

A MULTIDISCIPLINARY PROCESS FOR INTEGRATED ROTORCRAFT DESIGN

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Abstract

This paper presents the activities of the German Aerospace Center (DLR) on integrated rotorcraft design. The process and design environment presented here are the results of two DLR internal projects carried out by the Institute of Flight Systems, the Institute of Aerodynamics and Flow Technology and the Institute of Structures and Design. The fundamental features of this process are distributed computation on the servers of the different institutes, analysis tools with increasing physical fidelity along the progress of the design process and a high modularity inside the software framework. The tools cover the phases of conceptual and largely preliminary design. The design process is initialized by a statistical concept study, providing the initial configuration for the following sizing and optimization task. The methods of the tools applied range from blade element theory over vortex panel theory, to finite element methods for structural sizing. This design toolbox is the foundation necessary for the DLR to tackle upcoming research tasks dealing with future unconventional rotorcraft configurations.

NOTATIONS

Symbols

a	speed of sound, m/s
C_T	thrust coefficient, -
$C_{l\alpha}$	lift curve slope, -
c_{MR}	main rotor blade chord length, m
E_{rot}	rotational energy, J
E_{trans}	translational energy, J
g	gravitational acceleration, m/s ²
J_β	blade flap moment of inertia, kg·m ²
J_c	blade lag moment of inertia, kg·m ²
l_{fus}	fuselage length, m
m_{BEM}	basic empty mass, kg
m_F	fuel mass, kg
m_{MTOM}	maximum take-off mass, kg
m_{OEM}	operating empty mass, kg
m_{OM}	operators mass, kg
m_p	payload mass, kg
m_{bl}	mass of rotor blade, kg
m_{fus}	fuselage structural mass, kg
m_{propu}	propulsion mass inclusive drivetrain, kg
m_{struc}	structural mass, kg
m_{syst}	systems mass, kg
$Ma_{tip,MR}$	main rotor tip Mach number, -
$N_{bl,MR}$	number of blades per main rotor, -
$N_{rot,MR}$	number of main rotors, -
P_{ind}	induced power, W

P_{inst}	maximum installed power, W
R_{MR}	main rotor blade radius, m
s_{RNG}	flight range, m
s_{shaft}	rotor shaft spacing, m
T_{MR}	main rotor thrust force, N
v_h	horizontal flight speed, m/s
$v_{tip,MR}$	main rotor tip speed, m/s
γ_{MR}	main rotor Lock number, -
κ_{ov}	overlapping factor, -
κ_{cutout}	cut-out ratio of the rotor blade, -
μ	advance ratio, -
Λ_{MR}	main rotor blade aspect ratio, -
ρ	air density, kg/m ³
σ_{MR}	rotor density of main rotor, -
Ω_{MR}	main rotor rotational speed, rad/s

Abbreviations

AFDD	U.S. Army Aeroflightdynamics Directorate
BEM	Basic empty mass
CPACS	Common Parametric Aircraft Configuration Scheme
DLR	German Aerospace Center (Deutsches Zentrum für Luft- und Raumfahrt)
EDEN	Evaluation and Design of Novel Rotorcraft Concepts
FEM	Finite Element Method
FSD	Fully stressed design
HOST	Helicopter Overall Simulation Tool
MDO	Multidisciplinary Design and Optimization
MTOM	Maximum take-off mass
OEM	Operating empty mass
RCE	Remote Component Environment
RIDE	Rotorcraft Integrated Design and Evaluation
TLAR	Top Level Aircraft Requirement

1 INTRODUCTION

The integrated design of aerial vehicles is one of the most complex engineering processes. In this context the design of rotorcraft is even more complex than the design of fixed wing aircraft for general aviation and commercial transport. The sizing process of rotorcraft is very sensitive to small variations of input parameters resulting in an abrupt change of the overall design. The impact of uncertainties on matching the Top Level Aircraft Requirements (TLARs) during the different phases of design with different levels of computation is very crucial.

The goal of a design project for aerial vehicles is to increase the accuracy of the virtual configuration with every step in the sizing process. Therefore, a suitable arrangement of the design phases was introduced in the traditional engineering process. In the majority of the references, the engineering process is divided into three design phases; as specified by Raymer [1] or Nicolai [2] and shown FIG 1.

- The conceptual phase performs the sizing and optimization task for a consistent external configuration. A consistent configuration balances the external dimensions with the mass breakdown and TLARs.
- The preliminary design focuses on the internal arrangement and the solution of all aerodynamic problems. The result is a complete definition of the outer hull of the vehicle as well as the component arrangement and an evaluation of the load cases.
- The detailed design proceeds which fits the correct construction into the outer hull. Fatigue, vibration and crash problems have to be solved. The prototype is usually a part of the detailed design. This implies an enormous increase of the required personal resources. The results of this phase are all drawings and information required for industrial production.

The three phases typically consist of a concept or design study at the beginning of the conceptual design phase and a proposal at the end of the preliminary design phase. The design study points out the possibilities for future projects and initiates the process. The proposal is the last point that the decision whether to proceed with the project or to save the conclusions and continue a different project can be made. Sometimes this process is extended to five phases, see Layton [3].

In the traditional approach, the configuration is frozen after each design phase. Modifications to the design were only made within exactly defined parameters. For example, more sophisticated aerodynamic calculations are performed in the preliminary design resulting in the final airfoil and twist distribution of the rotor blades. A change in the rotor radius is not allowed without going back to the conceptual design phase. Until the early 2000's unfreezing and returning to the conceptual design was not considered as an option. Nevertheless, the complex interactions of the rotorcraft components, like rotors, fuselage, wings, stabilizers, etc. exhibit an

inherent danger of design changes due to uncertainties in the conceptual phase. If information from the preliminary design were available in the conceptual design, sizing decisions could be different. Therefore, the conceptual design in terms of configuration aerodynamics and sizing is still strongly knowledge driven and limits the design of unconventional configurations.

Phase	Results	Grade of definition
Conceptual Design	geometry basic 3-Side View	65%
	mass and balance centre of gravity component masses	
	flight performance drag levels & performance	
	cost factors & benefits	
Preliminary Design	basic internal arrangement component positions and volume demand	85%
	complete external configuration airfoil and blade twist local flow problems solved	
	major loads, stresses & deflections load cases are solved	
Detailed Design	detail design mechanisms joints, fittings & attachments damage assessment	100%
	prototype flight tests	
	design refinements as result of tests & operation	

FIG 1 Three phase design process applied to rotorcraft proposed by Raymer [1] and Nicolai [2] and the corresponding grades of definition

The timeframe of the first two design phases in relation to the complete product lifecycle is rather short. FIG 2 shows the impact of the grade of definition of a design and consequently the impact on the lifecycle costs according to Roskam [4].

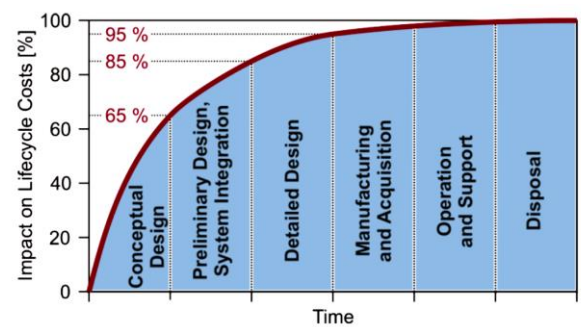


FIG 2 Impact on lifecycle costs over timeline according to Roskam [1]

Even though the time covered by conceptual and preliminary design is short, the impact on the design is massive. By processing the conceptual phase almost two thirds of the design and its resulting costs are determined. By completing the preliminary design and freezing the external configuration for the detailed design the grade of definition goes up to 85%. This demonstrates the need for new design chains that combine as many multidisciplinary design and optimization (MDO) tools as possible into one process.

