Rotorcraft Engineering – Preliminary Design

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General information

• **Lecture:**	Monday, 13.00 – 14.30 Uhr, MW2701m
• **Exercise:**	Monday, 14.45 – 15.30 Uhr, MW2701m
• **Prüfung:**	written, 90min (short questions + calculation tasks)
• **Sprechzeiten:**	by arrangement
• **Kontakt:**	Office Institute of Helicopter Technology Room MW2703 Tel.: 089-289-16300 Fax: 089-289-16354 Email: info@ht.mw.tum.de

Sources and Literature

- Stepniewski, W.Z.; Keys, C.N. – Rotary-Wing Aerodynamics
- Prouty, R.W. – Helicopter Performance, Stability and Control
- Johnson, W. – Helicopter Theory
- Johnson, W. – Rotorcraft Aeromechanics
- Leishman, J.G. – Principles of Helicopter Aerodynamics
- Leishman, J.G. – The Helicopter: Thinking Forward, Looking Back

Content of the Lecture

- Overview and Introduction
- Design Requirements
- Preliminary Sizing
- Flight Performance
- Cost and Mass Estimation
- Architecture and Component Design

1 Overview and Introduction

1 Overview and Introduction

1.1 Course Outline

1.2 The Helicopter Design Task

1.3 Conceptual Study

1.4 Comparing Configurations

1 Overview and Introduction

1.1 Course Outline

1.1 Course Outline

Objective of the Lecture:

- Helicopter Engineering requires an interaction between a number of different disciplines like aerodynamics, flight mechanics, structural engineering, design etc.
- This lecture addresses the fundamental correlations that play a role in preliminary design. It will be shown how to dimension main helicopter components based on mission requirements. Building on this, the iteration steps to further refining of the design will be explained.
- At the end of this lecture the participants will have basic knowledge for the preliminary design of a helicopter. This includes the estimation of empty weight and maximum take-off weight, installed power, flight envelope as well as the main- and tail-rotor sizing.

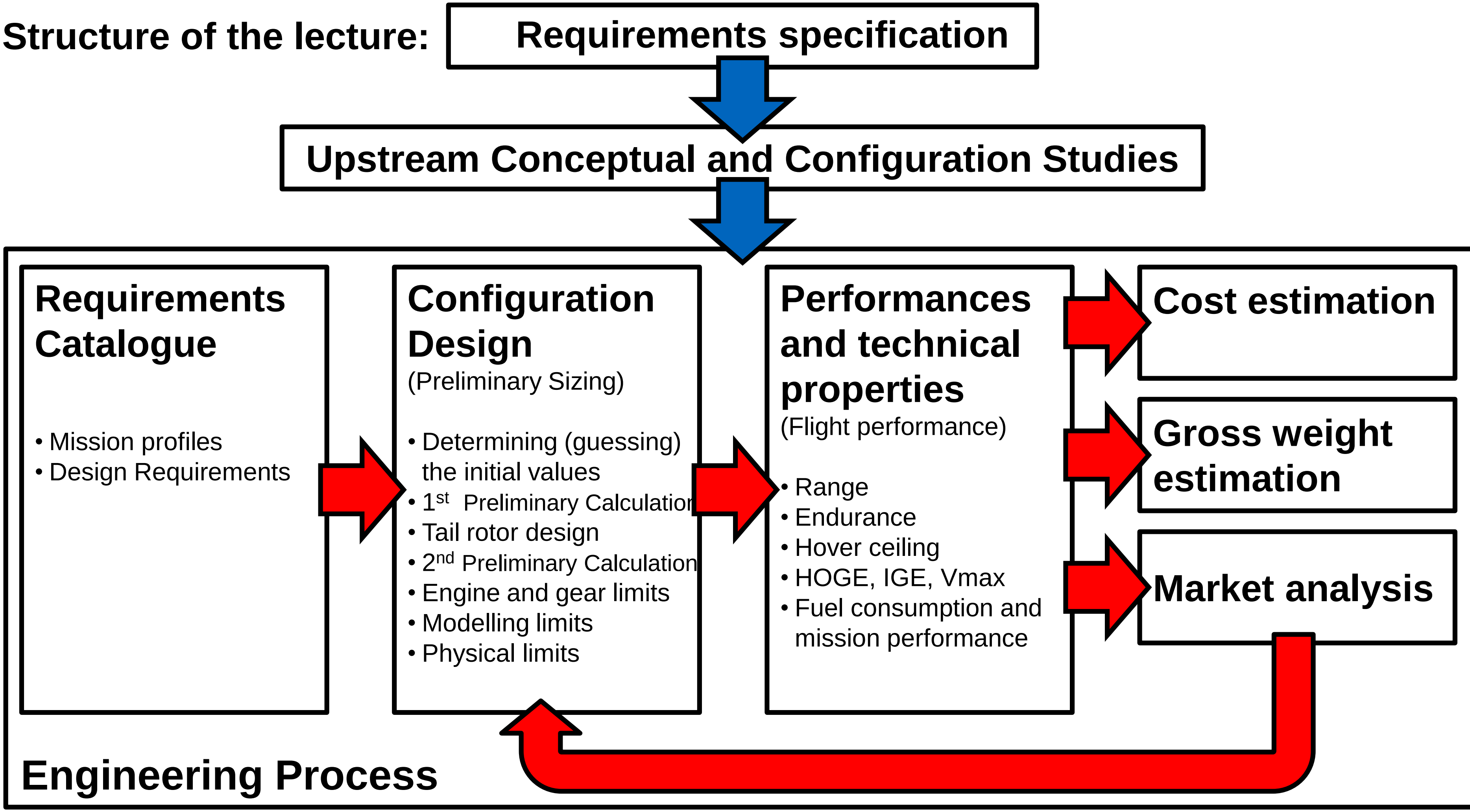
1.1 Course Outline

Learning content of the lecture:

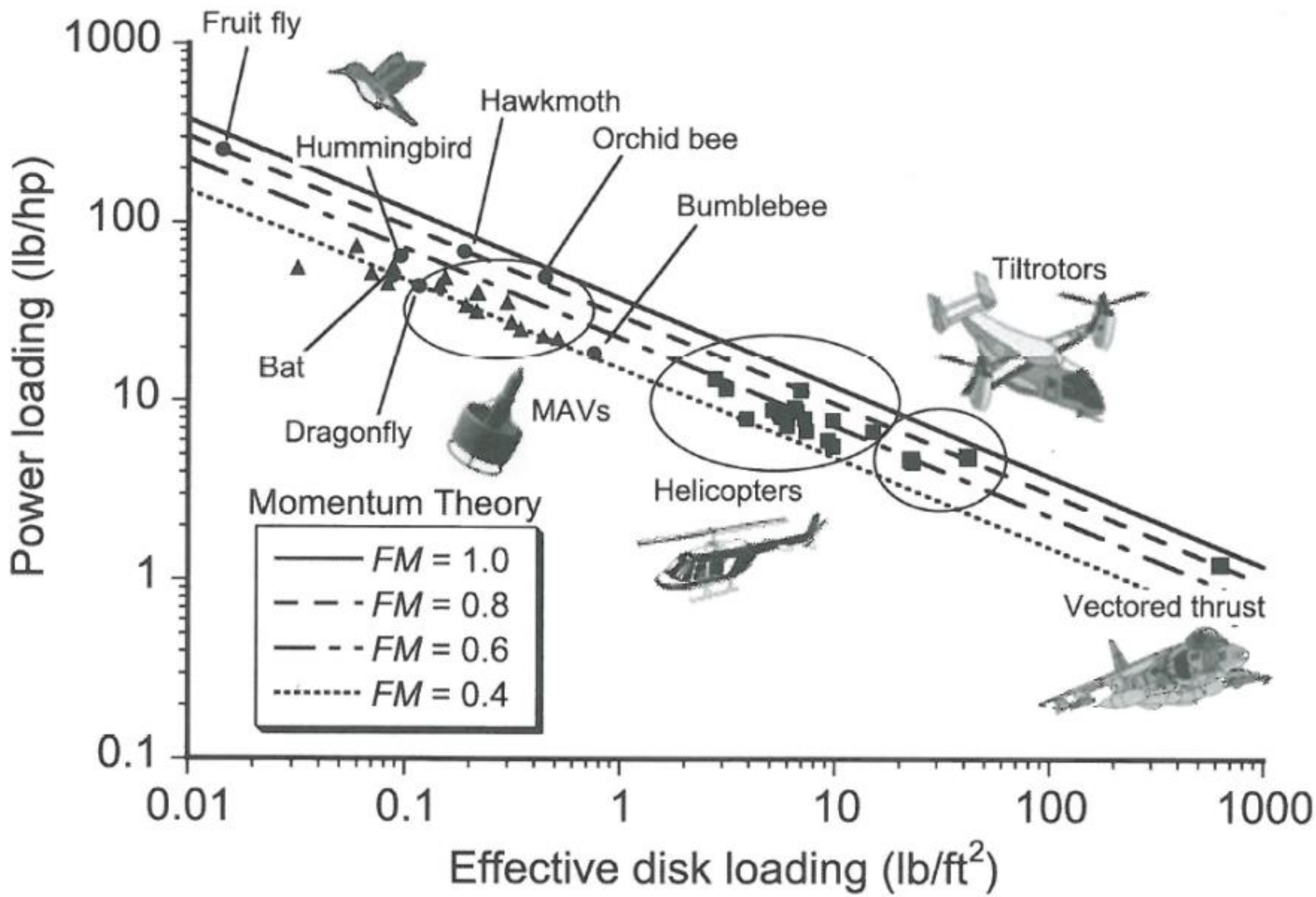
- Providing an insight into the process of helicopter engineering and the requirements of a modern design.
- Creating an understanding for trade-offs and physical correlations.
- Acquiring the capability to make use of engineering methods to assess different designs with respect to their expected flight performance.
- Communicating methods for estimating the cost and mass of individual helicopter components as well as identifying the decisive parameters.
- Building basic knowledge about design aspects in helicopter construction.

1.1 Course Outline

Structure of the lecture:



1.1 Course Outline



1.1 Course Outline

Structure of the lecture:

- After giving a short insight into the task description relevant to the design process, the lecture addresses the Conceptual and Configuration Studies.

- The main part of the lecture is dedicated to the iterative design process, which begins with listing the requirements followed by a preliminary sizing and flight performance which allows for a cost and gross weight estimation. Based on these estimates a modification to the design can be undertaken in order to repeat the iteration loop.

- The evaluation of component designs aspects and the detailed examination of commonly used designs round off the lecture.

1 Overview and Introduction

1.2 The Helicopter Design Task

1.2 The Helicopter Design Task

The foundation for designing a helicopter is the **Requirements Definition**, this means

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Tender (military design):

- Detailed mission profiles


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Performance Specification (civil design):

- Definition of the desired capabilities by a civil client or internally (in-house development)

1.2 The Helicopter Design Task

In the first instance a conceptual study is carried out based on the requirements catalogue. In order to do this, the design task has to be translated into technical requirements which are then weighted according to their priority:

Military Design:

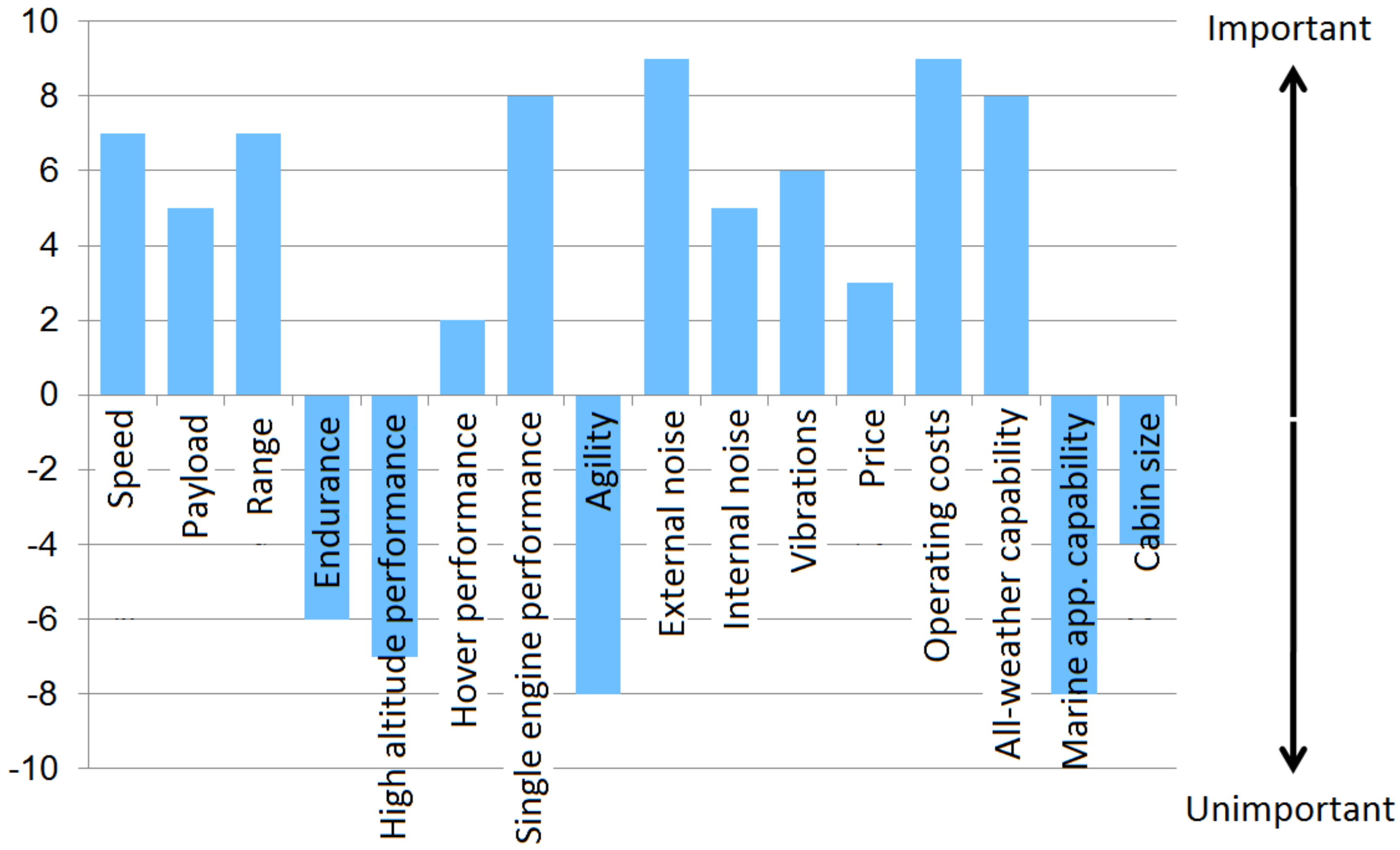
- The technical requirements and their prioritization are defined by the missions to be performed or are available as system capability requirements.

Civil Design:

- Making a requirements catalogue for a civil design is significantly more difficult and is mostly based on a **requirements analysis**. The various technical requirements are put into context and weighted according to their priority.

1.2 The Helicopter Design Task

Example of a requirements analysis:



1.2 The Helicopter Design Task

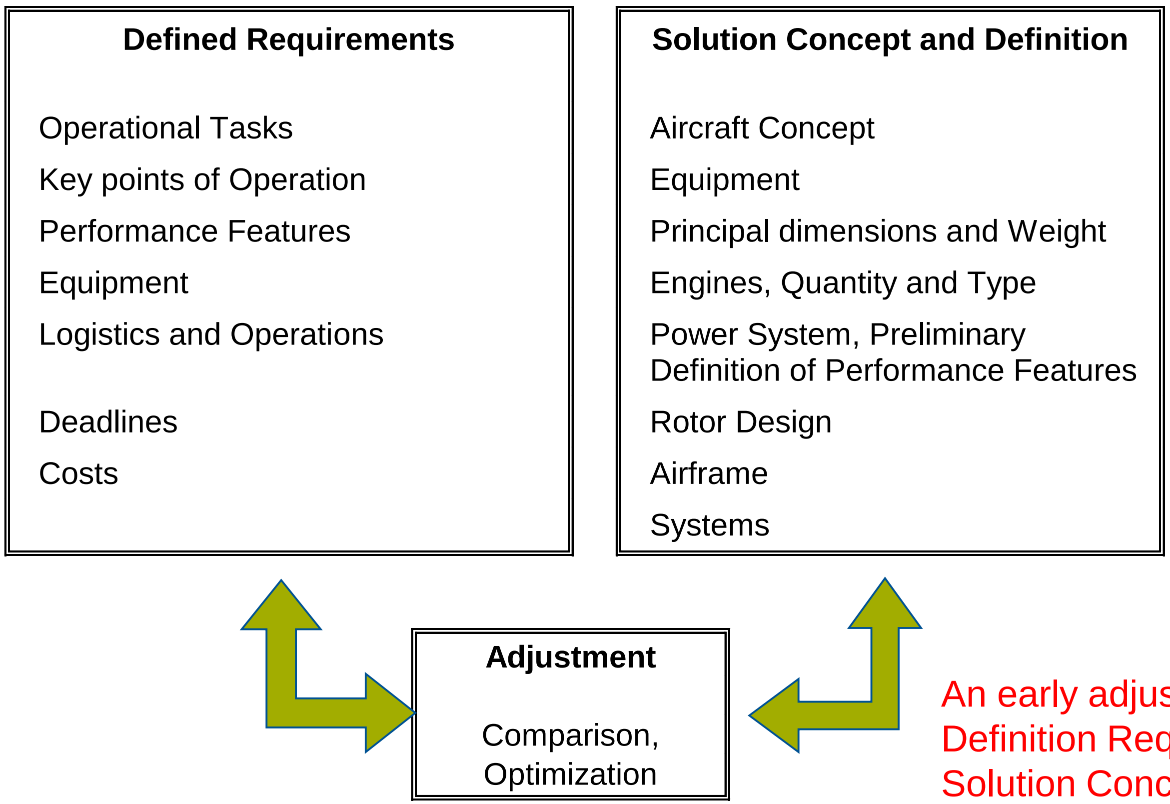
- Based on the requirement analysis the existing information has to be analyzed and if necessary completed in order to be able to formulate a technical statement.

- In general a requirements specification should cover every area and design point of a project i.e. besides knowing the operational tasks the key performance features, operational requirements etc. should also be known.

- The **requirements from the specification** and the **solution concept** should always be compliant with one another.

1.2 The Helicopter Design Task

Ensuring compliance between requirements and the solution concept:



An early adjustment of Problem Definition Requirements and the Solution Concept is fundamental for an optimal design!

1.2 The Helicopter Design Task

Requirements Definition – Operational Tasks (Mission profiles):

Declaration of:

- Payload/Range
- Passenger capacity
- Cabin equipment
- Take off conditions
- …

1.2 The Helicopter Design Task

Requirements Definition– Key points of Operation:

Definition of extraordinary operating conditions, i.e.:

- Take-off/Landing on heli decks
- Operation in icing conditions
- Definition of max. load capacity
- Maximum Range
- …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1.2 The Helicopter Design Task

Requirements Definition– Performance Features:

Fixation of performance features and flight property demands (Maneuverability requirements, maximum load factors. ,…)

Demands significant to the design:

- Hovering conditions
  - Take-off altitude
  - Temperature
  - Power reserve
  - Engine fail conditions
- Speed
  - Cruising speed, Maximum speed
- Payload/Range

1.2 The Helicopter Design Task

Requirements Definition– Equipment:

Demands to:

- Avionics
- Hydraulics and pneumatics
- Cabin Interior design
- Specialized equipment (winches, emergency floats)

Significance for the Design:

Making provisions for space, weight reserves and attachment points already at the design stage (Equipment alone makes up 30% of the payload)


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1.2 The Helicopter Design Task

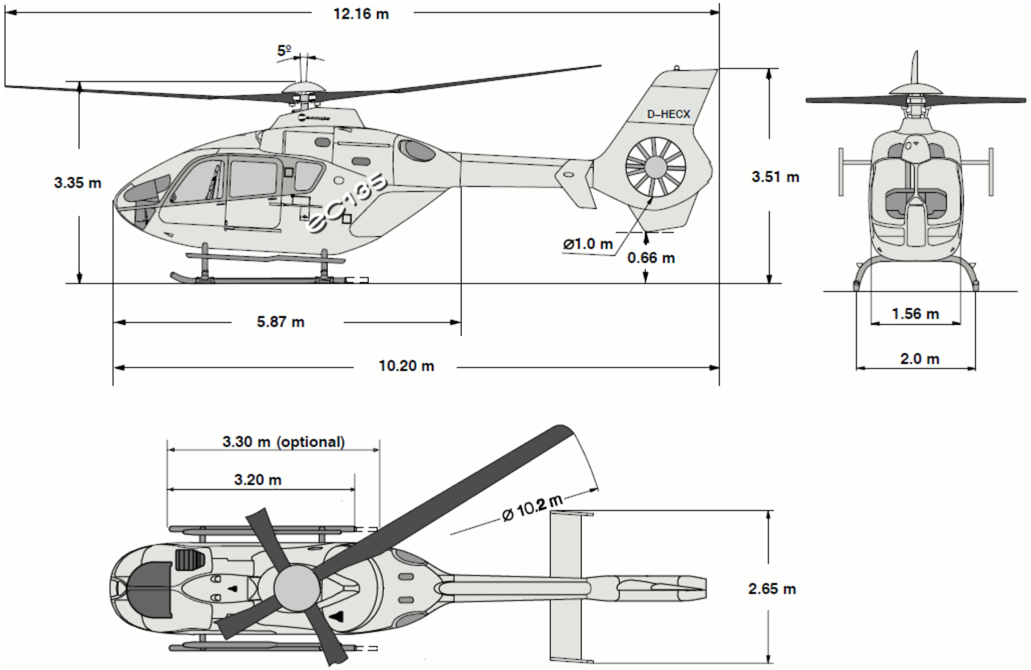
Requirements Definition – Logistics and Operation:

Logistic Demands:

- Fuel used
- Oil types used
- …

Operational Demands:

- Reliability
- Downwash in hover flight
- Max. dimensions
- …



1.2 The Helicopter Design Task

Requirements Definition – Deadlines and Costs:

Deadlines:

- Commitment to development time frame
- Definition of Milestones
- …

Costs:

Specification of Life Cycle Cost (LCC),

Particularly

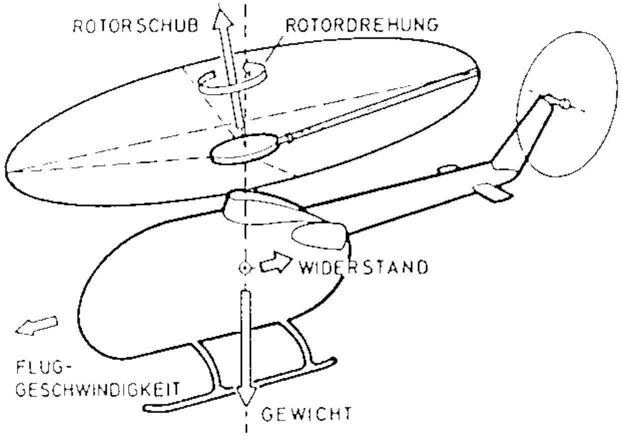
- Operator maintenance costs
- Production and development costs as part of the total-cost calculation of the manufacturer.

$\Rightarrow$ In civil as well as in military aviation today, the economical factors play a vastly increasing role. Therefore great attention should be devoted to an economical design from the very beginning of the project!

1 Overview and Introduction
1.3 Conceptual Study

1.3 Conceptual Study

For tasks requiring vertical take-off and landing the conventional helicopter concept is mostly used. This concept incorporates one or more rotors providing both lift and propulsion.



1.3 Conceptual Study

Requirements leading to alternative concepts:

When the mission requirements exceed the conventional helicopter's capabilities, conceptual studies and comparisons must be carried out in order to determine, which solutions can meet the customer requirements.

Requirements such as:

- Significantly higher cruising speed (>250 kts)
- Cruising altitude 7500 m ("above the weather")
- Higher range (>600 nm)
- Improved reliability (Failure $<10^{-7}$/Fh)
- Improved CAT A performance (OEI - one engine inoperative)
- Improved comfort (noise and vibrations)
- Improved environmental impact (outside noise, NO_x, ...)

1.3 Conceptual Study

Requirements leading to alternative concepts :

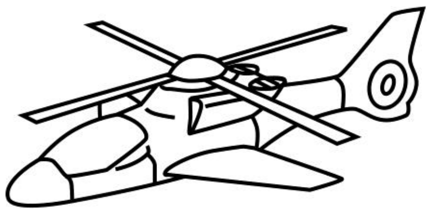
- Longer service life (more important, expensive components)
- Lower production and development costs
- Lower DMC / DOC (direct maintenance costs / direct operating costs)
- Seat kilometers max. 20% more expensive than on an airplane

Important criteria for the concept choice:

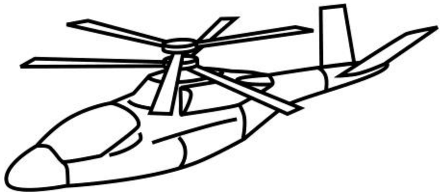
- Mission
- Performance requirement
- Disc loading
- Flight mechanics
- Hover flight performance
- Speed
- Range
- Transport capacity
- Costs

1.3 Conceptual Study

Alternative Concepts:

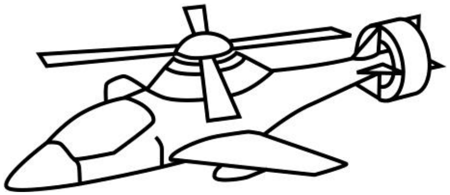


Helicopter with Wings (Lift Compound)



ABC - Advancing Blade Concept

- 2 counter rotating, very stiff rotors

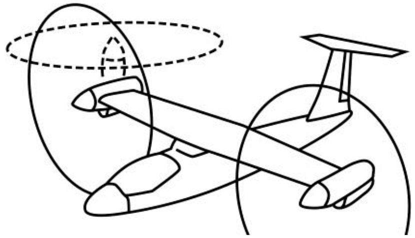


Thrust Compound

- Helicopter with additional propeller

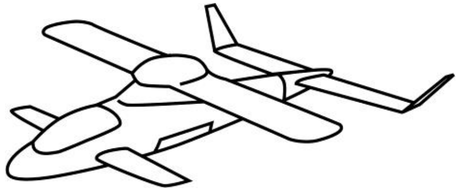
1.3 Conceptual Study

Alternative Konzepte:



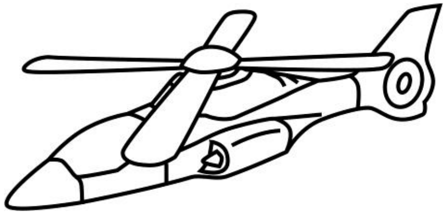
Tilt-Rotor

- Rotors tilt forward for flight mode



CRW (Canard Rotary Wing)

- For the transition from helicopter to airplane mode the rotor stops in the wing position



X-Wing

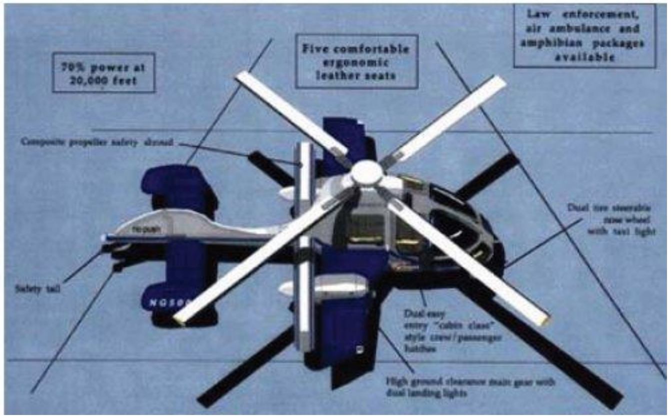
- Rotor stops rotating in airplane mode and acts as fixed wing

1.3 Conceptual Study

Examples– ABC-Concept and Helicopter with Wings:

1.3 Conceptual Study

Examples– Compound - Helicopters:

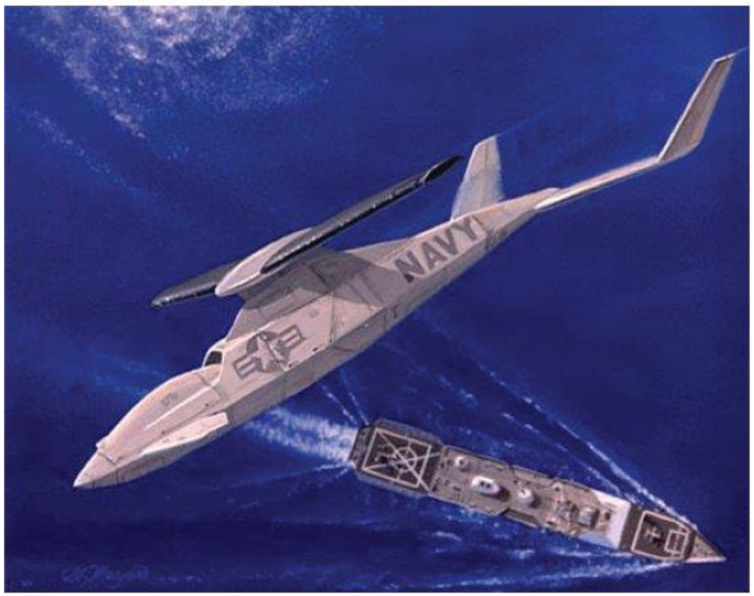


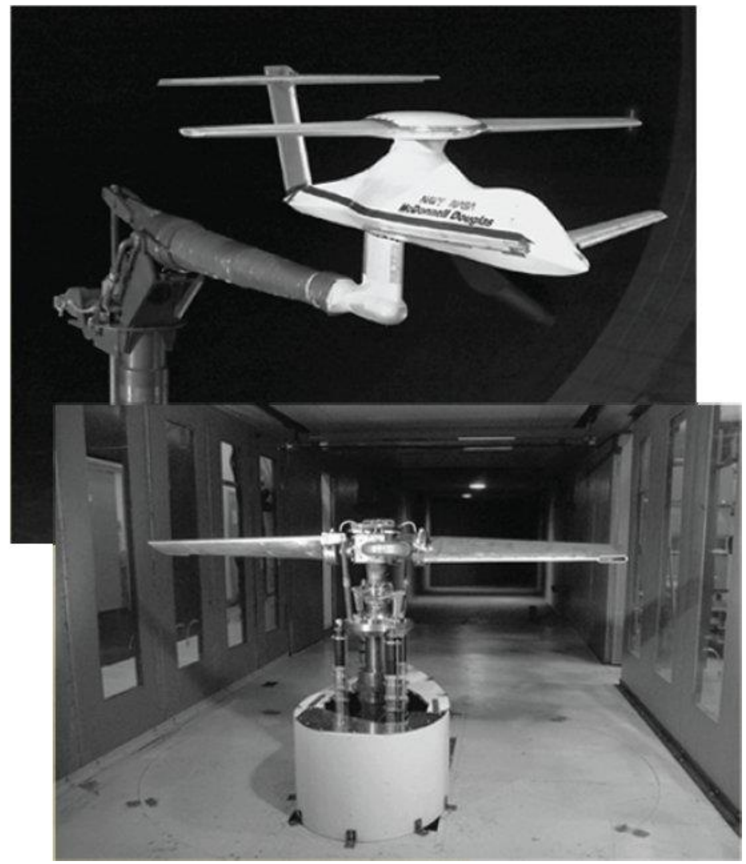
1.3 Conceptual Study

Examples – Tilt-Rotor:

1.3 Conceptual Study

Examples – Canard Rotary Wing:



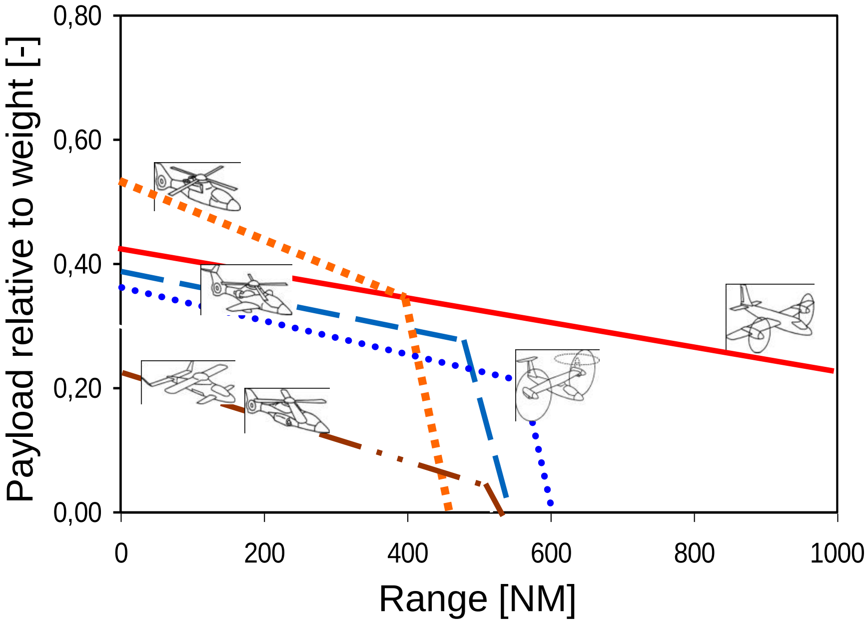


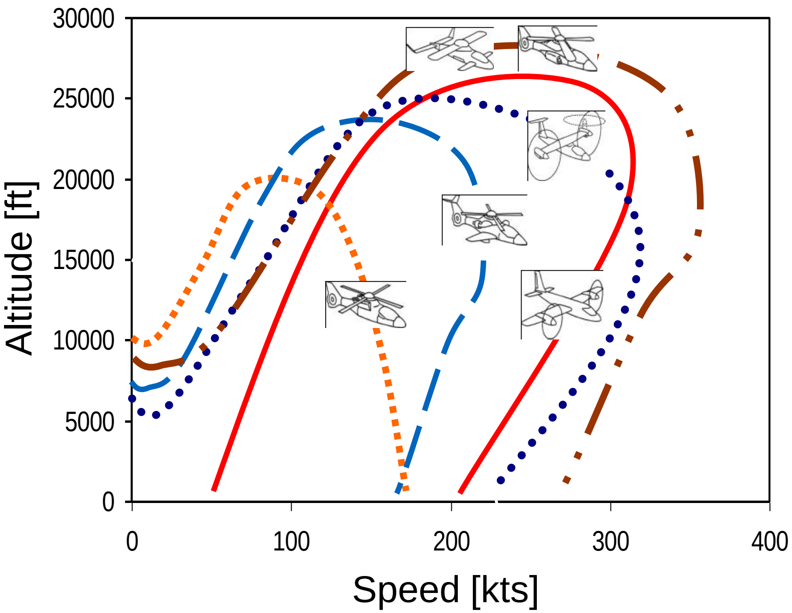
1.3 Conceptual Study

Examples – EVTOLs:

1.3 Conceptual Study

Concept Choice based on selected Criteria:





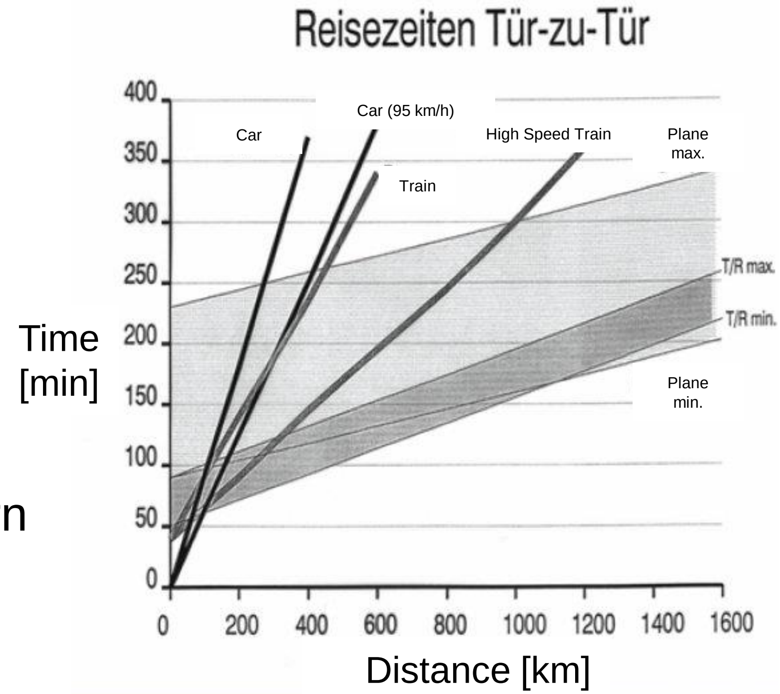
- **Transport capacity** (payload vs. range)

- Depending on the demands, the maximum **Speed** at a certain altitude is decisive. It has a wide range depending on the chosen configuration.

