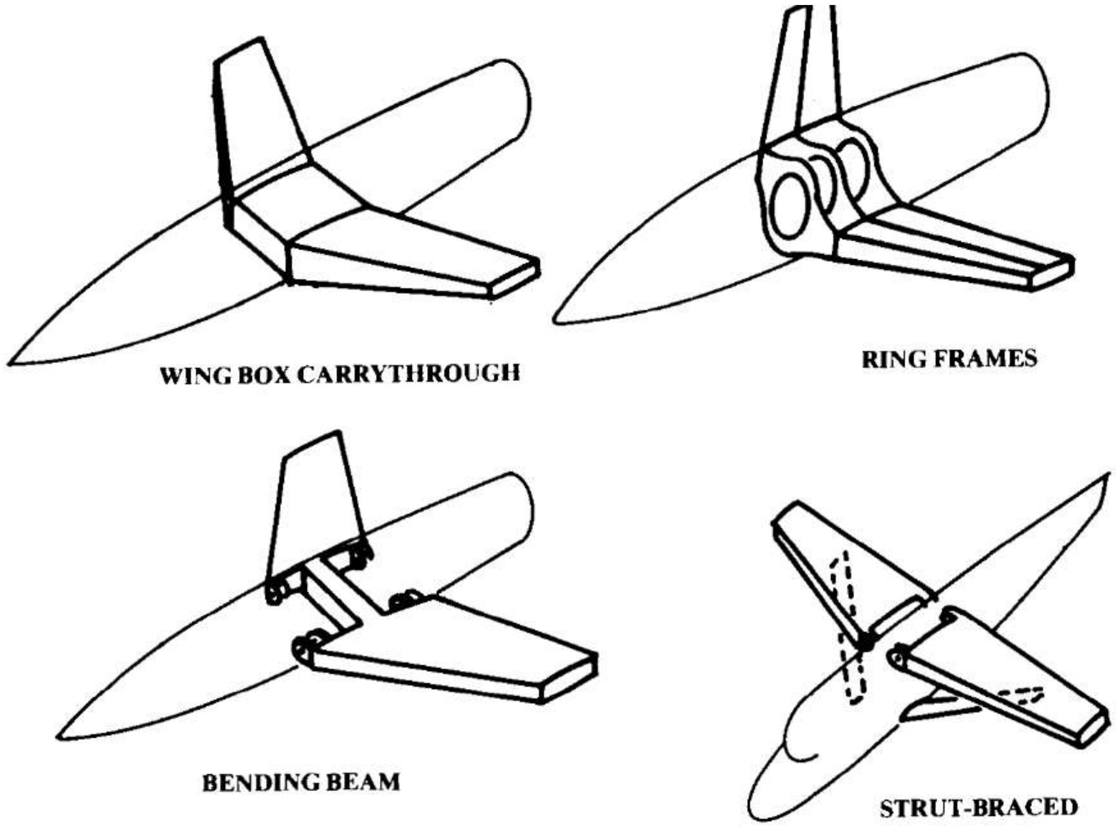
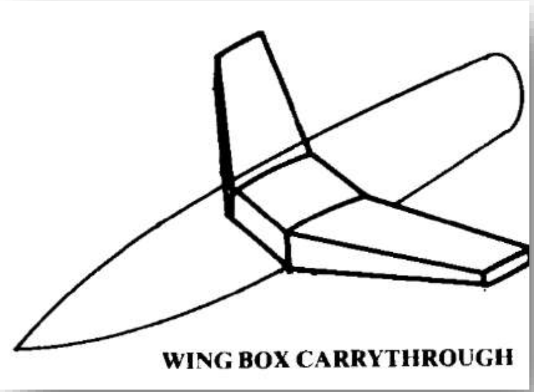


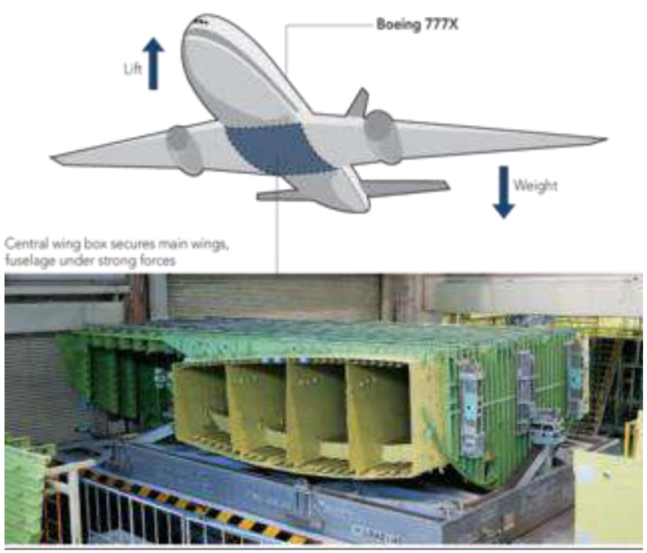
Fig. 8.8 Wing carrythrough structure.

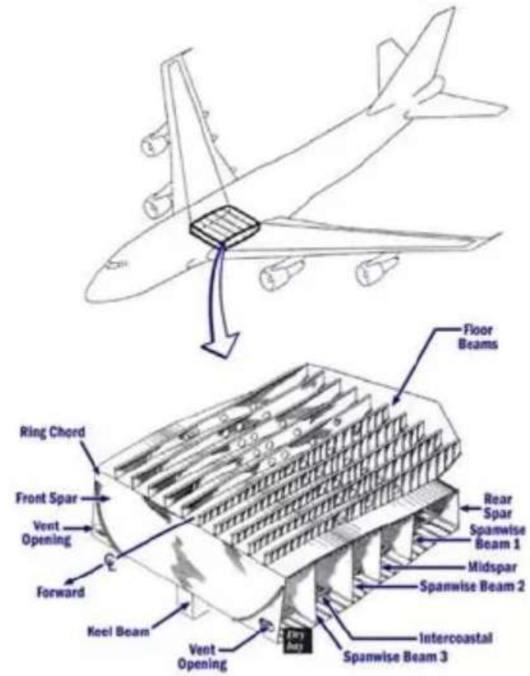
Special Considerations in Configuration Layout

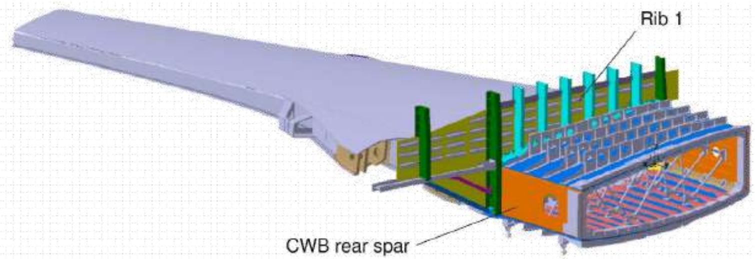


- The "box carry through" is virtually standard for high-speed transports and general-aviation aircraft.
- The box carry through simply continues the wing box through the fuselage.
- The fuselage itself is not subjected to any of the bending moment of the wing, which minimizes fuselage weight.
- However, the box carrythrough occupies a substantial amount of fuselage volume, and tends to add cross-sectional area at the worst possible place for wave drag, as discussed above.
- Also, the box carrythrough interferes with the longeron load-paths.

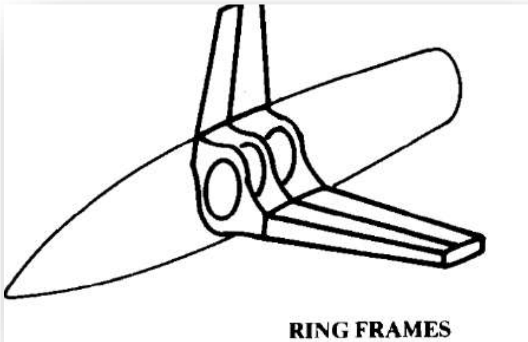
Special Considerations in Configuration Layout

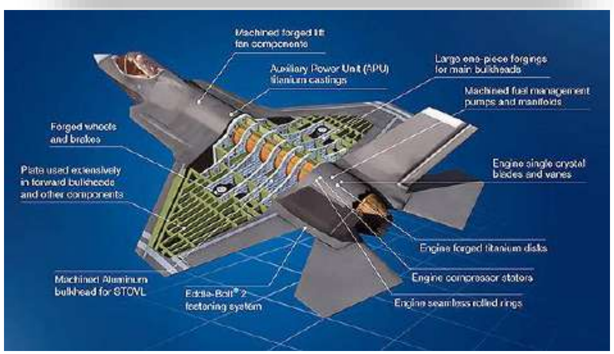






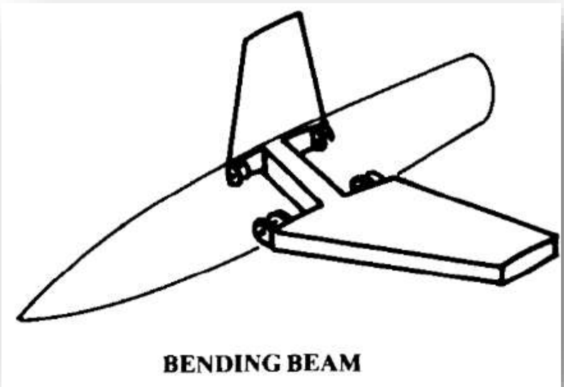
Special Considerations in Configuration Layout

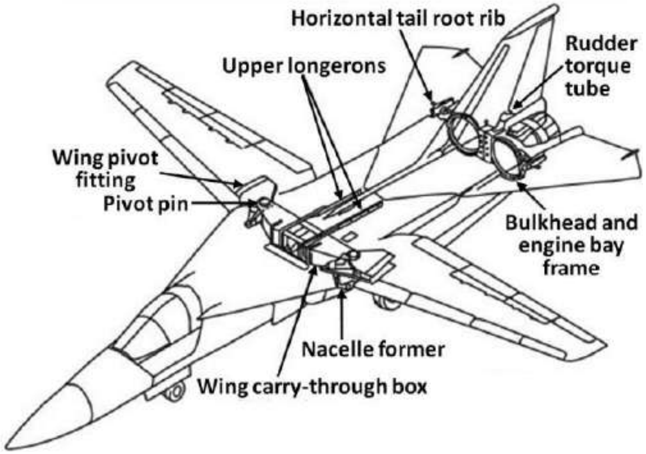




- The "ring-frame" approach relies upon large, heavy bulkheads to carry the bending moment through the fuselage.
- The wing panels are attached to fittings on the side of these fuselage bulkheads.
- While this approach is usually heavier from a structural viewpoint,
- The resulting drag reduction at high speeds has led to the use of this approach for most modern fighters.

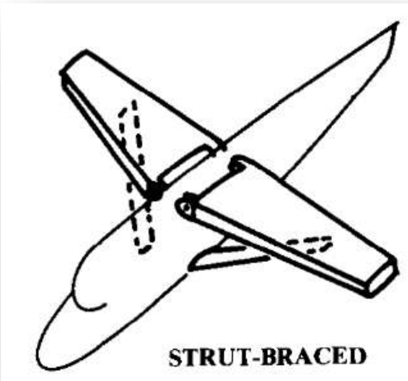
Special Considerations in Configuration Layout





- The "bending beam" carrythrough can be viewed as a compromise between these two approaches.
- Like the ring-frame approach, the wing panels are attached to the side of the fuselage to carry the lift forces.
- However, the bending moment is carried through the fuselage by one or several beams that connect the two wing panels.
- This approach has less of a fuselage volume increase than does the box-carrythrough approach.

Special Considerations in Configuration Layout



- Many light aircraft and slower transport aircraft use an external strut to carry the bending moments.
- While this approach is probably the lightest of all, it obviously has a substantial drag penalty at higher speeds.

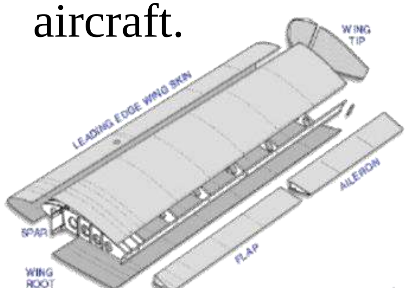
Special Considerations in Configuration Layout

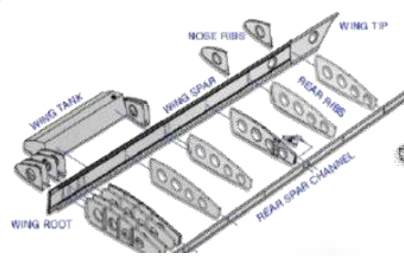
- Aircraft wings usually have the front spar at about 20-30% of the chord back from the leading edge.

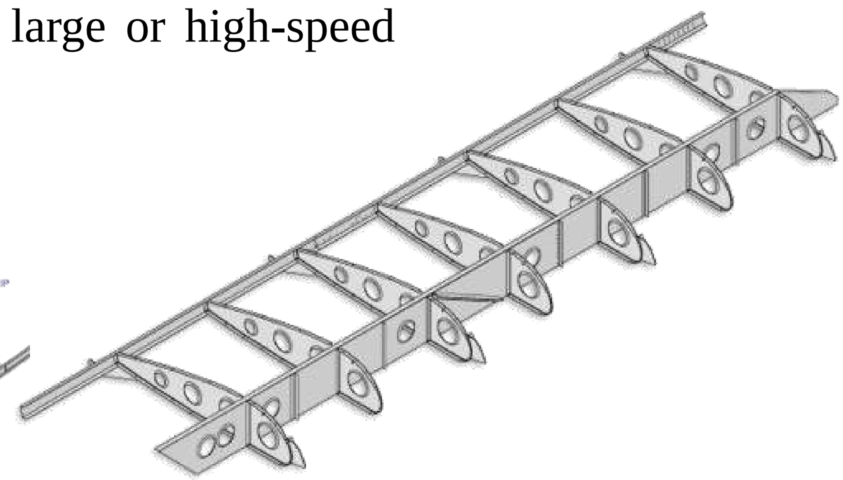
- The rear spar is usually at about the 60-75% chord location.

- Additional spars may be located between the front and rear spars forming a "multispar" structure.

- Multispar structure is typical for large or high-speed aircraft.







Special Considerations in Configuration Layout

- Another form of wing structure, the "multirib" or "stringer panel" box, has only two spars, plus a large number of spanwise stringers attached to the wing skins.

- Numerous ribs are used to maintain the shape of the box under bending.

- 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 in Fig. 8.9.

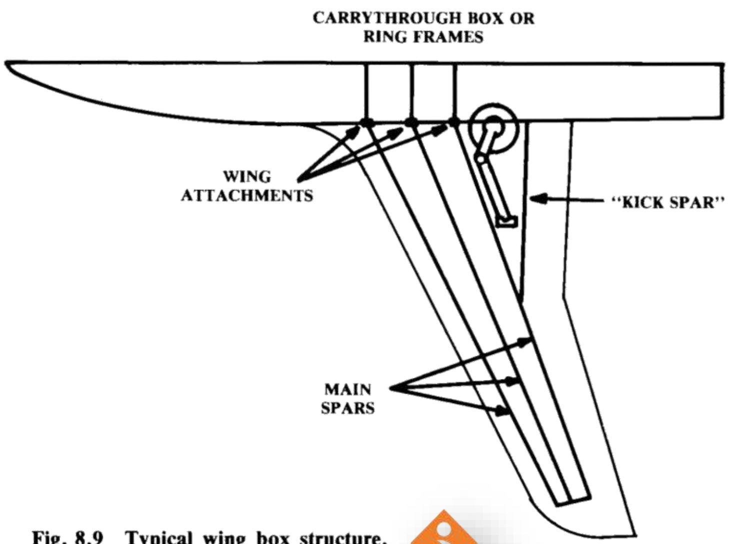


Fig. 8.9 Typical wing box structure.

Special Considerations in Configuration Layout

- For initial layout purposes the designer must guess at the amount of clearance required for structure around the internal components.
- A large airliner will typically require about 4 in. of clearance from the inner wall of the passenger compartment to the outer skin ("moldline").
- The structure of a conventional fighter fuselage will typically require about 2 in. of offset from the moldline for internal components.
- For a small general aviation aircraft, 1 in. clearance or less may be acceptable.

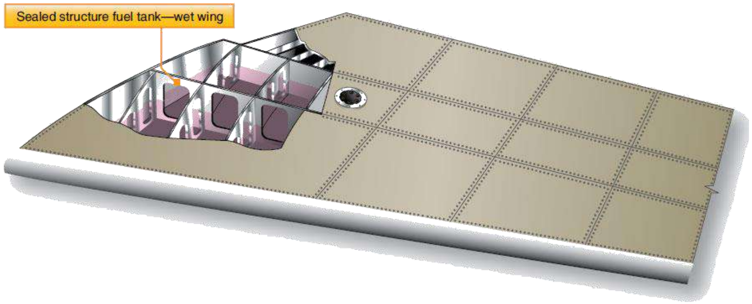
Special Considerations in Configuration Layout

- The type of internal component will affect the required clearance.

- A jet engine contained within an aluminum or composite fuselage will require perhaps an additional inch of clearance to allow for a heat shield.

- The heat shield may be constructed of titanium, steel, or a heat-proof matting.

- On the other hand, an "integral" fuel tank in which the existing structure is simply sealed and filled with fuel will require no clearance other than the thickness of the skin.



Special Considerations in Configuration Layout

Radar Detectability

- 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.
- Usually the transmitter and receiver antennas are~ collocated ("monostatic radar"), although some systems have them m different locations ("bistatic radar").

Special Considerations in Configuration Layout

- 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.

  - Chaff obscures the actual location of the aircraft, but does not allow the aircraft to pass unnoticed.

  - To avoid detection, the aircraft must return such a low amount of the transmitted radio beam that the receiver antenna cannot distinguish between it and the background radio static.

Special Considerations in Configuration Layout

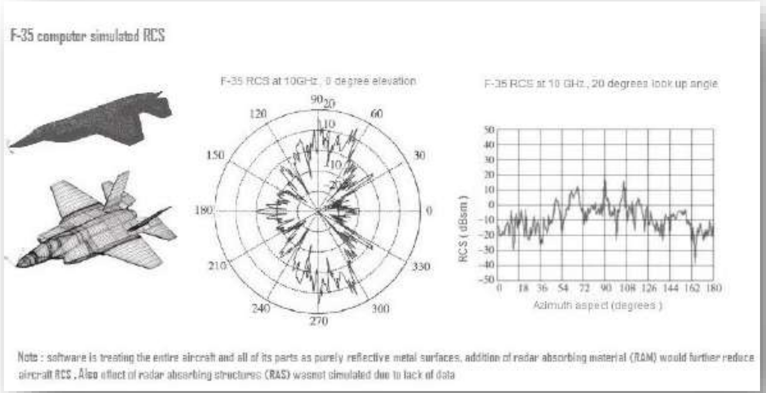
- The extent to which an object returns electromagnetic energy is the object's "Radar Cross Section" (RCS).

- RCS is usually measured in square meters or in decibel square meters.

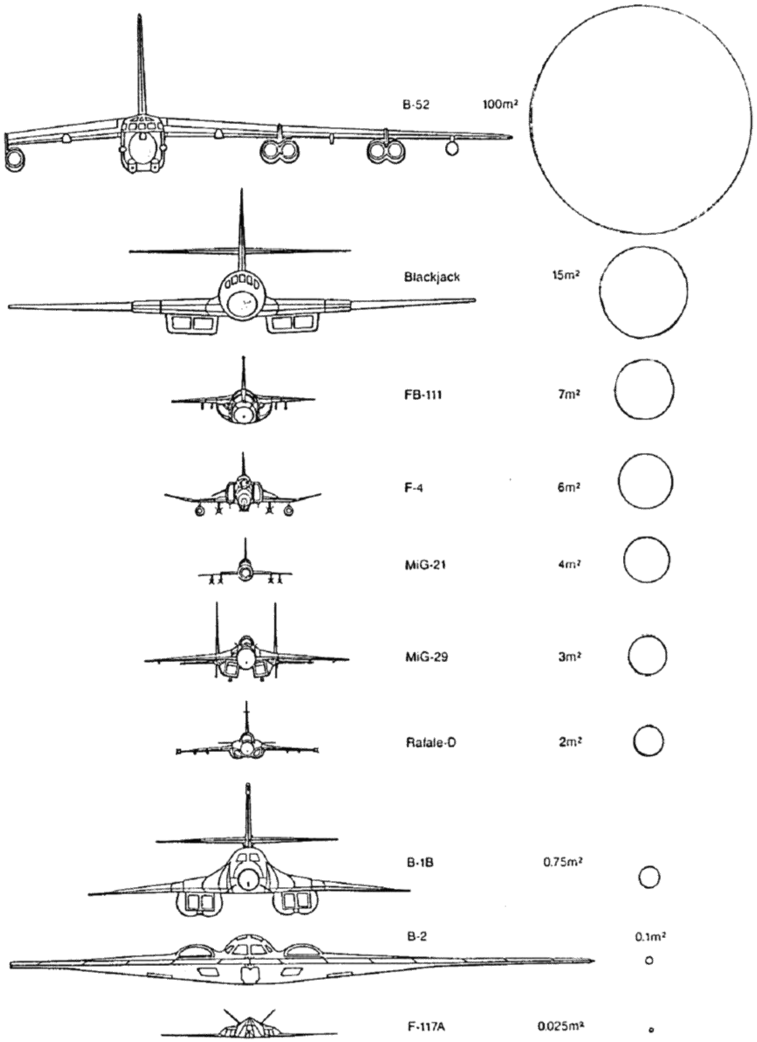
- Actually, the RCS of an aircraft is not a single number.

- The RCS is different for each "look-angle" (i.e., direction from the threat radar).

- Also, the RCS varies depending upon the frequency and polarization of the threat radar.



Special Considerations in Configuration Layout



- There are many electromagnetic phenomena that contribute to the RCS of an aircraft.

- These require different design approaches for RCS reduction,

- and can produce conflicting design requirements.

- Figure 8.1 illustrates the major RCS contributors for a typical, untreated fighter aircraft.

Special Considerations in Configuration Layout

- 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.

- To prevent these RCS "spikes," the designer may slope the fuselage sides, angle the vertical tails, and so on,

- so that there are no flat surfaces presented towards the radar

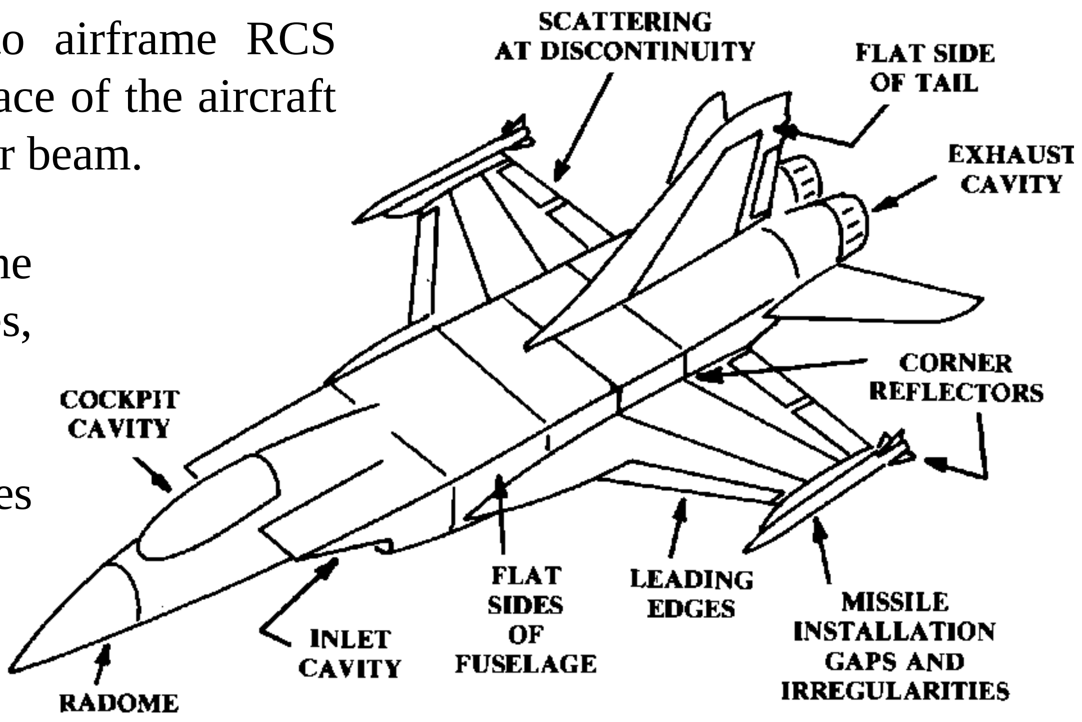


Fig. 8.10 Major RCS contributors.

Special Considerations in Configuration Layout

- 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.

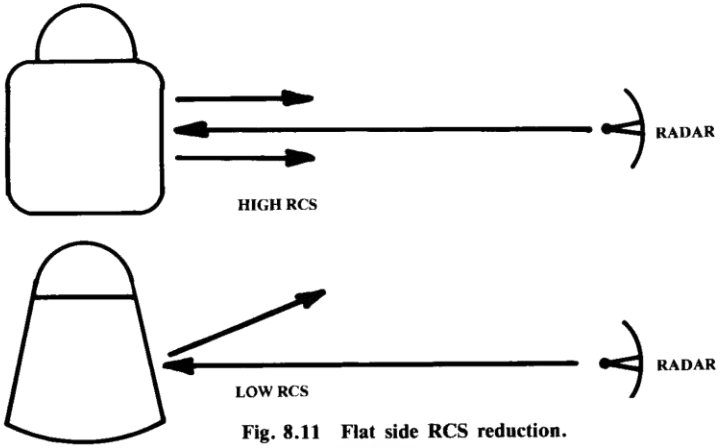


Fig. 8.11 Flat side RCS reduction.

Special Considerations in Configuration Layout

- Another contributor to airframe RCS occurs due to the electromagnetic currents that build up on the skin when illuminated by a radar.

- These currents flow across the skin until they hit a discontinuity such as at a sharp trailing edge, a wing tip, a control surface, or a crack around a removable panel or door.

- At a discontinuity, the currents "scatter," or radiate electromagnetic energy, some of which is transmitted back to the radar (Fig. 8.12).

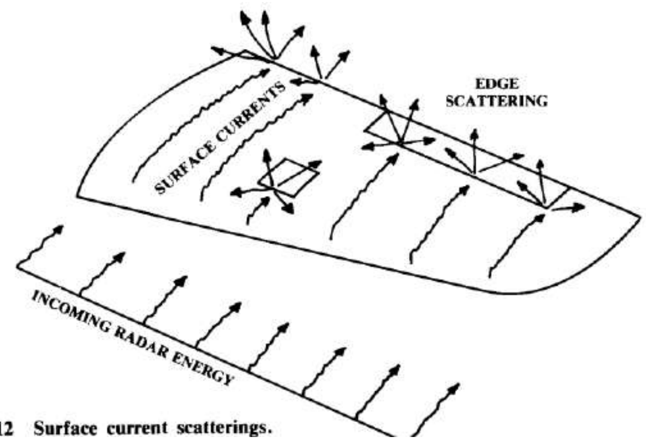


Fig. 8.12 Surface current scatterings.

Special Considerations in Configuration Layout

- 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 sawtooth-shaped edges on every door, access plate, and other discontinuity on the aircraft, as seen on the B-2 and F-117.

Special Considerations in Configuration Layout

- First-generation stealth designs such as the Lockheed F-117 relied upon faceted shaping in which the aircraft shape is constructed of interlocking flat triangles and trapezoids.

- This has advantages in ease of construction and signature analysis,

- but offers a large number of sharp edges to create diffraction returns, and so is no longer in favor

Special Considerations in Configuration Layout

- Current stealth design begins by configuring the aircraft such that all "big" returns, such as from perpendicular bounces, are "aimed" in just a few directions.

- For example, if the leading edges of the wings and tails are all straight and set at the same angle, there would be a huge radar return from that angle direction,

- but little return from other directions.

- This would presumably offer a small probability that the aircraft and threat radar would be mutually oriented in exactly the angle of high return,

- and the aircraft would be undetectable from all other angles.

Special Considerations in Configuration Layout

- It is also common practice to "aim" the wing leading-edge return in the same direction as the edge diffraction return from the trailing edge.

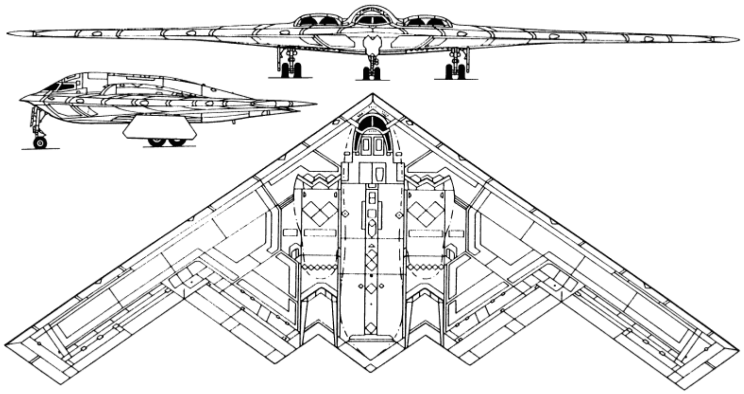
- This is done either by using identical sweep angles at the leading and trailing edges,

- or by aligning the left wing leading edge at the same sweep angle as the right wing trailing edge.

a diamond wing as on the F-23

Special Considerations in Configuration Layout

- Once all wing and tail returns are "aimed" in the same direction,



- the returns from doors, access panels, and other discontinuities can be "aimed" in the same direction by alignment of their edges.

- This is clearly seen on the B-2 where virtually every feature on the aircraft,

- including weapons, bay doors, gear doors, inlets, nozzles, and access panels, is constructed using only lines which are parallel to a wing leading edge.

Special Considerations in Configuration Layout

- RCS can also be reduced simply by eliminating parts of the aircraft.

- Modern computerized flight controls combined with the use of vectored-thrust engines can solve many of the difficulties of the tailless configuration.

- Similarly, RCS can be reduced if the nacelles can be eliminated through the use of buried engines,

- or better yet, by eliminating the entire fuselage through the use of the flying-wing concept.

- This approach is used in the Northrop B-2.

Special Considerations in Configuration Layout

- In addition to reshaping the aircraft, detectability can be reduced through the use of skin materials that absorb radar energy.

- Such materials, called "radar absorbing materials" (RAM), are typically composites such as fiberglass embedded with carbon or ferrite particles.

- These particles are heated by the radar electromagnetic waves, thus absorbing some of the energy.

- This will reduce (not eliminate!) the radar return due to perpendicular bounce,

- and can also reduce the surface currents and thus reduce the RCS due to scattering at sharp edges.

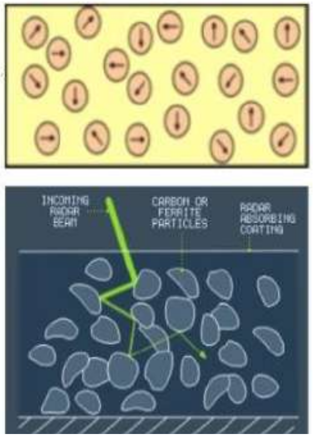
Special Considerations in Configuration Layout

- As there are many types of RAM and similar treatments, no quick estimate for the weight impact of their use can be provided here.

- However, one can probably assume that such use will reduce or eliminate any weight savings otherwise assumed for the use of composite materials.

2. Radar Absorbing Material(RAM)

- Nanostructural composite material, absorbing without reflection radar wave.
- R.A.M. is basically paint with which external surfaces are coated with.
- These paints are Magnetic ferrite-based substance having ingredients dielectric, such as carbon.
- RAM reduces the radar cross section making the object appear smaller.



Special Considerations in Configuration Layout

- For most existing aircraft, the airframe is not the largest contributor to RCS, especially nose-on.

- A conventional radome, covering the aircraft's own radar, is transparent to radar for obvious reasons.

  - Therefore, it is also transparent to the threat radar, allowing the threat radar's beam to bounce off the forward bulkhead and electronic equipment within the radome.

  - Even worse, the aircraft's own radar antenna, when illuminated by a threat radar, can produce a radar magnification effect much like a cat's eye.

  - These effects can be reduced with a "bandpass" radome, which is transparent to only one radar frequency (that of the aircraft's radar).

Special Considerations in Configuration Layout

- Other huge contributors to the RCS for a conventional aircraft are the inlet and exhaust cavities.
- The best solution for reducing these RCS contributions is to hide them from the expected threat locations.
- For example, inlets can be hidden from ground-based radars by locating them on top of the aircraft.
- Exhausts can be hidden through the use of two-dimensional nozzles.
- Cockpits provide a radar return for a similar reason.
- One solution for this is to thinly coat the canopy with some conductive metal such as gold, causing the canopy to reflect the radar energy away.

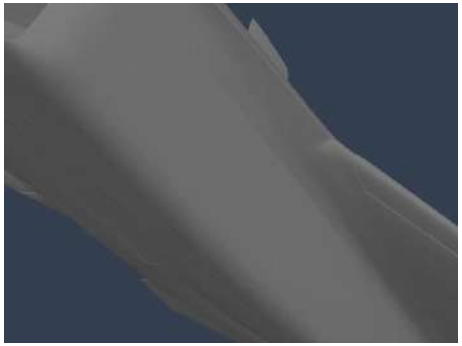
Special Considerations in Configuration Layout

- Metallized canopy,
- Sloped fuselage sides,
- No external weapons,
- Top-mounted inlet,
- Radar absorbing material coating,
- Shielded nozzle,
- Eliminated horizontal tail,
- Canted/inclined verticals,
- No corner reflectors…

F 117

Special Considerations in Configuration Layout

- Finally, the aircraft's weapons can have a major impact on RCS.
- Missiles and bombs have fins that form natural corner reflectors.
- The carriage and release mechanisms have numerous corner reflectors, cavities, and surface discontinuities.



- Gun ports present yet another kind of cavity.
- The only real solution for these problems is to put all the weapons inside, behind closed doors.
- However, the weight, volume, and complexity penalties of this approach must be carefully considered.

Special Considerations in Configuration Layout

Infrared Detectability

- Infrared (IR) detectability also concerns the aircraft designer.
- Many short-range air-to-air and ground-to-air missiles rely upon IR seekers.
- Modern IR sensors are sensitive enough to detect not only the radiation emitted by the engine exhaust and hot parts,
- but also that emitted by the whole aircraft skin due to aerodynamic heating at transonic and supersonic speeds.

Special Considerations in Configuration Layout

- Of several approaches for reduction of IR detectability, one of the most potent reduces engine exhaust temperatures through the use of a high bypass-ratio engine.

- Another approach hides the nozzles from the expected location of the threat IR sensor.

- For example, the H-tails of the A-10 hide the nozzles from some angles.

- Unfortunately, the worst-case threat location is from the rear, and it is difficult to shield the nozzles from that direction

Special Considerations in Configuration Layout

- IR missiles can sometimes be tricked by throwing out a flare which burns to produce approximately the same IR frequencies as the aircraft.

- However, modern IR seekers are getting better at identifying which hot source is the actual aircraft.

Special Considerations in Configuration Layout

Visual Detectability

- The human eyeball is still a potent aircraft-detection sensor.
- On a clear day, an aircraft or its contrail may be spotted visually before detection by the on-board radar of a typical fighter.
- Visual detection depends upon the size of aircraft and its color and intensity contrast with the background.
- Background contrast is reduced primarily with camouflage paints, using colors and surface textures that cause the aircraft to reflect light at an intensity and color equal to that of the background.
- Current camouflage paint schemes are dirty blue-grey for sky backgrounds and dull, mottled grey-greens and grey-browns for ground backgrounds.

Special Considerations in Configuration Layout

Aural Signature

- Aural signature (noise) is important for civilian as well as military aircraft.

- Aircraft noise is largely caused by airflow shear layers, primarily due to the engine exhaust.

- Well-designed engine mounts, mufflers, and insulation materials can be used to reduce the noise.

- Jet engines mounted on the aft fuselage (DC-9, B727, etc.) should be located as far away from the fuselage as structurally permitted to reduce cabin noise.

DC-9

Special Considerations in Configuration Layout

Vulnerability Considerations

- Vulnerability concerns the ability of the aircraft to sustain battle damage, continue flying, and return to base.

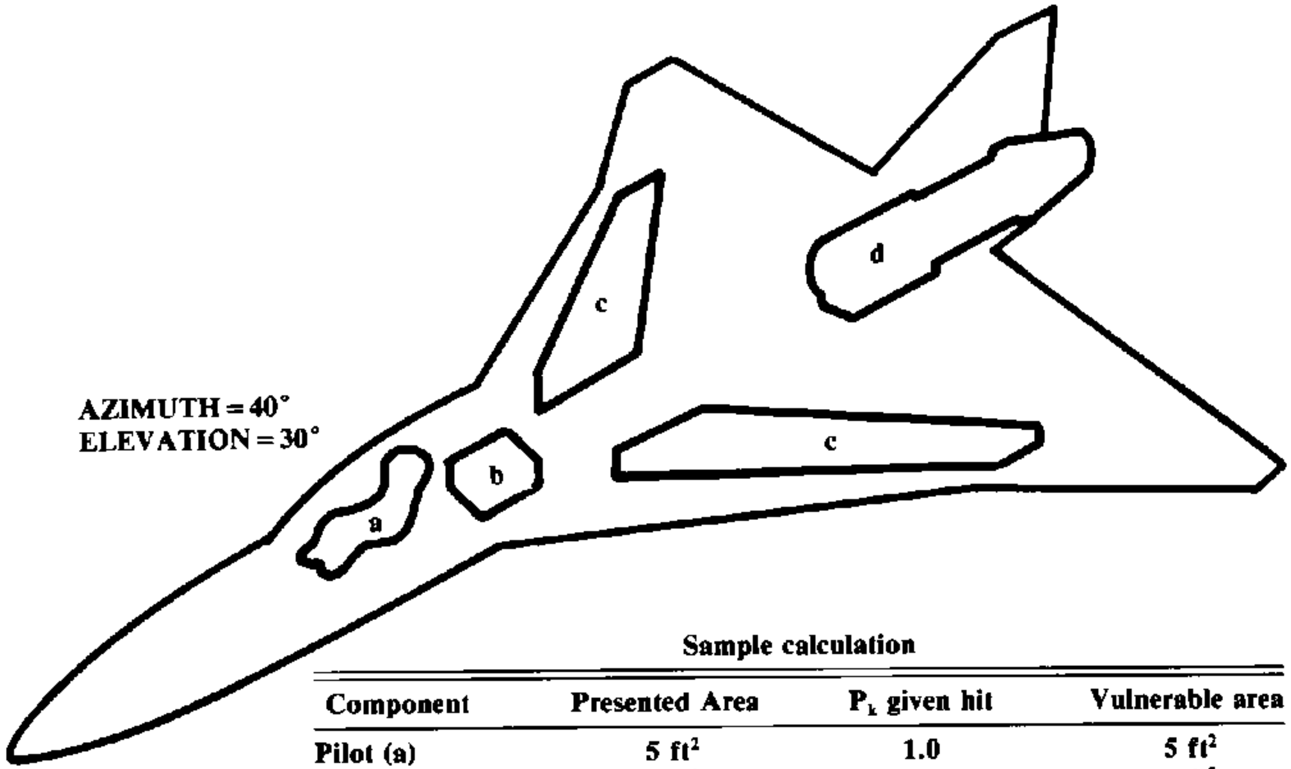
- "Vulnerable area" is a key concept.

- This refers to the product of the projected area of the aircraft components, times the probability that each component will, if struck, cause the aircraft to be lost.

- Typical components with a high aircraft kill probability (near 1.0) are the crew compartment, engine (if single-engined), fuel tanks (unless self-sealing), and weapons.

- Figure 8.14 shows a typical vulnerable area calculation.

Special Considerations in Configuration Layout



Sample calculation

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

Fig. 8.14 Vulnerable area calculation.

Special Considerations in Configuration Layout

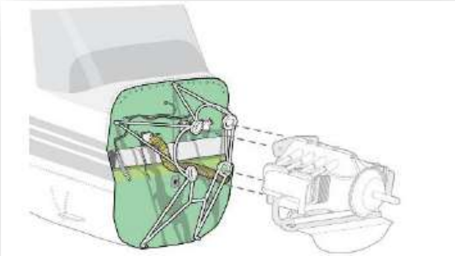
- During initial configuration layout, the designer should strive to avoid certain features known to cause vulnerability problems.

- Fire is the greatest danger to a battle-damaged aircraft.

- If at all possible, fuel should not be located over or around the engines and inlet ducts.

- Firewalls should be used to prevent the spread of flames beyond a burning engine bay.

- Engine bays, fuel bays, and weapon bays should have a fire suppression system.



Special Considerations in Configuration Layout

- Also, a twin-engine aircraft should have enough separation between engines to prevent damage to the good engine.
- If twin engines are together in the fuselage, a combined firewall and containment shield should separate them.
- This requires at least 1 foot of clearance between engines.
- Avoid placing guns, bombs, or fuel near the crew compartment.
- Fuel should not be placed in the fuselage of a passenger plane.

Special Considerations in Configuration Layout

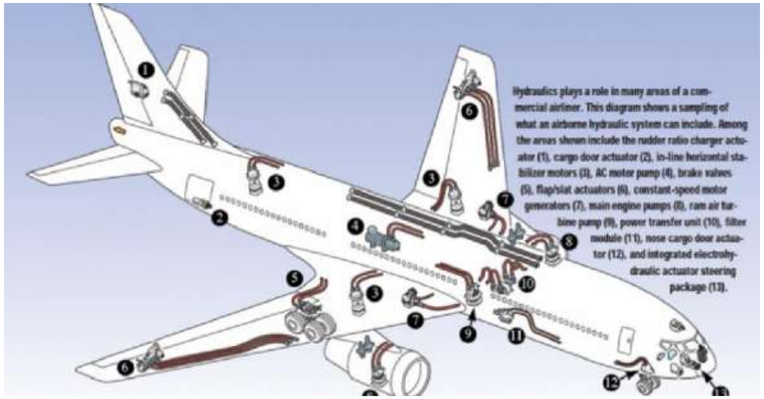
- Propeller blades can fly off either from battle damage or during a wheels up landing.
- Critical components, especially the crew and passenger. compartments, shouldn't be placed within a 5-deg arc of the propeller disk.

Special Considerations in Configuration Layout

- 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.



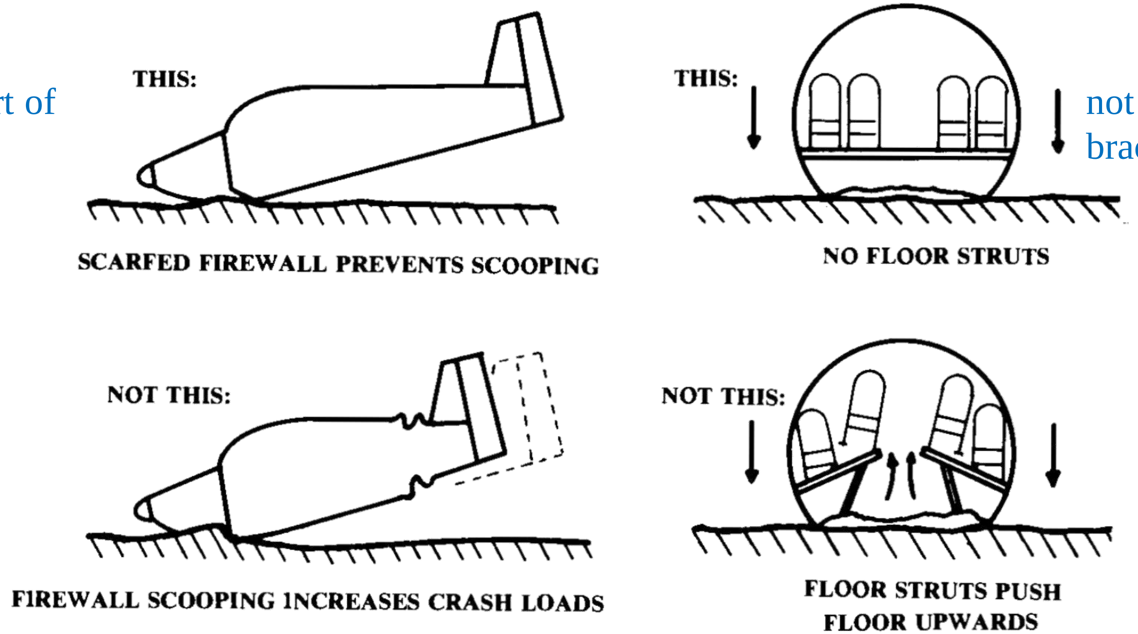
Special Considerations in Configuration Layout

Crashworthiness Considerations

- Airplanes crash. Careful design can reduce the probability of injury in a moderate crash.

- Figure 8.15 shows several other design suggestions which were learned the hard way.

Sloping the lower part of the firewall



not be supported by braces

Fig. 8.15 Crashworthiness design.

Special Considerations in Configuration Layout

- One should also consider secondary damage.
- For example, landing gear and engine nacelles will frequently be ripped away during a crash.
- If possible, they should be located so that they do not rip open fuel tanks in the process.

Special Considerations in Configuration Layout

Producibility Considerations

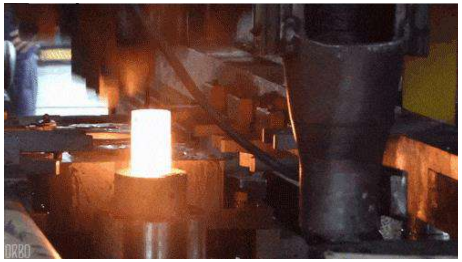
- It is often said that aircraft are bought "by the pound."
- While it is true that aircraft cost is most directly related to weight,
- there is also a strong cost impact due to the materials selected, the fabrication processes and tooling required (forging, stamping, molding, etc.), and the assembly man-hours.
- The greatest impact the configuration designer has upon producibility is the extent to which **flat-wrap** structure is incorporated.

meaning the have a curve in one direction only.

- This has a major impact upon the tooling costs and fabrication man-hours

Special Considerations in Configuration Layout

- Part commonality can also reduce production costs.
- If possible, the left and right main landing gear should be identical (left-right common). It may be desirable to use uncambered horizontal tails to allow left-right commonality.
- Forgings are the most expensive type of structure in common usage, and are also usually the longest-lead-time items for production tooling.
- Forgings may be required whenever a high load passes through a small area.
- Forgings are used for landing-gear· struts, wing-sweep pivots, and all-moving tail pivots (trunnions).
- The designer should avoid, if possible, such highly loaded structure.



Special Considerations in Configuration Layout

- Installation of internal components and routing of hydraulic lines, electrical wiring, and cooling ducts comprises another major production cost due to the large amount of manual labor required.

- Routing can be simplified through provision of a clearly defined "routing tunnel."

- This can be internal or, as shown in Fig. 8.16, an external and nonstructural fairing that typically runs along the spine of the aircraft.

- However, if all routing is concentrated in one area the aircraft vulnerability will be drastically worsened.

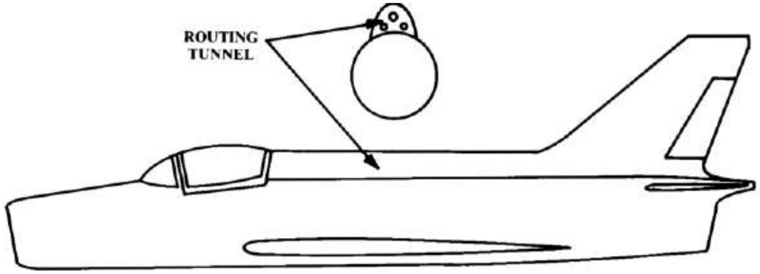


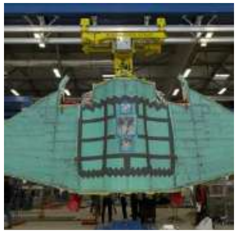
Fig. 8.16 External routing tunnel.

Special Considerations in Configuration Layout

- Another factor for producibility concerns manufacturing breaks.
- Aircraft are built in subassemblies.
- Typically, a large aircraft will be built up from a cockpit, an aft-fuselage, and a number of mid-fuselage subassemblies.
- A small aircraft may be built from only two or three subassemblies.
- It is important that the designer consider where the subassembly breaks will occur, and avoid placing components across the convenient break locations.

Special Considerations in Configuration Layout

Fig. 3 F-35 global supply chain.



Special Considerations in Configuration Layout

- Figure 8.17 shows a typical fighter with a fuselage production break located just aft of the cockpit.

- This is very common because the cockpit pressure vessel should not be broken for fabrication.

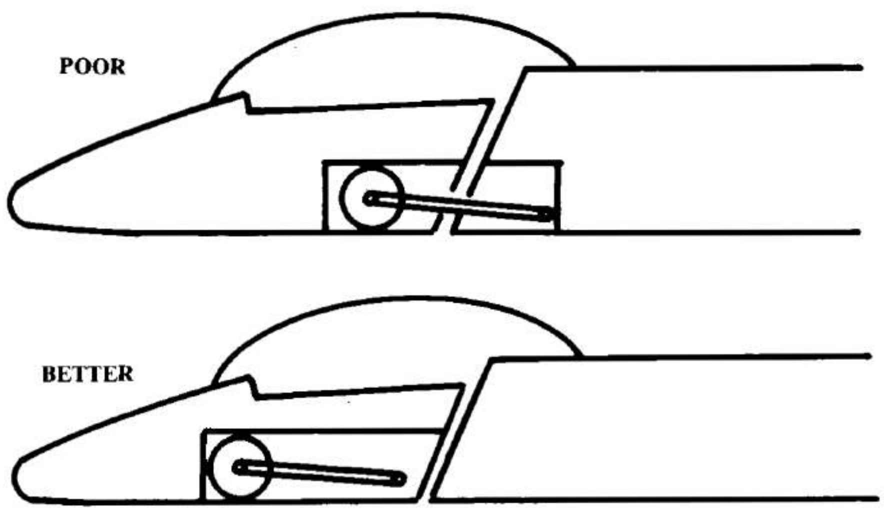


Fig. 8.17 Production breaks.

Special Considerations in Configuration Layout

Maintainability Considerations

- 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 Man-hours Per Flight hour" (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.

- Getting at the internal components frequently takes longer than fixing them!

Special Considerations in Configuration Layout

- 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.

- As a general rule, the best access should be provided to the components that break the most often or require the most routine maintenance.

Special Considerations in Configuration Layout

- The worst feature an aircraft can have for maintainability is a requirement for major structural disassembly to access or remove a component.

- For example, the V/STOL AV-8B Harrier requires that the entire wing be removed before removing the engine.

- Similarly, the designer should avoid placing internal components such that one must be removed to get to another.

- In the F -4 Phantom, an ejection seat must be removed to get to the radio equipment.

Aircraft Design
AE 405

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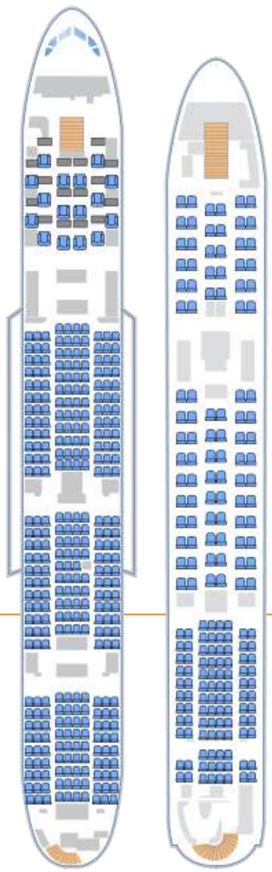
Contents



Chapter 9 – 10

- Crew Station, Passengers, and Payload.
- Propulsion and Fuel System Integration.

Crew Station, Passengers, and Payload

- At the conceptual design level it is not necessary to go into the details of crew-station design.

- However, the basic geometry of the crew station and payload/passenger compartment must be considered,

- so that the subsequent detailed cockpit design and payload integration efforts will not require revision of the overall aircraft.

- This chapter presents dimensions and "rule-of-thumb" design guidance for conceptual layout of
  - aircraft crew stations,
  - passenger compartments,
  - payload compartments,
  - and weapons installations.

Crew Station, Passengers, and Payload

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.

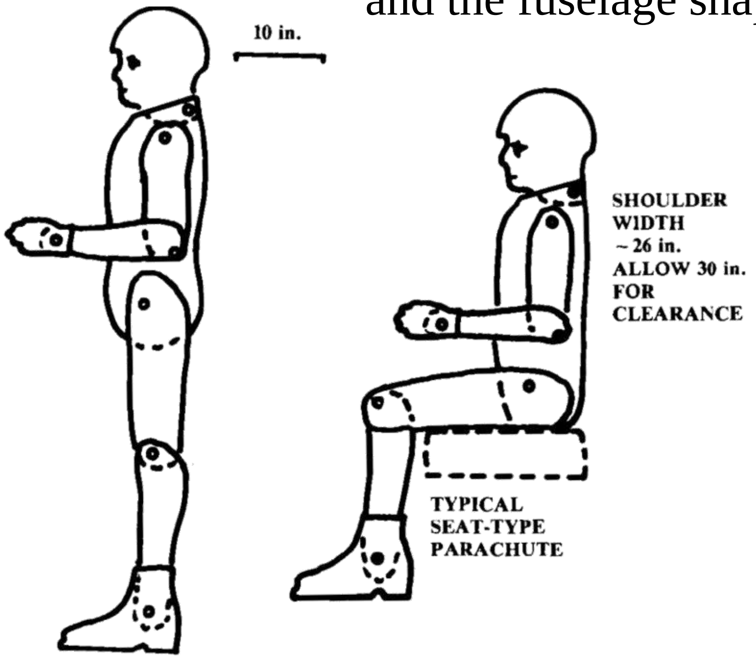


Fig. 9.1 Average 95th percentile pilot.

- When laying out an aircraft's cockpit, it is first necessary to decide what range of pilot sizes to accommodate (a pilot height range of 65.2-73.1 in).

- Figure 9.1 shows a typical pilot figure useful for conceptual design layout.

Crew Station, Passengers, and Payload

- Dimensions for a typical cockpit sized to fit the 95th-percentile pilot are shown in Fig. 9.2.

- The two key reference points for cockpit layout are shown.

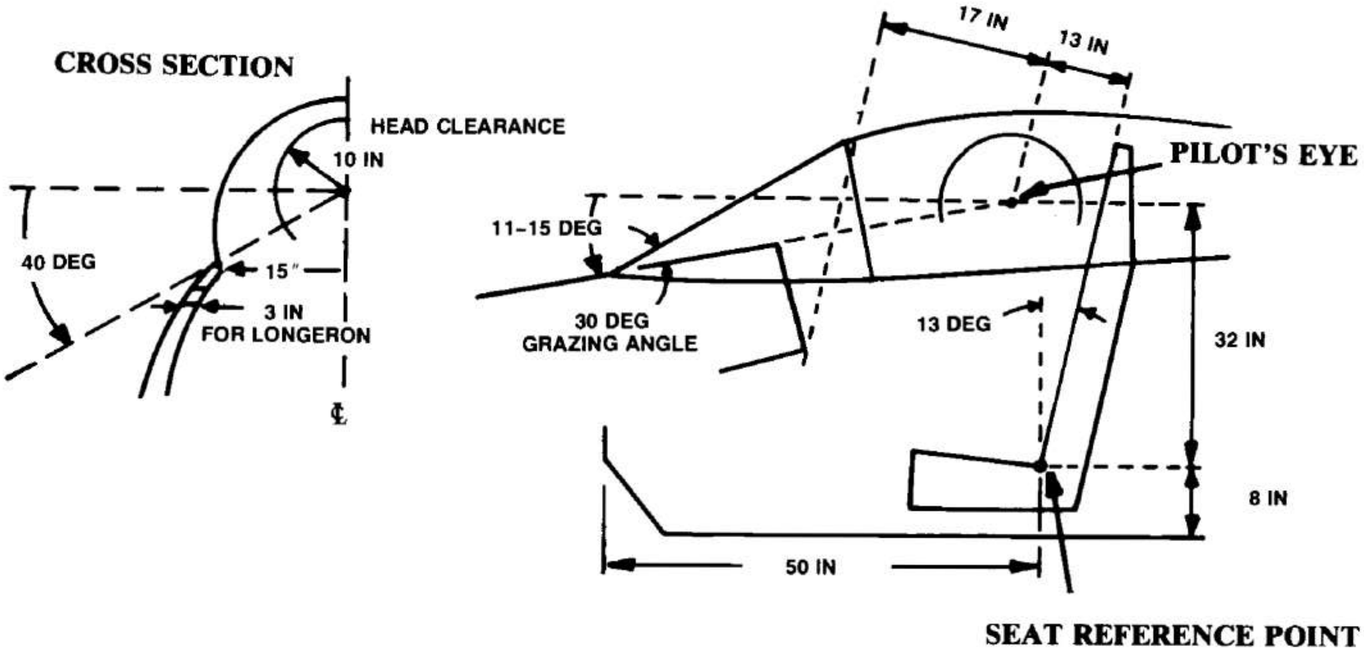


Fig. 9.2 Typical fighter cockpit.

Crew Station, Passengers, and Payload

- The seat reference point, where the seat pan meets the back, is the reference for the floor height and the legroom requirement.

- The pilot's eye point is used for defining the overnose angle, transparency grazing angle, and pilot's head clearance (10-in. radius).

- 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 in outside vision for the pilot, but can improve his ability to withstand high-g turns and also can reduce drag because of a reduction in the cockpit height.

Crew Station, Passengers, and Payload

- For initial layout, Eq. (9.1) is a close approximation for determining the required overnose angle, based upon the aircraft angle of attack during approach and the approach speed.

$$\alpha_{\text{overnose}} \cong \alpha_{\text{approach}} + 0.07\, V_{\text{approach}} \qquad (9.1)$$

where V_{approach} is in knots.

- Figure 9.2 shows an over-the-side vision requirement of 40 deg, measured from the pilot's eye location on centerline.

- This is typical for fighters and attack aircraft.

- For bombers and transports, it is desirable that the pilot be able to look down at a 35-deg angle without head movement, and at a 70-deg angle when the pilot's head is pressed against the cockpit glass.

Crew Station, Passengers, and Payload

- The vision angle looking upward is also important.
- Transport and bomber aircraft should have unobstructed vision forwards and upwards to at least 20 deg above the horizon.
- Fighters should have completely unobstructed vision above and all the way to the tail of the aircraft.
- The transparency grazing angle shown in Fig. 9.2 is the smallest angle between the pilot's line of vision and the cockpit windscreen.
- If this angle becomes too small, the transparency of the glass or plexiglass will become substantially reduced,
- and under adverse lighting conditions the pilot may only see a reflection of the top of the instrument panel instead of whatever is in front of the aircraft!

the minimum grazing angle of 30 deg

Crew Station, Passengers, and Payload

- The cockpit of a transport aircraft must contain anywhere from two to four crew members as well as provisions for radios, instruments, and stowage of map cases and overnight bags.
- Suggestions: an overall length of about 150 in. for a four-crewmember cockpit, 130 in. for three crewmembers, and 100 in. for a two-crewmember cockpit.
- The cockpit dimensions shown in Fig. 9.2 will provide enough room for most military ejection seats.
- An ejection seat is required for safe escape when flying at a speed which gives a dynamic pressure above about 230 psf (equal to 260 knots at sea level).
- At speeds approaching Mach 1 at sea level (dynamic pressure above 1200), even an ejection seat is unsafe and an encapsulated seat or separable crew capsule must be used.

FB-111

separable crew capsule

Crew Station, Passengers, and Payload

Passenger Compartment

- The actual cabin arrangement for a commercial aircraft is determined more by marketing than by regulations.
- Figure 9.3 defines the dimensions of interest.
- Table 9.1 provides typical dimensions and data for passenger compartments with first-class, economy, or high-density seating.

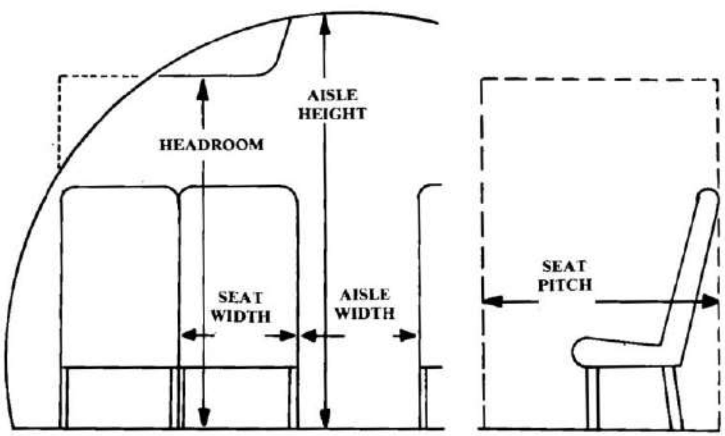


Fig. 9.3 Commercial passenger allowances.

Table 9.1 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

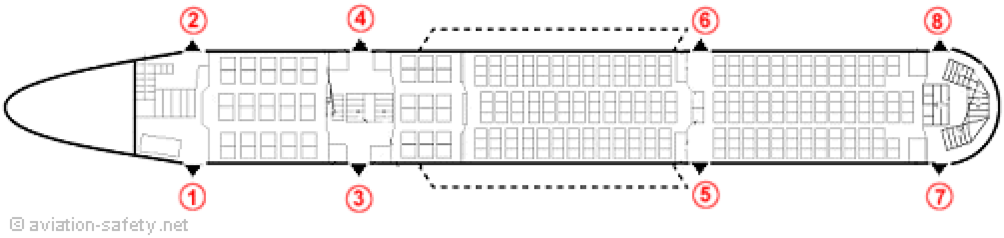
Crew Station, Passengers, and Payload

- There should be no more than three seats accessed from one aisle, so an aircraft with more than six seats abreast will require two aisles.

- Also, doors and entry aisles are required for approximately every 10-20 rows of seats.

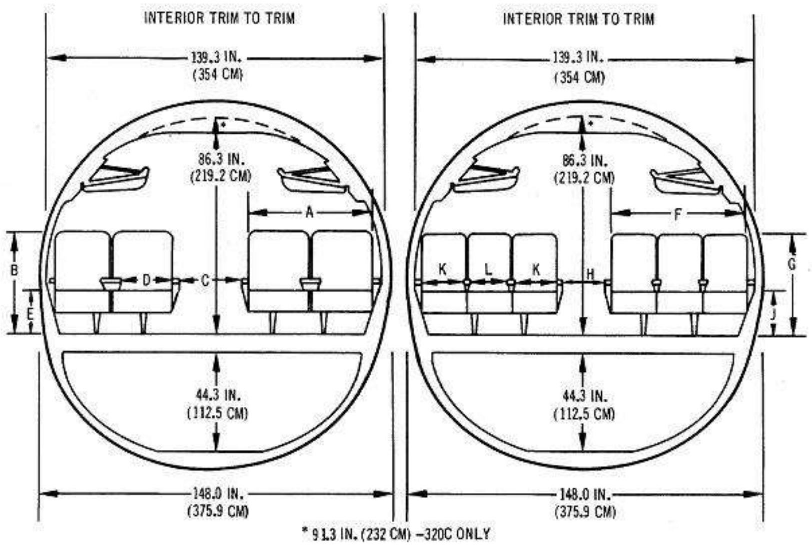
- These usually include closet space, and occupy 40-60 in. of cabin length each.

- Passengers can be assumed to weigh an average of 180 lb (dressed and with carry-on bags), and to bring about 40-60 lb of checked luggage.



Crew Station, Passengers, and Payload

- The cabin cross section and cargo bay dimensions are used to determine the internal diameter of the fuselage.
- The fuselage external diameter is then determined by estimating the required structural thickness.
- This ranges from 1 in for a small business or utility transport to about 4 in for a Jumbo Jet.



Crew Station, Passengers, and Payload

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.

- Two of the more widely used cargo containers are shown in Fig. 9.4.

Crew Station, Passengers, and Payload

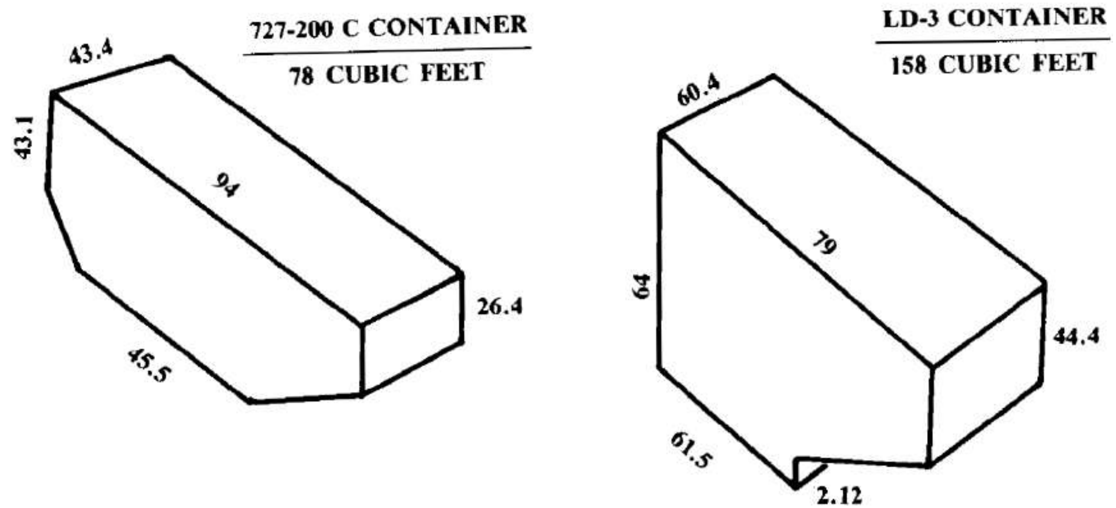


Fig. 9.4 Cargo containers.

Crew Station, Passengers, and Payload

- To accommodate these containers, the belly cargo compartments require doors measuring approximately 70 in. on a side.
- The cargo volume per passenger of a civilian transport ranges from about 8.6-15.6 ft^3 per passenger.
- The smaller number represents a small short-haul jet (DC-9).
- The larger number represents a transcontinental jet (B-747).
- The DC-10, L-1011, Airbus, and B-767 all have about 11 ft^3 per passenger.
- Smaller transports don't use cargo containers, but instead rely upon hand-loading of the cargo compartment.

Crew Station, Passengers, and Payload

- Military transports use flat pallets to pre-load cargo.
- Cargo is placed upon these pallets, tied down, and covered with a tarp.
- The most common pallet measures 88 by 108 in.
- Military transports must have their cargo compartment floor approximately 4-5 ft off the ground to allow direct loading and unloading of Cargo from a truck bed at air bases.

C-5

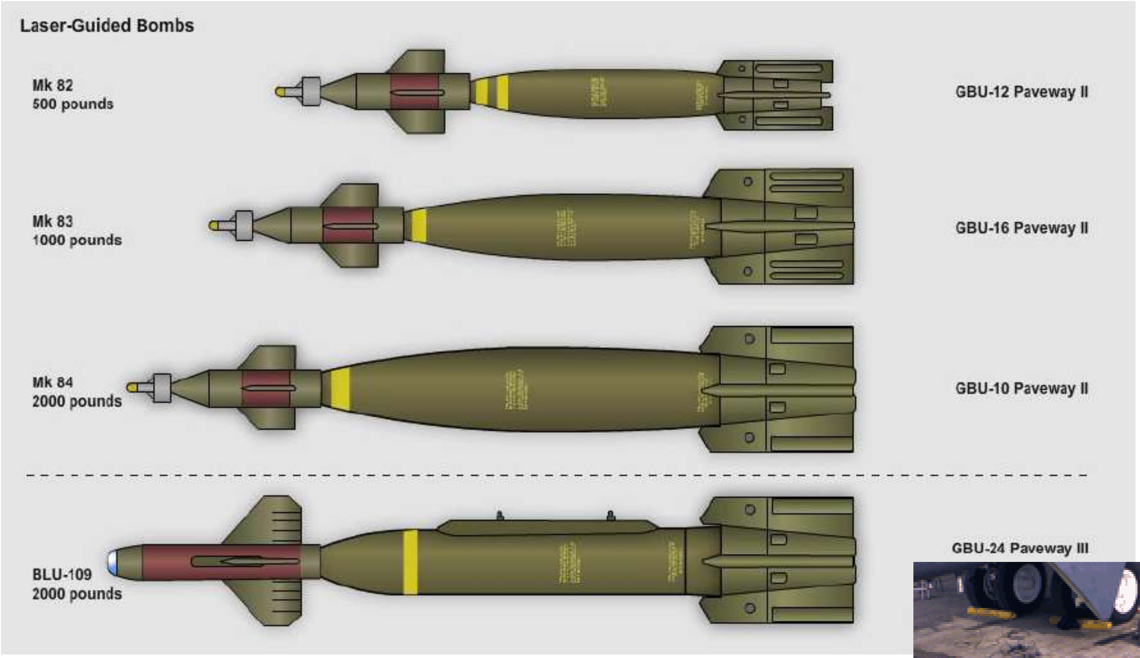
C-130

Crew Station, Passengers, and Payload

Weapons Carriage

- Carriage of weapons is the purpose of most military aircraft.
- Traditional weapons include guns, bombs, and missiles.
- 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 from bombs primarily in that missiles are powered.

Crew Station, Passengers, and Payload



- Bombs can also be ejected, or can simply be released and allowed to fall free of the aircraft.

Crew Station, Passengers, and Payload

- 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 on Fig. 9.5.
- For launch, the missile motor powers the missile down the rail and free of the aircraft.

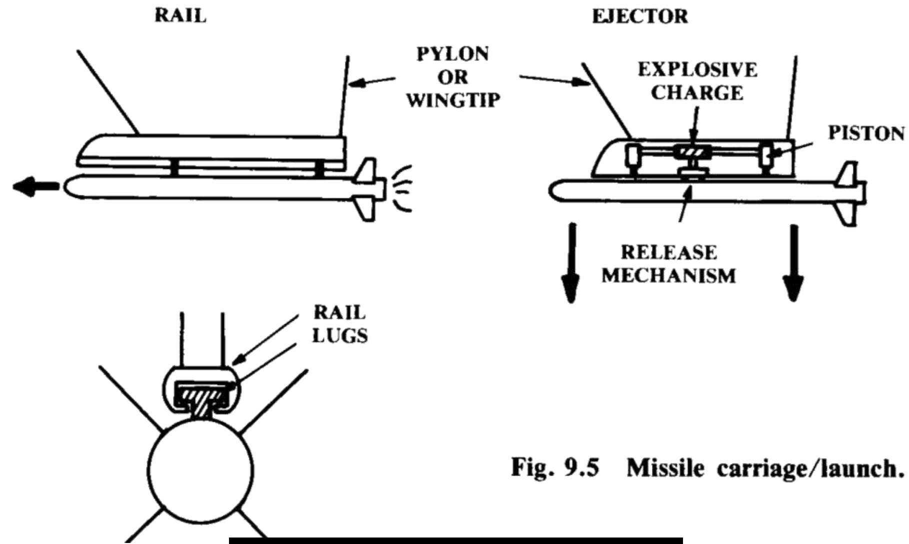


Fig. 9.5 Missile carriage/launch.

Crew Station, Passengers, and Payload

- Ejection-launch is used mainly for larger missiles.
- The missile is attached to the aircraft through hooks which are capable of quick-release, powered by an explosive charge.
- This explosive charge also powers two pistons that shove the missile away from the aircraft at an extremely high acceleration.

- The missile motor is lit after it clears the aircraft by some specified distance.

Crew Station, Passengers, and Payload

- There are four options for weapons carriage.

- External carriage is the lightest and simplest, and offers the most flexibility for carrying alternate weapon stores.

- Externally-carried weapons have extremely high drag.

- At near-sonic speeds, a load of external bombs can have more drag than the entire rest of the aircraft.

- Supersonic flight is virtually impossible with pylon-mounted external weapons, due to drag and buffeting. (Wing tip-mounted missiles are small, and have fairly low drag.)

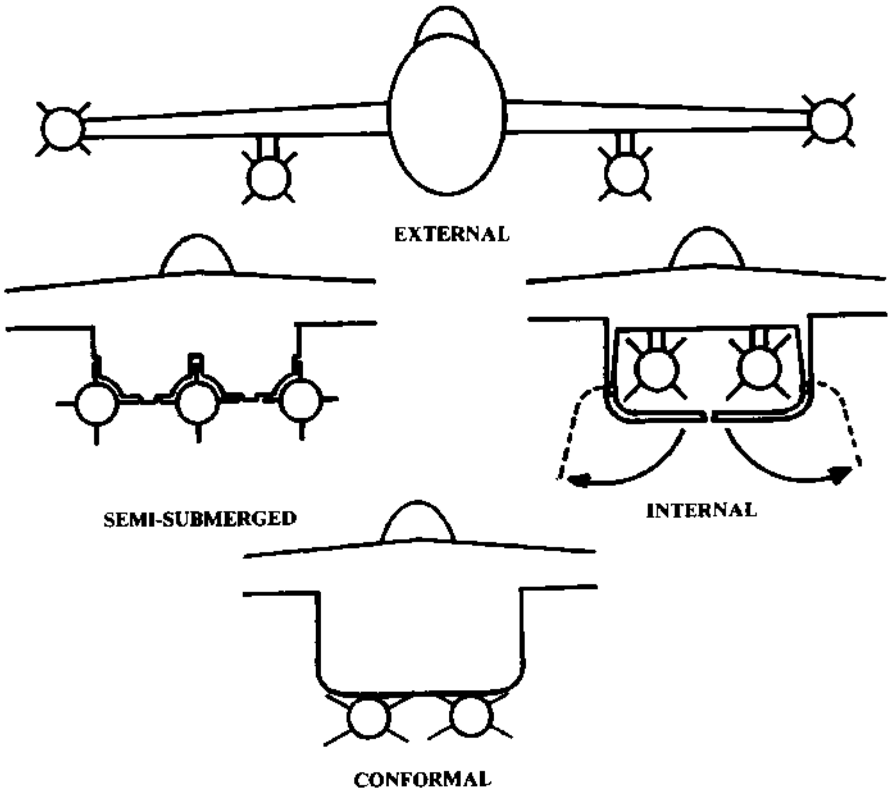


Fig. 9.6 Weapon carriage options.

Crew Station, Passengers, and Payload

- To avoid these problems, semisubmerged or conformally-carried weapons may be used.
- Conformal weapons mount flush to the bottom of the wing or fuselage.
- Semisubmerged weapons are half-submerged in an indentation on the aircraft.
- This is seen on the F-4 for air-to-air missiles.

Crew Station, Passengers, and Payload

- The lowest-drag option for weapons carriage is internal.
- An internal weapons bay has been a standard feature of bombers for over fifty years, but has been seen on only a few fighters and fighter-bombers, such as FB-111.
- This is partly due to the weight penalty imposed by an internal weapons bay and its required doors,
- but is also due to the prevalent desire to maximize dogfighting performance at the expense of alternate mission performance.
- However, only an internal weapons bay can completely eliminate the weapons' contribution to radar cross section
- so the internal weapons bay may become common for fighters as well as bombers.

Crew Station, Passengers, and Payload

Crew Station, Passengers, and Payload

- Clearance around the missiles and bombs is also important for safety.
- To insure that the weapons never strike the ground, the designer should provide at least a 3-in. clearance to the ground in all aircraft attitudes.
- This includes the worst-case bad landing in which one tire and shock-strut are completely flat.
- If weapons are mounted near each other, there should be a clearance on the order of 3 in. between them.
- There should also be a foot or more clearance between weapons and a propeller disk.

- For an ejector-launched or free-fall released weapon,
    - there should be a fall line clearance of 10 deg off the vertical down from any part of the missile to any part of the aircraft or other weapons,
    - there should be at least a 10-deg cone of clearance between any part of the aircraft and the launch direction of the missile

as shown in Fig. 9. 7.

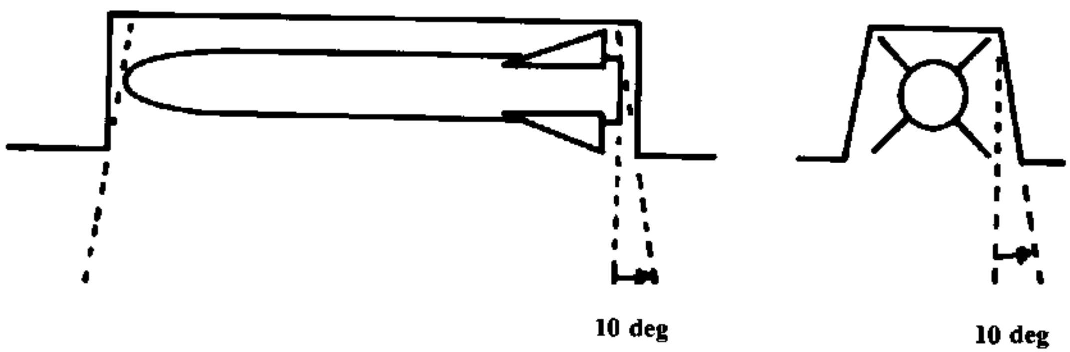
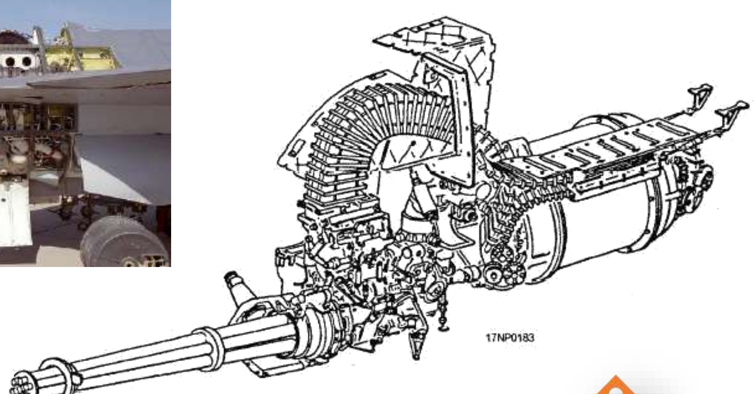


Fig. 9.7 Weapon release clearance.

Crew Station, Passengers, and Payload

Gun Installation

- The gun has been the primary weapon of the air-to-air fighter since the first World War I.
- The standard U.S. air-to-air gun today is the M61Al "Vulcan" six-barrel gatling gun, shown in Fig. 9.9.
- This is used in the F-15, F-16, F-18, and others.
- The door to loading chute must be accessible from the ground.



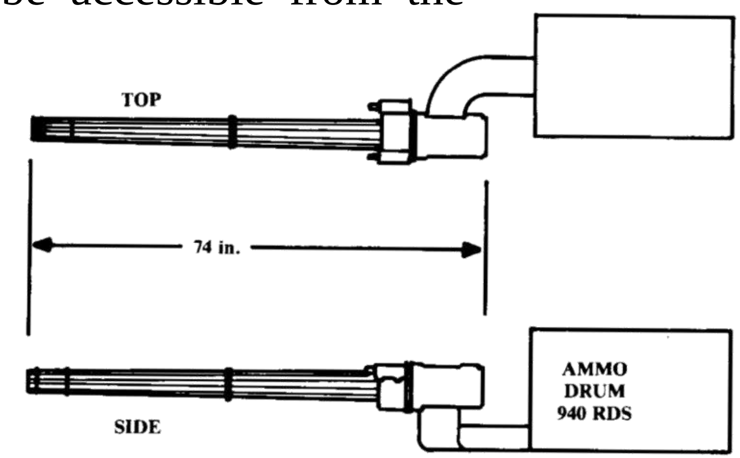


Fig. 9.9 M61 "VULCAN" gun.

Crew Station, Passengers, and Payload

- An air-to-air gun such as the M61Al can produce a recoil force on the order of two tons.
- A large anti-tank gun such as the GAU-8 used in the A-10 can produce recoil force five times greater.
- To avoid a sudden yawing motion from firing, guns should be located as near as possible to the centerline of the aircraft.
- When a gun is fired, it produces a bright flash and a large cloud of smoke.
- The gun muzzle should be located so that these do not obscure the pilot's vision.

Crew Station, Passengers, and Payload

- Also, being very noisy, a gun should be located away from the cockpit.

- The cloud of smoke produced by a gun can easily stall a jet engine if sucked into the inlet. This should also be considered when locating a gun.

Propulsion and Fuel System Integration

- This section treats the integration and layout of the propulsion system into the overall vehicle design.

- To develop the propulsion system layout it is necessary to know the actual dimensions and installation requirements of the engine as well as its supporting equipment such as

  - inlet ducts,
  - nozzles,
  - or propellers.

- Also, the fuel system including the fuel tanks must be defined.

Propulsion and Fuel System Integration

Propulsion Selection

- Figure 10.1 illustrates the major options for aircraft propulsion.

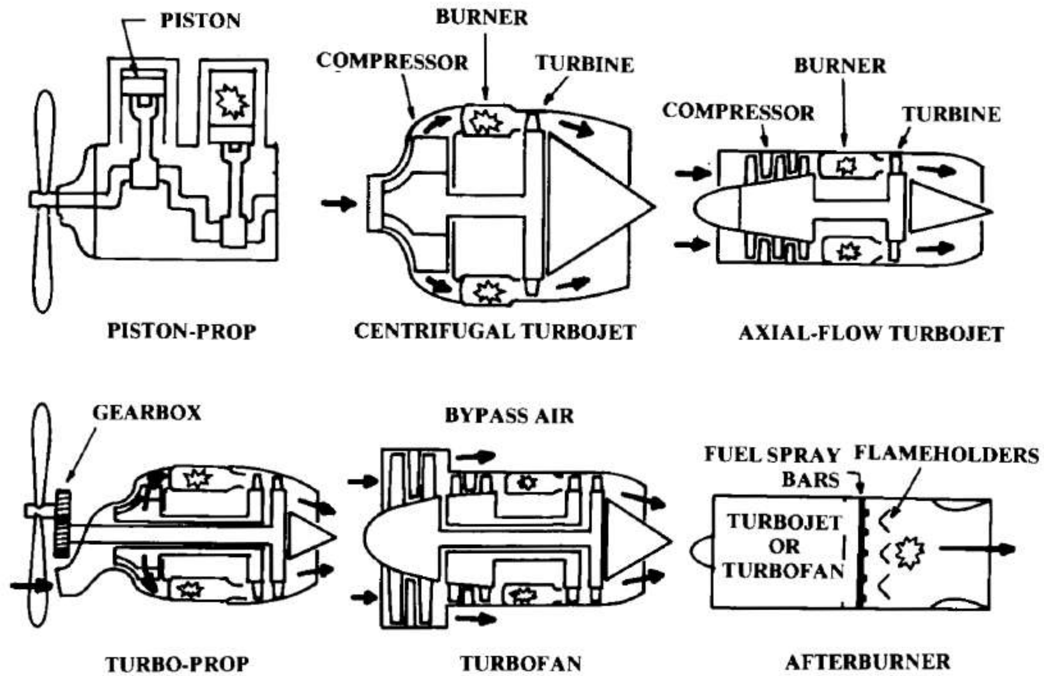
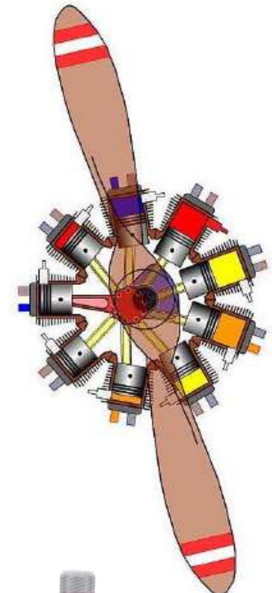


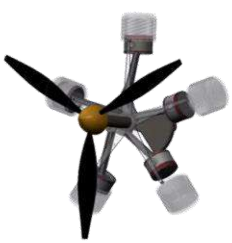
Fig. 10.1 Propulsion system options.

- 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.

Propulsion and Fuel System Integration

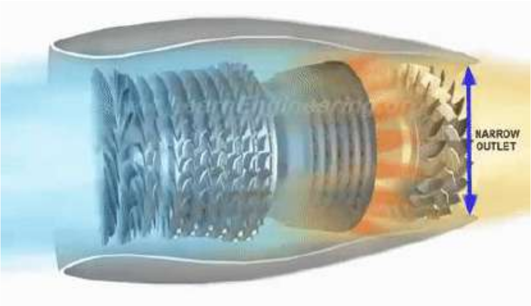


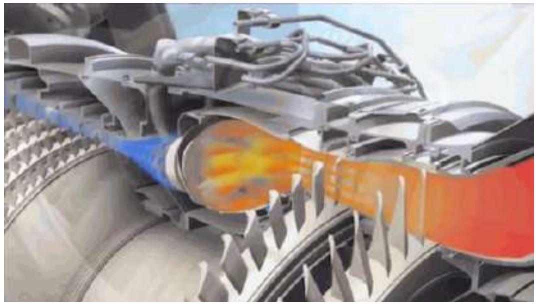
- The piston-prop was the first form of aircraft propulsion.
- By the dawn of the jet era, a 5500-hp piston-prop engine was in development.
- Today piston props are mainly limited to light airplanes and some agricultural aircraft.
- Piston-prop engines have two advantages. They are cheap, and they have 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.

Propulsion and Fuel System Integration

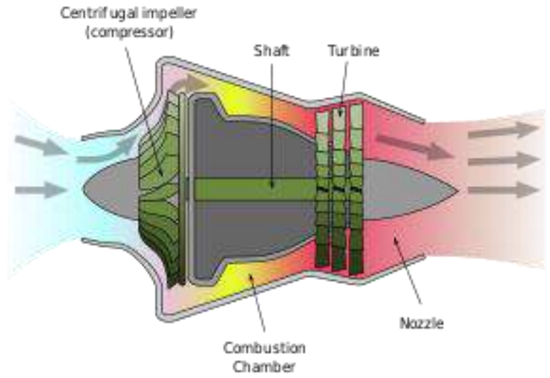


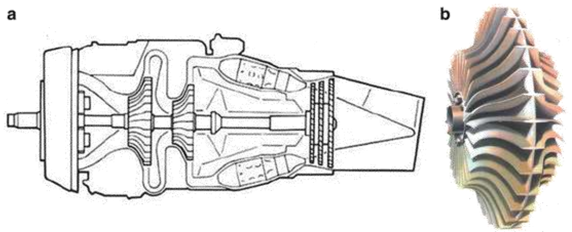


- The turbine engine consists of a "compressor," a "burner," and a "turbine."
- The compressor takes the air delivered by the inlet system and compresses it to many times atmospheric pressure.
- This compressed air passes to the burner, where fuel is injected and mixed with the air and the resulting 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.

Propulsion and Fuel System Integration

- There are two types of compressors.
- The centrifugal compressor relies upon centrifugal force to "fling/propel" the air into an increasingly narrow channel, which raises the pressure.
- In contrast, an axial compressor relies upon blade aerodynamics to force the air into an increasingly narrow channel.
- An axial compressor typically has about six to ten "stages," each of which consists of a "rotor" (i.e., rotating) disk of blades and a "stator" (i.e., stationary) disk of blades.
- The rotors tend to swirl the air, so the stators are used to remove the swirl.





Propulsion and Fuel System Integration

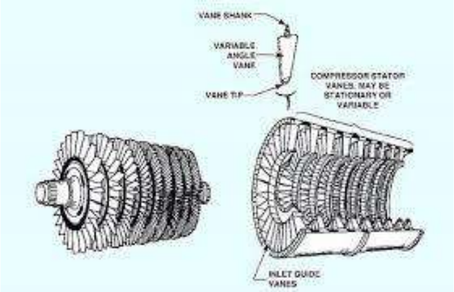
- The axial compressor, relying upon blade aerodynamics, is intolerant to distortions in the incoming air such as swirl or pressure variations.

- These distortions can stall the blades, causing a loss of compression and a possible engine flame-out.

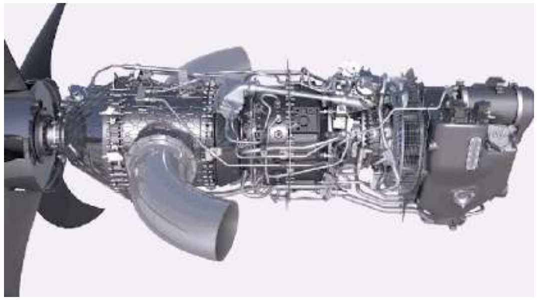
- The centrifugal compressor is much more forgiving of inlet distortion,

- 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.



Propulsion and Fuel System Integration



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

Propulsion and Fuel System Integration

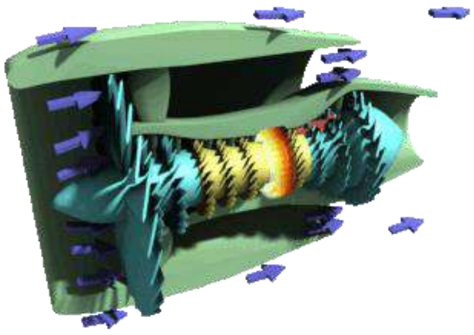
- For the turbofan engine, the air is accelerated with a ducted fan of one or several stages.

- This accelerated air is then split, with part remaining in the engine for further compression and burning,

- and the remainder being "bypassed" around the engine to exit unburned.

- The "bypass ratio" is the mass-flow ratio of the bypassed air to the air that goes into the core 'of the engine.

- Bypass ratio ranges from as high as 6 to as low as 0.25 (the so-called "leaky turbojet").



Propulsion and Fuel System Integration

- The ideal turbine engine would inject enough fuel to completely combust all of the compressed air, producing maximum thrust for a given engine size.

- Unfortunately, this air/fuel mixture ratio of about 15 to 1 produces temperatures far greater than the capabilities of known materials,

- and would therefore burn up the turbine blades.

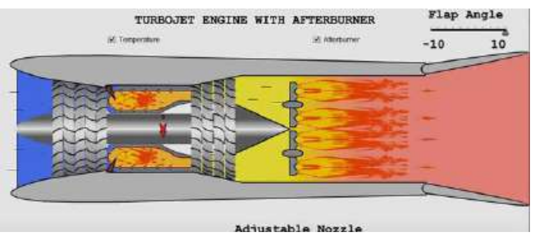
- Currently engines are limited to a turbine temperature of about 2000-25000F, which requires an air/fuel mixture ratio of about 60 to 1.

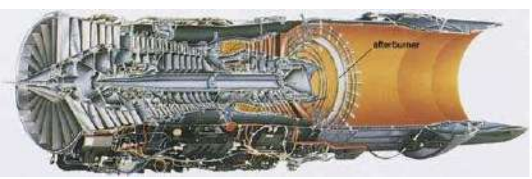
- Thus, only about a quarter of the captured and compressed air is actually used for combustion. The exhaust is 75% unused hot air.

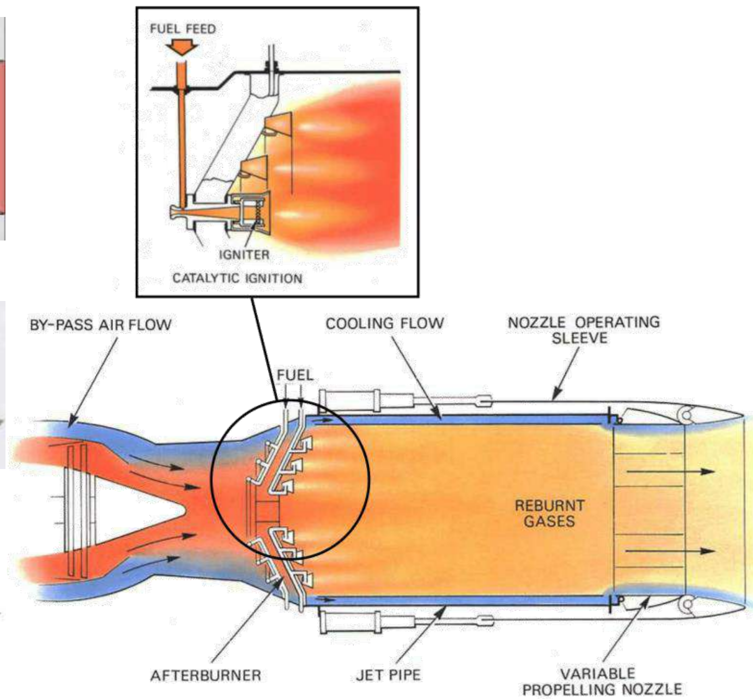
Propulsion and Fuel System Integration

- If fuel is injected into this largely-uncombusted hot air, it will mix and burn.
- This will raise the thrust as much as a factor of two, and is known as "afterburning."
- Unfortunately, afterburning is inefficient in terms of fuel usage.
- The fuel flow required to produce a pound of thrust in afterburner is approximately double that used to produce a pound of thrust during normal engine operations.
- Afterburner will approximately double the length of a turbojet or turbofan engine.