On the basis of the demand for new rotorcraft configurations several research organizations have recently presented different approaches to integrated design. Well known is NDARC by Johnson [5] which is the center of many design projects. For instance, Lawrence [6] showed the use of NDARC for resizing with respect to handling qualities in conceptual design. Basset [7] presented the CREATION project which is the approach of the ONERA. In 2011 the DLR started its own activities on rotorcraft design. Starting with project RIDE (Rotorcraft Integrated Design and Evaluation, 2011 to 2013) followed by project EDEN (Evaluation and DEsign of Novel rotorcraft concepts, 2014 to 2017) a multidisciplinary toolbox was developed by different specialists within DLR. These projects mark the first collaboration on rotorcraft design by the Institute of Flight Systems, the Institute of Aerodynamics and Flow Technology, both in Braunschweig, and the Institute of Structures and Design in Stuttgart. The objective of this project team was to generate a multidisciplinary design process in order to size different configurations by starting from scratch with only mission requirements. This paper presents the basic thoughts and new approaches of the multidisciplinary rotorcraft design environment developed by DLR.

2 VIRTUAL DESIGN ENVIRONMENT

The design tasks discussed in this paper cover the phase of conceptual and partly preliminary design. Most problems of system integration, detailed design, building of wind tunnel models and prototypes are beyond the scope of this project team. In the following paragraph some general agreements are explained bordering the design process.

2.1 Design Process

In the first project (RIDE) a process chain was developed including tools for conceptual design and applying distributed computation on the servers at different locations. The focus was put on the arrangement of the tool chain and the data exchange in the process. This first project only covered the conventional main rotor / tail rotor configuration. The knowledge gathered in this project was the basis for the development of the process to a multidisciplinary and multi configuration design toolbox. Within the following project (EDEN) the tools were extended by preliminary design level tools including flight mechanic simulation, applying complete trim and performance calculation. The possible configurations of the design process were extended. FIG 3 shows the structure tree for the

development of rotorcraft configurations aiming at the field of high speed rotorcraft. Besides helicopters, the sum of rotorcraft also includes gyrocopters and convertible rotorcraft with tilting discs, which are not covered by this process. In addition to the standard configuration, with one main rotor and tail rotor for compensation of the torque, helicopters with more than one main rotor are considered. Here the tandem configuration features a partly overlapping area of the rotor discs with possibly intermeshing discs and the coaxial configuration where the discs are vertically separated and completely overlapping. Other types of helicopters like the intermeshing or Flettner configuration as well as the side-by-side configuration are mentioned here but are not included into the process. By adding compound components like a thrust generator and wings to the helicopter, every helicopter can be extended to a compound configuration in order to achieve higher flight speeds.

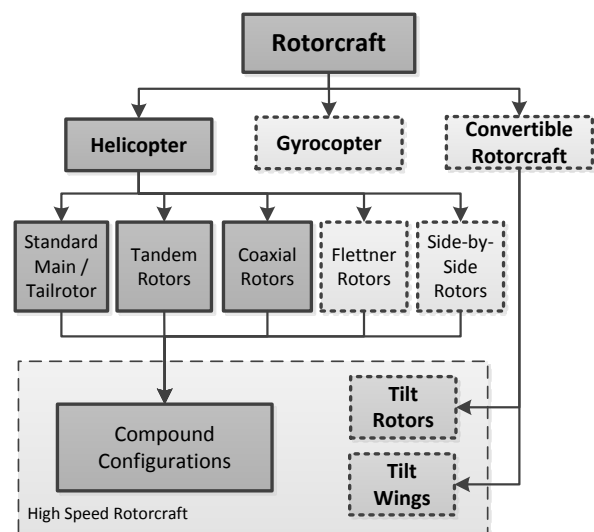


FIG 3 Hierarchical structure of rotorcraft development to high speed configurations

The overall process is outlined in FIG 4. The process starts with the initialization of the dataset. The arrangement of the helicopter components for this initial configuration follows the possible configurations marked in the rotorcraft tree, see FIG 3. For the primary design loop, a traditional approach is used by iterating the maximum take-off mass (MTOM). The loop consists of a design leg and an analysis leg. The design leg performs the scaling and optimizing of the external configuration according to the TLARs and engineering boundaries like aerodynamic and flight mechanical limitations, safety, etc. The analysis leg performs the recalculation of the sizing input by predicting the MTOM. In the first step, the fuel mass is calculated by a flight mechanic simulation. In the next step, the basic empty mass is calculated by estimating the individual component masses. If the MTOM is not in close margin to the MTOM from the previous step the loop is repeated. The result of a converged sizing loop is a consistent external configuration taking into account the level of applied physical modelling.

After completing the conceptual design, the preliminary design focuses on the internal configuration, higher order flow problems and a larger amount of flight conditions. The calculations in this phase are more complex and require more computation time. In the ideal case the results of these tools do not require an adaption of the external configuration. Hence, the design is frozen after the conceptual phase in the traditional process. In practice, there has always been some influence on external configuration by preliminary design. Typically this influence was given by higher order aerodynamic calculations. However major changes were neglected and the uncertainties of the lower order tools were bypassed by knowledge based sizing approaches. For future configurations with more complex interactions, a lack of empirical knowledge for non-existing designs and more sophisticated MDO approaches a possible unfreezing and resizing of the external configuration is required.

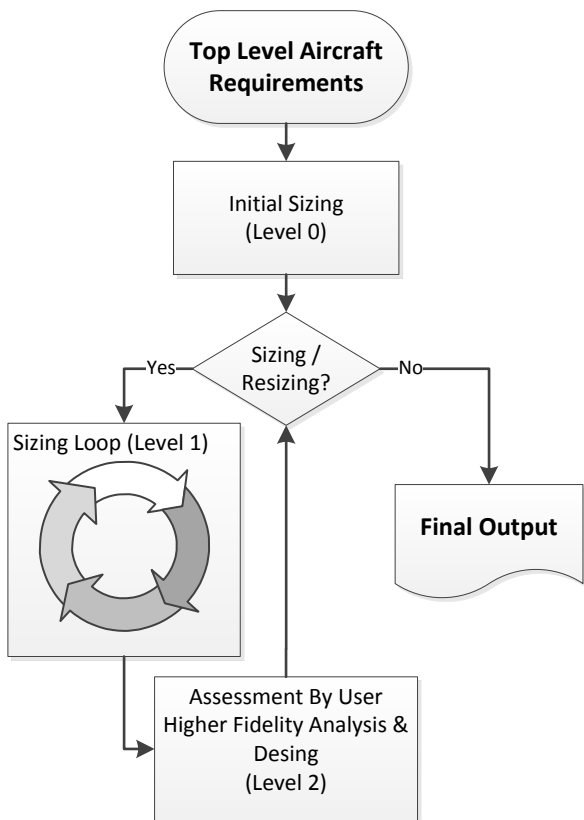


FIG 4 Flowchart of the main functions of the virtual design environment

On this account the user needs a breakpoint where he has to decide whether to repeat the sizing process with updated parameters or proceed with the latest dataset.

2.2 Tool classification

The Rotorcraft Virtual Engineering Conference 2016 highlighted the similarities and common problems of different engineering teams dealing with rotorcraft design. The greatest harmony in the different approaches for virtual engineering of rotorcraft was

achieved with the classification of the design tools (see Ries und Schimke [8], Sinsay [9], Basset [10] and Weiland [11]). There will never be an integrated process taking all tools with all levels of accuracy into one loop. A staging of the design phases and tool levels is the consequence.

