TU Delft

Summary

AE4240 - Advanced Aircraft Design I

January 12, 2015

Contents

This is a summary including both the chapters from the book and the lecture slides. The summary is structured according to the lectures, thus section 1 refers to lecture 1 and the chapters treated during this lecture, section 2 refers to lecture 2 and the chapters treated during this lecture, etc. In each section the chapters treated will be mentioned.

1 Design objectives

(chapter 1-6, 43)

There are two steps which have to be taken into account before the design phase of an aircraft can be started.

Step one is to make a specification of the market and its operational requirements, and the economic situation.

Step two is finding a balance between the aircraft's performance and the capabilities and complexity (costs), which determines the main requirements and main objective of the aircraft.

For example; for a transport aircraft the main objective is: To transport a payload A over a distance B between airports of category C against minimum costs (i.e. at an optimum speed D). And some driving parameters can be: Lift/drag ratio, $C_{L_{max}}$, weight, safety, Stability and control, CS-25, etc.

When designing an aircraft the **certification regulations** (which describe the minimum requirements with which a design must comply in order to convince airworthiness authorities that an acceptable safety standard is achieved) and **design rules** (which are goals and relevant regulations incorporated due to a design organisations previous experience, they are not necessarily achieved completely) have to be taken into account.

The design process itself is structured into three main phases:

- Conceptual design (weeks to months)
  - Definition of the performance goals
  - evaluation of possible competing concepts
  - generation of many possible concepts
  - selection of a baseline design
- Preliminary design (months to years)
  - Refined sizing of the baseline design concept
  - Parametric studies
  - Global design frozen with the possibility to change only few details
- Detail design (years)
  - Detailed design of the whole vehicle down to each single detail
  - Accurate evaluation of performances
  - Fine tuning of the design
  - Release of drawings for production

1.1 Conceptual Design

When the top-level requirements are determined the conceptual design can be started. The conceptual design consists of the Class I and the Class II design. For the class I design first the aerodynamic shape is determined using three design methods:

- **Empirical method** Handbooks which are a collection of graphs and equations - give a relation between elementary parameters of the geometry of the aircraft and the desired characteristics of the aircraft.
- **Analytical method** - The aerodynamic characteristics of the detailed geometry are obtained by physical insight (pressure distributions).
- **Computational fluid dynamics** - obtain the intended characteristics of the aircraft by directly determining the required detailed aerodynamic shape through the use of fluid dynamics and the associated pressure distribution.

During the conceptual design a balance has to be found between required volumes, weight distribution, main dimensions, and engine performance without being too optimistic about those parts, since being too optimistic can cause high costs because the design has to be changed. Therefore, the designer should have detailed knowledge about (almost) every aspect of the design.

Usually a thrust to weight versus wing loading diagram is made which determines the design point. In this diagram the take-off, climb, cruise, descent, landing, and stall conditions are taken into account. From this design point to find an external shape that optimizes the lift and drag characteristics attention has to be paid to the tailplane, fin, and control surfaces to get satisfactory flight handling characteristics.

The conceptual design finally gives requirements on geometry (volume and configuration), weight, aerodynamics, handling characteristics, and cost (lift cycle and operating) which have to be used in the preliminary design.

1.2 Preliminary design

During the preliminary design the aircraft geometry is determined. This leads to an aerodynamic design loop. The aerodynamic design loop consists of the airplane characteristics, which determine the external shape which determines the aerodynamic characteristics which again determines the airplane characteristics.

Taken into account are the aerodynamic design goals, which are to minimize drag and postpone separation.

The aircraft geometry is dependent on:

- Internal geometry constraints
  - Required volume (cargo, seats, legroom, headroom, in-flight entertainment)
  - Shape of the fuselage (usefulness of the volume)
  - Access to the available volume
- Internal geometry constraints related to structural consideration
  - thickness distribution along the span of the wing and tail surface torsion boxes
  - Thickness of flaps
  - Thickness of engine struts and pylons
- External geometry w.r.t. aerodynamic considerations
- External geometry w.r.t. Manufacturability and cost control

The external geometry should lead to optimum aircraft performance, satisfactory flight handling characteristics while fulfilling the requirements on the internal geometry and producibility.

The crucial question in aerodynamic design is: **What is the proper design (=target) pressure distribution?**

In order to postpone flow separation and minimize drag the following goals should be kept in mind:

- for components that do not have to produce resultant forces, local super velocities should be minimised

- for components that do need to produce resultant forces (wing, rudder), the pressure distribution at the relevant flight conditions should be optimised such that the momentum loss in the boundary layer and behind the shock wave is minimal.
- for components that must tolerate a large variation in local flow direction, leading-edge shapes and design pressure distributions must be found, which cope with this variation.

2 Boundary layer effects and subsonic cruise drag

(chapter 7, 8, 9, and 40)

Flow in real life displays viscosity (friction)

- The flow "sticks" to the surface
- there is a friction acting on the surface (wall-shear stress)
- Development of a boundary layer along the airfoil surface

Due to the boundary layer, skin-friction drag and pressure drag have to be taken into account. Also, due to the boundary layer, flow transition (from laminar to turbulent) and flow separation is occurring.

While the boundary layer increases, the slope of the velocity distribution becomes more shallow. This is directly related to the shear stress, which ties to the friction coefficient, and hence the friction coefficient decreases over the slope of a flat plate.

The shape factor is the factor necessary to scale from a flat plate to a certain body. The shape factor (H) is dependent on both the displacement thickness (δ^*) and the momentum thickness (θ):

$$\delta^* \cdot U \cdot \rho = \rho \int_{y=0}^{y=\infty} (U - u) dy \tag{1}$$

$$\theta \cdot U \cdot \rho \cdot U = \rho \int_{y=0}^{y=\infty} u(U - u) dy \tag{2}$$

$$H = \frac{\theta}{\delta^*} \tag{3}$$

Flow separation occurs when the slope of the velocity distribution at the wall is zero and becomes negative. This is often due to a fast increasing pressure distribution.

Boundary layer transition depends on the Reynolds number, pressure gradient and receptivity, where a laminar boundary layer separates earlier than a turbulent boundary layer because a turbulent boundary layer has extra momentum near the wall, which makes it withstand an unfavourable pressure gradient without separating.

2.1 Subsonic cruise drag

(chapter 40)

Cruise drag is a major driver to fuel consumption and determines the range and economics for the operator. When the cruise drag is predicted too low the required range cannot be reached, but if the cruise drag is predicted too high there is no advantage relative to the aircraft in service.

Each component produces drag, this drag consists of:

- Zero lift drag

- Friction drag - The friction drag of a smooth flat plate with the same length as the component is determined. The shape factor then determines the ratio between this drag and the actual drag of the component.

  Contributors to the friction drag are:

  * Wetted area
  * Surface roughness/imperfections
  * Shape (local supervelocities and pressure drag components)
- Pressure drag
- Excrescences
- Interference drag

- Lift dependent drag - consists of form drag and induced drag. Where the form drag consists of friction drag and pressure drag.
- Compressibility drag
- Trim drag

2.2 Friction drag

Friction drag is the largest drag component in climb/cruise. However, the friction coefficient decreases with Reynolds number, therefore larger aircraft have lower friction drag coefficients. This is only up to a certain Reynolds number, after this critical Reynolds number the friction drag stays the same.

This is because: with increasing Reynolds number the boundary layer decreases. At this point their additional drag is friction drag, which stays the same, and therefore, since the boundary layer decreases, the drag decreases. When the boundary layer becomes thin enough that the grains of the material begin to protrude from the boundary layer, a wake is produced, after which this wake will increase while the boundary layer decreases, keeping the total drag measured constant. The critical Reynolds number depends on the relative equivalent sand-roughness height (k_s/l). Since the cruise Reynolds number is often higher than the critical Reynolds number, reducing the grain size leads to reduced drag.

The equivalent skin friction drag is determined by the wetted area (often called parasite drag or equivalent flat plate area).

2.3 Drag prediction

The drag of an aircraft can be predicted using the following methods:

- Form factor and flat plate analysis (for zero-lift drag)
- Trim drag calculated from moment distribution
- Bookkeeping gives thrust/zero-lift drag/induced drag

CFD methods do not always give better results than classical methods. When wind-tunnel testing is an option to determine shape factors which then don't have to be estimated.

3 Relation between geometry and pressure distribution

(chapter 7, 10, and 11)

If no (lift) forces should be generated by an aircraft part, shock waves, separation and friction drag should be reduced or eliminated completely.

If (lift) forces have to be generated, the pressure distribution has to be optimized and the overall characteristics of the aircraft have to be determined ($C_L - \alpha$, $C_L - C_M$, etc.).

C_p and ΔV are linked by the following equation:

$$C_p = -2\frac{\Delta V}{V_\infty} \tag{4}$$

This is related to geometry by the following equation:

$$\frac{\rho V^2}{r} = \frac{dp}{dn} \tag{5}$$

For a concave wall: $\frac{dp}{dn} < 0$, $C_p > 0$, and $\Delta V < 0$.

For a convex wall $\frac{dp}{dn} > 0$, $C_p < 0$, and $\Delta V > 0$.

However, these relations do not hold at the leading and the trailing edge. Besides, the effect of compressibility has not been accounted for.

For the flow around transonic airfoils, the following relation holds (however, this relation does not take entropy increase due to shock-waves into account):

$$M_{local}^2 = \frac{2}{\gamma - 1}\left[\frac{1 + \frac{\gamma-1}{2}M^2}{\left(1 + \frac{1}{2}\gamma M^2 c_p\right)^{\frac{\gamma-1}{\gamma}}} - 1\right] \tag{6}$$

Away from the stagnation areas a direct relationship exists between surface curvature and pressure coefficient. The stronger the (convex) curvature, the higher the (positive) supervelocities and the (negative) C_p-values.