Propulsion and Fuel System Integration



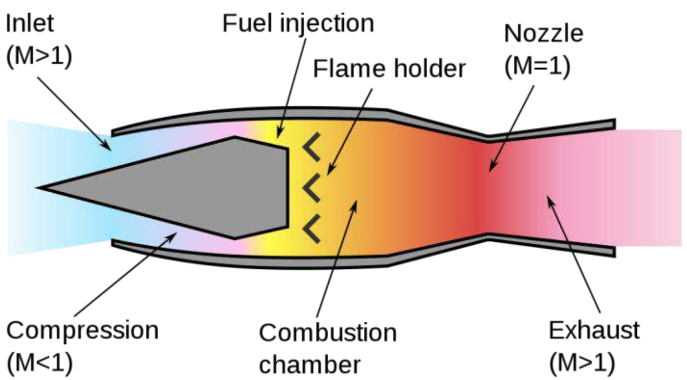


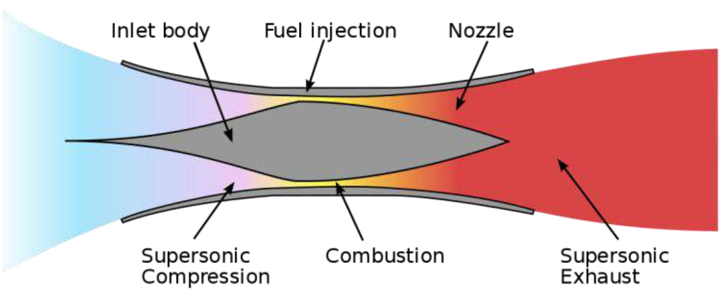


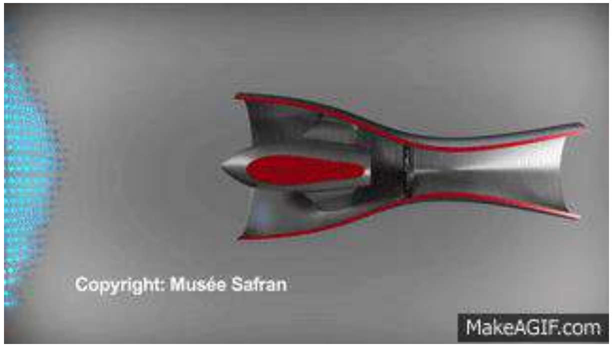
Propulsion and Fuel System Integration

- If the aircraft is traveling fast enough, the inlet duct alone will compress the air enough to burn if fuel is added.
- This is the principle of a "ramjet."
- Ramjets must be traveling at above Mach 3 to become competitive with a turbojet in terms of efficiency.
- A "scramjet" is a ramjet that can operate with supersonic internal flow and combustion.
- Scramjets are probably suitable only for operation above Mach 5 or 6.
- Ramjets and scramjets require some other form of propulsion for takeoff and acceleration to the high Mach numbers they require for operation.

Propulsion and Fuel System Integration







Propulsion and Fuel System Integration

- The selection of the type of propulsion system-piston-prop, turboprop, turbofan, turbojet, ramjet-will usually be obvious from the design requirements.

- Aircraft maximum speed limits the choices, as shown in Fig. 10.2.

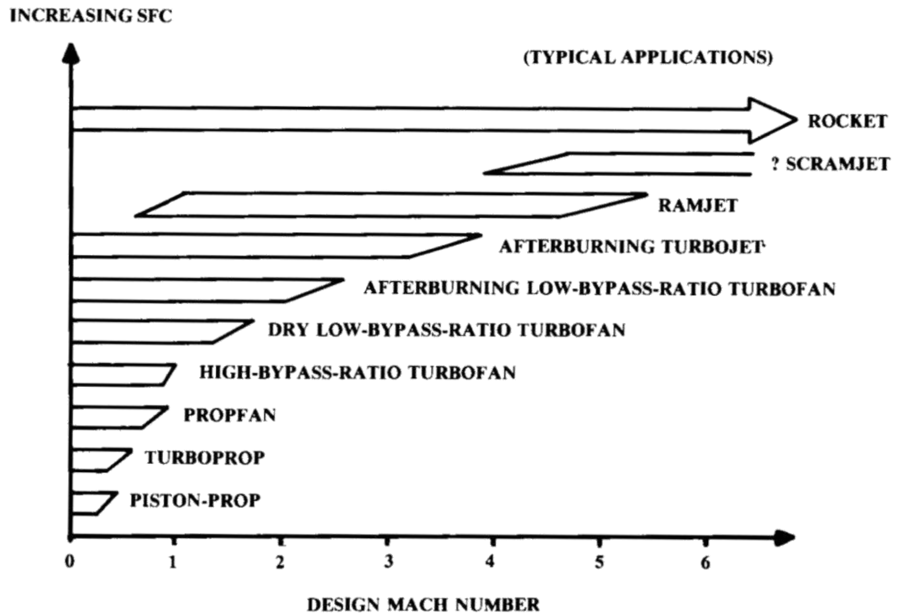


Fig. 10.2 Propulsion system speed limits.

Propulsion and Fuel System Integration

Jet-engine Integration

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

Engine Dimensions

- Figure 10.3 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 (10.1-10.3) Show how length, diameter, and weight vary with the scale factor for the typical jet engine.

$$L = L_{\text{actual}}(\text{SF})^{0.4} \quad (10.1)$$

$$D = D_{\text{actual}}(\text{SF})^{0.5} \quad (10.2)$$

$$W = W_{\text{actual}}(\text{SF})^{1.1} \quad (10.3)$$

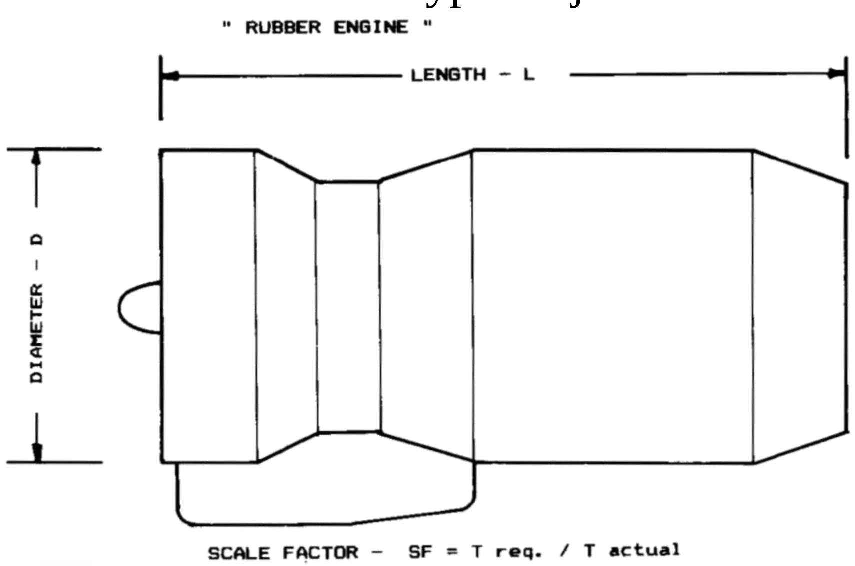
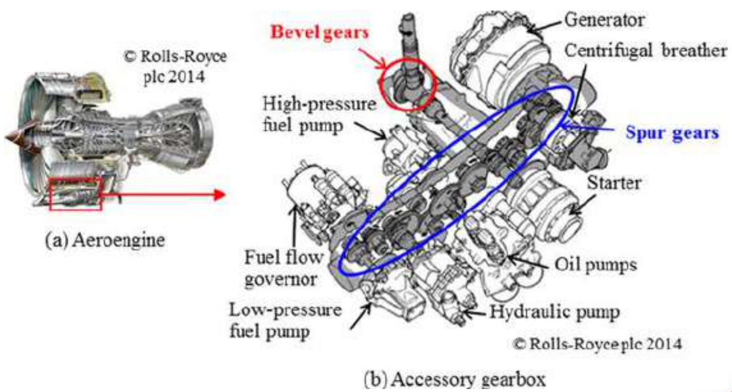


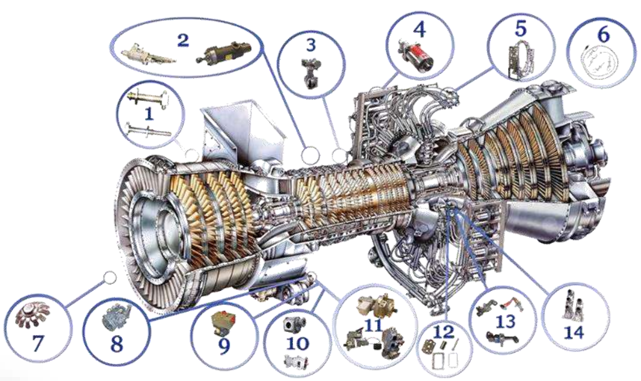
Fig. 10.3 Engine scaling.

Propulsion and Fuel System Integration

- Note the engine-accessories package beneath the engine.
- The accessions include fuel pumps, oil pumps, power-takeoff gearboxes, an engine control boxes.
- The location and size of the accessory package vanes widely for different types of engines.
- In the absence of a drawing: the accessory package can be assumed to extend below the engine to a radius of about 20-40% greater than the engine radius.



(a) Aeroengine (b) Accessory gearbox



Propulsion and Fuel System Integration

- 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.

- Equations (10.4-10.15) define two first-order statistical jet-engine models;

Nonafterburning engines:

$$W = 0.084T^{1.1}e^{(-0.045\text{ BPR})} \tag{10.4}$$

$$L = 2.22T^{0.4}M^{0.2} \tag{10.5}$$

$$D = 0.393T^{0.5}e^{(0.04\text{ BPR})} \tag{10.6}$$

$$\text{SFC}_{\max T} = 0.67e^{(-0.12\text{ BPR})} \tag{10.7}$$

$$T_{\text{cruise}} = 0.60T^{0.9}e^{(0.02\text{ BPR})} \tag{10.8}$$

$$\text{SFC}_{\text{cruise}} = 0.88e^{(-0.05\text{ BPR})} \tag{10.9}$$

Propulsion and Fuel System Integration

Afterburning engines:

$$W = 0.063T^{1.1}M^{0.25}e^{(-0.81\ \mathrm{BPR})} \tag{10.10}$$

$$L = 3.06T^{0.4}M^{0.2} \tag{10.11}$$

$$D = 0.288T^{0.5}e^{(0.04\ \mathrm{BPR})} \tag{10.12}$$

$$\mathrm{SFC}_{\max T} = 2.1e^{(-0.12\ \mathrm{BPR})} \tag{10.13}$$

$$T_{\mathrm{cruise}} = 1.6T^{0.74}e^{(0.023\ \mathrm{BPR})} \tag{10.14}$$

$$\mathrm{SFC}_{\mathrm{cruise}} = 1.04e^{(-0.186\ \mathrm{BPR})} \tag{10.15}$$

where

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

Propulsion and Fuel System Integration

- One model is for subsonic nonafterburning engines such as found on commercial transports, and covers a bypass-ratio range from 0 to about 6.

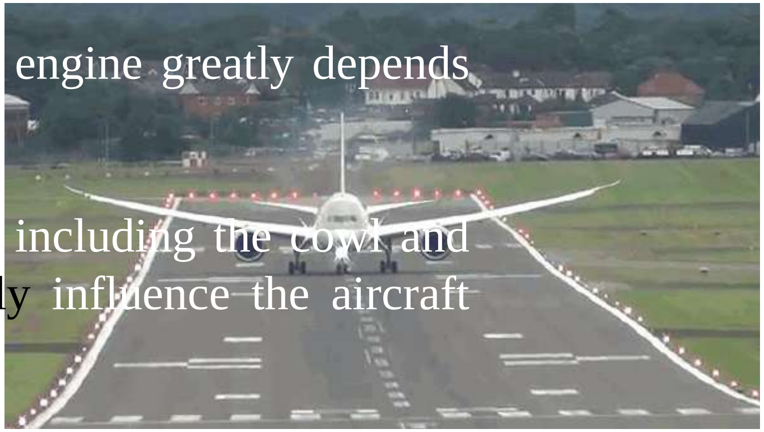
- The other model is for afterburning engines for supersonic fighters and bombers (M <2.5), and includes bypass ratios from zero to just under 1.

- For a post-1995 engine this author recommends, as a crude approximation, a 20% reduction in SFC, weight, and length for a given maximum thrust.

Propulsion and Fuel System Integration

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.
- Also, the inlet's external geometry including the cowl and boundary-layer diverter will greatly influence the aircraft drag.



- There are four basic types of inlet, as shown in Fig. 10.4.

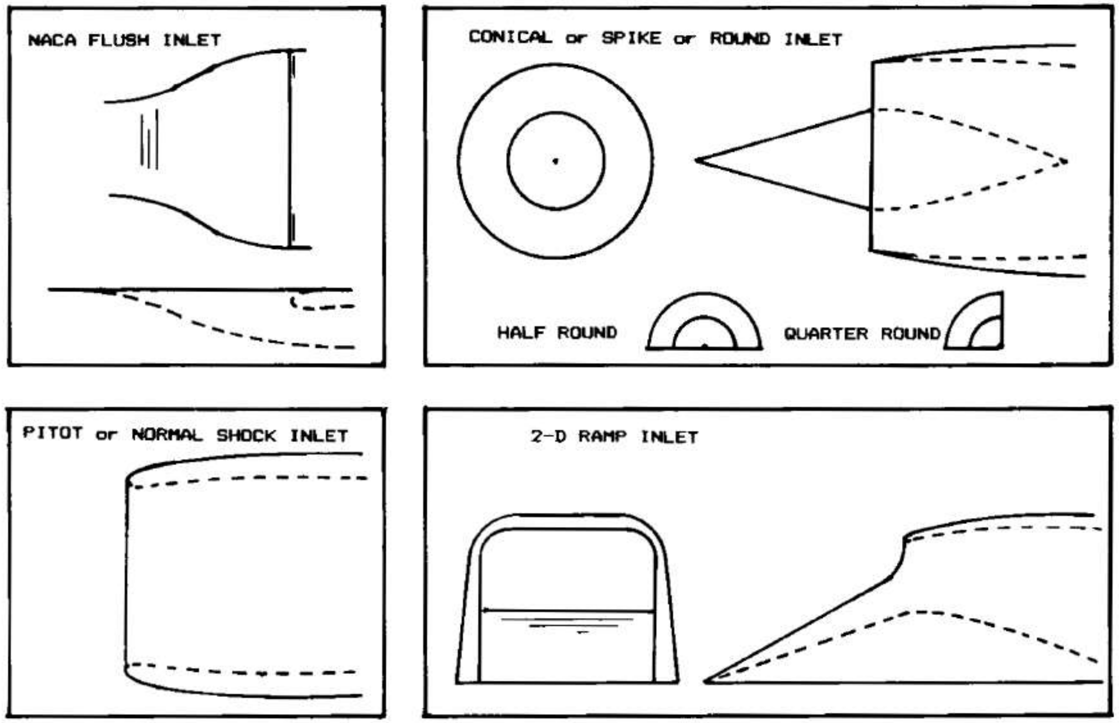


Fig. 10.4 Inlet types.

Propulsion and Fuel System Integration

- The NACA flush inlet was used by several early jet aircraft but is rarely seen today for aircraft propulsion systems due to its poor pressure recovery (i.e., large losses).

- However, the NACA inlet tends to reduce 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 auxiliary power units.

- Figure 10.5 and Table 10.1 provide dimensions for laying out a good NACA flush inlet.

Propulsion and Fuel System Integration

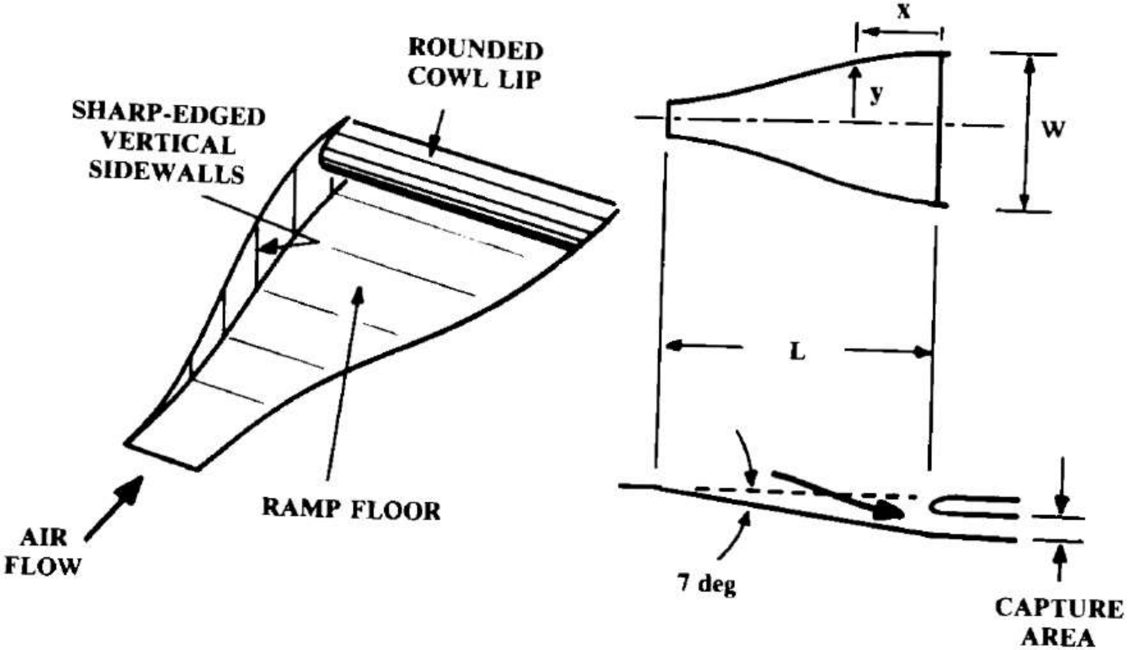


Fig. 10.5 Flush inlet geometry.

Table 10.1 Flush inlet wall geometry

x/L	$\frac{y}{W/2}$
1.0	0.083
0.9	0.160
0.8	0.236
0.7	0.313
0.6	0.389
0.5	0.466
0.4	0.614
0.3	0.766
0.2	0.916
0.1	0.996
0.0	1.000

Propulsion and Fuel System Integration

- The pitot inlet is simply a forward-facing hole.
- It works very well subsonically 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 10.6 gives design guidance for pitot inlets.

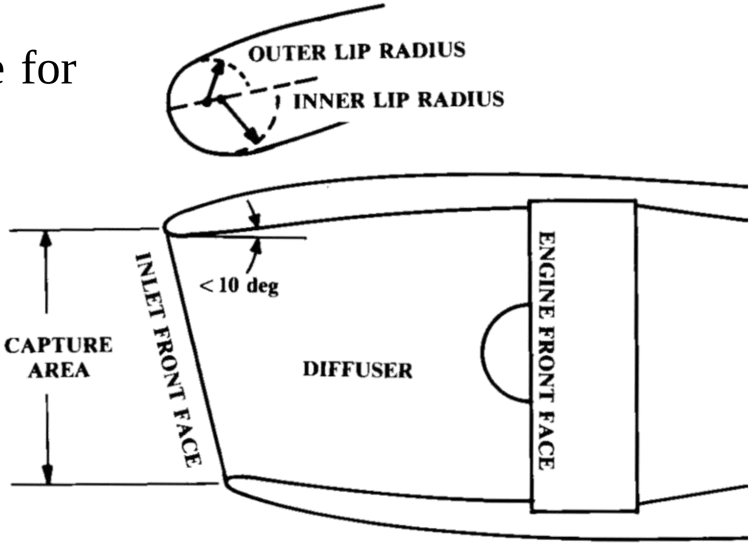


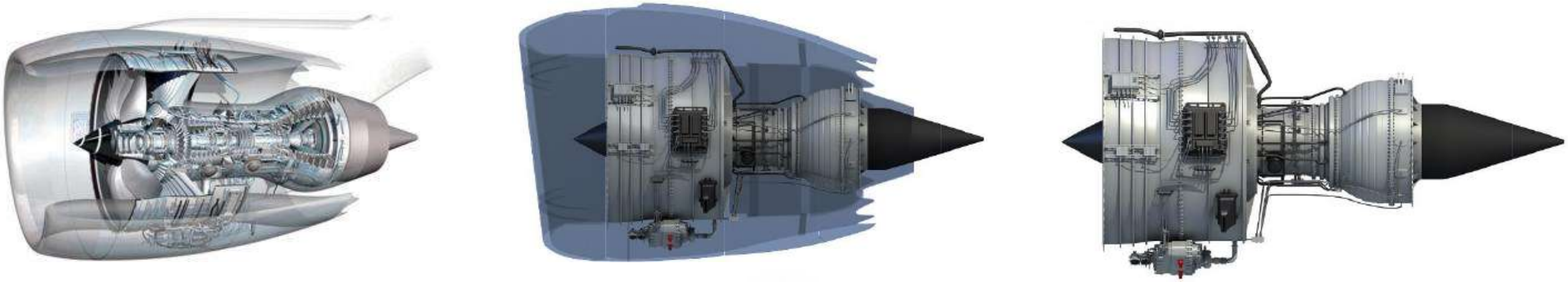
Fig. 10.6 Pitot (normal shock) inlet layout.

Propulsion and Fuel System Integration

- The cowl lip radius has a major influence upon engine performance and aircraft drag.
- For supersonic jets, the cowl lip should be nearly sharp.
- Typically the lip radius will be about 3-5% of the inlet front face radius.
- For subsonic jets, the lip radius ranges from 6-10% 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 8% inner radius and a 4% outer radius.
- Also, a number of aircraft have a lip radius on the lower part of the inlet up to 50% greater than that on the upper lip.
- This reduces the effects of angle of attack during takeoff and landing.

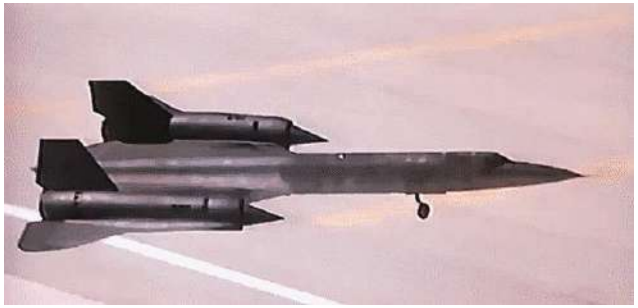
Propulsion and Fuel System Integration

- 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.



Propulsion and Fuel System Integration

- The remaining inlet types shown in Fig. 10.4 are for supersonic aircraft,
- and offer improvements over the performance of the normal shock inlet at higher supersonic speeds.
- The conical inlet (also called a spike, round, or axisymmetric inlet) exploits the shock patterns created by supersonic flow over a cone.
- Similarly, the two-dimensional ramp inlet (also called a "D-inlet") uses the flow over a wedge.



Propulsion and Fuel System Integration

- The spike inlet is typically lighter and has slighter better pressure recovery (1.5%),

- but has higher cowl drag and involves much more complicated mechanisms to produce variable geometry.

- The ramp inlet tends to be used more for speeds up to about Mach 2, while the spike inlet tends to be used above that speed.

- Any inlet must slow the air to about half the speed of sound before it reaches the engine.

- The final transition from supersonic to subsonic speed always occurs through a normal shock.

- The pressure recovery through a shock depends upon the strength of the shock, which is related to the speed reduction through the shock.

Propulsion and Fuel System Integration

- A series of shock waves including oblique shocks and a normal shock is preferred to reduce the speed of fl

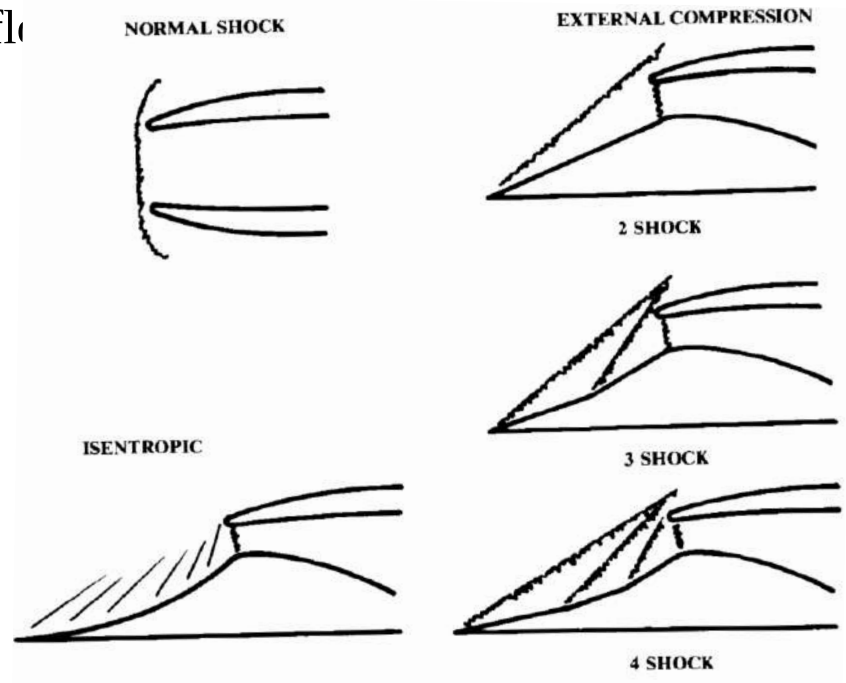


Fig. 10.7 Supersonic inlets—external shocks.

- This illustrates the principle of the external-compression inlet shown in Fig. 10.7.

- The above example is a two-shock system, one external and one normal.

- The greater the number of oblique shocks employed, the better the pressure recovery.

- The pure isentropic ramp inlet works properly at only its design Mach number, and is seen only rarely.

- However, isentropic ramps are frequently used in combination with flat wedge ramps.

Concorde

Propulsion and Fuel System Integration

- Figure 10.8 illustrates a typical three-shock external-compression inlet.
- This illustration could be a side view of a 2-D inlet or a section view through a spike inlet.
- Note that the second ramp has a variable angle, and can collapse to open a larger duct opening for subsonic flight.

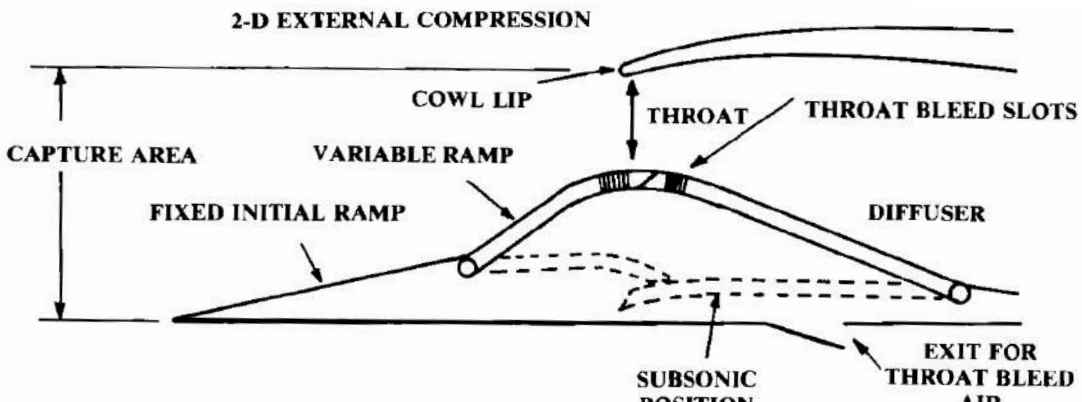


Fig. 10.8 Variable inlet geometry.

Propulsion and Fuel System Integration

- For initial layout, the overall length of the external portion of the inlet can be estimated by assuming an initial ramp angle (10-20 deg)

- and determining the shock wave angle for the design Mach number using standard shock wave charts.

- The cowl lip should be placed just aft of the shock.

- The throat area should be about 70-80% of the engine front-face area.

Propulsion and Fuel System Integration

- The "mixed compression inlet" as shown in Fig. 10.9 uses both external and internal compression to provide high efficiency over a wide Mach number range,
- with an acceptable amount of external flow turning.
- Typically one or more external oblique shocks will feed a single internal oblique shock, followed by a final normal shock.

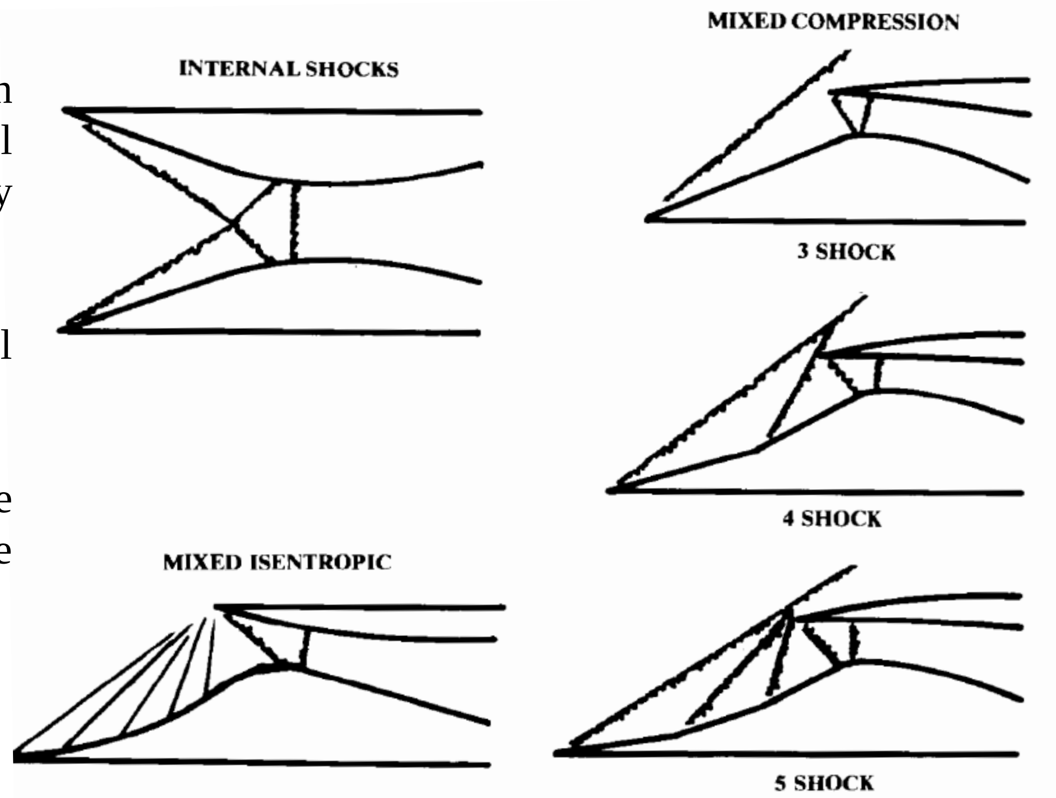
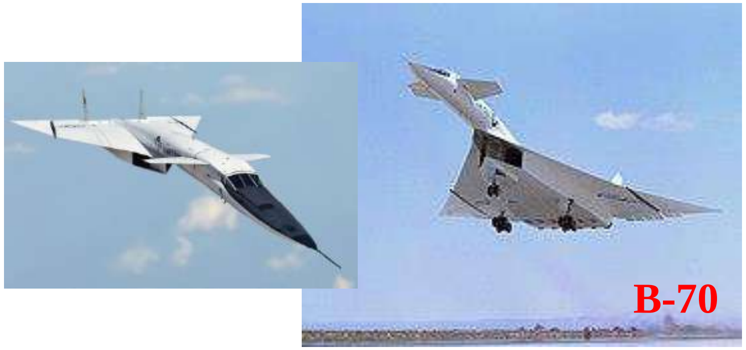


Fig. 10.9 Supersonic inlets—internal and mixed.



Propulsion and Fuel System Integration

- Figure 10.10 summarizes the selection criteria for different inlets, based upon design Mach number.

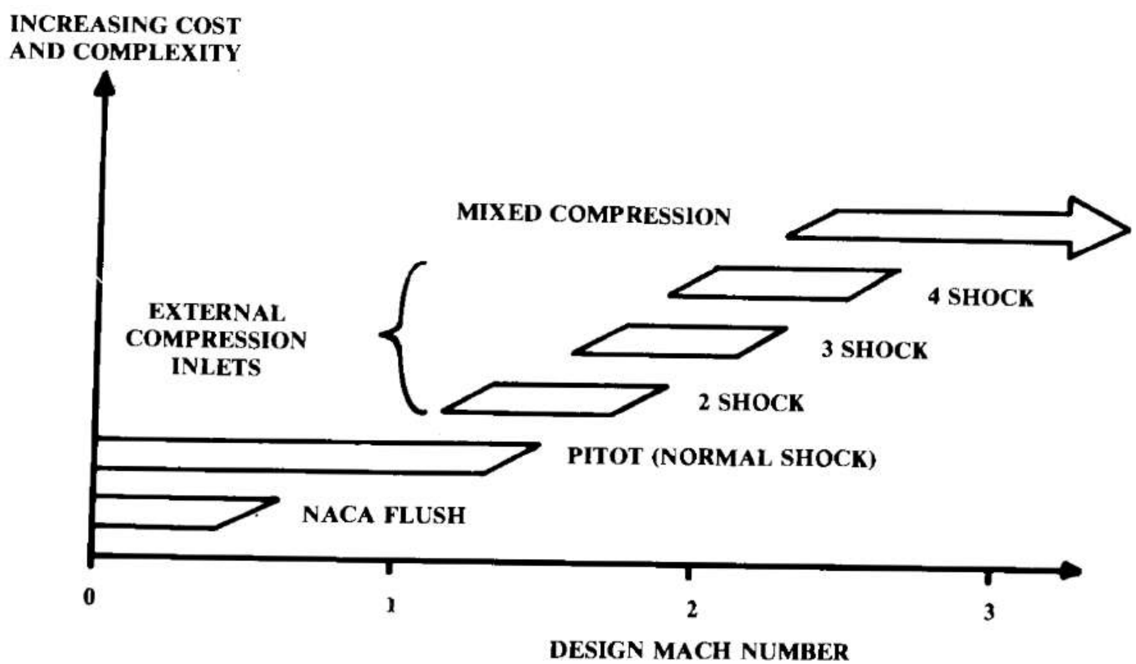


Fig. 10.10 Inlet applicability.

Propulsion and Fuel System Integration

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 a vortex off the fuselage or a separated wake from a wing, the resulting inlet - flow distortion can stall the engine.
- The F-111 had tremendous problems with its inlets, which were tucked up under the intersection of the wing and fuselage.
- The A-10 required a fixed slot on the inboard wing leading edge to cure a wake-ingestion problem.

Propulsion and Fuel System Integration

- Figures 10.11 and 10.12 illustrate the various options for inlet location.

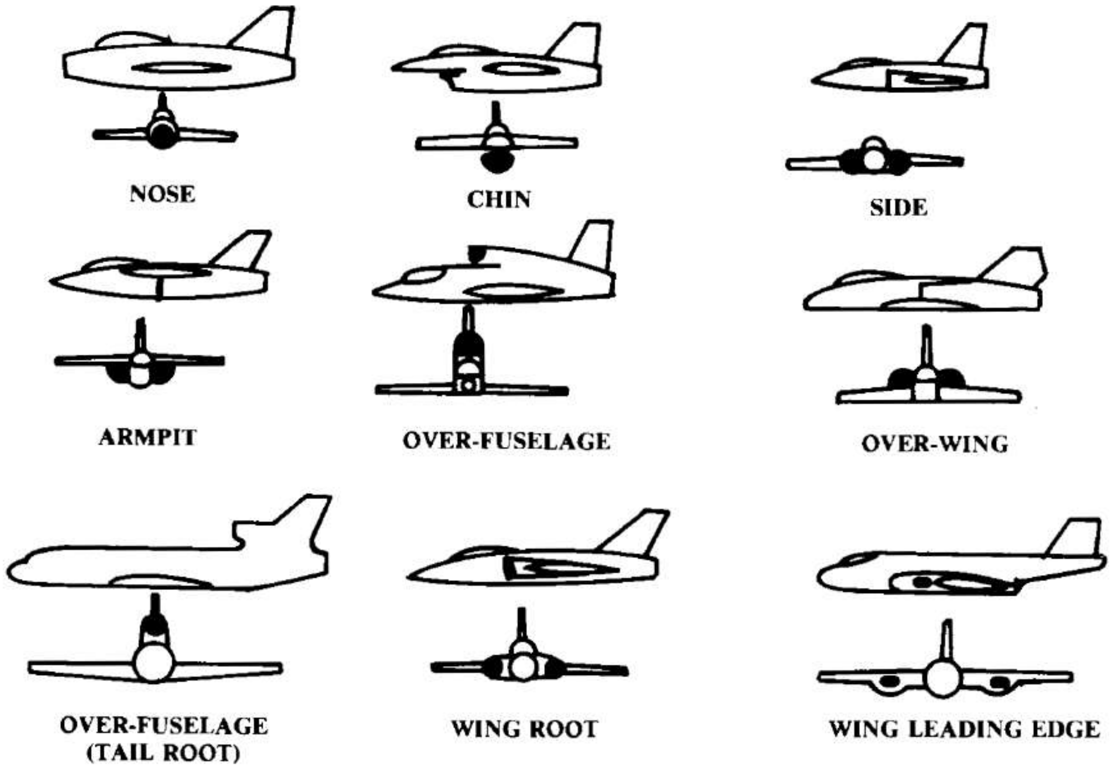


Fig. 10.11 Inlet locations—buried engines.

Propulsion and Fuel System Integration

- The nose location offers the inlet a completely clean airflow, and was used in most early fighters including the F-86 and MiG 21

- as a way of insuring that the fuselage would not cause distortion problems.

- However the nose inlet requires a very long internal duct, which is heavy, has high losses, and occupies much of the fuselage volume.
- The chin inlet as seen on the F-16 has most of the advantages of the nose inlet but a shorter duct length.

- The chin inlet is especially good at high angle of attack because the fuselage forebody helps to turn the flow into it.

Propulsion and Fuel System Integration

- The location of the nose landing gear is a problem.
- If two engines are used, twin inlets can be placed in the chin position with the nose wheel located between them.
- This was used on the North American Rockwell proposal for the F-15, and is seen on the Su-27.

Propulsion and Fuel System Integration

- Another problem with the chin inlet is foreign-object ingestion by suction.

- As a rule of thumb, all inlets should be located a height above the runway equal to

  - at least 80% of the inlet's height if using a low bypass ratio engine,
  - and at least 50% of the inlet's height for a high-bypass-ratio engine.

Propulsion and Fuel System Integration

- Side-mounted inlets are now virtually standard for aircraft with twin engines in the fuselage.
- Side inlets provide short ducts and relatively clean air.
- Side-mounted inlets can have problems at high angles of attack due to the vortex shed off the lower corner of the forebody.
- If side-mounted inlets are used with a single engine, a split duct must be used.
- A side inlet at the intersection of the fuselage and a high wing is called an "armpit" inlet. It is risky!

Propulsion and Fuel System Integration

- An over-fuselage inlet is much like 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.

Propulsion and Fuel System Integration

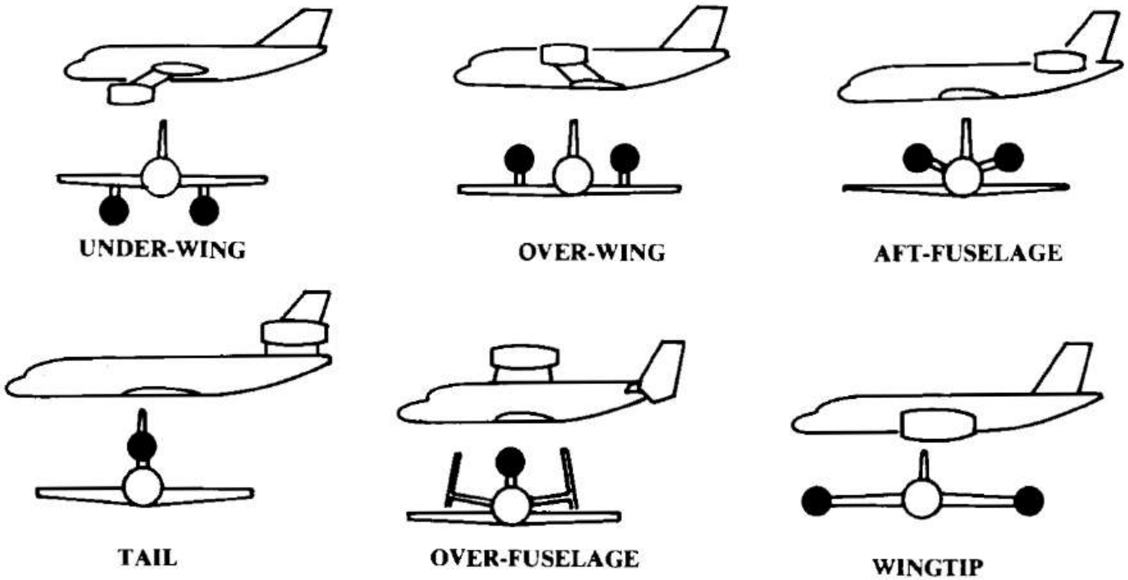


Fig. 10.12 Inlet locations—podded engines.

Propulsion and Fuel System Integration

- A podded engine has higher wetted area than a buried engine, but offers substantial advantages that have made it standard for commercial and business jets.
- Podded engines place the inlet away from the fuselage, providing undisturbed air with a very short inlet duct.
- Podded engines produce less noise in the cabin because the engine and exhaust are away from the fuselage.
- Podded engines are usually easier to get to for maintenance.
- Most are mounted on pylons, but they can also be mounted conformal to the wing or fuselage.

Propulsion and Fuel System Integration

- The wing-mounted podded engine is the most commonly used engine installation for jet transports.
- The engines are accessible from the ground and well away from the cabin.
- The weight of the engines out along the wing provides a "span-loading" effect, which helps reduce wing weight.
- The jet exhaust can be directed downward by flaps which greatly increases lift for short takeoff.
- On the negative side, the presence of pods and pylons can disturb the airflow on the wing, increasing drag and reducing lift.
- To minimize this, the pylons should not extend above and around the wing leading edge

Propulsion and Fuel System Integration

- 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.

Propulsion and Fuel System Integration

- To reduce foreign-object ingestion by suction, the inlet of a high-bypass engine should be located about half a diameter above the ground.

- This requirement increases the required landing-gear height of the under-wing arrangement.

- The other standard engine installation for jet transports is the aft fuselage mount, usually with a T-tail.

- This eliminates the wing-interference effects of wing-mounted engines, and allows a short landing gear.

- However, it increases the cabin noise at the rear of the aircraft.

- Also, aft-mounting of the engines tends to move the center of gravity aft, which requires shifting the entire fuselage forward relative to the wing.

Propulsion and Fuel System Integration

B-727

Tupolev Tu-22

Capture-Area Calculation

- Figure 10.13 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.

- If mass flow is not known, it may be estimated as 0.18 times the square of the engine front-face diameter in inches, or as 0.12 times the square of the maximum engine diameter.

- To determine the required capture area, the mass flow is multiplied by the value from Fig. 10.13.

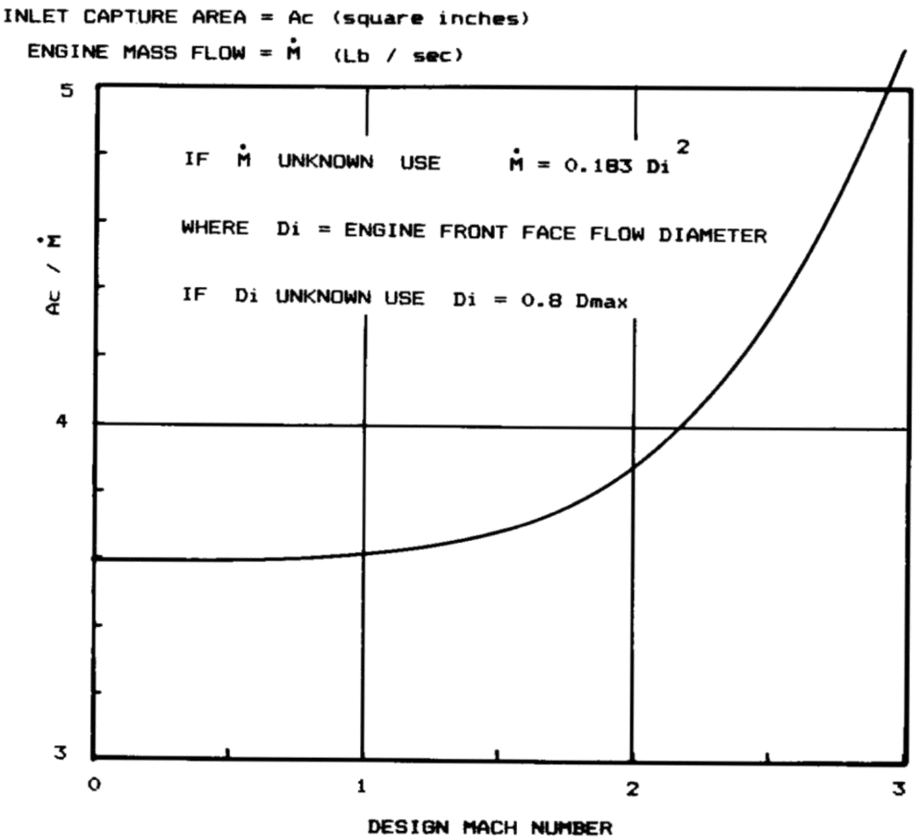


Fig. 10.13 Preliminary capture area sizing.

Propulsion and Fuel System Integration

- In a jet propulsion system, the engine is the boss.
- 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.
- Figure 10.14 defines the capture area for a subsonic inlet.
- A typical subsonic jet inlet is sized for cruise at about Mach 0.8-0.9, and the inlet must slow the air to about Mach 0.4 for most engines.

Propulsion and Fuel System Integration

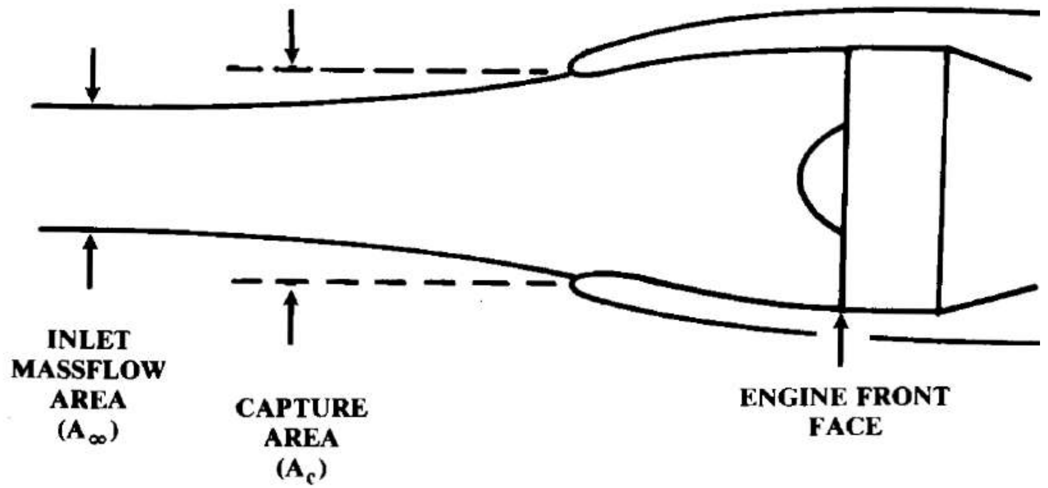


Fig. 10.14 Subsonic inlet capture area.

- The area at the inlet front face can be calculated from the following isentropic compressible flow relationships:

$$\frac{A_{\text{throat}}}{A_{\text{engine}}} = \frac{(A/A^*)_{\text{throat}}}{(A/A^*)_{\text{engine}}} \tag{10.16}$$

$$\frac{A}{A^*} = \frac{1}{M}\left(\frac{1 + 0.2M^2}{1.2}\right)^3 \tag{10.17}$$

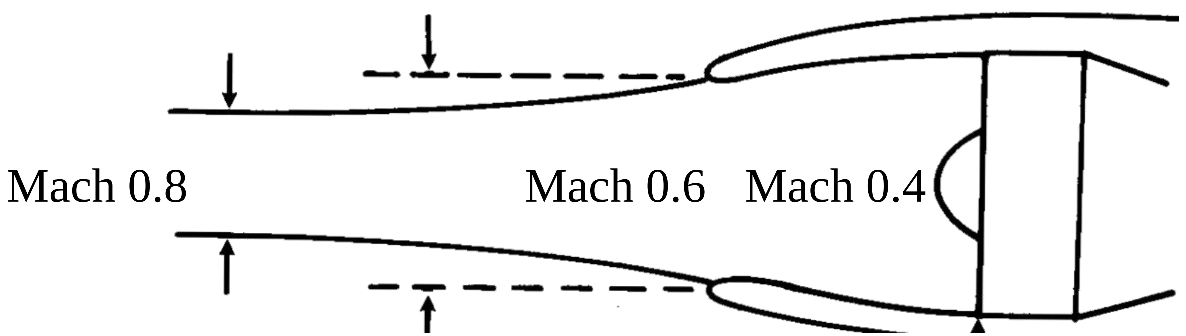
where A^* is the area of the same flow at sonic speed.

Propulsion and Fuel System Integration

- 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.

- Equations (10.16) and (10.17) give the ratio between throat area and engine front-face area as 1.188/1.59, or 0. 75 (for this example).



Propulsion and Fuel System Integration

- Equations (10.16) and (10.17) may be used to determine the capture area for a supersonic pitot inlet with negligible bleed or secondary airflow by finding the Mach number behind the normal shock.

- 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.

- The secondary airflow requirements are accurately determined by an evaluation of the aircraft's subsystems such as environmental control.

- For initial capture-area estimation, Table 10.2 provides secondary airflow as a fraction of engine mass flow.

Propulsion and Fuel System Integration

Table 10.2 Secondary airflow (typical)[26]

	$\dot{m}_s/\dot{m}_e$
Engine	
Nacelle cooling	0–0.04
Oil cooling	0–0.01
Ejector nozzle air	0.04–0.20
Hydraulic system cooling	0–0.01
Environmental control system cooling air (if taken from inlet)	0.02–0.05
Typical totals	
Fighter	0.20
Transport	0.03

$$\dot{m} = g\rho VA, \text{ lb/s} \tag{10.18}$$

- The required engine mass flow is provided by the engine manufacturer, and is a function of the Mach number, altitude, and throttle setting

Propulsion and Fuel System Integration

- Figure 10.15 defines the capture area geometry for a supersonic ramp or cone inlet.
- This inlet is shown at the design case, which is known as "shock on-cowl."
- At this Mach number and ramp angle, the initial oblique shock is almost touching the cowl lip.
- If the auxiliary doors are shut and the shock is on cowl, the geometric capture area is providing exactly the right amount of air for the engine, bleed, and secondary flow.

Propulsion and Fuel System Integration

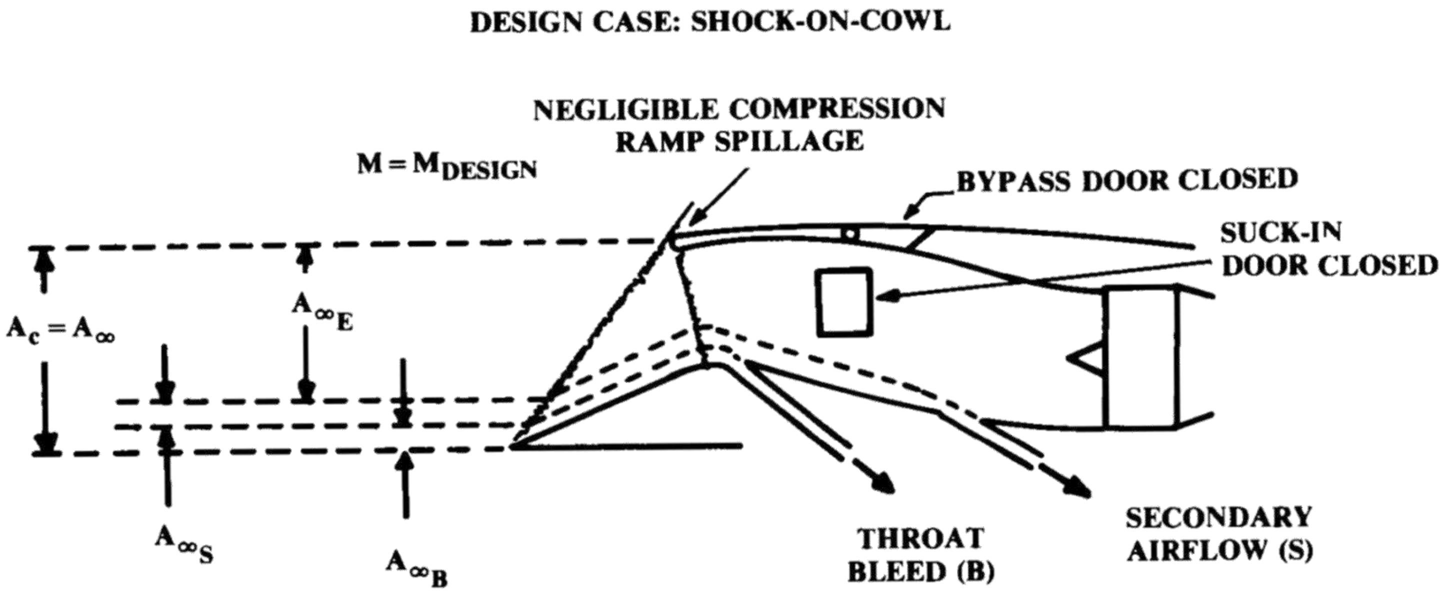


Fig. 10.15 Supersonic inlet capture area—on design.

- Inlet boundary layer bleed should also be determined analytically, but can be approximated using Fig. 10.16.

- This estimates the required extra capture area for bleed as a percent of the capture area required for the engine and secondary airflow.

- The capture area is therefore determined as in Eq. (10.19), using Table 10.2 and Fig. 10.16.

$$A_{\text{capture}} = \left[\frac{\dot{m}_E(1+\dot{m}_S/\dot{m}_E)}{g\rho_\infty V_\infty}\right]\left(1+\frac{A_B}{A_c}\right) \qquad (10.19)$$

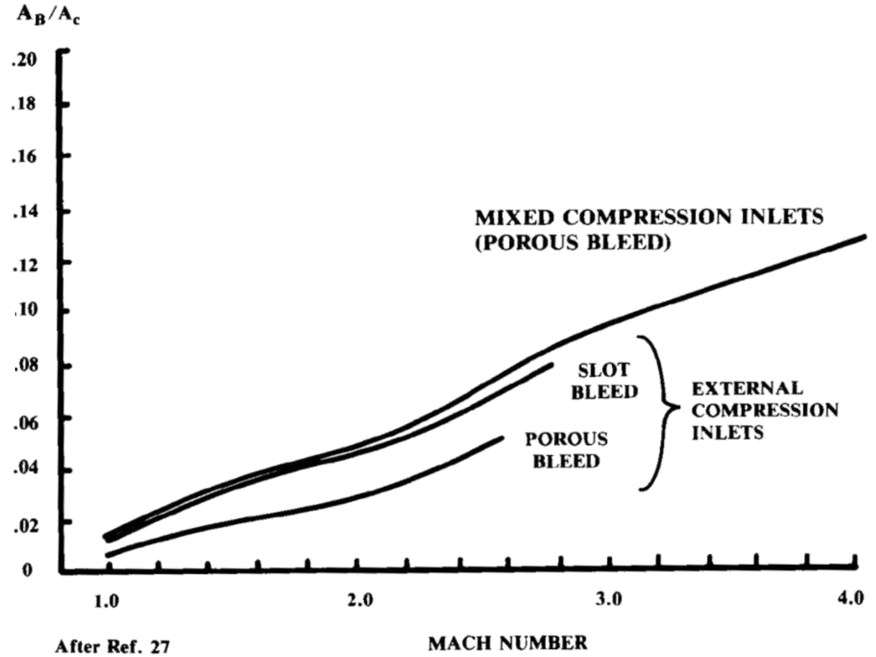


Fig. 10.16 Typical boundary layer bleed area.

Propulsion and Fuel System Integration

Boundary-Layer Diverter

- Any object moving through air will build up a boundary layer on its surface.

- The aircraft's forebody builds up its own boundary layer. If this low-energy, turbulent air is allowed to enter the engine,

- it can reduce engine performance subsonically and prevent proper inlet operation supersonically.

- The four major varieties of boundary-layer diverter are shown in Fig. 10.18.

Propulsion and Fuel System Integration

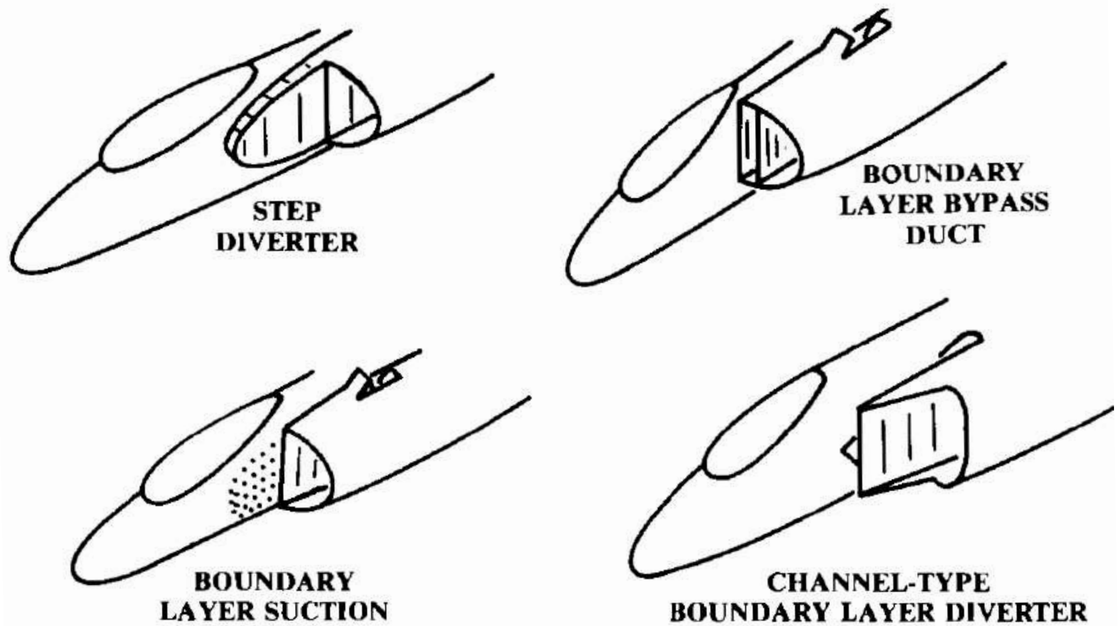
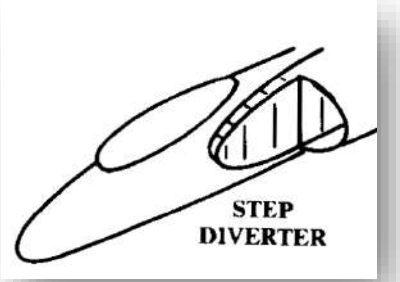


Fig. 10.18 Boundary layer removal.

Propulsion and Fuel System Integration

- The step diverter s suitable only for subsonic aircraft and relies upon the boundary layer itself for operation.
- The boundary layer consists of low-energy air, compared to the air outside of the boundary layer.
- The step diverter works by forcing the boundary-layer air to climb the step, pushing aside high-energy air outside the boundary layer.

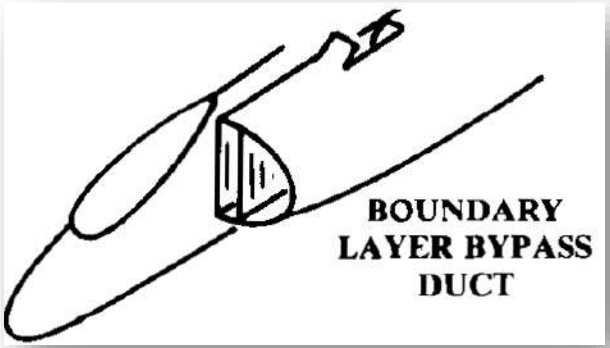


- The step diverter should have an airfoil-like shape that is faired smoothly to the nacelle.
- The diverter should extend about one inlet diameter forward of the inlet, and should be have a depth equal to roughly 2-4% of the forebody length ahead of the inlet.

Propulsion and Fuel System Integration

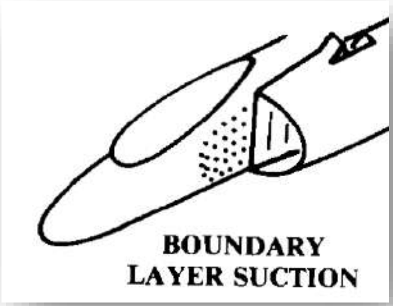
- The boundary-layer bypass duct (simply a separate inlet duct) admits the boundary-layer air and ducts it to an aft-facing hole.

- The internal duct shape should expand roughly 30% from intake to exit to compensate for the internal friction losses.



Propulsion and Fuel System Integration

- The suction form of boundary-layer diverter is similar.
- The boundary layer air is removed by suction through holes or slots just forward of the inlet and ducted to an aft-facing hole.
- This type of diverter does not benefit from the ram impact of the boundary-layer air, and therefore does not work as well.



Propulsion and Fuel System Integration

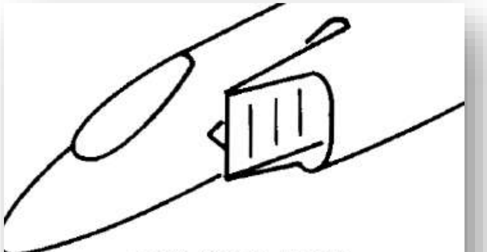
- The channel diverter is the most common boundary-layer diverter for supersonic aircraft.

- It provides the best performance and the least weight in most cases.

- The inlet front face is located some distance away from the fuselage, with a splitter plate to insure that the boundary-layer air does not get into 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.



CHANNEL-TYPE
BOUNDARY LAYER DIVERTER

Propulsion and Fuel System Integration

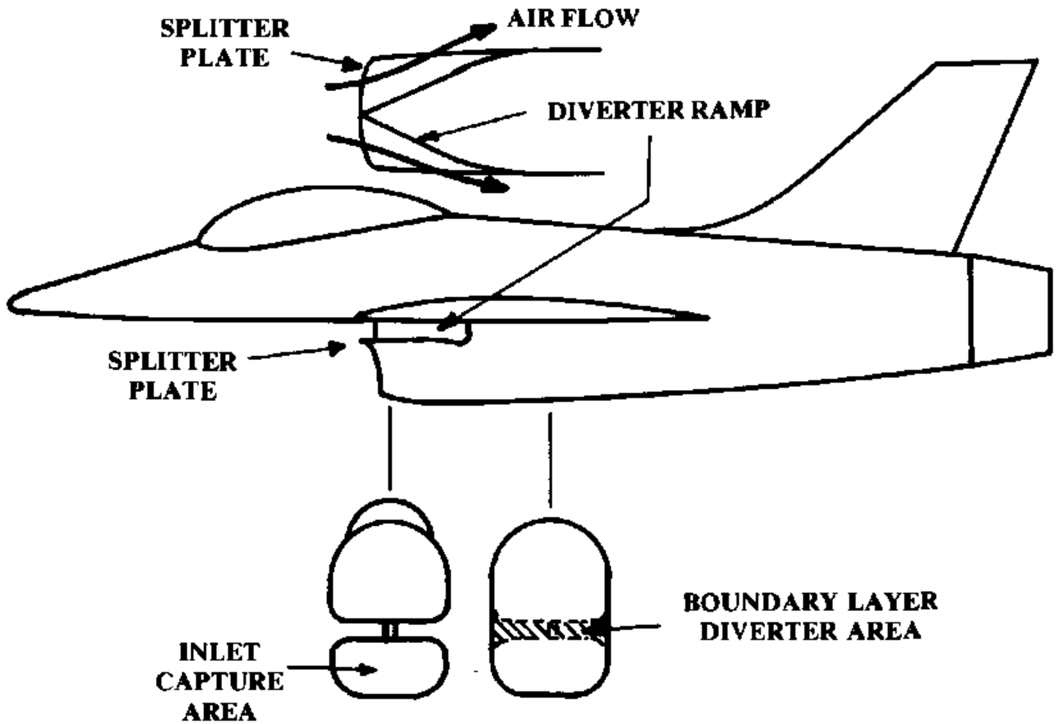


Fig. 10.19 Boundary layer diverter.

- 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.

Propulsion and Fuel System Integration

- We have an additional option for boundary layer problem.
- That is diverterless inlet design.
- It consists of a "bump" and a forward-swept inlet cowl, which work together to divert boundary layer airflow away from the aircraft's engine.
- This eliminates the need for a splitter plate,
- The DSI bump functions as a compression surface and creates a pressure distribution that prevents the majority of the boundary layer air from entering the inlet at speeds up to Mach 2

Propulsion and Fuel System Integration

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.

Propulsion and Fuel System Integration

- Typical nozzles are shown in Fig. 10.20.

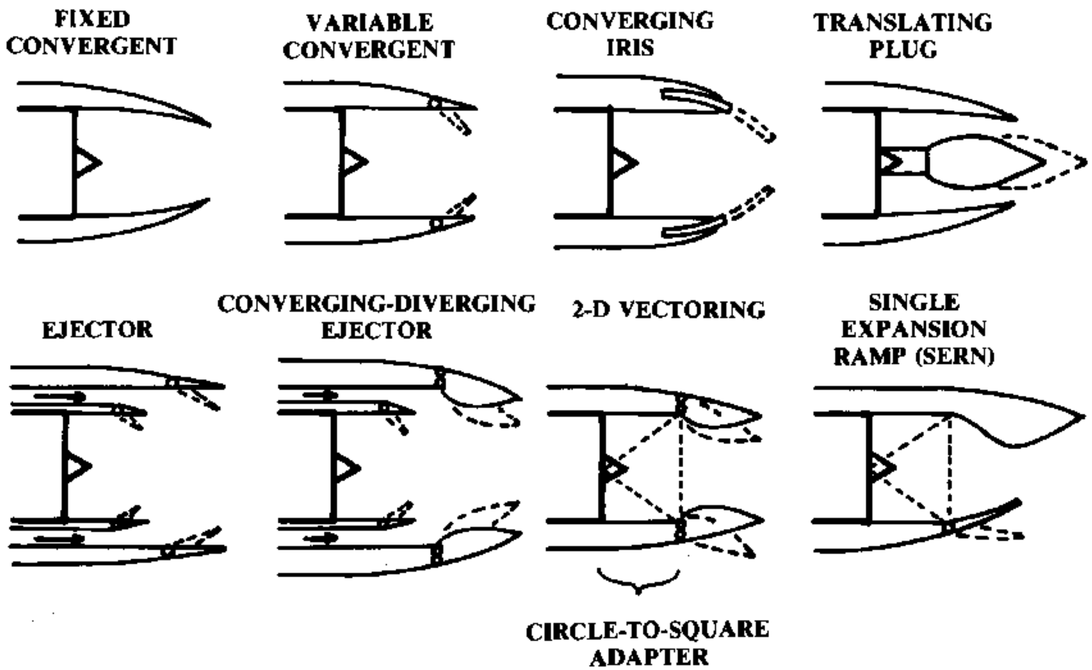


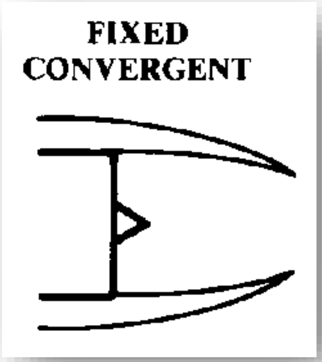
Fig. 10.20 Types of nozzles.

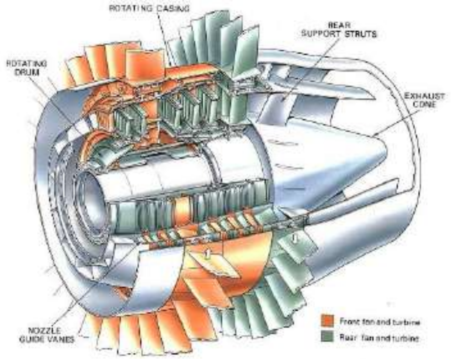
Propulsion and Fuel System Integration

- The fixed convergent nozzle is almost universally used for subsonic commercial turbojet and turbofan engines.

- The nozzle exit area is selected for cruise efficiency, resulting in a loss of theoretical performance at lower speeds.

- However, the gain in simplicity and weight reduction of the fixed nozzle more than makes up for the performance loss in most subsonic applications.

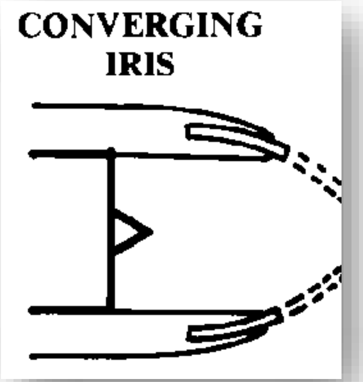




Propulsion and Fuel System Integration

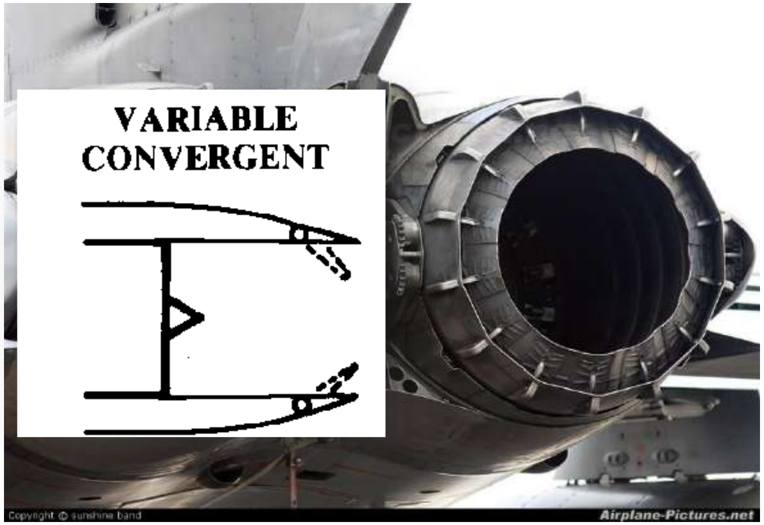
- For an aircraft which occasionally flies at high-subsonic to low-supersonic speeds, a variable-area convergent nozzle allows a better match.

- The nozzle shown has a fixed outer surface, which causes a "base" area when the nozzle inside is in the closed position.



- Such a nozzle was used on many early transonic fighters, but is not typically used today.

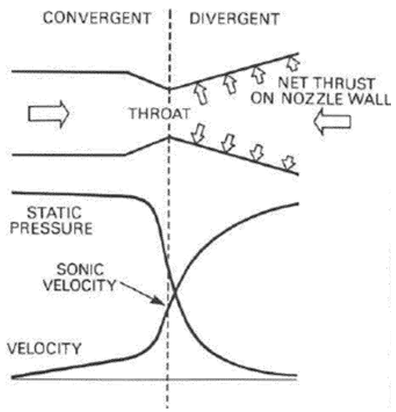
- Instead, the convergent-iris nozzle is used to vary the area of a convergent nozzle without introducing a base area.



Propulsion and Fuel System Integration

- 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.

Propulsion and Fuel System Integration



- It allows varying the throat and exit areas separately for maximum engine performance throughout the flight envelope.

F 16 and F 100 engine

Propulsion and Fuel System Integration

- For initial design layout, a reasonable approximation can be made based upon the estimated capture area.

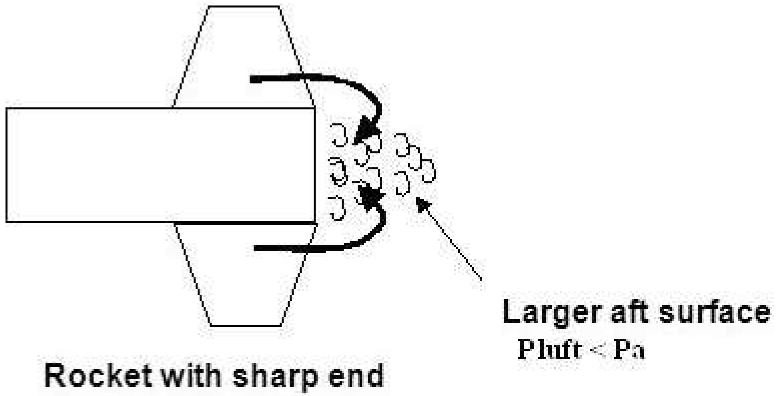
- For a subsonic convergent nozzle or a convergent-divergent nozzle in the closed position, the required exit area is approximately 0.5-0. 7 times the capture area.

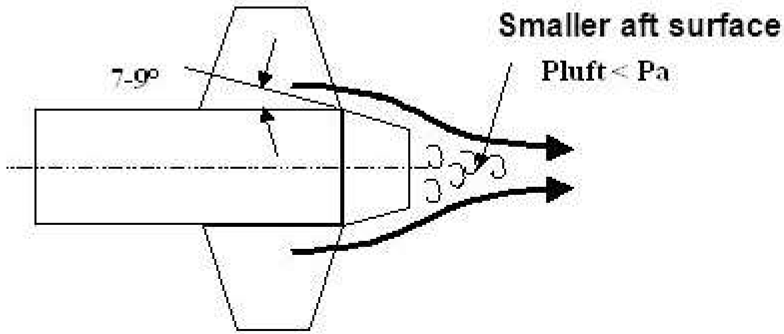
- For maximum supersonic afterburning operation, the required exit area is about 1.2-1.6 times the capture area.

Propulsion and Fuel System Integration

- As mentioned, nozzle arrangement can have a substantial effect on boat-tail drag.
- This is the drag due to separation on the outside of the nozzle and aft fuselage.
- To reduce boat-tail drag to acceptable levels, the closure angles on the aft fuselage should be kept below 15 deg,
- and the angles outside of the nozzle should be kept below 20 deg in the nozzle-closed position.
- Jet engines mounted next to each other produce an interference effect that reduces net thrust.
- To minimize this, the nozzles should be separated by about one to two times their maximum exit diameter.

Propulsion and Fuel System Integration





boat-tail drag

Twin engines

Propulsion and Fuel System Integration

Propeller-engine Integration

- The actual details of the propeller design such as the blade shape and twist are not required to lay out a propeller-engine aircraft.

- But the diameter of the propeller, the dimensions of the engine, and the required inlets and exhausts must be determined.

Propeller Sizing

  - Generally speaking, the larger the propeller diameter, the more efficient the propeller will be.

  - The limitation on length is the propeller tip speed, which should be kept below sonic speed.

  - Tip speed is the vector sum of the rotational speed [Eq. (10.21)] and the aircraft's forward speed as defined in Eq. (10.22).

Propulsion and Fuel System Integration

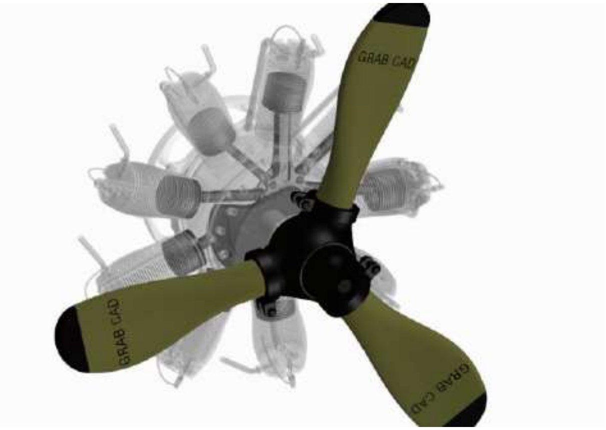
$$(V_{\text{tip}})_{\text{static}} = \pi n d/60 \qquad (\text{ft/s}) \tag{10.21}$$

where

n = rotational rate (rpm) obtained from engine data
d = diameter

$$(V_{\text{tip}})_{\text{helical}} = \sqrt{V_{\text{tip}}^2 + V^2} \tag{10.22}$$

- At sea level the helical tip speed of a metal propeller should not exceed 950 fps.
- A wooden propeller, which must be thicker, should be kept below 850 fps.
- If noise is of concern, the upper limit for metal or wood should be about 700 fps during takeoff.



Propulsion and Fuel System Integration

- Equations (10.23-10.25) provide an estimate of the propeller diameter as a function of horsepower.

- The propeller diameters obtained from these equations should be compared to the maximum diameters obtained from tip-speed considerations,

- and the smaller of the two values used for initial layout.

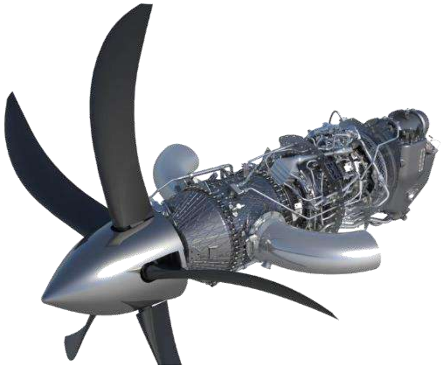
Two blade: $$d = 22 \sqrt[4]{\mathrm{Hp}} \tag{10.23}$$

Three blade: $$d = 18 \sqrt[4]{\mathrm{Hp}} \tag{10.24}$$

Three blade (agricultural): $$d = 20 \sqrt[4]{\mathrm{Hp}} \tag{10.25}$$

Propulsion and Fuel System Integration

- Most aircraft propellers have a "spinner," a cone- or bullet-shaped fairing at the hub.
- The inner part of the propeller contributes very little to the thrust.
- A spinner pushes the air out to where the propeller is more efficient.
- Also, a spinner streamlines the nacelle.
- Ideally, the spinner should cover the propeller out to about 25% of the radius, although most spinners are not that large.



Propulsion and Fuel System Integration

Propeller Location

- A matrix of possible propeller locations is shown in Fig. 10.21.
- A tractor installation has the propeller in front of its attachment point (usually the motor).
- A pusher location has the propeller behind the attachment point.

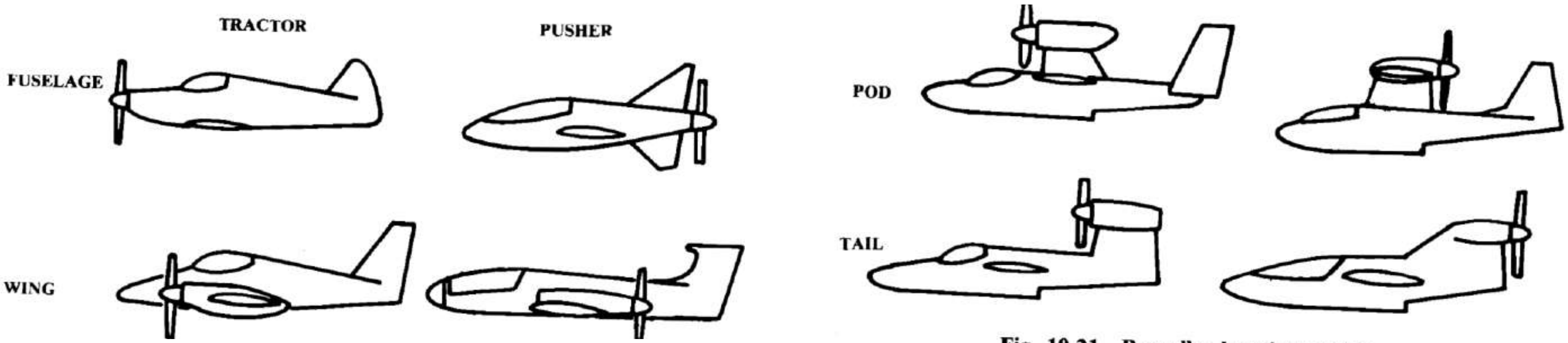
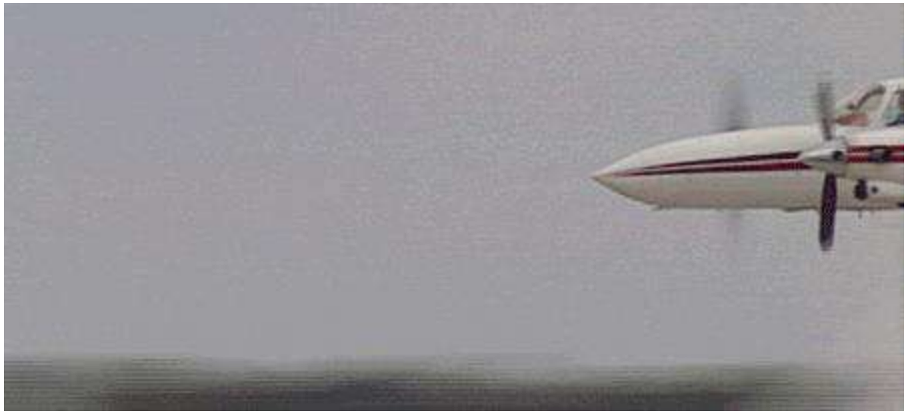


Fig. 10.21 Propeller location matrix.

Propulsion and Fuel System Integration

- The tractor location has been standard for most of the history of aviation.
- The conventional tractor location puts the heavy engine up front, which tends to shorten the forebody, allowing a smaller tad area and improved stability.
- The tractor location also provides a ready source of cooling air, and places the propeller in undisturbed air.



Propulsion and Fuel System Integration

- The pusher location can reduce aircraft skin friction drag because the pusher location allows the aircraft to fly in undisturbed air.

- The inflow caused by the propeller allows a much steeper fuselage closure angle without flow separation than otherwise possible.

- The canard-pusher combination is especially favorable because the canard requires a shorter tail arm than the aft tail.

- The pusher propeller reduces cabin noise because the engine exhaust is pointed away from the cabin

- Also, the pusher arrangement usually improves the pilot's outside vision.

Propulsion and Fuel System Integration

- The pusher propeller may require longer landing gear because the aft location causes the propeller to dip closer to the runway as the nose is lifted for takeoff.

- The propeller should have at least 9 in. of clearance in all attitudes.

- The pusher-propeller is also more likely to be damaged by rocks thrown up by the wheels.

- The Cessna Skymaster and Rutan Defiant use a combination of pusher and tractor engines on the fuselage.

Propulsion and Fuel System Integration

- Wing mounting of the engines is normally used for multiengine designs.
- Wing mounting of engines reduces wing structural weight through a spanloading effect, and reduces fuselage drag by removing the fuselage from the propeller wake.
- Wing mounting of engines introduces engine-out controllability problems that force an increase in the size of the rudder and vertical tail.
- Also, care must be taken to insure that the crew compartment is not located within plus or minus 5 deg of the propeller disk, in case a blade is thrown through the fuselage.

Propulsion and Fuel System Integration

- Most twin-engine aircraft are of low-wing design.
- For these, the location of the engine and propeller on the wing requires a longer landing gear.
- Frequently the propeller will be raised above the plane of the wing to reduce landing-gear height.
- This causes additional interference between the wing and propeller.

Propulsion and Fuel System Integration

- Upper fuselage pods and tail-mounted pods tend to be used only for seaplane and amphibian designs,
- which need a huge clearance between the water and the propeller (minimum of 18 in., preferably one propeller diameter).
- The high thrust line can cause undesirable control characteristics in which application of power for an emergency go-around produces a nose down pitching moment.

Propulsion and Fuel System Integration

Engine-Size Estimation

- If a production engine is to be used, dimensional and installation data can be obtained from the manufacturer.
- If a rubber-engine is to be used, an existing engine can be scaled using the scaling equations defined in Table 10.3.
- Alternatively, the statistical models defined in Table 10.4 can be used to define a nominal engine.

Table 10.3 Scaling laws for piston and turboprop engines

$X_{scaled} = X_{actual} SF^{b}$; b from table values

$SF = \text{bhp}_{scaled}/\text{bhp}_{actual}$

	Piston engines			Turboprop
X	Opposed	In-line	Radial	
Weight	0.78	0.78	0.809	0.803
Length	0.424	4.24	0.310	3.730
Diameter	*	*	0.130	0.120

*Width and height vary insignificantly within ±50% horsepower.

Propulsion and Fuel System Integration

Table 10.4 Piston and turboprop statistical models

$X = a(\text{bhp})^b$ (lb or in.)

	Piston engines						Turboprop	
X	Opposed		In-line		Radial			
	a	b	a	b	a	b	a	b
Weight	5.47	0.780	5.22	0.780	4.90	0.809	1.67	0.803
Length	3.86	0.424	5.83	0.424	6.27	0.310	4.14	0.373
Diameter	Width 32–34 in. Height 22–25 in.		Width 17–19 in. Height 24–26 in.		20.2	0.130	9.48	0.120
Typical propeller rpm	2770		2770		2300			
Applicable bhp range	60–500		100–300		200–2000		400–5000	

Propulsion and Fuel System Integration

Piston-Engine Installation

- Piston engines have special installation requirements that can greatly affect the configuration layout.
- These are illustrated in Fig. 10.22.
- Cooling is a major concern.
- Up to 10% of the engine's horsepower can be wasted by the drag associated with taking in cooling air, passing it over the engine, and exiting it.
- As a rough rule of thumb, the cooling-air intake should be about 30-50% of the engine frontal area.
- The exit should be about 30% larger, and may be variable in area ("cowl flaps") to better control cooling airflow.

Propulsion and Fuel System Integration

- For tractor engines, the cooling-air intake is usually located directly in front of the engine cylinders.
- The air is diverted over the top of the engine by "baffles," which are flat sheets of metal that direct the airflow within the engine compartment.
- The air then flows down through and around the cylinders into the area beneath the engine,
- and then exits through an aft facing hole below the fuselage. This is referred to as "down-draft" cooling.

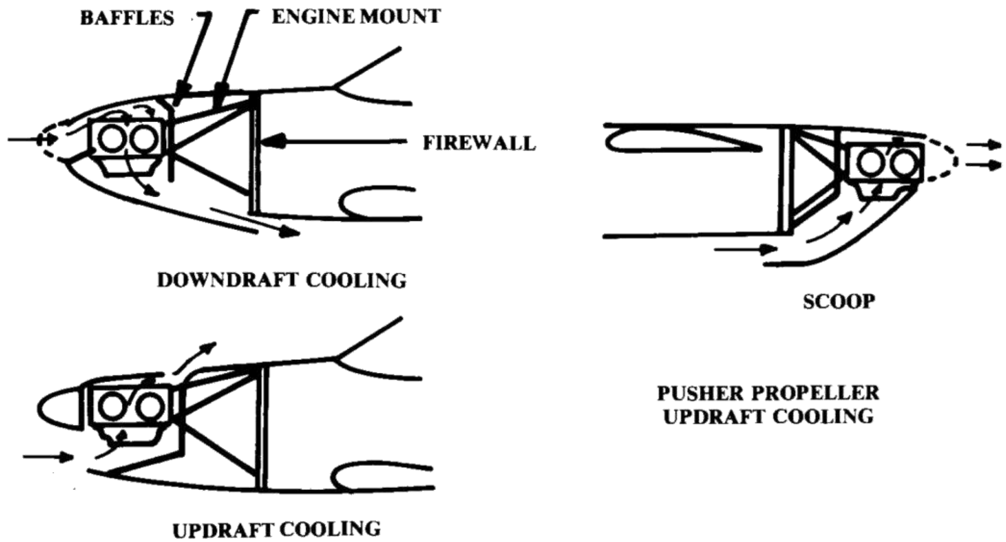
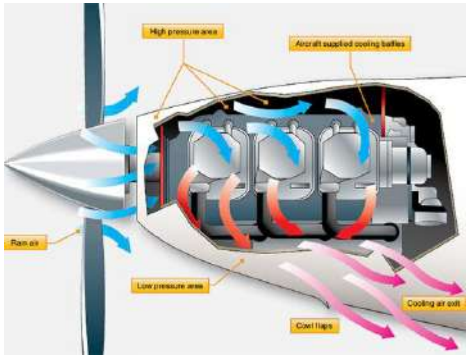


Fig. 10.22 Piston engine installation.



Propulsion and Fuel System Integration

- Down-draft cooling exits the air beneath the fuselage, which is a high pressure area and therefore a poor place to exit air.
- "Up-draft" cooling flows the cooling air upwards through the cylinders and exits it into low pressure air above the fuselage,
- creating more efficient cooling flow due to a suction effect.
- However, updraft cooling dumps hot air in front of the windscreen; this can heat up the cabin.
- An engine oil leak can coat the windscreen with black oil.
- Aircraft engines have the exhaust pipes below the cylinders, so updraft cooling causes the cooling air to be heated by the exhaust pipes before reaching the cylinders.

Propulsion and Fuel System Integration

- For pusher engines cooling is much more difficult.
- For these reasons virtually all piston-pushers use updraft cooling with a large scoop mounted below the fuselage.
- Also, internal fans are sometimes used to improve cooling on pusher configurations.

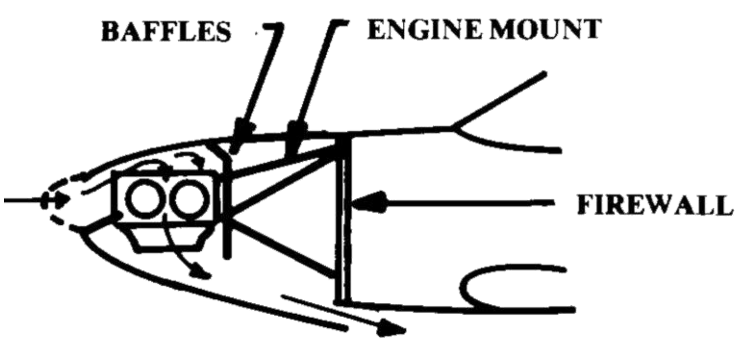
Propulsion and Fuel System Integration

- Figure 10.22 also shows the motor mount and firewall.

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

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

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



Propulsion and Fuel System Integration

firewall.

motor mount

- The firewall is typically a 0.015-in. steel sheet (stainless or galvanized) attached to the first structural bulkhead of the fuselage or nacelle.

- Its purpose is to prevent a fire in the engine compartment from damaging the aircraft structure or spreading into the rest of the aircraft.

Propulsion and Fuel System Integration

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.

- 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.



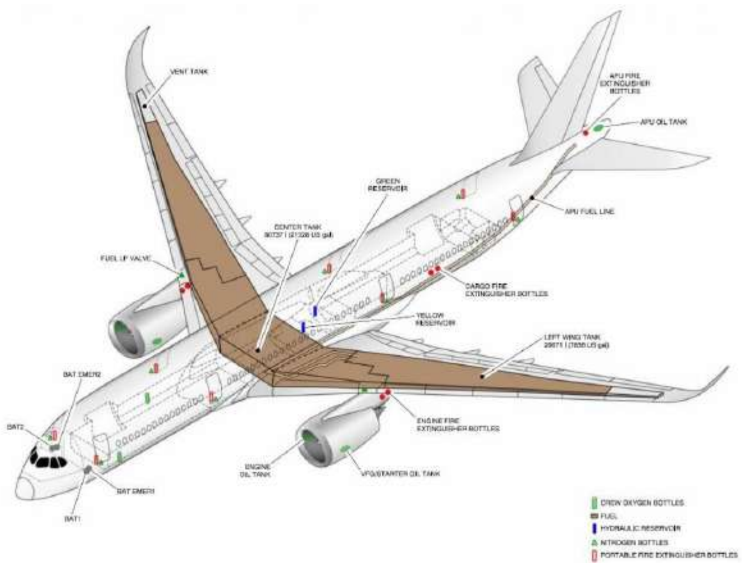
Propulsion and Fuel System Integration

- Discrete tanks are normally used only for small 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.

Propulsion and Fuel System Integration

- Integral tanks are cavities within the airframe structure that are sealed to form a fuel tank.

- Ideally, an integral tank would be created simply by sealing existing structure such as wing boxes and cavities created between two fuselage bulkheads.



Propulsion and Fuel System Integration

- Despite years of research, integral tanks are still prone to leaks as witnessed upon the introduction of the B-1B into service.

- Due to the fire hazard in the event of a leak or battle damage, integral tanks should not be used near personnel compartments, inlet ducts, gun bays, or engines.

- The fire hazard of an integral tank can be reduced by filling the tank with a porous foam material, but some fuel volume is lost.

- The required volume of the fuel tanks is based upon the total required fuel, as calculated during the mission sizing. Densities for various fuels are provided in Table 10.5.

Propulsion and Fuel System Integration

Table 10.5 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

one gallon occupies 231 cubic in.

- A rule of thumb is to assume that 85% of the volume measured to the external skin surface is usable for integral wing tanks, and 92% is usable for integral fuselage tanks.
- If bladder tanks are used, the values become 77% for wing tanks and 83% for fuselage tanks.

Propulsion and Fuel System Integration

- KC-135R Stratotanker Refuels a KC-10 Extender Aircraft

Aircraft Design
AE 405

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Contents

Chapter 11

- Landing Gear and Subsystems

Landing Gear And Subsystems

- Of the many internal components that must be defined in an aircraft layout, the landing gear will usually cause the most trouble.

