PrandtlPlane High Lift System Preliminary Aerodynamic Design

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Abstract

An analysis of low-speed aerodynamics for an unconventional aircraft configuration has been carried out. This configuration, named Prandtlplane, implements Prandtl's Best Wing System for low induced drag.

The state of the art of high-lift systems on civil aircraft and their historical trends have been reviewed in order to correctly identify the issues to be addressed. Lift requirements suitable to the investigated unconventional configuration have been assessed and several prospective high lift system layouts have been screened against these criteria.

A suite of aerodynamic design tools and procedures has been assembled, thoroughly validated and applied to the analysis and modification of the selected low speed configurations.

Preliminary results concerning the peculiar high-lift aerodynamics of the Prandtlplane have been summarized and recommendations for further investigations have been made.
1 Introduction

1.1 Why Unconventional Configurations?
The general consensus in the aeronautics world is that the traditional transport configuration has nearly reached an optimum. The swept wing aircraft with underwing podded jet engines, whose origins date back to a bomber (the Boeing B-47), on the scale of technological evolution (Figure 1-1), is in the terminal plateau of an evolutionary cycle, where performance improvement are minimal at the expense of great efforts.

To face the new challenges of the air transport a leap (innovation) is needed. In recent years innovation has often applied to less visible items (composites have expanded from a few details to whole fuselage sections) but no one has still dared to change the "shape" of the whole aircraft, not just the "skin". It is a tough task as a whole set of concepts and truths are to be critically addressed; even the Cayley's Paradigm, which states that each function has a distinct implementation on the aircraft (e.g. Lift-Wing, Thrust-Engine, Volume for Payload-Fuselage, Directional Stability-Fin) might be re-thought whereas an integrated approach could lead to significant gains.

The drivers for evolution are also changing their nature: from the commercial ones ("faster" and "further") focused on airline revenues to more passenger-centred requirements. The European Union tries to synthesise the issues facing the air transport world in a paper
compiled by a panel of pre-eminent aeronautical experts. Their "Vision for 2020" (1) sets a whole new set of ambitious targets for the years to come:

1. 80% reduction of NOx emissions
2. Halving perceived aircraft noise
3. Five-fold reduction in accidents
4. 50% cut in CO\textsubscript{2} emissions per passenger-km
5. 99% of all flights within 15 minutes of timetable

Tackling these demanding issues is quite daunting for a conventional aircraft, given its slow evolution pace (L/D increases of 0.25% per year (2)); there is therefore a strong need for a major breakthrough (innovation), such as an unconventional aircraft configuration, as only it can provide the required improvements in the given timeframe. This explains the renewed interest in novel configuration research. In the EU 5th Framework Program there were already three projects concerning this topic (3):

- ROSAS (Figure 1-2) addressed noise issues with engines mounted above the wing or the tailplane and thus shielded towards the ground

![Figure 1-2 ROSAS configuration (ref. 4)](image)

- VELA (Figure 1-3) investigated Blended Wing Bodies
NEFA (Figure 1-4) sought to improve knowledge of the V-tail

In the EU 6th Framework Program these research activities have been integrated in the NACRE program encompassing the whole gamut of innovative concepts for a novel transport aircraft.
1.2 Why the PrandtlPlane?
But, by the way, what is a PrandtlPlane? A PrandtlPlane is an aircraft employing Prandtl's Best Wing System concept of ref. 7 (Figure 1-5). In this paper Prandtl describes a lifting system featuring the lowest theoretical induced drag for a given span and lift.

![Figure 1-5 Best Wing System](image)

Figure 1-5 Best Wing System
Figure 1-6 shows its figures of merit (relative to the equivalent monoplane represented by the horizontal straight line for $D_B/D_M=1$) against the non-dimensional vertical gap $G/b$ between the wings ($G$ and $b$ are defined in Figure 1-5).

![Figure 1-6 Induced Drag of Biplane and Best Wing System as fractions of the equivalent monoplane](image)
The figure illustrates also that the Best Wing System can be thought of as the limit for \( n \to \infty \) of an "n-plane"; this multiplane with an infinite number of wings can be approximated by a biplane with two surfaces connecting the tips; these vertical panels substitute for tip vortices of the intermediate (infinite) wings between the upper and lower ones.

**Figure 1-7 Lift distribution for the Best Wing System**

A closed form solution of Prandtl's problem is given in ref. 8, along with an expression for its lift distribution, shown in Figure 1-7. It consists of an elliptical one plus a constant on the wings and a linearly varying one on the vertical panels. Why a Prandtl Plane? Because it grants the lowest induced drag, and this component of the drag accounts for 40% of the total in cruise configuration. For the range of practical \( G/b \) (0.1÷0.2) the gain in induced drag is 30% thus making the above-mentioned points 1 and 4 of the "Vision 2020" targets (concerned with emission reductions) look more close, since every decrease in drag translates usually in less fuel burned and consequently lower pollution.

The configuration is very flexible and the transport aircraft is just one possible implementation; more developments can be found in ref. 9.
1.3 Design and Development of the PP250

A prospective configuration for medium-long haul has been developed (10), focused on satisfying Vision 2020 requirements. Figure 1-8 shows its potential also for improved airport operations: the cargo bay is not broken by a wing carry-through box thus quickening loading and unloading, while the fuselage with a square-shaped cross-section has been aptly designed to optimise its combined passenger/freighter role.

![Figure 1-8 PP250 showing its embarking/disembarking features](image)

The peculiar rear wing junction with fins and fuselage can be seen in Figure 1-9. The fuselage tailcone is flattened (thus resembling an airfoil trailing edge) to improve the flow in the resulting duct completed by the wing lower surface and the fins. This duct is functional in improving stability: one of the issues associated with a PrandtlPlane is its low stability but, if the rear wing is not aerodynamically buried in the fuselage, it adds a righting moment due to its additional lift in the centre section, and the resulting configuration is satisfactorily stable.
1.4 **High-Lift Design and the PrandtlPlane**

The design of ref. 10 has been confined to the transonic range (Figure 1-10) and further investigation on low speed characteristics must be undertaken.
The aim of this thesis is to synthesise clear low-speed requirements pertaining to the PP250 mission, define a prospective layout (Figure 1-11) of the high-lift system, find reliable means of predicting its performance, and modifying the design where needed. It focuses more specifically on tools and methods so as to establish a capability to quickly analyse low-speed configurations, correctly assess their performance and eventually adjust their aerodynamic shape to improve the overall stall behaviour.

Figure 1-11 PrandtlPlane flapped configuration
2 High Lift Systems

2.1 High-lift Systems on Commercial Airliners

2.1.1 The Need for High-Lift Systems

High-lift systems are necessary in order to let an aircraft cope with low speed flight conditions, such as takeoff and landing, without detrimental effects on the high speed wing. Indeed, cruise demands an optimised wing area which, for a given weight, decreases with design speed, whereas a much larger area would be necessary in order to meet airfield performances without a high-lift system; but a careful trade-off between effectiveness and complexity must be made when designing the low speed wing, as a more sophisticated flap/slat arrangement adds weight and costs and could thus overwhelm the benefits it provides.

After refs 1,3, the following constraints leading to lower takeoff and landing speeds are identified:

- Economical limits to runway length (higher approach speeds would mean longer stopping distance which are seldom available or achievable)
- Safety limits to takeoff, landing and approach speeds (there is an evident correlation between these and accident rate (11))
- Speed limits due to tire wear
- Community noise limits

The same ref.’s give also clear figures (valid for a generic large twin engine transport) of the design sensitivities to high-lift system parameters:

1. A 0.10 increase in lift coefficient at constant angle of attack is equivalent to reducing the approach attitude by about one degree. For a given aft body-to-ground clearance angle, the landing gear may be shortened resulting in a weight savings of 1400 lb.
2. A 1.5% increase in maximum lift coefficient is equivalent to a 6600 lb increase in payload at a fixed approach speed.

3. A 1% increase in take-off L/D is equivalent to a 2800 lb increase in payload or a 150 nm increase in range.

4. On the Boeing B777 a 1% change in maximum lift coefficient is worth 4400 pounds in payload on landing, so a small loss in maximum lift coefficient could have a very dramatic effect on maximum landing weight.

5. High-lift system may amount up to 11% of total production cost

These examples show how even a small change in low speed characteristics can significantly affect overall aircraft weight, performances and costs, and explain why this topic is currently object of considerable research.

### 2.1.2 Some Historical Trends

High-lift system evolution is strictly related to cruise speed raise. Biplanes were quite slow and their cruise/takeoff speed ratio did not exceed 2:1, therefore an unflapped aerofoil was deemed adequate, but the subsequent speed increase (Figure 2-1) and the analogous wing loading trend (Figure 2-2) led to higher stalling speeds (Figure 2-3) which were sometimes unacceptable. On Figure 2-2 two notable case are highlighted: Both the Martin B-26 (a twin engine medium bomber) and the Boeing B-29 (a four engine strategic bomber) had quite high wing loading but, whereas the former exhibited an unusually high stalling speed (Figure 2-3), source of a historically known elevated accident rate at landing, the latter succeeded in keeping good low speed performances thanks to its sophisticated (at the time) Fowler flap granting it a $C_{L_{\text{max}}}$ value in excess of 2 (Figure 2-4).
Figure 2-1 Historical maximum speed trends (from ref 15)

Figure 2-2 Historical wing loading trends (from ref 15)
Figure 2-3 Historical stalling speed trends (adapted from ref 15)

Figure 2-4 Historical Maximum lift coefficient trends (from ref 15)
This trend towards more complicated devices saw a fast transition from split flap (as on the Douglas DC3, 1935) to the first double-slotted flap on the Douglas A-26 (Figure 2-5, 1945), and was further boosted by swept wings, which appeared shortly after WWII to address compressibility phenomena, but had detrimental effects on high-lift system effectiveness. Finally, complexity reached a peak on the Boeing 747, with Krueger flap at the leading edge (with variable camber on intermediate sections, Figure 2-30) and triple slotted fowler flap at the trailing edge.