The tools integrated into this design environment have different grades of physical modelling, required input and computation time. To perform a suitable arrangement of the tools a classification is made, dividing the tools into four levels ranging from 0 to 3. The principle of the tool levels is shown in FIG 5 in conjunction with the computation time and the uncertainties of the physical modelling.

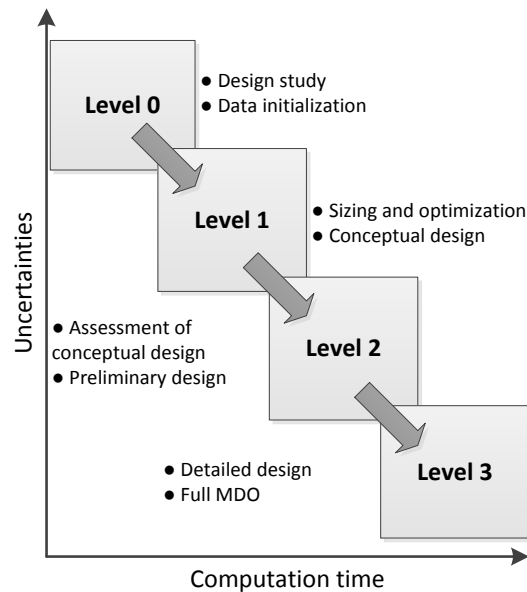


FIG 5 Uncertainties of the design tool as a function of the computation time

The tools classified as Level 0 start with the TLARs. They initialize the dataset and deliver the input for the primary sizing loop using empirical methods and featuring simple physical models. The computation time is certainly low, but the uncertainties at this level are the highest. Level 1 tools are used for highly iterative procedures like scaling and optimization in the primary sizing loop. They are completely based on physical models with only small empirical corrections. Their computation time is still fast enough for iterative sizing but their uncertainties are small enough to deliver consistent design results for conventional standard configurations without considering more sophisticated interaction problems. Level 2 tools have a higher order physical modelling. The resulting computation time is too long for an implementation in a conceptual sizing loop but their pre- and post-processing procedures are fully compatible with the data framework of the conceptual sizing. They are introduced in the preliminary design phase and arranged in a queue without performing recursive loops. The need for

repeated executions is controlled by the user. Level 3 tools have the most accurate physical modelling but either their computation time violates the timeframe or their pre- and post-processing procedures cannot be performed automatically. Numerical methods including complex discretisation are common methods in this group. These tools are most often used in the end phase of preliminary design and in the detailed design and are not covered by this design environment.

The design approaches from Sinsay [9], Basset [10] and Weiland [11] show a highly comparable classification of the tool levels 1 to 3. The works of DLR and ONERA even show a group of Level 0 tools, which are excluded from the level 1 group and serve as a fast initialization of the data set. Nevertheless, the three organisations use different infrastructures and software frameworks resulting in recognizable differences of borders between the tool levels. For example, the implementation of flight mechanic simulations including model building in a sizing loop, as well as distributed computing result in different boundaries for the tool arrangement.

2.3 Distributed Computation

There are two possible ways to integrate a higher variety of tool levels into one process. The first is a collection of all tools on one central server. The access for the computation will be easy, but the maintenance by the responsible specialist is difficult. The second way is to apply distributed computation on the servers of the individual specialist and integrate the overall toolbox into a network based framework. For this purpose DLR uses its in-house development Remote Component Environment (RCE). RCE is a program for collaboration in design processes allowing specialists at different locations to publish their tools to authorized partners (see Bachmann [12], Litz [13] and Seider [14]). The tools are located on different servers, where further development and maintenance takes place, data transfer is ensured via secure connection. Users at different locations can run the tools but can neither open nor edit the program code.

2.4 Data model and exchange format

A requirement for efficiently combining different disciplines and computing at different locations is a uniform data structure within the design environment. Therefore, DLR developed the CPACS data model (Common Parametric Aircraft Configuration Scheme) and refined it for integrated design of fixed wing aircraft (see Bachmann [15], Liersch [16] and Nagel [17]). With experience from aircraft design, this XML data structure was extended to rotorcraft design. The data model features a strong hierarchical structure that allows the storage of the complete vehicle definition in one file. The uniform exchange format together with RCE enables the full toolbox characteristics of the rotorcraft design environment. For every tool, input and output files are identical. Individual tools can easily be replaced and workflows can be rearranged without massive reprogramming.

2.5 Flight mechanic simulation

One other very important external resource is the flight mechanical simulation tool HOST (Helicopter Overall Simulation Tool). HOST was developed by Eurocopter at the very beginning of the 1990's. It is generally based on three main functions: trim calculation, time domain simulation and the calculation of linear equivalent system (see Benoit [18]).

As part of the DLR tool chain, the trim calculation is used for the flight mechanical simulation. All required input values for a simulation are contained in a CPACS file. For a HOST calculation of a typical helicopter configuration they are, for example: masses, inertia, rotor characteristics or aerodynamic polar of fuselage and stabilizers. Out of this file the values are written in Python scripts. These scripts generate the required HOST modules in order to initialize a trim calculation. Each file presents a certain helicopter component. For a simple conventional helicopter configuration these modules consist of a main-rotor file, tail-rotor file, fuselage cell file, swashplate and transmission file, blade file, aerodynamic files of the fuselage and stabilizers and a general HOST file that merges all other files. After a calculation the CPACS file is updated with the results. By integrating HOST into the CPACS framework, it is possible to perform iteration calculations with HOST during the tool chain.

HOST additionally allows the calculation of rotor-rotor interaction. All supported main rotor configurations contain this interaction. Models of super positioned induced velocity fields were developed and applied with HOST for this reason. The first calculation is performed without mutual rotor interaction and the results of the velocity field will be used as an overlay to perform a second trim calculation with the combined overlay (see Krenik [19]). The factors for the induced velocity increase were derived from Johnson [20].

FIG 6 shows the induced velocities during hover and forward flight of a coaxial configuration. In the hover case the lower rotor is influenced by the full downwash of the upper rotor. It can be seen as a concentric disc (red). During forward flight the downwash shifts back downstream. Thereby the influenced area of the lower rotor gets reduced. This rotor-rotor interaction calculation can be performed for any arbitrary configuration in the design process. For simple trim and performance calculations HOST can be considered as a level 1 tool. For more complex simulations and for a higher amount of flight conditions it has to be considered as a level 2 tool.

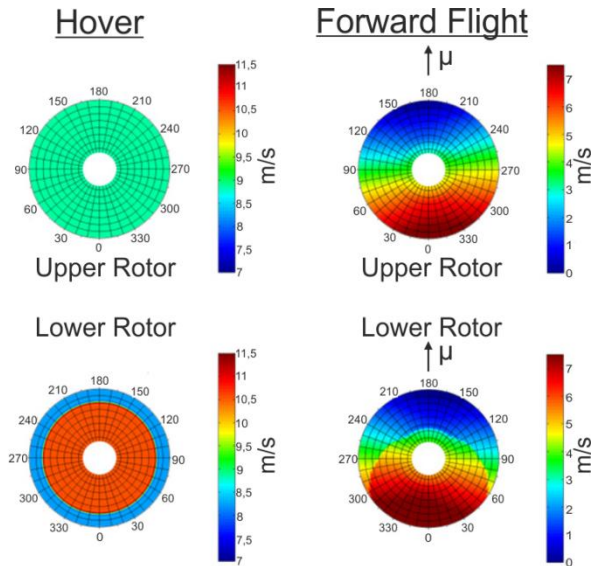


FIG 6 Induced velocity field during hover (left) and forward flight (right) at $\Omega=424$ rpm, $\mu=0.08$, $m_{MTOM}=2200$ kg for the coaxial configuration

3 CONCEPTUAL DESIGN

The objective of the conceptual design phase is to create the external configuration of the vehicle. In the following paragraph the steps from the initialization of the dataset by level 0 tools, up to the sizing and optimization by level 1 tools is explained, including a small design study.