- Landing gear must be placed in the correct down position for landing, and must somehow retract into the aircraft without chopping up the structure.

Landing Gear Arrangements

- The common options for landing-gear arrangement are shown in Fig. 11.1.

Landing Gear And Subsystems

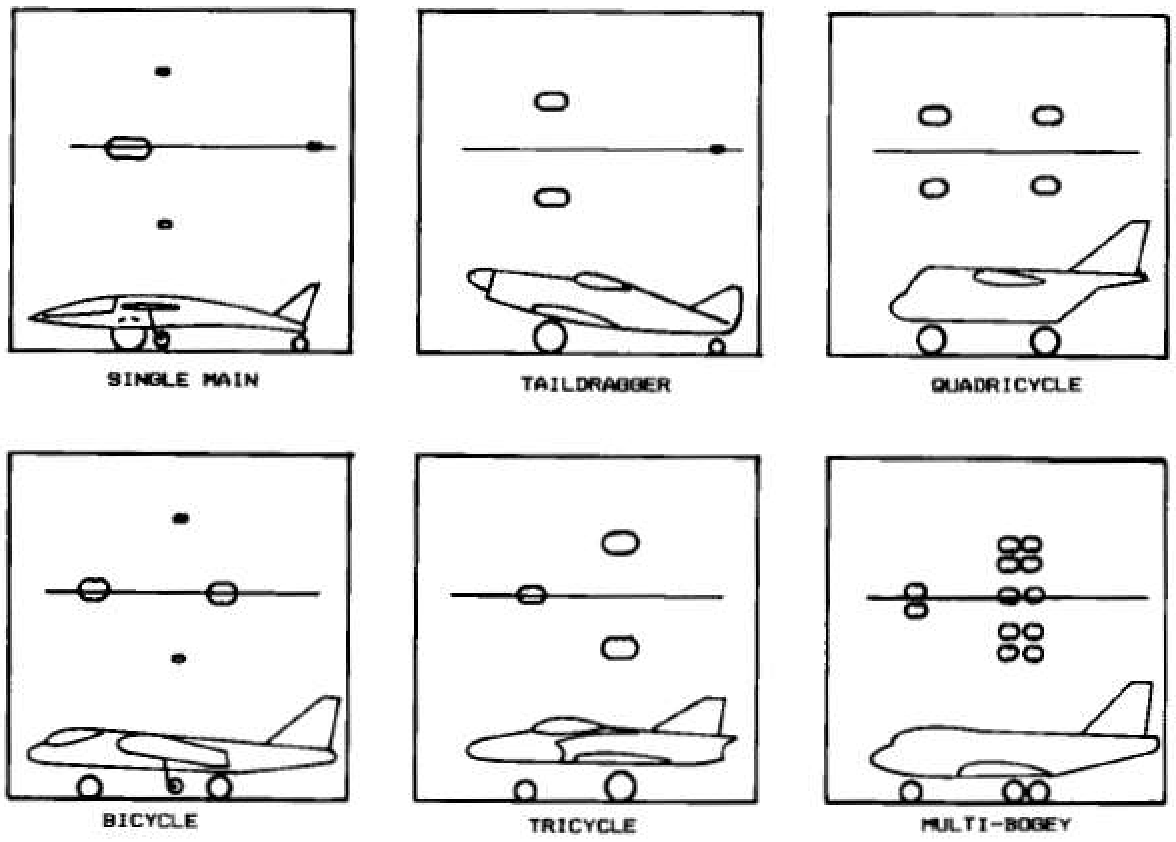
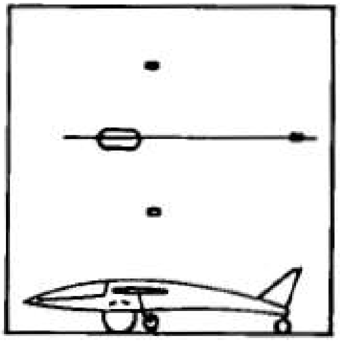
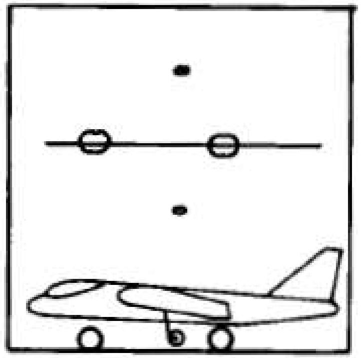


Fig. 11.1 Landing gear arrangements.

Landing Gear And Subsystems

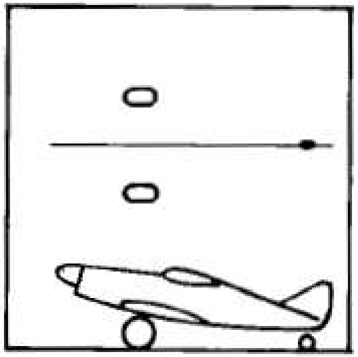


- The single main gear is used for many sailplanes because of its simplicity.
- The wheel can be forward of the center of gravity (c.g.), as shown here, or can be aft of the c.g. with a skid under the cockpit.
- "Bicycle" gear has two main wheels, fore and aft of the c.g.. with small "outrigger" wheels on the wings to prevent the aircraft from tipping sideways.
- The bicycle landing gear has the aft wheel so far behind the c.g. That the aircraft must takeoff and land in a flat attitude.
- Bicycle gear has been used mainly on aircraft with narrow fuselage and wide wing span such as the B-47.



BICYCLE

Landing Gear And Subsystems

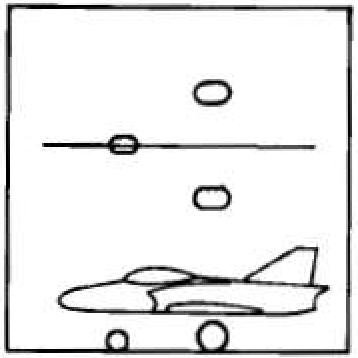


TAILDRAGGER

- The "taildragger" landing gear has two main wheels forward of the c.g. and an auxiliary wheel at the tail.
- Taildragger gear is also called "conventional" landing gear, because it was the most widely used arrangement.
- Taildragger gear provides more propeller clearance, has less drag and weight, and allows the wing to generate more lift for rough-field operation than does tricycle gear.

- However taildragger landing gear is inherently unstable in turnings on the ground (ground loop).

Landing Gear And Subsystems



TRICYCLE

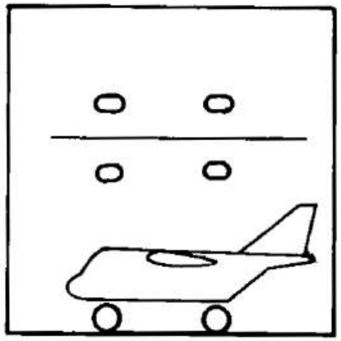
- The most commonly used arrangement today is the "tricycle" gear, with two main wheels aft of the c.g. and an auxiliary wheel forward of the c.g.

- With a tricycle landing gear, aircraft is stable on the ground and can be landed at a fairly large crab angle (i.e., nose not aligned with the runway).

- Also, tricycle landing gear improves forward visibility on the ground,

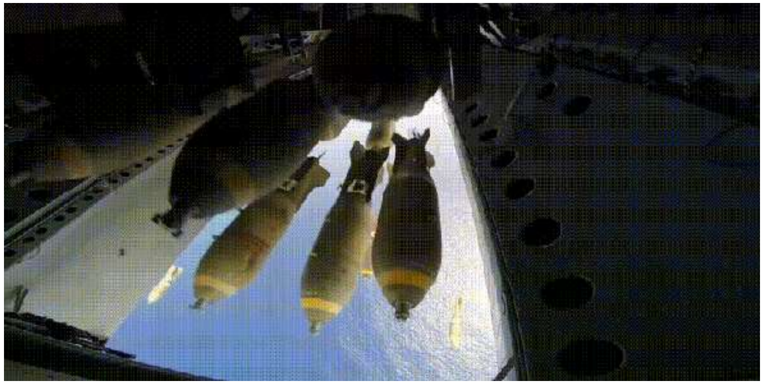
- and permits a flat cabin floor for passenger and cargo loading.

Landing Gear And Subsystems



QUADRICYCLE

- Quadricycle gear is much like bicycle gear but with wheels at the sides of the fuselage.
- Quadricycle gear also requires a flat takeoff and landing attitude.
- It is used on the B-52 and several cargo planes where it has the advantage of permitting a cargo floor very low to the ground.



Landing Gear And Subsystems

- The gear arrangements described above are also seen with two, four, or more wheels in place of the single wheels shown in Fig. 11.1.
- As aircraft weights become larger, the required wheel size for a single wheel capable of holding the aircraft's weight becomes too large.
- Then multiple wheels are used to share the load between reasonably-sized tires.
- Similarly, multiple main wheels (i.e., total of four or more) are desirable for safety.
- When multiple wheels are used in tandem, they are attached to a structural element called a "bogey/bogie," or "truck," which is attached to the end of the shock-absorber strut.

Landing Gear And Subsystems

- Typically an aircraft weighing under about 50,000 lb will use a single main wheel per strut.

- Between 50,000 and 150,000 lb, two wheels per strut are typical.

- Two wheels per strut are sometimes used for aircraft weighing up to about 250,000 lb.

- Between aircraft weights of about 200,000 and 400,000 lb the four-wheel bogey is usually employed.

- For aircraft over 400,000 lb, four bogeys, each with four or six wheels, spread the total aircraft load across the runway pavement.

Landing Gear And Subsystems

- Except for light aircraft and a few fighters, most aircraft use twin nose wheels to retain control in the event of a flat nose tire.
- Carrier-based aircraft must use twin nose-wheels at least 19 inches in diameter to straddle the catapult-launching mechanism.
- The massive C-5 employs four nose-wheels to spread the tire load, permitting operation off of relatively soft fields.

Landing Gear And Subsystems

- Guidelines for layout of a bicycle landing gear are shown in Fig. 11.2.
- The c.g. should be aft of the midpoint between the two wheels.

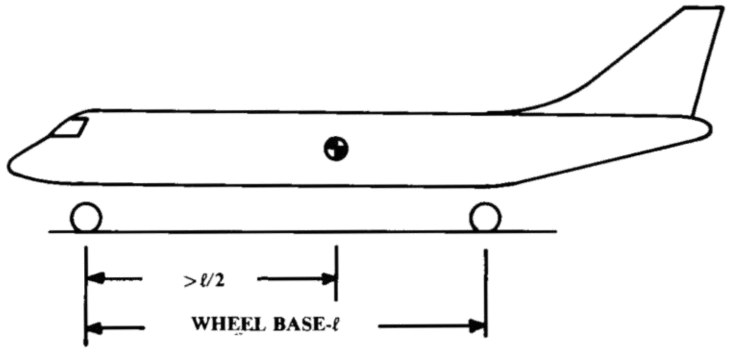


Fig. 11.2 Bicycle landing gear.

- The requirements for taildragger gear are shown in Fig. 11.3.
- The tail-down angle should be about 10-15 deg with the gear in the static position.
- The c.g. should fall between 16-25 deg back from vertical measured from the main wheel location.
- If the c.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.

Landing Gear And Subsystems

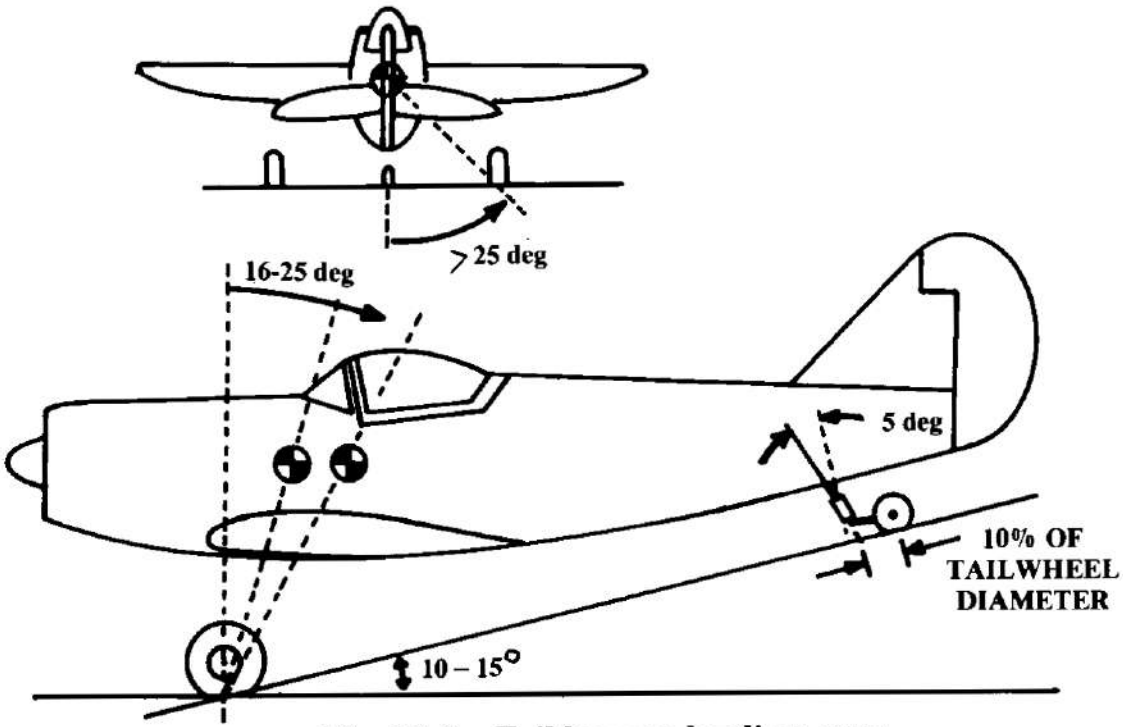
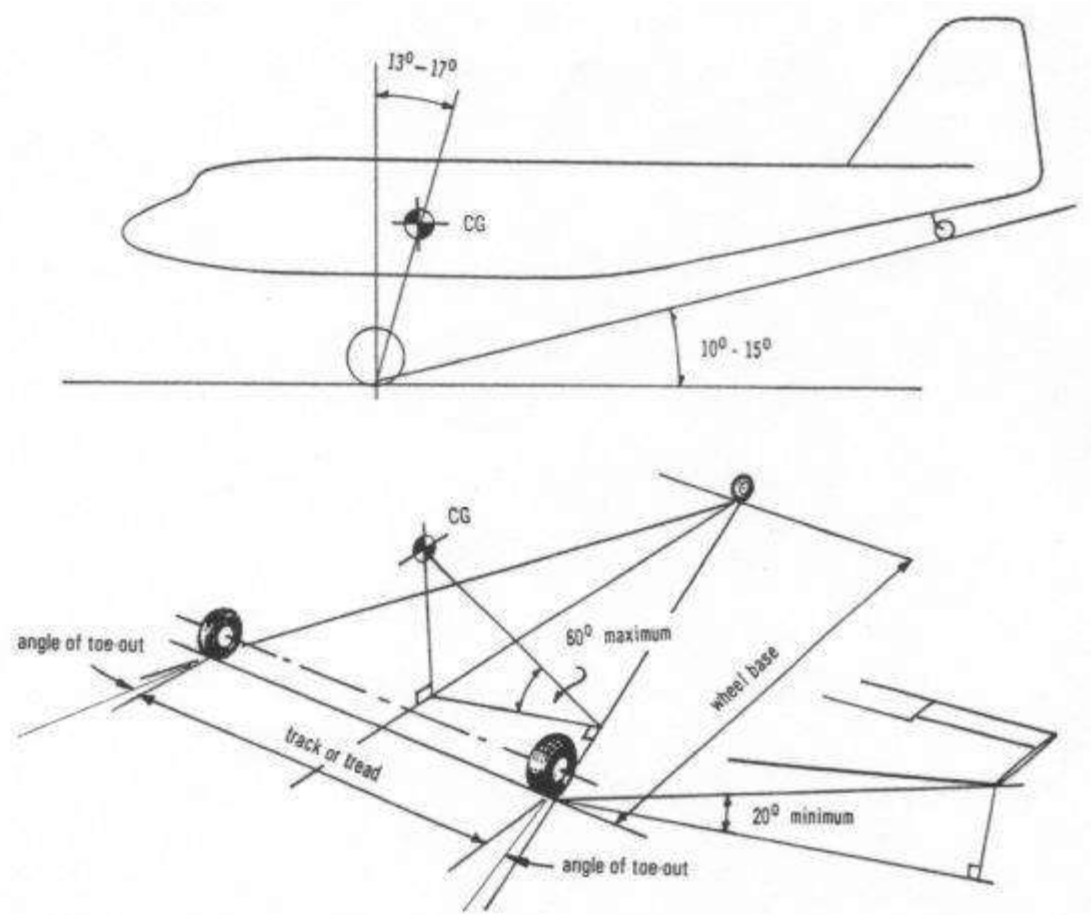


Fig. 11.3 Taildragger landing gear.

- To prevent the aircraft from overturning the main wheels should be laterally separated beyond a 25 deg angle off the c.g., as measured from the rear in a tail-down attitude.

Landing Gear And Subsystems



Landing Gear And Subsystems

- The layout of tricycle landing gear as shown in Fig. 11.4 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 90% of the maximum lift.

- This ranges from about 10-15 deg for most types of aircraft.

Landing Gear And Subsystems

- The "tipback 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 c.g. should be greater than the tipback angle or 15 deg.
- For carrier-based aircraft this angle frequently exceeds 25 deg, implying that the c.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.

Landing Gear And Subsystems

- If the nose wheel is carrying over 20% of the aircraft's weight, the main gear 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.

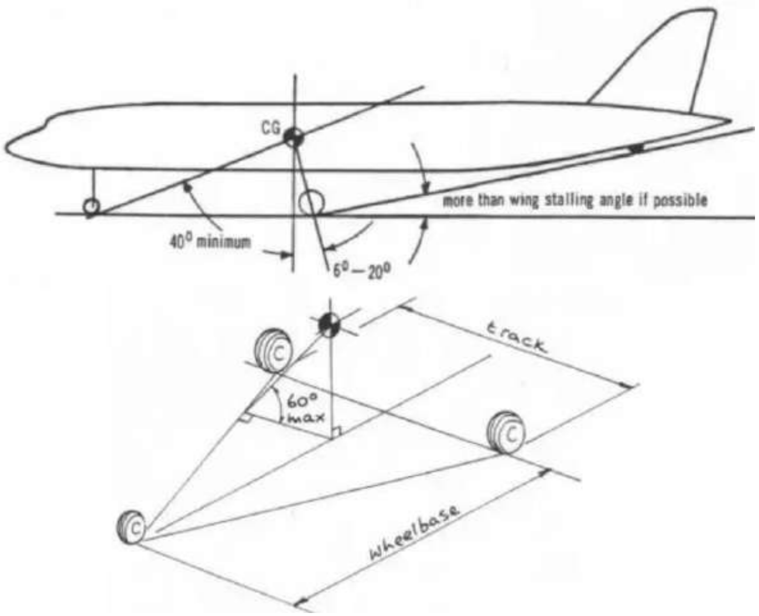
- The optimum range for the percent of the aircraft's weight which is carried by the nose wheel is about 8-15%.

Landing Gear And Subsystems

- The "overturn angle" is a measure of the aircraft's tendency to overturn when taxied around a sharp corner.

- This is measured as the angle from the c.g. to the main wheel, seen from the rear at a location where the main wheel is aligned with the nose wheel.

- For most aircraft this angle should be no greater than 63 deg (54 deg for carrier-based aircraft).



- Figure 11.4 also shows the desired strut-travel angle as about 7 deg.

- This optimal angle allows the tire to move upwards and backwards when a large bump is encountered, thus tending to smooth out the ride.

- However, any strut-travel angle from purely vertical to about 10 deg aft of vertical is acceptable.

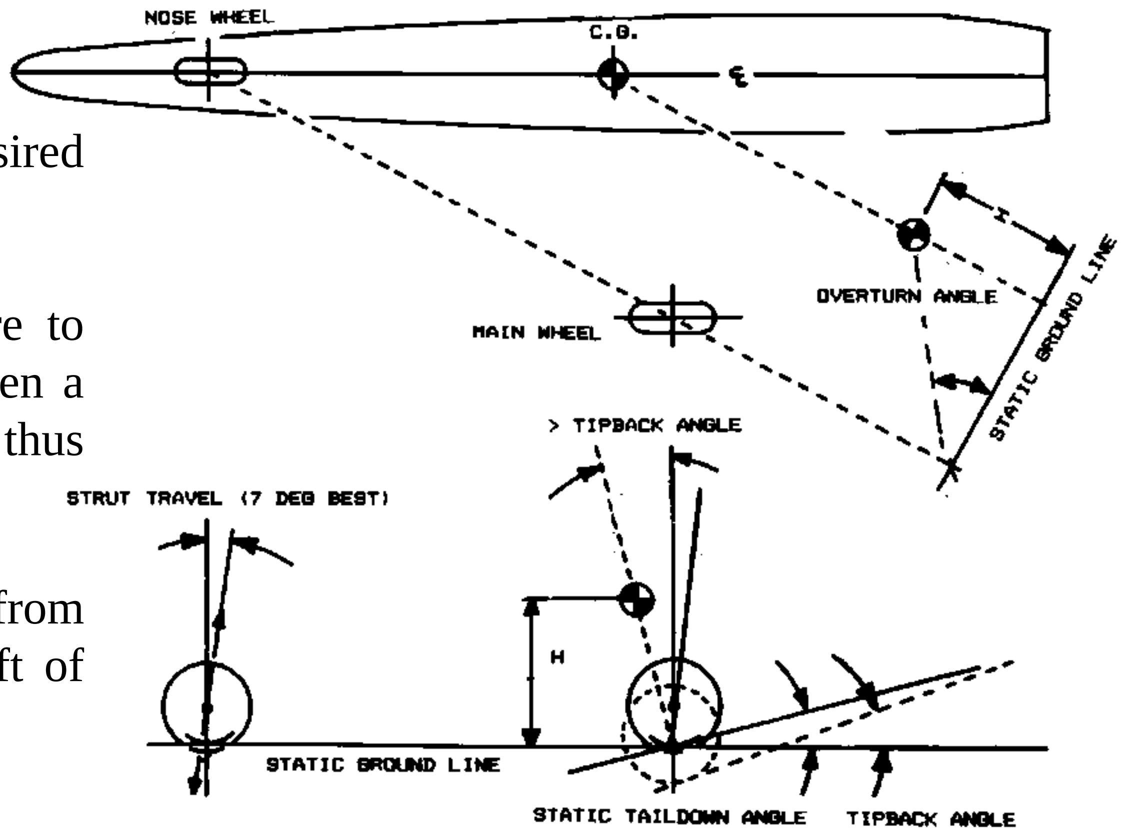


Fig. 11.4 Tricycle landing gear geometry.

Tire Sizing

- Strictly speaking, the "wheel" is the circular metal object upon which the rubber "tire" is mounted.
- The tires are sized to carry the weight of the aircraft.
- Typically the main tires carry about 90% of the total aircraft weight.
- Nose tires carry only about 10% of the static load but experience higher dynamic loads during landing.
- For early conceptual design, the engineer can copy the tire sizes of a similar design or use a statistical approach.
- Table 11.1 provides equations for rapidly estimating main tire sizes (assuming that the main tires carry about 90% of the aircraft weight).

Landing Gear And Subsystems

Table 11.1 Statistical tire sizing

Main wheels diameter or width (in.) $= A\ W_W^B$

	Diameter		Width	
	A	B	A	B
General aviation	1.51	0.349	0.7150	0.312
Business twin	2.69	0.251	1.170	0.216
Transport/bomber	1.63	0.315	0.1043	0.480
Jet fighter/trainer	1.59	0.302	0.0980	0.467

W_W = Weight on Wheel

- These calculated values for diameter and width should be increased about 30% if the aircraft is to operate from rough unpaved runways.

- Nose tires can be assumed to be about 60-100% the size of the main tires.

- The front tires of a bicycle or quadricycle-gear aircraft are usually the same size as the main tires.

Landing Gear And Subsystems

- Taildragger aft tires are about a quarter to a third the size of the main tires.

- For a finished design layout, the actual tires to be used must be selected from a manufacturer's catalog.

- This selection is usually based upon the smallest tire rated to carry the calculated static and dynamic loads.

- Calculation of the static loads on the tires is illustrated in Fig. 11.5 and Eqs. (11.1-11.3).

- The additional dynamic load on the nose tires under a 10 fps per second braking deceleration is given in Eq. (11.4).

- Note that these loads are divided by the total number of main or nose tires to get the load per tire (wheel) "W_w," which is used for tire selection.

Landing Gear And Subsystems

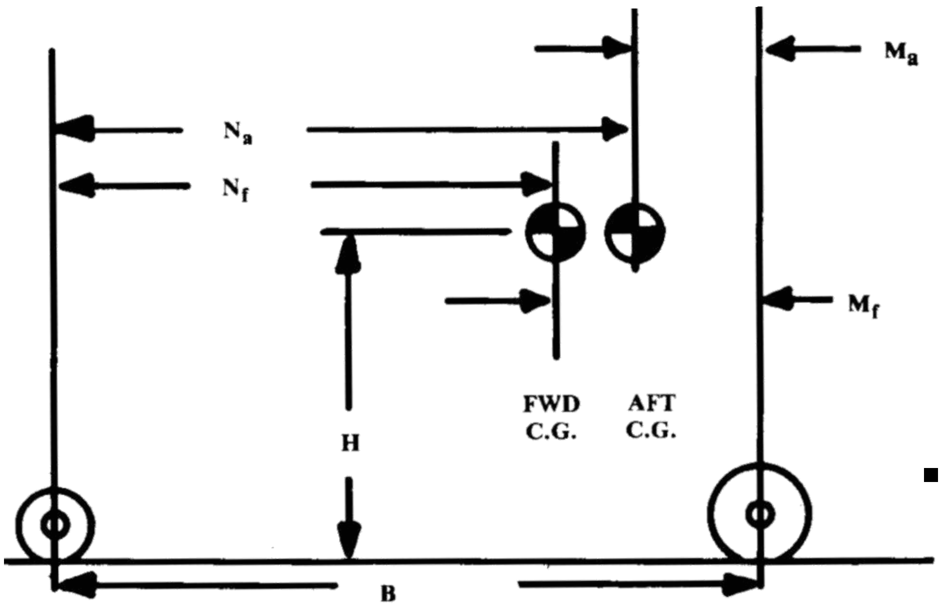


Fig. 11.5 Wheel load geometry.

$$(\text{Max Static Load}) = W\frac{N_a}{B} \tag{11.1}$$

$$(\text{Max Static Load})_{\text{nose}} = W\frac{M_f}{B} \tag{11.2}$$

$$(\text{Min Static Load})_{\text{nose}} = W\frac{M_a}{B} \tag{11.3}$$

$$(\text{Dynamic Braking Load})_{\text{nose}} = \frac{10HW}{gB} \tag{11.4}$$

- Equation (11.4) assumes a braking coefficient(μ) of 0.3, which is typical for hard runways.
- This results in a deceleration of 10 ft/s^2.
- To insure that the nose gear is not carrying too much or too little of the load, the parameter *(M_a / B)* should be greater than 0.05,
- and the parameter *(M_f / B)* should be less than 0.20 (0.08 and 0.15 preferred).

Landing Gear And Subsystems

- If an airplane is to be operated under FAR 25 provisions, a 7% margin should be added to all calculated wheel loads.

- Also, it is common to add an additional 25% to the loads to allow for later growth of the aircraft design.

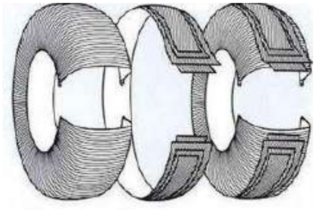
- Table 11.2 summarizes design data for typical tires, including

  - maximum load ratings,
  - inflation pressures at that load,
  - maximum landing speed,
  - tire width, diameter,
  - and "rolling radius" (i.e., radius when under load, typically two-thirds of tire radius).

Landing Gear And Subsystems

Table 11.2 Tire data

Size	Speed, mph	Max load. lb	Infl, psi	Max width, in.	Max diam, in.	Rolling radius	Wheel diam	Number of plies
Type III								
5.00-4	120	1,200	55	5.05	13.25	5.2	4.0	6
5.00-4	120	2,200	95	5.05	13.25	5.2	4.0	12
7.00-8	120	2,400	46	7.30	20.85	8.3	8.0	6
8.50-10	120	3,250	41	9.05	26.30	10.4	10.0	6
8.50-10	120	4,400	55	8.70	25.65	10.2	10.0	8
9.50-16	160	9,250	90	9.70	33.35	13.9	16.0	10
12.50-16	160	12,800	75	12.75	38.45	15.6	16.0	12
20.00-20	174kt	46,500	125	20.10	56.00	22.1	20.0	26
Type VII								
16 × 4.4	210	1,100	55	4.45	16.00	6.9	8.0	4
18 × 4.4	174kt	2,100	100	4.45	17.90	7.9	10.0	6
18 × 4.4	217kt	4,350	225	4.45	17.90	7.9	10.0	12
24 × 5.5	174kt	11,500	355	5.75	24.15	10.6	14.0	16
30 × 7.7	230	16,500	270	7.85	29.40	12.7	16.0	18
36 × 11	217kt	26,000	235	11.50	35.10	14.7	16.0	24
40 × 14	174kt	33,500	200	14.00	39.80	16.5	16.0	28
46 × 16	225	48,000	245	16.00	45.25	19.0	20.0	32
50 × 18	225	41,770	155	17.50	49.50	20.4	20.0	26
Three Part Name								
18 × 4.25-10	210	2,300	100	4.70	18.25	7.9	10.0	6
21 × 7.25-10	210	5,150	135	7.20	21.25	9.0	10.0	10
28 × 9.00-12	156kt	16,650	235	8.85	27.60	11.6	12.0	22
37 × 14.0-14	225	25,000	160	14.0	37.0	15.1	14.0	24
47 × 18-18	195kt	43,700	175	17.9	46.9	19.2	18.0	30
52 × 20.5-23	235	63,700	195	20.5	52.0	21.3	23.0	30



Plies are layers of fabric (Kevlar, nylon, polyester)

Radial

- A Type III tire, used for most piston-engined aircraft, has a wide tread and low internal pressure.
- The identifying numbers for a Type III tire, such as 8.50-10, refer to the approximate tire width (8.2-8.7 in.) and wheel rim diameter (10 in.).
- The tire outside diameter must be obtained from a tire book

Landing Gear And Subsystems

- Type VII tires, used by most jet aircraft, operate under higher internal pressures, which reduces their size.
- Also, the Type VII tires are designed for higher landing speeds.
- They are identified by their approximate external dimensions.
- For example, an 18 x 5.7 tire has an outside diameter of 17.25-17.8 in. and a width of 5.25-5.6 in.
- A tire supports a load almost entirely by its internal pressure.
- The load carrying ability of the sidewalls and tread can be ignored.

Landing Gear And Subsystems

- The weight carried by the tire (W_w) is simply the inflation pressure (P) times the tire's contact area with the pavement (A_p, also called "footprint area"),

- as shown in Fig. 11.6 and defined in Eq. (11.5).

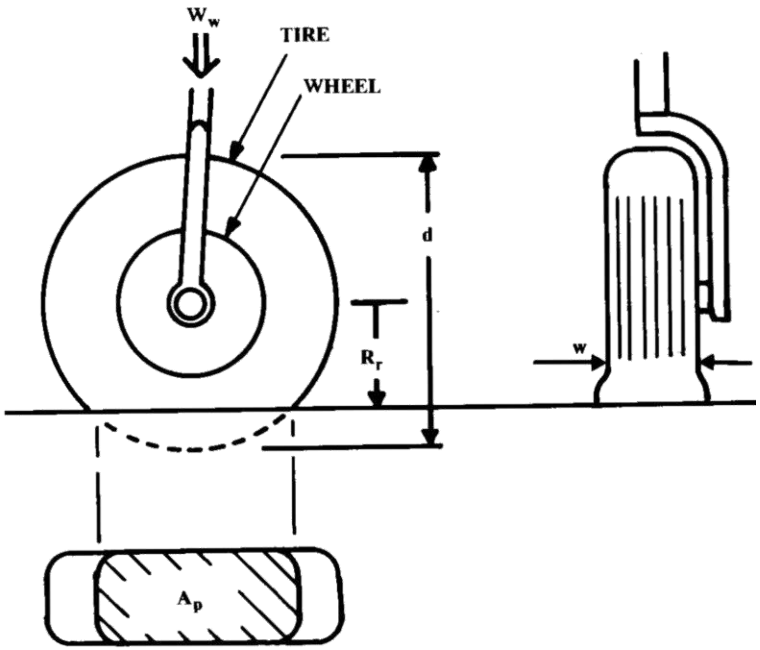


Fig. 11.6 Tire contact area.

$$W_W = PA_p \tag{11.5}$$

$$A_p = 2.3\sqrt{wd}\left(\frac{d}{2} - R_r\right) \tag{11.6}$$

- As a rough estimate, tires should be sized to keep internal pressures below the values in Table 11.3 for the desired application.

Landing Gear And Subsystems

Table 11.3 Recommended tire pressures

Surface	Maximum pressure, psi
Aircraft carrier	200 +
Major military airfield	200
Major civil airfield	120
Tarmac runway, good foundation	70–90
Tarmac runway, poor foundation	50–70
Temporary metal runway	50–70
Dry grass on hard soil	45–60
Wet grass on soft soil	30–45
Hard packed sand	40–60
Soft sand	25–35

- Sometimes the diameter of the tires is set by the braking requirements.

- The brakes must absorb the kinetic energy of the aircraft at touchdown, aerodynamic drag and thrust reversing can be ignored.

$$KE_{\text{braking}} = \frac{1}{2} \frac{W_{\text{landing}}}{g} V_{\text{stall}}^2 \qquad (11.7)$$

- Energy must be divided by the number of wheels with brakes to get the kinetic energy that must be absorbed by each brake.
- The landing weight should be approximated as 80-100% of the takeoff weight.
- Figure 11. 7 provides a statistical estimate of the required wheel rim diameter to provide a brake that can absorb a given amount of kinetic energy.

Landing Gear And Subsystems

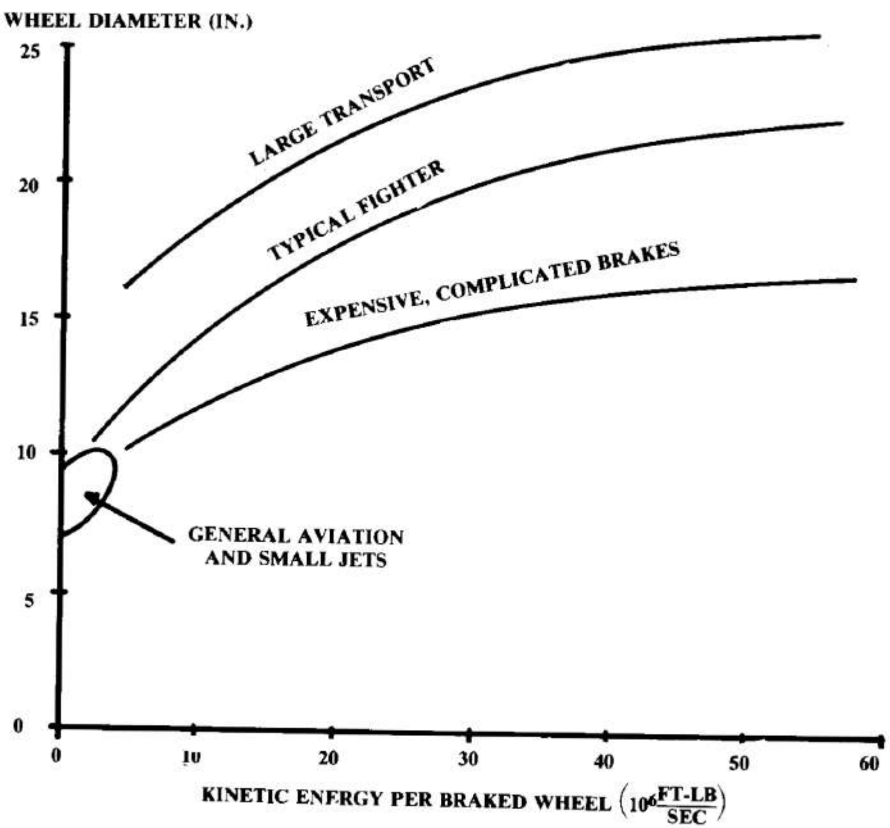


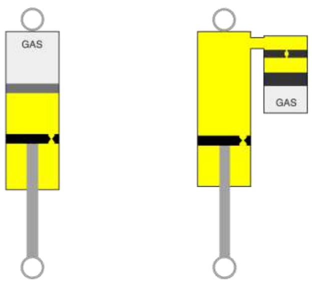
Fig. 11.7 Wheel diameter for braking.

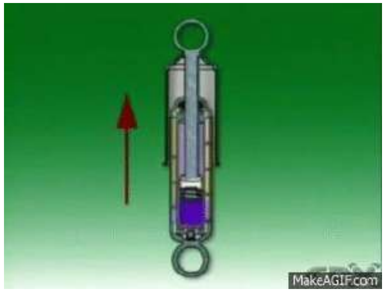
Landing Gear And Subsystems

Shock Absorbers

- 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 Fig. 11.8.





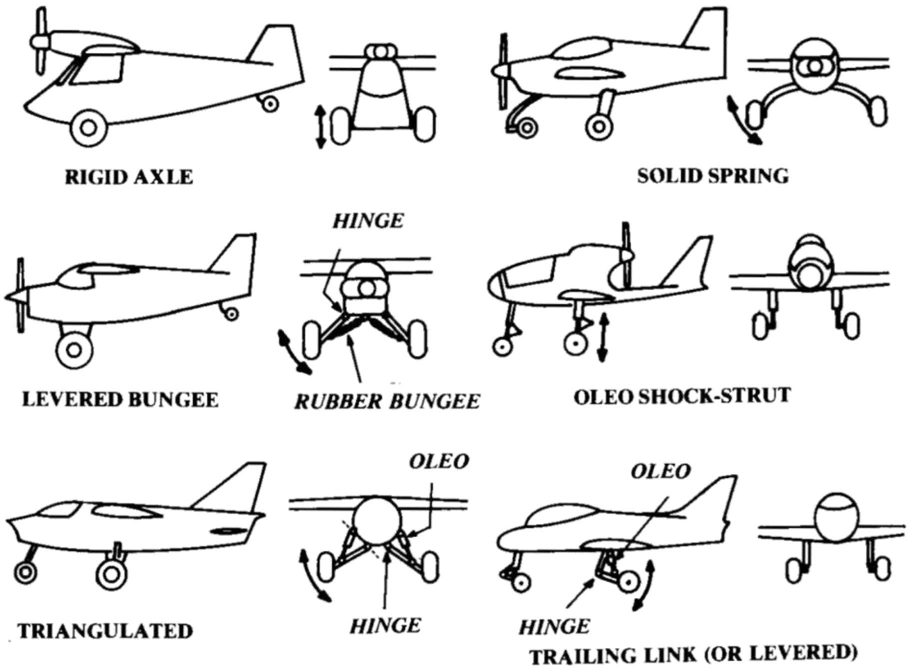
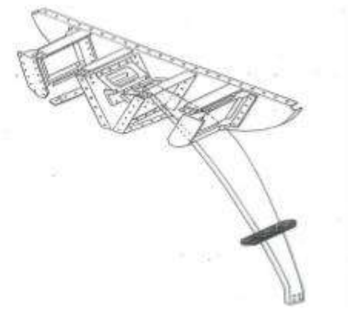


Fig. 11.8 Gear/shock arrangments.

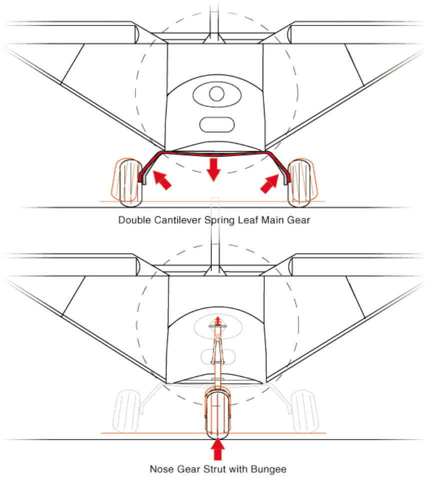
Landing Gear And Subsystems

- The tires themselves provide some shock-absorbing ability by deflecting when a bump is encountered.
- The solid spring gear is used in many general-aviation aircraft (especially Cessna products). The solid spring is as simple as possible, but is slightly heavier than other types of gear.
- The solid spring has no damping other than scrubbing action.
- The aircraft thus tends to bounce a lot.



Landing Gear And Subsystems

- The levered bungee-chord gear was very common in early light aircraft such as the Piper Cub.
- The gear leg is pivoted at the fuselage.
- Rubber bungee chords underneath the gear are stretched as the gear deflects upward and outwards.



Landing Gear And Subsystems

- The oleo pneumatic shock strut, or "oleo," is the most common type of shock - absorbing gear in use today (Fig. 11.9).

- The oleo combines a spring effect using compressed air with a damping effect using a piston which forces oil through small hole (orifice).

- For maximum efficiency, many oleos have a mechanism for varying the size of the orifice as the oleo compresses ("metered orifice").

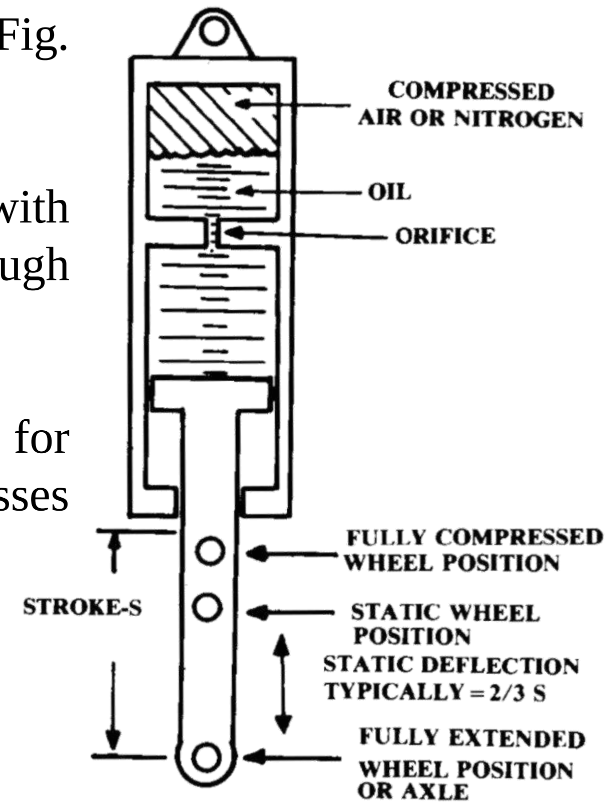
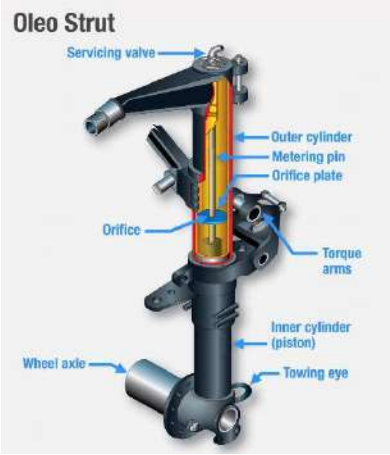


Fig. 11.9 Oleo shock absorber (most simple type).



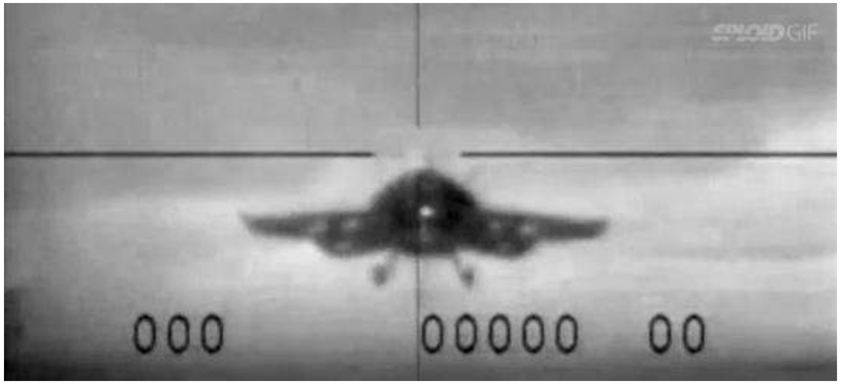
Landing Gear And Subsystems

- 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 can be 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.

Landing Gear And Subsystems

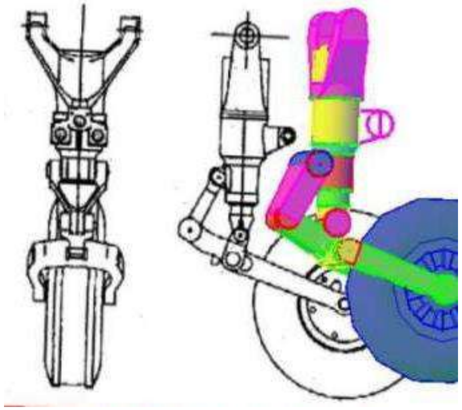
- The trailing-link, or levered, gear resembles the triangulated gear, but with the solid gear leg running aft rather than laterally.

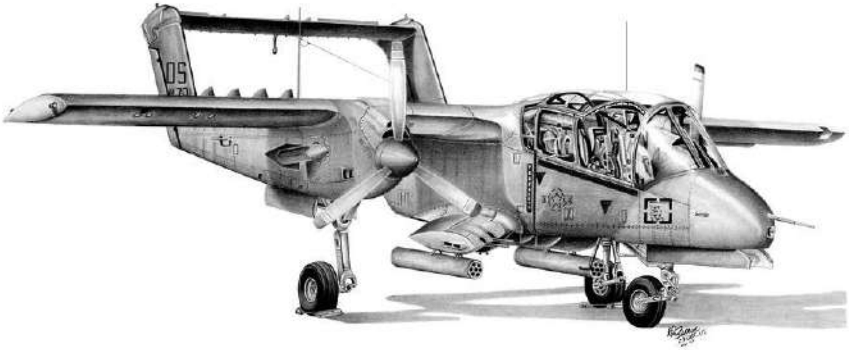
- This gear is common for carrier-based aircraft such as the F-18 where it provides the large amounts of gear travel required for carrier landings.



Landing Gear And Subsystems

- The levered gear allows the wheel to travel aft as it deflects.
- This is very desirable for operations on rough fields.
- If a large rock or other obstacle is encountered, the aft motion of the wheel gives it more time to ride over the obstacle.
- For this reason the levered gear was used on the North American Rockwell OV-10.





Landing Gear And Subsystems

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 a rough rule-of-thumb, the stroke in inches approximately equals the vertical velocity at touchdown in (ft/s).

- The vertical velocity (or "sink speed") at touchdown is established in various specifications for different types of aircraft.
- Most aircraft require 10 ft/s vertical velocity capability.
- This is substantially above the 4-5 ft/s that most passengers would consider a "bad" landing.

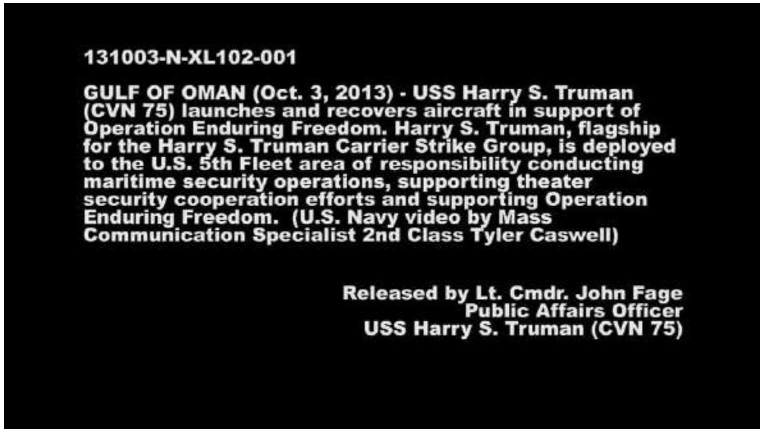
Landing Gear And Subsystems

- While most Air Force aircraft require only 10 ft/s, Air Force trainer aircraft require 13 ft/s.

- Due to steeper descent angles, it is suggested 15 ft/s for short takeoff and landing ("STOL") aircraft.

- Carrier-based naval aircraft require 20 or more ft/s vertical velocity, which is much like a controlled crash!

- This is the reason that carrier-based aircraft tend to use triangulated or levered gear, which provide longer strokes than shock-strut gear.



Landing Gear And Subsystems

- The vertical energy of the aircraft, which must be absorbed during the landing, is defined in Eq. (11.8).

- This kinetic energy is absorbed by the work of deflecting the shock absorber and tire.

$$KE_{\text{vertical}} = \left(\frac{1}{2}\right)\left(\frac{W_{\text{landing}}}{g}\right) V^2_{\text{vertical}} \tag{11.8}$$

where

W = total aircraft weight
g = 32.2 ft/s^2

- 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, as provided in Table 11.4.

Table 11.4 Shock absorber efficiency

Type	Efficiency η
Steel leaf spring	0.50
Steel coil spring	0.62
Air spring	0.45
Rubber block	0.60
Rubber bungee	0.58
Oleopneumatic	
– Fixed orifice	0.65–0.80
– Metered orifice	0.75–0.90
Tire	0.47

- The actual energy absorbed by deflection is defined in Eq. (11.9).

$$KE_{\text{absorbed}} = \eta L S \tag{11.9}$$

where

η = shock-absorbing efficiency
L = average total load during deflection (not lift!)
S = stroke

Landing Gear And Subsystems

- For tires it is assumed that the tire deflects only to its rolling radius, so the "stroke" *(ST)* of a tire is equal to half the diameter minus the rolling radius.

- Combining Eqs. (11.8) and (11.9) and assuming that the shock absorber and tire both deflect to absorb the vertical kinetic energy yields:

$$\left(\frac{1}{2}\right)\left(\frac{W_{\text{landing}}}{g}\right)V_{\text{vertical}}^2 = (\eta L S)_{\substack{\text{shock}\\\text{absorber}}} + (\eta_T L S_T)_{\text{tire}} \tag{11.10}$$

- The shock absorbers and tires act together to decelerate the aircraft from the landing vertical velocity to zero vertical velocity.

- The vertical deceleration rate is called the "gear load factor" (N_{gear}).

Landing Gear And Subsystems

- Gear load factor is the average total load summed for all of the shock absorbers divided by the landing weight, and is assumed to be constant during touchdown.

- N_{gear} is defined in Eq. (11.11) and typically equals three.

$$N_{gear} = L/W_{landing} \tag{11.11}$$

- The gear load factor determines how much load the gear passes to the airframe,

- which affects the airframe structural weight as well as crew and passenger comfort during the landing.

- Table 11.5 provides typical gear load factors permitted for various types of aircraft:

Landing Gear And Subsystems

Table 11.5 Gear load factors

Aircraft type	N_{gear}
Large bomber	2.0–3
Commercial	2.7–3
General aviation	3
Air Force fighter	3.0–4
Navy fighter	5.0–6

- Substituting Eq. (11.11) into Eq. (11.10) yields Eq. (11.12) for shock absorber stroke.

$$S = \frac{V_{\text{vertical}}^2}{2g\,\eta N_{\text{gear}}} - \frac{\eta_T}{\eta} S_T \tag{11.12}$$

- Note that the equation for stroke does not include any terms containing the aircraft weight.

- For the same required landing vertical velocity and gear load-factor, an airliner and an ultralight would require the same stroke!

- The stroke calculated by Eq. (11.12) should be increased by about 1 in. As a safety margin.

- Also, a stroke of 8 in. is usually considered a minimum and at least 10-12 in. is desirable for most aircraft.

Landing Gear And Subsystems

- Nose wheel stroke is generally set equal to or slightly larger than mainwheel stroke to provide a smooth ride while taxiing.

- Note that the stroke defined by Eq. (11.12) is a vertical distance.

- If a type of gear is used which produces some lateral motion of the wheel, the total distance the wheel moves must provide the required stroke in a vertical direction.

- Equation (11.12) defines the total stroke required.

- If a levered or triangulated gear is used, the required stroke of the oleo, bungee, or rubber block is reduced.

Oleo Sizing

- Oleo diameter is determined by the load carried by the oleo.

- The main wheel oleo load is the static load found from Eq. (11.1) divided by the number of main-wheel oleos (usually two).

- The nose-wheel oleo load is the sum of the static and dynamic loads [Eqs. (11.2) and (11.4)].

- The oleo carries its load by the internal pressure of compressed air, applied across a piston.

- Typically an oleo has an internal pressure ("P") of 1800 psi.

- Internal diameter is determined from the relationship which states that force equals pressure times area.

- The external diameter is typically 30% greater than the piston diameter, so the external oleo diameter can be approximated by Eq. (11.13).

$$D_{\text{oleo}} = 1.3\sqrt{\frac{4L_{\text{oleo}}}{P\pi}} \cong 0.04\sqrt{L_{\text{oleo}}} \qquad (11.13)$$

where L_{oleo} = load on the oleo in pounds.

Solid-Spring Gear Sizing

- Figure 11.10 illustrates the deflection geometry for a solid-spring gear leg.

- The total stroke as determined by Eq. (11.12) is the vertical component of the deflection of the gear leg.

Landing Gear And Subsystems

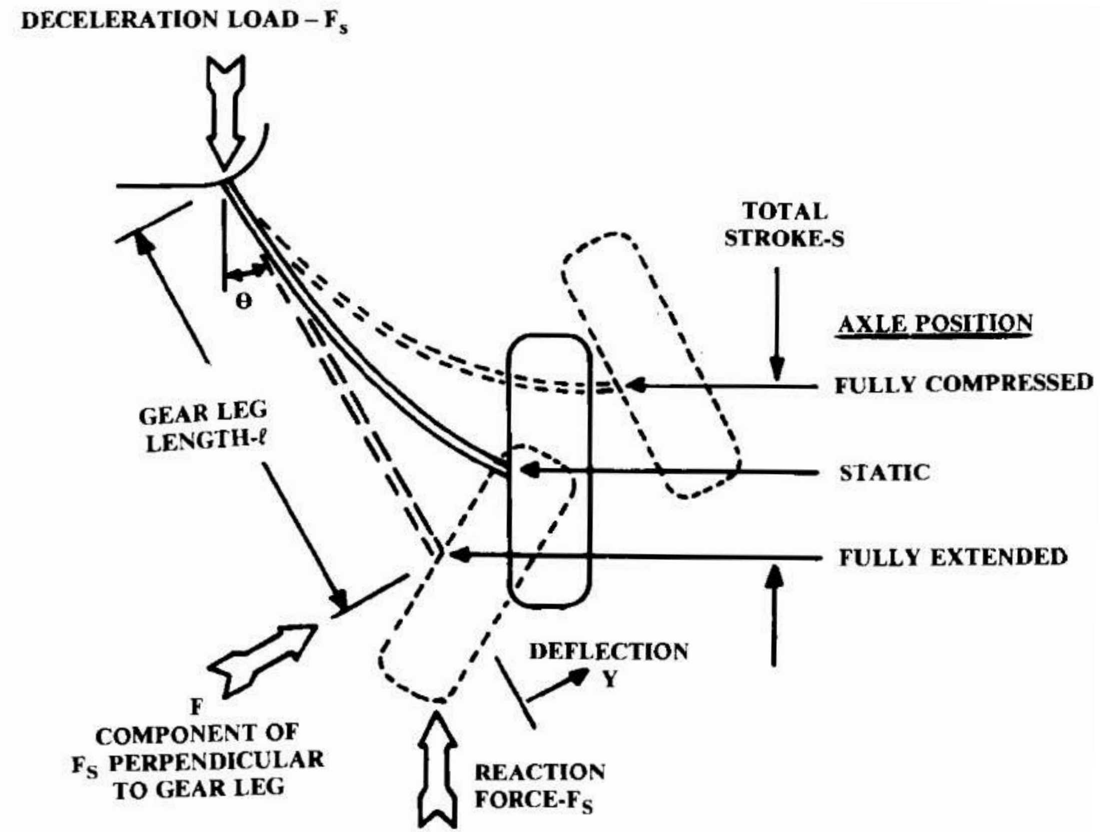


Fig. 11.10 Solid spring gear deflection.

Landing Gear And Subsystems

- The load on the gear leg in the fully deflected position is the force required to produce the gear load factor, N_{gear}.
- Assuming there are two gear legs yields Eq. (11.14).
- The component of the load on the gear that is perpendicular to the gear leg is defined by Eq. (11.15).

$$F_s = WN_{gear}/2 \qquad (11.14)$$

$$F = F_s(\sin\theta) \qquad (11.15)$$

- The deflection "y" perpendicular to the gear leg is related to the stroke by Eq. (11.16), and is calculated by the structural bending-beam equation, Eq. (11.17).

$$S = y\,\sin\theta \qquad (11.16)$$

$$y = Fl^3/3EI \qquad (11.17)$$

- Substituting Eqs. (11.14-11.16) into Eq. (11.17) yields the equation for the stroke *"S"* of a solid-spring gear leg:

$$S = F_s (\sin^2\theta) \frac{l^3}{3EI} \tag{11.18}$$

where

I = beam's moment of inertia
E = material modulus of elasticity (psi)
(approximately 30 million for steel; 10 million for aluminum)

- For a rectangular-cross-section gear leg,

$$I = \frac{wt^3}{12}$$

Castoring-wheel Geometry

- For ground steering, a nose wheel or tailwheel must be capable of being castored (turned).

- The castoring can introduce static and dynamic stability problems causing "wheel shimmy," a rapid side-to-side motion of the wheel that can tear the landing gear off the airplane.

Landing Gear And Subsystems

- Prevention of shimmy is accomplished by selection of the "rake angle« and "trail," as shown in Fig. 11.11.

- In some cases a frictional shimmy damper is also used to prevent shimmy.

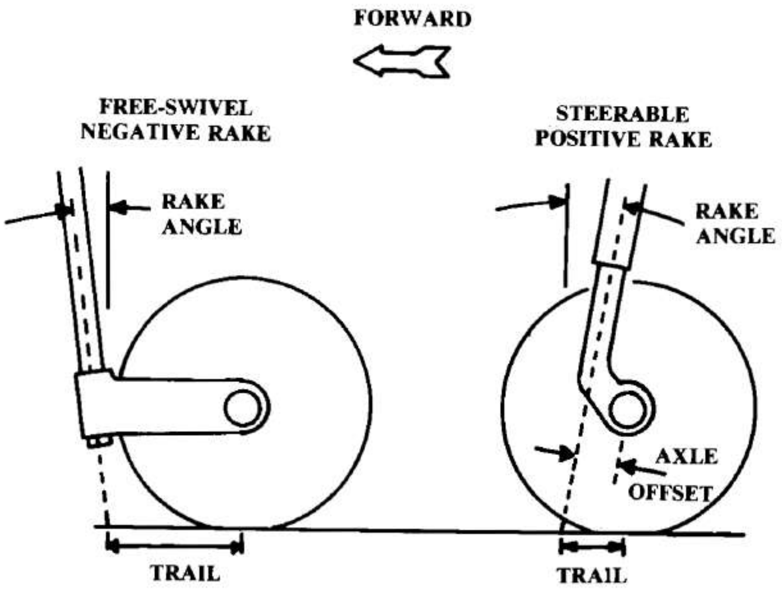


Fig. 11.11 Castoring wheel geometry.

Landing Gear And Subsystems

- If the castoring wheel is free to swivel, shimmy can be prevented by using a small negative angle of rake (4-6 deg),

- and trail equal to 0.2-1.2 times the tire radius (0.2 is typical for tailwheels).

- For a large aircraft with a steerable nose-wheel, the rake angle should be about 7 deg positive, and the trail should be at least 16% of the tire radius.

- For smaller aircraft, rake angles up to 15 deg and trail of about 20% are used.

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.
- Figure 11.12 shows the options for main-landing-gear retracted positions.
- Locating the gear in the wing, in the fuselage, or in the wing-fuselage junction produces the smallest aerodynamic penalty but tends to chop up the structure.

Landing Gear And Subsystems

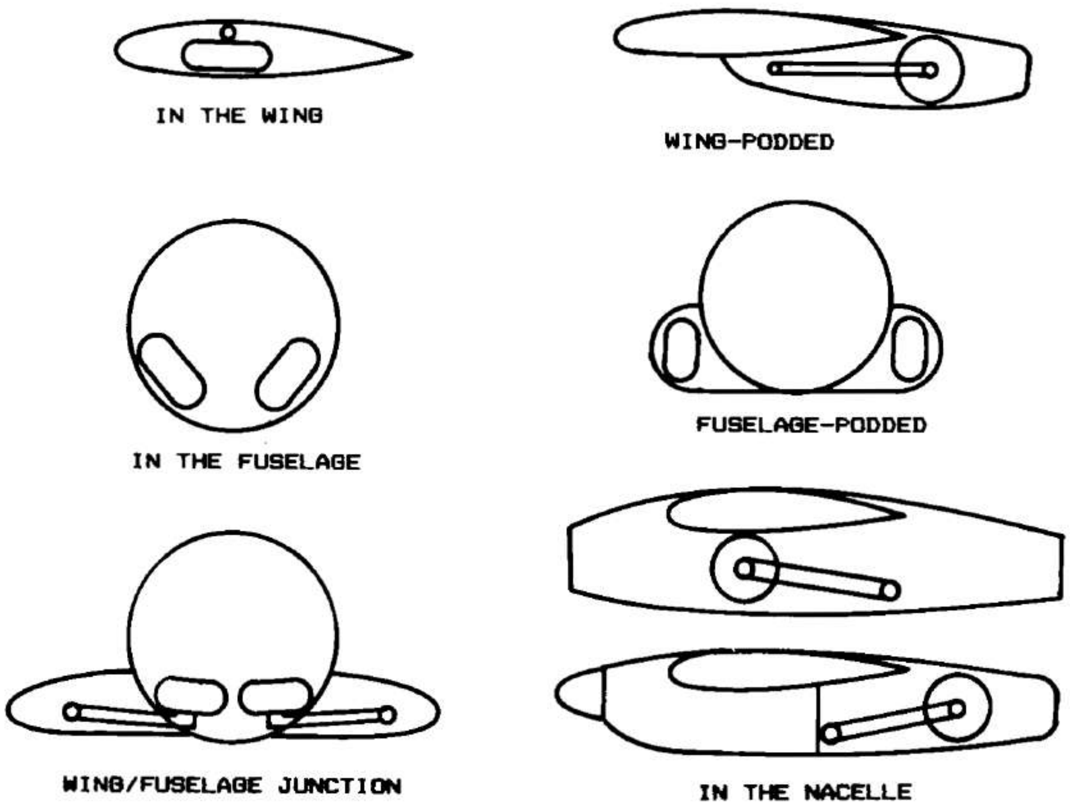
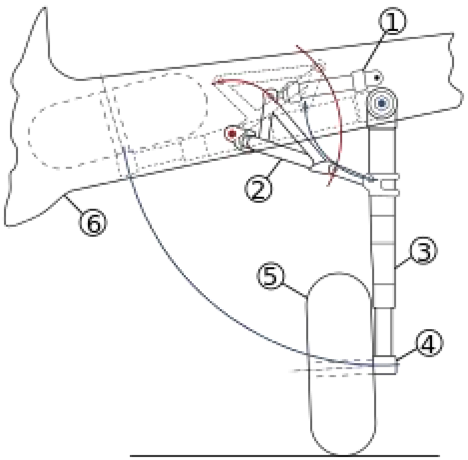
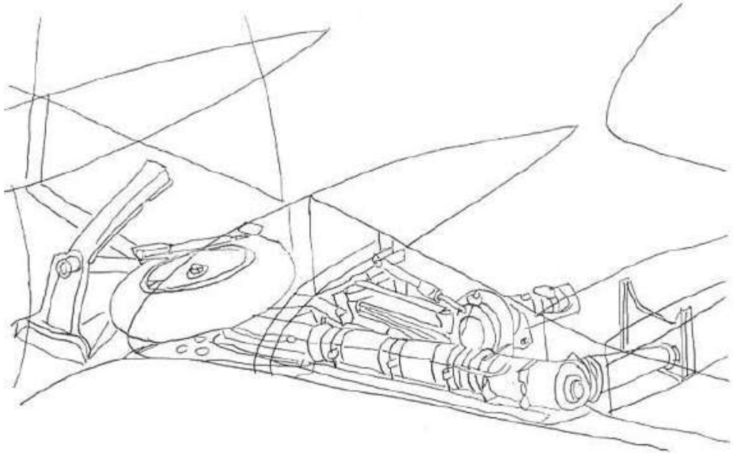


Fig. 11.12 "A home for the gear."

Landing Gear And Subsystems

- Gear in the wing reduces the size of the wing box, which increases weight and may reduce fuel volume.
- Gear in the fuselage or wing-fuselage junction may interfere with the longerons.
- However, the aerodynamic benefits of these arrangements outweigh the drawbacks for higher-speed aircraft.



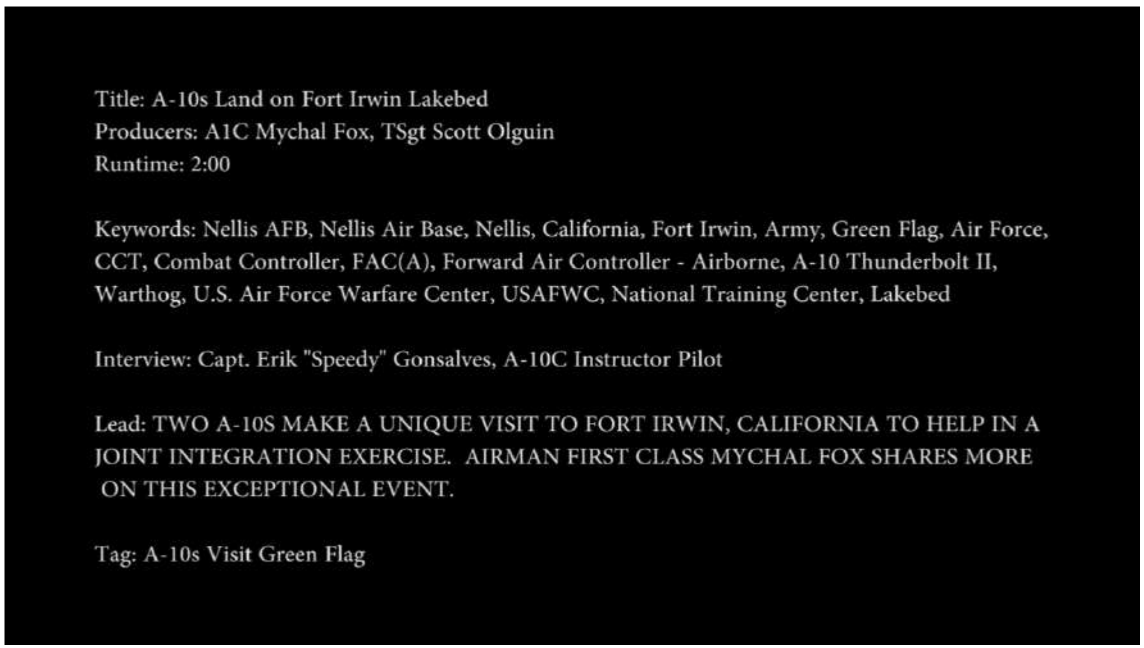


Landing Gear And Subsystems

- Virtually all civilian jet transports retract the gear into the wing-fuselage junction.
- Most low-wing fighters retract the gear into the wing or wing-fuselage junction, while mid- and high-wing fighters retract the gear into the fuselage.
- Some aircraft retract the gear into the nacelles or a separate gear pod.
- This reduces weight significantly because the wing and fuselage structure is uninterrupted.

Landing Gear And Subsystems

- The wing-podded arrangement is rarely seen in Western aircraft designs (A-10), but is used in Soviet designs even for jet transports and bombers.



Landing Gear And Subsystems

- The fuselage-podded arrangement is common for high-winged military transports where the fuselage must remain open for cargo.
- The drag penalty of the pods can be substantial.
- Retraction of the gear into the nacelles behind the engine is typical for propeller-driven aircraft.

Landing Gear And Subsystems

- Most mechanisms for landing-gear retraction are based upon the "four bar linkage."

- This uses three members (the fourth bar being the aircraft structure) connected by pivots.

- The four-bar linkage provides a simple and light weight gear because the loads pass through rigid members and simple pivots.

- Several variations of four-bar linkage landing gear are shown in Fig. 11.13.

Landing Gear And Subsystems

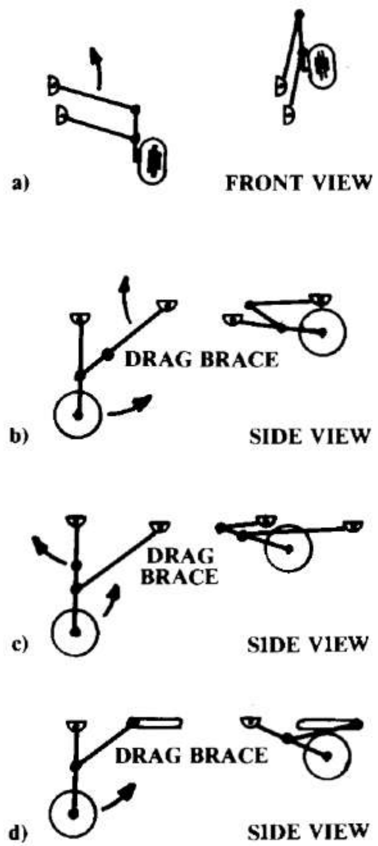
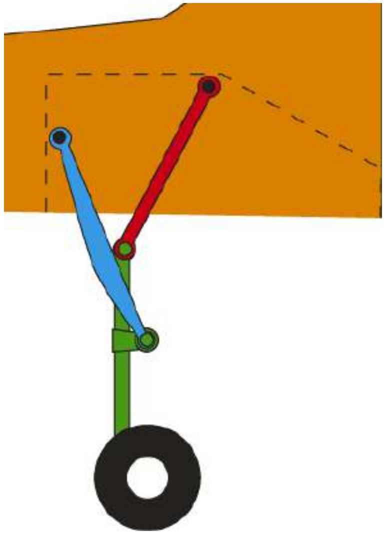


Fig. 11.13 Landing gear retraction.

- The oldest form of four-bar linkage for landing gear retraction is shown in front view (Fig. 11.13a), which in turn attach to the fuselage.
- The gear retracts by pivoting the arms upward and inward.
- This was widely used during the 1930's, and is seen today in modified form on the MiG-23.

Landing Gear And Subsystems



- Figure 11.13b shows the typical retraction arrangement for nose wheels.
- The diagonal arm is called a "drag brace" because it withstands the aerodynamic loads (as well as braking loads).
- The drag brace breaks at the middle for retraction.
- The drag brace may be behind the wheel with the gear retracting rearward or it may be in front of the wheel with the gear retracting forward.
- The latter is preferable because the air loads will blow the gear down in the event of a hydraulic failure.

Landing Gear And Subsystems

- In Fig. 11.13c the vertical gear member breaks for retraction instead of the drag brace.

- This has the advantage of reducing the length of the retracted
- gear, but is usually heavier.

- This gear was used on the DC-3 and several World War II bombers.

Landing Gear And Subsystems

- Figure 11.13d shows the use of a sliding pivot rather than a four-bar linkage.
- The sliding motion is frequently provided by a wormscrew mechanism that is rotated to retract the gear.
- This is usually heavier than a four bar linkage because the entire length of the wormscrew must be strong enough for the landing-gear loads.
- However, this gear is very simple and compact.

Landing Gear And Subsystems

- The retraction concepts as shown in Fig. 11.13 are for nose or main wheels that retract in a fore or aft direction.
- However, the same basic concepts can be used for main wheels that retract inwards or outwards.
- These illustrations then become front views, and the tires are redrawn accordingly.
- The gear members labeled "drag brace" become "sway/ move slowly side by side braces" because they provide lateral support for the gear in this arrangement.
- There are dozens of additional geometries for gear retraction based upon the four-bar linkage and other concepts.

Landing Gear And Subsystems

- The landing-gear leg is attached to the aircraft at the "pivot point."

- This is determined as shown in Fig. 11.14.

- The pivot point can lie anywhere along the perpendicular bisector to the line connecting the up and down positions of the wheel.

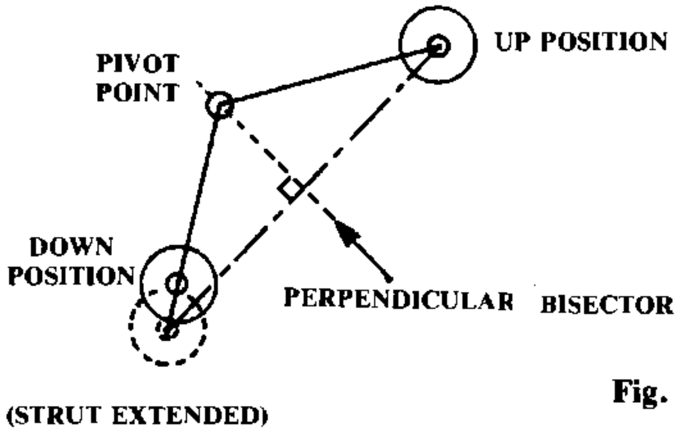


Fig. 11.14

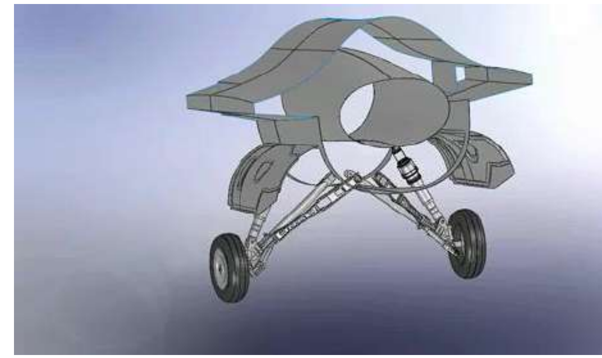
Landing Gear And Subsystems

- It is also possible to provide a "rotator" mechanism or "planing link" that will change the angle between the gear leg and the wheel axis when the gear is retracted.

- This is sometimes required to permit the wheel to lie flat inside the wheel well when retracted.

- This is fairly simple and is seen on many military aircraft such as the F-16.

- However, all such mechanisms should be avoided if possible due to the increased weight, complexity, and maintenance.



Seaplanes

- Seaplanes were important during the early days of aviation because of the limited number of good airports.
- Today the seaplane concept is largely restricted to sportplanes, bush planes, and search-and-rescue aircraft.
- The bottom of seaplane is fairly flat, allowing the aircraft to skim (plane) on top of the water at high speeds.
- A step breaks the suction on the after body.
- A vertical discontinuity, as shown in Fig. 11.15, the step can be straight in planview, or it can have an elliptical shape in planview to reduce aerodynamic drag.

Landing Gear And Subsystems

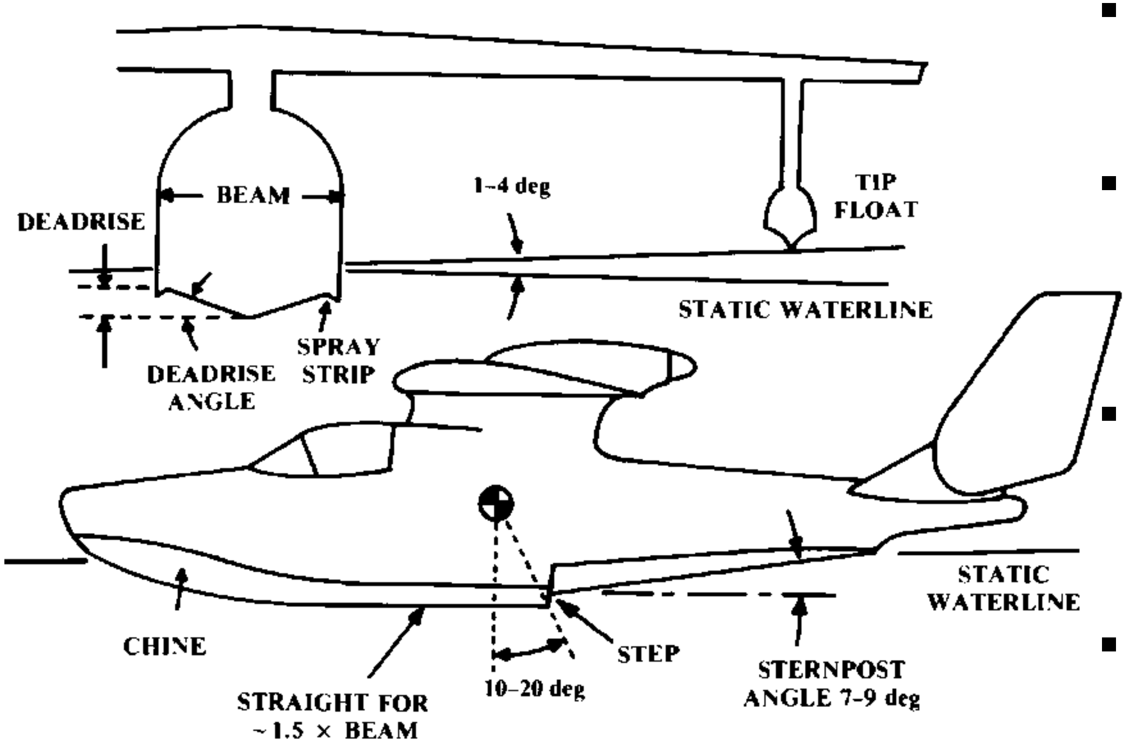


Fig. 11.15 Seaplane geometry.

- Most use a V-shaped bottom to reduce the water-impact loads.
- The height of the *V* is called the "deadrise," and the angle is the "deadrise angle."
- Deadrise angle must be increased for higher landing speeds, and should roughly follow Eq. (11.20).
- Deadrise angle is increased toward the nose to about 30-40 deg to better cut through waves.

$$\alpha_{\text{deadrise}} \cong \frac{V - 20}{2}, \text{ deg} \tag{11.20}$$

where V = stall speed in miles per hour.

Landing Gear And Subsystems

- To reduce water spray, spray strips can be attached to the edges of the bottom, as shown.

- These are angled about 30 deg below the horizon.

Landing Gear And Subsystems

- The ratio between the waterline length and "beam" (width) has a strong effect upon water resistance and landing impact.

- A wider hull has a lower water resistance due to its better planing ability but suffers a higher landing impact.

- Length-to-beam ratios vary from about six for a small seaplane to about 15 for a large one.

- The step height should be about 5% of the beam.

- The step should be located on an angle about 10-20 deg behind the c.g.

Landing Gear And Subsystems

- The bottom of the hull forward of the step should not be curved for a distance about equal to 1.5 times the beam.
- This is to reduce porpoising/ rise and submerge move tendencies.
- Also, the hull bottom aft of the step (the "sternpost") should angle upward about 8 deg.
- For a true "flying boat" (i.e., seaplane with a boat-like fuselage), lateral stability on the water is usually provided by wing-mounted pontoons.
- These should be located such that they contact the water when the aircraft tips sideways about 1 deg.

Landing Gear And Subsystems

- The fuselage submerged volume is multiplied by the density of water (62.4 lb/ft^3) to determine the weight of aircraft which can be supported by that amount of displaced water.
- The centroid of the area on the volume is the center of buoyancy, which should coincide with the c.g.
- Water-resistance drag is very difficult to estimate.
- It depends upon the mechanics of wave production, and can vary widely for similar hull shapes.
- Also, water-resistance drag varies with speed.
- A seaplane hull can have a maximum water resistance at "hump speed" equal to 20% or more of the aircraft's weight.

Landing Gear And Subsystems

Subsystems

- Aircraft subsystems include the hydraulic, electrical, pneumatic, and auxiliary/emergency power systems.
- Also, the avionics can be considered a subsystem.
- 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.

Hydraulics

- A simplified hydraulic system is shown in Fig. 11.16.
- Hydraulic fluid, a light oil-like liquid, is pumped up to some specified pressure and stored in an "accumulator" (simply a holding tank).

Landing Gear And Subsystems

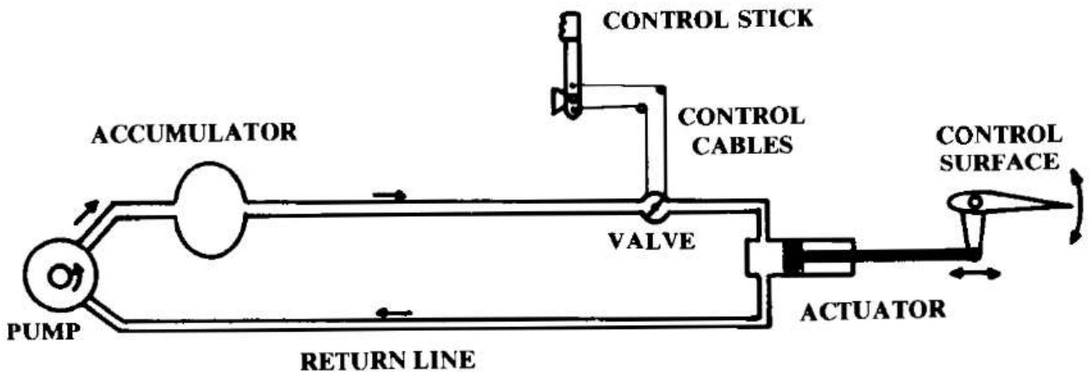


Fig. 11.16 Simplified hydraulic system.

- 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.

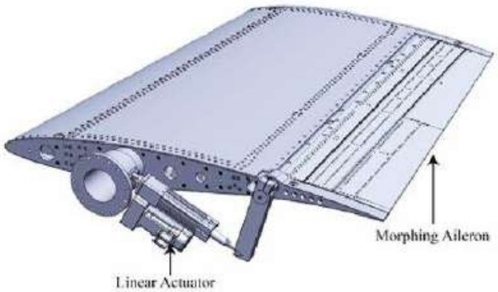
- To move the control surface the other direction, an additional valve (not shown) admits hydraulic fluid to the back side of the piston.

- The hydraulic fluid returns to the pump by a return line.

Landing Gear And Subsystems

- 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.



(b)

Landing Gear And Subsystems

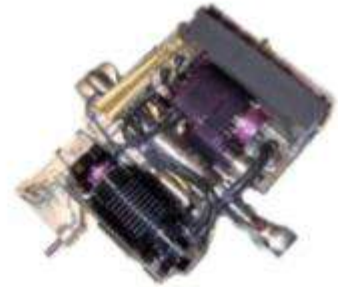
(a) Mechanically-signaled/hydraulically-supplied duplex-parallel aileron actuator for dassault falcon 900 private jet

(b) Electrically-signaled, hydraulically-supplied (HSA)simplex actuator for rudder actuation of boeing B777

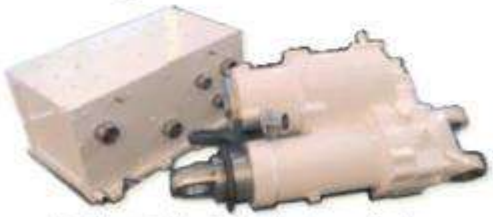
(c) Digitally-signaled, hydraulically-supplied,smart HSA aileron actuator for airbus A350

(d) Electrically-signaled and supplied EHA for the JSF F-35 fighter

(e) Electrically-signaled and supplied EBHA for the airbus A350



(f) EHA actuator for airbus A400M



(g) Electrically signaled and supplied, EMA for 4 boeing B787 spoilers

(h) Electrically signaled and supplied, EMA for the first stage of vega european launcher

Landing Gear And Subsystems

- 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").

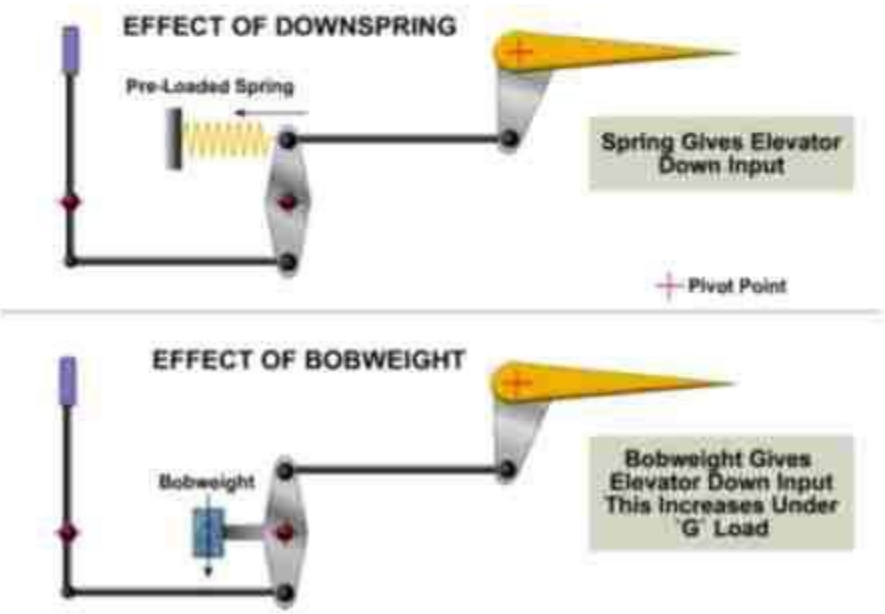
- Hydraulics are used for aircraft flight control as well as actuation of the flaps, landing gear, spoilers, speed brakes, and weapon bays.

Landing Gear And Subsystems

- 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, bobweights, and air bellows.



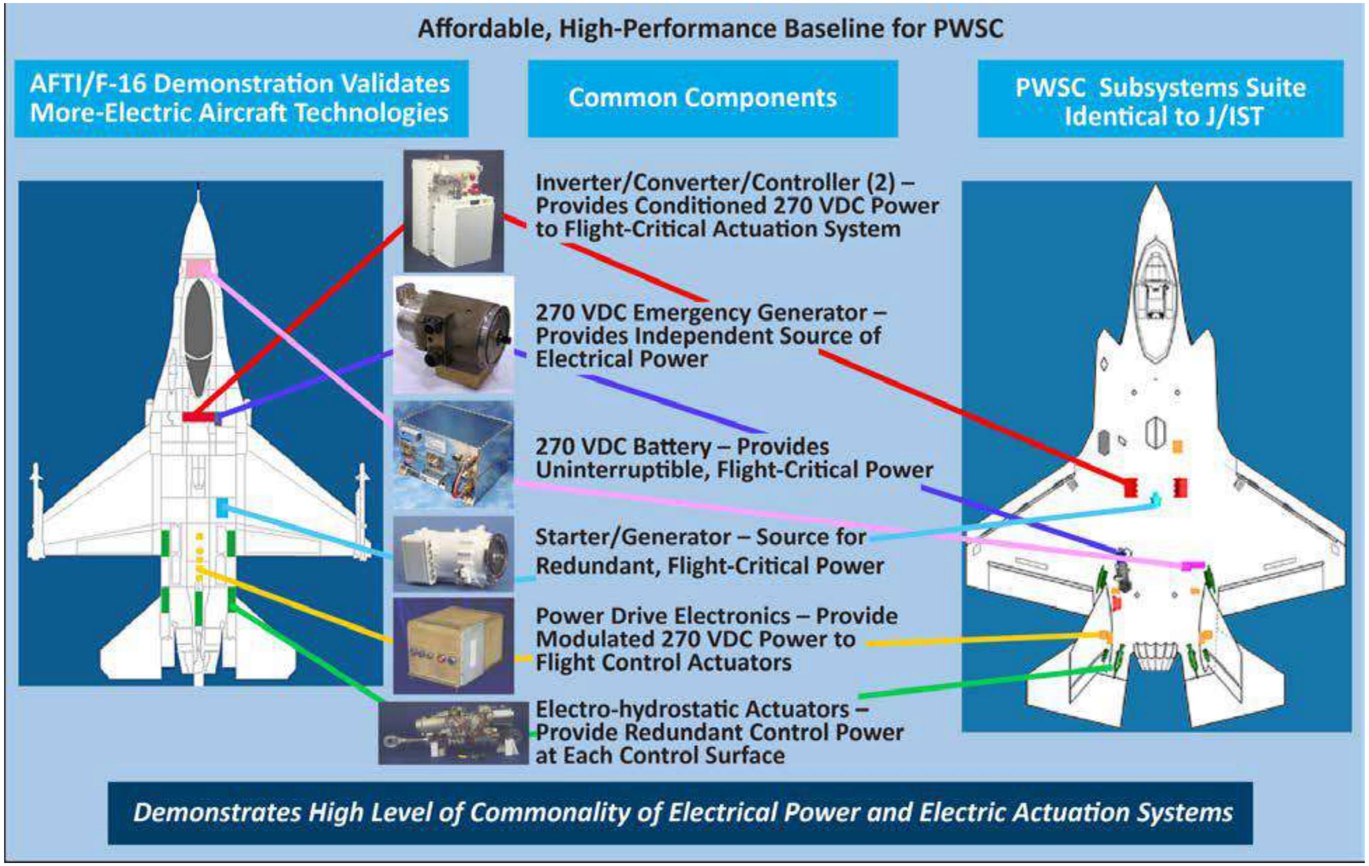
- In most cases the hydraulic system will impact the aircraft conceptual design only in the provision of space for the hydraulic pumps, which are usually attached to the engines.

- These should be copied from a similar aircraft if better information is not available.

Electrical System

- An aircraft electrical system provides electrical power to the avionics, hydraulics, environmental-control, lighting, and other subsystems.
- 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.

Landing Gear And Subsystems



Landing Gear And Subsystems

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.

Landing Gear And Subsystems

- 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.

Landing Gear And Subsystems

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.

Landing Gear And Subsystems

- 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.

Landing Gear And Subsystems

- The monopropellant EPU uses a monopropellant fuel such as hydrazine to drive a turbine.
- The available monopropellants are all toxic and caustic/corrosive, 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.
- 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.

Landing Gear And Subsystems

- Most commercial transports and an increasing number of military aircraft use a jet-fuel auxiliary power unit ("APU").

- An APU is much like an EPU but is designed and installed to allow continuous operation.

- The APU is actually another jet engine, and its installation must receive attention in the earliest design layout.

Landing Gear And Subsystems

- The APU requires its own inlet and exhaust ducts, and must be contained in a firewalled structure.

- APU's have fairly high maintenance requirements so access is important.

- To avoid high levels of noise, the inlet and exhaust of an APU should be directed upward.

- For in-flight operation of the APU, the inlet should ideally be in a high pressure area and the exhaust in a low-pressure area.

- Also, the inlet should not be located where the exhaust of the jet engines or APU can be ingested.

- The exhaust of an APU is hot and noisy, and should not impinge upon aircraft structure or ground personnel.

- Transport aircraft usually have the APU in the tail, as shown in Fig. 11.17.

- This removes the APU from the vicinity of the passenger compartment to reduce noise.

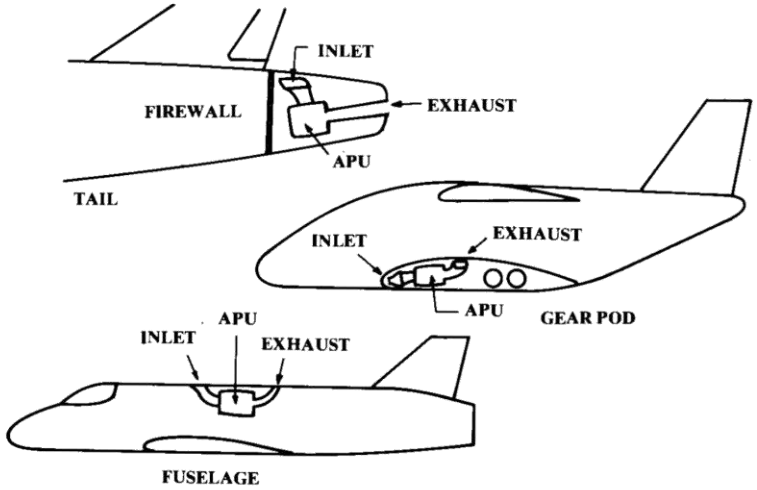


Fig. 11.17 APU installation.

Landing Gear And Subsystems

- The APU firewall is of minimum size, and the APU is easily accessible from a workstand.
- Military transports with the landing gear in fuselage-mounted pods can place the APU in the pod.
- This provides ground-level access to the APU, but requires increased firewall area.
- Fighters usually have the APU in the fuselage, near the hydraulic pumps and generators.
- This requires a firewall that completely encloses the APU.

Landing Gear And Subsystems

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/ft^3
- The required avionics weight can be estimated from the aircraft empty weight (W_e), which is known at this point.
- Table 11.6 provides ratios between avionics weight and aircraft empty weight.