Since then research has focused on improved simpler systems (Figure 2-6) which reduce weights and costs, while retaining the same performance of previous configurations.
2.2 Design Requirements for High Lift Systems

The main objectives of the high-lift system design are:

1. Meeting field length requirements for takeoff and landing
2. Keeping approach speed below reasonable limits for safety
3. Having sufficient climb gradient

2.2.1 Takeoff

A commercial airliner must have a lift-off speed $V_{LOF}$ equal or greater than 1.1 times $V_{MU}$, where $V_{MU}$ is the minimum speed allowing the aircraft to safely take off with one engine inoperative. A lower $V_{MU}$ (achieved through a higher $C_{L\text{\text{max}}}$) is sought, as it means a shorter take-off length, but there are geometrical constraints, such as fuselage upsweep angle, which limit ground rotation at take-off and could be a critical issue for derivative aircraft with stretched fuselage (the Airbus A321 is an example (14)).
After lift-off and landing gear retraction, a safe climb speed $V_2$ greater than 1.2 times $V_{S\text{dyn}}$ (dynamic stall speed) must be maintained, while during the subsequent second climb segment, a constant climb rate gradient of 2.3 (twin-engine aircraft) or 3 (four-engine) is required by FAR and JAR regulations.

Climb gradient $\gamma$ is

$$\gamma = \frac{T}{W} - \left( \frac{L}{D} \right)^{-1}$$

hence a trade-off between $C_{L\text{max}}$ and $L/D$, which are conflicting features, must be achieved. In Figure 2-7 a $C_L-\alpha$ curve for three flap settings is shown along with the ground rotation limit ($\alpha_{\text{limit}}$). The setting named TO III (higher flap deflection) is better for ground roll (lower $V_{\text{MU}}$ and shorter take-off run), but in Figure 2-8 it is clearly seen that it implies a low $L/D$ ratio and probably will not meet the climb gradient requirement.

Figure 2-7 $C_L-\alpha$ curve at several Take-Off configurations (11)
2.2.2 Landing

Landing condition is deemed the most critical for high-lift system performances on modern turbofan-equipped aircraft, as a low approach speed is sought for safety and economic considerations ($V_{\text{approach}}$ or $V_3$ must be greater than 1.3 times $V_{\text{Smin}}$). Moreover, cockpit visibility requirements demand a limit on angle of attack. Figure 2-9 shows a typical situation, where a single slotted flap is unable to provide the $C_{L_{\text{appr}}}$ at a reasonable attitude, and a more powerful double slotted system must be adopted.
2.3 Flow Physics of High Lift

The flow around a wing in high-lift configuration is extremely complex as it involves distinct flow regimes cohabiting and different phenomena interacting. Viscous and compressibility effects cannot be neglected as they can control stall in landing and take-off configurations respectively (17).

In the following a brief description of major flow phenomena governing high lift aerodynamics is given.
2.3.1 Smith’s Analysis of Slotted Aerofoil Aerodynamics

Smith identified five effects governing the aerodynamics of multi-element slotted aerofoils and improving their high-lift behaviour through separation avoidance. He examined mainly the outer inviscid flow and how its changes affect the boundary layer. It must be noted that, although distinction is often made between slat, flap and the main aerofoil, these concepts are general and can apply to generic upstream and downstream elements as well. It must also be stressed that these principles are valid only in the presence of slots or gaps and do not apply to plain flaps and droop-nose devices.

Slat Effect
The circulation of the slat (substituted by a vortex in Figure 2-10) induces a velocity on the leading edge of the main aerofoil opposed to the prevalent flow in that zone. The main effect is a reduction of the local speed on the main element upper side and consequently of the suction peak.
An alternative explanation of the phenomenon is that the angle of attack of the main aerofoil is lowered near the aerofoil nose; the flow is thus allowed to make a less sharp turn around the leading edge with less acceleration and suction. This lowered peak implies that the boundary layer has a lower pressure gradient to stand up to the trailing edge, and therefore stays attached up to higher angles of attack. The $C_L$-$\alpha$ curve is extended accordingly (Figure 2-11).
**Figure 2-11** $C_L - \alpha$ curve changes with slat and flap deployment (19)

*Circulation Effect*

The upwash exerted by a flap on the main aerofoil causes the total lift of the upstream element to increase (Figure 2-12) but the suction peak at the leading edge is increased as well, leading to higher lift for same angle of attack but also lower stall angle as shown in Figure 2-11.

*Figure 2-12* Circulation effect (adapted from ref. 18)
Dumping Effect

The velocity induced by the downstream element (a flap in Figure 2-13) increases the tangential speed at the trailing edge. The speed at which the flow leaves the element is named “dumping speed” and its higher value is beneficial to boundary layer as it has to stand a lower pressure gradient.

![Figure 2-13 Dumping effect on an airfoil-flap configuration (adapted from ref. 18)](image)

Figure 2-13 Dumping effect on an airfoil-flap configuration (adapted from ref. 18)

Figure 2-14 shows the phenomenon on a slat/main aerofoil configuration: the slat dumping speed is nearly three times as high as the main aerofoil one, and exemplifies why slats can be so highly loaded before stalling.

![Figure 2-14 Dumping effect on a slat-airfoil configuration (adapted from ref. 18)](image)

Figure 2-14 Dumping effect on a slat-airfoil configuration (adapted from ref. 18)
Off-the-Surface Pressure Recovery

The boundary layer continues recovery even after leaving the trailing edge, as it undergoes a pressure field similar to the downstream surface one, but the pressure rise does not take place in contact with a wall allowing higher decelerations in shorter lengths.

Fresh Boundary Layer Effect

Subdividing the aerofoil in two or more segments allows on each of them thinner boundary layers which separate at higher angles of attack.

2.3.2 Viscous Effects

Figure 2-15 Viscous effects on a multi-element airfoil (adapted from ref. 20)
**Laminar Bubbles**
A laminar bubble is likely to appear on the slat at lower Reynolds numbers, possibly induced by a shock, and may govern the stall particularly at take-off, where the forward element is the most loaded and the first to separate.
A second laminar bubble may be found on the main element and may accentuate the separation tendency at the main element trailing edge.

**Confluent Boundary Layers**
Wakes of upstream elements may merge with downstream boundary layers and cause an overall lift loss. Although systems optimised for maximum lift exhibit unmerged boundary layers up to the flap trailing edge, in practice merging of slat/main aerofoil wakes upstream of the flap shroud is often difficult to avoid.

**Viscous Wake Interaction**
The displacement effect of the wake leaving the main element tends to suppress the pressure on the flap; the suction at the start of the recovery is consequently reduced and the boundary layer on the flap has a lower pressure gradient to stand. Thicker wakes (as may be encountered at lower Reynolds numbers or higher incidences) mean higher displacements and are therefore a positive feature (Figure 2-16); it explains why a reattachment is often observed on flaps at higher angles of attack, whereas the flow is separated at a lower incidence as in curve B of Figure 2-17 compared with curve A (flow attached at lower incidence with higher lift but premature separation).
Relaminarization/Attachment Line Transition

Even with a turbulent attachment line, the flow may become laminar while passing from the lower side to the upper one (Figure 2-15), due to the favourable pressure gradient at the leading edge. A laminar boundary layer is a positive feature up to the point of maximum
suction, as it allows the subsequent turbulent flow to be thinner at the beginning of recovery and therefore to stand higher pressure gradients.

### 2.3.3 3D Phenomena (conventional aircraft)

Wing stalling is usually addressed in a quasi-2D fashion (Figure 2-18), by checking wing sections exhibiting the highest section lift coefficient compared to local $C_{L_{\text{max}}}$. $C_{L_{\text{max}}}$ may vary spanwise due to different aerofoil families employed at root and tip, and sometimes also due to different local Reynolds number (changing because of taper).

![Figure 2-18 Separation detection on wings (from ref. 19)](image)

But care should be given also to truly 3D phenomena, as vortex flows originating at engine nacelle(s) and at wing/fuselage junction (Figure 2-19), since stall is often due to them rather than to conventional 2D separation at maximum local $C_{L_{\text{max}}}$.

![Figure 2-19 3D phenomena of aircraft stalling (from ref. 14)](image)
Pylon-nacelle-wing interaction

Current airliners employs nacelles close-coupled to the wing often implying a slat cut-out which could significantly lower $C_{L_{\text{max}}}$ due to local separation (Figure 2-20 middle).