Hence, local shape changes affect the pressure coefficient and the effect on the pressure coefficient depends on the Mach number of the flow. Gradual changes in shape result in a smoother pressure distribution reducing friction drag and/or eliminating separation. Finally, the Mach number has an effect on the region of influence of a body, the pressure coefficient over the body, the sensitivity of the forces with respect to the inflow angle, the drag of the body, the effectiveness of control surfaces, and the handling characteristics of the aircraft.

4 Interference effects and area ruling

(chapter 12)

4.1 Area rule

In order to reduce the drag of an aircraft the area rule is used. The area rule states that two airplanes with the same longitudinal cross-sectional area distribution have the same wave drag, independent of how the area is distributed laterally (i.e. in the fuselage or in the wing). Furthermore, to avoid the formation of strong shock waves, this total area distribution must be smooth. As a result, aircraft have to be carefully arranged such that at the location of the wing, the fuselage is narrowed or "waisted", such that the total area doesn't change much. Similar but less pronounced fuselage waisting is used at the location of a bubble canopy and perhaps the tail surfaces.

Also, according to the area rule the wave drag produced by a slender body of arbitrary dimensions in supersonic flow may be modelled by an equivalent body that is axi-symmetric if:

- The body ends with an axi-symmetric portion
- The body ends in a point
- The body ends in a cylindrical portion parallel to the free stream

A related concept is the Sears-Haack body, which is the shape with the minimum wave drag for a given length and a given volume. However, the Sears-Haack body shape is derived starting with the Prandtl-Glauert equation, which governs small disturbance supersonic flows, but this equation is not valid for transonic flows where the area rule applies. So although the Sears-Haack body shape, being smooth, will have favourable wave drag properties according to the area rule, it is not the theoretically optimum.

Most jet airliners have a cruising speed between Mach 0.8 and 0.85. For aircraft operating in the transonic regime (Mach 0.8-1.0), wave drag can be minimized by having a cross-sectional area which changes smoothly along the length of the aircraft. This is known as the area rule, and is the operating principle behind the design of anti-shock bodies. Reducing wave drag improves fuel economy.

Wave drag is only dependent on cross-sectional area distribution. It is independent of the actual shape of components. And it is dependent on the second derivative of area; hence small changes in area gradient yield low wave drag and large changes in area gradient yield high wave drag. However, shortcomings are that no lift is considered, the flow is assumed irrotational (no shockwaves are present) and no viscous effects are considered.

An anti-shock body, (also known as Whitcomb body or Küchemann carrot) is a pod positioned on the leading edge or trailing edge of an aircraft's aerodynamic surfaces to reduce wave drag at transonic speeds (Mach 0.8âĂŞ1.0). They promote isentropic compression of the supersonic flow and postpone shock-induced separation

4.2 Interference drag

In order to optimize an aircraft body the interference between parts has to be minimized. When having a locally convex curvature the pressure decreases while a locally concave curvature increases the pressure. This also means that higher local convex curvatures lead to higher supervelocities.

On aircraft parts which are not intended to generate aerodynamic forces local supervelocities should be minimized:

- Front fuselage including cockpit canopies
- Centre and rear fuselage sections
- Engine struts and pylons
- Fins in cruise flight
- Tailplane-fin fairings
- Wing-body interference

interference issues are:

- Every curvature generates supervelocities
- Multiple surfaces close together (nacelle, pylon, wing/body/VT)
- Summation of supervelocities on each component
- High supervelocities can cause interference drag
- In high-subsonic conditions interference drag is more pronounced than in low subsonic conditions

Using CFD to change the shape of components, the local supervelocities can be minimized.

4.2.1 Wing-body interference

When the wing and the body are considered together, supervelocities of the individual components are added. The lift over the wing is increased due to the presence of the fuselage. Due to the presence of supervelocities the lift distribution over the wing is altered.

4.2.2 Empennage interference

The horizontal tail produces a down-force during cruise to balance the aircraft. The vertical tail has to minimize drag during cruise. Interference between those two can lead to lower pressure on the top surface, reduced down force and a nose-down pitching moment.

A solution of this can be to allow the flow to expand by waisting the fuselage (hence local area ruling).

A way to reduce T-tail interference is to add a fairing between the horizontal tail and the vertical tail.

4.2.3 Nacelle interference

Constraints are:

- Attached inflow in all conditions
- Structural rigidity of pylon and nacelle
- Sufficient space for structure and systems

Inboard supervelocities of the nacelle are always higher than outboard supervelocities.

Nacelle-wing interference can be reduced by placing the nacelle at a certain position from the wing. By using CFD techniques it has become possible to place the nacelle closer to the wing, even to previously thought to be unacceptable positions.

5 Pressure distribution about airfoils

(chapter 15)

The characteristics of a pressure distribution are:

- A stagnation point at or near the leading edge ($C_p = 1$)
- The height and location of the maximum supervelocity ($C_{p_{min}}$)
- The ratio between $C_{p_{min}}$ and C_p^*
- The pressure gradient $\frac{dp}{dx}$ behind $C_{p_{min}}$
- The trailing edge pressure (for inviscid flow $C_{p_{TE}} \leq 1$)

To determine the overall characteristics of the pressure distribution over an airfoil section, the characteristics of the boundary layer have to be taken into account (δ^*, θ, and c_f).

The velocity gradient is linked to the friction coefficient, a lower velocity gradient means a lower friction coefficient.

When comparing pressure distribution over airfoils, the following can be noted:

- At $C_l = 0$ ($\alpha = 0$) $C_{p_{min}}$ is proportional to the sections relative thickness

- With increasing α, the leading edge suction peak increases much faster and is followed by a much stronger adverse pressure gradient for the airfoil section with the small leading edge radius than the section with the large leading edge radius.
- On the thick section, boundary layer effects are much stronger than on the thin section, and flow separation is approached much quicker
- Increasing Reynolds number causes boundary layer effects to decrease.

Lift on an airfoil next to each other can be increased in two ways:

1. Increasing camber by control surface deflection - the pressure distribution over that control surface will change (this is more effective on thin airfoil sections because at thick sections this can lead to trailing edge flow separation).
2. Increasing the angle of attack

In order to avoid drag creep, but have an as thick as possible airfoil, a sonic rooftop shape can be selected. This is characterized by a constant $C_p = C_p^*$ from the leading edge up to a certain point.

When the freestream Mach number increases the shock moves downstream.

A stream through a nozzle has the following characteristics:

- If $A_t \leq A^*$ the flow is "chocked" and sonic at the throat
- Increasing back pressure reduces M_e to subsonic
- Shock wave initially at the exit
- Large Mach number in front of shock wave
- Large pressure jump over shock wave
- Increasing back pressure reduces M_e even more
- Shock position moves closer towards the throat
- Mach number in front of shock reduces
- Pressure jump across shock reduces

Lower M in front of shock means the shock is pulled forward
Higher M in front of shock means the shock pushed back

The position of the shock is dominated by:

- Local area ratio
- Ratio between back pressure and inlet pressure of stagnant air

The result is a relation between geometry, Mach number in front of the shock, pressure jump and Mach number behind the shock.

The NACA 6-series is developed for pressure distributions which are favourable for developing a laminar boundary layer. However, as soon as the speed of sound is surpassed locally this laminar boundary layer has unfavourable characteristics. This is because the supervelocities reach a maximum near mid-chord. A small increase in Mach number then produces a sharp increase in supersonic local velocity, resulting in a strong shock wave and flow separation at the foot of the shock.

Therefore the NACA 4-series was modified such that in sub-critical flow:

- conditions the highest supervelocities were concentrated near the leading-edge (suction peak)
- followed by an area with almost constant pressure
- rapid deceleration behind the pressure peak

- Behind the leading edge curvature distribution has to be such that in transonic conditions the interactions between expansion waves, the sonic line and the reflected compression waves occur behind the suction peak, leading to a weak shockwave at the end of the supersonic region.

6 Reynolds effects

(Chapter 15 and 18)

6.1 Supercritical airfoils

Characteristics of supercritical airfoils are:

- Small drag penalty at design Mach number
- High lift coefficient
- Sensitive to Reynolds number and Mach variations

For supercritical airfoils first the upper surface is designed after which the lower surface is designed to modify spanwise lift and moment distribution.

For supercritical airfoils the following is valid:

- The leading edge can be modified to reduce drag creep and postpone the divergence drag. This is due to the partial isentropic recompression on the upper surface behind the leading-edge suction peak and the weaker shock wave terminating the area of supersonic flow. Hereby a small radius means less drag creep and a large radius more drag creep.
- The cusp at the trailing edge allows off the surface pressure recovery, hence a less steep adverse pressure gradient. If a sharp or a blunt trailing edge is chosen depends on the design Mach number, for high Mach numbers the drag will be lower for a blunt trailing edge, and for low Mach numbers the drag will be lower for a sharp trailing edge.
- The drag divergence Mach number is dependent on the thickness ratio of the airfoil. High thicknesses mean lower drag divergence Mach numbers.
- The sonic rooftop prevents drag creep and postpones divergence drag. The design is such that $C_p = C_p^*$ from leading edge to around 0.3-0.6 x/c. This keeps the maximum local velocity around $M = 1$ and thus no mixed subsonic/supersonic flow will occur.

Relations between airfoil shape, pressure distribution, lift coefficient, and design Mach number are:

- When adding thickness to the airfoil at constant camber, higher local velocities will occur due to the stronger surface curvature. For a given lift coefficient this will lead to a lower design or drag divergence Mach number, or for an increase in design Mach number this will lead to a thinner section.
- adding thickness only to the lower part of the airfoil results in an increase in supervelocities on the lower surface. This may only marginally decrease the lift coefficient in the design condition but may appreciably lower the drag divergence Mach number at lower lift coefficients
- Decreasing the aft loading by thickening the rear part of the airfoil results in a lower lift coefficient at the design Mach number. A positive effect his modification leads to a less negative zero-lift pitching moment coefficient.