Table 11.6 Avionics weights

	Typical values $\frac{W_{\text{avionics}}}{W_{\text{empty}}}$
General aviation—single engine	.01–.03
Light twin	.02–.04
Turboprop transport	.02–.04
Business jet	.04–.05
Jet transport	.01–.02
Fighters	.03–.08
Bombers	.06–.08
Jet trainers	.03–.04

Landing Gear And Subsystems

- Estimation of radar size is very complex, and depends upon the desired detection range, radar cross section of the threat aircraft, and radar frequency.

- For initial design, the radar from a similar design should be used until the avionics group provides a definition of the required radar.

- In the absence of better information, it can be assumed that
  - a bomber will use a 40-in. radar,
  - a large fighter will use a 35-in. radar,
  - and a small fighter will use a 22-in. radar.

- Transport-aircraft radars are only for weather avoidance.

- They are small relative to the size of the aircraft nose and can be initially ignored.

Landing Gear And Subsystems

F-4 Phantom

F-15 Eagle

Aircraft Design
AE 405

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Contents

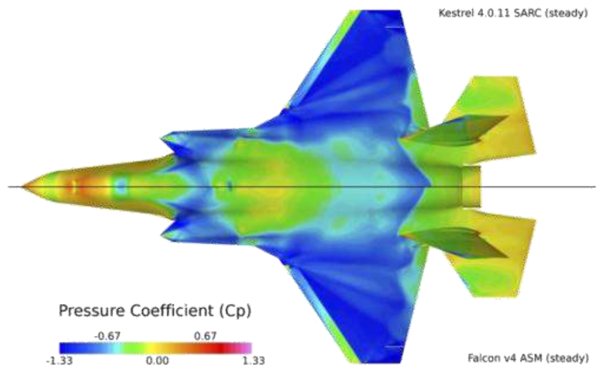


Figure 6. Top surface Cp contours: Kestrel SA-RC (top), Falcon ASM (bottom).

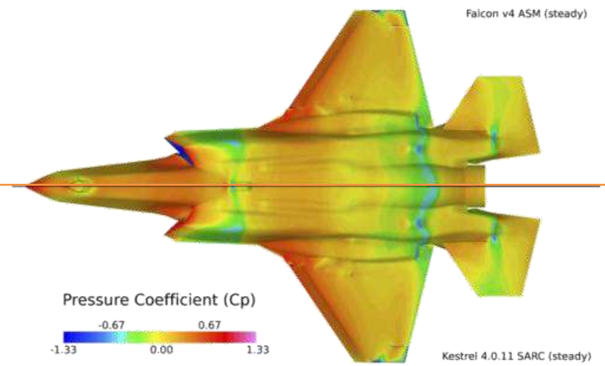
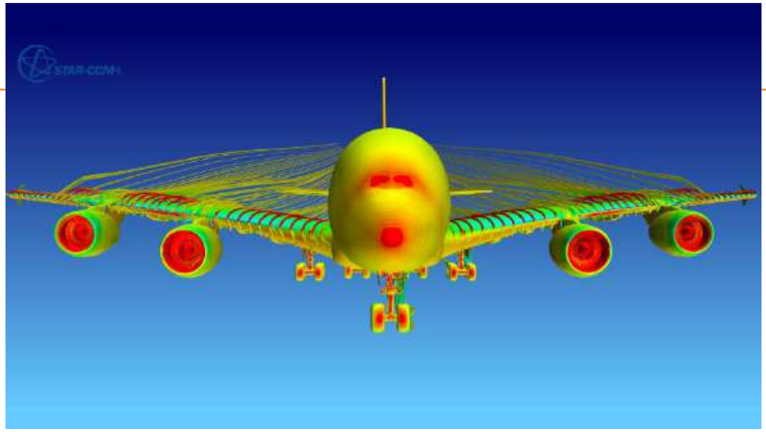


Figure 7. Bottom surface Cp contours: Kestrel SA-RC (bottom), Falcon ASM (top).

Chapter 12

Aerodynamics



Aerodynamics

- The previous chapters have presented methods for the design layout of a credible aircraft configuration.

- Initial sizing, wing geometry, engine installation, tail geometry, fuselage internal arrangement, and numerous other
- design topics have been discussed.

- The initial sizing was based upon rough estimates of the aircraft's aerodynamics, weights, and propulsion characteristics.

- At that time we could not calculate the actual characteristics of the design because the aircraft had not been designed yet!

- Now the aircraft design can be analyzed "as-drawn" to see if it actually meets the required mission range.

Aerodynamics

- The analysis techniques presented in these chapters, all approximate methods, illustrate the major parameters to be determined and provide realistic trends for trade studies.

- In many cases these are not the methods employed by the major aircraft companies, whose methods are highly computerized and cannot be presented in any single textbook.

- Also, each company uses many proprietary methods that are simply unavailable to students.

Aerodynamic Forces

- Figure 12.1 shows the only two ways that the airmass and the airplane can act upon each other.

- As the aircraft moves forward, the air molecules slide over its skin.

Aerodynamics

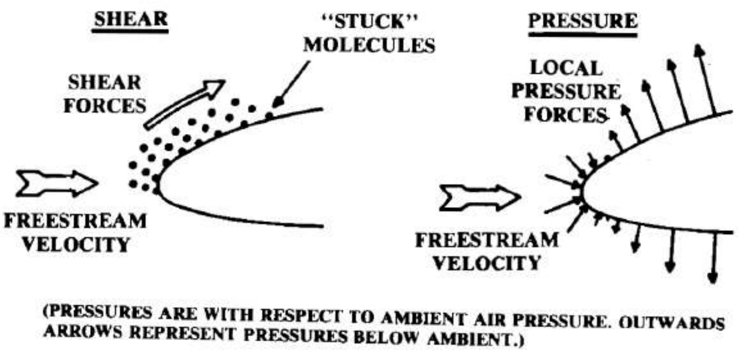


Fig. 12.1 Origin of aerodynamic forces.

- All aerodynamic lift and drag forces result from the combination of shear and pressure forces.

- Figure 12.2 presents the various drag terminologies using a matrix
  - that defines the drag type based upon the origin of the drag force (shear or pressure)
  - and whether or not the drag is strongly related to the lift force being developed.

Aerodynamics

	SHEAR FORCES	PRESSURE FORCES		
		SEPARATION	SHOCK	CIRCULATION
PARASITE DRAG	SKIN FRICTION	VISCOUS SEPARATION	WAVE DRAG	
	SCRUBBING DRAG	SHOCK-INDUCED SEPARATION "DRAG RISE"		
	INTERFERENCE DRAG			
	PROFILE DRAG			
		CAMBER DRAG		
INDUCED DRAG [f(Lift)]	SUPERVELOCITY EFFECT ON SKIN FRICTION	SUPERVELOCITY EFFECT ON PROFILE DRAG – i.e., LANDING GEAR, ETC.		DRAG DUE TO LIFT
				TRIM DRAG
			WAVE DRAG DUE TO LIFT	
REFERENCE AREA:	S_{wetted}	Max. Cross Section	(Volume Distribution)	S_{ref}

Fig. 12.2 Drag terminology matrix.

Aerodynamics

- Drag forces not strongly related to lift are usually known as parasite drag or zero-lift drag.
- In subsonic cruising flight of a well-designed aircraft, the parasite drag consists mostly of skin-friction drag, which depends mostly upon the wetted area.
- Scrubbing drag is an increase in the skin-friction drag due to the propwash or jet exhaust impinging upon the aircraft skin.
- This produces a higher effective air velocity and assures turbulent flow, both of which increase drag.
- It is for this reason that pusher-propellers are desirable

Aerodynamics

- There are three separate origins of the drag-producing pressure forces.
- The first, viscous separation.
- Viscous separation drag, also called "form drag," depends upon the location of the separation point on the body.
- Turbulent air has more energy than laminar air, so a turbulent boundary layer actually tends to delay separation.
- "vortex generators" are can used on wing and tail surfaces.

Boeing 737 wing

C 130 ramp

Aerodynamics

- 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.

- 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.

Aerodynamics

- "Wave drag" is the drag caused by the formation of shocks at supersonic and high subsonic speeds.

- 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.

- To counter the pitching moment of the wing, the tail surfaces produce a lift force generally in the downward direction.

- The induced drag of the tail is called "trim drag."

Aerodynamics

- Most preliminary drag-estimation methods do not actually use the airfoil profile drag data to determine total wing drag.

- Instead, the drag for an idealized wing with no camber or twist is determined, and then a separate "camber drag" is estimated.

- Often the camber drag term is included statistically in the drag-due-to-lift calculation.

- Changing the lift on the wing changes the velocities above and below it.

- This change in local airflow velocity causes a small change in skin-friction drag.

- Sometimes called a "supervelocity" effect, this is minor and is usually ignored.

Aerodynamics

Aerodynamic Coefficients

- Lift and drag forces are usually treated as nondimensional coefficients as defined in Eqs. (12.1) and (12.2).

$$L = qSC_L \tag{12.1}$$

$$D = qSC_D \tag{12.2}$$

$$q = \tfrac{1}{2}\rho V^2$$

- Drag is normally spoken of as so many "counts" of drag, meaning the four digits to the right of the decimal place.

- For example, 38 counts of drag mean a drag coefficient of 0.0038.

Aerodynamics

- Figure 12.3 illustrates the drag polar, which is the standard presentation format for aerodynamic data used in performance calculations.

- The drag polar is simply a plot of the coefficient of lift vs the coefficient of drag.

- Angle of attack (α) is indicated here by tic marks along polar curve.

- This is not standard practice, but is useful for understanding the relationship between lift, drag, and angle of attack.

Aerodynamics

$$\text{Uncambered:} \quad C_D = C_{D_0} + KC_L^2 \tag{12.4}$$

$$\text{Cambered:} \quad C_D = C_{D\,\text{min}} + K(C_L - C_{L\,\text{min drag}})^2 \tag{12.5}$$

- The point at which a line from the origin is tangent to the drag polar curve is the point of maximum lift-to-drag ratio.
- Note that this is not the point of minimum drag.

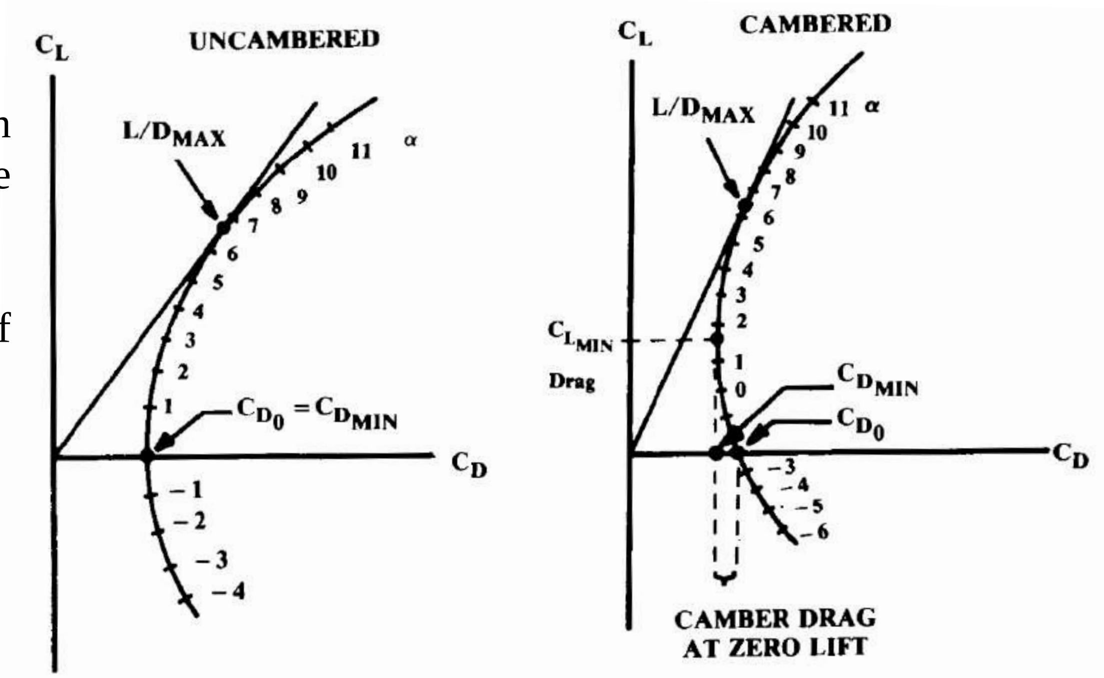


Fig. 12.3 Drag polar.

Aerodynamics

Lift

- Figure 12.4 shows typical wing lift curves.

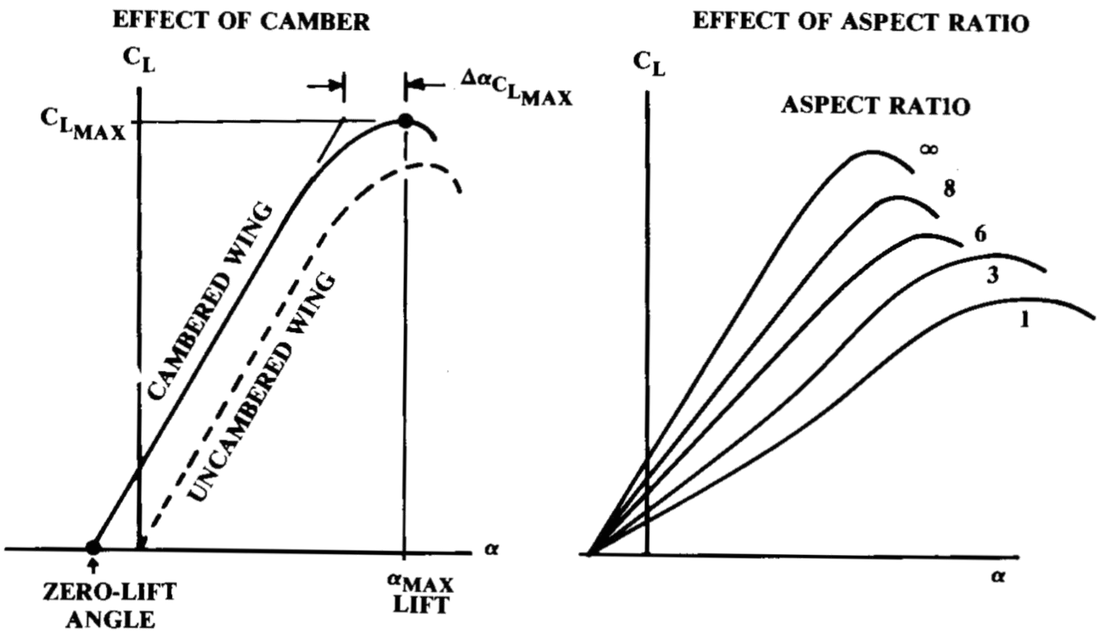


Fig. 12.4 Wing lift curve.

- Maximum lift is obtained at the "stall" angle of attack, beyond which the lift rapidly reduces.

- When a wing is stalled, most of the flow over the top has separated.

Aerodynamics

- Figure 12.4 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 π (per radian).
- Actual airfoils have lift-curve slopes between about 90 and 100% of the theoretical value.
- This percentage of the theoretical value is sometimes called the "airfoil efficiency".
- Also note that the lift curve becomes nonlinear for very low aspect ratios.

Aerodynamics

- Increasing the wing sweep has an effect similar to reducing the aspect ratio. A highly-swept wing has a lift-curve slope much like the aspect-ratio three curve shown.

- The effect of Mach number on the lift-curve slope is shown in Fig. 12.5.

- The 2-D airfoil lines represent upper boundaries for the no-sweep, infinite aspect-ratio wing.

- Real wings fall below these curves as shown.

Aerodynamics

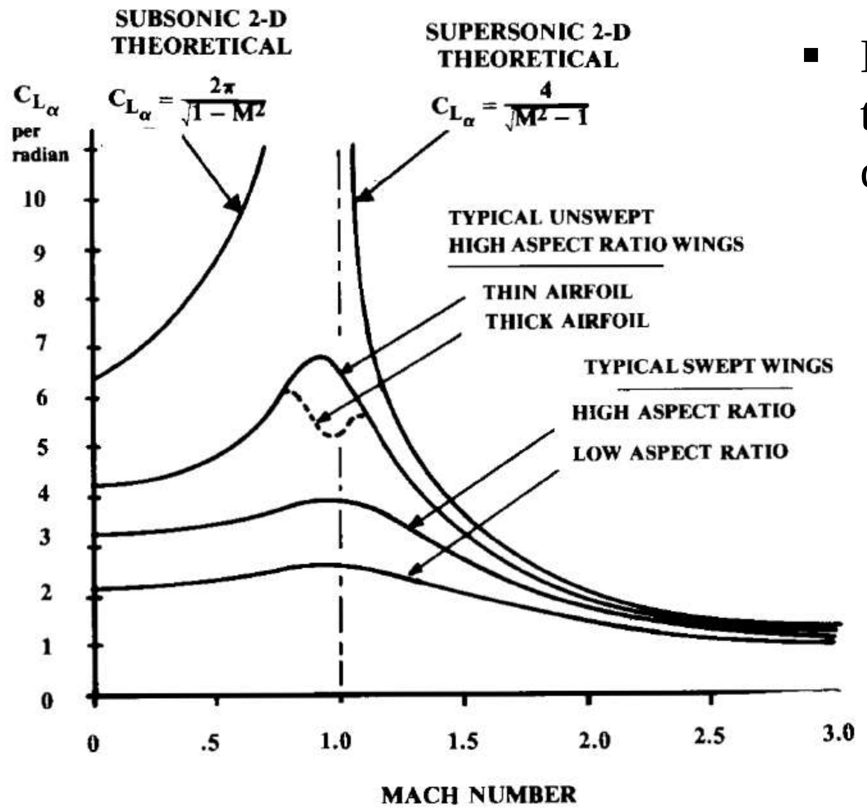


Fig. 12.5 Lift curve slope vs Mach number.

- Note that a fat and unswept wing loses lift in the transonic regime whereas swept wing does not.

Aerodynamics

- 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.

- Secondly, the methodology for calculating drag-due-to-lift for high-performance aircraft uses the slope of the lift curve, as will be seen.

- The third use of the lift curve slope in conceptual design is for longitudinal-stability analysis

Subsonic Lift-Curve Slope

- Equation (12.6) is a semi-empirical formula 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.

$$C_{L_\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2\beta^2}{\eta^2}\left(1 + \frac{\tan^2\Lambda_{\max t}}{\beta^2}\right)}}\left(\frac{S_{\text{exposed}}}{S_{\text{ref}}}\right)(F) \tag{12.6}$$

$$\beta^2 = 1 - M^2 \tag{12.7}$$

$$\eta = \frac{C_{\ell_\alpha}}{2\pi/\beta} \tag{12.8}$$

Aerodynamics

- $\Lambda_{max,t}$ is the sweep of the wing at the chord location where the airfoil is thickest.

- If the airfoil lift-curve slope as a function of Mach number is not known the airfoil efficiency can be approximated as about 0.95.

- "$S_{exposed}$" is the exposed wing planform, i.e., the wing reference area less the part of the wing covered by the fuselage.

- *"F"* is the fuselage lift factor [Eq. (12.9)] that accounts for the fact that the fuselage of diameter *"d"* creates some lift due to the "spill-over" of lift from the wing.

$$F = 1.07(1 + d/b)^2 \qquad (12.9)$$

Aerodynamics

- The wing aspect ratio *"A "* is the geometric aspect ratio of the complete reference planform.

- The effective aspect ratio will be increased by wing endplates or winglets.

$$\text{Endplate:} \quad A_{\text{effective}} = A(1 + 1.9\, h/b) \tag{12.10}$$

where h = endplate height.

$$\text{Winglet:} \quad A_{\text{effective}} \cong 1.2A \tag{12.11}$$

- These effective aspect ratios should also be used in the induced drag calculations below.

- Typically, a wing with higher aspect ratio will obtain less improvement by the use of winglets.

Aerodynamics

Supersonic Lift-Curve Slope

- For a wing in purely supersonic flow, the lift-curve slope is ideally defined by Eq. (12.12).

- A wing is considered to be in purely supersonic flow when the leading edge is "supersonic," i.e., when the Mach cone angle is greater than the leading-edge sweep.

$$C_{L_\alpha} = 4/\beta \tag{12.12}$$

$$\beta = \sqrt{M^2 - 1} \tag{12.13}$$

$$M > 1/\cos\Lambda_{LE} \tag{12.14}$$

- The actual lift-curve slope of a wing in supersonic flight is difficult to predict without use of a sophisticated computer program.

Transonic Lift-Curve Slope

- In the transonic regime (roughly Mach 0.85-1.2 for a swept wing) there are no good initial-estimation methods for slope of the lift curve.
- It is suggested that the subsonic and supersonic values be plotted vs Mach number, and that a smooth curve be faired between the subsonic and supersonic values similar to the curves shown in Fig. 12.5.

Maximum Lift (Clean)

- 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

Aerodynamics

- The prediction of max. lift coefficient is difficult.

- 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 90% of the airfoil's maximum lift.

- 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 [Eq. (12.15)].

- This equation is reasonably valid for most subsonic aircraft of moderate sweep.

$$C_{L_{\max}} = 0.9 C_{\ell_{\max}} \cos \Lambda_{0.25c} \tag{12.15}$$

Aerodynamics

- An arbitrary "leading-edge sharpness parameter" has been defined as the vertical separation between the points on the upper surface,

- which are 0.15% and 6% of the airfoil chord back from the leading edge (Fig. 12.7).

- The leading-edge sharpness parameter (or "Δy") as a function of thickness ratio for various airfoils is provided in Table 12.1.

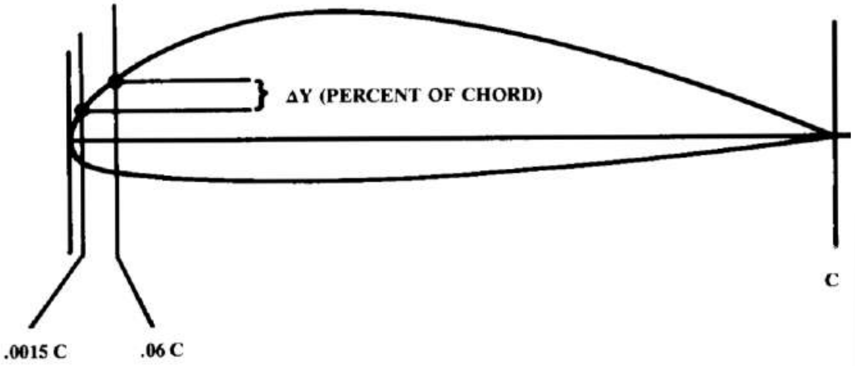


Fig. 12.7 Airfoil leading edge sharpness parameter.

Table 12.1 ΔY for common airfoils

Airfoil type	ΔY
NACA 4 digit	26 t/c
NACA 5 digit	26 t/c
NACA 64 series	21.3 t/c
NACA 65 series	19.3 t/c
Biconvex	11.8 t/c

Aerodynamics

- The leading-edge sharpness parameter has been used to develop methods for the construction of the lift curve up to the stall, for low- or high-aspect-ratio wings.

- For high-aspect-ratio wings, Eq. (12.16) is used along with Figs. 12.8 and 12.9.

- The first term of Eq. (12.16) represents the maximum lift at Mach 0.2, and the second term represents the correction to a higher Mach number.

High Aspect Ratio: $$C_{L_{\max}} = C_{\ell_{\max}} \left(\frac{C_{L_{\max}}}{C_{\ell_{\max}}} \right) + \Delta C_{L_{\max}} \qquad (12.16)$$

Aerodynamics

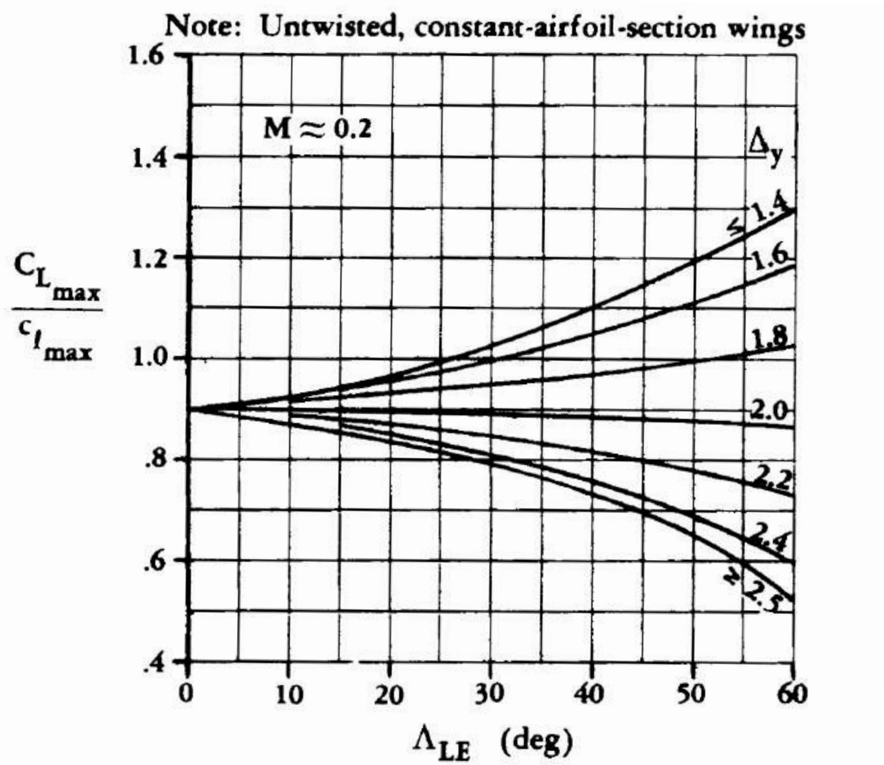


Fig. 12.8 Subsonic maximum lift of high-aspect-ratio wings. (Ref. 37)

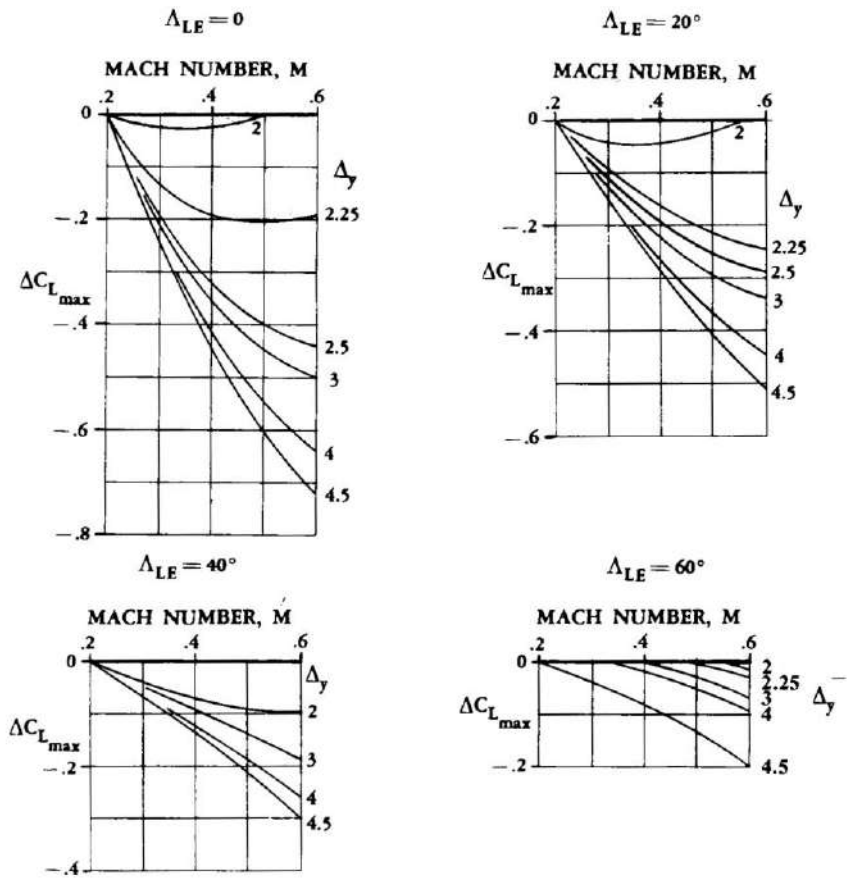


Fig. 12.9 Mach-number correction for subsonic maximum lift of high-aspect-ratio wings. (Ref. 37)

Aerodynamics

- The angle of attack for maximum lift is defined in Eq. (12.17) with the help of Fig. 12.10.

High Aspect Ratio: $$\alpha_{C_{L\max}} = \frac{C_{L\max}}{C_{L_\alpha}} + \alpha_{0L} + \Delta\alpha_{C_{L\max}} \qquad \textbf{(12.17)}$$

- 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 for a cambered airfoil.

- If the wing is twisted, the zero-lift angle is approximately the zero lift angle at the mean chord location.

- The third term in Eq. (12.17) is a correction for the nonlinear effects of vortex flow.

Aerodynamics

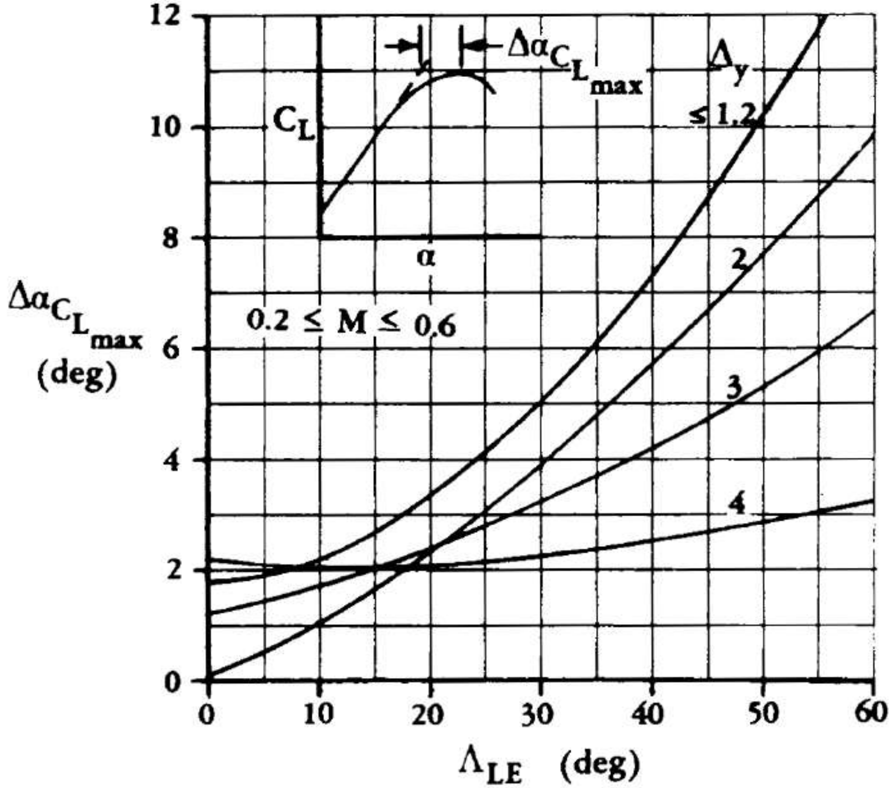


Fig. 12.10 Angle-of-attack increment for subsonic maximum lift of high-aspect-ratio wings. (Ref. 37)

- A different set of charts is used for a low-aspect-ratio wing, where vortex flow dominates the aerodynamics.

- For use of these charts, low aspect ratio is defined by Eq. (12.18), which uses the parameter C1 from Fig. 12.11.

S = wing area

AR = aspect ratio = $\frac{b^2}{S} = \frac{b}{c}$

λ = taper ratio = $\frac{c_{tip}}{c_{root}}$

Λ = sweep of c/4 line

θ = wing twist angle

$$\text{Low Aspect Ratio if:} \quad A \leq \frac{3}{(C_1 + 1)(\cos\Lambda_{LE})} \qquad (12.18)$$

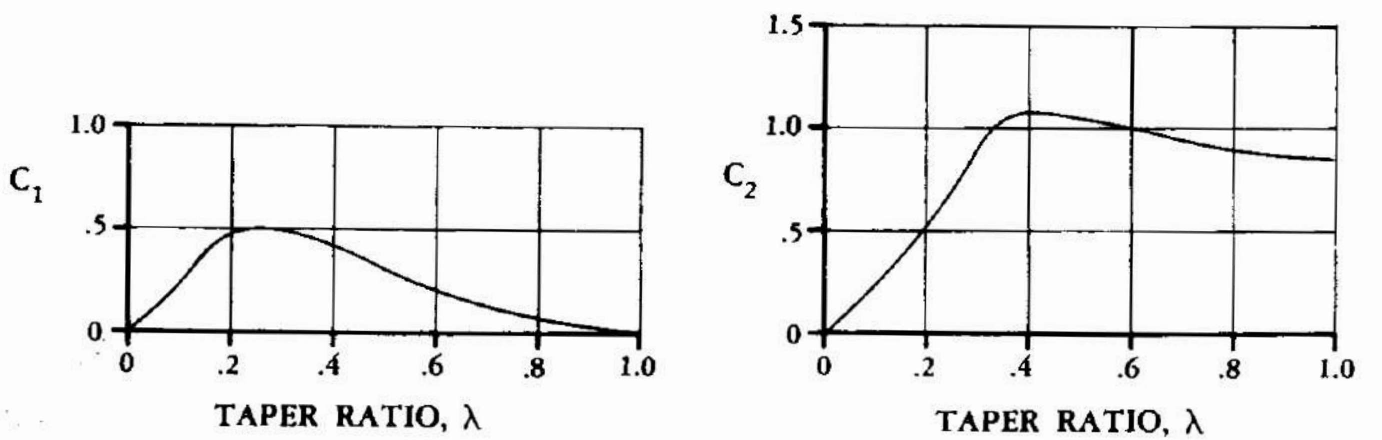


Fig. 12.11 Taper-ratio correction factors for low-aspect-ratio wings. (Ref. 37)

Aerodynamics

- Maximum lift of a low-aspect-ratio wing is defined by Eq. (12.19) using Figs. 12.12 and 12.13.

- The angle of attack at maximum lift is defined by Eq. (12.20) using Figs. 12.14 and 12.15.

$$\text{Low Aspect Ratio:} \quad C_{L_{\max}} = (C_{L_{\max}})_{\text{base}} + \Delta C_{L_{\max}} \tag{12.19}$$

$$\alpha_{C_{L_{\max}}} = (\alpha_{C_{L_{\max}}})_{\text{base}} + \Delta\alpha_{C_{L_{\max}}} \tag{12.20}$$

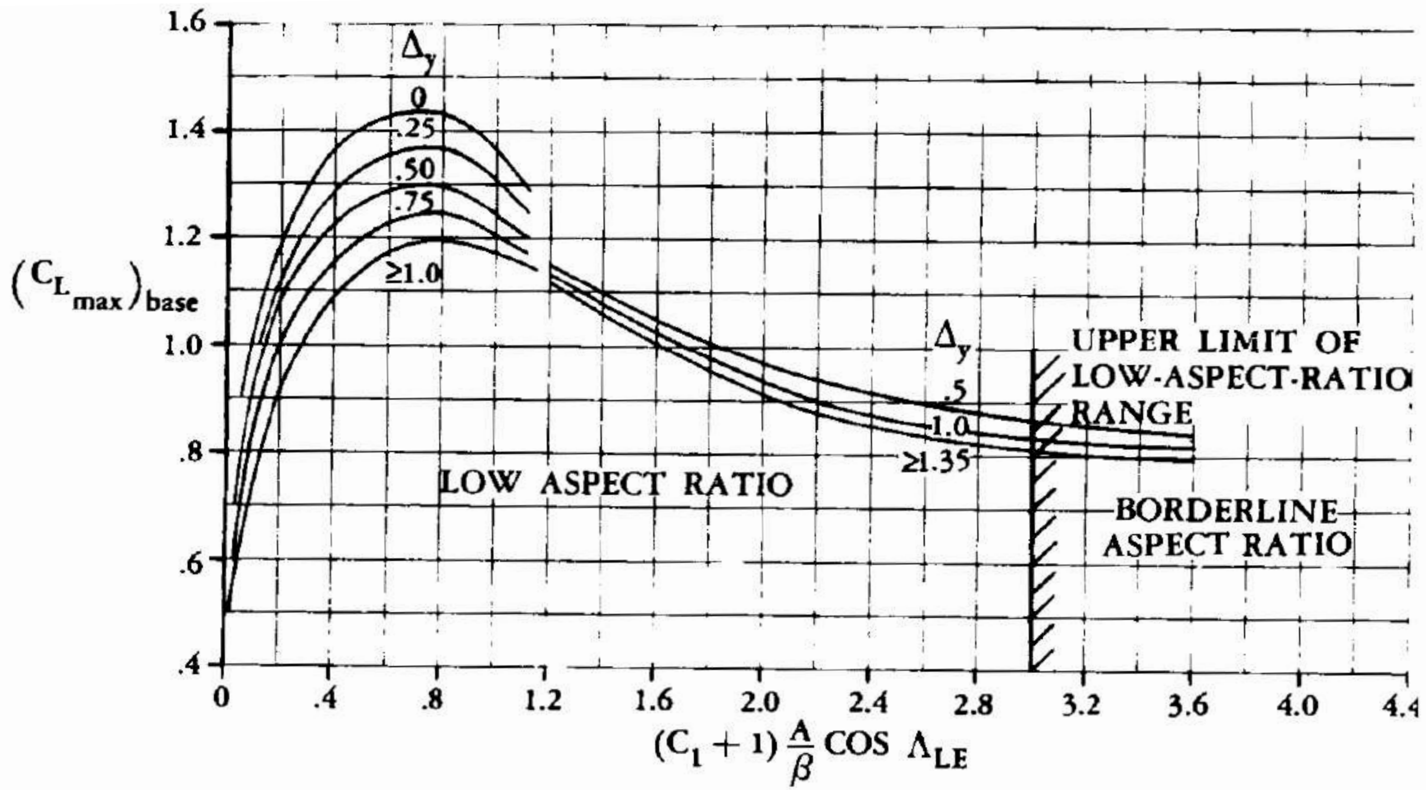


Fig. 12.12 Maximum subsonic lift of low-aspect-ratio wings. (Ref. 37)

Aerodynamics

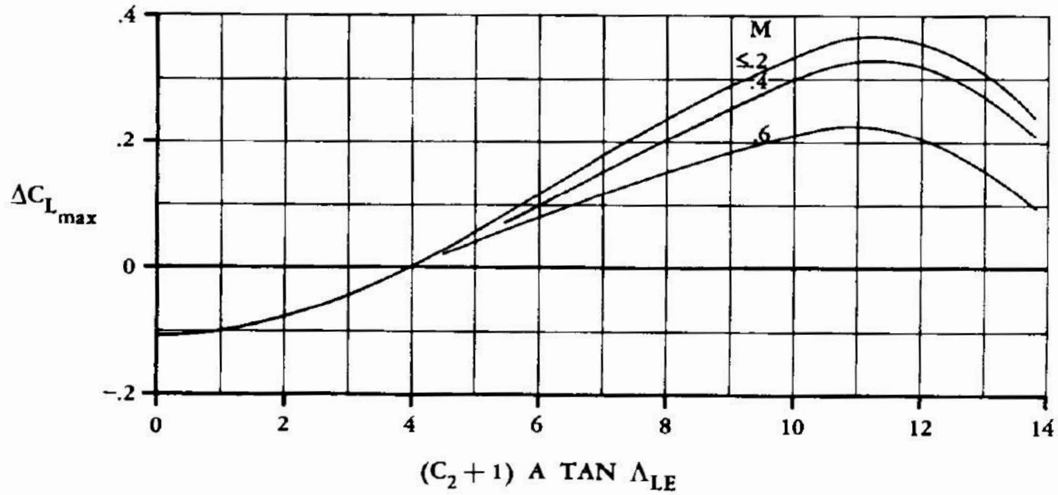


Fig. 12.13 Maximum-lift increment for low-aspect-ratio wings. (Ref. 37)

Aerodynamics

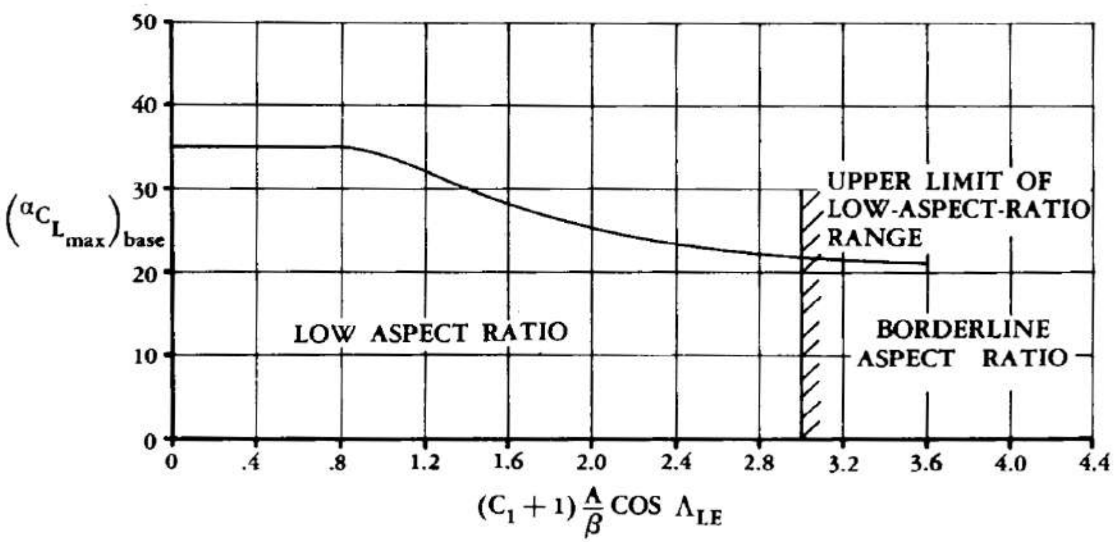


Fig. 12.14 Angle of attack for subsonic maximum lift of low-aspect-ratio wings. (Ref. 37)

Aerodynamics

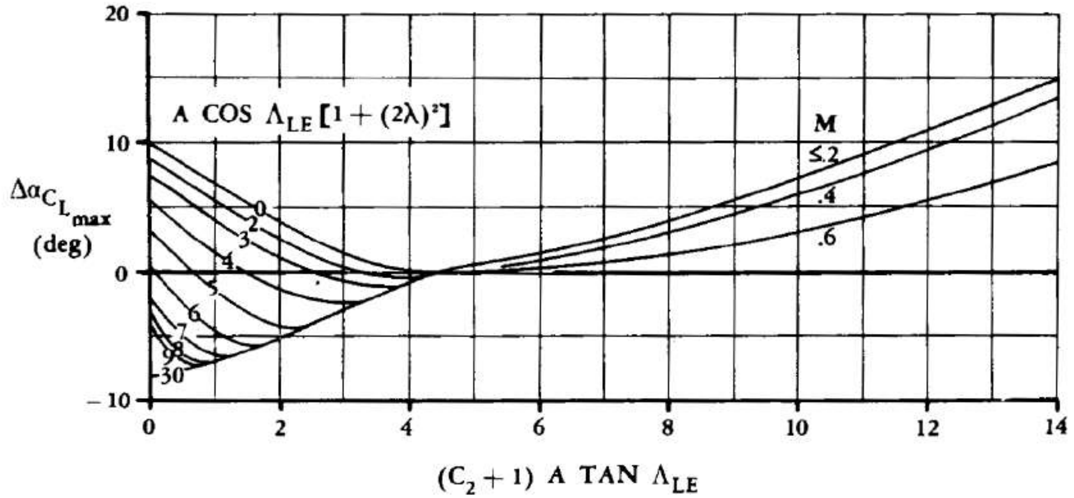


Fig. 12.15 Angle-of-attack increment for subsonic maximum lift of low-aspect-ratio wings. (Ref. 37)

Aerodynamics

- At transonic and supersonic speeds, the maximum lift a wing can achieve is usually limited by structural considerations rather than aerodynamics.

Maximum Lift with High-Lift Devices

- Figure 12.16 illustrates the commonly used high lift flaps.
- The plain flap is simply a hinged portion of the airfoil, typically with a flap chord "C_f" of 30% of the airfoil chord.
- The plain flap increases lift by increasing camber.
- For a typical airfoil, the maximum lift occurs with a flap deflection angle of about 40-45 deg.
- Note that ailerons and other control surfaces are a form of plain flap.

Aerodynamics

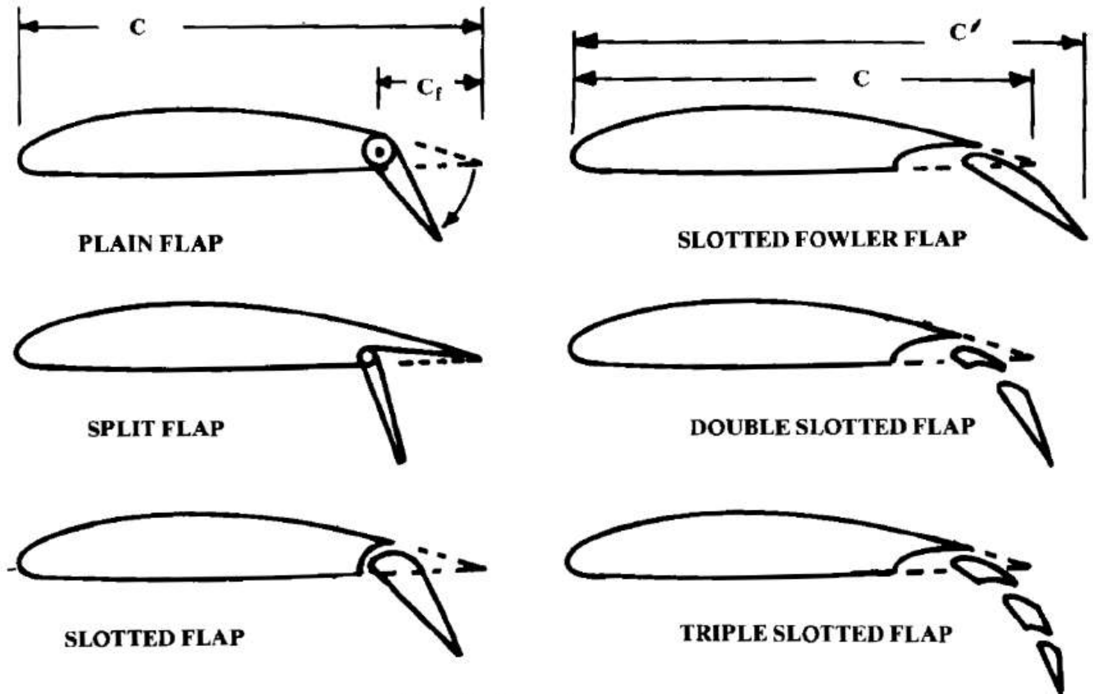
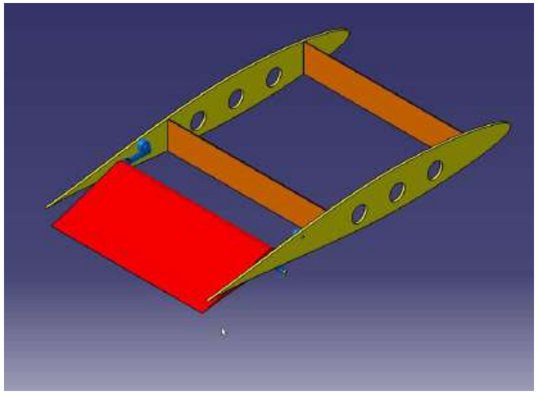


Fig. 12.16 Flap types.

Aerodynamics

- The slotted flap is a plain flap with a slot between the wing and the flap.
- This permits high-pressure air from beneath the wing to exit over the top of the flap, which tends to reduce separation.
- This increases lift and reduces drag.
- The "Fowler-type" flap is like a slotted flap, but mechanized to slide rearward as it is deflected.
- This increases the wing area as well as the camber.



Aerodynamics

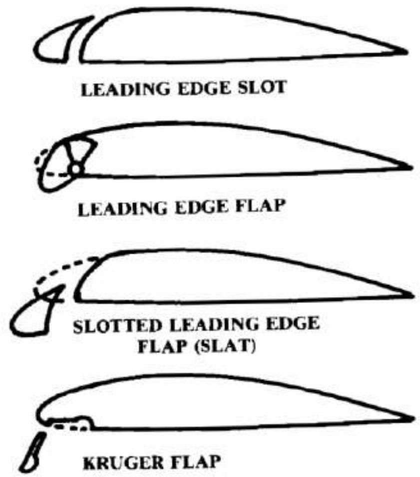
- To further improve the airflow over the Fowler flap, double- and even triple-slotted flaps are used on some airliners.

- These increase lift but at a considerable increase in cost and complexity.

Aerodynamics

- Aft flaps do not increase the angle of stall.
- In fact, they tend to reduce the stall angle by increasing the pressure drop over the top of the airfoil, which promotes flow separation.
- To increase the stall angle, some form of leading-edge device is required, as shown in Fig. 12.17.
- The leading-edge slot is simply a hole which permits high-pressure air from under the wing to blow over the top of the wing, delaying separation and stall.
- Usually such a slot is fixed, but may have closing doors to reduce drag at high speeds.

Aerodynamics



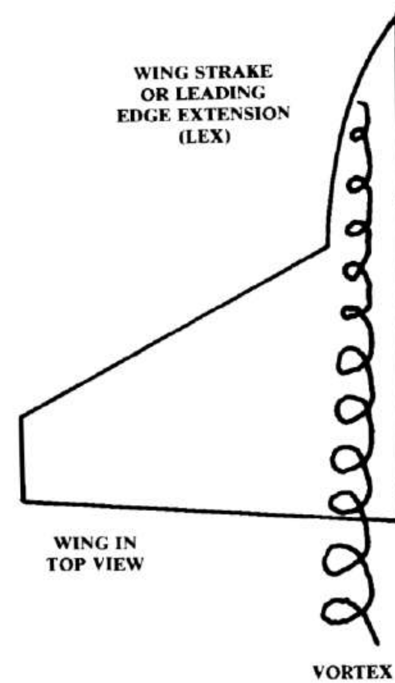


Fig. 12.17 Leading edge devices.

- The wing strake, or "Leading Edge Extension (LEX), is similar to the dorsal fin used on vertical tails.
- Like dorsal fins, the LEX at high angle of attack produces a vortex that delays separation and stall.
- Unfortunately, a LEX tends to promote pitch-up tendencies and so must be used with care.

Aerodynamics

- Figure 12.18 illustrates the effects these high-lift devices have upon the lift curve of the wing.

- There are many complex methods for estimating the effects of high-lift devices.

- For initial design, Eqs. (12.21) and (12.22) provide a reasonable estimate of the increase in maximum lift

- and the change in the zero-lift angle for flaps and leading-edge devices when deployed at the optimum angle for high lift during landing.

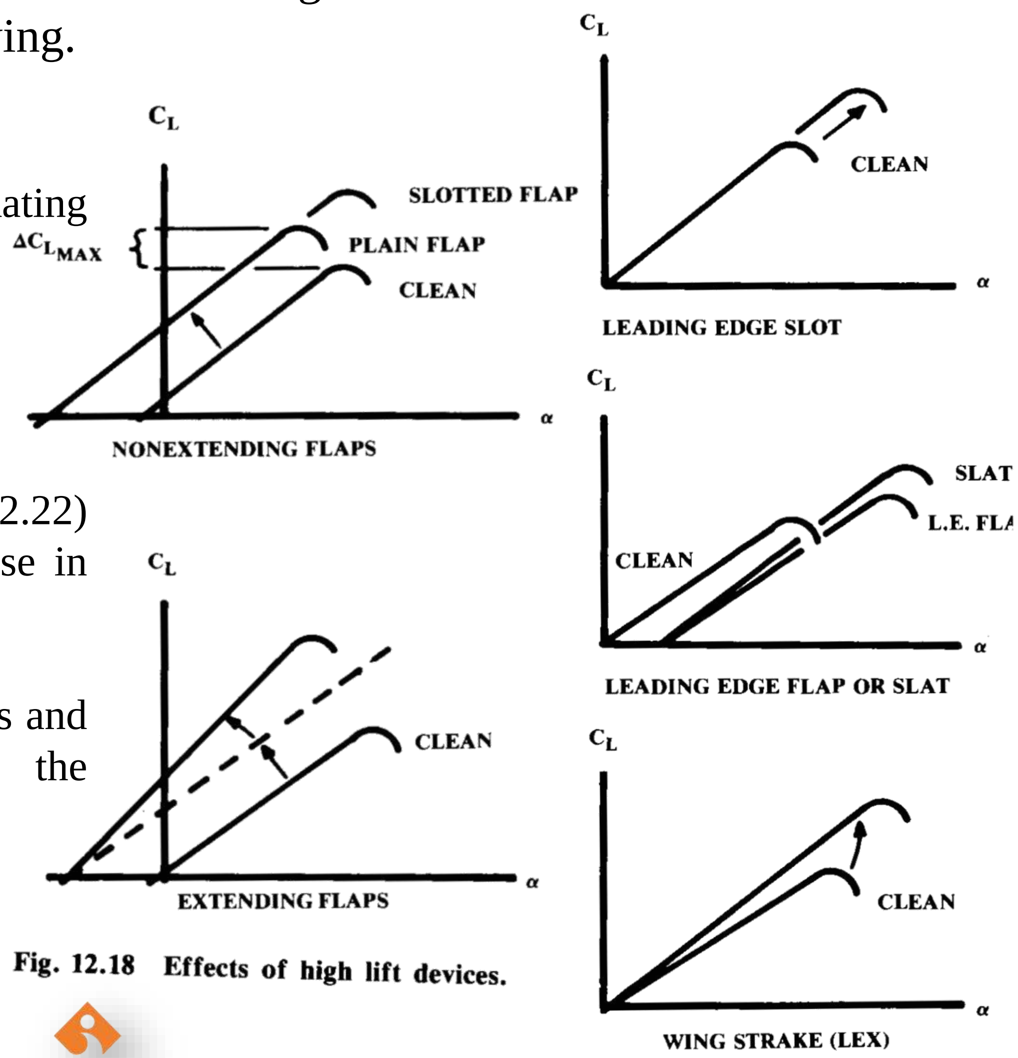


Fig. 12.18 Effects of high lift devices.

Aerodynamics

$$\Delta C_{L_{max}} = \Delta C_{\ell_{max}} \left(\frac{S_{\text{flapped}}}{S_{\text{ref}}} \right) \cos \Lambda_{\text{H.L.}} \tag{12.21}$$

$$\Delta \alpha_{OL} = (\Delta \alpha_{OL})_{\text{airfoil}} \left(\frac{S_{\text{flapped}}}{S_{\text{ref}}} \right) \cos \Lambda_{\text{H.L.}} \tag{12.22}$$

$\Delta C_{\ell_{max}}$ values should be obtained from test data for the selected airfoil, or may be approximated from Table 12.2.

- For takeoff flap settings, lift increments of about 60-80% of these values should be used.
- The change in zero lift angle for flaps in the 2-D case is approximately -15 deg at the landing setting, and -10 deg at the takeoff setting.

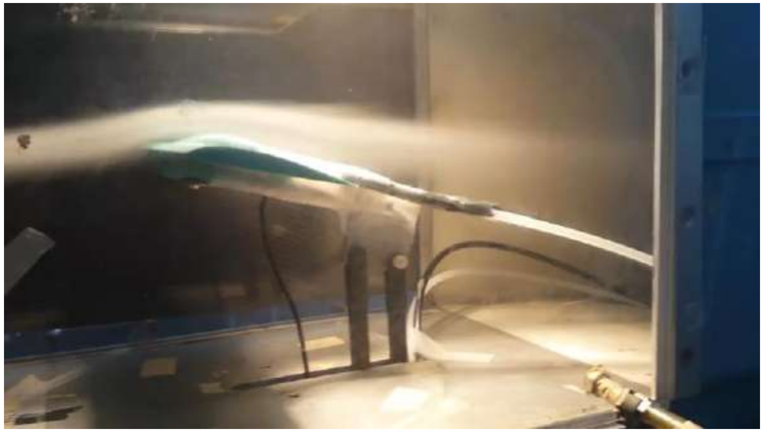
Aerodynamics

Table 12.2 Approximate lift contributions of high-lift devices

High-lift device	$\Delta C_{\ell_{max}}$
Flaps	
Plain and split	0.9
Slotted	1.3
Fowler	1.3 c′/c
Double slotted	1.6 c′/c
Triple slotted	1.9 c′/c
Leading edge devices	
Fixed slot	0.2
Leading edge flap	0.3
Kruger flap	0.3
Slat	0.4 c′/c

Aerodynamics

- In Eqs. (12.21) and (12.22), "H.L." refers to the hinge line of the high lift surface.
- "$S_{flapped}$" is defined in Fig. 12.19.
- The lift increment for a leading-edge extension may be crudely estimated as 0.4 at high angles of attack.
- Other methods for increasing the lift coefficient involve active flow control using either suction or blowing.



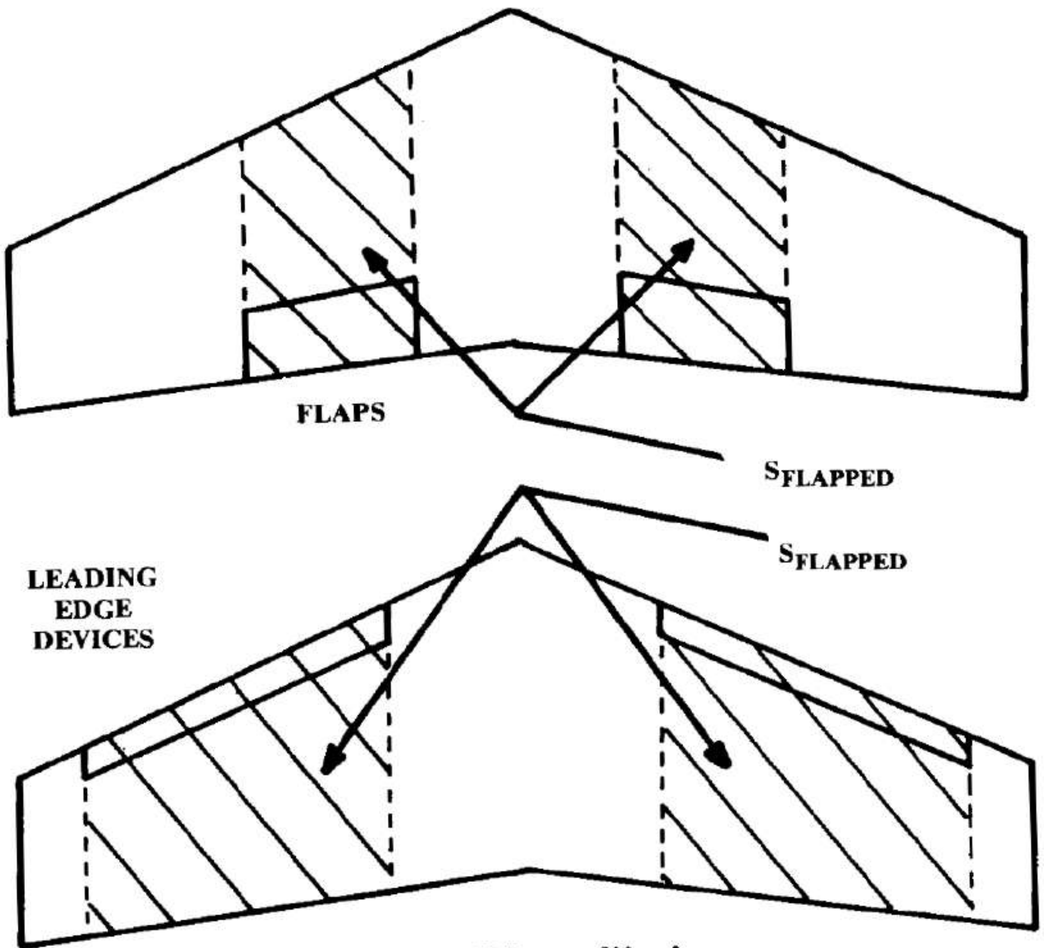


Fig. 12.19 "Flapped" wing area.

Parasite (Zero-lift) Drag

- Two methods for the estimation of the parasite drag (C_{D0}) 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.

Aerodynamics

Equivalent Skin-Friction Method

- C_{fe} is multiplied by the aircraft's wetted area to obtain an initial estimate of parasite drag.
- This estimate [Eq. (12.23) and Table 12.3] is suitable for initial subsonic analysis

$$C_{D_0} = C_{f_e}\frac{S_{\text{wet}}}{S_{\text{ref}}} \qquad (12.23)$$

Table 12.3 Equivalent skin friction coefficients

$C_{D_0} = C_{fe}\frac{S_{\text{wet}}}{S_{\text{ref}}}$	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

Component Buildup Method

- The component buildup method estimates the subsonic parasite drag of each component of the aircraft

  - using a calculated flat-plate skin friction drag coefficient (C_f)
  - and a component form factor (*FF*) that estimates the pressure drag due to viscous separation.

- Then the interference effects on the component drag are estimate as a factor *Q* and the total component drag is determined as the product of the wetted area, (C_f), *FF*, and *Q*.

Aerodynamics

- Miscellaneous drags (C_{Dmisc}) for special features of an aircraft such as flaps, unretracted landing gear, an unswept aft fuselage are then estimated

- and added to the total, along with estimated contributions for leakages and protuberances ($C_{DL\&P}$).

- Subsonic parasite-drag buildup is shown in Eq. (12.24) where the subscript '*c*' indicates that those values are different for each component.

$$(C_{D0})_{\text{subsonic}} = \frac{\Sigma(C_{f_c} FF_c Q_c S_{\text{wet}_c})}{S_{\text{ref}}} + C_{D_{\text{misc}}} + C_{D_{\text{L\&P}}} \qquad (12.24)$$

Aerodynamics

- For supersonic flight, the skin-friction contribution is simply the flat plate skin friction coefficient times the wetted area.

- All supersonic pressure drag contributions (except base drag) are included in the wave-drag term, which is determined from the total aircraft volume distribution.

Flat-Plate Skin Friction Coefficient

- The flat-plate skin friction coefficient C_f depends upon the Reynolds number, Mach number, and skin roughness.

- At a local Reynolds number of one million, a surface with turbulent flow will have a friction drag coefficient as much as three times the drag coefficient of a surface with laminar flow.

Aerodynamics

- Most current aircraft have turbulent flow over virtually the entire wetted surface, although some laminar flow may be seen towards the front of the wings and tails.

- A typical current aircraft may have laminar flow over perhaps 10-20% of the wings and tails, and virtually no laminar flow over the fuselage.

- A carefully designed modern composite aircraft such as the Piaggio GP180 can have

  - laminar flow over as much as 50% of the wings and tails,
  - and about 20-35% of the fuselage.

Aerodynamics

- For the portion of the aircraft that has laminar flow, the flat-plate skin friction coefficient is expressed by Eq. (12.25).

$$\text{Laminar:} \quad C_f = 1.328/\sqrt{R} \tag{12.25}$$

$$R = \rho V \ell / \mu \tag{12.26}$$

- The "*l*" in Eq. (12.26) is the characteristic length.
  - For a fuselage, *l* is the total length.
  - For a wing or tail, *l* is the mean aerodynamic chord length.

- For turbulent flow, which in most cases covers the whole aircraft, the flat-plate skin friction coefficient is determined by Eq. (12.27).

Aerodynamics

$$\text{Turbulent:} \quad C_f = \frac{0.455}{(\log_{10} R)^{2.58}\,(1 + 0.144 M^2)^{0.65}} \qquad (12.27)$$

- If the surface is relatively rough, the friction coefficient will be higher than indicated by this equation.

- This is accounted for by the use of a "cut-off Reynolds number,"

- which is determined from Eq. (12.28) or (12.29) using the characteristic length *l* (feet) and a skin-roughness value "*k*" based upon Table 12.4.

- The lower of the actual Reynolds number and the cut-off Reynolds number should be used in Eq. (12.27).

Aerodynamics

$$\text{Subsonic:} \quad R_{\text{cutoff}} = 38.21(\ell/k)^{1.053} \tag{12.28}$$

$$\text{Transonic or Supersonic:} \quad R_{\text{cutoff}} = 44.62(\ell/k)^{1.053} M^{1.16} \tag{12.29}$$

Table 12.4 Skin roughness value (k)

Surface	k (ft)
Camouflage paint on aluminum	3.33×10^{-5}
Smooth paint	2.08×10^{-5}
Production sheet metal	1.33×10^{-5}
Polished sheet metal	0.50×10^{-5}
Smooth molded composite	0.17×10^{-5}

- Once laminar and turbulent flat-plate skin friction coefficients have been calculated, an "average" coefficient can be calculated as the weighted average of the two.

- This requires estimation of the percentage of laminar flow which can be attained.

Aerodynamics

Component Form Factors

- Form factors for subsonic-drag estimation are presented in Eqs. (12.30-12.32).

- These are considered valid up to the drag divergent Mach number.

Wing, Tail, Strut, and Pylon:

$$FF = \left[1 + \frac{0.6}{(x/c)_m}\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4\right]\left[1.34M^{0.18}(\cos\Lambda_m)^{0.28}\right] \tag{12.30}$$

Fuselage and Smooth Canopy:

$$FF = \left(1 + \frac{60}{f^3} + \frac{f}{400}\right) \tag{12.31}$$

Nacelle and Smooth External Store:

$$FF = 1 + (0.35/f) \tag{12.32}$$

where

$$f = \frac{\ell}{d} = \frac{\ell}{\sqrt{(4/\pi)\,A_{max}}} \tag{12.33}$$

Aerodynamics

- In Eq. (12.30), the term *"$(x/c)_m$"* is the chordwise location of the airfoil maximum thickness point.

- For most low-speed airfoils, this is at about 0.3 of the chord.

- For high-speed airfoils this is at about 0.5 of the chord.

- Λ_m refers to the sweep of the maximum-thickness line.

- A tail surface with a hinged rudder or elevator will have a form factor about 10% higher than predicted by Eq. (12.30)

- due to the extra drag of the gap between the tail surface and its control surface.

Aerodynamics

- Equation (12.31) is mainly used for estimation of the fuselage form factor,

- but can also be used for a blister or fairing such as a pod used for landing-gear stowage.

- A square-sided fuselage has a form factor about 40% higher than the value estimated with Eq. (12.31) due to additional separation caused by the corners.

- This can be somewhat reduced by rounding the corners.

- A flying boat hull has a form factor about 50% higher, and a float has a form factor about three times the estimated value.

Aerodynamics

- Equation (12.31) will predict the form factor for a smooth, one-piece fighter canopy such as seen on the F-16.

- For a typical two-piece canopy with a fixed but streamlined windscreen (i.e., F-15), the form factor calculated with Eq. (12.31) should be increased by about 40%.

- A canopy with a flat sided windscreen has a form factor about three times the value estimated with Eq. (12.31).

Aerodynamics

- The external boundary-layer diverter for an inlet mounted on the fuselage can have a large drag contribution.
- Equations (12.34) and (12.35) estimate the form factors to use for a double-wedge and single-wedge diverter,
- where the Reynolds number is determined using *l* and the wetted area is defined as shown in Fig. 12.20.
- Remember to double the drag if there are two inlets.

$$\text{Double Wedge:} \quad FF = 1 + (d/\ell) \tag{12.34}$$

$$\text{Single Wedge:} \quad FF = 1 + (2d/\ell) \tag{12.35}$$

Aerodynamics

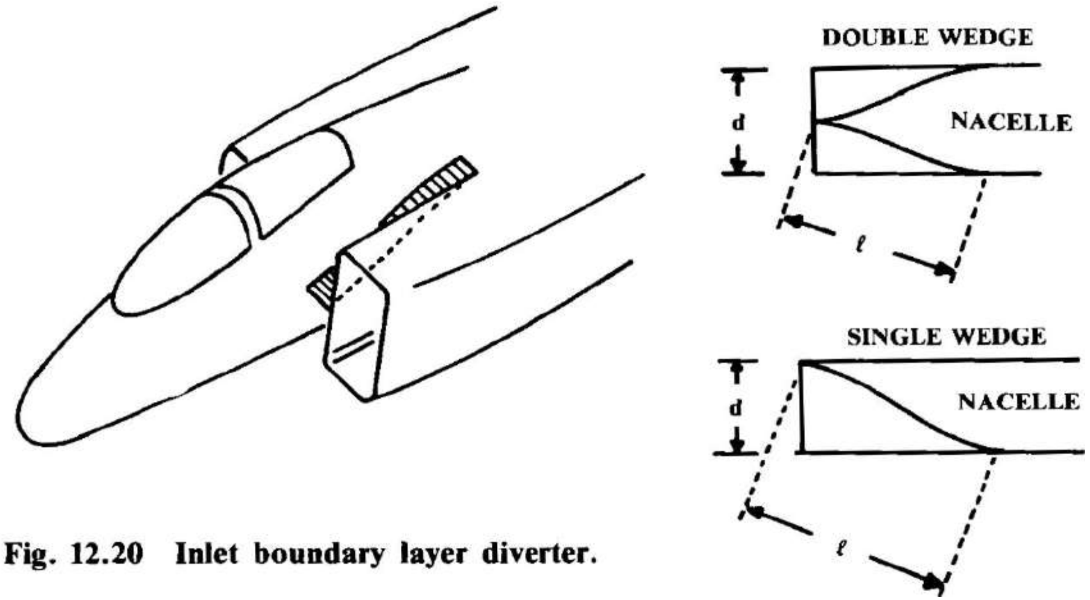


Fig. 12.20 Inlet boundary layer diverter.

Aerodynamics

Component Interference Factors

- Parasite drag is increased due to the mutual interference between components.
- For a nacelle or external store mounted directly on the fuselage or wing, the interference factor *Q* is about 1.5.
- If the nacelle or store is mounted less than about one diameter away, the *Q* factor is about 1. 3.
- If it is mounted much beyond one diameter, the *Q* factor approaches 1.0.
- Wing tip-mounted missiles have a *Q* factor of about 1.25.

Aerodynamics

- For a high-wing, a mid-wing, or a well filleted low wing, the interference will be negligible so the Q factor will be about 1.0.

- An unfilletted low wing can have a Q factor from about 1.1-1.4.

- The fuselage has a negligible interference factor (Q = 1.0) in most cases.

- Also, Q = 1.0 for a boundary-layer diverter.

- For tail surfaces, interference ranges from about three percent (Q = 1.03) for a clean V-tail to about eight percent for an H-tail.

- For a conventional tail, four to five percent may be assumed.

Aerodynamics

- Component parasite drags can now be determined using Eq. (12.24) and the skin-friction coefficients, form factors, and interference factors.

$$(C_{D_0})_{\text{subsonic}} = \frac{\Sigma(C_{f_c} FF_c Q_c S_{\text{wet}_c})}{S_{\text{ref}}} + C_{D_{\text{misc}}} + C_{D_{\text{L\&P}}} \qquad (12.24)$$

Miscellaneous Drags

- The drag of miscellaneous items can be determined separately using a variety of empirical graphs and equations, and then adding the results to the parasite drags determined above.

- While the drag of smooth external stores can be estimated using Eq. (12.32), the majority of external stores are in fact not very smooth.

- Figures 12.21 and 12.22 provide drag estimates for external fuel tanks and weapons, presented as drag divided by dynamic pressure (D/q).

Aerodynamics

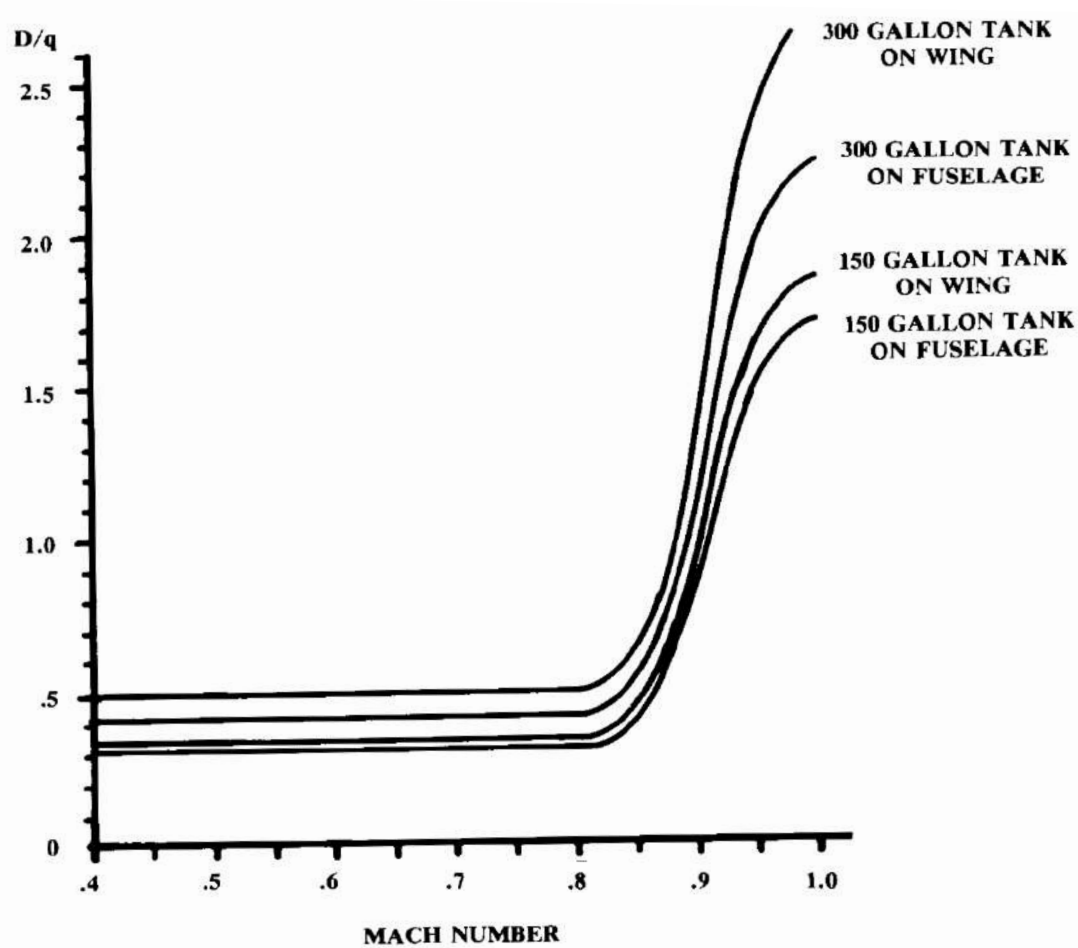


Fig. 12.21 External stores drag-fuel tanks.

Aerodynamics

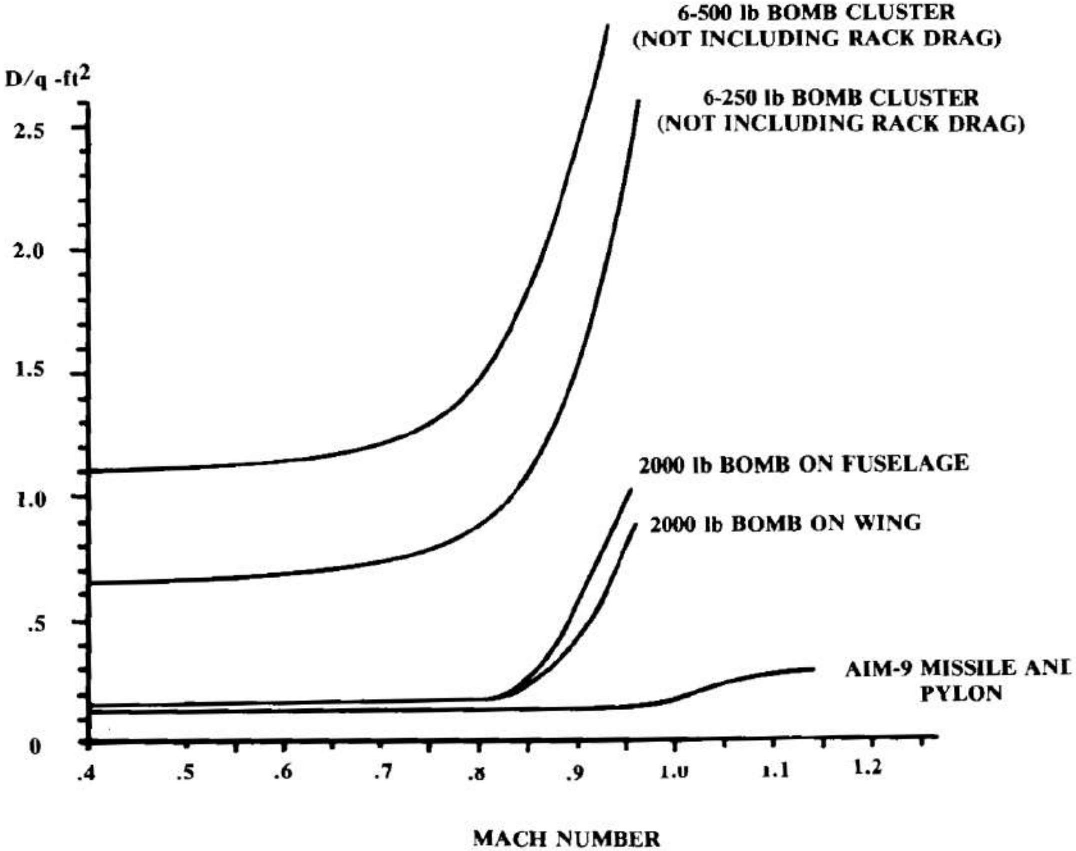


Fig. 12.22 Bomb and missile drag.

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Aerodynamics

- *D/q* has units of square feet, and so is sometimes called the "drag area.«

- *D/q* divided by the wing reference area yields the miscellaneous parasite drag coefficient.

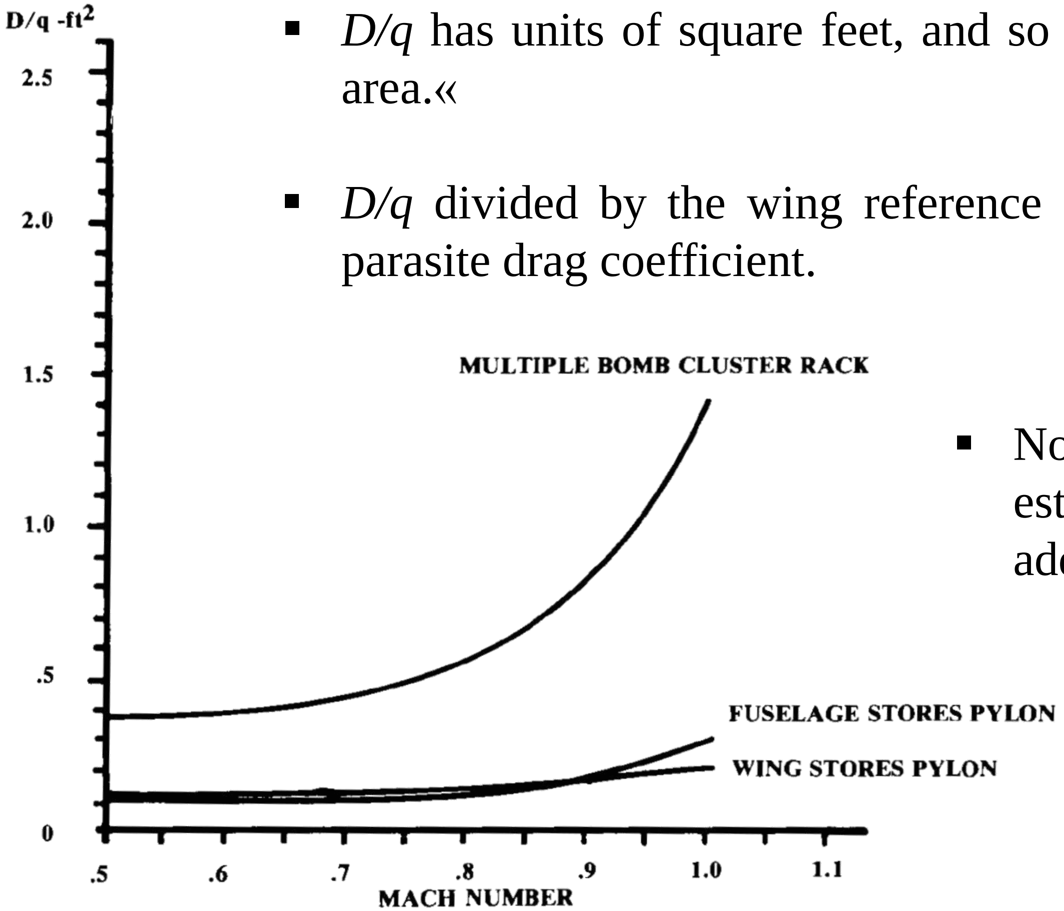


Fig. 12.23 Pylon and bomb rack drag.

- Note that pylon and bomb-rack drag as estimated using Fig. 12.23 must be added to the store drag.

Aerodynamics

- Most transport and cargo aircraft have a pronounced upsweep to the aft fuselage (Fig. 12.24).

Fig. 12.24 Fuselage upsweep.

- This increases the drag beyond the value calculated using Eq. (12.31).

- This extra drag is a complicated function of the fuselage cross-sectional shape and the aircraft angle of attack,

- but can be approximated using Eq. (12.36) where "*u*" is the upsweep angle (radians) of the fuselage centerline and A_{max} is the maximum cross-sectional area of the fuselage.

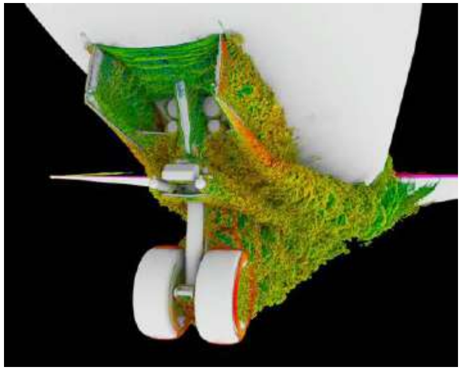
$$D/q_{\text{upsweep}} = 3.83u^{2.5}A_{\max} \tag{12.36}$$

Aerodynamics

- The landing-gear drag is best estimated by comparison to test data for a similar gear arrangement.
- If such data is not available, the gear drag can be estimated as the summation of the drags of the wheels, struts, and other gear components using the data in Table 12.5

Table 12.5 Landing gear component drags

	$\frac{D/q}{\text{Frontal area (Ft}^2\text{)}}$
Regular wheel and tire	0.25
Second wheel and tire in tandem	0.15
Streamlined wheel and tire	0.18
Wheel and tire with fairing	0.13
Streamline strut $(1/6 < t/c < 1/3)$	0.05
Round strut or wire	0.30
Flat spring gear leg	1.40
Fork, bogey, irregular fitting	1.0–1.4



Aerodynamics

- These values times the frontal area of the indicated component yield *D/q* values, which must be divided by the wing reference area to obtain parasite drag coefficients.

- To account for mutual interference it is suggested that the sum of the gear component drags be multiplied by 1.2.

- Also, the total gear drag should be increased by about 7% for a retractable landing gear in which the gear wells are left open when the gear is down.

Aerodynamics

- Flaps affect both the parasite and induced drag.
- The flap contribution to parasite drag is caused by the separated flow above the flap, and can be estimated using Eq. (12.37) for most types of flap.

$$\Delta C_{D_{0_{\text{flap}}}} = 0.0023 \frac{\text{flap span}}{\text{wing span}} \delta_{\text{flap}} \qquad (12.37)$$

where δ_{flap} is in deg.

- Note that this is referenced to wing area.
- Typically the flap deflection is about 60-70 deg for landing and about 20-40 deg for takeoff.
- Light aircraft usually take off with no flaps.

Aerodynamics

- Many aircraft have some form of speed brake.
- Typically these are plates which extend from the fuselage or wing.
- Fuselage-mounted speed brakes have a *D/q* of about 1.0 times the speed-brake frontal area,
- while wing mounted speed brakes have a *D/q* of about 1.6 times their frontal area if mounted at about the 60% of chord location.

Aerodynamics

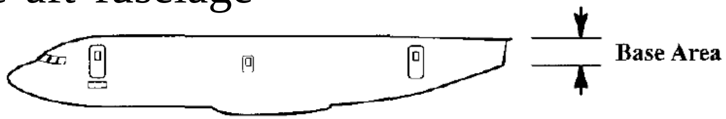
- Speed brakes mounted on top of the wing will also disturb the airflow and spoil the lift, and so are called "spoilers."

A 320 speed brakes

Aerodynamics

- Base area produces a drag according to Eqs. (12.38) and (12.39).

- "A_{base}" includes any aft-facing flat surfaces as well as the projected aft-facing area for any portions of the aft fuselage that experience highly-separated airflow.



- Roughly speaking, this should be expected any place where the aft fuselage angle to the freestream exceeds about 20 deg.

- As previously mentioned, a pusher propeller may prevent aft-fuselage separation despite an aft fuselage angle of 30 deg or more.

$$\text{Subsonic:} \quad (D/q)_{\text{base}} = [0.139 + 0.419(M - 0.161)^2]A_{\text{base}} \tag{12.38}$$

$$\text{Supersonic:} \quad (D/q)_{\text{base}} = [0.064 + 0.042(M - 3.84)^2]A_{\text{base}} \tag{12.39}$$

Aerodynamics

- Fighter-type canopies have been discussed above.
- For transport and light aircraft windshields that smoothly fair into the fuselage, an additional *D/q* of about 0.07 times the windshield frontal area is suggested.
- A sharp-edged, poorly-faired windshield has an additional *D/q* of about 0.15 times its frontal area.
- An open cockpit has a *D/q* of about 0.50 times the windshield frontal area.
- For an aircraft with an unenclosed cockpit, such as a hang-glider or ultralight, a seated person has a *D/q* of about 6 ft^2.
- This reduces to a *D/q* of 1.2 ft^2 in the prone position.

Aerodynamics

- An arresting hook for carrier operation adds a *D/q* of about 0.15 ft2.
- The smaller emergency arresting hook for Air Force aircraft adds a *D/q* of about 0.10 ft^2.
- Machine-gun ports add a *D/q* of about 0.02 ft^2 per gun.
- A cannon port such as for the M61 adds a *D/q* of about 0.2 ft^2.

Aerodynamics

Leakage and Protuberance Drag

- Leaks and protuberances add drag that is difficult to predict by any method.

- Leakage drag is due to the tendency of an aircraft to "inhale" through holes and gaps in high-pressure zones, and "exhale" into the low pressure zones.

- The momentum loss of the air "inhaled" contributes directly to drag, and the air "exhaled" tends to produce additional airflow separation.

- Protuberances include antennas, lights, and such manufacturing defects as protruding rivets and rough or misaligned skin panels.

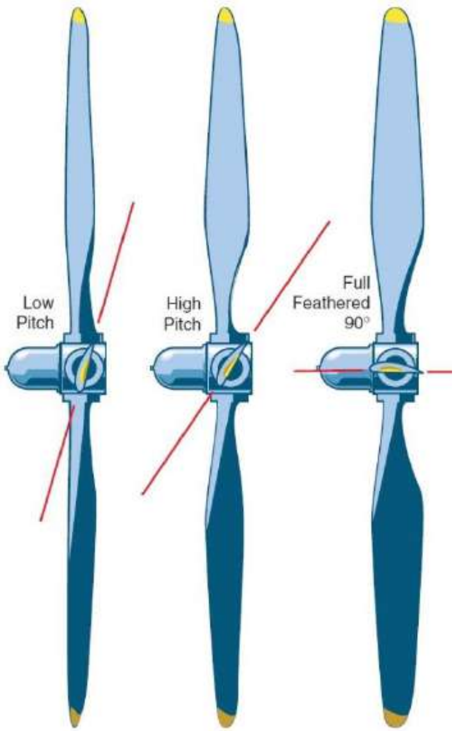
- Typically these drag increments are estimated as a percent of the total parasite drag.

Aerodynamics

- For a normal production aircraft, leaks and protuberance drags can be estimated as
  - about 2-5% of the parasite drag for jet transports or bombers,
  - 5-10% for propeller aircraft, and
  - 10-15% for current- design fighters (5-10% for new-design fighters).
- An aircraft with variable-sweep wings will have an additional protuberance drag of about 3 % due to the gaps and steps of the wing pivot area.

Stopped-Propeller and Windmilling Engine Drags

- The specifications for civilian and military aircraft require takeoff and climb capabilities following an engine failure.
- Not only does this reduce the available thrust, but the drag of the stopped propeller or windmilling engine must be considered.
- Data on the drag of a stopped or windmilling propeller are normally obtained from the manufacturer.



Feathered that the blades align with the airflow

Aerodynamics

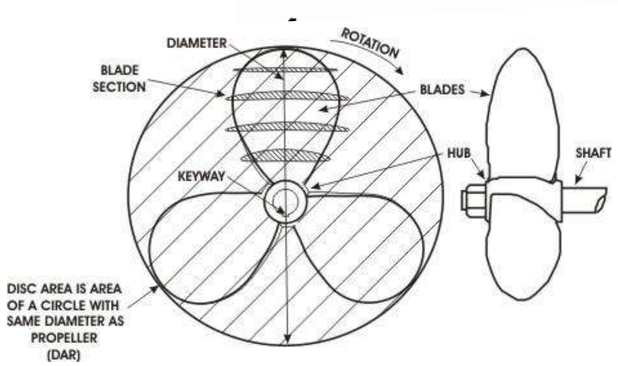
- Drag of a feathered propeller can be roughly estimated by Eq. (12.40).

$$(D/q)_{\text{feathered prop}} = 0.1\sigma A_{\text{propeller disk}} \qquad (12.40)$$

- For an unfeathered, stopped propeller, 0.1 term is replaced by 0.8.

- For jet engines, subsonic drag coefficient of a windmilling turbojet engine will be about

$$D/q)_{\text{windmilling jet}} = 0.3A_{\text{engine front face}} \qquad (12.41)$$



- The propeller "solidity" (σ), the ratio between the total blade area and the propeller disk area. This can be shown to equal the number of blades divided by the blade aspect ratio and π.

Supersonic Parasite Drag

- The supersonic parasite drag is calculated in a similar fashion to the subsonic drag, with two exceptions.
- First, the supersonic skin-friction drag does not include adjustments for form factors or interference effects (i.e., *FF = Q = 1.0*).
- Second, a new term, wave drag, is added.
- This accounts for the pressure drag due to shock formation.
- Supersonic parasite-drag buildup is defined in Eq. (12.42):

$$C_{D0_{\text{supersonic}}} = \frac{\Sigma(C_{f_c} S_{\text{wet}_c})}{S_{\text{ref}}} + C_{D_{\text{misc}}} + C_{D_{\text{L\&P}}} + C_{D_{\text{wave}}} \qquad (12.42)$$

Aerodynamics

- The wave drag in supersonic flight will often be greater than all the other drag put together.
- Wave drag is pressure drag due to shocks, and is a direct result of the way in which the aircraft's volume is distributed.
- An ideal volume distribution is produced by the Sears-Haack body, which was shown in Fig. 8.2.
- A Sears-Haack body, as defined by Eq. (12.43), has a wave drag as in Eq. (12.45).
- This is the minimum possible wave drag for any closed-end body of the same length and total volume.

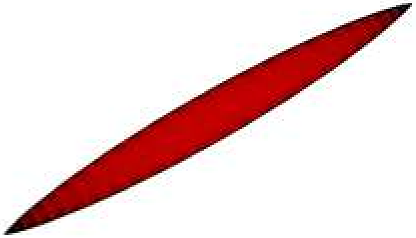
Aerodynamics

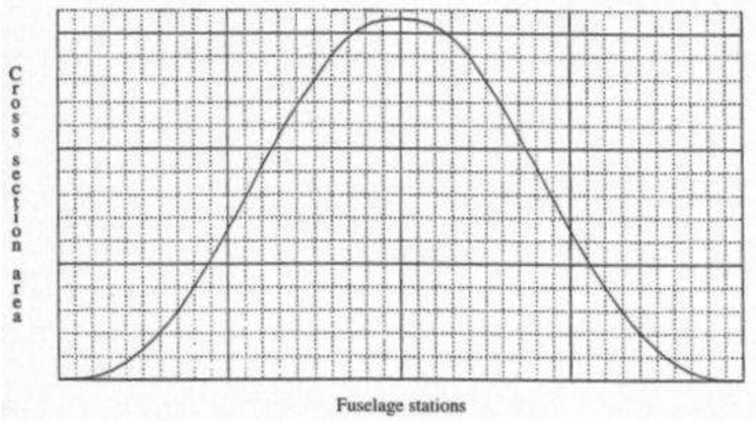
$$\frac{r}{r_{\max}} = \left[1 - \left(\frac{x}{\ell/2}\right)^2\right]^{0.75} \qquad (12.43)$$

$$-\ell/2 \le x \le \ell/2 \qquad (12.44)$$

where

r = the cross-section radius
ℓ = the longitudinal dimension





Aerodynamics

$$(D/q)_{\text{wave}} = \frac{9\pi}{2}\left(\frac{A_{\max}}{\ell}\right)^2 \qquad (12.45)$$

where $A_{\max}$ is the maximum cross-sectional area.