This drawback can be “naturally” circumvented by proper control of nacelle vortex flow; these high incidence vortices shed at nacelle sides (Figure 2-20 bottom) tend to reduce local angle of attack and separation on critical zones through their downwash component with beneficial effects on $C_{L_{\text{max}}}$ (Figure 2-21).
Further control is achieved by fixing shedding locations (and therefore shaping vortex flow) with strakes (Figure 2-22) with dramatic reduction of separated flow (Figure 2-23).

**Figure 2-21 Nacelle effects on $C_L-\alpha$ curve (20)**
Figure 2-22 Nacelle strake vortex flow (20)

Flap 1 & Flap 2 flow always separated - only Flap 2 flow (3/8) improves slightly at 17°
Strakes off

TUFT ACTIVITY KEY
a = active
va = very active
Sep = separated flow

Re = 6.5 x 10^6

Boundaries for \( \Delta \alpha = \alpha - (\text{C}_l^{\text{max}} \text{ at } \text{Re} = 1.5 \times 10^6) \)
= \( \alpha = \text{17°} \)
M = 0.2

Figure 2-23 Effect of nacelle strake on wing separation (20)
Slat End/Fuselage Side Problem Area

Another major source of separation is at the wing/fuselage junction. Proper shaping of slat end seems to tackle the problem (Figure 2-24).
Figure 2-24 Slat shaping effect on $C_L - \alpha$ (20)
Stability and Control Issues

High lift system may affect stability and control as well: Figure 2-25 shows how an incorrectly rigged flap causes separation on the inboard flap starting a chain of events which degrades lateral trim.

![Figure 2-25 Effect of flap separation on lateral trim (22)](image_url)

How these vortices might interact with the back wing in a prandtlplane configuration is a quite interesting issue deserving further investigation.

2.4 Review of “State-of-the-Art” on Conventional Configurations

Contemporary tendency in high-lift design is towards simpler and lighter systems due to the following advantages(11):

A simpler high-lift systems have higher L/D at takeoff which in turn gives :

- The chance to increase takeoff weight adding payload or fuel
• Reduction in aerodynamic noise and engine noise due to lower power settings

The reduction in complexity also allow lower costs in:
• Manufacturing
• Maintenance
• Spare parts logistics

This trend is exemplified by the Airbus family in fig. Figure 2-26 and Figure 2-27, where the trailing edge system has evolved from double slotted (fowler) to single slotted (fowler) and the leading edge from slat to droop-nose (on the inboard wing, which stalls first). It must be noticed that, from A320 onwards, thrust-gate is dropped in favour of a continuous trailing edge. A similar tendency is seen on Boeing airplanes, from triple slotted flap on the 727 (Figure 2-28) to single slotted on the 767 (Figure 2-29), and from leading edge Krüger flap on the 747 (Figure 2-30) to three-position slat on the 777 (Figure 2-31).
Figure 2-27 High-lift devices on Airbus A380

Figure 2-28 Triple slotted Fowler flap on the Boeing B727 (11)
Figure 2-29 Single slotted Fowler flap on the Boeing B767 (11)

Figure 2-30 Krüger flap on the Boeing B747

Figure 2-31 Three-position slat on the Boeing B777
3 Preliminary Design of the High-Lift System for a Prandtlplane Aircraft

An investigation about preliminary sizing of the high-lift system for a the PP-250 Prandtlplane aircraft has been carried out by calculating the related requirements, choosing the planform layout and checking whether its performances are adequate.

3.1 Requirements Prediction through Several Methods
Several semi-empirical methods, relying basically on statistical data, have been employed with inputs from Table 3-1 in order to find design requirements for low-speed flight; the main aim was to identify the most demanding conditions for CLMAX which will eventually drive the choice of high-lift system features and its complexity.

<table>
<thead>
<tr>
<th>W_{TO}</th>
<th>208804 kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Takeoff Runway Length L_{TO}</td>
<td>6000 nm</td>
</tr>
<tr>
<td>Range</td>
<td>3000 m (~9900 ft)</td>
</tr>
<tr>
<td>Take-off Airport Altitude</td>
<td>0 m (sea level)</td>
</tr>
<tr>
<td>Landing Runway Length L_{LAND}</td>
<td>2000 m</td>
</tr>
<tr>
<td>Approach Speed V_3</td>
<td>140 kts</td>
</tr>
<tr>
<td>Cruise Altitude</td>
<td>10500 m</td>
</tr>
<tr>
<td>\left( \frac{T}{W} \right)_{TO}</td>
<td>0.254</td>
</tr>
<tr>
<td>\left( \frac{W}{S} \right)_{TO}</td>
<td>575 kg/m^2 (~90 lb/ft^2)</td>
</tr>
<tr>
<td>C_{D0}</td>
<td>0.0225</td>
</tr>
<tr>
<td>S_{REF}</td>
<td>362.6 m^2</td>
</tr>
</tbody>
</table>

| Table 3-1 Aircraft and Mission Data[16] |
3.1.1 A note on "dynamic" and "1-g" values

In ref.'s 24, 19 and 11 the procedure outputs must be corrected in order to be compared with the values obtained by design methods in par. 3.3 and chapt. 4: in fact, regulations refer to a stalling speed \( V_s \) which is achieved in a manoeuvre (Figure 3-1), while aerodynamic prediction methods refer to a steady "1-g" situation (the one usually encountered in a wind tunnel).

A suitable correction factor must therefore be identified and used to transform dynamic data into static ones. Torenbeek suggests, citing BCAR requirements, that \( V_{S_{dyn}} \) may be taken less or equal to 0.94 times \( V_{S_{1-g}} \), and latest issues of JAR regulations (25) implicitly acknowledge the same value\(^1\) thus \( C_{L_{MAX}} \)'s calculated in the following have been corrected, where necessary, according to eq. (3.1)

\[
C_{L_{MAX_{1-g}}} = (0.94)^2 C_{L_{MAX_{dyn}}} \quad (3.1)
\]

\(^1\)See, for example JAR 25.107(b)(1), where the corrected 1.13 speed ratio is used instead of the former uncorrected 1.2
3.1.2 Raymer’s Method

This method (24) focuses on a statistical correlation between the Balanced Field Length (BFL) and the Take Off Parameter (TOP) defined in (3.2).

\[
TOP = \frac{\left(\frac{W}{S}\right)_{TO}}{\sigma\left(\frac{T}{W}\right)_{TO} C_{L_{TO}}} \quad (3.2)
\]

Figure 3-2 TOP chart, with PP-250 entry (red line)

Take-off Runway Length is substituted to BFL and the related TOP (Figure 3-2) gives the \(C_{L_{TO}}\) required at take-off. From this value, which is related to a take-off speed \(V_{TO}=1.1V_{S}\), the corresponding \(C_{L_{MAX}}^{TO_{dyn}}\) is derived through (3.3), corrected with (3.1) and displayed in Table 3-4.
\[ C_{L_{\text{MAX},\text{TO}}} = (1.1)^2 C_{L_{\text{TO}}} \] (3.3)

A further check is carried out on Landing distance through (3.4), to verify whether the \( C_{L_{\text{MAX},\text{TO}}} \) is adequate to meet also landing requirements or a higher value is needed.

\[ L_{\text{LAND}} = \frac{0.5}{\sigma \cdot C_{L_{\text{MAX}}}} \left( \frac{W}{S} \right)_{\text{LAND}} + 305 \] (3.4)

### 3.1.3 Torenbeek’s Method

This method, adapted from ref. (19), relies on statistical data as well, and uses equation (3.5) (for take-off) and (3.6) (landing) which resemble a more detailed version of the ones in 3.1.2.

\[
\left( \frac{W}{S} \right)_{\text{TO}} = \rho g C_{L_2} \left[ \frac{1.159}{L_{\text{TO}} - \frac{\Delta S_{\text{TO}}}{\sqrt{\sigma}}} \left( 1 + 2.3 \Delta \gamma_2 \right) \right] - h_{\text{TO}} \] (3.5)

- \( h_{\text{TO}} = 10.7 \text{m} \) obstacle height
- \( \mu' = 0.01 C_{L_{\text{MAX},\text{TO}}} + 0.02 \) equivalent friction coefficient
- \( ?S_T = 200\text{m} \) inertia distance
- \( ?\gamma_2 = \gamma_2 \) climb gradient margin
- \( \gamma_2 \) minimum gradient (from Regulations)

\[ \gamma_2 = \left( \frac{T}{W} \right)_{\text{TO}} - \frac{C_{\rho_2}}{C_{L_2}} \] actual climb gradient

\[ C_{\rho_2} = C_{\rho_0} + \frac{C_{L_2}^2}{\pi \cdot AR \cdot e} \]

- \( e = 0.7 \) Oswald’s coefficient in take-off configuration
- \( C_{L_2} = (0.94)^2 (1.2)^2 C_{\text{L_{MAX,TO}}} \) lift coefficient at \( V_2 \)

\[ 0.85 \left( \frac{W}{S} \right)_{\text{TO}} = \left( \frac{W}{S} \right)_{\text{LAND}} = \left( \frac{L_{\text{LAND}}}{f_{\text{LAND}} \cdot h_{\text{LAND}}} - 10 \right) \frac{h_{\text{LAND}} \rho g C_{L_{\text{MAX}}}}{1.52 \cdot a/g} \] (3.6)
\[ f_{\text{LAND}} = \frac{5}{3} \] landing field length factor

\[ h_{\text{LAND}} = 15.3 \text{ "screen" height} \]

\[ a = 0.6g \text{ average deceleration during landing} \]

Notice that in eq. (3.6) the landing weight is taken as 0.85 times take-off weight, as suggested by ref. (25) for structural analysis.