1.3 Conceptual Study

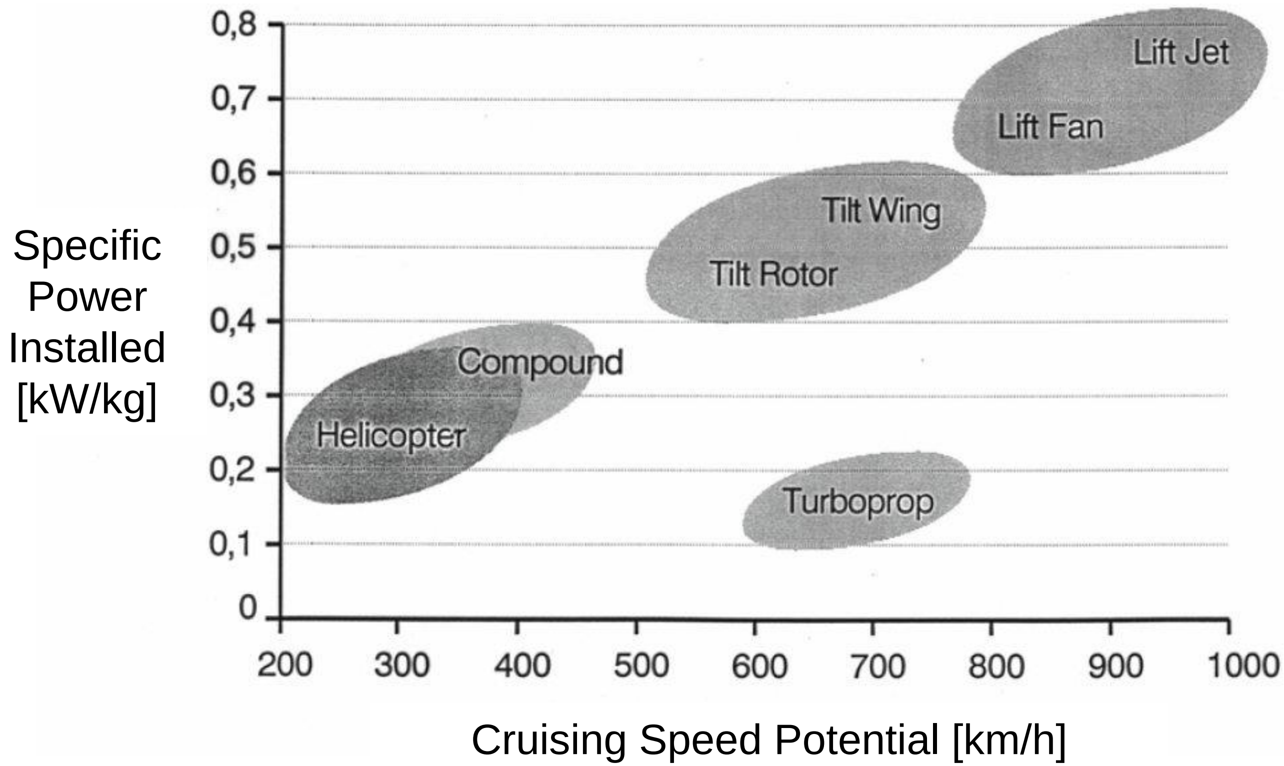
Mission Profile Criterion:

- **Civil Passenger Transport :**
  Objective: Short "door to door"- times, 400km, Block flying time 60min
  Criteria: Seat kilometer cost, comfort, Speed/Range
- **Search And Rescue:**
  Objective: Cruise≈370km, 3,5 h Search - Rescue - 370km high speed return to hospital
  Criteria: Speed, Range
- **Military Transport:**
  Objective: Logistic Transport within the new EU boarders
  Criteria: Range, Transport Capacity (Speed x Weight)



1.3 Conceptual Study

Power Requirement Criterion:



1.3 Conceptual Study

Flight Mechanics Criteria:

- Helicopters, Compounds and Tilt-Rotors can operate continuously within the entire flight envelope in a steady state
- Tilt-Wings in transition from helicopter mode to airplane mode operate in an unsteady state.
- With Helicopters and Compounds autorotation is possible
- With Tilt-Rotors autorotation is possible only at high descent speeds with difficult flare out
- With CRW and Xwing autorotation is not possible due to small propellers, but landing like an airplane is possible on an airfield

1.3 Conceptual Study

Conclusions of the Conceptual Study:

- After having chosen a concept and proven its feasibility, the process of **Comparing Configurations** can be started.

- Taking into account the planned payload, cargo bay space and on board equipment a preselection of the dynamic components, such as the rotor system, the power system and the engines, can be performed.

1.3 Conceptual Study

Technology Trends – Gyrocopters:

www.auto-gyro.com

www.ul-schule.de

High Speed VTOL Concept

www.groenbros.com

www.groenbros.com

www.cartercopters.com

1.3 Conceptual Study

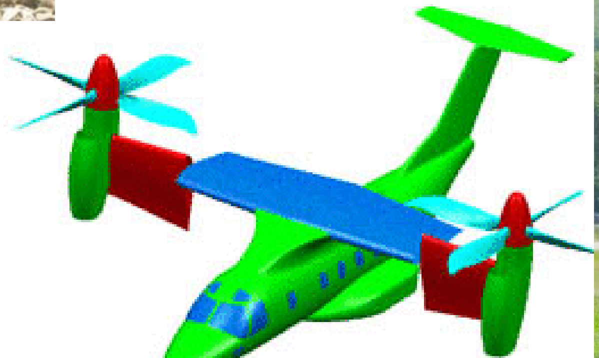
Technology Trends – Alternative Concepts:


Quad-Tiltrotor for
Joint Heavy Lift


Stop-Rotor UAV


Compound Resurrection
Maiden flight 29.6.2007

BA609


ERICA


X2 Concept (ABC)
Maiden flight
28.8.2008

1.3 Conceptual Study

High-speed, long range Hybrid Helicopter (H3) Concept

Technology Trends –

Eurocopter X3:

- Maiden flight 06.09.10
- **Power System:**
  - 2 turboshafts
  - 5 blade main rotor
  - 2 propellers on stubby wings
- **Cruising Speed** > 407km/h
- **Civil Operational Areas:** long distance search and rescue (SAR), coast guard patrol, border patrol, passenger transport, inter-city-shuttle
- **Military Operational Areas :** deployment of special forces , infantry transportation, combat SAR, evacuation missions

1 Overview and Introduction

1.4 Comparing Configurations

1.4 Comparing Configurations

- The following slides will refer to the „Helicopter“ concept as basis for various configurations.
- Taking into account the desired payload, cargo bay space and on board equipment the following dynamic components are prechosen:

$\Rightarrow$ **Rotor system**

$\Rightarrow$ **Power system**

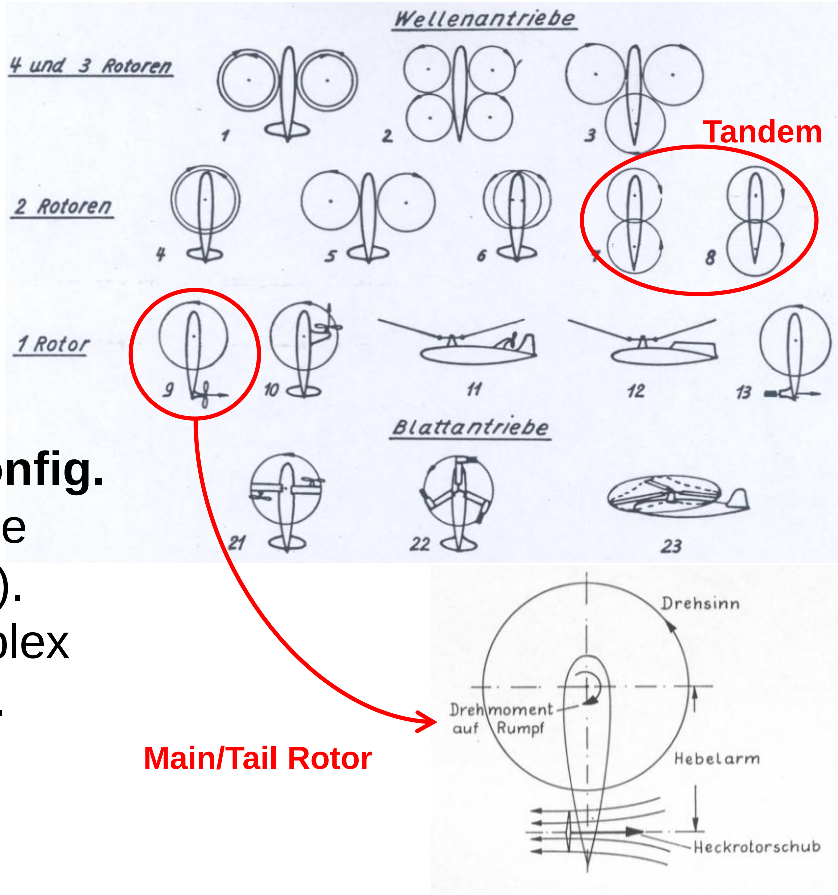
$\Rightarrow$ **Engine selection**

1.4 Comparing Configurations

Rotor System – Main and Tail Rotor vs. Tandem Configuration:

An analysis of helicopters currently in service shows that:

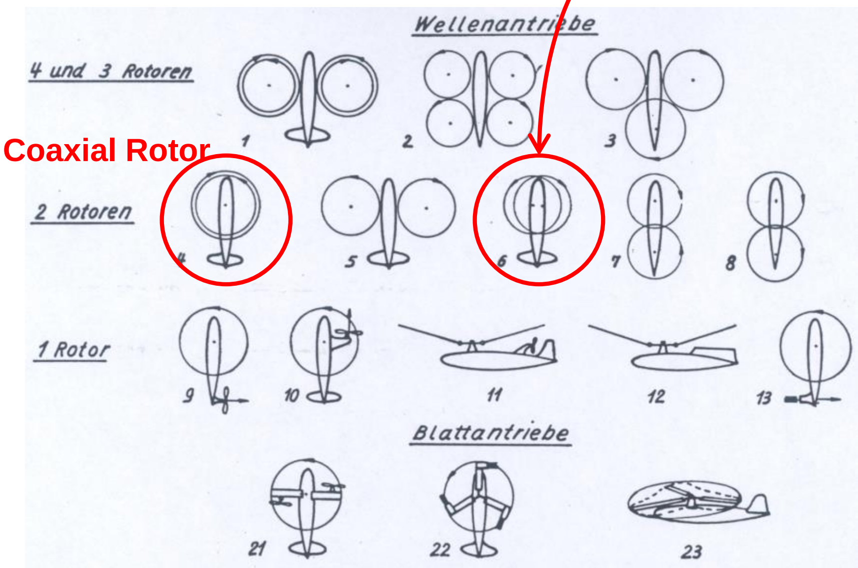
- among light und mid sized helicopters the **Single rotor Config.** with **Tail rotor** torque compensation (10-14% power consumption, 3% empty weight contribution) has established itself.
- among heavy helicopters the **Tandem Config.** prevails. It consumes less power for torque compensation (15% for heavy helicopters). However the control system is more complex as well as the gear mechanism is heavier.



1.4 Comparing Configurations

Rotor System – Coaxial Rotors und Intermeshing Rotors:

Besides having the conventional Main/Tail rotor and Tandem configurations, there is also the Coaxial and Intermeshing rotor configuration. Depending on company philosophy these configurations may be found throughout all weight categories.



The performance loss due to mutual interference of the two main rotors accounts for 20% of total power required which corresponds to the losses that a conventional tail rotor configuration induces.

1.4 Comparing Configurations

Rotor System – Examples:

1.4 Comparing Configurations

Power System:

The power system, in addition to the rotor system, is a main component of a helicopter. It consists of the motor itself as well as the functional elements for transmitting power and force/torque. Two different systems can be distinguished:

- **Shaft driven:** Conventional power transmission from the motor via shafts and gearing mechanisms to the rotors. Relatively low power loss (2-3% of total power).
- **Reaction propulsion:**
  - Blade tip nozzles (supply of compressed air/mixed gas/exhaust through hollow rotor blades)
  - Combustion jets (supply of air and fuel to be burnt at the blade tip): Ram jet, pulse jet or rockets on the blade tip

1.4 Comparing Configurations

Power System:

Reaction propulsion driven rotors do not require torque compensation. Apart from the Sud Aviation Djinn these rotors only ever made it as prototypes (i.e. MBB Heidelberg-Rotor), because they

- are very complicated in terms of power adjustment
- do not allow multiple engines due to their complexity
- have a low efficiency
- generate too much noise

Due these reasons the mechanical power system has asserted itself!

1.4 Comparing Configurations

Engine Selection– Estimating Maximum Take-Off Weight:

- Depending on the helicopter, one, two or three engines might be employed.
- The first step in selecting engines is estimating the maximum power requirement based on **Maximum Take-Off Weight (MTOW)**.
- The MTOW is composed of **Empty Weight (EW)** and **Useful Load (UL)**:

$$\boxed{MTOW = EW + UL}$$

$$\boxed{UL = m_{crew} + m_{payload} + m_{usable\,fuel} + \underbrace{m_{drainable\,oil}}_{\textbf{Negligible}}}$$

1.4 Comparing Configurations

Engine Selection– Estimating Maximum Take-Off Weight:

Definitions:

Manufacturer's Empty Weight (MEW)	A/C structure and parts necessary for flight
Empty Weight (EW)	MEW + permanently installed equipment
Basic Weight (BW)	EW + hydraulic fluid, oil and residual fuel
Basic Operating Weight (BOW)	BW + crew and luggage, drinking water, additional equipment
Zero Fuel Weight (ZFW)	BOW + payload (passengers und luggage, cargo,…) **excluding** usable fuel!
Take of Weight (TOW) or **Maximum Take-Off Weight (MTOW)**	total A/C weight at take off including maximum fuel and additional load

1.4 Comparing Configurations

Engine Selection – Estimating Maximum Take-Off Weight:

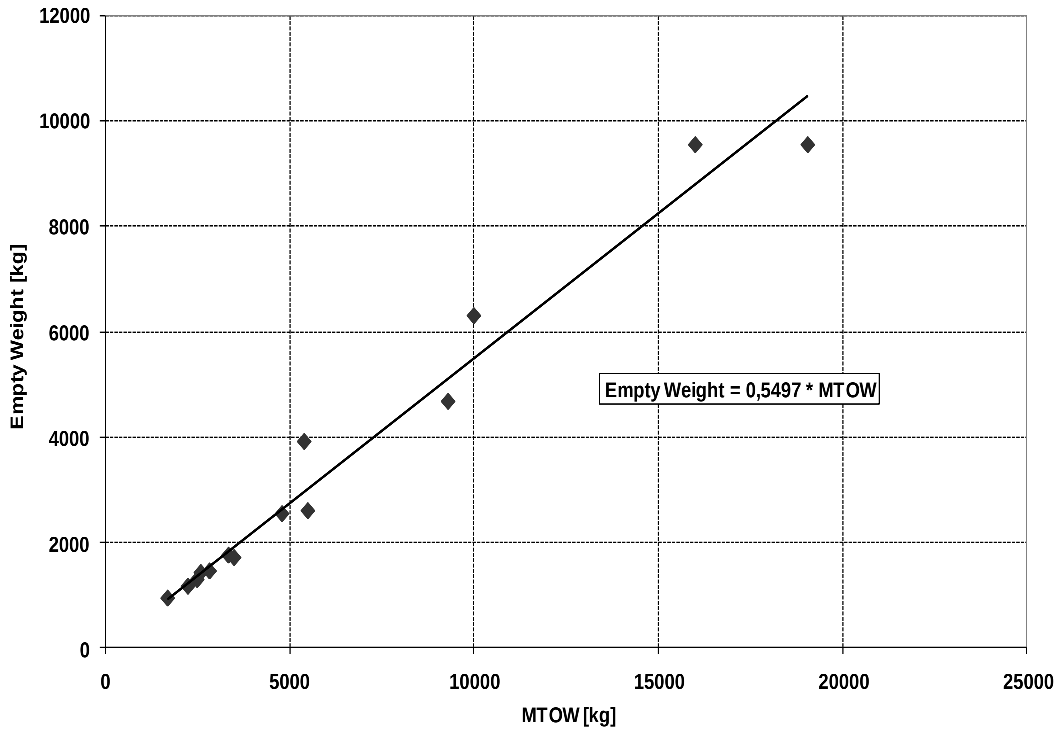
- For the initial estimate of MTOW for a given payload the following relationship is used

$$MTOW = EW + UL$$

- According to empirical data there seems to be a linear relationship between MTOW and EW

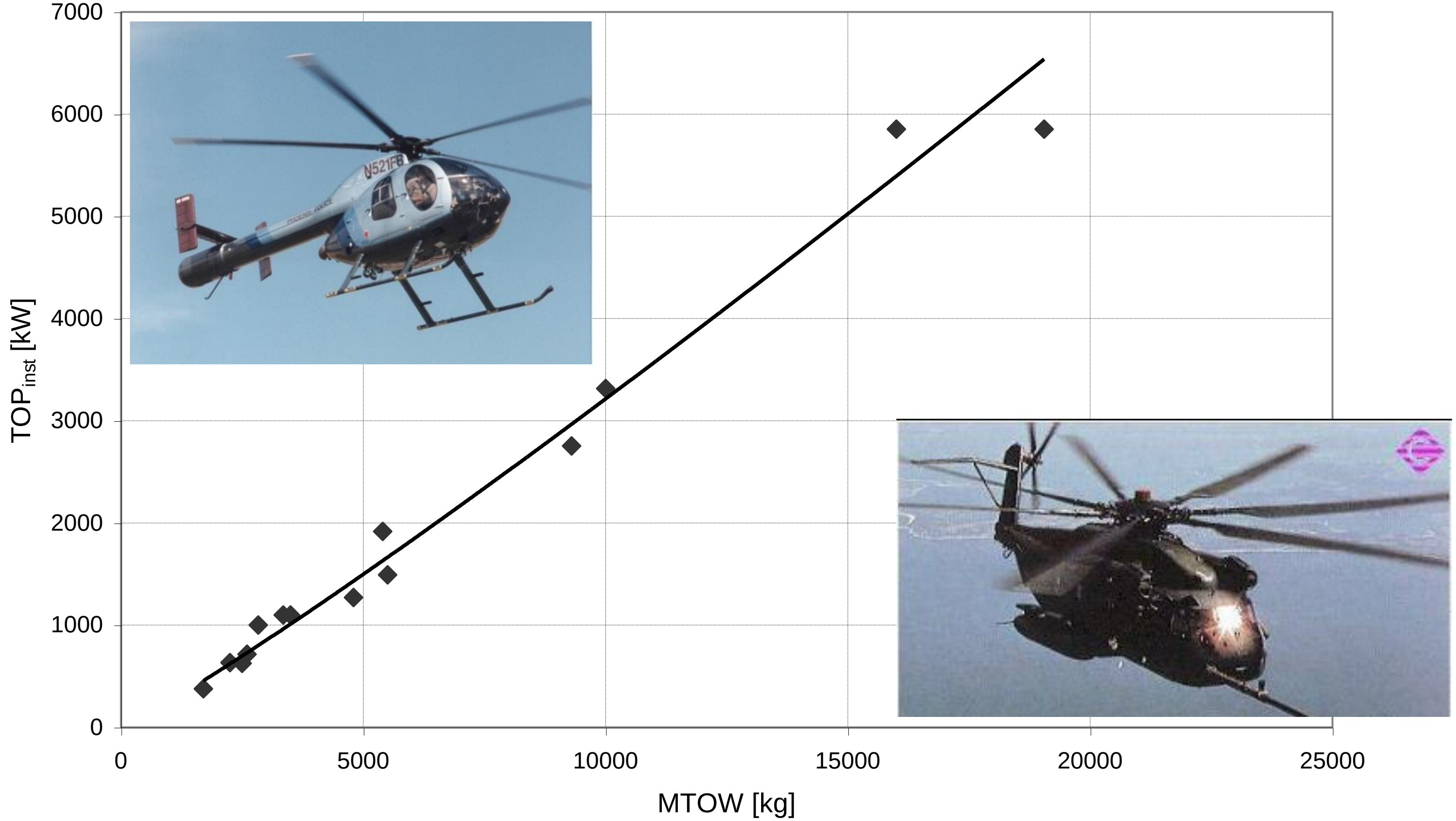
$$\boxed{\begin{aligned} EW/MTOW &= 0{,}5 - 0{,}55 \\ \Rightarrow UL/MTOW &= 0{,}45 - 0{,}50 \end{aligned}}$$

$\Rightarrow$ Empirical approach for estimating MTOW at a given UL



1.4 Comparing Configurations

Engine Selection – Estimating Power Requirement (Take-Off Power, TOP) in Relationship to MTOW:



1.4 Comparing Configurations

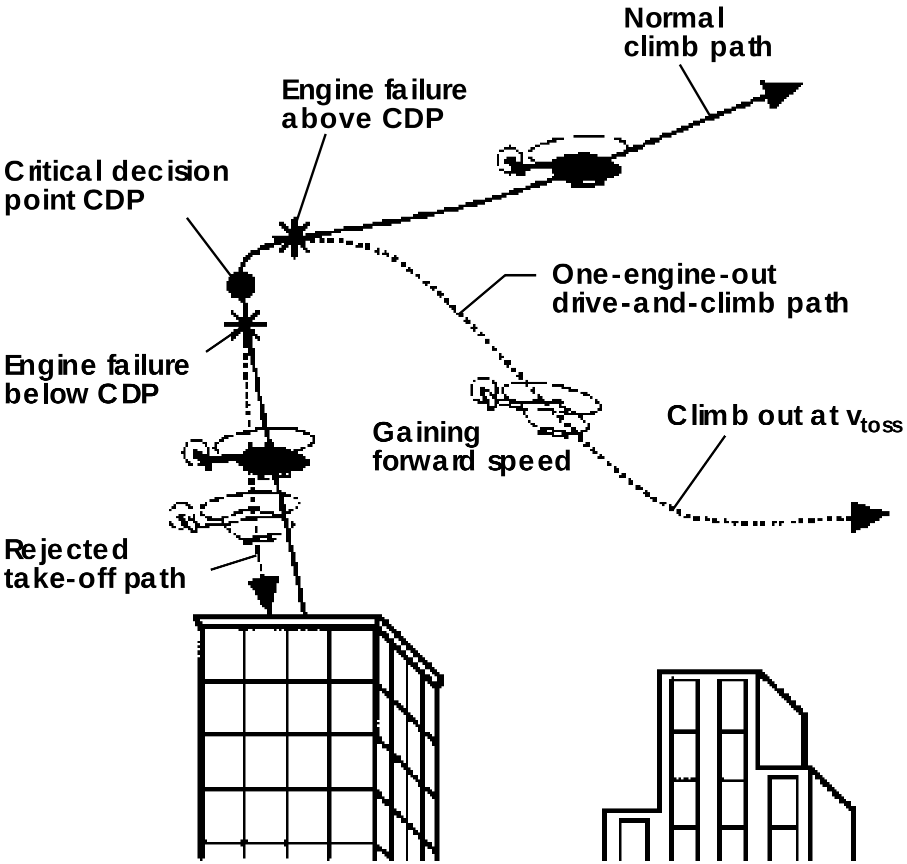
Engine Selection – Further Criteria for the Selection:

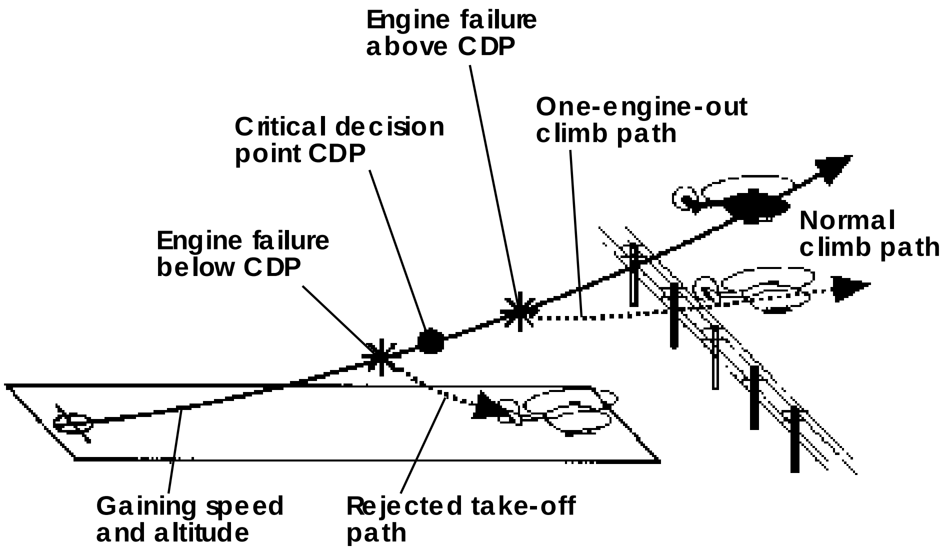
- Operation site with its take-off and landing conditions
- Nature of operation with its corresponding legal regulations (CAT. A)
- Costs of manufacturing and operation
- Extreme requirements on safety and performance. The military sector demands three engines for the heavy helicopter category

1.4 Comparing Configurations

Engine Selection – Influence from Operation Sites and Regulations:

**The use of some heliports (i.e. on hospital buildings) Requires a certification according to CAT. A.
It includes take-off and landing maneuvers, that can only be achieved with multiple engines.**





1.4 Comparing Configurations

Engine Selection – Influence from Safety and Performance Requirements and Regulations:

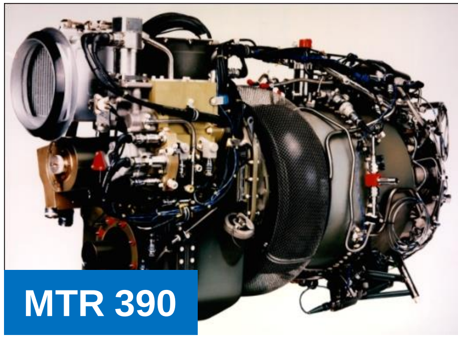
- CAT. A certification requires two or more installed engines. Single-engined helicopters do not comply with CAT. A and therefore can only be used for missions during which an engine failure wouldn't pose a danger.

- The necessity for an additional third engine arises in the light of extreme CAT. A requirements and costs for heavy helicopters. Example: Operation at sea, the success of which is absolutely essential.

1.4 Comparing Configurations

Engine Selection – Trends:

- Excellent high power output
- High power reserves for OEI(one engine inoperative)
- Compliance with CAT. A certification requirements
- High operational safety/reliability
- Low operational costs



1.4 Comparing Configurations

Technology Trends in Civil Configurations:

- **Alternative Airframe Concepts** (modular airframe structure, family concept for the cockpit, use of common interchangeable parts, improved crash safety)
- **Mission Equipment** (active/passive climate control, active vibration reduction, modular equipment, active/passive noise reduction)
- **Modular Control System** (interchangeable hydraulics components, 4-axis-autopilot)
- **Power System** (5-blade bearingless main rotor (BMR), adaptive rotor control

1.4 Comparing Configurations

Technology Trends in Military Configurations :

- **New Rotor Concepts** (Number of rotor blades ≥ 6, reduced vulnerability, improved maintainability, tilting rotor plane, folding rotor, alternative anti-ice system)
- **New Transmission/Engine Concepts** (transmission with high RPM-variation, adaptive engine control, modular structure of transmission/engines, adjustable main rotor mast tilt angle)
- **Alternative Tail Rotor Concepts** (electric motor, alternative actuators, ducted rotor systems, folding concepts)
- **Alternative Airframe Concepts** (modular mission equipment, reduced vulnerability, improved crash safety)

2 Design Requirements

2 Design Requirements

2.1 Helicopter Pre-Design
2.2 Identification of Requirements
2.3 Example Cases

2 Design Requirements

2.1 Helicopter Pre-Design

2.1 Helicopter Pre-Design

Main Objectives of the Preliminary Design:

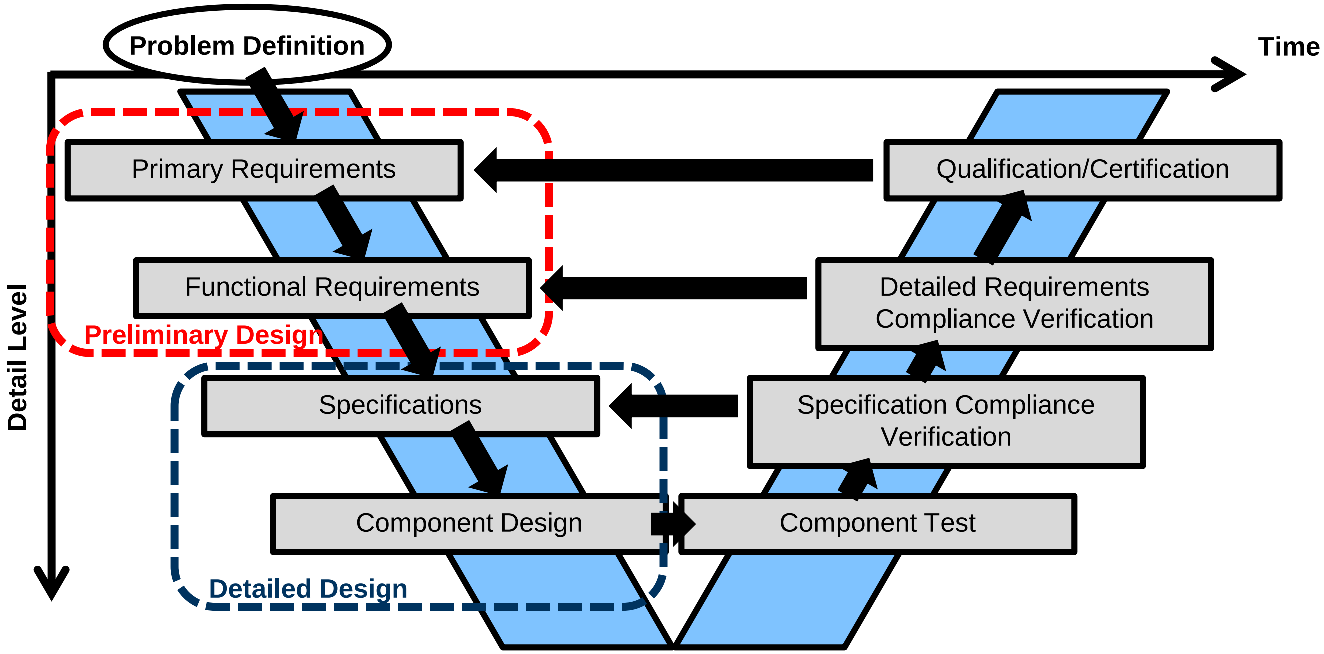
To determine the significant design parameters early on in the process, in order to provide a balanced (optimal) design that complies with all component requirements.

⇒ The **preliminary design** is a part of the design process which as a result leads to a **formulation of specifications for the detailed design**, all in accordance with the initial Defined Requirements.

2.1 Helicopter Pre-Design

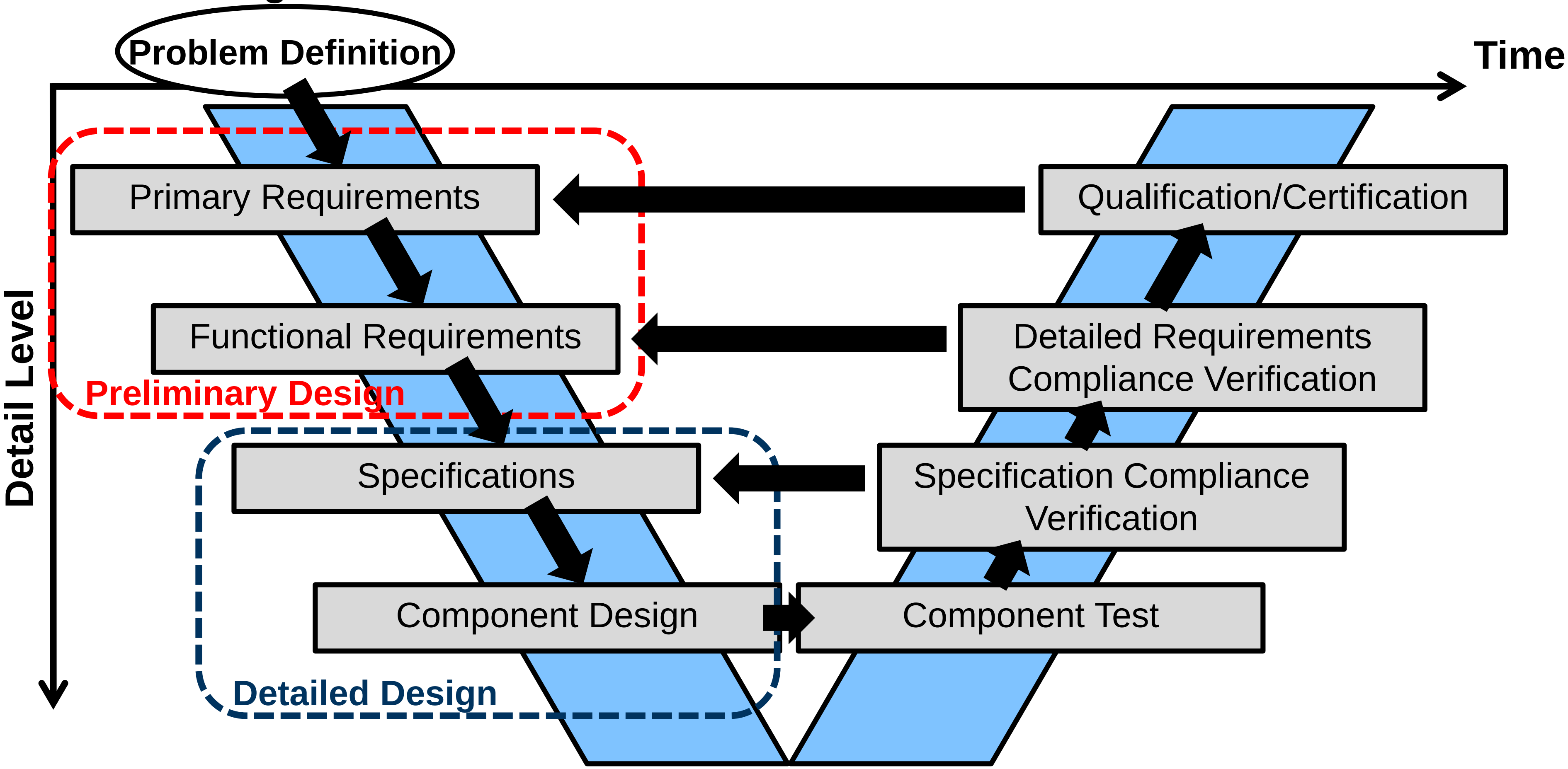
Placement of the Preliminary Design in the Design Process – The V-Model:

- Procedure model for planning and the execution of system development projects
- V-shaped presentation project elements: specification und disassembly in descending branch, realization and integration in the ascending branch



2.1 Helicopter Pre-Design

The Design Process in a V-Model:



2.1 Design Requirements

Classification of the Preliminary Design within the Design Process–

The Design Process in a V-Modell :

- The design process is iterative; every iteration step converges towards a balanced and optimal design (see chapter 1.1).
- By using the V-Model it is possible to break down the process by time and level of detail, which can be used for the purpose of project management.
- The V-Modell in helicopter engineering divides the design process into the development of an increasingly detailed design in the descending branch and the validation of the design (not covered in this lecture) in the ascending branch.

⇒ The **Preliminary Design's** place in the V-Model is in the descending branch before the Detailed Design. It serves as a means of **finding a design** based on the **Problem Definition** as well as the **primary** and **functional requirements**.

2 Design Requirements

2.2 Identification of Requirements

2.2 Identification of Requirements

The Design Process in a V-Modell – Problem Definition:

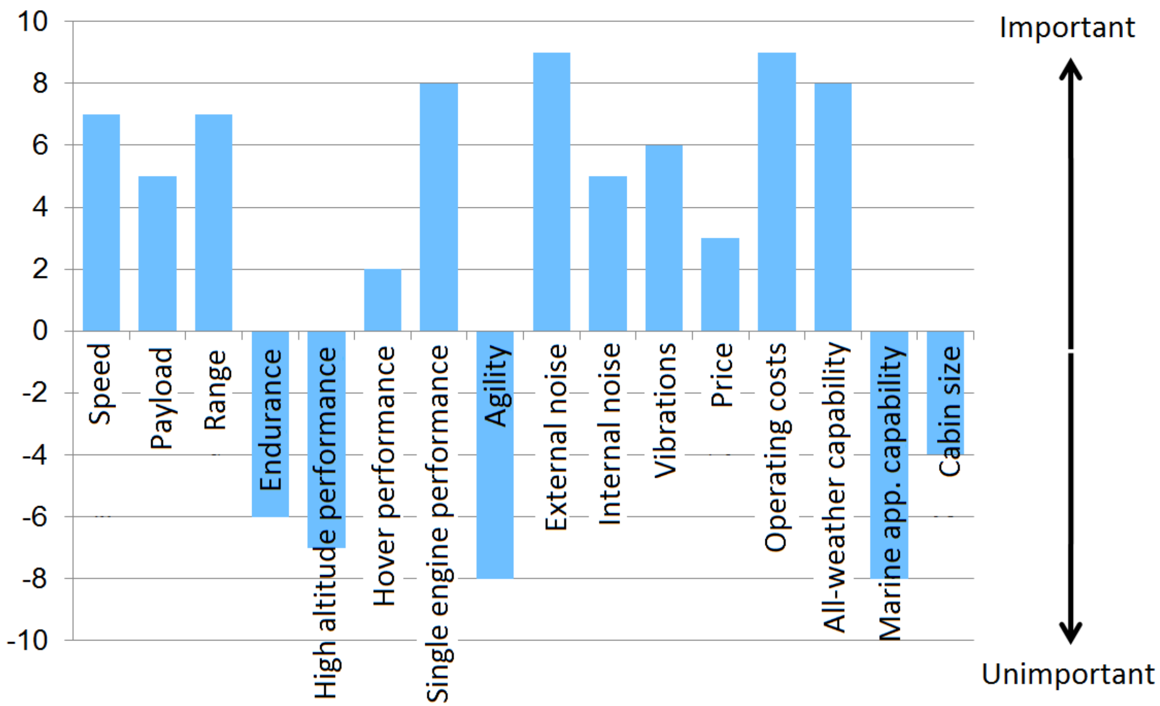
An Analysis of the Problem Definition should result in a Requirement Catalogue:

- What is the **main field of application** of the helicopter being developed?
- What type of **missions** should the helicopter carry out? (i.e. transport, surveillance, combat,…)
- **Who operates** the helicopter? (i.e. police, military, offshore-operator, civilian operator,…)
- Which **requirements** arise out of **mission profiles**? (i.e. exceptionally high power requirement, special equipment, size, environmental conditions,…)
- Should an **existing helicopter** be replaced or its weaknesses eliminated?