- Wave drag at Mach 1.0 is minimized when the aircraft has a volume distribution identical to that of a Sears-Haack body.
- Drag is reduced when the volume distribution is changed to more resemble the Sears-Haack's.
- At Mach 1.0, the wave drag is based upon the aircraft's cross-sectional areas found by the intersection of the aircraft
- and an infinite plane set at an angle perpendicular (90 deg) to the freestream direction.

Aerodynamics

- At speeds higher than Mach 1.0, the wave drag still depends upon the volume distribution as before, but with one major exception.
- At higher Mach numbers the volume distribution is based upon aircraft cross sections that are determined by intersecting the aircraft with "Mach planes,"
- set at the appropriate Mach angle to the freestream direction.
- A Mach plane may be rolled about the freestream direction to any roll angle.
- Figure 12.25 shows two roll angles.
- Note that the different Mach plane roll angles produce entirely different volume-distribution plots.

Aerodynamics

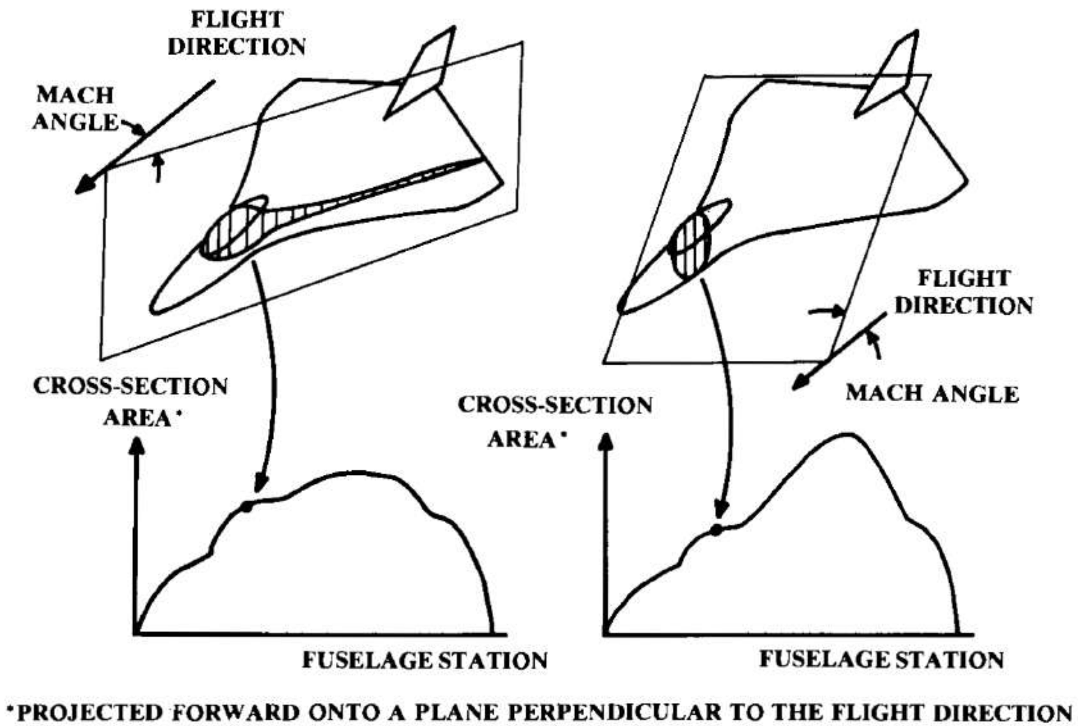


Fig. 12.25 Mach-plane cut volume distribution (two roll angles).

Aerodynamics

- For each Mach-plane roll angle, a volume-distribution plot can be prepared by taking Mach-plane cuts at a number of longitudinal locations.

- According to linear wave drag theory the supersonic wave drag at Mach numbers greater than 1.0 is determined by

- averaging the wave drags of the Mach-plane-cut volume distributions for different roll angles.

- For preliminary wave drag analysis at $M \geq 1.2$, without use of a computer, a correlation to the Sears-Haack body wave drag is presented in Eq. (12.46),

$$(D/q)_{\text{wave}} = E_{\text{WD}} \left[1 - 0.386(M - 1.2)^{0.57} \left(1 - \frac{\pi \Lambda_{\text{LE-deg}}^{0.77}}{100} \right) \right] (D/q)_{\text{Sears-Haack}} \tag{12.46}$$

- where the Sears-Haack D/q is from Eq. (12.45).

Aerodynamics

- The maximum cross-sectional area, A_{max}, is determined from the aircraft volume-distribution plot.
- Inlet capture area should be subtracted from A_{max} .
- The length term l is the aircraft length except that any portion of the aircraft with a constant cross-sectional area should be subtracted from the length.
- E_{WD} is an empirical wave drag efficiency factor.

Aerodynamics

- A very clean aircraft with a smooth volume distribution E_{WD} may be 1.2.
- A more typical supersonic fighter has an E_{WD} of about 1.4-2.0.
- For poor supersonic design, E_{WD} may be 2-3 (F-15 has 2.6).
- Note that efficiency factor is less important than the fitness ratio as represented by (A_{max}/l).
- This term is squared.

SST: silent supersonic transport

Aerodynamics

Transonic Parasite Drag

- The definition of what speed constitutes M_{DD} is arbitrary, and several definitions are in use.
- The Boeing definition is that M_{DD} is where the drag rise reaches 20 counts (0.002).
- M_{DD} (Boeing) is usually about 0.08 Mach above the critical Mach number.
- The Douglas definition also used by USAF, is that M_{DD} is the Mach number at which the rate of change in parasite drag with Mach number first reaches 0.1. $(\mathrm{d}C_{D_0}/\mathrm{d}M)$

Aerodynamics

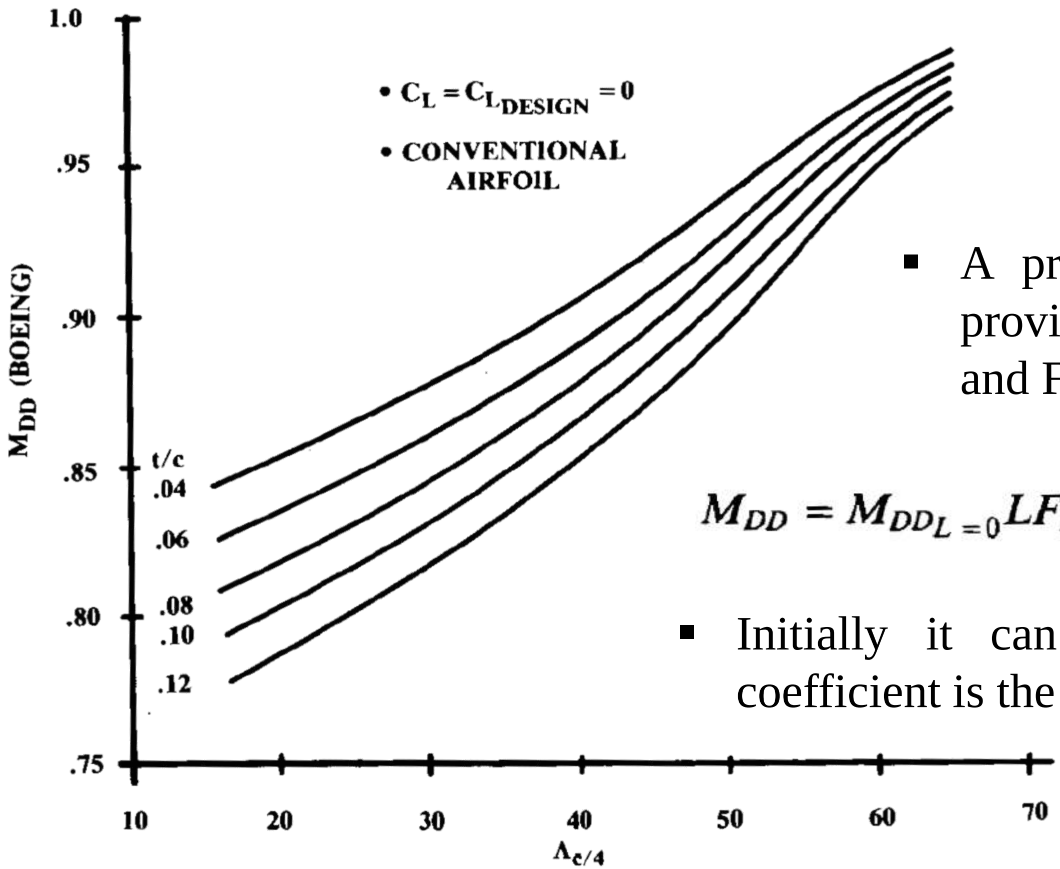


Fig. 12.26 Wing drag-divergence Mach number.

- A preliminary estimate of wing M_{DD} is provided by Eq. (12.47) using Fig. 12.26 and Fig. 12.27.

$$M_{DD} = M_{DD_{L=0}} LF_{DD} - 0.05 C_{L_{\text{design}}} \tag{12.47}$$

- Initially it can be assumed that the design lift coefficient is the same as the lift coefficient at cruise.

Aerodynamics

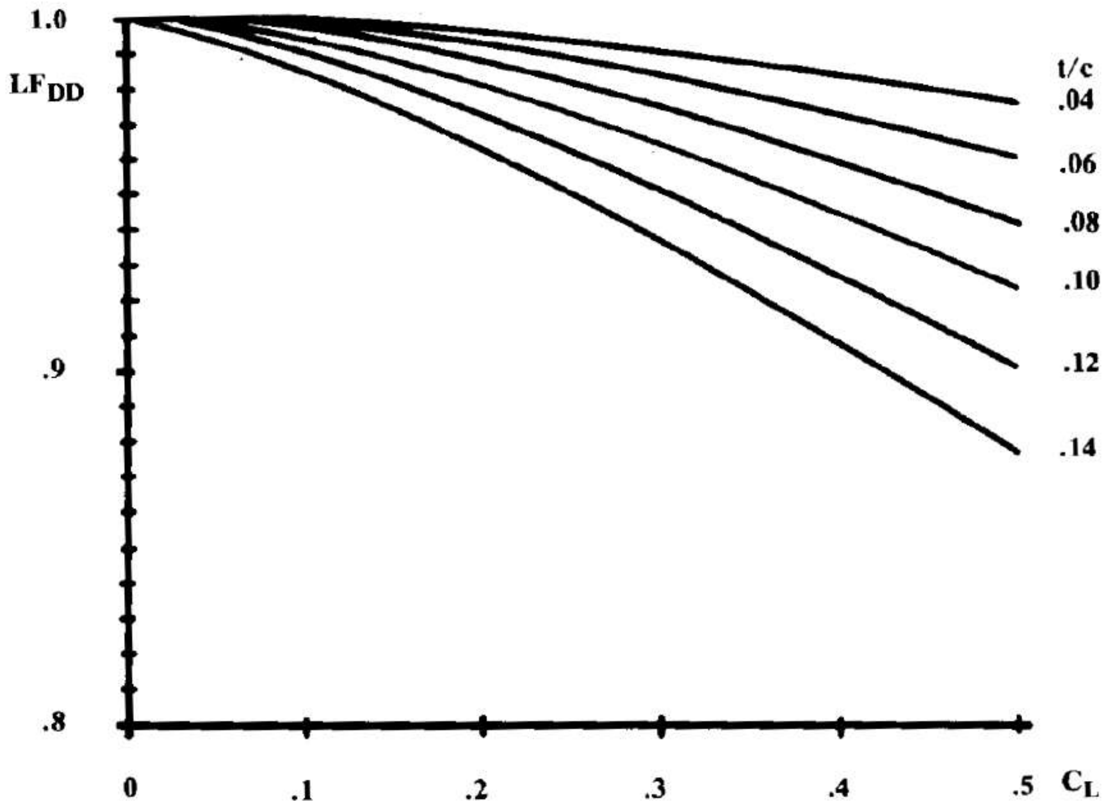


Fig. 12.27 Lift adjustment for M_{DD}.

Aerodynamics

- If the fuselage is relatively blunt it will experience shock formation before the wing does.
- In this case, M_{DD} is set by the shape of the forebody.
- Body M_{DD} can be estimated using Fig. 12.28, where "L_n" is the length from the nose to the longitudinal location at which the fuselage cross section becomes essentially constant.
- d is the body diameter at that location.
- If the fuselage is noncircular, d is an equivalent diameter based upon the fuselage cross-sectional area.
- Determine both wing and fuselage M_{DD}, and use the lower value.

Aerodynamics

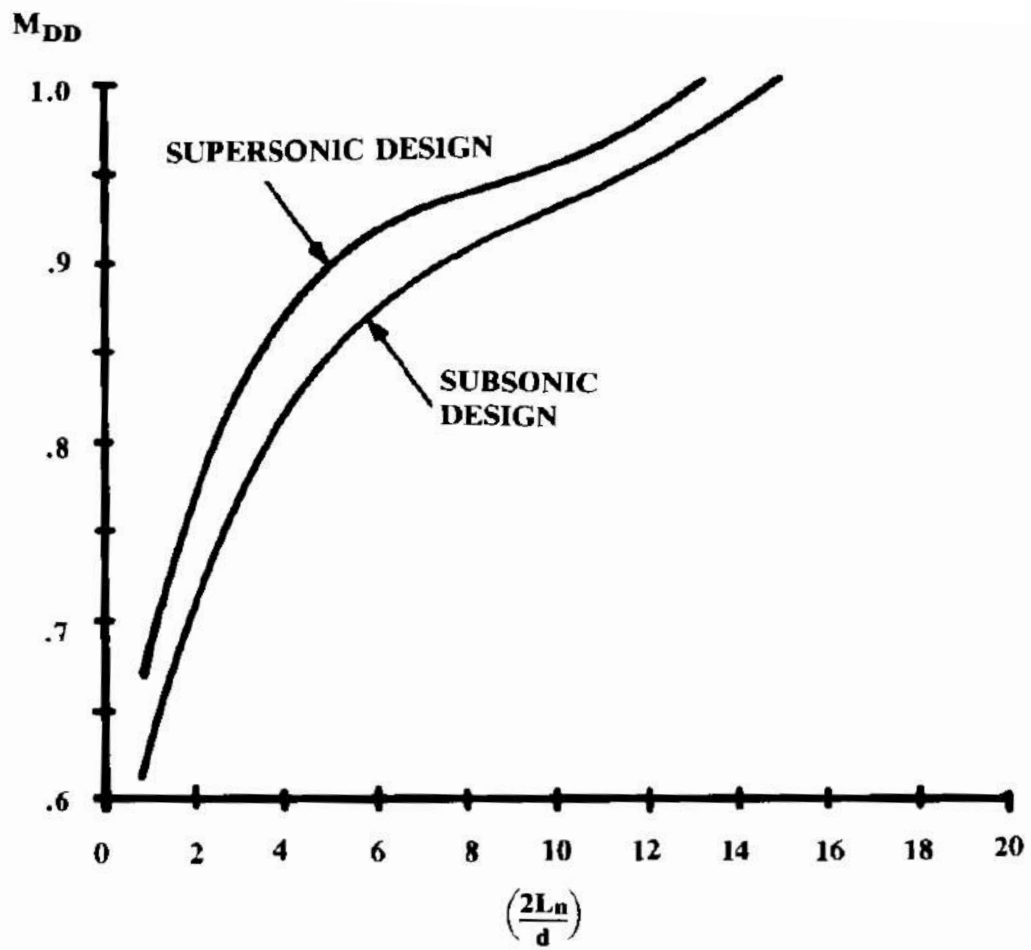


Fig. 12.28 Body drag-divergent Mach number.

Aerodynamics

- However, for initial analysis the drag rise may be graphically estimated using a few rules of thumb, as shown in Fig. 12.29.
- The drag at and above Mach 1.2 (labeled A in the figure) is determined using Eq. (12.46) (divided by wing reference area).
- The drag at Mach 1.05 (labeled B) is typically equal to the drag at Mach 1.2.
- The drag at Mach 1.0 (labeled C) is about half of the Mach 1.05 value.

Aerodynamics

M_{DD}: *Drag Divergence Mach number*
M_{CR} :*Critical Mach number*

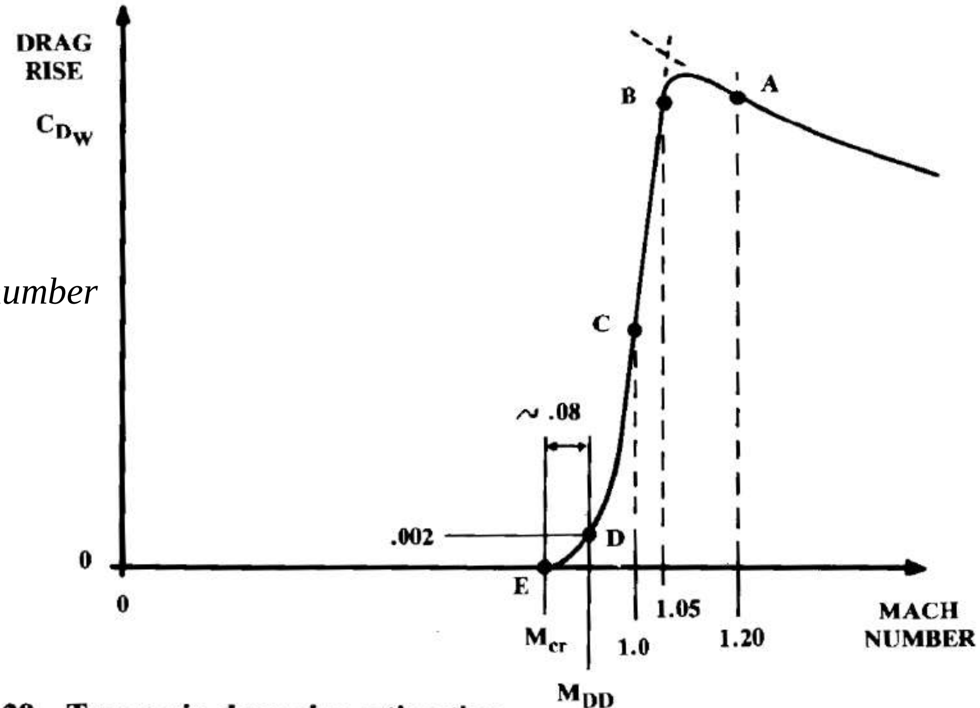


Fig. 12.29 Transonic drag rise estimation.

- The drag rise at M_{DD} is 0.002 by definition (labeled D).
- M_{CR}, the beginning of drag rise, is roughly 0.08 slower in Mach number than M_{DD} and is labeled E.

- To complete the transonic-drag-rise curve from these points, draw a straight line through points B and C, extending almost to the horizontal axis.
- Then, draw a curve from M_{CR}, through M_{DD} which fairs smoothly into the straight line as shown.
- If a smooth curve cannot be drawn, the M_{CR} point (E) should be moved until an approximately circular arc can be drawn.
- Finally, draw a smooth curve connecting B to A.
- The supersonic wave drag (point B) is determined from Eq. (12.46)

Aerodynamics

Complete Parasite-Drag Buildup

- Figure 12.30 illustrates the complete buildup of parasite drag vs Mach number for subsonic, transonic, and supersonic flight.

- The subsonic drag consists of the skin-friction drag including form factor and interference, plus miscellaneous drag and leak and protuberance drag.

- The supersonic drag includes the flat-plate supersonic skin-friction drag, miscellaneous drag, leak and protuberance drag, and wave drag.

Aerodynamics

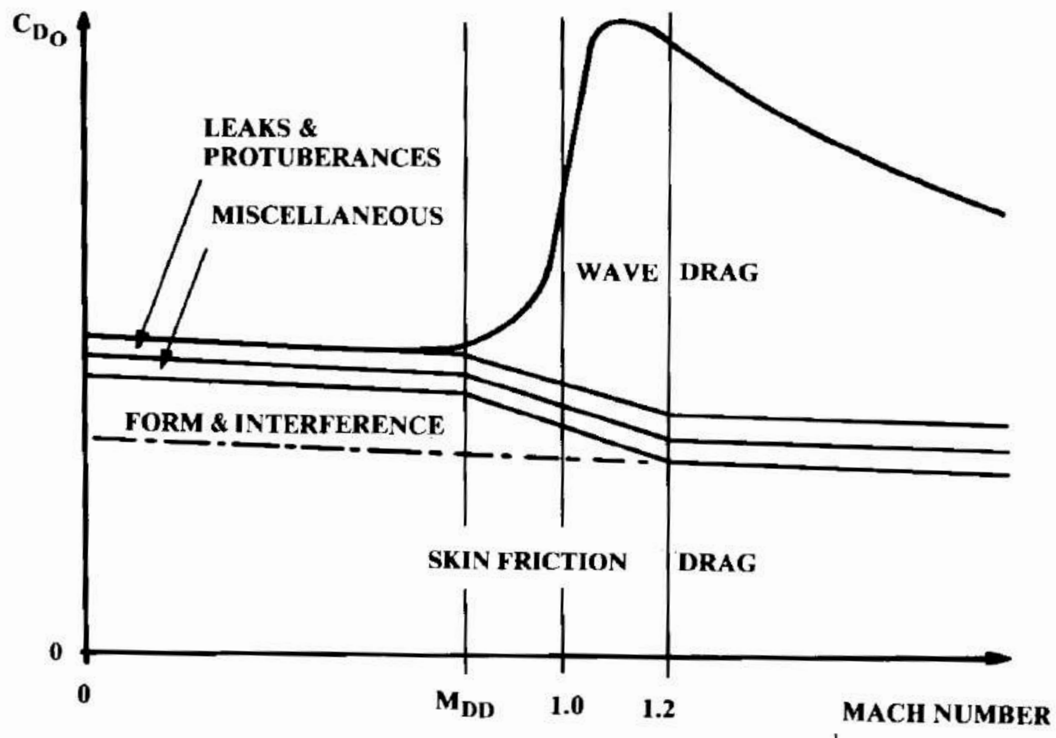


Fig. 12.30 Complete parasite drag vs Mach number.

Aerodynamics

- In Fig. 12.31, the actual parasite drag and drag rise is shown for a number of aircraft.

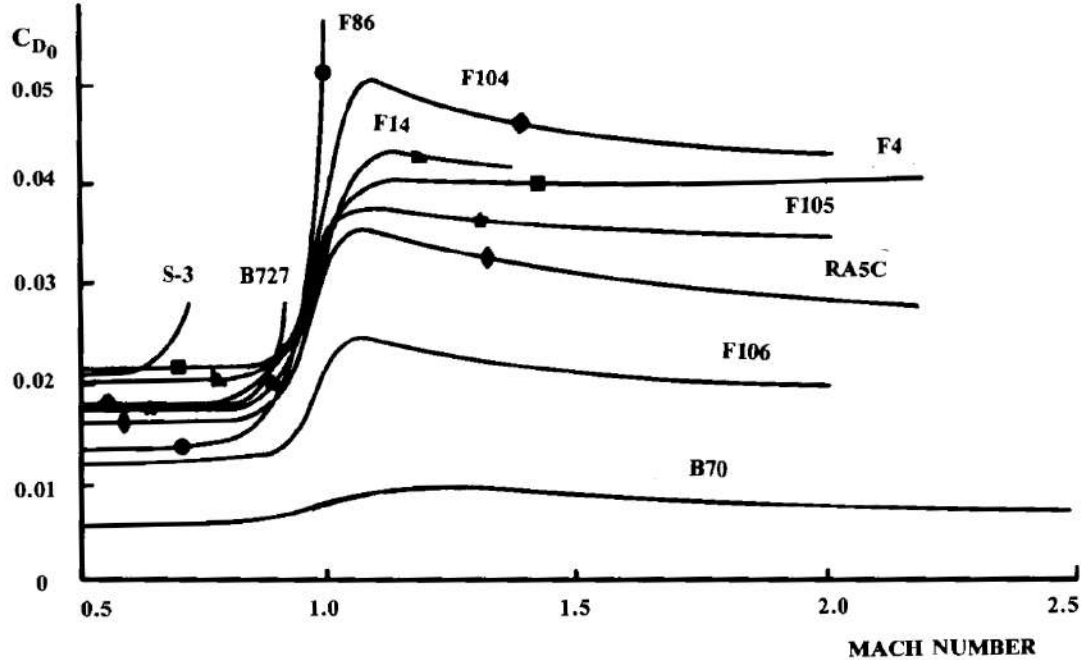


Fig. 12.31 Parasite drag and drag rise.

F-4 Phantom

Aerodynamics

Drag Due To Lift (Induced Drag)

- The induced-drag coefficient at moderate angles of attack is proportional to the square of the lift coefficient
- with a proportionality factor called the "drag-due-to-lift factor," or K.
- Two methods of estimating *K* will be presented.
- The first is the classical method based upon e, the Oswald span efficiency factor.
- The second method for the estimation of *K* is based upon the concept of leading-edge suction and provides, for high-speed designs, a better estimate of *K*.

Aerodynamics

Oswald Span Efficiency Method

$$K = \frac{1}{\pi A e} \tag{12.48}$$

- The Oswald efficiency factor is typically between 0.7 and 0.85.
- Numerous estimation methods for *e* have been developed over the years.

$$\text{Straight-Wing Aircraft: } e = 1.78(1 - 0.045A^{0.68}) - 0.64 \tag{12.49}$$

$$\text{Swept-Wing Aircraft: } e = 4.61(1 - 0.045A^{0.68})(\cos\Lambda_{LE})^{0.15} - 3.1 \tag{12.50}$$

$$(\Lambda_{LE} > 30 \text{ deg})$$

- If the wing has end-plates or winglets, the effective aspect ratio from Eq. (12 10) or (12.11) should be used in Eq. (12.48).

Aerodynamics

- At supersonic speeds, the drag-due-to-lift factor *(K)* increases substantially.

- In terms of Oswald efficiency factor, e is reduced to approximately 0.3-0.5 at Mach 1.2.

- Equation (12.52) provides a quick estimate of K at supersonic speeds.

Supersonic Speeds: $$K = \frac{A(M^2 - 1)}{4A\sqrt{M^2 - 1} - 2} \cos\Lambda_{LE} \qquad (12.52)$$

Aerodynamics

Leading-Edge-Suction Method

- A semi-empirical method for estimation of K allows for the variation of K with lift coefficient and Mach number.

- The method below for calculating K for high-speed aircraft is based upon an empirical estimate of the actual percent of leading-edge suction attainable by a wing.

- The percent of leading-edge suction a wing attains is called S.

$$K = SK_{100} + (1 - S)K_0 \tag{12.58}$$

- for K_0

$$K = \frac{\alpha}{C_L} = \frac{1}{C_{L_\alpha}}$$

- for K_{100}

$$K = \frac{1}{\pi A}$$

Aerodynamics

- When the leading-edge becomes supersonic, the suction goes to zero so the *K* value equals the 0% *K* value.
- For initial analysis, the supersonic behavior of the 100% K line may be approximated by a smooth curve, as shown in Fig. 12.34.
- This shows the typical behavior of the 100 and 0% *K* values vs Mach number.

Aerodynamics

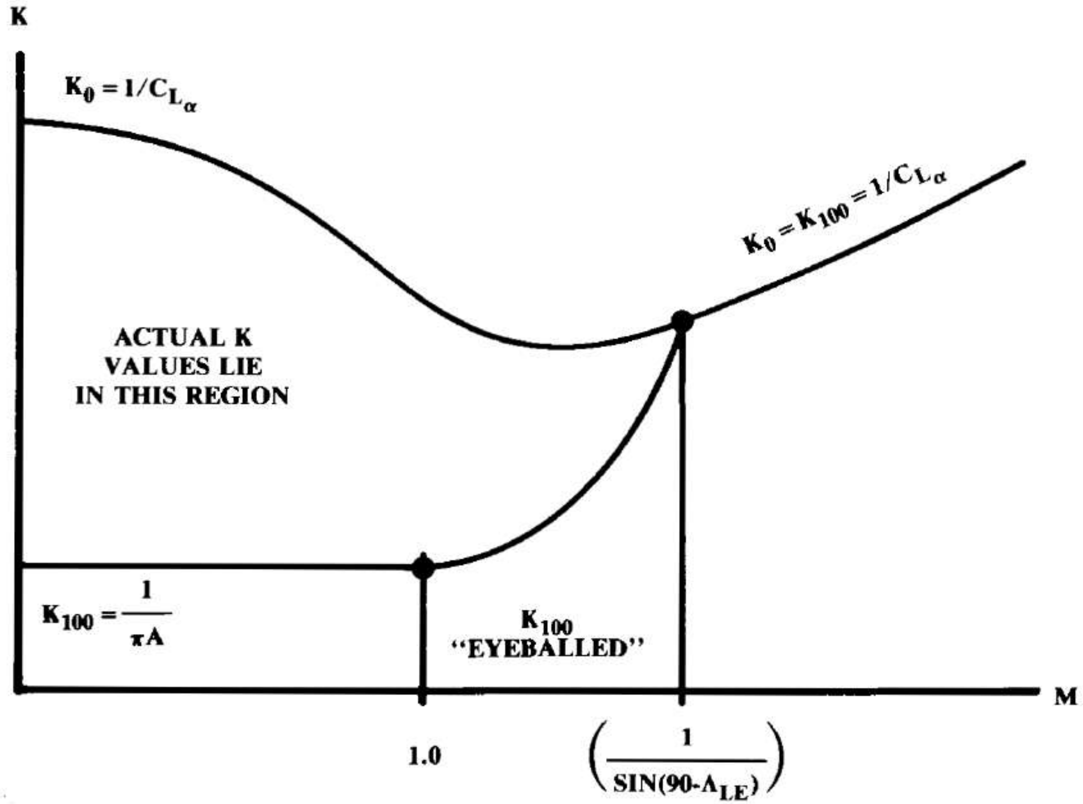


Fig. 12.34 0% and 100% K vs Mach number.

Aerodynamics

- The only unknown remaining is the value of *S*.
- *S* depends largely upon the leading-edge radius, and is also affected by the sweep and other geometric parameters.
- S is also a strong function of the wing design lift coefficient and the actual lift coefficient.
- For any wing, the value of S is at a maximum when the wing is operating at the design lift coefficient.
- For most wings, *S* equals approximately 0.9 when operating at the design lift coefficient.

Aerodynamics

- Figure. 12.35 provides a first-order estimate of the percent of leading-edge suction for a typical supersonic aircraft's wing,
- given the actual lift coefficient and the design lift coefficient

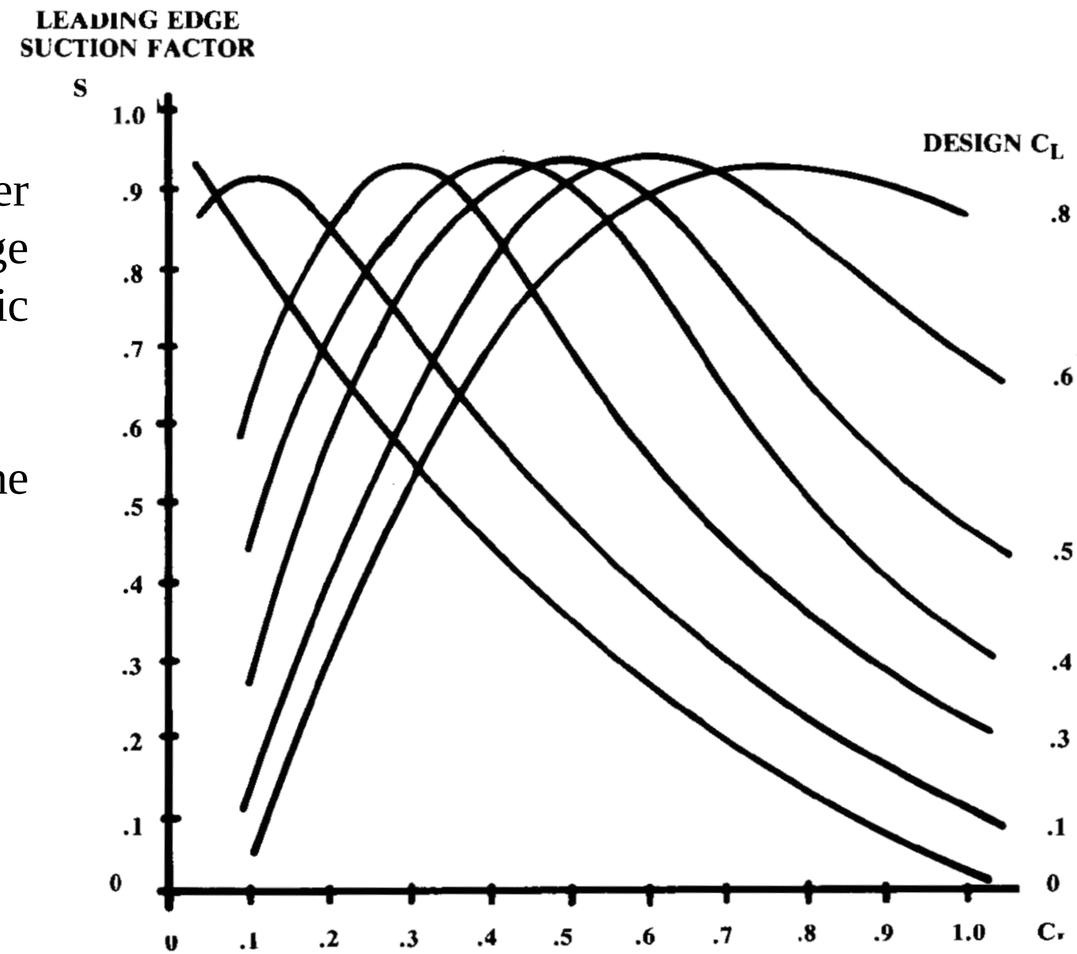


Fig. 12.35 Leading edge suction vs C_L.

Aerodynamics

- These are then used for total drag estimation via Eq. (12.4).

$$C_D = C_{D_0} + KC_L^2 \qquad (12.4)$$

Ground Effect

- When a wing is near the ground, say less than half the span away the drag due to lift can be substantially reduced.

- This effect is accounted for by multiplying K by the factor calculated in Eq. (12.61)

$$\frac{K_{\text{effective}}}{K} = \frac{33(h/b)^{1.5}}{1+33(h/b)^{1.5}} \qquad (12.61)$$

where h is wing height above ground.

Aerodynamics

- The aerodynamic methods presented above do not reflect current industry practice.

- Aircraft companies rely upon computer codes.

- The current state of the art for complex aircraft configurations the "Reynolds-Averaged Navier-Stokes," RANS has turbulence modeled statistically.

- Reynolds-Averaged codes can handle most of the complex flow phenomena.

Aircraft Design
AE 405

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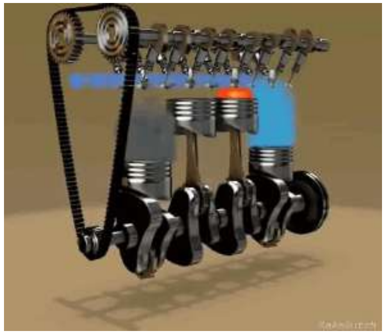
Contents

Chapter 13

Propulsion



Propulsion

- 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.

Jet-engine Thrust Considerations

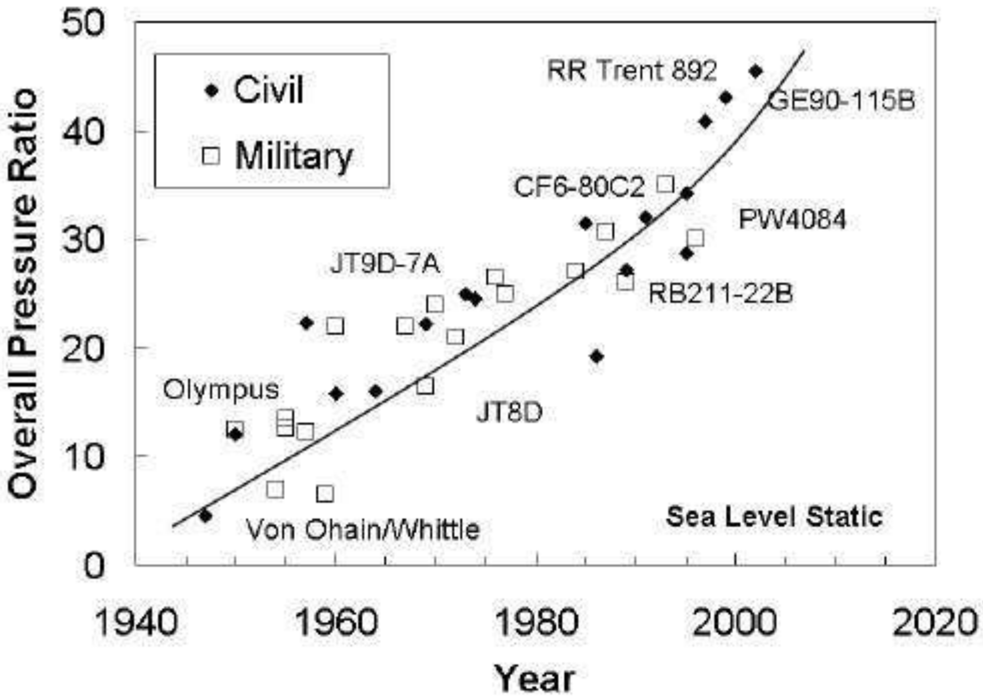
- Thrust and propulsive efficiency are strongly affected by the engine's overall pressure ratio (OPR).

- OPR is the ratio of the pressures at the engine exhaust plane and inlet front face.

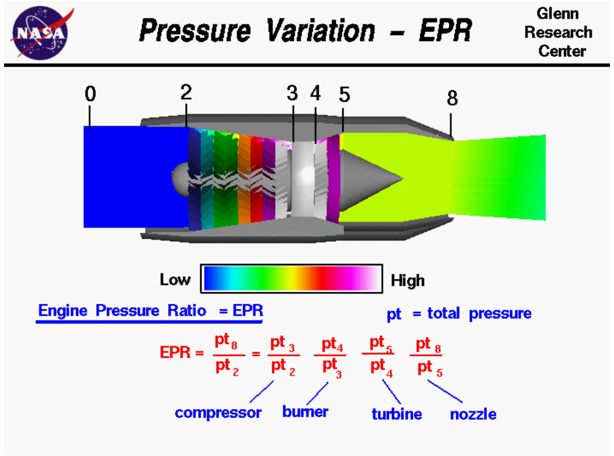
- This pressure ratio is a measure of the engine's ability to accelerate the exhaust, which produces thrust.

- OPRs usually range from about 15 to 1 to about 50 to 1.

Propulsion



Overall Pressure Ratio
Civil
Military
RR Trent 892
GE90-115B
CF6-80C2
PW4084
JT9D-7A
RB211-22B
Olympus
JT8D
Von Ohain/Whittle
Sea Level Static
0
10
20
30
40
50
1940
1960
1980
2000
2020
Year



NASA
Pressure Variation – EPR
Glenn Research Center
0
2
3
4
5
8
Low
High
Engine Pressure Ratio = EPR
pt = total pressure
EPR = pt8/pt2 = pt3/pt2 pt4/pt3 pt5/pt4 pt8/pt5
compressor
burner
turbine
nozzle

Propulsion

- Another key parameter which currently limits turbine engine performance is the turbine inlet temperature (TIT).

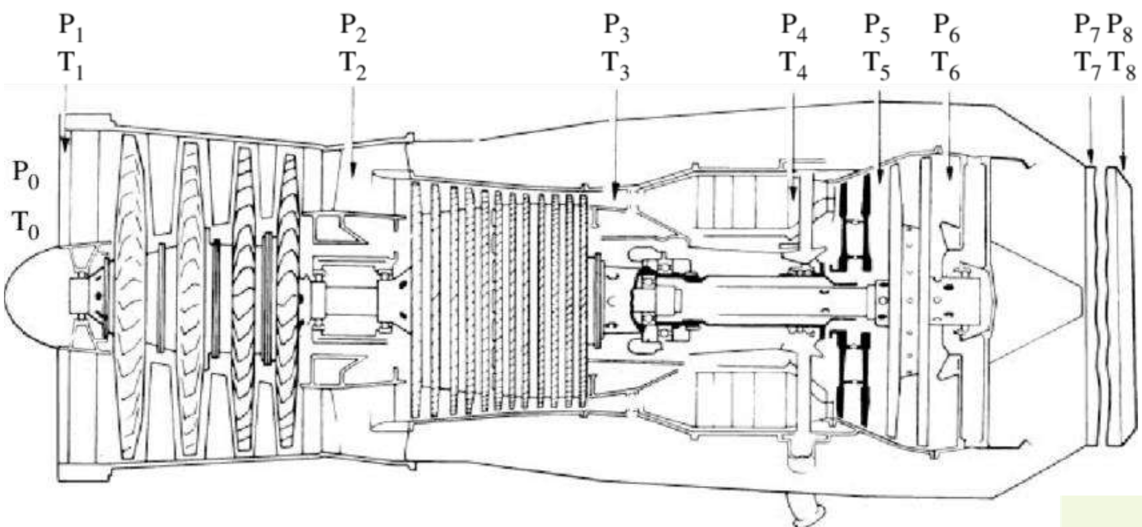
- As mentioned earlier, it would be desirable for maximum thrust and efficiency to combust at the stoichiometric air-fuel ratio of about 15 to 1.

- This produces temperatures far too high for current turbine materials, even using the best available cooling techniques.

- Instead, a "lean" mixture of about 60 to 1 (air to fuel) is used, with the extra air holding down the TIT to about 2000-2500°F.

- This results in less thrust and thermal efficiency, and so a key objective in propulsion technology development has always been the increase in allowable TIT.

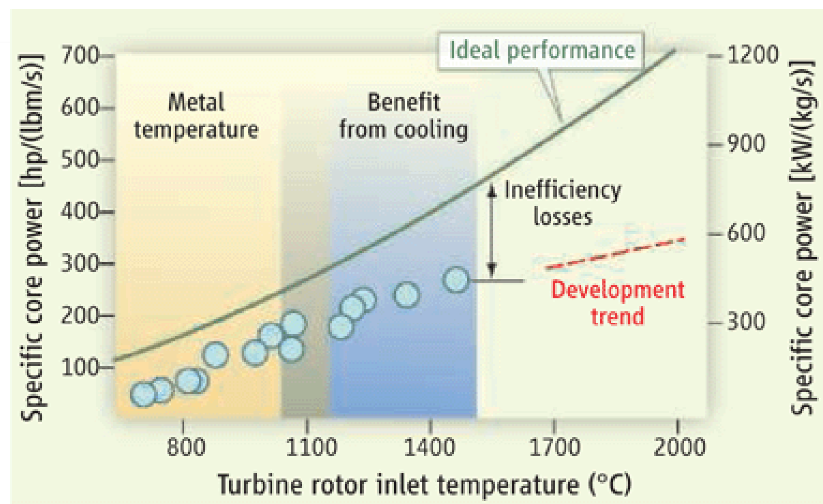
Propulsion



- Higher-temperature materials are needed to shift performance back toward the ideal limits.

P_0 T_0 ambient
P_1 T_1 inlet
P_2 T_2 low-pressure compressor delivery
P_3 T_3 high-pressure compressor delivery

P_4 T_4 turbine entry TIT
P_5 T_5 high-pressure turbine exit
P_6 T_6 low-pressure turbine exit
P_7 T_7 exhaust
P_8 T_8 propelling nozzle



Propulsion

Turbojet Installed Thrust

- Chapter 10 described statistical methods for estimating installed thrust and specific fuel consumption for jet engines.
- These are suitable for initial sizing and performance estimation.

Thrust-drag Bookkeeping

- The interactions between thrust and drag are so complex that only a bookkeeping-like approach can assure that all forces have been counted once and only once.

Propulsion

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 bookkeeping system.

- This is depicted m Fig. 13.2.

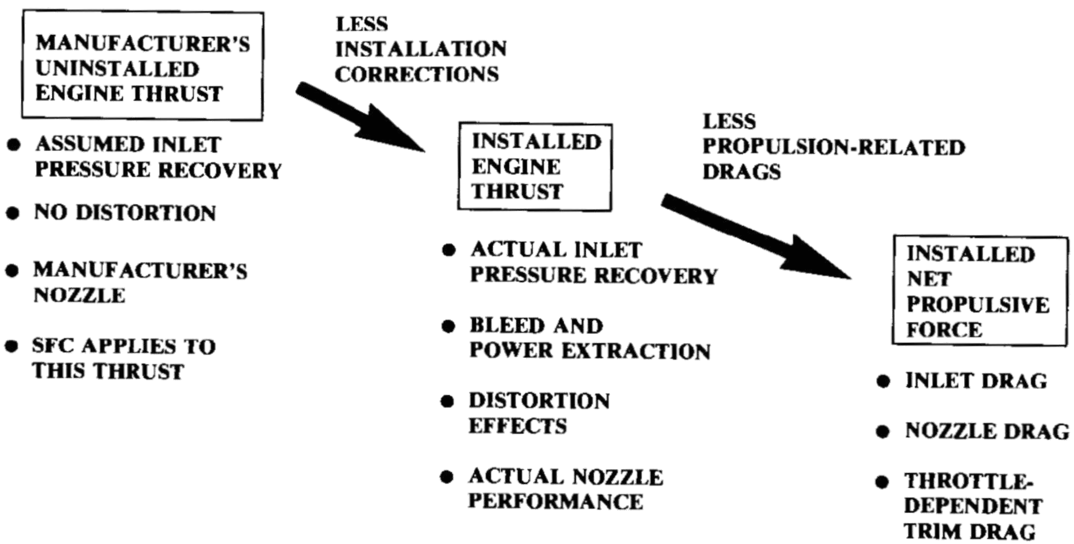


Fig. 13.2 Installed thrust methodology.

Propulsion

- Inlet pressure recovery (P_1/P_0) is the total pressure at the engine front face (1) divided by the total pressure in the freestream (0).

- For subsonic engines, it is frequently assumed that the inlet pressure recovery is perfect:

$$P_1/P_0 = 1.0$$

- For supersonic military aircraft a MilSpec formula is used.

- The "installed-engine thrust" is the actual thrust generated by the engine 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.

Propulsion

- "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.
- The installed net propulsive force is the installed engine thrust minus the inlet, nozzle, and throttle-dependent trim drags.

Installed Engine Thrust Corrections

- The manufacturer's uninstalled engine thrust is based upon an assumed inlet pressure-recovery.
- For a subsonic engine, it is typically assumed that the inlet has perfect recovery, i.e., 1.0.

Propulsion

- Supersonic military aircraft engines are usually defined using an inlet pressure-recovery of 1.0 at subsonic speeds and the inlet recovery of Eq. (13.5) (MIL-E-5008B) at supersonic speeds.

$$\left(\frac{P_1}{P_0}\right)_{\text{ref}} = 1 - 0.075(M_\infty - 1)^{1.35} \qquad (13.5)$$

- The pressure recovery loss due to internal flow in the inlet duct itself.

- Typically this adds 2-3% to the losses.

- Figure 13.3 shows this reference inlet pressure-recovery plotted vs Mach number,

- compared to the recovery available for a normal-shock inlet and external compression inlets with one, two, and three ramps.

Propulsion

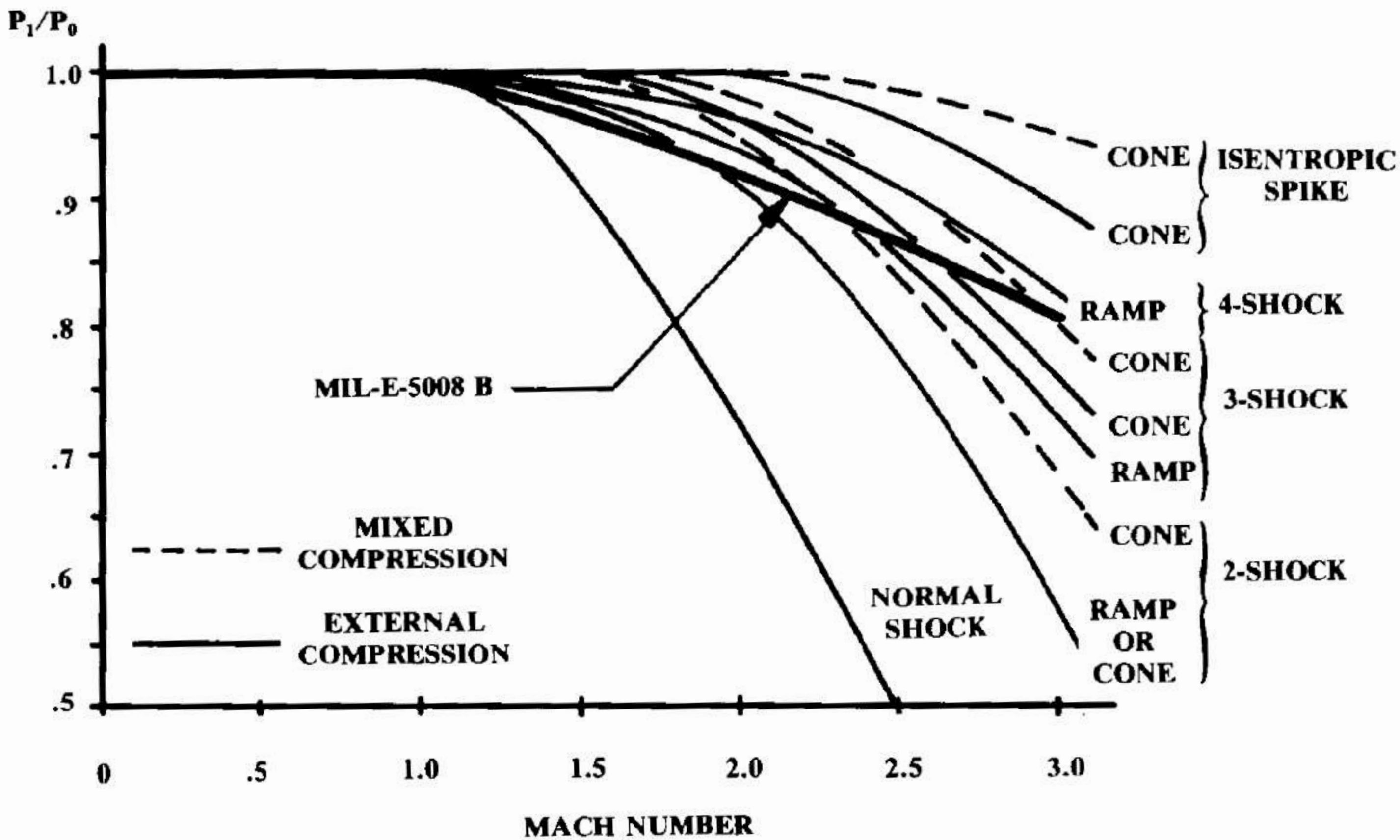


Fig. 13.3 Reference and available inlet pressure recovery.

Propulsion

- The pressure losses inside the inlet duct must also be accounted for.
- These losses are determined by the length and diameter of the duct, the presence of bends in the duct, and the internal Mach number.
- For initial evaluation of a typical inlet duct, an internal pressure recovery of 0.96 for a straight duct and 0.94 for an S duct may be used.
- The short duct of a subsonic podded nacelle will have a pressure recovery of 0.98 or better.
- Figure 13.4 provides the actual inlet pressure recoveries of some existing designs.

Propulsion

- This figure may be used for pressure-recovery estimation during early design studies.

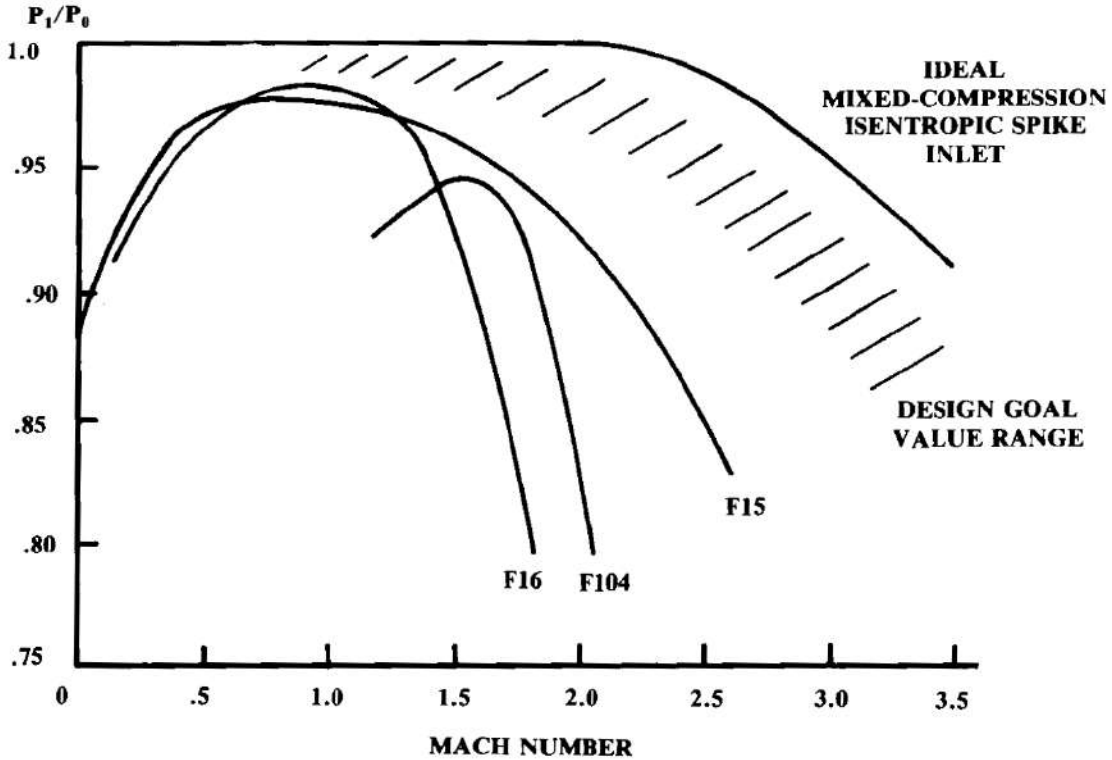


Fig. 13.4 Actual inlet pressure recoveries.

Propulsion

- Reducing inlet pressure recovery has a greater-than-proportional effect upon the engine thrust, as shown in Eq. (13.6).

$$\text{Percent thrust loss} = C_{\text{ram}}\left[\left(\frac{P_1}{P_0}\right)_{\text{ref}} - \left(\frac{P_1}{P_0}\right)_{\text{actual}}\right] \times [100] \qquad (13.6)$$

- The inlet "ram recovery correction factor (C_{ram})" is provided by the manufacturer.

- If the manufacturer's data is not available, C_{ram} may be approximated as 1.35 for subsonic flight and by Eq. (13.7) for supersonic flight.

$$\text{Supersonic:} \quad C_{\text{ram}} \cong 1.35 - 0.15(M_\infty - 1) \qquad (13.7)$$

Propulsion

- High-pressure air is bled from the engine compressor for cabin air anti-icing, and other uses.

- This engine bleed air exacts a thrust penalty that is also more-than-proportional to the percent of the total engine mass flow extracted as bleed air.

- Equation (13.8) illustrates this, where the "bleed correction factor (C_{b1eed})" is provided by the manufacturer for various flight conditions.

- For initial analysis, C_{b1eed} can be approximated as 2.0.

- The bleed mass flow typically ranges from 1-5% of the engine mass-flow.

$$\text{Percent thrust loss} = C_{\text{bleed}}\left(\frac{\text{bleed mass flow}}{\text{engine mass flow}}\right) \times [100] \qquad (13.8)$$

Propulsion

- Installed engine thrust is also affected by horsepower extraction.
- Jet engines are equipped with rotating mechanical shafts turned by the turbines.
- The electrical generators, hydraulic pumps, and other such components connect to these shafts.
- This extraction is typically less than 200 hp for a 30,000-lb-thrust engine, and can be ignored for initial analysis.
- For initial design, the distortion effect may be eliminated.
- Additionally, it is rare to use a nozzle other than that provided by the manufacturer.

Propulsion

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 different 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.

Propulsion

- When the inlet is providing exactly the amount of air the engine demands (mass flow ratio equals 1.0), the inlet drag is negligible.
- The inlet must be sized to provide enough air at the worst-case condition, when the engine demands a lot of air.
- This sets the capture area.
- Most of the time the engine demands less air than an inlet with this capture area would like to provide (i.e., mass flow ratio is less than 1.0).
- When the mass flow ratio is less than 1.0, the excess air must either be spilled before the air enters the inlet
- or bypassed around the engine via a duct that dumps it overboard (Fig. 13.5).

Propulsion

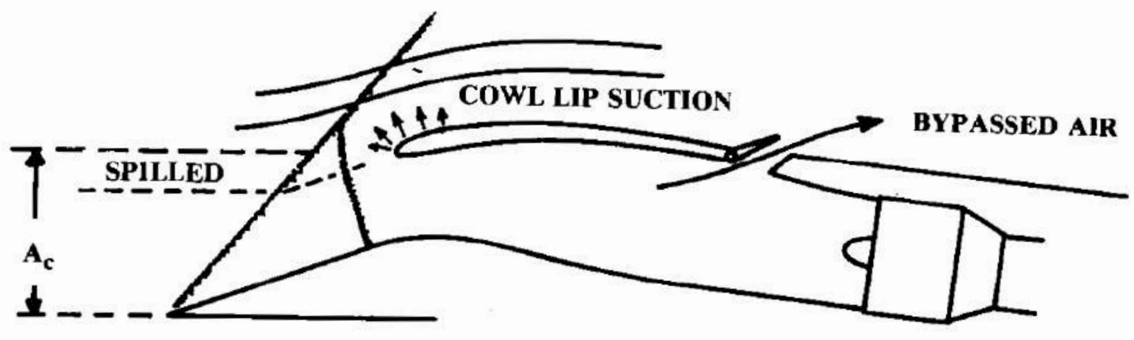


Fig. 13.5 Additive drag, cowl lip suction, and bypass subcritical operation.

- The drag from air spilled before entering the inlet is called "spillage," or additive drag.

- 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.

Propulsion

- 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.

Propulsion

- This air is dumped overboard out an aft-facing discharge exit, which is usually located a few feet behind the inlet.
- *Note:* don't confuse inlet boundary-layer bleed with the inlet boundary layer diverter.
- Calculation of bleed, bypass, and additive drag including cowl-lip suction is a complicated procedure combining analytical and empirical methods.
- To permit rapid initial analysis and trade studies, Fig. 13.6 provides a "ballpark" estimate / rough approximation of inlet drag for a typical supersonic aircraft.

Propulsion

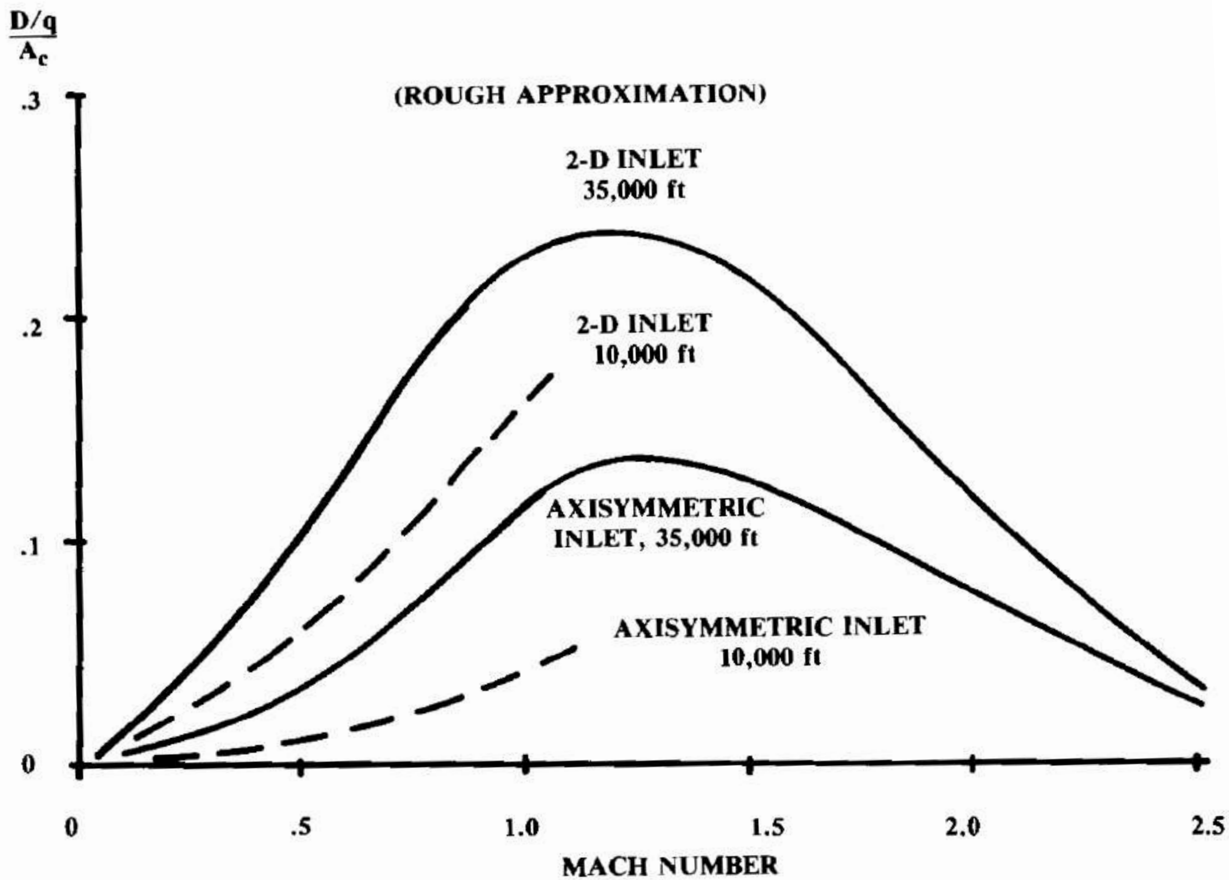


Fig. 13.6 Inlet drag trends.

Propulsion

- Nozzle drag varies with nozzle position as well as with the flight condition.
- To properly determine nozzle drag the actual nozzle geometry as a function of throttle setting and flight condition must be known,
- and the drag calculated by taking into account the overall aircraft flow field.

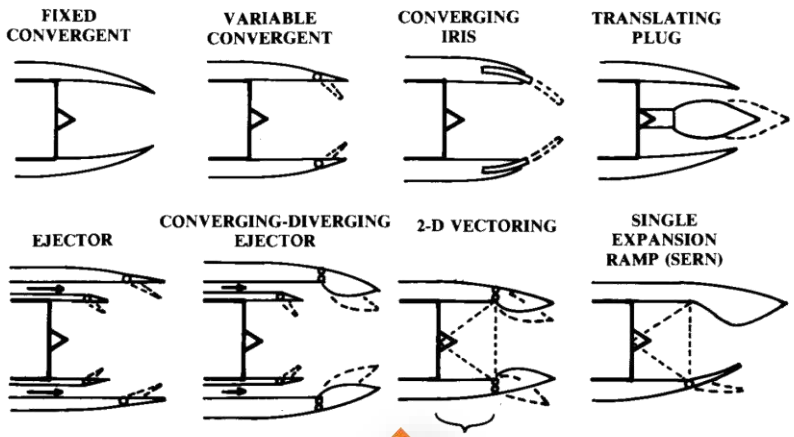
- As an initial approximation, the effect of nozzle positon may be ignored
- and the nozzle drag estimated by the typical subsonic values shown in Table 13.1 for the nozzle types shown in Fig. 10.20.
- For a subsonic, podded nacelle, the nozzle drag is ignored.

Propulsion

Table 13.1 Nozzle incremental drag[10]

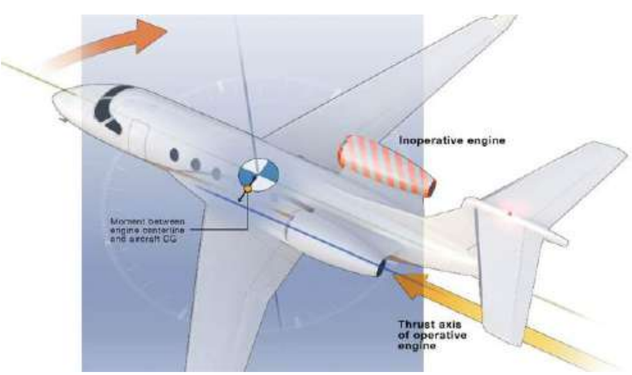
Nozzle type	Subsonic $\Delta C_{D_{Fuselage}}$ *
Convergent	.036–.042
Convergent iris	.001–.020
Ejector	.025–.035
Variable ejector	.010–.020
Translating plug	.015–.020
2-D nozzle	.005–.015

*Referenced to fuselage maximum cross-section area.



Propulsion

- The remaining propulsion system drag is the variation of trim drag with throttle setting.
- If the engine thrust axis is not through the center of gravity, any thrust change will cause a pitching moment.
- The trim force required to counter this moment is charged to the propulsion in most thrust-drag book-keeping systems.
- For initial analysis this may be ignored unless the thrust line is substantially above or below the aircraft centerline .



Trim drag is the component of aerodynamic **drag** on an aircraft created by the flight control surfaces, mainly elevators and trimable horizontal stabilizers, when they are used to offset changes in pitching moment and center of gravity during flight.

Propulsion

Piston-engine Performance

- For design purposes the most important thing to know about the piston engine is that the horsepower produced is directly proportional to the mass flow of the air into the intake manifold.

- In fact, horsepower is approximately 620 times the air mass flow (Lb/s).

- Mass flow into the engine is affected by the outside air density (altitude, temperature, and humidity) and intake manifold pressure.

- Equation (13.9) accounts for the air-density effect upon horsepower,

$$\mathrm{bhp} = \mathrm{bhp}_{\mathrm{SL}} \left(\frac{\rho}{\rho_0} - \frac{1 - \rho/\rho_0}{7.55} \right) \tag{13.9}$$

Propulsion

- The intake manifold is usually at atmospheric pressure.
- A forward-facing air-intake scoop can provide some small increase in manifold pressure at higher speeds.
- Large increases in manifold pressure require mechanical pumping via a "supercharger" or "turbosupercharger.«
- The supercharger is a centrifugal air compressor mechanically driven by a shaft from the engine.
- The amount of air compression available is proportional to engine RPM.
- The turbosupercharger, or "turbocharger," is driven by a turbine placed in the exhaust pipe.

Propulsion

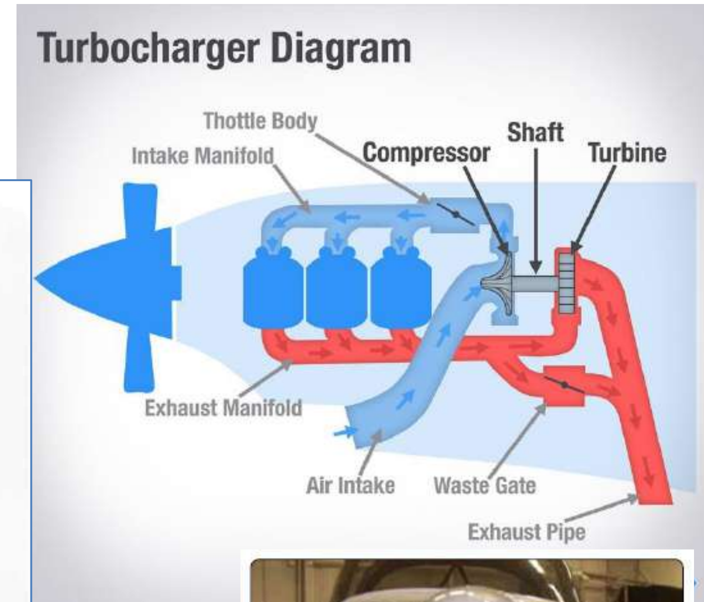


Figure 3-1. *Inlet scoop in engine cowling.*

Propulsion

- Supercharging or turbocharging is usually used to maintain sea level pressure in the intake manifold as the aircraft climbs.
- Typically the sea-level pressure can be maintained up to an altitude of about 15,000-20,000 ft.
- Above this altitude the manifold pressure, and hence the horsepower, drops.
- Figure 13.7 shows typical engine performance for nonsupercharged, supercharged, and turbocharged engines.

Propulsion

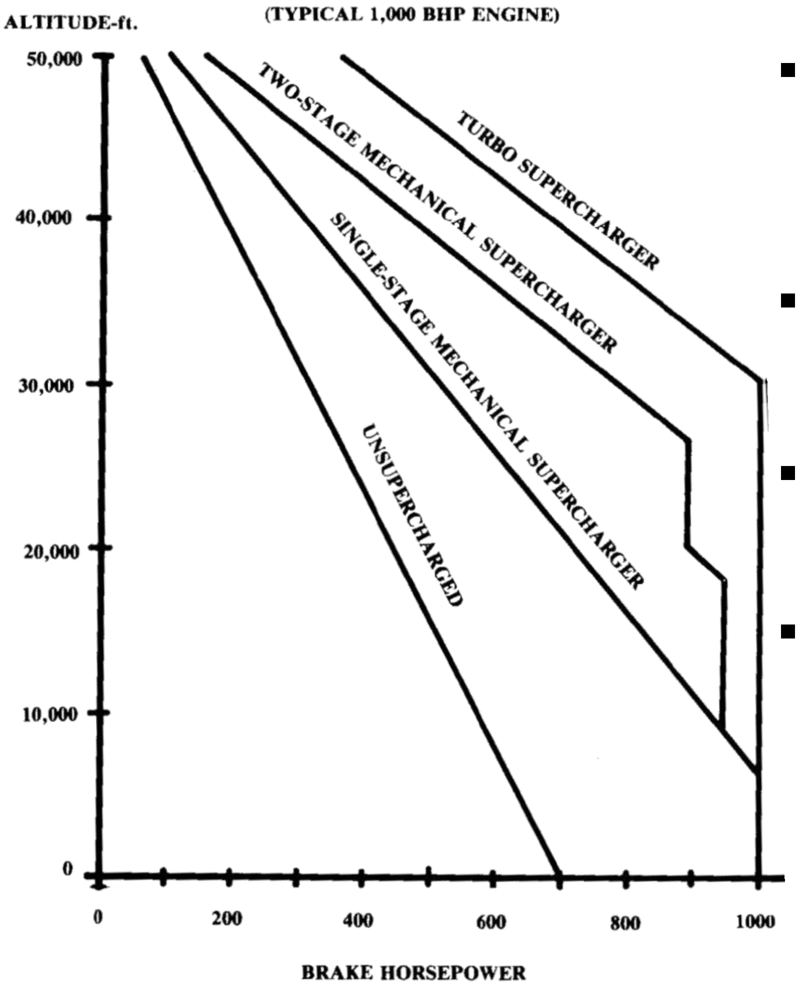


Fig. 13.7 Effects of supercharging.

- Supercharging or turbocharging may also be used to raise the intake manifold pressure above the sea-level value
- to provide additional horsepower from a given engine.
- However, the increased internal pressures require a heavier engine for structural reasons .
- Piston engine performance charts are provided by the manufacturer as a function of manifold pressure, altitude, and RPM .

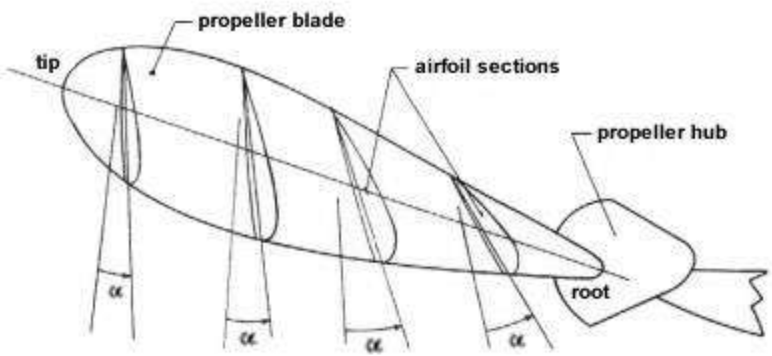
Propulsion

Propeller Performance

- Like a wing, the propeller is designed to a particular flight condition.

- Since the tangential velocities of the propeller airfoil sections increase with distance from the hub the airfoils must be set at progressively reduced pitch angles going from root to tip.

- The overall pitch of a propeller refers to the blade angle at 70-75% of the radius.



Propulsion

- The advance ratio (equivalent to the wing angle of attack) is related to the distance the aircraft moves with one turn of the propeller.

- Advance ratio is sometimes called the "slip function" or "progression factor."

$$\text{Advance Ratio:} \quad J = V/nD$$

- The power and thrust coefficients are nondimensional measures of those quantities, much like the wing lift-coefficient.

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

$$\text{Thrust Coefficient:} \quad c_T = T/\rho n^2 D^4$$

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

Propulsion

- The speed-power coefficient is defined as the advance ratio raised to the fifth power divided by the power coefficient.

- The speed-power coefficient is nondimensional and does not involve the propeller diameter, which is useful for comparison between propellers of different size.

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

- The activity factor is a measure of the amount of power being absorbed by the propeller.

- Activity factors range from about 90-200, with a typical light-aircraft activity factor being 100 and a typical large turboprop having an activity factor of 140.

- The final expression in Eq. (13.14) is the activity factor for a straight-tapered propeller blade of taper ratio λ.

Propulsion

$$\text{AF}_{\text{per blade}} = \frac{10^5}{D^5} \int_{0.15R}^{R} cr^3 \, \mathrm{d}r = \frac{10^5 c_{\text{root}}}{16D} [0.25 - (1 - \lambda)0.2] \tag{13.14}$$

- Equation (13.15) relates the propeller efficiency, to the advance ratio and the ratio of the thrust coefficient to the power coefficient.

- This ratio is used in Eq. (13.16) to determine the thrust at static conditions when the velocity is zero.

$$\text{Propeller Efficiency:} \quad \eta_p = \frac{TV}{P} = \frac{TV}{550 \text{ bhp}} = J\frac{c_T}{c_P} \tag{13.15}$$

$$\text{Thrust:} \quad T = \frac{550 \text{ bhp } \eta_p}{V} = \frac{c_T}{c_P} \frac{550 \text{ bhp}}{nD} \tag{13.16}$$

Propulsion

- Figures 13.8 and 13.9 have been chosen as typical of propellers used for modern light and business aircraft.

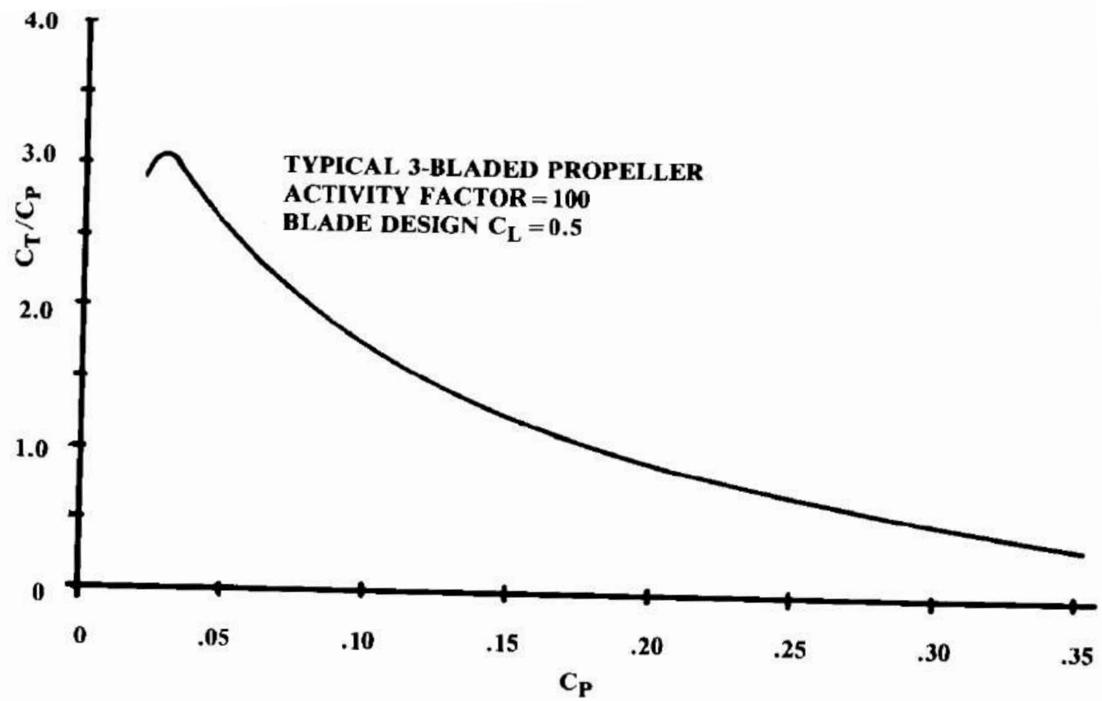


Fig. 13.8 Static propeller thrust. (after Ref. 50)

Propulsion

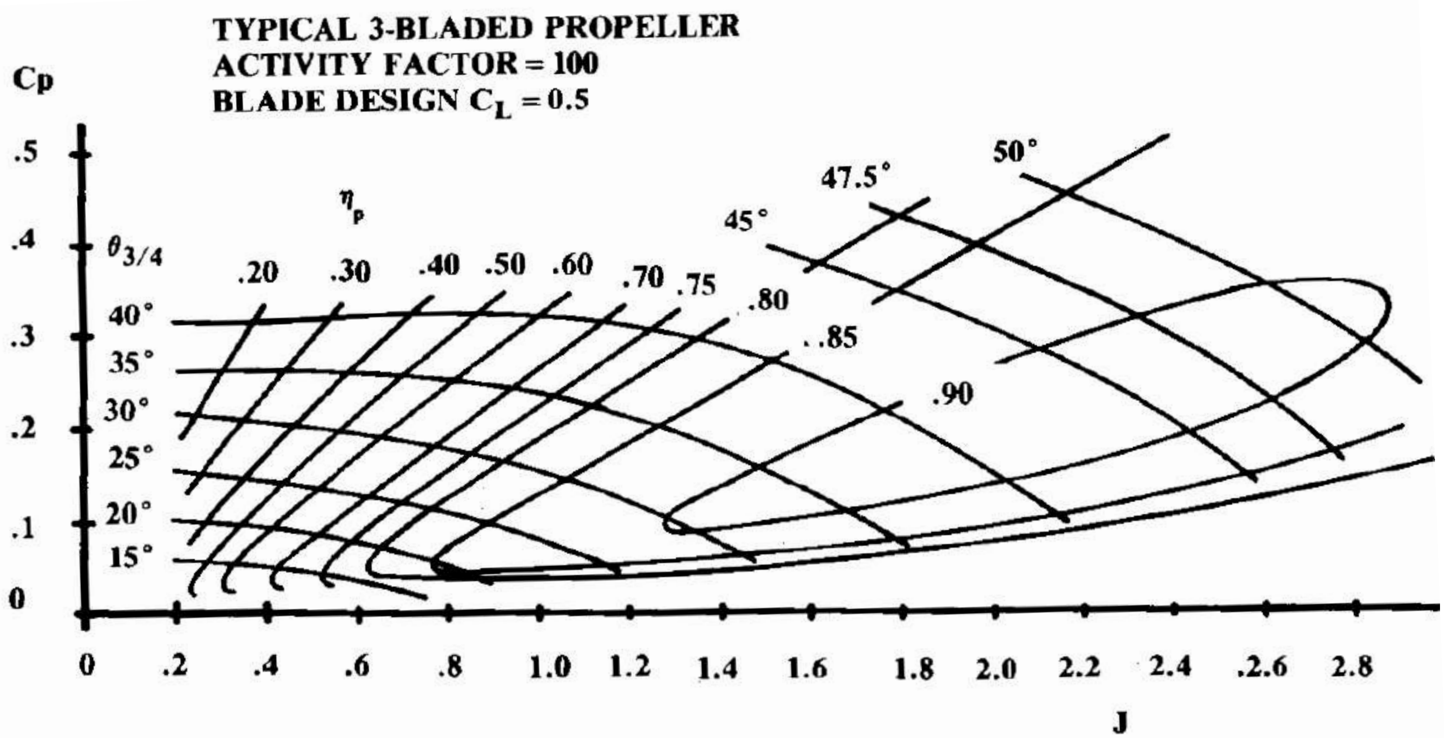


Fig. 13.9 Forward flight thrust and efficiency. (after Ref. 50)

Propulsion

- Figure 13.9 could be used to determine the thrust from a fixed-pitch propeller by following the appropriate line for the selected blade angle.

- However, it is simpler to use the approximate method of Fig. 13.10 unless actual propeller data is available.

- Figure 13.10 relates the fixed-pitch propeller efficiency at an off-design velocity and RPM to the on-design efficiency, which is attained by the propeller at some selected flight condition.

Propulsion

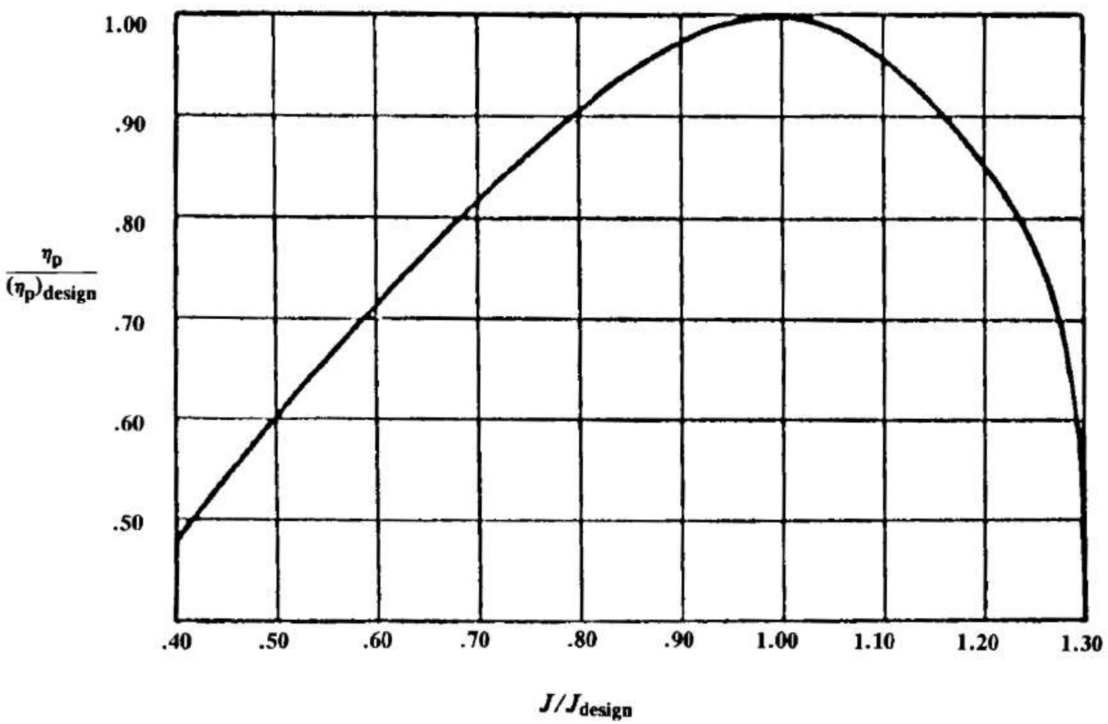


Fig. 13.10 Fixed-pitch propeller adjustment.

Propulsion

- For a two-bladed propeller, the forward-flight efficiencies are about 3% better than shown in Fig. 13.9,

- but the static thrust is about *5%* less than shown in Fig. 13.8.

- The reverse trends are true for a four-bladed propeller.

- Also, a wooden propeller has an efficiency about 10% lower due to its greater thickness.

- These charts provide useful rough estimations of propeller performance, but actual charts for the selected propeller should be obtained from the manufacturer for any serious design effort.

Propulsion

- As with jet engines, there are several engine-related drag items that must be considered, namely, scrubbing drag, cooling drag, and engine miscellaneous drag.

- Scrubbing drag is the increase in aircraft drag due to the higher velocity and turbulence experienced by the parts of the aircraft within the propwash.

- If the parasite-drag coefficient for the propwashed parts of the
- aircraft cannot be determined, 0.004 is a reasonable estimate.

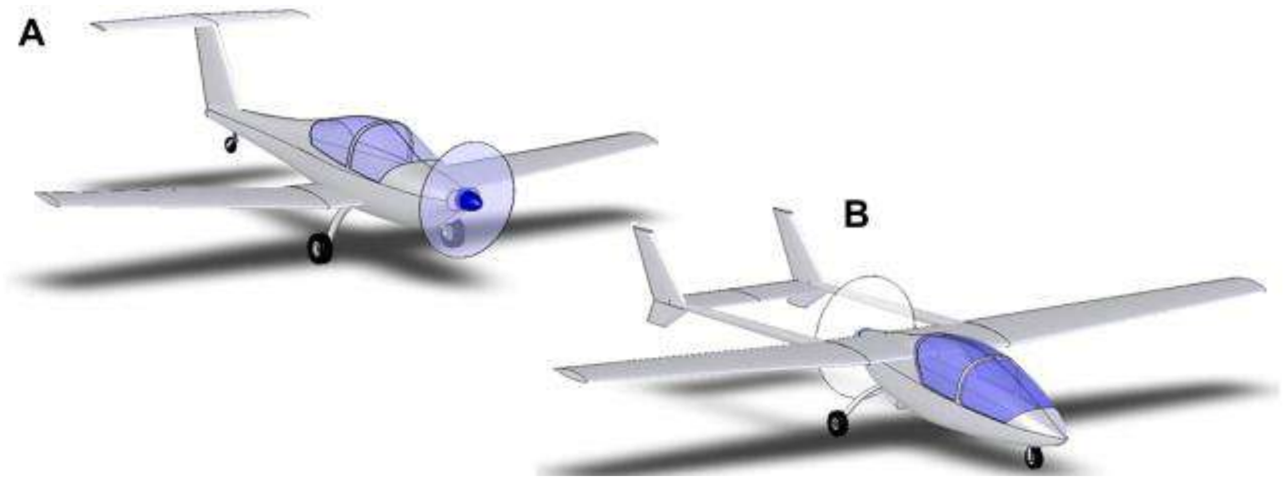
- A simpler approach adjusts the propeller efficiency as in Eq. (13.17).

- The subscript "washed" refers to the parts of the aircraft which lie within the propwash.

$$\eta_{p_{\text{effective}}} = \eta_p \left[1 - \frac{1.558}{D^2} \frac{\rho}{\rho_0} \Sigma (C_{f_e} S_{\text{wet}})_{\text{washed}} \right] \qquad (13.17)$$

Propulsion

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



Propulsion

- Cooling drag represents the momentum loss of the air passed over the engine for cooling.

- This is highly dependent upon the detail design of the intake, baffles, and exit.

- Miscellaneous engine drag includes the drag of the oil cooler, air intake, exhaust pipes, and other parts.

- Cooling and miscellaneous drags for a well designed engine installation can be estimated by Eqs. (13.18) and (13.l9).

- However, a typical light aircraft engine installation may experience cooling and miscellaneous drag levels 2-3 times the values estimated by these equations.

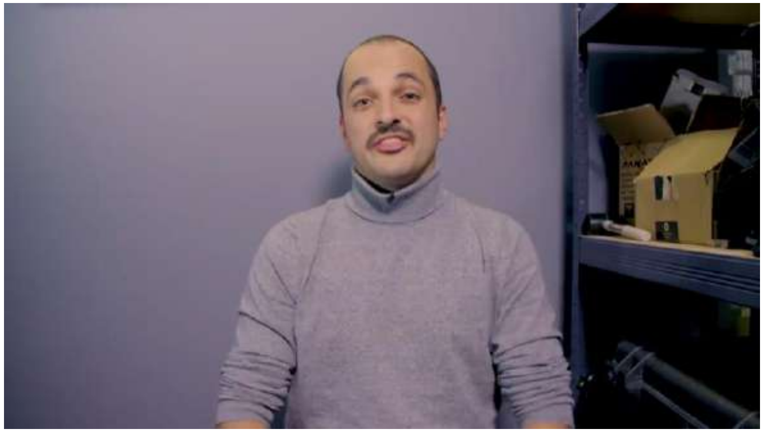
Propulsion

$$(D/q)_{\text{cooling}} = (4.9 \times 10^{-7}) \frac{\text{bhp} \cdot T^2}{\sigma V}, \text{ ft}^2 \tag{13.18}$$

$$(D/q)_{\text{misc}} = (2 \times 10^{-4}) \text{ bhp, ft}^2 \tag{13.19}$$

where

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



Propulsion

Turboprop Performance

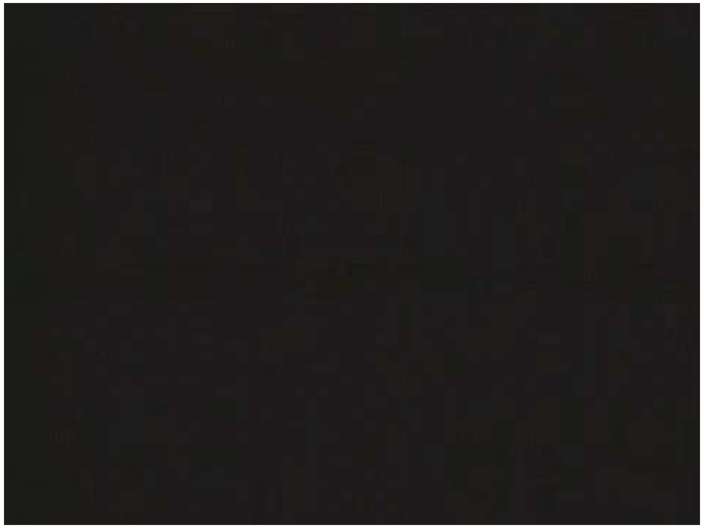
- A turboprop is a jet engine that drives a propeller using a turbine in the exhaust.
- The jet exhaust retains some thrust capability, and can contribute as much as 20% of the total thrust.
- Analysis of the turboprop is a hybrid between the jet and the piston-prop analysis.
- The engine is analyzed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

Propulsion

- The conventional turboprop, like the piston-prop, is limited by tip Mach number to about Mach 0.7.
- The turboprop has higher efficiency than the piston-prop at Mach numbers greater than about 0.5 due to the residual jet thrust,
- but the conventional turboprop is no match for a turbofan engine at the higher subsonic speeds.
- Recently, a new type of advanced propeller has been developed that offers good efficiencies up to about Mach 0.85.

Propulsion

- These are known as propfans" or "unducted fans (UDF)."
- They are smaller in diameter than the regular propellers and feature numerous wide, thin, and swept blades.
- Test programs to date indicate that a well-designed propfan can retain propeller efficiencies of over 0.8 at speeds on the order of Mach 0.85.



Aircraft Design
AE 405

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Contents

Chapter 14 - 15

- Structures and Loads
- Weights

Structures and Loads

- In a large aircraft company, the conceptual designer may never do any structural analysis.

- The conceptual designer relies upon an experienced eye to insure that sufficient space is provided for the required structural members.

- The only direct impact of structures during the initial stages of conceptual design is in the weights estimation.

- Before the actual structural members can be sized and analyzed, the loads they will sustain must be determined.

- Aircraft loads estimation, a separate discipline of aerospace engineering, combines aerodynamics, structures, and weights.

Structures and Loads

- Today's computer programs have mechanized much of the time-consuming work in loads estimation.

- Modern aerodynamic panel programs determine the air loads as an intermediate step toward determining aerodynamic coefficients.

- Modern finite element methods are used to analysis aircraft structures.

- This chapter introduces the concepts of loads estimation and summarizes the subjects of aircraft materials and structural analysis.

Structures and Loads

Loads Categories

- Table 14.1 lists the major load categories experienced by aircraft.

- Civil and military specifications (FAR 23 & 25, MIL-A-8860/8870) define specific loading conditions for these categories.

- For each structural member of the aircraft, one of the loads listed in Table 14.1 will dominate.

- Fig. 14.1 and 14.2 show typical critical loads for a fighter and a transport.

- Note that the lifting surfaces are almost always critical under the high-g maneuver conditions.