**Figure 3-3 dashed lines refer to take-off requirements, solid lines to landing, the circle is PP-250 design point at take-off**

Figure 3-3 is essentially a picture of the design space for \( \left( \frac{T}{W} \right)_{TO}, \left( \frac{W}{S} \right)_{TO} \) variables. The intent is to identify the required \( C_{\text{LMAX}} \) by tracing the constant-\( C_{\text{LMAX}} \) loci, which bound the allowed configurations subspace (the top-left corner), and finding the closest one to the
actual aircraft design point (the circle in Figure 3-3). On this basis, it can be seen that the most demanding $C_{L_{\text{MAX}}}$ is at landing with a figure of less than 1.9

### 3.1.4 Method devised from NASA CR 4746

Ref. (14) suggests that high-lift system complexity is usually governed by landing maximum lift requirements, while Rudolph in NASA CR 4746 (11) identifies the approach speed $V_3$ as the crucial parameter in this situation. It seems therefore reasonable to assume that $C_{L_{\text{MAX}}}$ can be derived by choosing an approach speed and then applying regulation limits in order to find the related stalling speed.

![Figure 3-4 Approach Chart adapted from ref. 11- red square is Prandtlplane landing condition](image)

A 140 kts approach speed has been selected as Figure 3-4 shows that airplanes in the same class as the PP-250 exhibit figures between 135 and 145 kts. From this the and the
are subsequently derived with eqs. (3.7) and (3.8). It should also be noticed that commercial airliner are sized for an approach weight close to 0.75 times $W_{TO}$.

$$V_3 = 1.3 \cdot V_{\text{dyn}} \quad (3.7)$$

$$C_{L\text{max-g}} = (0.94)^2 (1.3)^2 \cdot C_{\text{approach}} \quad (3.8)$$

### 3.1.5 Definitive requirement definition

Table shows the $C_{L \text{MAX}}$’s found from the previous methods. It should be noticed that values agree well but are quite different from the usual ones (see for example Figure 4 of ref. 14, where figures are usually greater than 2.5); this is due to the reference area which is comprised of all lifting surfaces in the Prandtlplane, whereas on a conventional aircraft it neglects the tailplane.

<table>
<thead>
<tr>
<th>Method</th>
<th>CLmax</th>
<th>Sizing Condition</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Raymer</td>
<td>2.03</td>
<td>Take-off</td>
<td>Landing Dist.$=1490m &lt; 2000m$ (mission req.)</td>
</tr>
<tr>
<td>Torenbeek</td>
<td>$&lt; 1.9$</td>
<td>Landing</td>
<td></td>
</tr>
<tr>
<td>NASA CR-4746</td>
<td>2.05</td>
<td>Landing</td>
<td>$C_{L\text{approach}}=1.36$</td>
</tr>
</tbody>
</table>

*Table 3-2 results*

The sizing $C_{L \text{MAX}}$ (the highest one) is 2.05 and corresponds, as expected, to a landing condition.

### 3.2 High Lift System Layout

Following from requirements, a suitable arrangement for high-lift and control surfaces must be devised. A previous design, developed for a 600 passenger Prandtlplane configuration
(26) and described in par. 3.2.1, has been taken as a starting point and adapted to the present case.

3.2.1 PP-600 Layout

The PP-600 (Figure 3-5) features a combination of double and single slotted fowler flap on both wings to meet a demanding $C_L_{\text{MAX}}$ requirement of around 2.5, while elevators are placed at the inboard sections; these latter devices (which, among their capabilities, have the possibility of being scheduled in conjunction to give a direct lift or pure moment output) have been extensively analyzed in ref. (26) and are credited with providing an adequate longitudinal control power. Ailerons, lastly, are positioned at the outboard of the rear wing in order to keep them effective also in a stall, as wing tip is the last part to separate on forward swept wing.

Figure 3-5 Control and High-Lift Surface Layout on PP-600(26) - 1,5 elevator 2,3,6,7 double-slotted flap 4 single-slotted flap 8 aileron
3.2.2 PP-250 Layout

The PP-250 high-lift and control system (Figure 3-6) is based mainly on PP-600 with some notable modifications. The elevators configuration has been retained without changes, as it is deemed effective in ref. (26), whereas the aileron span has been slightly increased up to 0.27% of the total wingspan. Since the $C_{L_{\text{MAX}}}$ requirement is considerably lower than the PP-600 one (2.05 against 2.5), the high-lift system has been extensively simplified with single-slotted fowler flap on both wings. Inboard front wing slat has been also retained and a similar device has been added in front of rear wing elevator; their task, apart from increasing overall $C_{L_{\text{MAX}}}$, is to protect elevator from local separation and thus keep them effective up to higher angles of attack. The front wing would also feature a thrust-gate on the kink with a wing mounted engine (this gap avoids drag caused by interference between engine jet and flaps, but also lowers flap effectiveness); in the case of rear mounted engines it is obviously not necessary.
3.3 Preliminary Analysis

In order to have a first estimate of the aircraft low-speed performances, a semi-empirical method on has been developed, validated against experimental data and applied to the prandtplane configuration under scrutiny.

3.3.1 Outline of the Method
The method is mainly based on appendix G of ref. 19, with some minor elements taken from (24). The aim is to obtain a $C_L\alpha$ curve of the aircraft in high-lift configuration starting from aerofoil data in Table 3-3.

<table>
<thead>
<tr>
<th>Wing Station</th>
<th>Aerofoil</th>
<th>Twist [deg]</th>
<th>$C_{\text{MAX}}$</th>
<th>$a_0$ [deg]</th>
<th>$C_L$ [1/rad]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>NASA SC 20714</td>
<td>+2</td>
<td>2</td>
<td>-5</td>
<td>6.28</td>
</tr>
<tr>
<td>2</td>
<td>NASA SC 20714</td>
<td>+1.8</td>
<td>2</td>
<td>-5</td>
<td>6.28</td>
</tr>
<tr>
<td>3</td>
<td>Grumman K-2</td>
<td>+4</td>
<td>2</td>
<td>-1.32</td>
<td>6.28</td>
</tr>
<tr>
<td>4</td>
<td>Grumman K-2</td>
<td>0</td>
<td>2</td>
<td>-1.32</td>
<td>6.28</td>
</tr>
<tr>
<td>5</td>
<td>Grumman K-2</td>
<td>+1.6</td>
<td>2</td>
<td>-1.32</td>
<td>6.28</td>
</tr>
<tr>
<td>6</td>
<td>NASA SC 20412</td>
<td>+1.6</td>
<td>2</td>
<td>-5</td>
<td>6.28</td>
</tr>
<tr>
<td>7</td>
<td>Grumman K-2</td>
<td>0</td>
<td>2</td>
<td>-1.32</td>
<td>6.28</td>
</tr>
</tbody>
</table>

Table 3-3 Aerofoils on PP-250 wing  Corresponding stations shown in Figure 3-7

2D characteristics are converted to 3D through eq. (3.9) and (3.10), where AR is wing aspect ratio, $\beta$ is compressibility correction factor, $k$ is an effectiveness factor, $\gamma_{.5c}$ and $\gamma_{.25c}$ are sweep angles calculated at half and quarter chord respectively. Notice that $C_{\text{MAX}}$ is

$^2$ This value is taken from aerofoil with similar features
multiplied by 0.9 to allow for spanwise variation of local $C_L$, which causes the wing to stall at a lower angle of attack (when first section stalls).

\[
C_{L_{\text{unflapped}}} = \frac{C_{L_b} \cdot AR}{2 + \left( \frac{AR^2 \beta^2}{k^2 \left( 1 + \tan^2 \frac{\Lambda_{55}}{\beta^2} \right)} + 4 \right)}
\]  
(3.9)

\[
C_{L_{\text{MAX,unflapped}}} = 0.9 \cdot C_{L_{\text{MAX}}} \cdot \cos \Lambda_{25e}
\]  
(3.10)

\[
\beta = \sqrt{1 - M^2}
\]  
(3.11)

\[
k = \frac{C_{L_b}}{2\pi}
\]  
(3.12)

Results are then modified for upwash and downwash effects with eq. (3.13) and eq. (3.14), where $\delta$ is the taper ratio of the wing subject to downwash, and $m$ and $r$ are its vertical and horizontal distance from the inducing lifting surface (the front wing in this case).

\[
\frac{\partial \epsilon}{\partial \alpha} = -0.18 \quad \text{(upwash)} \quad (3.13)
\]

\[
\frac{\partial \epsilon}{\partial \alpha} = \frac{1.75}{\pi \cdot AR \cdot (1 + |m|) \sqrt{\lambda \cdot r}} \quad \text{(downwash)} \quad (3.14)
\]