2.2 Identification of Requirements

The Design Process in a V-Modell – Problem Definition:

For every element of the Requirements Catalogue there can be only one **weighting factor** (based on customer requirements) and also so called **design drivers** can be identified.



2.2 Identification of Requirements

The Design Process in a V-Modell – Primary Requirements:

With the help of the Requirements Catalogue and the weighting factors one may now determine:

- which **requirements** arise for the payload, range, equipment,… of the helicopter being developed from its **field of application** and its **mission profiles.**
- whether there are **critical mission requirements** that outweigh standard requirements such as range etc.
- whether **requirements are governed by trends,** based on technology and strategy development (or will change in the future).

Based on these observations and especially on the customer's weighting factors the **Primary Requirements** for the design can be derived.

2.2 Identification of Requirements

The Design Process in a V-Modell – Functional Requirements:

- Performance benchmarks: speed, range, rate of climb,…
- Agility: load factor, control response, yaw rate,…
- Payload requirements: volume, surface load, exterior loads,…
- Environment: icing, poor visibility, sand, sea water,…
- Equipment requirements: sensors, weaponry, rescue equipment,…

⇒ Functional requirements are demands towards what the design can do and its properties, not technical solutions!

2 Design Requirements

2.3 Example Cases

2.3 Example Cases

Problem definition:

Development of a replacement for the light transport helicopter Bell UH-1, the most successful helicopter in the world with over 10000 units produced.

© TSgt. Rod Prouty, USAF

Technical Details Bell UH-1:

Max. load capacity:	1200kg (2 pilots + 12 soldiers)
Max. endurance:	2h 30min
Max. range:	500km (without reserve)
MTOW:	4309kg
Ø Rotor:	14,50m
Power installed:	808kW

2.3 Example Cases

Problem definition:

- The UH-1 is a cost-effective, single engine helicopter of the all-purpose transport helicopter class.
- Weaknesses: poor engine performance in high ambient temperature, low cruising speed, high noise and level of vibrations

$\Rightarrow$ **Possible options for the design of a replacement:**

- – Cost-efficient transporter with moderate performance
- – High performance design with increased power available
- – …

2.3 Example Cases

Problem definition– Market situation:

- Already in operation as replacement: **Sikorsky UH-60 Blackhawk**

© SSgt. Suzanne M. Jenkins, USAF

⇒ Two engines with high power reserves at high ambient temperatures (more than double the power)

⇒ Folding rotor, stabilizers and tail boom suitable for transport in nearly any transport aircraft in the USA

⇒ High cruising speed (≈300km/h) and low silhouette for better transportability

2.3 Example Cases

Problem definition– Market situation :

- Eurocopter (formerly Aérospatiale) offers the twin-engined **SA 330 Puma** and its performance-enhanced upgrade **AS 332 Super Puma MK2 an.**

© Molch-Entertainment

© SSgt. Fernando Serna, USAF

2.3 Example Cases

Problem definition– Comparison of technical Details (light and medium-weight transport helicopters, 4-10t):

	UH-1D	SA 330 G Puma	AS 332 Super Puma MK II	UH-60 Blackhawk
Main dimensions				
Length (with rotor)	17,40m	18,22m	19,50m	19,76m
Fuselage length	12,77m	16,00m	16,79m	15,25m
Total width	2,76m	3,00m	3,86m	4,38m
Fuselage width	2,61m	2,00m	2,00m	2,34m
Height (with tail rotor)	4,48m	5,14m	4,97m	5,13m
Rotor diameter	14,50m	15,00m	16,20m	16,36m
Cabin/Cargo hold				
Length	2,34m	4,68m	7,87m	3,84m
Width	2,44m	1,80m	1,80m	2,34m
Height	1,33m	1,55m	1,55m	1,37m
Volume	6,2m³	11,4m³	15,5m³	11,6m³
Engines	1x Lycoming T53-L11	2x Turmo IVC	2x Makila 1A2	2x GE T701
Power (TOP)	808kW	1115kW	1376kW	1342kW
Weight				
Take-off weight	4309kg	7000kg	9300kg	9980kg
Empty weight (with equipment)	2300kg	3766kg	4660kg	5343kg
Load capacity	2009kg	3234kg	4640kg	4637kg
Propellant	665kg	1220kg	1595kg	1075kg
Payload	1344kg	2014kg	3045kg	3562kg
Range (without reserve)	511km	600km	780km	585km

2.3 Example Cases

Primary Requirements– Payload, Range, Equipment:

The UH-1D replacement will take over the task of airborne and commando operations with groups of 10-12 soldiers. Things to take into account:

- The amount of equipment carried by soldiers increases continuously
- Airborne assault troops may be provided with light vehicles
- Material is to be transported to the deployment location without transshipment (range with reserves min. 500km)
- The threat of being attacked during a mission necessitates the application of night vision equipment as well as effective defense systems

Possible **Payload**:

- 14 soldiers with combat packs and body armor (at 110kg each) 1540kg
- Light tactical vehicle with crew 2000kg
- Range of ammuntion 1000kg

2.3 Example Cases

Primary Requirements– Missions:

Fulfilling specific mission requirements is crucial to the helicopter design. For example: a short mission with a high payload may be more critical than long range.

Possible Missions:

- Transport of 14 soldiers for a mission duration of 2h 30min
- Range of 500km with a 30min reserve at 2000kg payload


© EUROCOPTER, Jérome Deulin

2.3 Example Cases

Functional Requirements:

Performance benchmarks: **At reference conditions 1000m, ISA+15K and mission weight**

- Range >500km
- Maximum top speed: >300km/h
- Power reserve at HOGE with TOP: ≈20 %
- OEI rate of climb at intermediate power: ≈1m/s

Payload requirement:

- Max. 2000kg or 14 soldiers with luggage, vehicle shipment

Agility requirement:

- Load factor n_z at v_h ≈1,5 g

ISA	Intern. Standard Atmosphere	**HOGE**	Hover out of ground effect
TOP	Take-off power	**OEI**	One engine inoperative
n_z	Load factor in z-axis	**v_h**	Flight speed at maximum continuous power

2.3 Example Cases

Functional Requirements:

Environmental Requirements:

- Operation at sea (salt water, floaters)

Equipment Requirements:

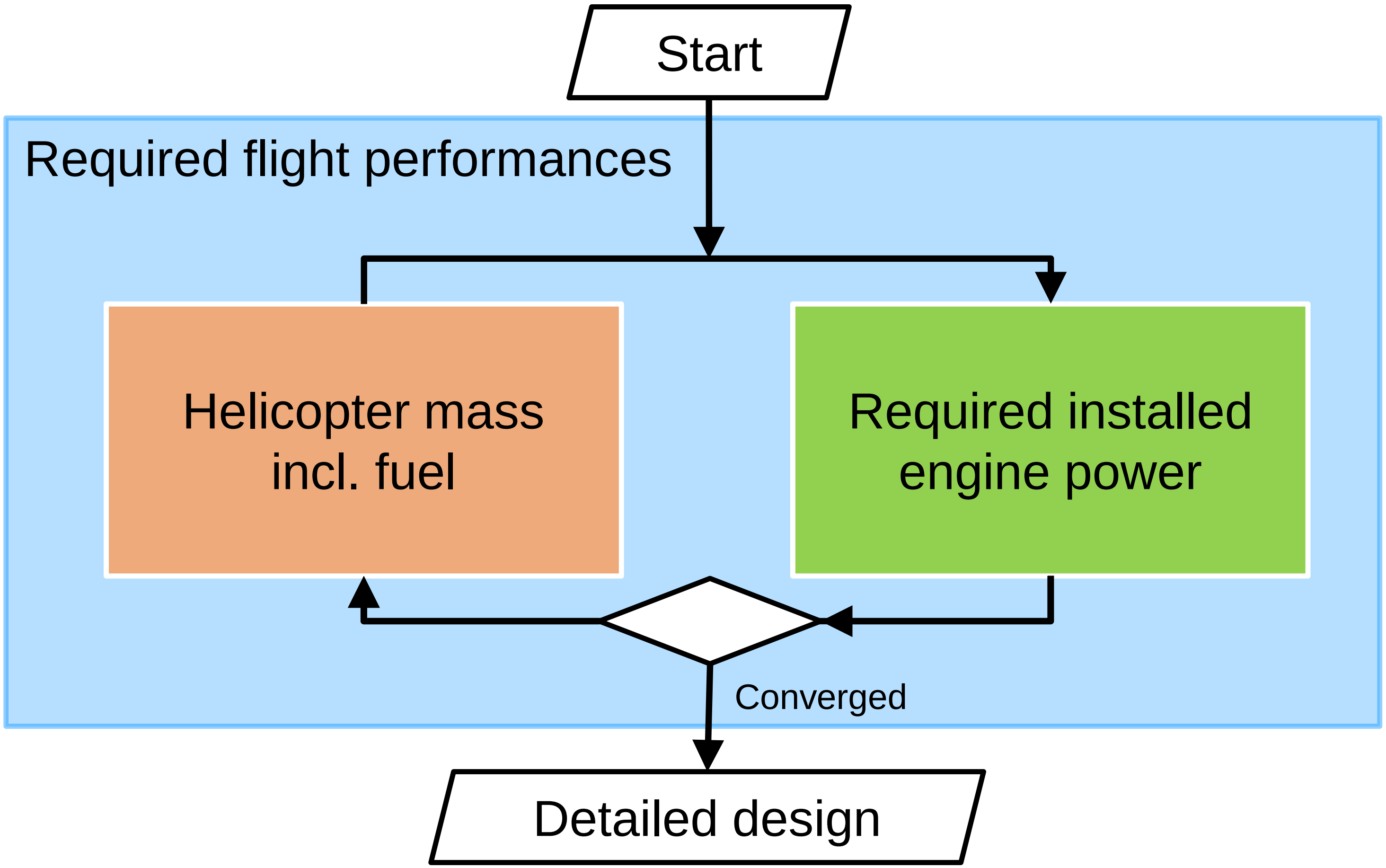
- Autonomous navigation, Instruments for low altitude flight at night, Defense systems


© EUROCOPTER, Eric Raz

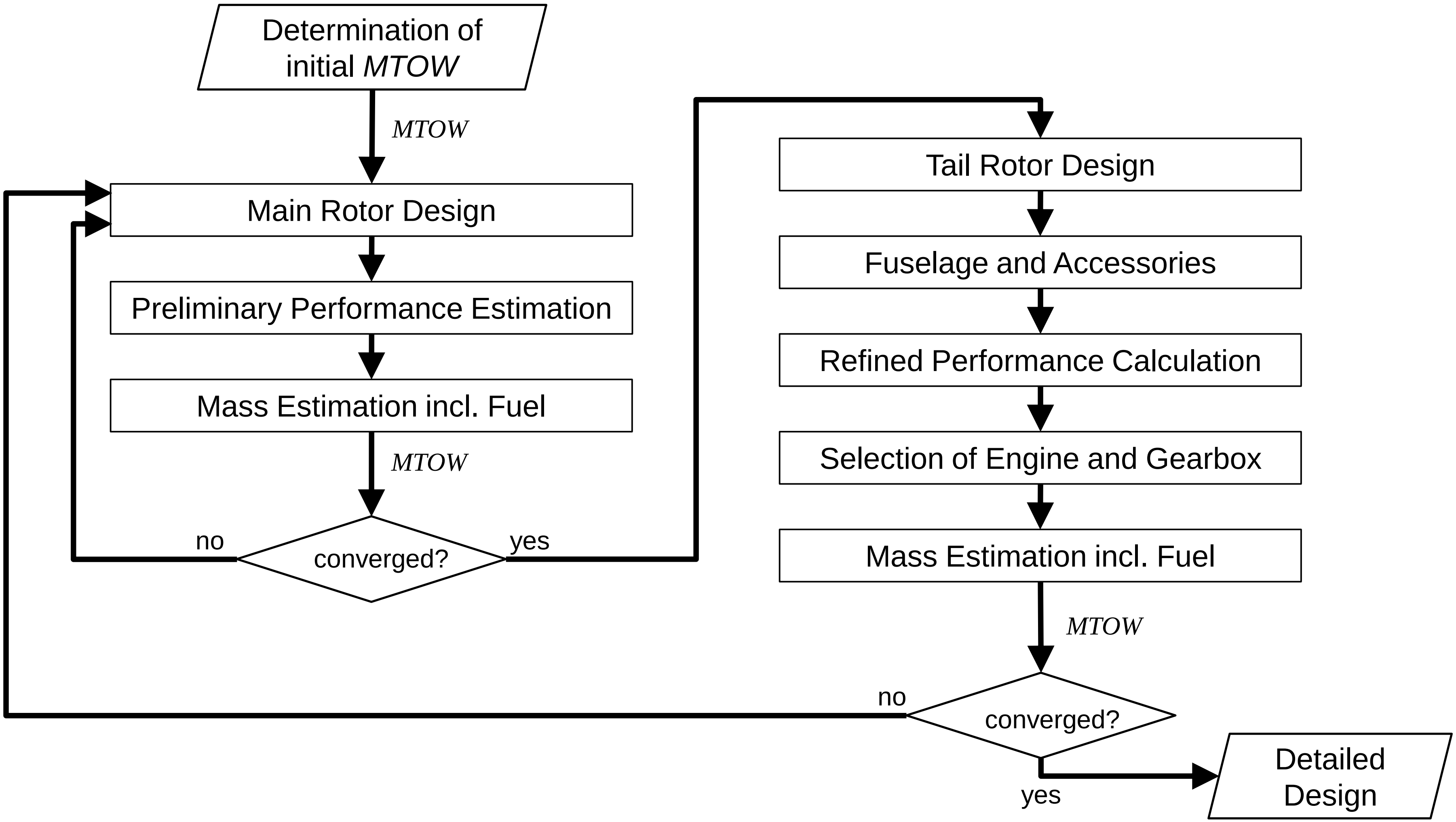
3 Sizing

3 Sizing - Overview

Iterative process in preliminary design phase



3 Sizing - Overview



3 Sizing

3.1 Determination of initial MTOW
3.2 Main Rotor Design
3.3 Preliminary Performance Estimation
3.4 Tail Rotor Design
3.5 Fuselage and Accessories
3.6 Refined Performance Calculation
3.7 Selection of Engine and Gearbox

3 Sizing

3.1 Determination of initial MTOW

3.1 Determination of initial MTOW

Sought is the assumed helicopter mass, which is used to start the design process. The required *maximum take off weight* $MTOW$ will be therefore at first estimated empirically.

The result of the Sizing will not be influenced by the choice of the start value. A favourable selected start value however reduces the number of necessary iterations in the design process.

For initial estimates of the $MTOW$, values of helicopters with similar tasks can be used for instance. Alternatively, required payload and mission endurance can be considered already for the $MTOW$ start value, like illustrated subsequently (according to Prouty).

3.1 Determination of initial MTOW

1. Estimate of installed power P_{av} on the basis of existing helicopter models with similar mission,
   e.g. with similar requirements concerning payload, maximum speed, hover ceiling, range, maximum endurance

2. Estimate of required mass of fuel with an assumed mission time $t_{mission}$:

$$m_{Fuel} = 0{,}25 \frac{kg}{kWh} \cdot P_{av} \cdot t_{mission}$$

3. Determination of *useful load* UL from masses for required payload, crew and fuel (mass for oils and fuel remains disregarded):

$$UL = m_{Payload} + m_{Fuel} + m_{Crew}$$

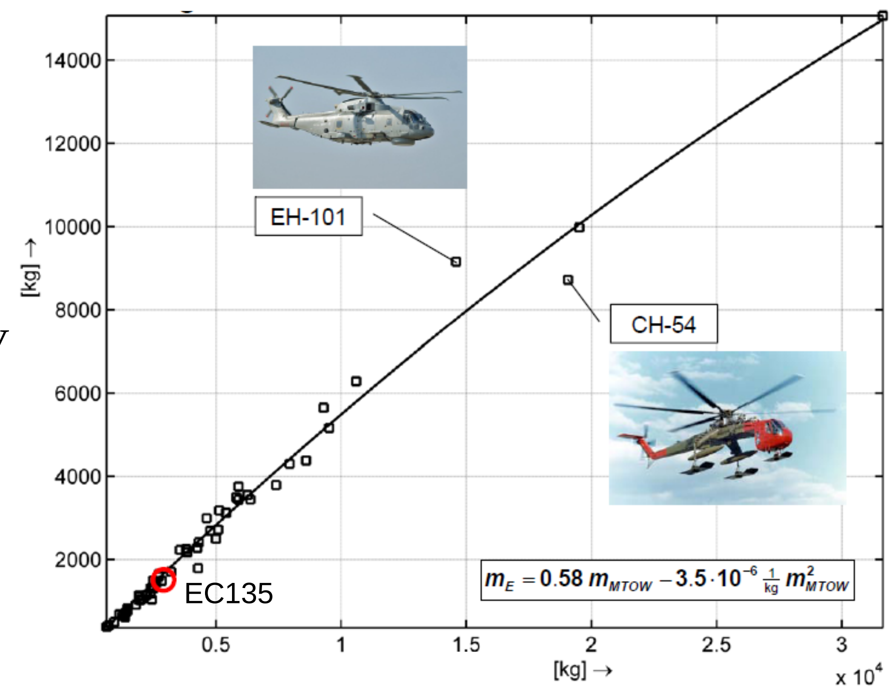
3.1 Determination of initial MTOW

3. Estimate of empty weight ratio $\frac{EW}{MTOW}$ of existing helicopter types

$$\frac{EW}{MTOW} \approx 0{,}5$$

with *empty weight* EW

EW



$MTOW$

4. Calculation of start value

$$MTOW = UL + MTOW \cdot \frac{EW}{MTOW}$$

$$MTOW = \frac{UL}{1 - \frac{EW}{MTOW}}$$

Figure: DLR, Dr. Keßler

3 Sizing

3.2 Main Rotor Design

3 Sizing

3.2. Main Rotor Design

- Parameters
- Radius of main rotor
- Rotational speed
- Solidity
- Number of blades
- Blade geometry
- Airfoil selection

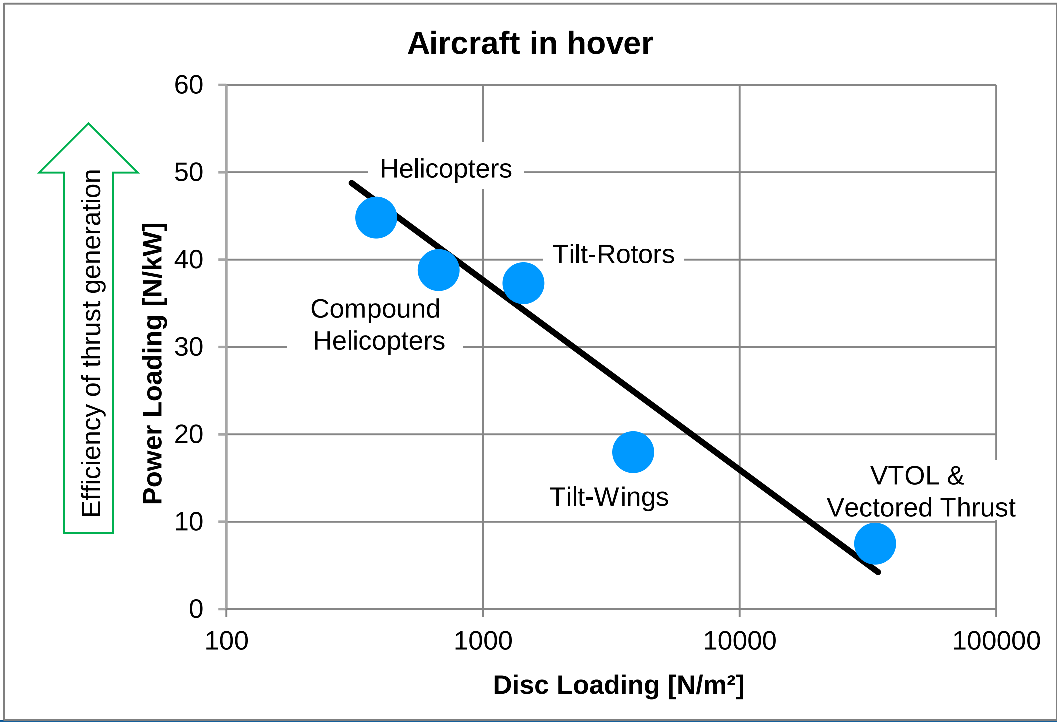
3.2. Main Rotor Design: Parameters

- Disc loading

$$DL = \frac{thrust}{disc\ area} = \frac{T}{A} = \frac{T}{\pi \cdot R^2}$$

- Power loading

$$PL = \frac{thrust}{power\ consumption} = \frac{T}{P}$$



Data from Leishman: *Principles of Helicopter Aerodynamics*

3.2. Main Rotor Design: Parameters

- *Tip speed*

$$V_{TIP} = R \cdot \Omega$$

- *Thrust coefficient*

$$C_T = \frac{T}{\rho \cdot A \cdot V_{TIP}^2}$$

- *Power coefficient*

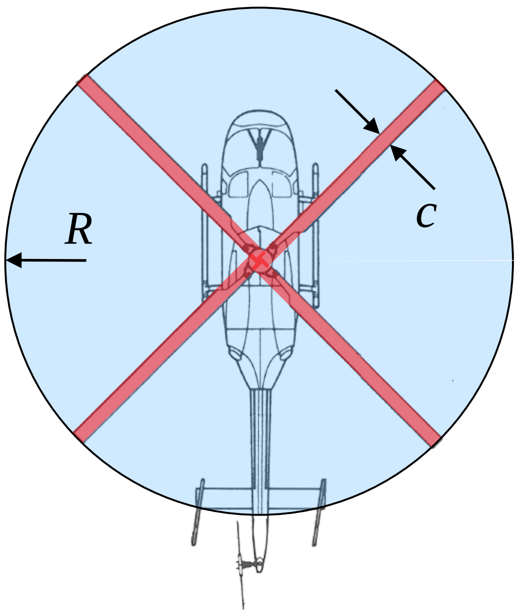
$$C_P = \frac{P}{\rho \cdot A \cdot V_{TIP}^3}$$

3.2. Main Rotor Design: Parameters

- *Solidity*

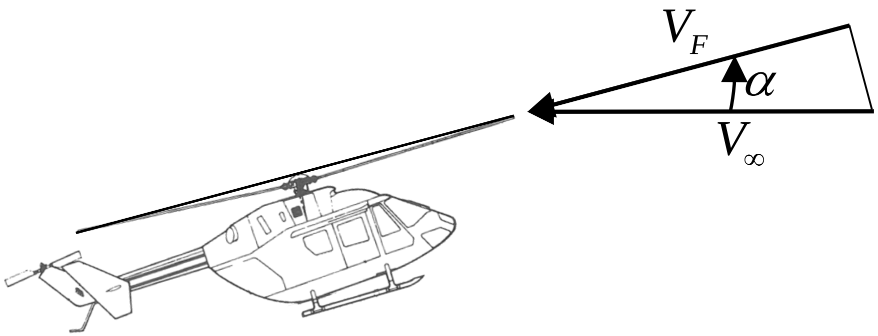
$$\sigma = \frac{blade\,area}{disc\,area} = \frac{N_b \cdot c \cdot R}{A} = \frac{N_b \cdot c}{\pi \cdot R}$$

with number of rotor blades N_b



- *Advance ratio*

$$\mu = \frac{V_F}{V_{TIP}} = \frac{V_\infty \cdot \cos\alpha}{V_{TIP}}$$



3 Sizing

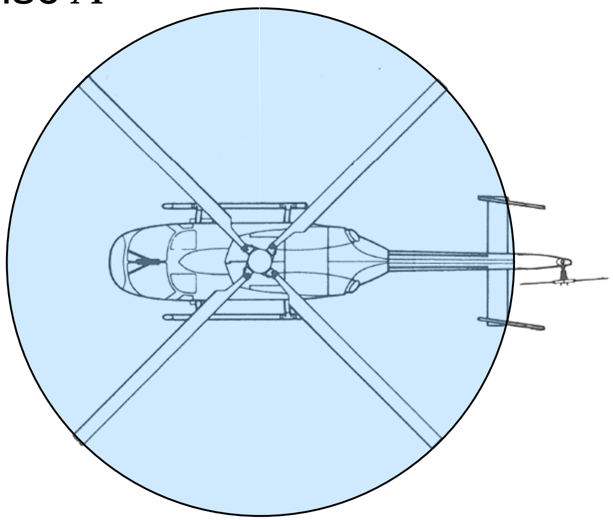
3.2. Main Rotor Design

- ✓ Parameters
- ➔ Radius of main rotor
- Rotational speed
- Solidity
- Number of blades
- Blade geometry
- Airfoil selection

3.2. Main Rotor Design: Radius

The rotor radius R defines the area of the rotor disc A

$$A = \pi \cdot R^2$$

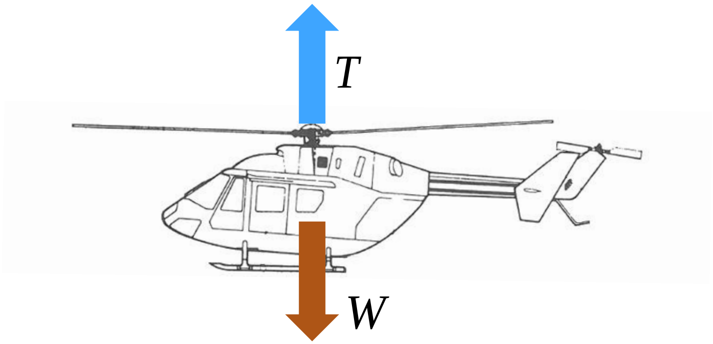


with the required thrust T it results in the *disk loading* DL

$$DL = \frac{T}{A}$$

For the steady flight is assumed that the required thrust equals the weight of the helicopter W

$$DL = \frac{T}{A} = \frac{W}{A}$$



3.2. Main Rotor Design: Radius

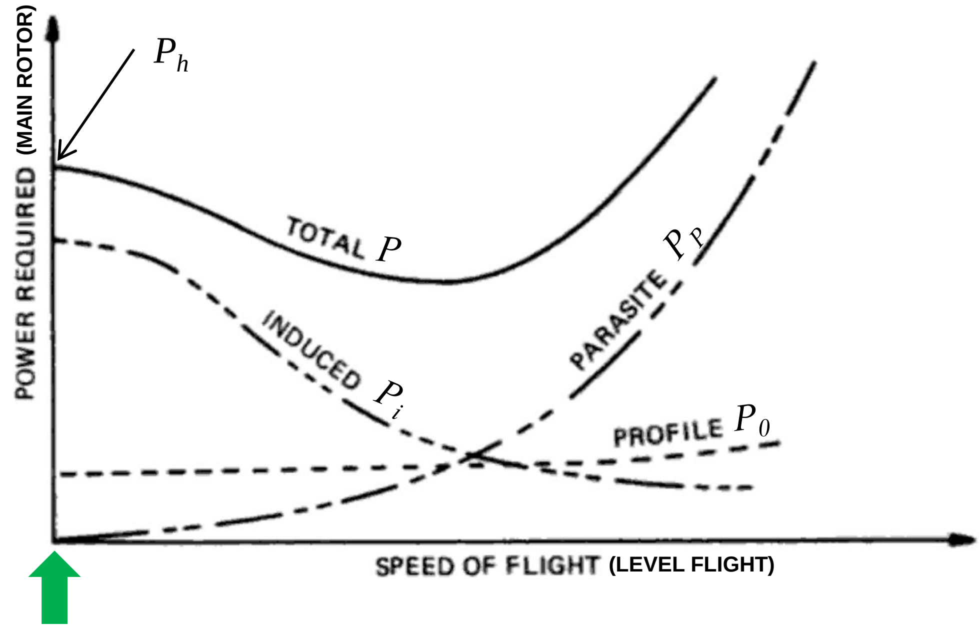
Requirement of low power consumption in hover P_h

- Total power P over speed of flight features a local maximum in hover

- Power requirement defined by induced power P_i and profile power P_0

$$P_h = P_i + P_0$$

- Parasite power P_P is irrelevant here, is only slightly influenced by Main Rotor Design



➔ High power loading PL favoured

$$PL = \frac{T}{P} \overset{\text{hover}}{=} \frac{W}{P}$$

Figure from Stepniewski: *Rotary-Wing Aerodynamics*

3.2. Main Rotor Design: Radius

The **induced power** $\boldsymbol{P_i}$, which flows off with the stream as kinetic energy, regarding the elementary theory of the screw propeller amounts in hover ideally ($P_{i_{id}}$) to:

$$P_{i_{id}} = T \cdot v_h = \sqrt{\frac{T^3}{2\rho\pi R^2}} = \sqrt{\frac{W^3}{2\rho\pi R^2}} = \sqrt{\frac{W^3}{2\rho\pi}} \cdot \frac{1}{R}$$

Rotor flow velocity (hover)

$$v_h = \sqrt{\frac{T}{2\rho A}}$$

The actual induced power is higher because of various effects:

- Flowed through area smaller than πR^2:
  rotor hub zone for thrust generation ineffective, tip ineffective
- Rotation of rotor introduces swirl
- Inhomogeneous induced velocity

➔ Implementation of an empiric correction factor κ for the induced power

$$P_i = \kappa \cdot P_{i_{id}} = \kappa \cdot \sqrt{\frac{T^3}{2\rho\pi}} \cdot \frac{1}{R}$$

typically $\kappa \approx 1{,}15$

➔ Rotor radius R⇧ for low P_i

3.2. Main Rotor Design: Radius

Besides the reduction of the induced power, low induced downwash velocities v_i offer further advantages:

- Low download of helicopter components (fuselage, fittings…) in rotor stream
- Facilitates the work in rotor downwash (e.g. rescue hoist operation)
- Favourable for starts and landing on loose surface (reduction of *brownout*)

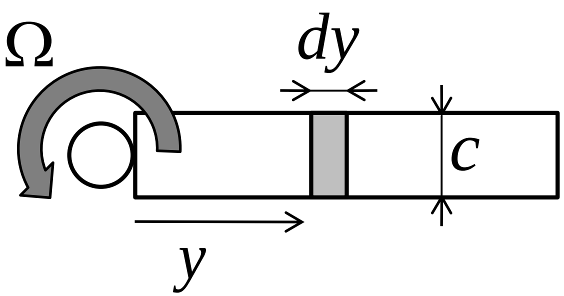
So with $v_h = \sqrt{\frac{DL}{2\rho}}$ is also here a low disc loading beneficial

➔ Rotor radius R⇧

Photos: Eurocopter, Rémy Michelin; 3rd Marine Aircraft Wing Public Affairs

3.2. Main Rotor Design: Radius

The **profile power** P_0 for overcoming the blade drag is calculated in first approximation via a constant airfoil drag coefficient C_{d0} and the rotatory motion in the rotor disc. It is assumed that the profiled blade with constant chord length c extends to the rotation axis.



$$P_0 = \Omega \cdot N_b \cdot \int_0^R \frac{\rho}{2} \cdot (\Omega \cdot y)^2 \cdot c \cdot C_{d0} \cdot y \, dy$$

$$= \Omega^3 \cdot N_b \cdot \frac{\rho}{2} \cdot c \cdot C_{d0} \cdot \int_0^R y^3 \, dy$$

$$= \frac{1}{8} \cdot \rho \cdot N_b \cdot \Omega^3 \cdot c \cdot C_{d0} \cdot R^4$$

tip speed $V_{TIP} = \Omega \cdot R$

solidity $\sigma = \dfrac{N_b \cdot c}{\pi \cdot R}$

$$P_0 = \frac{1}{8} \cdot \rho \cdot V_{TIP}{}^3 \cdot \sigma \cdot C_{d0} \cdot \pi \cdot R^2$$

➔ Rotor radius R⇩ for low P_0 at constant V_{TIP} and σ

3.2. Main Rotor Design: Radius

When summing up the approximations for induced power P_i and profile power P_0 we get the total power in hover P_h

$$P_h = P_i + P_0$$

$$P_h = \kappa \cdot \sqrt{\frac{T^3}{2\rho\pi}} \cdot \frac{1}{R} + \frac{1}{8} \cdot \rho \cdot V_{TIP}{}^3 \cdot \sigma \cdot C_{d0} \cdot \pi \cdot R^2$$

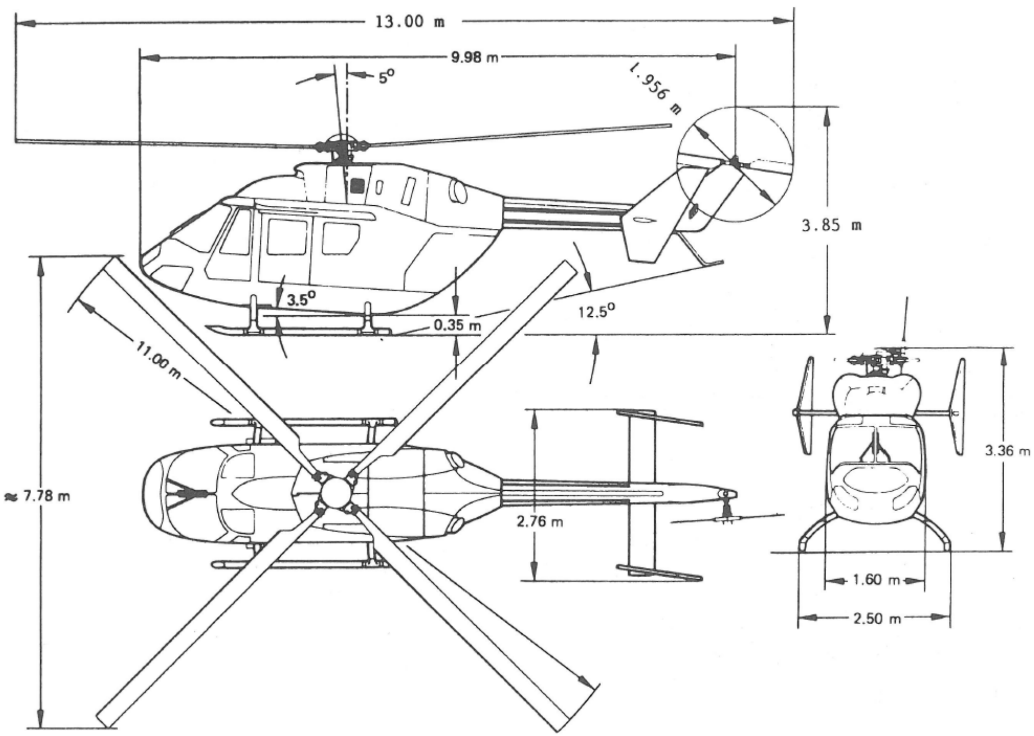
With constant tip speed V_{TIP} and constant solidity σ we can calculate the radius $R_{P_{h\min}}$ which leads to a minimal power requirement

$$P_{h\min} = \kappa \cdot \sqrt{\frac{T^3}{2\rho\pi}} \cdot \frac{1}{R_{P_{h\min}}} + \frac{1}{8} \cdot \rho \cdot V_{TIP}{}^3 \cdot \sigma \cdot C_{d0} \cdot \pi \cdot R^2_{P_{h\min}}$$

$$R_{P_{h\min}} = \frac{1}{V_{TIP}} \cdot \sqrt{\frac{2 \cdot T}{\rho \cdot \pi}} \cdot \sqrt[3]{\frac{\kappa}{\sigma \cdot C_{d0}}}$$

