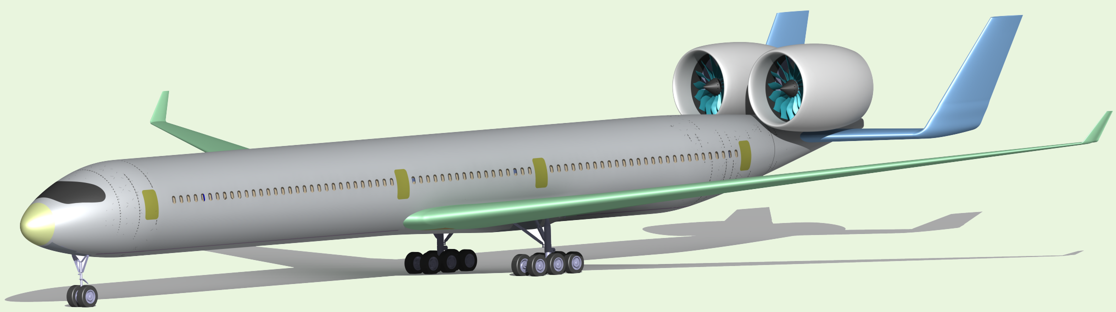
NG SLR420

Institut für Flugzeugbau
Institute of Aircraft Design

NG SLR420: Design towards sustainable long-range air travel

Aircraft Design Seminar #20



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Postprint (accepted version)

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List of Abbreviations

Abbreviations	Meaning
AC	aircraft
AoA	angle of attack
BWB	Blended Wing Body
CAD	Computer Aided Design
CG	center of gravity
CS	EASA Certification Specifications
DOC	direct operating cost
EASA	European Union Aviation Safety Agency
EIS	entry into service
FL	flight level
HTP	horizontal tail plane
IFB	Institute of Aircraft Design University of Stuttgart
MAC	mean aerodynamic chord
MTOW	maximum takeoff weight
MW	medium wide
NACA	National Advisory Committee for Aeronautics
NASA	National Aeronautics and Space Administration
NP	neutral point
OEI	one engine inoperative
SC	super critical
SFC	specific fuel consumption
SPP	standard passenger payload
TO	takeoff
VTP	vertical tail plane

List of Symbols

Abbreviations	Unit	Meaning
α_{0L}	[°]	zero lift angle of attack
α	[°]	angle of attack
b	[m]	wing span
$b_{1/2}$	[m]	half wing span
c	[m]	chord length
c_{TL}	[lb/hr/lbf]	specific fuel consumption
c_d	[-]	airfoil drag coefficient (2D)
c_D	[-]	wing drag coefficient (3D)
$c_{D,flap}$	[-]	flap drag coefficient
$c_{D,gear}$	[-]	landing gear drag coefficient
$c_{D,L\&P}$	[-]	leakage and protuberance drag coefficient
$c_{D,misc}$	[-]	miscellaneous drag coefficient
$c_{D,Trim}$	[-]	trim drag coefficient
c_f	[-]	friction coefficient
c_{fp}	[m]	flap chord length
Γ	[-]	circulation
c_l	[-]	airfoil lift coefficient (2D)
c_{lmax}	[-]	maximum lift coefficient (2D)
c_L	[-]	wing lift coefficient (3D)
$c_{L,ldg}$	[-]	lift coefficient for landing
c_{Lmax}	[-]	maximum lift coefficient (3D)
c_m	[-]	airfoil moment coefficient (2D)
c_M	[-]	wing moment coefficient (3D)
c_{mac}	[m]	chord length of mean aerodynamic chord
c_r	[m]	chord length wingtip
c_t	[m]	chord length wingroot
C_{HTP}	[-]	volume coefficient horizontal tail plane
C_{VTP}	[-]	volume coefficient vertical tail plane
δ	[°]	flap deflection angle
η_{AC}	[-]	relative aircraft efficiency
f_{CalE}	[-]	calibration factor for weight estimation
F	[N]	force
FF	[-]	form factor
L	[N]	lift force
L_{HTP}	[m]	horizontal tail plane arm length
L_{VTP}	[m]	vertical tail plane arm length
$\Lambda_{H.L.}$	[°]	sweep of the flap hinge line
Λ_m	[°]	sweep of the maximum thickness line
l/d	[-]	fineness ratio
L/D	[-]	lift over drag
m_0	[kg]	maximum takeoff mass
$\dot{m}_{Jet}$	[kg/s]	fuel mass flow
M	[Nm]	Moment
Ma	[-]	Mach number

m_{crew}	[kg]	crew mass
m_e	[kg]	empty mass
m_f	[kg]	fuel mass
m_{MTOW}	[kg]	maximum takeoff weight
$m_{payload}$	[kg]	payload mass
ν	[m^2/s]	kinematic viscosity
N_{crit}	[-]	turbulence factor
q	[N/m^2]	dynamic pressure
Q	[-]	interference factor
R	[m]	range
ρ	[kg/m^3]	density
Re	[-]	Reynolds number
$S_{flapped}$	[m^2]	flapped wing area
S_{HTP}	[m^2]	horizontal tail plane area
S_{ref}	[m^2]	wing area
S_{tra}	[m^2]	trapezoid area
S_{VTP}	[m^2]	vertical tail plane area
S_W	[m^2]	wing area
S_{wet}	[m^2]	wetted surface area
t/c	[-]	thickness-to-chord ratio
T_{TO}	[N]	takeoff thrust
v	[m/s]	velocity
v_{cruise}	[m/s]	cruise velocity
W	[kg]	weight
x/c	[-]	chordwise location of the airfoil maximum thickness point
Δx_N	[m]	position neutral point behind wing neutral point

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1 Aircraft Configuration & Fuselage Sizing

In the beginning of this project to develop an aircraft with the goal of more sustainable long range travel is the discussion where the main potentials in aircraft design are. As the engine technology is very defined by conventional fuel and the Entry Into Service by 2030 a very restricted variety of possible solution is seen. To evaluate a promising design in terms of the given sustainability goals a process with different steps was applied. This process included

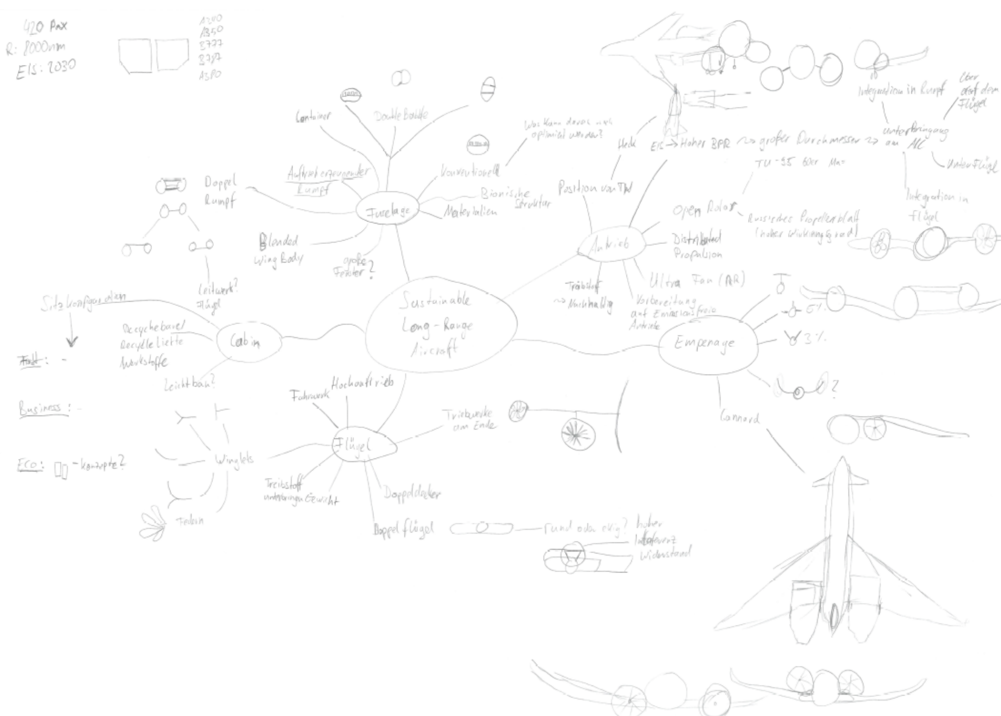


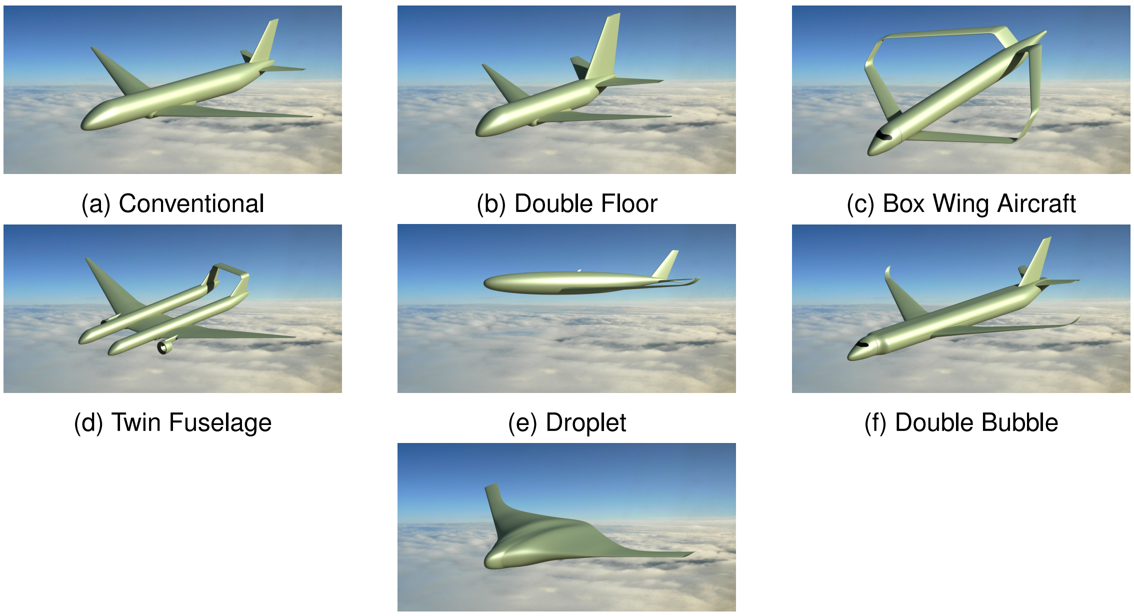
Figure 1.1: Mindmap of questioned configuration possibilities

the analysis of the aircraft and possibilities of improvement, formulation of criteria to evaluate different component configurations and also definition of the significance of these different criteria. These different steps and the final selection of configurations are further outlined in the next sections.

In fig. 1.1 a variety of possible design options is shown. The created mindmap was the first step in the draft of the aircraft design and was created in the first meeting. It should help to define possible configurations of the different options.

1.1 Initial collection of ideas

Based on a simple brainstorming process a bunch of options were collected. This included the shape of main components like fuselage wing and empennage and also minor design elements like winglets or different engine configurations. On this basis the fuselage was identified as the main part which directs or influences subsequent possibilities of options to improve the economic impact of a long-range aircraft. Therefore different fuselage shapes were identified and literature research was carried out for each option of fuselage design.



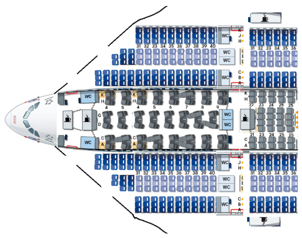
(a) Conventional (b) Double Floor (c) Box Wing Aircraft

(d) Twin Fuselage (e) Droplet (f) Double Bubble

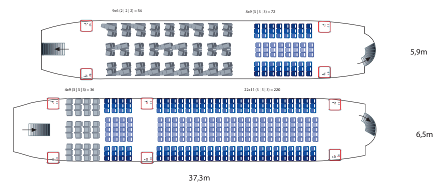
(g) Blended Wing Body

Figure 1.2: Initial collection of aircraft configurations.

Also a CAD-Model was created for every option, which was sized mainly with the given passenger number for possible visual comparison and to discuss the reasonableness of the first draft, created by these initial estimations. Further a first plain design for wing and empennage were executed by referring to specification of the reference aircraft or simple estimation by volume coefficient method. Evaluation of other promising options of component design were suspended to later design phases as they had not as much impact of the overall design. There was not a focus for literature research on simpler options like empennage design or positioning of landing gear, as the already acquired knowledge of lectures was considered sufficient.
Cabin layouts for different options of fuselage configurations are presented in fig. 1.3. To obtain the sizes of these different layout a few requirements for cabin design have been taken into consideration. The length and width depend mainly by the seat pitch and the chosen seat abreast. But also necessary space of galleys and lavatories was included. Another parameter was that the layout also should contain enough emergency exits to meet requirements. Therefore the door uniform rule according airworthiness standards was applied.

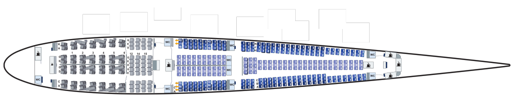


(a) Blended Wing Body

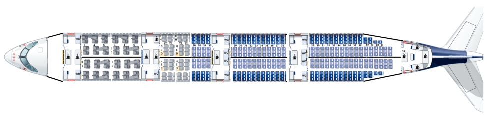


(b) Double Floor

(c) Twin Fuselage



(d) Droplet



(e) Double Bubble

Figure 1.3: Initial collection of aircraft configurations.

		Seat Width			Seat Pitch				Aisle Width		
Class	Seats Abreast	Raymer	Roskam	Lufthansa	Raymer	Roskam	Airbus	Lufthansa	Raymer	Airbus	Roskam
Business	2-2-2	20-28"	20"	20"	38-40"	38"-40"	42"	64"	20-28"	15"	15"
Premium Eco	2-4-2	17-22"	17"	18"	34-36"	34"-36"	38"	38"	18-20"	15"	15"
Economy	3-5-3	16-18"	16.5"	17"	30-32"	30"-32"	31"	31"	>=12"	15"	15"

Table 1.1: Seat dimensions.

1.2 Definition of Design Criteria

To tackle the problem of the design to invent a more environmentally friendly aircraft but also to consider other aspects. Other aspects are for example the difficulty of Entry Into Service by 2030 or to ensure competitive ability. For each component an assessment of different design is applied. Therefore a list of different criteria was made, because each component has to perform different functions. Also other criteria, which do not relate to functions but efficiency, were included. These were characteristics like required structural weight or drag. Also the cost efficiency had to be considered, which included direct operating costs and developing costs.

	Wing		Empennage		Fuselage		Gear		Engine Integration
1	low Drag	1	Low interference Drag	1	low Drag	1	minimum Weight	1	low drag
2	Structural Effort	2	operatability at stall	1	Direct Operating Costs	2	Direct Operating Costs	1	Direct Operating Costs
3	high c_L	2	Direct Operating Costs	3	Safety	3	Clearance provided	3	safety
3	Stall behaviour (soft stall)	4	structural weight	3	minimum Weight	3	minimum noise	3	non/less extra moment with OEI
5	Direct Operating Costs	5	Synergien	3	Ground Handling	5	Ground Handling	5	low interference drag
6	Manufacturing Costs	6	Trimmable HTP	6	Evacuation	6	low Drag	6	uncomplicated air intake
7	integratability High-Lifting Devices	7	Developing Costs	7	low Turn-around time	6	Synergien	7	low noise
7	Synergien	8	Stabilization Properties	8	Wing integration	6	Developing Costs	8	Developing Costs
7	Developing Costs	9	Redundancy	8	Pressure Shot Integration	6	Manufacturing Costs	8	Manufacturing Costs
10	Airport Sizing	10	Behavior on shorter AC	8	Synergien	10	easy mechanism	10	maintainability
10	Ground Handling	10	Manufacturing Costs	8	Manufacturing Costs	11	configuration	10	Synergien
12	Fuel Integration	12	Complexity of Flight Mechanics	12	Landing Grear integration	12	no extra space in fuselage	12	ground/service interference
13	Engine Ground Clearance	12	Manufacturing	12	Developing Costs	*13*	*safety*	13	available space
14	Maintainability	*14*	minimum Stall risk	14	Additional Lift by shape			13	visibility
15	Flight Stability			14	small wetted Surface				
16	Gear Integration			16	Maintainability				
17	Manufacturing			17	Familiy Concept possible				
18	Ground effect			17	Innovative Concept				
18	Effects on Cabin			19	Cabin Space/Comfort				
				20	Space for Crew Rest				
				21	Manufacturing				
				22	Amount of Cargo				
				23	Second Life prep Cargo				
				24	Alternative Usage				

Table 1.2: Design Criteria

1.3 Criteria Weighting

After the collecting of the different functions and efficiency criteria the importance of each one of these had to be ascertained. Therefore a rating method based on the VDI2225 were applied. This contains a criteria weighting with a pairwise comparison were applied. The method includes the use of pairwise comparison. The comparison method starts with setting up a matrix system. In the first row and first column every criteria is written. In the first step the diagonal is filled with ones, since every criterion is equal to itself. Then one criterion in the first row is chosen and compared with every other criterion in the first column starting with the first criterion under the diagonal. When the selected criteria is rated more important the field in the matrix is filled with a 2, with a 0 when rated less important and with 1 when both criteria are equal. At the end every rating in one column is summed up and then divided with the overall sum of all given points. With this last step the weighting factor for each criteria is obtained. The matrix for each component is shown in the Appendix 10. The rating process which filled out the matrix was performed by a discussion within the group. An example of a criteria weighting matrix can be seen in table 1.3.

Empennage

	Low interference Drag	Behavior on shorter AC	low structural weight	minimum Stall risk	operatability at stall	Stabilization Properties	Complexity of Flight Mechanics	Trimmable HTP	Redundancy	Manufacturing Difficulty	possible Synergies	Developing Costs	Manufacturing Costs	Direct Operating Costs	Overall		Quantifyable Parameter
Low interference Drag	1	0	1		1	0	0	1	0	0	1	0	0	1			c_Di
Behavior on shorter AC	2	1	2		2	1	1	1	2	0	1	1	1	2			
low structural weight	1	0	1		1	1	0	1	1	0	1	0	0	1			m_empennage
minimum Stall risk				*1*													
operatability at stall	1	0	1		1	0	0	0	0	0	1	1	1	1			
Stabilization Properties	2	1	1		2	1	1	1	1	0	1	1	1	2			
Complexity of Flight Mechanics	2	1	2		2	1	1	2	1	1	2	2	2	2			
Trimmable HTP	1	1	1		2	1	0	1	0	0	1	1	1	1			
Redundancy	2	0	1		2	1	1	2	1	2	1	1	1	1			
Manufacturing Difficulty	2	2	2		2	2	1	2	0	1	2	2	1	2			
possible Synergies	1	1	1		1	1	0	1	1	0	1	0	0	2			
Developing Costs	2	1	2		1	1	0	1	1	0	2	1	0	1			
Manufacturing Costs	2	1	2		1	1	0	1	1	1	2	2	1	2			
Direct Operating Costs	1	0	1		1	0	0	1	1	0	0	1	0	1			
Sum	20	9	18	*1*	19	11	5	15	10	5	16	13	9	19	**170**		
Rank	1	10	4	14	2	8	12	6	9	12	5	7	10	2			
	0.117647059	0.052941176	0.105882353	0.005882353	0.111764706	0.064705882	0.029411765	0.088235294	0.058823529	0.029411765	0.094117647	0.076470588	0.052941176	0.111764706			

Table 1.3: Criteria Weighting of component fuselage. From this Matrix the ranking in criteria importance can be obtained and also the weighting factor for the later component evaluation.

1.4 Evaluation of Aircraft Components

With the obtained weighting factors the rating of the single components can be executed. The list of minded component options is given in table 1.4

As mentioned earlier different sources of information where considered. For example in the rating of the fuselage the different design option were written in the first column of a table, the chosen design criteria in the first row and the associated weighting factor in the second row. For the evaluation a rating scale from zero to 10 was chosen. A score 0 was equivalent with the worst option and ten for the best. Criteria like "little wetted surface" was obtained from the first coarse CAD model, as other criteria like "safety", "evacuation", "ground handling" or "direct operating costs" were derived from existing aircraft concepts or certification requirements. If possible, also data from scientific papers were considered, but this was a very limited option. The initial score was than multiplied with the weighting factor and the score for one criterion was then obtained. Lastly all scores for one design option where summed up and the final overall ranking for each option was obtained. Another ranking, an "eco-ranking" was performed by only considering the criteria which quantify the efficiency of a component, like minimum weight, drag and lift. The choice of the design option mainly depends on the overall raking but can be also influenced how different options of components can be combined. For example a low wing engine integration is harder to combine with a low wing configuration. The awarding of points were then executed in a discussion within the entire design group. An example matrix of the component empennage is given in table 1.5.

The results (shown in appendix 10) indicate that not too innovative design option are preferred. This follows from the fact that the criteria selection also includes sufficient weight for

Wing	Empennage	Fuselage	Gear	Engine Integration
Shoulder Wing	T-Tail	Twin-Fuselage	Fuselage/Belly	Under Wing
Low Wing	U-Tail	Droplet	Under Wing	In Wing
Mid Wing	V-Tail	Double Bubble	In Wing	Upper Wing
C Wing	Standard	Double Floor	In Engine Cowling	Aft Fuselage
Box Wing	Canard	Conventional		Front Fuselage
Canard		Blended Wing Body		On Top of Fuselage
Blended Wing Body		Extra Wide Body		
Strut Braced		Geodesic Structure		
Foldable Wingtip		3D Bionic Structure		

Table 1.4: Design Options

		Empennage														
				T-Tail			V-Tail			U-Tail			Standard			Canard
Low interference Drag	0.117647059	4%	6	0.705882353	3%	8	0.941176471	3-4%	7	0.823529412	5%	5	0.588235294	5%	5	0.588235294
Behavior on shorter AC	0.052941176		5	0.264705882		5	0.264705882		5	0.264705882		5	0.264705882		5	0.264705882
low structural weight	0.105882353		3	0.317647059		6	0.635294118		6	0.635294118		4	0.423529412		4	0.423529412
minimum Stall risk	0.005882353			0			0			0			0			0
operatability at stall	0.111764706		5	0.558823529		6	0.670588235		5	0.558823529		5	0.558823529		7	0.782352941
Stabilization Properties	0.064705882		5	0.323529412		4	0.258823529		5	0.323529412		5	0.323529412		3	0.194117647
Complexity of Flight Mechanics	0.029411765		5	0.147058824		3	0.088235294		4	0.117647059		5	0.147058824		3	0.088235294
Trimmable HTP	0.088235294		4	0.352941176		2	0.176470588		6	0.529411765		8	0.705882353		8	0.705882353
Redundancy	0.058823529		5	0.294117647		6	0.352941176		7	0.411764706		5	0.294117647		5	0.294117647
Manufacturing Difficulty	0.029411765		4	0.117647059		6	0.176470588		4	0.117647059		7	0.205882353		6	0.176470588
possible Synergies	0.094117647		6	0.564705882		5	0.470588235		6	0.564705882		5	0.470588235		6	0.564705882
Developing Costs	0.076470588		5	0.382352941		3	0.229411765		4	0.305882353		8	0.611764706		5	0.382352941
Manufacturing Costs	0.052941176		5	0.264705882		6	0.317647059		4	0.211764706		7	0.370588235		6	0.317647059
Direct Operating Costs	0.111764706		4	0.447058824		5	0.558823529		5	0.558823529		5	0.558823529		5	0.558823529
				4.741176471			5.141176471			5.423529412			5.523529412			5.341176471
				5			4			2			1			3
"Green Factors"																
				1.023529412			1.576470588			1.458823529			1.011764706			1.011764706
				3			1			2			4			4

Table 1.5: Evaluation of component fuselage, ecologic factors are highlighted in green for an extra indication of the components sustainability

the competitiveness of the future product to the discussion and also the effort of research for more innovative and unusual concepts has to be considered.

1.5 Assembly of Aircraft Configuration variation

After the evaluation of each component option a ranking were obtained. For the further design process some options were sorted out after this step. This is reasoned hence the obtained score of some options far inferior to the best. The best options are shown in table 1.2.

The elimination of the other options during the component evaluation process, presented in section 1.4, have various reasons. First it could be a technology which is not researched and developed at this time, so that an Entry Into Service would cause an enormous developing effort. Another difficulty of developing a more sustainable aircraft is the uncertainty of the future circumstances by legislation and also future airport infrastructure layout. The blended wing body is an example which combines both facets. Secondly and simpler the reason can be, that the design option does not seem that promising in more ecological friendly development,

1.5. ASSEMBLY OF AIRCRAFT CONFIGURATION VARIATION

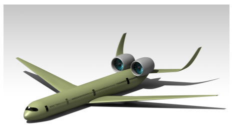
Best Ranked Components				
	Wing	Empennage	Fuselage	Engine
1	Low Wing	Standard	Conventional	Under Wing
2	Shoulder Wing	U-Tail	Extra Wide Body	Upper Wing
3	Canard	Canard	Double Floor	Aft Fuselage
4	C-Wing	V-Tail		

Table 1.6: Component Ranking.

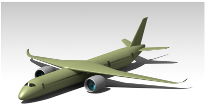
for instance the Twin-Fuselage design has a promising advantage in lower structural weight but was rated lower in terms of low drag and especially of wetted surface [4].

With the best ranked components, shown in table 1.6, were then variations of configurations assembled. In the assembling of these configuration the suitability of the component designs to each other were considered. Combination of different component can be a question of installation space, for instance it will become much harder to fit a modern Ultra-Fan engine under a low Wing. Another assembly criterion can be aerodynamic, that the empennage should not be in the wake of a component upstream. Also the old aphorism "If it looks good, it flies good" played a role in this task.

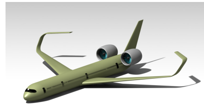
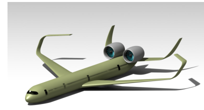
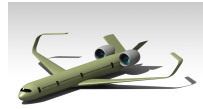
The collection of assembled configurations is given in the following:

Version 1	1.1
Extra wide body Low wing U-tail Aft fuselage engine	

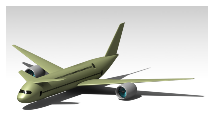
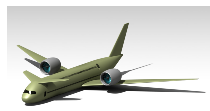
Version 1 of assembled a combination of an aft-fuselage engine integration and u-tail empennage with an extra wide body fuselage. The advantages are low interference drag on engine and empennage, low weight due to possible shorter landing gear, good ground clearance for engines, a utilization of boundary layer ingestion on fuselage. Also good noise shielding and higher comfort are positive aspects of this configuration. Negative aspects are hereby possible developing issues, unkown flight characteristics due to rearward position of the center of gravity. Also higher weight penalty according the engine position and the use of the u-tail can be expected.

Version 2	2.1
Conventional fuselage Shoulder wing Standard Empennage Under Wing Engine Integration	

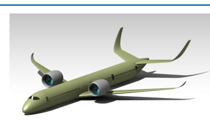
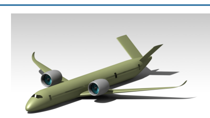
The basic attribute of version 2 is the shoulder wing which features good ground clearance for the under wing integration of the engines. Beside these facts the configuration also provides already experienced concepts and an undisturbed engine air intake. A further efficiency advantage could be slightly higher lift due to shoulder wing section. An disadvantageous characteristic is a high weight penalty as a result of increased structural effort for the wing integration and higher wetted surface integration due to the integration of the gear in the belly. Aerodynamically unfavourable is the location of the empennage in the wake of the wing.

Version 3	3.1	3.2	3.3
Extra Wide Body C-Wing Standard/V/U-Tail Aft Fuselage Engine Integration			

The assembled configuration version 3 is mainly characterized by the use of a C-Wing and an aft fuselage engine integration. The most promising advantages of this configuration are in general offered by C-Wing shape, which reduces induced drag by C-Wing shape and also enables a smaller horizontal tail plane or even an omission of it. Also enough space for modern ultra fan engines would be provided due to the aft fuselage integration and no disadvantages would occur for ground handling. The general problem for this configuration would be the yet not developed wing concept and complicated line service for engines. Specific unfavourable characteristic of the 3.1 and 3.3 versions would be higher interference between engines and empennage. Version 3.2 would fix this problem but with the use of an U-tail empennage but this type of tail would also not be ideal due to the use of the C-Wing.

Version 4	4.1	4.2
Double Floor Low Wing Canard & Standard Tail Upper/Under Wing Engine		

The primary quality of this version 4 configuration would be clearly the use of a double floor fuselage and a canards. This would provide the positive aspects of a shorter fuselage than conventional and lower needed wing area which leads to reduced wingspan. General disadvantages are hereby statically longitudinal instability and complications to fulfill turnaround time requirements. A special unfavourable attribute of version 4.1 would be the space for engine integration. Version 4.2 would remedy this but a complication could be that the wake of the canard interferes with the upper wing installed engine

Version 5	5.1	5.2
Conventional Low wing V/U-Tail Upper Wing Engine Integration		

Configuration Version 5 primarily focuses and the integration of a modern ultra fan engine. Therefore an upper wing location for the engine is defined. Benefits of this configuration are higher efficiency of control surfaces due to Coanda effect provided by the engine exhaust and noise shielding from wing. Disadvantageous are the untypical engine position and thermal shielding form engine exhaust.

1.6 Evaluation of selected configuration

After assembling of the best ranked component options the next step was to evaluate the selected possible configurations in section 1.5. This was mainly performed by the use of another evaluation matrix was set up in which every component option is compared with each option of other components and rated. The rating was given after a discussion how one option fits best to a combined use of other component options. The matrix is given in table 1.7. A evaluation of the shoulder wing was renounced due to the lack of promising attributes of the version 2 configuration viewed in section 1.5.

	Wing			Empennage				Engine Position			Fuselage		
	Upper Wing	C-Wing	Low Wing	V-Tail	U-Tail	Standard	Canarad	Upper Wing	Lower Wing	Aft Fuselage	Double Floor	Extra Wide Body	Convetional
Upper Wing													
C-Wing													
Low Wing													
V-Tail	1	0.5	1										
U-Tail	1	1	1										
Standard	1	∼	1										
Canard	1	X	1										
Upper Wing	?	∼	1	2	2	1	∼						
Lower Wing	1	X	X	1	1	1	1						
Aft Fuselage	1	1	1	∼	∼	1	1						
Double Floor	1	1	1	1	1	1	1	1	1	1			
Extra Wide Body	1	1	1	1	1	1	1	1	1	1			
Convetional	1	1	1	1	1	1	1	1	1	1			

Table 1.7: Evaluation matrix for variation of components.

With the obtained ratings from the evaluation matrix in table 1.7 a small selection of the most promising aircraft configurations was defined. It was decided that configuration versions 3.1, 5.1 and 5.2 are promising. Therefore a conventional or C-Wing and extra wide body fuselage with aft fuselage or upper wing engine integration will be examined in later tasks to obtain more knowledge.

1.7 Fuselage Sizing

With the obtained selection of aircraft configuration, it was also defined that the cabin layout will be quite close to conventional, either the fuselage will have an extra wide body or a conventional layout. Therefore a concrete fuselage design was performed.
The main parameter was the required passenger number of 420 and the demanded seat distribution in Business, Premium Economy and Economy class. The dimensions for seat width, seat abreast, seat pitch and aisle width were derived from market analyses. In these analyses the dimensions of different airlines were investigated and an average value were chosen in each on of them. In the sizing also enough space for lavatories and galleys had to be provided. These required numbers were also obtained from market analyses. Another important property of fuselage sizing was the definition of emergency exits. In case of emergency an commercial aircraft it must be guaranteed that the maximum passenger number including the required crew number can be evacuated within 90 seconds. To fulfill this requirement defined in EASA's certification specification CS25.803 an uniform distribution rule can be applied. It provides a counting method which indicates if the doors are distributed in a way that the evacuation requirements can be fulfilled. In the specifications the allowed increase of passenger number is defined for each type of door. Then the further adaption of the distribution rule includes two countings, from the front to the rear and vice versa. Within these countings the maximum increase provided by the door and the actual seats are considered. The target is, that the number of seats must not exceed the maximum increase, defined by the door.
Different investigated designs of cabin layouts are shown in appendix 10. To obtain knowl-

edge how suitable the investigated cabin designs were the slenderness ratio of every design was calculated and then compared with empirical data. This should indicate the aerodynamic efficiency. Results of this study are shown in fig. 1.4.

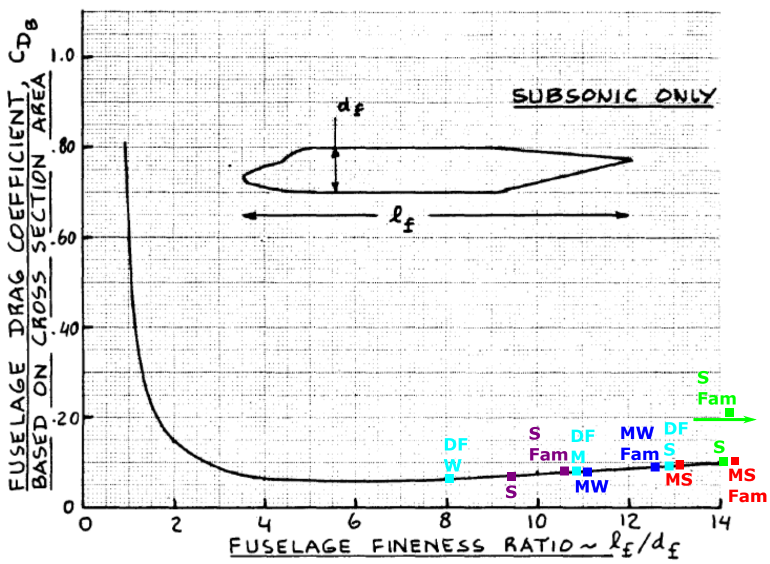


Figure 1.4: Study of fuselage drag [1]

Also layouts for the family version for passenger increase of 20% were carried out. Selected were the MW-version (Medium Wide). Cabin layouts are shown in fig. 1.5 for the basis projected aircraft and in fig. 1.6 for the stretched family version. The cabin layout in this version considered a number of 60 passengers in the business class, which is more then in the requirements, but this still guarantees compliance, since the space of these additional business class seats can be swapped with a row of economy or premium economy class.

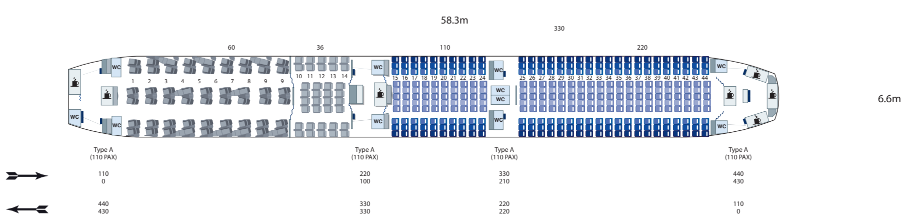


Figure 1.5: Cabin Layout of the projected basis aircraft.

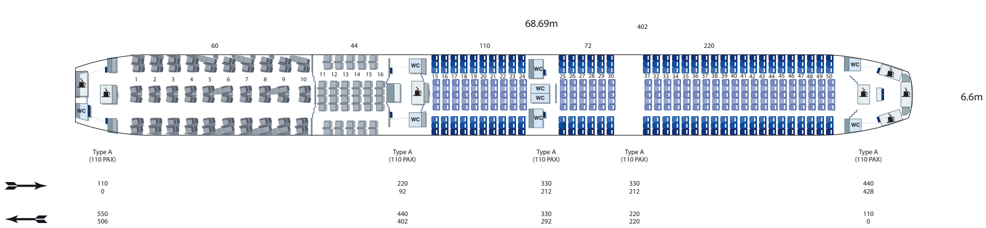


Figure 1.6: Cabin Layout of the stretched family version aircraft.

2 Initial weight estimation and flight performance

2.1 Initial Weight Estimation

For the initial weight estimation the formula according to Raymer [5] should be applied, given in eq. (2.1).

$$m_0 = \frac{m_{\text{crew}} + m_{\text{payload}}}{1 - \frac{m_f}{m_0} - \frac{m_e}{m_0}} \tag{2.1}$$

Before the use of this estimation equation it was calibrated to the provided reference aircraft data given in the seminar's task description. The derived data is given in following table 2.1:

Mass	[kg]
$m_{0\ \text{Ref Ac}}$	377000
$m_{\text{Max payload}}$	55000
Fraction	**[-]**
(m_f/m_0)	0.358
(m_e/m_0)	0.499

Table 2.1: Reference Aircraft Data

With the given data a calibration factor f_{CalW} of 1.024 was obtained. This factor was then used for the actual weight estimation of the projected aircraft. Before the necessary operational fuel fractions had to be selected according [6] seen in table 2.2 and for the mission fuel calculated with the Breguet range equation [5], presented in eq. (2.2)

$$R = \frac{v_{\text{cruise}}}{c_{\text{TL}}} \cdot \frac{c_{\text{L}}}{c_{\text{D}}} \cdot \ln m_{5\_4} \tag{2.2}$$

Which provides:

$$m_{5\_4} = \exp\left(\frac{R \cdot c_{\text{TL}}}{v_{\text{cruise}}} \cdot \frac{c_{\text{D}}}{c_{\text{L}}}\right) \tag{2.3}$$

Mission segment	Fuel fraction	[-]
Engine start / Warm up	m_{1_0}	0.99
Taxi	m_{2_1}	0.99
Take off	m_{3_2}	0.99
Climb	m_{4_3}	0.995
Cruise [Breguet]	m_{5_4}	0.66
Descent	m_{6_5}	0.98
Missed approach and climb	m_{7_6}	0.988
Landing / Taxi / Shut down	m_{10_9}	0.992

Table 2.2: Reference Aircraft Data

To obtain the m_{MTOW} for the projected aircraft 2.1 is combined with a formula for empty weight estimation according Raymer [5], given in eq. (2.4). This two equations add up to an iterative system. Regarding the literature the values 1.02 for A and -0.06 for C are chosen.

$$(m_e/m_0) = A \cdot m_{\text{MTOW}}^C \tag{2.4}$$

Also eq. (2.4) were calibrated with a factor of $f_{CalE} = 1.082$.
A difference in the projected aircraft to the reference aircraft was the selection of 3 pilots, since this is common in long range air travel. This resulted in a crew number of 18. The first step then in the iteration process was to define an empty weight fraction which enables a calculation of m_{MTOW} with eq. (2.1) and then a new empty weight fraction can be calculated. The result of this first initial weight estimation process was a m_{MTOW} of 377 656.92 kg.

2.2 Aircraft Performance

To evaluate the performance of the projected aircraft the IFB provided the "Flight performance tool". It is an Excel-Sheet were all data, defined by the Top Level Requirements of the task description or obtained by the further design process outlined in the following sections, can be entered. The first result with use of the $m_{\text{MTOW}} = 377656.92$kg and an Ultra Fan with expected thrust of 140lbf (622.8kN) at take off conditions is presented in fig. 2.1
In the final design the projected aircraft contained the design parameters presented in table 2.3. These values derived from the executed tasks outlined in the the following chapters. The design point was then selected, that the family concept version (MTOW 353 618 kg) is still in the design window. Furthermore the design point of the actual projected basis aircraft with MTOW 294 681 kg should be not too far from the wing loading for maximum range. The sizing charts for projected and family concept aircraft are shown in fig. 2.2 and fig. 2.3.

Wing surface area	521.7 m^2
Static sea level thrust	600 kN
MTOW	294 681 kg
MTOW family concept	353 618.0 kg

Table 2.3: Data of projected aircraft

2.2. AIRCRAFT PERFORMANCE

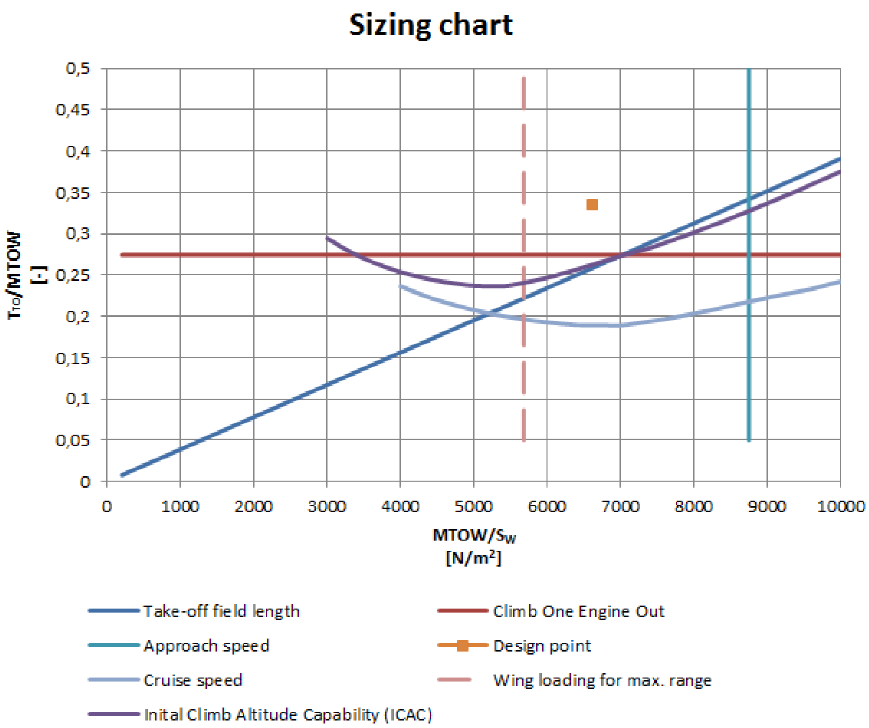


Figure 2.1: Sizing chart of Flight Performance Tool with first own design inputs of wing loading ($m_{\text{MTOW}} = 377\,656.92$ kg and reference aircraft wing area) and Thrust to Weight Ration (with 140lbf of Ultra Fan engine.