Final lift for each "clean" wing is:

\[
C_L = C_{L_{\text{unflapped}}} \cdot \left[ 1 - \frac{\partial \epsilon}{\partial \alpha} \right] \left( \alpha - \alpha_0 \right)
\]  
(3.15)
Next step is taking into account flap and slat contribution to lift \(\Delta C_{L_{\text{MAX}}}, \Delta C_{L_0}\) and \(\Delta C_{L_{\alpha}}\) integrating them into the following expressions, where \(S_i\) is the percentage of wing area influenced by the i-th high-lift device (see Figure 3-8).

\[
\Delta C_{L_{\text{MAX}}} = \Delta C_{L_{\text{MAX,unflapped}}} + 0.9 \cdot \sum_i \Delta C_{i_{\text{MAX}}} \cdot \frac{S_i}{S_{\text{ref}}} \cos \Lambda_{25c} \quad (3.16)
\]

\[
\Delta C_{L_0} = \Delta C_{L_{0,unflapped}} + 0.9 \cdot \sum_i \Delta C_{i_{L_0}} \cdot \frac{S_i}{S_{\text{ref}}} \cos \Lambda_{25c} \quad (3.17)
\]

\[
\Delta C_{L_{\alpha}} = \Delta C_{L_{\alpha,unflapped}} \cdot \left[ \sum_i \Delta C_{i_{\alpha}} \cdot \frac{S_i}{S_{\text{ref}}} \right] \quad (3.18)
\]

\[\text{Figure 3-8 An example of "flapped" wing area, in this case under the influence of the inboard trailing edge flap}\]

Additional downwash on rear wing (due to front wing flap deflection) is modelled with eq. (3.19), where all terms in the right side are the inducing surface ones and \(\text{coeff}\) depends on whether the flap is slotted or not.
\[ \Delta c = \frac{\text{coeff} \cdot \Delta C_{L_{\text{rear}}}}{AR \left( \frac{b_f}{b/2} \right)} \] (3.19) from ref. [15]

The final estimate for aircraft lift is done with eq.'s (3.20) and (3.21).

\[ L_{\text{total}} = L_{\text{front-wing}} + L_{\text{rear-wing}} \] (3.20)

\[ C_{L_{\text{total}}} = \left( \frac{1}{S_{\text{ref}}} \right) \left( \frac{S_{\text{front-wing}} \cdot C_{L_{\text{front-wing}}} + S_{\text{rear-wing}} \cdot C_{L_{\text{rear-wing}}}}{S_{\text{ref}}} \right) \] (3.21)

### 3.3.2 Validation

A validation must be undertaken in order to check precision and limits of the method applied. Two cases have been analyzed: the first, a swept wing tested at NACA (Figure 3-9, ref. (28)), has been used to examine generic \( C_{L_{\alpha}} \) and \( C_{\text{LMAX}} \) prevision capabilities on flapped configurations, whereas the second, taken from CFD results of ref. 10, constitutes the only set of data available for a Prandtlplane aircraft of comparable size.
Figure 3-9 Geometry of configuration tested in NACA TN 3040(28)

Figure 3-10 Semi-empirical Method Validation against Experimental Data from ref. (28)
As shown in Figure 3-10, the method predict quite well $C_{L\alpha}$ trend and $C_{L\text{MAX}}$ numerical value which does not seem to need the 0.9 factor (which has been retained anyway for the sake of safety), but lacks precision on the $C_{L0}$. The latter has been corrected accordingly (blue line) and the method outcome for the second case (Figure 3-11) is satisfactory. This correction has been extended to the later analysis of section 3.3.3.

![Figure 3-11 Validation through CFD output for cruise configuration (CFD data from (10))](image)

**3.3.3 Results concerning the Prandtlplane Aircraft**

Next step in high-lift system performance prediction is to examine several flap settings so as to gain a better understanding of maximum lift behaviour, and to find the most suitable schedule.
Figure 3-12 shows that wings alone (with no high-lift devices) do not meet the requirements, as it could have been expected since the "clean" wing is optimized for cruise. It must be underlined that the aircraft is considered to stall when just one wing stalls.

![Wing System Lift - "clean" wings](image)

**Figure 3-12 Lift for clean configuration**

<table>
<thead>
<tr>
<th>Configuration</th>
<th>$d_f$ front [deg]</th>
<th>$d_r$ rear [deg]</th>
<th>$\alpha$ [deg]</th>
<th>$C_{L\text{ MAX}}$ front</th>
<th>$C_{L\text{ MAX}}$ rear</th>
<th>$C_{L\text{ MAX}}$ total</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>35</td>
<td>35</td>
<td>0</td>
<td>2.2024</td>
<td>2.4701</td>
<td>2.1794</td>
</tr>
<tr>
<td>B</td>
<td>35</td>
<td>35</td>
<td>9</td>
<td>2.2024</td>
<td>2.5099</td>
<td>2.2184</td>
</tr>
<tr>
<td>C</td>
<td>55$^3$</td>
<td>35</td>
<td>0</td>
<td>2.2623</td>
<td>2.4701</td>
<td>2.1390</td>
</tr>
<tr>
<td>D</td>
<td>55$^2$</td>
<td>55$^2$</td>
<td>0</td>
<td>2.2623</td>
<td>2.5650</td>
<td>2.2321</td>
</tr>
<tr>
<td>E</td>
<td>35</td>
<td>55$^2$</td>
<td>9</td>
<td>2.2623</td>
<td>2.6048</td>
<td>2.2711</td>
</tr>
</tbody>
</table>

Table 3-4 $C_{L\text{ MAX}}$ results for several flap settings (slat deflection is 23° for all configurations)

$^3$ Double slotted flap
A configuration featuring single slotted fowler flap at their maximum allowable deflection of 35° (configuration A) has been found adequate and provides a maximum lift in excess of 2.15 (Figure 3-13). Notice that the system adopted (single slotted fowler flap) is simpler than the common standard for this class of aircraft, as Boeing 767 and 777 feature a mix of single and double slotted, and Airbus A330-340 whose highly optimized low-speed wing has single slotted flaps but no thrust-gate; the better performance of the Prandtlplane is due to the fact that both lifting surfaces lift upwards, whereas a conventional aircraft features downforce on the tailplane in many flight conditions. It must be stressed also that the semiempirical model relies on rather old data (‘60s and ‘70s, when double and triple slotted high-lift systems were the standard on commercial airliners), which allow to think that further improvements could be obtained on an optimized Prandtlplane configuration.

![Figure 3-13 Lift for low-speed wings, configuration A](image-url)
Notice also the strong downwash of front wing flap, which lowers rear wing lift and therefore maximum lift too. This effect is shown more dramatically on configuration C (Figure 3-14), where a 55° deflection of double slotted Fowler flaps on front wing causes the overall $C_{L_{MAX}}$ to decrease to an even lower value than the less flapped configuration A (see Table 3-4).

Figure 3-14 Lift for low-speed wings, configuration C

Figure 3-15 shows performance achieved by Configuration A compared to requirements, and also that the approach angle of attack is slightly greater than 5°, meaning that a further check on cockpit visibility should be carried out (11).
Further configurations have been analyzed, and it can be seen that configuration D (Figure 3-16), with double slotted flaps on both wings, would meet also a very stringent requirement on aircraft approach attitude with less than 5° (Figure 3-17).
From these preliminary results, a few warnings and design guidelines can be inferred:

- As we deal with two lifting surfaces in a non-linear aerodynamic range, we cannot simply do a weighted sum of each wing's lift; we must instead check the situation at every angle of attack.
- The first wing to stall causes the entire configuration to stall.
- Interaction between the lifting surface must be carefully watched.
- The front wing must stall first by reason of stability (the rear one would provide a righting moment improving stall recovery (29)).
- The high-lift contribution must be carefully divided between the wings:
  - the difference in $C_{L_{\text{MAX}}}$ between the wings must be kept to a minimum margin of safety for stability, as it is useless to have a wing lifting up to higher angles of attack when the other one has already stalled (this situation would mean that somewhere the high-lift devices have been oversized with an unjustified increase in cost and weight).
An attempt to define a procedure to design a high-lift system for prandtl plane with wings of comparable size could be as follows:

- size front wing system for a $C_{L\text{MAX}}$ slightly greater than aircraft requirement
- size rear wing system so that it achieves a $C_{L\text{MAX}}$ slightly greater than the front one, and at a higher angle of attack ($a_{\text{stall-rear}}>a_{\text{stall-front}}$)
- make provision for a high $C_{L0}$ on the rear wing, as it will pull upwards the total lift curve where it is more needed, before front wing stall angle
4 Final Aerodynamic Design of the High Lift System

4.1 Design Procedure

At this stage the need for a procedure to design and analyze wings in details at low speed has arisen. It should enable the engineer to rapidly evaluate different configurations and how changes affect their performances. A Navier-Stokes analysis of the wing is still not viable at this preliminary phase as it requires huge computational facilities and has a low turnaround time. A simplified and effective one has been proposed by Brune and McMasters in ref. 21. It decouples viscous effects from phenomena which are instead more easily caught by potential codes [31].

The first step is therefore an inviscid analysis of the $C_l$ span distribution in order to detect possible critical areas (Figure 4-1). This operation can be aptly done with a panel method, a vortex lattice method or even a lifting line method (but the latter has more difficulties in correctly modelling partial flapped sections), since spanwise lift distribution is largely dominated by potential phenomena.