Structures and Loads

Table 14.1 Aircraft loads

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		

Structures and Loads

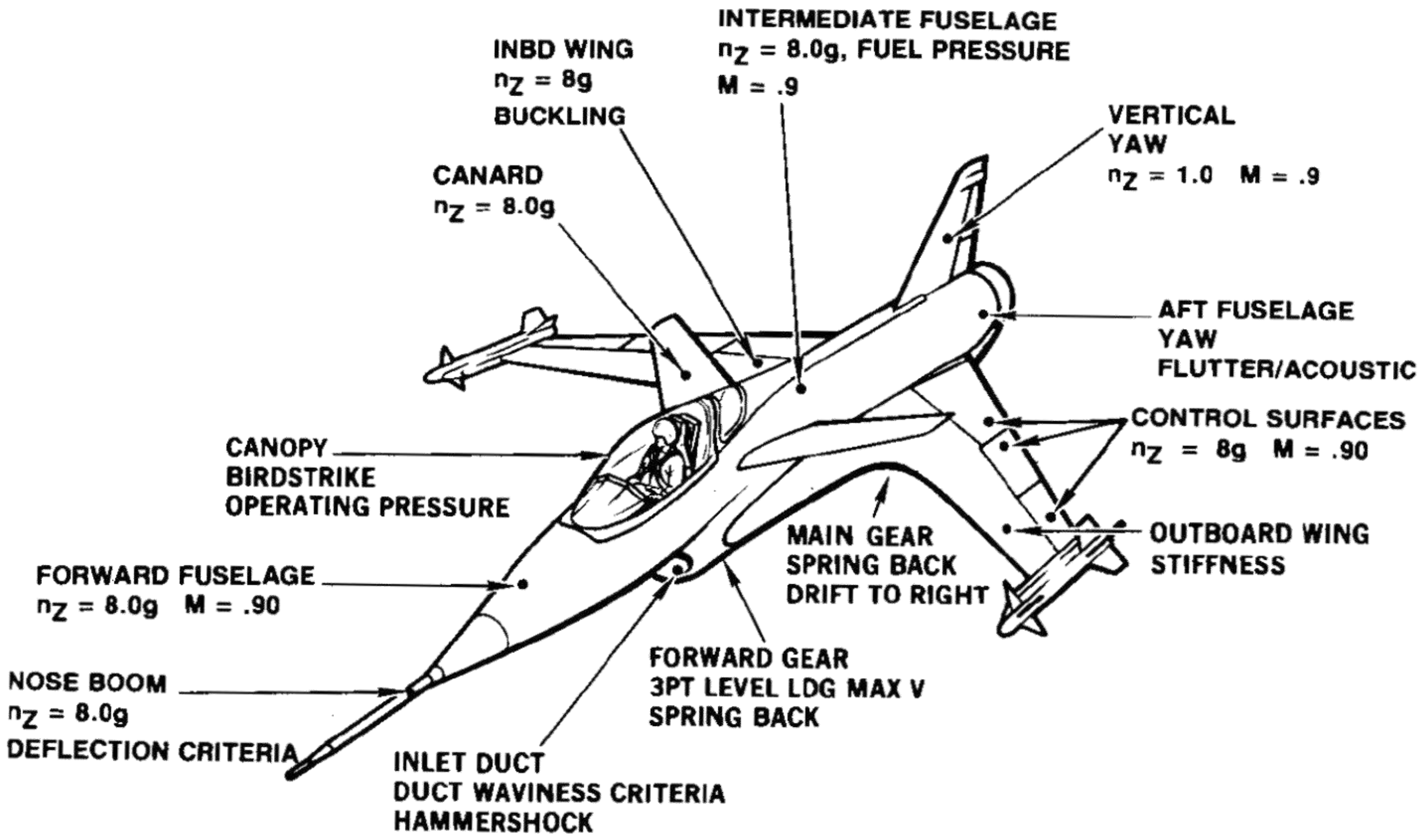


Fig. 14.1 Typical fighter limit loads.

Structures and Loads

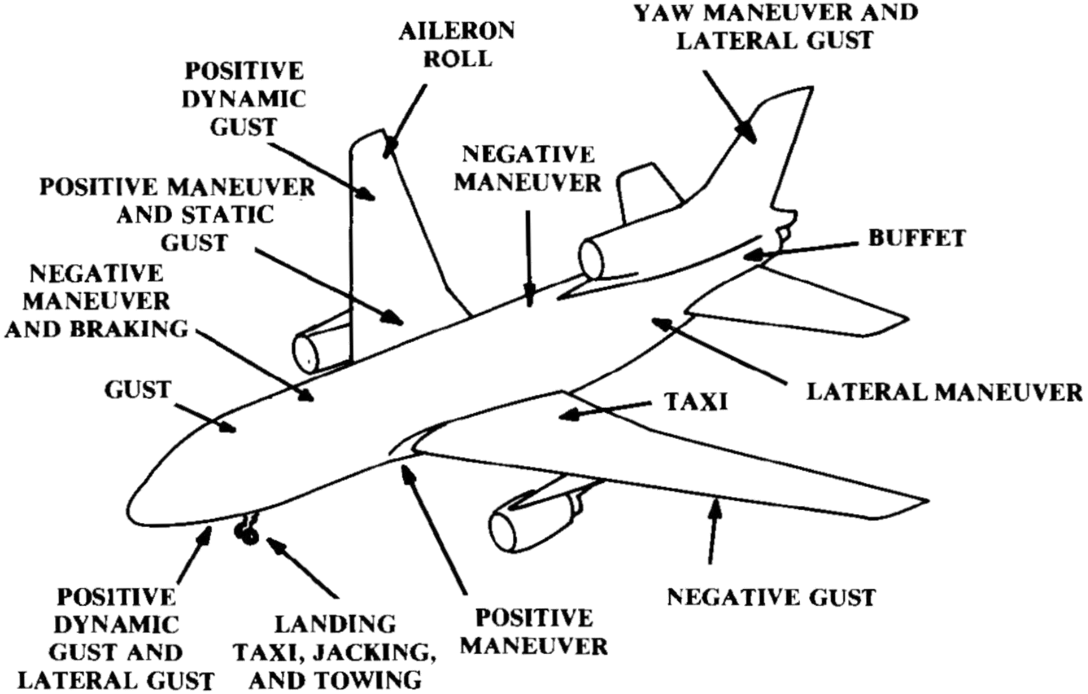


Fig. 14.2 L1011 critical loads.

Structures and Loads

- The largest load the aircraft is actually expected to encounter is called the "limit," or "applied," load.
- For the fighter of Fig. 14.1, 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.

Structures and Loads

- Since the 1930's the factor of safety has usually been 1.5.

- For the fighter in Fig. 14.1, the design load for the wing structure would then be based upon a 12-g maneuver, above which the wing would break.

Air Loads - 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).
- Table 14.2 lists typical limit load factors.
- Note that the required negative load factors are usually much less in magnitude than the positive values.

Structures and Loads

Table 14.2 Typical limit load factors

	$n_{positive}$	$n_{negative}$
General aviation—normal	2.5 to 3.8	−1 to −1.5
General aviation—utility	4.4	−1.8
General aviation—aerobatic	6	−3
Homebuilt	5	−2
Transport	3 to 4	−1 to −2
Strategic bomber	3	−1
Tactical bomber	4	−2
Fighter	6.5 to 9	−3 to −6

- 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.

Structures and Loads

- 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.

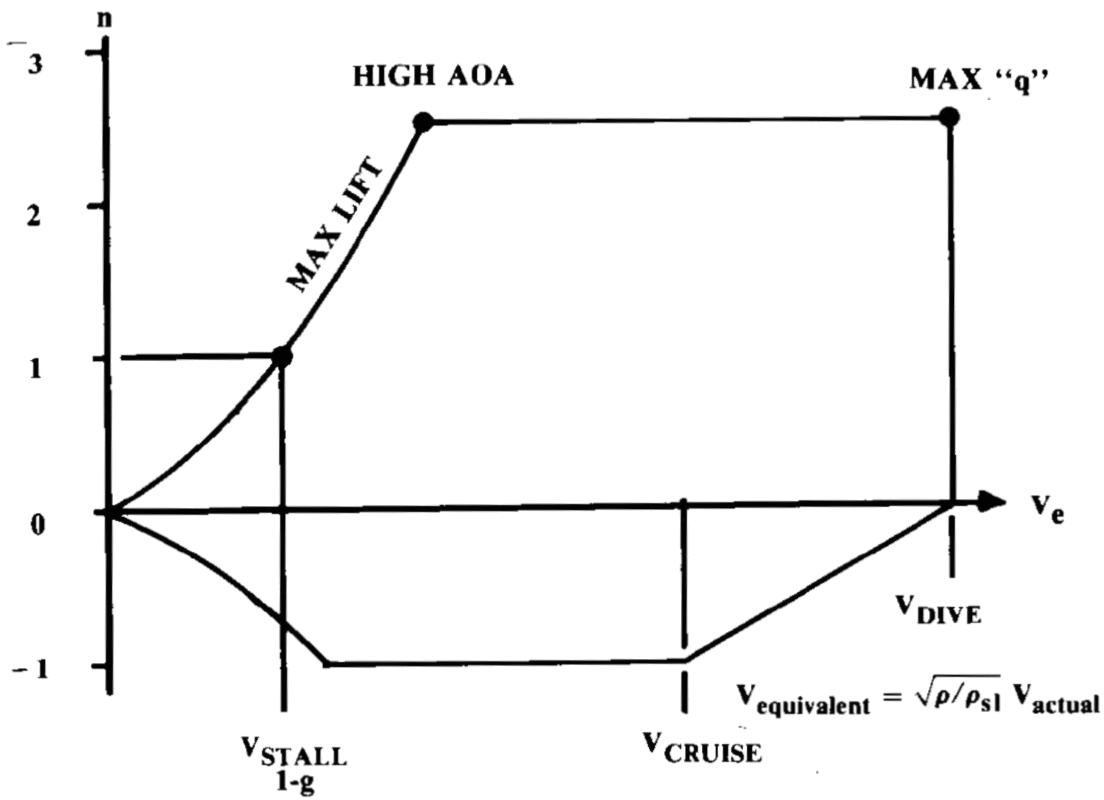


Fig. 14.3 V-n diagram (manuever).

- 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.

Structures and Loads

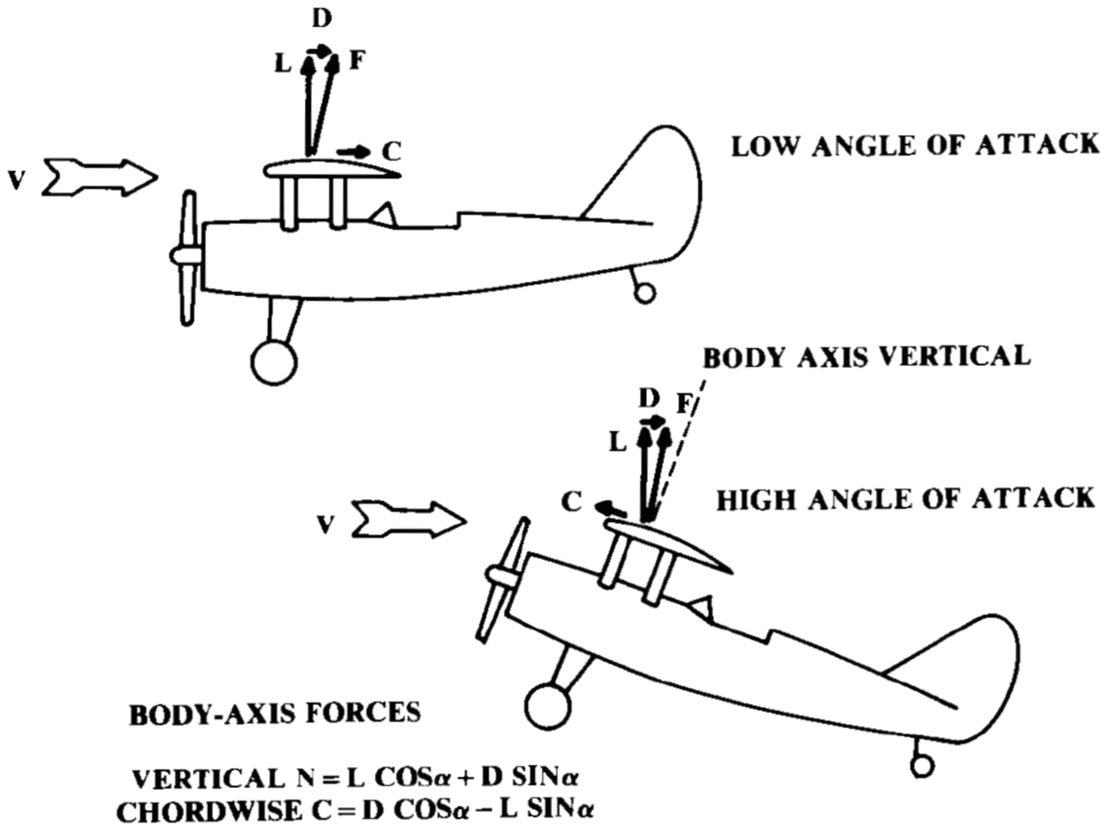


Fig. 14.4 Wing load direction at angle of attack.

Structures and Loads

- The point representing maximum q and maximum load factor is clearly important for structural sizing.
- At this condition the aircraft is at a fairly low angle of attack because of the high dynamic pressure, so the load is approximately vertical in the body axis.
- For a subsonic aircraft, maximum or dive speed is typically 50% higher than the level-flight cruise speed.
- For a supersonic aircraft the maximum speed is typically about Mach 0.2 faster than maximum level-flight speed,
- although many fighters have enough thrust to accelerate past their maximum structural speed.

Structures and Loads

- For loads estimation, V_e is a convenient measure of velocity because it is constant with respect to dynamic pressure regardless of altitude.

$$V_e = \sqrt{\rho/\rho_{SL}}\,(V_{actual}) = \sqrt{\sigma}\,(V_{actual}) \qquad (14.1)$$

Gust Loads

a sudden strong rush of wind

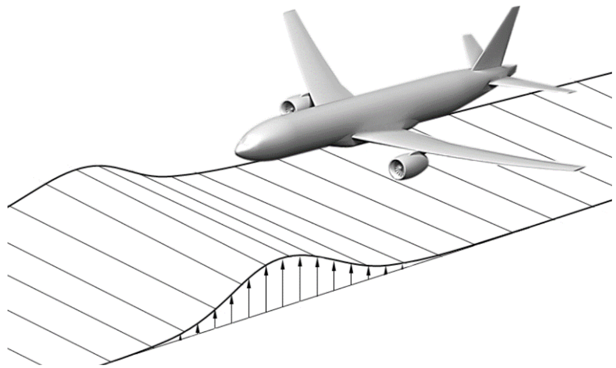
- The loads experienced when the aircraft encounters a strong gust can exceed the maneuver loads in some cases.

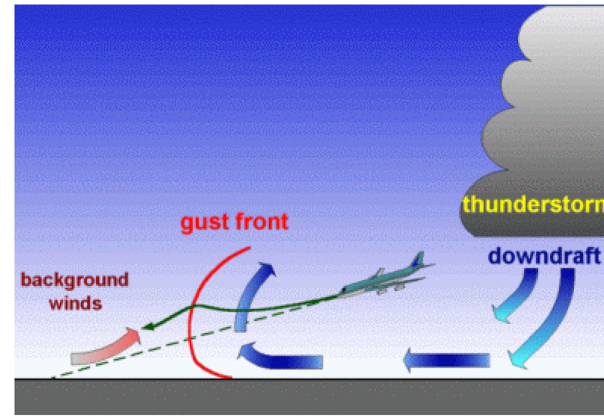
- For a transport aircraft flying near thunderstorms or encountering high-altitude "clear air turbulence,"

- 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.

Structures and Loads





Structures and Loads

- Figure 14.5 illustrates the geometry for an upward gust of velocity U.

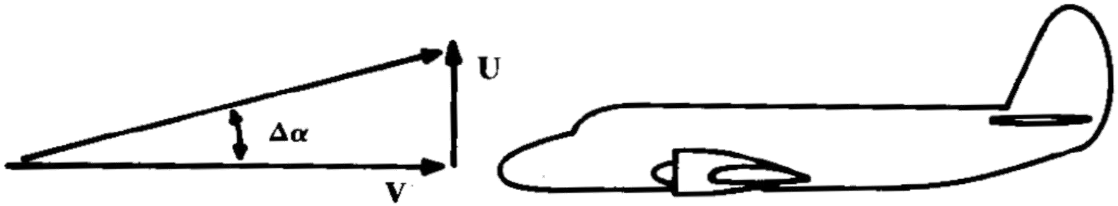


Fig. 14.5 Gust encounter.

- The change in angle of attack, as shown in Eq. (14.2), is approximately U divided by V, the aircraft velocity.

$$\Delta\alpha = \tan^{-1}\frac{U}{V} \cong \frac{U}{V} \tag{14.2}$$

- The change in aircraft lift in Eq. (14.3) to be proportional to the gust velocity.

- The resulting change in load factor is derived in Eq. (14.4).

Structures and Loads

$$\Delta L = \tfrac{1}{2}\rho V^2 S (C_{L_\alpha} \Delta\alpha) = \tfrac{1}{2}\rho V S C_{L_\alpha} U \tag{14.3}$$

$$\Delta n = \frac{\Delta L}{W} = \frac{\rho U V C_{L_\alpha}}{2W/S} \tag{14.4}$$

- Fig. 14.5 and Eq. (14.4) assume that the aircraft instantly encounters the gust and it instantly affects the entire aircraft.
- These assumptions are not realistic.
- Gusts tend to follow a cosine-like intensity increase as the aircraft flies through, allowing it more time to react to the gust.
- To account for this effect a statistical guest alleviation factor (K) has been devised and applied to measured gust data (U_{de}).
- The gust velocity in Eq. (14.4) can be defined in the following terms

Structures and Loads

$$U = KU_{de} \tag{14.5}$$

Subsonic: $$K = \frac{0.88\mu}{5.3 + \mu} \tag{14.6}$$

Supersonic: $$K = \frac{\mu^{1.03}}{6.95 + \mu^{1.03}} \tag{14.7}$$

where

Mass Ratio: $$\mu = \frac{2(W/S)}{\rho g \bar{c} C_{L_\alpha}} \tag{14.8}$$

- For many years the standard vertical gust U_{de} has been 30 ft/s (positive or negative).
- For higher speeds iy may be assumed that U_{de} drops linearly to 15 ft/s.
- A more detailed requirement for U_{de} is shown in Fig. 14.6.

Structures and Loads

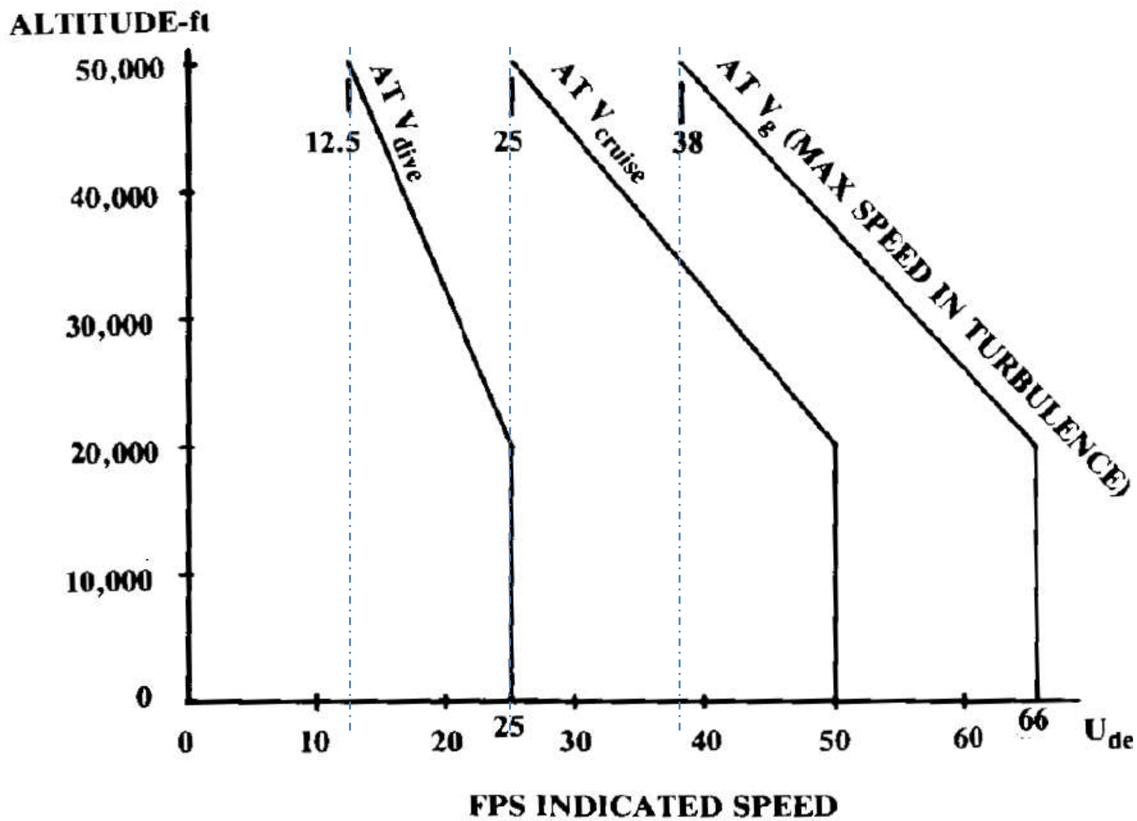


Fig. 14.6 Derived equivalent gust velocities (transport).

Structures and Loads

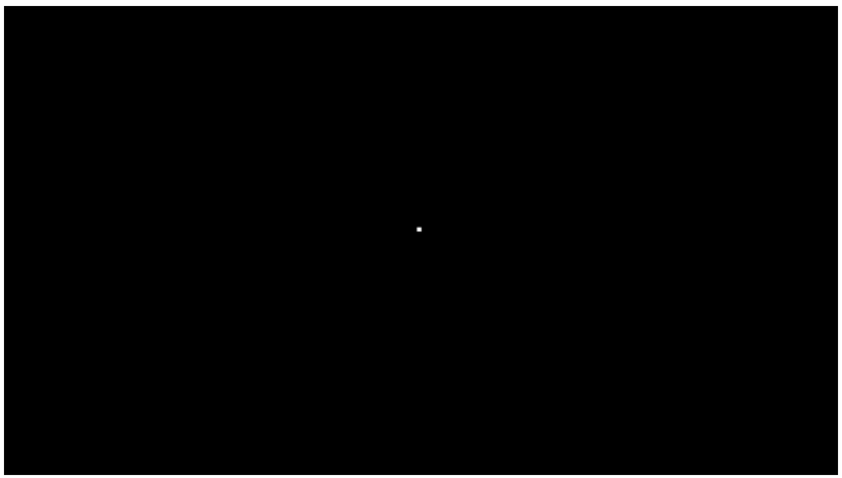
- One interesting point concerning gusts is that, as shown in Eq. (14.4), the load factor due to a gust increases if the aircraft is lighter.

- This is counter to the natural assumption that an aircraft is more likely to have a structural failure if it is heavily loaded.

- The gust load factors as calculated with Eqs. (14.4-14.8) and using the appropriate U_{de} (positive and negative) can then be plotted on a V-n diagram as shown in Fig. 14. 7.

- It is assumed that the aircraft is in 1-g level flight when the gust is experienced.

Structures and Loads



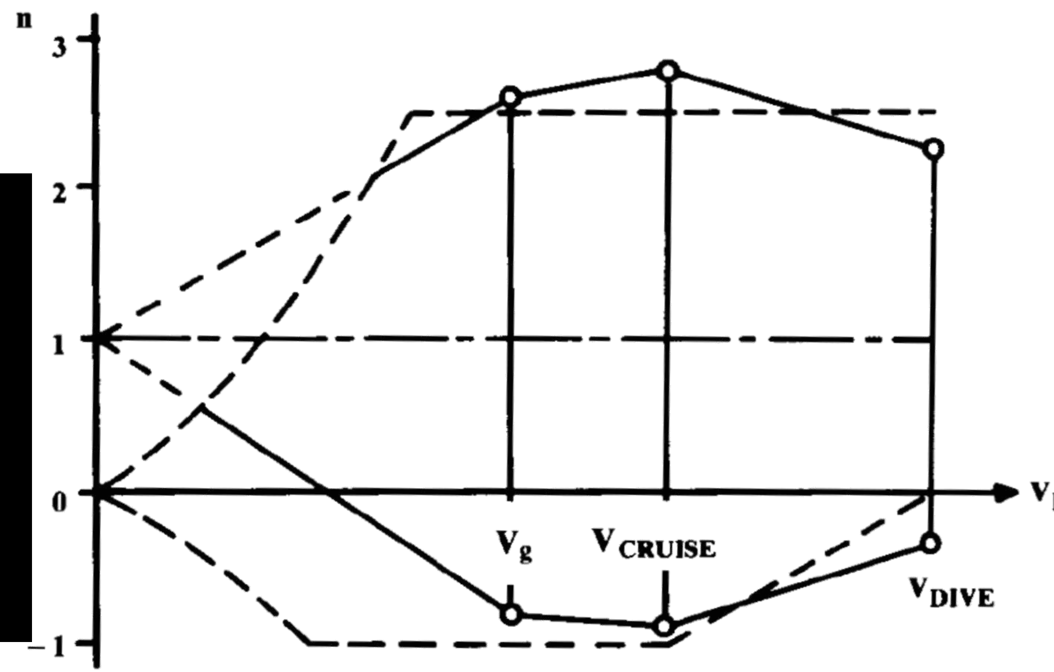


Fig. 14.7 V-n diagram (gust).

Structures and Loads

Air Loads on Lifting Surfaces

- Now that the V-n diagram is complete, the actual loads and load distributions on the lifting surfaces can be determined.
- In most cases this needs to be done only at the "high A.0.A." and "max q" velocities (see Fig. 14.3)
- and any velocities where the gust load factor exceeds the assumed limit load factor.
- 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.

Structures and Loads

- 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 spanwise and chordwise load distributions can be determined (Fig. 14.9).

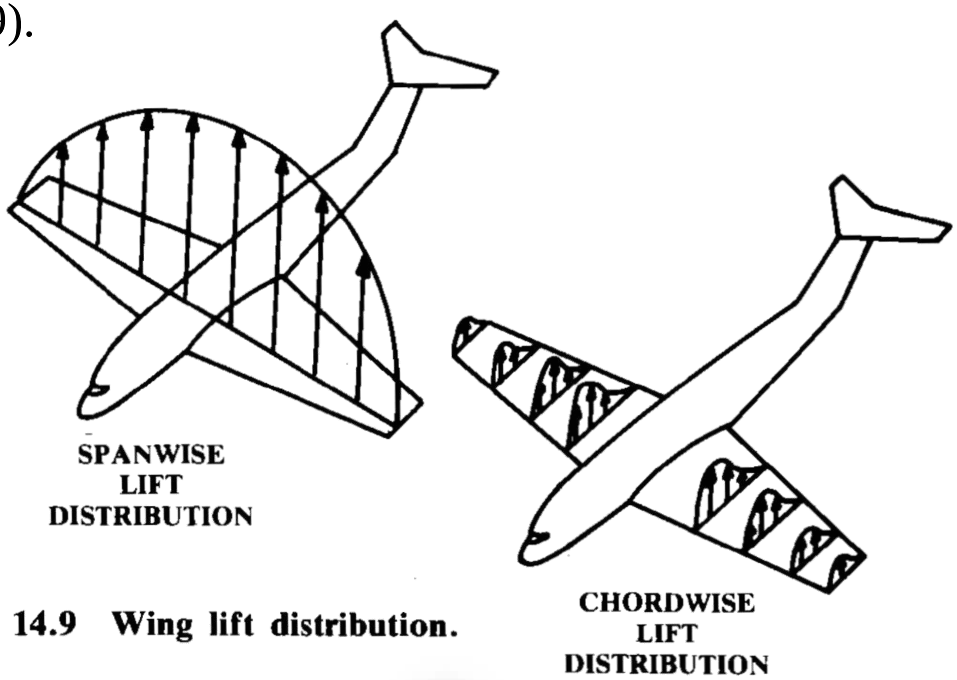


Fig. 14.9 Wing lift distribution.

Structures and Loads

- According to classical wing theory, the spanwise lift (or load) distribution is proportional to the circulation at each span station.
- A vortex lifting-line calculation will yield the spanwise lift distribution.
- For an elliptical planform wing, the lift and load distribution is of elliptical shape.
- For a nonelliptical wing, a good semi-empirical method for spanwise load estimation is known as Schrenk's Approximation.
- This method assumes that the load distribution on an untwisted wing or tail has a shape that is the average of the actual planform shape and an elliptic shape of the same span and area (Fig. 14.10).

Structures and Loads

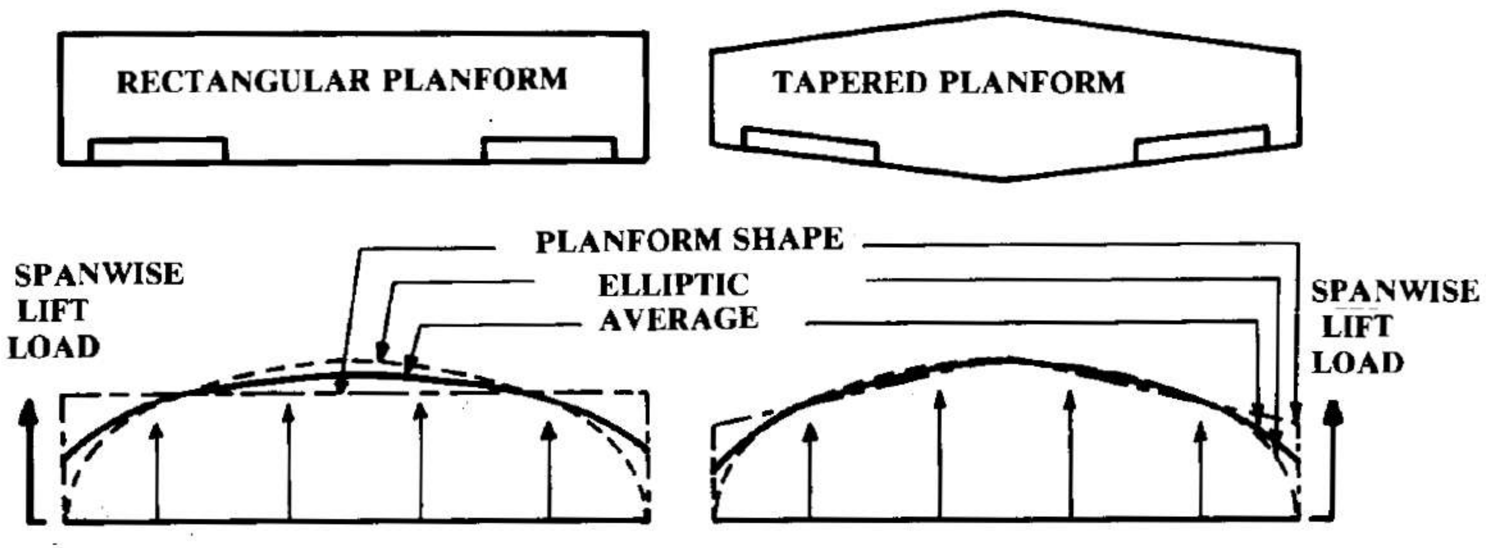


Fig. 14.10 Schrenk's approximation.

- The total area under the lift load curve must sum to the required total lift.
- Equations (14.9) and (14.11) describe the chord distributions of a trapezoidal and elliptical wing.

Structures and Loads

Trapezoidal Chord: $$C(y) = C_r\left[1 - \frac{2y}{b}(1-\lambda)\right] \qquad (14.9)$$

Elliptical Chord: $$C(y) = \frac{4S}{\pi b}\sqrt{1-\left(\frac{2y}{b}\right)^2} \qquad (14.11)$$

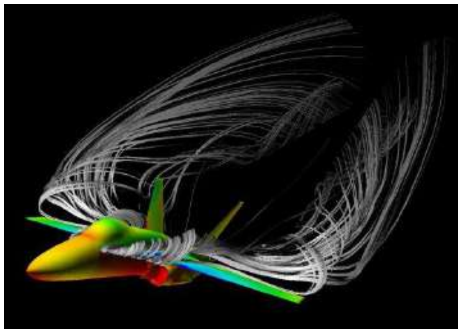
$$S = \frac{b}{2}C_r(1+\lambda)$$

- If substantial dihedral is used, the perpendicular load on the wing is grater than the lift.
- Divide the lift by the cosine of the dihedral angle to get the perpendicular load.
- Schrenk's Approximation does not apply to highly swept planforms experiencing vortex flow.
- Loads for such a planform must be estimated using computers
- and wind tunnels.

Structures and Loads

- The spanwise distribution of drag loads must also be considered, especially for fabric-covered aircraft in which drag loads are carried by internal "drag wires."
- Drag loads tend to be greatest near the wingtips, and should be determined from wind-tunnel or aerodynamic panel program data.
- As a first approximation the spanwise distribution of drag loads can be roughly approximated
  - as a constant 95% of the average drag loading from the root to 80% of the span,
  - and 120% of the average loading from 80% of span to the wingtip.

Structures and Loads



- A vortex from a leading-edge strake can cause vibrational stresses on any component of the aircraft it touches.
- The F-18 had a problem with vertical tail fatigue for this reason.
- A similar problem can occur due to propeller propwash.
- These effects are difficult to predict, but must be considered during conceptual design.

Structures and Loads

- Once the spanwise load distribution is known, the wing or tail bending stress can be determined.

- To determine torsional stresses, the airfoil moment coefficient is applied to spanwise strips and the total torsional moment is summed from tip to root.

- Actual chordwise pressure distributions for a NACA 4412 airfoil at various angles of attack are shown in Fig. 14.11.

- Figure 14.12 shows a rough approximation for the lift load distribution on a conventional airfoil if wind-tunnel data is unavailable.

Boeing 787 Wingflex

Structures and Loads

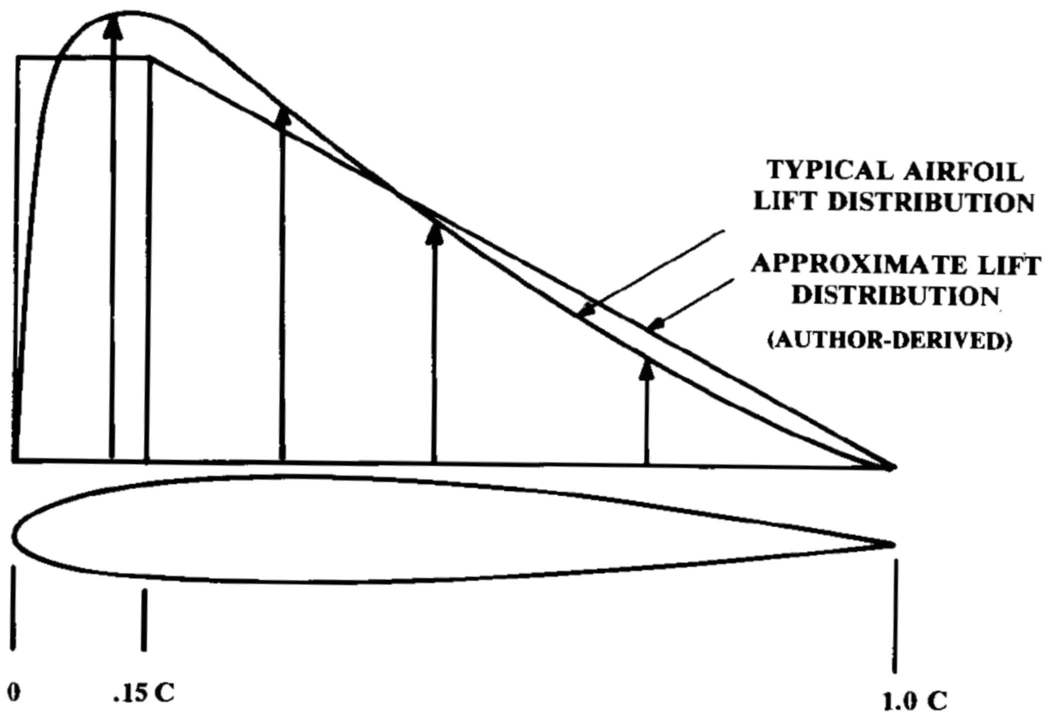


Fig. 14.12 Approximate airfoil lift distribution.

Structures and Loads

Airloads due to Control Deflection

- Operation of the control surfaces produces airloads in several ways.
- The greatest impact is in the effect of the elevator on angle of attack and hence load factor.
- The rudder's effect on yaw angle can also impose large loads.
- Deflection of control surfaces produces additional loads directly upon the wing or tail structure.
- "Maneuver speed," or "pull-up speed V_p," is the maximum speed at which the pilot can fully deflect the controls without damaging either the airframe or the controls themselves.
- For most aircraft the maneuver speed is less than the maximum level cruise speed V_L .

Structures and Loads

- Maneuver speed is established in the design requirements or may be selected using an old empirical relationship, Eq. (14.12).

- Stall speed V_s is with high lift devices deployed.

- The factor K_p is estimated in Eq. (14.13),

- But should not be allowed to fall below 0.5 or above 1.0. (For general aviation aircraft, K_p usually doesn't exceed 0.9.)

$$V_p = V_s + K_p(V_L - V_s) \tag{14.12}$$

$$K_p = 0.15 + \frac{5400}{W + 3300} \tag{14.13}$$

- The airloads imposed upon the structure can be determined via maneuver speed.

Structures and Loads

- Fig. 14.13 shows an approximate distribution of the additional airloads imposed directly upon a conventional airfoil by the deflection of a control surface.

- Control deflection will typically provide a change in section lift coefficient of about 0.8-1.1 at 25 deg deflection.

- In the absence of better data, the change in airfoil moment coefficient can be estimated as -0.01 times the control deflection in degrees.

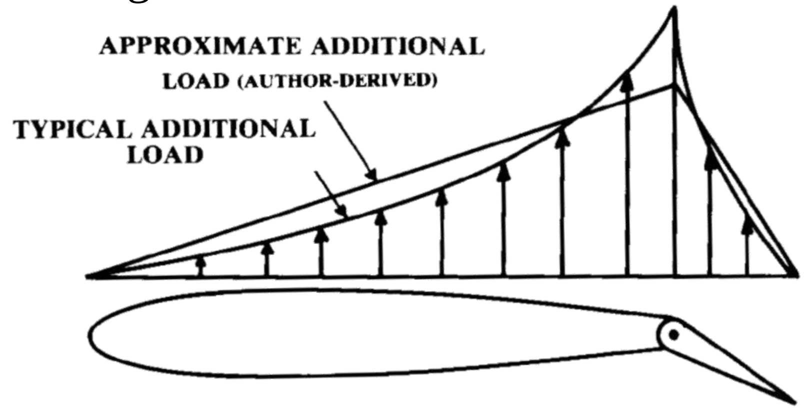


Fig. 14.13 Approximate additional load due to control deflection.

Structures and Loads

- Figure 14.14 shows the loading distribution used for the special case of a horizontal tail consisting of a fixed stabilizer and a moving elevator.
- Under some combinations of angle of attack and elevator position the stabilizer and elevator will actually have loads in opposite directions, as shown.
- For design purposes, the elevator load *P* is assumed to equal 40% of the total required tail load *T*, but in the opposite direction.
- The distributed load shown on the stabilizer must then equal 140% of the tail load.
- The spanwise load distribution is usually assumed to be proportional to chord length.

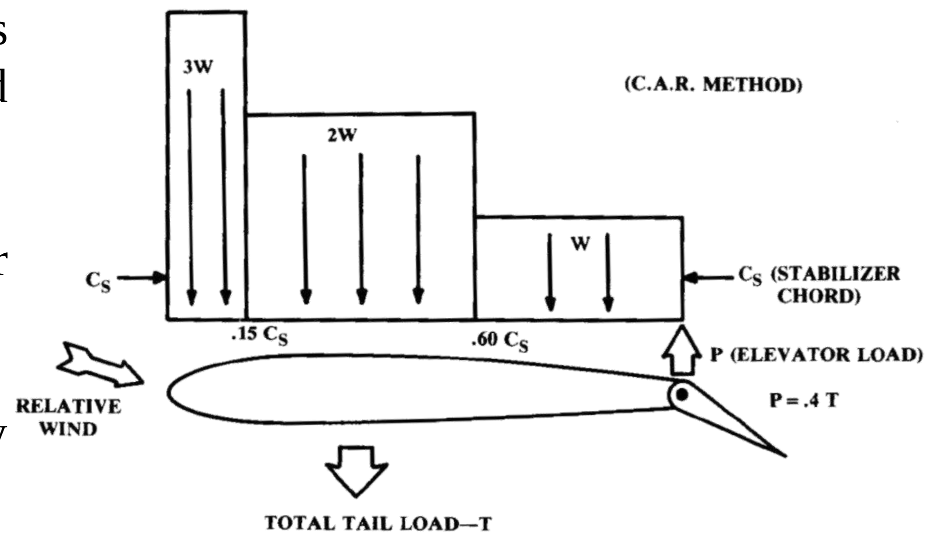


Fig. 14.14 Loads on fixed stabilizer with elevator deflection.

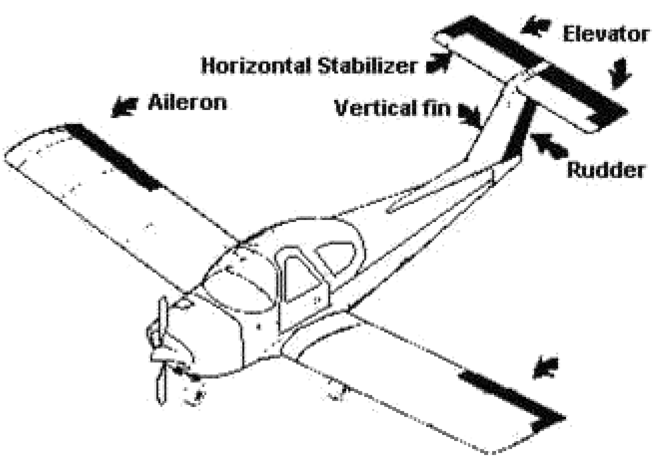
Structures and Loads

- For an aircraft with a manual flight-control system, the control loads may be limited by the strength of the pilot.

- For a stick-controlled aircraft, the pilot strength is limited to 167 lb for the elevator and to 67 lb for the ailerons.

- For a wheel-controlled aircraft, the pilot strength is limited to 200 lb for the elevator and to 53 (times the wheel diameter) in.-lbs for the ailerons.

- The rudder force is limited to 200 lb.



Structures and Loads

Inertial Loads

- Inertial loads reflect the resistance of mass to acceleration.
- The various accelerations due to maneuver and gust, described above, establish the stresses for the aerodynamic surfaces.

- Every object in the aircraft experiences a force equal to the object's weight times the aircraft load factor.
- This creates additional stresses throughout the aircraft, which must be determined.

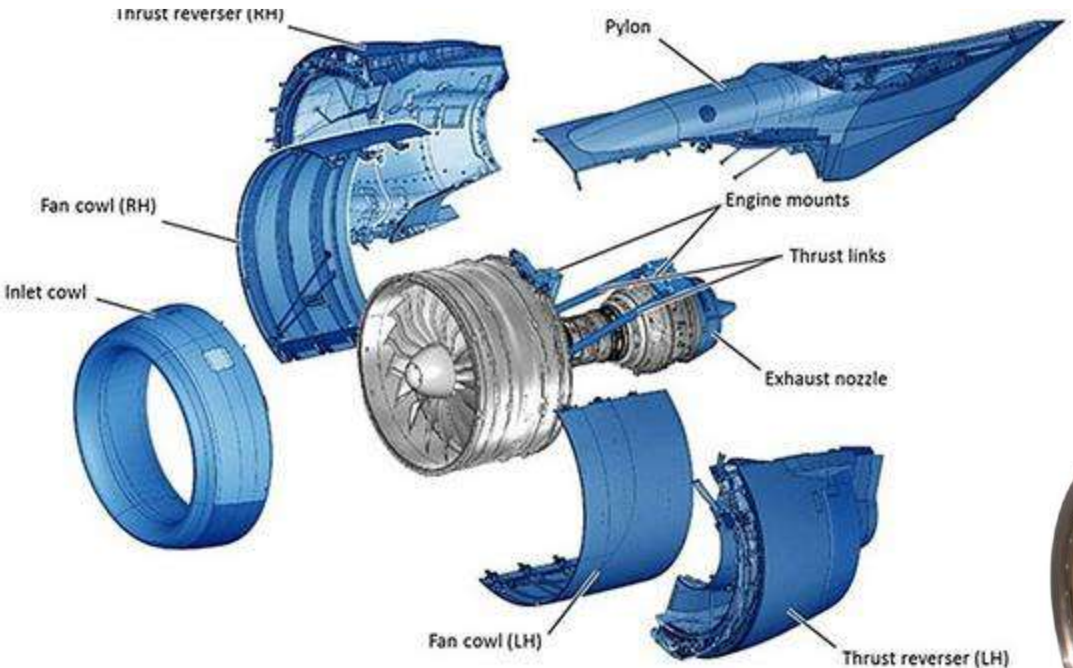
- Inertial loads due to rotation must also be considered. For example, the tip tanks of a fighter rolling at a high rate will experience an outward centrifugal force.
- Calculation of these loads goes beyond this book.

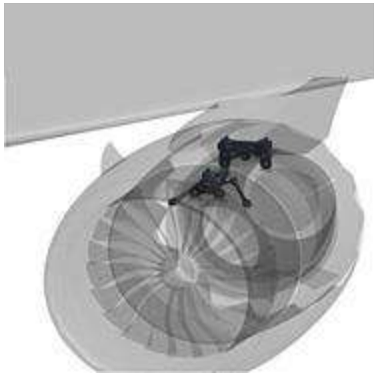
Structures and Loads

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 windmilling.
- 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 third 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.

Structures and Loads

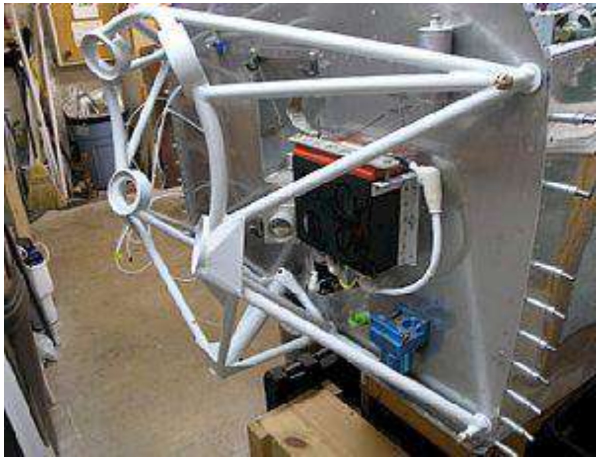


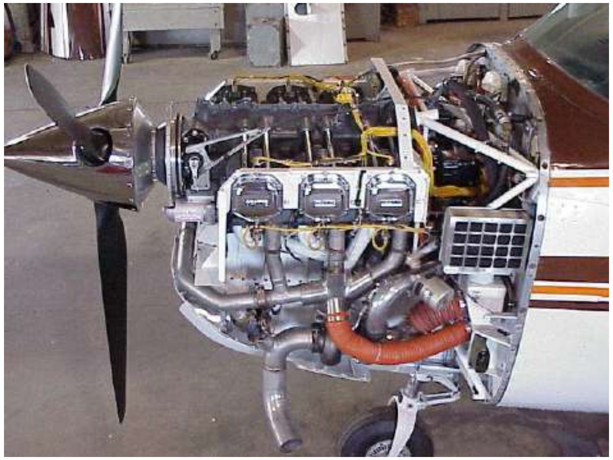


Structures and Loads

- 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.
- For 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 in normal operation to obtain the design torque for the engine mounts.

Structures and Loads





Structures and Loads

- 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 "hammershock" is especially severe.

Engine surges are associated with one or more heavy pessue pulses, generally known as hammer shock.

During the 1960s development of the Concorde Supersonic Transport (SST) a major incident occurred when a compressor surge caused a structural failure in the intake.

The hammershock which propagated forward from the compressor was of sufficient strength to cause an inlet ramp to become detached and expelled from the front of the intake. The ramp mechanism was strengthened and control laws changed to prevent a re-occurrence

Structures and Loads

Landing-gear Loads

- 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.

Structures and Loads

- 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 airloads 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.

Structures and Loads

Material Selection

- A number of properties are important to the selection of materials for an aircraft.

- Factors to be considered include

  - yield and ultimate strength, stiffness, density,
  - fracture toughness,
  - fatigue crack resistance,
  - creep, corrosion resistance, temperature limits,
  - producibility, repairability, cost, and availability.

- 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.

Structures and Loads

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

- This "fatigue" effect is 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 vibrations of the engine and propeller.

Structures and Loads

- Creep is the tendency of some materials to slowly and permanently deform under a low but sustained stress.
- For most aerospace materials, creep is a problem only at elevated temperatures.
- However, some titaniums, plastics, and composites will exhibit creep at room temperatures.
- Creep deformation data is presented in materials handbooks as a function of time, temperature, and stress loading.
- Corrosion of aircraft materials has been a major problem since the early days of aviation.
- Aircraft materials are exposed atmospheric moisture, salt-water spray, aircraft fuel, oils, hydraulic fluids, battery acid, engine exhaust products, missile plumes, gun gases, and even leaking toilets.

Structures and Loads

- Corrosion of materials is greatly accelerated when the materials experience a sustained stress level.
- The corrosion products at the surface tend to form a protective coating that delays further corrosion.
- When the material is subjected to a tension stress, however, cracks in the protective coating are formed that accelerate the corrosion.
- Once corrosion begins, it tends to follow cracks opened in the material by the stress.
- This "stress corrosion" can cause fracture at a stress level one tenth the normal ultimate stress level.
- For this reason it is important to avoid manufacturing processes that leave residual tension stresses.

Structures and Loads

Structures and Loads

- Operating temperature can play a major role in determining material suitability.
- Stainless steel or some other high-temperature material must be used as a firewall around the engine.
- For high-speed aircraft, aerodynamic heating may determine what materials may be used.
- Figure 14.21 shows typical skin temperatures at speeds of Mach 2.2 and 3.0.

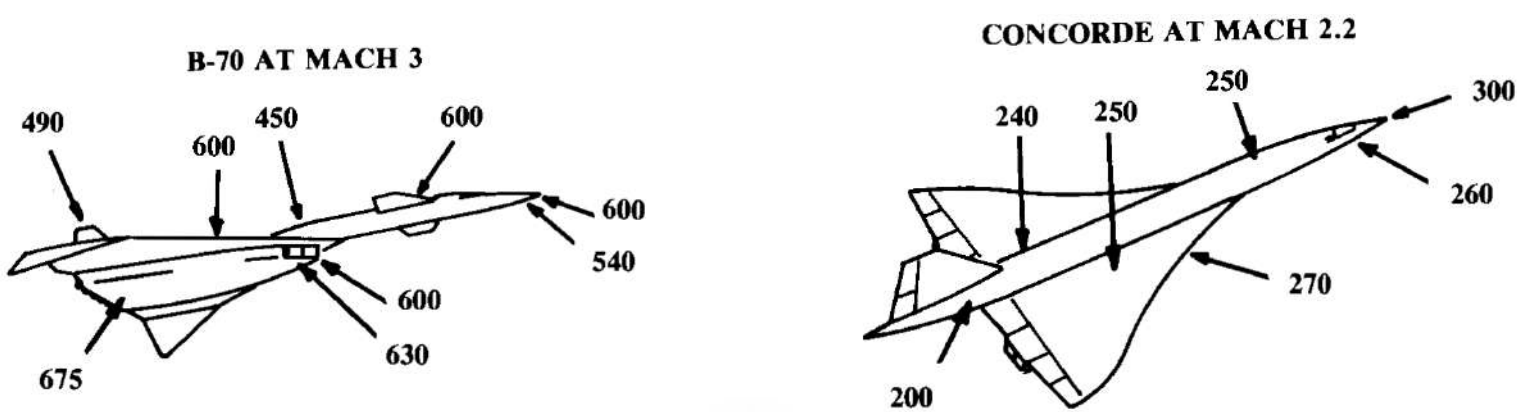


Fig. 14.21 Supersonic skin temperatures (°F).

Structures and Loads

- The stagnation (total) temperature is the highest possible temperature due to aerodynamic heating [Eq. (14.20)].

$$T_{\text{stagnation}} = T_{\text{ambient}}(1 + 0.2M^2) \tag{14.20}$$

- Actual skin temperatures are difficult to calculate because they depend upon the airflow conditions, surface finish, and atmospheric conditions.
- Figure 14.22 provides a reasonable estimate of the expected skin temperatures over most of the airframe

Structures and Loads

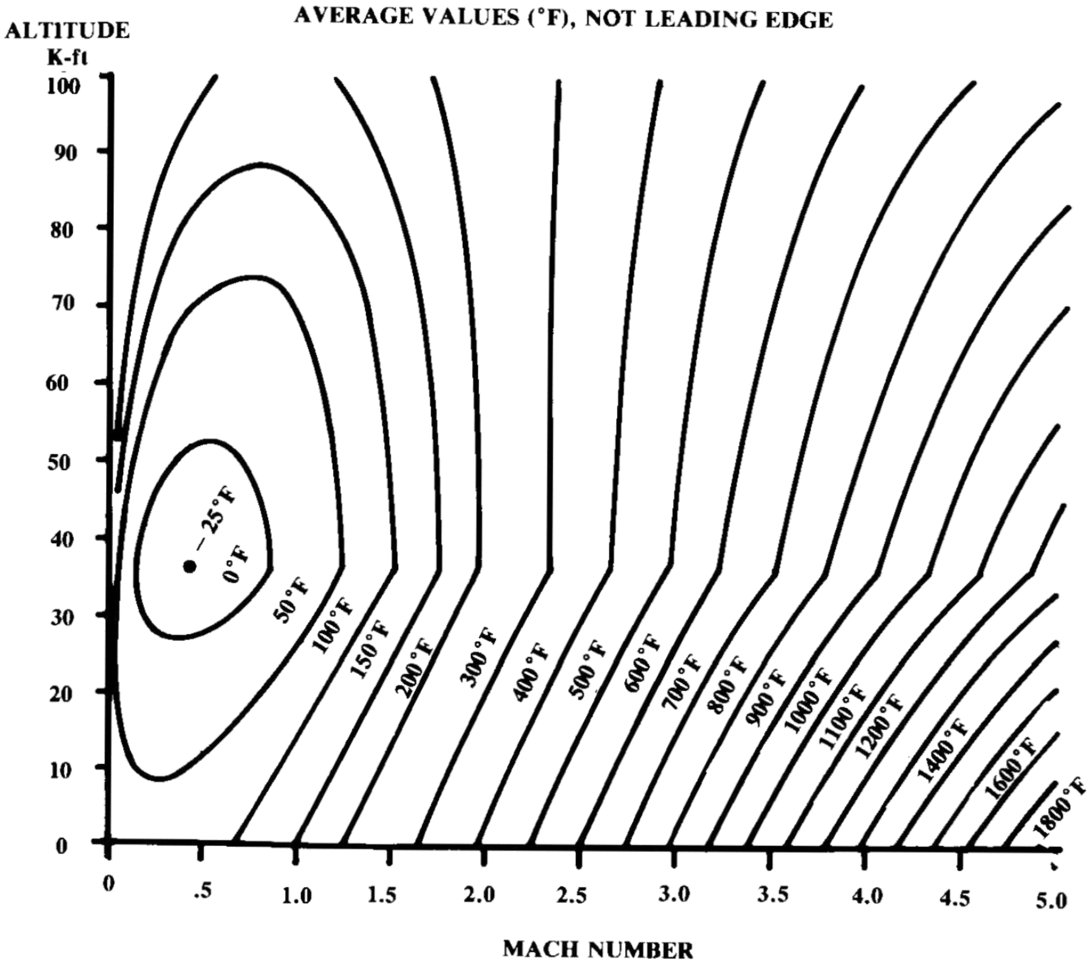


Fig. 14.22 Skin temperature estimate.

Structures and Loads

- Producibility and repairability are also important in material selection.
- As a rule, the better the material properties, the more difficult it is to work with.
- For example, a major difficulty in the development of the SR-71 was in learning how to work with the selected titanium alloy.
- Similarly, composite materials offer a large reduction in weight, but pose problems both in fabrication and repair.
- Cost is also important in material selection, both for raw material and fabrication. The better the material, the more it usually costs.

Structures and Loads

- Another factor to consider is material availability.
- Titanium and some of the materials used to produce high-temperature alloys are obtained from unfriendly or unstable countries, and it is possible that the supply may someday be cut off.
- Figures 14.23, 14.24, and 14.25 illustrate the materials selected for the Rockwell proposal for the X-29.
- These are typical of current fighter design practice.
- Note the stainless-steel heat shield and nozzle interface and the aluminum-honeycomb access doors.

Structures and Loads

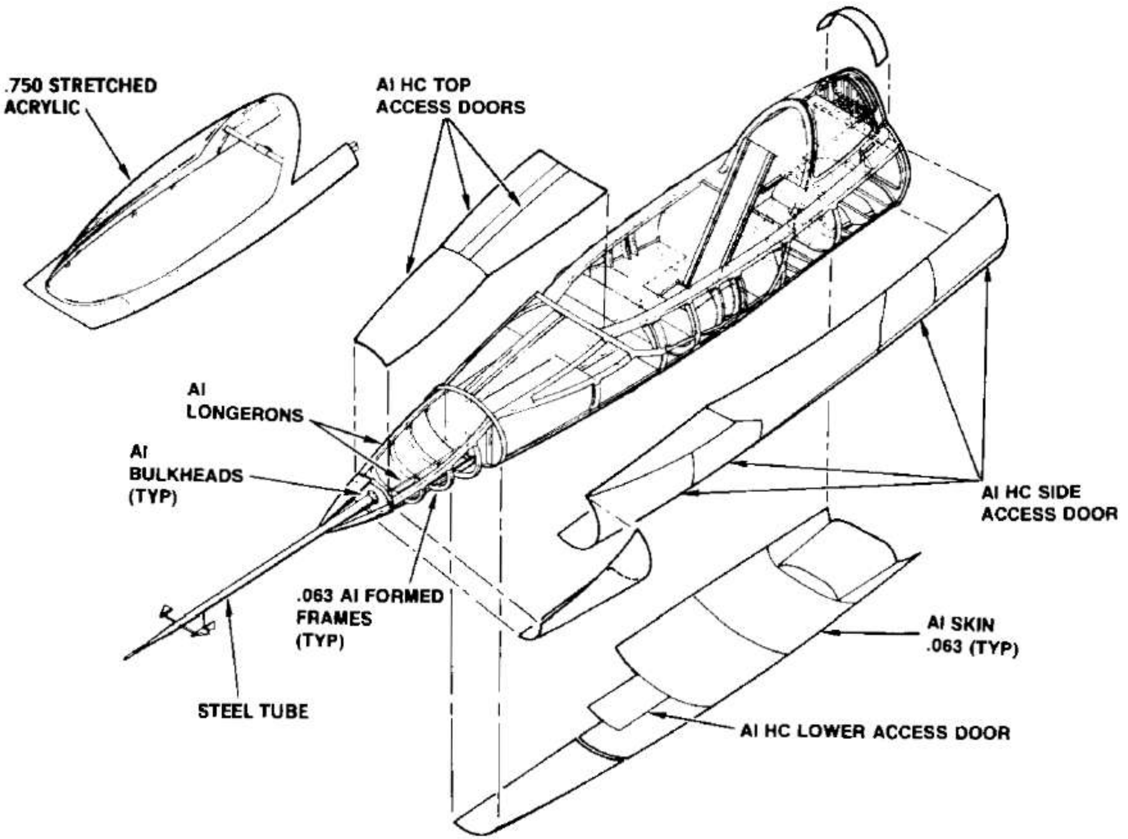


Fig. 14.23 Materials – forebody.

Structures and Loads

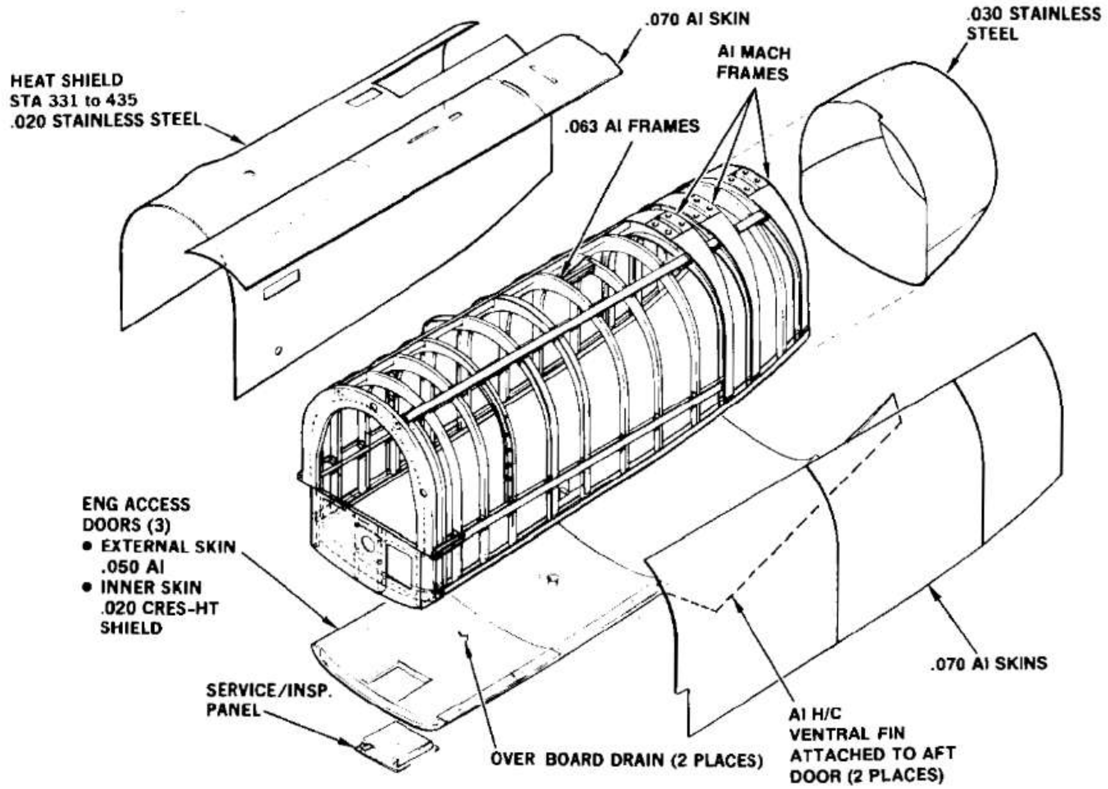


Fig. 14.24 Material selection—aft fuselage.

Structures and Loads

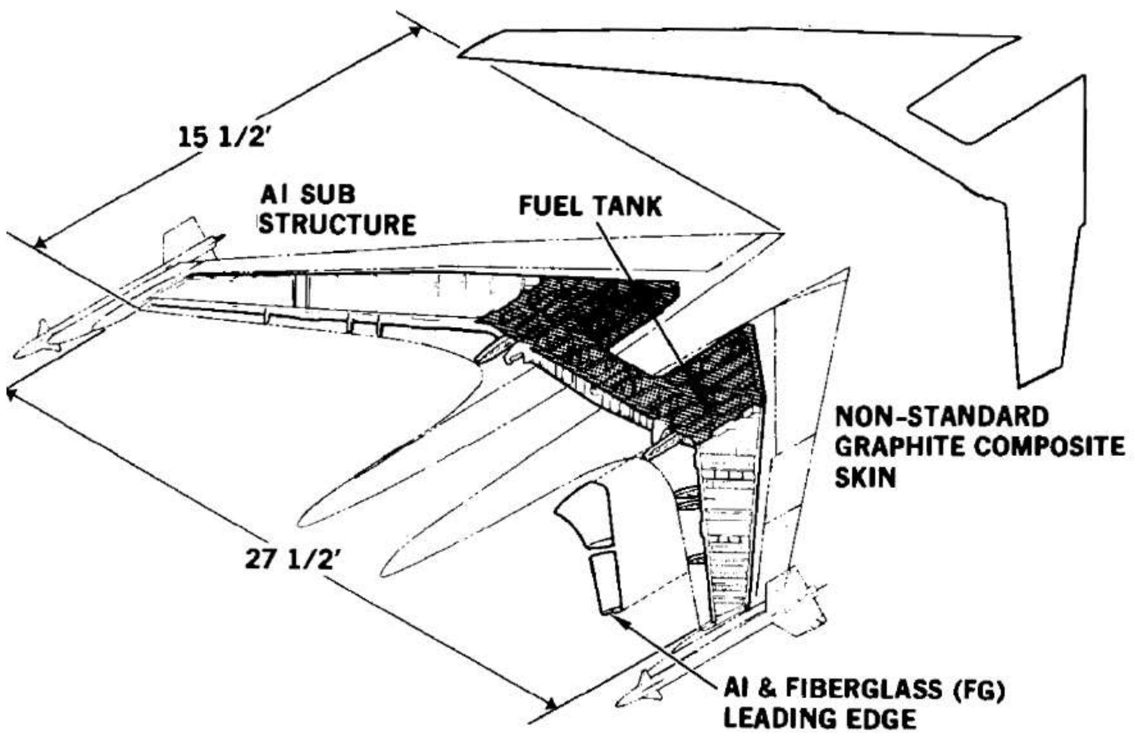


Fig. 14.25 Wing materials.

Structures and Loads

Material Properties

- This section covers various commonly-used aircraft materials.

Wood

- Wood is rarely used today in production aircraft.
- Wood offers good strength-to-weight ratio and is easy to fabricate and repair.
- It is actually much like composite materials in that it has different properties in different directions.
- Wood makes a natural bending beam for wing spars because of the lengthwise fibers.
- The disadvantages of wood are its sensitivity to moisture and its susceptibility to rot and insect damage.

Structures and Loads

- Today wood is used largely in homebuilt and specialty, low-volume production aircraft.
- However, the use of foam core and fiberglass-epoxy has largely replaced wood in homebuilt aircraft.

Aluminum

- Aluminum remains by far the most widely used aircraft material.
- It has an excellent strength-to-weight ratio, is readily formed, is of moderate cost, and is resistant to chemical corrosion.
- Aluminum is the most abundant metal in the Earth's crust.
- Being relatively soft, pure aluminum is alloyed with other metals for aircraft use.

Structures and Loads

- The most common aluminum alloy is "2024 (or 24ST)," sometimes called "duralumin."

- 2024 consists of 93 .5% aluminum, 4.4% copper, 1.5% manganese, and 0.6% magnesium.

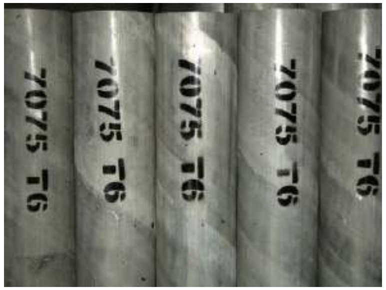
- For high-strength applications, the 7075 alloy is widely used.

- 7075 is alloyed with zinc, magnesium, and copper.

- Since the corrosion resistance is lessened by alloying, aluminum sheet is frequently "clad" with a thin layer of pure aluminum.

- Newer alloys such as 7050 and 7010 have improved corrosion resistance and strength.

2024 Duralumin



7075 Alloy

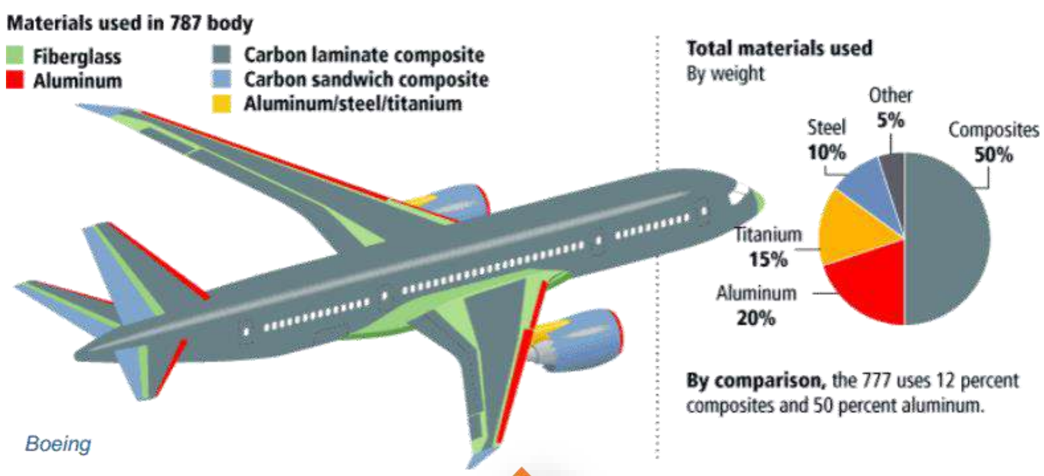
Structures and Loads

Steel

- Today steel is used for applications requiring high strength and fatigue resistance, such as wing attachment fittings.
- Also, steel is used wherever high temperatures are encountered such as for firewalls and engine mounts.
- Steel is primarily an alloy of iron and carbon, with the carbon adding strength to the soft iron.
- As carbon content increases, strength and brittleness increase.
- Typical steel alloys have about 1% of carbon.
- Other materials such as chromium, molybdenum, nickel, and cobalt are alloyed with steel to provide various characteristics.
- The "stainless steel" alloys are commonly used where corrosion resistance is important.

Structures and Loads

- The properties of steel are strongly influenced by heat treatment and tempering.
- The same alloy can have moderate strength and good ductility or can have much higher strength but at the expense of brittleness, depending upon the heat treatment and tempering employed.
- Steel is very cheap, costing about one-sixth what aluminum does.
- Steel is also easy to fabricate.



Structures and Loads

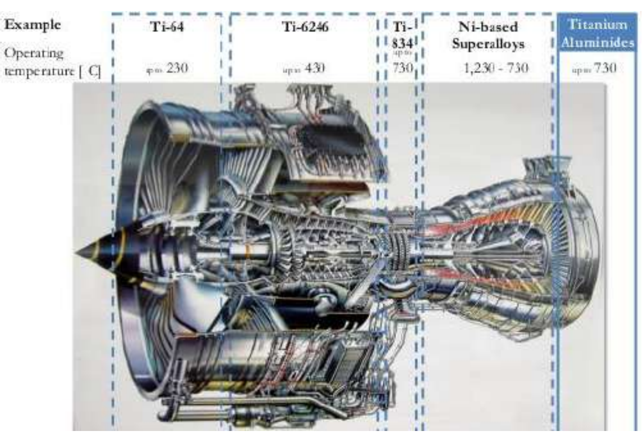
Titanium

- Titanium would seem to be the ideal aerospace material.
- It has a better strength-to-weight ratio and stiffness than aluminum, and is capable of temperatures almost as high as steel.
- Titanium is also corrosion-resistant.
- However, titanium is extremely difficult to form for these same reasons.
- Most titanium alloys must be formed at temperatures over 1000°F and at very high forming stresses.
- Titanium is very expensive, costing about five to ten times as much as aluminum.
- Much of the titanium comes from the Russia and China.

Top Titanium Producers
1.China.
2.Russia.
3.Japan

Structures and Loads

- To handle the aerodynamic heating of Mach 3 + flight, the structure of the SR-71 is about 93% titanium.



- Titanium is extensively used in jet-engine components,

- Engines parts manufactured from titanium include discs, blades, shafts and casings for the front fan to the rear end of the engine.

- and is also used in lower-speed aircraft for such high-stress airframe components as landing-gear beams and spindles for all-moving tails.

Structures and Loads

Magnesium

- Magnesium has a good strength-to-weight ratio, tolerates high temperatures, and is easily formed, especially by casting, forging, and machining.

- It has been used for engine mounts, wheels, control hinges, brackets, stiffeners, fuel tanks, and even wings.

- However, magnesium is very prone to corrosion and must have a protective finish.

- Furthermore, it is flammable.

Structures and Loads

High-Temperature Nickel Alloys

- Inconel, Rene 41, and Hastelloy are high-temperature nickel-based alloys suitable for hypersonic aircraft and re-entry vehicles.
- Inconel was used extensively in the X-15, and Rene 41 was to have been used in the X-20 Dynasoar.
- Hastelloy is used primarily in engine parts.
- These alloys are substantially heavier than aluminum or titanium, and are difficult to form.

X-15

X-20

Structures and Loads

Composites

- The greatest revolution in aircraft structures since the all-aluminum Northrop Alpha has been the ongoing adoption of composite materials for primary structure.
- In a typical aircraft part, the direct substitution of graphite-epoxy composite for aluminum yields a weight savings of 25%.
- Figure 14.26 shows the two major composite forms,
    - filament-reinforced
    - and whisker-reinforced.

Northrop Alpha

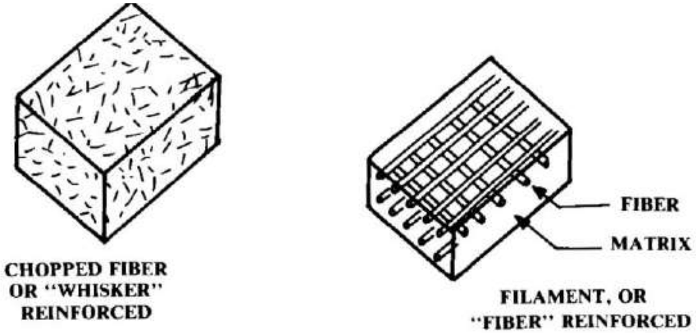


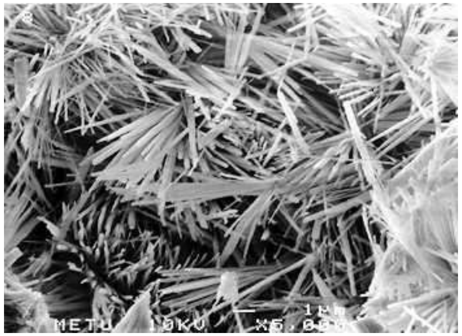
Fig. 14.26 Composite material types.

Structures and Loads

- In the whisker-reinforced composite, short strands of the reinforcing material are randomly located throughout the matrix.

- The most common example of this is chopped fiberglass, which is used for low-cost fabrication of boats.

- Whisker reinforcing is sometimes used in advanced metal matrix composites such as boron-aluminum.



boron-aluminum

- Most of the advanced composites used in aircraft structure are of the filament reinforced type because of outstanding strength-to-weight ratio.

- Also, filament composites may have their structural properties tailored to the expected loads in different directions.

Structures and Loads

- Metals and whisker-reinforced composites are isotropic, having the same material properties in all directions.
- Filament composites, like wood, are strongest in the direction the fibers are running.
- If a structural element such as a spar cap is to carry substantial load in only one direction, all the fibers can be oriented in that direction.
- This offers a tremendous weight savings.
- Figure 14.27 shows four common arrangements for tailoring fiber orientation.

Structures and Loads

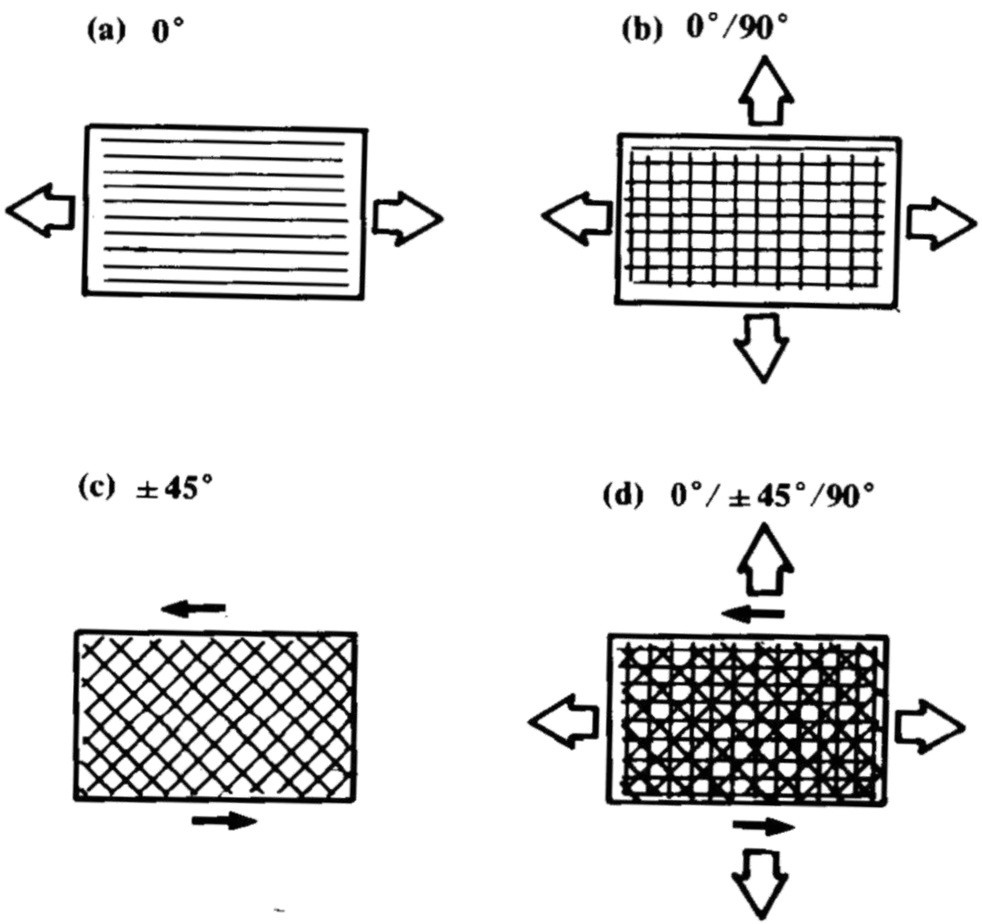
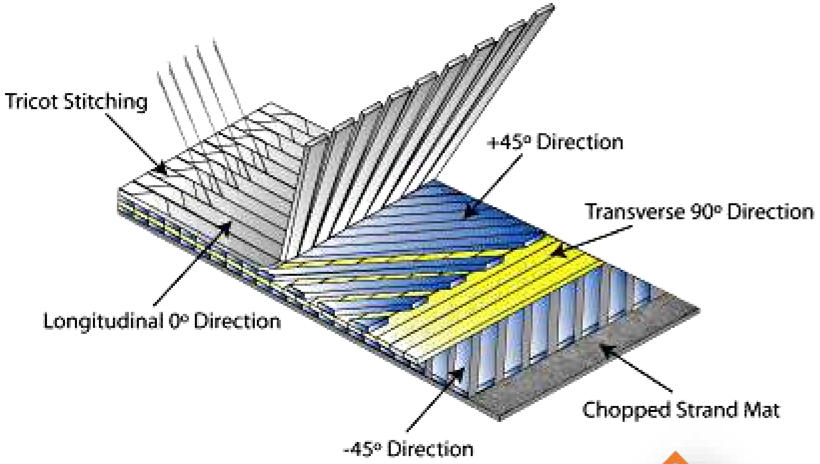


Fig. 14.27 Composite ply tailoring.

Structures and Loads

- In (c), the fibers are at 45-deg angles with the principle axis.
- This provides strength in those two directions, and also provides good shear strength in the principle-axis direction.
- For this reason, this arrangement is commonly seen in a composite-wing-box shear web.
- Also, the 45-deg orientation is frequently used in structure that must resist torque.



Structures and Loads

Prepreg Carbon Fiber

fiberglass-epoxy Radome

- There are a number of fiber and matrix materials used in composite aircraft structure.
- Fiberglass with an epoxy-resin matrix has been used for years for such nonstructural components as radomes and minor fairings.
- More recently, fiberglass-epoxy has been used by homebuilders.
- While fiberglass-epoxy has good strength characteristics, its excessive flexibility (tensile E) prevents its use in highly loaded structure in commercial or military aircraft.
- However, it is cheap and easy to form, and is suitable for some applications.

Structures and Loads

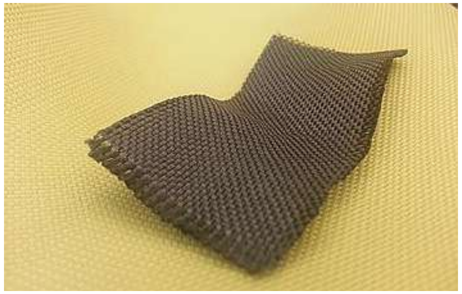
- The most commonly used advanced composite is graphite-epoxy, called "carbon-fiber composite".
- Graphite-epoxy composite has excellent strength-to-weight ratio and is not difficult to mold.
- It is substantially more expensive than aluminum,
- but unlike metals, little material is wasted in manufacturing operations such as milling and cutting from flat patterns.
- Boron-epoxy was initially used for complete part fabrication.
- An F-111 horizontal tail and F-4 rudder were built of boron-epoxy.

F-4

Structures and Loads

- Aramid, sold under the trade name "Kevlar," is used with an epoxy matrix in lightly-loaded applications.
- A graphite-aramid-epoxy hybrid composite offers more ductility than pure graphite-epoxy.
- It is used in the Boeing 757 for fairings and landing-gear doors.
- Composites using epoxy as the matrix are limited to maximum temperatures of about 350°F, and normally aren't used in applications where temperatures will exceed 260°F.
- For higher-temperature applications, several advanced matrix materials are in development.
- Furthermore, composites are difficult to repair because of the need to match strength and stiffness characteristics.

Kevlar is a heat-resistant and strong synthetic fiber



Structures and Loads

Sandwich Construction

- Sandwich construction has special characteristics and is very important to aircraft design.
- A structural sandwich is composed of two "face sheets" bonded to and separated by a "core" (Fig. 14.29).
- The face sheets can be of any material, but are typically aluminum, fiberglass-epoxy, or graphite-epoxy.
- The core is usually an aluminum or phenolic honeycomb material for commercial and military aircraft, but various types of rigid foam are used as the core in some cases.

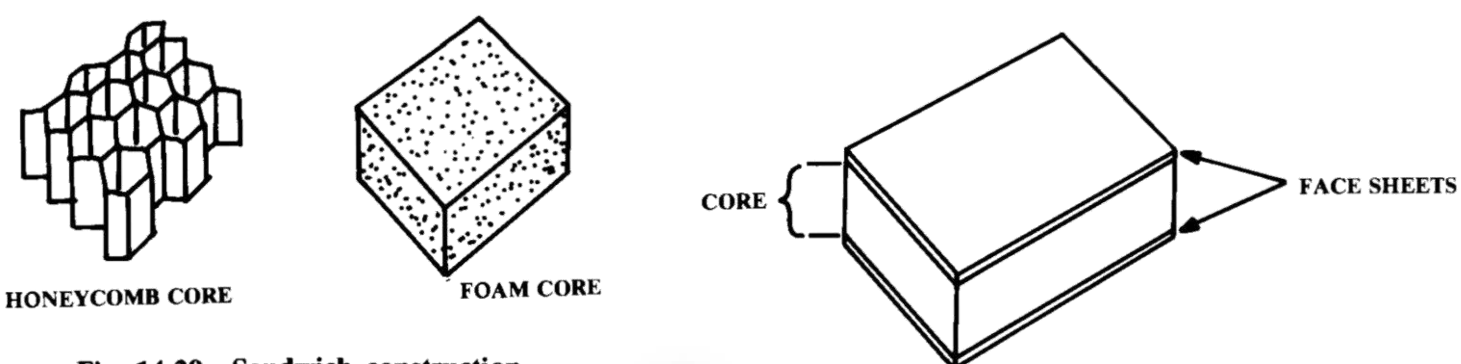


Fig. 14.29 Sandwich construction.

Structures and Loads

- In a sandwich, the face sheets carry most of the tension and compression loads due to bending.

- The core carries most of the shear loads as well as the compression loads perpendicular to the skin.

Structures and Loads

Material-Property Tables

- Tables 14.3, 14.4 provide typical material properties for various metals, composites, and woods.

- Note that these are typical values only.

Table 14.3 Typical metal properties (room temperature)

Material	Density lb/in.^{3}	Temp limit °F	F_{tu} 10^3 psi	F_{ty} 10^3 psi	F_{cy} 10^3 psi	F_{su} 10^3 psi	E 10^6 psi	G 10^6 psi	Comments
Steel									
Aircraft steel (5 Cr-Mo-V)	0.281	1000	260	220	240	155	30	11	Heat treat to 1850°F
Low carbon steel (AISI 1025)	0.284	900	55	36	36	35	29	11	Shop use only today
Low alloy steel (D6AC-wrought)	0.283	1000	220	190	198	132	29	11	–
Chrom-moly steel (AISI 4130)									
sheet, plate, & tubing	0.283	900	90	70	70	54	29	11	Widely used
wrought	0.283	900	180	163	173	108	29	11	–
Stainless steel (AM-350)	0.282	800	185	150	158	120	29	11	Good corrosion resistance
Stainless (PH 15-7 Mo-sheet & plate)	0.277	600	190	170	179	123	29	11	B-70 honeycomb material
Aluminum									
Aluminum-2017	0.101	250	55	32	32	33	10.4	3.95	–
Clad 2024 (24 st)-(sheet & plate)	0.100	250	61	45	37	37	10.7	4.0	Widely used, weldable
extrusions	0.100	250	70	52	49	34	10.8	4.1	
Clad 7178-T6 (78 st)									
-(sheet & plate)	0.102	250	80	71	71	48	10.3	3.9	High strength, not weldable, subject to stress corrosion
extrusions	0.102	250	84	76	75	42	10.4	4.0	
Clad 7075-T6-(sheet)	0.101	250	72	64	63	43	10.3	3.9	High strength, not weldable, common in high-speed aircraft
(forgings)	0.101	250	74	63	66	43	10.0	3.8	
(extrusions)	0.101	250	81	72	72	42	10.4	4.0	
Magnesium									
Magnesium-HK 31A	0.0674	700	34	24	22	23	6.5	2.4	High-temperature, high strength-to-weight, subject to corrosion
-HM 21A	0.0640	800	30	21	17	19	6.5	2.4	

Structures and Loads

Table 14.3 (continued) Typical metal properties (room temperature)

Material	Density lb/in.^{3}	Temp limit °F	F_{tu} 10^3 psi	F_{ty} 10^3 psi	F_{cy} 10^3 psi	F_{su} 10^3 psi	E 10^6 psi	G 10^6 psi	Comments
Titanium									Most-used titanium, including
Titanium-Ti-6A1-4V	0.160	750	160	145	154	100	16.0	6.2	B-70
-Ti-13V-11Cr-3Al	0.174	600-1000	170	160	162	105	15.5	–	SR-71 titanium
High temperature nickel alloys									
Inconel X-750	0.300	1000-1500	155	100	100	101	31.0	11.0	X-15
Rene 41	0.298	1200-1800	168	127	135	107	31.6	12.1	X-20, very difficult to form
Hastelloy B	0.334	1400	100	45	–	–	30.8	–	Engine parts

Table 14.4 Typical composite material properties (room temperature)

Material	Fiber orientation	Fiber % volume	Density lb/in^3	Temp. limit °F	$F_{tu}(L)$ 10^3 psi	$F_{tu}(T)$ 10^3 psi	$F_{cu}(L)$ 10^3 psi
High-strength Graphite/epoxy	0	60	0.056	350	180.0	8.0	180.0
	±45	60	0.056	350	23.2	23.2	23.9
High-modulus Graphite/epoxy	0	60	0.056	350	110.0	4.0	100
	±45	60	0.058	350	16.9	16.9	18
Boron/epoxy	0	50	0.073	350	195	10.4	353
Graphite/polyimide	0	–	–	–	204	4.85	111
S-Fiberglass/epoxy	0	–	0.074	350	219	7.4	73.9
E-Fiberglass/epoxy	0	45	0.071	350	105	10.2	69
Aramid/epoxy	0	60	.052	350	200	4.3	40

L – Longitudinal direction; T = transverse direction; F_{isu} = interlaminate shear stress (ultimate); t = tension; c = compression.

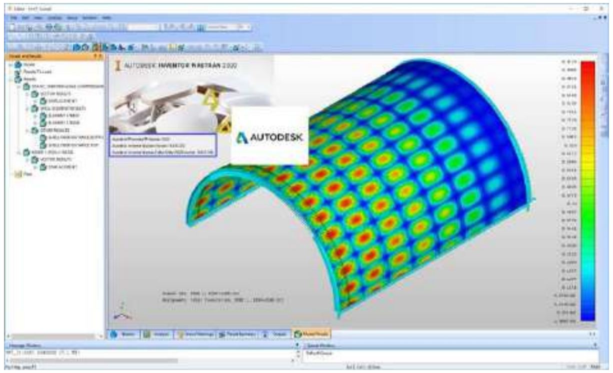
Structures and Loads

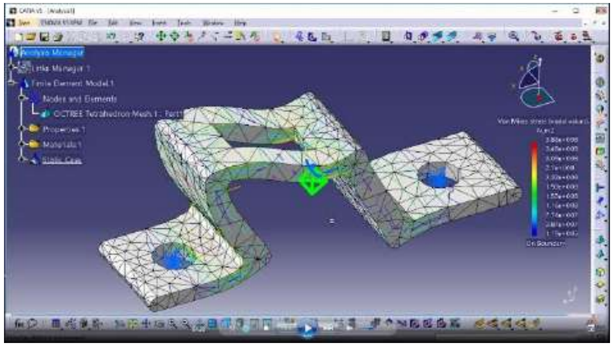
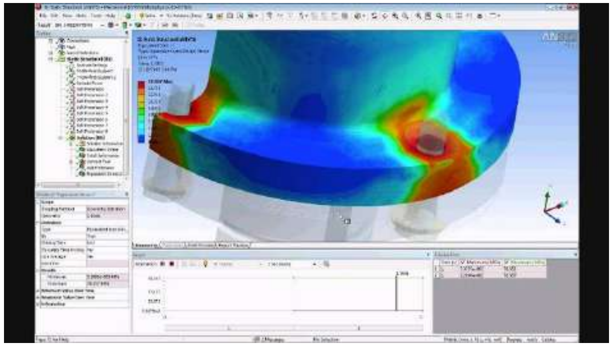
Table 14.4 (contd.) Typical composite material properties (room temperature)

$F_{cu}(T)$ 10^3 psi	$F_{su}(LT)$ 10^3 psi	F_{isu} 10^3psi	$\epsilon_{tu}(L)$ in/in	$\epsilon_{tu}(T)$ in/in	$E_t(L)$ 10^6 psi	$E_t(T)$ 10^6 psi	$E_C(L)$ 10^6 psi	$E_C(T)$ 10^6 psi	$G(LT)$ 10^6 psi
30.0	12	13	0.0087	0.0048	21.00	1.70	21.00	1.70	0.65
23.9	65.5	–	0.022	0.022	2.34	2.34	2.34	2.34	5.52
20	9.0	10	0.0046	0.0025	25.00	1.70	25.00	1.70	0.65
18	43.2	–	0.012	0.012	2.38	2.38	2.38	2.38	6.46
40	15.3	13	0.0065	0.004	30	2.7	30	2.7	0.70
18.5	8.5	–	–	0.0036	20	1.35	17.4	1.4	0.84
22.4	–	11	–	–	7.70	2.70	6.80	2.5	–
33	7.9	–	0.025	0.019	4.23	1.82	4.43	1.8	0.51
20	9	–	0.018	0.006	11	0.8	11	0.8	0.3

Structures and Loads

- Today, virtually all major structural analysis is now performed using finite-element computer programs.
- The industry-standard FEM program is the "NASTRAN (NAsa STRuctural ANalysis)" program, developed years ago for NASA.
- Another commercial programs are Ansys, Catia…



Weights

- The estimation of the weight of a conceptual aircraft is a critical part of the design process.
- The first is a crude component buildup based upon planform areas, wetted areas, and percents of gross weight.
- This technique is useful for initial balance calculations and can be used to check the results of the more detailed statistical methods.
- The second uses detailed statistical equations for the various components.
- This technique is sufficiently detailed to provide a credible estimate of the weights of the major component groups.
- Those weights are usually reported in groupings as defined by MIL-STD-1374

Weights

- At the conceptual level the weights are reported via a "Summary Group Weight Statement."
- A typical summary format appears as Table 15.1, where the empty weight groups are further classified into three major groupings (structure, propulsion, and equipment).

Table 15.1 Group weight format

Group	Group
STRUCTURES GROUP	EQUIPMENT GROUP
Wing	Flight controls
Tail-horizontal/canard	APU
vertical	Instruments
ventral	Hydraulic
Body	Pneumatic
Alighting gear-main	Electrical
auxiliary	Avionics
arresting gear	Armament
catapult gear	Furnishings
Nacelle/engine section	Air conditioning/ECS
Air induction system	Anti-icing
	Photographic
	Load and handling

Weights

PROPULSION GROUP

- Engine—as installed
- Accessory gearbox and drive
- Exhaust system
- Cooling provisions
- Engine controls
- Starting system
- Fuel system/tanks

TOTAL WEIGHT EMPTY

USEFUL LOAD GROUP

- Crew
- Fuel-usable
  - -trapped
- Oil
- Passengers
- Cargo/baggage
- Guns
- Ammunition
- Pylons and racks
- Expendable weapons
- Flares/chaff

TAKEOFF GROSS WEIGHT

- Flight design gross weight
- Landing design gross weight
- DCPR weight

"DCPR" stands for "Defense Contractors Planning Report." The DCPR weight is important for cost estimation,

Weights

- In a Group Weight Statement, the distance to the weight datum (arbitrary reference point) is included, and the resulting moment is calculated.

- These are summed and divided by the total weight to determine the actual center of gravity (c.g.) location.

- The c.g. varies during flight as fuel is burned off and weapons expended.