6.2 Reynold number effects

For supercritical airfoil sections operating near their design point the upper surface has a region of supersonic flow coupled with an almost constant static pressure. The Reynolds effects are much stronger than for subsonic flow, this has two causes:

1. The boundary layer may be laminar up to the shock wave at low Reynolds numbers. The shock wave terminating the supersonic region will then be a lambda shock wave which the boundary layer can negotiate much easier than a straight shock wave associated with a turbulent boundary layer. With increasing Reynolds number transition will occur progressively further ahead of the shock wave.
2. Increasing the Reynolds number when the boundary layer is turbulent will decrease the displacement thickness and therefore effectively increase the flow curvature near the section surface. This will produce a more aft shock wave position at constant Mach number and angle of attack. This again may cause an increase in shock wave strength and even a second shock wave.

The effects of the Reynolds number increase on an airfoil with turbulent boundary layer are:

- Aft moving shock wave, leading to an increase in lift coefficient at constant angle of attack due to the larger area of supersonic flow
- Before flow separation occurs the linear part of the pitching moment versus lift coefficient curve increases with increasing Reynolds number due to the rear movement of the shock, leading to a more negative pitching moment coefficient at high Reynolds numbers
- The drag rise number increases
- Buffet is postponed

There are two types of separation: Shock induced separation bubble (low dependence on Reynolds number), and rear separation (strong dependence on Reynolds number).

At high Mach numbers the Reynolds number has an increasing effect on the pressure distribution of the airfoil. Also the lift and moment coefficients rise with increasing Mach number.

Increasing Reynold number decreases the displacement thickness of the turbulent boundary layer, leading to:

- effective increase in curvature
- Higher supervelocities
- More aft position of the shockwave
- May cause increase in shock strength or second shock

7 Low speed and high speed stall

(Chapter 17 and 19)

When the flow separates a plateau is visible in the pressure distribution.

There are three types of stall

1. Leading edge stall - characteristics are:
    - Abrupt stall
    - Flow separation over the entire airfoil
    - Present on airfoils with moderate leading-edge radii
    - Leading edge radius has significant effect on maximum lift coefficient

2. Trailing edge stall - characteristics are:
    - Gradual stall
    - Flow separation moves forward with angle of attack
    - occurs on airfoils with large leading edge radius and strong upper surface curvature
3. Thin-airfoil stall - characteristics are:
    - Gradual stall at low angle of attack
    - Occurs of airfoils with a sharp leading edge or low Reynolds numbers
    - Development of a laminar separation bubble - this lowers the suction peak and thus gives a lower adverse pressure gradient.

From the pressure distribution it can be concluded if the flow is separated or not. If the flow is separated the pressure coefficient at the trailing edge is negative. The point of separation is when the pressure distribution shows a plateau (or a straight distribution curve).

7.1 Maximum lift

Up until a Reynolds number of 10 million an increase in Reynolds number for a constant Mach number means a thinner relative boundary layer, which tolerates larger unfavourable pressure gradients, increasing the maximum lift.

The Reynolds number also decreases the minimum pressure coefficient until the critical pressure coefficient, after this value shock waves will occur and the minimum pressure coefficient will increase again. This is because of:

- Larger Re_c increases tolerance to strong adverse pressure gradient behind pressure peak
- Airfoil can achieve higher angles of attack without leading edge stall
- The result of this is that higher lift coefficients can be achieved
- Increasing Re_c moves the trailing edge separation point upstream
- The result is that the lift coefficient decreases

Up until the Mach number where the flow at the leading edge reaches a local Mach number of 1 the maximum lift coefficient and minimum pressure coefficient are independent of Mach number.

Hence, for $M_{loc} < 1$ the Reynolds number is limiting for the maximum lift coefficient, for $M_{loc} \leq 1$ the Mach number is limiting for the maximum lift coefficient.

7.2 thickness and camber

Increasing the leading edge radius leads to decreased overspeeds, which reduces the adverse pressure gradient and thus postpones leading-edge separation.

When an airfoil section has a sharp leading edge with a small leading edge radius the flow has, at large angles of attack, to deal with a strong curvature leading to high velocities. When the leading-edge radius is increased without altering the upper surface curvature distribution the peak velocities can, at the same angle of attack, be lowered, due to less severe curvature, without affecting the high-speed characteristics. Consequently the angle of attack can be increased to higher values than the basic section before the suction peak collapses, leading to a higher maximum lift coefficient.

The shape of the lower surface section has no effect on the maximum lift coefficient.

7.3 Buffet onset

Buffet is a form of airframe vibration caused by pressure fluctuations in separated flow felt by the occupants in the cockpit and cabin. Buffet can appear in different forms:

- Low speed buffet due to flow separation close to stall
- Buffet due to lift dumper or speed-brake extensions
- Buffet due to local flow separation
- High speed buffet due to flow separation caused by shock waves

High speed buffet occurs when a separation bubble, which starts at the foot of the shock wave and gradually increases rearwards, reaches the trailing edge or when the boundary layer separates near the trailing edge before the shock wave becomes strong enough to cause separation at its foot.

Trailing edge pressure divergence is used in wind tunnel tests as an indication of buffet onset on the real aircraft.

High speed buffet develops gradually, in particular with increasing angle of attack. If this flow regime is penetrated too far it may result in structural damage.

There is a margin between normal flight conditions and buffet onset both with regard to speed and to angle of attack because:

- Manoeuvre in cruise flight
- Deal with disturbances due to turbulence, be it in speed or in normal load factor
- Deal with upsets due to aircraft system failures

Buffet onset occurs when when the local Mach number in front of the shockwave reaches a given value ($M_{loc} = 1.35 - 1.50$).

The buffet determines:

- Maximum CL at certain M
- Maximum M at certain CL
- ceiling at which the aircraft can fly at a certain M

8 Airfoil with high-lift devices

(chapter 25)

The high lift devices are installed to:

- Increases the maximum lift coefficient for landing with acceptable drag penalty
  - Reduces the stall speed of the aircraft
  - Allows for shorter landing length
- Increases the maximum lift coefficient with acceptable drag penalty during take-off
  - Reduces the minimum unstick speed
  - Reduces the take-off field length
- Reduce the pitch angle at low speeds (for trailing-edge devices)

The result is that the wing sizing is done based on:

- Cruise performance
- Climb performance
- Field performance
- Volume considerations

Three effects determine the increase in maximum lift by the use of high-lift devices:

- increase in camber
  - Higher camber produces more lift but also more drag (hence variable camber = high lift devices)
  - More camber leads to more circulation at the trailing edge and thus in more lift at constant angle of attack. (where the angle of attack is the angle between the chord line of the basic section and the direction of the undisturbed flow)
- Increase in effective chord/effective wing area
- Mutual interaction effect

This is for a continuous camber line, if the various components are decoupled and moved a small distance away from each other, the lift at the leading and trailing edge of each part will be zero. However, the mutual interaction of the vortices will remain and the total chordwise lift distribution over the compound airfoil section will be similar that on a single (highly curved) section.

The insertion of slats gives the following characteristics:

- Suppress the pressure peaks on the proceeding components through mutual interference.
- Create a new boundary layer on each component postponing stall
- Generate additional drag

8.1 mutual interference

Each lifting airfoil can be represented by a vortex.

The main body (centre wing) experiences downwash of the preceding vortex (slat) and upwash of the succeeding vortices (flaps), this leads to a decrease of lift and suction peak on the front part and increase of lift on the rear part. Hence the slat, which experiences only upwash, has a high increase in lift. And thus the last flap component has very little lift (however, the effect on the preceding components is large, increasing the overall lift).

The upwash creates a velocity vector on the succeeding vortex. Due to the vertical component of this velocity vector the velocities on the lower side of the airfoil are reduced and the lift is increased. Due to this upwash effect on the lower side the stagnation point shifts either aft to the lower side of the airfoil, resulting in a higher pressure peak, higher supervelocities on the upper side, and lower supervelocities on the lower side. Or aft on the upper side, resulting in a lower pressure peak, lower supervelocities on the upper side, and higher supervelocities on the lower side.

The mutual interaction of the vortices of the individual components ensures that the total chordwise lift distribution over the compound airfoil section is similar to that on a single, highly curved section. The gaps between the individual components postpone separation to higher angles of attack and are therefore highly desirable.

By separating each component, every component will form its own boundary layer resulting in wake. This new boundary layer postpones separation. The wake of the elements grow if:

$$\frac{1}{1-C_p}\frac{dC_p}{dx} > \frac{0.007}{\delta^*} \tag{7}$$

In practice this usually means that no wake instabilities occur before the boundary layer of the underlying component separates.

Due to earlier separation merged boundary layers should be avoided.

The combined effects on each component of being placed in a flow field with upwash or downwash, the smaller chord length and the dumping velocity make that compound airfoil sections can reach much higher $C_{L_{max}}$ than single airfoil sections.

The highest $C_{L_{max}}$ is reached when the wake of the preceding component and the boundary layer of the succeeding component do not merge. However, if merging occurs the lift-drag ratio improves.

8.2 stall

Types of high-lift device separation:

- Trailing edge of the flap at small angle of attack
- Trailing edge separation just in front of the flap
- Leading edge separation - abrupt stall.

Flow separation when slats are added:

- stall on the slat
- Trailing edge separation on the main wing
- leading edge of the main wing

8.3 design for stall characteristics

The most effective high-lift configuration is one where separation is reached on all components simultaneously over the full span of the wing. On an aircraft this may produce unacceptable flying characteristics.

In order to find the highest maximum lift, the largest flap setting should be selected with or without only limited flow separation on the flap at low angles of attack. The slat however should be set at such an angle that trailing edge separation occurs near the trailing edge on the main component, just prior to flow separation on the slat.

At high Reynolds numbers on modern airfoil sections with effective high-lift devices this limiting peak Mach number occurs at low free-stream Mach numbers, which lie in the range of Mach numbers where on actual aircraft the maximum lift coefficient is determined. The maximum lift coefficient becomes then a function of aircraft weight and flight altitude.

Reynolds and Mach effects:

- High Reynolds numbers allow high $C_{P_{min}}$ values
- A high $C_{P_{min}}$ implies high supervelocities over the leading edge
- Combined with high take-off and landing speeds this may lead to local supersonic flow
- As a general rule $M_{loc} < 1.58$, which limits $C_{L_{max}}$
- Minimum air pressure may then be less than 30% of ambient
- This result can be properly translated to 3D wings.