3.2. Main Rotor Design: Radius

Example: MBB/Kawasaki BK117



Tip speed

$$V_{TIP} = 221 \frac{m}{s}$$

Solidity

$$\sigma = 0.074$$

Thrust

$$T = 3200 kg \cdot 9{,}81 \frac{m}{s^2}$$

Air density (1000m ISA)

$$\rho = 1{,}11 \frac{\mathrm{kg}}{\mathrm{m}^3}$$

Assumption:
Empirical correction factor:

$$\kappa = 1{,}15$$

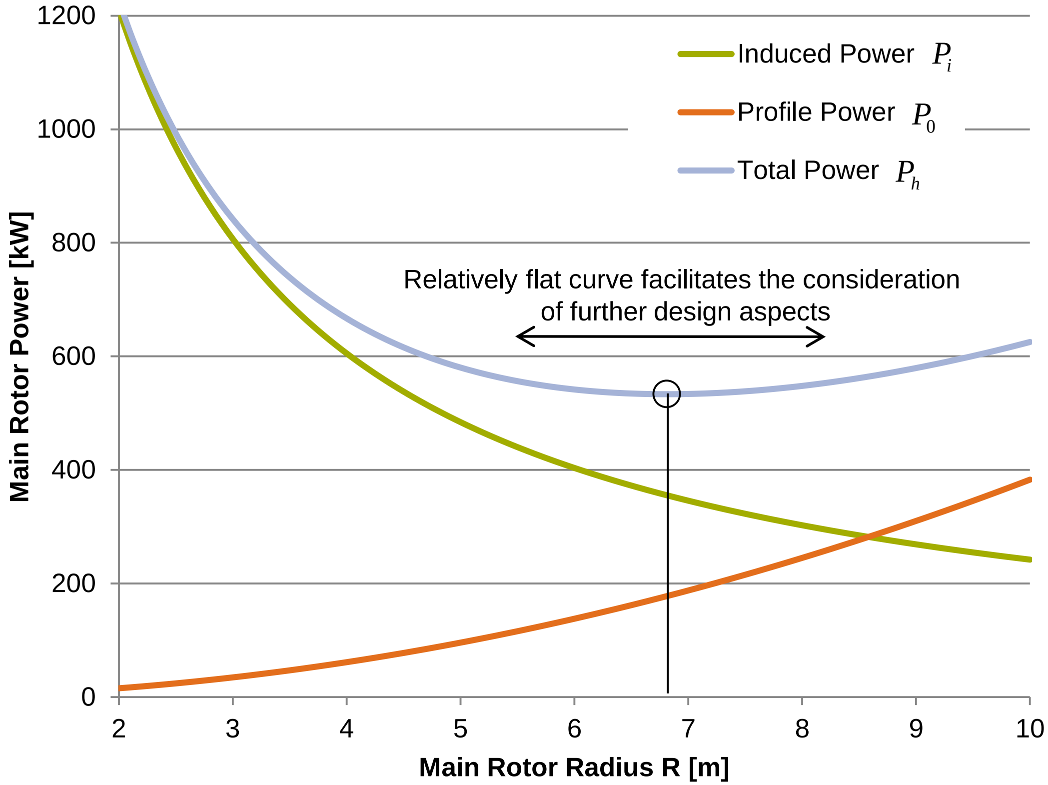
Airfoil Drag Coefficient

$$C_{d0} = 0{,}011$$

Data from Prouty: *Helicopter Performance, Stability and Control*

3.2. Main Rotor Design: Radius

Example: MBB/Kawasaki BK117



$$P_i = \kappa \cdot \sqrt{\frac{T^3}{2\rho\pi}} \cdot \frac{1}{R}$$

$$P_0 = \frac{1}{8} \cdot \rho \cdot \pi \cdot R^2 \cdot V_{TIP}{}^3 \cdot \sigma \cdot C_{d0}$$

$$P_h = P_i + P_0$$

$\rightarrow R_{P_{h\min}} = 6{,}81m$

$\rightarrow DL = \frac{3200kg \cdot 9{,}81\frac{m}{s^2}}{\pi \cdot R^2_{P_{h\min}}} = 215\frac{N}{m^2}$

3.2. Main Rotor Design: Radius

Autorotation

Large rotor radii provide good autorotation characteristics:

- Stored kinetic energy due to high moment of inertia I_R
  - Long reaction time to the transition in the autorotation at power loss
  - Safe flare out to minimize the rate of descent before touchdown

- Low disc loading DL for small steady rate of descent

Evaluation via autorotation index AI (here regarding Sikorsky)

$$AI = \frac{I_R \Omega^2}{2W} \cdot \frac{1}{DL}$$

Large values indicate favourable autorotation characteristics

➔ Rotor radius R⇧

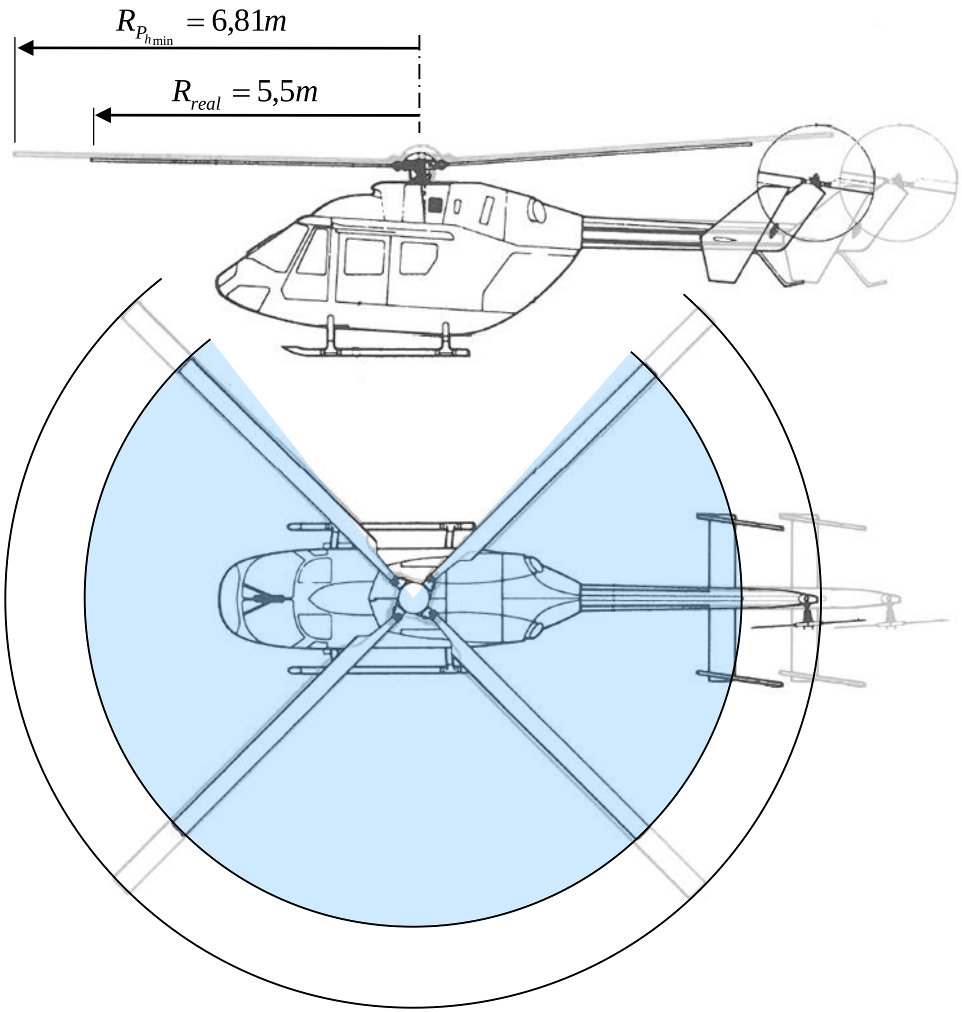
Autorotation AS350

3.2. Main Rotor Design: Radius

Constructional aspects

- Compact dimensions for storage and transport (e.g. aboard ships and aircraft, in hangars)
- Low structure weight
- Position of centre of gravity
- Low manufacturing costs
- Less gearbox weights due to higher revolution speed and lower torques at same V_{TIP}

➔ Rotor radius R ⇩

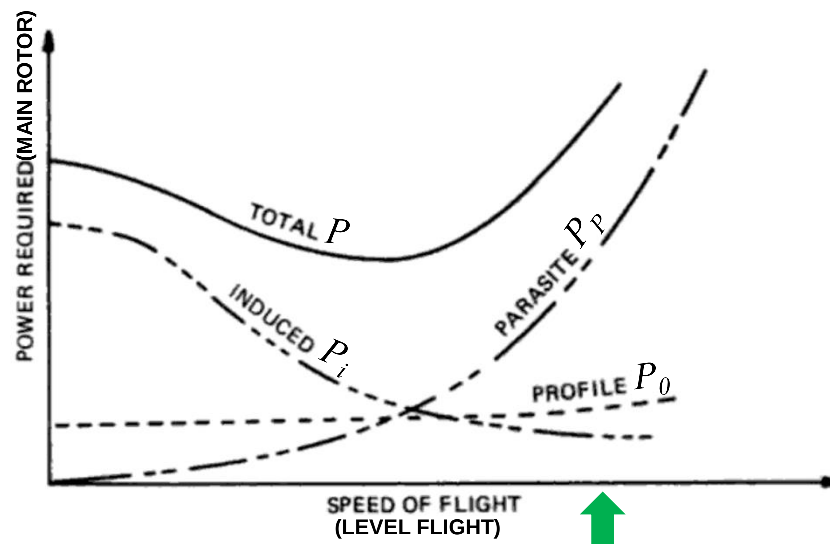


3.2. Main Rotor Design: Radius

More effects:

- Aerodynamic layout of the rotor hub highly limited because of functional requirements
  - ➔ Causes ≈25% of the parasite power P_P
  - ➔ small rotor hubs favourable in forward flight
- Reduction of the profile power P_0 also at high speeds
- Higher agilities by small mass and inertia of the rotor

➔ Rotor radius R ⇩



Figures from Stepniewski: *Rotary-Wing Aerodynamics;* Padfield: *Helicopter Flight Dynamics*

3.2. Main Rotor Design: Radius - Summary

Main rotor radius

R ⇧

- Low induced power
- Low downwash velocity
- Good autorotation characteristics

R ⇩

- Low profile power
- Compact dimensions
- Low weight
- Low manufacturing costs
- High agility
- Small rotor hub drag

Photo: Eurocopter, Rémy Michelin

3.2. Main Rotor Design: Radius

Trend helicopter mass ⇨ rotor radius

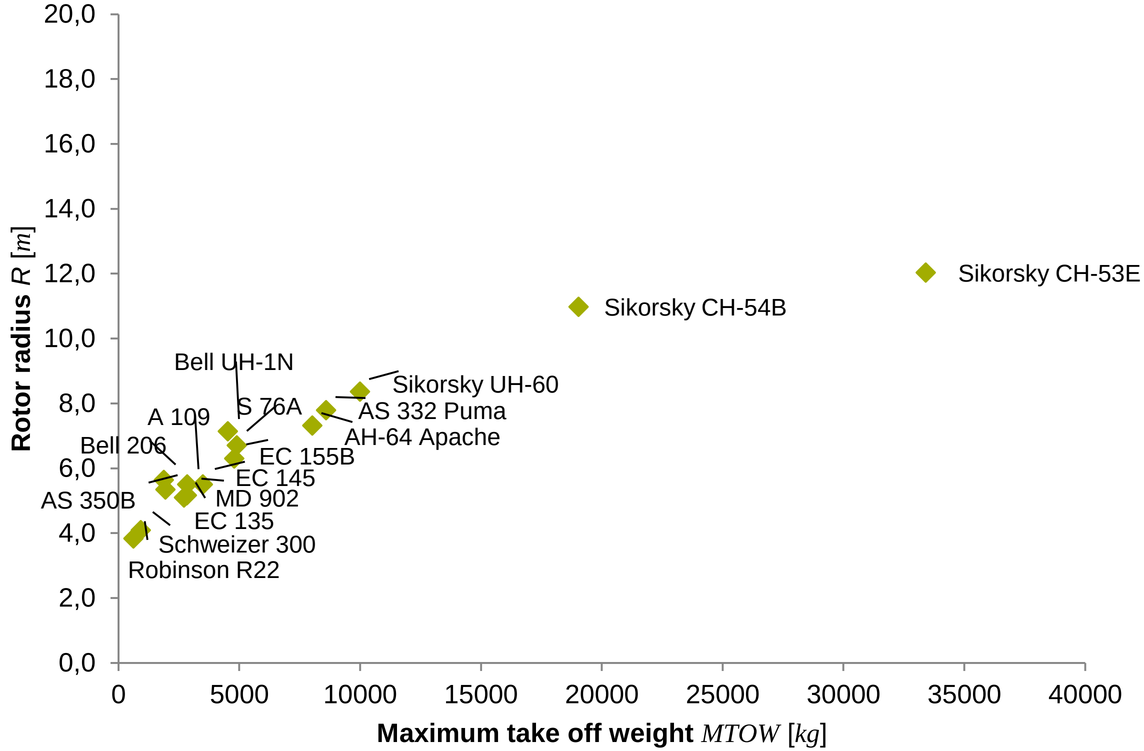
Correlation after power analysis in hover

$$R_{P_{h_{\min}}} \sim \sqrt{T} = \sqrt{W} \quad \text{with} \quad \sigma, \rho, V_{TIP} \text{ const.}$$

Actual implemented rotors follow rather a geometrical scale under consideration of the numerous other boundary conditions.

Assuming constant density of the parts and characteristic length R, here applies

$$W \sim R^3 \quad R \sim \sqrt[3]{W}$$



Data from Leishman: *Principles of Helicopter Aerodynamics;* Layton: *Helicopter Conceptual Design;* Eurocopter

3.2. Main Rotor Design: Radius

Trend helicopter mass ⇨ rotor radius

Correlation after power analysis in hover

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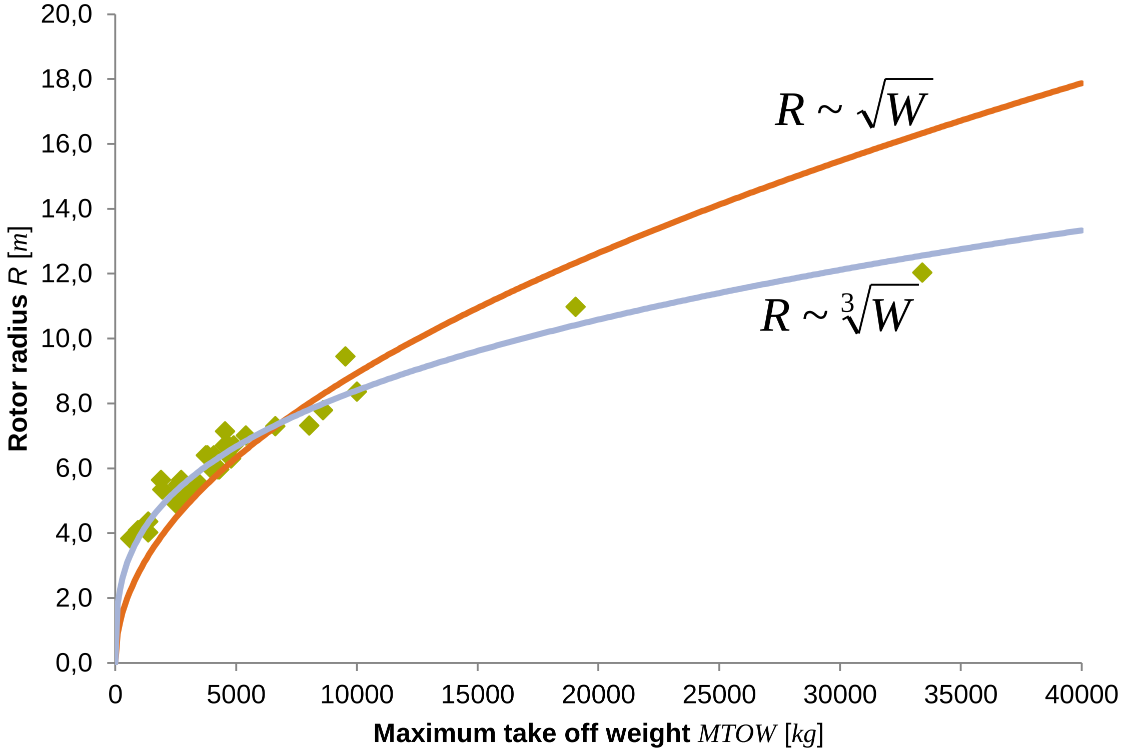


Data from Leishman: *Principles of Helicopter Aerodynamics;* Layton: *Helicopter Conceptual Design;* Eurocopter

3.2. Main Rotor Design: Radius

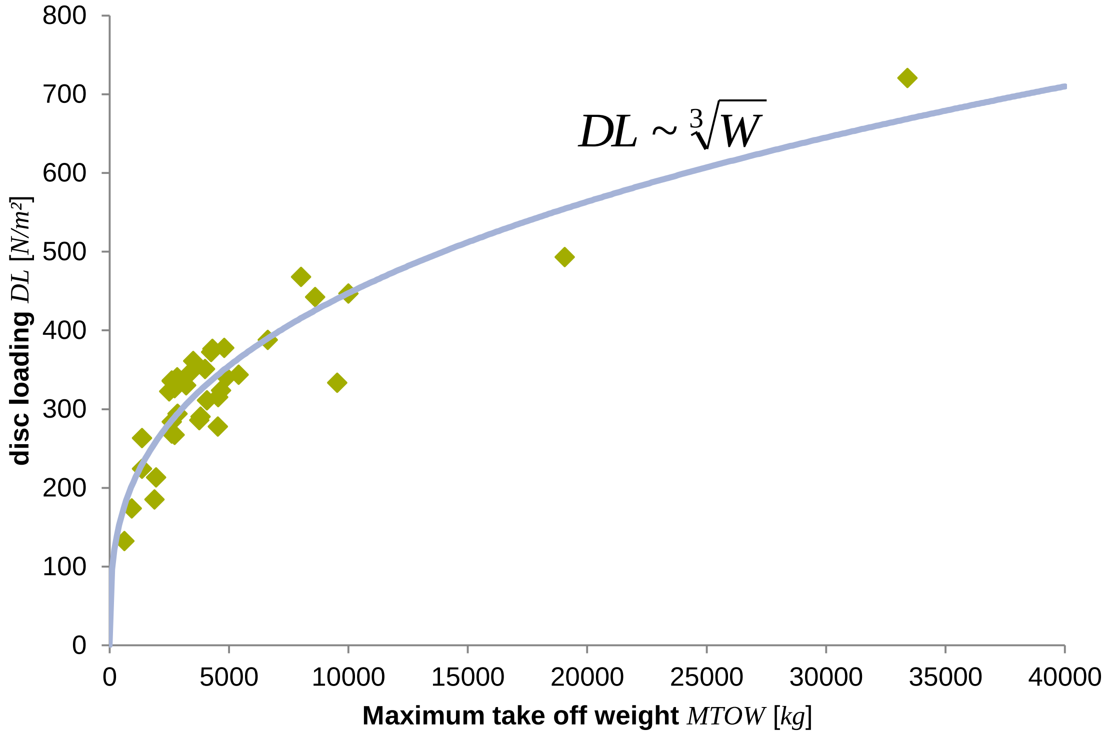
Trend helicopter mass ⇨ disc loading

$$DL = \frac{T}{A} \overset{\substack{hover\\ \downarrow}}{=} \frac{W}{A} = \frac{W}{\pi \cdot R^2} \sim \frac{W}{R^2}$$

Observed correlation between maximum take off weight and rotor radius

$$R \sim \sqrt[3]{W}$$

$$\rightarrow \quad DL \sim \frac{W}{W^{\frac{2}{3}}} = \sqrt[3]{W}$$



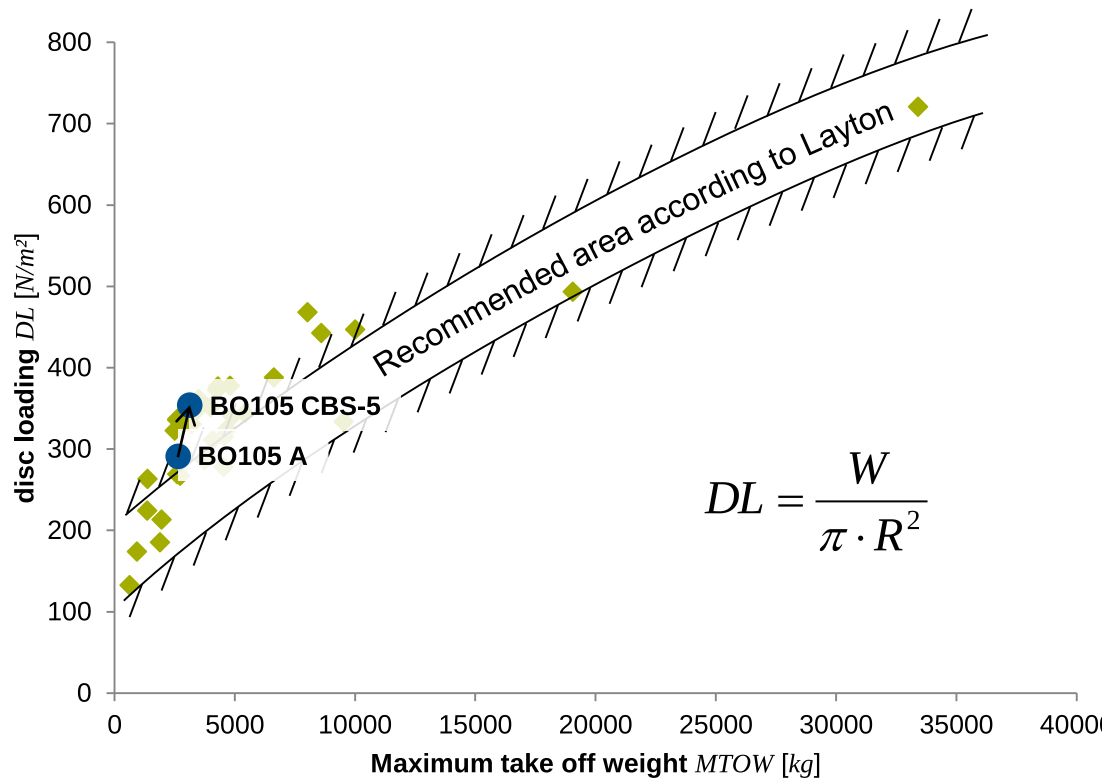
Data from Leishman: *Principles of Helicopter Aerodynamics;* Layton: *Helicopter Conceptual Design;* Eurocopter

3.2. Main Rotor Design: Radius

Definition of rotor radius via disc loading

For the first definition of the rotor radius it is helpful to conform to disc loadings of existing helicopter types in the correspondent weight class

Generally the smallest radius, which complies with all the specified requirements, is chosen.



$$DL = \frac{W}{\pi \cdot R^2}$$

Data from Layton: *Helicopter Conceptual Design;* Leishman: *Principles of Helicopter Aerodynamics;* Eurocopter

3 Sizing

3.2. Main Rotor Design

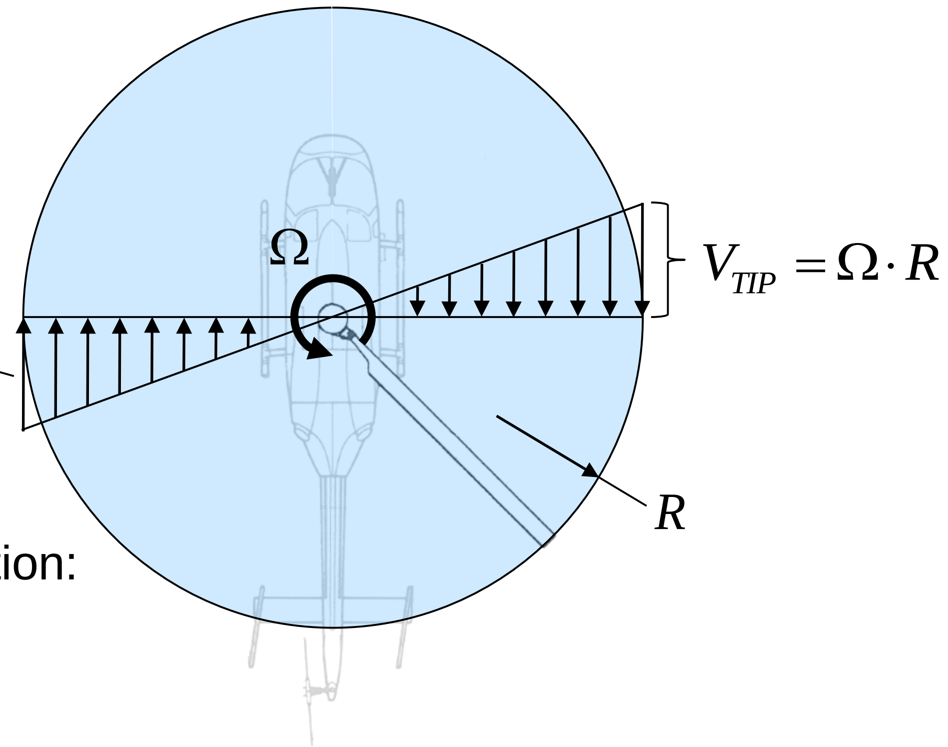
- ✓ Parameters
- ✓ Radius
- ➔ Rotational speed
- Solidity
- Number of blades
- Blade geometry
- Airfoil selection

3.2. Main Rotor Design: Rotational speed

Hover

At given main rotor radius R the rotor turning rate Ω defines directly the tip speed in hover V_{TIP}

The tangential flow velocity on the blade cross sections depends in this case only from the radial position and is constant across the revolution



Established main rotor rotating direction:

France, Russia 

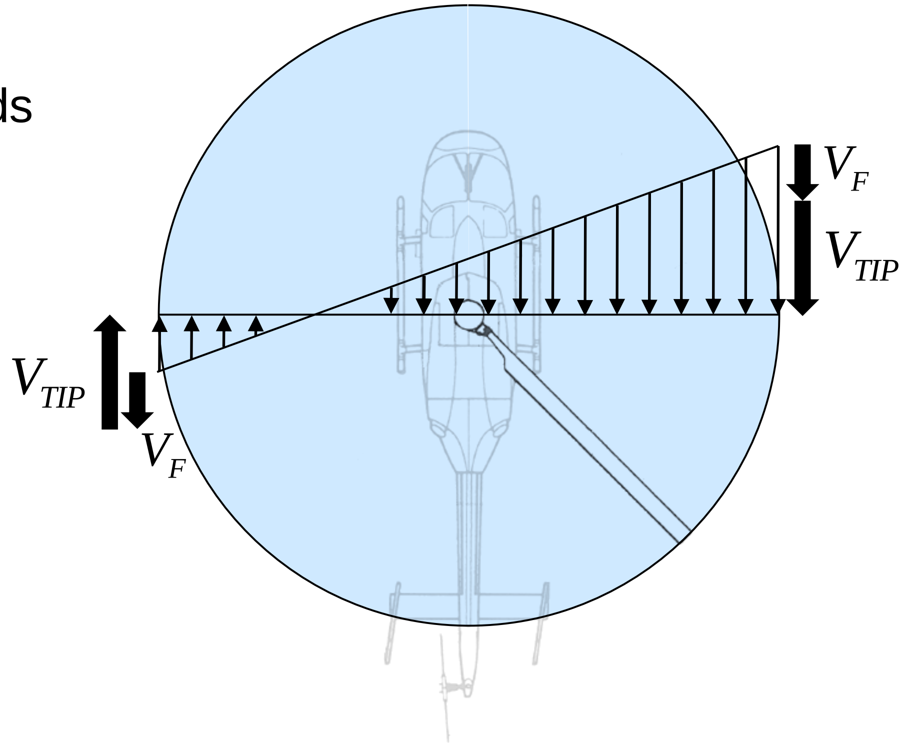
Rest of the world 

3.2. Main Rotor Design: Rotational speed

Forward flight

The rotational speed of the blades is now additionally superimposed by the translational speed of the helicopter in the rotor plane V_F .

With increasing forward velocity this leads to a distinctive asymmetry of the flow conditions on the blade across the revolution.



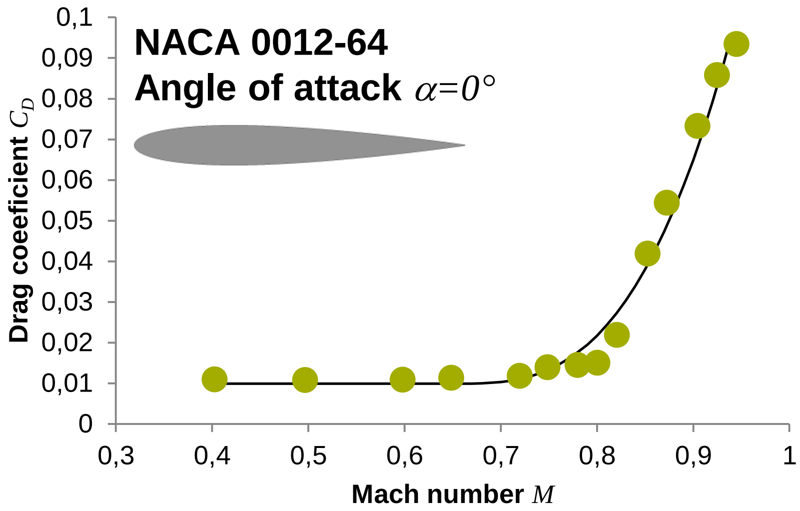
A dimensionless representation of the forward velocity in the rotor plane is the advance ratio μ

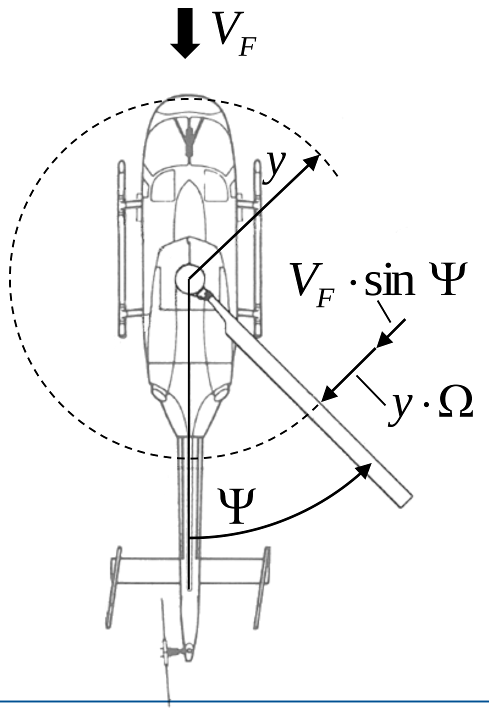
$$\mu = \frac{V_F}{V_{TIP}}$$

3.2. Main Rotor Design: Rotational speed

Due to **compressibility effects**

the drag of airfoils increases from
a certain mach number M_{dd} drastically
(*drag divergence*).





Disregarding the axial stream, for the mach number in the profile cross section applies

$$M_{r,\Psi} = \frac{y \cdot \Omega + V_F \cdot \sin \Psi}{a}$$

with the sonic speed a

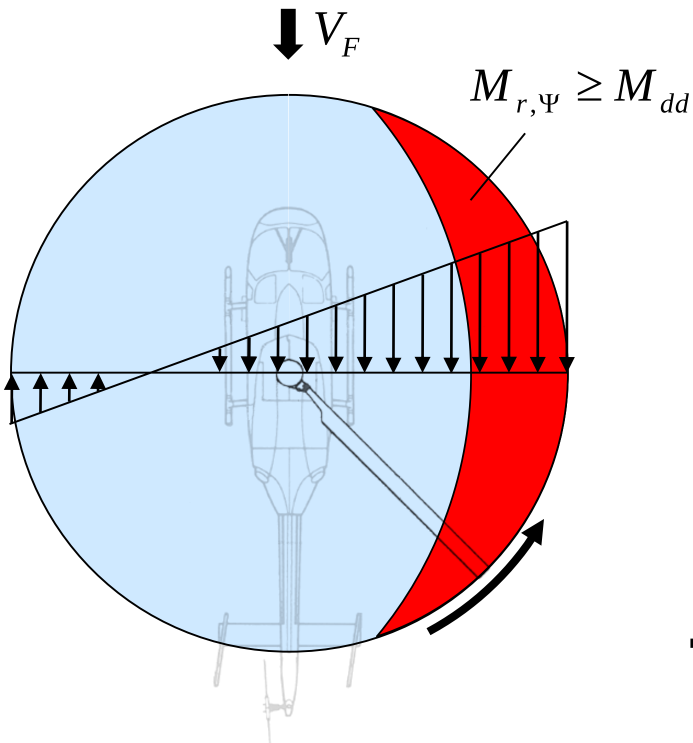
Data from A. Ferri: *NACA Wartime Report L-143*

3.2. Main Rotor Design: Rotational speed

The combination of forward velocity and rotational speed can create an area at the advancing blade, in which the mach number of the stream in direction of the profile exceeds the mach number M_{dd}

➔ strong rise of the profile power P_0

➔ also *Mach Tuck* at even higher mach numbers



$$M_{r,\Psi} = \frac{y \cdot \Omega + V_F \cdot \sin \Psi}{a}$$

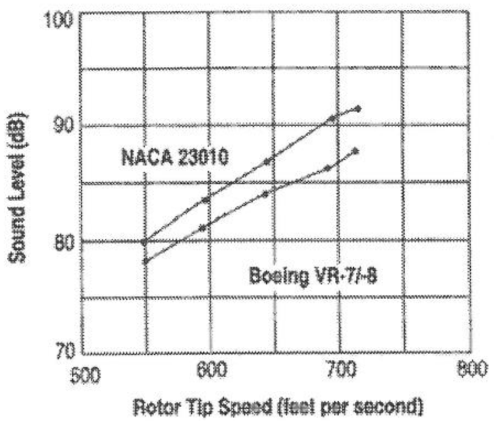
A low rotational speed Ω thus allows higher forward velocities, before compressibility effects become considerable

➔ Rotational speed Ω ⇩

3.2. Main Rotor Design: Rotational speed

The demand of a lower **sound level** restricts the upper area of possible tip speeds.

Rotors with tip speeds V_{TIP} above circa $230\,\frac{m}{s}$ are perceived as unacceptable.



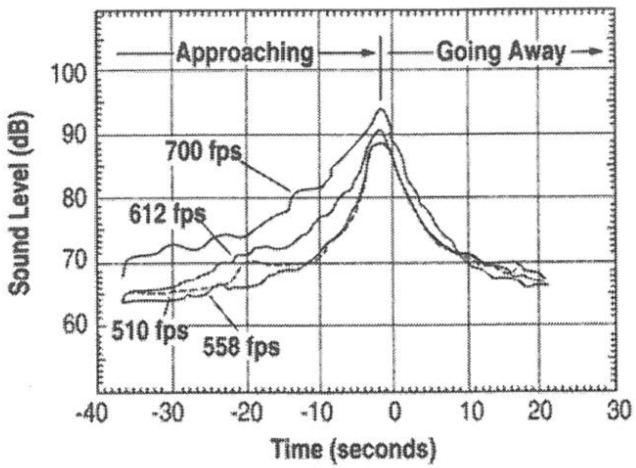


Figure 79-3 Helicopter Noise At Various Rotor Tip Speeds MD 500E During 80-Knot Flyover

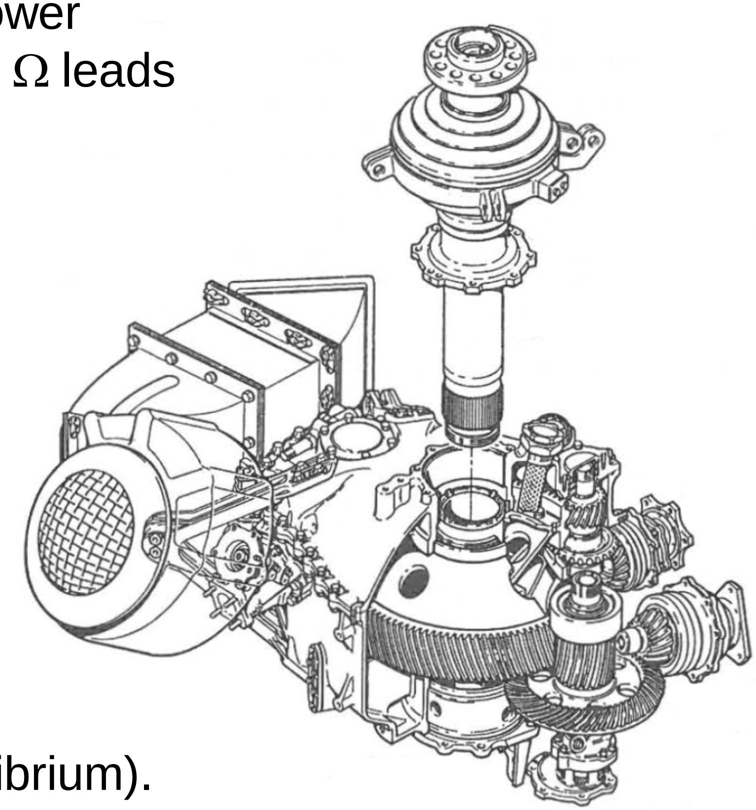
➔ Rotational speed Ω ⇩

Figures from Prouty: *Helicopter Aerodynamics Volume I*

3.2. Main Rotor Design: Rotational speed

With the assumption of constant input power of the main rotor, a high rotational speed Ω leads to a low **drive torque** Q

$$Q = \frac{P}{\Omega}$$



The required torque defines the gear box design and thereby its weight decisively.

Furthermore at the same rotor distance, the required thrust of the tail rotor gets lower (moment equilibrium).