3.1 Initialization

3.1.1 Take-off mass and fuselage dimensions

The objective of the initialization is to create the first data set depending on the TLARs demanded by the user. The tools applied in this step belong to the level 0 group. The result of this initial step can be considered a rough concept study. The fundamental characteristic of this computation is to produce as much output as possible with as little input as possible. This happens in a strict top down approach starting with the overall mass fractions given by equation(1).

$$(1) \quad m_{MTOM} = m_{BEM} + m_{OM} + m_P + m_F$$

The payload mass m_P is already prescribed by the TLARs. The initial basic empty mass m_{BEM} and the initial fuel mass m_F are estimated using statistical databases. The statistical background for this initial sizing is a database of 159 existing helicopters. The basic empty mass and the operator's mass m_{OM} form the operating empty mass m_{OEM} . The initial take-off mass m_{MTOM} can now be estimated by equation(1).

The minimum TLARs required to start the initialization are given in TAB 1. Payload, cruise speed and range span the basic triangle of transport performance.

Name	Type of parameter	Unit
payload mass	continuous	kg
cruise speed	continuous	m/s
range	continuous	m
number of main rotor blades	discrete	-
main rotor configuration	selection	-

TAB 1 Minimum TLARs to start the process

The basic concept of the rotor arrangement has to be specified. For a full automated initial sizing, the options for a standard main rotor / tail rotor arrangement are available as well as the coaxial and tandem rotor arrangement. FIG 7 shows one exemplary 3D output for each of the three configurations.

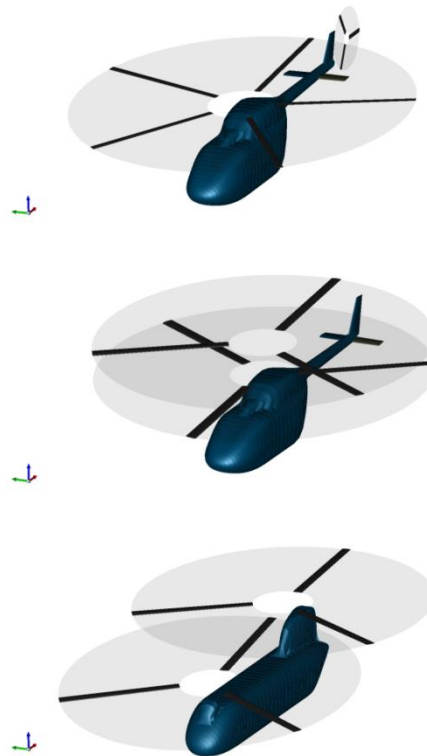


FIG 7 Three examples for the baseline configurations (above the standard configuration, middle the coaxial configuration and below the tandem configuration)

Experience has shown that the number of blades per main rotor is an important discrete design parameter. On one hand the number of blades is a big cost factor and on the other hand the number has to be considered wisely with respect to rotor dynamics, vibrations and the implementation of new technologies like higher harmonic control. In addition to the minimum required TLARs given in TAB 1, optional requirements may be added. The most important additional TLARs are the length, width and height of the cabin with respect to the inner fuselage, as given in TAB 2.

Name	Type of parameter	Unit
cabin height	continuous	m
cabin width	continuous	m
cabin length	continuous	m
cargo hold payload fraction	continuous	-

TAB 2 Optional TLARs which are not (absolutely) necessary for the initial sizing

Another important parameter is the cargo hold payload fraction. In case of a required stern ramp the rear part of the fuselage will not carry any fraction of the payload. If no stern ramp is required, the rear fuselage carries a certain percentage of the overall payload. From experience, this value goes up to 20% and influences the mass and balance considerations of the fuselage arrangement under the rotors. If the dimensions of the cabin are not specified by the user, the fuselage is sized by statistics. The most problematic step in statistical sizing of aerial vehicles is to find a reliable relation between the mass fraction and the spatial measurement. FIG 8 shows the statistics of the fuselage length over the maximum take-off mass with a suitable regression curve.

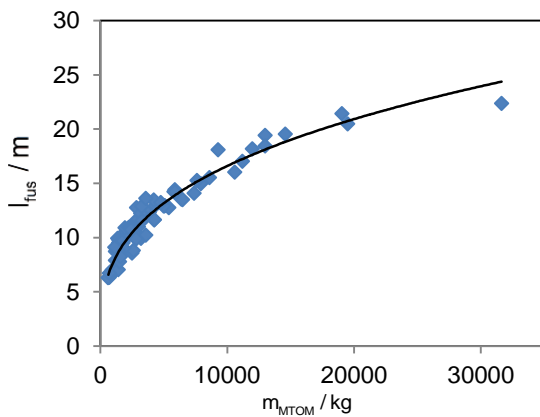


FIG 8 Statistics of the fuselage length over the maximum take-off mass with a regression curve

The concept of a generic fuselage is applied. Here the fuselage is divided into several parts in order to individually scale aside the junction of the parts. FIG 9 shows the principle separation of the fuselage into several parts.

The starting point is the cross section of the cabin which extends the cabin length resulting in the inner fuselage given either by user input or empirical estimation. Additional space for structural components and equipment has to be considered on the sides as well as on the upper and lower part. The initial dimensions are scaled by using design parameters. These values can be reconsidered for more unconventional configurations. The length of the cockpit and the rear fuselage is determined by the equipment, cargo loading and drag boundaries. Retractable landing gears as well as higher performance weather radars require more construction volume in the nose. Cargo loading and

unloading over a stern ramp requires specific dimensions of the rear part which are opposed by considerations for drag reduction due to flow separation on the back of the ramp. For instance, the default ratio for the rear part length including a stern ramp is 1.25 times the height of the center fuselage part. The front, center and rear part form the fuselage body. The body is now arranged under the rotors with respect to mass and balance considerations. The most suitable arrangement is with the center of payload mass at the center of operating empty mass. In case of a standard and coaxial configuration, the length of the tail boom is given by the distance between fuselage body and the tail plane. In case of a tandem configuration, the position of the fuselage front and rear part is given by the position of the rotors. The overlapping ratio of the rotors is usually increased up to about two third of the rotor radius. Nevertheless, lengthening of the cabin might become necessary to fit the fuselage body under the rotors, resulting in a higher payload volume with constant payload mass.