9 Swept wing concept

(Chapter 20)

9.1 Wing design requirements

Wing design requirements are:

- At cruise Mach number and cruise altitude
  - Design lift coefficient at top of climb
  - Lift-to-Drag ratio at design lift coefficient
  - Pitching moment coefficient at design lift coefficient
  - Pitching moment distribution with span (to limit structural warping)
- At low speed conditions
  - $C_{L_{max}}$ for all aircraft configurations (clean, take-off, landing)
  - Stalling characteristics for all aircraft configurations over the complete c.g. range (pitch and roll)
  - Maximum lift-to-drag ratio (to meet climb requirement)
  - Lift curve slope and maximum angle of attack in ground effect (for rotation requirement)
- Stability and control (everywhere in the flight envelope)
  - Pitching moment due to angle of attack
  - rolling moment due to sideslip
  - Rolling moment due to aileron deflection/spoiler deflection
- Around the boundaries of cruise flight conditions
  - Margin (ΔC_L and ΔM) between design point and buffet onset boundary
  - Acceptable stability and control (pitch and roll) between buffet onset and maximum buffet penetration boundary (ΔC_L and ΔM)
  - Acceptable stability and control between M_{MO} and M_D.
- Structural constraints
  - Wing weight
  - Tank volume
  - Landing gear volume
  - Space for systems: kinematic systems for wing movables, anti-icing systems, fuel systems, hydraulic system, electric system
  - Lightning strike protection
  - Fixation points for: engines

The prime characteristics of wing design are:

1. Aerodynamic: M_{design} and $C_{L_{design}}$.
2. Geometric: Aspect ratio (A), taper ratio (λ), twist distribution (θ), sweep angle (Λ), and airfoil section in the outboard wing.

9.2 Wing sweep

At the start of aviation, swept wings were not in use, mainly due to not large enough wind tunnels to do correct tests. The first swept wing was implemented not to reduce drag, but to account for a miscalculation in the centre of gravity position.

The main reason for applying sweep is to increase the drag-divergence Mach number. Wing sweep also affects other aerodynamic parameters such as lift curve slope.

Swept-wing aircraft will have a lower maximum lift coefficient and will therefore have to pay more attention to low speed performance and often require high-lift devices.

Wing sweep is introduced because it decreases the effective curvature, which leads to lower overspeeds, which makes that strong shock waves occur at higher Mach numbers, which postpones the drag divergence to higher Mach numbers.

Wing sweep influences the flow over the wing in the following ways:

- Near the wing root the pressure distribution deviates increasingly from the prediction according to simple sweep theory
- The increasing supervelocities over the aft part of the wing root region may lead to the formation of shock waves and flow separation.
- These may then spread further outboard and lead to a sharp drag rise much earlier than would be expected from simple sweep theory
- At $M = 0.95$, sweep theory looses meaning.

At low speed the sweep lowers maximum section lift coefficient of the outboard wing/tip, this leads to a reduction of the wing maximum lift coefficient, therefore effective high-lift devices are required for landing and take-off.

From the relation between the effective and the streamwise lift coefficients it can be concluded that the lift coefficient based on the free-stream velocity will be lower than the section CL if wing sweep is applied. This means that high-speed aircraft with highly swept wings require more attention to their performance in the low-speed regime than aircraft with straight wings because their maximum lift coefficient is lower.

As a consequence a swept-wing aircraft will often require powerful high-lift devices in order to show satisfactory take-off and landing performance for a given wing area.

Flow from infinity will initially move outboard at the stagnation point. Following the contour along the top surface, the flow will experience an acceleration inboard which reduces along the chord, this bends the streamlines inboard. This is due to an increase in effective velocity on the upper side.

On the lower side the effective velocity decreases, this curves the streamlines outboard. For a wing with forward sweep the situation is reversed.

Inside the boundary layer the kinetic energy of the air particles is reduced towards the surface, but the pressure gradient remains the same. The boundary layer air will therefore tend to move the lower pressure regions, = perpendicular to the isobars. On an aft swept wing this will force the boundary layer outboard. Hence the boundary layer on the inboard wing is thinner, and on the outboard wing thicker than on the comparable straight wing. This suggests unfavourable aerodynamic characteristics because flow near the tip would separate earlier than expected based on the airfoil section characteristics.

At the wing the pressures are higher over the forward part of the section and lower over the rear part. The reverse occurs at the wing tip, this effects becomes stronger with increasing Mach number.

10 Tip stall and aeroelastic effects

(chapter 20 and 21)

Swept wings sometimes have fences in order to:

- Align the streamlines in high-speed conditions
- Prevent tip stall by locally renewing the boundary layer
- Postpone stall through mutual beneficial interference

The oldest device for preventing tip stall is a vertical plate fitted on the wing upper surface in a streamwise direction, thus forming a physical barrier for the boundary-layer crossflow, the full-chord fence.

A fence placed on the wing provides a physical barrier for the boundary layer crossflow and creates an increase in supervelocities inboard and a decrease outboard of the fence thus altering the shape of the isobars and causing early flow separation inboard of the fence.

high supervelocities can lead to premature separation inboard of the fence. Optimizing the shape and spanwise position of a fence is done experimentally. No sufficiently accurate computer models exist.

The lift coefficient at which the pitch-up tendency starts (and thus the maximum usable lift coefficient) is increased considerably over the complete Mach number range investigated.

A disadvantage of wing fences is the increase in drag. Although their surface area may be small they may produce as much as 20 drag counts. For this reason on modern aircraft wing fences are only applied when at a late stage in the development or during flight testing stalling characteristics are found to be unsatisfactory.

Instead of fences also "shark" or "dog teeth" (which are local leading edge extensions), "saw cuts", or leading edge boundary layer fences can be placed. These create a streamwise vortex at high angles of attack such that the boundary-layer cross flow on the inboard wing is swept inboard.

In order to tailor the stalling characteristics the leading edge radius is increased without changing the upper surface, this as not to change the high-speed characteristics.

The effects of wing sweep are:

- Increases drag divergence Mach number
- Reduces maximum lift coefficient
- Reduces lift-curve slope
- Shifts aerodynamic centre and centre of pressure aft

The result of this is:

- Fancy high-lift devices required to increase maximum lift coefficient
- Geometrical modifications to prevent tip stall required

10.1 first generation swept wing aircraft

The first swept wings were designed along the same principles as straight wings, this led to unsatisfactory stalling characteristics, and pitch-up tendencies both at high and low speeds due to tip stall.

10.2 Aeroelastic effects

In a swept wing is the main load-carrying component the torsion box consisting of front and rear spar, ribs and upper and lower skin panels. In flight the lift forces will, partly counteracted by the weight of the wing itself, the fuel, and the engines make the wing flex upward.

The torsional deformation about the aeroelastic axis is limited.

When ailerons are deflected the extra lift applies behind the aeroelastic axis of the torsion box producing a torsion moment. A downward aileron deflection leads to a leading-edge-down torsion moment twisting the torsion box an producing a decrease in local angle of attack. The resulting change in lift on the outer wing counteracts the lift due to aileron deflection lowering the aileron efficiency. At high dynamic pressures the structural deformation may lead to aileron reversal.

10.3 forward sweep

The advantages of forward sweep are:

- Not prone to tip stall
  - No asymmetric wing drop
  - Aileron control up to high angle of attack
- Higher sweep angle for shock wave for given $\Lambda_{c/4}$
  - Either less geometric sweep is desired for the same shock sweep
  - or for the same geometric sweep there is less wave drag
- Possibility of NLF
  - Reduction of leading edge sweep which leads to a reduced attachment line instability.

The disadvantages are:

- Highly swept trailing edge: reduces effectiveness of high-lift devices
- Could reduce divergence speed (or have a heavier wing)
- Reduced stability in Dutch roll mode
- Root stall can cause rapid loss in lift (abrupt stall) and pitch-up

In order to postpone root stall a close coupled canard can be sued to decrease the effective angle of attack near the root.

11 Root and Tip effects

(chapter 22 and 40)

On a swept wing with constant airfoil section and near-constant wing-fuselage junction, in subsonic flow the isobars tend to curve rearward at the wing root and forwards at the wing tip.

When a shockwave occurs, this occurs further rearwards at the root than at the tip, which results in lower drag rise Mach number than would be expected.

For an infinite 45° swept wing the isobars curve across the plane of symmetry perpendicular, causing a decrease in supervelocities over the forward part of the centerwing and an increase over the rear part. These changes lead to an unfavourable pressure gradient, increasing the possibility of boundary layer separation.

Tip shape for swept wings are disadvantageous from a drag point of view because suction forces on the outer wing are decreased. High-speed characteristics are however improved by delaying shockwaves and increasing the drag-rise number.

The amplitude of the pressure distribution can be scaled by the thickness. The thickness should be increased to increase the amplitude of the pressure distribution.

Hence, in order to improve the velocity distribution due to thickness at the root, the thickest point should move forward and the overall thickness should be increased.

On a swept forward wing there is boundary layer inflow towards the root, which causes the wing root to stall first and produce a pitch-up. Especially with the T-tail situation may occur where the aircraft cannot recover from stall.

Near a straight sided fuselage wall the flow over a wing is similar to the flow over the centre part of the wing alone, for swept-back wings.

Vortices on a swept-back wing will show similar patterns as isobars near the wing root, but not identical. Vortices tend to curve backward and across the plane of symmetry perpendicular. This causes a decrease in lift at the wing root.

Taper and sweepback increase the loading of a high transonic cruise optimized wing in the outboard part, this compromises the design for high maximum lift.

improving the velocity distribution due to lift at the root;

- Lift adds velocity distribution to thickness-induced velocity distribution
- Sweep concentrates lift near trailing edge

The solutions are:

- Reduce camber of root profile to shift pressure peak more forward and also reduces lift.
- Increase root incidence to increase lift coefficient to match desired pressure distribution on top surface

In practice combine thickness, camber and incidence measures to have an acceptable isobar pattern on the top surface at the design condition (Cl and M) The result could be too much lift inboard for elliptical lift distribution.