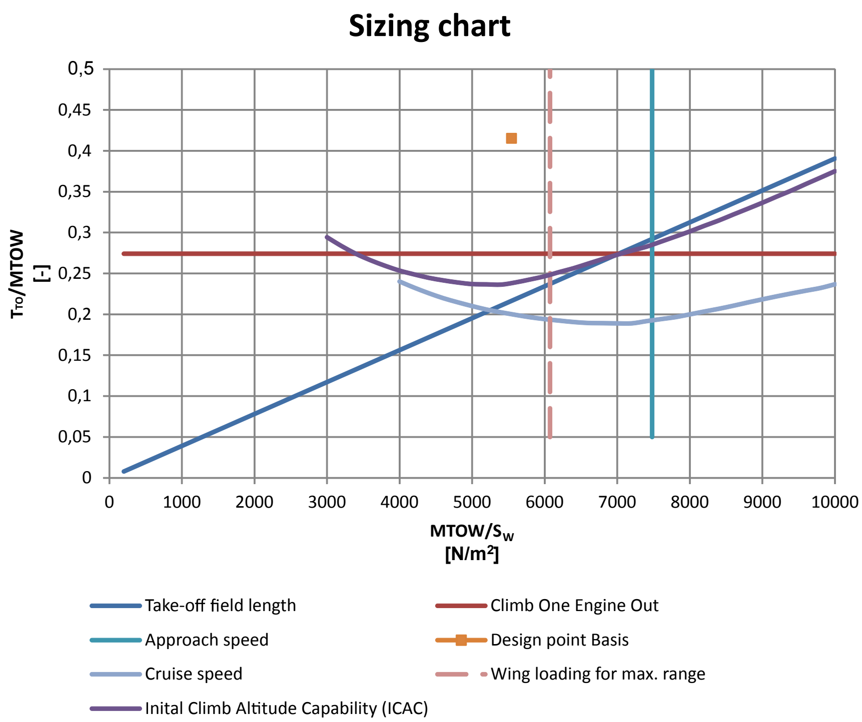


Figure 2.2: Sizing chart of Flight Performance Tool with projected design inputs. (MTOW 294 681 kg)

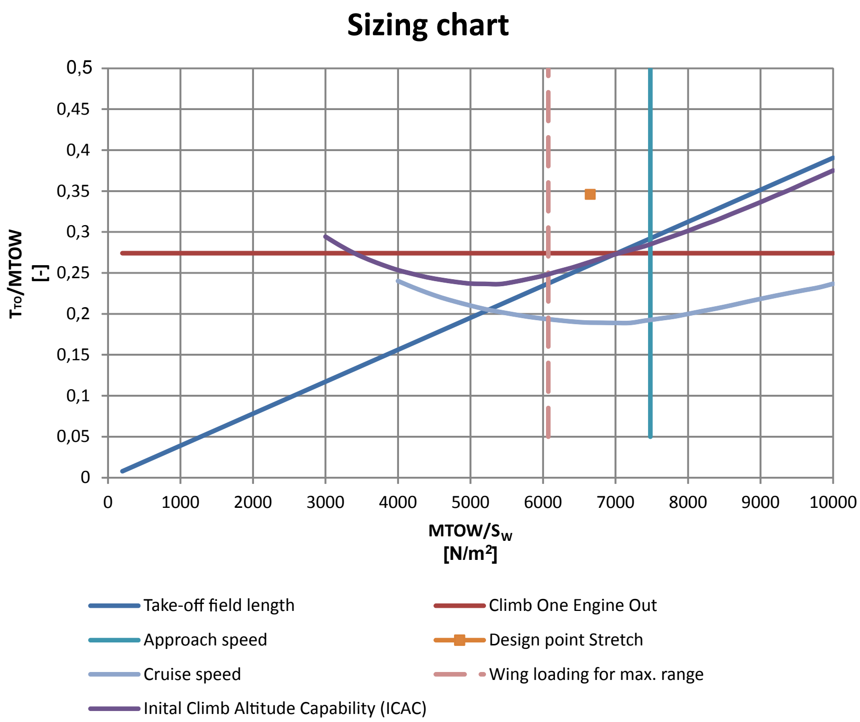


Figure 2.3: Sizing chart of Flight Performance Tool with family concept design inputs. (MTOW 353 618 kg)

Aerodynamics & flight mechanics

3

In this chapter, the aerodynamic investigations and flight mechanics considerations are carried out.

3.1 Selection of Airfoils

The first step was to find suitable airfoils for the wing, horizontal and vertical stabilizers. A large collection of airfoils is provided on the website `airfoiltools.com`. This website also has a built-in analysis function, which provides polar plots of each airfoil. All the polar diagrams currently available have been produced using Xfoil. The diagrams can be put into comparison to determine the best fitting profile.
The airfoils have been analysed at Reynolds number of one million, which is the maximum available of the built-in system, a Turbulence factor N_{crit} of nine and a Mach number of zero. This settings do not represent the expected flow conditions, but at least provide a qualitative statement.
All of the profiles examined have in common that they are supercritical airfoils. A supercritical airfoil is an airfoil with a flattened upper surface and a highly cambered aft section. This causes the compression shock to occur further back and weaker and therefore the wave drag is reduced. This scheme is shown in figure 3.1.
In order to evaluate the airfoils, several criteria are considered. First, the airfoil should have good stall characteristics, that is the lift coefficient drop after the maximum point should not be abrupt. The same should also be true for the moment coefficient. Second, the airfoil should not have a high moment coefficient. A higher moment from the wing would require bigger counter moment from the tail, which leads to a higher trim drag. The airfoils being compared here are listed in table 3.1 for both wing and empennage and shown in figure 3.2 for the wing with the corresponding polars in figure 3.3 and in figure 3.4 and 3.5 for the empennage, respectively.

[1]Source: `https://www.nasa.gov/centers/armstrong/news/FactSheets/FS-044-DFRC.html` 04.01.20

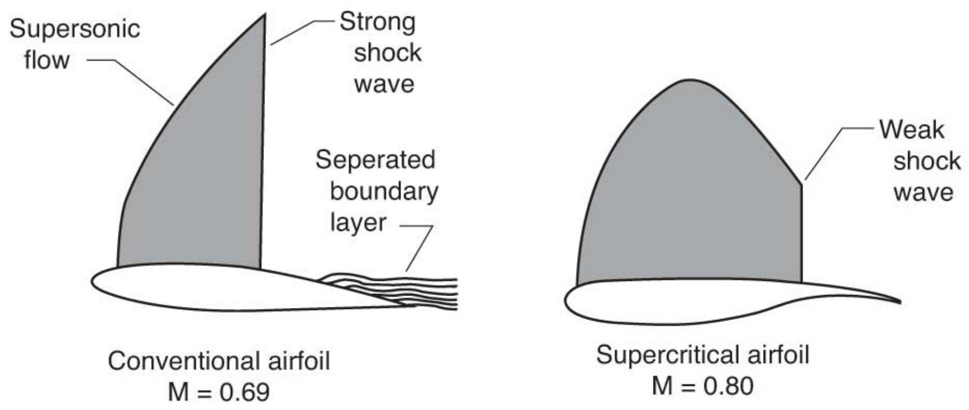


Figure 3.1: Airflow over different Airfoil types[1]

Table 3.1: Compared Airfoils

Airfoil	max thickness	max camber
Reference Aircraft/CRM.eta65.unswept31.5deg.sharp.te	11.4% at 39.1% c	1.8% at 82% c
NYU/Grumman K-1	11.6% at 33% c	1.9% at 75% c
NASA SC(2)-0712 (sc20712)	12% at 37% c	2.2% at 81% c
NASA SC(2)-0714 (sc20714)	14% at 37% c	2.5% at 81% c
Whitcomb Integral Supercritical	11% at 35% c	2.4% at 82.5% c
NACA 0012	12% at 30% c	symmetrical
NASA SC(2)-0012	12% at 37% c	symmetrical
NASA 0011 sc	11% at 40% c	symmetrical

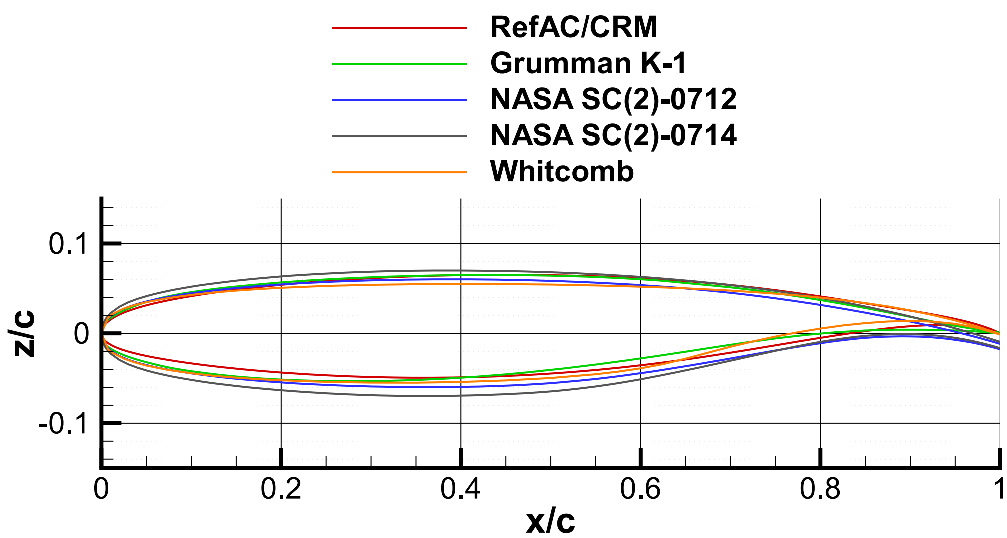
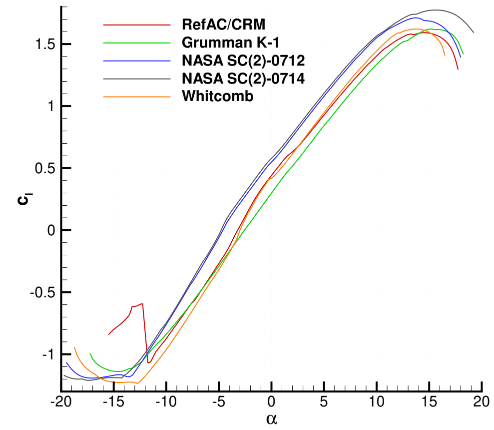


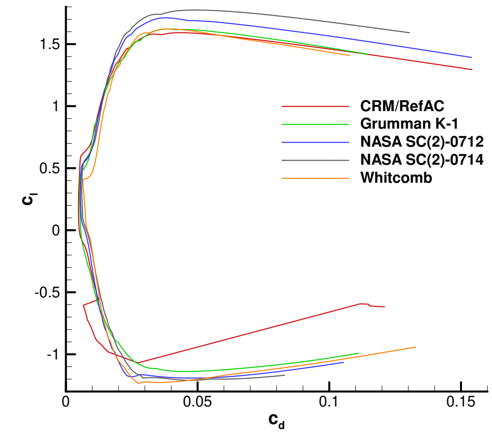
Figure 3.2: Compared Airfoils for the wing

These polars are taken from `airfoiltools.com` except the one of the Reference Aircraft. This one was calculated using XFOIL. The reason for the jags and outliers is that the program has not found a convergent solution for some points.

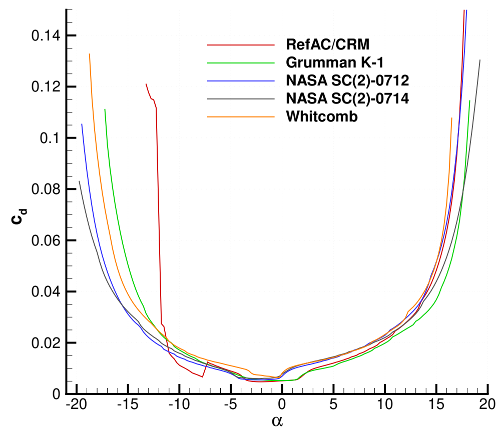
The diagrams show that the selected airfoils have roughly similar characteristics and each has advantages and disadvantages. At a closer look, the Whitcomb profile shows the worst



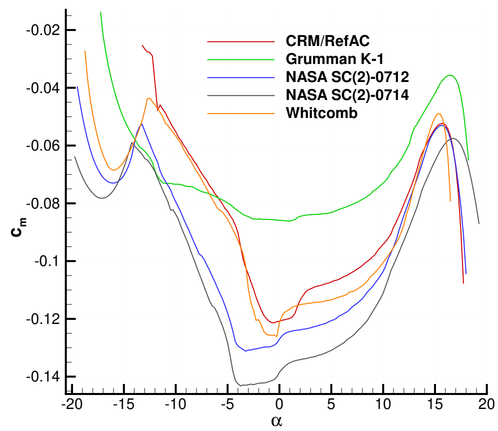
(a) c_l-α



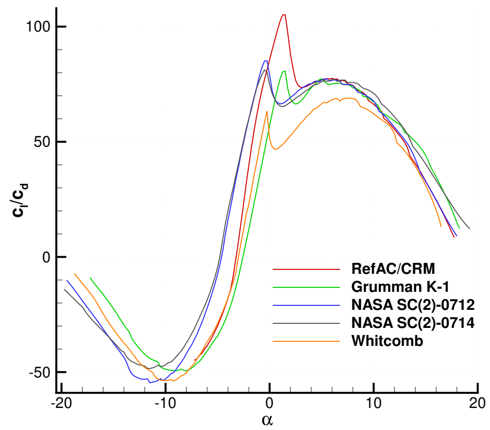
(b) c_l-c_d



(c) c_d-α



(d) c_m-α



(e) c_l/c_d-α

Figure 3.3: Airfoil comparison diagrams for the wing

performance in almost all areas, whereas the Reference Aircraft or NASA airfoils tend to have the best properties. Taking into account that, according to Raymer [5], the wing structural

weight varies approximately inversely with the square-root-of-the-thickness ratio, the NASA SC(2)-20714 profile provides an advantage because it is the thickest.
A good compromise in most aspects seems to be the NASA SC(2)-20712 airfoil, therefore it is used for the further design of the wing.

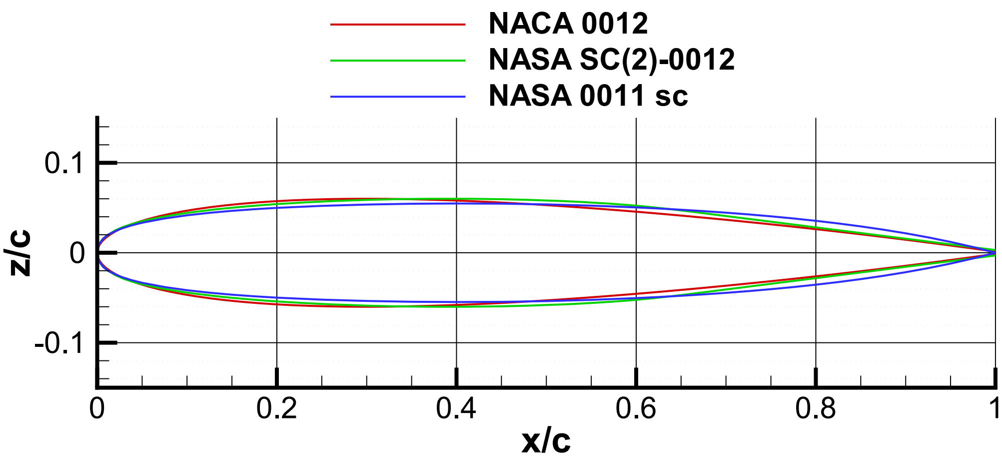
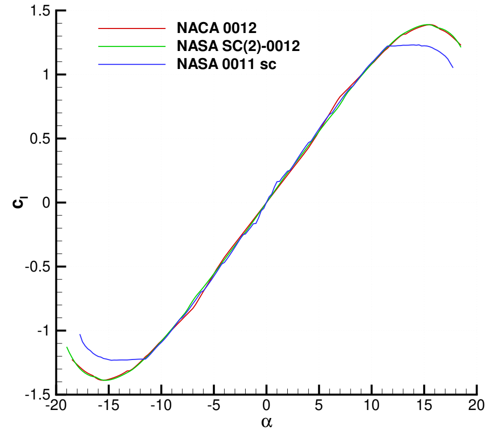
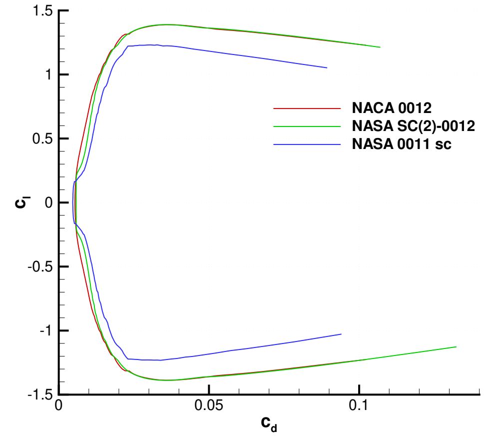


Figure 3.4: Compared Airfoils for the empennage

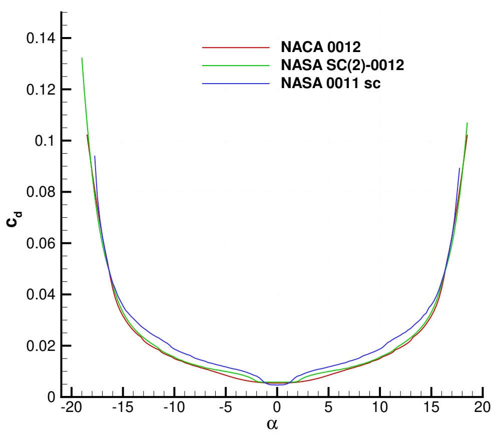
The same considerations have been done for the empennage. The profiles compared show almost the same lift curve with the difference that the NASA 0011 sc airfoil has a lower $c_{l_{max}}$ value. The same applies to the drag curve. Here the NASA 0011 sc shows higher values for almost every angle of attack. This results in worse values for the c_l-c_d-Polar. The NACA 0012 and NASA SC(2)-0012 show almost the same characteristics in all areas. This may be because the diagrams are calculated with a Mach number of zero. At higher Mach numbers the NASA SC(2)-0012 will may have advantages considering drag because it was designed for it. Therefore it is used for the further design of the empennage.



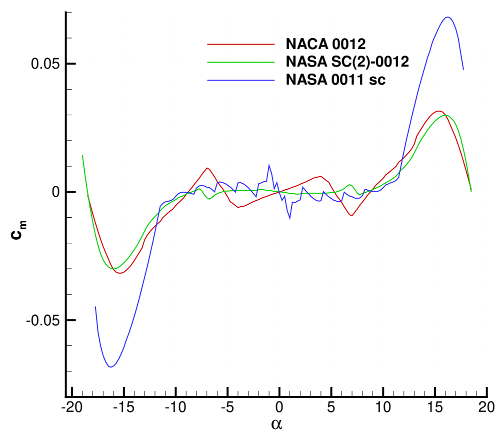
(a) c_l-α



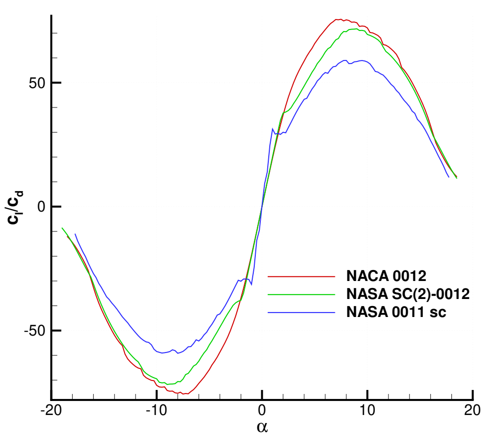
(b) c_l-c_d



(c) c_d-α



(d) c_m-α



(e) c_l/c_d-α

Figure 3.5: Airfoil comparison diagrams for the empennage

3.2 Inflow conditions and requirements from Flight Performance Tool

The calculations for cruising should been carried out under typical cruising conditions that are listed in table 3.2 for flight level 330. The calculation of this conditions can be found in the appendix 12.

Table 3.2: Inflow condition for cruising

Mach number	0.8
Reynolds number	64.17×10^6
kinematic viscosity	$3.53 \times 10^{-5}\ \mathrm{m^2\,s^{-1}}$
Cruising speed	$239.407\ \mathrm{m\,s^{-1}}$

Unfortunately, the Aero tool is not able to calculate with this settings with activated compressibility correction for that high Mach number. After trying out, a Mach number of 0.4 delivers results without error messages. The input parameters for the Aero tool for cruising conditions are depicted in figure 3.6 . The calculations were carried out for angles of attacks from -0.172 up to 4.172, as the program does not provide convergent solutions for larger areas. Anyway this covers the typical cruising flight conditions.

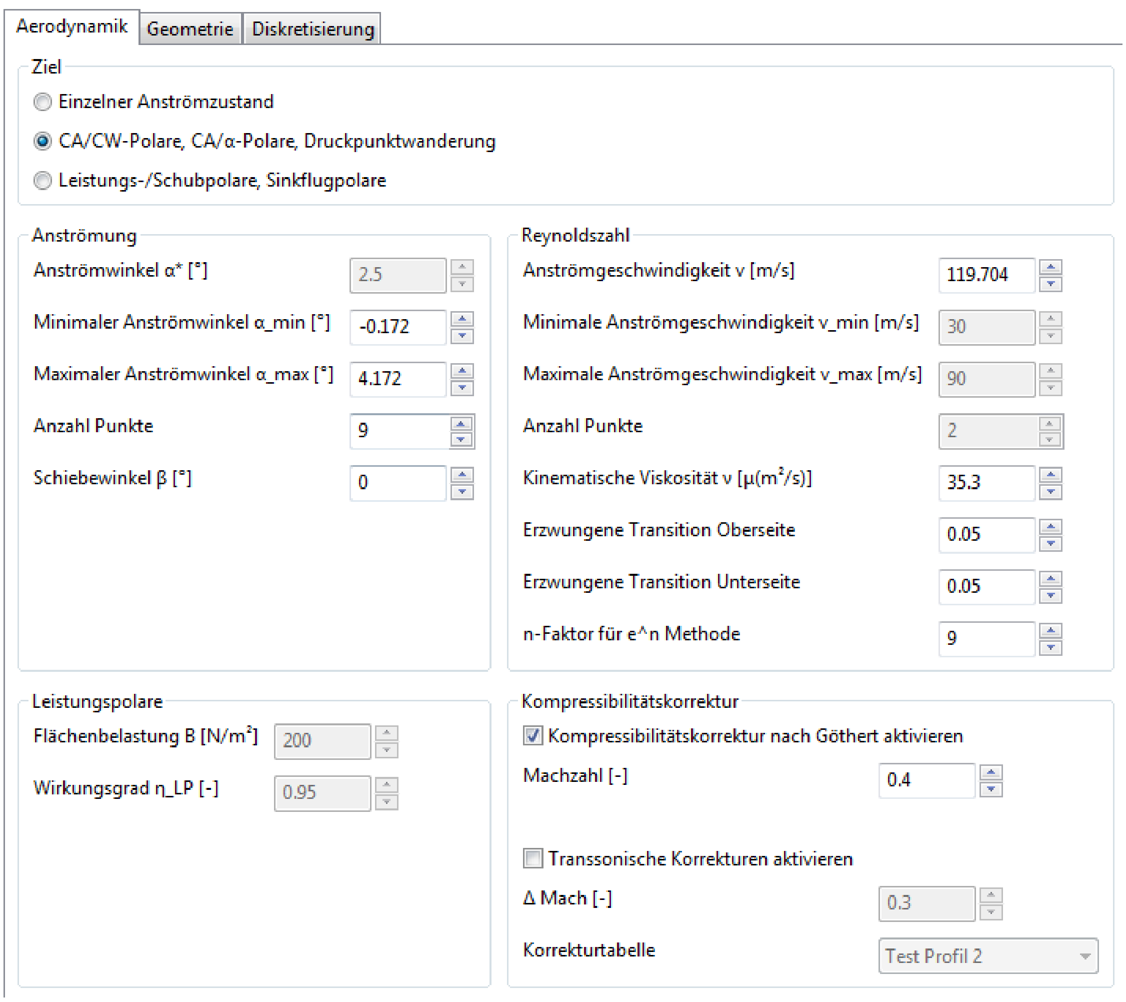


Figure 3.6: Input parameters Aero tool

For takeoff and landing, the typical inflow conditions at sea level are listed in table 3.3. A

Mach number of 0.22 was assumed as this covers the stallspeed and is below the required approach speed of 150 kn.

Table 3.3: Inflow condition for takeoff and landing

Mach number	0.22
Reynolds number	48.52×10^{6}
kinematic viscosity	$1.46 \times 10^{-5}\,\mathrm{m^2\,s^{-1}}$
Cruising speed	$74.88\,\mathrm{m\,s^{-1}}$

Also the Flight Performance Tool delivers data, which is important for the aerodynamic analysis. This includes the wing area and the lift coefficient for cruising speed. Wing area is derived from the estimated maximum take off weight and the defined wing loading at the design point. Lift coefficient at cruise is settled by the required cruise speed, and then also dependent by the estimated weight and the wing area, resulting from the design point. These number can be seen as targets, but will most likely change during the design process, also caused from results in the aerodynamic investigation.

Table 3.4: Start parameters provided from the Flight Performance Tool

Wing Area m_{MTOW}	$560\,\mathrm{m^2}$
Lift coefficient cruise $c_{\text{L cruise}}$	0.58

The start parameters in table 3.4 are defined by the initial weight estimation and data from the reference aircraft, to be exact by its Drag-Lift polar and wing area.

3.3 Drag calculation

In order to find out the calibration factor for the calculations, the drag polar of the reference aircraft was recalculated and compared to the polar given in the task sheet.
The parasite (zero-lift) drag will be estimated using the component buildup method according to Raymer [5]. With this method, the drag for the individual aircraft components is determined and then added up to the total parasite drag.

$$c_{D_0,subsonic} = \frac{\sum(c_{f_c} FF_c Q_c S_{wet_c})}{S_{ref}} + c_{D,misc} + c_{D,L\&P} \tag{3.1}$$

The component friction can be calculated with the flat-plate skin-friction coefficient, which includes a Mach number correction

$$C_{f,turb} = \frac{0.455}{(\log(Re))^{2.58} \cdot (1 + 0.144 \cdot Ma^2)^{0.65}} \tag{3.2}$$

As the flat-plate skin-friction is dependent of the flow state, we assume the flow to be fully turbulent. This is a conservative assumption, since not all areas have a turbulent flow. In order to take the pressure drag through flow separation into account, a form factor for wing, tail, strut and pylon is determined by

$$FF = \left(1 + \frac{0.6}{(x/c)}\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4\right)\left(1.34 \cdot Ma^{0.18} \cdot (\cos\Lambda_m)^{0.28}\right) \tag{3.3}$$

and for the fuselage by

$$FF = \left(0.9 + \frac{5}{(l/d)^{1.5}} + \frac{(l/d)}{400}\right) \tag{3.4}$$

Typical literature values were used for the interference factors Q. Miscellaneous drags are left out and will be considered later at landing configuration. Leakage and protuberance drag is estimated by adding 2% on the overall parasite drag. According to Torenbeek [7], the wave resistance was taken into account with a value of Δc_D=0.0015.
For the estimation of trim drag the force system on the aircraft had to be examined. Therefore by subtracting the entire aircraft's weight force from the lift force of the wing the quantity of the needed downforce of the horizontal tail plane can be obtained. The needed calculations can be derived on the one hand from the equilibrium of forces in the perpendicular direction of stationary horizontal flight (3.5) and the definition of the lift force (3.6).

$$\sum_i F_i = L_{wing} + W + L_{HTP} = 0 \tag{3.5}$$

$$L_i = \frac{\rho}{2} \cdot v_{cruise}^2 \cdot S_i \cdot c_{L_i} \tag{3.6}$$

With the determined lift force of the horizontal tailplane its lift coefficient $c_{L_{HTP}}$ can be obtained. With the known $c_{L_{HTP}}$ the drag coefficient of the horizontal tail plane $c_{D_{HTP}}$ in cruise flight can be identified by the c_L-c_D-polar of the reference aircrafts HTP presented in section 3.9 in fig. 3.24. As this polar already contains the zero-lift drag of the horizontal tail plane, it must not be calculated separately. But the interference drag must be added and is considered with a 5% value in this case.
Lastly the $c_{D_{HTP}}$ had to be referred to the reference area by (3.7) to obtain actual trim drag of the aircraft.

$$c_{D_{Trim}} = c_{D_{HTP}} \cdot \frac{S_{HTP}}{S_{ref}} \tag{3.7}$$

The drag breakdown from the first iteration can be found in table 3.5. The difference between the Reference Aircrafts c_{D_0}=0.013 and the one calculated with the described textbook methods is about 14.352%. This seems reasonable because the formulas deliver a conservative estimate. For the further design, the parasite drag will be decreased by this calibration value. The additional drag of the landing gear and flaps is considered separately in the subsection 3.7.1.

Table 3.5: Drag breakdown Reference Aircraft

Component	S_{wet} $[m^2]$	L $[m]$	Re [-]	c_f	FF	Q	c_{D_0}
Wing	1018.888	9.461	6.42E+07	2.14E-03	1.475	1	0.005743
VTP	110.708	5.6495	3.83E+07	2.31E-03	1.518	1.05	0.000727
Fuselage	1277.186	71.394	4.84E+08	1.63E-03	1.069	1	0.003964
Pylons	15	3.5	2.37E+07	2.48E-03	1.5	1.1	0.000110
Nacelles	146	5.812	3.94E+07	2.30E-03	1.3	1.2	0.000934
Sum							0.011478
Including wave drag with Δc_D=0.0015							0.012978
Including trim and HTP interference drag with $c_{D_{Trim}}$=0,001596							0.014574
Including leakage and protuberance drag with 2%							0.014866
c_{D_0} exclusive wing and wave drag, calibrated							0.006529

For the final iteration the parasite drag was calculated with the own component values. The breakdown for that is provided in table 3.6.

Table 3.6: Drag breakdown own calculation

Component	S_{wet} $[m^2]$	L $[m]$	Re [-]	c_f	FF	Q	c_{D_0}
Wing	932	8.537	5.79E+07	2.17E-03	1.475	1	0.005332
VTP	112	5	3.39E+07	2.35E-03	1.518	1.04	0.000742
Fuselage	1322	74	5.02E+08	1.62E-03	1.063	1	0.004059
Pylons	16	4	2.71E+07	2.43E-03	1.5	1.1	0.000115
Nacelles	218	6.4	4.34E+07	2.27E-03	1.3	1.2	0.001376
Sum							0.011623
Including wave drag with Δc_D=0.0015							0.013123
Including trim and HTP interference drag with $c_{D_{Trim}}$=0,001596							0.014704
Including leakage and protuberance drag with 2%							0.014998
c_{D_0} exclusive wing and wave drag, calibrated							0.006994

3.4 Polars from Reference Aircraft/Calibration Factors

The drag polar of the wing can be obtained from the Aero Tool. This polar already contains the parasite drag of the wing, of course. In order to receive the polar for the overall aircraft, the calibrated parasite drag of the remaining aircraft components excluding wave drag must be added. Each angle of attack is then compared to the data from the reference aircraft to determine the calibration factors. These factors take into account, among other things, the problem that the calculations were carried out with a reduced inflow velocity, as described in section 3.2. They remain the same for each calculation and are listed in appendix 12 in table 12.1.
Now the polars could be recalculated. This was done with the wing geometry of the reference aircraft but the airfoil is changed to the ones described in section 3.1. The two airfoils which show the best performance are displayed in figure 3.7 with the Grumman-K1 airfoil and in figure 3.8 for the NASA SC 20712.

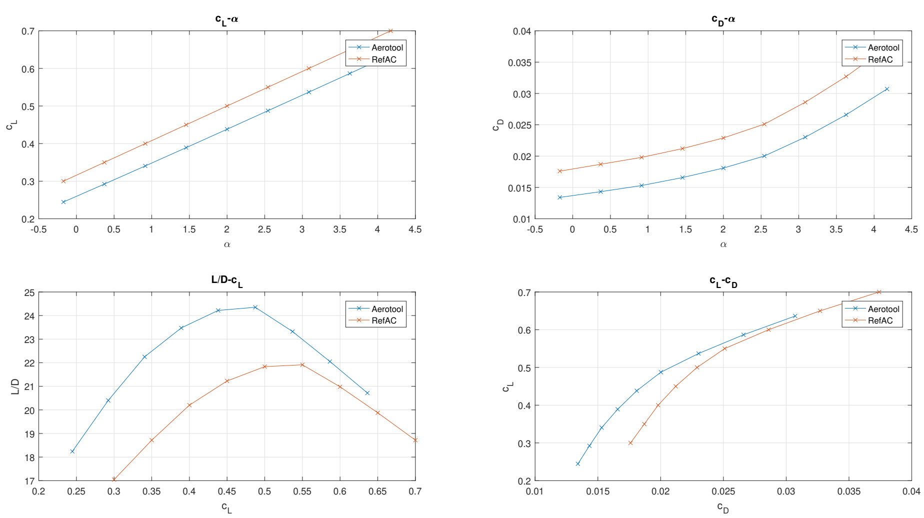


Figure 3.7: Reference Aircraft polar with the Grumman K-1 airfoil

It can be seen that the Grumman-K1 airfoil has better characteristics at lower lift coefficient values whereas the NASA SC 20712 shows better properties at higher values. Therefore a combination of both airfoils is planned. The Grumman-K1 at the wing root and the NASA SC20712 at the remaining crosssections.

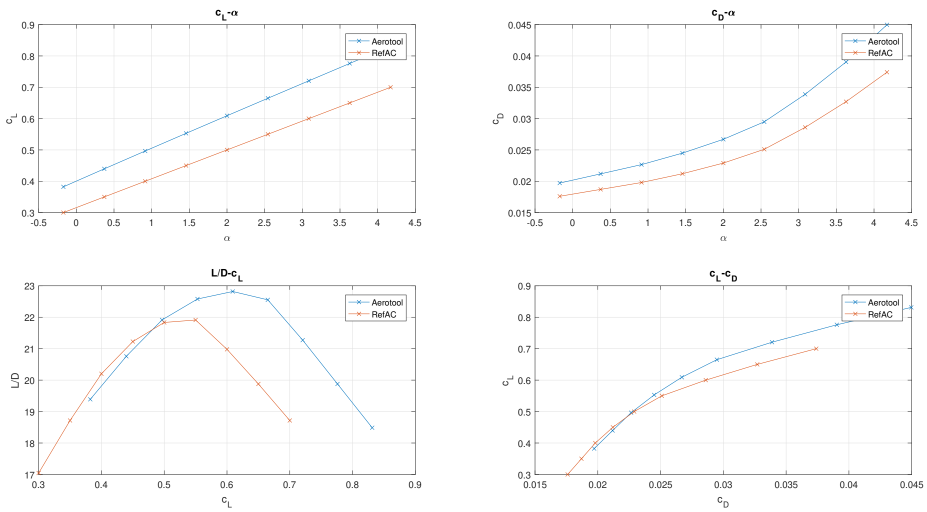


Figure 3.8: Reference Aircraft polar with the NASA SC 20712 airfoil

3.5 Determination of suitable planforms

The design of the wing planform based on the geometry of the reference aircraft provided. In order to achieve a better performance, the wing span was increased to 80 m with slightly decreased wing area. In addition, the leading edge sweep has been set to 30° and winglets were applied as figured out in section 7.4. The entire wing is shown in figure 3.9. The red line in this picture marks the mean aerodynamic chord (MAC). The calculation for this is explained in the next section. The geometric data is provided in table 3.7.

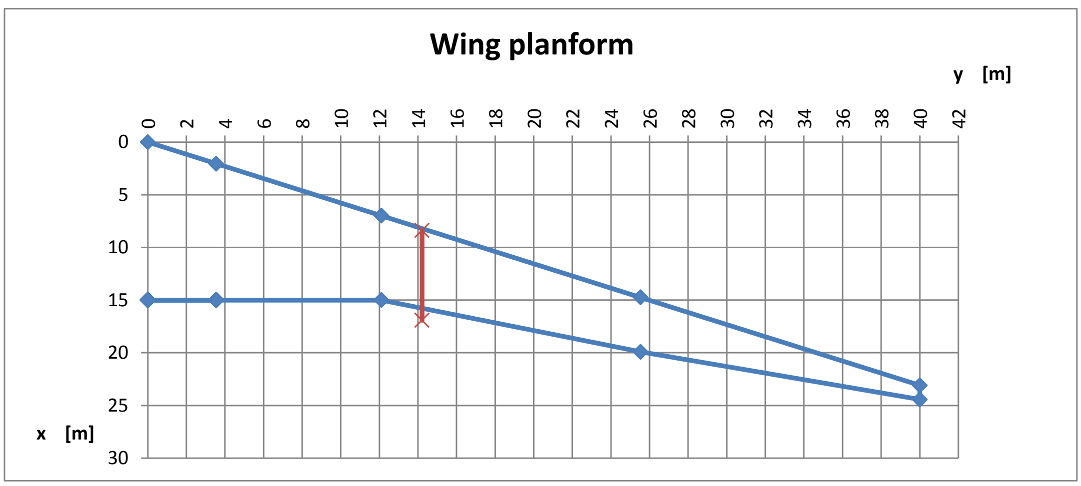


Figure 3.9: Wing planform with MAC from the first iteration

3.5.1 Mean Aerodynamic Chord (MAC)

For the calculation of the MAC the wing planform was divided into four trapezial segments. For each trapezoid, the area is calculated using the formula

$$S_{tra,i} = \frac{(c_r + c_t)}{2} \cdot b_{1/2} \tag{3.8}$$

Table 3.7: Geometric wing data

wing span	80 m
wing area	521.7 m^2
aspect ratio	12.27
taper ratio	0.108
sweep leading edge	30°
sweep 1/4 chord	26.21°
MAC	8.511 m

in which c_r represents the root chord length of the wing segment, c_t the tip chord length and $b_{1/2}$ the length of the segment.
According to Scholz [8], the MAC of a trapezium is governed by the equation

$$c_{mac,i} = \frac{2}{3} \cdot \frac{(c_r^2 + c_r \cdot c_t + c_t^2)}{c_r + c_t} \tag{3.9}$$

Further, the spanwise location of the MAC is determined by

$$y_{mac,i} = \frac{b_{1/2}}{3} \cdot \frac{2c_t + c_r}{c_t + c_r} \tag{3.10}$$

and the chordwise location by

$$x_{mac,i} = \frac{a}{3} \cdot \frac{2c_t + c_r}{c_t + c_r} \tag{3.11}$$

where a is the sweep of the leading edge as the distance from the reference point to the tip of the segment.
The entire MAC can be calculated from the individual sections as follows:

$$c_{mac} = \frac{1}{S_{tra}} \cdot \sum_i c_{mac,i} \cdot S_{tra,i} \tag{3.12}$$

where

$$S_{tra} = \sum_i S_{tra,i} \tag{3.13}$$

The same applies to the spanwise and chordwise location:

$$y_{mac} = \frac{1}{S_{tra}} \cdot \sum_i y_{mac,i} \cdot S_{tra,i} \tag{3.14}$$

$$x_{mac} = \frac{1}{S_{tra}} \cdot \sum_i x_{mac,i} \cdot S_{tra,i} \tag{3.15}$$

where $y_{mac,i}$ and $x_{mac,i}$ always refers to the reference point of the wing.
This principle of calculating the MAC and the variables are depicted in figure 3.10

[2] Source: `http://walter.bislins.ch/blog/index.asp?page=Bild%3AWingFofY%2Epng` 21.01.20

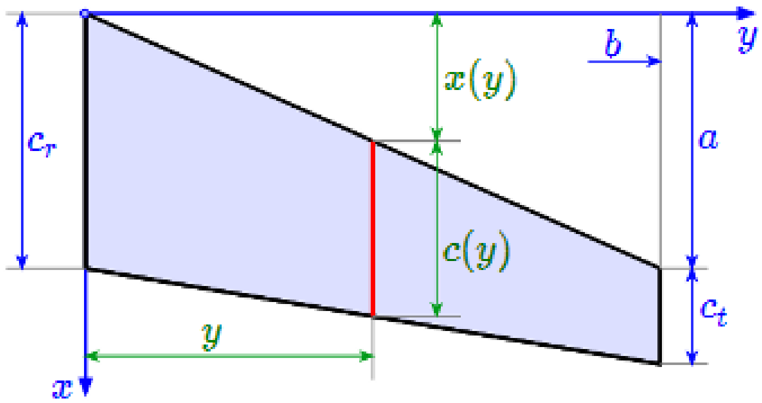


Figure 3.10: MAC calculation of a trapezoidal wing[2]

3.6 High lift devices, slats and ailerons

From the flight performance tool, a maximum lift coefficient of $c_{L,ldg}$ =2.5 for landing approach was obtained. In order to achieve this value there are some high lift devices necessary. As the design goal is to reduce fuel burn, there is scope for improvement in this area. Therefore, we use a new technology called morphing structures. The aim of morphing structures is to improve aerodynamics. On the other hand, one disadvantage is that the weight will increase due to more complicated actuator technology. This will be discussed later in chapter 6 Structure and loads. Morphing structures are considered in detail in [9] and [10], for example.

In order to estimate the areas required for the flap design, they are treated as a Plain flap. According to Raymer [5], the change in maximum lift coefficient and zero-lift angle is given by

$$\Delta c_{Lmax} = 0.9 \cdot \Delta c_{lmax} \cdot \frac{S_{flapped}}{S_{ref}} \cdot \cos \Lambda_{H.L.} \tag{3.16}$$

and

$$\Delta \alpha_{0L} = (\Delta \alpha_{0L})_{airfoil} \cdot \frac{S_{flapped}}{S_{ref}} \cdot \cos \Lambda_{H.L.} \tag{3.17}$$

with

$$\Delta c_{lmax} = 0.9 \tag{3.18}$$

The flaps extend over almost the entire span and are divided in 3 sections to provide redundancy. Between the fuselage and the inner section there is some space left out to avoid collision. The outer area works simultaneously as aileron, a so called flaperon. It ends 2 meters before the wing tip as this last segment would provide little control effectiveness due to the vortex flow at the wing tips. The segmentation is pictured in figure 3.11.

In total, this system delivers the values given in table 3.8.

A sketch about how this morphing flap structure could look like is shown in figure 3.12 and 3.13.

3.6. HIGH LIFT DEVICES, SLATS AND AILERONS

Figure 3.11: flap and slat location

Table 3.8: Lift contribution of high lift devices

	takeoff	landing
Δc_{Lmax}	0.5762	0.5762
lift increment % Δc_{Lmax}	60 %	90 %
Δc_{Lmax}	0.3457	0.5186
$+\Delta c_{Lflaps}$	0.4	0.4
totalΔc_{Lmax}	0.7457	0.9186
$(\Delta\alpha_{0L})_{airfoil}$	$-10°$	$-15°$
$\Delta\alpha_{0L}$	$-7.114°$	$-10.67°$

Figure 3.12: flap structure: morphed up (left hand side), morphed down (right hand side) (taken from [10])

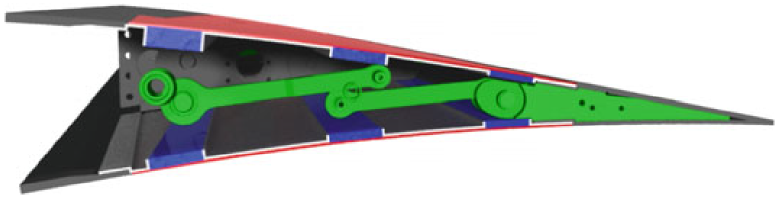


Figure 3.13: flap structure mechanism (taken from [10])

Morphing structures that function as slats are also provided for the leading edges. As depicted in figure 3.11, they extend almost over the whole leading edge. Possible problems and synergies with the de-icing system have to be mentioned at this point, but are not considered any further. According to Raymer [5] the lift increment for a leading-edge extension can be crudely estimated as 0.4 at high angles of attack. A possible concept for morphing slats is shown in figure 3.14.

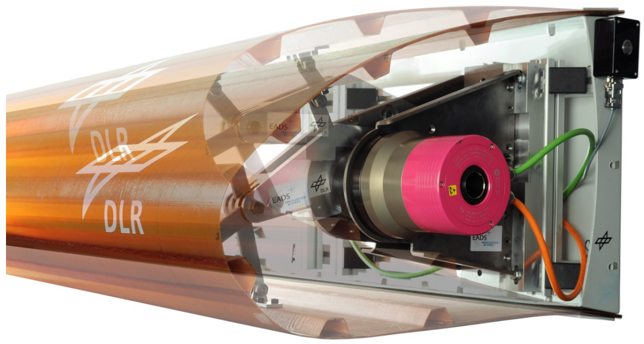


Figure 3.14: Example of morphing mechanism inside leading edge (taken from [10])

3.7 Aerodynamic aircraft characteristics

For the cruise condition, the calculation was carried out with the Aero Tool and calibrated using the data of the reference aircraft, as explained in section 3.4. Unfortunately there is no comparative data given from the reference aircraft for takeoff and landing conditions except the $c_{L_{max}}$. Hence, the polars are not calibrated for this and can only be regarded as very rough estimates. In addition, the Aero tool does not make any reasonable predictions for the stall. Therefore, textbook methods are applied to determine the maximum lift coefficient.