➔ Rotational speed Ω ⇧

Figure from Bittner: *Flugmechanik der Hubschrauber*

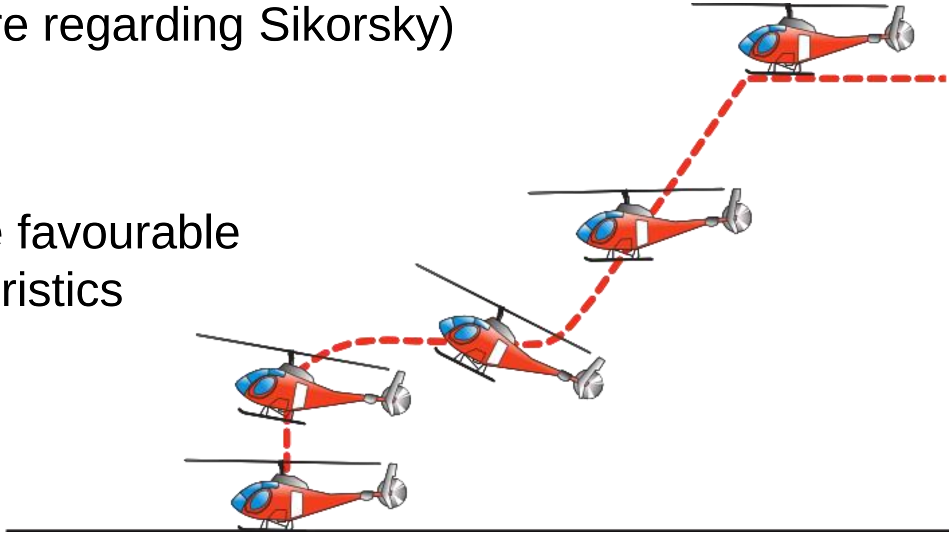
3.2. Main Rotor Design: Rotational speed

Autorotation

High rotational speeds Ω provide good autorotation characteristics due to stored kinetic energy:

- Long reaction time to the transition in the autorotation at power loss
- Safe flare out to minimize the rate of descent before touchdown

Evaluation via autorotation index AI (here regarding Sikorsky)



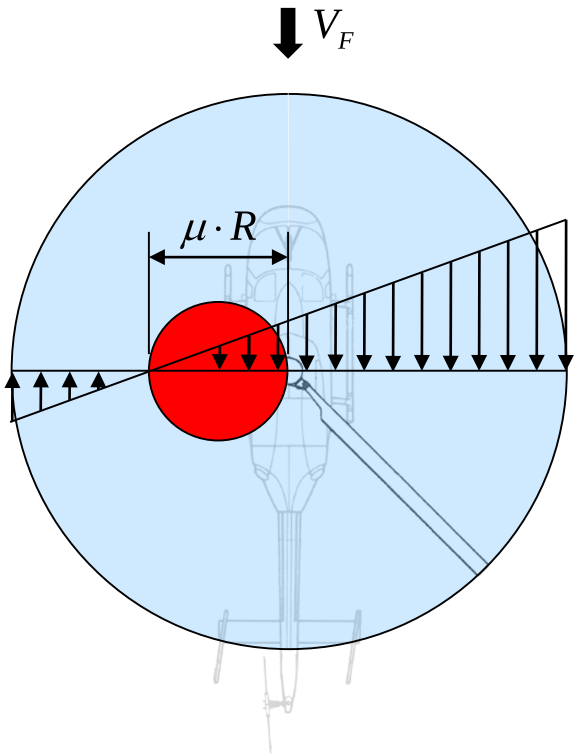
$$AI = \frac{I_R \Omega^2}{2W} \cdot \frac{1}{DL}$$

Large values indicate favourable autorotation characteristics

➔ Rotational speeds Ω ⇧

Figure from FAA: *Rotorcraft Flying Handbook*

3.2. Main Rotor Design: Rotational speed



Reverse flow at retreating blade

In forward flight there exists a zone, in which
The stream in the rotor disc flows backwards
over the blade profile.

- No contribution to rotor thrust
- Can provide downforce
- Diameter $\mu \cdot R$

In order to keep the zone small the requirement of
a low advance ration μ applies for a given forward velocity V_F

$$\mu = \frac{V_F}{V_{TIP}} = \frac{V_F}{R \cdot \Omega}$$

➔ Rotational speed Ω ⇧

3.2. Main Rotor Design: Rotational speed

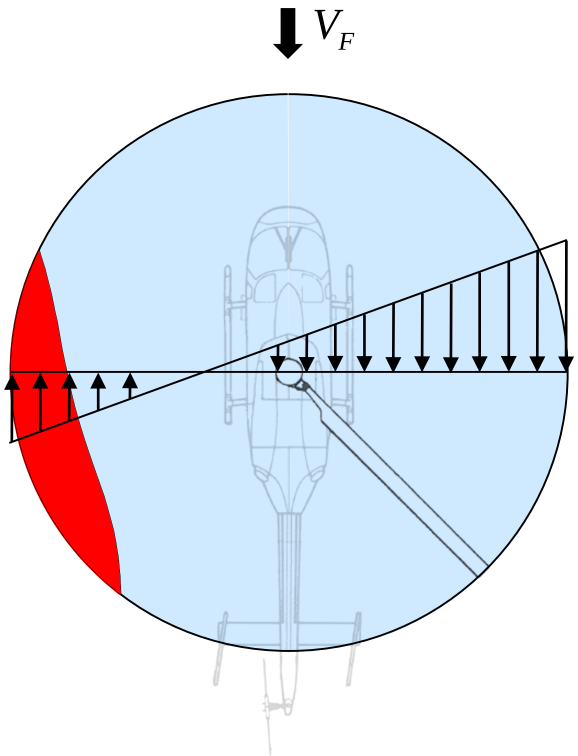
Stall at retreating blade

Reduced flow velocities at
the retreating blade in forward flight require
increased angles of attack in order to maintain
the lift at the blade.

From a certain angle this leads to a stall,
starting from the blade tip (*retreating blade stall*)

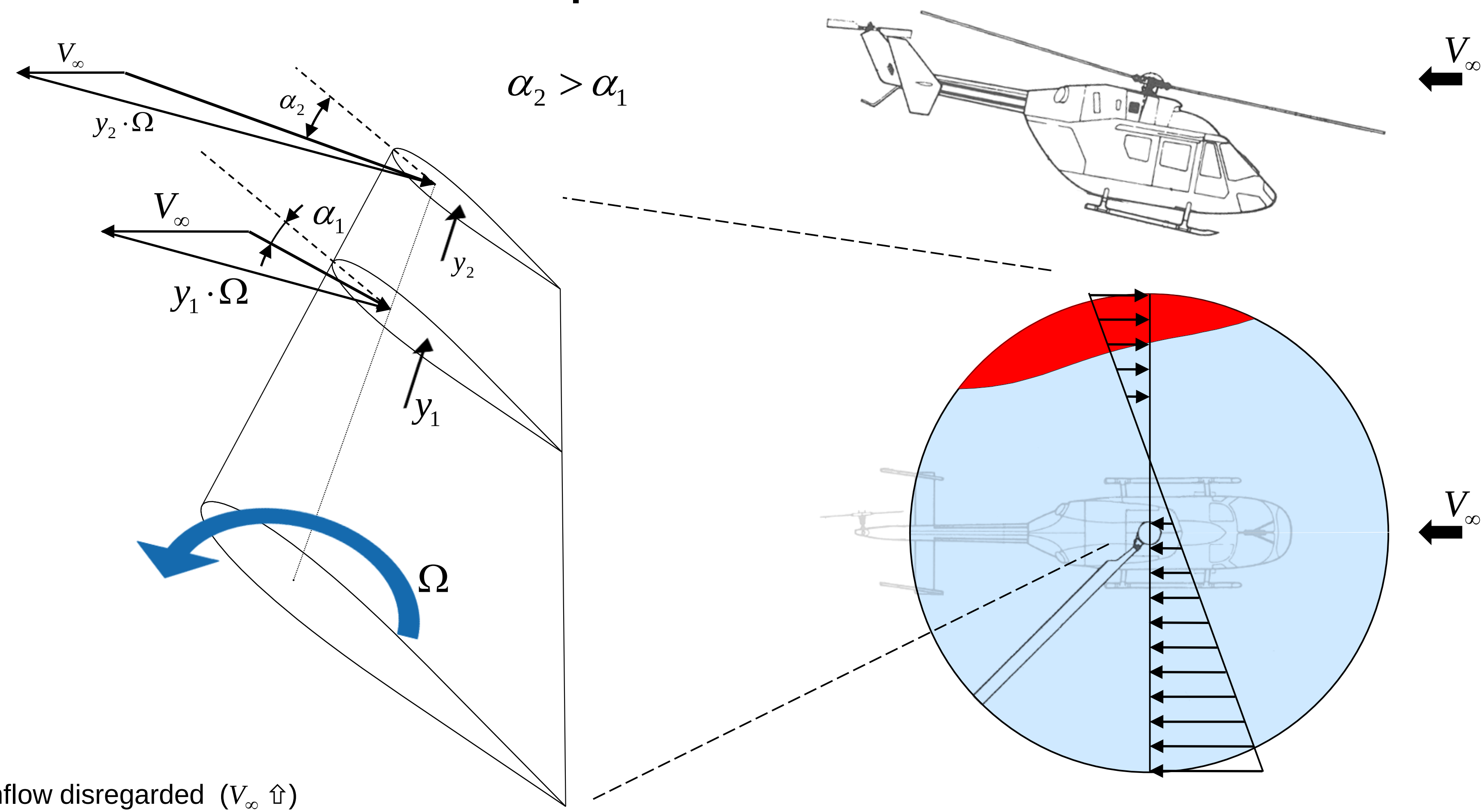
High rotational speeds shift the initiation of the
stall to higher forward velocities.

➔ Rotational speed Ω ⇧



3.2. Main Rotor Design: Rotational speed

Stall occurs first at the blade tip



Induced inflow disregarded (V_∞ ⇧)

3.2. Main Rotor Design: Rotational speed - Summary

Rotational speed

Ω ⇧

- Low gear box weights
- Good autorotation characteristics
- Favourable flow conditions at the retreating blade in forward flight

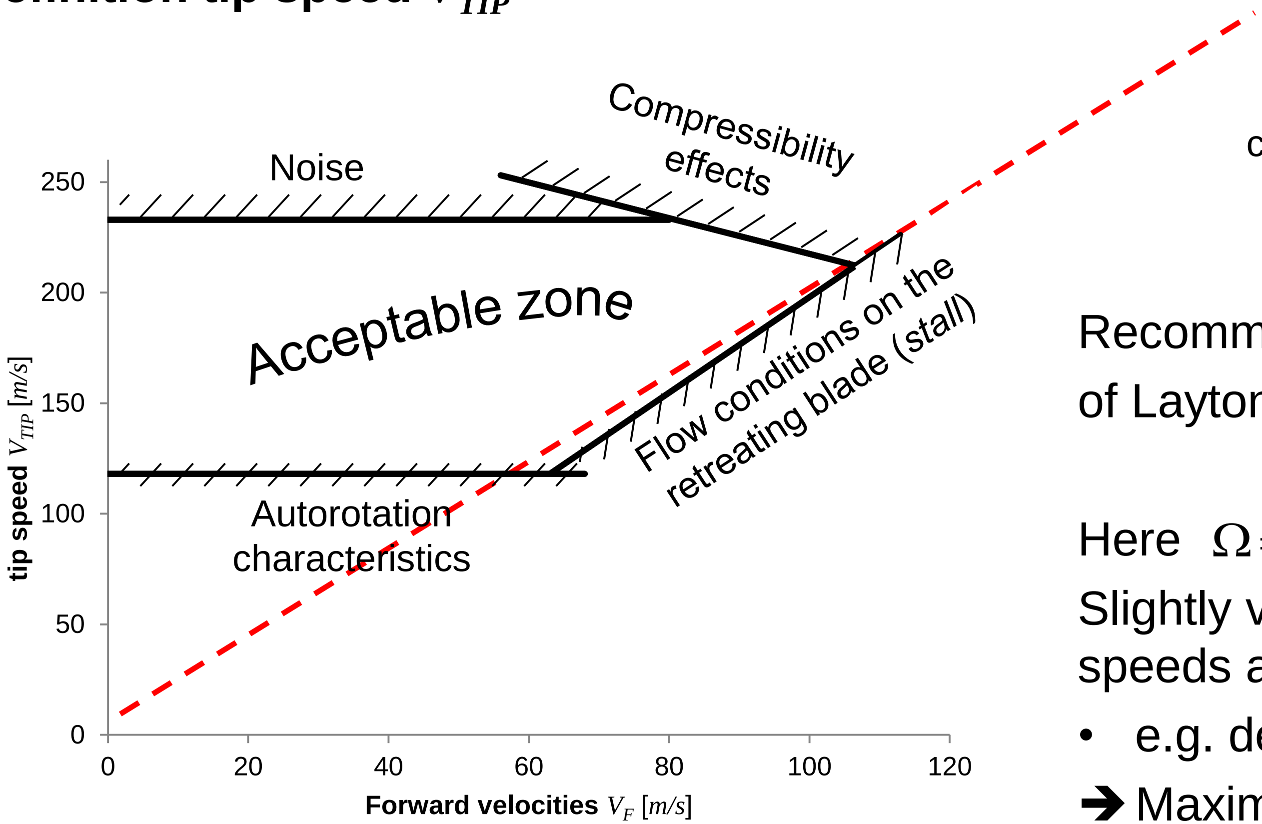
Ω ⇩

- Low noise level
- Compressibility effects at advancing blade only at higher velocities

Photo: Eurocopter, Christophe Guibbaud

3.2. Main Rotor Design: Rotational speed

Definition tip speed V_{TIP}



Maximum achievable advance ratio for conventional helicopters

$$\mu_{\max} \approx 0{,}5$$

Recommendation of Layton: $V_{TIP} \approx 213 \frac{m}{s}$

Here $\Omega = \text{const.}$

Slightly variable rotational speeds are yet possible:

- e.g. dependent on ρ

➔ Maximum thrust (Tail Rotor)

➔ Noise

3.2. Main Rotor Design: Rotational speed

From tip speed and rotor radius results the required rotational speed:

$$\Omega = \frac{tip\ speed}{rotor\ radius} = \frac{V_{TIP}}{R}$$

	V_{TIP} $[m/s]$	R $[m]$	Ω $[rad/s]$
Robinson R22 Beta	213,1	3,8	55,6
MD 500E	207,3	4,0	52,3
MBB BO 105 CB	218,0	4,9	44,3
Agusta A109	221,6	5,5	40,3
Bell 412	237,7	7,0	33,9
Sikorsky UH-60A	221,0	8,2	27,1
Sikorsky CH-53E	223,1	12,0	18,5

Data from Prouty: *Helicopter Performance, Stability and Control;* Photos: Robinson Helicopter Company; Alan Radecki

Derschmidt-Rotor

Rotor with enforced pivoting of the blades

BO-46

Photo: Flightglobal

3 Sizing

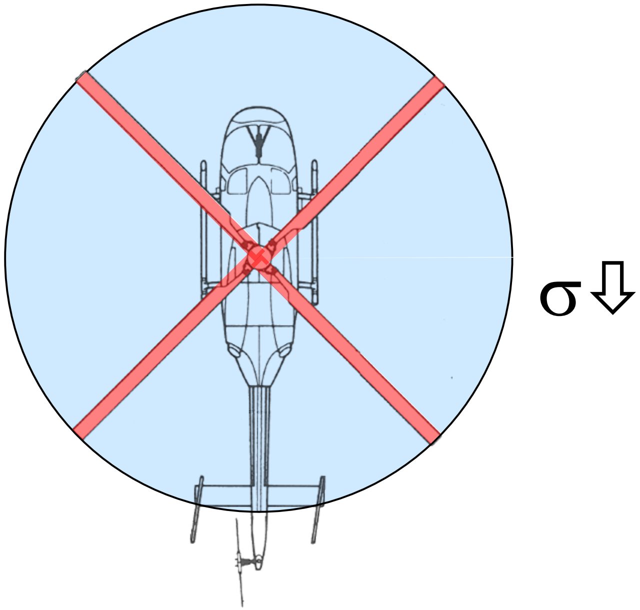
3.2. Main Rotor Design

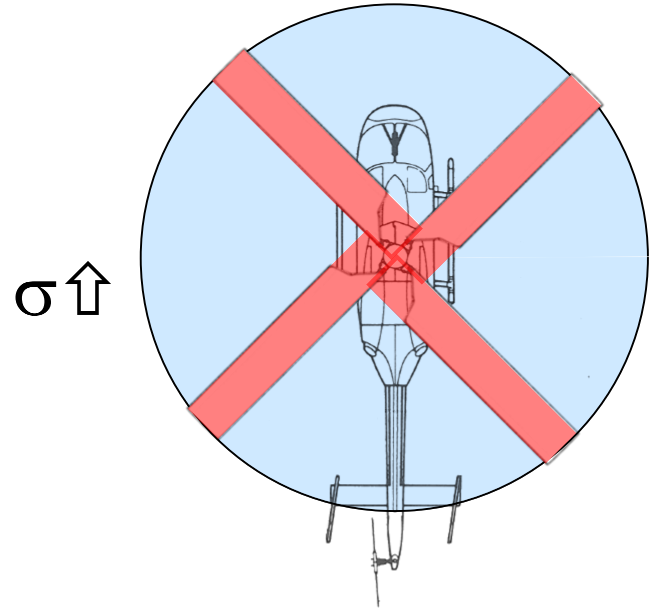
- ✓ Parameters
- ✓ Radius
- ✓ Rotational speed
- ➔ Solidity
- Number of blades
- Blade geometry
- Airfoil selection

3.2. Main Rotor Design: Solidity

At given radius R and known number of blades N_b the solidity σ defines the mean chord length on the blade. Assuming a constant chord length c it applies:

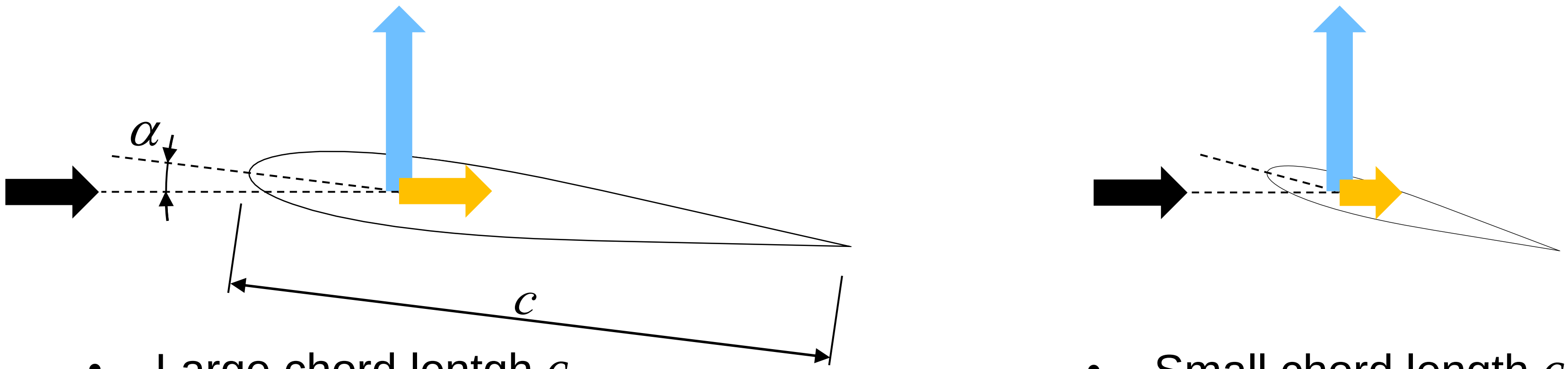
$$\sigma = \frac{blade\ area}{disc\ area} = \frac{N_b \cdot c \cdot R}{\pi \cdot R^2} = \frac{N_b \cdot c}{\pi \cdot R}$$





3.2. Main Rotor Design: Solidity

For a given profile the required lift in the blade cross section can be achieved differently:

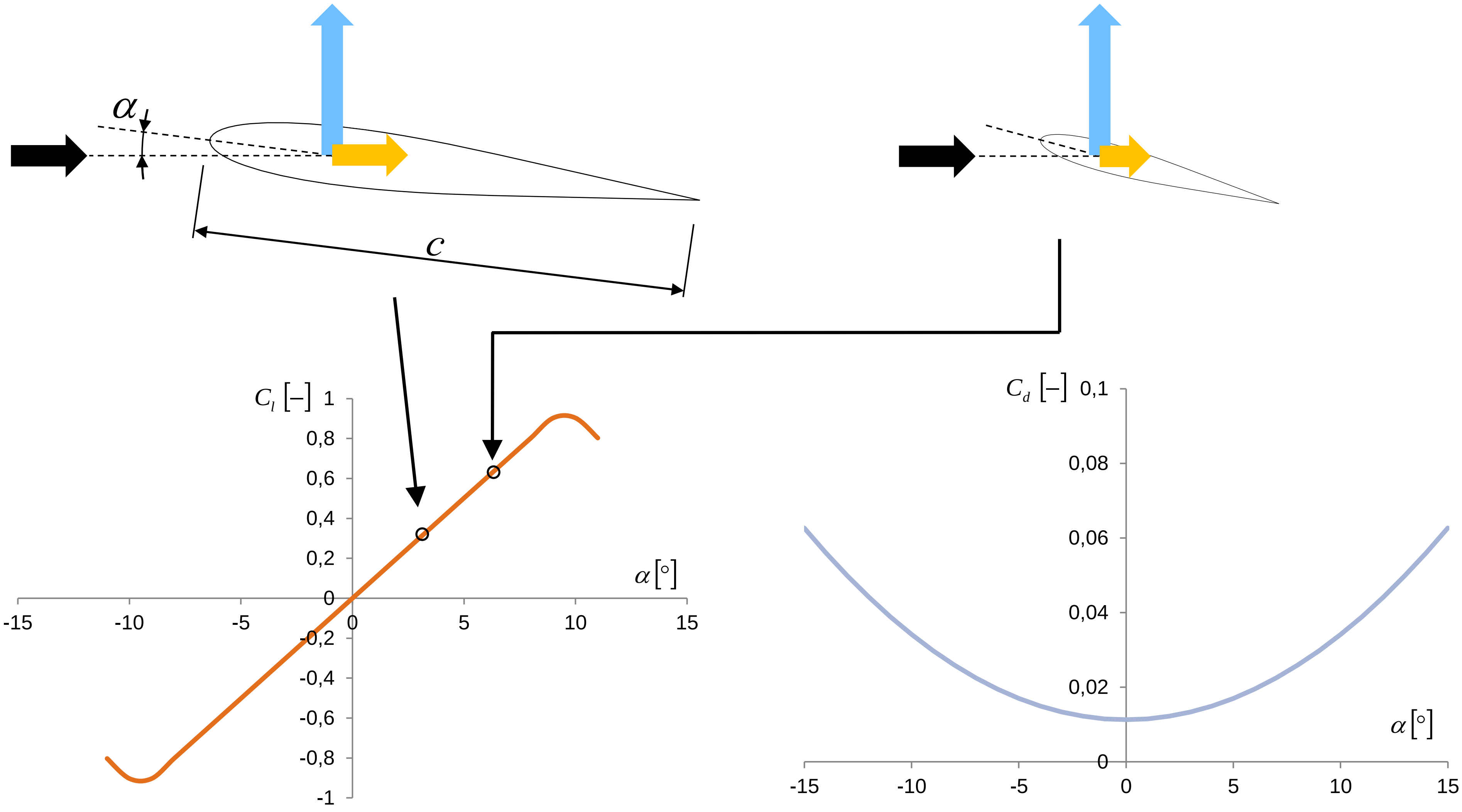


- Large chord lentgh c
- Small angle of attack α

- Small chord length c
- High angle of attack α

So with the variation of solidity the angle of attack level of the rotor blades for a required thrust can be adjusted. There is a combination of c and α depending on the profile, for which the drag is minimized.

3.2. Main Rotor Design: Solidity



3.2. Main Rotor Design: Solidity

Blade loading

$$\frac{C_T}{\sigma} = \frac{T}{\rho \cdot V_{TIP}^2 \cdot N_b \cdot c \cdot R}$$ ➔ Angle of attack level

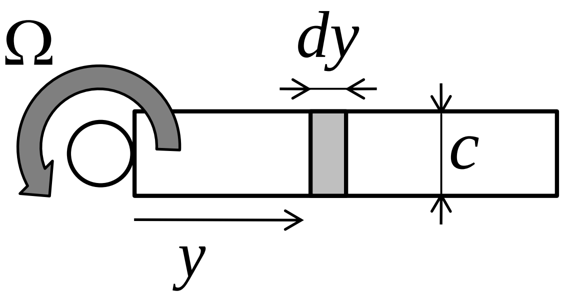
Correlation between blade loading $\frac{C_T}{\sigma}$ and an assumed mean lift coefficient $\overline{C}_L$ in all blade cross sections:

In hover, disregarding the induced inflow

$$T = N_b \cdot \frac{1}{2} \cdot \rho \cdot c \cdot \overline{C}_L \cdot \Omega^2 \int_0^R y^2 dy$$

$$C_T = \frac{N_b \cdot c \cdot \overline{C}_L \cdot \Omega^2}{2 \cdot \pi \cdot R^2 \cdot V_{TIP}^2} \cdot \int_0^R y^2 dy$$

$$C_T = \frac{N_b \cdot c \cdot \overline{C}_L}{6 \cdot \pi \cdot R} = \frac{\sigma \cdot \overline{C}_L}{6}$$ ➔ $$\frac{C_T}{\sigma} = \frac{\overline{C}_L}{6}$$



3.2. Main Rotor Design: Solidity

Blade loading

For hovering it is possible to determine a solidity, at which the profile cross sections with ideal angle are operated.

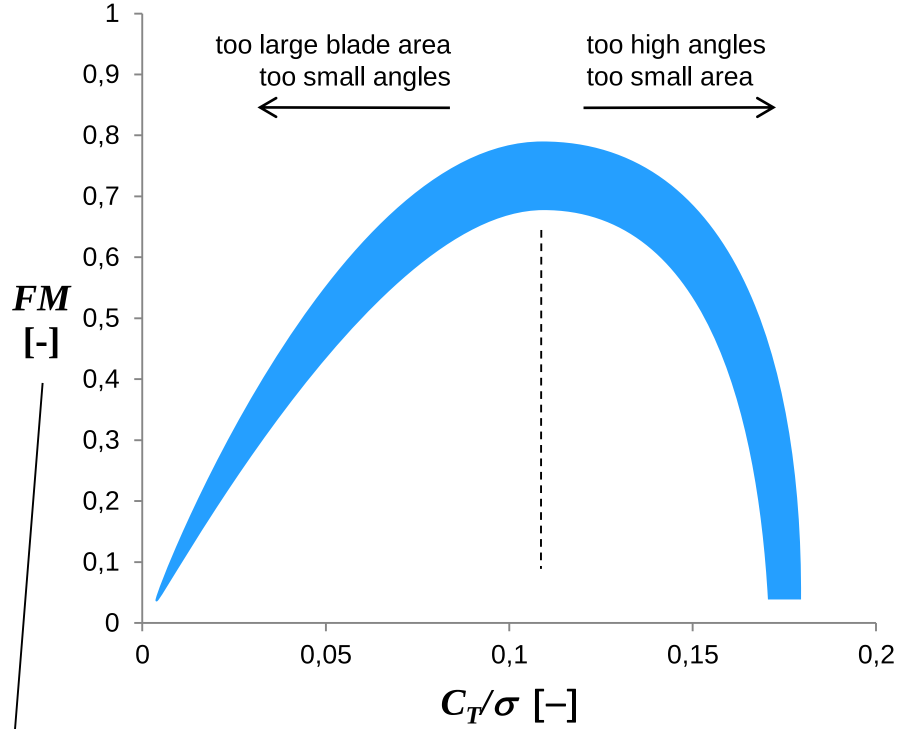


Figure of Merit $FM = \frac{P_{ideal}}{P_{real}} < 1$

The suitable blade loading is typically at $\frac{C_T}{\sigma} \approx 0{,}1$

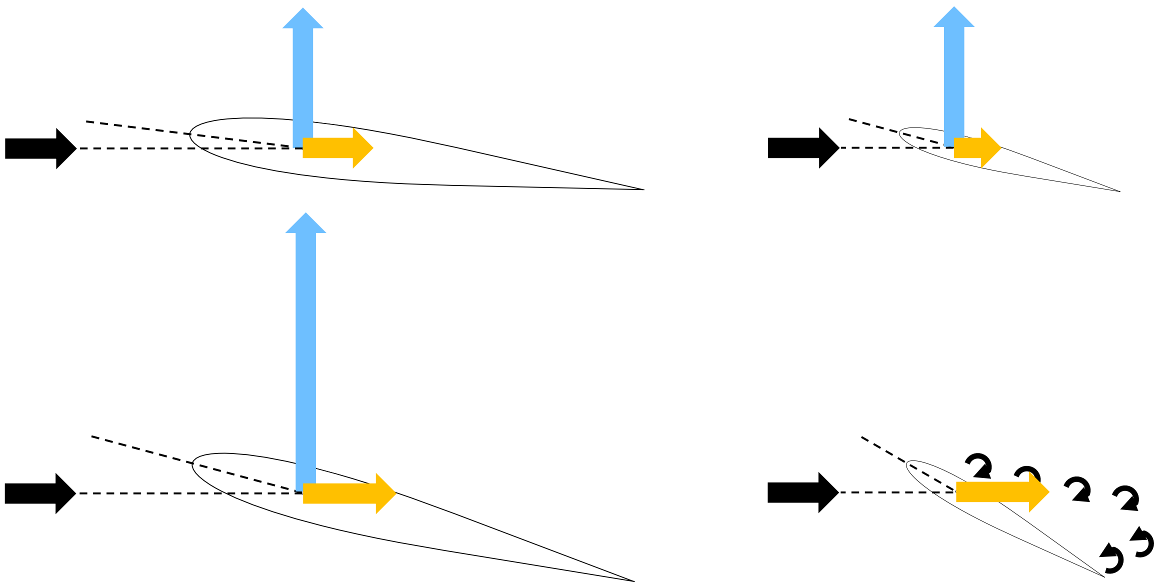
With $\frac{C_T}{\sigma} = \frac{\overline{C}_L}{6}$ ➔ $\overline{C}_L \approx 0{,}6$

For current profiles this equals an average angle of attack of $\overline{\alpha} \approx 6°$

Graph according to Prouty: *Helicopter Aerodynamics I*

3.2. Main Rotor Design: Solidity

The *stall* limits the maximum achievable lift in the profile cross section.



An ideal chord length for hover can be too small, in order to also comply with increased lift requirements.

3.2. Main Rotor Design: Solidity

The occurrent angles of attack on the blade sections are increased compared to hover due to

- Lower air density
- Manoeuvring loads
- Forward flight
  - Propelling force necessary
  - Unfavourable flow conditions on retreating blade

➔ Higher blade area as in hover requires $\sigma \Uparrow$

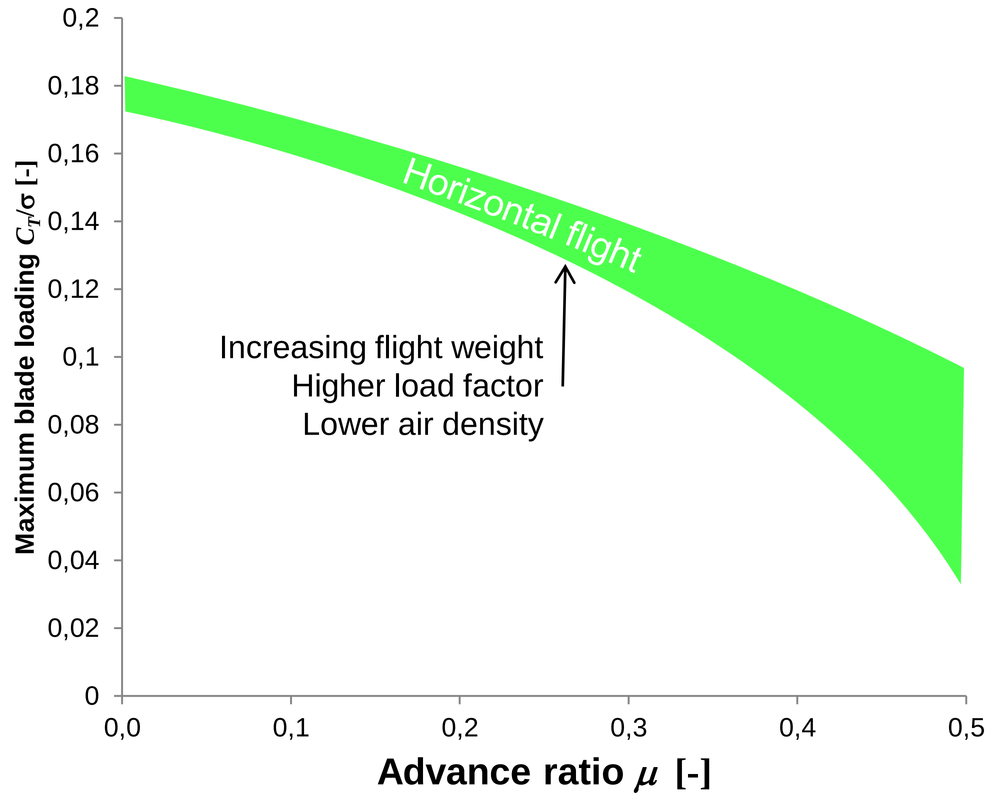

Photos: Eurocopter, Patrick Penna; Red Bull; AgustaWestland

3.2. Main Rotor Design: Solidity

Diagram of maximum blade loading versus advance ratio

Experimentally determined graphs indicate, from which values for $\frac{C_T}{\sigma}$ *stall* effects get unacceptable:

- Vibration
- Component load
- Control forces
- Manoeuvrability



Often it is assumed that the rotor thrust is only defined by the helicopter weight and load factor n_L .

$$C_T \approx \frac{n_L \cdot W}{\rho \cdot A \cdot V_{TIP}^2}$$

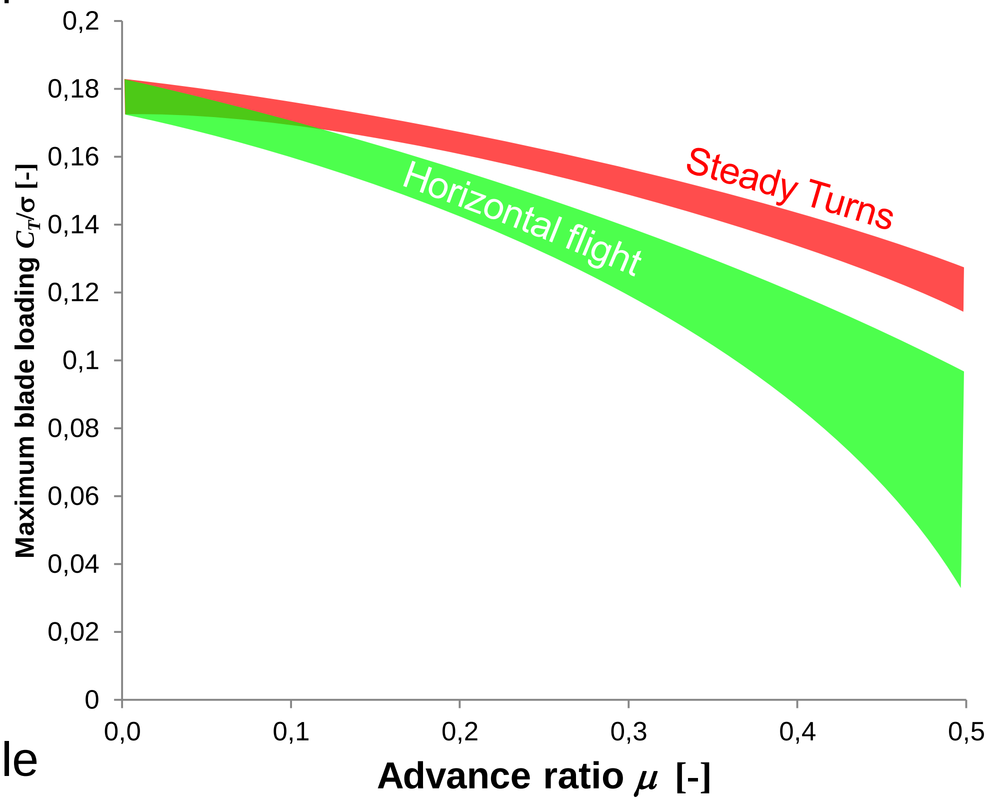
Graph according to Prouty: *Helicopter Aerodynamics I*

3.2. Main Rotor Design: Solidity

Diagram of maximum blade loading versus advance ratio

In steady turning flight the maximum blade loading is higher than in straight and level flight:

- Tracking of the rotor disc with constant pitch rate requires lower lift at the critical retreating blade
  - ➔ Higher angle of attack level possible



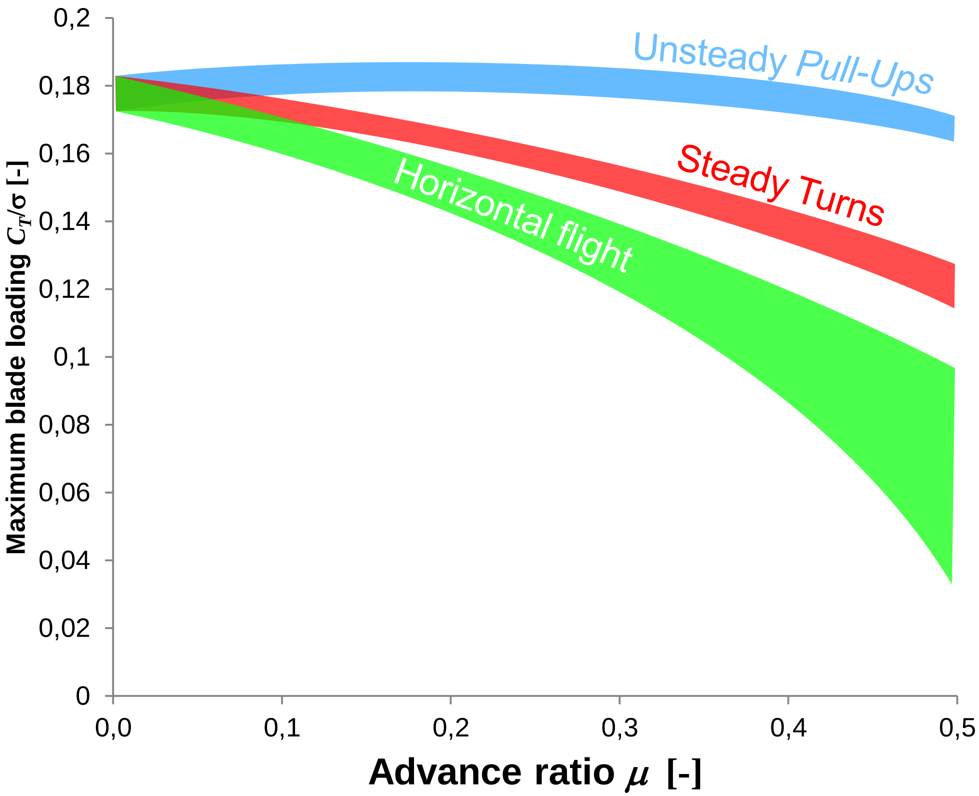
- Turning flight limited in time
  - ➔ Stronger vibrations endurable

Graph according to Prouty: *Helicopter Aerodynamics I*

3.2. Main Rotor Design: Solidity

Diagram of maximum blade loading versus advance ratio

Unsteady manoeuvres
can temporarily tolerate large
blade loadings also at
high velocities.