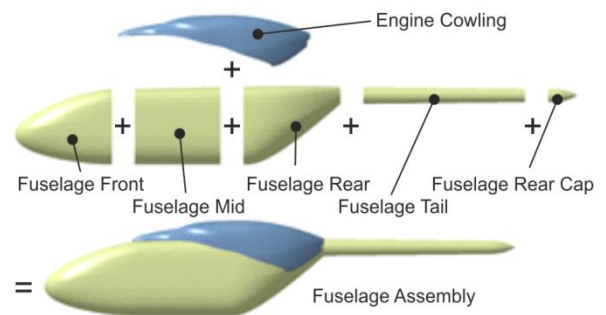


FIG 9 Components of a generic fuselage model for the assembly of the 3D-model

3.1.2 Rotor sizing

During the level 0 initialization, typically no component masses are required along the general mass fractions. With respect to the flight mechanic simulation in the sizing loop, the mass and moment of inertia of the rotor blades are a prerequisite and need to be available at the beginning of the loop. A reliable approach is a backward calculation. This is achieved by starting with reasonable aeromechanical similarities. Two important parameters defining the flight properties of main rotors are the Lock number given by equation (2) and the hinge off-set.

$$(2) \quad \gamma_{MR} = \frac{Q \cdot C_{l\alpha} \cdot C_{MR} \cdot R_{MR}^4}{J_{\beta}}$$

The Lock number represents the ratio of the gyroscopic forces to the inertia forces. Reasonable values for the initial rotor geometry, considering aspect ratio and stall offset, are given by equations (3) and (4). The general definition of the rotor density is given with equation (5) connecting radius, chord and number of blades.

$$(3) \quad R_{MR} = 0,226 \cdot m_{MTO}^{0,392}$$

$$(4) \quad \sigma_{MR} = 0,012 \cdot m_{MTOM}^{0,221}$$

$$(5) \quad \sigma_{MR} = \frac{c_{MR} \cdot N_{bl,MR}}{\pi \cdot R_{MR}}$$

The values for equation (2) can range from 4 to 10 and increase with decreasing blade mass. Older articulated rotors show values from 4 to 6. Modern hingeless rotors consist of composite materials and show values of clearly more than 6. Autorotation capabilities require a certain amount of inertia for the main rotor to avoid strong deceleration and stalling of the blades during transition to autorotation. Also modern coaxial lift off-set rotors with high stiffness can have Lock numbers of about 5.3 (see Johnson [21]). However, these rotors must proof a limited flapping motion between the two counter rotating discs. Values of around 10 are reasonable for high agility combat helicopters. From equation (2) the moment of inertia is given by equation (6).

$$(6) \quad J_{\beta} = \frac{Q \cdot C_{l\alpha} \cdot c_{MR} \cdot R_{MR}^4}{\gamma_{MR}}$$

According to the data given by van der Wall [22], a BO105 main rotor blade has a radius of 4.91 m with a mean chord of 0.27 m and a Lock number of 8.159. The lift slope is given by 2π . By assuming standard ambient conditions, equation (6) gives a moment of inertia of 148 kgm² which is 9.7 % higher than the reference data. These results are reasonable in order to build up a flight mechanic model directly from a concept study. The default Lock number is 6 but the value for modern rescue helicopters around 3 t m_{MTOM} ranges from 7 to 8.5. In order to make a reasonable trim and performance calculation, these design parameters can be adapted for different types of rotorcraft configurations.

3.1.3 Power estimation

The prediction of the performance for the initial engine sizing is difficult in this early phase of design. The uncertainties of estimating the installed power from the available data are large. A semi-empirical approach is very functional. Equation (7) calculates the ratio of the installed and ideal induced power of a rotorcraft in hover. FIG 10 shows some exemplary values for existing helicopters.

$$(7) \quad \frac{P_{inst}}{P_i} = \frac{P_{inst}}{\sqrt{\frac{(m_{MTOM} \cdot g)^3}{2 \cdot Q \cdot \pi \cdot R_{MR}^2}}}$$

The installed power includes the 30 sec emergency power in case one engine is inoperative. The power ratings for take-off and maximum continuous are respectively lower. Most values range from 1.7 to 2.7 with modern configurations laying around 2.4. The S-97 (3.44) is a compound configuration with extended flight speed and higher performance. Special requirements

by operators like emergency cases for the EC135 (3.57) and survivability aspects for the Tiger (3.79) bring the values out of this frame. Considering such special configurations and boundaries the design parameters might need to be adapted.

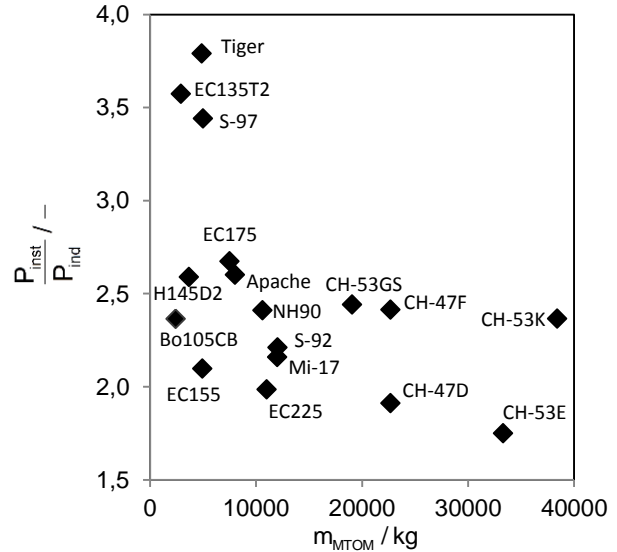


FIG 10 Ratio of the installed and induced power in hover over MTOM for existing helicopters

3.2 Sizing loop

The primary loop consists of a design leg and an analysis leg. FIG 11 shows the principle arrangement of the level 1 tools for conceptual design. The initial dataset from level 0 pre-sizing is now the basis for the optimization of the rotor plane. On the basis of a more precise sizing and positioning of the rotors, a 3D model is generated and the aerodynamic polars of the fuselage are calculated. Additional sizing tools can be integrated here. Proceeding with the analysis, the required fuel mass is calculated for trimmed flight conditions and the OEM is predicted by component mass estimation. The loop iterates the MTOM until convergence is reached. The loop shown in FIG 11 is the core component of the design environment for the sizing of the level 0 concept study data, as well as the resizing of returning data from level 2 calculations.

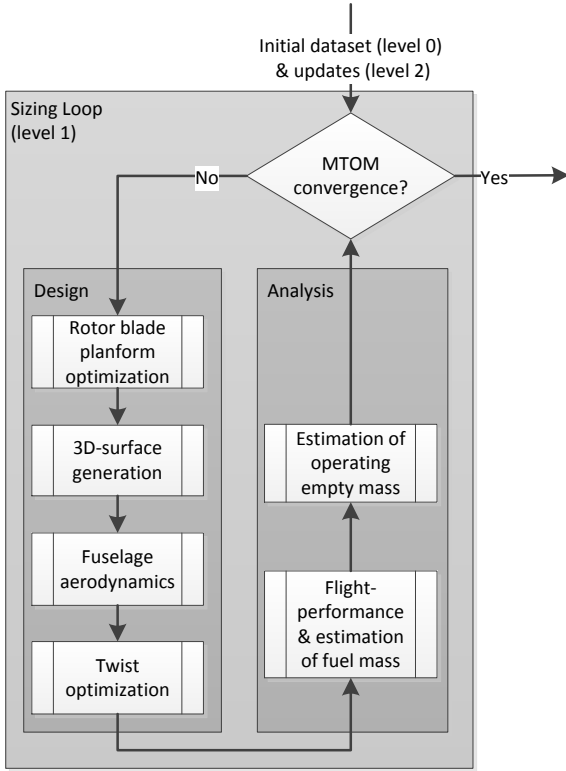


FIG 11 Flowchart of the sizing loop applying level 1 tools for the conceptual design

3.2.1 Rotor blade planform optimization

The optimization of the planform is a knowledge based procedure which is described in detail by Krenik [19]. To optimize the radius, chord length and angular speed of the rotor, a series of characteristic rotor parameters are calculated and used as optimization constraints. RCE as a software framework already provides the COBYLA (Constrained Optimization BY Linear Approximation) algorithm from Powell [23]. The characteristic parameters are shown in equations (8) to (14). Here, the aspect ratio and rotor density equations (8) and (9) give basic information about the rotor and blade planform geometry. The aspect ratio ranges from 12 to 25 and the rotor density from 0.03 to 0.15.