3.7.1 Landing gear and flap drag

The additional drag from the landing gear and the flaps is calculated according to Sun, 2018 [11].
Flap drag can be estimated with the formula

$$\Delta c_{D,flap} = 0.9 \cdot \left(\frac{c_{fp}}{c}\right)^{1.38} \left(\frac{S_{flapped}}{S_{ref}}\right) \cdot \sin^2 \delta \tag{3.19}$$

With a flap-to-wing-chord ratio of $\frac{c_{fp}}{c}$ = 0.3 and a flap-to-wing-surface ratio of $\frac{S_{flapped}}{S_{ref}}$ = 0.75 follow the values given in table 3.9. The values of the flap deflection angle are predicted by Raymer [5]. The values of the landing gear drag are estimated from Sun, 2018 [11]. The reason for different landing gear drag at takeoff and landing are different flow conditions around it caused by different $c_{L_{max}}$.

Table 3.9: Flap and landing gear drag

	takeoff	landing
flap deflection angle	40°	60°
$\Delta c_{D,flap}$	0.05295	0.09612
$\Delta c_{D,gear}$	0.019	0.013

The zero-lift drag of the aircraft is now added to these two drag components. This is done using the same method as described in section 3.3, but for the inflow condition during takeoff and landing. The drag breakdown for the final iteration can be found in table 3.10.

Table 3.10: Drag breakdown takeoff and landing

Component	S_{wet} $[m^2]$	L $[m]$	Re [-]	c_f	FF	Q	c_{D_0}
Wing	932	8.537	4.38E+07	2.38E-03	1.169	1	0.004640
VTP	112	5	2.56E+07	2.58E-03	1.203	1.04	0.000646
Fuselage	1322	74	3.80E+08	1.77E-03	1.063	1	0.004437
Pylons	16	4	2.05E+07	2.67E-03	1.5	1.1	0.000126
Nacelles	218	6.4	3.28E+07	2.49E-03	1.3	1.2	0.001511
Sum							0.0113614
Including trim and HTP interference drag with $c_{D_{Trim}}$=0,001596							0.0129422
Including leakage and protuberance drag with 2%							0.0132011
c_{D_0} exclusive wing, calibrated							0.0073322

The total there results a value for the calibrated parasite drag without wings but with flaps and landing gear of $c_{D_{0,AC,TO}}$ = 0.079282 for takeoff and $c_{D_{0,AC,LD}}$ = 0.116447 for landing configuration.

3.7.2 Polars

In fig. 3.15 the polars from the final iteration for the projected aircraft in comparison with the reference aircraft is shown. These polars derive directly from the calibration with given points of the reference aircraft. It indicates a more desirable characteristic in every polar. The $(c_L/c_D)_{max}$ is derived from this fig. 3.15, with a value of 27.27, where the improvement of the morphing wing technology is already included. The improvement due to this technology is taken into consideration with a value of $4.3\,\%$ as suggested from Suleman [12].
Needed Parameters for generation of additional polars in takeoff and landing configuration are shown in table 3.11. The value of c_{Lmax} of the wing is determined according to Roskam [6]. c_{Lmax} for takeoff and landing is then obtained by adding the total Δc_{Lmax} from section 3.6. $c_{D0,AC,TO}$ and $c_{D0,AC,LD}$ for take off and landing are determined in section 3.7.1. Then calculating the polar from Aero Tool for the projected wing in takeoff and landing conditions was the next step. With the wing polar and the parameters shown in table 3.11 the takeoff and landing polars can be derived.
To get a qualitative insight of the projected aircraft, these polars were extrapolated to the maximum lift coefficients given in table 3.11. The detailed calculation of this extrapolation can be found in appendix 12.
With fig. 3.16 it can be indicated the no tail strike is expected since the maximum angle of the projected configuration is $11°$, as can be seen in figure 3.17.
The aircraft characteristics shown in fig. 3.18 and fig. 3.19 are supposed to show the qualitative characteristics of the projected aircraft. This is hence the fact that the given range of

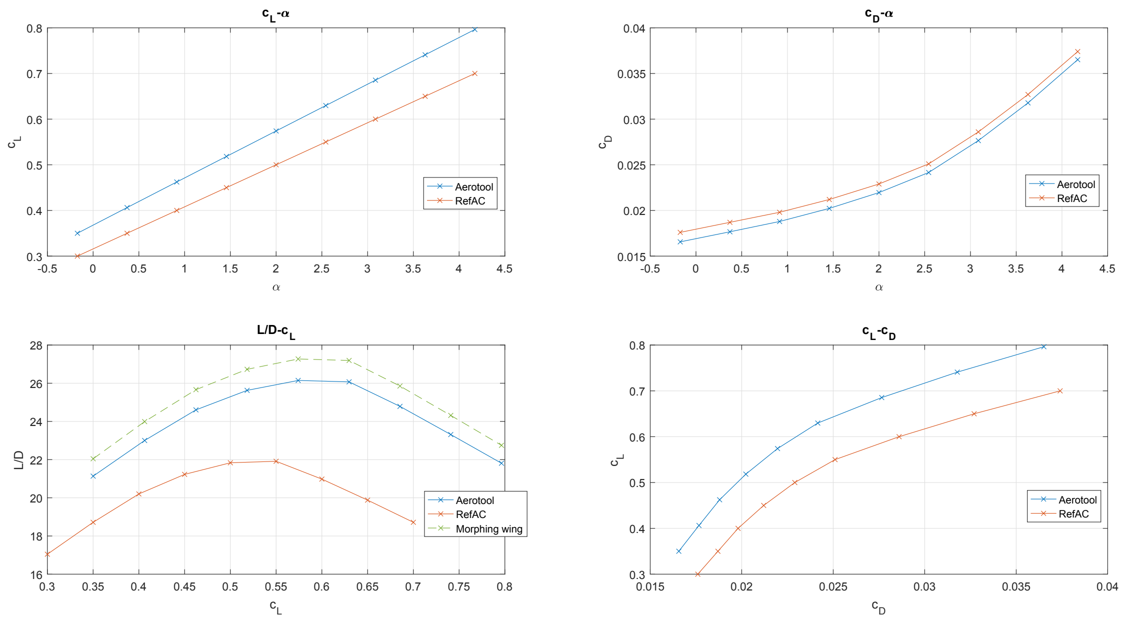


Figure 3.15: Polars in cruise condition

Table 3.11: Parameters for takeoff and landing polar

	takeoff	landing
c_{lmax}	1.6	1.6
c_{Lmax}	2.346	2.519
$c_{D0,AC}$	0.079282	0.116447
$\Delta\alpha_{0L}$	$-7.114°$	$-10.67°$

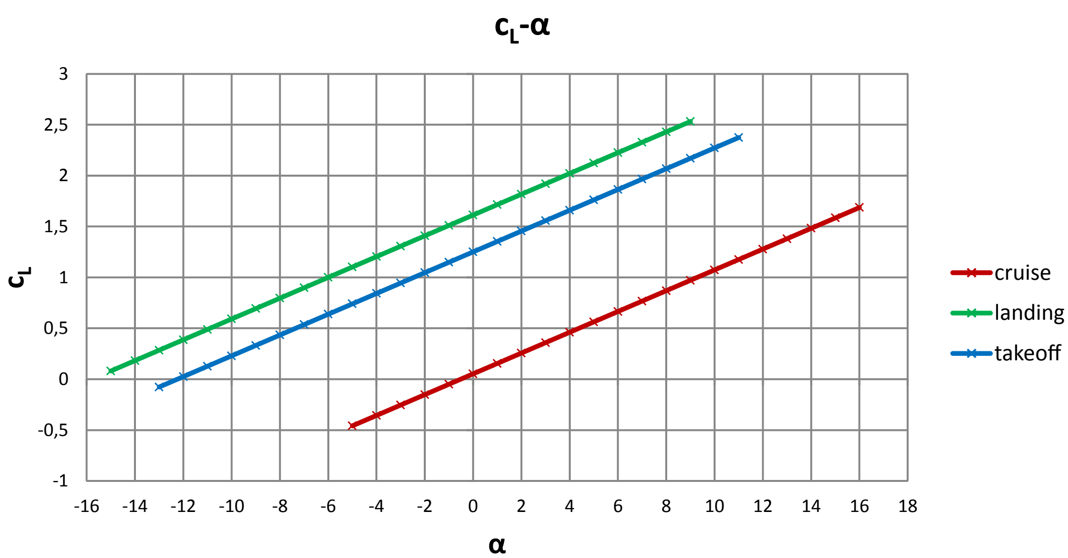


Figure 3.16: Lift slope for cruise, landing and takeoff configuration

reference aircraft aerodynamic data is in a very short range and the extrapolation is calculated in a much wider range. Therefore also the $(c_L/c_D)_{max\_takeoff}$ is shows a lower value than assumed. Because a calibration in this region of c_L would cause a higher $(c_L/c_D)_{max\_takeoff}$ the value of the reference aircraft is taken, with $(c_L/c_D)_{max\_takeoff}$=12.

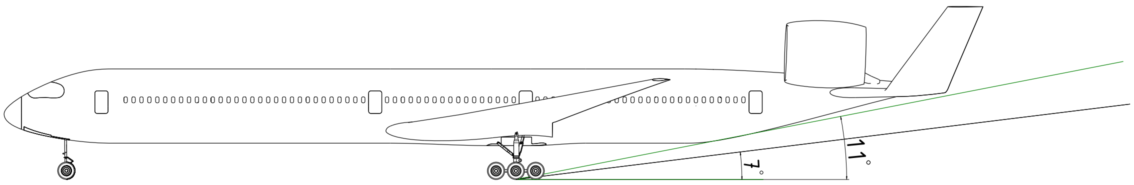


Figure 3.17: Examination of tailstrike

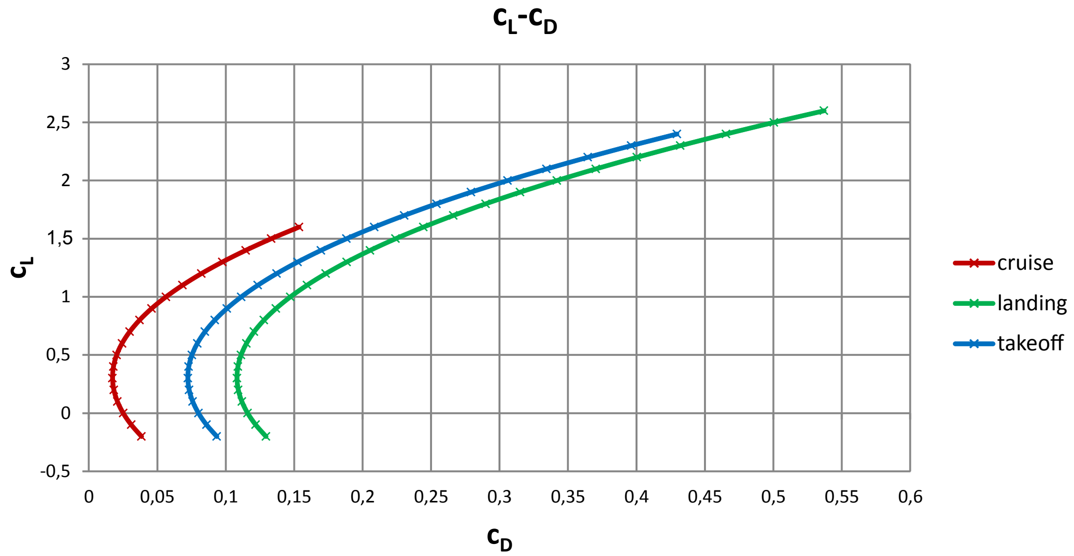


Figure 3.18: c_L-c_D polar for cruise, landing and takeoff

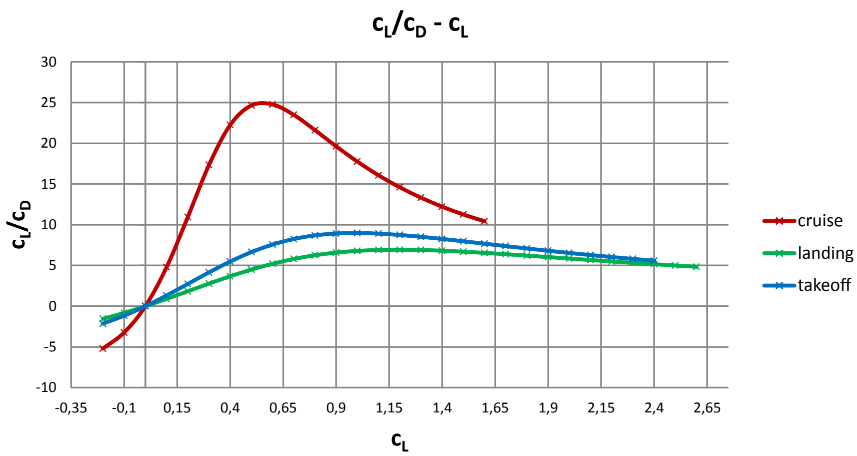


Figure 3.19: c_L/c_D polar for cruise, landing and takeoff

3.7.3 Lift distribution

The distribution of the local lift coefficient c_l over the wing span for an angle of attack of $\alpha = 0.371°$ is shown in figure 3.20. It can be seen that the values in the inner area are slightly higher and decrease minimal towards the outside. This is a desired behavior, since the maximum lift coefficient is reached faster inside and the flow detaches there earlier, whereas the control surfaces at the outer areas remain still operational.
The circulation distribution over the wing span for a lift coefficient of $c_L \approx 0.55$ is shown in figure 3.21. Because of the relation

$$dL(y) = \rho \cdot v \cdot \Gamma(y) \cdot dy \tag{3.20}$$

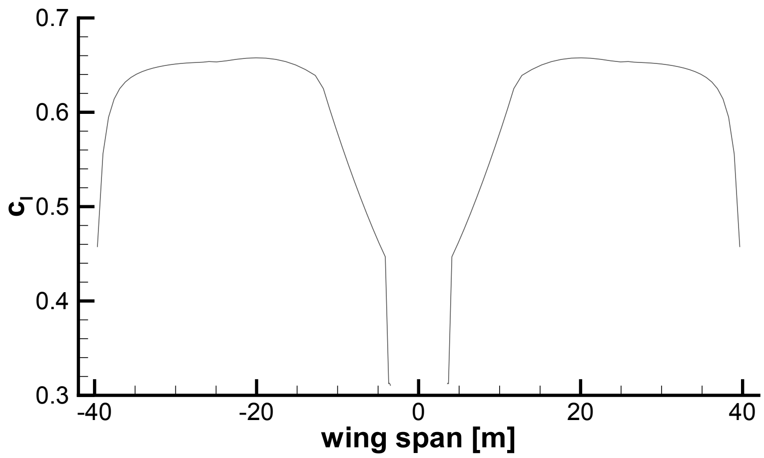


Figure 3.20: Lift coefficient distribution in spanwise direction for $\alpha = 0.371°$

the local lift is direct proportional to the local circulation. The circulation distribution can therefore be regarded as a measure of the lift distribution.

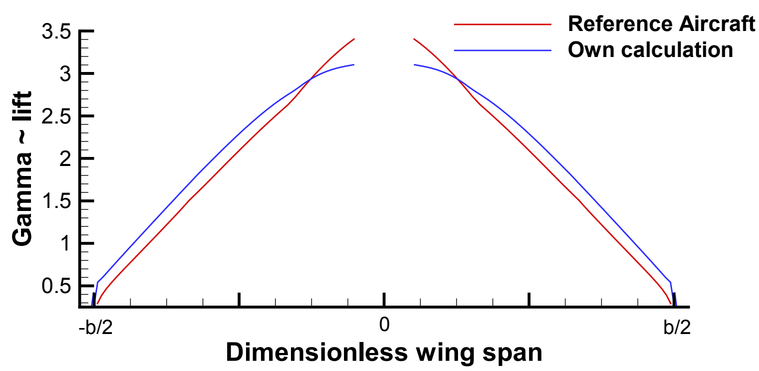


Figure 3.21: Gamma distribution in spanwise direction for $c_L \approx 0.55$

In order to evaluate the induced drag the K-factor is regarded as measure. For the same lift coefficient, a value of 1.499 was obtained from the Aero Tool for the reference wing and a value of 1.393 for the projected design with winglets. As the last value is lower, the induced drag is expected reduced. This seems reasonable because increasing the wing span and using winglets leads to reduced induced drag.

3.8 Applied computational mesh

The plan view of the applied computational mesh is plotted in figure 3.22. In flow direction there are 41 points distributed over the cross section which are condensed towards the leading edge. The distribution in spanwise direction could only be adapted for each segment separately. It is specified in table 3.12. The grid has been adjusted that the point density increases towards the wing tip to capture the higher flow gradients expected there.

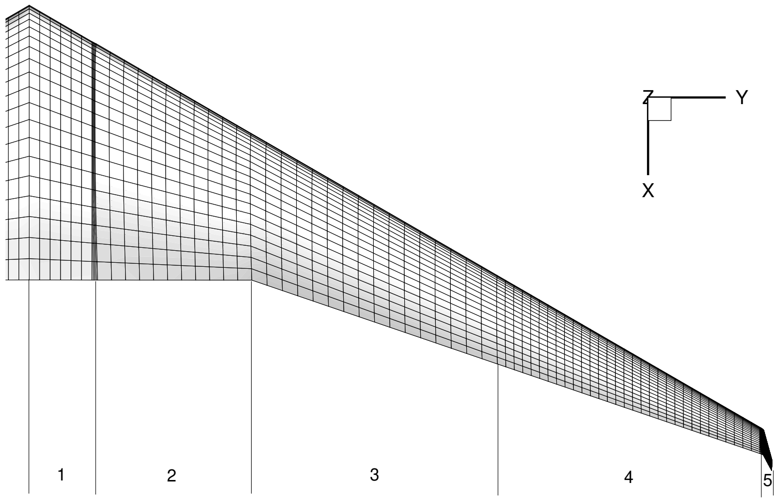


Figure 3.22: Computational mesh

Table 3.12: Point distribution in spanwise direction

segment	cells
1	6
2	11
3	16
4	31
5	5

3.9 Empennage

The first step in the design process of the empennage was to evaluate the suitability of the in section 3.1 presented symmetrical airfoils. Therefore the given empennage of the reference aircraft is equipped with these three airfoils, NACA 0011sc, NACA 0012 and NACA 0012sc. Then calculations with the Aero Tool were performed with the conventional empennage and its dimensions of the reference aircraft. This should ensure insight which airfoil is best suited for the further empennage design. The polars obtained by the Aero Tool are given in fig. 3.24 Because the polars in fig. 3.24 don't indicate any advantages again by the use of an super-critical airfoil, the projected empennage will also be equipped with the NACA 0012 airfoil. Additionally the polars given in fig. 3.24 and every further presented polar is correlated to the reference surface, which is the surface of the wing's planform.

To fulfil an appropriate arrangement of the aft fuselage installed engines and the empennage the use of a U-tail empennage was projected. The main geometric dimensions for a U-tail derive from the needed dimensions of a conventional empennage. Additional the surface of the VTP is divided by two, for each VTP on left and right side of the U-Tail. Therefore the common volume coefficient formulas are applied (eq. (3.21), eq. (3.22)), according to Raymer [5]. To perform calibration of these formulas the dimensions of the reference aircraft were used. The resulting values of the coefficients then also provide the calibration.

$$C_{VTP} = \frac{S_{VTP} L_{VTP}}{b S_W} \tag{3.21}$$

$$C_{HTP} = \frac{S_{HTP} L_{HTP}}{c_{mac} S_W} \tag{3.22}$$

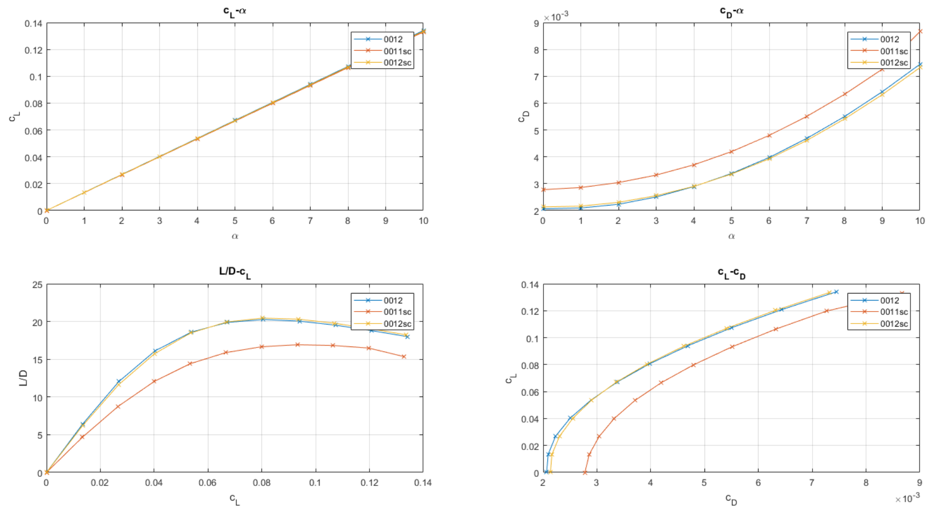


Figure 3.23: Polars obtained by Aero Tool for different airfoils on reference aircraft's empennage

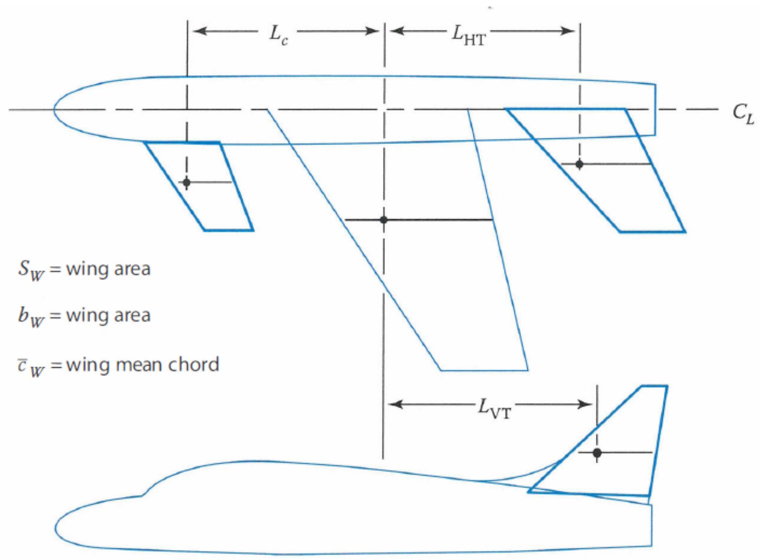


Figure 3.24: Explanation of dimensions used in (3.21) and (3.22), [5]

For first estimations data of the reference aircraft where used on the one hand. On the other L_{HTP} and L_{VTP} are estimated to have the same length and because of the aft fuselage engine installation to be 45% of the fuselage length. In further phases of the design process these values are harmonized with obtained data of projected wing, fuselage and neutral point calculations.Also the distance between the neutral points of HTP and VTP of the U-tail were considered and added to tail arm length for surface area determination of the VTP.

Calculations with eq. (3.21) and eq. (3.22) then delivered the values for surface areas of HTP and VTP. These surface areas were the main parameters for the planform design in the WingNet tool. The main parameters like chord length and wing span were then chosen to obtain the calculated surface area. In the case of an U-tail the chord length of the HTP has to be longer to get a considerable chord length for the VTP root. The projected planforms for the HTP and VTP are depicted in figure 3.25 and 3.26, respectively.

In fig. 3.27 the polars of the empennage are shown. Compared with the conventional empen-

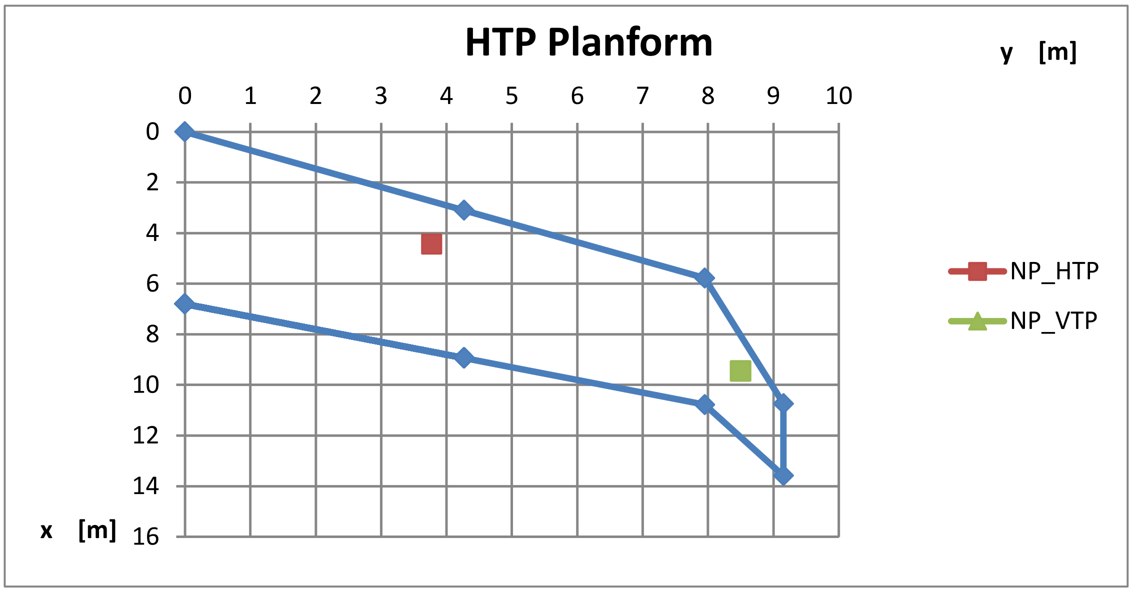


Figure 3.25: The planform for the HTP for the U-tail which also presents a part of the VTP.

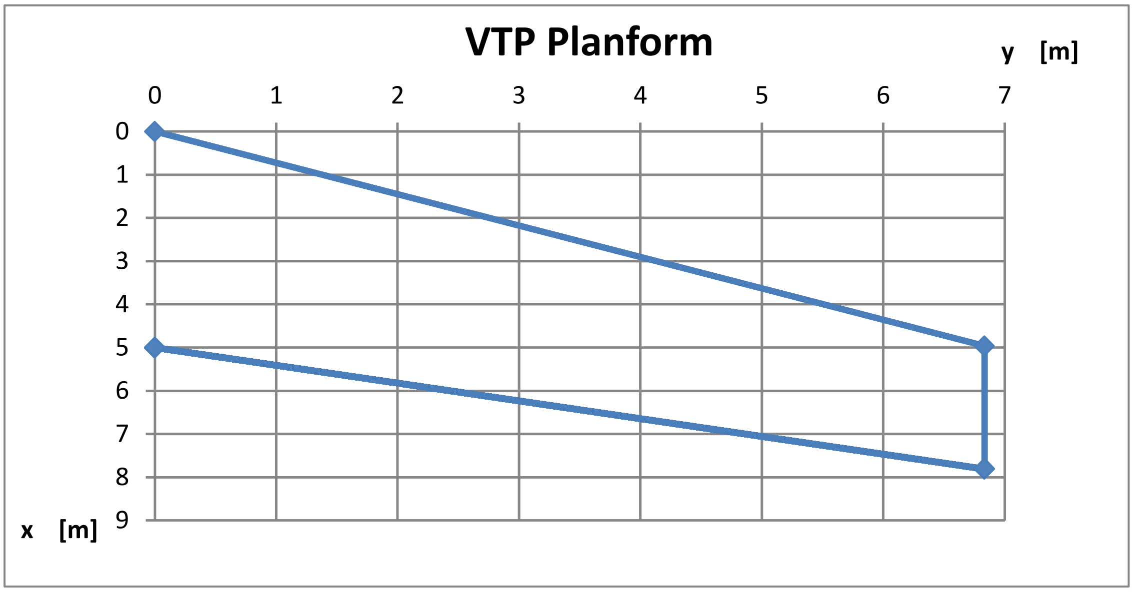


Figure 3.26: U-Tail VTP planform

nage layout, the U-tail indicates better characteristics in induced drag over change in angle of attack and a slightly higher zero lift drag.

To obtain these polars the induced drag were calculated over presented range of the angle of attack and the minimum drag was obtained by a calculation with no angle of attack. This seems considerable due to the fact that in transonic flow conditions the form drag does not varies in a high magnitude.

The applied computational mesh is shown in fig. 3.28 with the point distribution in shown in table 3.13. The usual rules for point distribution were applied, the conceive higher resolution in areas of higher flow gradients, like nose and tip.

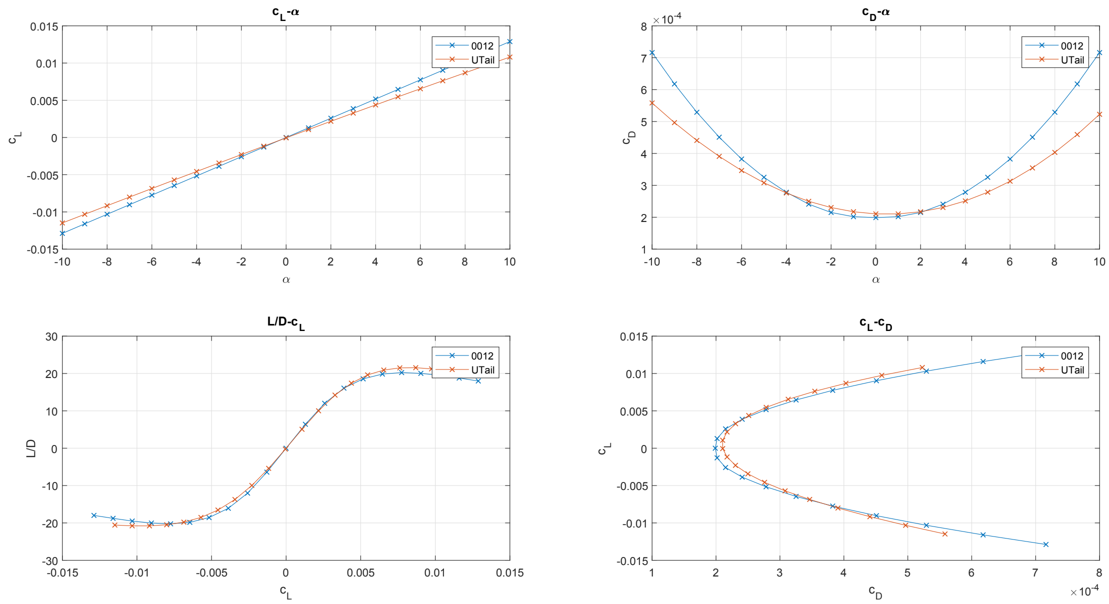


Figure 3.27: Empennage Polars of the U-Tail obtained by Aero Tool, with the reference of a conventional T-tail.

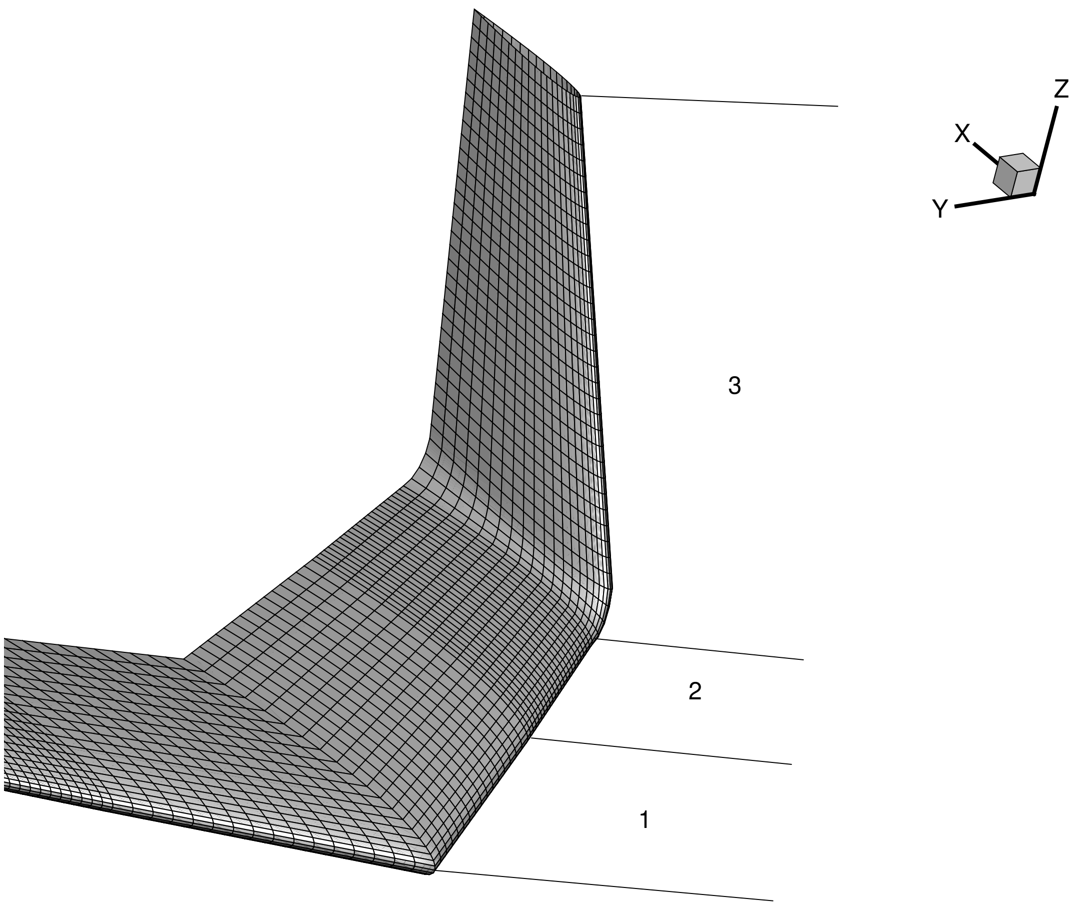


Figure 3.28: Applied computational mesh for the empennage calculations with AeroTool

Table 3.13: Point distribution of empennage in spanwise direction

segment	cells
1	15
2	22
3	30

3.10 Neutral Point

To identify the neutral point of the aircraft the equilibrium of moments about this point is set up.

$$\sum_i M_{i_{NP_{aircraft}}} = \Delta L_{wing} \cdot \Delta x_N - \Delta L_{HTP} \cdot (\Delta x_{AC_{wingHTP}}) = 0 \tag{3.23}$$

The needed force system is outlined in fig. 3.29.

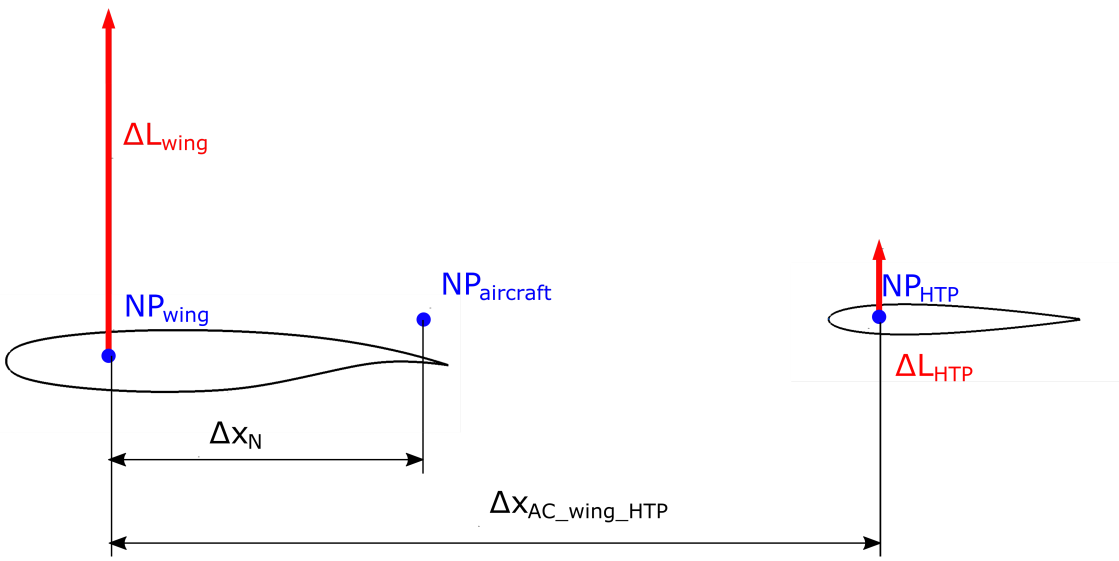


Figure 3.29: Force system which indicates the position of the aircraft's neutral point.

Definition of the lift written out in (3.24) and (3.25) and further determined lift coefficient due to angle of attack in (3.26) and (3.30)

$$\Delta L_{wing} = q \cdot S_{wing} \cdot \Delta c_{A_{wing}} \tag{3.24}$$

$$\Delta L_{HTP} = q_H \cdot S_{HTP} \cdot \Delta c_{A_{HTP}} \tag{3.25}$$

$$\Delta c_{A_{wing}} = c'_{A_{wing}} \cdot \Delta\alpha \tag{3.26}$$

$$\Delta c_{A_{HTP}} = c'_{A_{HTP}} \cdot (\Delta\alpha(1 - \frac{d\epsilon_h}{d\alpha}) \tag{3.27}$$

The equations (3.24) to (3.30) can then be combined to the following formula to obtain the neutral point of the entire airplane:

$$\Delta x_N = \frac{q_H}{q} \cdot \frac{S_{HTP}}{S_{wing}} \cdot \frac{c'_{A_{HTP}}}{c'_{A_{Wing}}} \cdot (1 - \frac{d\epsilon_h}{d\alpha}) \cdot (\Delta x_{AC_{wingHTP}} - \Delta x_N) \tag{3.28}$$

Table 3.14: Parameters for Neutral point calculation

Parameters from aircraft:	
Tail arm HTP	$\Delta x_{AC_{wingHTP}}$
Surface area wing	S_{wing}
Lift slope wing	$c'_{A_{Wing}}$
Lift slope HTP	$c'_{A_{HTP}}$
Parameters from literature	
Dynamic pressure ratio	$\frac{q_H}{q} = 0.85$
Downwash gradient	$\frac{d\epsilon_h}{d\alpha} = 0.40$

Parameters for the given equation are given in table 3.14
Surface areas of wing and HTP are simply defined by the selected planforms, lift slopes are obtained by calculations of the Aero Tool. Values for the parameter from literature are taken from [13] for low tail arrangement.

From equation (3.28) the final formula for the position of the neutral point can be defined with:

$$Q = \frac{q_H}{q} \cdot \frac{S_{HTP}}{S_{wing}} \cdot \frac{c'_{A_{HTP}}}{c'_{A_{Wing}}} \cdot (1 - \frac{d\epsilon_h}{d\alpha}) \tag{3.29}$$

$$\Delta x_N = \frac{Q}{1+Q} \cdot \Delta x_{AC_{wingHTP}} = 1.5\,\mathrm{m} \tag{3.30}$$

4 Component weights and weight & balance

Considering using statistical Class I methods for weight estimation would only have provided a very coarse weight estimation of overall aircraft components. This would have resulted in a weight breakdown which wouldn't be in the expected degree of details. In the end, this iteration step would not have provided a significantly improved weight estimation. Additionally, a more detailed list of component weights for the reference aircraft was already provided. In relation to that component breakdown a structured comprehensive list of aircraft components and their sections was composed.

Aircraft Component Breakdown

Structure	Systems	Powerplant	Furnishing	Operator Items
Wing	Bleed Air System	Engine	Furnishing	Unusable Fuel
HTP	Fuel Systems	- 2x UHBR	Lavatories	Flight Crew Items
VTP	Hydraulic System	APU	Lighting	LD3 Container
Landing Gear	Fire Protection	- 1x APU	Emergency Oxygen	Passenger Seats
- Front Landing Gear	Electric Systems		Water Installation	Catering
- Main Landing Gear	Air Conditioning			Additional Items
Pylon	Communication			Emergency Equipment
	Navigation			

Table 4.1: Aircraft Component Breakdown

For each of these components a weight was calculated using Class II or other methods giving a more detailed prediction. By using the calculated and actual weights, given in the task description, a correction factor was calculated. This correction factor represents on the one hand the technology development of aircraft over time and on the other hand improving the weight prediction as most of the equations were created using statistical data from very old aircraft. For each component the values for the reference aircraft, the first iteration of the own aircraft and the final iteration of the own aircraft are named.

In a next step for every component a center of gravity was determined using some handbook methods as well as arguing about a certain position.
Using both, weight and CG, the overall aircraft CG was calculated and the loading diagrams were plotted.
The detailed calculations can be found in Appendix 13.

4.1 Structure

All weights from basic structure of the airplane were divided into wing, empennage, landing gear and engine pylon. For calculation the different structure components equations from Raymer [5] were used. Only for calculating the pylon weight Raymer [5] does not provide any equation instead an equation from Torenbeek [7] was used.

4.1.1 Wing

4.1.1.1 Weight Estimation

The formula for wing weight calculation from Raymer [5] reads as follows:

$$W_{wing} = 0.0051 \cdot (W_{dg} \cdot N_z)^{0.557} \cdot S_w^{0.649} \cdot A^{0.5} \cdot (t/c)^{-0.4} \cdot (1+\lambda)^{0.1} \cdot (cos(\Lambda))^{-1.0} \cdot S_{csw}^{0.1} \quad (4.1)$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
W_{dg}	flight design gross weight, [lb]	831134.199	718568.199	294683.866
N_z	ultimate load factor	3.75	3.75	3.75
S_w	trapezoidal wing area, [ft^2]	6027.79	5709.627	5615.499
B	wing span, [ft]	251.31	262.47	262.47
A	aspect ratio	10.478	12.065	12.2676
t/c	thickness-to-chord ratio	0.114	0.114	0.118
λ	taper ratio	0.096	0.104	0.108
Λ	wing sweep at 25% MAC, [°]	29.7	29.7	26.2
S_{csw}	control surface area, [ft^2]	775.0016	775.0016	1031.39

Table 4.2: Parameter for wing weight estimation

The ultimate load factor is calculated by multiplication of 1.5 and the limit load factor $N_z = 1.5 \cdot N_{lff}$. For the limit load factor, CS25.337 gives the equation

$$N_{lff} = 2.1 + (52910.94/(W_{dg} + 22046.23) \quad (4.2)$$

Because the result of this is smaller than 2.5, C25.337 determines N_{lff} to 2.5.
The aspect ratio calculates by dividing the wing span squared by the wing area.

The taper ratio calculates by dividing the length of the outside of the wing by the length of the inside of the wing.