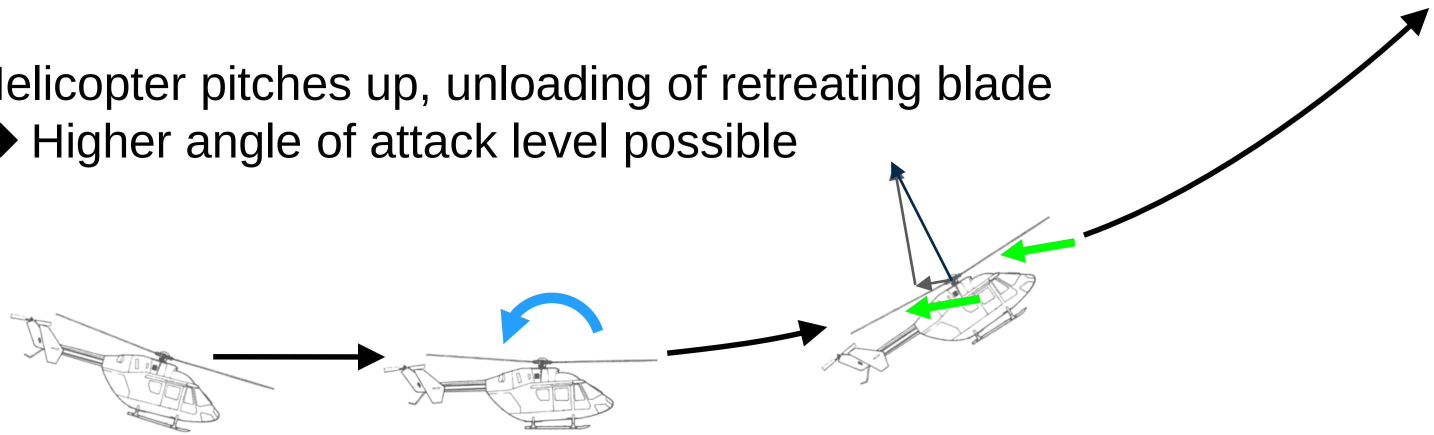
Graph according to Prouty: *Helicopter Aerodynamics I*

3.2. Main Rotor Design: Solidity

Unsteady manoeuvres can temporarily tolerate large blade loadings also at high velocities.

Example *Pull-Up*:

Helicopter pitches up, unloading of retreating blade
➔ Higher angle of attack level possible



Flow through rotor disc from below
➔ More favourable angle of attack allocation on the rotor disc, profile drag provides thrust contribution

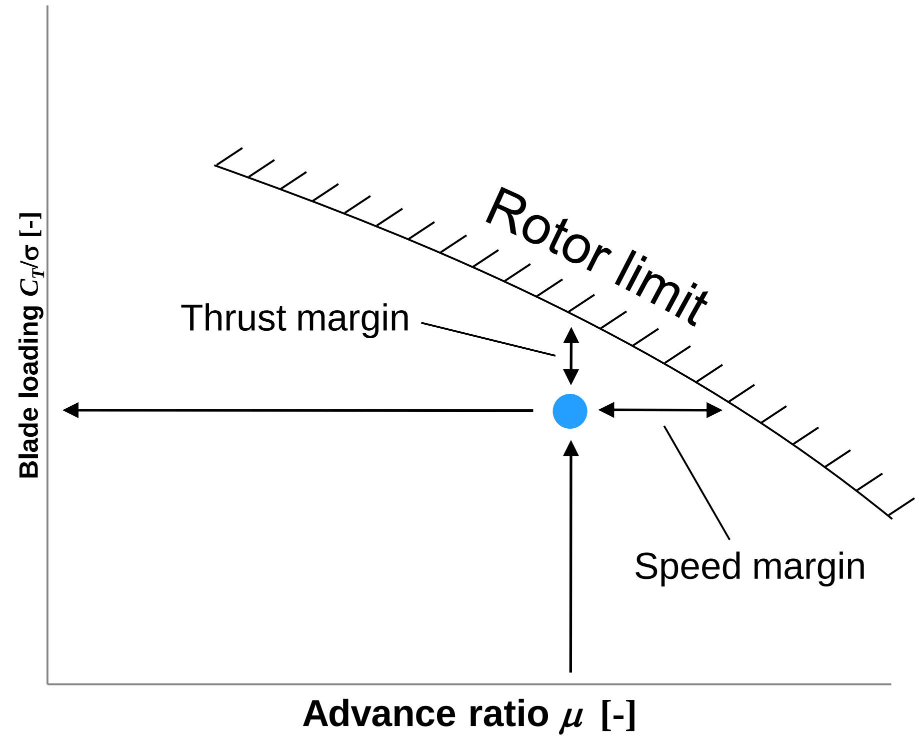
Manoeuvre limited in time
➔ Stronger vibrations endurable

3.2. Main Rotor Design: Solidity

Selection of blade loading C_T/σ

For given advance ratio with distance to *stall* limit of rotor:

- Ensuring controllability
  (also in case of gusts…)
- Growth possible
- Reduction of vibration and structural stress of the rotor



For Sizing based on
permitted maximum speed v_{NE}

- Thrust margin
  > ~15%
- Speed margin
  > ~10%

Details per Certification Specification

3.2. Main Rotor Design: Solidity

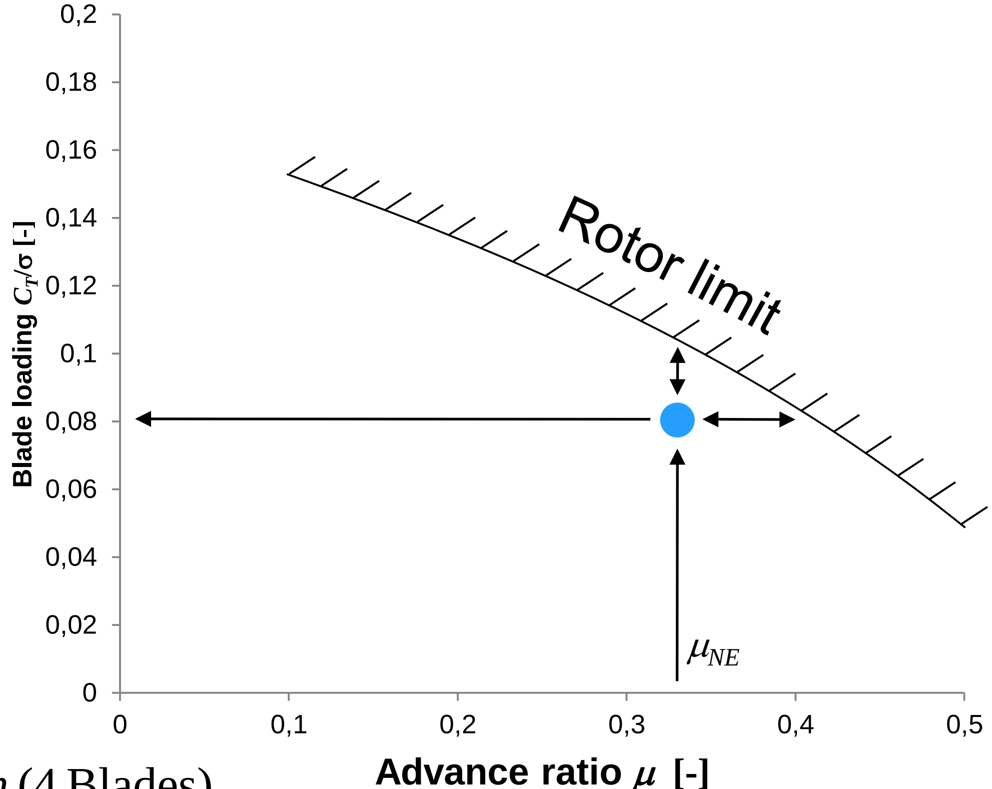
Example: BK117-C1, v_{NE}, $0m$ altitude in standard atmosphere (*ISA*), $MTOW$

- $\mu_{NE} = 0{,}33$

➔ $C_T/\sigma = 0.081$

- $R = 5{,}5m$
- $\rho = 1{,}225 \frac{kg}{m^3}$
- $V_{TIP} = 221 \frac{m}{s}$
- $T = 3550kg \cdot 9{,}81 \frac{m}{s^2} \cdot n_L$

  Load factor $n_L = 1$



➔ $$C_T = \frac{T}{\rho \cdot \pi \cdot R^2 \cdot V_{TIP}^2} = 0.0058$$

➔ $$\sigma = \frac{C_T}{0.081} = 0.0714$$ ➔ $c = 0{,}31m$ (4 Blades)

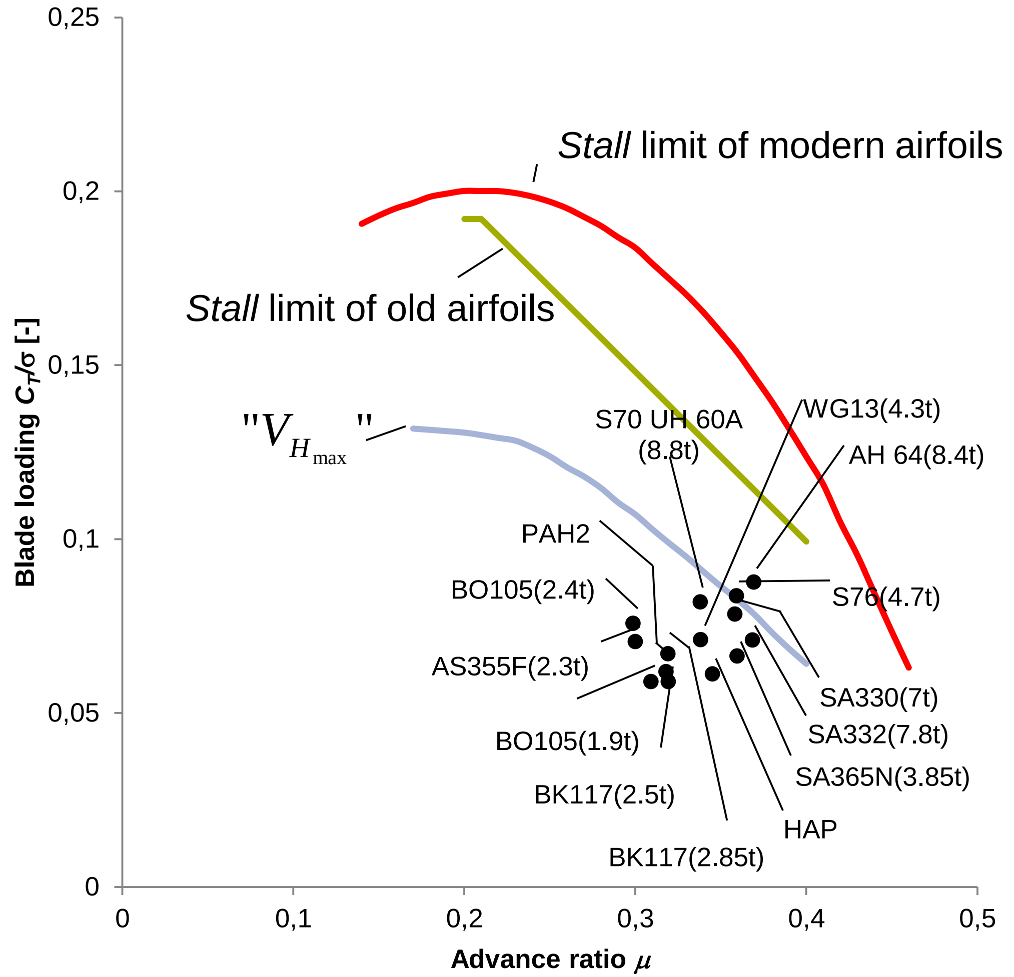
Data: Eurocopter

3.2. Main Rotor Design: Solidity

Selection of blade loading C_T/σ

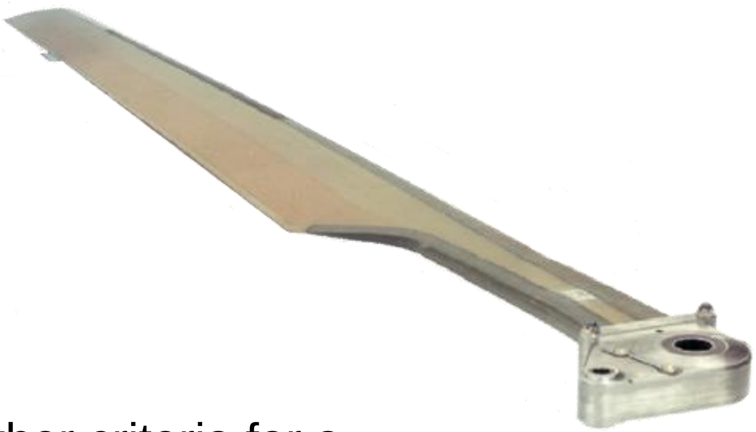
Blade loadings of different helicopter types at V_H

V_H : Maximum speed in horizontal flight at *maximum continuous power (MCP)*.



Data: Eurocopter

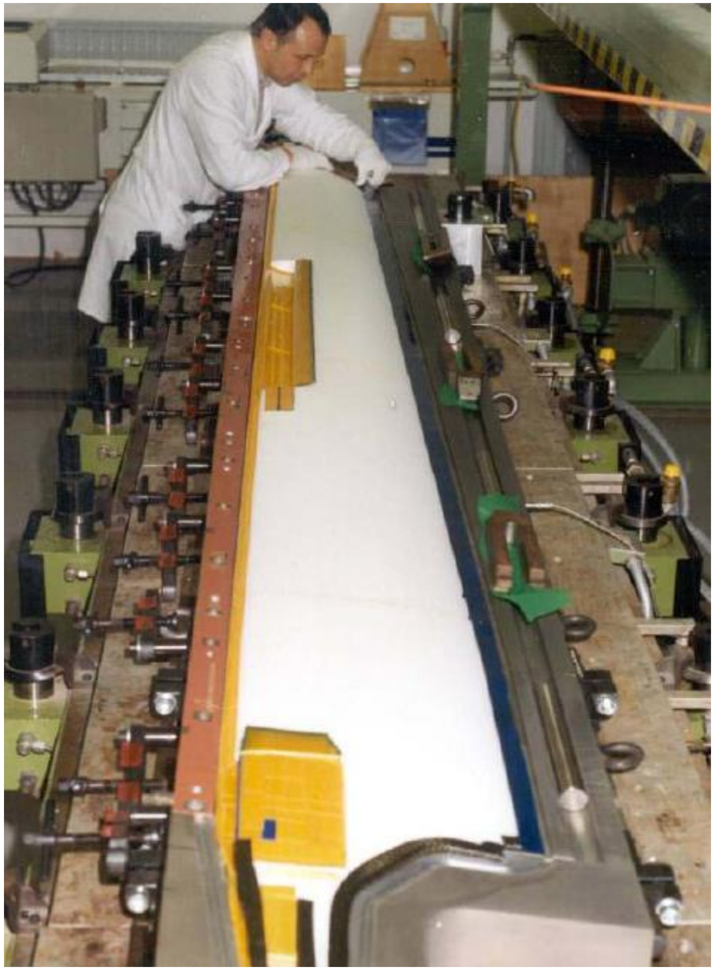
3.2. Main Rotor Design: Solidity



Further criteria for a low solidity:

- Low rotor weight
- Low costs

➔ σ ⇩



Photos: Eurocopter

3.2. Main Rotor Design: Solidity - Zusammenfassung

Solidity

σ ⇧

- High flight speed
- Large load factors (Manoeuvre)
- Flight in low air density
- Possibility of flight weight increase

σ ⇩

- Low weight
- Low costs
- Hover efficiency (Optimum at low σ)

Photo: Bell Helicopter Textron

3.2. Main Rotor Design: Solidity

Trend over *MTOW*

$$C_T = \frac{T}{\rho \cdot \pi \cdot R^2 \cdot V_{TIP}^2}$$

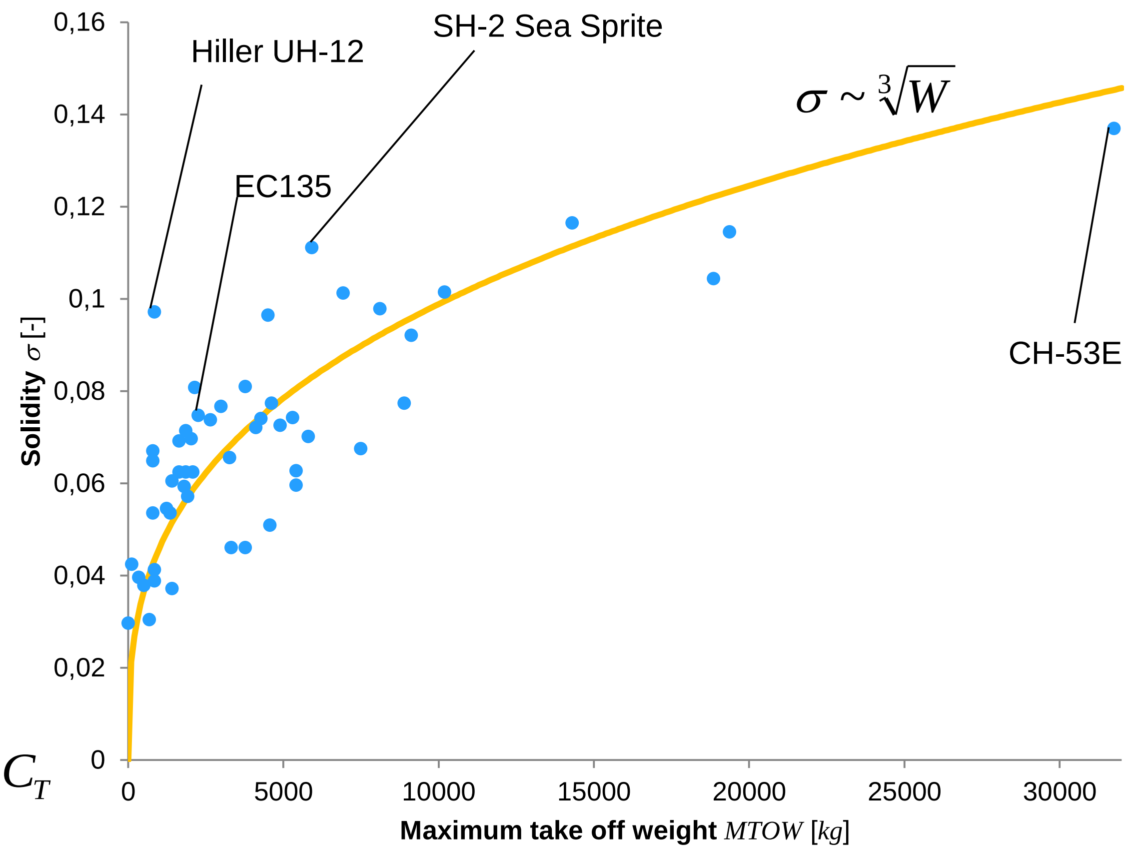
With

- $T \sim W \sim MTOW$
- $V_{TIP} = const.$
- $R \sim \sqrt[3]{W}$

$$\rightarrow C_T \sim \frac{W}{W^{\frac{2}{3}}} = \sqrt[3]{W}$$

Assumption $\frac{C_T}{\sigma} = const. \rightarrow \sigma \sim C_T$

$$\rightarrow \sigma \sim \sqrt[3]{W}$$



Data: DLR, Dr. Christoph Keßler

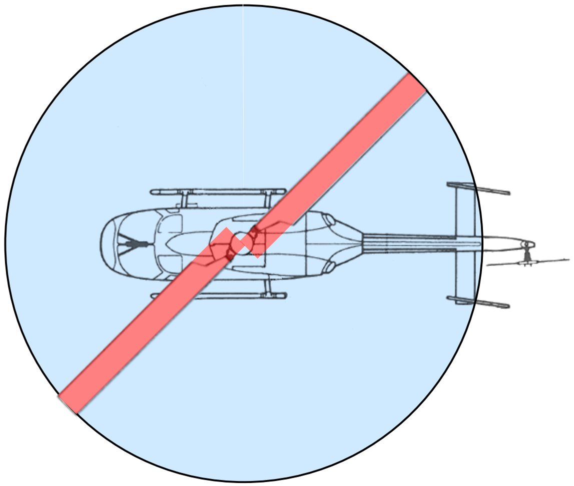
3 Sizing

3.2. Main Rotor Design

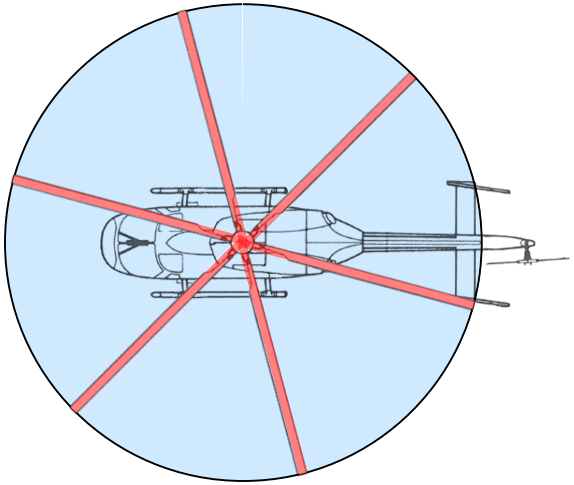
- ✓ Parameters
- ✓ Radius
- ✓ Rotational speed
- ✓ Solidity
- ➔ Number of blades
- Blade geometry
- Airfoil selection

3.2. Main Rotor Design: Number of blades

Following the definition of the solidity,
the blade area is distributed to a certain
number of blades N_b.



It is mainly defined by dynamic
aspects and constructional requirements.



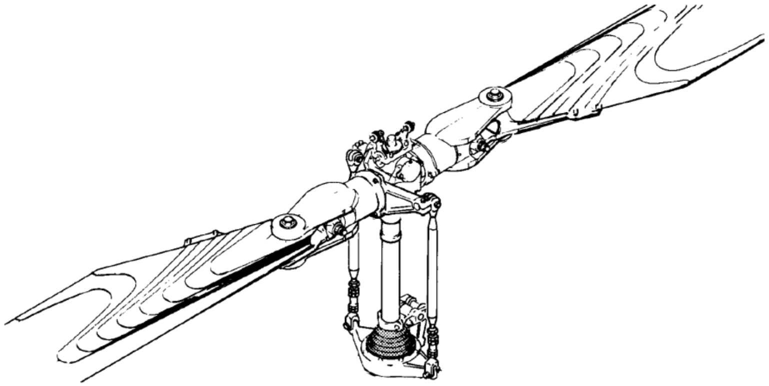
Aerodynamic effects and the influence
on the flight performance are
only of minor relevance.

3.2. Main Rotor Design: Number of blades

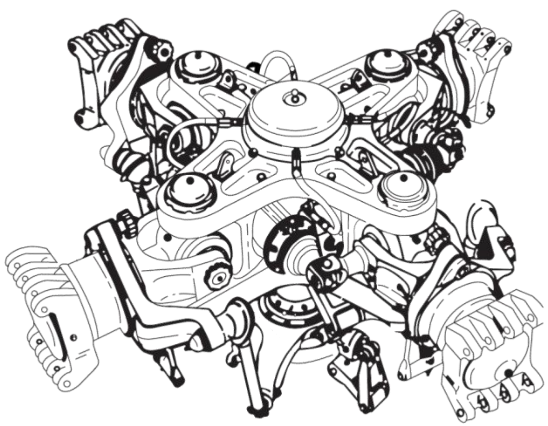
Small number of rotor blades:

- Simple setup of the rotor hub
- Small required space
- Folding mechanism easy to implement
- Low weight of the rotor system
- Easy to maintain because of low number of parts
- Low costs
- Torsionally rigid blades by the use of large chord length

➔ Number of blades N_b ⇩



Rotor hub Bell 206B



Rotor hub Westland Wessex

Figures: Bell Helicopter Textron; Bramwell: *Bramwell's Helicopter Dynamics*

3.2. Main Rotor Design: Number of blades

Large number of rotor blades:

- Low weight of single blades,
  good manageability (replacement…)

- Load distribution on many blades,
  favourable vibration characteristics
  (higher frequencies and smaller
  amplitudes)

➔ Number of blades N_b ⇧

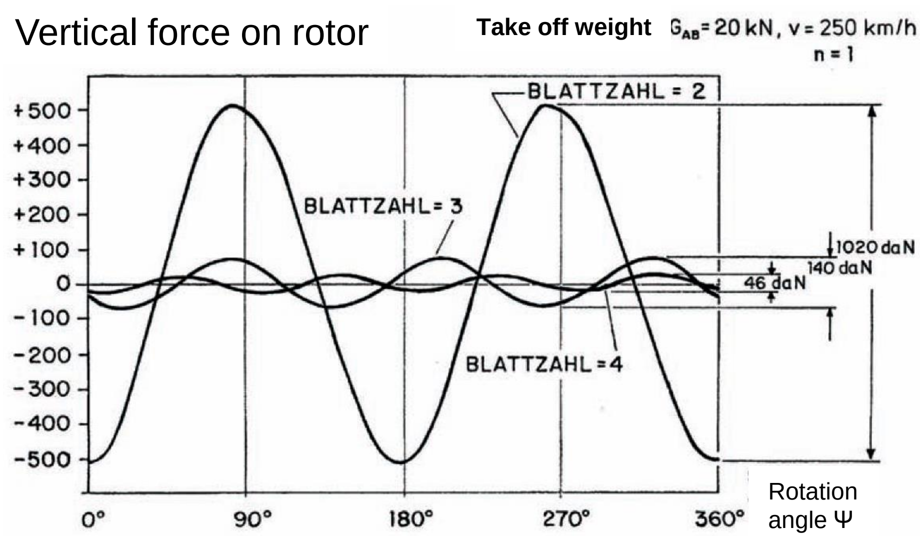
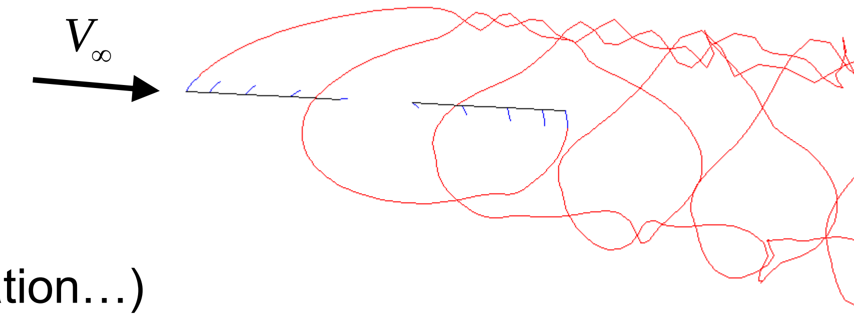


Photo: Eurocopter; Graph: Prof. Peter Richter

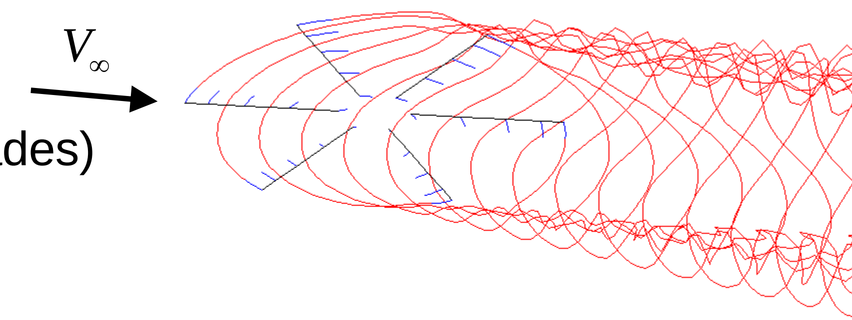
3.2. Main Rotor Design: Number of blades

Aerodynamic aspects



Small number of blades:

- Low rotor hub drag
- Few strong vortices in rotor wake
  (Influence on blade loadings, noise, vibration…)



Large number of blades:

- Low tip losses due to
  high aspect ratio (but unfavourable
  Reynolds number effects for very thin blades)
- Many weak vortexes in rotor wake
- Thin blades, lower *thickness noise*

➔ No general trends
Aerodynamic effects usually irrelevant for design

3.2. Main Rotor Design: Number of blades

Trend over *MTOW*

	$MTOW$ [kg]	N_b
BO103	390	1
R22	621	2
MD 530F	1406	5
EC120	1715	3
206L-4	2018	2
EC135	2835	4
MD900	2836	5
EC145	3585	4
UH-1D	4310	2
S-76A	4671	4
Tiger	6000	4
NH90	10500	4
EH101	15600	5
CH-53G	19050	6
CH-53E	33300	7
CH-53K	38400	7
Mil-26	56000	8

Data: DLR, Dr. Keßler; Helicopter museum Bückeburg; Jane's Encyclopedia of Aviation; Photos: EADS; Andrés Tonini

3 Sizing

3.2. Main Rotor Design

- ✓ Parameters
- ✓ Radius
- ✓ Rotational speed
- ✓ Solidity
- ✓ Number of blades
- ➔ Blade geometry
- • Airfoil selection

3.2. Main Rotor Design: Blade geometry

Ideal rotor in hover

- Requirement of minimal induced power P_i for a Sizing thrust T

➔ Uniform induced velocity over the rotor disc

➔ Triangular thrust distribution on the blade

- At every blade position optimum angle of attack of airfoil

➔ Maximum $\dfrac{C_l}{C_d}$

Variable parameters, in order to meet the requirements at every blade cross section over the radius:

- Twist
- Local chord length

3.2. Main Rotor Design: Blade geometry

Ideal rotor in hover: $v_i = const.$

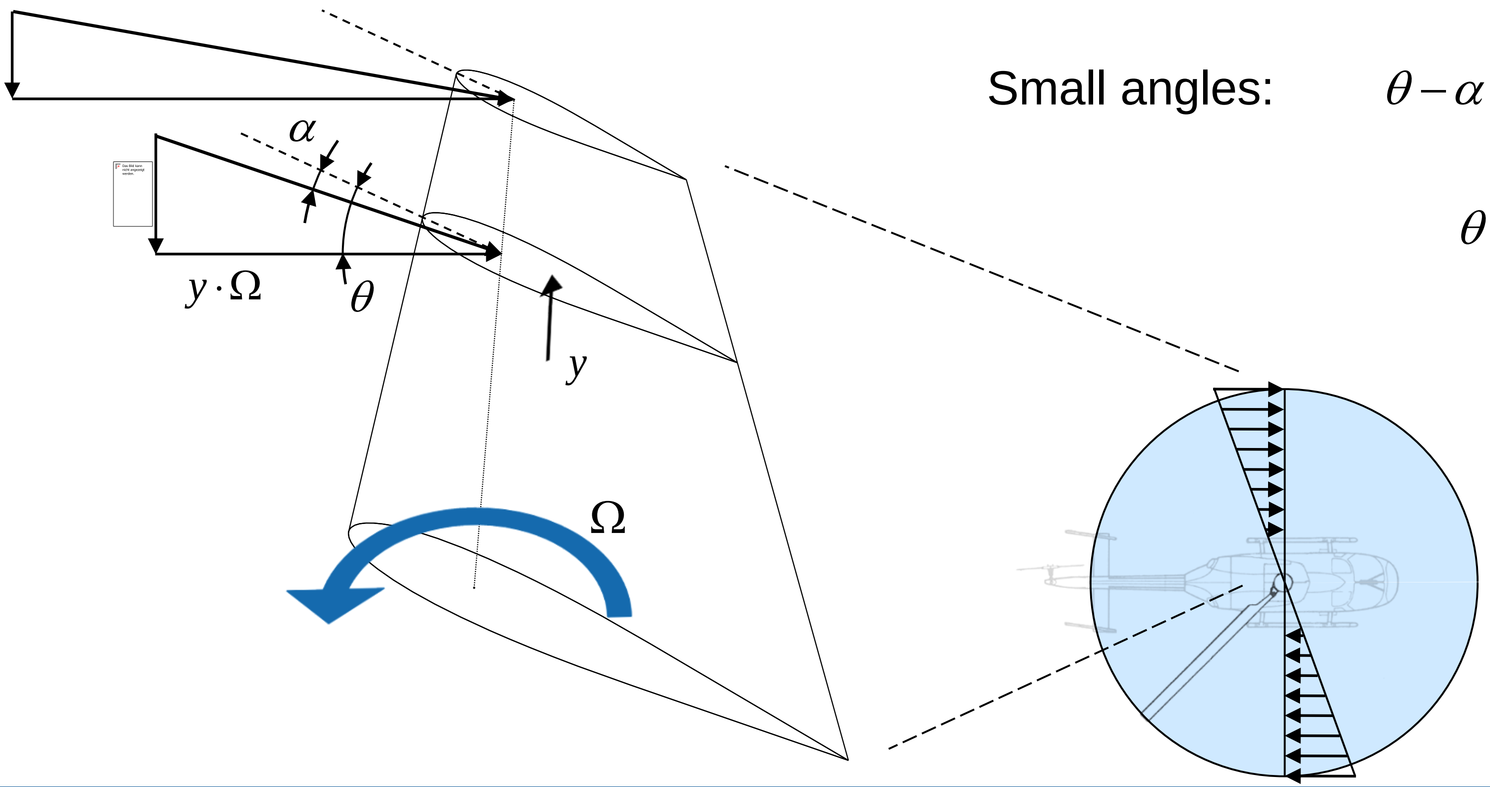
At every blade position same (optimum) angle of attack of airfoil: $\alpha = const.$

$$\tan(\theta - \alpha) = \frac{v_i}{\Omega \cdot y}$$

Small angles: $$\theta - \alpha = \frac{v_i}{\Omega \cdot y}$$

$$\theta = \frac{v_i}{\Omega \cdot y} + const.$$

Hyperbolic trend of θ vs y for constant α



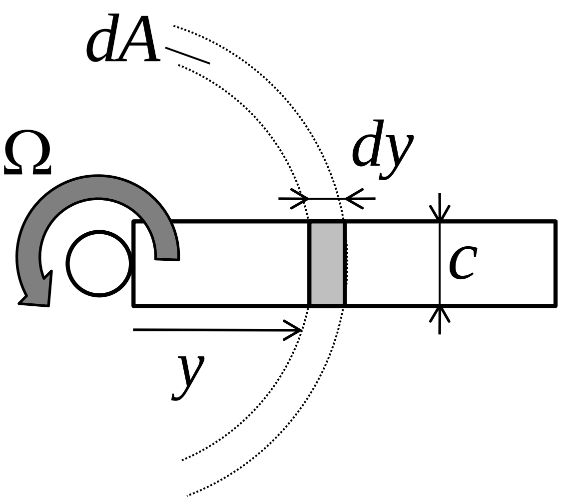
3.2. Main Rotor Design: Blade geometry

Ideal rotor in hover

The area covered by a rotor section dy depends linearly of the radial position y . $dA = 2\pi \cdot y \cdot dy$

Constant flow $v_i = const.$ requires constant thrust per area

➔ Lift on blade also needs to grow linearly with y



$$dL = \frac{1}{2}\rho C_L \Omega^2 \cdot y^2 \cdot c \cdot dy \overset{!}{\sim} y$$ (v_i disregarded)

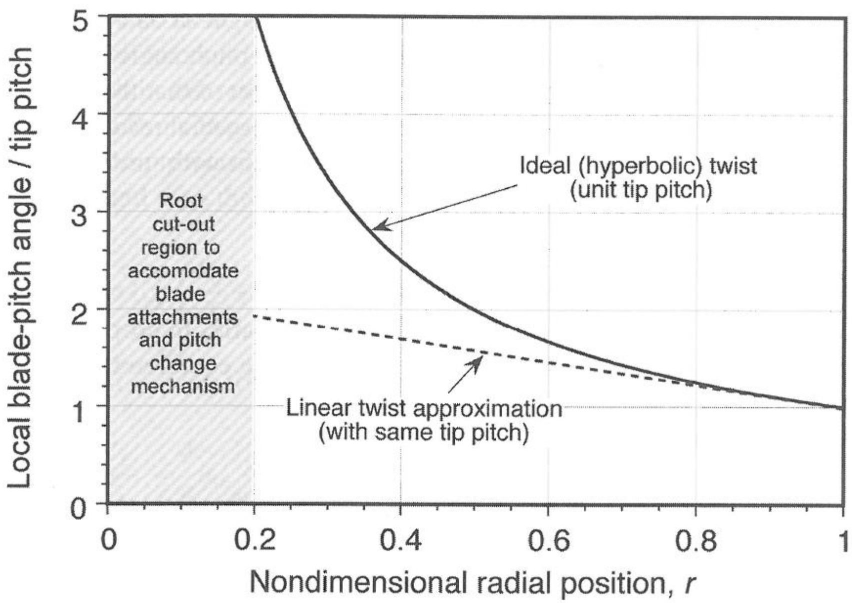
- For constant C_L at optimum $\frac{C_L}{C_D}$

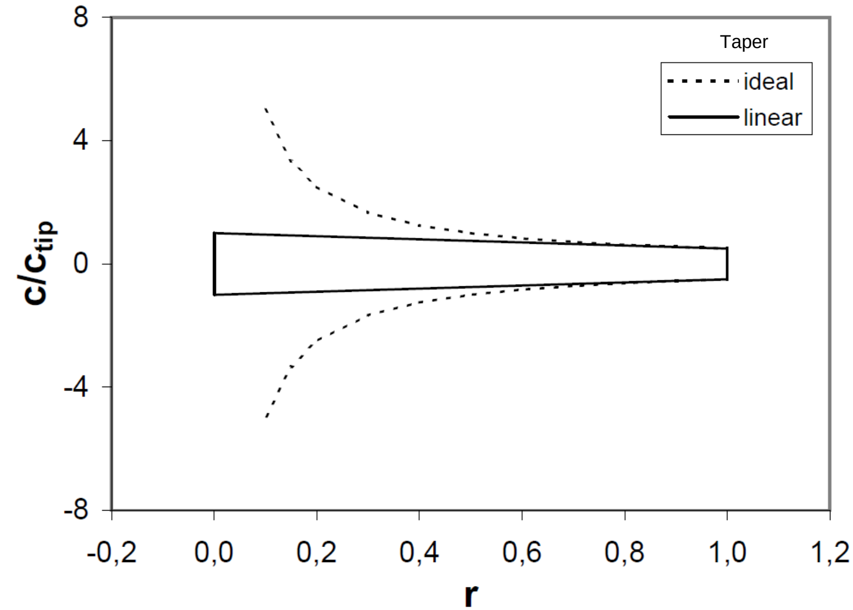
$$y^2 c \overset{!}{\sim} y \qquad c \sim \frac{1}{y}$$

➔ Also the ideal chord length c runs hyperbolic versus y

3.2. Main Rotor Design: Blade geometry

Ideal rotor in hover





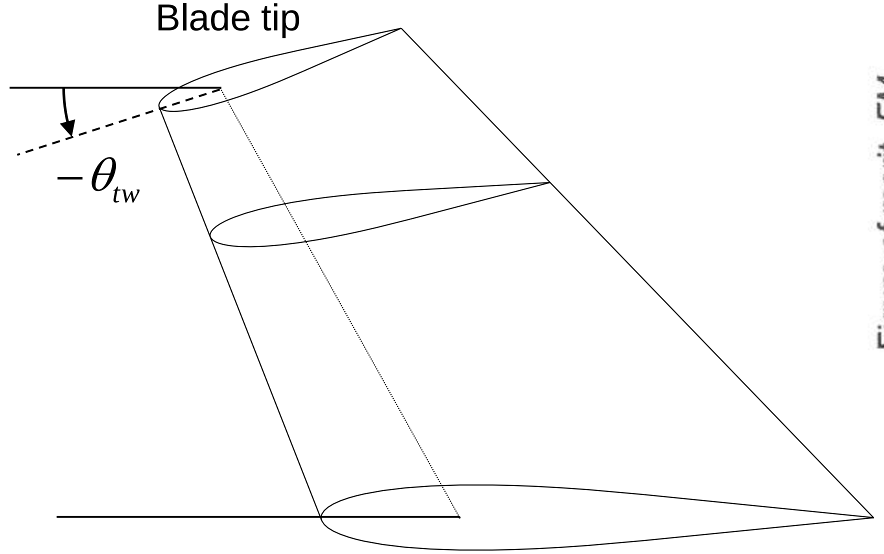
The linear trends are often used for the twist and taper along the blade length. The ideal trends can be thereby approximated well in the important outer blade sector.