$$(8) \quad \Lambda_{MR} = \frac{C_{MR}}{R_{MR}}$$

$$(9) \quad \sigma_{MR} = \frac{N_{bl,MR} \cdot C_{MR}}{\pi \cdot R_{MR}}$$

The influence of the blade loading on the rotorcraft, given by equation (10), is crucial. With increasing flight altitude, the density is decreasing, resulting in an increase of the blade loading. Values of 0.12 and more indicate a first flow separation somewhere on the rotor disc. Therefore, values between 0.08 and 0.09 at sea-level are good for most rotorcraft. For special hot and high considerations, with low air density, higher rotor solidity and consequently a lower blade loading is

reasonable. Low values of the blade loading imply a high parasite drag and lead to a bad Figure of Merit.

$$(10) \quad \left(\frac{C_T}{\sigma} \right)_{MR} = \frac{T_{MR}}{Q \cdot v_{tip,MR}^2 \cdot N_{bl,MR} \cdot C_{MR} \cdot R_{MR}}$$

The general speed characteristics of the rotor are considered by equations (11) to (13). The cruise tip Mach number given with equation (11) represents the state of the art transonic blade aerodynamics for the advancing side of the rotor. It may range from 0.82 to 0.88.

$$(11) \quad Ma_{tip,cruise,MR} = \frac{v_{tip,MR} + v_{h,cruise}}{a}$$

The advance ratio given by equation (12) represents the capability of dealing with reverse flow on the retreating side of the rotor. For conventional twist and airfoil distribution it should not exceed 0.35.

$$(12) \quad \mu = \frac{v_h}{v_{tip,MR}}$$

The tip speed in hover given by equation (13) usually ranges from 210m/s to 225m/s.

$$(13) \quad v_{tip,MR} = \Omega_{MR} \cdot R_{MR}$$

Equation (14) takes into account the autorotation capabilities of the configuration. The energy ratio has shown to be a suitable design parameter. In equation (14), the ratio of the rotational energy of all rotors over the translational energy of the complete vehicle is calculated for the case of vertical autorotation (see Krenik [19]).

$$(14) \quad \frac{E_{rot}}{E_{trans}} = \frac{N_{rot} \cdot N_{bl} \cdot J_c \cdot \Omega_{MR}^2}{1.62 \cdot \frac{m_{MTOM} \cdot g}{Q \cdot \kappa_{ov} \cdot \pi \cdot R_{MR}^2}}$$

Values for the energy ratio may range from 1 to 3. Helicopters like the UH-1 with values higher than 2.5 show good autorotation capabilities. Suitable values for recent helicopters are between 1.2 and 2.0. For tandem and coaxial configuration, the shaft spacing has to be considered. An overlapping factor has to be calculated according to Johnson [20] by means of equation (15) and (16). For configurations with one main rotor, the factor is 1. In case of the coaxial configuration with full overlapping discs the factor is $\sqrt{2}$.

$$(15) \quad \kappa_{ov} = \sqrt{\frac{2}{2-m}}$$

$$(16) \quad m = \frac{2}{\pi} \left[\arccos \left(\frac{s_{shaft}}{2 \cdot R_{MR}} \right) - \frac{s_{shaft}}{2 \cdot R_{MR}} \sqrt{1 - \left(\frac{s_{shaft}}{2 \cdot R_{MR}} \right)^2} \right]$$

The Lock number, given by equation (2) is the last constraint. As mentioned before, it may vary with respect to flapping behavior and autorotation. This boundary has to be set carefully. Setting a target value is also a credible approach.

The default range of values given here has to be considered as a generalization of contemporary rotorcraft. For more sophisticated and unconventional designs these values must be reconsidered with caution.

3.2.2 3D-surface generation

The generation of a three dimensional surface is performed using a complex approach by Kunze [24]. This geometry generation module is connected to the commercial software CATIA V5 via its programming interface to build a CAD model of the helicopter fuselage.

For the generation of a CAD model, as shown in FIG 12 for the standard configuration, the fuselage is divided into individual segments. The model is sized by the individual segments shown in FIG 9. For each segment multiple templates are available which can be assembled to build a complete fuselage. Each segment is parameterized to allow a high flexibility for the sizing and design of the fuselage.

During the conceptual design process no detailed knowledge of the fuselage design is available, thus generic fuselage models are implemented into the geometry generation module. These generic models contain all the required parameters to create a helicopter fuselage and to get a first impression of the fuselage. With this fuselage, an estimation of its aerodynamic properties can be obtained.

Generic fuselage templates are currently available for standard, coaxial and tandem configurations. These templates can be adapted to the current helicopter design by the definition of its length, width and height. Optionally the fuselage cabin can be modified by the definition of segment lengths for the nose, cabin, and rear-segments.

3.2.3 Fuselage aerodynamics

After the fuselage geometry is generated, the aerodynamic properties of the fuselage are calculated by a module (see Kunze [24]) using the commercial software VSAERO (see Nathman [25]). VSAERO is based on the potential flow theory using an incompressible and inviscid approach. Compressibility corrections can be calculated by various methods, for instance, Prandtl-Glauert or Karman-Tsien. To account for viscous effects, the calculation is coupled with an integral boundary layer formulation which can also calculate boundary layer transition and separation. A sample result of the flow on a fuselage is shown in FIG 12 where the streamlines are colorized by the shape factor H so that the calculated locations of the transition and separation can be seen.

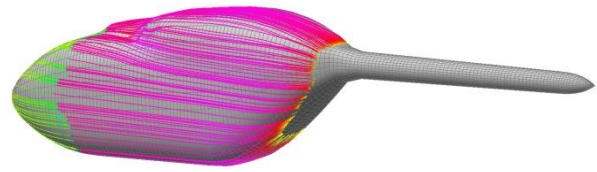


FIG 12 Example of streamlines with estimated transition line on the fuselage up to the separation line

Beginning with the CPACS (see Nagel [26] and Böhnke [27]) definition of the fuselage geometry, the aerodynamic module generates a structured surface mesh on the aerodynamic fuselage definition and invokes VSAERO for different flight conditions (e.g. angle of attack, sideslip angle, Mach and Re-number). Afterwards, a first calculation of lift and drag due to the pressure distribution is performed. A pressure equal to the average pressure at the separation points is assumed on the panels with separated flow to calculate a pressure drag. The combined results are written back to the CPACS file.

This module currently only calculates the aerodynamics of a clean fuselage without rotor hub, empennage, landing gears, or any attachments like holds or antennas. The drag generated by these parts must be calculated separately by generic models.

3.2.4 Twist optimization

The optimization of the blade twist is performed at the end of the design leg. For every optimization step, a trim calculation is conducted with HOST. The default flight condition for optimization is cruise speed with low level altitude, but can be altered by the user. A linear twist distribution is iterated beginning with the first blade section at the cut-out and ending at the blade tip. The cost function for this optimization is to minimize the required power for the considered flight condition.

3.2.5 Estimation of fuel mass

The fuel mass is calculated during the tool chain in a nested loop at the beginning of the analysis leg. The loop starts with a trim calculation performed by HOST considering steady flight condition. For every flight segment two calculations are performed. The first is at the beginning of the segment and the second at the end. The results are the mean fuel flow and the actual range in contrast to the required range.

The next step is to compare the actual range and the required range. The fuel mass is corrected in order to match the actual range with the required range. The updated fuel mass is written to the CPACS file. This iteration runs until the fuel mass converges and the required range is met.