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{wing} = 1.035 \quad (4.3)$$

So after conversion of the result into metric units and multiplication with the correction factor the final wing weight of the new aircraft is:

$$W_{wing} = 45339.228kg|43495.663kg \quad (4.4)$$

In chapter 7.4 the addition of a winglet is discussed, the weight of the winglet is added to the final wing weight with $W_{winglet} = 0.3\% \cdot W_{wing} = 129.7$ kg
In chapter 6 the use of a geodesic structure is explained, this technique causes a weight saving of 15% (for more detail see chapter 6). These extras are just calculated for the final wing weight:

$$W_{wing\_withExtras} = 0.85(W_{wing} + W_{winglet}) = 37082.228kg \quad (4.5)$$

4.1.1.2 CG Estimation

To calculate the longitudinal position (x-direction) of the center of gravity of the wings, a formula from Roskam Part V [14] chapter eight is used. Its says: "Swept wing: 70 percent of the distance between the front and rear spar behind the front spar at 35 percent of the semi-span". These values are measured in Catia:
$x_{CG\_wing} = 45.295m|41.161m$ and $y_{CG\_wing} = 14m|14m$

4.1.2 Empennage

4.1.2.1 Horizontal Tail Plane

4.1.2.1.1 Weight Estimation The formula for calculating the horizontal tail plane (htp) from Raymer [5] reads as follows:

$$W_{htp} = 0.0379 \cdot K_{uht} \cdot (1 + F_w/B_h)^{-0.25} \cdot W_{dg}^{0.639} \cdot N_z^{0.10} \cdot S_{ht}^{0.75} \cdot L_t^{-1.0} \cdot K_y^{0.704} \cdot (cos(\Lambda_{ht}))^{-1.0} \cdot A_h^{0.166} \cdot (1 + S_e/Sht)^{0.1} \tag{4.6}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
K_{uht}	1.143 for all moving horizontal tail	1.143	1.143	1.143
F_w	fuselage width at horizontal tail intersection, [ft]	11.3189	10.53	10.53
B_h	horizontal tail span, [ft]	69.554	57.04	52.165
S_{ht}	horizontal tail area, [ft^2]	1119	1123.75	1009.66
L_t	tail length, [ft]	50.668	58.7	58.7
K_y	A/C pitching radius of gyration, $= 0.3 \cdot L_t$, [ft]	15.2	17.61	17.61
Λ_{ht}	wing sweep horizontal tail at 25% MAC, [°]	36	36	36
A_h	aspect ratio horizontal tail	4.322	2.9	2.695
S_e	elevator area, [ft^2]	215.278	215.278	265.187

Table 4.3: Parameter for HTP weight estimation

The aspect ratio calculates by dividing the htp wing span squared by the htp wing area. The calculation with the values from the reference aircraft got a correction factor of

$$cf_{wing} = 0.571 \tag{4.7}$$

So after conversion of the result into metric units and multiplication with the correction factor the final horizontal tail plane weight of the new aircraft is:

$$W_{htp} = 2535.573kg|2172.16kg \tag{4.8}$$

4.1.2.1.2 CG Estimation To calculate the longitudinal position (x-direction) of the center of gravity of the htp, a formula from Roskam Part V [14] chapter eight is used. Its says: "Regardless of sweep angle: 42 percent chord from the L.E.(leading edge) at 38 percent of the semi-span". These values are measured in Catia:
$x_{CG\_htp} = 71.266m|71.266m$

4.1.2.2 Vertical Tail Plane

4.1.2.2.1 Weight Estimation The formula for the vertical tail plane (vtp) from Raymer [5] reads as follows:

$$W_{vtp} = 0.002 \cdot (1 + H_{t/v})^{0.225} \cdot W_{dg}^{0.556} \cdot N_z^{0.536} \cdot L_t^{-0.5} \cdot S_{vt}^{0.5} \cdot K_z^{0.875} \cdot (cos(\Lambda_{vt}))^{-1} \cdot A_v^{0.35} \cdot (t/c)^{-0.5} \tag{4.9}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
$H_{t/v}$	0.0 for conventional, 1.0 for T tail	0	0	0
S_{vt}	vertical tail area, [ft^2]	645.835	315.909	288.473
K_z	A/C yawning radius of gyration, $= L_t$, [ft]	50.668	58.7	58.7
Λ_{vt}	wing sweep vertical tail at 25% MAC, [°]	38	38	36
A_v	aspect ratio vertical tail	1.88	1.174	1.743

Table 4.4: Parameter for VTP weight estimation

The aspect ratio calculates by dividing the vtp wing span squared by the vtp wing area. The calculation with the values from the reference aircraft and the fact that the chosen U-Tail has two vertical tail planes with one on each side it got a correction factor of

$$cf_{vtp} = 1.399 \cdot 2 = 2.798 \tag{4.10}$$

So after conversion of the result into metric units and multiplication with the correction factor the final vertical tail plane weight of the new aircraft is:

$$W_{vtp} = 2496.277kg|3005.388kg \tag{4.11}$$

4.1.2.2.2 CG Estimation To calculate the longitudinal position (x-direction) of the center of gravity of the vtp, a formula from Roskam Part V [14] chapter eight is used. Its says: "Regardless of sweep angle: 42 percent chord from the L.E. at 38 percent vertical tail span from the root chord". These values are measured in Catia:
$x_{CG\_vtp} = 76.331m|76.331m$

4.1.3 Fuselage

4.1.3.1 Weight Estimation

The formula for the fuselage from Raymer [5] reads as follows:

$$W_{fuselage} = 0.3280 \cdot K_{door} \cdot K_{lg} \cdot (W_{dg} \cdot N_z)^{0.5} \cdot L^{0.25} \cdot S_f^{0.302} \cdot (1 + K_{ws})^{0.04} \cdot (L/D)^{0.10} \tag{4.12}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
K_{door}	1.12 if two cargo doors	1.12	1.12	1.12
K_{lg}	1.12 if fuselage mounted landing gear	1.12	1.12	1.12
L	fuselage structural length, [ft]	229.6588	242.7822	242.7822
S_f	wetted area fuselage, [ft^2]	13747.516	14230.449	14383.92
K_{ws}	wingsweep factor	0.505	0.506	0.438
D	fuselage structural depth, [ft]	21.818	22.966	22.966

Table 4.5: Parameter for fuselage weight estimation

The wingsweep factor is calculated by the following equation:

$$K_{ws} = 0.75 \cdot ((1 + 2 \cdot \lambda)/(1 + \lambda)) \cdot (B/L) \cdot tan(\Lambda) \tag{4.13}$$

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{fuselage} = 1.289 \tag{4.14}$$

So after conversion of the result into metric units and multiplication with the correction factor the final fuselage weight of the new aircraft is:

$$W_{fuselage} = 36023.548kg|34300.249kg \tag{4.15}$$

In chapter 6 the use of a geodesic structure is explained, this technique causes a weight saving of 15% (for more detail see chapter 6). These structure are calculated just for the final fuselage weight:

$$W_{fuselage\_withExtras} = 0.85 \cdot W_{fuselage} = 30870,224kg \tag{4.16}$$

4.1.3.2 CG Estimation

To calculate the longitudinal position (x-direction) of the center of gravity of the fuselage, a formula from Roskam Part V [14] chapter eight is used. Its says: "Jet transports: 0.47-0.50 (rear fuselage mounted engines)". So the center of gravity of the fuselage is calculated as follows:

$$x_{CG\_fuselage} = 0.5 \cdot L = 37m|37m \tag{4.17}$$

4.1.4 Landing Gear

The calculation of the weight of the landing gear is split up into two different calculations. After these calculations the two weight are summed up to get overall landing gear weight.

4.1.4.1 Main Landing Gear

4.1.4.1.1 Weight Estimation The formula for the main landing gear (mlg) from Raymer [5] reads as follows:

$$W_{mlg} = 0.0106 \cdot K_{mp} \cdot W_l^{0.888} \cdot N_l^{0.25} \cdot L_m^{0.4} \cdot N_{mw}^{0.321} \cdot N_{mss}^{-0.5} \cdot V_{stall}^{0.1} \tag{4.18}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
K_{mp}	1.126 if kneeling gear, 1.0 otherwise	1.0	1.0	1.0
W_l	landing design gross weight, [lb]	630522.1	630522.1	630522.1
N_l	ultimate load factor, $N_{gear} \cdot 1.5$	3	3	3
L_m	extended length of main landing gear, [in]	170.2437	140.5118	153.5433
N_{mw}	number of main wheels	12	12	12
N_{mss}	number of main gear shock struts	2	2	2
V_{stall}	stall speed, [kt]	127.658	118.6989	153.723

Table 4.6: Parameter for Main Landing Gear weight estimation

So after conversion of the result into metric units the main landing gear weight of the new aircraft is:

$$W_{mlg} = 16361.466kg|17396.371kg \tag{4.19}$$

4.1.4.1.2 CG Estimation Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the main landing gear. In this case a well educated estimation has to be made. In this case it is determined that the CG is at 50% of landing gear length:

$$x_{CG\_mlg} = 45.542m|41.391m \tag{4.20}$$

4.1.4.2 Front Landing Gear

4.1.4.2.1 Weight Estimation The formula for the front landing gear (flg) from Raymer [5] reads as follows:

$$W_{flg} = 0.032 \cdot K_{np} \cdot W_l^{0.2} \cdot L_n^{0.5} \cdot N_{nw} 0.45 \tag{4.21}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
K_{np}	1.15 if kneeling gear, 1.0 otherwise	1.0	1.0	1.0
Ln	extended nose gear length, [in]	194.5644	105.9843	120.0095
N_{nw}	number of nose wheels	2	2	2

Table 4.7: Parameter for front landing gear weight estimation

So after conversion of the result into metric units the front landing gear weight of the new aircraft is:

$$W_{flg} = 1418.845kg|1509.81kg \tag{4.22}$$

4.1.4.2.2 CG Estimation Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the front landing gear. In this case a well educated estimation has to be made. In this case it is determined that the CG is at 50% of landing gear length:

$$x_{CG\_flg} = 5.056m|4.970m \tag{4.23}$$

4.1.4.3 Landing gear overall

To get the overall landing gear (lg) weight, both weights have to be summed up:

$$W_{ldg} = W_{mlg} + W_{flg} \tag{4.24}$$

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{ldg} = 0.822 \tag{4.25}$$

So after conversion of the result into metric units and multiplication with the correction factor the final landing gear weight of the new aircraft is:

$$W_{ldg} = 14713.63kg|15540.88kg \tag{4.26}$$

To get the overall center of gravity of the landing gear, both CGs have to be summed up:

$$x_{CG\_ldg} = (x_{CG\_mlg} \cdot W_{mlg} + x_{CG\_flg} \cdot W_{flg})/W_{ldg} \tag{4.27}$$

4.1.5 Pylon

4.1.5.1 Weight Estimation

The formula for the pylon from Torenbeek [7] reads as follows:

$$W_{pylon} = 0.405 \cdot \sqrt{v_D} \cdot S_{wet}^{1.3} \tag{4.28}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
v_D	design dive speed, [m/s]	320	320	320
S_{wet}	total wetted area nacelle, [m^2]	171.06	201.675	201.675

Table 4.8: Parameter for pylon weight estimation

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{pylon} = 0.661 \tag{4.29}$$

So after conversion of the result into metric units and multiplication with the correction factor the final pylon weight of the new aircraft is:

$$W_{pylon} = 4745.469kg|4745.469kg \tag{4.30}$$

4.1.5.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the pylon. In this case a well educated estimation has to be made. In this case it is determined that the CG is at 50% of engine length:

$$x_{CG\_pylon} = 66.243m|66.243m \tag{4.31}$$

4.1.6 Structure weight overall

To get the overall structure weight, all weights from chapter 4.1 have to be summed up:

$$W_{structure} = W_{wing} + W_{htp} + W_{vtp} + W_{ldg} + W_{pylon} = 105110kg|93416kg \tag{4.32}$$

4.1.7 Structure CG overall

To get the overall structure center of gravity in x-direction, all CGs from chapter 4.1 have to be summed up:

$$\begin{aligned} x_{CG\_structure} = &(x_{CG\_wing} \cdot W_{wing} + x_{CG\_htp} \cdot W_{htp} + x_{CG\_vtp} \cdot W_{vtp} \\ &+ x_{CG\_ldg} \cdot W_{ldg} + x_{CG\_pylon} \cdot W_{pylon})/W_{structure} \\ &= 43.313m|42.184m \end{aligned} \tag{4.33}$$

4.2 Systems

The section "systems" contains all weights from systems of the airplane like bleed air system, fuel system, hydraulic system, fire protection, electric systems, air conditioning, flight controls, communication and navigation. The formulas for the different structure components are taken from Raymer [5].

4.2.1 Bleed Air System

4.2.1.1 Weight Estimation

Unfortunately no equation to calculate the specific weight of bleed air system (basys) was provided by literature. Because of that it was decided to proceed by taking the same bleed air system weight as the reference aircraft.

$$W_{basys} = 1050kg|1050kg \tag{4.34}$$

4.2.1.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the bleed air system. In this case a well educated estimation has to be made. It is determined that the CG is at in the area of $0.1 \cdot MAC$ up to $0.2 \cdot MAC$:

$$x_{CG\_basys} = 45.035m|40.982m \tag{4.35}$$

4.2.2 Fuel System

4.2.2.1 Weight Estimation

The formula for the fuel system (fsys) from Raymer [5] reads as follows:

$$W_{fsys} = 2.405 \cdot V_t^{0.606} \cdot (1 + V_i/V_t)^{-1.0} \cdot (1 + V_p/V_t) \cdot N_t^{0.5} \tag{4.36}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
V_t	total fuel volume, [gal]	50192.69	50192.69	48953.72
V_i	integral tanks volume, [gal]	30000	30000	30000
V_p	self sealing tanks volume, [gal]	5000	5000	5000
N_t	number of fuel tanks	6	6	6

Table 4.9: Parameter for fuel system weight estimation

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{fsys} = 0.601 \tag{4.37}$$

So after conversion of the result into metric units and multiplication with the correction factor the final fuel system weight of the new aircraft is:

$$W_{fsys} = 1297.764kg|762.771kg \tag{4.38}$$

4.2.2.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the fuel system. In this case a well educated estimation has to be made. It is determined that the center of gravity lies between the leading edge of the wing where most of the tanks are located and the middle of the engines, where the fuel is needed.

$$x_{CG\_fsys} = 51.183m|51.183m \tag{4.39}$$

4.2.3 Hydraulic System

4.2.3.1 Weight Estimation

The formula for the hydraulic system (hsys) from Raymer [5] reads as follows:

$$W_{hsys} = 0.2673 \cdot N_f \cdot (L + B_w)^{0.937} \tag{4.40}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
N_f	number of different functions performed by surface controls	7	7	7

Table 4.10: Parameter for hydraulic weight estimation

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{hsys} = 9.014 \tag{4.41}$$

So after conversion of the result into metric units and multiplication with the correction factor the final hydraulic system weight of the new aircraft is:

$$W_{hsys} = 2611.378kg|2611.378kg \tag{4.42}$$

4.2.3.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the hydraulic system. In this case a well educated estimation has to be made. It is determined that the CG is at 50% of fuselage length:

$$x_{CG\_hsys} = 37m|37m \tag{4.43}$$

4.2.4 Fire Protection

4.2.4.1 Weight Estimation

Unfortunately no equation to calculate the specific weight of fire protection system (fpsys) was provided by literature. Because of that it was decided to proceed by taking the same fire protection system weight as the reference aircraft.

$$W_{fpsys} = 360\,\text{kg}|360\,\text{kg} \tag{4.44}$$

4.2.4.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the fire protection system. In this case a well educated estimation has to be made. It is determined that the CG is at 50% of fuselage length:

$$x_{CG\_fpsys} = 37\,\text{m}|37\,\text{m} \tag{4.45}$$

4.2.5 Electric System

4.2.5.1 Weight Estimation

The formula for the electric system (esys) from Raymer [5] reads as follows:

$$W_{esys} = 7.291 \cdot R_{kva}^{0.782} \cdot L_a^{0.346} \cdot N_{gen}0.10 \tag{4.46}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
R_{kva}	system electrical rating, typically 40-60	60	60	60
L_a	electrical routing distance, generator to avionics to cockpit, [ft]	500	1000	1000
N_{gen}	number of generators, N_{eng}+RAM+APU	4	4	4

Table 4.11: Parameter for electric system weight estimation

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{esys} = 3.112 \tag{4.47}$$

So after conversion of the result into metric units and multiplication with the correction factor the final electrical system weight of the new aircraft is:

$$W_{esys} = 6990.395kg|6990.395kg \tag{4.48}$$

4.2.5.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the electric system. In this case a well educated estimation has to be made. It is determined that the CG is at 50% of fuselage length:

$$x_{CG\_esys} = 37\,\mathrm{m}|37\,\mathrm{m} \tag{4.49}$$

4.2.6 Air Conditioning

4.2.6.1 Weight Estimation

The formula for the air conditioning system (acsys) from Raymer [5] reads as follows:

$$W_{acsys} = 62.36 \cdot N_p^{0.25} \cdot (V_{pr}/1000)^{0.604} \cdot W_{uav}^{0.10} \tag{4.50}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
N_p	number of all personel on board (crew+passengers)	440	445	445
V_{pr}	volume of pressurized section, [ft_3]	66653.242	58061.485	58061.485
W_{uav}	uninstalled avionics weight, typically 800-1400, [lb]	1400	1400	1400

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{acsys} = 0.621 \tag{4.51}$$

So after conversion of the result into metric units and multiplication with the correction factor the final air condition system weight of the new aircraft is:

$$W_{acsys} = 1938.465kg|1938.465kg \tag{4.52}$$

4.2.6.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the air conditioning system. In this case a well educated estimation has to be made. It is determined that the CG is at the leading edge of the wing:

$$x_{CG\_acsys} = x_{LE\_wing} = \tag{4.53}$$

4.2.7 Flight Controls

4.2.7.1 Weight Estimation

The formula for the flight controls system (fcsys) from Raymer [5] reads as follows:

$$W_{fcsys} = 145.9 \cdot N_f^{0.554} \cdot (1 + N_m/N_f)^{-1.0} \cdot S_{cs}^{2.0} \cdot (I_{yaw} \cdot 10^{-6})^{0.07} \tag{4.54}$$

It contains the following elements:

symbol	description	value RefAC	value 1^{st} iteration	value final iteration
N_m	number of surface contents driven by mechanical actuation instead of hydraulics	1	1	1
S_{cs}	total area of control surfaces, [ft^2]	1101.713	1101.713	1408.07
I_{yaw}	yawning moment of inertia, [lb-ft^2]	1000000	1000000	1000000

The calculation with the values from the reference aircraft got a correction factor of

$$cf_{fcsys} = 4.053 \tag{4.55}$$

So after conversion of the result into metric units and multiplication with the correction factor the final flight controls system weight of the new aircraft is

$$W_{fsys} = 2799.678kg|2940.487kg \tag{4.56}$$

In chapter 2 the use of morphing structure is explained, this technique causes a weight increase of x2 in flight controls (for more detail see chapter 6). These extras are just calculated for the final flight control system weight:

$$W_{fsys\_withExtras} = W_{fsys} \cdot 2 = 5880.975kg \tag{4.57}$$

4.2.7.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the flight controls system. In this case a well educated estimation has to be made. It is determined that the CG is at 50 % between the leading edge of the wing and the trailing edge of the htp:

$$x_{CG\_fcsys} = 57.847m|55.444m \tag{4.58}$$

4.2.8 Communication

4.2.8.1 Weight Estimation

Unfortunately no equation to calculate the specific weight of communication system (csys) was provided by literature. Because of that it was decided to proceed by taking the same communication system weight as the reference aircraft.

$$W_{csys} = 250kg|250kg \tag{4.59}$$

4.2.8.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the communication system. In this case a well educated estimation has to be made. It is determined that the CG is located under the cockpit in the avionics bay:

$$x_{CG\_csys} = 6.818m|6.818m \tag{4.60}$$

4.2.9 Navigation

4.2.9.1 Weight Estimation

Unfortunately no equation to calculate the specific weight of navigation system (nsys) was provided by literature. Because of that it was decided to proceed by taking the same navigation system weight as the reference aircraft.

$$W_{nsys} = 270kg|270kg \tag{4.61}$$

4.2.9.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the navigation system. In this case a well educated estimation has to be made. It is determined that the CG is located under the cockpit in the avionics bay:

$$x_{CG\_nsys} = 6.818m|6.818m \tag{4.62}$$

4.2.10 Systems weight overall

To get the overall system weight, all weights from chapter 4.2 have to be summed up:

$$\begin{aligned} W_{systems} = {} & W_{basys} + W_{fsys} + W_{hsys} + W_{fpsys} + W_{esys} \\ & + W_{acsys} + W_{fcsys} + W_{csys} + W_{nsys} = 19849.539kg|20113.994kg \end{aligned} \tag{4.63}$$

4.2.11 Systems CG overall

To get the overall systems center of gravity in x-direction, all CGs from chapter 4.2 have to be summed up:

$$\begin{aligned} x_{CG\_systems} = {} & (x_{CG\_basys} \cdot W_{basys} + x_{CG\_fsys} \cdot W_{fsys} + x_{CG\_hsys} \cdot W_{hsys} + x_{CG\_fpsys} \cdot W_{fpsys} \\ & + x_{CG\_esys} \cdot W_{esys} + x_{CG\_acsys} \cdot W_{acsys} + x_{CG\_fcsys} \cdot W_{fcsys} \\ & + x_{CG\_csys} \cdot W_{csys} + x_{CG\_nsys} \cdot W_{nsys})/W_{systems} = 40.25161m|40.365m \end{aligned} \tag{4.64}$$

4.3 Furnishing

The section "furnishing" contains all weights from furnishing of the airplane like furnishing of cabin, lavatories, lighting, emergency oxygen and water installation. The formulas for the different components are taken from [7] and the slides of the presentation held by airbus to the students of this seminar [15].
In this chapter no changes were made during the iterations of the aircraft so for every weight and center of gravity only one value is given.

4.3.1 Lavatories

4.3.1.1 Weight Estimation

The formula for the lavatories (lav) weight from [15] reads as follows:

$$W_{lav} = (N_{lav_business} + N_{lav_eco}) \cdot m_{lavatory} \tag{4.65}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
$N_{lav_business}$	number of lavatories in business class	3	3
N_{lav_eco}	number of lavatories in economy class	9	9
$m_{lavatory}$	mass per lavatory, [kg]	100	100

So the final weight for all lavatories installed in the aircraft is $W_{lav} = 1200kg$

4.3.1.2 CG Estimation

To calculate the longitudinal (x-direction) position of the center of gravity of all the lavatories, every lavatory CG is measured in Catia where the lavatories are build in schematically in the overall assembly. It is determined that the lavatory CG is $x_{CG\_lav} = 30.478$ m. It is considered that the lavatories are looked seperately in case of position of center of gravity but are taken into the weight of the furnishing.

4.3.2 Lighting

4.3.2.1 Weight Estimation

Unfortunately no equation to calculate the specific weight of lighting was provided by literature. Because of that it was decided to proceed by taking the same lighting weight as the reference aircraft.

$$W_{lighting} = 400kg \tag{4.66}$$

4.3.2.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the lighting. In this case a well educated estimation has to be made. It is determined that the CG is at 50% of fuselage lenght:

$$x_{CG\_lighting} = 0.5 \cdot L = 37\,\text{m} \tag{4.67}$$

4.3.3 Emergency Oxygen

4.3.3.1 Weight Estimation

The formula for the emergency oxygen weight (o) from [7] reads as follows:

$$W_o = 40 + 2.4 \cdot N_{pax} \tag{4.68}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
N_{pax}	number of passengers on board	424	424

The calculation with the values from the reference aircraft got a correction factor of

$$cf_o = 0.736 \tag{4.69}$$

So after conversion of the result into metric units and multiplication with the correction factor the final installed emergency oxygen weight of the new aircraft is

$$W_o = 353.38kg \tag{4.70}$$

4.3.3.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the emergency oxygen. In this case a well educated estimation has to be made. It is determined that the CG is at 50% of fuselage length:

$$x_{CG\_o} = 0.5 \cdot L = 37\,\text{m} \tag{4.71}$$

4.3.4 Water Installation

4.3.4.1 Weight Estimation

The formula for the water and waste (ww) weight from [15] reads as follows:

$$W_{ww} = m_{waterwaste} \cdot N_{pax} \tag{4.72}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration	
m_{water}	waste	mass of water and waste per passenger, [kg]	1.5	1.5

The calculation with the values from the reference aircraft got a correction factor of

$$cf_o = 0.952 \tag{4.73}$$

So after multiplication with the correction factor the final installed water and waste weight of the new aircraft is:

$$W_{ww} = 600kg \tag{4.74}$$

4.3.4.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the water installation. In this case a well educated estimation has to be made. The tanks which are the main components for weight of this part are placed in the rear of the aircraft:

$$x_{CG\_ww} = 51 \tag{4.75}$$

4.3.5 Furnishing

4.3.5.1 Weight Estimation

To get the weight of the furnishing only (fur_o), the following formula from [15] which consider lighting, oxygen, lining and stowage bins has to be used and components which are already calculated have to be subtracted so that they wont get counted two times. Lavatories are added to the weight.

$$W_{fur\_o} = 150 \cdot L_{cabin} + W_{lav} - W_o - W_{lighting} \tag{4.76}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
L_{cabin}	cabin length, [m]	65.8	65.8

4.3.5.2 CG Estimation

Unfortunately there is no formula to calculate the longitudinal (x-direction) position of the center of gravity of the furnishing only. In this case a well educated estimation has to be made. It is determined that the CG is at 50% of fuselage length:

$$x_{CG\_fur\_o} = 0.5 \cdot L = 37\,\text{m} \tag{4.77}$$

4.3.6 Furnishing weight overall

To get the overall furnishing weight, all relevant weights from chapter 4.4 have to be summed up:

$$W_{furnishing} = W_{lighting} + W_o + W_{ww} + W_{fur\_o} = 12110.45kg \tag{4.78}$$

4.3.7 Furnishing CG overall

To get the overall furnishing center of gravity in x-direction, all CGs from chapter 4.3 have to be summed up:

$$\begin{aligned} x_{CG\_furnishing} = (&x_{CG\_lav} \cdot W_{lav} + x_{CG\_lighting} \cdot W_{lighting} + x_{CG\_o} \cdot W_o \\ &+ x_{CG\_ww} \cdot W_{ww} + x_{CG\_fur_o} \cdot (W_{fur_o} - W_{lav}) = 37.343 \end{aligned} \tag{4.79}$$

4.4 Powerplant

4.4.1 Engines

The engines are an important part in the weight calculation for an aircraft. Not only are the engines itself heavy objects which make an impact to the center of gravity, the specific fuel consumption of the engines determines the quantity of fuel the aircraft has to load to reach the desired range.

4.4.2 Rolls-Royce UltraFan

As Rolls-Royce already released their plans for the next generation jet engine, the Ultra Fan seems a reasonable technology to be ready in 2030. As a matter of facts, Rolls-Royce plans on starting ground testing the Ultra Fan engine by 2022. Besides a higher diameter due to the larger Bypass ratio also a significantly reduced fuel burn can be estimated. As the current Trent XWB has a Fan Diameter of 3 metre it was assumed that the Ultra Fan has at least 4,2 metre. That large diameter makes it necessary to place the engines at the rear section of the fuselage.
Rolls-Royce announced:
"The first design, Advance, will offer at least 20 per cent better fuel burn and CO2 emissions than the first generation of Trent engine and could be ready from the end of this decade. The second, UltraFan™, a geared design with a variable pitch fan system, is based on technology that could be ready for service from 2025 and will offer at least 25 per cent improvement in fuel burn and emissions against the same baseline." [16]
So it is assumed for c_{TL} of the used engine:

$$c_{TL} = 0.75 \cdot (0.8 \cdot c_{TL\_Trent700}) = 0.75 \cdot (0.8 \cdot 0,562) = 0.3372 lb/hr/lbf \tag{4.80}$$

Unfortunately Rolls-Royce published no information about the weight of the new engine, so it has to be interpolated with the help of development of weight of Rolls-Royce engines over production years or over specific fuel consumption (SFC). For this interpolation the following engines are used:

Engine	Year of production	Weight [lb]	SFC [lb/hr/lbf]
Trent 700	1992	13580	0.562
Trent 900	2003	13770	0.522
Trent XWB	2015	16043	0.478

With the values given by Rolls-Royce, c_{TL} and year of production, two possible weights from the progress weight over year of production and weight over SFC two possible weights for the new engine are generated: $W_{EngInt\_Year} = 17400.5lb$ and $W_{EngInt\_SFC} = 19911.5lb$. For the estimated engine weight the mean of this two interpolations is taken.

$$W_{eng} = 18656.01lb = 8462.22kg \tag{4.81}$$

The static sea level thrust of the Ultra Fan is supposed at a value of 600kN.
It is assumed that the longitudinal (x-direction) position of the center of gravity of the engines is at 50% of nacelle. This value is measured in Catia:

$$x_{CG\_engine} = 65476.87m|65476.87m \tag{4.82}$$

4.4.3 Open Rotor Option

In order to reduce fuel consumption even more, Open Rotor engines are in discussion again. In comparison to the Ultra Fan, it was assumed that a rotor diameter would take up to 6 metre. Placing the Ultra Fan engines at the rear sections already enabled the path for a new engine option with Open Rotor engines.

4.4.4 Auxilary Power Unit

The standard auxilary power unit which is also taken by the reference aircraft has a weight of 500 kg. For an aircraft which goes into service in 2030 it is possible to take a far more

advanced auxiliary power unit which works by the principle of a fuel cell. A fuel cell which has to provide enough power for the designed aircraft has to have a big size. Fuel cells have a very good efficiency and by the time of 2030 it is possible to build in such a fuel cell.
Unfortunately a fuel cell will be clearly heavier than a conventional APU with $1400kg$, but under consideration of future technologies it can be estimated that it will only weight $1100kg$ [17] at 2030. This weight still is clearly heavier than a conventional APU but different aspects speak in favour of a fuel cell as APU.
Unless a fuel cell is heavier than a conventional APU which leads to higher operating weight, its operation is far more enviromentally friendly because no cerosine is burned. It is assumed that the oxygen and hydrogen which are needed for operation are produced with "green electricity" from wind power or solar panels.
Because a fuel cell is not fixed at the rear end of the aircraft like a conventional APU it can be placed at any point of the aircraft. To get a useful influence on the center of gravity the fuel cell is placed directly in front of the wing box. This action has no influence on other system weights because the engines are located in the rear of the aircraft, but the tail of the aircraft had no mandatory needed space it can be optimated in aerodynamic aspects.
The specifications for the new APU with fuel cell technology are:

$$\begin{aligned} W_{APU} &= 1100\,\mathrm{kg} \\ x_{CG\_APU} &= 32\,\mathrm{m} \end{aligned} \tag{4.83}$$

Moreover, when the APU is placed in front of the Wing Center Box the rear section of the fuselage, especially the last part, can be aerodynamically optimized. This might also lead to a small reduction of drag and hence compensating a bit of the heavier weight.

4.5 Operator Items

To get the aircraft in state where it can be used to transport passengers and fulfill its duty, the aircraft has to be equipped with multiple operator items. These items are customized for each airline and product. If all of the different weights of these items are summed up the operator items weight is calculated. To get the operator weight empty (OWE), operator items weight has to be added to the manufacturer weight empty (MWE).
In this chapter some values for operator items are calculated, all formulas are taken from [15]. Unfortunately not all operator items which are really equipped to the aircraft can be calculated so the rest of the items are depicted by a calibration factor from the given operator items weight of the reference aircraft.

4.5.1 Unusable Fuel

The formula for the unusable fuel (ouf) weight from [15] reads as follows:

$$W_{ouf} = 0.25\% \cdot W_{total\_fuel} = 345kg \tag{4.84}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
W_{total_fuel}	maximum total fuel, [kg]	141000	138000

4.5.2 Flight Crew

The formula for the flight crew (ofc) weight from [15] reads as follows:

$$W_{ofc} = (n_{flight\_crew} + ncabin\_crew) \cdot W_{Pax} = 2200kg \tag{4.85}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
n_{flight_crew}	number of flight crew	3	3
n_{cabin_crew}	number of cabin crew	18	18
W_{Pax}	Pax standard mass, [kg]	110	110

4.5.3 LD3 Container

The formula for the LD3 container weight (oc) from [15] reads as follows:

$$W_{oc} = W_{Container} \cdot n_{Container} = 900kg \tag{4.86}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
$W_{Container}$	container standard mass, [kg]	75	75
$n_{Container}$	number of container	12	12

To calculate the number of containers needed, it is assumed that one container contains the luggage of 40 Pax. With 420 Pax travelling with the aircraft, 10.5 containers are needed. Because it is not efficient to carry an uneven number of containers, the number of containers is adjusted upward to 12.

4.5.4 Passenger Seats

The formula for the passenger seats weight (ops) from [15] reads as follows:

$$\begin{aligned} W_{ops} = {} & W_{Business\_seat} \cdot n_{Pax\_Business} + W_{Premium\_eco\_seat} \cdot n_{Pax\_Premium\_Eco} \\ & + W_{Eco\_seat} \cdot n_{Pax\_Eco} = 5634kg \end{aligned} \tag{4.87}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
$W_{Business\_seat}$	business class seat mass, [kg]	21	21
$W_{Premium\_eco\_seat}$	premium economy class seat mass	16	16
W_{Eco_seat}	economy class seat mass	12	12
$n_{Pax\_Business}$	number of business class pax	60	60
$n_{Pax\_Premium\_Eco}$	number of premium economy class pax	45	45
n_{Pax_Eco}	number of economy class pax	319	319

4.5.5 Catering

The formula for the catering weight (oca) from [15] reads as follows:

$$\begin{aligned} W_{oca} = {} & W_{Catering\_Business} \cdot n_{Pax\_Business} + W_{Catering\_Prem\_Eco} \cdot n_{Pax\_Premium\_Eco} \\ & + W_{Catering\_Eco} \cdot n_{Pax\_Eco} = 5656kg \end{aligned} \tag{4.88}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
$W_{Catering\_Business}$	Catering business class mass per pax, [kg]	20	20
$W_{Catering\_Prem\_Eco}$	Catering premium econonmy class mass per pax, [kg]	18	18
$W_{Catering\_Eco}$	Catering economy class mass per pax, [kg]	12	12

4.5.6 Galley Structure

The formula for the galley structure weight (ogs) from [15] reads as follows:

$$\begin{aligned} W_{ogs} = {} & W_{Galley\_Business} \cdot n_{Pax\_Business} + W_{Galley\_Prem\_Eco} \cdot n_{Pax\_Premium\_Eco} \\ & + W_{Galley\_Eco} \cdot n_{Pax\_Eco} = 2606kg \end{aligned} \tag{4.89}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
$W_{Galley\_Business}$	Galley business class mass per pax, [kg]	7	7
$W_{Galley\_Prem\_Eco}$	Galley premium economy class mass per pax, [kg]	7	7
W_{Galley}	Galley economy class mass per pax, [kg]	6	6

4.5.7 Additions Options

The formula for the additional options (oao) weight from [15] reads as follows:

$$W_{oao} = 1\% \cdot W_{MWE} = 1436kg \tag{4.90}$$

It contains the following elements:

symbol	description	value 1^{st} iteration	value final iteration
W_{MWE}	maximum weight empty, [kg]	144736.079	143665.236

4.5.8 Operator items weight overall

To get the overall operator items weight, all weights from chapter 4.5 have to be summed up:

$$W_{Operator\_items} = W_{ouf} + W_{ofc} + W_{oc} + W_{ops} + W_{oca} + W_{ogs} + W_{oao} \tag{4.91}$$

As mentioned before the elements of this chapter do not consider all the operator items equipped in an aircraft. So with the help of the given weight of the operator items for the reference aircraft a correction factor is calculated:

$$cf_{Operator\_items} = 1.326 \tag{4.92}$$

So after multiplication with the correction factor the final operator items weight of the new aircraft is

$$W_{operator\_items} = 24430.787kg|24531.157kg \tag{4.93}$$

4.5.9 Operating weight empty

All previous values from this chapter add up to the operating weight empty (OWE) and a specific center of gravity in x-direction of the aircraft. This value is important, further calculations will be done with it. For the final iteration of the aircraft it is:

$$W_{OWE} = 168196kg \tag{4.94}$$

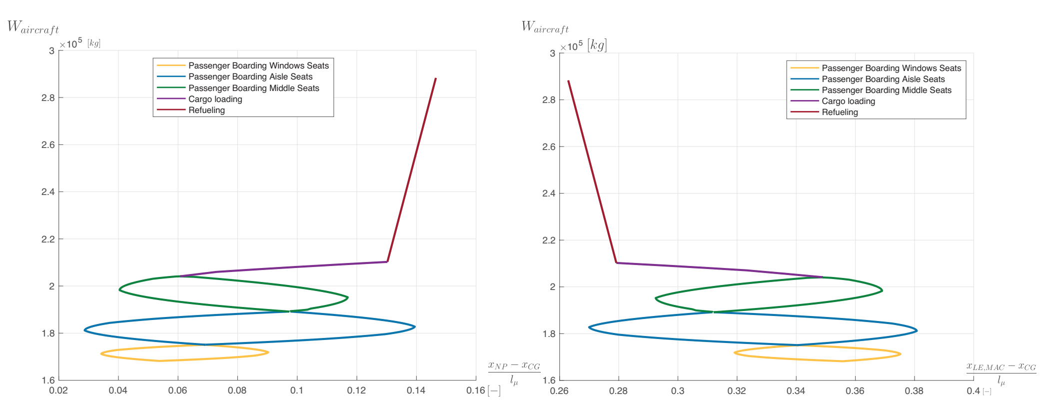
$$x_{CG\_OWE} = 40.607m \tag{4.95}$$

4.6 Loading diagram

In order to validate the longitudinal stability the loading diagram was created in order to observe CG shifts with added weight.

4.6.1 Passenger Loading

For passenger loading the standard assumption for passenger boarding was considered. This includes two procedures where passenger board from forward to rear as well as from rear to front. For each window seats are filled first, aisle seats second and middle seats at last. That boarding procedure is shown in figure 4.2 where yellow represent window seats, blue the aisle and green the middle seats.



(a) longitudinal stability for loading (b) Centre of gravity in percentage of MAC

Figure 4.1: Shift of CG due to loading

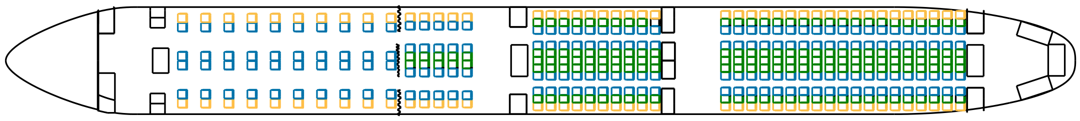


Figure 4.2: Passenger Loading Procedure

4.6.2 Refueling

After the wing design has been fixed, a wing tank was created by using connecting span wise 15% chord for the front spar and 65% chord for the rear spar. For each change in wing geometry a new point was added. With that a tank with approximately $258m^3$ can be estimated. Moreover, using CATIA CAD Model a CG was determined at $x_{CG\_WingTank} = 40880mm$. For refueling weight was added subsequently to the wing tank CG. Alternatively, the CG of the Wing could be used but as the wing also consists of other components the CAD estimated value was deemed more precise.

4.6.3 Cargo

As the LD3 Container, which are necessary to store SPP Pax's luggage, are positioned in the front section of the fuselage. Moreover, exact positions for each container have been set. Hence, when loading the cargo containers weight is added to that position. As mentioned in Chapter 4.5.3 each container has a mass of $75kg$ each and can accommodate luggage for 40 Pax. Considering the $30kg$ Pax standard luggage each container has an approximately weight of $1275kg$.

Payload and range 5

5.1 Range Estimation

5.1.1 Mission Profile

As a first step the mission profile which the aircraft flies is determined:

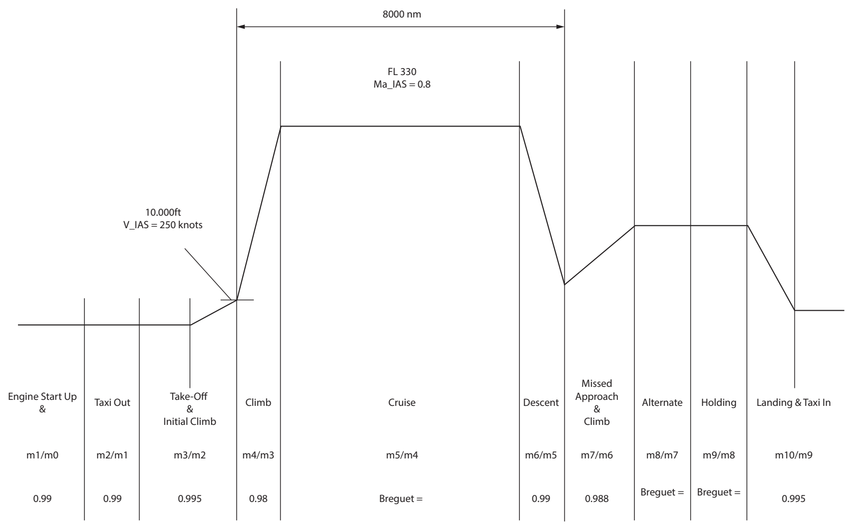


Figure 5.1: Mission Profile for 8000nm mission.

For each section a fraction of the weights at the beginning and at the end of the section is given from Roskam part I [6]. Some parts like cruise, holding or diversion to alternate are calculated with the Breguet formula which takes speed, specific fuel burn, L/D and the weights at the beginning and the end of the section into credit to calculate the range.
The detailed calculations can be found in Appendix 14.

5.1.2 MTOW and fuel weight for SPP

The following elements will be used for calculation:

symbol	description	value final iteration
m_{OWE}	operating weight empty, [kg]	168196.39
m_{crew}	crew weight, [kg]	2200
$m_{SPP\_payload}$	payload weight SPP mission, [kg]	46200
v_c	cruise speed, [m/s]	239.407
c_{TL}	specific fuel burn, [lb/hr/lbf]	0.3372
L/D	lift over drag	27.2697
v_{FL10}	speed while flying under 10000ft*), $[m/s]$	128.611

*) Air traffic limitations while flying under 10000ft do not allow to fly faster than 250 knots.

To calculate the range which is travelled during climb or descent the Breguet formula is used. It is assumed that the speed during climb and descent is the average between v_{FL10} and v_C: $v_{climb/descent} = \frac{(v_{FL10}+v_C)}{2} = 187.002\frac{m}{s}$. The specific fuel consumption is considered 20% higher than it is during normal cruise, so is assumed that:

$$c_{TL\_climb/descent} = c_{TL} \cdot 1.2 = 0.4046 \tag{5.1}$$

With all values filled in the ranges can be calculated:

$$R_{climb} = \frac{v_{climb}}{c_{TL\_climb}} \cdot L/D \cdot ln\left(\frac{m_3}{m_4}\right) = 901.909km = 486.992\text{nm} \tag{5.2}$$

$$R_{descent} = \frac{v_{descent}}{c_{TL\_descent}} \cdot L/D \cdot ln\left(\frac{m_6}{m_7}\right) = 448.677km = 242.266\text{nm} \tag{5.3}$$

These distances can be subtracted by the overall demanded range:

$$\begin{aligned} R_{new} = R - R_{climb} - R_{descent} &= 8000\text{nm} - 486.992\text{nm} - 242.266\text{nm} \\ &= 7270.742\text{nm} \end{aligned} \tag{5.4}$$

In the next step the missing mass fractions for diversion to alternate and holding have to be calculated. Based on general experience it is assumed that $R_{alternate} = 300\text{nm}$ and $R_{holding} = 100\text{nm}$. With that distances and a shifted form of the Breguet formula both mass fractions can be evaluated:

$$\frac{m_8}{m_7} = exp\left(-\frac{R_{alternate} \cdot c_{TL\_descent}}{v_{FL10} \cdot L/D}\right) = 0.982 \tag{5.5}$$

$$\frac{m_9}{m_8} = exp\left(-\frac{R_{holding} \cdot c_{TL\_descent}}{v_{FL10} \cdot L/D}\right) = 0.994 \tag{5.6}$$

Now the most important mass fraction $\frac{m5}{m4}$ for the mass loss during cruise flight over the new overall demanded range can be evaluated:

$$\frac{m_5}{m_4} = exp\left(-\frac{R_{new} \cdot c_{TL}}{v_C \cdot L/D}\right) = 0.777 \tag{5.7}$$

So the overall mass fraction from beginning to end of mission is:

$$\frac{m10}{m0} = \frac{m1}{m0} \cdot \frac{m2}{m1} \cdot \frac{m3}{m2} \cdot \frac{m4}{m3} \cdot \frac{m5}{m4} \cdot \frac{m6}{m5} \cdot \frac{m7}{m6} \cdot \frac{m8}{m7} \cdot \frac{m9}{m8} \cdot \frac{m10}{m9} = 0.746 \tag{5.8}$$

This fraction has to be calibrated with a factor which is get when you do the same calculation for the reference aircraft and compare it to the given value $cf_{RefAC} = 0.985$. So the new fraction is given with:

$$\frac{m_{10}}{m_0} = cf_{RefAC} \cdot \frac{m10}{m0} = 0.735 \tag{5.9}$$

This defines the fuel fraction of the aircraft:

$$\frac{m_{fuel}}{m0} = 1 - \frac{m10}{m0} = 0.265 \tag{5.10}$$

And the MTOW of the aircraft can be calculated:

$$m_{MTOW} = m_0 = 1/\frac{m10}{m0} \cdot (m_{OWE} + m_{crew} + m_{SPP\_payload}) = 294681.53kg \tag{5.11}$$

To ensure that the calculations are valid, the overall mass fraction $\frac{m10}{m0}$ can be set in the Breguet formula:

$$R_{validation} = \frac{v_C}{c_{TL}} \cdot L/D \cdot ln\left(\frac{m_0}{m_{10}}\right) = 11586.28\text{nm} \tag{5.12}$$

This value is far above the demanded 8000nm because in this case no other mission parts are considered.