The effect of wing taper and sweep on spanwise lift distribution:

- The upper surface pressure distribution is constant over the span
- The required reduction of lift on the inboard wing is achieved by increasing the root section thickness via the lower airfoil contour
- This has a significant benefit in wing weight, stiffness and fuel volume.

Minimizing root effects:

- Difficult to design a wing with straight isobars between fuselage and tip
  - Geometric sweep affects isobaric pattern
  - Thickness affects isobaric pattern
  - Circulation distribution affects isobaric pattern
- Near the root various design changes should be made
  - increase in thickness
  - thickest point moves forward
  - Increase in incidence angle
  - Decrease in camber (or even negative camber)

Rounded tip sweeps isobars aft: more form drag near the tip; less wave drag at high speeds.

To obtain a similar velocity distribution at the root and mid semi span the forward part of the root section must be thickened and the rear part made thinner. The opposite applies to the tip.

By giving a negative camber line the loss of lift in swept wings can be compensated

A constant percentage isobars can be achieved by increasing supervelocities on the lower surface of the inboard wing.

11.1 Winglets

End slats increase lift curve slope and decrease induced drag.

Tip vortex due to pressure difference on upper and lower wing. Induced angle of attack is produced, causes induced drag.

The classic winglet design involves:

- leading edge behind thickest point to prevent adverse interference effects
- Winglet has higher camber than wingtip
- Winglet has higher critical Mach number than wing tip
- winglet loading same as wing loading
- Elliptical lift distribution preserved

The shape of winglets:

- Large enough radius of inner curve to prevent interference drag
- Drag decrease dependent on toe in toe out angle with respect to interference
- Result: only drag decrease at particular lift coefficients

While an equivalent increase in wingspan would be more effective than a winglet of the same length, the bending force becomes a greater factor. A three-foot winglet has the same bending force as a one-foot increase in span, yet gives the same performance gain as a two-foot wing span increase.

Wingtip/winglets are influenced by:

- Root bending moment (depends on control surface deflection and roll manoeuvres)
- Flutter considerations
- Stability and control considerations

therefore: winglets could be problematic as add-on, wing should therefore be designed with winglets.

12 Examples of modern wing design

(chapter 23 and 24)

In order to get straight isobars over most of the wing upper surface, the following modifications can be made:

Furthermore the following has to be taken into account:

- $M_{loc} < 1.2$ to prevent drag creep
- If $M_{loc} = 1.35 - 1.45$: separation and buffet

mod. no.	Modification	Reason
1	Increase the thickness of the forward part of the root section. Decrease the thickness of the rear part of the root section.	To obtain similar chordwise upper surface velocity distributions due to thickness along the span.
2	Increase the thickness-chord ratio of the root section	To obtain identical chordwise upper surface velocity distributions due to thickness along the span.
3	Decrease the positive camber of apply negative camber on the root section.	To adapt the pattern of the chordwise upper-surface velocity distribution due to lift to that of the basic airfoil section.
4	Increase the incidence of the root section.	To obtain identical chordwise upper surface velocity distributions along the span.
5	Modify the wing lower surface along the span (mostly on the inner wing).	To obtain the desired spanwise distribution of the local lift coefficient.
6	Modify the lower surface velocity distribution on the root section regarding front and rear loading.	To minimise the wing pitching moment.
7	Modify the leading-edge region on the outer wing.	To obtain satisfactory stalling characteristics.

- Pressure gradient should never cause trailing edge separation - this limits the maximum amount of aft loading
- Shock wave position should not be too much aft
- The bottom side of the wing can be modified to get the target lift distribution
- High aft loading outboard results in highly negative C_m -Increased front loading near root can compensate this
- Increase in nose radius to postpone tip stall
- Low sweep trailing edge fro flap effectiveness

13 Control surface design

(chapter 33 and 34)

Control surfaces perform three functions:

1. Provide a means to achieve equilibrium (either trim or at non-zero control force)
2. Allow for manoeuvring in pitch roll and yaw in all flight conditions (cross wind landing)
3. counteract gusts for flight path tracking

Requirements are:

- High deflection without separation
- Linear behaviour over the entire range of deflections

The linearity between lift force and control surface angle decreases with increasing control-surface-chord-to-airfoil-chord ratio.

On a swept airfoil the root of the rear spar is a heavily loaded structural element. To maximise the spar height may therefore seem attractive. The resulting increase in the section tail angle will however decrease the aerodynamic effectiveness of the rudder. Therefore a compromise has to be made.

When designing an aileron the following has to be taken into account:

- Effect increases linearly with speed
- Linear effect of dynamic pressure
- Increasing the Mach number results in compressibility effects
- Beyond M_{mo} shock waves in combination with flow separation
- Aeroelastic deformation reduces aileron effectiveness
- The deformation is lessened at higher altitudes

Vortex generators on the wing can be an aerodynamic fix if the aileron effectiveness decreases due to separation or boundary layer: They can prevent:

- Leading edge separation
- Separation at the foot of the shock wave
- Separation at the hinge line

The type of aerodynamic balance on control surfaces has a large effect on the linearity of the hinge moment coefficient versus angle of attack and versus control surface deflection.

Dorsal fins:

- Increase the maximum sideslip angle which can be reached before control is lost
- Increase the sideslip angle where a significant rudder pedal force lightening and eventually rudder lock occurs

13.1 Spoilers

Spoilers have three functions:

- Roll control
- Speed brakes in flight
- Lift dumpers during decelerating ground runs in landings or aborted take-offs.

Wing mounted control surfaces spoil the flow over the wing, which reduces lift and increases drag.

Spoilers have three aerodynamic properties:

- decrease lift
- increase drag
- change pitching moment

However, for each of the three given functions only one of the three aerodynamic properties are necessary, thus:

- For roll control through spoiler deflection on one wing half also the overall lift of the aircraft is affected
- With symmetric deflection of spoiler panels as speed brakes also the wing lift is affected and the longitudinal behaviour of the aircraft.
- Spoiler deflection in the lift dumper function also produces a change in the pitching moment and hence may either increase or decrease the nose wheel load.

These unintended effects require often a complicated mixing of the various functions and highly non-linear relations between control wheel force and individual panel deflection angles.

- At touch down major portion of weight carried by wing
- This leads to low braking forces
- Lift spoiling allows high wheel loads and thereby higher deceleration.

For spoilers there are closed-shroud and open-shroud systems:

- Open shroud spoilers show strong non-linear behaviour
- The control response is dependent on angle of attack and flap angle
- Closed shroud spoilers aim to decouple the air-brake function from the other spoiler functions and from the lift generation
- Most current transport aircraft accept the disadvantages of open shroud spoilers because of reduced hardware complexity

Some design considerations of spoilers:

- Strong interaction between spoiler functions
- Forces and moments are non-linear with:
  - Spoiler deflection
  - Flap deflection
  - Angle of attack
  - Mach number
  - Dynamic pressure
- System redundancy requires multiple spoiler panels
- Complex mixing schedules are required

14 Horizontal tail design

(chapter 30 and 31)

Effect of planform parameters:

- Aspect ratio, A
  - High A $\rightarrow$ high $\frac{dC_L}{d\alpha}$, low α_{stall}
  - Low A $\rightarrow$ low $\frac{dC_L}{d\alpha}$, high α_{stall}
  - Little effect on $C_{L_{max}}$
- Taper ratio, λ
  - Little effect on $\frac{dC_L}{d\alpha}$
  - Effect on $C_{L_{max}}$ depends on $\Lambda_{c/4}$
    * High Λ_{le} yields vortex lift and high $C_{L_{max}}$
    * Low Λ_{le} and high λ results in tip stall due to low Re_{tip} and low $C_{L_{max}}$
- Sweep, $\Lambda_{c/4}$
  - high $\Lambda_{c/4}$ $\rightarrow$ low $\frac{dC_L}{d\alpha}$, high α_{stall}
  - low $\Lambda_{c/4}$ $\rightarrow$ high$\frac{dC_L}{d\alpha}$, low α_{stall}

- Effect on $C_{L_{max}}$ depends on formation of leading edge vortex

14.1 Horizontal tail design

Tail surfaces perform three functions

1. They provide static and dynamic stability
2. They enable aircraft control
3. They provide a state of equilibrium in each flight condition

An important driver of the horizontal tail design is: The ability to maintain a state-of-equilibrium, particular in extreme flight conditions.

Design requirements are:

1. They shall provide a sufficiently large contribution to static and dynamic longitudinal, directional, and (lateral) stability.

   Requires: high aspect ratio, minimum sweep (for high aspect ratio) and high $\frac{dC_{L_h}}{d\alpha_e} S_h$.
2. They shall provide sufficient control capability.

   Requires: high aspect ratio, minimum sweep (for high aspect ratio) and high $\frac{dC_{L_h}}{d\delta_h} S_h$.
3. Control shall be possible with acceptable control forces.

   Requires: High aspect ratio of the control surfaces, and $F = G_e C_h \frac{1}{2} \rho V^2 S_c \bar{c}_c$
4. Shall be able to handle high angles of attack.

   Requires: Low aspect ratio of the control surfaces, Sweep is beneficial.

   Limiting: High speeds in combination with deflected flaps, icing conditions
5. The tail surfaces shall be able to provide a maximum force sufficiently large to balance the total tail-off forces and moments so that static equilibrium is achieved in all flight conditions.

   Requires: Sufficiently large tail surface, sufficiently high maximum lift coefficient for both a range of control surface deflections and the effect of ice roughness.

   Static equilibrium is to be achieved for: all aerodynamic conditions and all centre of gravity positions.
6. They shall be able to handle high Mach numbers without flow separation.