![Critical Lift Areas](image)

**Figure 4-1 Critical Lift Areas**
In these areas, where the $C_i$ is higher, corresponding better low-speed performances are demanded to the airfoil; these performances are embodied by the local available $C_{i_{\text{MAX}}}$ which varies spanwise because of different local airfoil shape and Reynolds number. Once these section(s) are identified, a more detailed analysis of their characteristics can be carried out with a Navier-Stokes code (Figure 4-2 and Figure 4-3) or an inviscid-viscous interactional code (Figure 4-4).

**Figure 4-2 NLR 7301 flapped airfoil (turbulent kinetic energy flowfield) from ref.33**

**Figure 4-3 NLR 7301 flapped airfoil ($C_p$ distribution) from ref.33**
Care should be exerted when defining the airfoil to be analyzed: $C_l$ distribution is calculated for the streamwise airfoil (Figure 4-5), but on a swept wing the flow is really 2D only on the airfoil projected normal to the sweep line (Figure 4-6 and eq. (4.1)).
It is therefore necessary to calculate a new "normal" $C_l$ (eq.(4.3)) and assess its "normal" performances. The transforming equation are (4.1) to (4.3), where $M$ is Mach number.

\[ C_{\text{normal}} = C_{\text{streamwise}} \cdot \cos \Lambda \quad (4.1) \]

\[ M_{\text{normal}} = M_{\text{streamwise}} \cdot \cos \Lambda \quad (4.2) \]

\[ C^{\text{1normal}} = C^{\text{1streamwise}} \cdot \frac{1}{\cos^2 \Lambda} \quad (4.3) \]

### 4.2 Design Tools

#### 4.2.1 AVL

AVL is a vortex lattice method developed at the MIT Department of Aeronautics and Astronautics (34). It can model an aircraft with thin lifting surfaces and also fuselages, nacelles and general slender bodies with source+doublet segments. It has a large set of output variables, including lift, moment, lift distribution and aerodynamic derivatives. It can perform Trefftz's plane analysis too.
4.2.2 XFOIL
XFOIL is a program to analyse and design isolated airfoil by means of inviscid panel method (35,36). A boundary layer solver is also available to take viscous effects into account.

4.2.3 Fluent
Fluent is a multi-purpose program able to address a wide range of CFD and heat transfer problems (37). It features a complete set of Turbulence models and is capable of solving compressible and incompressible flows.

4.3 AVL Validation
A validation of the programs (30) is necessary in order to acquire a "sensitivity" (albeit forcefully limited) to the order of magnitude of their outputs and to be able to make a reality check on their results. It is quite impossible to make a "complete" validation of the software in a short time frame, it is therefore better to concentrate on selected topics functional to the context of the present work. For what AVL is concerned, three main issues are to be addressed: its capability to correctly predict lift distribution on lifting surfaces and force and moments on complex configurations, such as a flapped aircraft (with tail) and a PrandtlPlane.
These results will be subsequently employed as building blocks to a combination of them: a flapped configuration of a PrandtlPlane aircraft.

4.3.1 Lift and Lift Distribution
Experimental data are still taken from ref. 28 (Figure 4-7).

Figure 4-7 Geometry of NACA TN 3040 test case
It can be seen in Figure 4-8 that $C_L$ prediction are accurate up to separation angle of attack, and also local $C_l$ distribution is in fairly good agreement with experimental data for even mildly separated flow ($\alpha=8^\circ$ and $\alpha=12^\circ$ in Figure 4-9 and Figure 4-10 respectively).

![Figure 4-8 $C_L-\alpha$ curve for the NACA wing](image)

![Figure 4-9 $C_l$ spanwise distribution (slat retracted)](image)
4.3.2 Lift and Moment data on a Complex Configuration

A validation on a more complex configuration (ref. 38,) has been carried out so as to check whether AVL is able to assess direct lift and moments on a lifting system including interaction between lifting surface (here wing and tail), fuselage effect and flap deployment.
Figure 4-11 Civil aircraft configuration tested in ref.38

The configuration has been modelled with and without the source-doublet fuselage (Figure 4-13, Figure 4-14, Figure 4-19, Figure 4-20), for mid and low tail position (see Figure 4-11) and tailplane setting \( i_t = 2.6^\circ \) and \( i_t = 2.2^\circ \) (see Figure 4-12 for reference angles). Results for
the first tail position are shown in Figure 4-15 to Figure 4-18, and in Figure 4-21 to Figure 4-24 for the second one. Flap effects (available only for the second configuration) are shown in Figure 4-25 to Figure 4-28 and the static margin variation due to flap deployment is shown in Figure 4-29. Results are summarized in Table 4-1.

Figure 4-13 Aircraft model with doublet-source fuselage, mid tail

Figure 4-14 Aircraft model without fuselage, mid tail
Figure 4-15 Mid tail, $\alpha = 2.6^\circ$

Figure 4-16 Mid tail, $\alpha = 2.6^\circ$
Figure 4-17 Mid tail, it=2.6°

Figure 4-18 Mid tail, it=2.6°
Figure 4-19 Aircraft model with doublet-source fuselage, low tail

Figure 4-20 Aircraft model without doublet-source fuselage, low tail
Figure 4-21 Low tail, \( \alpha = 2.2^\circ \)

Figure 4-22 Low tail, \( \alpha = 2.2^\circ \)
Figure 4-23 Low tail, $\alpha=2.2^\circ$

Figure 4-24 Low tail, $\alpha=2.2^\circ$

Figure 4-25 Low tail $\alpha=2.2$ deg, flapped
Figure 4-26 low tail βt=2.2 deg, flapped

Figure 4-27 low tail βt=2.2 deg, flapped
Figure 4-28 low tail it=2.2 deg, flapped

Figure 4-29 Static Margin variation due to flap deployment
The following conclusions can be inferred: The doublet fuselage model tends overpredict lift and underpredict stability, therefore performs rather bad and should be excluded.

The model without doublet fuselage assesses lift and $\alpha_{\text{TRIM}}$ rather well, but the $C_m-\alpha$ curve is not as good (more stable than experiment). Variation in static margin is minimal in the experimental case (5-15% less stable), whereas for AVL it is nearly zero (at low angles of attacks it is anyway in the right verse (less stable)).

Fuselage effect has the same order of magnitude of the tail, but the doublet fuselage has a stronger unrealistic contribution.

<table>
<thead>
<tr>
<th></th>
<th>AVL with Doublet Fuselage</th>
<th>AVL - Lifting Surfaces only</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{\text{L}}$</td>
<td>Overpredicted by as high as 40% (unreliable)</td>
<td>Difference is less than 10% (higher than experiment)</td>
</tr>
<tr>
<td>$\alpha_0$</td>
<td>1 to 3 degree greater than experimental value</td>
<td>0±3° lower than experiment</td>
</tr>
<tr>
<td>$C_{\text{m}}$</td>
<td>Unreliable (too low)</td>
<td>Twice the experiment datum (unreliable)</td>
</tr>
<tr>
<td>$\alpha_{\text{TRIM}}$</td>
<td>Difference greater than 2°, unreliable</td>
<td>0±2° greater than experiment</td>
</tr>
<tr>
<td>Static Margin</td>
<td>Lower than experiment, on the verge of instability</td>
<td>Higher than experiment (nearly twice, coherent with $C_{\text{m}}$ data)</td>
</tr>
<tr>
<td>$\Delta$Static Margin due to flap deflection</td>
<td>Predicts nearly no variation compared to an experimental 10% decrease of stability</td>
<td>Predicts nearly no variation compared to an experimental 10% decrease of stability</td>
</tr>
</tbody>
</table>

**Table 4-1 AVL Output Compared to Experiment**
4.3.3 Lift and Moment data on a PrandtlPlane configuration (CFD data)

Comparison with PrandtlPlane data can be done only relative to CFD data. The complete configuration has been analysed at high (Figure 4-31) and low speed (Figure 4-30). The PrandtlPlane exhibits the usual variations due to shockwaves (see Figure 4-31, where they are highlighted with density gradient, obtaining a pseudo-Schlieren visualization) in lift and stability data (at transonic speed the aircraft is more stable and shows a higher $C_{L\alpha}$ derivative).

Figure 4-30 PP250 at M=0
The two familiar models (with and without fuselage) are employed in this case too (Figure 4-32 and Figure 4-33), and results of Figure 4-34 to Figure 4-36 are summarized as usual in Table 4-2.
Figure 4-33 PrandtlPlane – lifting surfaces only

Figure 4-34 PrandtlPlane $C_L$ vs $\alpha$ curve
Figure 4-35 PrandtlPlane $C_M$-$\alpha$ curve

Figure 4-36 PrandtlPlane Static Margin
<table>
<thead>
<tr>
<th></th>
<th>AVL with Doublet Fuselage</th>
<th>AVL - Lifting Surfaces only</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{L_{a}}$</td>
<td></td>
<td>AVL value 15% lower than Fluent Euler data</td>
</tr>
<tr>
<td>$\alpha_{o}$</td>
<td></td>
<td>2.5 degrees lower than Fluent Euler data</td>
</tr>
<tr>
<td>$C_{m_{a}}$</td>
<td></td>
<td>Values are actually coincident (difference below 2%)</td>
</tr>
<tr>
<td>$\alpha_{TRIM}$</td>
<td></td>
<td>2.3 degrees lower than Fluent Euler data</td>
</tr>
<tr>
<td>$Static Margin$</td>
<td>Not stable (doublets overestimate fuselage influence)</td>
<td>20% more stable</td>
</tr>
</tbody>
</table>

**Table 4-2 AVL Output Compared to Fluent CFD analysis**

The doublet fuselage model has been discarded as its stability outputs are highly unreliable.