Figures: Leishman: *Principles of Helicopter Aerodynamics*; B. van der Wall

3.2. Main Rotor Design: Blade geometry

Twist is favourable:

- Increase of *Figure of Merit* FM in hover of ~5% compared to untwisted blades possible
- Especially linear twist simple to manufacture



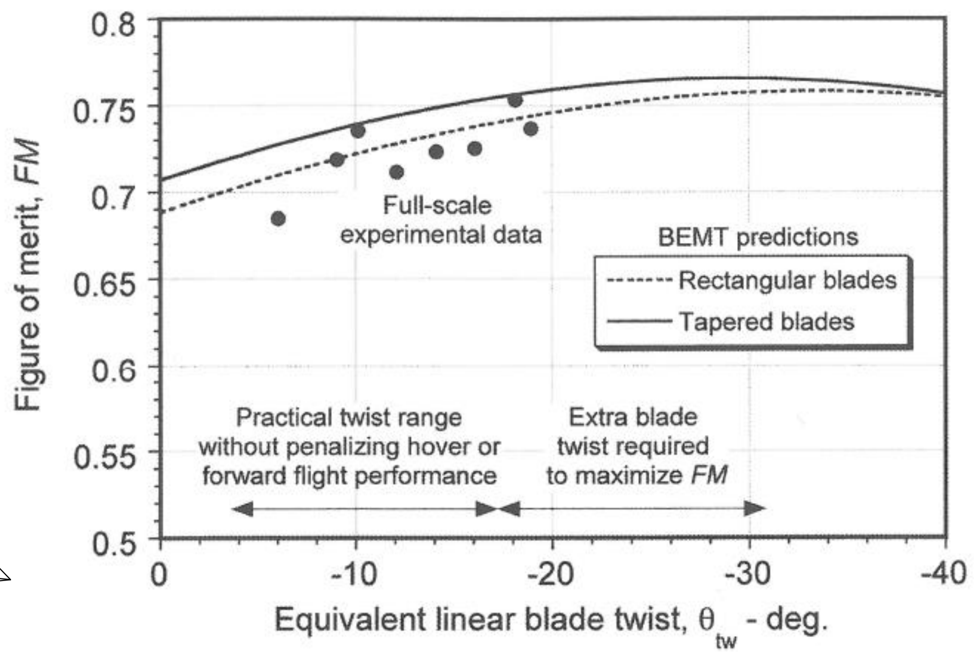
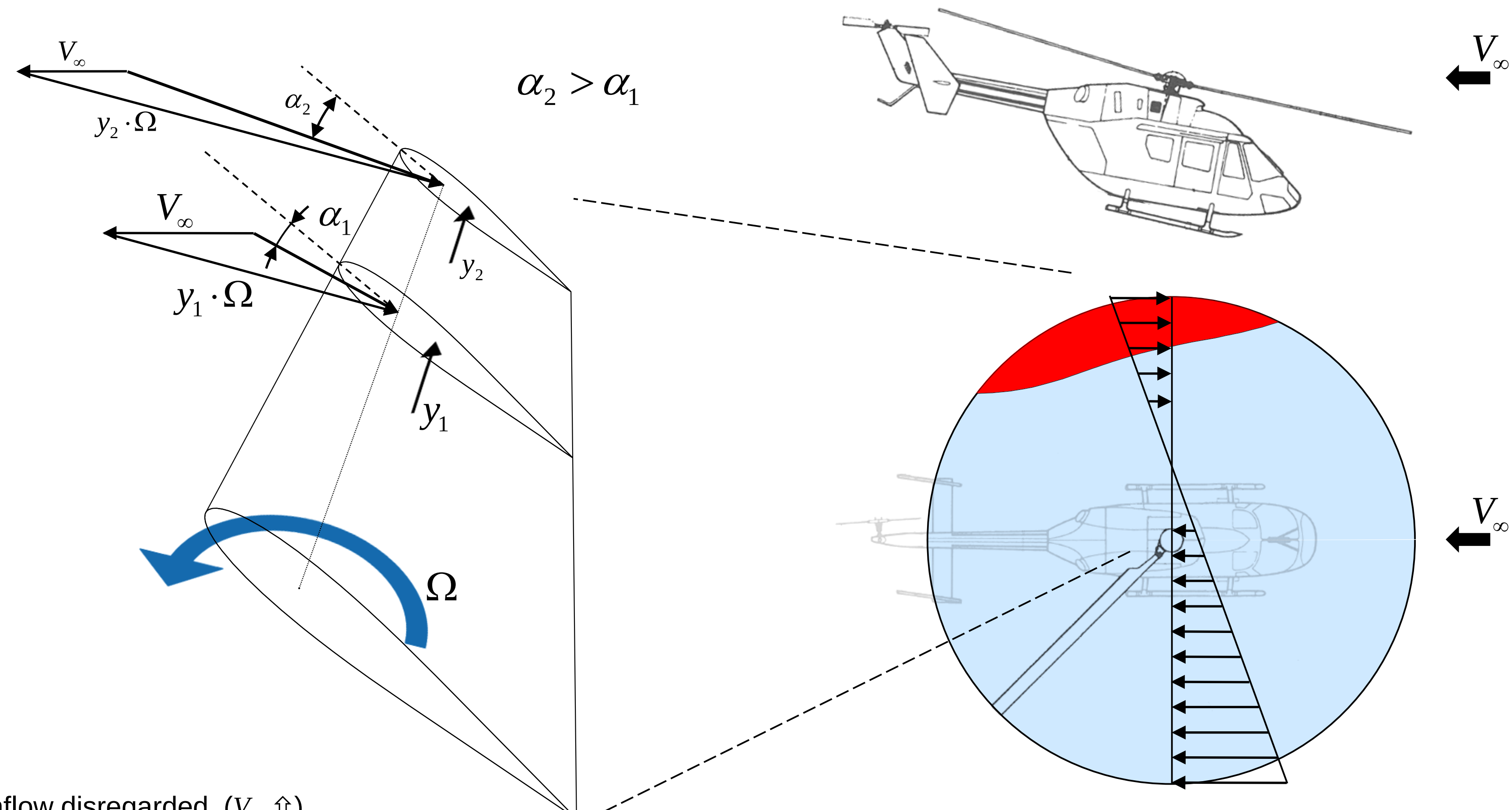


Figure from Leishman: *Principles of Helicopter Aerodynamics*

3.2. Main Rotor Design: Blade geometry

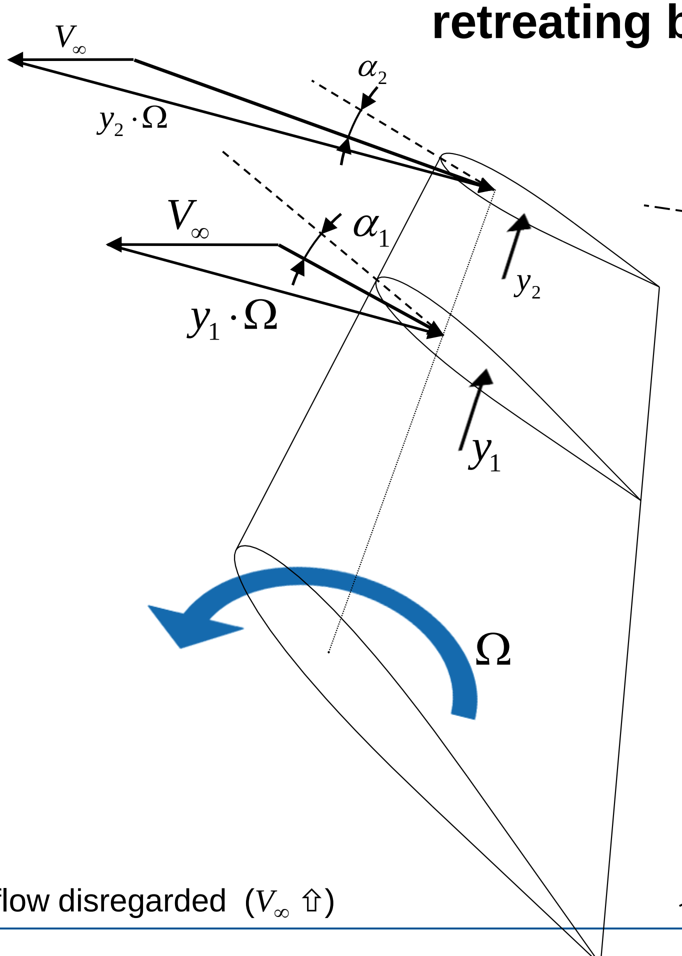
Blade without twist

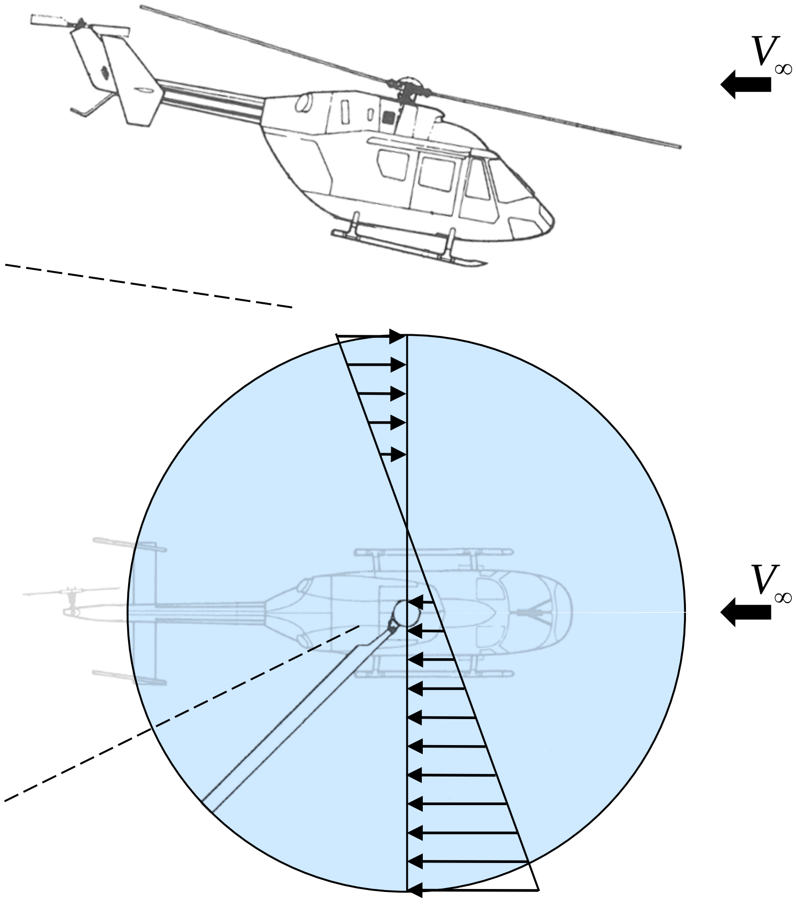


Induced inflow disregarded (V_∞ ⇧)

3.2. Main Rotor Design: Blade geometry

Negative twist provides an equalization of angle of attack at retreating blade



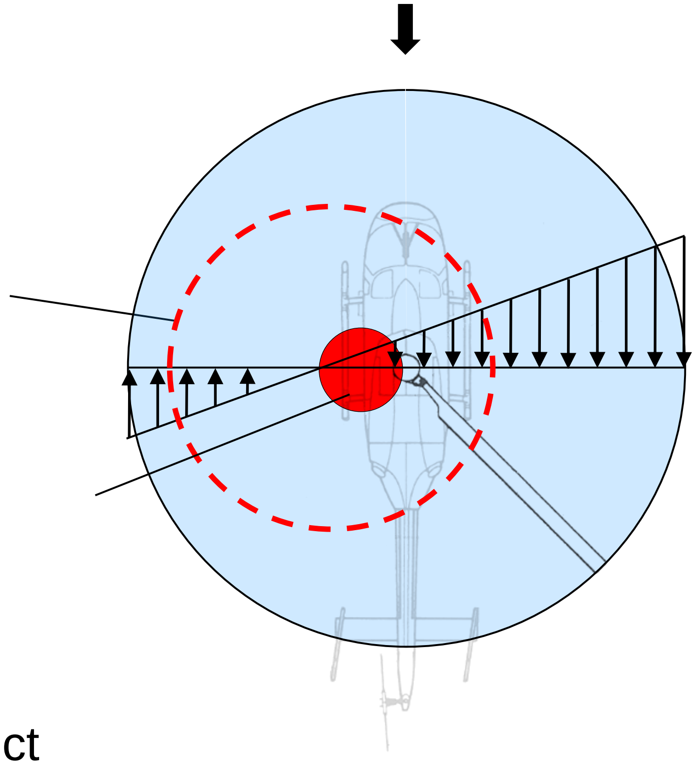


Induced inflow disregarded (V_∞ ⇧)

3.2. Main Rotor Design: Blade geometry

Twist is unfavourable:

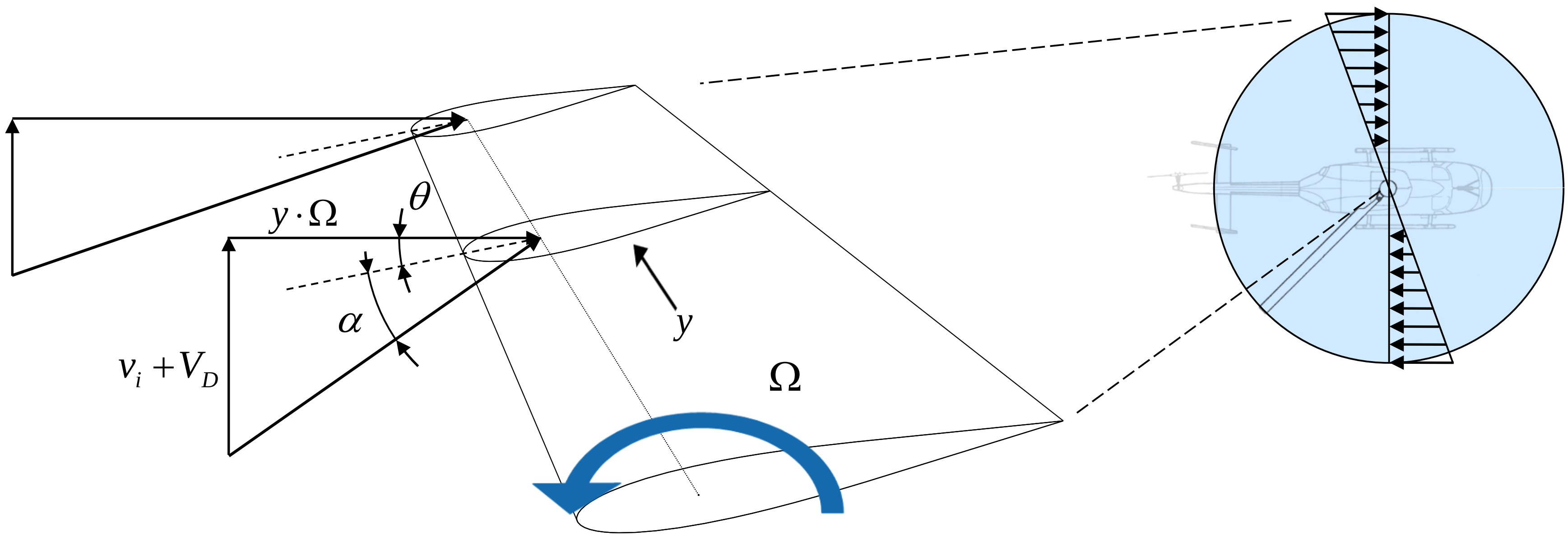
- In forward flight the twist increases the displacement of the centre of lift along the blade
  - ➔ Stronger vibration and blade loads
- Stronger downforce in area of reverse flow
- With strong twist the advancing tip can generate downthrust
  - ➔ Increased power demand
- Twist only optimum in design point, e.g. ideal twist for hover outside of ground effect too high for hover with ground effect
- Autorotation



3.2. Main Rotor Design: Blade geometry

Twist

For the fast descent, especially autorotation, the common negative twist is unfavourable. There the blade should „twist up“ to the tip.



3.2. Main Rotor Design: Blade geometry

Twist

- Common are linear twists from -8° to -15° , measured from the rotor axis to the tip.
- Especially at blade root and blade tip areas deviations can be designed, in order to diminish negative effects particularly in forward flight

Linear basic twist

Twist trend EC135

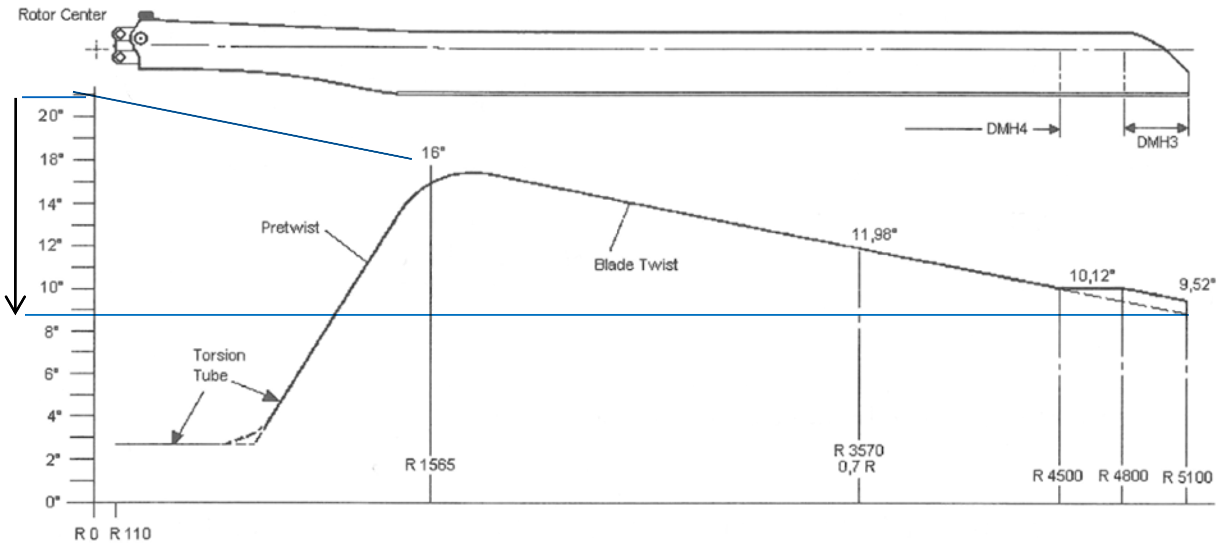


Figure from Bittner: *Flugmechanik der Hubschrauber*

3.2. Main Rotor Design: Blade geometry

Twist

Tilt Rotor:

Twist even bigger
than hover optimum

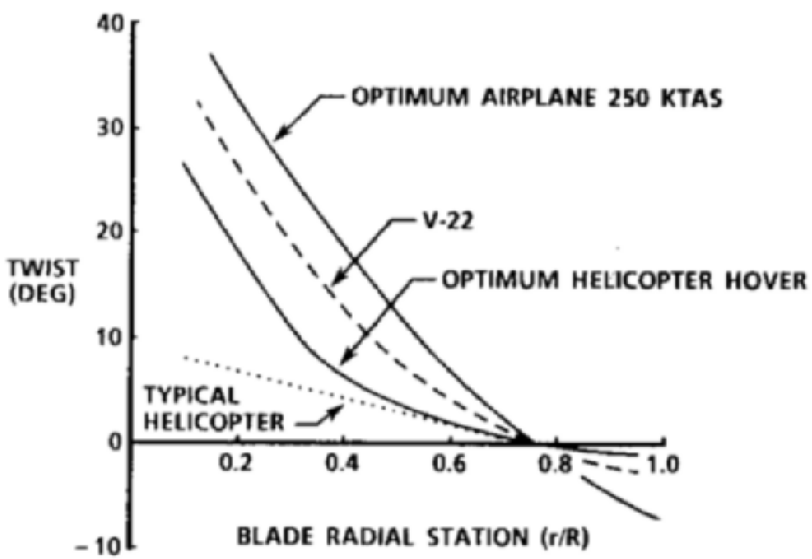


Figure from Michael K. Farrell: *Aerodynamic Design of the V-22 Osprey Proprotor*; Photo: Wikipedia, *MilborneOne*

3.2. Main Rotor Design: Blade geometry

Taper is unfavourable:

- Increase of *Figure of Merit* FM in hover of ~3% compared to twisted rectangular blades possible

- A linear taper in the outer blade section is comparatively simple to implement in production.

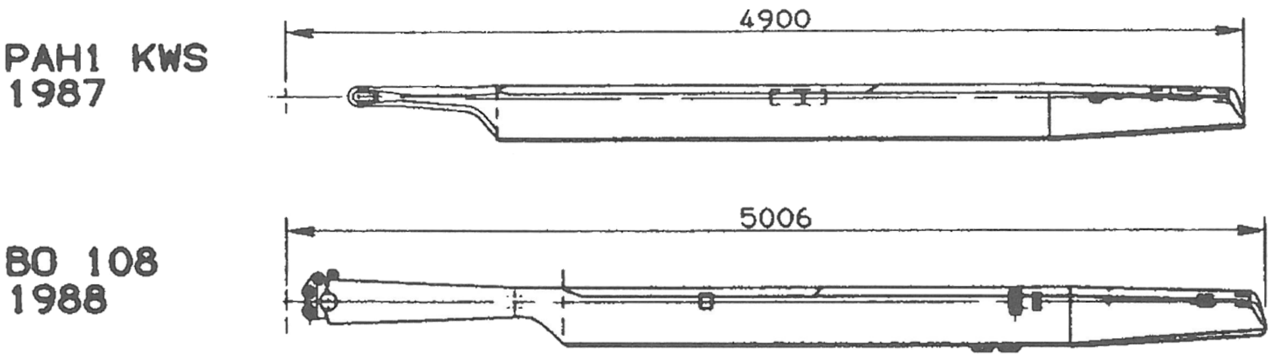


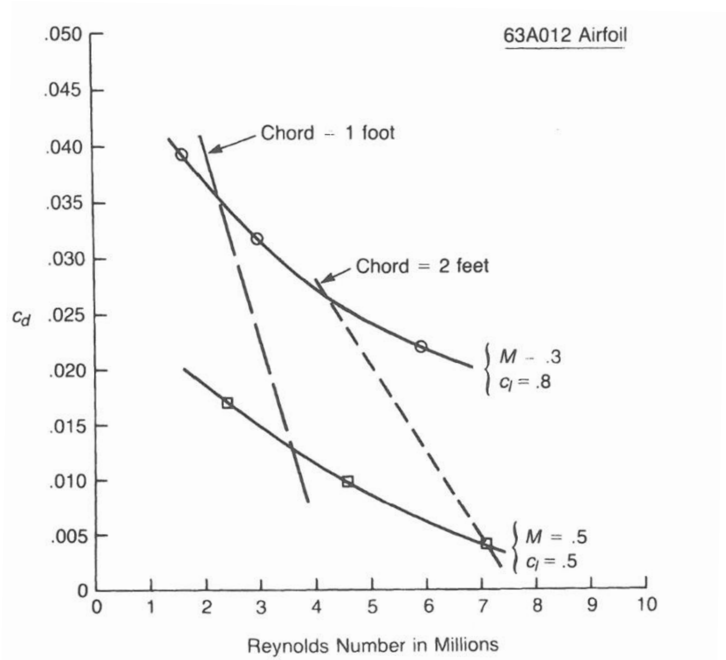
Figure from Bittner: *Flugmechanik der Hubschrauber*

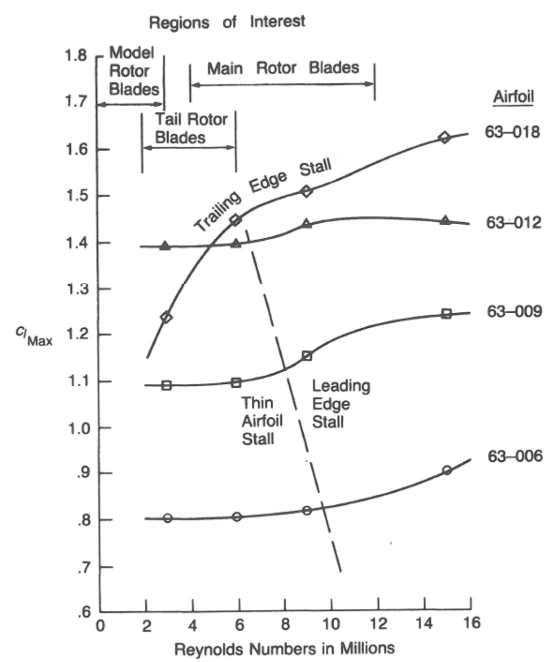
3.2. Main Rotor Design: Blade geometry

Taper is unfavourable:

- Structural design and production more complex, costs

- Small Reynolds numbers on very short chords reduce the maximum lift potential and increase the drag of the airfoil





Figures from Prouty: *Helicopter Performance, Stability and Control*

3.2. Main Rotor Design: Blade geometry

Taper is unfavourable:

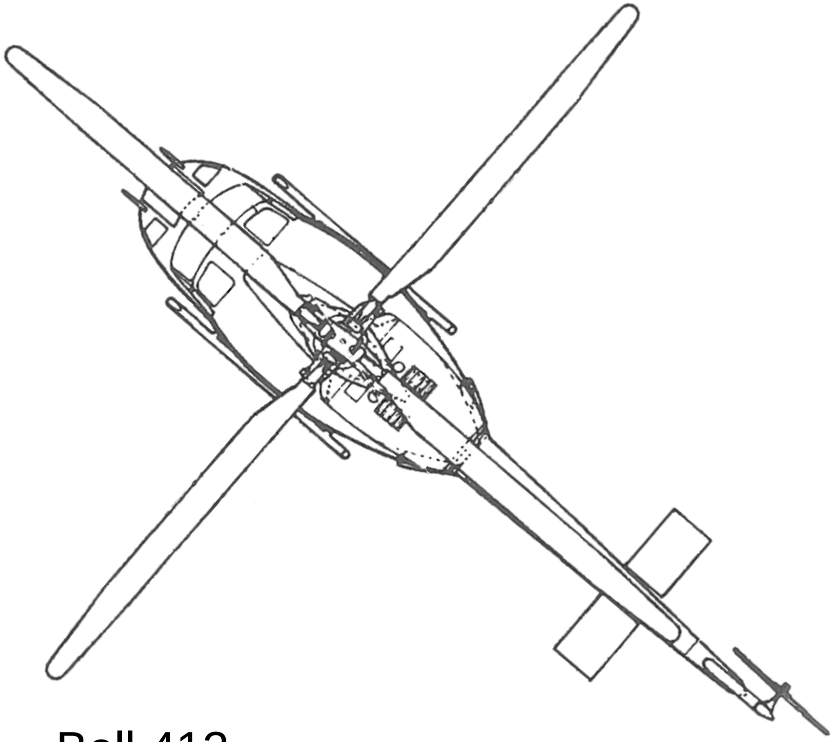
- Small blade cross section in outer sector:
  - Decreased inertia of the rotor deteriorates autorotation characteristics
  - Can be unfavourable for the blade dynamics (Resonance in 2nd bending self-oscillation)
  - Integration of weights difficult

- Taper reduces the blade area in the outer blade sector, which is crucial in forward flight.
  In order to favour this flight situation, an outward widening blade can be chosen (*inverse taper*).

3.2. Main Rotor Design: Blade geometry

Taper

Implemented blade planform



Bell 412

Eurocopter EC145

Figure from Prouty: *Helicopter Performance, Stability and Control;* Photo: *Eurocopter*

3 Sizing

3.2. Main Rotor Design

- ✓ Parameters
- ✓ Radius
- ✓ Rotational speed
- ✓ Solidity
- ✓ Number of blades
- ✓ Blade geometry
- ➔ Airfoil selection

3.2. Main Rotor Design: Airfoil selection

Retreating blade

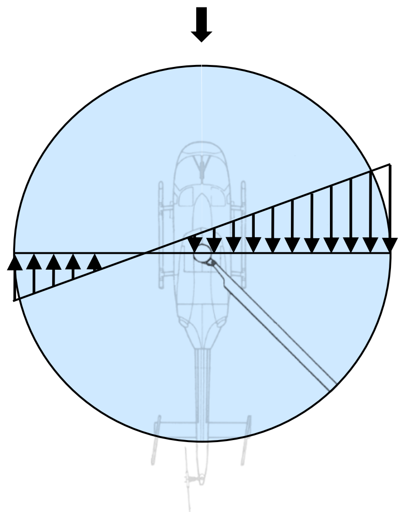
Low flow velocities up to
reverse flow in inner blade section

Requirement:

- High maximum lift coefficient C_{lmax}
- Large maximum angles of attack

Achievable due to:

- Thick profiles
- High camber



Advancing blade

High flow velocities,
outside up to transonic region

Requirement:

- Low drag at high mach numbers
- High mach number, from which rapid drag increase starts (*drag divergence)*

Achievable due to:

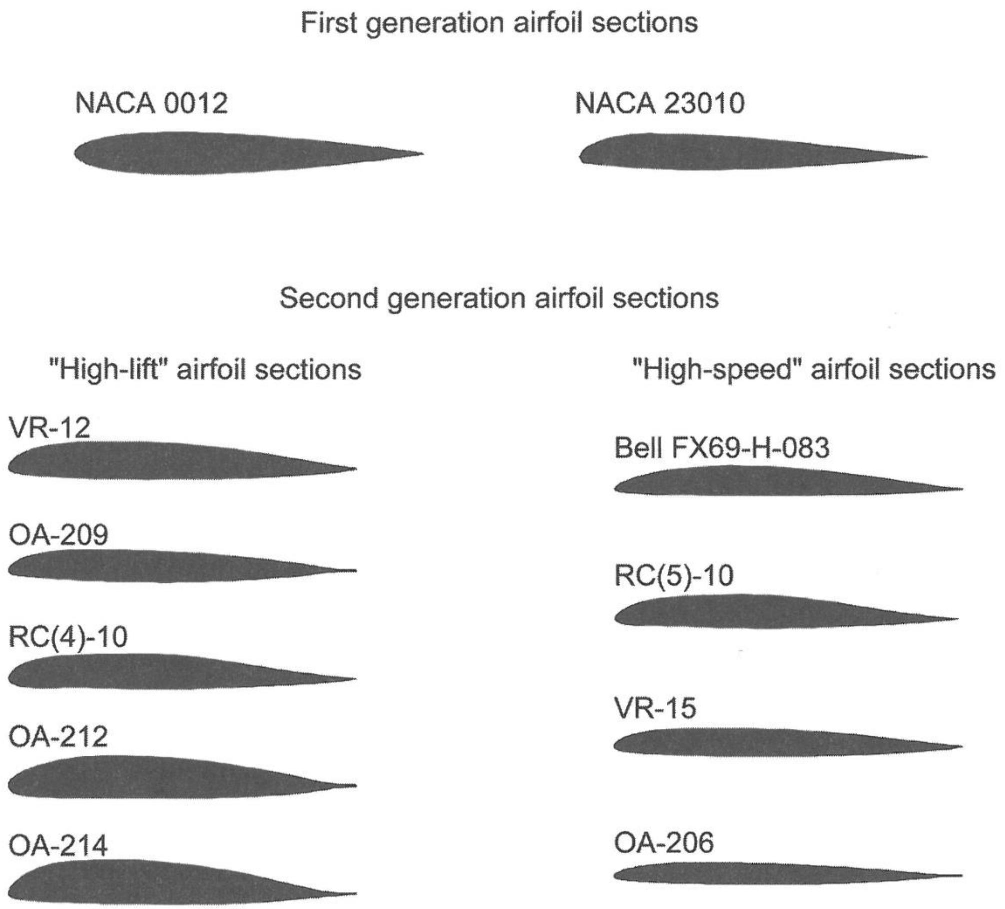
- Thin profiles
- Low camber

Additional requirements:

- Low profile pitch torques for low control loads and low elastic blade twist
- Favourable ratio C_l/C_d for typical angles of attack and mach numbers

3.2. Main Rotor Design: Airfoil selection

Implemented airfoils according to different requirements

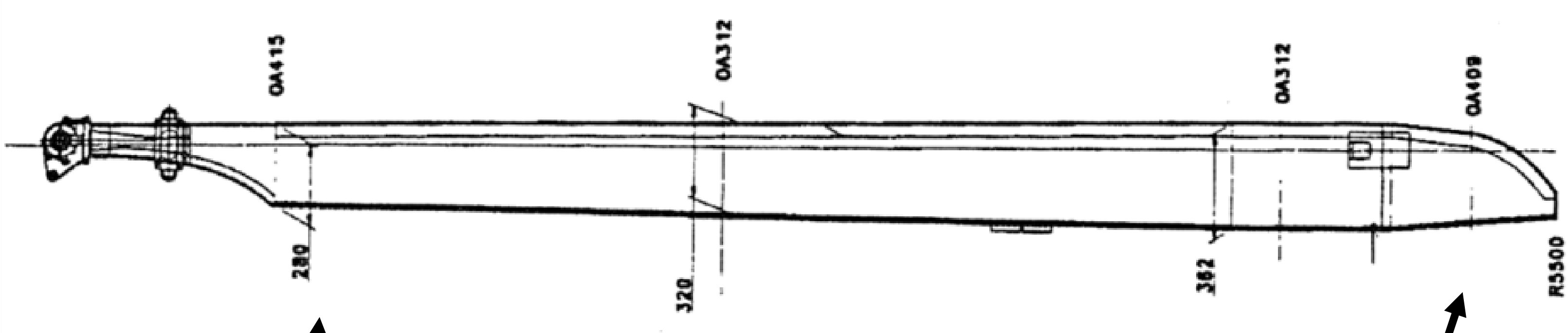


Figures from Leishman: *Principles of Helicopter Aerodynamics*

3.2. Main Rotor Design: Airfoil selection

Generally rotor blades are provided with different airfoils along the radius (Example EC145)

Increase of maximum mach number in forward flight



Thick, high camber airfoil at root,
also favourable regarding structural design aspects

Thin, flat airfoil at blade tip

Figure : DLR, C. Keßler

3 Sizing

3.3. Preliminary performance estimation

- Required hover power without ground effect
- Statistical weight estimate by components
- Validation of disc loading
- Revisal of Figure of Merit FM

3 Sizing

3.3. Preliminary performance estimation

➔ Required hover power without ground effect

- Statistical weight estimate by components
- Validation of disc loading
- Revisal of Figure of Merit FM

3.3. Preliminary performance estimation

Required hover power without ground effect

Adding the approximations for induced power P_i and profile power P_0, we get the total power in hover P_h

$$P_h = P_i + P_0$$

$$P_h = \kappa \cdot \sqrt{\frac{T^3}{2\rho\pi}} \cdot \frac{1}{R} + \frac{1}{8} \cdot \rho \cdot V_{TIP}{}^3 \cdot \sigma \cdot C_{d0} \cdot \pi \cdot R^2$$

With the preliminary parameters plus a mean airfoil drag cofficient C_{d0} and an empirical value for κ, the power consumption at an air density ρ can be determined.