   Requires: Mach number at which extensive flow separation starts above M_D for control deflection necessary to pull up at n = 1.5

Furthermore:

- high aspect ratio → higher weight, flutter analysis for T-tails.
- A few degrees anhedral (negative dihedral) has a very beneficial effect.
- High taper → could lead to premature tip stall, lower weight.
- Sweep → makes stall more gradual

C_{L_h} depends on:

- Flow parameters (Reynolds number, Mach number, Angle of attack)
- Shape parameters (planform shape, airfoil shape, control surface deflection, chord ratio, gap/overhang, tab deflection, horn geometry)

- For wings with the same Aspect ratio, increasing the sweep leads to increased stall angle of attack and increased maximum lift coefficient.

Main types of aero-elastic deformation are:

- Wing bending and torsion

  Result in:

  - Re-distribution of spanwise and chordwise loading
  - Reduction in wing lift curve slope
  - Reduction in aileron effectiveness
  - A forward shift of aerodynamic centre position on swept wings

- Fuselage bending and torsion

  Results in: reduction in stabiliser and fin lift curve slope.

  Solution: "down-rig" the elevator $\rightarrow$ leads to negative zero-lift pitching moment which tends to bend the rear fuselage and tailplane attachment upwards (increasing longitudinal stability)

- Stabiliser or horizontal tail bending and torsion
- Fin or vertical tailplane bending and torsion

  results in: lower fin lift curve slope, reduced rudder effectiveness

- Deformation of movable stabiliser attachment.

  Results in: reduction in stabiliser lift curve slope

Effect of ice:

- In certain atmospheric conditions ice may accrue on the leading edges of wings and tail surfaces
- This leads to a strong reduction of the stall angle of attack and thereby to a much reduced lifting capability
- Ice accretion on tail surfaces may therefore lead to sudden dangerous pitching moment changes
- Ice/snow accretion on the ground can lead to significant maximum lift loss, and must therefore be removed before flight.
- In certain flight conditions ice accretion on tail surfaces may lead to potentially catastrophic situations.
- Negative tailplane stall combined with elevator lock can lead to unrecoverable pitch-down situations
- Ice protection on the leading edges allows to size the tailplane closer to the clean tailplane stall limits
- Usually the vertical tailplane is much less critical (highest lift demand on take-off, or landing)

15 Vertical tail design

(chapter 32 and 33)

15.1 vertical fins

Vertical tail surface (fins) are installed to:

1. Provide directional stability in all flight conditions

2. Provide yaw control in all flight conditions
3. Ensure safe handling during engine failure conditions

Important design drivers are:

- Minimum control speed with OEI, right after take-off:
  - Relatively low speed
  - High trust setting
  - High angle of attack

  This leads to: high yaw rate and large side-slip angles when one engine fails.
- Maximum cross-wind capability
  - Low speed
  - High side-slip angle

Design requirements are:

- Shall provide a sufficiently large contribution to static and dynamic stability (directional and lateral)

  Requires: High aspect ratio, minimum sweep (for high A), high $\frac{dC_{L_v}}{d\beta_v} S_v$
- Shall provide sufficient directional control capability

  Requires: high aspect ratio, minimum sweep (for high A), high $\frac{dC_{L_v}}{d\delta_r} S_v$.
- Directional control shall be possible with acceptable control forces

  Requires: high aspect ratio of the control surfaces, $F = G_e C_h \frac{1}{2} \rho V^2 S_r \bar{c}_r$.
- Shall be able to handle high angles of sideslip

  Results in: low aspect ratios, large leading edge sweep or dorsal fins

  Limiting conditions are: Low speed and high cross wind, OEI conditions right after take-off.
- Shall provide sufficiently large forces to balance OEI moments and provide equilibrium

  Static equilibrium is to be achieved in all flight conditions (All aerodynamic conditions, all centre of gravity positions, allowing for control surface deflections, including the effects of ice)

  Requires: Sufficiently large tail surface, Measures to ensure a high maximum lift coefficient.
- Shall be able to handle high Mach numbers without flow separation.

  Requires: Mach number at which extensive flow separation stars is above M_D for control deflection necessary to correct for sideslip.

At a given angle of attack the side force on a vertical tail surface is only dependent on the fin height and planform is of secondary importance.

Thus for vertical tail design:

- High aspect ratio → higher weight, Flutter analysis should be considered for T-tails
- High taper ratio → lower weight, could lead to premature tip stall (can be reduced by sweep)

Dorsal fins:

- Generates leading edge vortices which stabilizes the flow and provides a low pressure region over the main surface
- Works identical to strakes at high α.

high yaw characteristics:

- To ensure safe flight characteristics at high yaw angles the vertical tailplane must have a large stalling angle
- This is obtained by a low aspect ratio and sweepback and/or dorsal fin
- Racy swept back tails on slow propeller aircraft may well be functional.

15.2 Rudder design

Some handling considerations:

- Control forces should be acceptable for reversible flight controls
- The force should vary linearly with deflection and side-slip angle

Control forces can be reduced by:

- Applying a horn balance
- Application of balance tabs

Some structural considerations w.r.t. sweep angle:

- Aft spar of Vertical tail highly loaded when rudder deflected
- Thicker spar beneficial
- Compressibility effects at cruise Mach numbers requires higher sweep of the aft spar
- Rudder effectiveness decreases with sweep
- Careful trade-off needs to be made to arrive at sweep angle

15.3 Control surface design

Control surfaces perform three functions

1. Control surfaces provide means to achieve a steady state of equilibrium (trim) either at zero or at non-zero control force
2. Control surfaces allow manoeuvring up to maximum roll, pitch and yaw rates. Also combinations of manoeuvres should be possible.
3. Control surfaces are used to counteract disturbances such as gusts, both small and large, which might otherwise cause the aircraft to deviate from its intended flight path

In order to obtain a maximum lift force due to control deflection a deep control surface may seem attractive, however, increasing the control-surface-chord-to-airfoil-chord ratio decreases the linearity between lift force and control-surface angle.

On a swept airfoil the root of the spar is a heavily loaded structural element. To maximise the span height may therefore seem attractive. The resulting increase in the section tail angle will however decrease the aerodynamic effectiveness of the rudder.

Dorsal fins not only increase the maximum sideslip angle which can be reached before control is lost. But, as both effects are caused by thickening of the boundary layer and flow separation, dorsal fins also increase the sideslip angle where a significant rudder pedal force lightening and eventually rudder lock occurs.

For ailerons:

- aeroelastic deformation may decrease the ailerons effectiveness

- at high lift coefficients large aileron deflections may cause significant flight path disturbances with large adverse yaw.

16 Propeller slipstream effects

(chapter 36)

Propellers generate thrust by adding momentum to a streamtube that flows through the propeller disc area. Each propeller blade may be compared to a wing and deflects a streamtube such that a resultant force is created in the desired direction. The propeller deflects the part of the streamtube behind the blade segment over the induced angle θ. This deflection does not cause induced drag but an induced thrust resulting from the increased impulse in flight direction. The deflection of the flow over the angle θ is experienced as a rotation or swirl with angle θ of the flow behind the propeller. The swirl is an energy loss and should therefore be minimised. The stream tube behind a propeller in which the velocity of the axial flow is higher than the undisturbed flow and a rotational velocity is present is called the propeller slipstream.

Components located in the slipstream experience the slipstream as a variation in the oncoming airflow. Therefore, the oncoming flow is no longer homogeneous with parallel streamlines and so the pressure distribution may differ considerably from a situation in which these components would be located outside the slipstream.

In reality there is more complexity:

- Radial velocity varies with blade span
- Finite blade span causes induced velocities
- Multiple blades cause induced velocities

Three power effects affect the longitudinal stability and control of the propeller aircraft

1. Propeller forces:

   The propeller thrust will increase when the velocity of the aircraft is reduced. This increase in thrust may destabilise depending on whether the thrust line is located beneath or above the centre of gravity of the aircraft. This is especially important at very low speeds, when the propeller thrust is at its maximum.

2. Increase in wing lift, tail-off pitching moment and downwash

   The increase in local wing lift due to the slipstream causes an increase in the downwash behind the wing. Therefore, the tail will experience a larger downwash, which will reduce the contribution of the tail to the longitudinal stability of the aircraft.

3. Change in horizontal tail lift due to the increased dynamic pressure in the slipstream

   If the horizontal tail is partly or completely immersed in the propeller slipstream it will experience a higher advantage dynamic pressure.This will result in a larger effective tailplane lift curve slope, and increases the tailplane contribution to stability.

When the flaps are deployed the slipstream will increase $C_{L_{max}}$ and increase C_{L_α}.

An increase in C_T has a destabilising effect. Two types of longitudinal stability exists for propeller aircraft:

1. The response to $\Delta\alpha$ at constant C_T
2. The response to $\Delta\alpha$ due to change in speed (and C_T)

The downwash gradient at constant C_T is:

$$\left(\frac{d\epsilon}{d\alpha}\right)_{C_T=constant} = \frac{d\epsilon}{dC_L}\left\{\left(\frac{dC_L}{d\alpha}\right)_{C_T=0}(1+kC_T)\right\} \tag{8}$$

This equation says that, when flying at constant airspeed and power setting, the higher the power setting is chosen and so the thrust, the more the contribution of the horizontal tailplane to the longitudinal stability of the aircraft decreases. This applies particular when the tailplane is outside the slipstream.