### 4.4 X-foil/Fluent Validation

In order to validate the 2D tools, as test-case a supercritical airfoil (ref.39, Figure 4-37) has been selected.

#### 4.4.1 Stall Behaviour of a Supercritical Airfoil

The airfoil has been analyzed in three conditions (Figure 4-38 to Figure 4-40) with two turbulence model in Fluent (Spalrt-Allmaras and k-ω SST) and with XFOIL.
Figure 4-37 Supercritical airfoil tested in ref. 39

Figure 4-38 Test condition 1
Figure 4-39 Test condition 2

Figure 4-40 Test condition 3
The Spalart-Allmaras and $k-\omega$ SST model have been chosen as they perform quite well for flows whose separation is due to pressure gradient (41), but the $k-\omega$ SST model probably needs accurate settings as it missed $C_l_{\text{MAX}}$ value in condition 3. This is probably due to an overproduction of turbulence in the leading edge region (Figure 4-42 and Figure 4-43) where the flow is assumed to be laminar instead. This is a typical drawback of the standard $k-\varepsilon$ model (Figure 4-44), which the SST is thought to have addressed. The SST model is actually a blending of $k-\varepsilon$ and $k-\omega$ equations, supposed to switch on each one in the field best suited to its characteristics, but probably in this case it requires a more expert operator.

**Figure 4-41 Reynolds Number trends**
Figure 4-42 Spurious turbulence shed by SST model

Figure 4-43 Spurious turbulence shed by SST model (leading edge close-up)
In the end the Spalart-Allmaras model is the first choice, as it enforces a "natural" transition from laminar to turbulent flow and is thus able to detect also the inversion in $\text{Re}-C_{\text{MAX}}$ shown in Figure 4-41.

### 4.5 Design Procedure Validation

The procedure outlined in par.4.1 must be checked against experiment data to see its ability to correctly predict trends due to design variables changes. The test-case (43) is a swept wing (Figure 4-46) with symmetrical airfoil and no spanwise twist, and its re-designed version (whose intent was to improve stalling characteristics) with a cambered airfoil and washout at the tip (Figure 4-45).

More specifically, the validation aims at establishing if the procedure is able to
- Detect the shift of the first section to separate
- Notice the variation in $C_{L_{\text{MAX}}}$ due to the aforementioned geometric and aerodynamic changes.
The plain wing (Figure 4-47) starts to separate (i.e. to deviate from theoretical attached-flow $C_l$ distribution) roughly at 65% of its semispan and its global $C_L$ at separation is about 0.4, but it exhibits an actual deviation from linear range in the $C_L$-$\alpha$ curve (Figure 4-49) only for a $C_L=0.7$. 

**Figure 4-46 Dimensions of the two models of ref. 43**
The cambered and twisted wing features a separation point more inboard (60% of semispan) and at a higher $C_L$ value ($C_L=0.6$). The actual departure from linearity is at $C_L=0.8$ (Figure 4-49). These discrepancies are probably due to the use of a lifting-line method, which has more difficulties in capturing $C_L$ distribution on flapped wings.

Figure 4-47 $C_L$ distribution for the plain wing

Figure 4-48 $C_L$ distribution for the twisted and cambered wing
The first step in applying the potential-viscous analysis to the same cases is to generate a vortex lattice model of the wing (Figure 4-50) and to calculate $C_l_{\text{MAX}}$ for root and tip airfoils (Figure 4-51 and Figure 4-52).

**Figure 4-49 Lift comparison between the wings**

**Figure 4-50 AVL model of the wings tested in ref. 43**
It is now possible to analyze the separation by drawing a $C_l_{\text{MAX}}$ line joining the root and tip values (Figure 4-53, notice the slight difference between the two geometrically similar airfoil, due to the different local Reynolds numbers). This line is assumed to be a separation threshold for the $C_l$ distribution curve (both values are depicted against non-dimensional semispanwise coordinate $\eta$): once the latter trespass the former, the section is stalled and the flow starts to separate. It must be noticed that the $C_l_{\text{MAX}}$ is not the 2D value represented in Figure 4-51 and Figure 4-52, but their streamwise-projected value, according to Figure 4-5 and the inverse of eq.(4.3).
The comparison between the situation at $C_L=0.6$ ($C_L$ is the figure for the whole wing, opposed to $C_l$, which is the local one pertaining to the airfoil) and $C_L=0.7$ (Figure 4-54) leads to the deduction that the flow is separated in the latter case and that this value correlates well with the separation $C_L$ of the experimental curve (Figure 4-55), better than the theoretical results of ref. 43 shown in Figure 4-47 since, as mentioned above, a low-fidelity potential method (Weissinger's lifting line) with slightly poorer performances has been employed in that case.
Figure 4-55 Experimental-numerical comparison – plain wing

The same procedure is applied to the twisted and cambered wing (Figure 4-56 to Figure 4-60).

Figure 4-56 $C_{L}$ curve for the cambered NACA 64-A810 employed at the root
Figure 4-57 $C_l - \alpha$ curve for the cambered NACA 64-A810 employed at the tip

Figure 4-58 $C_l$ distribution for the twisted and cambered wing at $C_L = 0.8$
**Twisted & Cambered Wing - CL = 0.9**

![Graph showing Cl distribution for the twisted and cambered wing at CL=0.9](image)

**Figure 4-59** $C_l$ distribution for the twisted and cambered wing at $C_L=0.9$

![Graph showing experimental-numerical comparison - twisted and cambered wing](image)

**Figure 4-60** Experimental-numerical comparison – twisted and cambered wing

In this case (Figure 4-61) the correlation is only on the $C_l$ value, meaning that the $\alpha$ related to separation is not the correct one. This difference is less important as the designer is concerned more with aircraft maximum lift than the corresponding $\alpha$. 
It must also be stressed that the $C_L_{\text{MAX}}$ value is significantly higher than the separation one, this difference being equivalent to a 20% and almost constant in this kind of swept wing in accordance with the findings of ref. 31 too. It is therefore a correct assumption to calculate the $C_L$ for separation and add this 20% difference in every analysis.

The validated design procedure is then

1. Calculate the spanwise distribution of $C_l$ with AVL
2. Calculate airfoil $C_l_{\text{MAX}}$ at the intermediate sections and at the local Reynolds numbers, project them streamwise (Figure 4-5) and join them with straight lines (they are considered a good engineering approximation of the real spanwise variation)
3. Compare the two curves and check at which station and at which wing $C_L$ there is separation
4. Add 20% to the previous $C_L$, in order to get the $C_L_{\text{MAX}}$

4.6 PP-250 Analysis / Original "Clean Wing" (Wing 1)

The previous analysis has been applied to the PP250 (Figure 4-62). The investigation has been concentrated on the front wing, being it the first to stall, for the aforementioned stalling recovery requirements (par. 3.3.3).
The simulations in Figure 4-63 and Figure 4-64 show that the separation occurs at a relatively low $C_L$ and starts at about 75% of semispan. The main factor in this behaviour is the low $C_l_{\text{MAX}}$ of the outboard and tip sections (station 3 and 4).
riferimento non è stata trovata.}, due to their lower Reynolds number and different airfoils employed.

![Figure 4-64 PP 250 – Front wing at separation (α=6°)](image_url)

Figure 4-65 and Figure 4-66 feature the rather abrupt stall of the aforementioned tip airfoil. The nature of the stall is not strictly a problem, as it is more important how the separation spreads over the wing rather than the pure 2D phenomena, but it involves a lower airfoil $C_l_{\text{MAX}}$ too (Figure 4-67) compared with SC 20714 root and kink airfoils; this lower figure has detrimental effects on the external sections of the wing.

![Figure 4-65 K2 airfoil on the verge of stall at α=12°](image_url)
It must be noticed that the airfoil analysed is not the actual K2, but the usual corresponding projection normal to the sweep at 25% of the chord (see again Figure 4-5).

This behaviour is caused mainly by a laminar bubble developing near the leading edge (denoted by a very small plateau in the pressure distribution of Figure 4-68).
A redesign of the wing is therefore needed in order to improve its performances. In wing 2 the external K2 airfoil have been substituted with a NASA SC20714; they are both supercritical airfoils, but the latter features a higher leading edge radius of curvature, which beneficially affects stall (Figure 4-69) with separation starting at the trailing edge (Figure 4-70) and a higher $C_l_{\text{MAX}}$ (Figure 4-71). A rounder leading edge may also improve the performances of the slat when it is deployed (46).
Figure 4-69 SC-20714 airfoil at start of separation

Figure 4-70 Close-up of separated trailing edge
The front wing now has higher $C_l_{\text{MAX}}$ on the external stations (Figure 4-72) and a consequently higher overall $C_L$ at separation and $C_{L_{\text{MAX}}}$.