5.1.3 Payload Variation

Now that the point of standard passenger payload (SPP) and maximum take of weight (MTOW) of the aircraft are known, two other points can be calculated: the range with maximum payload and the range with MTOW and maximum fuel. Both points can be calculated with the help of the Breguet formula.
The maximum payload is defined with $m_{maxPP} = 46200kg$. With this payload the range of the aircraft is reduced. The new range can be calculated with the Breguet formula in which m_4 and m_5 are calculated manually.

$$\begin{aligned} R_{maxPP} &= R_c + \frac{v}{c_{TL}} \cdot L/D \cdot ln\left(\frac{m_{MTOW} \cdot \frac{m_1}{m_0} \cdot \frac{m_2}{m_1} \cdot \frac{m_3}{m_2} \cdot \frac{m_4}{m_3}}{m_{10\_maxPP} \cdot \frac{m_9}{m_{10}} \cdot \frac{m_8}{m_9} \cdot \frac{m_7}{m_8} \cdot \frac{m_6}{m_7} \cdot \frac{m_5}{m_6}}\right) + R_d = \\ &= 7082.87\text{nm} \end{aligned} \tag{5.13}$$

In this case m_{fuel} is equal to $m_{MTOW} - m_{10\_maxPP} = 69285.14kg$
For the range and fuel weight with maximum fuel and maximum take off weight, a design decision has to be made. It has to be determined how much fuel the aircraft is possibly able to carry. Measurement of the wingbox of the new airplane got a value of $V_{fuel} = 185.31m^3$. With a density of $750\frac{kg}{m^3}$ for cerosine a theoretical mass of 138982kg of fuel is get. Because this mass would exceed the maximum take off weight, even when no payload is loaded, a lower value for the maximum fuel mass has to be chosen. It is decided that the maximum fuel mass should be 110 000 kg. Before the calculation starts $m_{10\_(maxfuel+MTOW)}$ and $m_{PP\_(maxfuel+MTOW)}$ have to be found:

$$m_{10\_maxfuel+MTOW} = m_{MTOW} - m_{maxfuel} = 184681.526kg \tag{5.14}$$

$$m_{PP\_maxfuel+MTOW} = m_{10\_maxfuel+MTOW} - m_{OWE} - m_{crew} = 14285.136kg \tag{5.15}$$

Now the range with maximum fuel weight and MTOW can be calculated:

$$\begin{aligned} R_{maxfuel+MTOW} &= R_c + \frac{v}{c_{TL}} \cdot L/D \cdot ln\left(\frac{m_{MTOW} \cdot \frac{m_1}{m_0} \cdot \frac{m_2}{m_1} \cdot \frac{m_3}{m_2} \cdot \frac{m_4}{m_3}}{m_{10\_maxfuel+MTOW} \cdot \frac{m_9}{m_{10}} \cdot \frac{m_8}{m_9} \cdot \frac{m_7}{m_8} \cdot \frac{m_6}{m_7} \cdot \frac{m_5}{m_6}}\right) + R_d = \\ &= 7082.87\text{nm} \end{aligned} \tag{5.16}$$

At last, the ferry range has to be calculated. At this point, no more payload is in the aircraft, only crew and maximum fuel ($m_{0\_ferry} = m_{OWE} + m_{crew} + m_{maxfuel}$). This is the maximum range of the aircraft:

$$\begin{aligned} R_{ferry} &= R_c + \frac{v}{c_{TL}} \cdot L/D \cdot ln\left(\frac{m_{MTOW} \cdot \frac{m_1}{m_{0\_ferry}} \cdot \frac{m_2}{m_1} \cdot \frac{m_3}{m_2} \cdot \frac{m_4}{m_3}}{m_{10\_ferry} \cdot \frac{m_9}{m_{10}} \cdot \frac{m_8}{m_9} \cdot \frac{m_7}{m_8} \cdot \frac{m_6}{m_7} \cdot \frac{m_5}{m_6}}\right) + R_d = \\ &= 15158.821\text{nm} \end{aligned} \tag{5.17}$$

5.2 Payload Range Diagram

All points needed for the payload range diagram are displayed in the following table:

name	range [nm]	payload weight [kg]	fuel weight [kg]
standard passenger payload (SPP)	8000	46200	78085
maximum passenger payload	6501	55000	69285
maximum fuel, MTOW	13999	14285	110000
ferry range	15158	0	110000

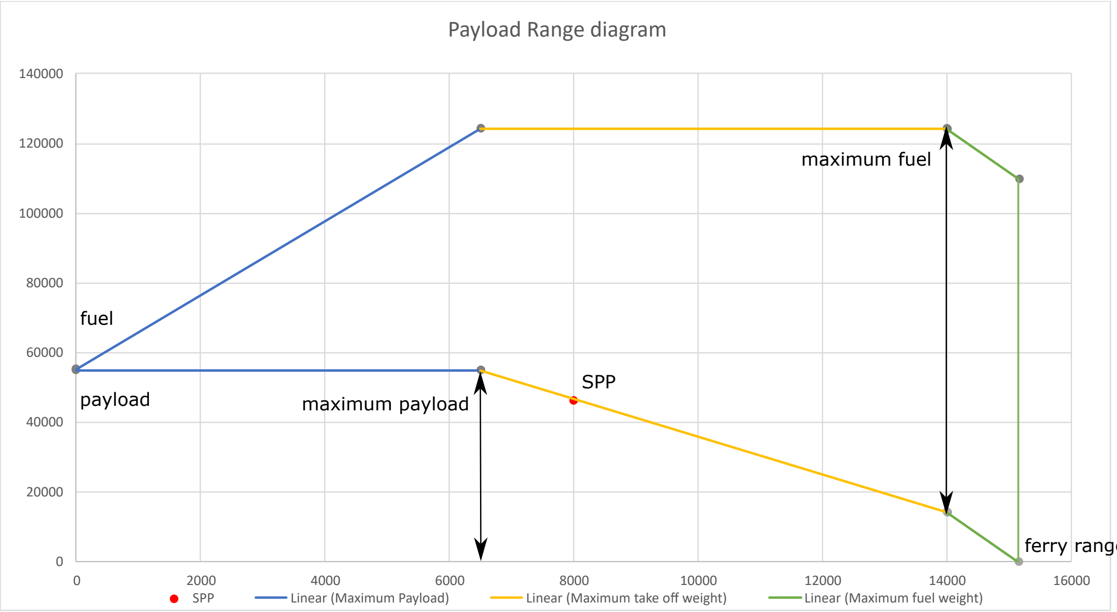


Figure 5.2: Payload range diagram

6 Structure and loads

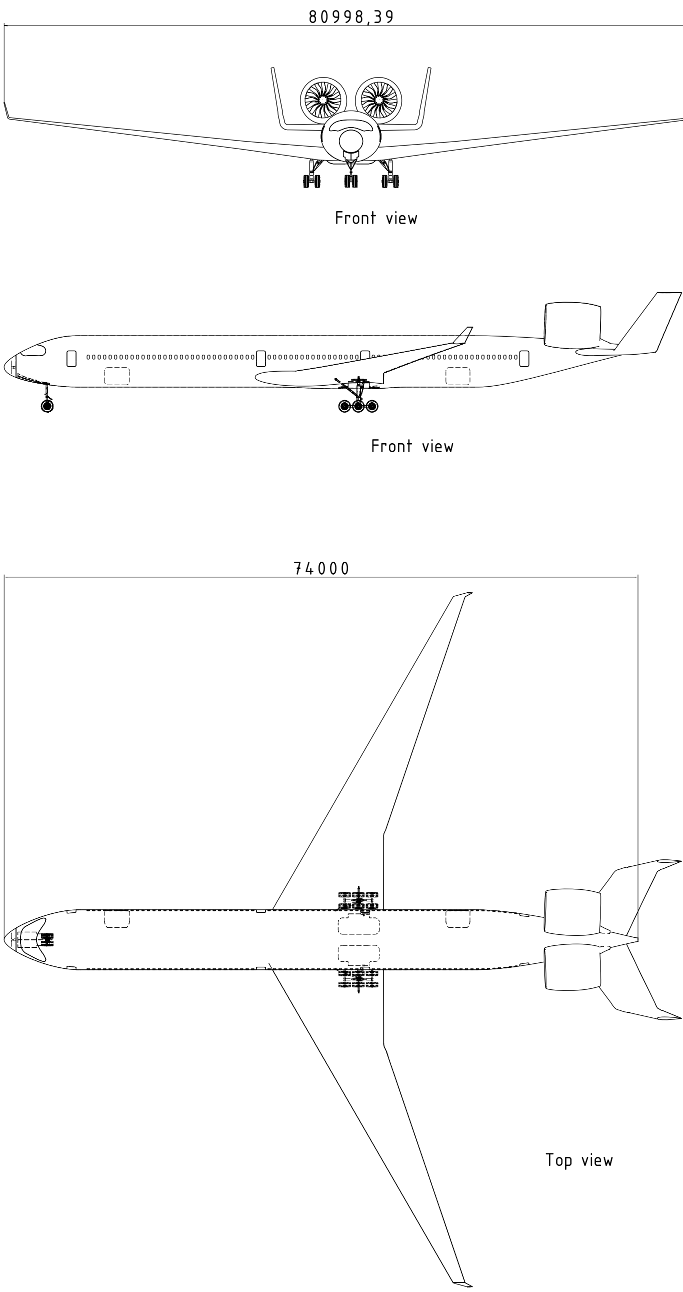


Figure 6.1: Aircraft 3 side view

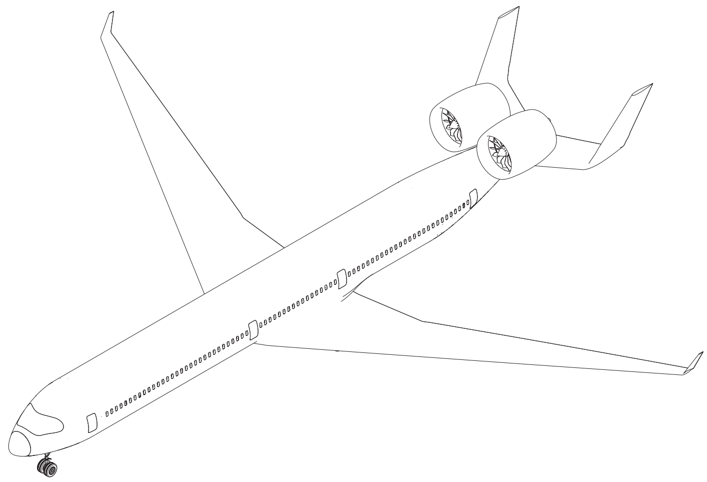


Figure 6.2: Isometric view

6.1 Fuselage

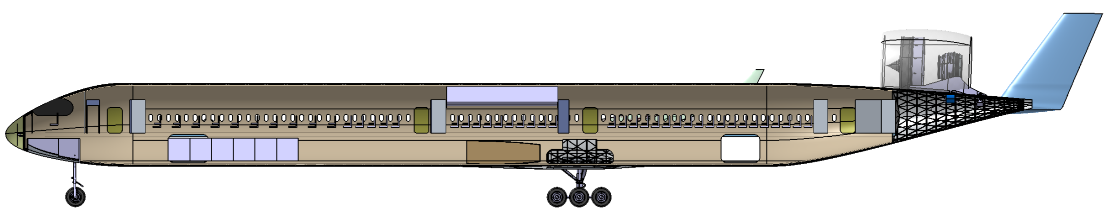
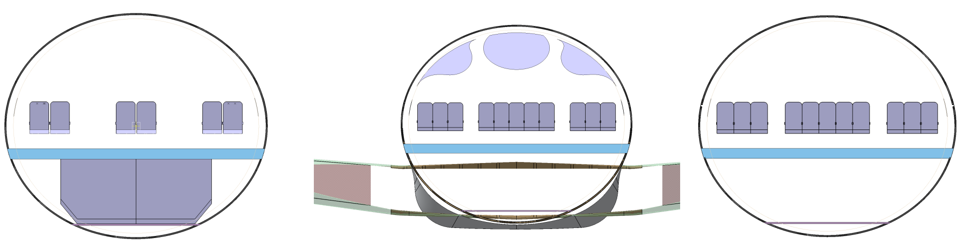


Figure 6.3: Fuselage half cut

As the central part of the whole fuselage it not only needs to bear the loads due to the pressure difference but also from other components such as the wings, landing gear, and empennage. Therefore, the fuselage needs to offer sufficient space and mountings for all components while accommodating the passengers and their luggage.



(a) BC cross section (b) PE cross section (c) Economy cross section

Figure 6.4: Fuselage Cross Sections

6.1.1 Main load bearing Structure

6.1.2 Geodesic Structure

In order to further investigate possible light weight structures high performance light weight composite materials were the obvious choice. For additional weight savings a research study was conducted. This concluded that a possible Geodetic helical stringer and frame construction of composite might be very promising. The paper "Development of Geodesic structures" from 2012 by Vasiliev et.al. concluded that with such structure a weight saving of 30-40% compared to a similar aluminium stringer frame structure[2]. As the reference Aircraft already consists mostly of composite material, the potential weight savings might be way lower than stated in the paper. Anyhow, there might be potential further weight saving as [2] states. After WWII aircraft with aluminium structures had a weight decrease of about 30% compared to conventional structural layouts. Therefore, if within the same material a 30% and Vasiliev et.al. conclude that from conventional aluminium structure to a geodetic composite a 30-40% weight margin is possible a structural weight saving of 20% from the reference aircraft might be realistic. This only take pure structural mass into account. Anyhow, the fuselage weight estimation includes more than just the basic structure. Moreover, considering that for the fuselage component weight half is purely structural. Hence, a prospective weight saving of 10% on overall fuselage weight seems realistic.

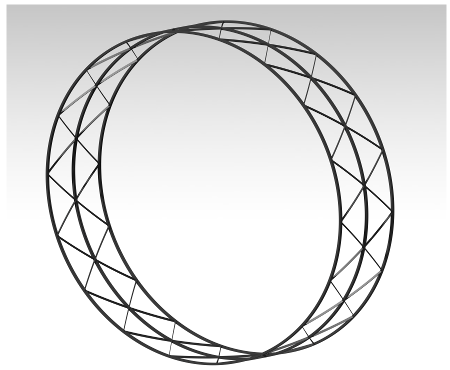
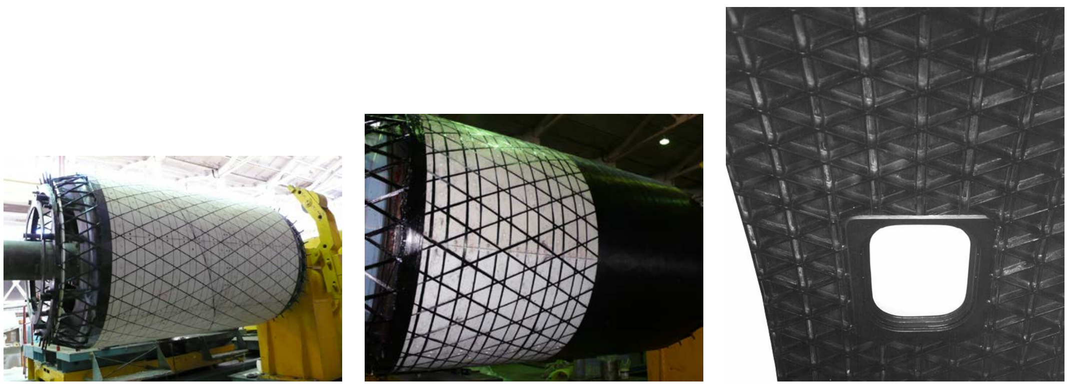


Figure 6.5: Example for geodesic frame-stringer sections



(a) Winding of carbon fibres on mandrel (b) Skin winding on top of base strucuture (c) Integral manufactured window

Figure 6.6: Manufacturing process of geodesic fuselage structure. Images from [2].

Furthermore, the wing structure of the new designed aircraft could also be realized using geodesic structures e.g. wing ribs, struts and further parts. As the structural fraction of the wing weight might be higher than for the fuselage 70-75% are considered purely structurally. This leads to the assumption that 15% weight saving with a geodetic structure might be plausible. The principle structure is shown for one exemplary section part in Figure 6.5. Additional weight might be added due to structure that becomes necessary for connecting and mounting main structures to the fuselage. For the fuselage skin Vasiliev et.al. recommends a thickness of 1.5 mm and 30-40 mm height for stringer and frames. For this design a skin thickness of 3mm and 40mm height was chosen in order to consider sufficient space. All stringers are manufactured with preimpregnated uni-directional carbon fibre. These are laid into a mandrel with groves. These plies forming

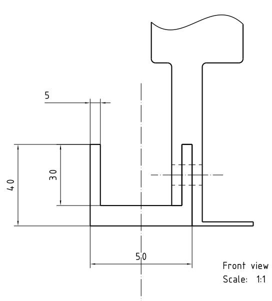


Figure 6.7: Connection to fuselage frames

6. STRUCTURE & LOADS

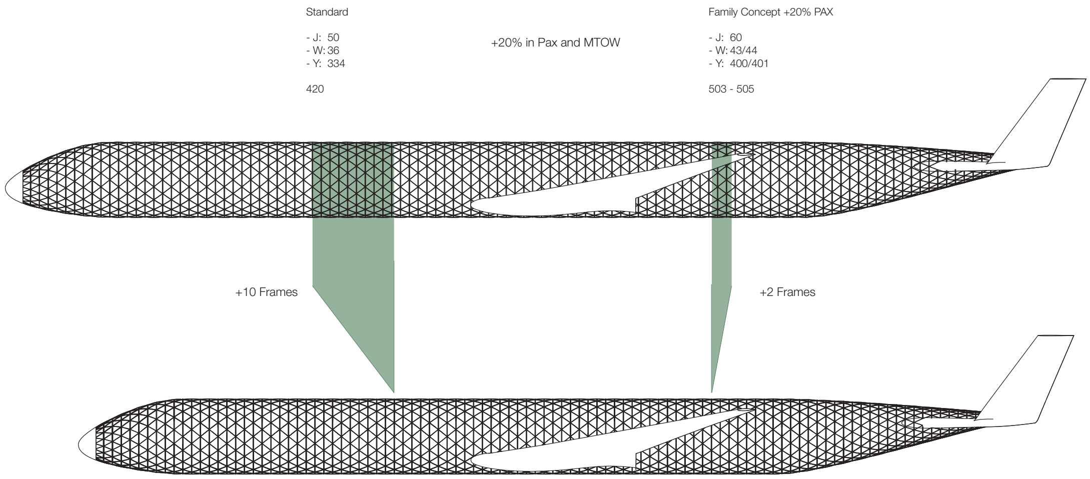
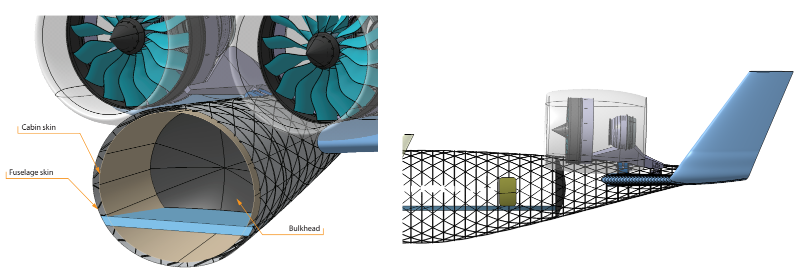


Figure 6.8: Stretch for family concept

the helical structure. The Frames are placed within the same step as shown in figure 6.6b. Subsequently the skin is winded onto that structure. Modern airliner such as the Boeing 787 already uses a similar technique for fuselage skin fibre placement. Therefore, also technology readiness was assumed to be ready for a 2030 aircraft. After the curing process the plastic mandrel is removed leaving the pure carbon fibre structure. As in conventional aircraft manufacturing the fuselage might be split into sections. The main difference is that the whole fuselage with all its cut-outs is manufactured as one integral part. These holes for Windows, doors, and cargo doors are manufactured within the same step. As an example a window from Vasiliev et. at. for a short range aircraft is shown in figure 6.6c. The cross section of the frames is an U shaped profile shown in figure 6.7. For connecting any load bearing structure to this a negative shaped extension. This structure can be connected to the frames with rivets. The larger upper section allows connecting larger bolts to the fuselage structure. As an example the connection is shown in figure 6.10. The cabin floor struts can be connected in a similar manner. Anyhow, the connecting frames don't need to be around the whole frame but only for small sections. First estimations gave that the stretched variant needs to be 8m longer in order to accommodate 506 or more passenger. Hence, for the family concept a stretch of 10 frames in front of the wing and 2 frames are added behind the wing. This is results in a total fuselage length of 82m for the family concept.

6.1.3 Bulkhead



(a) Pressure bulkhead

(b) Bulkhead in side view with geodetic structure

Figure 6.9: Bulkhead position and structure

Similar to the fuselage structure the bulkhead will also be manufactured with carbon composite materials. Sufficient space as considered at the aft section of the fuselage.

6.2 Wing Mounting

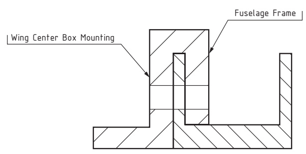
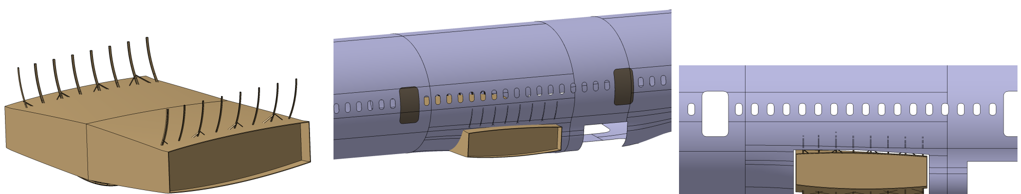


Figure 6.10: Connection of the frames with rivets

The new designed aircraft will also have a low wing. Therefore, mounting the wing will be achieved by connecting the wing box of each wing to a center wing box. The center wing box itself is a very rigid and massive structure which connects both wings to the fuselage.



(a) Centre Wing Box (b) Wing Center Box inside fuselage (c) Side view

Figure 6.11: Design of the wing box

6.2.1 Flight Controls

For the actuators of the morphing wing structure is sufficient space considered between the wing box and trailing as well as leading edge. As A. Concillio et al. states in [9] the detailed structure needs to be adjusted to the final wing and flight profile. Hence, for the preliminary design only sufficient space and twice as heavy flight controls were considered. Even though A. Concillio et al. argue that there might be significant weight increase which needs to be avoided be any means the technology is evaluated by them to be at TRL 9 in 10 to 15 years from 2015. Hence, making it feasible for this aircraft application.

6.3 Landing Gear

As the engines of the new design will be mounted to rear part of the fuselage the landing gear does not provide sufficient ground clearance. This leads to a significant size reduction of both front and main landing gears and resulting in a slight decrease in weight, at least for this component.
Additionally, no further technology improvements were considered. Over time landing gear technology will advance and come with potential weight savings. Anyhow,these were not further considered.
According to the Airbus slides 31t per wheel can be considered on large aircraft. As the MTOW is estimated to be between 310t and 380t 10 to 12 main landing gear wheels are necessary. For this design the Basic Version and the Family Stretch will have both 12 main landing gear wheels. With improved materials and new technologies it might also be possible

to even lower the amount to 8 wheels for the Basic Version. Additional weight savings could be achieved by considering different wheel configurations for the Basic Version.
The front landing gear will remain a conventional two wheel layout.

6.3.1 Front Landing Gear

Figure 6.12: Mounting of the front landing gear

6.3.1.1 Positioning

The front landing gear is positioned directly below the cockpit and behind the front nose cone. When retracted the wheels lay directly behind the fuselage nose cone. There two wheels at the front landing gear which are centered around the middle of the fuselage's x-axis.

6.3.1.2 Fuselage Mounting

6.3.1.3 Retraction Mechanism

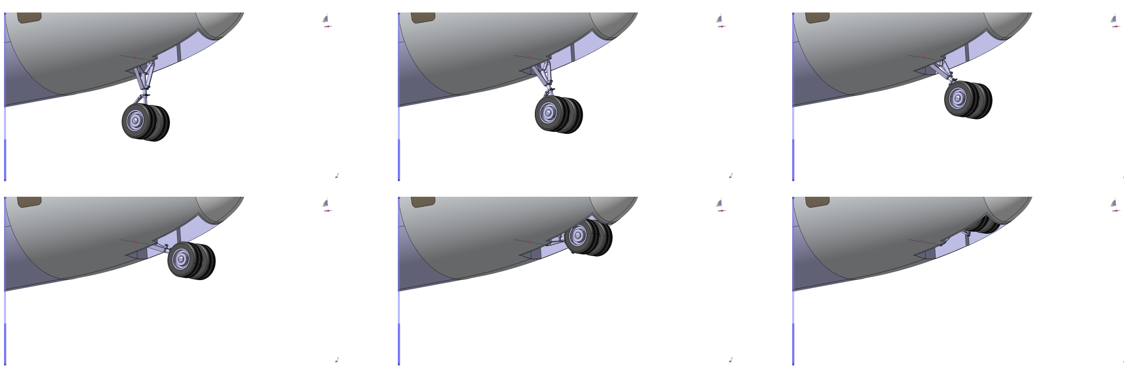
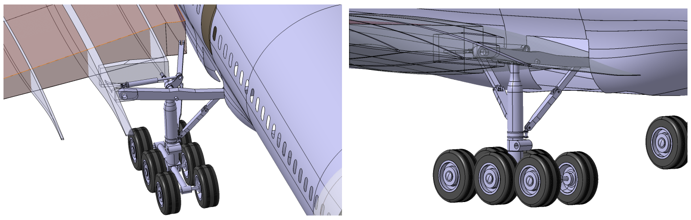


Figure 6.13: Retraction of front landing gear

6.3.2 Main Landing Gear

6.3.2.1 Positioning

The landing gear was positioned in a way so that it provides enough clearance at the rear section for a sufficient AoA. Moreover, it needs also to be in a 15°degree from the furthest



(a) Mounting of the main landing gear (b) Side view of the mounting structure

Figure 6.14: Main landing gear mounting

aft located CG. Span wise positioning was orientated so that the wheels wont intersect in the middle. A position below the wing was a good choice.

6.3.2.2 Wing/Fuselage Mounting of Center Beam

The front mounting of the main shock strut is connected to the rear Wing Box. The rear section is mounted to the Landing Gear Center Beam. The Center Beam is responsible for taking the vertical torque and thrust loads. Hence, this structure needs to be quite massive. The side torque links are also connected to either Center Beam or Wing Box.

6.3.2.3 Retraction Mechanism

For the retraction of the main landing gear one hydraulic actuator is considered. This actuator is fully extended when the gear is down and locked, allowing an easier gravity extension. The full concept is shown in Figure **??**.
When the Main Landing Gear is retracted it lays mainly in the belly of the fuselage directly behind the Center Wing Box. Struts and actuators are positioned inside the Wing. The belly fairing also allows a bigger cover of the Landing Gear.

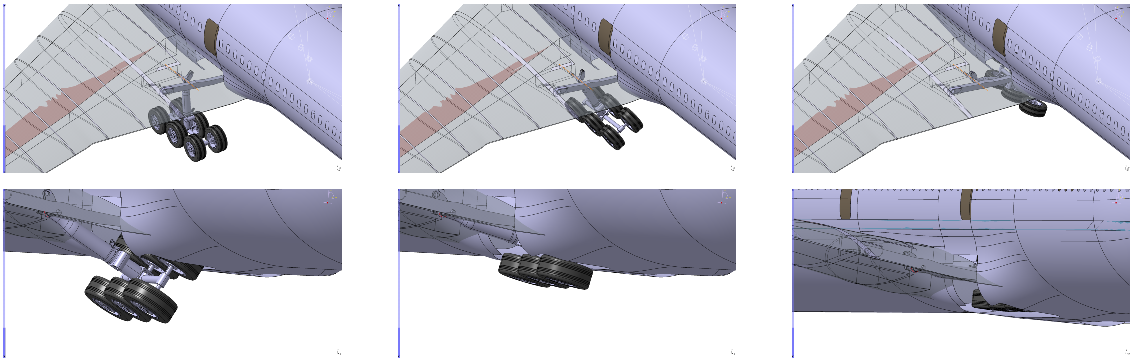
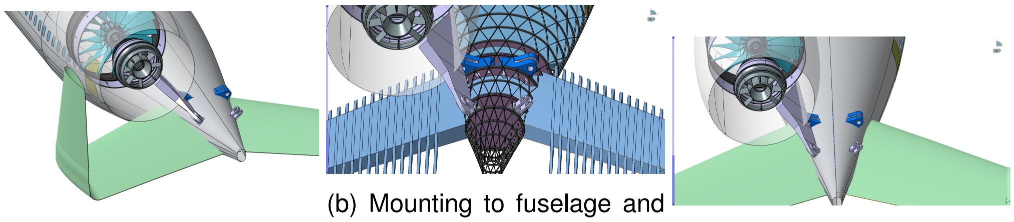


Figure 6.15: Retraction of main landing gear

6.4 Engine Mounting

The engines are mounted with a pylon in the aft section. The main connection will be made with large bolts connecting and holding in place the engines.

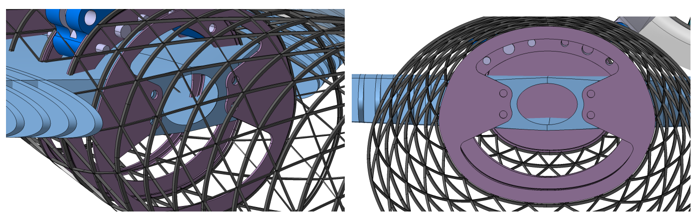


(a) Engine mounting (b) Mounting to fuselage and aft frames (c) Close up to the connection

Figure 6.16: Engine Mounting

6.5 Empennage Mounting

As the empannage will also use morphing wing technology a large hole needs to be considered for all electronics. Moreover, also all pipes for the engine need to pass through that frame. Therefore, the empennage will also have a large wing box with four holes on each side. Through these holes large bolts mount the empennage to the fuselage.



(a) Empennage Center Box with connections (b) Large hole for actuator and electronic supply

6.6 Gust and Manoeuvre loads

To display all the loads which are acting upon the aircraft a diagram with velocity over load needs to be created. CS 25.337 gives all relevant methods to calculate the different velocities and loads which are acting upon the aircraft in its mission.
The detailed calculations can be found in Appendix 15

6.6.1 Maneuver loads

6.6.1.1 Cruising speed

The assignment sets a fix cruising velocity of $Ma = 0.8$ in height of 33000ft. This velocity has to be converted into an absolute velocity with m/s as unit. To do this the temperature at 33000 ft has to be figured out. With the standard decline of temperature over altitude of 0.0065 K/m and the ISA standard atmosphere at 0ft it is calculated:

$$T_{FL33} = 288.15K - 0.0065\frac{K}{m} \cdot (33000ft \cdot 0.3048\frac{m}{ft}) = -50.38\,°\text{C} \tag{6.1}$$

With this temperature and the given Mach number a cruising velocity v_C can be calculated:

$$v_C = Ma \cdot \sqrt{\kappa \cdot R \cdot T} = 0.82 \cdot \sqrt{1.4 \cdot 287.15\frac{J}{kg \cdot K} \cdot 222.77K} = 239.407\frac{m}{s} \tag{6.2}$$

6.6.1.2 Limit load factor cruise

CS25.337 gives the formula for the positive limit load factor:

$$n_+ = 2.1 + (24000/(W_{dg} + 10000)) \tag{6.3}$$

For the own aircraft this load factor would $n_+ = 2.136$ This value is smaller than 2.5 so it has to be set to a least 2.5 and not greater than 3.8. It is decided to set it to $n_+ = 2.5$. The negative load factor should not be smaller than $n_- = -1.0$.

6.6.1.3 Stalling speed without highlifting devices

The stalling speed without any flaps extended is calculated as follows:

$$v_{S1} = \sqrt{(2 \cdot W_{dg} \cdot g)/\rho \cdot c_A \cdot S} = 75.195 \frac{m}{s} \tag{6.4}$$

6.6.1.4 Design Maneuvering speed

The desgin maneuvering speed is calculated as follows given by CS25.337:

$$v_A = v_{s1} \cdot \sqrt{n_+} = 118.893 \frac{m}{s} \tag{6.5}$$

6.6.1.5 Design dive speed

The design dive speed should not be smaller than the following value given by CS25.337:

$$v_D = v_C / 0.8 = 299.259 \frac{m}{s} \tag{6.6}$$

So it is determined to $v_D = 320 \frac{m}{s}$.

6.6.1.6 Maneuver loads overall

So after calculation of all necessary speeds and loads for maneuvering the following table can be generated, each line gives one point in the V-n-diagram:

speed	description	load	description
v_C	cruising speed	n_+	limit load factor positive
v_{S1}	stall speed without flaps	n_{f+}	load factor with flaps positive, defined as 1
v_A	design maneuvering speed	n_{A+}	load factor maneuvering positive, defined by diagram
v_D	design dive speed	n_+	limit load factor positive
v_C	cruising speed	n_-	limit load factor negative
v_{S1}	stall speed without flaps	n_{s-}	load factor with flaps negative, defined by diagram
v_A	design maneuvering speed	n_{A-}	load factor maneuvering negative, defined as -1
v_D	design dive speed	n_-	limit load factor negative, defined as 0

6.6.2 Gust loads

6.6.2.1 Gust velocity at cruising altitude

To calculate the velocity of gust at a specific cruising altitude, CS25.337 (5)(i) provides that the velocity of a gust at $15000 ft$ is $44 \frac{ft}{s}$ and at $60000 ft$ it is $20.86 \frac{ft}{s}$. Between this two altitudes the velocity of a gust decreases linearly. So a function with the help of these two points is created and the needed altitude fit in to get the needed velocity of a gust at the given cruising altitude:

$$U = \frac{20.86 - 44}{6000 - 15000} \cdot 33000 + 51.7133 = 34.744 \frac{ft}{s} \tag{6.7}$$

6.6.2.2 Gust factor K

To calculate the gust factor K which should represent the kind of cosine-like intensity which the gust has an effect on the aircraft CS25.337 gives the following formula:

$$K = \frac{0.88 \cdot \mu}{5.3 \cdot \mu} = 0.805 \tag{6.8}$$

μ is calculated by:

$$\mu = \frac{2 \cdot w}{\rho_{FL33} \cdot c \cdot a \cdot g} = 55.24 \tag{6.9}$$

It contains the following elements:

symbol	description	value
w	wing loading W_{dg}/S, $[lb/ft^2]$	115.69
ρ_{FL33}	density of air at cruising altitude, taken from ISA standard table, $[slugs/ft^3]$	$79.501 \cdot 10^{-5}$
c	mean geometric chord of the wing, $[ft]$	28.01
a	slope of the aeroplane normal force coefficient curve, $[1/rad]$	5.8443
g	acceleration due to gravity, $[ft/sec^2]$	32,185

6.6.2.3 Design speed for maximum gust intensity

The design speed for maximum gust intensity should not be smaller than the following value given by CS25.337:

$$v_B = v_{S1} \cdot (1 + \frac{K \cdot U \cdot v_C \cdot a}{498 \cdot w})^{0.5} = 222.49\frac{m}{s} \tag{6.10}$$

6.6.2.4 Load factors gust

The load factors from gust with different speeds are calculated as followed for v_C, v_D and v_B:

$$n_{C\_G+} = 1 + \frac{K \cdot U \cdot v_C \cdot a}{498 \cdot w} \quad n_{C\_G-} = 1 - \frac{K \cdot U \cdot v_C \cdot a}{498 \cdot w} \tag{6.11}$$

$$n_{B\_G+} = 1 + \frac{K \cdot U \cdot v_B \cdot a}{498 \cdot w} \quad n_{B\_G-} = 1 - \frac{K \cdot U \cdot v_B \cdot a}{498 \cdot w} \tag{6.12}$$

$$n_{D\_G+} = 1 + \frac{0.5 \cdot K \cdot U \cdot v_D \cdot a}{498 \cdot w} \quad n_{D\_G-} = 1 - \frac{0.5 \cdot K \cdot U \cdot v_D \cdot a}{498 \cdot w} \tag{6.13}$$

	n_{X_G+}	n_{X_G-}
v_C	2.317	-0.317
v_B	1.63	0.37
v_D	1.88	0.12

6.6.2.5 Gust loads overall

So after calculation of all necessary speeds and loads for gusts the following table can be generated, each line gives one point in the V-n-diagram:

speed	description	load	description
v_C	cruising speed	n_{C_G+}	load factor gust positive at v_C
v_B	design speed for maximum gust intensity	n_{B_G+}	load factor gust positive at v_B
v_D	design dive speed	n_{D_G+}	load factor gust positive at v_D
v_C	cruising speed	n_{C_G-}	load factor gust negative at v_C
v_B	design speed for maximum gust intensity	n_{B_G-}	load factor gust negative at v_B
v_D	design dive speed	n_{D_G-}	load factor gust negative at v_D

6.6.3 V-n-diagram

With all points from maneuver loads and gust loads overall a V-n-diagram can be created:

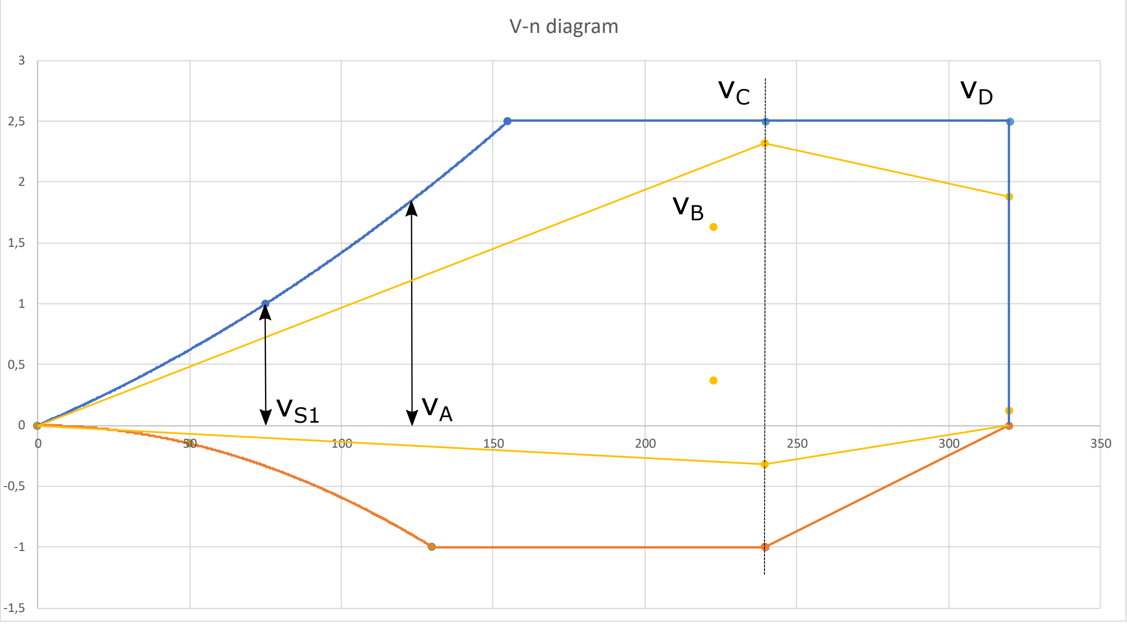


Figure 6.18: V-n-diagram

Trade Studies 7

7.1 Efficiency Parameters

For jet powered aircraft the fuel mass flow is depending on the thrust specific fuel consumption of the engines, the weight and the lift to drag ratio

$$\dot{m}_{Jet} = c_{TL} \cdot \frac{L}{D} \cdot W \tag{7.1}$$

In order to decrease fuel burn each parameter has an equal influence of the result. Hence, a 10% reduction of weight would be as good as 10% aerodynamic improvement. Even though, the lift-to-drag ratio varies in a range from 10 - 25 while the weight varies in much larger scope.

7.2 Identification of Trade Study Objects

The only considerable design parameters for our configuration which have an impact on the fuelburn as predicted by equation 7.1 was the variation of the wing shape. Therefore a variation of wingspan and the adaption of winglets were investigated.

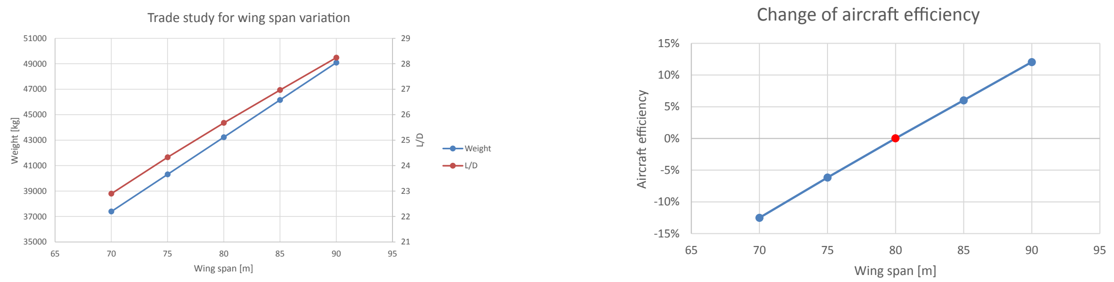
7.3 Variation of Wingspan

In this study the influence of a varied wingspan on the overall efficiency of the aircraft was investigated. Therefore different wingspans were selected by keeping the wing area constant. The method then included the calculation of the wing weight with empirical equations according chapter 4 and the calculation of $(L/D)_{max}$ with Aero Tool and subsequent calibration. Results for different wingspans are shown in fig. 7.1 and table 7.1.

Table 7.1: Trade study of varied wingspan

Wingspan [m]	Wing weight [kg]	$(L/D)_{max}$	Change [%]
70	37392.7	22.9	-12.5
75	40311.1	24.3	-6.1
80	43232.7	25.7	0
85	46158.0	27.0	6.0
90	43232.7	25.7	12.0

The efficiency of the aircraft derived from eq. (7.1) and is calculated by:



(a) (b)

Figure 7.1: Results of the wingspan trade study

$$\eta_{AC} = \left(\frac{(L/D)_{max_i}}{(L/D)_{max80\,\mathrm{m}}} \cdot \frac{m_{MTOW} - (m_{wing_{80\,\mathrm{m}}} - m_{wing_i})}{m_{MTOW}} - 1\right) \cdot 100\% \tag{7.2}$$

The results do not indicate any considerable design point, since the efficiency rises around projected aircraft reference wingspan (80 m) linearly. Therefore the design will stay in this configuration, because other boundary conditions like ground handling and airport sizing were already considered. That no sweet spot could be obtained can be founded in the fact that empirical formulas were used and again by the unratable results provided by the Aero Tool.

7.4 Adaption of Winglets

This study considers the adaption of winglets and its effect on $(L/D)_{max}$. This was simply obtained by a calculation of a wing with winglets, which was generated with the WingNet tool. Then a calculated with the Aero Tool was performed, but only for the induced drag because the drag prediction by XFOIL did not work. Subsequent the remaining drag of the wing from a calculation without winglets was added to the induced drag. With this provided data the comparison with the plain wing was carried out. It turned out that the wing with winglets has a higher $(L/D)_{max}$ of 26.15. According to [18], winglet weight can be estimated with a value of 0.3 % of the wing mass, which in our case leads to a weight penalty of 129.7 kg. Hence, this results in a value of 1.8 % for improving aircraft efficiency according to equation (7.1). Therefore, winglets are applied for the aircraft design as shown in figure 7.2.