On propeller driven aircraft two types of longitudinal stability at constant power setting exist:

1. The response of the aircraft to a change in angle of attack at constant speed such as in a turn or pull-up manoeuvre
2. The response of the aircraft to a change in angle of attack due to a change in speed in rectilinear flight.

negative tailplane stall:

- High power, large flap deflection and low speed causes large downwash angles
- Very large negative angles of attack at the horizontal tail
- Might lead to flow separation

Possible solutions are:

- Modify horizontal tail leading edge with negative camber
- Change stabilizer incidence with flap setting
- Change to T-tail configuration to increase the distance to the vortex sheet and reduce the negative angle of attack

Slipstream effects:

- **Propeller forces**
  - Increase in thrust may destabilizing depending on whether the thrust line is located beneath or above the centre of gravity of the aircraft. This is especially important at very low speeds when the propeller thrust is at its maximum.
  - The propeller normal forces (C_{np}) will become quite substantial if the propeller is at a high effective angle of attack. Since the point of application is usually ahead of the centre-of-gravity, these too will often have a destabilizing effect
- **Increase in wing lift, tail-off pitching moment and downwash**
  - The increase in local wing lift due to the slipstream causes an increase in the downwash behind the wing. Therefore, the tail (also a T-tail) will experience a larger downwash, which will reduce the contribution of the tail to the longitudinal stability of the aircraft
  - Usually the increase in wing lift, particularly when the flaps are deflected will increase the tail-off pitching moment in a nose-down sense
- **Change in horizontal tail lift due to the increased dynamic pressure in the slipstream**
  - If the horizontal tail is partly or completely immersed in the propeller slipstream, it will experience a higher average dynamic pressure. This will result in a larger effective tailplane lift curve slope, and increase the tailplane contribution to stability.
  - Whether the tail is actually in the slipstream, depends on angle of attack, flap deflection and power setting.

- **Oncoming flow at the horizontal tail differs from undisturbed flow due to**
  - **The wind and engine nacelle wake**, characterized by a lower average dynamic pressure than the undisturbed flow. A low-set stabilizer may, with the wing flaps retracted or at a take-off setting, exhibit a diminishing effectiveness at high angle-of-attack if the stabilizer is immersed in the wing wake
  - **The propeller slipstream**, characterized by swirl and an increased average dynamic pressure relative to the outer flow and by an inflow of the outer flow on the slipstream boundary. The latter may further complicate the tailplane contribution to stability and control
  - **The downwash**, which is a direct consequence of the wing producing lift and is primarily tied to the lift coefficient with or without the presence of propeller slipstream

Propeller slipstream does not only affect the flow over the horizontal tail but also over the vertical tail and therefore the directional stability and control of the aircraft. This effect is due to crossflow which is induced through asymmetric lift distribution which again is caused by propeller swirl.

When the outboard engine fails the cross flow due to the inboard-up engine increases which results in a moment imbalance. This induces a yaw rate and side slip and thus requires opposite rudder deflection.

Critical situation for directional equilibrium is with flaps down and the outboard-up engine fails. This creates a large cross flow which is independent of tail size. A large vertical tail will then cause a larger yawing moment at $\beta = 0$. Therefore, a balance must be such between moment from operative engine and moment generated by sideslip and rudder deflection.

16.1 lateral stability

power application will cause an asymmetric lift distribution and a rolling moment. When $\beta > 0$ the centre of lift shifts laterally, which causes the vertical tail to generate a force and therefore an opposite rolling moment occurs. For high power this can cause lateral instability.

17 Engine intakes, exhausts, and reversers

(chapter 37, 38 and 39)

Function of engine intake:

- Decelerate flow to approximately $M = 0.6$
- Raise the pressure in the flow in high-speed conditions
- Avoid shocks or separated flow
- Minimize loss in total energy of the flow (maximize total pressure recovery)

The engine intake must deliver large amounts of air to the engine over a large envelope of operating conditions, such as Mach 0 to 0.9 an angle of attack from -5 to +20 degrees and large mass flow variations.

- Mass flow and velocity determine streamtube diameter
- Fan RPS is a compromise between turbine and fan
- Fan tips may be supersonic
- High speed at the fan increases helical tip Mach number

Cruise design requirements

- Minimizing drag

  - As low as possible supervelocities
  - Mass flow ratio A_∞/A_{HL} as near as unity as other design requirements permit, such that the stagnation point is close to the highlight.

- The supervelocities on the outside of the intake will be critical which determines the intake area
- The cross section of the engine flow streamtube will be independent to the inlet shape
- Mass flow varies greatly with engine setting, altitude, and airspeed.
- The intake area should be tailored closely to the required mass flow
- The external cowling should prevent strong shock waves from occurring

Intake design requirements at low speeds:

- MFR > 1.0
- Highest supervelocities occur near the throat area
- Velocity distribution over the throat area is not uniform (highest velocities occur near the wall)
- If $M_{TH} > 0.8$ large decrease in total pressure recovery efficiency
- Maximum M_{TH} is critical at low speed because the required engine mass flow is largest at take-off and initial climb
- Average throat Mach number should be limited to $M_{TH} < 0.8$ to prevent strong shockwaves

In order to maximise the load-carrying capability of a transport aircraft after an engine has failed every effort should be made to minimise the extra drag resulting from this flight condition.

With engines on the wing there are then two additional drag sources and one possible drag source:

1. In order to balance the resulting yawing moment a large rudder deflection is required leading to extra drag from the vertical tailplane (asymmetric drag)
2. The failed engine will produce windmilling or blocked rotor drag. This is the internal drag of the engine itself
3. The resulting low mass flow ratio on the failed engine will produce higher supervelocities on the intake exterior surfaces and increased friction drag and possible some wave drag. In combination with the high angle of attack this may lead to flow separation, with additional drag, either on the lower inside or on the upper outside of the engine nacelle. The sum of these drag contributions there termed forebody drag. When the basic external drag at mass flow ratios close to one is added the total drag is sometimes called cowl drag.

To minimise shock wave strength in cruise and to prevent flow separation on an engine intake at take-off the intake is as far as possible aligned with the local flow. As intakes on wing-engines operate in the upwash in front of the wing the intake can operate at high angles of attack. At high angles of attack the stagnation point at the top has moved far inboard. This creates a leading edge suction peak comparable to the suction peak on an airfoil. On the other hand at the maximum thrust setting the operative engines will at low speeds operate at a high mass flow. this will move the stagnation point on the lower outside further aft with increasing angle of attack leading to high local velocities on the lower inboard nacelle.

In order to align engine intakes with the local flow in cruise and to maximise the intake efficiency at full thrust and minimize the added drag when an engine fails at high angles of attack at low speed engine intakes are usually drooped 3 to 5 degrees relative to the engine centre line.

17.1 Engine exhausts

Compared to basic one-dimensional jet engine theory the real gas flow through and over engine exhausts shows a number of differences:

- Both internal and external flow have boundary layers
- Because of the boundary layer and stream wise wall curvature the internal flow is not one-dimensional. The axial velocity varies over each cross-section
- Despite the application of nozzle guide vanes or stators to reduce swirl, a certain degree of rotation will be present in the exhaust. This also leads to the internal flow not being one dimensional
- Because of the difference in velocity between external and engine exhaust flow or, on a by-pass engine, between the fan and the core flow mixing occurs on the flow boundaries. This leads to an exchange of energy between external and internal flow and between fan and core flow.

To account for the above effects the following equations are used:

$$C_V = \frac{V_{actual}}{V_{ideal}} \tag{9}$$

$$V_{actual} = \frac{F_{actual}}{\dot{m}_{actual} V_{ideal}} \tag{10}$$

$$C_D = \frac{\dot{m}_{actual}}{\dot{m}_{ideal}} \tag{11}$$

$$C_T = \frac{F_{actual}}{\dot{m}_{ideal} V_{ideal}} = \frac{F_{actual}}{F_{ideal}} = C_V C_D \tag{12}$$

A high fan pressure ratio indicates high fan exhaust flow velocities. The fan flow is directed towards the core exhaust and flow suppression occurs on the core flow. this reduces the mass flow from the core and consequently the discharge coefficient deteriorates. This situation exists particular during take-off and initial climb.

17.2 Thrust reversers

The main function of thrust reversers is to decrease the ground roll distance, either after landing or in an aborted take-off. Thrust reversers are also used for taxiing backwards from the gate on reversers thrust. There are two basic types of thrust reversers in use:

1. Bucket- or target-type reversers
2. Cascade thrust reversers.

In the design of thrust reversers three design goals can be recognized:

1. The maximization of reverse thrust (usually not more than 50% reverse thrust is realised)
2. The minimization of the risk of ingestion of the hot exhaust gas and foreign objects
3. The minimization of adverse effects on stability and control

18 Stall characteristics

(chapter 26)

According to the CS-25 regulation:

$$C_{L_{max_{V_{min}}}} < (1.06)^2 C_{L_{max_{1-G}}} \tag{13}$$

where:

- For certification purposes both values should be determined from test data obtained at the most forward C.G. position
- Certification speed margins for 1-G and V_{min} are different to obtain comparable take-off performance

In order to determine the maximum lift coefficient of a full-scale aircraft, three different definitions of $C_{L_{max}}$ are used:

1. The maximum lift coefficient of the aircraft is steady, rectilinear flight:

$$C_{L_{max_{1-G}}} = \frac{nW}{\frac{1}{2}\rho V^2 S_W} \tag{14}$$

   This value of the maximum lift coefficient is usually considered at zero thrust ($T_C = 0$).

2. The maximum lift coefficient of the aircraft without tailplane in steady flight, $C_{L_{max_{TO}}}$. This value cannot be determined directly from flight tests but must be determined from $C_{L_{max_{1-G}}}$ by subtracting the tail contribution.

3. The maximum lift coefficient based on the minimum speed measured in a stall manoeuvre.

$$C_{L_{max_{V_{min}}}} = \frac{W}{\frac{1}{2}\rho V^2 S_W} \tag{15}$$

$C_{L_{max}}$ is reached because at some part on the wing upper surface the flow in the boundary layer can no longer negotiate the large adverse pressure gradients and separates. This results in a change in the distribution of forces over the wing and a consequent change in downwash characteristics. As a consequence, the aircraft develops apart from the lift loss a tendency towards abrupt attitude changes.

Medium-speed aircraft with straight or moderately-swept wings and thick leading edges may, even without leading-edge devices, reach maximum lift coefficients of around $C_{L_{max_{1-G}}} = 3.0$.

The stalling characteristics of an aircraft determine the response of the full-scale aircraft to the occurrence of significant flow separation on the wing. Satisfactory stalling characteristics means that the response to the flow separation should be either favourable (such as a strong nose-down pitch tendency which restores a condition of attached wing flow) or should be easily controlled by the pilot. Under no circumstances shall flow separation cause an abrupt pitch-up or yaw movement which may lead to auto rotation in a spin.