Figure 4-72 PP 250 – Front wing 2 at separation
4.8 PP-250 Analysis / Modified Front Wing (Wing 3 & 4)
The new airfoil has a higher camber and the \(C_l\) distribution shows consequently a pronounced "hill" at the 70% span station. In order to address this drawback, the setting twist of the local airfoil (Figure 4-73) has been changed from the previous +4° to 0° (Figure 4-74).

![Figure 4-73 Generic twist notation](image)

![Figure 4-74 PP 250 – Front wing 3 at separation](image)
A further decrease in airfoil setting (washout) is needed to shift the separation point inwards since, on an aircraft with tail engines only, a stall starting from the outer stations of the wing may cause an uncontrolled rolling moment (the stall is never truly symmetrical) (19). The separation section should not be too much inboard either, because the separated wake could be ingested by the engines (45).

![Figure 4-75 PP 250 – Front wing 4 at separation](image)

The resulting design, with outboard and tip airfoil rotated 1° downwards (wing 4), exhibits a separation slightly inboard (Figure 4-75), and a higher $C_L$.

### 4.9 PP-250 Modified Front Wings Comparison

Figure 4-76 details the lift characteristics for the four configurations; it can be observed that the development of the front wing has improved the aircraft overall high angle-of-attack performance; it must be underlined that $C_{L, \text{SEPARATION}}$'s are shown, whereas $C_{L, \text{MAX}}$'s are higher and correspond to the front wing reaching its own $C_L=1.2 \ C_{L, \text{SEPARATION}}$. 
Figure 4-76 $C_L$-$\alpha$ curves for the whole configuration with several different front wings

Table 4-3 summarizes modified features in the four analysed wings.

<table>
<thead>
<tr>
<th></th>
<th>Wing 1</th>
<th>Wing 2</th>
<th>Wing 3</th>
<th>Wing 4</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>tip airfoil</strong></td>
<td>K2</td>
<td>SC 20714</td>
<td>SC 20714</td>
<td>SC 20714</td>
</tr>
<tr>
<td><strong>angle of twist at tip station [deg]</strong></td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>-1</td>
</tr>
<tr>
<td><strong>intermediate airfoil</strong></td>
<td>K2</td>
<td>SC 20714</td>
<td>SC 20714</td>
<td>SC 20714</td>
</tr>
<tr>
<td><strong>angle of twist at intermediate station [deg]</strong></td>
<td>4</td>
<td>4</td>
<td>0</td>
<td>-1</td>
</tr>
</tbody>
</table>

Table 4-3 Modified front wing features

**4.10 PP-250 Analysis / "Flapped Wing"**

A similar procedure can be applied to the flapped configuration. The analysis is focused on the front wing as usual, as it is the first to stall. In this case a viscous analysis of the flapped airfoil is not viable in a preliminary design context, as it would need a careful tuning of slat and flap positioning relative to the main airfoil (Figure 4-77), a multivariable optimization problem, which is usually solved in the wind tunnel (20) or requires many computational
runs with a numerical code. It has therefore been adopted an hybrid procedure, with the local $C_{l_{\text{MAX}}}$ of the clean airfoil determined by viscous analysis, and the additional high-lift devices contribution calculated with the semi-empirical methods of ref. 19.

![Diagram](image1.png)

**Figure 4-77 Slat and flap settings, from ref. 20**

The first configuration to be analysed has the high-lift settings of layout B of par. 3.3.3, but the performances are lower than expected (Figure 4-78), as there is separation at a $C_{L}$ well below the one assessed in the conceptual study.

![Diagram](image2.png)

**Figure 4-78 Configuration B at separation**
A different setting, corresponding to configuration E (Figure 4-79), displays almost the same situation on the front wing, whereas the back one features a higher flap deflection (which will need in turn a more complex system, probably a double slotted Fowler flap). This configuration exploits more the back wing, whose flow is on the verge of separation (Figure 4-80) thus maximizing the contribution of both lifting surfaces, and making the system to achieve the best performance: the overall $C_L_{\text{MAX}}$ (corresponding to the stalled front wing with $C_{\text{L Front Wing MAX}}=1.2 C_{\text{L SEPARATION}}$) is 2.06, satisfying the requirement of a total $C_{\text{L MAX}}=2.05$ with a tiny margin.

![Figure 4-79 Configuration E at separation](image1)

![Figure 4-80 Configuration E at separation, showing back wing still attached](image2)
Finally Figure 4-81 shows the peculiar interactional behaviour of the PrandtlPlane: a higher deflection of flaps on the front wing has no positive effect; it is counterproductive instead as it increases downwash on the back wing with an overall lower $C_L$ at separation.

![Figure 4-81 Configuration D at separation](image)

Stability issues are illustrated in Figure 4-82, showing how flap deployment shifts the static margin backwards and makes the configuration more stable. From the validation it is known that AVL tends to overpredict stability, but the difference is such that the result may be retained as qualitatively adequate.

![Figure 4-82 Static margin of several flapped configurations](image)
5 Conclusions

This thesis has addressed the design of a high-lift system of a PrandtlPlane aircraft. It can be said that it just started the exploration of an "undiscovered country", with little or no previous bibliography on it, and much work yet to be done.

The analysis started with a survey of the state of the art and trends in high-lift systems, in order to identify the correct guidelines of what is currently achievable and what should be pursued.

Several procedures were used to get an idea of the required performances and a critical assessment of the results have been carried out with the aim of scrutinizing the PrandtlPlane numbers against the conventional ones.

A validation of semi-empirical prediction methods (originally developed in the conventional configuration context) has been undertaken to build up some confidence in their results; they have been extended to the PrandtlPlane domain, mainly by taking into account upwash, downwash and their variation due to flap deployment. It has been eventually found that they are reliable enough and able to give preliminary insights into the main trends for this kind of unconventional aircraft too.

Some prospective layouts have been identified and assessed. A possible gain in system simplicity (and consequently weight) has been identified and attributed to the fact that on a PrandtlPlane both surfaces bring lift, whereas on a conventional one the tail exhibits usually a downforce.

A more detailed design procedure pertaining to conventional configuration has been taken from bibliography and thoroughly validated. The aim has been, as in the semi-empirical case, to examine the single building blocks (prediction of airfoil stall, wing stall, downwash due to flap deployment, etc.) against available data and integrate them in order to analyse the PrandtlPlane high lift context which lacks any experimental or numerical data.

The procedure employs a potential method (AVL, a vortex lattice code) and viscous 2D codes (Fluent and XFOIL).

Fluent and XFOIL has been validated against experimental data concerning a supercritical airfoil. Two Fluent turbulence models, Spalart-Allmaras and k-ω SST, suited to address
stalling, have been employed and the first has been selected due to its ability to better resolve the flow physics. XFOIL is considerably faster but is not able to correctly detect Reynolds number trends in supercritical thick airfoil stall, therefore its use has been limited to known conventional airfoil whose behaviour can be reality-checked against empirical data.

AVL has been tested to check its doublet-source fuselage model, which has been found inaccurate, especially in the conventional case. The lifting-surfaces-only model has consequently been preferred instead, and its precision in calculating lift and moment data has been assessed.

The procedure exhibited accurate results in the analysis and design of conventional wings and has been confidently applied to the PrandtlPlane ones.

A re-design of the front wing has been done as its stall characteristics proved unsatisfactory already in the clean configuration. The reason behind this discrepancy with conceptual phase results can be attributed mainly to the inaccurate $C_{l,\text{MAX}}$ estimate of the Grumman K2 airfoil (for which no data were available). A different airfoil with a higher $C_{l,\text{MAX}}$ have been used in tip and outboard stations and the washout has been increased (i.e. the sections have been rotated further downwards).

A subsequent analysis of the flapped configurations has found that the previous best one, configuration B, having the simpler high-lift system (single slotted Fowler flaps on both wings) and still satisfying requirements, shows degraded performances; configuration E (with single slotted Fowler flaps on the front wing and double slotted Fowler flaps on the rear wing) has been chosen instead; it exhibits a nearly optimum situation, with front wing stalling first (beneficial to the stall recovery) and rear wing just on the verge of separation (therefore fully exploited). Finally, investigating configuration D confirmed an important finding of the conceptual design: more powerful devices on the front wing do not always improve performances, since the gain on the front surface may be lost due to the lower lift on rear wing because of the increased downwash.
6 Recommendations for Future Research

As stated before, the topic of high lift aerodynamics on PrandtlPlane aircraft is a rather unexplored domain, therefore there is plenty of issues to be investigated. The first suggestion is to do a wind tunnel campaign in order to gather experimental data on a flapped PrandtlPlane. These can be used to better understand mutual aerodynamic interference between the flapped wings, and to improve the validation of numerical tools. A more accurate fast design method can be applied to the design and assessment of this configuration, as the one of ref. 47, which will employ the results of the abovementioned experimental campaign to "tune" its procedure. The possibility of lower drag at takeoff for a PrandtlPlane has also been envisaged, and is worth a thorough investigation because of its implications in engine sizing. Finally, it is suggested to design both a PrandtlPlane and a conventional aircraft for the same mission with the same tools to exert a more consistent comparison between them.
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