3.2.6 Estimation of operating empty mass

Mass estimation is an essential part in the conceptual design stage since the mass of the aircraft determines and influences directly and indirectly other design aspects, e.g. rotor, fuel, flight performance, airframe, etc.. In general, at the early stage in the overall design process, the mass of components such as structure, systems and propulsion, can be estimated using a statistical approach that compares similar rotorcraft configurations. Depending on typical design parameters like fuselage surface or design gross weight, the subcomponent weights can be estimated and summed up to the MTOM by equation (1) where the OEM consists of the BEM (structure, propulsion, systems) and the operators mass, as broken down in equation (17).

$$(17) \quad m_{OEM} = m_{BEM} + m_{OM} = m_{struc} + m_{propu} + m_{syst} + m_{OM}$$

Therefore, statistical mass estimation methods have been integrated into the presented design process. Currently, the mass estimation according to the methods proposed by Beltramo [28], Palasis [29], Layton [3] and AFDD-models presented by Johnson [5] have been implemented. Johnson introduced the AFDD (U.S. Army Aeroflightdynamics Directorate) mass models that feature the highest coupling grade of geometric and performance characteristics. Moreover, his methods provide the use of so-called technology factors that can be used to scale component masses with respect to their compositions.

FIG 13 shows the statistical OEM estimations for a coaxial helicopter of medium class take-off mass. It can be observed that the four approaches differ by up to 30%.

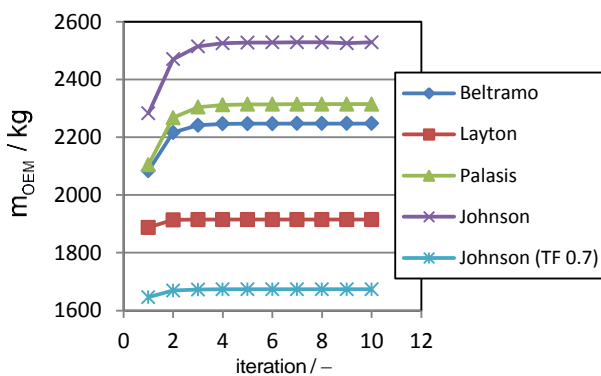


FIG 13 OEM over the computational iterations

The highest OEM is estimated by Johnson through using technology factors (TF) of 1.0 for all component masses. In the case that the technology factors are set to 0.7 (which is slightly less than done by Basset and Russel [30]) the estimated OEM drops significantly, resulting in a calibrated model deviation of less than 3%. Another reason for the highest mass estimated using the AFDD models is due to the estimation of the

systems mass. Some components such as furnishings, avionics, electrics, etc. are only specified as a range for medium weight class helicopters. Since helicopter design is strongly driven by functionality, a statistical approach for those systems is merely too difficult to realize and should be handled by providing a fixed input mass (as available) to avoid large deviations.

3.3 Examples for sizing of different configurations

The following section shows the results of three examples for different configurations. FIG 14 illustrates some parameters for converged sizing of the three configurations, which are the Mil Mi-8, the Boeing Vertrol CH-46 and the Kamov Ka-226. These are rather old helicopters but there is enough unclassified data available to use them as an example. The process was started with the minimum TLARs, listed in TAB 3 for every case, without any second level requirements and with all design parameters on default setting.

Parameter	Mi-8	CH-46	Ka-226
m_P / kg	4000	2757	860
v_h / m/s	225	241	190
S_{RNG} / km	425	356	555
$N_{bl,MR}$ / -	5	3	3
configuration	standard	tandem	coaxial

TAB 3 TLARs used as an input for the three example design studies

FIG 14 shows the deviations of several parameters from the reference. The reference values for FIG 14 are listed in TAB 4.

Parameter	Mi-8	CH-46	Ka-226
m_{MTOM} / kg	12000	9072	3400
m_{BEM} / kg	6800	5251	1952
m_F / kg	1200	1064	588
P_{inst} / kW	1790	1861	566
R_{MR} / m	10.65	7.62	6.5
c_{MR} / m	0.52	0.46	0.22
$v_{tip,MR}$ / m/s	214	211	193
$\left(\frac{C_T}{\sigma}\right)_{MR}$ / -	0.0757	0.0778	0.0856

TAB 4 Reference values of the three example configurations (Mi-8, CH-46 and Ka-226) for the case study

Here, the BEM of the standard configuration shows a good agreement with the reference. Other conventional main / tail rotor configuration also showed good results. As the prediction of the complete propulsion group including the drive train is difficult for a configuration with more than one main rotor, the component masses are underestimated.

Not all sources of drag are modelled adequately in the calculation of the aerodynamics and the predicted fuel. The estimated power is also too low.

The sizing of the rotor used the same default constraints for all three configurations. In general, a higher aspect ratio tendency can be observed. The calculated tip speeds are close to each other and near the reference values. The Mi-8 shows a slightly reduced blade loading. The configurations with more than one main rotor show this tendency to a greater extent. Due to the underestimated BEM, the optimizer needs to decrease the blade area more radically. The result is a quite low blade loading with very late flow separation but an adverse parasite drag.

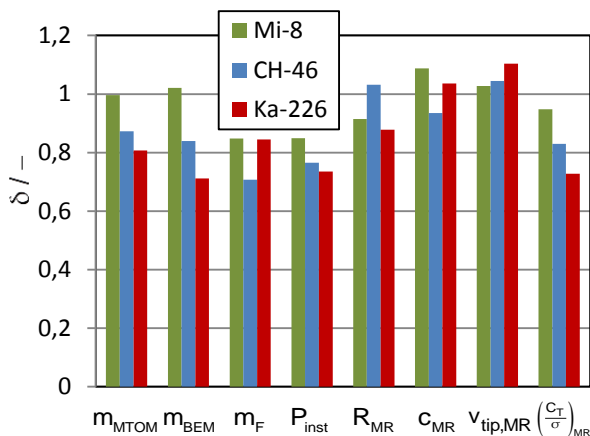


FIG 14 Parameter ratios after converged level 1 sizing compared with three reference configurations.

4 PRELIMINARY DESIGN

The calculations in this chapter are typically indicated as level 2. They are performed to get additional and more sophisticated data for the design. The preliminary design needs an external configuration to perform the internal arrangement inside the hull and the higher order solution of the flow field outside the hull.

4.1 Flight conditions and load cases

The computation of several load cases is not required for the level 1 sizing. Hence, the time-consuming calculations are conducted on level 2. An enhanced structural sizing requires a sophisticated calculation of the loads on the individual external components of the rotorcraft for a variety of maneuvers. The trim algorithm of HOST calculates the flight loads on the lifting surfaces (rotating and fixed surfaces) by applying blade element theory. This enables the consideration of a variety of cyclic and collective control inputs and the resulting loads on the rotor hubs. Typical flight and ground maneuvers that are considered taxiing on the ground, hovering, jump take-off, forward flight at different velocities as well as banked turns at different angles up to 45°.

4.2 Enhanced structural design and mass

Since the conceptual design deals with the determination of the outer mold line, the preliminary design phase determines i.a. the internal structural arrangement. Therefore, a module applying finite element theory (FEM) has been integrated into the design process that allows a more detailed analysis of the fuselage structure and thus, a more detailed fuselage mass estimation.

However, in a first step the primary structure (frames, stringers and panels) is distributed along the fuselage loft according to knowledge based design criteria, e.g. cutouts for doors, and stored in the CPACS file. In a subsequent step, an ANSYS based finite element tool is called either to generate the FE model, to conduct a static analysis or to size the fuselage structure according to specified static load cases of typical flight and ground maneuvers. The FE model is of global mesh quality where the masses and external loads are introduced as nodal loads and distributed over the airframe in user-specified regions, as exemplarily shown in FIG 15. Stringers are discretized using elastic beam elements (ANSYS Beam188) that feature arbitrary cross sections (as specified in CPACS) while frames and panels are discretized using elastic shell elements (ANSYS Shell 181).