- To determine if the c.g. remains within the limits established by an aircraft stability and control analysis, a "c.g.-envelope" plot is prepared (Fig. 15.1).

Weights

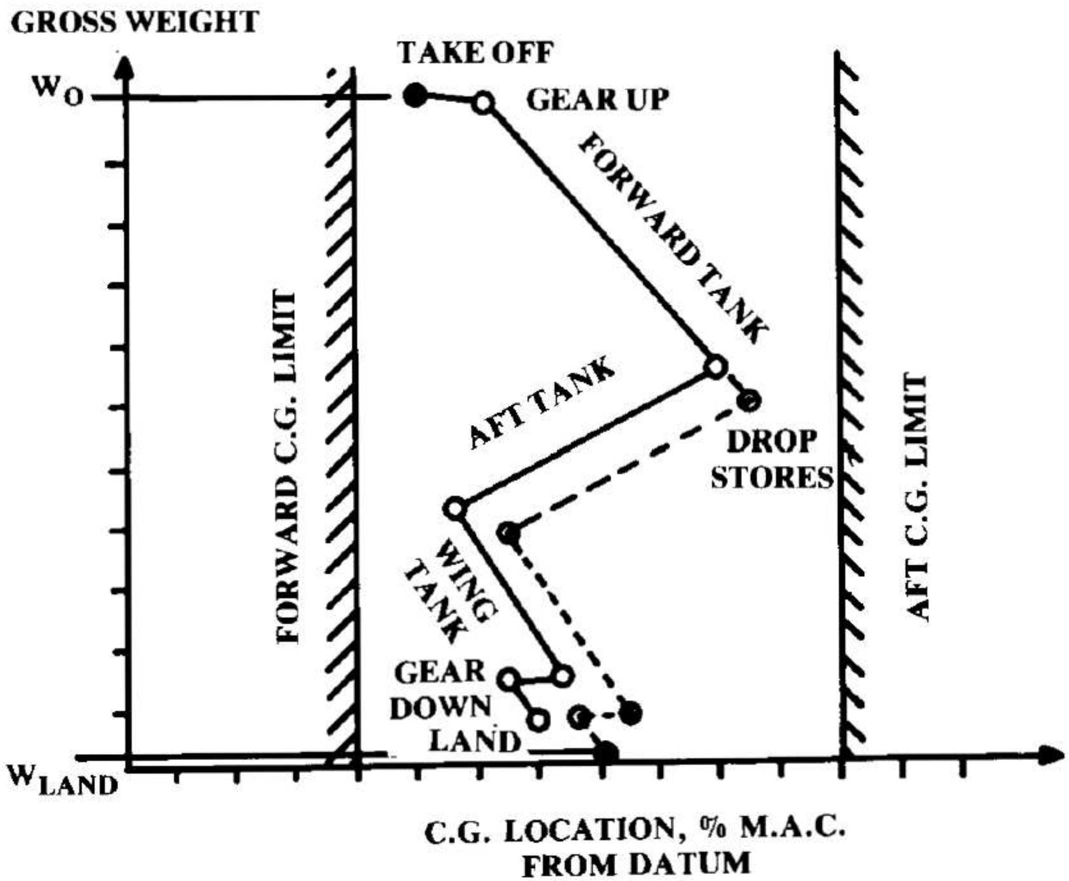
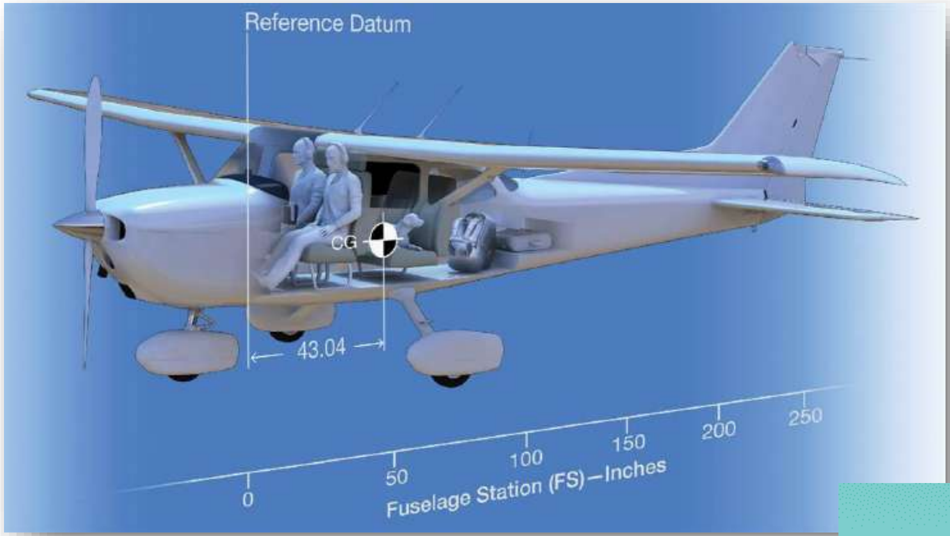
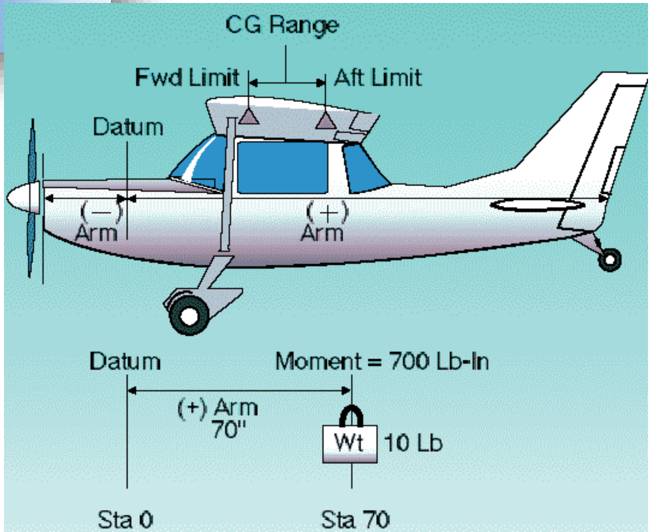


Fig. 15.1 C.G. envelope diagram.

Weights


Reference Datum
CG
43.04
0
50
100
150
200
250
Fuselage Station (FS)—Inches


CG Range
Fwd Limit
Aft Limit
Datum
(−)
Arm
(+)
Arm
Datum
Moment = 700 Lb-In
(+) Arm
70"
Wt
10 Lb
Sta 0
Sta 70

Weights

- The c.g. must remain within the specified limits as fuel is burned, and whether or not the weapons are expended.
- It is permissible to "sequence" the fuel tanks, selecting to burn fuel from different tanks at different times to keep the c.g. within limits.
- However, an automated fuel-management system must be used, and that imposes additional cost and complexity.
- The allowable limits on the c.g. vary with Mach number.
- At supersonic speeds the aerodynamic center moves rearward, so the forward c.g. limit may have to move rearward to allow longitudinal trim at supersonic speeds.
- However, the aft-c.g. limit is often established by the size of the vertical tail, which loses effectiveness at supersonic speeds.

Weights

Approximate Group Weights Method

- Early in design it is desirable to do a rough e.g. estimate.
- Otherwise, substantial rework may be required after the c.g. is properly estimated.
- A rough c.g. estimate can be done with a crude statistical approach as provided in Table 15.2.

example $Weight_{wing} = 9 * S_{exposed\ planform}$ pounds (fighter)

Table 15.2 Approximate empty weight buildup

Item	Fighters	Transports and bombers	General aviation	Multiplier[a]	Approximate location
Wing	9.0	10.0	2.5	$S_{exposed\ planform}$ ft^2	40% MAC
Horizontal tail	4.0	5.5	2.0	$S_{exposed\ planform}$ ft^2	40% MAC
Vertical tail	5.3	5.5	2.0	$S_{exposed\ planform}$ ft^2	40% MAC
Fuselage	4.8	5.0	1.4	$S_{wetted\ area}$ ft^2	40–50% length
Landing gear[b]	.033 .045 Navy	.043	.057	TOGW (lb)	–
Installed engine	1.3	1.3	1.4	Engine weight (lb)	–
"All-else empty"	.17	.17	.10	TOGW (lb)	40–50% length

[a]Results are in pounds.
[b]15% to nose gear; 85% to main gear.

Weights

- The wing and tail weights are determined from historical values for the weight per square foot of exposed planform area.
- The fuselage is similarly based upon its wetted area.
- The landing gear is estimated as a fraction of the takeoff gross weight.
- The installed engine weight is a multiple of the uninstalled engine weight.
- Finally, a catch-all weight for the remaining items of the empty weight is estimated as a fraction of the takeoff gross weight.

Weights

Statistical Group Weights Method

- A more refined estimate of the group weights applies statistical equations based upon regression analysis.

- Development of these equations represents a major effort, and each company develops its own equations.

- The equations presented below typify those used in conceptual design by the major airframe companies, and cover fighter/attack, transport, and general-aviation aircraft.

- It should be understood that there are no "right" answers in weights estimation until the first aircraft flies.

- However, these equations should provide a reasonable estimate of the group weights.

- It's a good idea to calculate the weight of each component using several different equations and then select an average, reasonable result.

Weights

- Table 15.3 tabulates various miscellaneous weights.

- When the component weights are estimated using these or similar methods, they are tabulated in a format similar to that of Table 15.1 and are summed to determine the empty weight.

Table 15.3 Miscellaneous weights (approximate)

Missiles	
Harpoon (AGM-84 A)	1200 lb
Phoenix (AIM-54 A)	1000 lb
Sparrow (AIM-7)	500 lb
Sidewinder (AIM-9)	200 lb
Pylon and launcher	$.12\ W_{\text{missile}}$
M61 Gun	
Gun	250 lb
940 rds ammunition	550 lb
Seats	
Flight deck	60 lb
Passenger	32 lb
Troop	11 lb

Weights

Instruments	
Altimeter, airspeed, accelerometer, rate of climb, clock, compass, turn & bank, Mach, tachometer, manifold pressure, etc.	1–2 lb each
Gyro horizon, directional gyro	4–6 lb each
Heads-up display	40 lb
Lavatories	
Long range aircraft	$1.11\ N_{pass}^{1.33}$
Short range aircraft	$0.31\ N_{pass}^{1.33}$
Business/executive aircraft	$3.90\ N_{pass}^{1.33}$
Arresting gear	
Air Force-type	$.002\ W_{dg}$
Navy-type	$.008\ W_{dg}$
Catapult gear	
Navy carrier-based	$.003\ W_{dg}$
Folding Wing	
Navy carrier based	$.06\ W_{wing}$

Weights

Fighter/Attack Weights

$$W_{\text{wing}} = 0.0103 K_{dw} K_{vs} (W_{dg} N_z)^{0.5} \; S_w^{0.622} A^{0.785} \; (t/c)_{\text{root}}^{-0.4} \times (1+\lambda)^{0.05} \; (\cos\Lambda)^{-1.0} \; S_{csw}^{0.04} \quad (15.1)$$

$$W_{\text{horizontal tail}} = 3.316 \left(1 + \frac{F_w}{B_h}\right)^{-2.0} \left(\frac{W_{dg} N_z}{1000}\right)^{0.260} S_{ht}^{0.806} \quad (15.2)$$

$$W_{\text{vertical tail}} = 0.452 K_{rht} (1 + H_t/H_v)^{0.5} (W_{dg} N_z)^{0.488} S_{vt}^{0.718} M^{0.341} \times L_t^{-1.0} (1 + S_r/S_{vt})^{0.348} A_{vt}^{0.223} (1+\lambda)^{0.25} (\cos\Lambda_{vt})^{-0.323} \quad (15.3)$$

$$W_{\text{fuselage}} = 0.499 K_{dwf} W_{dg}^{0.35} N_z^{0.25} L^{0.5} D^{0.849} W^{0.685} \quad (15.4)$$

$$W_{\substack{\text{main landing}\\\text{gear}}} = K_{cb} K_{tpg} (W_l N_l)^{0.25} L_m^{0.973} \quad (15.5)$$

$$W_{\substack{\text{nose landing}\\\text{gear}}} = (W_l N_l)^{0.290} L_n^{0.5} N_{nw}^{0.525} \quad (15.6)$$

Weights

$$W_{\substack{\text{engine}\\ \text{mounts}}} = 0.013 N_{\text{en}}^{0.795} T^{0.579} N_z \tag{15.7}$$

$$W_{\text{firewall}} = 1.13 S_{\text{fw}} \tag{15.8}$$

$$W_{\substack{\text{engine}\\ \text{section}}} = 0.01 W_{\text{en}}^{0.717} N_{\text{en}} N_z \tag{15.9}$$

$$W_{\substack{\text{air induction}\\ \text{system}}} = 13.29 K_{\text{vg}} L_d^{0.643} K_d^{0.182} N_{\text{en}}^{1.498} (L_s/L_d)^{-0.373} D_e \tag{15.10}$$

where K_d and L_s are from Fig. 15.2.

$$W_{\text{tailpipe}} = 3.5 D_e L_{\text{tp}} N_{\text{en}} \tag{15.11}$$

$$W_{\substack{\text{engine}\\ \text{cooling}}} = 4.55 D_e L_{\text{sh}} N_{\text{en}} \tag{15.12}$$

Weights

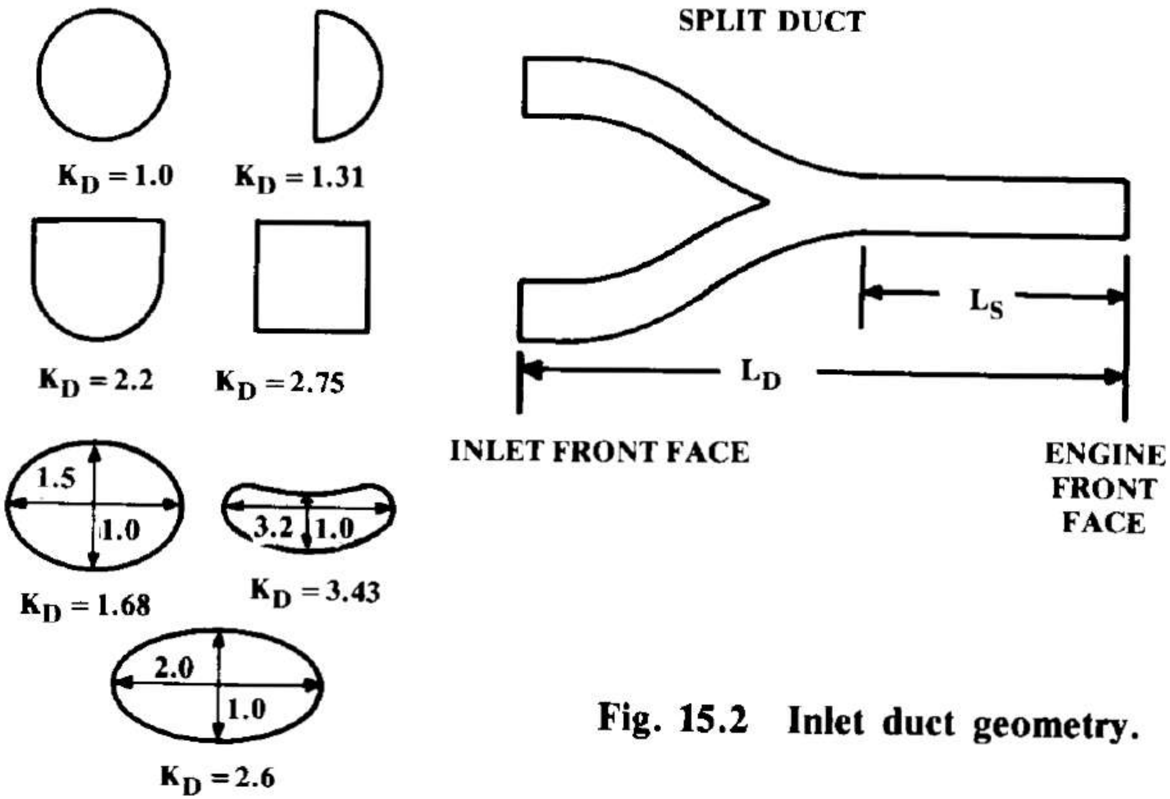


Fig. 15.2 Inlet duct geometry.

Weights

$$W_{\text{oil cooling}} = 37.82 N_{\text{en}}^{1.023} \tag{15.13}$$

$$W_{\substack{\text{engine}\\\text{controls}}} = 10.5 N_{\text{en}}^{1.008} L_{\text{ec}}^{0.222} \tag{15.14}$$

$$W_{\substack{\text{starter}\\\text{(pneumatic)}}} = 0.025\, T_e^{0.760} N_{\text{en}}^{0.72} \tag{15.15}$$

$$W_{\substack{\text{fuel system}\\\text{and tanks}}} = 7.45\, V_t^{0.47} \left(1 + \frac{V_i}{V_t}\right)^{-0.095} \left(1 + \frac{V_p}{V_t}\right) N_t^{0.066} N_{\text{en}}^{0.052} \left(\frac{T \cdot \text{SFC}}{1000}\right)^{0.249} \tag{15.16}$$

$$W_{\substack{\text{flight}\\\text{controls}}} = 36.28 M^{0.003} S_{\text{cs}}^{0.489} N_s^{0.484} N_c^{0.127} \tag{15.17}$$

$$W_{\text{instruments}} = 8.0 + 36.37 N_{\text{en}}^{0.676} N_t^{0.237} + 26.4(1 + N_{ci})^{1.356} \tag{15.18}$$

Weights

$$W_{\text{hydraulics}} = 37.23\, K_{\text{vsh}} N_u^{0.664} \tag{15.19}$$

$$W_{\text{electrical}} = 172.2\, K_{mc} R_{\text{kva}}^{0.152} N_c^{0.10} L_a^{0.10} N_{\text{gen}}^{0.091} \tag{15.20}$$

$$W_{\text{avionics}} = 2.117\, W_{\text{uav}}^{0.933} \tag{15.21}$$

$$W_{\text{furnishings}} = 217.6 N_c \tag{15.22}$$

$$W_{\substack{\text{air conditioning}\\ \text{and anti-ice}}} = 201.6[(W_{\text{uav}} + 200\, N_c)/1000]^{0.735} \tag{15.23}$$

$$W_{\substack{\text{handling}\\ \text{gear}}} = 3.2 \times 10^{-4}\, W_{\text{dg}} \tag{15.24}$$

Weights

Cargo/Transport Weights

$$W_{\text{wing}} = 0.0051 (W_{dg} N_z)^{0.557} S_w^{0.649} A^{0.5} (t/c)_{\text{root}}^{-0.4} (1+\lambda)^{0.1} \times (\cos\Lambda)^{-1.0} S_{csw}^{0.1} \quad (15.25)$$

$$W_{\substack{\text{horizontal}\\ \text{tail}}} = 0.0379 K_{uht} (1 + F_w/B_h)^{-0.25} W_{dg}^{0.639} N_z^{0.10} S_{ht}^{0.75} L_t^{-1.0} \times K_y^{0.704} (\cos\Lambda_{ht})^{-1.0} A_h^{0.166} (1 + S_e/S_{ht})^{0.1} \quad (15.26)$$

$$W_{\substack{\text{vertical}\\ \text{tail}}} = 0.0026 (1 + H_t/H_v)^{0.225} W_{dg}^{0.556} N_z^{0.536} L_t^{-0.5} S_{vt}^{0.5} K_z^{0.875} \times (\cos\Lambda_{vt})^{-1} A_v^{0.35} (t/c)_{\text{root}}^{-0.5} \quad (15.27)$$

$$W_{\text{fuselage}} = 0.3280 K_{door} K_{Lg} (W_{dg} N_z)^{0.5} L^{0.25} S_f^{0.302} (1 + K_{ws})^{0.04} (L/D)^{0.10} \quad (15.28)$$

$$W_{\substack{\text{main landing}\\ \text{gear}}} = 0.0106 K_{mp} W_l^{0.888} N_l^{0.25} L_m^{0.4} N_{mw}^{0.321} N_{mss}^{-0.5} V_{stall}^{0.1} \quad (15.29)$$

$$W_{\substack{\text{nose landing}\\ \text{gear}}} = 0.032 K_{np} W_l^{0.646} N_l^{0.2} L_n^{0.5} N_{nw}^{0.45} \quad (15.30)$$

Weights

$$W_{\substack{\text{nacelle}\\ \text{group}}} = 0.6724 K_{ng} N_{Lt}^{0.10} N_w^{0.294} N_z^{0.119} W_{ec}^{0.611} N_{\text{en}}^{0.984} S_n^{0.224} \quad \text{(includes air induction)} \tag{15.31}$$

$$W_{\substack{\text{engine}\\ \text{controls}}} = 5.0 N_{\text{en}} + 0.80 L_{\text{ec}} \tag{15.32}$$

$$W_{\substack{\text{starter}\\ \text{(pneumatic)}}} = 49.19 \left(\frac{N_{\text{en}} W_{\text{en}}}{1000} \right)^{0.541} \tag{15.33}$$

$$W_{\substack{\text{fuel}\\ \text{system}}} = 2.405\, V_t^{0.606} (1 + V_i/V_t)^{-1.0} (1 + V_p/V_t) N_t^{0.5} \tag{15.34}$$

$$W_{\substack{\text{flight}\\ \text{controls}}} = 145.9 N_f^{0.554} (1 + N_m/N_f)^{-1.0} S_{\text{cs}}^{0.20} (I_y \times 10^{-6})^{0.07} \tag{15.35}$$

$$W_{\substack{\text{APU}\\ \text{installed}}} = 2.2\, W_{\substack{\text{APU}\\ \text{uninstalled}}} \tag{15.36}$$

$$W_{\text{instruments}} = 4.509 K_r K_{\text{tp}} N_c^{0.541} N_{\text{en}} (L_f + B_w)^{0.5} \tag{15.37}$$

$$W_{\text{hydraulics}} = 0.2673 N_f (L_f + B_w)^{0.937} \tag{15.38}$$

Weights

$$W_{\text{electrical}} = 7.291\,R_{\text{kva}}^{0.782} L_a^{0.346} \mathrm{N}_{\text{gen}}^{0.10} \tag{15.39}$$

$$W_{\text{avionics}} = 1.73\,W_{\text{uav}}^{0.983} \tag{15.40}$$

$$W_{\text{furnishings}} = 0.0577\,N_c^{0.1} W_c^{0.393} S_f^{0.75} \tag{15.41}$$

$$W_{\substack{\text{air}\\ \text{conditioning}}} = 62.36\,N_p^{0.25} (V_{pr}/1000)^{0.604} W_{\text{uav}}^{0.10} \tag{15.42}$$

$$W_{\text{anti-ice}} = 0.002\,W_{\text{dg}} \tag{15.43}$$

$$W_{\substack{\text{handling}\\ \text{gear}}} = 3.0 \times 10^{-4}\,W_{\text{dg}} \tag{15.44}$$

$$W_{\substack{\text{military cargo}\\ \text{handling system}}} = 2.4 \times (\text{cargo floor area, ft}^2) \tag{15.45}$$

Weights

General-Aviation Weights

$$W_{\text{wing}} = 0.036 S_w^{0.758} W_{\text{fw}}^{0.0035} \left(\frac{A}{\cos^2 \Lambda} \right)^{0.6} q^{0.006} \lambda^{0.04} \left(\frac{100\ t/c}{\cos \Lambda} \right)^{-0.3} (N_z W_{\text{dg}})^{0.49} \tag{15.46}$$

$$W_{\substack{\text{horizontal}\\ \text{tail}}} = 0.016 (N_z W_{\text{dg}})^{0.414} q^{0.168} S_{\text{ht}}^{0.896} \left(\frac{100\ t/c}{\cos \Lambda} \right)^{-0.12} \times \left(\frac{A}{\cos^2 \Lambda_{\text{ht}}} \right)^{0.043} \lambda_h^{-0.02} \tag{15.47}$$

$$W_{\substack{\text{vertical}\\ \text{tail}}} = 0.073 \left(1 + 0.2 \frac{H_t}{H_v} \right) (N_z W_{\text{dg}})^{0.376} q^{0.122} S_{\text{vt}}^{0.873} \left(\frac{100\ t/c}{\cos \Lambda_{\text{vt}}} \right)^{-0.49} \times \left(\frac{A}{\cos^2 \Lambda_{\text{vt}}} \right)^{0.357} \lambda_{\text{vt}}^{0.039} \tag{15.48}$$

$$W_{\text{fuselage}} = 0.052\ S_f^{1.086} (N_z W_{\text{dg}})^{0.177} L_t^{-0.051} (L/D)^{-0.072} q^{0.241} + W_{\text{press}} \tag{15.49}$$

$$W_{\substack{\text{main landing}\\ \text{gear}}} = 0.095 (N_l W_l)^{0.768} (L_m/12)^{0.409} \tag{15.50}$$

Weights

$$W_{\substack{\text{nose landing}\\ \text{gear}}} = 0.125\,(N_l W_l)^{0.566}(L_n/12)^{0.845} \tag{15.51}$$

$$W_{\substack{\text{installed engine}\\ \text{(total)}}} = 2.575\,W_{\text{en}}^{0.922}N_{\text{en}} \tag{15.52}$$

$$W_{\text{fuel system}} = 2.49\,V_t^{0.726}\left(\frac{1}{1+V_i/V_t}\right)^{0.363}N_t^{0.242}N_{en}^{0.157} \tag{15.53}$$

$$W_{\substack{\text{flight}\\ \text{controls}}} = 0.053\,L^{1.536}B_w^{0.371}(N_z W_{\text{dg}}\times 10^{-4})^{0.80} \tag{15.54}$$

$$W_{\text{hydraulics}} = 0.001\,W_{\text{dg}} \tag{15.55}$$

$$W_{\text{electrical}} = 12.57\,(W_{\text{fuel system}} + W_{\text{avionics}})^{0.51} \tag{15.56}$$

$$W_{\text{avionics}} = 2.117\,W_{\text{uav}}^{0.933} \tag{15.57}$$

$$W_{\substack{\text{air conditioning}\\ \text{and anti-ice}}} = 0.265\,W_{\text{dg}}^{0.52}N_p^{0.68}W_{\text{avionics}}^{0.17}M^{0.08} \tag{15.58}$$

$$W_{\text{furnishings}} = 0.0582\,W_{\text{dg}} - 65 \tag{15.59}$$

Weights

Weights Equations Terminology

A	= aspect ratio
B_h	= horizontal tail span, ft
B_w	= wing span, ft
D	= fuselage structural depth, ft
D_e	= engine diameter, ft
F_w	= fuselage width at horizontal tail intersection, ft
H_t	= horizontal tail height above fuselage, ft
H_t/H_v	= 0.0 for conventional tail; 1.0 for "T" tail
H_v	= vertical tail height above fuselage, ft
I_y	= yawing moment of inertia, lb-ft^2 (see Chap. 16)
K_{cb}	= 2.25 for cross-beam (F-111) gear; = 1.0 otherwise
K_d	= duct constant (see Fig. 15.2)
K_{door}	= 1.0 if no cargo door; = 1.06 if one side cargo door; = 1.12 if two side cargo doors; = 1.12 if aft clamshell door; = 1.25 if two side cargo doors and aft clamshell door
K_{dw}	= 0.768 for delta wing; = 1.0 otherwise
K_{dwf}	= 0.774 for delta wing aircraft; = 1.0 otherwise
K_{Lg}	= 1.12 if fuselage-mounted main landing gear; = 1.0 otherwise
K_{mc}	= 1.45 if mission completion required after failure; = 1.0 otherwise

Weights

K_{mp} = 1.126 for kneeling gear; = 1.0 otherwise
K_{ng} = 1.017 for pylon-mounted nacelle; = 1.0 otherwise
K_{np} = 1.15 for kneeling gear; = 1.0 otherwise
K_p = 1.4 for engine with propeller or 1.0 otherwise
K_r = 1.133 if reciprocating engine; = 1.0 otherwise
K_{rht} = 1.047 for rolling tail; = 1.0 otherwise
K_{tp} = 0.793 if turboprop; = 1.0 otherwise
K_{tpg} = 0.826 for tripod (A-7) gear; = 1.0 otherwise
K_{tr} = 1.18 for jet with thrust reverser or 1.0 otherwise
K_{uht} = 1.143 for unit (all-moving) horizontal tail; = 1.0 otherwise
K_{vg} = 1.62 for variable geometry; = 1.0 otherwise
K_{vs} = 1.19 for variable sweep wing; = 1.0 otherwise
K_{vsh} = 1.425 if variable sweep wing; = 1.0 otherwise
K_{ws} = $0.75[1 + 2\lambda)/(1 + \lambda)]\ (B_w \tan\Lambda/L)$
K_y = aircraft pitching radius of gyration, ft ($\cong 0.3L_t$)
K_z = aircraft yawing radius of gyration, ft ($\cong L_t$)
L = fuselage structural length, ft (excludes radome, tail cap)
L_a = electrical routing distance, generators to avionics to cockpit, ft
L_d = duct length, ft
L_{ec} = length from engine front to cockpit—total if multiengine, ft
L_f = total fuselage length

Weights

L_m = length of main landing gear, in.
L_n = nose gear length, in.
L_s = single duct length (see Fig. 15.2)
L_{sh} = length of engine shroud, ft
L_t = tail length; wing quarter-MAC to tail quarter-MAC, ft
L_{tp} = length of tailpipe, ft
M = Mach number
N_c = number of crew
N_{ci} = 1.0 if single pilot; = 1.2 if pilot plus backseater; = 2.0 pilot and copassenger
N_{en} = number of engines
N_f = number of functions performed by controls (typically 4–7)
N_{gen} = number of generators (typically = N_{en})
N_l = ultimate landing load factor; = $N_{gear} \times 1.5$
N_{Lt} = nacelle length, ft
N_m = number of mechanical functions (typically 0–2)
N_{mss} = number of main gear shock struts
N_{mw} = number of main wheels
N_{nw} = number of nose wheels

Weights

N_p = number of personnel onboard (crew and passengers)
N_s = number of flight control systems
N_t = number of fuel tanks
N_u = number of hydraulic utility functions (typically 5–15)
N_w = nacelle width, ft
N_z = ultimate load factor; = 1.5 × limit load factor
q = dynamic pressure at cruise, lb/ft^2
R_{kva} = system electrical rating, kv · A (typically 40–60 for transports, 110–160 for fighters & bombers)
S_{cs} = total area of control surfaces, ft^2
S_{csw} = control surface area (wing-mounted), ft^2
S_e = elevator area, ft
S_f = fuselage wetted area, ft^2
S_{fw} = firewall surface area, ft^2
S_{ht} = horizontal tail area
S_n = nacelle wetted area, ft^2
S_r = rudder area, ft^2
S_{vt} = vertical tail area, ft^2
S_w = trapezoidal wing area, ft^2

Weights

SFC	= engine specific fuel consumption—maximum thrust
T	= total engine thrust, lb
T_e	= thrust per engine, lb
V_i	= integral tanks volume, gal
V_p	= self-sealing "protected" tanks volume, gal
V_{pr}	= volume of pressurized section, ft^3
V_t	= total fuel volume, gal
W	= fuselage structural width, ft
W_c	= maximum cargo weight, lb
W_{dg}	= design gross weight, lb
W_{ec}	= weight of engine and contents, lb (per nacelle), $\cong 2.331\, W_{\text{engine}}^{0.901} K_p K_{tr}$
W_{en}	= engine weight, each, lb
W_{fw}	= weight of fuel in wing, lb
W_l	= landing design gross weight, lb
W_{press}	= weight penalty due to pressurization, $= 11.9 + (V_{pr} P_{\text{delta}})^{0.271}$, where P_{delta} = cabin pressure differential, psi (typically 8 psi)
W_{uav}	= uninstalled avionics weight, lb (typically = 800–1400 lb)
Λ	= wing sweep at 25% MAC

Weights

Additional Considerations In Weights Estimation

- These statistical equations are based upon a database of existing aircraft (2000s…).
- They work well for a "normal" aircraft similar to the various aircraft in the database.
- However, use of a novel configuration (canard pusher) or an advanced technology (composite structure) will result in a poor weights estimate when using these or similar equations.
- To allow for this, weights engineers adjust the statistical-equation results using "fudge/adapt factors" (defined as the variable constant that you multiply your answer by to get the right answer!)

Weights

- Fudge factors for composite-structure, wood or steel-tube fuselages, braced wings, and flying-boat hulls are provided in Table 15.4.

- These should be viewed as rough approximations.

Table 15.4 Weights estimation "fudge factors"

Category	Weight group	Fudge factor (multiplier)
Advanced composites	Wing	0.85
	Tails	0.83
	Fuselage/nacelle	0.90
	Landing gear	0.95
	Air induction system	0.85
Braced wing	Wing	0.82
Wood fuselage	Fuselage	1.60
Steel tube fuselage	Fuselage	1.80
Flying boat hull	Fuselage	1.25

Weights

- Figure 15.3 shows the empty-weight growth of a number of aircraft.
- In the past, a weight growth of 5% in the first year was common.
- Today's better design techniques and analytical methods have reduced that to less than 2% in the first year.

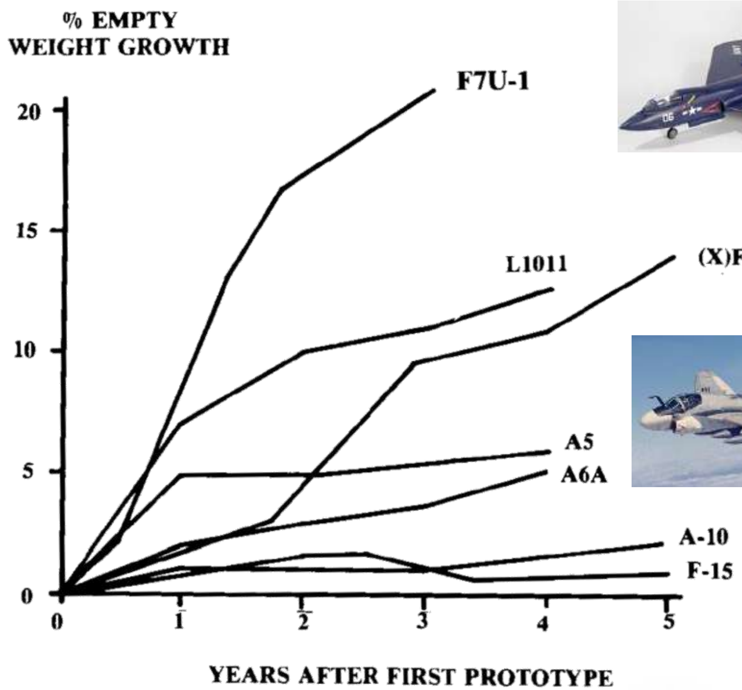


Fig. 15.3 Aircraft weight growth.

Aircraft Design
AE 405

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Contents

Chapter 16-17

- Stability, Control, and Handling Qualities.
- Performance and Flight Mechanics.

Stability, Control, and Handling Qualities

- During early conceptual design, the requirements for good stability, control, and handling qualities are addressed

  - through the use of tail volume coefficients and
  - through location of the aircraft center of gravity (e.g.) at some percent of the wing mean aerodynamic chord (MAC).

- In larger aircraft companies, the aircraft is then analyzed by the controls experts,

- probably using a six-degree-of-freedom (6-DOF) aircraft dynamics computer program to determine the required c.g. location and the sizes of the tails and control surfaces.

Stability, Control, and Handling Qualities

- The basic concept of stability is simply that a stable aircraft, when disturbed, tends to return by itself to its original state.

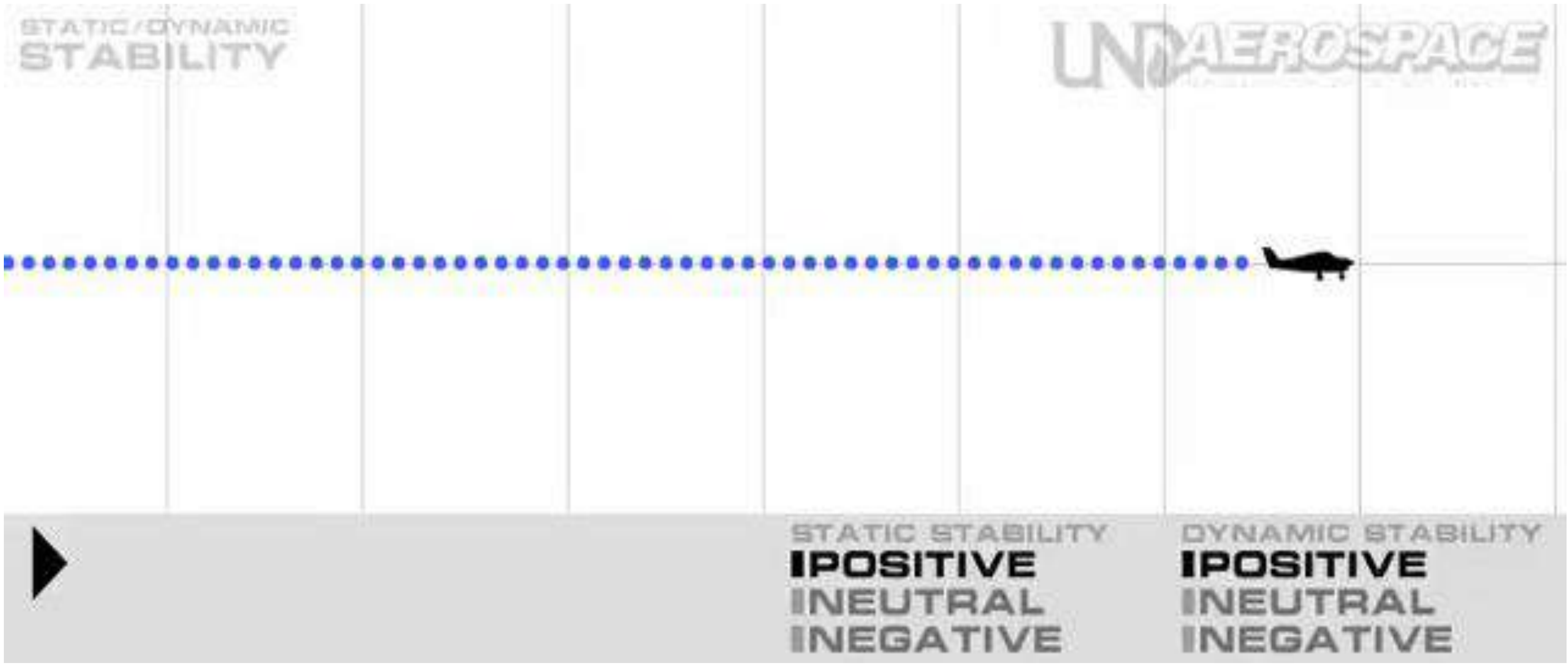
- "Static stability" is present if the forces created by the disturbed
- state push in the correct direction to return the aircraft to its original state.

- If these restoring forces are too strong the aircraft will overshoot the original state and will oscillate with greater and greater amplitude until it goes completely out of control.

- Although static stability is present, the aircraft does not have "dynamic stability."

- Dynamic stability is present if the dynamic motions of the aircraft will eventually return the aircraft to its original state.

Stability, Control, and Handling Qualities



- Figure 16.1 illustrates these concepts for an aircraft disturbed in pitch.

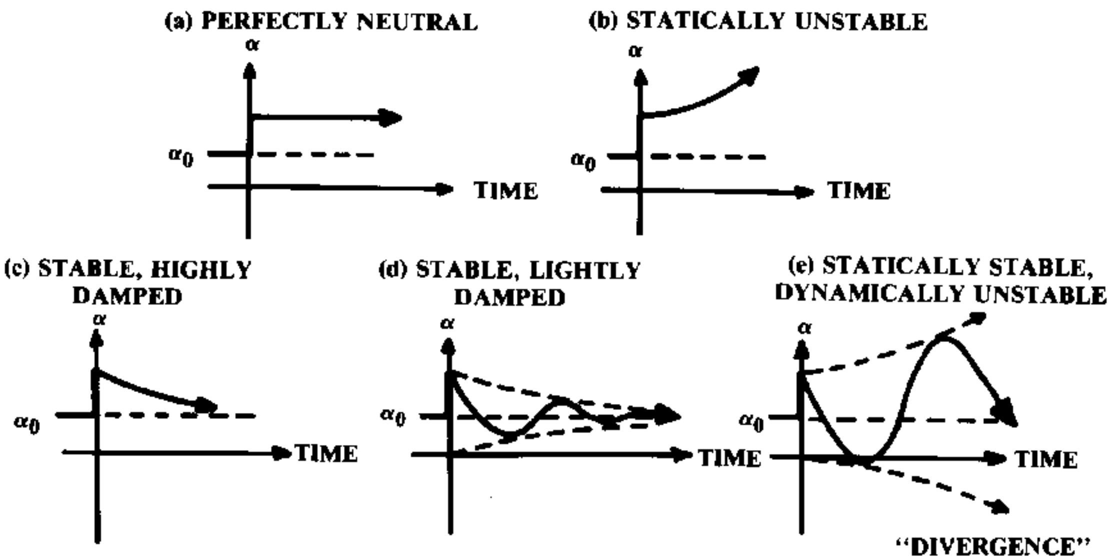
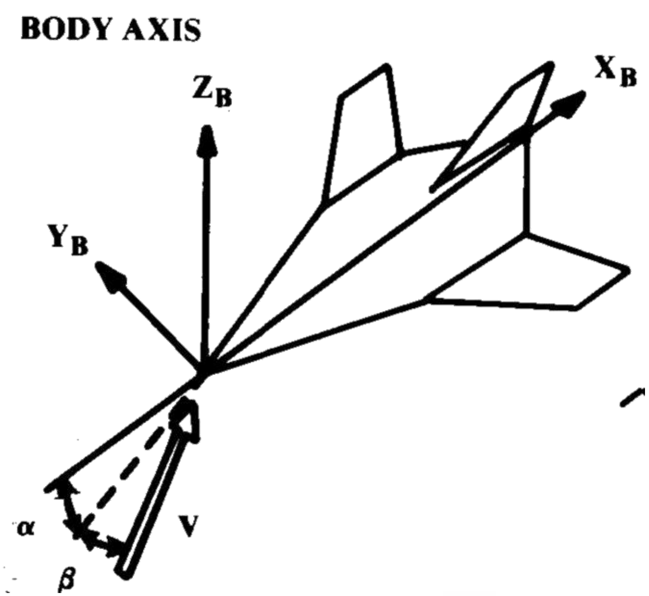


Fig. 16.1 Static and dynamic stability.

- For conventional aircraft configurations, satisfaction of static-stability requirements will probably give acceptable dynamic stability in most flight modes.

Coordinate Systems and Definitions

- Figure 16.2 defines the two axis systems commonly used in aircraft analysis.
- The "body-axis system" is rigidly fixed to the aircraft, with the *X* axis aligned with the fuselage and the *Z* axis upward.
- The origin is at an arbitrary location, usually the nose or c.g.



aircraft angle of attack α
sideslip, β

Stability, Control, and Handling Qualities

- The "wind axis" system orients the X-axis into the relative wind regardless of the aircraft's angle of attack or sideslip.

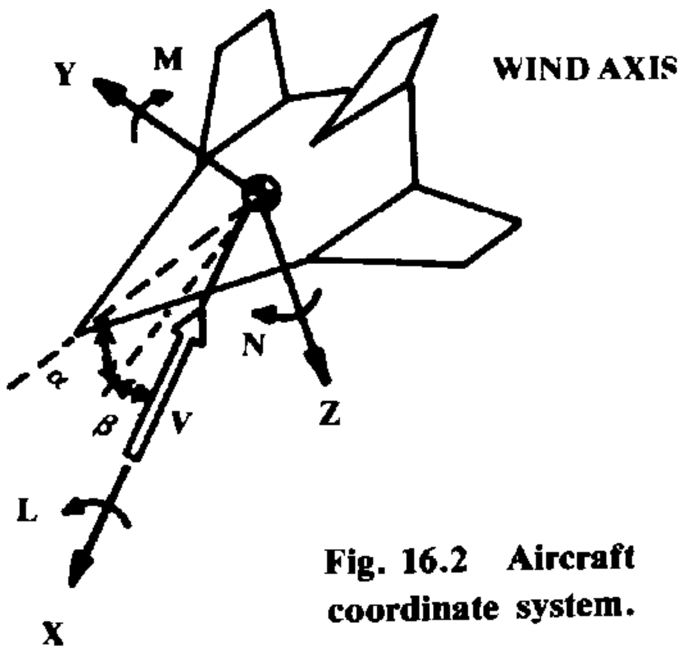


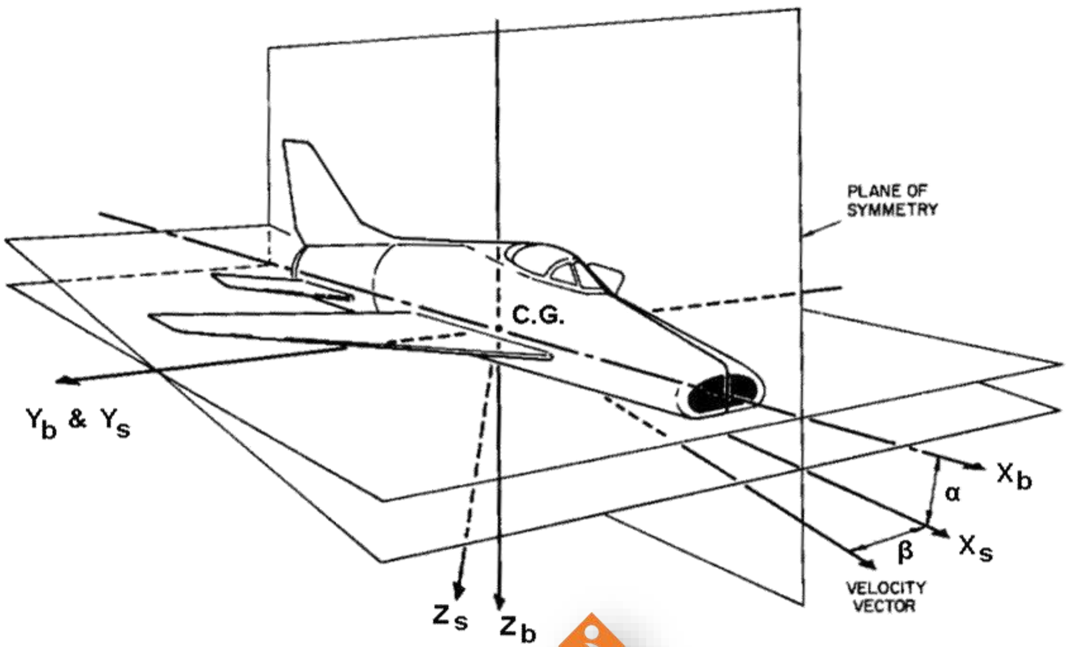
Fig. 16.2 Aircraft coordinate system.

Stability, Control, and Handling Qualities

- The "stability" axis system, commonly used in stability and control analysis, is a compromise between these two.

- The X-axis is aligned at the aircraft angle of attack, as in the wind axis system, but is not offset to the yaw angle.

- Directions of X, Y, and Z are as in the wind axis system.



Stability, Control, and Handling Qualities

- Note that the rolling moment is called L.
- This is easily confused with lift.
- Also, the yawing moment is called N, which is the same letter used for the normal-force coefficient.
- Wing and tail incidence angles are denoted by i, which is relative to the body-fixed reference axis.
- The aircraft angle of attack α is also with respect to this reference axis,
- so the wing angle of attack is the aircraft angle of attack plus the wing angle of incidence.

Stability, Control, and Handling Qualities

- For stability calculations, the moments about the three axes (M, N, and L) must be expressed as nondimensional coefficients.

$$c_m = M/qS\bar{c} \tag{16.1}$$

$$c_n = N/qSb \tag{16.2}$$

$$c_\ell = L/qSb \tag{16.3}$$

- Stability analysis is largely concerned with the response to changes in angular orientation,
- so the derivatives of these coefficients with respect to angle of attack and sideslip are critical.
- Subscripts are used to indicate the derivative.

Stability, Control, and Handling Qualities

- For example, $C_{n\beta}$ is the yawing moment derivative with respect to sideslip, a very important parameter in lateral stability.

- Similarly, subscripts are used to indicate the response to control deflections, indicated by δ.

- Thus, $c_{m\delta_e}$, indicates the pitching-moment response to an elevator deflection.

Stability, Control, and Handling Qualities

Longitudinal Static Stability and Control

- Most aircraft being symmetrical about the centerline, moderate changes in angle of attack will have little or no influence upon the yaw or roll.

- This permits the stability and control analysis to be divided into longitudinal (pitch only) and lateral-directional (roll and yaw) analysis.

- Figure 16.3 shows the major contributors to aircraft pitching moment about the c.g., including the wing, tail, fuselage, and engine contributions.

- The wing pitching-moment contribution includes the lift through the wing aerodynamic center and the wing moment about the aerodynamic center.

- Remember that the aerodynamic center is defined as the point about which pitching moment is constant with respect to angle of attack.

Stability, Control, and Handling Qualities

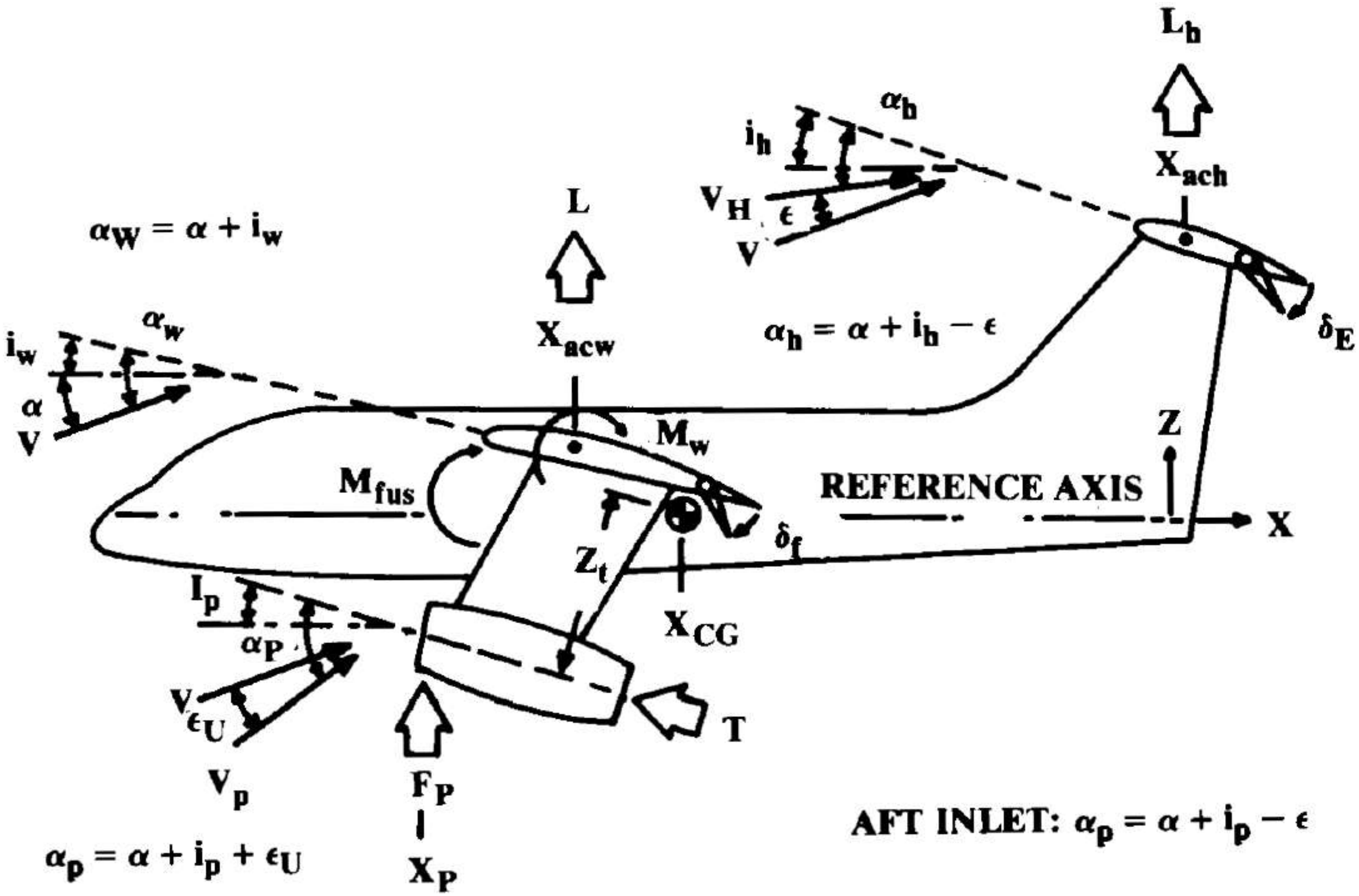


Fig. 16.3 Longitudinal moments.

the vertical force F_p produced at the propeller disk or inlet front face

- Equation (16.4) expresses the sum of moments about the c.g.
- Equation (16.5) expresses the moments in coefficient form by dividing all terms by ($qS_w c$) and expressing the tail lift in coefficient form.

$$M_{cg} = L(X_{cg} - X_{acw}) + M_w + M_{w\delta_f}\delta_f + M_{fus} - L_h(X_{ach} - X_{cg}) - Tz_t + F_p(X_{cg} - X_p) \quad (16.4)$$

$$C_{m_{cg}} = C_L\left(\frac{X_{cg} - X_{acw}}{c}\right) + C_{m_w} + C_{m_{w\delta_f}}\delta_f + C_{m_{fus}} - \frac{q_h S_h}{q S_w} C_{L_h}\left(\frac{X_{ach} - X_{cg}}{c}\right) - \frac{Tz_t}{qS_w c} + \frac{F_p(X_{cg} - X_p)}{qS_w c} \quad (16.5)$$

Stability, Control, and Handling Qualities

- For a static "trim" condition, the total pitching moment must equal zero.
- For static trim, the main flight conditions of concern are during the takeoff and landing with flaps and landing gear down and during flight at high transonic speeds.
- Usually the most forward c.g. position is critical for trim.
- Aft-c.g. position is most critical for stability.
- Equation (16.5) can be set to zero and solved for trim by varying some parameter,
- typically tail area, tail lift coefficient (i.e., tail incidence or elevator deflection), or sometimes c.g. position.

Stability, Control, and Handling Qualities

- For static stability to be present, any change in angle of attack must generate moments which oppose the change.

- In other words, the derivative of pitching moment with respect to angle of attack [Eq. (16.8)] must be negative.

$$C_{m_\alpha} = C_{L_\alpha}(\overline{X}_{cg} - \overline{X}_{acw}) + C_{m_{\alpha\text{fus}}} - \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} (\overline{X}_{ach} - \overline{X}_{cg})$$

$$+ \frac{F_{p_\alpha}}{q S_w} \frac{\partial \alpha_p}{\partial \alpha} (\overline{X}_{cg} - \overline{X}_p) \qquad (16.8)$$

- The magnitude of the pitching-moment derivative [Eq. (16.8)] changes with c.g. location.

- For any aircraft there is a c.g. location that provides no change in pitching moment as angle of attack is varied.

- This is an aircraft aerodynamic center," or neutral point X_{np} represents neutral stability (Fig. 16.1a) and is the most-aft c.g. location before the aircraft becomes unstable.

- Equation (16.9) solves Eq. (16.8) for the neutral point ($c_{m\alpha} = 0$).

- Equation (16.10) expresses the pitching moment derivative in terms of the distance in percent MAC from the neutral point to the c.g.

- This percents distance, called the "static margin," is the term in parenthesis in Eq. (16.10).

$$\overline{X}_{np} = \frac{C_{L_\alpha}\overline{X}_{acw} - C_{m_{\alpha_{\text{fus}}}} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} \overline{X}_{ach} + \frac{F_{p_\alpha}}{qS_w} \frac{\partial \alpha_p}{\partial \alpha} \overline{X}_p}{C_{L_\alpha} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} + \frac{F_{p_\alpha}}{qS_w}} \tag{16.9}$$

$$C_{m_\alpha} = -C_{L_\alpha}(\overline{X}_{np} - \overline{X}_{cg}) \tag{16.10}$$

Stability, Control, and Handling Qualities

- If the c.g. is ahead of the neutral point (positive static margin), the pitching-moment derivative is negative so the aircraft is stable.

- At the most-aft c.g. position, a typical transport aircraft has a positive static margin of 5-10%.

- Current fighters typically have positive static margins of about 5%.

- New fighters such as the F-16 have (zero to -15%) is coupled with a computerized flight control system that deflects the elevator to provide artificial stability.

- Trim requires that the total moment about the e.g. [Eq. (16.5)] equals zero.

Typical Static Margin Values

Transports & Consumer AC: 0.05 to 0.20	Cessna 172	0.19	
	Learjet 35	0.13	More Stable
	Boeing 747	0.27	
Fighters: 0 to 0.05	P-51 Mustang	0.05	More Maneuverable
	F-106	0.09	
Fighters - FBW	F-16A (early)	-0.02	Much More Maneuverable
	F-16C	0.01	
	X-29	-0.33	Stabilized by AFCS

Stability, Control, and Handling Qualities

Lateral-directional Static Stability and Control

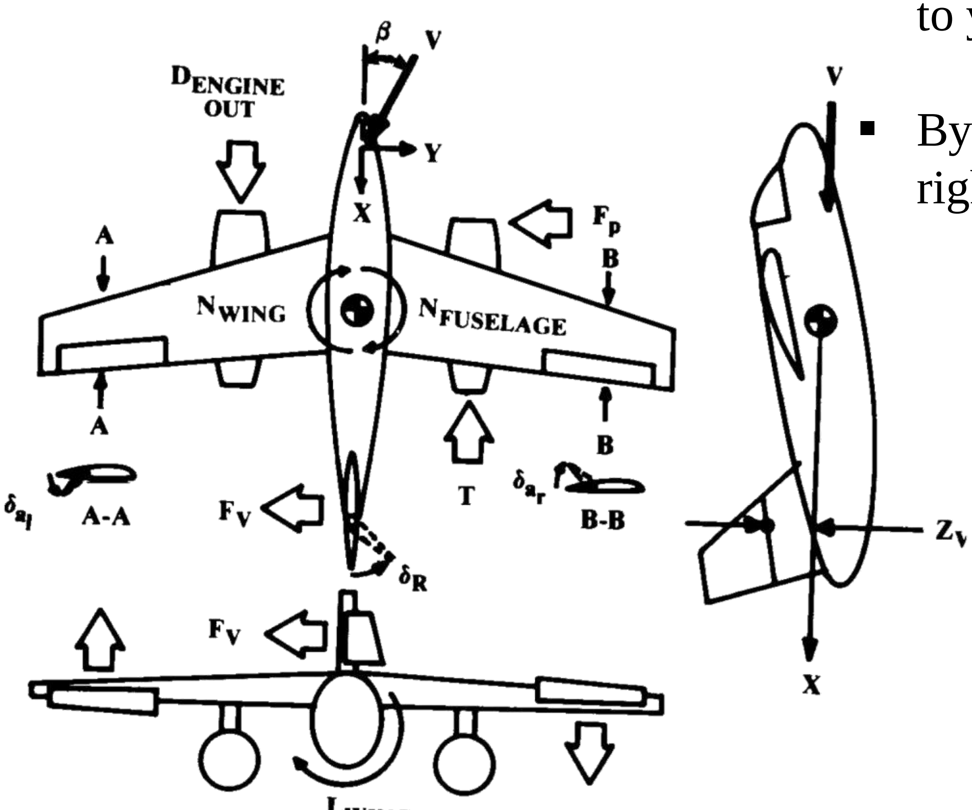


Fig. 16.19 Lateral geometry.

- The geometry for lateral analysis is illustrated in Fig. 16.19, showing the major contributors to yawing moment *N* and rolling moment *L*.
- By definition, yaw and roll are positive to the right.
  - The major yawing moment is due to the lateral lift of the vertical tail, denoted by F_v.
  - This counteracts the fuselage yawing moment.
  - Rudder deflection acts as a flap to increase the lateral lift of the vertical tail.

Stability, Control, and Handling Qualities

- In many ways the lateral-directional analysis resembles the longitudinal analysis.

- However, the lateral-directional analysis really embraces two closely-coupled analyses: the yaw (directional) and the roll (lateral).

- We will not deal with lateral-directional analysis…

- For further analysis, refer to AE 412.

Handling Qualities

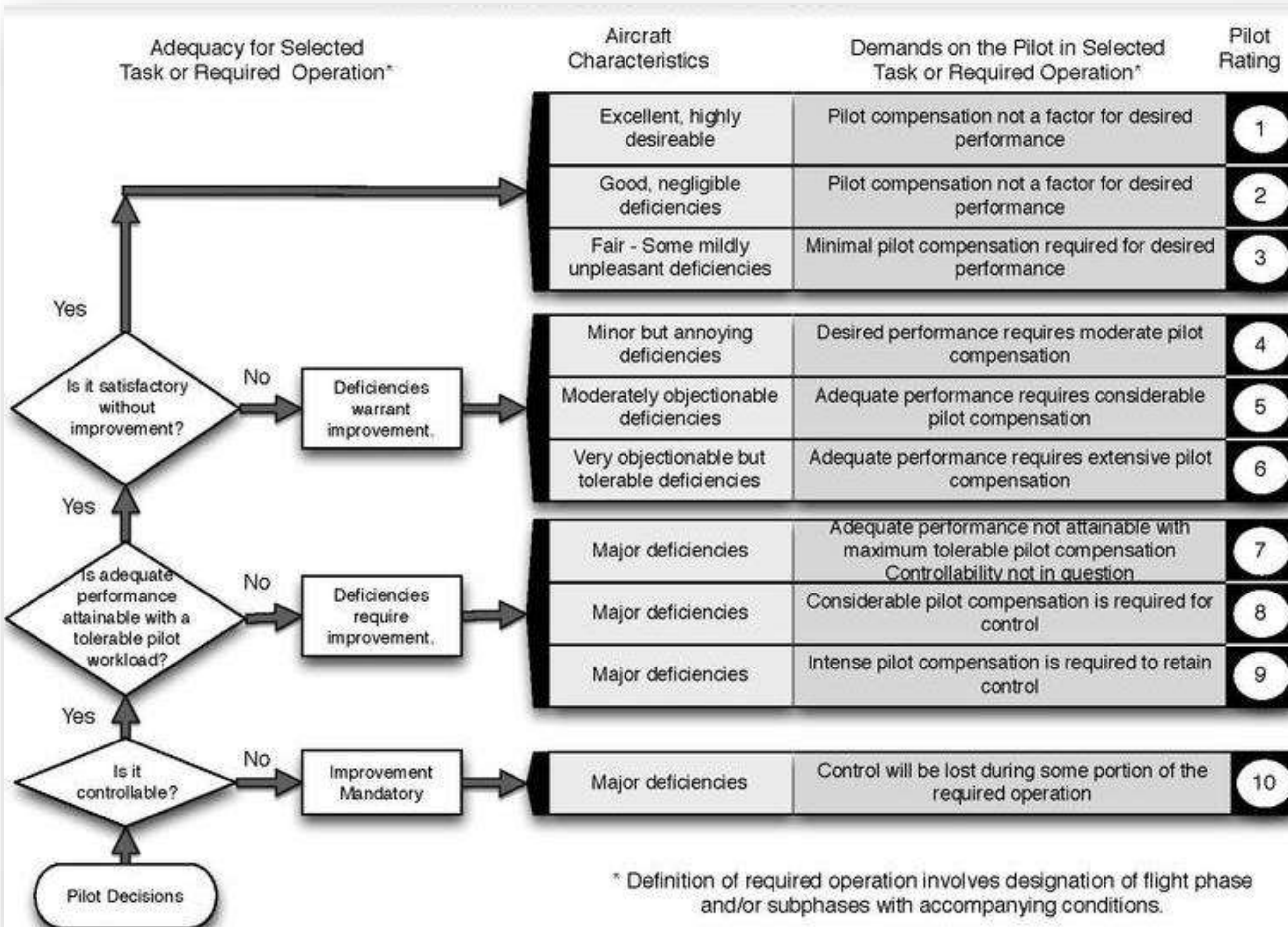
- Aircraft handling qualities are a subjective assessment of the way the plane feels to the pilot.

- These handling qualities criteria are generally considered later in the design cycle.

- Figure 16.27 illustrates the Cooper-Harper Handling Qualities Rating Scale, which is used by test pilots to categorize design deficiencies.

Stability, Control, and Handling Qualities

Cooper-Harper Handling Qualities Rating Scale



Performance and Flight Mechanics

- The geometry for flight mechanics is shown in Fig. 17.1.

- Summing forces in the *Xs* and *Zs* directions yields Eqs. (17 .1) and (17 .2).

- The resulting accelerations on the aircraft in the *Xs* and *Zs* directions are determined as these force summations divided by the aircraft mass (*W/g*).

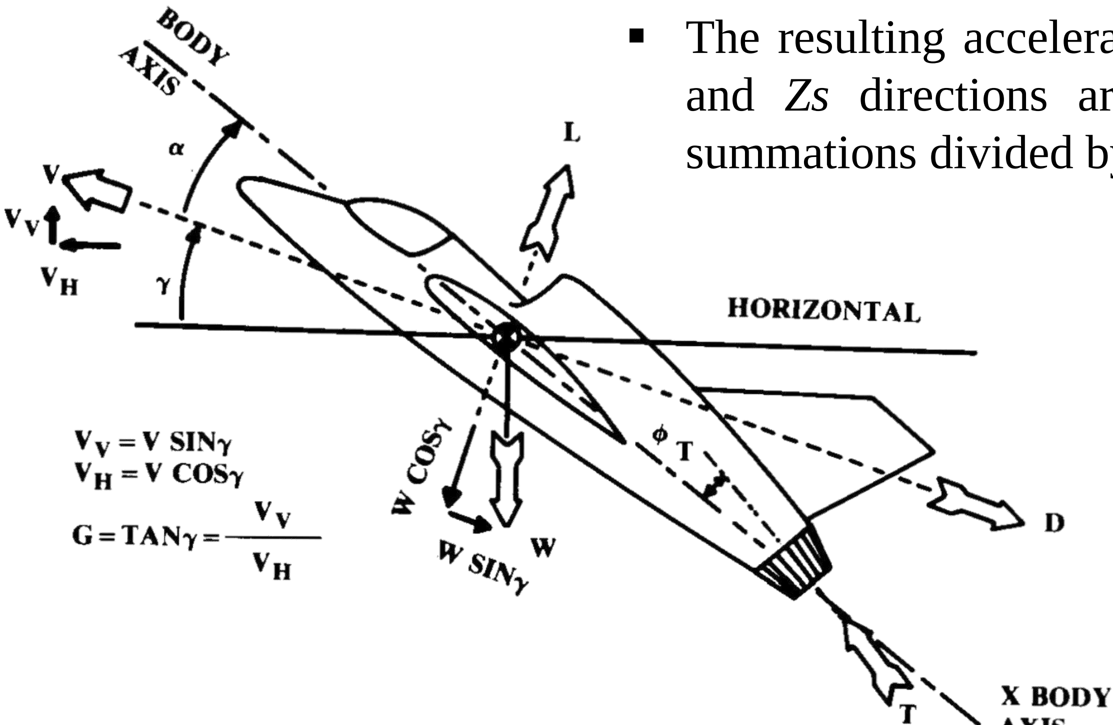


Fig. 17.1 Geometry for performance calculation.

$$\Sigma F_x = T\cos(\alpha + \phi_T) - D - W\sin\gamma$$

$$\Sigma F_z = T\sin(\alpha + \phi_T) + L - W\cos\gamma$$

$$\dot{W} = -CT$$

$$C = C_{bhp}\,\frac{V}{550\,\eta_p}$$

$$T = 550\ \text{bhp}\ \eta_p / V$$

Performance and Flight Mechanics

- These simple equations are the basis of the most detailed sizing and performance programs used by the major airframe companies.

- For further analysis, refer to AE 304.

Aircraft Design
AE 405

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Contents

Chapter 18

Cost Analysis

Cost Analysis

- Aircraft cost estimation occupies the fuzzy gray area between science, art, and politics.
- Cost estimation is largely statistical, and in the final analysis we predict the cost of a new aircraft based on the actual costs of prior aircraft.
- Figure 18.2 shows the elements which make up aircraft life cycle cost (LCC).
- The sizes of the boxes are roughly proportional to the magnitude of the costs for a typical aircraft.

Cost Analysis

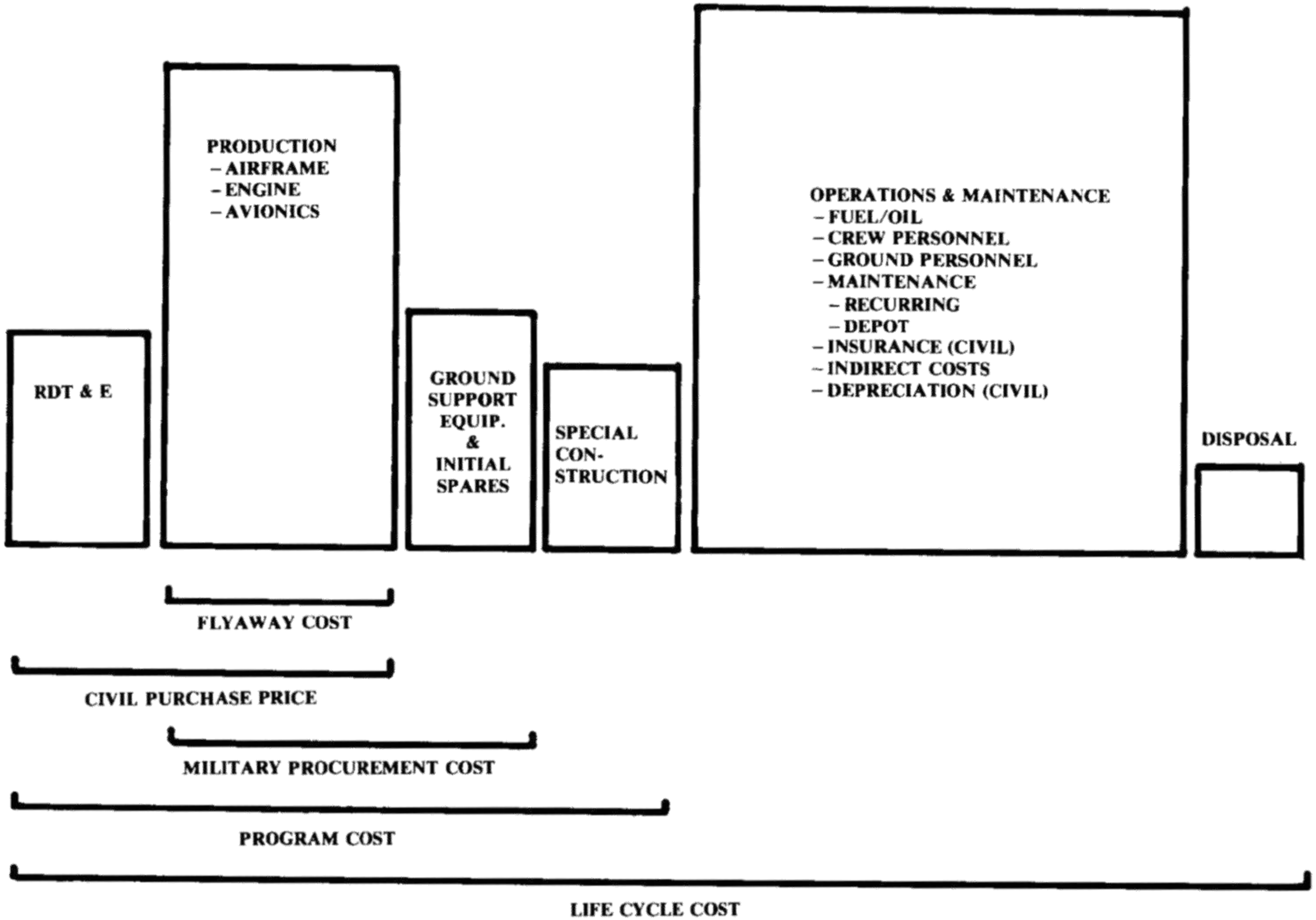

Fig. 18.2 Elements of life cycle cost.

Cost Analysis

- "RDT&E" stands for research, development, test, and evaluation,
- which includes all the technology research, design engineering, prototype fabrication, flight and ground testing, and evaluations for operational suitability.
- The cost of aircraft conceptual design as discussed in this course is included in the RDT &E cost.
- RDT&E includes certification cost for civil aircraft.
- For military aircraft, RDT &E includes the costs associated with the demonstration of airworthiness, mission capability, and compliance with Mil-Specs.
- RDT&E costs are essentially fixed ("nonrecurring") regardless of how many aircraft are ultimately produced.

Cost Analysis

- The aircraft "flyaway" (production) cost covers the labor and material costs to manufacture the aircraft, including airframe, engines, and avionics.
- This cost includes production tooling costs.
- Note that "cost" includes the manufacturer's overhead and administrative expenses.
- Production costs are "recurring" in that they are based upon the number of aircraft produced.
- The cost per aircraft is reduced as more aircraft are produced due to the learning curve effect.

Cost Analysis

- The purchase price for a civil aircraft is set to recover the RDT &E and production costs, including a fair profit.
- For military aircraft, the RDT &E costs are paid directly by the government during the RDT &E phase, so these costs need not be recovered during production.
- Military-aircraft "procurement cost" (or "acquisition cost") includes
  - the production costs
  - as well as the costs of required ground support equipment, such as flight simulators and test equipment,
  - and the cost of the initial spare parts during operational deployment.
- For civil aircraft, these are normally purchased separately.

Cost Analysis

- "Program cost" covers the total cost to develop and deploy a new aircraft into the military inventory.

- Some aircraft require special ground facilities for operational deployment.

- For example, a fighter/attack aircraft with a large wing span may not fit into the existing bombproof shelters in Europe.

- The cost of constructing new shelters would be included in the total program cost along with the RDT&E and procurement costs.

Cost Analysis

- "Operations and Maintenance" (O&M) costs are usually much larger than development and production costs.

- O&M covers fuel, oil, aircrew, maintenance, and various indirect costs.

- For civil aircraft, insurance will be part of operations cost.

- The final element making up the total life-cycle cost concerns "disposal."

- Obsolete military aircraft are flown one last time to Arizona for "pickling" and storage.

- The expense of this is not large, so it is frequently ignored in LCC estimation.

Cost Analysis

- Civil aircraft have a negative disposal cost because they are worth something on the resale market (typically 10% of purchase price).

Cost-estimating Methods

- Cost estimation during conceptual design is largely statistical.

- Cost data for a number of aircraft are analyzed using curve-fit programs to prepare cost estimating relations (CER) for the various cost elements.

- The output of a CER is either cost or labor hours (engineering, production, etc.), which are converted to cost by multiplying by the appropriate hourly rate.

Cost Analysis

- A set of CERs for conceptual aircraft design developed by the RAND Corporation is known as DAPCA (the Development and Procurement Costs of Aircraft (DAPCA) model).

- DAPCA estimates the hours required for RDT&E and production by the engineering, tooling, manufacturing, and quality control groups.

- These are multiplied by the appropriate hourly rates to yield costs.

- Modified DAPCA IV Cost Model (costs in constant 1986 dollars):

$$\text{Eng hours} = 4.86 W_e^{0.777} V^{0.894} Q^{0.163} = H_E \tag{18.1}$$

$$\text{Tooling hours} = 5.99 W_e^{0.777} V^{0.696} Q^{0.263} = H_T \tag{18.2}$$

$$\text{Mfg hours} = 7.37 W_e^{0.82} V^{0.484} Q^{0.641} = H_M \tag{18.3}$$

Cost Analysis

Quality Control

$$\left.\begin{aligned}\text{QC hours} &= 0.076\ (\text{mfg hours}) \quad \text{if cargo airplane}\\ &= 0.133\ (\text{mfg hours}) \quad \text{otherwise}\end{aligned}\right\} = H_Q \qquad (18.4)$$

$$\text{Devel support cost} = 45.42 W_e^{0.630} V^{1.3} = C_D \qquad (18.5)$$

$$\text{Flt test cost} = 1243.03 W_e^{0.325} V^{0.822} FTA^{1.21} = C_F \qquad (18.6)$$

$$\text{Mfg materials cost} = 11.0 W_e^{0.921} V^{0.621} Q^{0.799} = C_M \qquad (18.7)$$

$$\begin{aligned}\text{Eng production cost} &= 1548[0.043 T_{\max} + 243.25 M_{\max}\\ &\quad + 0.969 T_{\text{turbine inlet}} - 2228] = C_{\text{eng}}\end{aligned} \qquad (18.8)$$

Cost Analysis

$$\text{RDT\&E} + \text{flyaway} = H_E R_E + H_T R_T + H_M R_M + H_Q R_Q + C_D + C_F + C_M + C_{\text{eng}} N_{\text{eng}} + C_{\text{avionics}} \tag{18.9}$$

where

We	= empty weight (lb)
V	= maximum velocity (knots)
Q	= production quantity
FTA	= number of flight test aircraft (typically 2-6)
N_{eng}	= total production quantity times number of engines per aircraft
$T_{\max}$	= engine maximum thrust (lb)
$M_{\max}$	= engine maximum Mach number
$T_{\text{turbine inlet}}$	= turbine inlet temperature (Rankine)
C_{avionics}	= avionics cost

Cost Analysis

- The hours estimated by DAPCA are based upon the design and fabrication of an aluminum aircraft.

- For aircraft which are largely fabricated from other materials the hours must be adjusted to account for the more – difficult design and fabrication.

- Based upon minimal information, the following "fudge factors" are recommended:

aluminum	1.0
graphite-epoxy	1.5–2.0
fiberglass	1.1–1.2
steel	1.5–2.0
titanium	1.7–2.2

Cost Analysis

- The hours estimated with this model are multiplied by the appropriate hourly rates to calculate the labor costs.

- These hourly rates are called "Wrap Rates.

- Average 1986 Wrap rates are as follows:

engineering	$\$59.10 = R_E$
tooling	$\$60.70 = R_T$
quality control	$\$55.40 = R_Q$
manufacturing	$\$50.10 = R_M$

- Predicted costs are then rationed by some inflation factor to the selected year's constant dollar.

Cost Analysis

- DAPCA does not estimate avionics costs.
- They must be estimated from data on similar aircraft or from vendors' quotations.
- Avionics costs can be approximated as $2000 per pound in 1986 dollars.
- Predicted aircraft costs will be multiplied by an "investment cost factor" to determine the purchase price to the customer.
- The investment cost-factor includes the cost of money and the contractor profit; it is considered highly proprietary by a company.
- Investment cost-factor may be roughly estimated as 1.1-1.2.

Cost Analysis

Operations and Maintenance Costs

- The main O&M costs are fuel, crew salaries, and maintenance.
- For a typical military aircraft,
  - the fuel totals about 15% of the O&M costs,
  - the crew salaries about 35%,
  - and the maintenance most of the remaining 50%.
- Over one-third of U.S. Air Force manpower is dedicated to maintenance.
- For commercial aircraft,
  - the fuel totals about 38% of O&M costs,
  - the crew salaries about 24%,
  - and the maintenance about 25%.
- The depreciation of the aircraft purchase price is about 12% of total O&M costs, and the insurance is the remaining 1%.

Cost Analysis

- Table 18. I provides some rough guidelines for flight hours (FH) per year (YR) and other LCC (Life Cycle Cost) parameters.

Table 18.1 LCC parameter approximations

Aircraft class	FH/YR/AC	Crew ratio	MMH/FH
Light aircraft	500–1000	–	1/4–1
Business jet	500–2000	–	3–6
Jet trainer	300–500	–	6–10
Fighter (modern)	300–500	1.1	15–20
Bomber	300–500	1.5	25–50
Military transport	700–1400	1.5 if FH/YR < 1200 2.5 if 1200 < FH/YR < 2400 3.5 if 2400 < FH/YR	20–40
Civil transport	2500–4500	–	5–15

AC : Aircraft
MMH : Maintenance Man Hour

- For military aircraft, crew costs are determined by estimating how many flight-crew members will have to be kept on the active-duty to operate the aircraft.

- This is the number of aircraft times the number of crew members per aircraft, times the "crew ratio."

Aircraft Design
AE 405

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Contents

Chapter 20

VTOL Aircraft Design

VTOL Aircraft Design

- This chapter introduces the essential concepts and technologies of vertical takeoff and landing (VTOL) aircraft design.

- Although similar in many respects to conventional aircraft design, VTOL presents some key differences and pitfalls to avoid.

- The operational benefits of an ability to take off and land vertically are self-evident.

- Conventional aircraft must operate from a relatively small number of airports or airbases with long paved runways.

VTOL Aircraft Design

- The first type of VTOL heavier-than-air aircraft was the helicopter.

- The helicopter rapidly proved its worth for rescue operations and short range point-to-point transportation, but its inherent speed and range limitations restricted its application.

- For propeller-powered aircraft, the tilt-rotor concept as tested in the Bell XV-15 seems to offer the best compromise between helicopter-like vertical flight and efficient wing-borne cruise.

- The tilt-rotor concept is the basis of the V-22 Osprey.

- Helicopters and tilt-rotors go beyond the scope of this course.

Bell XV-15

VTOL Aircraft Design

- For jet VTOL aircraft, a clear "best" solution for vertical lift has yet to emerge.
- In the past, there have only been a few operational jet VTOL designs-the British Harrier and the Russian YAK-36.
- These are both subsonic aircraft.
- The state of art VTOL aircraft is F-35B, supersonic stealth aircraft.

Harrier

YAK-36

Mirage III V

VTOL Aircraft Design

- *Fundamental Problems of VTOL Design*

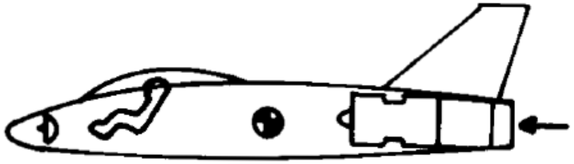
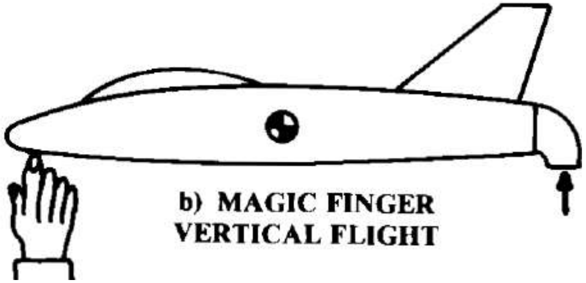


Fig. 20.1 The balance problem.

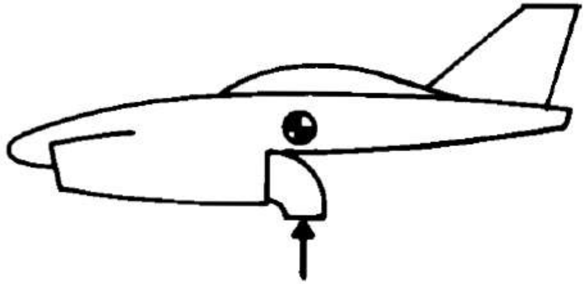
- Two fundamental problems stand out because they tend
- to have the greatest impact upon the selection of a VTOL propulsion concept
- and upon the design and sizing of the aircraft: balance and thrust matching.
- Modern supersonic jet fighters have a T/W exceeding 1.0, so it would seem fairly easy to point the jet exhaust downward and attain vertical flight.
- Unfortunately, this is complicated by the balance problem.
- Figure 20.la illustrates this traditional (and usually optimal) layout.

VTOL Aircraft Design

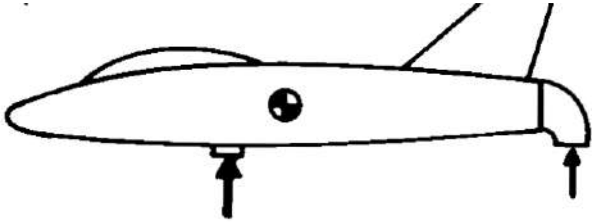
- If the aircraft's thrust exceeds its weight, vertical flight could be obtained simply by deflecting the thrust downward, as shown in Fig. 20.lb.
- However, a "magic finger" must hold up the nose in order to balance the vertical thrust force at the tail.
- This balance problem is possibly the single most important driver of the design of the VTOL jet fighter.
- There are really only two conceptual approaches to solving the balance problem.
- Either the thrust can somehow be moved to the c.g. or an additional thrust force can be located near the nose.
- Both of these approaches will tend to compromise the aircraft away from the traditional and usually optimal layout.



b) MAGIC FINGER VERTICAL FLIGHT



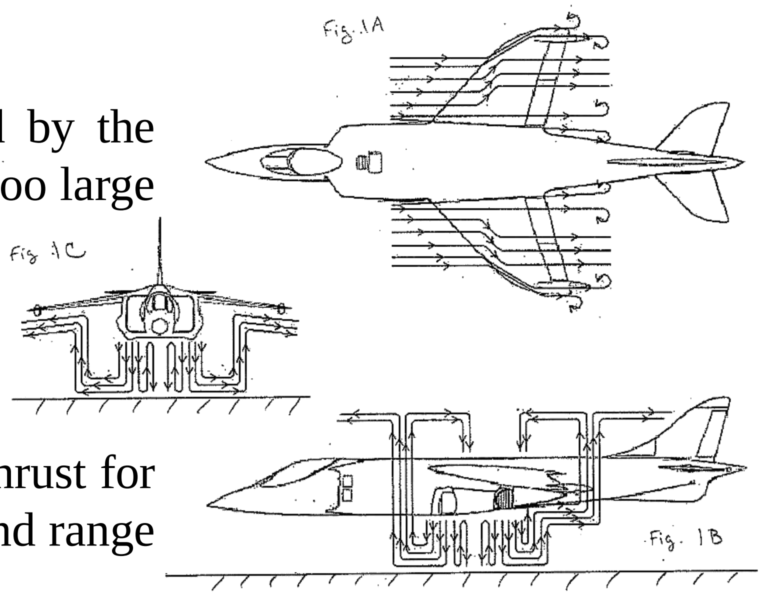
c) THRUST LOCATION MOVED



d) BALANCED THRUST

VTOL Aircraft Design

- For cruise-dominated VTOL aircraft such as transports, a more severe problem involves thrust matching.
- If the thrust required for vertical flight is provided by the same engines used for cruise, the engines will be far too large for efficient cruise.
- Aircraft range is directly proportional to SFC.
- The mismatch between thrust for vertical flight and thrust for cruise will produce a tremendous fuel consumption and range penalty.
- There are numerous other problems associated with VTOL aircraft design including transition, control, suckdown, hot gas ingestion, FOD, inlet flow matching, and ground erosion.



suckdown

VTOL Jet-propulsion Options

- The major options for jet VTOL propulsion systems are notionally derived in Fig. 20.2.
- Broadly speaking, jet VTOL concepts can be divided into
  - those that utilize fairly conventional engines,
  - those that use engines modified so that the fan and core air are split, with the fan air ducted and exhausted from some place separate from the core air.

VTOL Aircraft Design

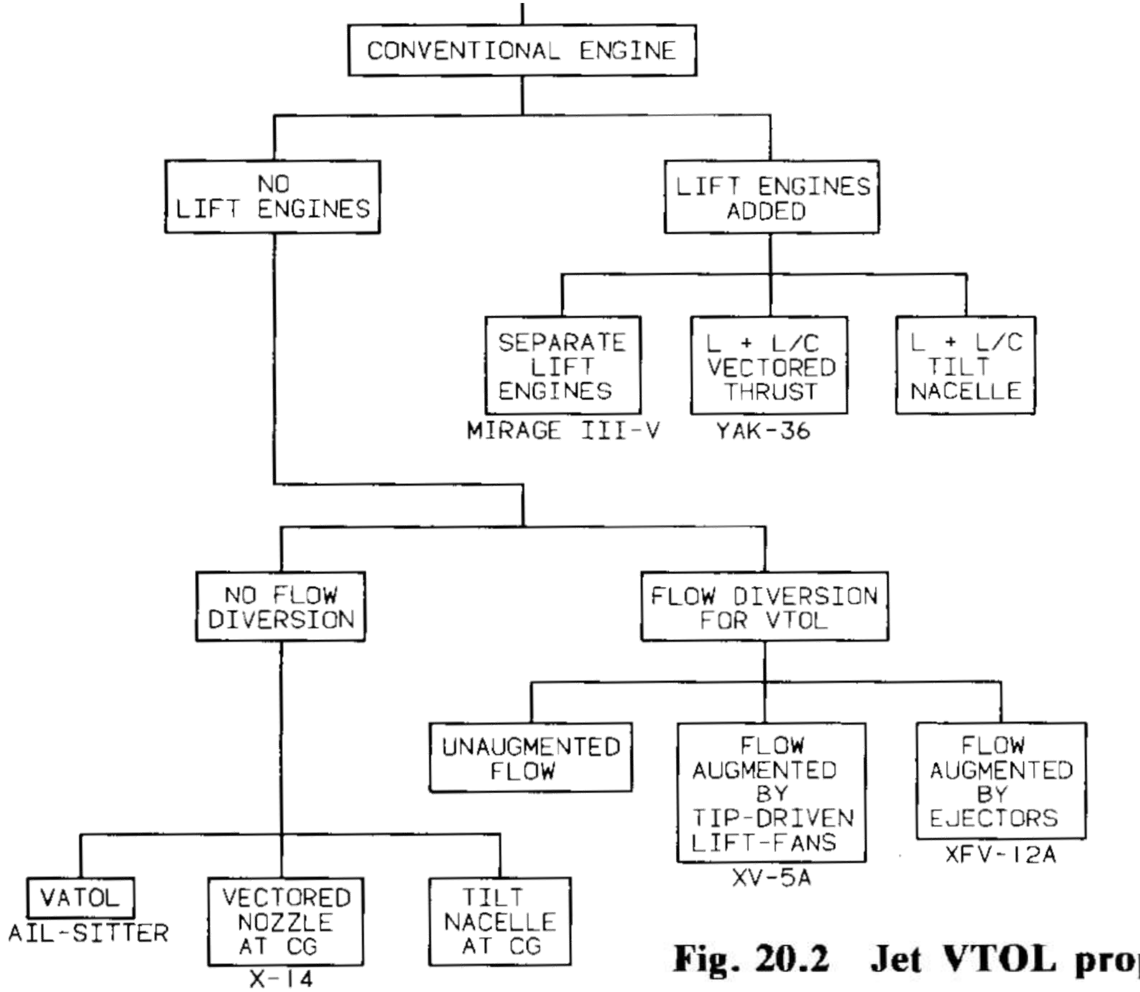


Fig. 20.2 Jet VTOL propulsion options.

VTOL Aircraft Design

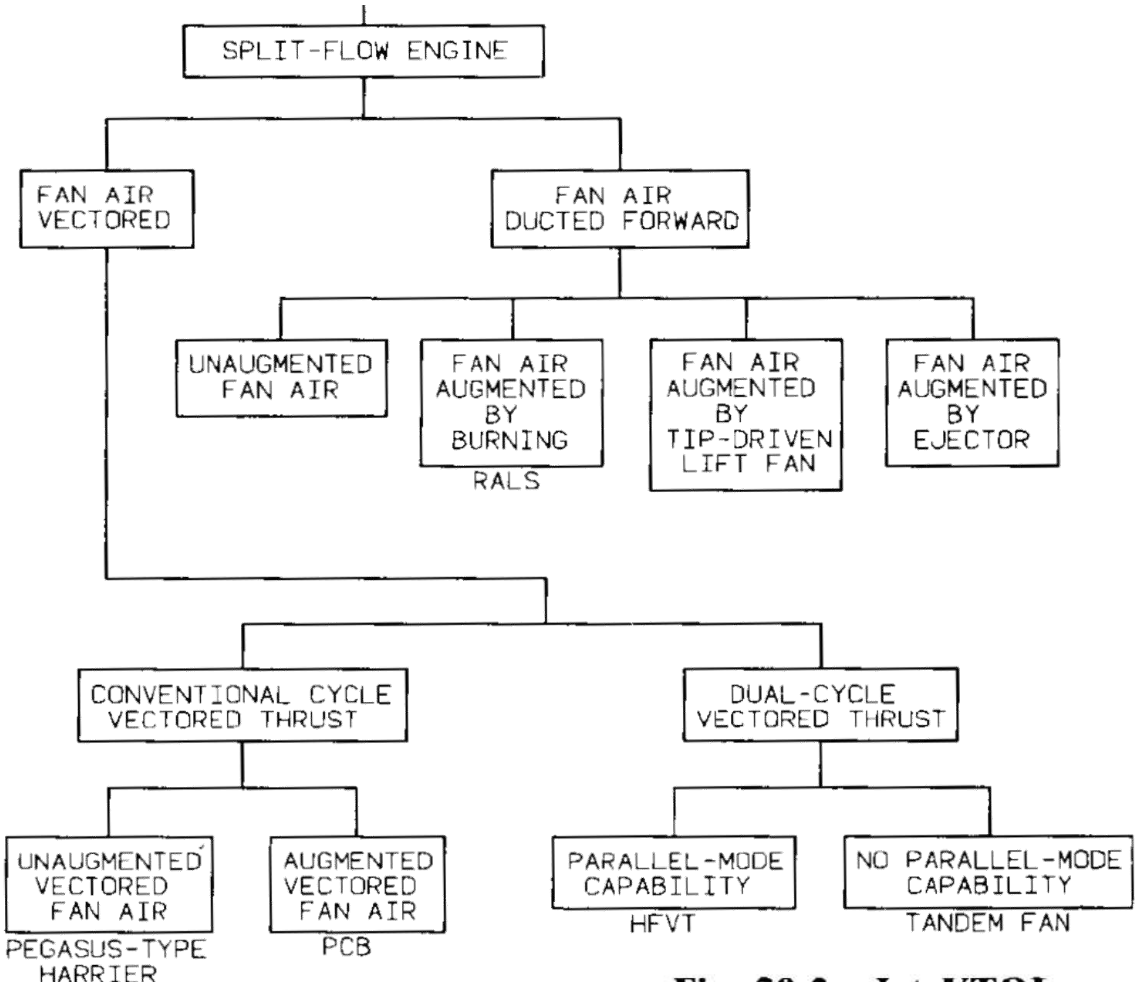


Fig. 20.2 Jet VTOL propulsion options.