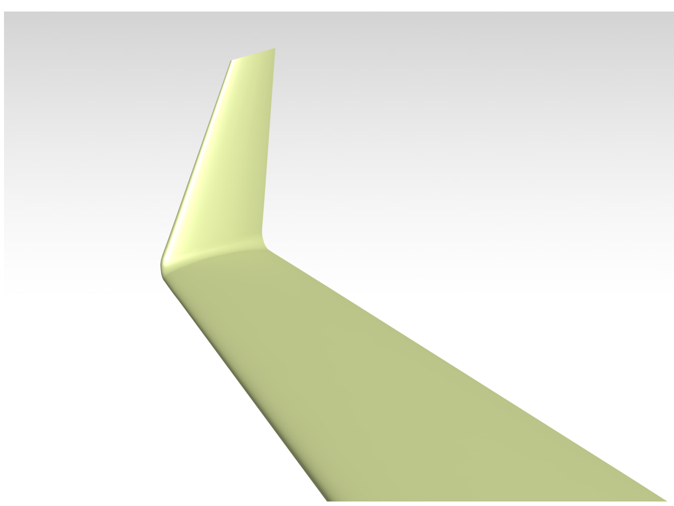


Figure 7.2: Applied winglets for projected wing

Direct Operating Costs and Turnaround 8

8.1 Direct Operating Costs

For calculating the Direct Operating Costs (DOC) the method developed by Thorbeck, TU Berlin [19], was used. This method uses simplifies equations for an estimation of DOC. All calculations were first made with data of the reference aircraft and hence calibrating, in a similar manner as before, each equation for higher result precision. As the method by Thorbeck does not provide an equation for Catering Costs, it is assumed that this stays the same as the reference aircraft. In order to calibrate the method a correction factor for the total. As for the Costs of Ownership are not stated for the reference aircraft this method is not calibrated and just used for a rough estimation.
The yearly flight cycles (FC) for the reference aircraft is given by the task with $600a^{-1}$.
The yearly flight cycles for the own aircraft is calculated as follows:

$$FC = \frac{T_{OT_{p.a.}}}{T_{FT} + T_{BTS}} = \frac{T_{ThOT_{p.a.}} - T_{DT_{p.a.}}}{T_{FT} + T_{BTS}} = 644 \tag{8.1}$$

symbol	description	unit	New Design
$T_{OT_{p.a.}}$	Yearly operating time	[hrs]	-
$T_{ThOT_{p.a.}}$	Theoretical yearly operating time	[hrs]	8760
$T_{DT_{p.a.}}$	Yearly forced down time	[hrs]	2746.8
T_{FT}	Flight time per flight	[hrs]	8.33
T_{BTS}	Block time supplement per flight	[hrs]	1

The yearly forced downtime is a value of experience taken from Thorbeck [19].
Using diagrams from Thorbeck the flight time per year correspondes to 5000 hours for a 4000 nm mission[19]. Dividing the flight time per year by the flight cycles per year of the reference aircraft of 600 results in an average flight time of $FT = 8.332hrs$.
The block time is one hour, a value demanded by the task.

8.1.1 Fuel Costs

??

$$DOC_{Fuel} = W_{BlockFuel} \cdot K_{Fuel} \cdot FC \tag{8.2}$$

symbol	description	unit	RefAC Value	New Design
$W_{BlockFuel}$	Block Fuel	[kg]	65850	42492
K_{Fuel}	Fuel cost	[€/kg]	0.8	0.8

After calculating the DOC_{Fuel} for the reference aircraft and comparing it with the given value of 50000$ for fuel per trip with consideration of an exchange rate of 1€=0.91$ a correction factor is obtained:

$$cf_{DOC\_fuel} = 0.86 \tag{8.3}$$

$$DOC_{fuel} = cf_{DOC\_fuel} \cdot DOC_{fuel} = 18,914,582.94€ \tag{8.4}$$

8.1.2 Airframe Maintenance Cost

$$DOC_{Maint,AF,Mat} = [W_{OWE} \cdot (0.21 \cdot FT + 13.7) + 57.5] \cdot FC \tag{8.5}$$

symbol	description	unit	RefAC Value	New Design
W_{OWE}	Operating Weight Empty	[t]	188.1	168.196
FT	Flight Time	[hrs]	8.333	

$$\begin{aligned} DOC_{Maint,AF,Pers} = LR \cdot (1 + B) \cdot [(0.655 + 0.01 \cdot W_{OWE} \cdot FT) + 0.254 \\ + 0.01 \cdot W_{OWE} \cdot FC \end{aligned} \tag{8.6}$$

symbol	description	unit	RefAC Value	New Design
LR	Labour Rate	[€/h]	50[19]	
B	Burden	[-]	2[19]	

So the airframe maintenance cost adds up to:

$$DOC_{Maint,AF} = DOC_{Maint,AF,Mat} + DOC_{Maint,AF,Pers} \tag{8.7}$$

After calculating the $DOC_{Maint,AF}$ for the reference aircraft and comparing it with the given value of 6000$ for airframe maintenance per trip with consideration of an exchange rate of 1€=0.91$ a correction factor is obtained:

$$cf_{DOC\_Maint,AF} = 0.952 \tag{8.8}$$

$$DOC_{Maint,AF} = cf_{DOC\_Maint,AF} \cdot DOC_{Maint,AF} = 3,155,648.43€ \tag{8.9}$$

8.1.3 Engine Maintenance Cost

$$DOC_{Maint,Engine} = [N_{engine} \cdot (1.5 \cdot F_{SLS} + 30.5 \cdot FT + 10.6) \cdot FC] \tag{8.10}$$

symbol	description	unit	RefAC Value	New Design
N_{Engine}	Number of engines	[-]	2	2
F_{SLS}	Sea-level static thrust	[t]	57.5	61.162

After calculating the $DOC_{Maint,Engine}$ for the reference aircraft and comparing it with the given value of 10000$ for engine maintenance per trip with consideration of an exchange rate of 1€=0.91$ a correction factor is obtained:

$$cf_{DOC\_Maint,Eng} = 12.963 \tag{8.11}$$

$$DOC_{Maint,Engine} = cf_{DOC\_Maint,Eng} \cdot DOC_{Maint,Engine} = 5,952,240.28€ \tag{8.12}$$

8.1.4 Crew Cost

$$DOC_{Crew} = N_{Crew} \cdot (K_{CC} + K_{FA} * N_{FA}) \tag{8.13}$$

symbol	description	unit	RefAC Value	New Design
N_{Crew}	Number of crew complement	[-]	5	5
K_{CC}	Average salary of cockpit crew per year	[€/yr]	300000[19]	
K_{FA}	Average salary of one flight attendant per year	[€/yr]	60000[19]	
N_{FA}	Number of flight attendants	[-]	18	18

After calculating the DOC_{Crew} for the reference aircraft and comparing it with the given value of 15000$ for crew per trip with consideration of an exchange rate of 1€=0.91$ a correction factor is obtained:

$$cf_{DOC\_Crew} = 1.187 \tag{8.14}$$

$$DOC_{Crew} = cf_{DOC\_Crew} \cdot DOC_{Crew} = 8,190,300€ \tag{8.15}$$

8.1.5 Landing Cost

$$DOC_{Landing} = (K_{ldg} \cdot W_{MTOW}) \cdot FC \tag{8.16}$$

symbol	description	unit	RefAC Value	New Design
K_{ldg}	Landing Fee rate	[€/kg]	0.01[19]	
W_{MTOW}	Maximum Take-Off weight	[kg]	377000	294681.53

After calculating the $DOC_{Landing}$ for the reference aircraft and comparing it with the given value of 3000$ for landing costs per trip with consideration of an exchange rate of 1€=0.91$ a correction factor is obtained:

$$cf_{DOC\_Landing} = 0.724 \tag{8.17}$$

$$DOC_{Landing} = cf_{DOC\_Landing} \cdot DOC_{Landing} = 1,373,970.30€ \tag{8.18}$$

8.1.6 Navigation Cost

For European domestic flights 100€/km, transatlantic flights 70€/km, far east flights 60€/km assuming average navigation costs for design, 76,666 = 77€/km

$$DOC_{Navigation} = K_{nav} \cdot \frac{R}{100} \cdot \sqrt{\frac{W_{MTOW}}{50}} \cdot FC \tag{8.19}$$

symbol	description	unit	RefAC Value	New Design
K_{nav}	Unit rate for navigation	[€/kg]	77[19]	
R	Range	[km]	7408	
W_{MTOW}	Maximum Take-Off weight	[t]	377	294.681

After calculating the $DOC_{Navigation}$ for the reference aircraft and comparing it with the given value of 6000$ for navigation costs per trip with consideration of an exchange rate of 1€=0.91$ a correction factor is obtained:

$$cf_{DOC\_Navigation} = 0.349 \tag{8.20}$$

$$DOC_{Navigation} = cf_{DOC\_Navigation} \cdot DOC_{Navigation} = 3,112,393.12€ \tag{8.21}$$

8.1.7 Catering Cost

Unfortunately no formula for catering cost is given by Thorbeek [19]. So the catering has to be considered in the overall correction factor for COC.

8.1.8 Pax and Ramp Handling Cost

$$DOC_{Ground} = (K_{ground} \cdot W_{DOC_PP}) \cdot FC \tag{8.22}$$

symbol	description	unit	RefAC Value	New Design
K_{ground}	Ground handling fee rate	[€/kg]	0.1[19]	
W_{DOC_PP}	Payload	[kg]	46200	46200

After calculating the DOC_{Ground} for the reference aircraft and comparing it with the given value of 15000$ for ground handling costs per trip with consideration of an exchange rate of 1€=0.91$ a correction factor is obtained:

$$cf_{DOC\_Ground} = 2.955 \tag{8.23}$$

$$DOC_{Ground} = cf_{DOC\_Ground} \cdot DOC_{Ground} = 8,791,952.40€ \tag{8.24}$$

8.1.9 Cash operating cost

If all previous DOCs are summed up, the cash operating cost is get. As mentioned before no formula for catering is available so the missing costs are pictured by calculating a correction factor with the values from the reference aircraft for the cash operating cost. To do this all calibrated values from before are added up.

$$\begin{aligned} cf_{cash\_operating\_cost} = 113000\$ * (&DOC_{fuel\_RefAC} + DOC_{Maint,AF\_RefAC} \\ &+ DOC_{Maint,Engine\_RefAC} + DOC_{Crew\_RefAC} + DOC_{Landing\_RefAC} \\ &+ DOC_{Navigation\_RefAC} + DOC_{Ground\_RefAC})^{-1} = 1.076 \end{aligned} \tag{8.25}$$

$$\begin{aligned} COC = cf_{cash\_operating\_cost} * (&DOC_{fuel} + DOC_{Maint,AF} \\ &+ DOC_{Maint,Engine} + DOC_{Crew} + DOC_{Landing} \\ &+ DOC_{Navigation} + DOC_{Ground}) = 53,261,836.98€ \end{aligned} \tag{8.26}$$

8.1.10 Capital cost

$$DOC_{capital} = (K_{OWE} * (W_{OWE} - W_{eng} * N_{eng}) + K_{eng} * W_{eng} * N_{eng}) * (A + Ins) \tag{8.27}$$

symbol	description	unit	RefAC Value	New Design
K_{OWE}	cost per kg OWE	[€/kg]	1150[19]	
W_{OWE}	operating weight empty	[kg]	188100	168196.39
W_{eng}	weight per engine	[kg]	11550	8462.22
K_{eng}	price per kg engine	[€/kg]	2500[19]	
Ins	insurance rate, % price A/C	[-]	0.5%[19]	
IR	intrest rate, % price A/C	[-]	5%[19]	
f_{RV}	residual value factor, % price A/C	[-]	10%[19]	
DP	depreciation	[a]	14[19]	
A	annual rate	[-]	0.0959	0.0959

$$A = IR * \frac{1 - f_{RV} * \left(\frac{1}{1+IR}\right)^{DP}}{1 - \left(\frac{1}{1+IR}\right)^{DP}} = 0.0959 \tag{8.28}$$

With the values of the final iteration of the new aircraft design the capital costs can be calculated as $DOC_{capital} = 21,826,696.28€$ per year.

8.1.11 Direct operating cost overall

To get the direct operating cost cash operating cost and capital cost have to be summed up. With $FC = 644$ they can be figured per flight or per year.

$$DOC_{total} = COC + DOC_{captial} = 75,088,533.26€ \tag{8.29}$$

$$DOC_{per\_flight} = \frac{COC}{FC} + fracDOC_{capital}FC = 116,592.71€ \tag{8.30}$$

In the summary form it is demanded to state different DOC values as a value of $€/(passenger * 100km)$. To get this value it is necessary to convert the different calculated DOCs from this chapter into this unit. This is realised by dividing by flight cycles per year (FC), a full aircraft with 424 Pax (N_{Pax}, and the range of the aircraft in 100km steps:

$$DOC_{example\_newUnit} = DOC_{example} * \frac{1}{FC} * \frac{1}{N_{pax}} * \frac{100}{R * 1.852} \tag{8.31}$$

symbol	unit	value	contents
$DOC_{fuel\_newUnit}$	[€/(passenger*100km)]	0.51	DOC_{fuel}
$DOC_{maintenance\_newUnit}$	[€/(passenger*100km)]	0.25	$DOC_{Maint,AF} + DOC_{Maint,Engine}$
$DOC_{fees\_newUnit}$	[€/(passenger*100km)]	0.35	$DOC_{Landing} + DOC_{Navigation} + DOC_{Ground}$
$DOC_{Personal\_newUnit}$	[€/(passenger*100km)]	0.22	DOC_{Crew}
$DOC_{Capital\_newUnit}$	[€/(passenger*100km)]	0.58	$DOC_{Capital}$
$DOC_{total\_newUnit}$	[€/(passenger*100km)]	2.00	DOC_{total}

8.2 Turnaround

As the reference aircraft is similar to a A350-Type the Airport and Maintenance Planning document of the A350 was used for estimating turnaround times of the newly design aircraft. Within that document typical times for ramp activities were used for calculating a standard turn round. Moreover, for this turnaround a full servicing is considered.
Furthermore, for the design mission was assumed, that the aircraft is loaded with the SPP, including 420 Pax and their luggage.

8.2.0.1 Passenger Handling

Passenger handling usually starts with equipment positioning and opening of doors, which takes approximately 3 min. Same time is allocated for door closing and equipment removal after boarding. Deplaning and boarding are taking place through doors 1L and 2L, which are each Type A doors allowing 25 Pax/min and 15 Pax/min respectively.[20] Due to the class configuration it is assumed that the front cabin will use door 1L while the rest may use door 2L. This corresponds to 96 Pax for 1L and 330 Pax for 2L for deplaning and boarding. Additional 4 minutes are added after the normal boarding time for the Last Pax Seating (LPS) allowance and head counting.

$$T_{PaxHandling1L} = 3min + \frac{\partial Pax}{\partial T_{deplaning}} \cdot n_{Pax_D 1L} + \frac{\partial Pax}{\partial T_{boarding}} \cdot n_{Pax_D 1L} + 4min + 3min \tag{8.32}$$

This results in a deplaning time for door 1L of 3 min 45 seconds so roughly 4 minutes. While deplaning on door 2L might take 13 minutes when considering all economy passenger only use this door.

As the 96 business and premium economy would quite quickly disembark either further service could start or, more likely, the remaining economy passengers would also use door 1L. Therefore, after 5 minutes 125 economy passengers have disembarked through door 2L and the remaining 205 passenger start using both doors. Hence, reducing the remaining, estimated time for disembarking to only 4.1 minutes so roughly 5 minutes.
Boarding with the rate of 15 Pax/min at each door results in 6 minutes 30 seconds for door 1L and 22 minutes for door 2L. Making the economy passenger boarding time the most critical time as it consumes almost most of the 1h.

8.2.0.2 Cargo

As for the design mission only luggage of the passengers needs to be transported and stowed. As calculated in Section 4.5.3, 12 LD3 Container are required for storing all passengers baggage.

Opening Doors and equipment positioning	=	2.5 min
Equipment removal and door closing	=	2.5 min
Container Unloading	=	1.2 min/Container
Total container unloading	=	14:30 min
Container loading	=	1.4 min/Container
Total container loading	=	17 min
Bulk Compartment unloading	=	110 kg/min
Bulk Compartment loading	=	95 kg/min

It was considered that all loading of luggage is done using LD3 Container. Hence, no bulk compartment loading was considered.

8.2.0.3 Refueling

As calculated earlier in **??** the fuel mass for a 4000 nm mission is 42492 kg. With a rate of 1400 litre/minute and two nozzles refueling takes about 19 minutes.

Required Fuel for Mission	=	42492kg
Fuel density	=	$845\frac{kg}{m^3} = 0.845\frac{kg}{l}$
Hydrant Positioning and connection	=	8 min
Disconnection and hydrant removal	=	8 min
Connected Nozzle	=	2
litre per minute fuel per Nozzle	=	1400 lpm
Required Mission Fuel	=	53115 l
Time for refueling	=	19 min

8.2.0.4 Cleaning

As during refueling no passengers might be allowed on board, it was assumed that all cleaning activities need to be performed during that time. This resulting available time needs to be sufficient for cleaning crews.

8.2.0.5 Catering

For catering it is assumed that three catering trucks will unload and load at the same time on doors 1R, 2R, and 4R.
As stated in the Airbus A350 the Airport and Maintenance Planning document 40 trolleys were considered for 315 passenger in business and economy class. Extrapolating from these fractions a similar amount was considered for the new designed aircraft and adding one for Full Size Trolley Equivalent (FSTE) for the premium economy. This results in 55 FSTE for the 420 passenger mission. Distributing these over the doors 1R, 2R, and 4R as well as considering a simultaneous loading brings us to further calculation of the required time.
Considering 12 FSTE for 60 Business Class passengers at door 1R, 20 FSTE for premium economy and first economy class section at door 2R, and the remaining 23 container for door 4L. For exchanging a FSTE it's assumed that about 1.5 min per FSTE is required, With which services times for each door were calculated.

Equipment positioning	=	5 min
Door closing and equipment removal	=	3 min

Door 1R	12 FSTE	18 min
Door 2R	20 FSTE	30 min
Door 4R	23 FSTE	34.5 min

With about 35 minutes the loading at door 4R is the most critical path. It needs to be ensured that it starts as early as possible in order to guarantee a turnaround time of 1 hour.

8.2.0.6 Turnaround Time Chart

With all calculated times from above a Gantt Chart for the whole ramp process was created. Therefore, to give a better overview of the whole process and critical time paths.

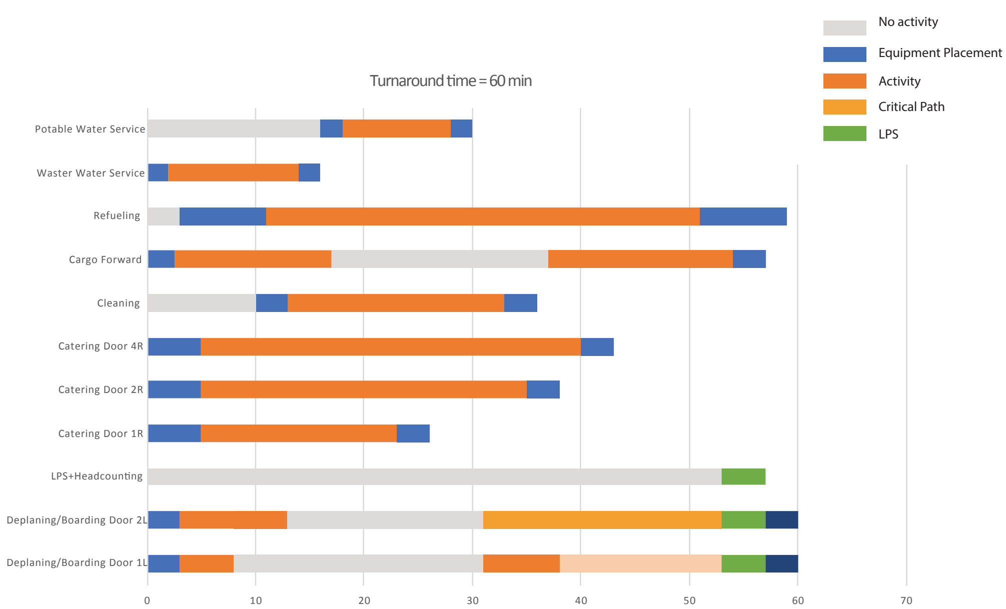


Figure 8.1: Turnaround time overview.

8.2.1 Aircraft Service Arrangements

For this section a turnaround only at the gate with two passenger bridges was considered. A ground handling on an open apron position might be possible as well. Anyhow, gate positions are a lot more space critical which is why this scenario was considered here.

8.2.1.1 Gate Layout

AC	-	Air Conditioning Unit
AS	-	Air Starter Unit
BULK	-	Bulk Train
CAT	-	Catering Truck
CB	-	Conveyor Belt
CLEAN	-	Cleaning Truck
FUEL	-	Fuel Hydrant dispenser or tanker
GPU	-	Ground Power Unit
LD CL	-	Lower Deck Cargo loader
LV	-	Lavatory Vehicle
PBB	-	Passenger Boarding Bridge
TOW	-	Towing Tractor
ULD	-	ULD Train
WV	-	Potable Water Vehicle

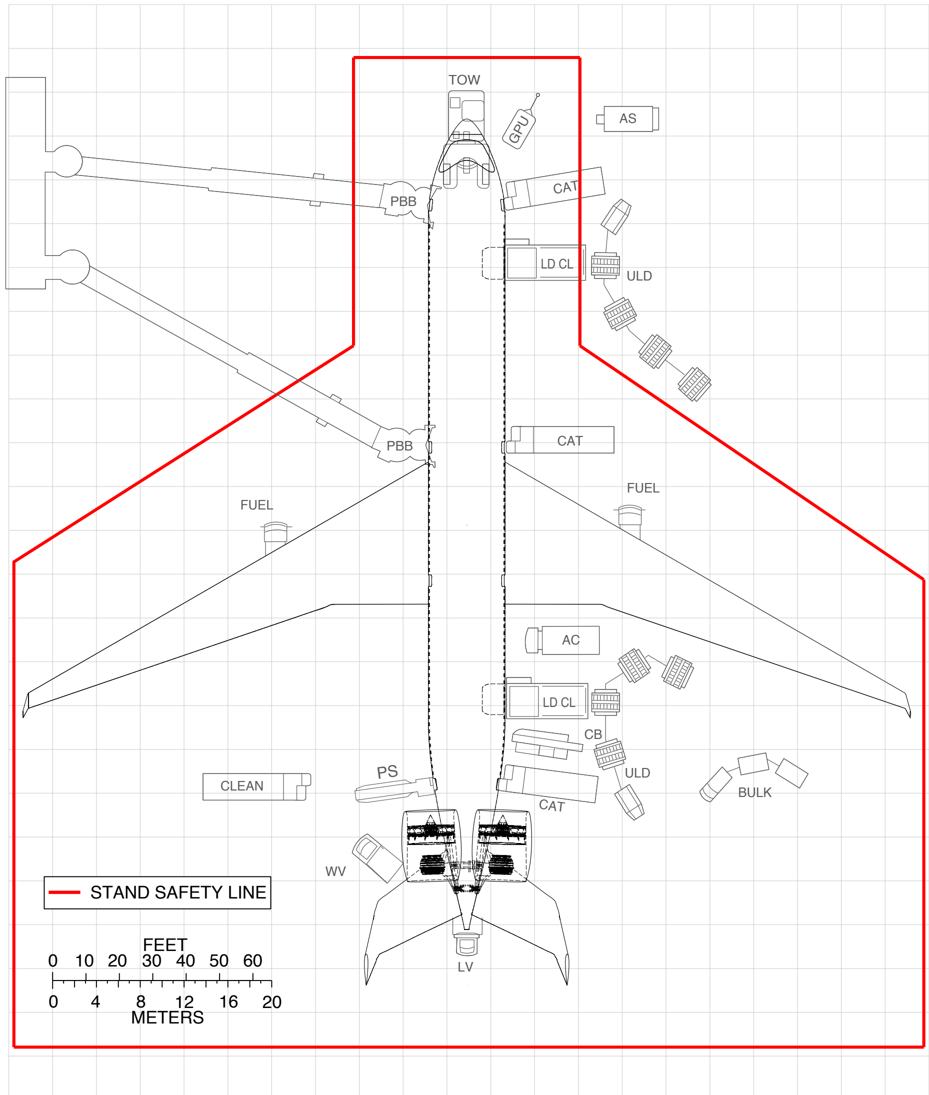


Figure 8.2: Turnaround time overview.

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Appendix A 10

10.1 Weighting of Criteria

10.1.1 Wing

		Wing																	
		Shoulder Wing		Low-Wing		Mid-Wing		C-Wing		Box-Wing		Canard		Blended Wing Body		Strut Braced		Foldable Wingtip	
Flight Stability	0.038062284	7	0.266435986	5	0.190311419	4	0.152249135	4	0.152249135	6	0.228373702	4	0.152249135	3	0.114186851	5	0.190311419	5	0.190311419
Engine Ground Clearance	0.048442907	9	0.435986159	2	0.096885813	4	0.193771626	2	0.096885813	7	0.339100346	4	0.193771626	7	0.339100346	6	0.290657439	3	0.14532872
high c_L	0.079584775	5	0.397923875	5	0.397923875	5	0.397923875	6	0.477508651	7	0.557093426	6	0.477508651	7	0.557093426	5	0.397923875	6	0.477508651
Ground effect	0		0		0		0		0		0		0		0		0		0
low Drag	0.089965398	4	0.359861592	4	0.359861592	3	0.269896194	7	0.629757785	8	0.719723183	6	0.539792388	8	0.719723183	3	0.269896194	6	0.539792388
Maintainability	0.041522491	3	0.124567474	7	0.290657439	4	0.166089965	3	0.124567474	2	0.083044983	5	0.207612457	2	0.083044983	3	0.124567474	3	0.124567474
Gear Integration	0.017301038	2	0.034602076	8	0.138408304	6	0.103806228	7	0.121107266	3	0.051903114	6	0.103806228	8	0.138408304	2	0.034602076	5	0.08650519
Fuel Integration	0.051903114	4	0.207612457	6	0.311418685	5	0.259515571	5	0.259515571	3	0.155709343	5	0.259515571	8	0.415224913	4	0.207612457	4	0.207612457
Structural Effort	0.083044983	6	0.498269896	4	0.332179931	3	0.249134948	2	0.166089965	4	0.332179931	6	0.498269896	6	0.498269896	5	0.415224913	3	0.249134948
Effects on Cabin	0		0		0		0		0		0		0		0		0		0
Airport Sizing	0.058823529	5	0.294117647	5	0.294117647	5	0.294117647	5	0.294117647	4	0.235294118	5	0.294117647	3	0.176470588	4	0.235294118	7	0.411764706
integratability High-Lifting Devices	0.065743945	8	0.525951557	5	0.328719723	5	0.328719723	5	0.328719723	3	0.197231834	5	0.328719723	7	0.460207612	4	0.262975779	5	0.328719723
Manufacturing effort	0.010380623	4	0.041522491	7	0.07266436	4	0.041522491	4	0.041522491	2	0.020761246	4	0.041522491	2	0.020761246	4	0.041522491	3	0.031141869
Ground Handling	0.058823529	7	0.411764706	4	0.235294118	4	0.235294118	4	0.235294118	3	0.176470588	3	0.176470588	2	0.117647059	3	0.176470588	6	0.352941176
possible Synergies	0.065743945	5	0.328719723	5	0.328719723	4	0.262975779	7	0.460207612	7	0.460207612	6	0.394463668	6	0.394463668	4	0.262975779	3	0.197231834
Developing Costs	0.065743945	4	0.262975779	9	0.591695502	2	0.131487889	4	0.262975779	1	0.065743945	2	0.131487889	0	0	3	0.197231834	3	0.197231834
Manufacturing Costs	0.069204152	3	0.207612457	8	0.553633218	3	0.207612457	3	0.207612457	2	0.138408304	4	0.276816609	1	0.069204152	3	0.207612457	4	0.276816609
Direct Operating Costs	0.076124567	6	0.456747405	5	0.380622837	4	0.30449827	5	0.380622837	3	0.228373702	3	0.228373702	1	0.076124567	4	0.30449827	5	0.380622837
Stall behaviour (soft stall)	0.079584775	5	0.397923875	5	0.397923875	5	0.397923875	5	0.397923875	5	0.397923875	7	0.557093426	4	0.3183391	5	0.397923875	5	0.397923875
Weighted Ranking			5.252595156		5.301038062		3.996539792		4.636678201		4.387543253		4.861591696		4.498269896		4.017301038		4.595155709
Place			2		1		9		4		7		3		6		8		5
"Green Factors"																			
Weighted Ranking			1.256055363		1.089965398		0.916955017		1.273356401		1.60899654		1.515570934		1.775086505		1.083044983		1.266435986
Place			6		7		9		4		2		3		1		8		5

10.1.2 Empennage

		Empennage														
				T-Tail			V-Tail			U-Tail			Standard			Canard
Low inteference Drag	0.117647059	4%	6	0.705882353	3%	8	0.941176471	3-4%	7	0.823529412	5%	5	0.588235294	5%	5	0.588235294
Behavior on shorter AC	0.052941176		5	0.264705882		5	0.264705882		5	0.264705882		5	0.264705882		5	0.264705882
low structural weight	0.105882353		3	0.317647059		6	0.635294118		6	0.635294118		4	0.423529412		4	0.423529412
minimum Stall risk	0.005882353			0			0			0			0			0
operatability at stall	0.111764706		5	0.558823529		6	0.670588235		5	0.558823529		5	0.558823529		7	0.782352941
Stabilization Properties	0.064705882		5	0.323529412		4	0.258823529		5	0.323529412		5	0.323529412		3	0.194117647
Complextity of Flight Mechanics	0.029411765		5	0.147058824		3	0.088235294		4	0.117647059		5	0.147058824		3	0.088235294
Trimmable HTP	0.088235294		4	0.352941176		2	0.176470588		6	0.529411765		8	0.705882353		8	0.705882353
Redundancy	0.058823529		5	0.294117647		6	0.352941176		7	0.411764706		5	0.294117647		5	0.294117647
Manufacturing Difficulty	0.029411765		4	0.117647059		6	0.176470588		4	0.117647059		7	0.205882353		6	0.176470588
possible Synergies	0.094117647		6	0.564705882		5	0.470588235		6	0.564705882		5	0.470588235		6	0.564705882
Developing Costs	0.076470588		5	0.382352941		3	0.229411765		4	0.305882353		8	0.611764706		5	0.382352941
Manufacturing Costs	0.052941176		5	0.264705882		6	0.317647059		4	0.211764706		7	0.370588235		6	0.317647059
Direct Operating Costs	0.111764706		4	0.447058824		5	0.558823529		5	0.558823529		5	0.558823529		5	0.558823529
				4.741176471			5.141176471			5.423529412			5.523529412			5.341176471
				5			4			2			1			3
"Green Factors"																
				1.023529412			1.576470588			1.458823529			1.011764706			1.011764706
				3			1			2			4			4

10.1.3 Fuselage

		Fuselage																	
		Twin Fuselage		Droplet		Double Bubble		Double Floor		Conventional		Blended Wing Body		Extra Wide Body		Geodesic Structure		3D Bionic Structure	
Familiy Concept possible	0.036458333	7	0.255208333	3	0.109375	7	0.255208333	7	0.255208333	7	0.255208333	1	0.036458333	7	0.255208333	7	0.255208333	4	0.145833333
Innovative Concept	0.036458333	7	0.255208333	8	0.291666667	7	0.255208333	6	0.21875	3	0.109375	9	0.328125	4	0.145833333	8	0.291666667	9	0.328125
Evacuation	0.057291667	5	0.286458333	7	0.401041667	6	0.34375	6	0.34375	7	0.401041667	2	0.114583333	7	0.401041667	5	0.286458333	5	0.286458333
Safety	0.060763889		0		0		0		0		0		0		0		0		0
Additional Lift by shape	0.041666667	5	0.208333333	7	0.291666667	7	0.291666667	5	0.208333333	5	0.208333333	10	0.416666667	5	0.208333333	5	0.208333333	5	0.208333333
Cabin Space/Comfort	0.032986111	7	0.230902778	5	0.164930556	5	0.164930556	5	0.164930556	5	0.164930556	3	0.098958333	4	0.131944444	5	0.164930556	5	0.164930556
low Drag	0.067708333	4	0.270833333	8	0.541666667	4	0.270833333	5	0.338541667	5	0.338541667	8	0.541666667	4	0.270833333	5	0.338541667	5	0.338541667
minimum Weight	0.060763889	6	0.364583333	4	0.243055556	3	0.182291667	4	0.243055556	5	0.303819444	4	0.243055556	5	0.303819444	9	0.546875	10	0.607638889
little wetted Surface	0.041666667	2	0.083333333	4	0.166666667	4	0.166666667	4	0.166666667	5	0.208333333	4	0.166666667	4	0.166666667	(X) 5	0.208333333	(X) 5	0.208333333
Amount of Cargo	0.008680556	5	0.043402778	4	0.034722222	5	0.043402778	4	0.034722222	5	0.043402778	3	0.026041667	4	0.034722222	(X) 5	0.043402778	(X) 5	0.043402778
Wing integration	0.046875	5	0.234375	5	0.234375	5	0.234375	5	0.234375	5	0.234375	9	0.421875	5	0.234375	(X) 5	0.234375	(X) 5	0.234375
Landing Grear integration	0.045138889	7	0.315972222	4	0.180555556	6	0.270833333	5	0.225694444	5	0.225694444	7	0.315972222	5	0.225694444	(X) 5	0.225694444	(X) 5	0.225694444
Preassure Shot Integration	0.046875	6	0.28125	4	0.1875	4	0.1875	5	0.234375	5	0.234375	2	0.09375	5	0.234375	(X) 5	0.234375	(X) 5	0.234375
Manufacturing effort	0.022569444	3	0.067708333	2	0.045138889	3	0.067708333	4	0.090277778	6	0.135416667	1	0.022569444	4	0.090277778	6	0.135416667	2	0.045138889
low Turn-around time	0.048611111	3	0.145833333	4	0.194444444	5	0.243055556	4	0.194444444	5	0.243055556	2	0.097222222	5	0.243055556	(X) 5	0.243055556	(X) 5	0.243055556
Maintainability	0.039930556	3	0.119791667	2	0.079861111	3	0.119791667	4	0.159722222	7	0.279513889	1	0.039930556	4	0.159722222	2	0.079861111	1	0.039930556
Ground Handling	0.060763889	3	0.182291667	5	0.303819444	5	0.303819444	5	0.303819444	6	0.364583333	2	0.121527778	6	0.364583333	(X) 5	0.303819444	(X) 5	0.303819444
Developing Costs	0.045138889	4	0.180555556	2	0.090277778	3	0.135416667	6	0.270833333	8	0.361111111	0	0	5	0.225694444	4	0.180555556	1	0.045138889
Manufacturing Costs	0.046875	4	0.1875	1	0.046875	3	0.140625	5	0.234375	8	0.375	0	0	6	0.28125	3	0.140625	1	0.046875
Direct Operating Costs	0.067708333	4	0.270833333	4	0.270833333	4	0.270833333	5	0.338541667	6	0.40625	2	0.135416667	5	0.338541667	(X) 5	0.338541667	(X) 5	0.338541667
Alternative Usage	0.005208333	7	0.036458333	3	0.015625	4	0.020833333	4	0.020833333	8	0.041666667	2	0.010416667	6	0.03125	4	0.020833333	6	0.03125
Second Life prep Cargo	0.006944444	6	0.041666667	3	0.020833333	6	0.041666667	4	0.027777778	8	0.055555556	2	0.013888889	7	0.048611111	(X) 5	0.034722222	(X) 5	0.034722222
Space for Crew Rest	0.026041667	4	0.104166667	7	0.182291667	7	0.182291667	7	0.182291667	7	0.182291667	6	0.15625	7	0.182291667	(X) 5	0.130208333	(X) 5	0.130208333
possible Synergies	0.046875	6	0.28125	5	0.234375	5	0.234375	6	0.28125	6	0.28125	5	0.234375	6	0.28125	4	0.1875	8	0.375
			4.447916667		4.331597222		4.427083333		4.772569444		5.453125		3.635416667		4.859375		4.833333333		4.659722222
			4		6		5		3		1		7		2				
"Green Factors"																			
			0.927083333		1.243055556		0.911458333		0.956597222		1.059027778		1.368055556		0.949652778		1.302083333		1.362847222
			6		2		7		4		3		1		5				

10.1.4 Engine Integration

		Engine Integration							
		Under Wing		In Wing		Upper Wing		Aft Fuselage	
low noise	0.076530612	3	0.229591837	6	0.459183673	7	0.535714286	4	0.306122449
low drag	0.102040816	4	0.408163265	6	0.612244898	4	0.408163265	4	0.408163265
uncomplicated air intake	0.086734694	6	0.520408163	5	0.433673469	7	0.607142857	5	0.433673469
available space	0.025510204	3	0.076530612	3	0.076530612	7	0.178571429	6	0.153061224
safety	0.096938776	6	0.581632653	4	0.387755102	5	0.484693878	7	0.678571429
non/less extra moment with OEI	0.096938776	4	0.387755102	4	0.387755102	4	0.387755102	7	0.678571429
maintainability	0.06122449	7	0.428571429	2	0.12244898	4	0.244897959	4	0.244897959
low intereference drag	0.091836735	5	0.459183673	7	0.642857143	5	0.459183673	3	0.275510204
visibility	0.025510204	8	0.204081633	6	0.153061224	5	0.12755102	5	0.12755102
ground/service interference	0.030612245	2	0.06122449	5	0.153061224	7	0.214285714	7	0.214285714
Synergien	0.06122449	4	0.244897959	4	0.244897959	6	0.367346939	6	0.367346939
Developing Costs	0.071428571	6	0.428571429	2	0.142857143	4	0.285714286	3	0.214285714
Manufacturing Costs	0.071428571	8	0.571428571	2	0.142857143	4	0.285714286	4	0.285714286
Direct Operating Costs	0.102040816	6	0.612244898	4	0.408163265	5	0.510204082	5	0.510204082
			5.214285714		4.367346939		5.096938776		4.897959184
			1		4		2		3
"Green Factors"									
			0.867346939		1.255102041		0.867346939		0.683673469
			2		1		2		4

10.2 Criteria Weighting of Components

10.2. CRITERIA WEIGHTING OF COMPONENTS

10.2.1 Wing

Wing

	Flight Stability	Engine Ground Clearance	high c_L	Ground effect	low Drag	Maintainability	Gear Integration	Fuel Integration	Structural Effort	Effects on Cabin	Airport Sizing	integratability High-Lifting Devices	Manufacturing effort	Ground Handling	possible Synergies	Developing Costs	Manufacturing Costs	Direct Operating Costs	Stall behaviour (soft stall)	Overall	Quantifyable Parameter
Flight Stability	1	2	1		2	1	0	1	2		2	2	0	1	2	2	2	1	1		
Engine Ground Clearance	0	1	1		2	1	1	1	2		1	1	0	1	1	2	2	1	2		
high c_L	1	1	1		1	0	0	0	1		1	1	0	1	0	0	1	1	1		cL
Ground effect																					
low Drag	0	0	1		1	0	0	0	1		1	0	0	0	1	0	1	1	1		cD
Maintainability	1	1	2		2	1	0	1	2		1	1	1	2	1	2	1	1	2		
Gear Integration	2	1	2		2	2	1	2	2		2	2	0	2	1	2	2	2	2		
Fuel Integration	1	1	2		2	1	0	1	1		1	1	0	1	1	2	1	2	1		V_Fuel_max
Structural Effort	0	0	1		1	0	0	1	1		0	0	1	0	1	1	1	1	1		Root Bending Moment
Effects on Cabin																					
Airport Sizing	0	1	1		1	1	0	1	2		1	2	0	1	2	1	1	1	1		MTOW, Size
integratability High-Lifting Devices	0	1	1		2	1	0	1	2		0	1	0	1	1	1	1	1	1		
Manufacturing effort	2	2	2		2	1	2	2	1		2	2	1	2	2	2	2	2	2		
Ground Handling	1	1	1		2	0	0	1	2		1	1	0	1	1	1	1	1	2		
possible Synergies	0	1	2		1	1	1	1	1		0	1	0	1	1	0	1	1	2		
Developing Costs	0	0	2		2	0	0	0	1		1	1	0	1	2	1	1	2	1		
Manufacturing Costs	0	0	1		1	1	0	1	1		1	1	0	1	1	1	1	2	1		
Direct Operating Costs	1	1	1		1	1	0	0	1		1	1	0	1	1	0	0	1	1		
Stall behaviour (soft stall)	1	0	1		1	0	0	1	1		1	1	0	0	0	1	1	1	1		
Sum	11	14	23	0	26	12	5	15	24	0	17	19	3	17	19	19	20	22	23	**289**	
Rank	15	13	3	18	1	14	16	12	2	18	10	7	17	10	7	7	6	5	3		
	0.038062284	0.048442907	0.079584775	0	0.089965398	0.041522491	0.017301038	0.051903114	0.083044983	0	0.058823529	0.065743945	0.010380623	0.058823529	0.065743945	0.065743945	0.069204152	0.076124567	0.079584775		

10.2.2 Empennage

	Empennage															
	Low inteference Drag	Behavior on shorter AC	low structural weight	minimum Stall risk	operatability at stall	Stabilization Properties	Complextity of Flight Mechanics	Trimmable HTP	Redundancy	Manufacturing Difficulty	possible Synergies	Developing Costs	Manufacturing Costs	Direct Operating Costs	Overall	Quantifyable Parameter
Low inteference Drag	1	0	1		1	0	0	1	0	0	1	0	0	1		c_Di
Behavior on shorter AC	2	1	2		2	1	1	1	2	0	1	1	1	2		
low structural weight	1	0	1		1	1	0	1	1	0	1	0	0	1		m_empennage
minimum Stall risk				*1*												
operatability at stall	1	0	1		1	0	0	0	0	0	1	1	1	1		
Stabilization Properties	2	1	1		2	1	1	1	1	0	1	1	1	2		
Complextity of Flight Mechanics	2	1	2		2	1	1	2	1	1	2	2	2	2		
Trimmable HTP	1	1	1		2	1	0	1	0	0	1	1	1	1		
Redundancy	2	0	1		2	1	1	2	1	2	1	1	1	1		
Manufacturing Difficulty	2	2	2		2	2	1	2	0	1	2	2	1	2		
possible Synergies	1	1	1		1	1	0	1	1	0	1	0	0	2		
Developing Costs	2	1	2		1	1	0	1	1	0	2	1	0	1		
Manufacturing Costs	2	1	2		1	1	0	1	1	1	2	2	1	2		
Direct Operating Costs	1	0	1		1	0	0	1	1	0	0	1	0	1		
Sum	20	9	18	*1*	19	11	5	15	10	5	16	13	9	19	**170**	
Rank	1	10	4	14	2	8	12	6	9	12	5	7	10	2		
	0.117647059	0.052941176	0.105882353	0.005882353	0.111764706	0.064705882	0.029411765	0.088235294	0.058823529	0.029411765	0.094117647	0.076470588	0.052941176	0.111764706		