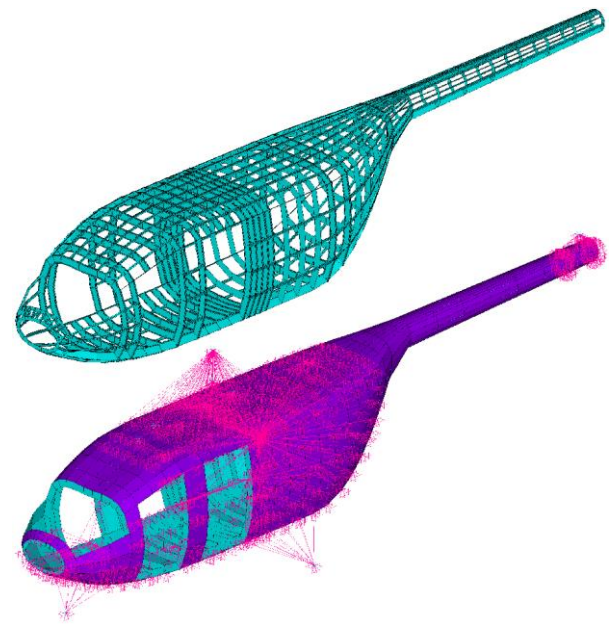


FIG 15 Airframe model showing the stringer and frame distribution (above) and with constrained nodal masses (below)

Static sizing is conducted by employing a sizing module which is an APDL (ANSYS Parametric Design Language) based tool originally developed for sizing of aircraft wings (see Nagel [16]) using the ANSYS solver. In subsequent developments, this module has been extended to allow sizing of aircraft fuselages (see Scherer [31]). Later it was adapted for the structural sizing of rotorcraft fuselages (see Schwinn [32]).

Sizing of the fuselage means finding the minimum acceptable thicknesses of the airframe according to all expected loads during the rotorcraft life. Minimization of airframe mass improves flight performance, reduces fuel consumption and increases payload and/or range. The strength evaluation implemented in this sizing module is based on fully stressed design (FSD) principles, i.e. for each specified load case the minimum required (shell) element thickness is calculated by the ratio of actual and maximum acceptable stress. Considering all defined load cases the maximum required sheet thickness is stored for each element. Beam elements are sized by scaling their geometric section in width and height. The employment of the FSD method allows for optimal material utilization.

FIG 16 shows a generic sized rotorcraft fuselage. On the top the critical load cases for each shell element are shown. The corresponding required thicknesses are presented below (top, side and bottom view). It can be seen that in front of and behind the cabin the required thicknesses are the minimum thicknesses while the cabin itself, where most masses and loads are introduced into the structure, requires thicknesses of up to $t=4.5$ mm. This is owed to the mounting of the structure.

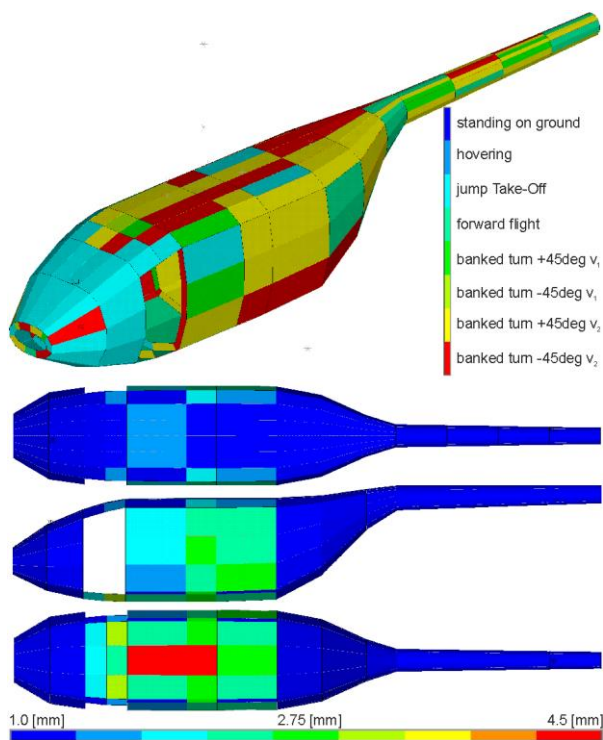


FIG 16 Airframe sizing: Critical load cases (above) and resulting thicknesses (below)

A comparison of the sized fuselage (as depicted in FIG 16) to the fuselage masses estimated statistically is presented in FIG 17. The loads that have been introduced into the FE computation are derived from the final masses (level 1 mass estimation) using the methods proposed by Johnson. It can be observed that the estimated masses are higher than those computed by the sizing module. However, when using the results

of the calculation applying the technology factors as specified earlier, both masses are very close.

Discrepancies, however, result from an incomplete load envelope, i.e. structurally highly stressing load cases (pull-up maneuver, asymmetric flight maneuvers) which are not yet included in the load case list like lateral and rearward flight maneuvers. Moreover, the examination of different mass distributions as well as joints and the crashworthiness proof are not yet included in the sizing loop, thus, potentially increasing the fuselage mass.

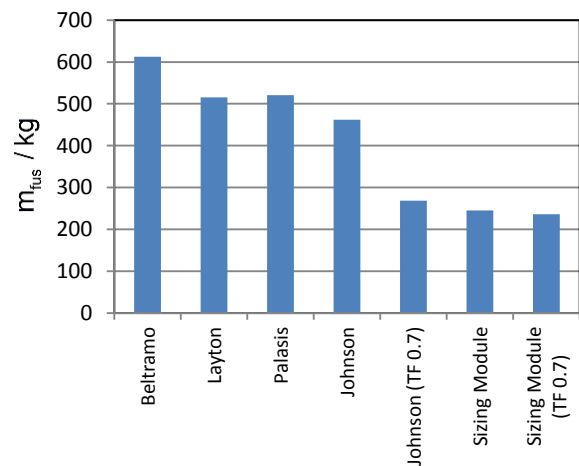


FIG 17 Comparison of the statistical mass estimation of the fuselage and the enhanced FEM based mass computation

4.3 Rescaling the external configuration

Results of the level 2 computations may require resizing of the external configuration. This is an issue especially for novel configurations with a lack of experience for initial data. The level 1 sizing loop starts with the uniform CPACS data model as an input. At this stage of the design process the input data comes from the level 0 initialization or the level 2 higher order computations. If the sizing loop is restarted with a CPACS file coming from level 2 the filling ratio of the dataset is slightly higher but the data exchange between the tools is equal. The information from level 2 overwrites the values from previous calculations. In order to keep the knowledge gathered by the last calculations, the level 1 technology factors have to be updated. These technology factors are applied for component masses as well as for aerodynamic parameters. The information from level 2 is transported over the several level 1 iterations until the new external configuration converges.

5 CONCLUSIONS AND OUTLOOK

This paper has shown the approach behind the rotorcraft design environment developed by DLR. For the last one and a half years the process was completely rebuilt using the lessons learned from the previous years. This toolbox now features some special

capabilities required for the research activities of the coming years:

- The process connects an arbitrary number of partners with tools located on different servers into one design process.
- The design process has a uniform data model (CPACS) that makes the tools exchangeable and rearrangeable. This modularity is necessary for the demanded toolbox characteristics.
- Distributed computation and a uniform data model allow the optional change between conceptual and preliminary design giving the ability to redesign the external configuration based on preliminary design results.

DLR will focus now on the integration of new technologies into the design environment. The influence of new technologies on novel configuration and the design process itself is an important factor for bringing forward future concepts and to evolve rotorcraft virtual engineering. The new configurations will range from urban aerial mobility vehicles, considering electrical propulsion to high speed compound configurations, considering the unloading of the rotor disc by wings and thrust generators.

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