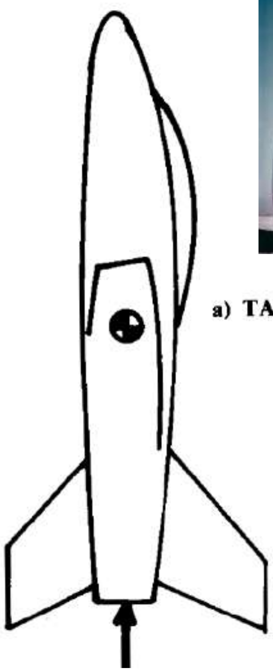
VTOL Aircraft Design

- The conventional-engine VTOL concepts that do not use additional lift engines for vertical flight must have a net takeoff T/W in excess of 1.0.

- If the jet exhaust is not diverted to some other location for vertical flight, the aircraft must either be a tail sitter (VATOL),

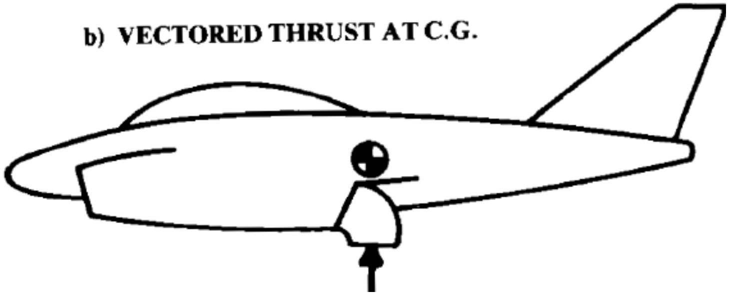
- or have the engine exhaust at the aircraft c.g. and capable of vectoring downward for vertical flight.

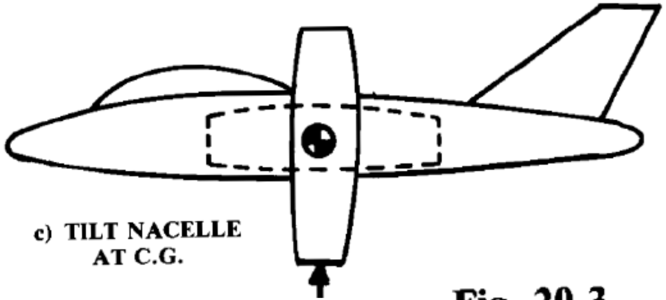
- This can be accomplished by using a vectoring nozzle or nacelles which tilt (Fig. 20.3).



a) TAIL SITTER

b) VECTORED THRUST AT C.G.





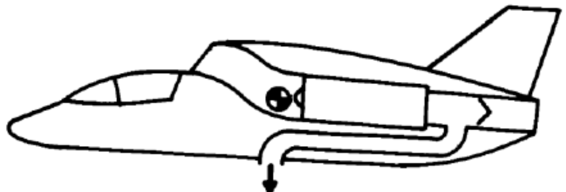
c) TILT NACELLE AT C.G.

Fig. 20.3

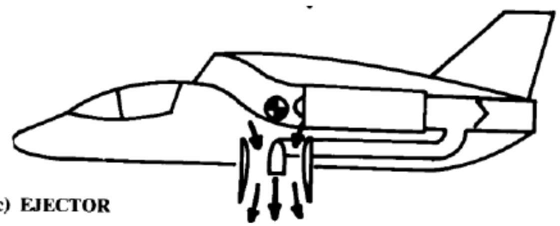
VTOL Aircraft Design

- Some VTOL concepts provide a means of diverting the exhaust flow to gain vertical lift.
- This is generally done by a retracting blocker device in the engine that shuts off the flow through the rearward-facing nozzle.
- The flow is then diverted forward through internal ducting (Fig. 20.4).

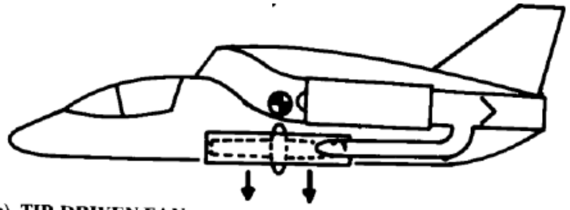
Ryan XV-5A Vertifan



a) UNAUGMENTED FLOW



c) EJECTOR



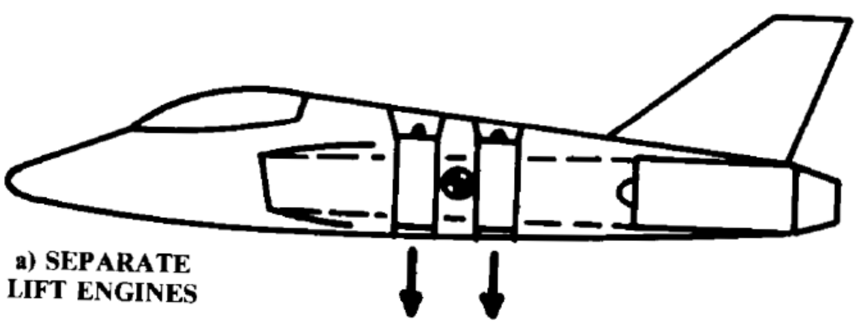
b) TIP-DRIVEN FAN

XFV-12A

Fig. 20.4 Conventional engine, no lift engines, flow diversion used.

VTOL Aircraft Design

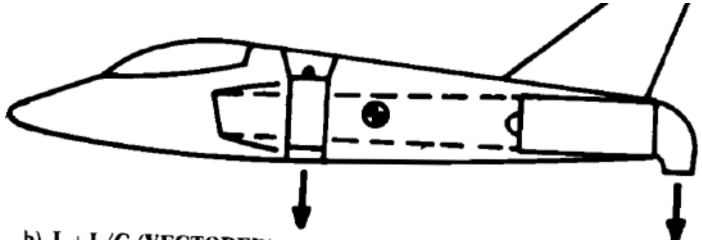
- One of the simplest ways of providing VTOL capability adds lift engines to an essentially conventional aircraft (Fig. 20.5).
- This brute-force approach was used in the Mirage III-V.
- Obviously, the separate lift engines add considerable weight and volume to the design, but the forward-flight engine can be sized for efficient cruise,
- thus solving the thrust-matching problem.



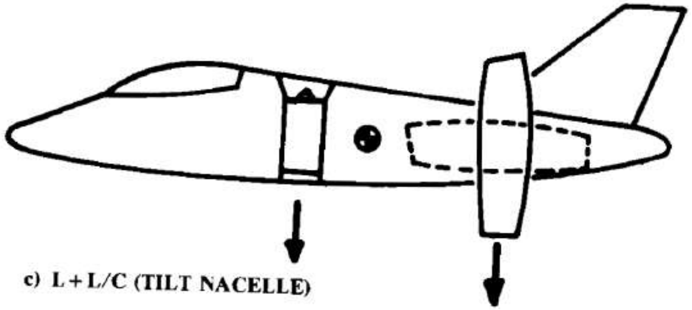
Mirage III-V

VTOL Aircraft Design

- A more subtle/nice approach than the use of separate engines for lift and cruise is to size the forward-flight engine for efficient cruise, but also provide a means of vectoring its thrust downward for vertical flight.
- The vertical-thrust shortfall is made up by the addition of lift engines.
- This is known as a "lift plus lift/cruise (L + L/C)" approach.



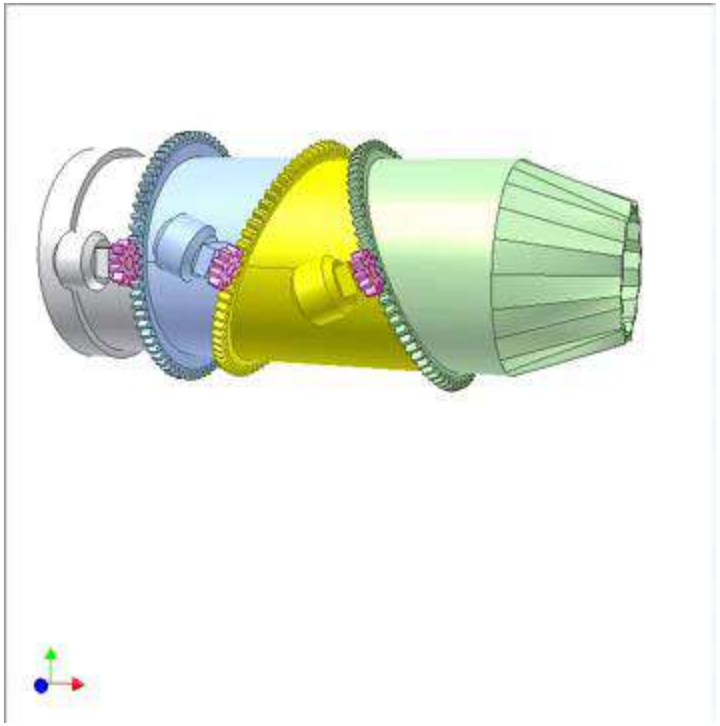
b) L + L/C (VECTORED)



c) L + L/C (TILT NACELLE)

- Since the forward-flight engine is providing some vertical thrust, the thrust required from the lift engines is reduced.
- The forward-flight thrust can be vectored by a vectoring nozzle as in the YAK-36, F-35B or by tilt nacelles.

VTOL Aircraft Design



F-35 swivel nozzle

VTOL Aircraft Design

- A major problem with the L + L/C approach is the transition from vertical to forward flight.

- During the transition period, the lift/cruise engine thrust is being vectored rearward, decreasing the vertical component of thrust.

- Since the lift / cruise engine is at the back of the aircraft, additional thrust is required to avoid a nose-up pitching moment.

- Related to the L + L/C concept is the "shaft-driven lift fan" (SDLF).

- This offers many of the benefits of L + L/ C but without some of the problems.

VTOL Aircraft Design

- In the SDLF concept, a driveshaft runs from the engine to a separate lift fan positioned where the lift engine on the L + L/C concept is located.

- The driveshaft is powered by the main engine through a modified or supplemental turbine, and spins the lift fan to provide vertical thrust.

- This avoids the need to develop a complete new lift engine, although the fan, driveshaft, gearbox, and turbine must be developed.

- Also, SDLF has a cooler front exhaust since the forward lift exhaust is not combusted.

VTOL Aircraft Design

shaft-driven lift fan

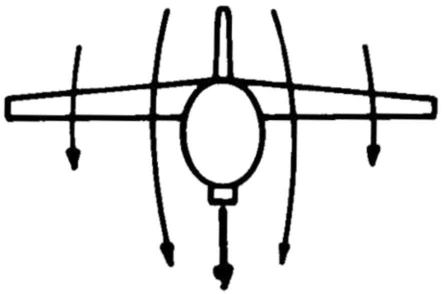
F-35B

Thrust vectoring & shaft-driven lift fan

VTOL Aircraft Design

Suckdown and Fountain Lift

- The VTOL aircraft in hover is not in stagnant air.
- The jet exhaust that supports the aircraft also accelerates the airmass around it.
- This entrainment is due to viscosity and is strongest near the exhaust plume, producing a downward flow field about the aircraft (Fig. 20. l0a).
- This downward flow field pushes down on the aircraft with a "vertical drag" force equivalent to a loss of typically 2-6% of the lift thrust.
- The magnitude of this vertical drag force depends largely upon the relative locations of the exhaust nozzles and the wing.

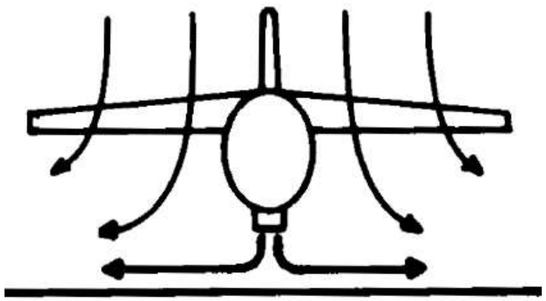


a) FREE AIR ENTRAINMENT

Fig. 20.10

VTOL Aircraft Design

- Figure 20. l0b shows the effect of the ground on the entrained flow field.
- The jet exhaust strikes the ground and spreads outward.
- This increases the mixing between the jet exhaust and the adjacent air, which increases the entrainment effect.
- The entrained download (or "suckdown") therefore increases as the ground is approached.
- A single-jet VTOL concept can experience a 30% reduction in effective lift due to suckdown.



b) SINGLE JET GROUND EFFECTS

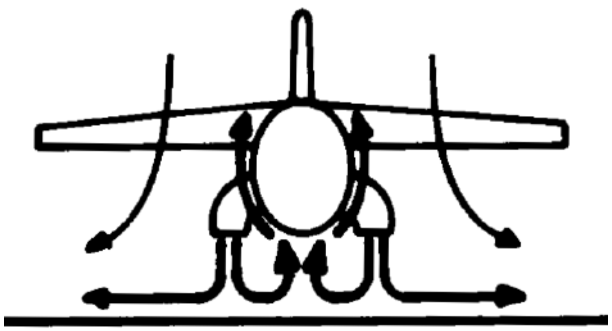
VTOL Aircraft Design

- Figure 20. l0c shows a VTOL concept with widely separated multiple nozzles near the ground.

- The jet exhausts strike the ground and spread outward.

- The exhausts meet in the middle.

- Since there is nowhere else to go, they merge and rise upward, forming a "fountain" under the aircraft.



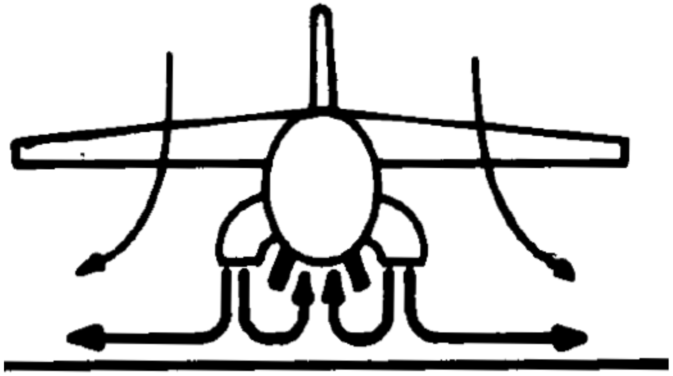
c) MULTIPLE JET GROUND EFFECTS—FOUNTAIN LIFT

  - This fountain pushes upward on the aircraft with a magnitude that will often cancel the suckdown force.

  - The strength of the fountain lift depends upon the exact arrangement of the nozzles and the shape of the fuselage.

VTOL Aircraft Design

- The fountain lift can be increased even more by the use of Lift Improvement Devices (LIDS).
- These are longitudinal strakes located along the lower fuselage corners which capture the fountain (Fig. 20. l0d).
- LIDS added to the AV-8B increased the net vertical lift over 6%.



d) LIDS—FOUNTAIN LIFT

VTOL Aircraft Design

Recirculation and Hot-gas Ingestion

- A VTOL aircraft hovering near the ground tends to "drink its own bath water.
- The hot exhaust gases find their way back into the inlet, causing a significant reduction in thrust.
- Also, this "recirculated" air can include dirt and other erosion particles that can damage or destroy the engine.
- In some cases the dirt kicked up by a hovering VTOL aircraft can completely obscure the pilot's vision.
- Figure 20.11 shows the three contributors to exhaust recirculation: buoyancy, fountain, and relative wind.

VTOL Aircraft Design

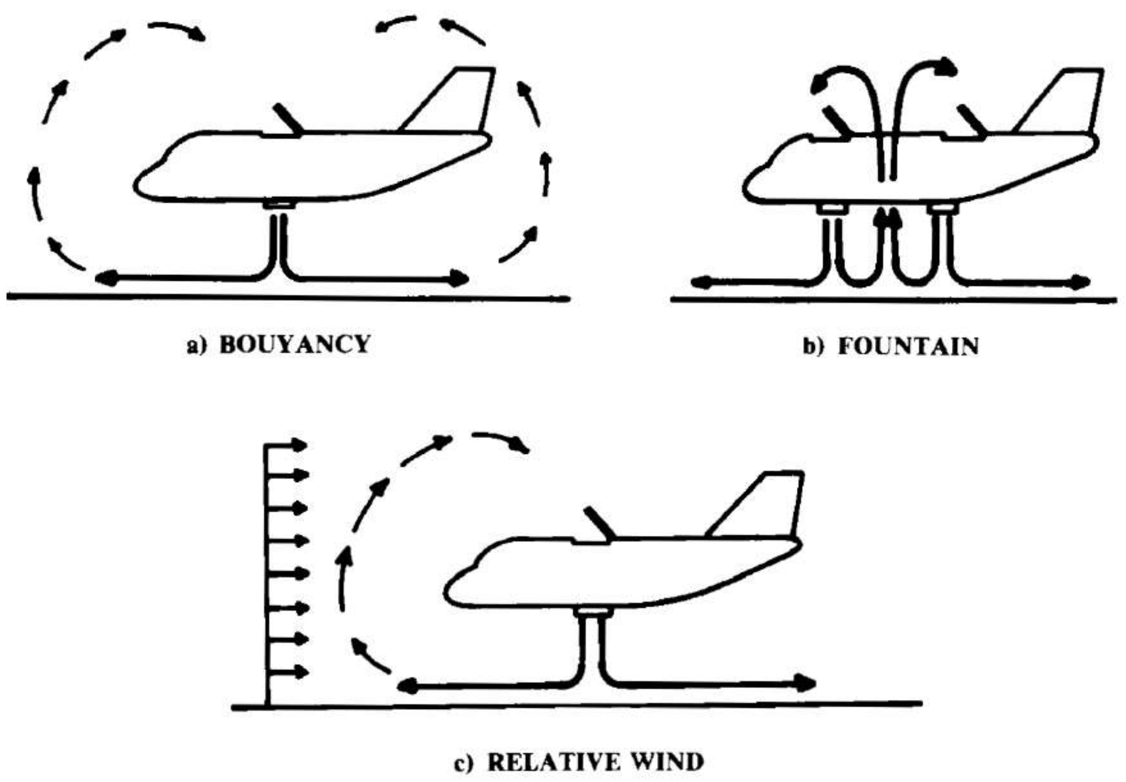


Fig. 20.11 Recirculation.

VTOL Aircraft Design

- Buoyancy refers to the natural tendency of hot gases to rise.
- The buoyancy effect takes time.
- It takes about 30 seconds in hover for the air around the Harrier to heat up by 5°C.
- This 5°C increase in air temperature entering the inlet reduces the engine thrust by about 4%.
- If the nozzle arrangement produces a fountain, the recirculation will be greatly increased.
- This causes additional hot-gas ingestion (HGI) in addition to the buoyancy effect.

VTOL Aircraft Design

- Unlike the buoyancy effect, the fountain effect takes little time to increase the temperature of the air entering the inlet.

- The Harrier experiences a l0°C temperature rise due to the fountain effect. This reduces thrust by about 8%.

- The third contributor to recirculation, relative wind, can be due to atmospheric wind or to aircraft forward velocity.

- Essentially, the relative wind pushes back on the spreading exhaust gases, forcing them up.

VTOL Aircraft Design

VTOL Footprint

- The "footprint" of a VTOL aircraft refers to the effect of the exhaust upon the ground.
- The exhaust of a turbojet VTOL aircraft can be of such high pressure and temperature that it can erode a concrete landing pad if the aircraft is hovered in one spot for too long.

VTOL Control

- The VTOL aircraft in hover and transition must be controlled by some form of thrust modulation.
- Most VTOL concepts use a reaction control system (RCS), in which high-pressure air is ducted to the wing tips and the nose and/or tail.
- This high-pressure air can be expelled through valve-controlled nozzles to produce yaw, pitch, and roll control moments.

VTOL Aircraft Design

- The high-pressure air for the RCS is usually bled off the engine compressor, causing a reduction in thrust.
- The Harrier loses roughly 10% of its lift thrust due to RCS bleed air.
- In addition to three-axis control (roll, pitch, and yaw), a VTOL needs vertical-velocity control ("heave/lift" control).
- This is done by varying the lifting thrust.
- For an aircraft with fixed nozzle-exit area (such as the Harrier), the lifting thrust is varied by engine throttle setting.
- An engine with variable nozzle-exit area can change its lifting thrust more rapidly by changing exit area.

Aircraft Design
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Contents

Chapter 21

Conceptual Design Example

Conceptual Design Example

- This final chapter offers design example that illustrate the concepts and methods presented in this course.
- The design example is a lightweight supercruise dogfighter designed to replace the F-16 as the "low" end of a future "high-low" mix of advanced fighters.
- Design requirements for this aircraft were assumed based upon data for similar aircraft.
- These design requirements were then treated as if they were mandated by some customer and used as the starting point for the design effort.

Conceptual Design Example

- The Turkish Air Force currently operates the F-4 and F-16 as dogfighters.
- The Turkish Air Force is developing the National Fighter Aircraft (MMU: Milli Muharip Uçak) as a replacement for the F-4.
- The new fighter is then likely to be a replacement for the F-16, which is also as old as the F-4.
- This design example presents such an F-16 follow-on design.
- Design requirements are based upon assumed improvements to published F-16 capabilities, with the addition of a required capability for sustained supersonic cruise ("supercruise") on dry power (thrust without the use of afterburners).
- Also, relatively short takeoff and landing requirements are imposed.

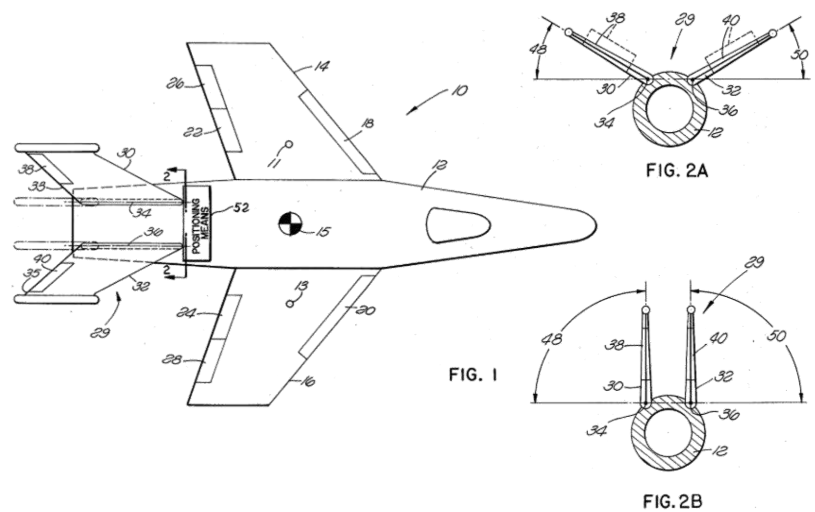
Conceptual Design Example

- The selected design incorporates one unproven technology, the variable dihedral vertical tail.

- This patented concept (by D. P. Raymer) purports to control the rearward shift in aerodynamic center as the aircraft accelerates to supersonic flight

- by converting from a "V" tail subsonically to upright vertical tails supersonically.

- This should reduce trim drag and enhance maneuverability.



Initial Sizing Example

Requirements

- SINGLE - SEAT ; SINGLE - ENGINE (NEW, i.e., "RUBBER")

PRIMARY MISSION : AIR-TO-AIR

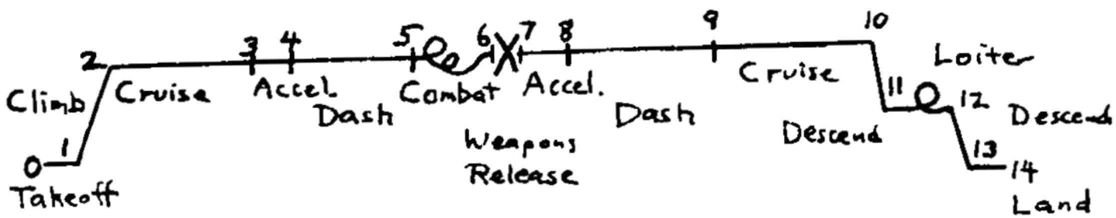
"Dash speed" means a speed that is unsustainable in normal operations but necessary for mission requirements.

For military operations, it refers to supersonic flight to escape a hostile area, travelling at a power setting that will only be available for a short period of time.

In civil airliner operations, it refers to high-speed cruise that is normally uneconomic.

Initial Sizing Example

PRIMARY MISSION: AIR-TO-AIR



3&10 (Cruise): 200 nm at Best Cruise Mach and Altitude (BCA/BCM)
5&9 (Dash): 50 nm at M 1.4 at 35,000 ft.
6 (Combat): 3 min at Maximum Thrust, M0.9 at 20,000 ft
12 (Loiter): 20 min at Sea Level, Best Loiter Speed
7 (Weapons Release): 400 Lb (Missiles only)

PAYLOAD: 2 ADVANCED MISSILES (200 Lb, 5in x92in)
ADVANCED GUN (400 Lb), 750 ROUNDS AMMO (440 Lb)
PILOT (220 Lb)

Initial Sizing Example

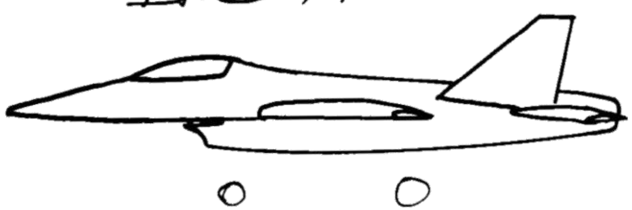
PERFORMANCE REQUIREMENTS

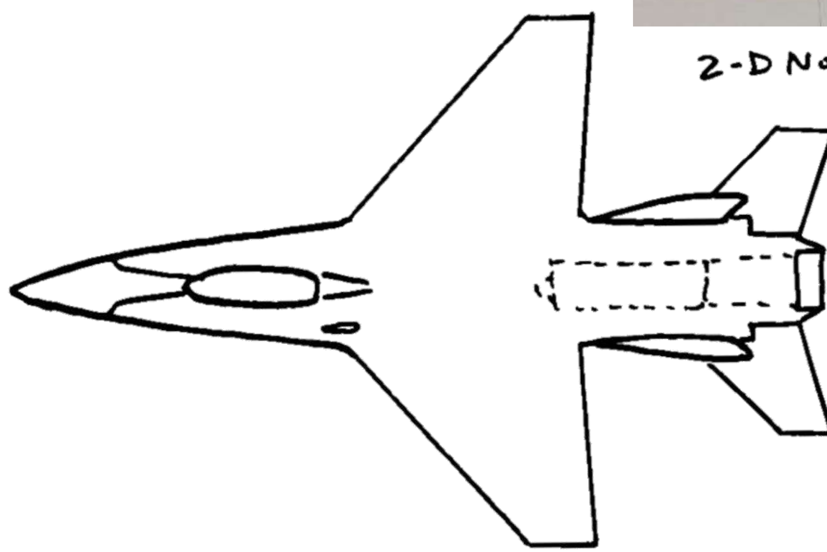
- TAKEOFF AND LANDING: 1,000 ft GROUND ROLL
- APPROACH SPEED $\leq$ 130 kts
- MAXIMUM MACH $\geq$ 1.8 (A/B) ; $\geq$ M1.4 (Dry)
- ACCELERATE M0.9 to M1.4 in 30 sec at 35,000 ft.
- $P_s = 0$ at 5g at 30,000 ft. at M0.9 and M1.4
- $\dot{\psi} \geq 20\,°/\text{sec}$ at 350 kts at 20,000 ft.

2-D Nozzle

Option 1

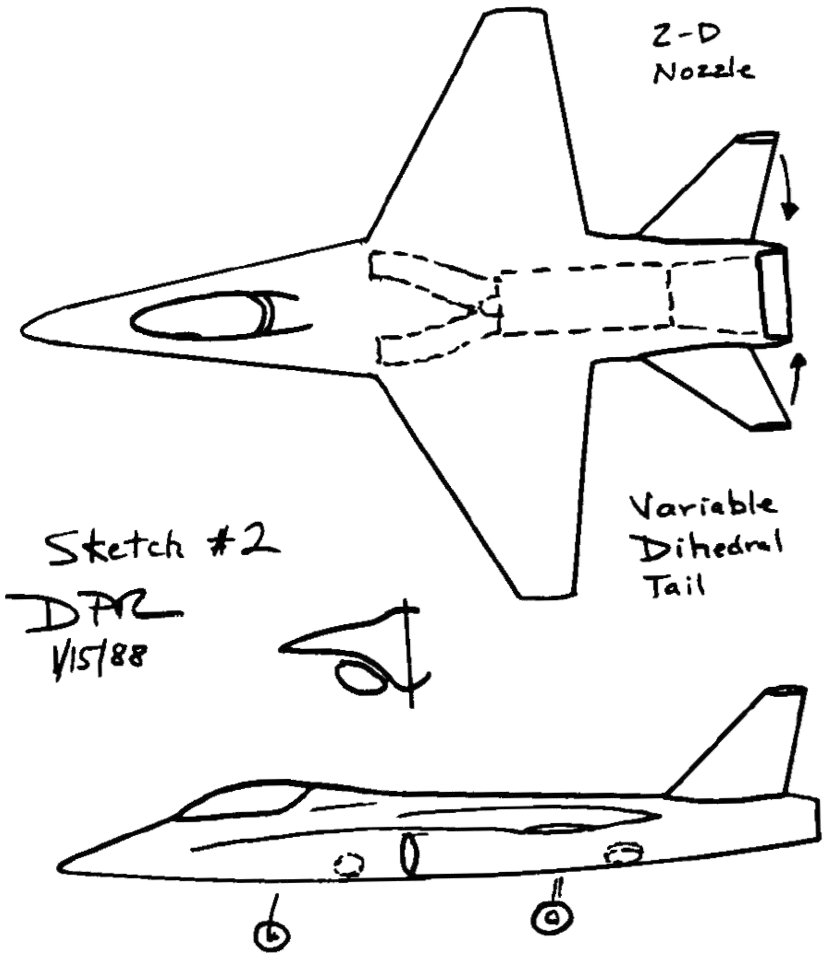
Sketch 1 – CONVENTIONAL CONCEPT
DPR 1/15/88





Initial Sizing Example

Option 2



Initial Sizing Example

WING GEOMETRY

$A = 5.416\,(1.8)^{-.622} = 3.8$

$\Lambda_{LE} = 48^\circ \quad (\Lambda_{c/4} \cong 40^\circ)$

CHECK V.S. FIG. INDICATES TRANSONIC PITCHUP! CHANGE TO: $\left\{ \begin{array}{l} A = 3.5 \\ \Lambda_{c/4} = 30^\circ \end{array} \right\}$

$(\text{SO } \Lambda_{LE} \cong 40^\circ)$

SELECT: $\lambda = 0.25$

$t/c = 6\%$

Airfoil: 64A006 (Initially)

- MAXIMUM MACH ≥ 1.8

- Most swept wings have a taper ratio of about λ 0.2-0.3.

Table 4.1 Aspect ratio

Equivalent aspect Ratio $= aM_{max}^{C}$

Jet aircraft	a	C
Jet trainer	4.737	−0.979
Jet fighter (dogfighter)	5.416	−0.622

Initial Sizing Example

WING GEOMETRY

$$A = 5.416\,(1.8)^{-.622} = 3.8$$

$$\Lambda_{LE} = 48^\circ \quad (\Lambda_{c/4} \cong 40^\circ)$$

CHECK V.S. FIG. TRANSONIC INDICATES PITCHUP! CHANGE TO:

$$\left\{ \begin{array}{l} A = 3.5 \\ \Lambda_{c/4} = 30^\circ \end{array} \right\}$$

$$(\text{so } \Lambda_{LE} \cong 40^\circ)$$

SELECT: $\lambda = 0.25$

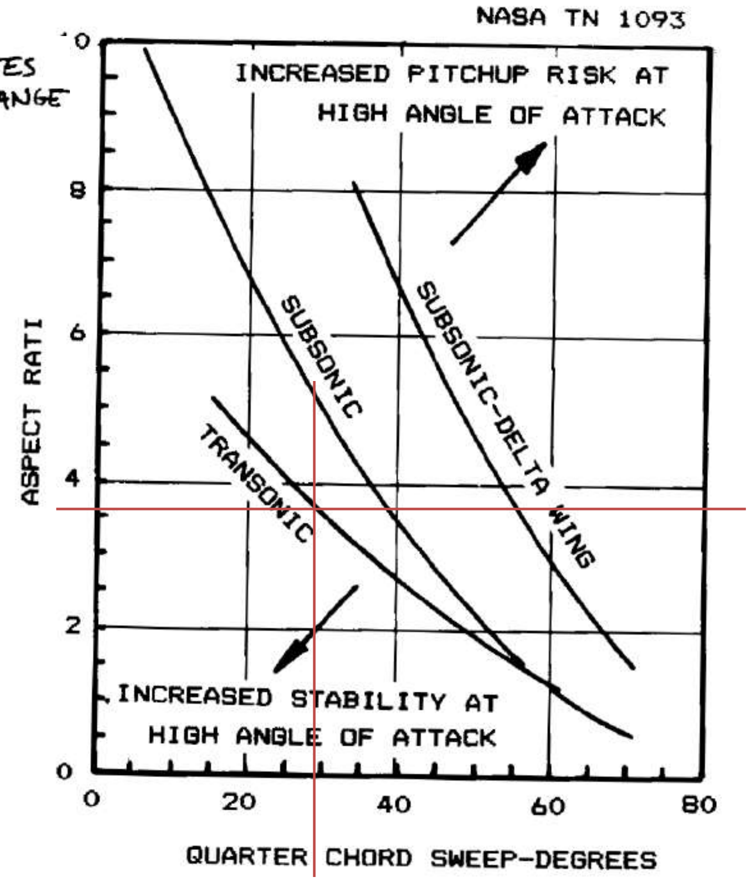


Fig. 4.20 Tail-off pitchup boundaries.

Initial Sizing Example

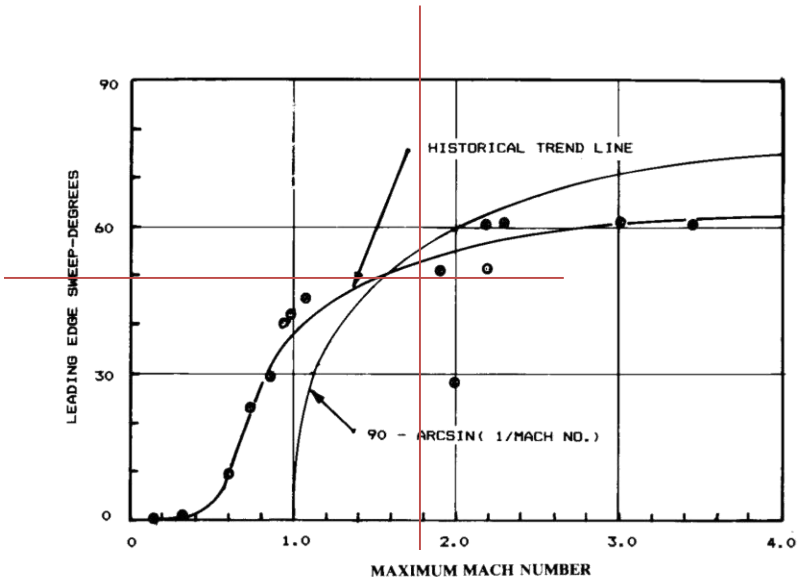


Fig. 4.19 Wing sweep historical trend.

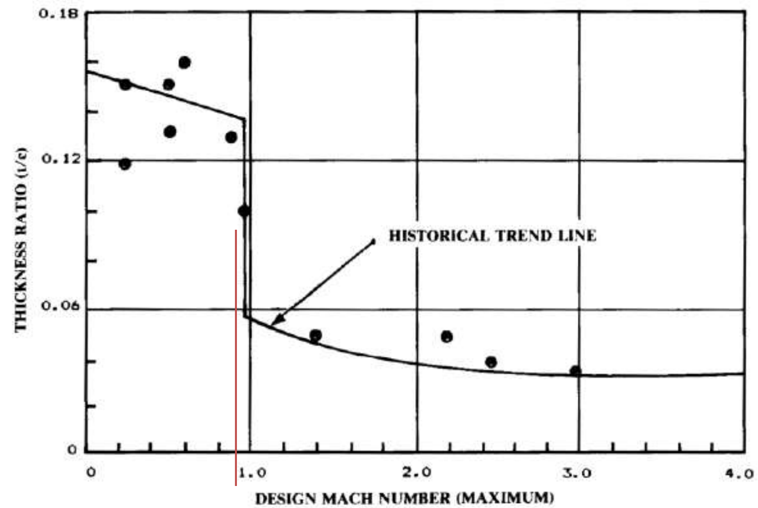


Fig. 4.14 Thickness ratio historical trend.

$\Lambda_{LE} = 48^\circ$ $(\Lambda_{c/4} \cong 40^\circ)$

$t/c = 6\%$

Airfoil: 64A006 (Initially)

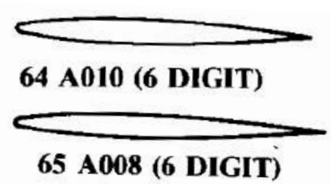


Fig. 4.6 Typical airfoils.

Initial Sizing Example

ENGINE : POST-2000 "RUBBER" ENGINE. APPROXIMATE WITH APPEN. A.4-1 ENGINE WITH 20% SFC REDUCTION.

T/W

table 5.3) $T/W_{takeoff} = 0.648\,(1.8)^{.594} = .92$ (Use Initially)

Table 5.3 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

Initial Sizing Example

W/S

Stall: $V_{approach} \leq 130$ KTS $= 220$ ft/sec

$V_{stall} \leq V_{approach}/1.2 = 183$ ft/sec

$W/S \leq q C_{L_{max}}$ at stall

fig 5.3) $C_{L_{max}} \cong 1.5 + .3$ (L.E. FLAP) $\cong 1.8$

so $W/S \leq 72$ Lb/ft² (at Sea Level)

air density 0.0023769 slug/(cu ft)

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

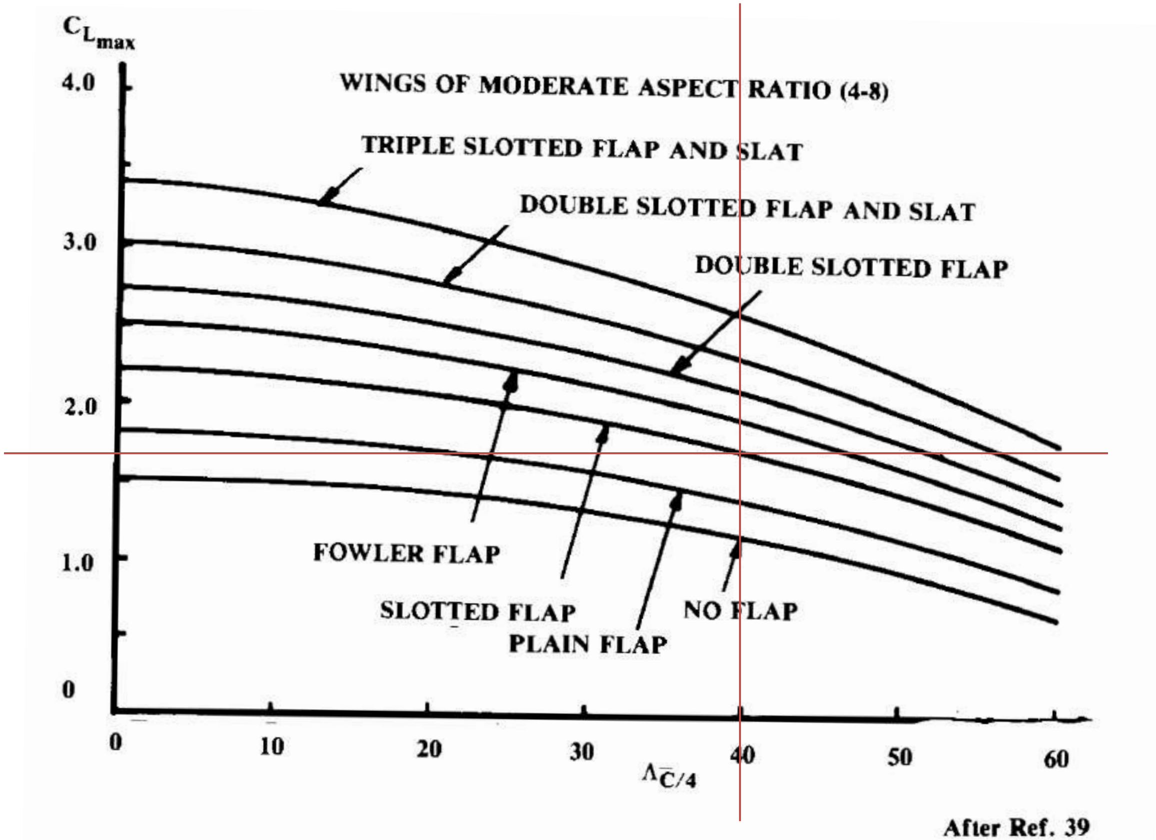


Fig. 5.3 Maximum lift coefficient.

Initial Sizing Example

Landing:

from eq $S_{\substack{\text{landing}\\\text{ground}\\\text{roll}}} = 80 \frac{W}{S}\left(\frac{1}{\sigma C_{L_{max}}}\right) \leq 1{,}000$

$\sigma = 1$ *sea level*

so $W/S \leq 22.5$ (!)

(MUCH TOO LOW FOR A FIGHTER! WE WILL IGNORE THIS INITIALLY AND USE THRUST REVERSING TO LAND.)

$$S_{\text{landing}} = 80\left(\frac{W}{S}\right)\left(\frac{1}{\sigma C_{L\,\text{max}}}\right) + S_a \qquad (5.11)$$

where

S_a = 1000 (airliner-type, 3-deg glideslope)
= 600 (general aviation-type power-off approach)
= 450 (STOL, 7-deg glideslope)

Initial Sizing Example

Takeoff:

fig 5.4) $TOP \cong 80$

eq 5.9) $W/S = TOP\left(\frac{C_{L_{max}}}{1.21}\right) T/W = 80\left(\frac{1.8}{1.21}\right)(.92) = 109$

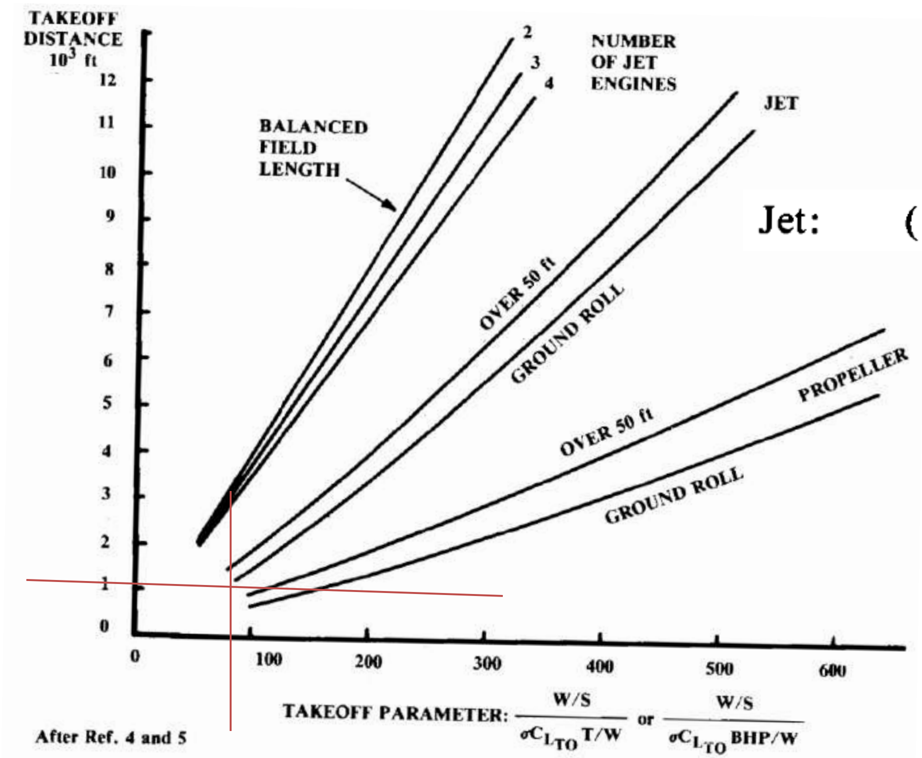


Fig. 5.4 Takeoff distance estimation.

Jet: $$(W/S) = (\mathrm{TOP})\sigma C_{L\,\mathrm{TO}}(T/W) \qquad (5.9)$$

- The aircraft takes off at about 1.1 times the stall speed so the takeoff lift coefficient is the maximum takeoff lift coefficient divided by 1.21

Initial Sizing Example

<u>Cruise:</u> Table 12.3 $C_{fe} = .0035$

assume $S_{wet}/S_{ref} \cong 4$, so $C_{D_0} \cong .014$ (eq 12.23)

eq. 12.50) $e = 4.61\left(1 - .045(3.5)^{.68}\right)(\cos 40°) - 3.1 = 0.86$

At M.9 and 35,000 ft (assumed BCM/BCA), $q = 284$ Lb/ft²

so

$$\left(W/S\right)_{\substack{opt.\\ cruise}} = 284\sqrt{\frac{\pi \times 3.5 \times 0.86 \times 0.014}{3}} = 59.6$$

$$\left(W/S\right)_{takeoff} \cong \frac{59.6}{.97 \times .977} = 62.9$$

(Using typical values for $\frac{W_1}{W_0}$ and $\frac{W_2}{W_1}$)

- This must be converted to takeoff conditions by dividing the cruise wing loading by the ratio of the average cruise weight to the takeoff weight.

$$\text{Swept-Wing Aircraft: } e = 4.61(1 - 0.045A^{0.68})(\cos\Lambda_{LE})^{0.15} - 3.1 \qquad (12.50)$$

$$(\Lambda_{LE} > 30 \text{ deg})$$

$$\text{Maximum Jet Range: } W/S = q\sqrt{\pi A e C_{D_0}/3} \qquad (5.14)$$

Initial Sizing Example

Table 12.3 Equivalent skin friction coefficients

$C_{D_0} = C_{fe} \dfrac{S_{\text{wet}}}{S_{\text{ref}}}$	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

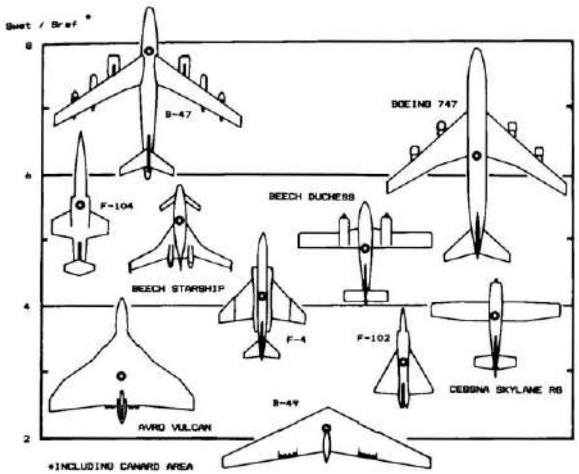


Fig. 3.5 Wetted area ratios.

- C_{D0} is the zero lift drag coefficient, and equals approximately 0.015 for a jet aircraft, 0.02 for a clean propeller aircraft, and 0.03 for a dirty, fixed-gear propeller aircraft.

- The Oswald efficiency factor e is a measure of drag due to lift efficiency, and equals approximately 0.6 for a fighter and 0.8 for other aircraft.

or

$$C_{D_0} = C_{fe} \frac{S_{\text{wet}}}{S_{\text{ref}}} \tag{12.23}$$

Initial Sizing Example

Instantaneous Turn: At 350 kts and 20,000 ft, $q = 222 \text{ lb/ft}^2$

$$\dot{\psi} = \frac{20}{57.3} \geq \frac{32.2\sqrt{n^2-1}}{350 \times 1.689} \text{ ; so } n \geq 6.5 \text{ g's } \left(= qC_L/_{W/S}\right)$$

Assume $C_{L_{max\ maneuver}} \cong 1.4$

then $(W/S)_{combat} = \frac{222 \times 1.4}{6.5} = 48$

$(W/S)_{takeoff} \cong 48/.85 = \underline{56}$

- For a fighter with a complex system of leading- and trailing edge flaps which can be deployed during combat, a maximum usable lift coefficient of about 1.0-1.5 is attainable.
- Again, the resulting wing loading must be divided by the ratio of combat weight to takeoff weight to obtain the required takeoff wing loading.
- This is approximately 0.85 times the takeoff weight for most fighters.

$$\dot{\psi} = \frac{g\sqrt{n^2-1}}{V} \tag{5.17}$$

- $\dot{\psi} \geq 20$ °/sec at 350 kts at 20,000 ft.

$$n = \frac{qC_L}{W/S} \tag{5.18}$$

Initial Sizing Example

Sustained Turn: At M.9 and 30,000 ft ; V= 895 ft/s ; q=357 lb/ft²

$C_{D_0}=.014$; assume $e=.6$ (reduced during high-g turns)

$$(T/W)_{combat} = 0.92\left(\frac{1}{.85}\right)\left(\frac{16000}{30000}\right) = .58$$

↑ Actual and S.L.S. values from A.4-1

↑ Typical (W_{combat}/W_0)

$$(W/S)_{combat} = \left[.58+\sqrt{.58^2-\frac{4\times5^2\times.014}{\pi\times3.5\times.6}}\right] \Big/ \left[\frac{2\times5^2}{357\times\pi\times3.5\times.6}\right] = 44$$

$$(W/S)_{takeoff} = 44/.85 = 52$$

- At high angles of attack the e value may be reduced by 30% or more.

• $P_S=0$ at 5g at 30,000 ft. at M0.9 and M1.4

T/W

table 5.3) $T/W_{takeoff} = 0.648\,(1.8)^{.594} = .92$ (Use Initially)

$$\frac{W}{S} = \frac{(T/W) \pm \sqrt{(T/W)^2 - (4n^2C_{D0}/\pi Ae)}}{2n^2/q\pi Ae} \qquad (5.25)$$

Initial Sizing Example

use of $W/S = 56$ **Instantaneous Turn:**

$$(W/S)_{combat} = .85 \times 56 = 48$$

eq. 5.24) $$(T/W)_{combat} = \frac{357 \times .014}{48} + 48\left(\frac{5^2}{357 \times \pi \times 3.5 \times .6}\right) = .614$$

$$(T/W)_{takeoff} = .614(.85)\left(\frac{30000}{16000}\right) = \underline{.98}$$ (Use This)

$$\frac{T}{W} = \frac{qC_{D0}}{W/S} + \frac{W}{S}\left(\frac{n^2}{q\pi Ae}\right) \qquad (5.24)$$

Initial Sizing Example

INITIAL SIZING

Empty Weight Fraction: (Assume composite structure)

table 6.1) $\frac{W_e}{W_0} = \left[-.02 + 2.16\, W_0^{-.1} \times 3.5^{.2} \times .98^{.04} \times 56^{-.1} \times 1.8^{.08}\right] \times 0.9$

$$\frac{W_e}{W_0} = 1.75\, W_0^{-.1} - .018$$

Mission Segment Weight Fractions:

Warmup & Takeoff: eq 6.8) $\frac{W_1}{W_0} = .98$

Climb: eq 6.9) $\frac{W_2}{W_1} = 1.0065 - .0325(.9) = .977$

$$W_1/W_0 = 0.97\text{–}0.99$$

$$W_i/W_{i-1} = 1.0065 - 0.0325M$$

Table 6.1 Empty weight fraction vs W_0, A, T/W_0, W_0/S, and M_{max}

$$W_e/W_0 = (a + bW_0^{C1} A^{C2} (T/W_0)^{C3} (W_0/S)^{C4} M_{max}{}^{C5}) K_{vs}$$

	a	b	$C1$	$C2$	$C3$	$C4$	$C5$
Jet fighter	−0.02	2.16	−0.10	0.20	0.04	−0.10	0.08

Initial Sizing Example

Cruise: (Assume M.9 at 35,000 ft. for BCA/BCM)
V= 876 ft/sec ; q= 283 Lb/ft²

$$W/S = 56 \times .98 \times .977 = 54$$

eq 6.13) $$L/D = \frac{1}{\frac{283 \times .014}{54} + \frac{54}{283\pi \times 3.5 \times .86}} = 10.7$$

SFC:

- A.4-1, Partial Power at M.9 ; 36,000 ft gives: C=1.07
- Increase this 10% to approximate installation: C = 1.18
- Reduce this 20% for advanced technology: C = 0.94

so;

eq 6.11) $$\frac{W_3}{W_2} = e^{-\frac{(200 \times 6076)(.94/3600)}{876 \times 10.7}} = .967$$

$$\frac{L}{D} = \frac{1}{\frac{qC_{D0}}{W/S} + \frac{W}{S}\frac{1}{q\pi Ae}} \quad (6.13)$$

3810(Cruise): 200 nm

1 nautical m to ft : 6076

$$\text{Jet:} \quad \frac{W_i}{W_{i-1}} = \exp\frac{-RC}{V(L/D)} \quad (6.11)$$

Initial Sizing Example

Acceleration:

eq 6.10) $\left(\frac{W_4}{W_3}\right)_{M.1 \to 1.4} = .9616$

eq 6.9) $\left(\frac{W_4}{W_3}\right)_{M.1 \to .9} = .9773$

$$\left(\frac{W_4}{W_3}\right)_{M.9 \to 1.4} = \frac{.9616}{.9773} = .984$$

Subsonic: $W_i/W_{i-1} = 1.0065 - 0.0325M$ (6.9)

Supersonic: $W_i/W_{i-1} = 0.991 - 0.007M - 0.01M^2$ (6.10)

Initial Sizing Example

$$K = \frac{A(M^2-1)}{4A\sqrt{M^2-1}-2}\cos\Lambda_{LE} \qquad (12.52)$$

5#9 (Dash)': 50 nm at M 1.4 at 35,000 ft.

$$\frac{L}{D} = \frac{1}{\frac{qC_{D0}}{W/S} + \frac{W}{S}\frac{1}{q\pi Ae}}$$

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

<u>Dash</u>: M1.4 at 35,000 ft.; V = 1362 ft/sec; q = 685 Lb/ft²

$W/S = 56 \times .98 \times .977 \times .967 \times .984 = 51$

Looking at fig. 12.31, we roughly estimate;

$(C_{D_0})_{M1.4} \cong 2 \times (C_{D_0})_{\text{subsonic}} = .028$ (crude !)

eq 12.52) $(K)_{M1.4} \cong \left(\frac{3.5(1.4^2-1)}{4\times 3.5\sqrt{1.4^2-1}-2}\right)\cos 40° = .22$

or $e = \frac{1}{\pi A K} = .414$

$L/D = \frac{1}{\frac{685\times .028}{51} + \frac{51\times .22}{685}} = 2.55$ (!)

SFC:
A.4-1, Military (Max. Dry) Power $C = 1.2$
Increase 10% (Installation)
Reduce 20% (Advanced Technology) $\Big\} C = 1.06$

eq. 6.11) $\frac{W_5}{W_4} = e^{-\frac{(50\times 6076)(1.06/3600)}{1362\times 2.55}} = .975$

Initial Sizing Example

Combat $\quad d = 3$ min.

$$(T/W)_{combat} = .98 \times \left(\frac{16000}{30000}\right) \Big/ (.98 \times .977 \times .967 \times .984 \times .975) = .588$$

SFC: A.4-1 for Max. thrust at M.9 at 20,000 ft : $C = 1.78$

Increase 10% for installation
Reduce 20% for advanced technology $\Big\}$ $C = 1.57$

$$\frac{W_6}{W_5} = 1 - [(1.57/3600)(.588)(3 \times 60)] = .954$$

$$W_i/W_{i-1} = 1 - C(T/W)(d) \qquad (6.16)$$

Initial Sizing Example

Weight Drop: Ignore for initial sizing

Accelerate: $\frac{W_8}{W_7} \cong \frac{W_4}{W_3} = .984$

Dash: $\frac{W_9}{W_8} \cong \frac{W_5}{W_4} = .975$

Cruise: $\frac{W_{10}}{W_9} \cong \frac{W_3}{W_2} = .967$

Descent: Ignore, assuming range credit

Initial Sizing Example

Loiter: $E = 20$ min ; Sea Level

$$\left(W/S\right)_{loiter} = 56 \times .98 \times .977 \times .967 \times .984 \times .975 \times .954 \times .984 \times .975 \times .967 = 44$$

$$\text{eq (17.13)} \quad V_{\substack{best\\loiter}} = \sqrt{\frac{2 \times 44}{\rho}\sqrt{\frac{1}{.014 \times \pi \times 3.5 \times .86}}} = 319 \text{ ft/sec}$$

so $q = 121$ lb/ft^2

$$L/D = \frac{1}{\frac{121 \times .014}{44} + \frac{44}{121\,\pi \times 3.5 \times .86}} = 13$$

SFC: Adjusted from A.4-1 : $C = .906$

$$\frac{W_{12}}{W_{11}} = e^{-\frac{(20 \times 60)(.906/3600)}{13}} = .977$$

$$\frac{L}{D} = \frac{1}{\frac{qC_{D0}}{W/S} + \frac{W}{S}\frac{1}{q\pi Ae}}$$

$$\text{Jet:} \quad \frac{W_i}{W_{i-1}} = \exp\frac{-EC}{L/D} \qquad (6.14)$$

where E = endurance or loiter time.

Initial Sizing Example

Descent for Landing: $\frac{W_{13}}{W_{12}} = .993$

Land: $\frac{W_{14}}{W_{13}} = .995$ (eq 6.23)

TOTAL MISSION WEIGHT FRACTION

$$\frac{W_{14}}{W_0} = .98 \times .977 \times .967 \times .984 \times .975 \times .954 \times .984 \times .975 \times .967 \times .977 \times .993 \times .995 = .7586$$

FUEL FRACTION

$$\frac{W_f}{W_0} = 1.06(1 - .7586) = .256$$

$$W_i/W_{i-1} = 0.990 \text{ to } 0.995 \qquad (6.22)$$

$$W_i/W_{i-1} = 0.992 \text{ to } 0.997 \qquad (6.23)$$

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right) \qquad (6.2)$$

Initial Sizing Example

SIZING

Need to adjust Empty Weight equation for impact of variable dihedral tails. Assume $\Delta W_e = 200$ Lb at initial guess $W_0 = 20{,}000$.

Then $W_e = \left[\left[1.75\,(20{,}000)^{-.1} - .018\right] 20{,}000\right] + 200 = 12{,}841$ Lb

At other $W_{0\,\text{guess}}$ values, use;

eq 19.13) $W_e = 12841 \left[\dfrac{W_0}{20000}\right]^{.9}$

SIZING ITERATIONS

$W_{\text{crew + payload}} = 220 + (2 \times 200) + 400 + 440 = 1460$ Lb

$$\frac{W_e}{W_0} = 1.75\, W_0^{-.1} - .018$$

FUEL FRACTION

$$\frac{W_f}{W_0} = 1.06\,(1 - .7586) = .256$$

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

$$W_e = W_{e\,\text{as drawn}} \left[\frac{W_0}{W_{0\,\text{as drawn}}}\right]^{(1+c)} \qquad (19.13)$$

C typically equals (-0.1),

Initial Sizing Example

***** SIZING ITERATIONS **********

WOG	WF	WE	WOCALC
20000.0	5117.7	12841.0	19418.7
19534.9	4998.7	12572.0	19030.6
19131.5	4895.4	12338.0	18693.5
18781.1	4805.8	12134.5	18400.2
18476.4	4727.8	11957.1	18145.0
18211.3	4660.0	11802.6	17922.6
17980.3	4600.9	11667.8	17728.7
17779.0	4549.4	11550.2	17559.5
17603.4	4504.4	11447.5	17411.9
17450.2	4465.2	11357.8	17283.0
17316.4	4431.0	11279.4	17170.4
17199.6	4401.1	11210.8	17071.9
17097.4	4375.0	11150.9	16985.9
17008.2	4352.1	11098.5	16910.6
16930.1	4332.1	11052.6	16844.8
16861.9	4314.7	11012.5	16787.2
16802.1	4299.4	10977.4	16736.8
16749.9	4286.0	10946.7	16692.7
16704.1	4274.3	10919.8	16654.1
16664.1	4264.1	10896.2	16620.3
16629.0	4255.1	10875.6	16590.7
16598.4	4247.3	10857.5	16564.8
16571.5	4240.4	10841.7	16542.1
16548.0	4234.4	10827.8	16522.2
16527.4	4229.1	10815.7	16504.8
16509.3	4224.5	10805.1	16489.5
16493.5	4220.4	10795.8	16476.2
16479.6	4216.9	10787.6	16464.5

Conceptual Design Example

$$b = \sqrt{AS} \qquad (7.5)$$

$$C_{root} = \frac{2S}{b(1+\lambda)} \qquad (7.6)$$

$$C_{tip} = \lambda C_{root} \qquad (7.7)$$

$$\bar{C} = \left(\frac{2}{3}\right) C_{root} \frac{1+\lambda+\lambda^2}{1+\lambda} \qquad (7.8)$$

$$\bar{Y} = \left(\frac{b}{6}\right)\left(\frac{1+2\lambda}{1+\lambda}\right) \qquad (7.9)$$

LAYOUT DATA

$W_0 \cong 16480$ Lb. $\quad W_f = .256 \times W_0 \cong 4220$ Lb $= 94$ ft^3

Fuselage: table 6.3) $L \cong .93\,(16480)^{.39} = 41$ ft $= 492$ in

Wing: $S = 16480/56 = 294$ ft^2

$A = 3.5 \qquad \lambda = 0.25 \qquad \Lambda_{c/4} = 30°$

eq 7.5) $b = \sqrt{3.5 \times 294} = 32$ ft $= 384$ in

eq 7.6) $C_{root} = \frac{2 \times 294}{32\,(1+.25)} = 14.7$ ft $= 176$ in

eq 7.7) $C_{tip} = 176 \times .25 = 44$ in

eq 7.8) $\bar{C} = 123$ in

eq 7.9) $\bar{Y} = 76.8$ in

Table 6.3 Fuselage length vs W_0

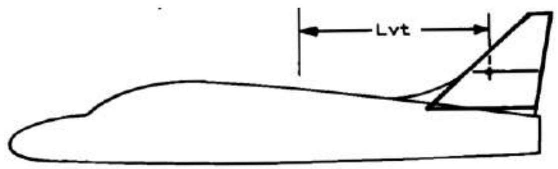
Length $= aW_0^C$	a	C
Jet fighter	0.93	0.39

Conceptual Design Example

$$c_{VT} = \frac{L_{VT} S_{VT}}{b_W S_W} \qquad (6.26)$$

$$c_{HT} = \frac{L_{HT} S_{HT}}{\bar{C}_W S_W} \qquad (6.27)$$

wings, the tail arm is about 50–55% of the fuselage length. For aft-mounted engines the tail arm is about 45–50% of the fuselage length. A sailplane has



tails. The tail dihedral angle should be set to the arctangent of the square root of the ratio between the required vertical and horizontal tail areas. This

Tails: (see ch.4) Lay out "V" tail such that the total tail area equals the sum of the required vertical and horizontal tail areas determined by the volume coefficient method. (Assume $L_t \cong 200$ in)

$$\left.\begin{aligned} &\text{Vertical Tail: } S_{vt} = .07 \frac{b_w S_w}{L_t} = 39\ ft^2 \\ &\text{Horizontal Tail: } S_{ht} = .4 \frac{\bar{c}_w S_w}{L_t} = 72\ ft^2 \end{aligned}\right\} \text{Sum} = 111\ ft^2$$

- If the tails met at the aircraft centerline, $\Gamma_t = \tan^{-1}\left(\frac{39}{72}\right) = 28.4°$
- We will lay out the projected planform for top view as a horizontal equivalent, using $\Gamma_t = 28.4°$ and $S_{t_{true}} = 111\ ft^2$.

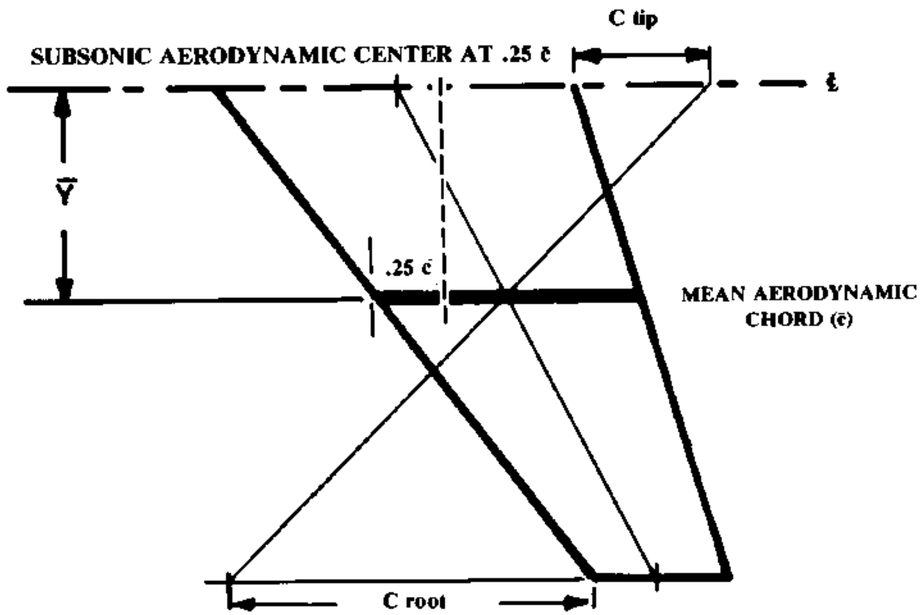
$$\left.\begin{aligned} &(S_{ht})_{projected} = 111 \cos 28 = 97.6\ ft^2 \\ &\text{Using } A = 3.5;\ \lambda = .25;\ \Lambda_{\bar{c}/4} = 30° \end{aligned}\right\} \begin{aligned} &b_h = 18.5\ ft = 222\ in \\ &C_{root} = 8.4\ ft = 101\ in \\ &C_{tip} = 25.3\ in \end{aligned}$$

- True tail geometry will be graphically determined.

Table 6.4 Tail volume coefficient

	Typical values	
	Horizontal c_{HT}	Vertical c_{VT}
Jet fighter	0.40	0.07

Conceptual Design Example



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

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

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

Fig. 4.17 Mean aerodynamic chord.

Conceptual Design Example

APPENDIX A

A.4-1 Afterburning Turbofan

A.4-1 Afterburning turbofan characteristics

Sea-level static thrust, lb	30,000
Sea-level static TSFC, 1/hr	1.64
Sea-level static airflow, lbm/s	246
Bare-engine weight, lb	3,000
Engine length (including axisymmetric nozzle), in.	160
Maximum diameter, in.	44
Fan-face diameter, in.	40
Overall pressure ratio	22
Fan pressure ratio	4.3
Bypass ratio	0.41

$$L = L_{\text{actual}}(\text{SF})^{0.4} \tag{10.1}$$

$$D = D_{\text{actual}}(\text{SF})^{0.5} \tag{10.2}$$

$$W = W_{\text{actual}}(\text{SF})^{1.1} \tag{10.3}$$

ENGINE: $T = (T/W)W_0 = .98 \times 16480 = 16150.4$ Lb (SLS)

A.4-1 ; 100%-Sized Engine: $T = 30{,}000$ Lb

$L = 160$ in

$D = 44$ in

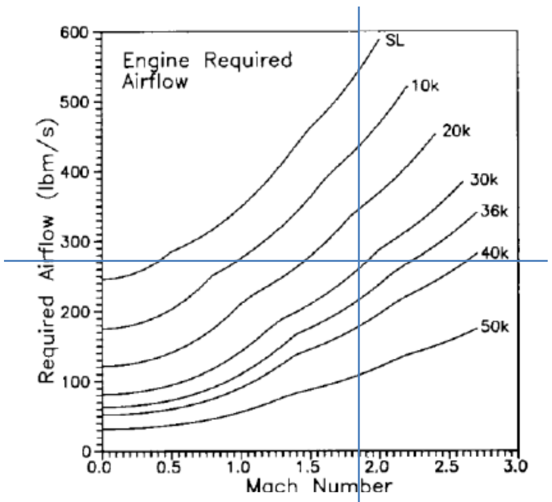
$W = 3{,}000$ Lb

so Scale Factor: $SF = \dfrac{16150.4}{30{,}000} = .538$

$$\left.\begin{aligned} L &= 160\,(.538)^{.4} = 125 \text{ in} \\ D &= 44\,(.538)^{.5} = 32 \text{ in} \\ W &= 3000\,(.538)^{1.1} = 1517 \text{ Lb} \end{aligned}\right\} \text{Engine with conventional nozzle}$$

TO PROVIDE PITCH CONTROL AT SUPERSONIC SPEEDS (WHEN THE TAILS ARE NEAR-VERTICAL), WE WILL USE A TWO-DIMENSIONAL VECTORING NOZZLE WITH THRUST REVERSING TO SHORTEN THE LANDING. THIS REQUIRES A CIRCLE-TO-SQUARE ADAPTER WHICH LENGTHENS THE ENGINE.

Conceptual Design Example



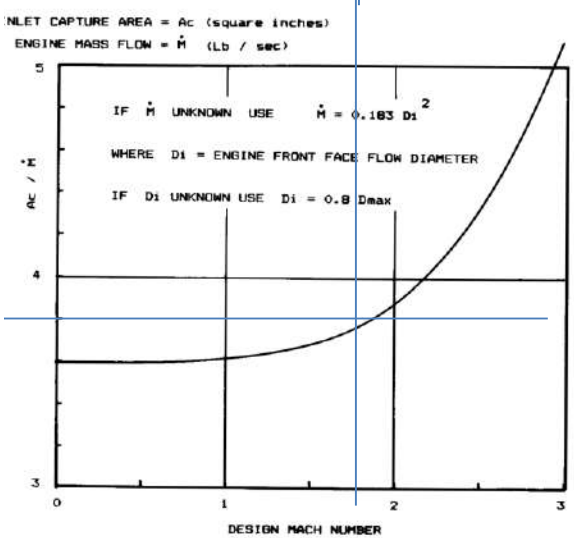


Fig. 10.13 Preliminary capture area sizing.

CAPTURE AREA SIZING

From A.4-1 at M1.8 at 30,000 ft, $\dot{m} = 270$ Lbm/s (mass flow)
Scale by Scale Factor: $\dot{m} = .538 \times 270 = 145.3$ Lbm/s

Fig 10.13) $A_c/\dot{m} = 3.8$ at M1.8, so $A_c = 3.8 \times 145.3 = \underline{552 \text{ in}^2}$

LANDING GEAR

$W_w \cong .9 \times \frac{16480}{2} = 7416$

MAIN: $D = 1.59\,(7416)^{.302} = 23$ in
$W = .098\,(7416)^{.467} = 6.3$ in

NOSE: $\left.\begin{array}{l} D = 18 \text{ in} \\ W = 5 \text{ in} \end{array}\right\}$ 80% of main gear

developed from data in Ref. 1 for rapidly estimating main tire sizes (assuming that the main tires carry about 90% of the aircraft weight).

Nose tires can be assumed to be about 60–100% the size of the main tires. The front tires of a bicycle or quadricycle-gear aircraft are usually the same size as the main tires. Taildragger aft tires are about a quarter to a third the

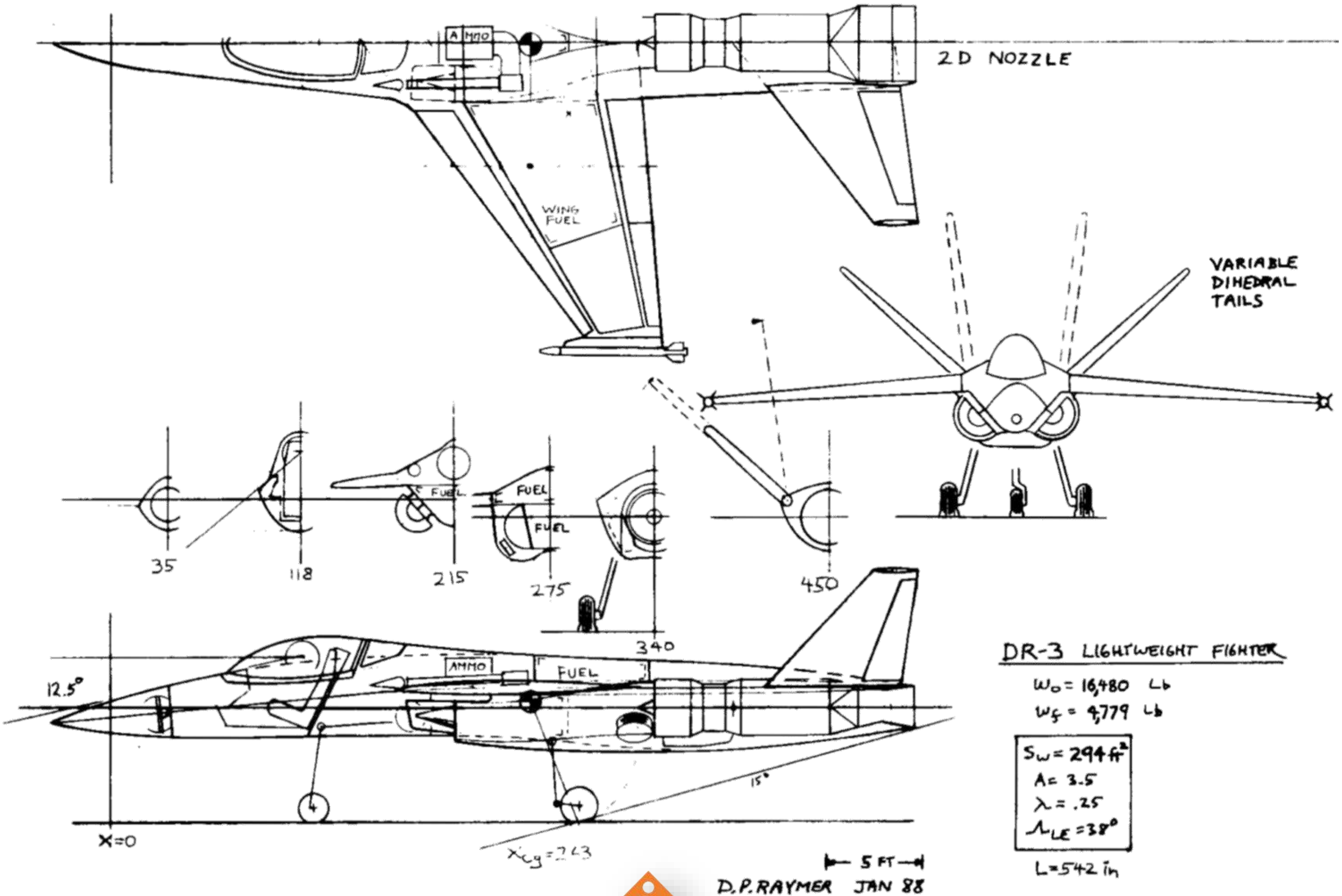
Table 11.1 Statistical tire sizing

Main wheels diameter or width (in.) $= A\ W_W^B$

	Diameter		Width	
	A	B	A	B
General aviation	1.51	0.349	0.7150	0.312
Business twin	2.69	0.251	1.170	0.216
Transport/bomber	1.63	0.315	0.1043	0.480
Jet fighter/trainer	1.59	0.302	0.0980	0.467

$W_W =$ Weight on Wheel

Conceptual Design Example



Conceptual Design Example

FUEL TANKS

$W_f = .256 \times W_0 \cong 4220$ Lb $= 94$ ft^3

Required: $W_f = .256 \times 16480 \cong 4220$ Lb

Assume: Integral wing tanks (85% Usable Volume)

Bladder fuselage tanks (83% Usable Volume)

Volumes Measured from Drawing:

Integral

WING $\begin{cases} \text{Total: } 61 \text{ ft}^3 \text{ at } X = 275 \text{ in} \\ \text{Usable: } 61 \times .85 = 52 \text{ ft}^3 = 2334 \text{ Lb} \end{cases}$

FORWARD FUSELAGE $\begin{cases} \text{Total: } 38 \text{ ft}^3 \text{ at } X = 240 \text{ in} \\ \text{Usable: } 38 \times .83 = 28 \text{ ft}^3 = 1257 \text{ Lb} \end{cases}$

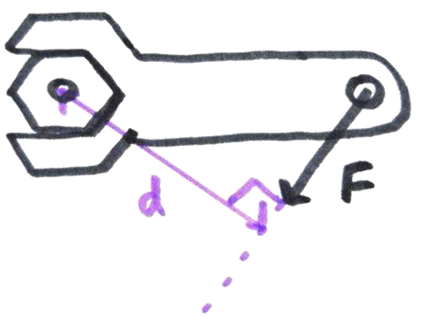
Bladder

AFT FUSELAGE $\begin{cases} \text{Total: } 32 \text{ ft}^3 \text{ at } X = 295 \text{ in} \\ \text{Usable: } 32 \times .83 = 26.5 \text{ ft}^3 = 1189 \text{ Lb} \end{cases}$

Fuel C.G. (desire near aircraft $X_{cg} = 265$ in)

Conceptual Design Example

Moment = F × d



Wing	2334	@	275 in
Fwd fus.	1257	@	240 in
Aft fus.	1189	@	295 in
Total:	4780 Lb	@	271 in

Too much!
Too far aft!

Reduce fuel in Aft fuselage tank:

Wing	2334 Lb	@	275 in
Fwd fus.	1257	@	240 in
Aft fus.	629	@	295 in
Total:	4220 Lb	@	267 in

(Good!)

Conceptual Design Example

AERO DATA FROM DRAWING

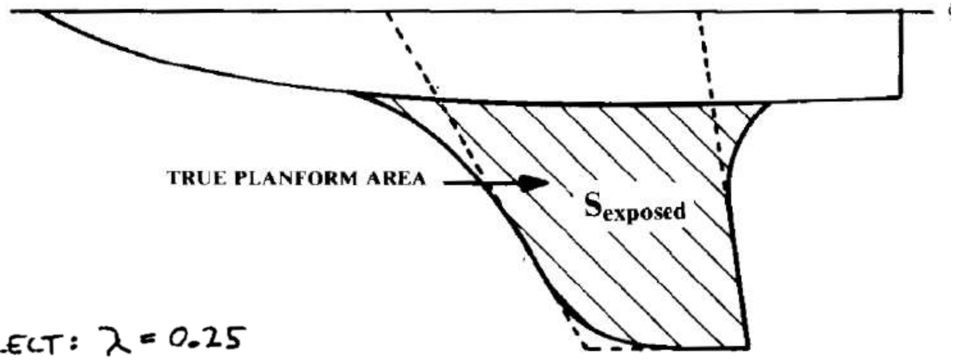
Wetted Areas:

Wing: $A_{exp} = 215\ ft^2$; $S_{wet} = 215(1.977 + (.52 \times .06)) = 432\ ft^2$

Tails: $A_{exp} = 90\ ft^2$; $S_{wet} = 90(1.977 + (.52 \times .06)) = 181\ ft^2$

If $t/c > .05$

$$S_{wet} = S_{exposed}[1.977 + 0.52(t/c)] \tag{7.11}$$



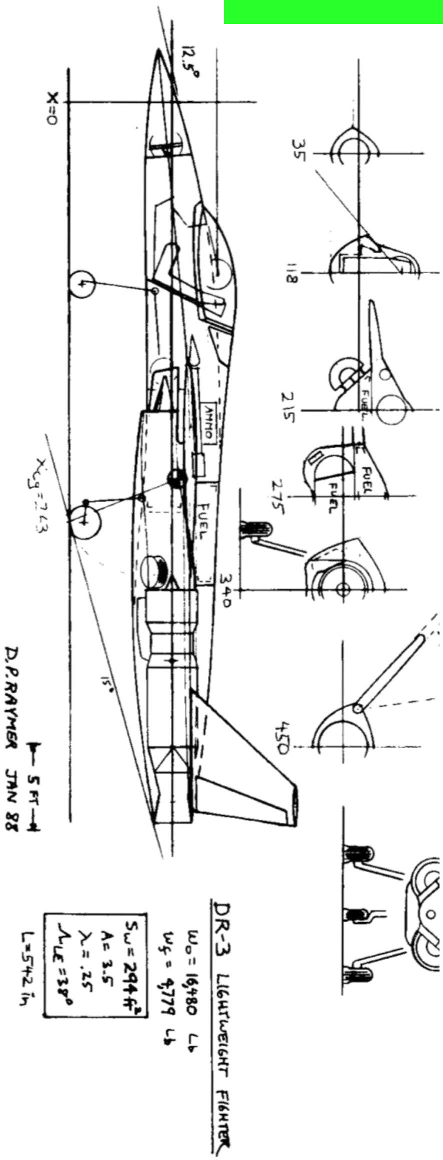
SELECT: $\lambda = 0.25$

$t/c = 6\%$

Airfoil: 64A006

Wing: $S = 16480/56 = 294\ ft^2$

Conceptual Design Example



Fuselage Wetted Area Plot:

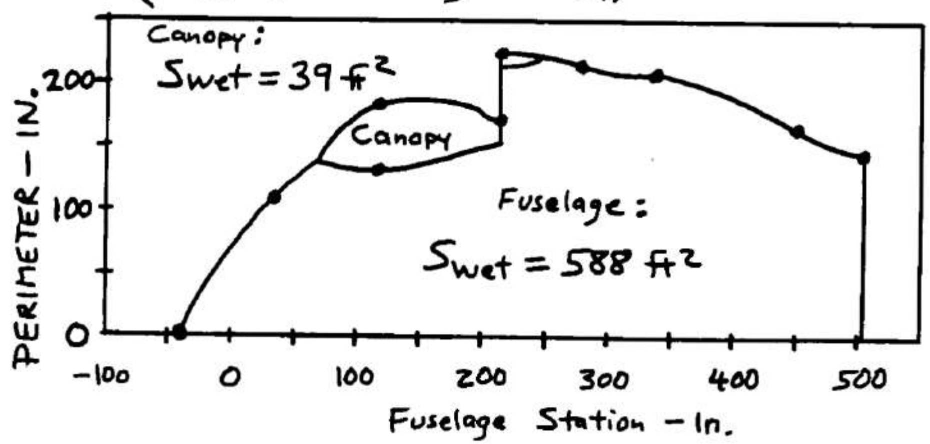
Perimeters measured from drawing:

X	Perimeter
-37	0 (in)
35	114
118	184 *
215	174 (222 with inlet)
275	216
340	212
450	164
505	150

(* 125 not including canopy)

Notes

Canopy starts at X=73
Wing starts at X=185
Canopy ends at X=240
Wing ends at X=370



Conceptual Design Example

ADDITIONAL DRAGS FOR TAKEOFF AND LANDING

FLAPS: (Only used for landing)

- Assume $\delta_{flap} = 60°$; from drawing $\left\{ \frac{\text{flap span}}{\text{wing span}} = .43 \right\}$
- Assume aileron used as flap also with $\delta_a = 30°$
  from drawing $\left\{ \frac{\text{aileron span}}{\text{wing span}} = .35 \right\}$

$$\Delta C_{D0_{\text{flap}}} = 0.0023 \frac{\text{flap span}}{\text{wing span}} \delta_{\text{flap}} \quad (12.37)$$

eq 12.37) $\Delta C_{D_0} = .0023\left((.43 \times 60) + (.35 \times 30)\right) = .0835$

Table 12.5

Conceptual Design Example

LANDING GEAR: (Using values from table 12.4)

	Frontal Area	× Table 12.4 factor	= D/q
Main wheels:	2.1 ft²	.25	.53
Main struts:	5.8 ft²	.30	1.74
Nose wheel:	.7 ft²	.25	.18
Nose strut:	3.2 ft²	.30	.96
		Subtotal	3.41

Increase 20% for interference
Increase 7% for open wheelwells } $D/q = 4.38$

$$\Delta C_{D_{o_{gear}}} = \frac{4.38}{294} = .0149$$

Table 12.5 Landing gear component drags

	D/q Frontal area (Ft²)
Regular wheel and tire	0.25
Second wheel and tire in tandem	0.15
Streamlined wheel and tire	0.18
Wheel and tire with fairing	0.13
Streamline strut (1/6<t/c<1/3)	0.05
Round strut or wire	0.30
Flat spring gear leg	1.40
Fork, bogey, irregular fitting	1.0–1.4

These values times the frontal area of the indicated component yield *D/q* values, which must be divided by the wing reference area to obtain parasite-drag coefficients. To account for mutual interference it is suggested that the

Conceptual Design Example

Weights

- Weights analysis was based upon the fighter equations in chapter 15, with adjustments for composite material usage as in table 15.4.

Table 15.4 Weights estimation "fudge factors"

Category	Weight group	Fudge factor (multiplier)
Advanced composites	Wing	0.85
	Tails	0.83
	Fuselage/nacelle	0.90
	Landing gear	0.95
	Air induction system	0.85
Braced wing	Wing	0.82
Wood fuselage	Fuselage	1.60
Steel tube fuselage	Fuselage	1.80
Flying boat hull	Fuselage	1.25

- The as-drawn takeoff weight of 16,480 lb. was used throughout, with ultimate load factor of Nz=7.33x1.5, or 11.

- Required dimensions and areas, such as the control surface area for the wing, were measured from the drawing.

- The V-tail was analyzed as a "horizontal" tail using the true area and aspect ratio of the surface (90 sq. ft., and 11).

Conceptual Design Example

- A 200 lb. weight penalty for the variable dihedral mechanism was added as a part of the miscellaneous empty weight.
- For the landing gear, it was assumed that landing weight equals takeoff weight, and gear load factor was assumed to be 4.
- For engine cooling it was assumed that a shroud covers the entire engine, so the shroud length is 14 ft.
- Engine control length was estimated from the drawing as 18.3 ft.
- To allow for the extra weight of the 2-D vectoring nozzle, an additional 400 lb. was added to the misc. empty weight.

Conceptual Design Example

- In the absence of better data, installed avionics weight of 990 lb. was guessed using table 11.6.

Table 11.6 Avionics weights

	Typical values $\frac{W_{\text{avionics}}}{W_{\text{empty}}}$
General aviation—single engine	.01–.03
Light twin	.02–.04
Turboprop transport	.02–.04
Business jet	.04–.05
Jet transport	.01–.02
Fighters	.03–.08
Bombers	.06–.08
Jet trainers	.03–.04

- The gun was assumed to always stay with the aircraft and so was treated as an addition to the misc. empty weight of 400 lbs.

- The following pages provide the complete weights assumptions and inputs, followed by the resulting summary weights statement.

- The empty weight for the as-drawn takeoff weight of 16,480 lbs. was determined to be 10947.2 lbs., somewhat above the preliminary prediction of 10788 used for initial sizing.

Conceptual Design Example

inputs

AIRCRAFT TYPE : FIGHTER/ATTACK

KEY AIRCRAFT DATA (Nz(ult) = 1.5 x Nz(design limit))

Wdg	=	16480.000
Nz (ultimate)	=	11.000
Sw	=	294.000
M	=	1.800
Nen	=	1.000

WING

X-Location	=	280.000
Kdw	=	1.000
Kvs	=	1.000
A	=	3.500
t/c	=	0.060
lambda=Ct/Cr	=	0.250
sweep c/4	=	30.000
Scsw	=	72.000
Fudge Factor	=	0.850

Conceptual Design Example

```
HORIZONTAL TAIL
     X-Location              =       470.000
     Rw                      =         4.700
     Ah                      =         6.500
     Sht                     =        90.000
     Fudge Factor            =         0.830

VERTICAL TAIL
     X-Location              =         0.000
     Krht                    =         0.000
     Ht/Hv                   =         0.000
     Svt                     =         0.000
     Lt                      =         0.000
     Sr                      =         0.000
     A(vt)                   =         0.000
     tail lambda             =         0.000
     tail sweep              =         0.000
     # vert tails            =         0.000
     Fudge Factor            =         0.000
```

Conceptual Design Example

FUSELAGE		
X-Location	=	260.000
Kdwf	=	1.000
L	=	39.000
D	=	4.000
W	=	5.400
Fudge Factor	=	0.900

LANDING GEAR		
X-Location	=	285.000
Kcb	=	1.000
Ktpg	=	1.000
Wl	=	16480.000
Nl	=	4.000
Lm (in)	=	46.000
Ln (in)	=	52.000
Nnw	=	1.000
NoseGear Loc	=	155.000
Fudge Factor	=	0.950

NACELLE		
X-Location	=	400.000
T per engine	=	16150.400
Sfw	=	52.000
Wen	=	1517.000
Kvg (inlet)	=	1.000
Ld	=	10.700
Kd	=	1.310
Ls	=	2.000
De	=	2.700
Inlet X-Loc	=	270.000
Fudge Factor	=	0.900

ENGINE INSTALLATION		
X-Location	=	400.000
Ltp	=	0.000
Lsh	=	14.000
Lec	=	18.300
Wt-Oil	=	50.000
Fudge Factor	=	0.000

Conceptual Design Example

FUEL SYSTEM

X-Location	=	267.000
Vt	=	703.300
Vi	=	389.000
Vp	=	314.300
Nt	=	3.000
SFC (max T)	=	1.900
Fudge Factor	=	0.000

CONTROLS & INSTRUMENTS

X-Location	=	260.000
Scs	=	94.000
Ns	=	4.000
Nc	=	1.000
Nci	=	1.000
Fudge Factor	=	0.000

HYDRAULICS & ELECTRICS

X-Location	=	260.000
Kvsh	=	1.000
Nu	=	10.000
Kmc	=	1.450
Rkva	=	120.000
La	=	25.000
Ngen	=	1.000
Fudge Factor	=	0.000

AIR CONDITIONING & FURNISHINGS

X-Location	=	100.000
(no inputs)	=	0.000
Fudge Factor	=	0.000

LOADS; MISC WT; & AVIONICS

Wuav	=	727.000
-Location	=	120.000
Wcrew	=	220.000
-Location	=	120.000
Wcargo	=	840.000
-Location	=	260.000
Wpassengers	=	0.000
-Location	=	0.000
Wmisc(empty)	=	1000.000
-Location	=	382.000
Wmisc(load)	=	0.000
-Location	=	0.000

Conceptual Design Example

outputs

```
------ FIGHTER/ATTACK GROUP WEIGHT STATEMENT : FILE   DR3.DWT ------
STRUCTURES GROUP :    4526.2          EQUIPMENT GROUP      :    3066.7
  Wing           :    1459.4            Flight Controls    :     655.7
  Horiz. Tail    :     280.4            Instruments        :     122.8
  Vert. Tail     :       0.0            Hydraulics         :     171.7
  Fuselage       :    1574.0            Electrical         :     713.2
  Main Lndg Gear :     631.5            Avionics           :     989.8
  Nose Lndg Gear :     171.1            Furnishings        :     217.6
  Engine Mounts  :      39.1            Air Conditioning   :     190.7
  Firewall       :      58.8            Handling Gear      :       5.3
  Engine Section :      21.0           MISC EMPTY WEIGHT   :    1000.0
  Air Induction  :     291.1          TOTAL WEIGHT EMPTY   :   10947.2

PROPULSION GROUP :    2354.3          USEFUL LOAD GROUP    :    5532.8
  Engine(s)      :    1517.0            Crew               :     220.0
  Tailpipe       :       0.0            Fuel               :    4422.8
  Engine Cooling :     172.0            Oil                :      50.0
  Oil Cooling    :      37.8            Cargo              :     840.0
  Engine Controls:      20.0            Passengers         :       0.0
  Starter        :      39.5            Misc Useful Load   :       0.0
  Fuel System    :     568.0          DESIGN GROSS WEIGHT:     16480.0

EMPTY CG= 285.3 LOADED-NO FUEL CG= 281.0 GROSS WT CG= 277.2
```

Aircraft Design
AE 405

Page: 634 in the book…
This is the end of the course AE 405…

I hope you enjoy and have knowledge…