10.2. CRITERIA WEIGHTING OF COMPONENTS

10.2.3 Fuselage

Fuselage	Familiy Concept possible	Innovative Concept	Evacuation	Safety	Additional Lift by shape	Cabin Space/Compfort	low Drag	minimum Weight	little wetted Surface	Amount of Cargo	Wing integration	Landing Grear integration	Pressure Shot Integration	Manufacturing effort	low Turn-around time	Maintainability	Alternative Usage	Second Life prep Cargo	Space for Crew Rest	possible Synergies	Ground Handling	Developing Costs	Manufacturing Costs	Direct Operating Costs	Overall
Familiy Concept possible	1	1	2	2	1	1	2	2	1	1	2	1	1	1	2	1	0	0	1	1	1	0	1	1	
Innovative Concept	1	1	2	2	1	1	2	2	1	0	1	1	1	1	1	1	0	0	1	1	2	1	1	2	
Evacuation	0	0	1	1	0	1	1	1	1	0	1	1	1	0	1	0	0	0	0	1	1	1	1	1	
Safety	0	0	1	1	0	0	1	1	1	0	1	1	1	0	1	0	0	0	0	0	1	1	1	1	
Additional Lift by shape	1	1	2	2	1	1	1	1	0	0	1	1	1	1	1	1	0	0	1	1	2	1	1	2	
Cabin Space/Compfort	1	1	1	2	1	1	2	2	2	0	1	1	1	1	1	1	0	0	1	1	2	2	2	2	
low Drag	0	0	1	1	1	0	1	1	0	0	0	0	1	0	0	0	0	0	1	0	1	0	0	1	
minimum Weight	0	0	1	1	1	0	1	1	0	0	1	1	1	0	0	0	0	0	0	1	1	1	1	1	
little wetted Surface	1	1	1	1	2	0	2	2	1	0	1	1	1	0	1	0	0	0	0	1	2	2	2	2	
Amount of Cargo	1	2	2	2	2	2	2	2	2	1	2	2	2	2	1	2	1	1	2	2	2	2	2	2	
Wing integration	0	1	1	1	1	1	2	1	1	0	1	1	1	0	1	1	0	0	1	1	1	1	1	2	
Landing Grear integration	1	1	1	1	1	1	2	1	1	0	1	1	1	0	1	1	0	0	1	1	1	1	1	2	
Preassure Shot Integration	1	1	1	1	1	1	1	1	1	0	1	1	1	0	1	1	0	0	0	1	2	1	1	2	
Manufacturing effort	1	1	2	2	1	1	2	2	2	0	2	2	2	1	2	1	0	1	1	2	2	2	1	2	
low Turn-around time	0	1	1	1	1	1	2	2	1	1	1	1	1	0	1	0	0	0	0	1	1	1	1	1	
Maintainability	1	1	2	2	1	1	2	2	2	0	1	1	1	1	2	1	0	0	0	1	1	0	1	1	
Alternative Usage	2	2	2	2	2	2	2	2	2	1	2	2	2	2	2	2	1	1	2	2	2	2	2	2	
Second Life prep Cargo	2	2	2	2	2	2	2	2	2	1	2	2	2	1	2	2	1	1	2	2	2	2	2	2	
Space for Crew Rest	1	1	2	2	1	1	1	2	2	0	1	1	2	1	2	2	0	0	1	2	2	2	2	2	
possible Synergies	1	1	1	2	1	1	2	1	1	0	1	1	1	0	1	1	0	0	0	1	1	0	1	2	
Ground Handling	1	0	1	1	0	0	1	1	0	0	1	1	0	0	1	1	0	0	0	1	1	0	1	1	
Developing Costs	2	1	1	1	1	0	2	1	0	0	1	1	1	0	1	2	0	0	0	2	2	1	0	2	
Manufacturing Costs	1	1	1	1	1	0	2	1	0	0	1	1	1	1	1	1	0	0	0	1	1	2	1	2	
Direct Operating Costs	1	0	1	1	0	0	1	1	0	0	0	0	0	0	1	1	0	0	0	0	1	0	0	1	
Sum	21	21	33	35	24	19	39	35	24	5	27	26	27	13	28	23	3	4	15	27	35	26	27	39	**576**
Rank	17	17	6	3	14	19	1	3	14	22	8	12	8	21	7	16	24	23	20	8	3	12	8	1	
	0.036458333	0.036458333	0.057291667	0.060763889	0.041666667	0.032986111	0.067708333	0.060763889	0.041666667	0.008680556	0.046875	0.045138889	0.046875	0.022569444	0.048611111	0.039930556	0.005208333	0.006944444	0.026041667	0.046875	0.060763889	0.045138889	0.046875	0.067708333	

10. APPENDIX A

10.2.4 Engine Integration

Gear

	low drag	minimum Weight	Clearance provided	configuration	easy mechanism	safety	no extra space in fuselage	minimum noise	Ground Handling	Synergies	Developing Costs	Manufacturing Costs	Direct Operating Costs	Overall
low Drag	1	2	2	0	0		0	1	1	2	1	1	2	
minimum Weight	0	1	1	0	0		0	0	1	0	0	0	1	
Clearance provided	0	1	1	0	1		0	1	1	0	1	1	1	
configuration	2	2	2	1	0		1	2	1	1	2	2	2	
easy mechanism	2	2	1	2	1		0	2	1	2	1	1	2	
safety						*1*								
no extra space in fuselage	2	2	2	1	2		1	2	2	2	2	2	2	
minimum noise	1	2	1	0	0		0	1	1	0	0	1	1	
Ground Handling	1	1	1	1	1		0	1	1	0	1	1	1	
Synergien	0	2	2	1	0		0	2	2	1	0	1	2	
Developing Costs	1	2	1	0	1		0	2	1	2	1	0	2	
Manufacturing Costs	1	2	1	0	1		0	1	1	1	2	1	2	
Direct Operating Costs	0	1	1	0	0		0	1	1	0	0	0	1	
Sum	11	20	16	6	7	*1*	2	16	14	11	11	11	19	**145**
Rank	6	1	3	11	10	13	12	3	5	6	6	6	2	
	0.075862069	0.137931034	0.110344828	0.04137931	0.048275862	0.006896552	0.013793103	0.110344828	0.096551724	0.075862069	0.075862069	0.075862069	0.131034483	

10.3 Cabin Layouts

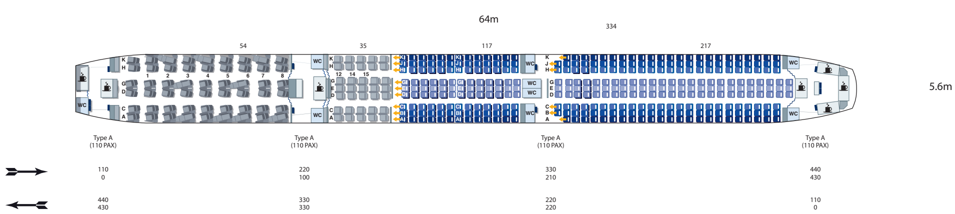


Figure 10.1: Cabin Layout version S (Small width)

10.3. CABIN LAYOUTS

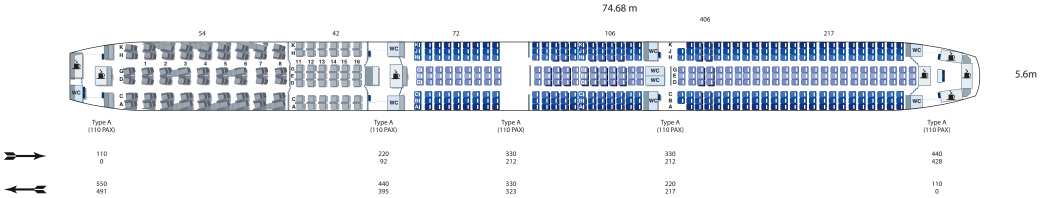


Figure 10.2: Cabin Layout version S Fam (Small width, stretched family concept)

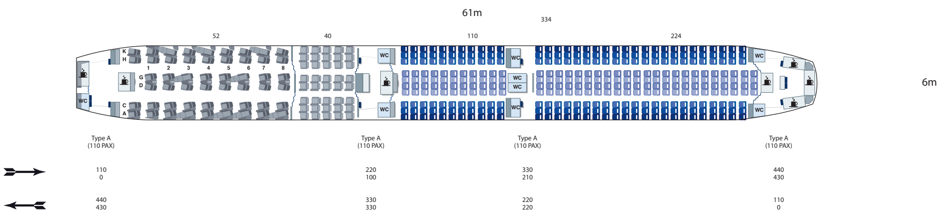


Figure 10.3: Cabin Layout version MS (Medium small width)

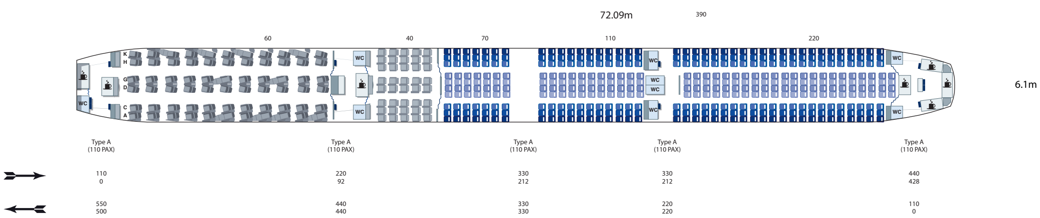


Figure 10.4: Cabin Layout version MS Fam (Medium small width, stretched family concept)

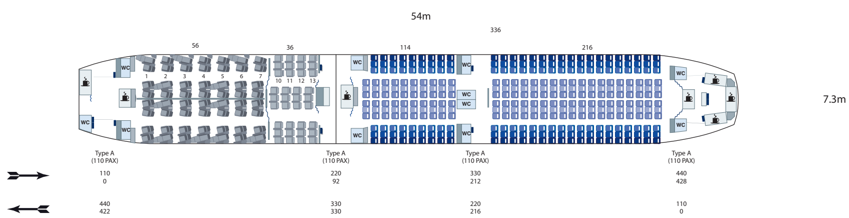


Figure 10.5: Cabin Layout version W (Wide width)

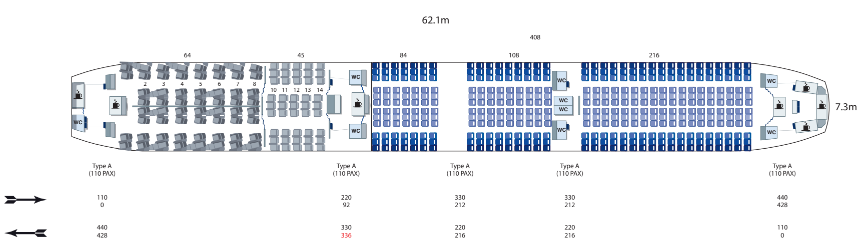


Figure 10.6: Cabin Layout version W Fam (Wide width, stretched family concept)

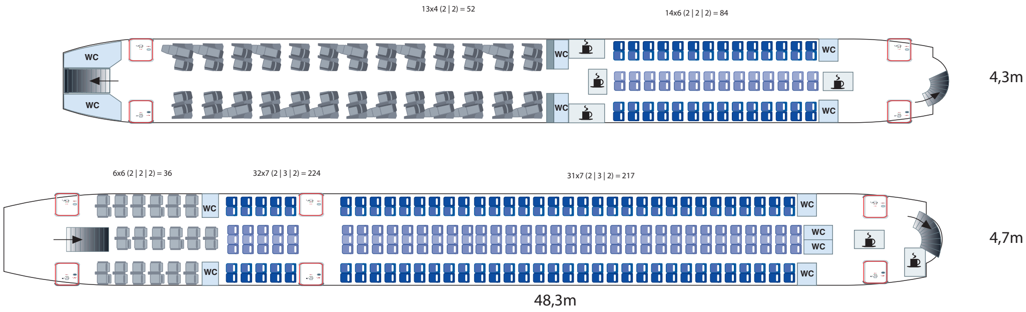


Figure 10.7: Cabin Layout version DF S (Double Floor small width)

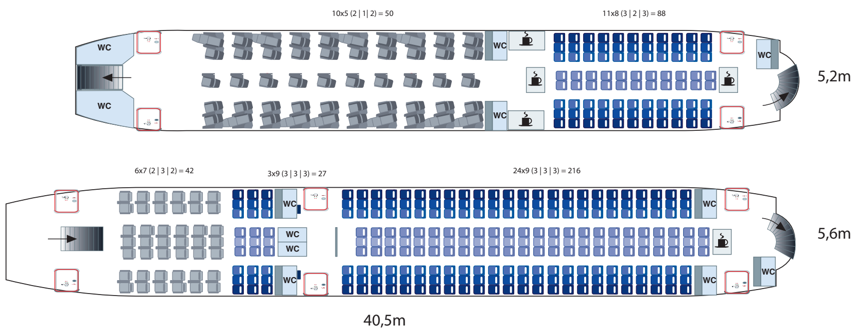


Figure 10.8: Cabin Layout version DF M (Double Floor medium width)

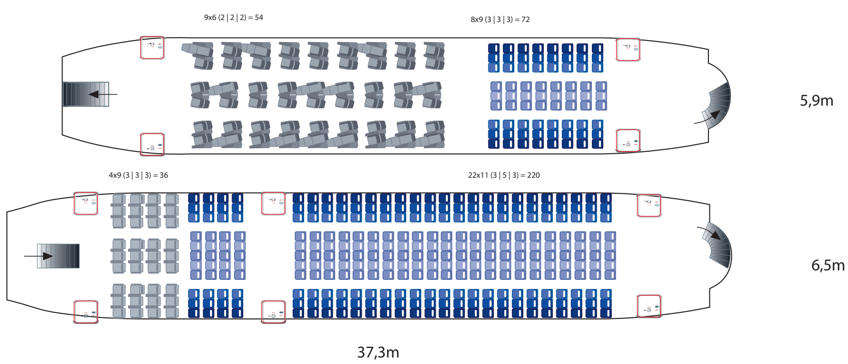


Figure 10.9: Cabin Layout version DF W (Double Floor wide width)

Appendix B 11

Requirements:

Passenger:

$n_{PAX} := 420$ $\quad$ $m_{PAX} := 110\ kg$

$n_{PAX\_Y} := 334$

$n_{PAX\_W} := 36$

$n_{Pax\_J} := 50$

Payload mass:

$m_{Payload\_SPP} := n_{PAX} \cdot m_{PAX} = 46200\ kg$ $\quad$ $m_{Max\_Payload} := 55000\ kg$

Design Range:

$R_{Design} := 8000\ nmi$

Crew estimation:

$n_{FlightCrew} := 3$ $\quad$ $n_{FlightCrew\_RefAC} := 2$

$n_{CabinCrew} := 18$

$m_{Crew} := \left(n_{FlightCrew} + n_{CabinCrew}\right) \cdot m_{PAX} = 2310\ kg$

$m_{Crew\_RefAC} := \left(n_{CabinCrew} + n_{FlightCrew\_RefAC}\right) \cdot m_{PAX} = 2200\ kg$

Reference Aircraft Parameters:

$m_{MTOW\_RefAC} := 377000\ kg$

$m_{MZFW\_RefAC} := 242100\ kg$

$m_{OWE\_RefAC} := 188100\ kg$

$m_{BlockFuel\_RefAC\_DRange} := 131700\ kg$

$$m_{0\_RefAC} := m_{OWE\_RefAC} + m_{BlockFuel\_RefAC\_DRange} + m_{Payload\_SPP} + m_{Crew\_RefAC} = 368200\ kg$$

Fuel Fractions:

Raymer, Part I, p. 12

Engine Start, Warm-up:

$m_1m_0 := 0.99$

Taxi:

$m_2m_1 := 0.99$

Take-off:

$m_3m_2 := 0.995$

Climb:

$m_4m_3 := 0.98$

Descent:

$m_6m_5 := 0.99$

Missed Approach and Climb:

$m_7m_8 := 0.988$

Landing, Taxi, Shut-Down:

$m_{10}m_9 := 0.992$

Cruise Fuel with Breguet:

Mission Parameter from Reference Aircraft:

Cruise Speed:

$Ma_{RefAC} := 0.82$

Specific Fuel Burn:

$c_{TL} := 0.53 \dfrac{\frac{lb}{hr}}{lb}$

Initial Cruise Altitude:

$H_{FlightLevel} := 33000\ ft$

Glide Ratio:

$c_Ac_W := 21.93$

ISA Standard atmosphere:

$$T_{FL330} := 15\ °C - 0.0065\ \frac{K}{m} \cdot H_{FlightLevel} = 222.77\ K$$

$$c_{FL330} := \sqrt{1.4 \cdot 287.15\ \frac{J}{K \cdot kg} \cdot T_{FL330}}$$

Cruise Speed Reference Aircraft:

$$v_{Cruise\_RefAC} := Ma_{RefAC} \cdot c_{FL330} = 245.392\ \frac{m}{s}$$

Cruise Fuel Fraction Reference Aircraft:

11. APPENDIX B

$$m_5m_{4\_RefAC} := \exp\left(-\frac{R_{Design} \cdot c_{TL}}{v_{Cruise\_RefAC} \cdot c_A c_W}\right) = 0.667$$

$$m_{FuelFraction} := 1 - m_1m_0 \cdot m_2m_1 \cdot m_3m_2 \cdot m_4m_3 \cdot m_5m_{4\_RefAC} \cdot m_6m_5 \cdot m_7m_8 \cdot m_{10}m_9 = 0.382$$

Alternative: Take Off Fuel:

$$m_{FuelFraction\_RefAC} := 1 - \frac{m_{0\_RefAC} - m_{BlockFuel\_RefAC\_DRange}}{m_{0\_RefAC}} = 0.358$$

$$m_{EmptyFraction\_RefAC} := \frac{m_{OWE\_RefAC}}{m_{0\_RefAC}} = 0.5109$$

according to Raymer, p. 12:

$$m_{MTOW\_RefAC\_Raymer} := \frac{m_{Payload\_SPP} + m_{Crew\_RefAC}}{1 - m_{FuelFraction\_RefAC} - m_{EmptyFraction\_RefAC}} = 368200 \ \boldsymbol{kg}$$

Calibration Factor for real MTOW:

$$cf_{MTOW\_Raymer} := \frac{m_{MTOW\_RefAC}}{m_{MTOW\_RefAC\_Raymer}} = 1.024$$

$$m_{MTOW\_RefAC\_Cal} := \frac{m_{Payload\_SPP} + m_{Crew\_RefAC}}{1 - m_{FuelFraction\_RefAC} - m_{EmptyFraction\_RefAC}} \cdot cf_{MTOW\_Raymer} = 377000 \ \boldsymbol{kg}$$

according to Raymer, p. 13:

$A := 1.02$ $\quad C := -0.06$

Calulating D as Manufacturing and Calibration Factor:

$$D_{RefAC} := \frac{m_{EmptyFraction\_RefAC}}{A \cdot m_{MTOW\_RefAC}{}^{C}} \cdot \boldsymbol{kg}^{-\frac{3}{50}} = 1.082$$

$$m_Lm_0RefAC := A \cdot m_{MTOW\_RefAC}{}^{C} \cdot D_{RefAC} = 0.5109 \ \frac{1}{\boldsymbol{kg}^{\frac{3}{50}}}$$

11. APPENDIX B

Iterative Calculation of MTOW:

$$m_{MTOW\_Raymer} := \frac{m_{Payload\_SPP} + m_{Crew}}{1 - m_{FuelFraction\_RefAC} - m_{EmptyFraction\_RefAC}} \cdot cf_{MTOW\_Raymer} = 377856.818\ kg$$

$$m_L m_0 Loop1 := A \cdot m_{MTOW\_Raymer}{}^{C} \cdot D_{RefAC} \cdot kg^{\frac{3}{50}} = 0.5108\ kg^{\frac{3}{50}} \cdot \frac{1}{kg^{\frac{3}{50}}}$$

$$m_{MTOW\_Raymer}Loop1 := \frac{m_{Payload\_SPP} + m_{Crew}}{1 - m_{FuelFraction\_RefAC} - m_L m_0 Loop1} \cdot cf_{MTOW\_Raymer} = 377656.916\ kg$$

$$\frac{\left| m_{MTOW\_Raymer}Loop1 - m_{MTOW\_Raymer} \right|}{m_{MTOW\_Raymer}} = 0.053\%$$

$$m_L m_0 Loop2 := A \cdot m_{MTOW\_Raymer}{}^{C} \cdot D_{RefAC} \cdot kg^{\frac{3}{50}} = 0.511\ kg^{\frac{3}{50}} \cdot \frac{1}{kg^{\frac{3}{50}}}$$

Thrust to weight ratio:

$$F_{0\_RefAC} := 2 \cdot 115.0 \cdot 1000 \cdot lbf = 1023090.972\ N$$

$$F_{0\_RefAC}m_{MTOW\_Raymer}Loop1 := \frac{F_{0\_RefAC}}{m_{MTOW\_Raymer}Loop1 \cdot g} = 0.276$$

Appendix C 12

Table 12.1: Calibration factors

alpha	c_L	c_D
-0.172	1.491266667	1.410711254
0.371	1.416400	1.403664068
0.914	1.360100	1.403460509
1.457	1.316200	1.389081041
2.0	1.280940	1.362817383
2.543	1.252000	1.318267652
3.086	1.227766667	1.22617196
3.629	1.207184615	1.135734498
4.172	1.189428571	1.051564654

Cruise Speed

$Ma := 0.8 \qquad Höhe := 33000\ ft = \left(1.006 \cdot 10^{4}\right)\ m$

$T_{FL330} := 15\ °C - 0.0065\ \frac{K}{m} \cdot Höhe = 222.77\ K$ ISA Standard atmosphere

$c_{FL330} := \sqrt{1.4 \cdot 287.15\ \frac{J}{K \cdot kg} \cdot T_{FL330}} = 299.259\ \frac{m}{s}$

$v := Ma \cdot c_{FL330} = 239.407\ \frac{m}{s}$

Reynolds Number

$MAC_{RefAC} := 9461\ mm = 9.461\ m$

kinematic viscosity from Raymer at 10 km altitude: $\nu_{cruise} := 3.53 \cdot 10^{-5}\ \frac{m^2}{s}$

$Re := v \cdot \frac{MAC_{RefAC}}{\nu_{cruise}} = 6.417 \cdot 10^{7}$

Takeoff and Landing

$Ma_{TO} := 0.22 \quad Höhe_{TO} := 0\ ft = 0\ m$

$T_{FL0} := 15\ °C - 0.0065\ \frac{K}{m} \cdot Höhe_{TO} = 288.15\ K$ ISA Standard atmosphere

$c_{FL0} := \sqrt{1.4 \cdot 287.15\ \frac{J}{K \cdot kg} \cdot T_{FL0}} = 340.352\ \frac{m}{s}$

$v_{TO} := Ma_{TO} \cdot c_{FL0} = 74.877\ \frac{m}{s}$

kinematic viscosity from Raymer at 0 km altitude: $\nu_{TO} := 1.46 \cdot 10^{-5}\ \frac{m^2}{s}$

$Re_{TO} := v_{TO} \cdot \frac{MAC_{RefAC}}{\nu_{TO}} = 4.852 \cdot 10^{7}$

Polar calculation:

Lift coefficient over angle of attack:

values for k and d are calculated with data points from Aero Tool with lowest and highest angle of attack ($-0.172°$ and $4.172°$)

Cruise:

$$c_{L\_cruise} = k \cdot (\alpha - \alpha_{L0}) = k \cdot \alpha + d$$

$k := 0.102141$ obtained by calibrated data for cruise from Aero Tool

$\alpha_{L0\_cruise} := -5.15$ obtained by calibrated data for cruise from Aero Tool in [deg]

$d := 0.525232$ obtained by data for cruise from Aero Tool

Landing:

$$c_{L\_landing} = k \cdot (\alpha - \alpha_{L0} - \Delta\alpha_{L0\_landing}) = k \cdot \alpha + d$$

$\Delta\alpha_{L0\_Landing} := -10.67$ acc Raymer in [deg]

$\alpha_{L0\_landing} := \Delta\alpha_{L0\_Landing} + \alpha_{L0\_cruise}$

$d := -\alpha_{L0\_landing} \cdot k$ obtained by new $\alpha_{L0\_landing}$ $d = -\alpha_{L0\_landing} \cdot k$

$d = 1.616$

Takeoff:

$$c_{L\_takeoff} = k \cdot (\alpha - \alpha_{L0} - \Delta\alpha_{L0\_takeoff}) = k \cdot \alpha + d$$

$\Delta\alpha_{L0\_takeoff} := -7.114$ acc Raymer in [deg]

$\alpha_{L0\_takeoff} := \Delta\alpha_{L0\_takeoff} + \alpha_{L0\_cruise}$

$d := -\alpha_{L0\_takeoff} \cdot k$ obtained by new $\alpha_{L0\_takeoff}$ $d = -\alpha_{L0\_takeoff} \cdot k$

$d = 1.253$

Lift coefficient over drag coefficient:

Calculation of wing polar with calibrated data from Aero Tool:

$$c_D = c_{D0} + K\left(c_L - c_{L\_min}\right)^2 = a \cdot c_L{}^2 + b \cdot c_L + c$$

$$c_D = c_{D0} + K \cdot c_L{}^2 - 2 \cdot K \cdot c_{L\_min} \cdot c_L + K \cdot c_{L\_min}{}^2$$

$$a = K \qquad b = -2 \cdot K \cdot c_{L\_min} \qquad c = c_{D0} + K \cdot c_{L\_min}{}^2$$

with three data points of the calibrated wing calculation in Aero Tool a, b and c can be obtained

Point 1: $\alpha_1 = -0.172\ deg$

$$0.01161 = a \cdot 0.34453^2 + b \cdot 0.34453 + c$$

Point 2: $\alpha_2 = 2\ deg$

$$0.01695 = a \cdot 0.56713^2 + b \cdot 0.56713 + c$$

Point 3: $\alpha_3 = 4.172\ deg$

$$0.03027 = a \cdot 0.788255^2 + b \cdot 0.788255 + c$$

$$M := \begin{bmatrix} 0.34453^2 & 0.34453 & 1 \\ 0.56713^2 & 0.56713 & 1 \\ 0.788255^2 & 0.788255 & 1 \end{bmatrix} \qquad Y := \begin{bmatrix} 0.01161 \\ 0.01695 \\ 0.03027 \end{bmatrix}$$

$$M^{-1} \cdot Y = \begin{bmatrix} 0.081691 \\ -0.050485 \\ 0.019307 \end{bmatrix}$$

$$a := 0.081691 \qquad b := -0.050485 \qquad c := 0.019307$$

for different flight conditions only the constant c has to be changed by adding remaining c_{D0} of the fuselage, engines etc., or respectivly c_{D0} of landing gear and flaps in landing and takeoff.

cruise:

$c_{D0r\_cruise} := 0.006994$ Drag of remaining aircraft.

$c_{cruise} := c + \frac{c_{D0r\_cruise}}{1.2} = 0.0251$ with average calibration factor 1.2 for c_D

$c_{D\_cruise} = a \cdot c_L^2 + b \cdot c_L + c_{cruise}$

landing:

$c_{D0r\_landing} := 0.006886$ Drag of remaining aircraft.

$\Delta c_{D0\_landing} := 0.10912$ Additional drag from langing gear and flaps

$c_{landing} := c + \frac{\left(c_{D0r\_landing}\right)}{1.2} + \frac{\Delta c_{D0\_landing}}{1.2} = 0.116$

$c_{D\_landing} = a \cdot c_L^2 + b \cdot c_L + c_{landing}$

takeoff:

$c_{D0r\_takeoff} := 0.006886$ Drag of remaining aircraft.

$\Delta c_{D0\_takeoff} := 0.07884$ Additional drag from langing gear and flaps

$c_{takeoff} := c + \frac{c_{D0r\_takeoff}}{1.41} + \frac{\Delta c_{D0\_takeoff}}{1.41} = 0.0801$ with highest calibration factor, to obtain reasonable valus at higher c_L

$c_{D\_takeoff} = a \cdot c_L^2 + b \cdot c_L + c_{takeoff}$

Empennage Dimensions

Calibration with reference aircraft:

$S_{wing\_Ref} := 560\ m^2$ $\quad S_{HTP\_Ref} := 104\ m^2$ $\quad S_{VTP\_Ref} := 60\ m^2$

$\Delta x_{AC\_wing\_HTP\_Ref} := 31.688\ m$ $\quad b_{Ref} := 76.6\ m$

$c_{w\_Ref} := 9.461\ m$

Formulas according Raymer

$$C_{HTP} := \frac{\Delta x_{AC\_wing\_HTP\_Ref} \cdot S_{HTP\_Ref}}{c_{w\_Ref} \cdot S_{wing\_Ref}} = 0.622$$

Raymer predicts C_HTP = 1
-> C_HTP is the calibration factor itself

$$C_{VTP} := \frac{\Delta x_{AC\_wing\_HTP\_Ref} \cdot S_{VTP\_Ref}}{b_{Ref} \cdot S_{wing\_Ref}} = 0.044$$

Raymer predicts C_VTP = 0.09
-> C_VTP is also calibrated with the value of 0.044

Fuselage length: $L_{Fuselage} := 74\ m$

Wing Area: $S_{wing} := 521.7\ m^2$

Mean Aerodynamic Chord: $c_w := 8.537\ m$

Wing Span: $b := 80.0\ m$

$\Delta x_{AC\_wing\_HTP} := L_{Fuselage} \cdot 0.45$ according Raymer P.160

$\Delta x_{AC\_wing\_HTP} := 29.5\ m$ tail-arm of projected aircraft

$\Delta x_{HTP\_VTP} := 5\ m$ distance between NP of HTP and VTP obtained by planform design in WingNet

U-Tail Dimensions

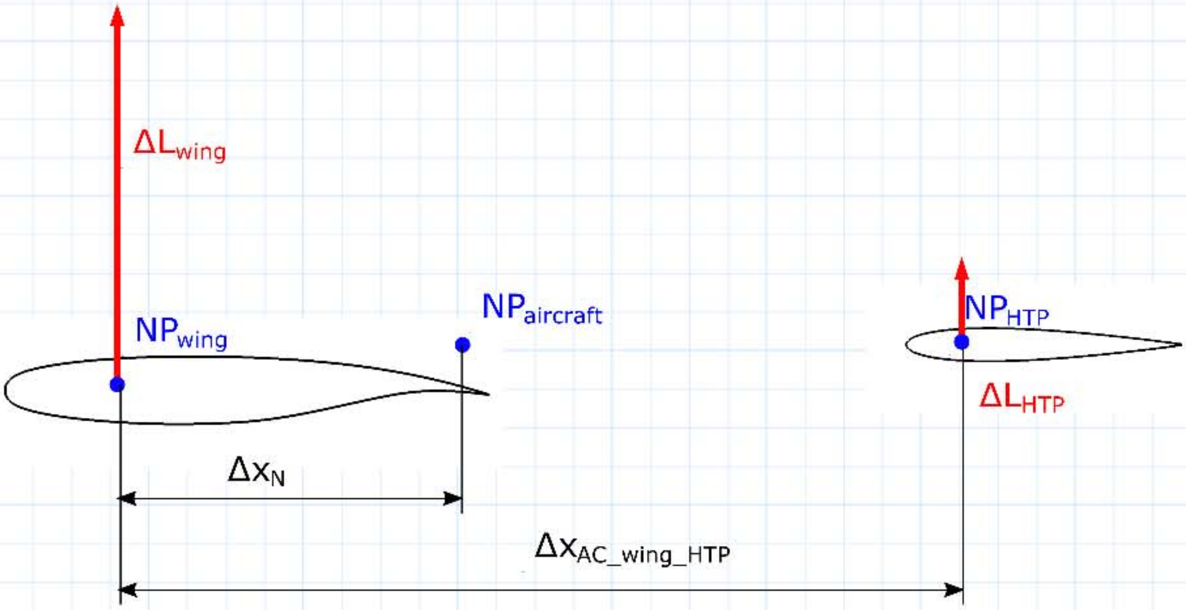
$$S_{HTP\_UTail} := \frac{c_w \cdot S_{wing} \cdot C_{HTP}}{\Delta x_{AC\_wing\_HTP}} = 93.909\ m^2$$

$$S_{VTP\_UTail} := \frac{b \cdot S_{wing} \cdot C_{VTP}}{\left(\Delta x_{AC\_wing\_HTP} + \Delta x_{HTP\_VTP}\right)} = 53.619\ m^2$$

Flight Stability

Aerodynamic Center of entire Aircraft Configuaration:

Equilibrium of Moments about the wing's aerodynamic center:



$$\Delta L_{wing} \cdot \Delta x_N = \Delta L_{HTP} \cdot (\Delta x_{AC\_wing\_HTP} - \Delta x_N)$$

with:

$$\Delta L_{wing} = q \cdot S_{wing} \cdot \Delta c_{A\_wing} \qquad \Delta c_{A\_wing} = c'_{A\_wing} \cdot \Delta\alpha$$

$$\Delta L_{HTP} = q_H \cdot S_{HTP} \cdot \Delta c_{A\_HTP} \qquad \Delta c_{A\_HTP} = c'_{A\_HTP} \cdot \left(\Delta\alpha\left(1 - \frac{d\varepsilon_h}{d\alpha}\right)\right)$$

leads to:

$$\Delta x_N = \frac{q_H}{q} \cdot \frac{S_{HTP}}{S_{wing}} \cdot \frac{c'_{A\_HTP}}{c'_{A\_wing}} \cdot \left(1 - \frac{d\varepsilon_h}{d\alpha}\right) \cdot (\Delta x_{AC\_wing\_HTP} - \Delta x_N)$$

$$\frac{q_H}{q} = q_{frac} \qquad q_{frac} := 0.85$$

according Torenbeek Advanced Aircraft Design P 188

$$S_{HTP} := 93.4\ m^2$$

Design

$$S_{wing} := 521.7\ m^2$$

Design

$$c'_{A\_HTP} := 0.057378$$

Aerotool

$$c'_{A\_wing} := 0.097629$$

Aerotool

$$\frac{d\varepsilon_h}{d\alpha} = \alpha_{frac} \qquad \alpha_{frac} := 0.4$$

according Torenbeek

$$Q := q_{frac} \cdot \frac{S_{HTP}}{S_{wing}} \cdot \frac{c'_{A\_HTP}}{c'_{A\_wing}} \cdot \left(1 - \alpha_{frac}\right)$$

$$L_{Fuselage} := 74\ m$$

$$\Delta x_{AC\_wing\_HTP} := L_{Fuselage} \cdot 0.45 = 33.3\ m$$

acc Raymer P.160; for first estimation

$$\Delta x_{AC\_wing\_HTP} := 29.5\ m$$

from projected aircraft

$$\Delta x_N := \frac{Q}{1+Q} \cdot \Delta x_{AC\_wing\_HTP} = 1.502\ m$$

behind wing's neutral point

Appendix D 13

13. APPENDIX D

Component Weights

Structure Weights

Wing

flight design gross weight, lb $W_{dg} := 294683.866 \cdot 2.2046 = 649660.051$

limit load factor $N_{llf} := 2.1 + \left(\frac{52910.94}{W_{dg} + 22046.23}\right) = 2.179$

-> CS25.337 $N_{llf1} := 2.5$

ultimate load factor, 1,5 $N_z := 1.5 \cdot N_{llf1} = 3.75$

trapezoidal wing area, ft^2 $S_w := 5615.499$

wingspan, ft $B := 262.467$

aspect ratio $A := \frac{(262.467\ ft)^2}{5615.499\ ft^2} = 12.26764$

thickness-to-chord ratio $t\_c := 0.118$

taper ratio $\lambda := \frac{1400}{12950.407} = 0.108$

wing sweep at 25% MAC $\Lambda := 26.2°$

control surface aera, ft^2 $S_{csw} := 1031.39$

Wing Weight:

$$W_{wing} := 0.0051 \cdot (W_{dg} \cdot N_z)^{0.557} \cdot S_w^{0.649} \cdot A^{0.5} \cdot t\_c^{-0.4} (1+\lambda)^{0.1} (\cos(\Lambda))^{-1.0} \cdot S_{csw}^{0.1} = 92647.863 \quad [lb]$$

$$W_{wing\_kg} := \frac{W_{wing}}{2.2046} = 42024.795 \quad [kg]$$

Correction factor wing $cf_{wing} := 1.035$

Calibrated weight wing $W_{wing\_cal} := cf_{wing} \cdot W_{wing\_kg} = 43495.663$
+Winglet: 1.003*W_wing_cal

CG wing
(Roskam Part V Chapter 8) *Swept wing: 70 percent of the distance between the front and rear spar behind the front spar at 35 percent of the semi-span*

$x_{cg\_wing} := 41160.832$ [mm] *measured from Catia*

$y_{cg\_wing} := 0.35 \cdot \frac{B}{2} \cdot 0.3048 = 14$ [m]

Horizontal Tail

1,143 for all moving hor. tail, 1,0 otherwise $K_{uht} := 1.143$

fuselage width at hor. tail intersection, ft $F_w := 10.53$ *from CAD*

hor. tail span, ft $B_h := 26.08268 \cdot 2 = 52.165$

hor. tail area, ft^2 $S_{ht} := 1009.655$

tail length, ft $L_t := 58.7$ *from CAD*

a/c pitching radius of gyration ~0,3*L_t, ft $K_y := 0.3 \cdot L_t = 17.61$

wing sweep hor. tail at 25% MAC $\Lambda_{ht} := 36°$

aspect ratio hor. tail $A_h := \dfrac{52.165^2}{1009.655} = 2.695$

elevator area, ft^2 $S_e := 265.187$

$$W_{horizontal\_tail} := 0.0379 \cdot K_{uht} \cdot \left(1 + \frac{F_w}{B_h}\right)^{-0.25} \cdot W_{dg}^{\ 0.639} \cdot N_z^{\ 0.10} \cdot S_{ht}^{\ 0.75} \cdot L_t^{\ -1.0} \cdot K_y^{\ 0.704} \left(\cos\left(\Lambda_{ht}\right)\right)^{-1.0} \cdot A_h^{\ 0.166} \cdot \left(1 + \frac{S_e}{S_{ht}}\right)^{0.1} = 8386.598 \quad \text{[lb]}$$

$$W_{horizontal\_tail\_kg} := \frac{W_{horizontal\_tail}}{2.2046} = 3804.136 \quad \text{[kg]}$$

Correction factor HTP $cf_{htp} := 0.571$

$$W_{HTP\_Corrected} := W_{horizontal\_tail\_kg} \cdot cf_{htp} = 2172.162$$

CG HTP
(Roskam Part V Chapter 8) *Regardless of sweep angle: 42 percent chord from the L.E. (leadding edge) at 38 percent of the semi-span*

$$x_{cg\_htp} := 71266.492$$

$$y_{cg\_htp} := 0.38 \cdot \frac{B_h}{2}\ ft = 3.021\ m$$

Vertical Tail

0,0 for conventional, 1,0 for T tail $H_{t\_v} := 0$

ver. tail area, ft^2 $S_{vt} := \frac{576.9456}{2} = 288.473$

a/c yawning radius of gyration ~L_t, ft $K_z := L_t = 58.7$

wing sweep hor. tail at 25% MAC $\Lambda_{vt} := 36°$

aspect ratio ver. tail $A_v := \frac{(6.834)^2}{\frac{53.6}{2}} = 1.743$

$$W_{vertical\_tail} := 0.002 \cdot \left(1 + H_{t\_v}\right)^{0.225} \cdot W_{dg}^{0.556} \cdot N_z^{0.536} \cdot L_t^{-0.5} \cdot S_{vt}^{0.5} \cdot K_z^{0.875} \cdot \left(\cos\left(\Lambda_{vt}\right)\right)^{-1} A_v^{0.35} \cdot t\_c^{-0.5} = 2368.005$$ [lb]

$W_{vtp\_kg} := \frac{W_{vertical\_tail}}{2.2046} = 1074.12$ [kg]

correction factor vertical tail $cf_{vtp} := 1.399 \cdot 2 = 2.798$

Calibrated weight vtp $W_{VTP\_cal} := cf_{vtp} \cdot W_{vtp\_kg} = 3005.388$

CG VTP

(Roskam Part V Chapter 8) *Regardless of sweep angle: 42 percent chord from the L.E. at 38 percent vertical tail span from the root chord*

$x_{cg\_htp} := 76331.857$

$y_{cg\_vtp} := 0$

13. APPENDIX D

Fuselage

1.12 if two cargo doors $K_{door} := 1.12$

1.12 if fuselage mounted landing gear $K_{lg} := 1.12$

wing span, ft $B_w := B$

fuselage structural lenght, ft $L := 242.7822$

fuselage wetted area, ft^2 $S_f := 14383.92$ *from CATIA Measurement*

wingsweep factor $K_{ws} := 0.75 \cdot \left(\frac{(1+2 \cdot \lambda)}{(1+\lambda)} \right) \cdot \left(\frac{B_w}{L} \right) \cdot (\tan(\Lambda)) = 0.438$

fuselage structural depth, ft $D := 22.96588$

$$W_{fuselage} := 0.3280 \cdot K_{door} \cdot K_{lg} \cdot (W_{dg} \cdot N_z)^{0.5} \cdot L^{0.25} \cdot S_f^{0.302} \cdot (1+K_{ws})^{0.04} \cdot \left(\frac{L}{D} \right)^{0.10} = 58664.336 \quad \text{[lb]}$$

$$W_{fuselage\_kg} := \frac{W_{fuselage}}{2.2046} = 26609.968 \quad \text{[kg]}$$

correction factor fuselage $cf_{fuselage} := 1.289$

Calibrated weight fuselage $W_{fuselage\_cal} := cf_{fuselage} \cdot W_{fuselage\_kg} = 34300.249$ [kg]

CG fuselage
(Roskam Part V Chapter 8)

Jet transports: 0.42-0.45 (wing mounted engines)
Jet transports: 0.47-0.50 (rear fuselage mounted engines)

$$x_{cg\_fuselage} := 0.50 \cdot L \cdot 0.3048 \cdot 1000 = 3.7 \cdot 10^4$$

13. APPENDIX D

Main Landing Gear

1.126 for kneeling gear, 1.0 otherwise $K_{mp} := 1.0$

landing design gross weight, lb $W_l := 630522.1$

ultimate load factor, =N_gear*1.5 $N_l := 3$

extended lenght of main ldg gear, in $L_m := 153.5433$ *from drawing*

number of main wheels $N_{mw} := 12$

number of main gear shock struts $N_{mss} := 2$

stall speed, kt $V_{stall} := 153.723$ *siehe Werte vn_Diagramm*

$$W_{main\_landing\_gear} := 0.0106 \cdot K_{mp} \cdot W_l^{0.888} \cdot N_l^{0.25} \cdot L_m^{0.4} \cdot N_{mw}^{0.321} \cdot N_{mss}^{-0.5} \cdot V_{stall}^{0.1} = 38352.04$$ [lb]

$$W_{main\_landing\_gear\_kg} := \frac{W_{main\_landing\_gear}}{2.2046} = 17396.371$$ [kg]

$$cf_{landing\_gear} := 0.822$$

CG main landing gear ???