Some general rules to prevent undesirable stalling characteristics are:

- Flow separation should start on the inboard wing so that spoilers and ailerons for roll control remain effective in the stall. This can for example be achieved by making sure that peak $C_{p_{min}}$-values at the leading edge (either on the wing or on the slat) occur on the inboard wing.
- When the stall occurs the tailplane and elevator shall not lose effectiveness due to becoming immersed in the wake of the separated flow on the wing.

For the spanwise distribution of the peak suction coefficient on the slat as a measure of the local lift coefficient for various angles of attack:

- The low $C_{p_{min}}$ near the root is due to the low local lift coefficient as a consequence of the strong tapering of the inboard wing
- Maximum lift is reached at $\alpha = 20.5°$.
- Above this angle $C_{p_{min}}$ decreases on the inboard wing.

On a T-tail the flow over the horizontal tail plane is initially largely undisturbed after the flow over the wing separates. Flow separation on the wing decreases the wing lift, so that the vertical balance of forces changes, making the aircraft lose altitude and increase the angle of attack. A strong nose-down pitching moment at large angles of attack is required to compensate the tendency to pitch-up at an increasing rate. But if this large-enough nose-down pitching moment is not produced the angle of attack will keep increasing until the tail plane becomes immersed in the wake of the wing. If the tailplane is immersed in the wake of the separated flow of the main wing, the aircraft will loose its longitudinal stability. It will remain unstable and pitch-up until a new equilibrium is found at a very high angle of attack. This may result in a "locked-in stall" or "deep stall", from which recovery may be extremely difficult. Next to deep stall the elevator will lose its effectiveness as well. Also, most of the aileron-effectiveness will be eliminated.

19 Take-off performance

(chapter 27, 44)

Take-off performance

- Certification tests are performed to determine the performance and flight handling characteristics of the aircraft in order to guarantee a satisfactory level of safety in its commercial day-to-day operation.
- A large part of the regulations defines the flight test required to be performed in order to obtain the basic data from which the take-off limitations can be calculated for inclusion in the official flight manual.

Engine failure during take-off has for may years been regarded as one of the most serious causes of accidents:

- During take-off the engine operates at its most critical condition
- If engine failure occurs the pilot has little time and space to react adequately

The use of high-lift devices during take-off and landing increases the maximum lift coefficient and thus decreases take-off and landing field length, approach speed and decreases the wing area as required for optimum cruise performance.

At high fields a high L/D is necessary because both the climbout and climb gradient depend on the L/D ratio:

$$s_{air} = \sqrt{\frac{W}{\sigma S C_L}\frac{T-D}{W}} \tag{16}$$

$$\sin\gamma = \frac{T}{W} - \frac{D}{L} \tag{17}$$

In the last years, due to limited maximum fuel volume, the interest of aircraft manufacturers has shifted from sophisticated high lift devices in order to minimise the wing area, to simpler systems with more emphasis on cost, reliability and maintainability.

Besides lift, high lift devices also increase the drag.

Because CFD has only limited applicability for high lift devices theory can be used fro three-dimensional configurations for certain particular applications:

- Analysing the spanwise lift distribution and the induced drag.
- Analysing chordwise pressure distributions and for smoothing irregularities in the pressure distribution.

The effects on flaps on the lift/drag polar is that field attitude might require different flap settings in order to comply with climb gradient requirements or maximum ground run.

The effect of slats is that they increase the L/D for a given flap setting and required C_L.

Take-off performance;

- Best one-engine-out climb requires a high L/D and high C_L

- Increase one, decrease the other
- Induced drag is the largest drag component
- High aspect ratio is beneficial

The boundary layer and some partial flow separation on the flap upper surface has a de-cambering effect on the flap causing the effective flap angle to be smaller than the nominal flap angle. If this effective flap angle is included in calculations based on lifting surface theory, good agreement between theory and experiment can be obtained in particular with small flap angles. At higher flap angles the agreement may be less good, particularly at low angles of attack. This is due to the rear fuselage being placed in a strong down wash resulting in vortices or flow separation on the lower fuselage and, with rear-engine configurations, on the lower surface or nacelles and pylons. For higher angles of attack the rear fuselage is better aligned with the local flow direction.

Non-planar lifting surface theory, combined with empirical data can be used to estimate lift-drag ratios for take-off and landing configurations when no wind tunnel data is available.

Large flap deflections in combination with low α result in bad predictions:

- Downwash makes large angles with fuselage
- Highly 3D flow in vortices

prediction versus wind tunnel tests:

- Prediction with empirical correction is fairly good
- No maximum lift coefficient can be predicted
- L/D can be estimated

How to determine L/D:

- Calculate induced drag
- Calculate friction drag
- Calculate trim drag
- Calculate form (pressure) drag due to lift at each section

To find the trim drag the tail-off pitching moment coefficient curves are calculated. These are compared with wind tunnel data for a centre of gravity position.

When the slope of the linearised C_L^2 versus C_D curve for the clean aircraft configuration is known, the average increase in the Oswald factor allow the estimation of the drag polar for speeds between $1.2V_s$ and $1.4V_s$ for every flap angle.

When no special measures are taken, boundary layer effects are not incorporated. This leads in the calculations to higher supervelocities than in real flow. One way of reducing this shortcoming is to compare theory and experiment not at the same angle-of-attack but at the same lift coefficient. This leads to a satisfactory comparison of the chordwise pressure distribution and the spanwise distribution of the local lift coefficient particularly when also aero-elastic deformation is taken into account.

19.1 Chapter 44

Engine failout during take-off has for many years been regarded as one of the most serious causes of accidents because:

1. During take-off the engine operates at its most critical condition
2. If an engine failout occurs the pilot has literally little time and space to react adequately

In order to maximise the operational flexibility transport aircraft are certified for a range of take-off flap settings. For each flap setting a complete free-flight and field performance test programme has to be executed.

In order to find the necessary runway length three tests are conducted:

1. Take-off with all engines operating, the necessary runway length is increased with 1.15 to account for non-ideal conditions.
2. Runway length while having a simulated engine failure just before V_1, and still be able to fly over 35 feet high obstacles
3. Runway length while having a simulated engine failure just before V_1, but the take-off is aborted, and the aircraft has to be able to come to a safe standstill

The largest of the three runway lengths necessary is the required runway length for that particular aircraft.

The minimum unstick speed, V_{MU}, is the minimum speed at which the aircraft is able to take-off within the certified take-off distance for the atmospheric and aircraft conditions under consideration. This speed is determined by three parameters:

1. Wing stall in ground effect with the main wheels still on the ground
2. Geometric limitations because the maximum usable C_L in ground effect is determined by the rear fuselage striking the runway
3. Through a limited elevator capacity (the aircraft rotates so late that due to the aircraft's speed and acceleration the aircraft leaves the ground before the maximum lift coefficient in ground effect is reached)

20 Flight beyond buffet onset and M_{MO}

The buffet onset boundary

- Just as the low speed stall the buffet onset boundary forms a limitation of the aircraft flight envelope
- Low-speed stall: trailing edge flow separation of the leading edge suction peak
- Buffet onset boundary: separation starts at the foot of the shock wave and creates oscillating pressure distribution

Airworthiness regulations require that in normal operation of civil transport aircraft the buffet onset boundary is not surpassed. Buffet may not intrude the normal operating envelope of the aircraft. Buffet can thus only appear during a pull-up or turn manoeuvre.

The certification regulations require the lift coefficient in operational cruise conditions to be limited such that a load factor of $n = 1.3$ can be reached without encountering buffet. This limits the maximum design lift coefficient.

Buffet and safety:

- The airworthiness requirements demand that if the buffet regime is penetrated this should be in fully controlled flight
- Flow separation on a swept wing should therefore not start too far outboard, to prevent strong roll or pitch-up tendencies
- For certification the buffet regime is deliberately penetrated in flight to demonstrate acceptable flight handling characteristics
- The maximum buffet penetration boundary or maximum demonstrated lift boundary has to be recorded

The buffet onset boundary is not a uniquely defined physical phenomenon. Flow separation leads to vibration and it is the latter which is recorded either by a human or by an accelerometer. This means that the intensity of the buffet as perceived by a human or recorded by a test equipment depends on their position relative to the nodes and loops of the vibrating structure.

In the wind tunnel the buffet onset boundary may be determined from a number of characteristics

1. Breaks in $C_L - \alpha$, $C_M - \alpha$ or $C_x - \alpha$.
2. Trailing-edge pressure divergence on the outboard wing
3. Divergence of dynamic wing root strain gauge recordings

The onset of buffet depends on the Reynolds number when the boundary layer is involved.

20.1 Chapter 29

Flight characteristics between M_{MO} and M_D:

- Civil transport aircraft normally do not exceed the Maximum Operating Mach Number (M_{MO})
- Requirement to comply with requirements on flying characteristics above M_{MO}
- Failures or severe atmospheric upsets cause the aircraft to pitch down and to exceed M_{MO} in a dive.
- Therefore the civil airworthiness requirements require that the aircraft demonstrates acceptable flight characteristics up to the design Dive Mach Number M_D ($M_D = M_{MO} + 0.05$ to 0.09).
- It is common practice to design the wing such that the design Mach number $M_{design} = M_{MO} - 0.03$ to 0.05.

When a jet transport aircraft is flying at or near its design condition and the flight speed is increased the pressure distribution on the wing changes rapidly. Initially the shock wave moves rearwards over a part of the wing and the pattern of the wing upper surface pressure distribution changes. When the speed is further increased the boundary layer will separate behind the shock wave and the suction forces over the front part of the wing will decrease further. On the lower surface the supervelocities increase in a regular pattern until, in particular at low lift coefficients, locally sonic speed is reached behind which point in the flow will further accelerate until a shock wave appears. This different development of upper and lower surface pressure distribution causes, in particular on swept wings, that the development of the tail-off pitching moment at different angles of attack or lift coefficients with a varying Mach number may show large variations for different wing geometries.