(guestimated) *at 0.50 of strutlength for gears with mostly vertical struts*

$$x_{cg\_main\_landing\_gear} := 41391.309$$

13. APPENDIX D

Nose Landing Gear

1.15 for kneeling gear, 1.0 otherwise $K_{np} := 1.0$

extended nose gear lenght, in $L_n := 120.00948$ *gemessen aus Zeichnung*

number of nose wheels $N_{nw} := 2$

$$W_{nose\_landing\_gear} := 0.032 \cdot K_{np} \cdot W_l^{0.646} \cdot N_l^{0.2} \cdot L_n^{0.5} \cdot N_{nw}^{0.45} = 3328.526 \quad [lb]$$

$$W_{nose\_landing\_gear\_kg} := \frac{W_{nose\_landing\_gear}}{2.2046} = 1509.809 \quad [kg]$$

$cf_{landing\_gear} := 0.822$

CG nose landing gear
(guestimated) at 0.50 of strutlength for gears with mostly vertical struts

$$x_{cg\_nose\_landing\_gear} := 4970.013$$

Landing Gear overall

$$W_{landing\_gear} := W_{main\_landing\_gear} + W_{nose\_landing\_gear} = 41680.566$$

$$W_{landing\_gear\_kg} := W_{main\_landing\_gear\_kg} + W_{nose\_landing\_gear\_kg} = 18906.181$$

correction factor landing gear $cf_{landing\_gear} := 0.822$

calibrated weight landing gear $W_{cal\_ldg} := cf_{landing\_gear} \cdot W_{landing\_gear\_kg} = 15540.881$

$$x_{cg\_landing\_gear} := \frac{\left(W_{main\_landing\_gear\_kg} \cdot x_{cg\_main\_landing\_gear} + W_{nose\_landing\_gear\_kg} \cdot x_{cg\_nose\_landing\_gear}\right)}{W_{landing\_gear\_kg}}$$

13. APPENDIX D

Pylon (Torenbeek)

design dive speed, m/s $v_D := 320$

total wetted area nacelle, m^2 $S_{wet} := 201.675$

$$W_{pylon} := 0.405 \cdot \sqrt{v_D} \cdot S_{wet}^{1.3} = 7179.227$$ [kg]

correction factor pylon $c_{pylon} := 0.661$

calibrated result $c_{pylon} \cdot W_{pylon} = 4745.469$

CG pylon
(guestimated)

$x_{cg\_pylon} := 66243.507$ mm CATIA

Total Structure Weight

$$W_{structure} := W_{wing\_kg} + W_{fuselage\_kg} + W_{horizontal\_tail\_kg} + W_{vtp\_kg} + W_{landing\_gear\_kg} + W_{pylon} = 99598.427$$

deviation total structure weight $d_{total\_structure} := \dfrac{112200}{W_{structure}} = 1.127$

System Weigths

Fuel System

total fuel volume, gal $V_t := 48953.72$ wing tank volume: 185.31m^3

integral tanks volume, gal $V_i := 30000$ *estimated*

self sealing tanks volume, gal $V_p := 5000$ *estimated*

number of fuel tanks $N_t := 6$ *estimated*

$$W_{fuel\_system} := 2.405 \cdot V_t^{0.606} \cdot \left(1 + \frac{V_i}{V_t}\right)^{-1.0} \cdot \left(1 + \frac{V_p}{V_t}\right) \cdot N_t^{0.5} = 2798.012 \quad \text{[lb]}$$

$$W_{fuel\_system\_kg} := \frac{W_{fuel\_system}}{2.2046} = 1269.17 \quad \text{[kg]}$$

correction factor fuel system $cf_{fuel\_system} := 0.601$

calibrated weight fuel system $W_{cal\_fuel\_system} := cf_{fuel\_system} \cdot W_{fuel\_system\_kg} = 762.771$

CG fuel system

$L_{engine\_cowling} := 6400$ [mm]
$x_{LE\_cowling} := 62276.87$ [mm]
$x_{LE\_wing} := 36889.14$ [mm]

$$x_{cg\_fuel\_system} := x_{LE\_wing} + \frac{\left(x_{LE\_cowling} + 0.5 \cdot L_{engine\_cowling} - x_{LE\_wing}\right)}{2} = 51183.005 \quad \text{[mm]}$$

Hydraulic System

number of diff func performed by surface controls $N_f := 7$

$$W_{hydraulics} := 0.2673 \cdot N_f \cdot \left(L + B_w\right)^{0.937} = 638.678$$

$$W_{hydraulics\_kg} := \frac{W_{hydraulics}}{2.2046} = 289.702$$

correction factor hydraulics $cf_{hydraulics} := 9.014$

calibrated weight hydraulics $W_{cal\_hydraulics} := cf_{hydraulics} \cdot W_{hydraulics\_kg} = 2611.377$

CG hydraulic system
(Roskam Part V Chapter 8) $x_{cg\_hydraulic} := 0.5 \cdot L \cdot 0.3048 = 37$ [m]

13. APPENDIX D

Electric System

system electrical rating, typically 40-60 $R_{kva} := 60$

electrical routing distance, generator to avionics to cockpit, ft $L_a := 1000$

number of generator, typically =N_eng $N_{gen} := 4$ (+RAM, +APU)

$$W_{electrical} := 7.291 \cdot R_{kva}^{\ 0.782} \cdot L_a^{\ 0.346} \cdot N_{gen}^{\ 0.10} = 2246.505 \quad \text{[lb]}$$

$$W_{electrical\_kg} := \frac{W_{electrical}}{2.2046} = 1019.008 \quad \text{[kg]}$$

correction factor electrical systems $cf_{electrical} := 6.86$

calibrated weight electrics $W_{cal\_electrical} := cf_{electrical} \cdot W_{electrical\_kg} = 6990.395$

CG electric system $x_{cg\_electric} := 0.5 \cdot L \cdot 0.3048 = 37$ [m]

Air Conditioning

number of personell on board (crew+passengers) $N_p := 424 + 3 + 18 = 445$

volume of pressurized section, ft^3 $V_{pr\_m3} := 1644.118$

$V_{pr} := 35.31467 \cdot V_{pr\_m3} = 58061.485$

uninstalled avionics weight typically 800-1400, lb $W_{uav} := 1400$ [lb]

$$W_{air\_conditioning} := 62.36 \cdot N_p^{\ 0.25} \cdot \left(\frac{V_{pr}}{1000}\right)^{0.604} \cdot W_{uav}^{\ 0.10} = 6870.642 \quad \text{[lb]}$$

$$W_{air\_conditioning\_kg} := \frac{W_{air\_conditioning}}{2.2046} = 3116.503 \quad \text{[kg]}$$

correction factor air conditioning $cf_{air\_conditioning} := 0.622$

calibrated weight air conditioning $W_{cal\_air\_conditioning} := cf_{air\_conditioning} \cdot W_{air\_conditioning\_kg} = 1938.465$

CG air conditioning $x_{cg\_air\_conditioning} := \frac{x_{LE\_wing}}{1000} = 36.889$ [m]

Flight Controls

number of surf. cont. driven by mechanical actuation instead of hydraulics $N_m := 1$

total area of control surfaces $S_{cs} := 1031.39 + 265.187 + 111.493 = 1408.07$

yawning moment of inertia, lb-ft^2 $I_{yaw} := 1000000$ *estimated*

$$W_{flight\_controls} := 145.9 \cdot N_f^{0.554} \cdot \left(1 + \frac{N_m}{N_f}\right)^{-1.0} \cdot S_{cs}^{0.20} \cdot \left(I_{yaw} \cdot 10^{-6}\right)^{0.07} = 1599.457 \quad \text{[lb]}$$

$$W_{flight\_controls\_kg} := \frac{W_{flight\_controls}}{2.2046} = 725.509 \quad \text{[kg]}$$

correction factor flight controls $cf_{flight\_controls} := 4.053 \cdot 2$

calibrated weight air conditioning $W_{cal\_flight\_controls} := cf_{flight\_controls} \cdot W_{flight\_controls\_kg} = 5880.975$

CG flight controls

$$x_{cg\_flight\_controls} := 55.444$$

To Be Determined:

$W_{bleedair\_kg} := 1050$ $W_{fireprotection\_kg} := 360$ $W_{communication\_kg} := 250$ $W_{navigation\_kg} := 27$

$x_{cg\_bleedair} := 45.035$

$x_{cg\_communication} := \frac{6818}{1000}$

$x_{cg\_fireprotection} := 0.5 \cdot L \cdot 0.3048 = 37$

$x_{cg\_navigation} := \frac{6818}{1000}$

Total Systems Weight

$$W_{systems} := W_{fuel\_system\_kg} + W_{hydraulics\_kg} + W_{electrical\_kg} + W_{air\_conditioning\_kg} + W_{flight\_controls\_kg} + W_{bleedair\_kg} + W_{fireprotection\_kg} + W_{communication\_kg} + W_{navigation\_kg} = 8349.892 \quad \text{[kg]}$$

deviation total system weight $dv_{total\_system} := \frac{15600}{W_{systems}} = 1.868$

13. APPENDIX D

Furnishing Weights

Furnishing (overall formula, Raymer)

Number of crew $N_C := 20$

Cargo weight, lb $W_C := 27780$ 420*30kg

$$W_{furnishing} := 0.0577 \cdot N_C^{\ 0.1} \cdot W_C^{\ 0.393} \cdot S_f^{\ 0.75} = 5702.608 \quad \text{[lb]}$$

$$W_{furnishing\_kg} := \frac{W_{furnishing}}{2.2046} = 2586.686 \quad \text{[kg]}$$

correction factor furnishing $c_{furnishing} := \dfrac{11200}{W_{furnishing\_kg}} = 4.33$

Lighting, Oxygen, Lining, Stowage Bins (formula from presentation)

cabin lenght, m $L_{cabin} := 65.8$

Mass per cabin Length $m_{cabin} := 150$

$$W_{lols} := L_{cabin} \cdot m_{cabin} = 9870$$

Lavatories:

Number of lavatories Business $n_{Lav\_Business} := 3$

Number of lavatories Economy $n_{Lav\_Eco} := 9$

Mass per Lavatory $m_{Lavatory} := 100$

$$W_{lavatories} := \left(n_{Lav\_Business} + n_{Lav\_Eco}\right) \cdot m_{Lavatory} = 1200$$

Water, Waste (formula from presentation)

Number of Passenger $N_{pax} := 424$

Mass Water Waste per PAX $m_{ww} := 1.5$

$$W_{ww} := m_{ww} \cdot N_{pax} = 636$$

correction factor water/waste $cf_{water\_waste} := 0.952$

CG water, waste
(Roskam Part V Chapter 8)

Emergency Oxygen (Torenbeek)

$$W_{oxygen} := 40 + 2.4 \cdot N_{pax} = 1057.6 \quad \text{[lb]}$$

$$W_{oxygen\_kg} := \frac{W_{oxygen}}{2.2046} = 479.724 \quad \text{[kg]}$$

correction factor oxygen $cf_{oxygen} := 0.736$

$$W_{oxygen\_kg} := W_{oxygen\_kg} \cdot cf_{oxygen} = 353.077$$

CG oxygen
(Roskam Part V Chapter 8)

Only Furnishing: $W_{furnishing\_o} := W_{lols} + W_{lavatories} - W_{oxygen\_kg} - 400 = 10316.923$

correction factor only furnising $cf_{furnishing\_o} := 1.056$

calibrated weight furnishing only $W_{cal\_furnishing\_only} := cf_{furnishing\_o} \cdot W_{furnishing\_o} = 10894.671$

Total Furnishing Weight

$$W_{total\_furnishing} := W_{lols} + W_{ww} + W_{lavatories} = 11706$$

deviation total system weight $d_{total\_system} := \dfrac{11200}{W_{total\_furnishing}} = 0.957$

Operator items

Seats:

Mass per Business Seat	$W_{BusinessSeat} := 21\ kg$	$n_{PAX\_Business} := 60$
Mass per PremEco Seat	$W_{PremEcoSeat} := 16\ kg$	$n_{PAX\_PremEco} := 45$
Mass per Economy Seat	$W_{EcoSeat} := 12\ kg$	$n_{PAX\_Eco} := 319$

$$W_{seats} := W_{BusinessSeat} \cdot n_{PAX\_Business} + W_{PremEcoSeat} \cdot n_{PAX\_PremEco} + W_{EcoSeat} \cdot n_{PAX\_Eco} = 5808\ kg$$

Unusable fuel

Total Fuel Weight	$W_{totalFuel} := 138000\ kg$

$$W_{unusable\_fuel} := 0.25\% \cdot W_{totalFuel} = 345\ kg$$

Flight crew

Number of Flight Crew	$n_{FlightCrew} := 3$
Number of Cabin Crew	$n_{CabinCrew} := 18$
Flight Crew Standard Mass	$W_{FC} := 85\ kg$

$$W_{FlightCrew} := n_{FlightCrew} \cdot W_{FC} = 255\ kg$$

Container

Container per Pax	$n_{PAXPerContainer} := 40$
Number of Pax	$n_{PAX} := 424$

$$n_{Container} := \frac{n_{PAX}}{n_{PAXPerContainer}} = 10.6 \qquad n_{Container} := 12$$

Weight per Container	$W_{PerContainer} := 75\ kg$

$$W_{container} := W_{PerContainer} \cdot n_{Container} = 900\ kg$$

Catering

13. APPENDIX D

Mass per Business Pax	$W_{CateringBusiness} := 20\ kg$
Mass per PremEco Pax	$W_{CateringPremEco} := 18\ kg$
Mass per Economy Pax	$W_{CateringEco} := 12\ kg$

$$W_{catering} := 50 \cdot 20 + 36 \cdot 16 + 334 \cdot 12 = 5584$$

$$W_{Catering} := W_{CateringBusiness} \cdot n_{PAX\_Business} + W_{CateringPremEco} \cdot n_{PAX\_PremEco} + W_{CateringEco} \cdot n_{PAX\_Eco} = 5838\ kg$$

Galley struct.

Galley Mass per Business Pax	$W_{GalleyBusiness} := 7\ kg$
Galley Mass per PremEco Pax	$W_{GalleyPremEco} := 7\ kg$
Galley Mass per Economy Pax	$W_{GalleyEco} := 6\ kg$

$$W_{galley\_struct} := W_{GalleyBusiness} \cdot n_{PAX\_Business} + W_{GalleyPremEco} \cdot n_{PAX\_PremEco} + W_{GalleyEco} \cdot n_{PAX\_Eco} = 2649\ kg$$

Additional Opt.

Maximum Weight Empty $W_{MWE} := 162600\ kg$

$$W_{add\_opt} := 1\% \cdot W_{MWE} = 1626\ kg$$

All Operator Items

$$W_{operator\_items} := W_{seats} + W_{unusable\_fuel} + W_{FlightCrew} + W_{container} + W_{Catering} + W_{galley\_struct} + W_{add\_opt} = 17421\ kg$$

$$W_{operator\_items\_RefAC} := 25500$$

correction factor Operator Items $cf_{OperatorItems} := \dfrac{W_{operator\_items\_RefAC}}{W_{operator\_items}} = 1.464\ \dfrac{1}{kg}$

Weights

				Method			Calibration Factor	Result	Calibrated result	Weight Difference	CoG Eq. Moment		Change	Technc		New Technol.	Weight Diff	EQ MM	x_i		y_i	z_i	V_i	m_i	Geodesic	
Structure	**Wing**	[kg]	49100	Raymer	Torenbeek				43626,14999		1795688631		X	X	X	37082,227	-6543,922498	1526335335,93	41160,832		14000				10%	39263,535
	Fuselage	[kg]	37800	Raymer	Torenbeek				34300,249		1269109213		X	X	X	30870,224	-3430,0249	1142198292	37000						15%	29155,2117
	HTP	[kg]	3110	Raymer	Torenbeek				2172,162		154802365,8		X	X		2172,162	0	154802365,8	71266,492							
	VTP	[kg]	2160	Raymer	Torenbeek				3005,388		229406847		X	X		3005,388	0	229406847	76331,857							
	LG	[kg]	16200		Torenbeek		0,822						X													
	Front Landing Gear	[kg]		Raymer				1509,81	1241,06382		6168103,319		X		!	1241,0638	0	6168103,319	4970,013							
	Main Landing Gear LEFT	[kg]		Raymer				8698,1855	7149,908481		295944078,4		X		!	7149,9085	0	295944078,4	41391,31							
	Main Landing Gear RIGHT	[kg]		Raymer				8698,1855	7149,908481		295944078,4		X		!	7149,9085	0	295944078,4	41391,31							
	Pylon	[kg]	3830		Torenbeek		0,661	7179,227	4745,469047		314356512		X			4745,469	0	314356512	66243,507							
	Total Structure	**[kg]**	**112200**			**0**			**103390,2988**	**8809,701182**	**4361419829**					**93416,351**	**-9973,947398**	**3965155613**	**42184,0335**							**68418,7466**
Systems	**System Weights Breakdown**																									
	Bleed Air System	[kg]	1050					1050	1050		47286928,5		-			1050	0	47286928,5	45035,17							
	Fuel System	[kg]	780	Raymer					762,771		39040911,91		o			762,771	0	39040911,91	51183,005							
	Hydraulic System	[kg]	2500	Raymer	Torenbeek				2611,378		96620986		o			2611,378	0	96620986	37000							
	Fire Protection	[kg]	360					360	360		13320000		-			360	0	13320000	37000							
	Electric System	[kg]	5500	Raymer	Roskam R5/Torenbeek				6990,4		258644800		o			6990,4	0	258644800	37000							
	Air Conditioning	[kg]	2100	Raymer	Roskam R5/Torenbeek				1938,465		71719599,46		o			1938,465	0	71719599,46	36998,14							
	Flight Controls	[kg]	2800	Raymer	Roskam R5/Torenbeek				2940,49		163034203,6		o			5880,98	2940,49	326068407,3	55444,57							
	Communication	[kg]	250					250	250		1704500		-			250	0	1704500	6818							
	Navigation	[kg]	270					270	270		1840860		-			270	0	1840860	6818							
	Total Systems	**[kg]**	**15600**			**0**		**1930**	**17173,504**	**-1573,504**	**693212789,5**					**20113,994**	**2940,49**	**856246993,1**	**40365,2504**							
Furnishing	**Furnishing Weight Breakdown**																									
	Furnishing	[kg]	9850		Airbus Folien		1,056		9660,931		357454447		X						37000							
	Lavatories per Unit	[kg]	**100**		Airbus Folien																					
	Lavatory 1	[kg]	100						100		629380		X						6293,8							
	Lavatory 2	[kg]	200		2 Lavatories				200		1879302		X						9396,51							
	Lavatory 3	[kg]	200		2 Lavatories				200		6232226		X						31161,13							
	Lavatory 4	[kg]	400		4 Lavatories				400		16080472		X						40201,18							
	Lavatory 5	[kg]	200		2 Lavatories				200		11752322		X						58761,61							
	Lighting	[kg]	400				1	400	400		14800000		-						37000							
	Emergency Oxygen	[kg]	350		Roskam R5/Torenbeek		0,735	475	349,125		12788448,75		o						36630							
	Water Intallation		600		Airbus Folien		0,953	630	600,39		30619890		o						51000							
	Total Furnishings	**[kg]**	**11200**			**0**		**1105**	**12110,446**	**-910,446**	**452236487,8**					**12110,446**		**452236487,8**	**37342,6782**							
Power Plant	APU	[kg]	500		Roskam R5/Torenbeek			500	500		34000000		X	X	X	1100	600	35200000	68000	32000						
	Engines (each)	[kg]	11550					7900	7900		1034534546		X	X	X	8462,2224	5374,44486	554079838	65476,87							
	Total Power Units	**[kg]**	**23600**			**0**		**8400**	**16300**	**7300**	**1068534546**					**18024,445**	**1724,44486**	**589279838**	**65554,2666**							
	MWE Evaluation	**[kg]**	**162600**			**0**			148974,2488	13625,75118	6575403652					143665,24	-5309,012539	5862918931	44137,8541							
Operator Items	Fuel								138000																	
	Unusable Fuel	[kg]	356,25		Airbus Slides			356,25	533,3318656		24927131,4		o			345		16124782,5	46738,5							
	Flight Crew	[kg]	255		Airbus Slides			255	381,7533354		1412869,094		o			381,75334		1412869,094	3701							
	LD 3 Container	[kg]	900		Airbus Slides			900																		
	LD 3 Container 1	[kg]	150					150	224,5607855		2805260,491		o			224,56079		2805260,491	12492,21							
	LD 3 Container 2	[kg]	150					150	224,5607855		3156509,489		o			224,56079		3156509,489	14056,37							
	LD 3 Container 3	[kg]	150					150	224,5607855		3507758,487		o			224,56079		3507758,487	15620,53							
	LD 3 Container 4	[kg]	150					150	224,5607855		3859007,485		o			224,56079		3859007,485	17184,69							
	LD 3 Container 5	[kg]	150					150	224,5607855		4210256,484		o			224,56079		4210256,484	18748,85							
	LD 3 Container 6	[kg]	150					150	224,5607855		4561505,482		o			224,56079		4561505,482	20313,01							
	Seats	[kg]	5634		Airbus Slides		Matlab Script	5808	8694,993615		318497616,1		o			8694,9936		318497616,1	36630							
	Catering	[kg]	5656		Airbus Slides			5584	8359,649509		351036419,8		o			8359,6495		351036419,8	41991,7629							
	Galley Structure	[kg]	2606		Airbus Slides			2649																		
	Galley 1	[kg]	372,285714					378,428571	566,5347818		3565656,609		o			566,53478		3565656,609	6293,8							
	Galley 2	[kg]	372,285714					378,428571	566,5347818		5323449,742		o			566,53478		5323449,742	9396,51							
	Galley 3	[kg]	372,285714					378,428571	566,5347818		17653863,98		o			566,53478		17653863,98	31161,13							
	Galley 4	[kg]	372,285714					378,428571	566,5347818		33290495,9		o			566,53478		33290495,9	58761,61							
	Galley 5	[kg]	372,285714					378,428571	566,5347818		35315982,38		o			566,53478		35315982,38	62336,83							
	Galley 6	[kg]	372,285714					378,428571	566,5347818		35315982,38		o			566,53478		35315982,38	62336,83							
	Galley 7	[kg]	372,285714					378,428571	566,5347818		36063128,45		o			566,53478		36063128,45	63655,63							
	Additional Options	[kg]	1626			0		1489,74249	2230,251622		82519310,03		o			1436,6524		53156137,42	37000							
		[kg]																								
	Mid Sum		17033,25			0			25513,08813	-8479,838133	967022203,8					24531,157		967022203,8	37902,9853							
	Calibration		25500				1,497071903																			
	Total Operator Items	**[kg]**	**25500**			**25500**				**25500**																
	OWE	**[kg]**	**188100**			**25500**			**174487,337**	**13612,66305**	**7542425856**	**43226,20763**				**168196,39**	**142696,3933**	**6829941135**	**40606,9417**							
	Max Strucutral Payload	[kg]	55000			55000																				
	MZFW	**[kg]**	**242100**			**80500**																				
	MLW	**[kg]**	**286000**			**286000**																				
	MTOW	**[kg]**	**377000**																							

Appendix E 14

Payload Range

$m_{MWE} := 153584\ kg$

$m_{OWE} := 168196.39\ kg$

$m_{crew} := 2200\ kg$

$m_{max\_payload} := 55000\ kg$

$m_{SPP\_payload} := 420 \cdot 110\ kg = 46200\ kg$

$m_{no\_payload} := 0\ kg$

$$m_{10} := m_{OWE} + m_{SPP\_payload} + m_{crew} = 216596.39\ kg$$

Cruise Speed:

$Ma := 0.8$

Specific Fuel Burn:

$$c_{TL} := 0.3372 \cdot \frac{\frac{lb}{hr}}{lb} = 0.000093667\ \frac{1}{s}$$

Initial Cruise Altitude:

$H_{FlightLevel} := 33000\ ft$

ISA Standard atmosphere:

$$T_{FL330} := 15\ °C - 0.0065\ \frac{K}{m} \cdot H_{FlightLevel} = 222.77\ K$$

$$c_{FL330} := \sqrt{1.4 \cdot 287.15\ \frac{J}{K \cdot kg} \cdot T_{FL330}}$$

Cruise Speed:

$$v := Ma \cdot c_{FL330} = 239.407\ \frac{m}{s}$$

Roskam, Part I, p. 12

Engine Start, Warm-up:

$m_1m_0 := 0.99$

Taxi:

$m_2m_1 := 0.99$

Take-off:

$m_3m_2 := 0.995$

Climb:

$m_4m_3 := 0.98$

Descent:

$m_6m_5 := 0.99$

Missed Approach and Climb:

$m_7m_6 := 0.988$

Landing, Taxi, Shut-Down:

$m_{10}m_9 := 0.992$

SPP

Range climb & descent

$v_{FL10} := 128.611 \frac{m}{s}$ $\quad$ $L\_D := 27.2697$

$v_{climb} := \frac{(v_{FL10} + v)}{2} = 184.009 \frac{m}{s}$ $\quad$ $v_{descent} := \frac{(v_{FL10} + v)}{2} = 184.009 \frac{m}{s}$

$c_{TL\_climb} := c_{TL} \cdot 1.2 = 0.40464 \frac{\frac{lb}{hr}}{lb}$ $\quad$ $c_{TL\_descent} := c_{TL} \cdot 1.2 = 0.405 \frac{\frac{lb}{hr}}{lb}$

$R_c := \frac{v_{climb}}{c_{TL\_climb}} \cdot L\_D \cdot \ln\left(\frac{1}{m_4 m_3}\right) = 486.992\ nmi$ $\quad$ $R_d := \frac{v_{descent}}{c_{TL\_descent}} \cdot L\_D \cdot \ln\left(\frac{1}{m_6 m_5}\right) = 242.266\ nmi$

$R_{new} := 8000\ nmi - R_c - R_d = 7270.742\ nmi$

$R_{alternate} := 300\ nmi$ $\quad$ $R_{holding} := 100\ nmi$

$m_8 m_7 := \exp\left(-\frac{R_{alternate} \cdot c_{TL\_descent}}{v_{FL10} \cdot L\_D}\right) = 0.982$

$m_9 m_8 := \exp\left(-\frac{R_{holding} \cdot c_{TL\_descent}}{v_{FL10} \cdot L\_D}\right) = 0.994$

$m_5 m_4 := \exp\left(-\frac{R_{new} \cdot c_{TL}}{v \cdot L\_D}\right) = 0.824$

$m_{10} m_0 := m_1 m_0 \cdot m_2 m_1 \cdot m_3 m_2 \cdot m_4 m_3 \cdot m_5 m_4 \cdot m_6 m_5 \cdot m_7 m_6 \cdot m_8 m_7 \cdot m_9 m_8 \cdot m_{10} m_9 = 0.746$
$cf := 0.984663$
$m_{10} m_0 := cf \cdot m_{10} m_0 = 0.735$

$m_{fuel} m_0 := 1 - m_{10} m_0 = 0.265$

MTOW: $m_{MTOW} := \frac{m_{10}}{m_{10} m_0} = 294681.526\ kg$ $\quad$ Fuel: $m_{fuel} m_0 \cdot \frac{m_{10}}{m_{10} m_0} = 7808$

$R_{test} := \frac{v}{c_{TL}} \cdot L\_D \cdot \ln\left(\frac{1}{m_{10} m_0}\right) = 11586.281\ nmi$

range with max PP

$m_{10\_maxPP} := m_{OWE} + m_{max\_payload} + m_{crew} = 225396.39\ kg$

$$R_{maxPP1} := \frac{v}{c_{TL}} \cdot L\_D \cdot \ln\left(\frac{0.984663 \cdot m_{MTOW} \cdot m_1m_0 \cdot m_2m_1 \cdot m_3m_2 \cdot m_4m_3}{m_{10\_maxPP} \cdot \frac{1}{m_{10}m_9} \cdot \frac{1}{m_9m_8} \cdot \frac{1}{m_8m_7} \cdot \frac{1}{m_7m_6} \cdot \frac{1}{m_6m_5}}\right) = 5771.932\ nmi$$

$R_{maxPP} := R_{maxPP1} + R_c + R_d = 6501.191\ nmi$

$m_{fuel\_maxPP} := m_{MTOW} - m_{10\_maxPP} = 69285.136\ kg$

Maximum fuel, MTOW

$m_{max\_fuel} := 110000\ kg$

design decision!

$m_{10\_maxfuel} := m_{MTOW} - m_{max\_fuel} = 184681.526\ kg$

$m_{PP\_max\_fuel} := m_{10\_maxfuel} - m_{OWE} - m_{crew} = 14285.136\ kg$

Range climb & descent

$v_{FL10} := 128.611\ \frac{m}{s}$

$v_{climb} := \frac{(v_{FL10} + v)}{2} = 184.009\ \frac{m}{s}$

$v_{descent} := \frac{(v_{FL10} + v)}{2} = 184.009\ \frac{m}{s}$

$c_{TL\_climb} := c_{TL} \cdot 1.2 = 0.40464\ \frac{\frac{lb}{hr}}{lb}$

$c_{TL\_descent} := c_{TL} \cdot 1.2 = 0.405\ \frac{\frac{lb}{hr}}{lb}$

$R_c := \frac{v_{climb}}{c_{TL\_climb}} \cdot L\_D \cdot \ln\left(\frac{1}{m_4m_3}\right) = 486.992\ nmi$

$R_d := \frac{v_{descent}}{c_{TL\_descent}} \cdot L\_D \cdot \ln\left(\frac{1}{m_6m_5}\right) = 242.266\ nmi$

$R_{alternate} := 300\ nmi$

$R_{holding} := 100\ nmi$

$m_8m_7 := \exp\left(-\frac{R_{alternate} \cdot c_{TL\_descent}}{v_{FL10} \cdot L\_D}\right) = 0.982$

$m_9m_8 := \exp\left(-\frac{R_{holding} \cdot c_{TL\_descent}}{v_{FL10} \cdot L\_D}\right) = 0.994$

$$R_{maxfuel1} := \frac{v}{c_{TL}} \cdot L\_D \cdot \ln\left(\frac{0.984663 \cdot m_{MTOW} \cdot m_1m_0 \cdot m_2m_1 \cdot m_3m_2 \cdot m_4m_3}{m_{10\_maxfuel} \cdot \frac{1}{m_{10}m_9} \cdot \frac{1}{m_9m_8} \cdot \frac{1}{m_8m_7} \cdot \frac{1}{m_7m_6} \cdot \frac{1}{m_6m_5}}\right) = 13269.86\ nmi$$

$R_{maxfuel} := R_{maxfuel1} + R_c + R_d = 13999.119\ nmi$

Ferry range

$m_{10\_ferry} := m_{OWE} + m_{crew} = 170396.39\ kg$

$m_{0\_ferry} := m_{10\_ferry} + m_{max\_fuel} = 280396.39\ kg$

Range climb & descent

$v_{FL10} := 128.611\ \frac{m}{s}$ $\quad L\_D := 27.2697$

$v_{climb} := \frac{(v_{FL10} + v)}{2} = 184.009\ \frac{m}{s}$ $\quad v_{descent} := \frac{(v_{FL10} + v)}{2} = 184.009\ \frac{m}{s}$

$c_{TL\_climb} := c_{TL} \cdot 1.2 = 0.40464\ \frac{\frac{lb}{hr}}{lb}$ $\quad c_{TL\_descent} := c_{TL} \cdot 1.2 = 0.405\ \frac{\frac{lb}{hr}}{lb}$

$R_c := \frac{v_{climb}}{c_{TL\_climb}} \cdot L\_D \cdot \ln\left(\frac{1}{m_4m_3}\right) = 486.992\ nmi$ $\quad R_d := \frac{v_{descent}}{c_{TL\_descent}} \cdot L\_D \cdot \ln\left(\frac{1}{m_6m_5}\right) = 242.266\ nmi$

$R_{alternate} := 300\ nmi$ $\quad R_{holding} := 100\ nmi$

$m_8m_7 := \exp\left(-\frac{R_{alternate} \cdot c_{TL\_descent}}{v_{FL10} \cdot L\_D}\right) = 0.982$ $\quad m_9m_8 := \exp\left(-\frac{R_{holding} \cdot c_{TL\_descent}}{v_{FL10} \cdot L\_D}\right) = 0.994$

$$R_{ferry1} := \frac{v}{c_{TL}} \cdot L\_D \cdot \ln\left(\frac{0.984663 \cdot m_{0\_ferry} \cdot m_1m_0 \cdot m_2m_1 \cdot m_3m_2 \cdot m_4m_3}{m_{10\_ferry} \cdot \frac{1}{m_{10}m_9} \cdot \frac{1}{m_9m_8} \cdot \frac{1}{m_8m_7} \cdot \frac{1}{m_7m_6} \cdot \frac{1}{m_6m_5}}\right) = 14429.563\ nmi$$

$$R_{ferry} := R_{ferry1} + R_c + R_d = 15158.821\ nmi$$

Appendix F 15

Speeds and loads v-n-diagram

$Ma := 0.8$

$Höhe := 33000\ ft = (1.006 \cdot 10^4)\ m$

$T := 15\ °C - 0.0065\ \frac{K}{m} \cdot Höhe = 222.77\ K$

$W_{dg} := 649661.5585$ [lb] $W_{dg\_kg} := 294681.526\ kg$ $W_{dg\_kg\_WU} := 294681.526$

$N_{llf1} := 2.1 + \left(\frac{24000}{W_{dg} + 10000}\right) = 2.136$ -> $N_{llf\_pos} := 2.5$ $N_{llf\_neg} := -1.0$

$\rho := 1.225\ \frac{kg}{m^3}$ $\rho_{WU} := 1.225$

$\rho_{FL33} := 0.4098$ [kg/m^3]

$c_A := 1.6$

$c_{A\_max} := 2.53$

$S := 521.7\ m^2$ $S_{WU} := 521.7$

$v_C := Ma \cdot \sqrt{1.4 \cdot 287.15\ \frac{J}{K \cdot kg} \cdot T} = 239.407\ \frac{m}{s}$ $v_{C\_WU} := 239.407$

$$v_{s1} := \sqrt[2]{\frac{2 \cdot W_{dg\_kg} \cdot 9.81\ \frac{m}{s^2}}{\rho \cdot c_A \cdot S}} = 75.195\ \frac{m}{s}$$

$$v_{s0} := \sqrt[2]{\frac{2 \cdot W_{dg\_kg} \cdot 9.81\ \frac{m}{s^2}}{\rho \cdot c_{A\_max} \cdot S}} = 59.798\ \frac{m}{s}$$ $v_F := 1.8 \cdot v_{s1} = 170.624\ \frac{m}{s}$ $n_F := 2$

$v_A := v_{s1} \cdot \sqrt{N_{llf\_pos}} = 118.893\ \frac{m}{s}$

$v_{D1} := \frac{v_C}{0.8} = 299.259\ \frac{m}{s}$ -> $v_D := 320\ \frac{m}{s}$ $v_{D\_WU} := 320$

$v_{D\_knots} := v_{D\_WU} \cdot 1.943844$

Gust load

$$w := \frac{W_{dg}}{\left(S_{WU} \cdot 10.76391\right)} = 115.69$$ [lb/ft^2]

$$U := \frac{20.86 - 44}{60000 - 15000} \cdot 33000 + 51.7133 = 34.744$$ [ft/s] CS25.337 (5) (i)

$a := 5.8443$ [1/rad] $c := 8.53747 \cdot 3.28084 = 28.01$ [ft]

$$\mu := \frac{2 \cdot w}{\left(\rho_{FL33} \cdot 0.00194\right) \cdot c \cdot a \cdot \left(9.81 \cdot 3.28084\right)} = 55.24$$

$$K := \frac{0.88 \cdot \mu}{5.3 + \mu} = 0.803$$

$$v_{C\_knots} := v_{C\_WU} \cdot 1.943844 = 465.37$$

$$v_{s1\_knots} := 75.195 \cdot 1.943844 = 146.167$$

$$v_{B\_knots} := v_{s1\_knots} \cdot \left(1 + \frac{K \cdot U \cdot v_{C\_knots} \cdot a}{498 \cdot w}\right)^{0.5} = 222.49$$

$$v_B := \frac{v_{B\_knots}}{1.943844} = 114.459$$

Roskam

$$n_{Cpos} := 1 + \frac{K \cdot U \cdot v_{C\_knots} \cdot a}{498 \cdot w} = 2.317$$

$$n_{Cneg} := 1 - \frac{K \cdot U \cdot v_{C\_knots} \cdot a}{498 \cdot w} = -0.317$$

$$n_{Bpos} := 1 + \frac{K \cdot U \cdot v_{B\_knots} \cdot a}{498 \cdot w} = 1.63$$

$$n_{Bneg} := 1 - \frac{K \cdot U \cdot v_{B\_knots} \cdot a}{498 \cdot w} = 0.37$$

$$n_{Dpos} := 1 + \frac{K \cdot 0.5 \cdot U \cdot v_{D\_knots} \cdot a}{498 \cdot w} = 1.88$$

$$n_{Dneg} := 1 - \frac{K \cdot 0.5 \cdot U \cdot v_{D\_knots} \cdot a}{498 \cdot w} = 0.12$$

Appendix G 16

DOC

FC (flight cycles per year)

$T_{OT} := 6011.2$ Yearly operating time, hrs
$T_{ThOT} := 8760$ Theoretical yearly operating time, hrs
$T_{DT} := 2748.8$ Yearly forced down time, hrs
$T_{FT} := 8.3333$ Flight time per flight, hrs
$T_{BTS} := 1$ Block time supplement per flight, hrs

$FC_{refAC} := 600$

$$FC := \frac{T_{ThOT} - T_{DT}}{T_{FT} + T_{BTS}} = 644.059$$

$FC := 644$

Fuel costs

$W_{BlockFuel} := 42492$ Block fuel, kg
$K_{fuel} := 0.8$ Fuel cost, €/kg

$DOC_{fuel1} := W_{BlockFuel} \cdot K_{fuel} \cdot FC = 21891878.4$ $cf_{fuel} := 0.864$
$DOC_{fuel} := cf_{fuel} \cdot DOC_{fuel1} = 18914582.938$

Airframe maintenance cost

$W_{OWE} := 168.19639$ operating weight empty, t
$FT := 8.333$ flight time, hrs
$LR := 50$ labor rate, €/hr
$B := 2$ cost burden

$$DOC_{MaintAFMaterial} := \left(W_{OWE} \cdot (0.21 \cdot FT + 13.7) + 57.5\right) \cdot FC = 1710542.859$$

$$DOC_{MaintAFPersonel} := LR \cdot (1 + B) \cdot \left(\left(0.655 + 0.01 \cdot W_{OWE} \cdot FT\right) + 0.254 + 0.01 \cdot W_{OWE}\right) \cdot FC = 1604213.89$$

$$DOC_{MaintAF1} := DOC_{MaintAFMaterial} + DOC_{MaintAFPersonel} = 3314756.752$$

$cf_{maintAF} := 0.952$

$$DOC_{MaintAF} := DOC_{MaintAF1} \cdot cf_{maintAF} = 3155648.428$$

Engine Maintenance cost

$N_{eng} := 2$ number of engines

$F_{SLS} := \frac{600}{9.81} = 61.162$ sea level static thrust, t

$$DOC_{eng1} := \left(N_{eng} \cdot \left(1.5 \cdot F_{SLS} + 30.5 \cdot FT + 10.6\right)\right) \cdot FC = 459171.51$$

$cf_{eng} := 12.963$

$$DOC_{eng} := DOC_{eng1} \cdot cf_{eng} = 5952240.279$$

Crew cost

$N_{crew} := 5$ number of crew complement

$K_{CC} := 300000$ average salary of cockpit crew per year, €

$K_{FA} := 60000$ average salary of one flight attendant per year, €

$N_{FA} := 18$ number of flight attendants

$$DOC_{crew1} := N_{crew} \cdot (K_{CC} + K_{FA} \cdot N_{FA}) = 6900000 \qquad cf_{crew} := 1.187$$

$$DOC_{crew} := DOC_{crew1} \cdot cf_{crew} = 8190300$$

Landing cost

$K_{ldg} := 0.01$ landing fee rate, €/kg

$W_{MTOW} := 294681.526$ maximum take off weight, kg

$$DOC_{landing1} := (K_{ldg} \cdot W_{MTOW}) \cdot FC = 1897749.027 \qquad cf_{landing} := 0.724$$

$$DOC_{landing} := DOC_{landing1} \cdot cf_{landing} = 1373970.296$$

Navigation cost

$K_{nav} := 77$ unit rate for navigation, €/kg

$R := 4000 \cdot 1.852$ range, km

$W_{MTOW\_t} := 10^{-3} \cdot W_{MTOW}$ maximum take of weight, t

$$DOC_{navigation1} := \left(K_{nav} \cdot \frac{R}{100} \cdot \sqrt[2]{\frac{W_{MTOW\_t}}{50}}\right) \cdot FC = 8918031.851$$

$$cf_{navigation} := 0.349$$

$$DOC_{navigation} := DOC_{navigation1} \cdot cf_{navigation} = 3112393.116$$

Ground handling cost

$K_{ground} := 0.1$ ground handling fee rate, €/kg

$W_{DOC\_PP} := 46200$ payload, kg

$$DOC_{ground1} := (K_{ground} \cdot W_{DOC\_PP}) \cdot FC = 2975280$$

$$cf_{ground} := 2.955$$

$$DOC_{ground} := DOC_{ground1} \cdot cf_{ground} = 8791952.4$$

Capital cost

$K_{OWE} := 1150$	cost per kg OWE, €/kg
$W_{OWE\_kg} := W_{OWE} \cdot 1000 = 168196.39$	operating weight empty, kg
$W_{eng} := 8462.22$	weight per engine, kg
$K_{eng} := 2500$	price per kg engine, €/kg
$Ins := 0.5\%$	insurance rate, % price a/c
$IR := 5\%$	intrest rate, % price a/c
$f_{RV} := 10\%$	residual value factor, % price a/c
$DP := 14$	depreciation, years
$A := IR \cdot \dfrac{1 - f_{RV} \cdot \left(\dfrac{1}{1+IR}\right)^{DP}}{1 - \left(\dfrac{1}{1+IR}\right)^{DP}} = 0.0959$	annual rate

$$DOC_{capital} := \left(K_{OWE} \cdot \left(W_{OWE\_kg} - W_{eng} \cdot N_{eng}\right) + K_{eng} \cdot W_{eng} \cdot N_{eng}\right) \cdot (A + Ins) = 21826696.278$$

$$DOC_{fees} := DOC_{navigation} + DOC_{landing} + DOC_{ground} = 1.328 \cdot 10^{7}$$

$$DOC_{maintenance} := DOC_{MaintAFMaterial} + DOC_{MaintAFPersonel} + DOC_{eng} = 9.267 \cdot 10^{6}$$

$$COC_{total} := DOC_{fuel} + DOC_{MaintAF} + DOC_{eng} + DOC_{crew} + DOC_{landing} + DOC_{navigation} + DOC_{ground} = 4.949 \cdot 10^{7}$$

EUR in $!

$$cf_{COC} := \frac{113000}{50000 + 6000 + 10000 + 15000 + 3000 + 6000 + 15000} = 1.076$$

$$COC := cf_{COC} \cdot COC_{total} = 53261836.977$$

$$DOC_{total} := COC + DOC_{capital} = 75088533.255$$

$$DOC_{perFlight} := \frac{COC}{FC} + \frac{DOC_{capital}}{FC} = 116597.101$$

Values for summary form

$N_{pax} := 424$

$$DOC_{fuel\_per100passkm} := DOC_{fuel} \cdot \frac{1}{FC} \cdot \frac{100}{R \cdot 1.852} \cdot \frac{1}{N_{pax}} = 0.505$$

$$DOC_{maintenance\_per100passkm} := DOC_{maintenance} \cdot \frac{1}{FC} \cdot \frac{100}{R \cdot 1.852} \cdot \frac{1}{N_{pax}} = 0.247$$

$$DOC_{fees\_per100passkm} := DOC_{fees} \cdot \frac{1}{FC} \cdot \frac{100}{R \cdot 1.852} \cdot \frac{1}{N_{pax}} = 0.354$$

$$DOC_{personal\_per100passkm} := DOC_{crew} \cdot \frac{1}{FC} \cdot \frac{100}{R \cdot 1.852} \cdot \frac{1}{N_{pax}} = 0.219$$

$$DOC_{capital\_per100passkm} := DOC_{capital} \cdot \frac{1}{FC} \cdot \frac{100}{R \cdot 1.852} \cdot \frac{1}{N_{pax}} = 0.583$$

$$DOC_{total\_per100passkm} := DOC_{total} \cdot \frac{1}{FC} \cdot \frac{100}{R \cdot 1.852} \cdot \frac{1}{N_{pax}} = 2.004$$