3 Sizing

3.3. Preliminary performance estimation

- ✓ Required hover power without ground effect
- ➔ Statistical weight estimate by components
- Validation of disc loading
- Revisal of Figure of Merit FM

3.3. Preliminary performance estimation

Statistical weight estimate by components

Empirical models can be used to estimate component masses, based on preliminary parametres and the calculated hover power out of gound effect.

So a refined estimate of the empty weight EW can be conducted.

<u>Examples according to Layton</u>

Rotor mass, helicopter below 1.4t MTOW:

$$W_1 = (408{,}562 \cdot \ln(A \cdot 0{,}3048^2) - 1142{,}917) \cdot 0{,}4535\,kg$$

Drive system, helicopter above 11t MTOW:

$$W_{6B} = 0.453 \cdot (P \cdot 1{,}3596)^{0{,}959}\,kg$$

3.3. Preliminary performance estimation

Statistical weight estimate by components

A new maximum take off weight $MTOW$ can be estimated with the updated EW (here according to Layton):

$$MTOW = EW + m_{Fuel} + m_{Payload} + m_{Crew}$$

- $m_{Payload}$: Payload by specification, for passengers 115-160 kg per person, dependent on equipment, luggage and emergency equipment (life raft…)

- m_{Crew}: Crew, ~115kg per person inclusive equipment

3.3. Preliminary performance estimation

Statistical weight estimate by components

A new maximum take off weight $MTOW$ can be estimated with the updated EW (here according to Layton):

$$MTOW = EW + m_{Fuel} + m_{Payload} + m_{Crew}$$

- m_{Fuel}: required fuel mass, conservative approach: $m_{Fuel} = SFC \cdot 1{,}1 \cdot P_h \cdot t_{mission}$
  - Specific fuel consumption $SFC = 0{,}38 \dfrac{kg}{kW \cdot h}$
  - Hover power P_h
  - Endurance t_{misson} from required range and cruise speed
  - +10% for reserve and a potentially higher power demand

Alternatively the fuel mass can also be determined by a comparison with helicopters of similar size.

3 Sizing

3.3. Preliminary performance estimation

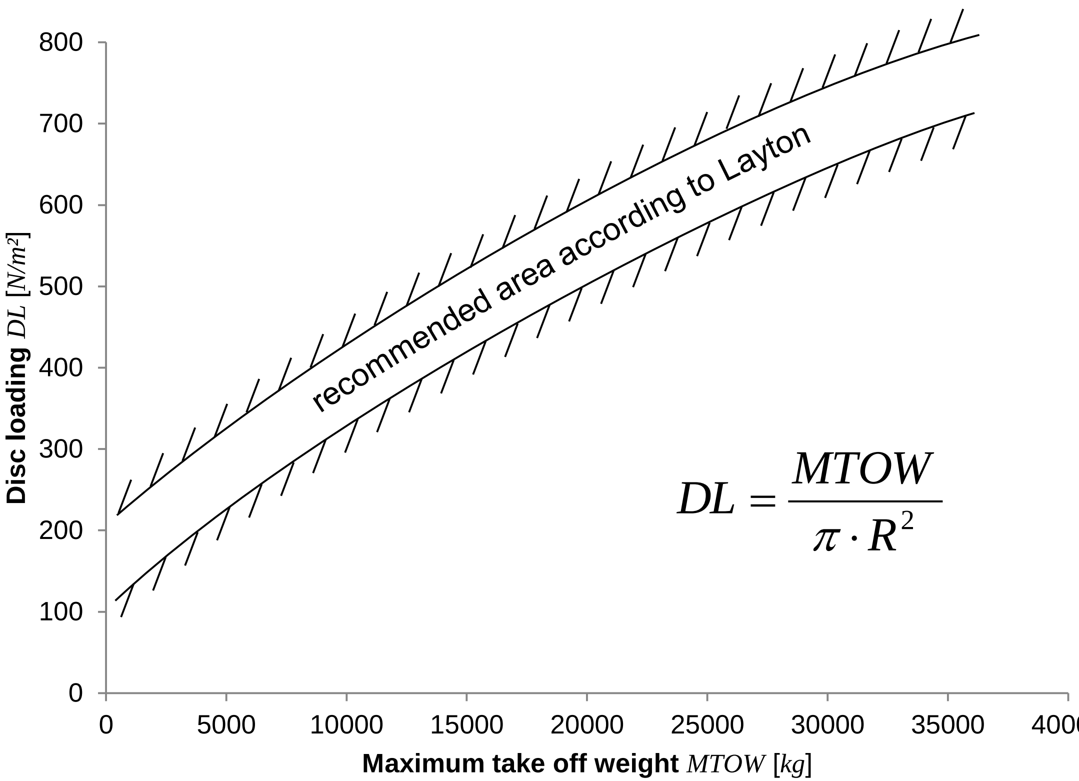
- ✓ Required hover power without ground effect
- ✓ Statistical weight estimate by components
- ➔ Validation of disc loading
- • Revisal of Figure of Merit FM

3.3. Preliminary performance estimation

Validation of disc loading

The disc loading gets reviewed with the newly determined maximum take off weight *MTOW* and the defined rotor radius.

In case the updated *MTOW* leads to an unacceptable *DL*, the iteration of the main rotor design with modified rotor radius has to be gone through again.



$$DL = \frac{MTOW}{\pi \cdot R^2}$$

Data from Layton: *Helicopter Conceptual Design*

3 Sizing

3.3. Preliminary performance estimation

- ✓ Required hover power without ground effect
- ✓ Statistical weight estimate by components
- ✓ Validation of disc loading
- ➔ Revisal of Figure of Merit FM

3.3. Preliminary performance estimation

Revisal of Figure of Merit FM

With the newly determined hover power P_h the figure of merit FM gets redefined.

$$FM = \frac{P_{i_{id}}}{P_h} = \frac{P_{i_{id}}}{P_i + P_0}$$

In case of hover with ideal rotor radius $R_{P_{h\min}}$ it applies

$$FM = \frac{2}{3 \cdot \kappa} \approx 0{,}6$$

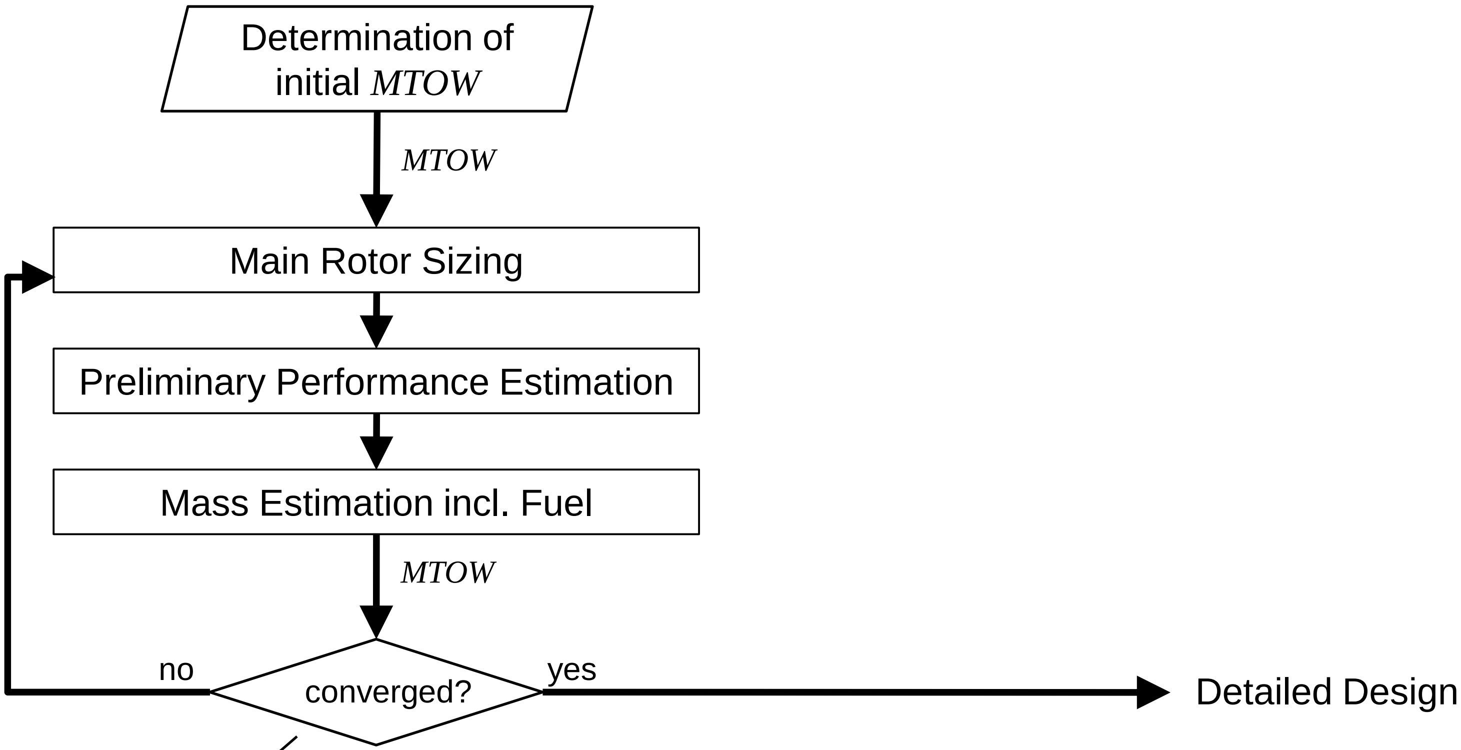
As the rotor radius is mostly chosen lower than the optimum for hover, the absolute value for the induced power P_i is higher and P_0 drops.
For balanced concepts therefore figures of merit of $FM \approx 0{,}7 - 0{,}8$ are common. Values in this range should be also expected for a new design.

3 Sizing

3.3. Preliminary performance estimation

- ✓ Required hover power without ground effect
- ✓ Statistical weight estimate by components
- ✓ Validation of disc loading
- ✓ Revisal of Figure of Merit FM

3. Sizing – First design loop



Run through till less than 10% *MTOW* difference between the iterations, accuracy limited due to mass estimate.

3 Sizing

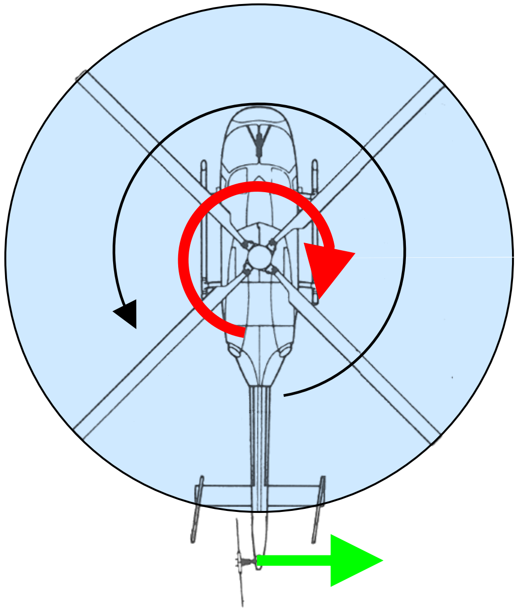
3.4. Tail rotor design

- Position
- *Pusher/Tractor*
- Direction of rotation
- Radius
- Rotational speed
- Solidity
- Airfoils
- Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

Helicopters in standard configuration require a mechanism to compensate the main rotor drive torque (*anti-torque device*).

It needs to meet all following requirements for combinations of flight weight, altitude and temperature:



- Compensation of main rotor drive torque at max. main rotor power (vertical climb)
- Thrust margin for change of direction (>10%)
- Flights in all directions up to a certain velocity
- Manoeuvrability also at autorotation

3.4. Tail rotor design

The most common solution represents the **conventional tail rotor**. This is an unducted rotor, which provides thrust in transverse helicopter direction and is located with distance to the main rotor axis.

The thrust of the tail rotor is controllable via collective blade pitch.

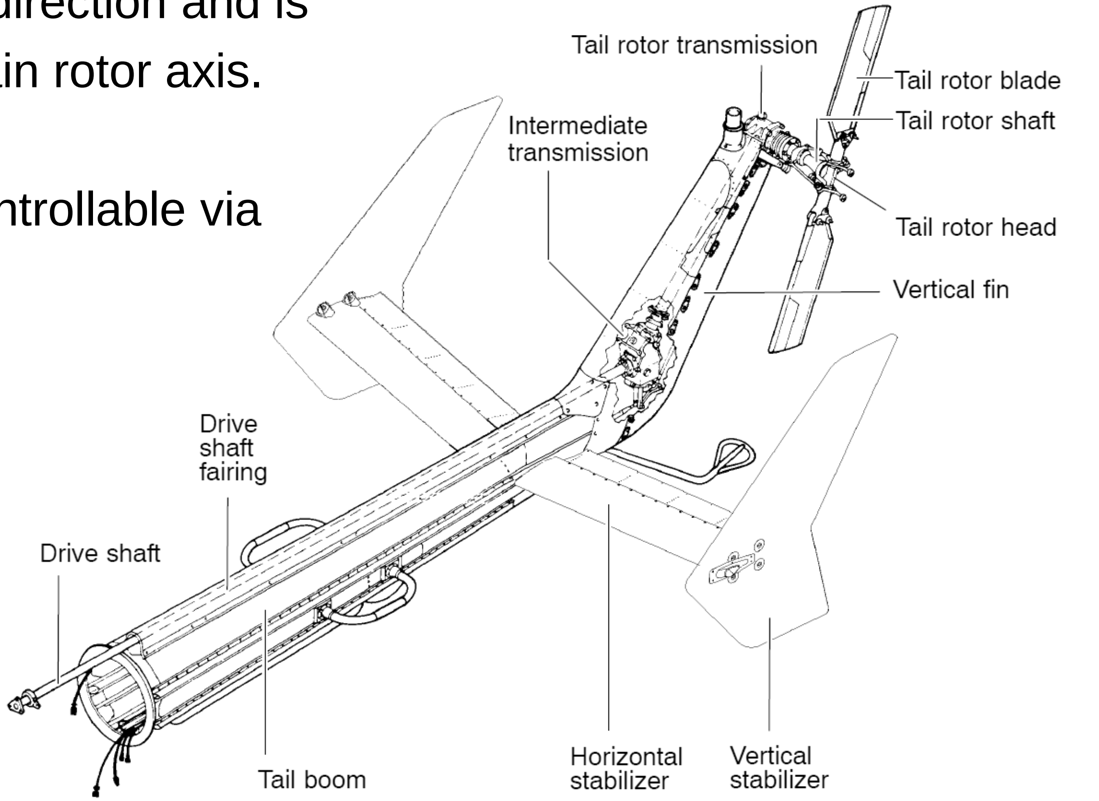


Figure: Eurocopter

3.4. Tail rotor design

The conventional tail rotor has three important functions

- Compensation of drive torque
- Directional control
- Yaw damping and wind vane stability

In contrary to the main rotor, from the tail rotor is demanded to be able to deliver thrust in both directions. As the free inflow can practically come from every direction and interacts aerodynamically with the main rotor, the flow conditions at the tail rotor are very complex.

The power consumption of the tail rotor is in hover ~10% of the required total power.

3 Sizing

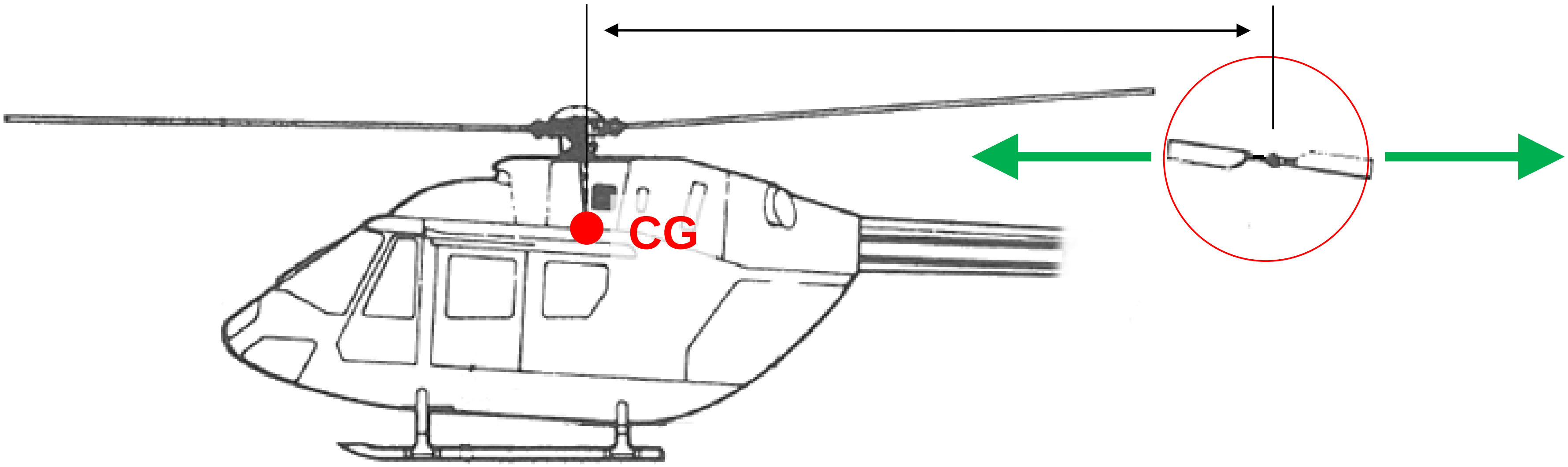
3.4. Tail rotor design

➔ Position

- *Pusher/Tractor*
- Direction of rotation
- Radius
- Rotational speed
- Solidity
- Airfoils
- Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

Position of tail rotor



- Preferably far away from the centre of gravity of the helicopter, in order to achieve the required torque with low thrust.
- Outside the movement area of the main rotor blades, in hover preferably not in main rotor downwash.
- Far in front, in order to keep the structural weight low and to keep the centre of gravity central

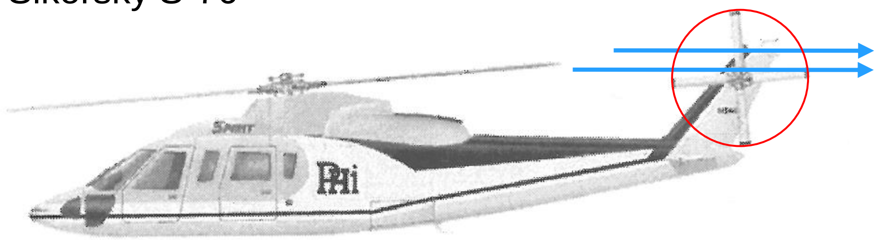
3.4. Tail rotor design

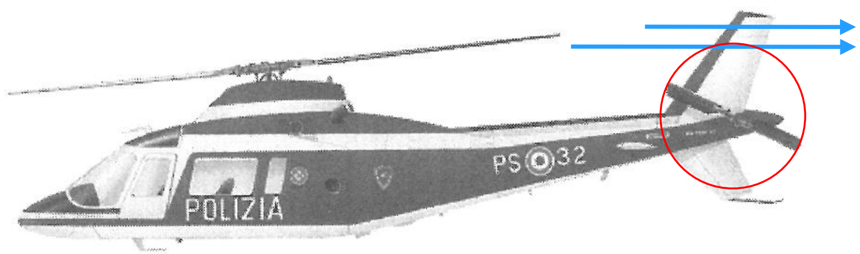
Position of tail rotor

- Low, in order to avoid vortices in the main rotor wake in cruise flight and so to reduce noise

- On the tail boom, in order to keep the drive system cost-efficient and light

Sikorsky S-76





Agusta A109

Figure from Prouty: *Helicopter Aerodynamics Volume II*

3.4. Tail rotor design

Position of tail rotor

- High, in order to create a large distance to ground, objects and people

- Level with rotor plane, in order to minimize roll coupling and to obtain a small bank angle in hover

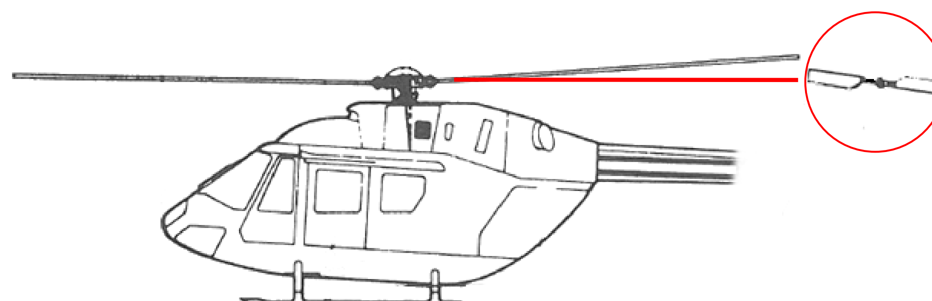
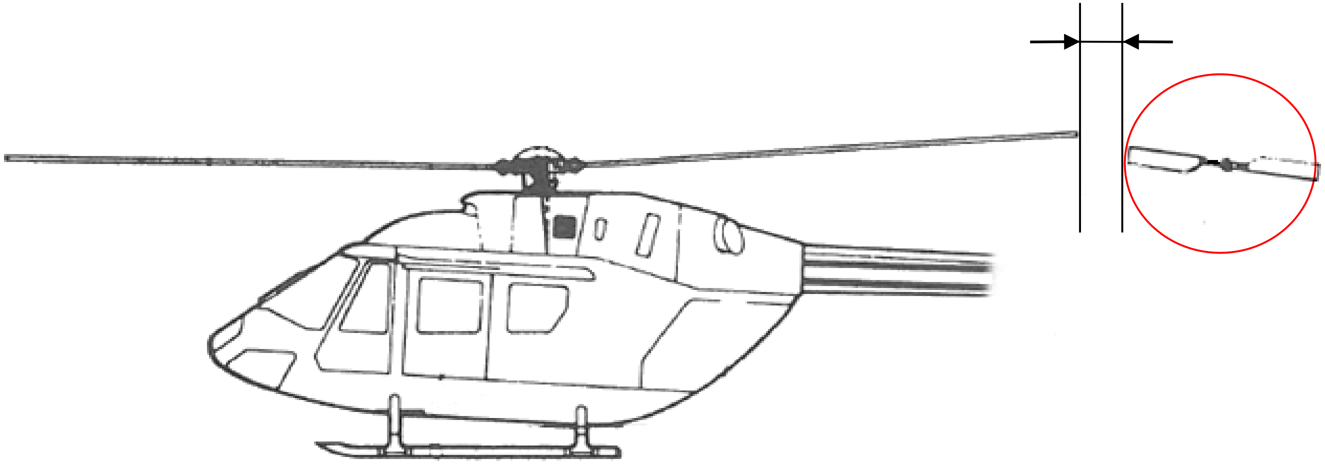


Photo: Eurocopter, Claes Axtral

3.4. Tail rotor design

Position of tail rotor– trend

Generally implemented tail rotors are located with small distance to the main rotor, however not overlapped.



According to Layton, a distance between the rotor systems in longitudinal direction of approximately $0.15\ m$ shall be provided.

3 Sizing

3.4. Tail rotor design

✓ Position

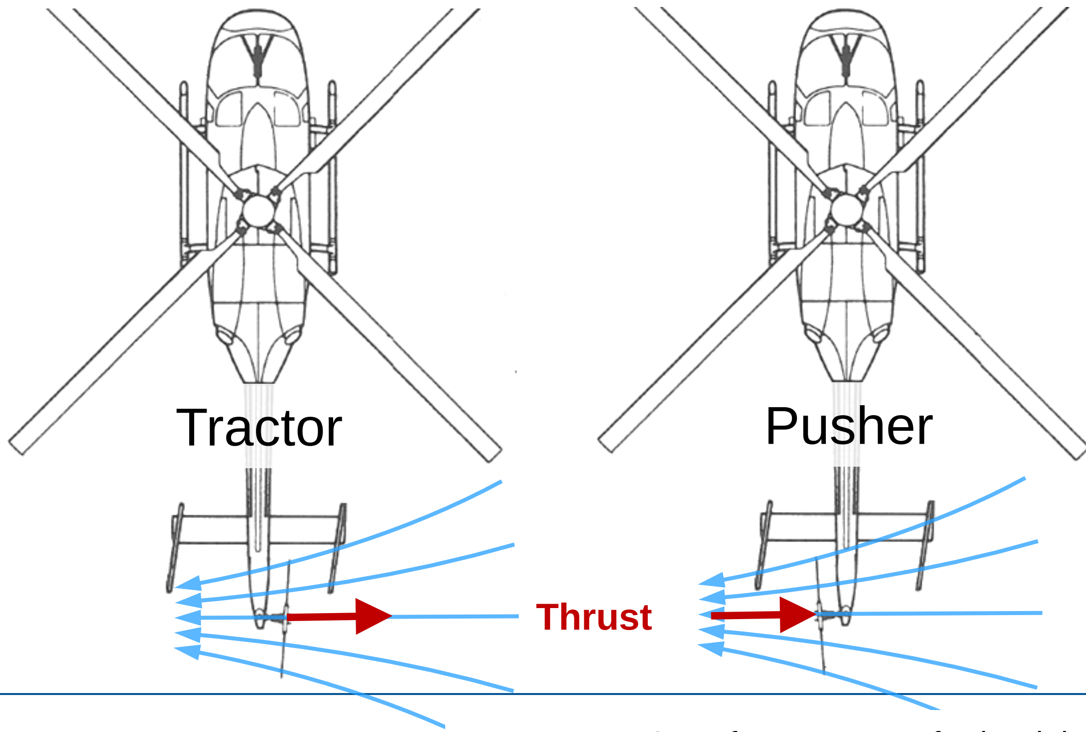
➔ *Pusher/Tractor*

- Direction of rotation
- Radius
- Rotational speed
- Solidity
- Airfoils
- Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

The position of the tail rotor regarding its load-bearing structure, defines ***pusher*** and ***tractor (puller)*** arrangement.

For biggest thrust profit a *pusher* design with large distance to the vertical fin shall be provided. In consideration of the required structure weight the distances to the fin however are mostly chosen small.



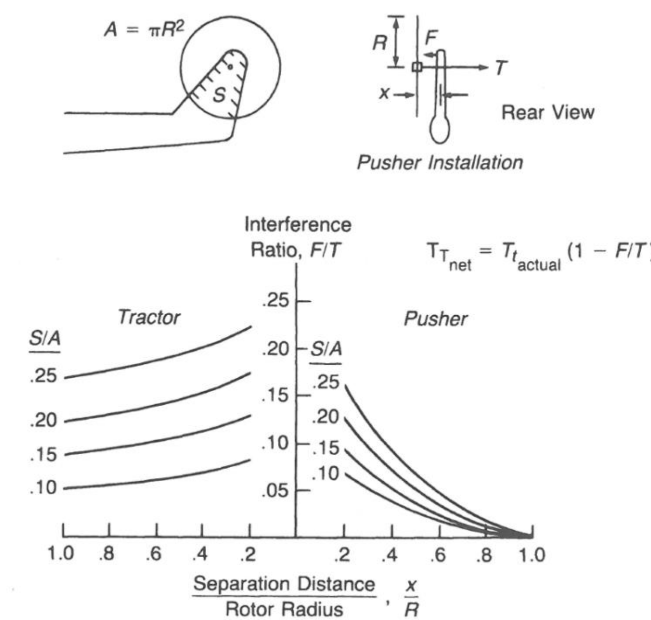


Figure from Prouty: *Helicopter Performance, Stability and Control*

3.4. Tail rotor design

Due to higher efficiency the *pusher* configuration is mostly chosen for new designs.

A speciality features the Sikorsky UH-60. Here, a *tractor* system was knowingly implemented:

- The tail rotor is canted in order to allow with a lift component positions of centre of gravity which are further behind. But as the fin is vertical, a *pusher* system would require an impractical hub spacing.

- Design requirement was the flight without tail rotor (e.g. after strafing). In case the detached parts keep their aerodynamic effectiveness, they will move away from the fuselage structure in *tractor* configuration .

Photo: *HH-60 Pave Hawk*, U.S. Air Force

3 Sizing

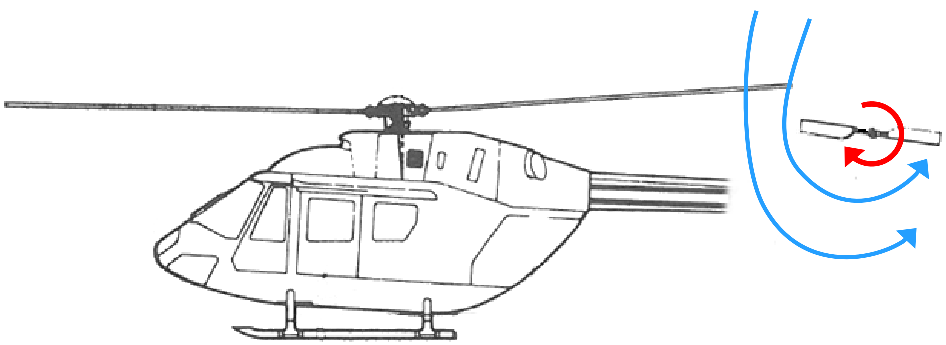
3.4. Tail rotor design

- ✓ Position
- ✓ *Pusher/Tractor*
- ➔ Direction of rotation
- Radius
- Rotational speed
- Solidity
- Airfoils
- Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

Direction of rotation of tail rotor

- Interaction with the main rotor downwash influences the tail rotor effectiveness, especially in sideward flight.
  The direction of rotation ‚**at the bottom forward**' is preferred.
- As the tail rotor blades in forward flight are striking slowlier at the top through the vortices of the main rotor, noise is reduced.
- A reason for ‚at the bottom backward' is that loose debris is catapulted away from the fuselage.



AH-56A, required pedal control inputs

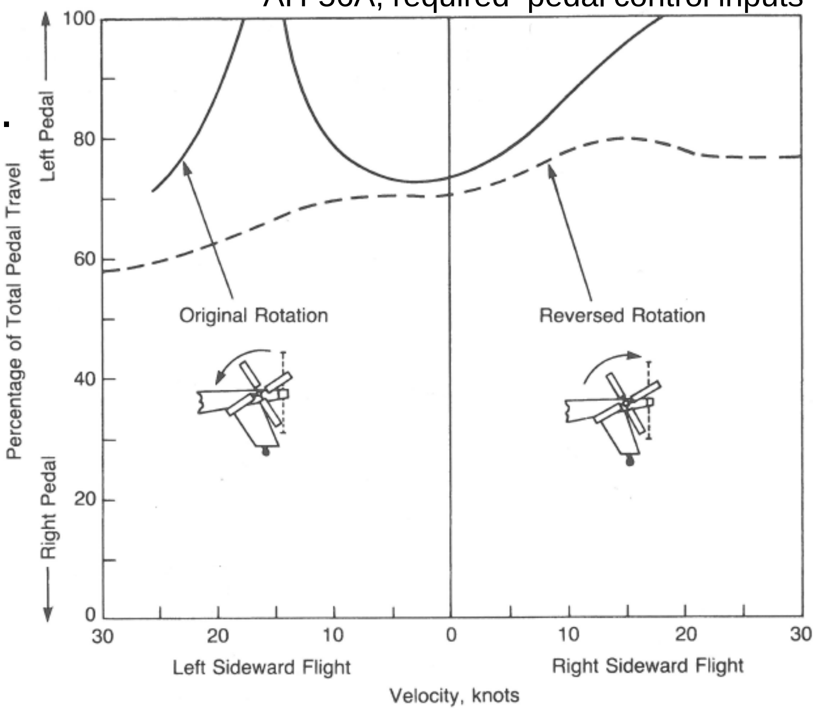


Figure from Prouty: *Helicopter Performance, Stability and Control*

3.4. Tail rotor design

As the importance of rotational direction of the tail rotor for the effective directional control became historically apparent only relatively late, an inversion to the ‚correct‘ direction of rotation was subsequently conducted at several helicopter types.

A simple constructional solution consists of reinstalling the existing tail rotor gear box after being turned by 180° . As the tail rotor thereby changed sides from one to the other, designed *pusher rotors so* turned to *tractor* rotors.

Bell 204

Bell 205A++

Photos: Alan Radecki

3 Sizing

3.4. Tail rotor design

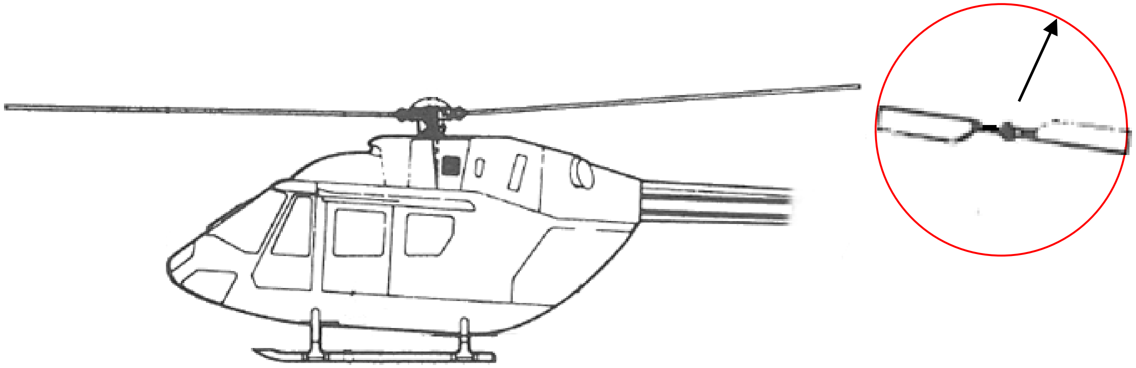
- ✓ Position
- ✓ *Pusher/Tractor*
- ✓ Direction of rotation
- ➔ Radius
- Rotational speed
- Solidity
- Airfoils
- Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

The **tail rotor radius** should be large

- In order to keep the power consumption for torque compensation and yaw control low

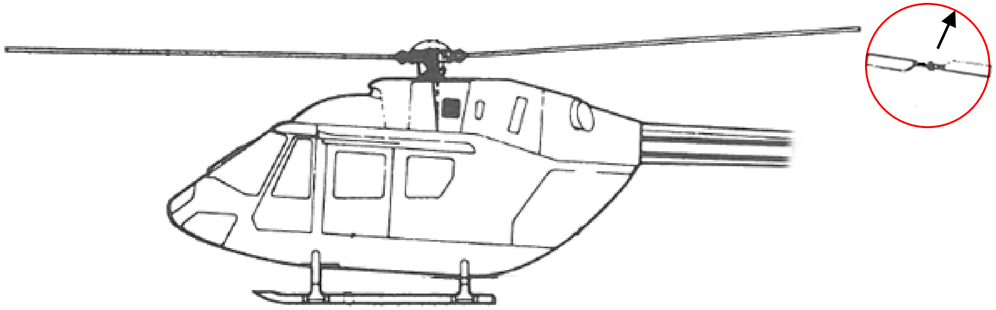
- For large wind vane stability and yaw damping at given solidity



3.4. Tail rotor design

The **tail rotor radius** should be small for following reasons:

- Low weight, favourable position of centre of gravity
- Low parasite drag of tail rotor hub
- Large distance to ground
- Small space of rotor axis to main rotor and empennage required
- High induced velocities of the tail rotor shift the appearance of the vortex ring state in sideward flight to high velocities



3.4. Tail rotor design

Tail rotor radius (R_{TR}) – trends

Approximation according to Layton:

$$R_{TR} = 0{,}4\,\mathrm{m} \cdot \sqrt{2{,}2 \cdot \frac{\mathrm{MTOW}}{1000\,\mathrm{kg}}}$$

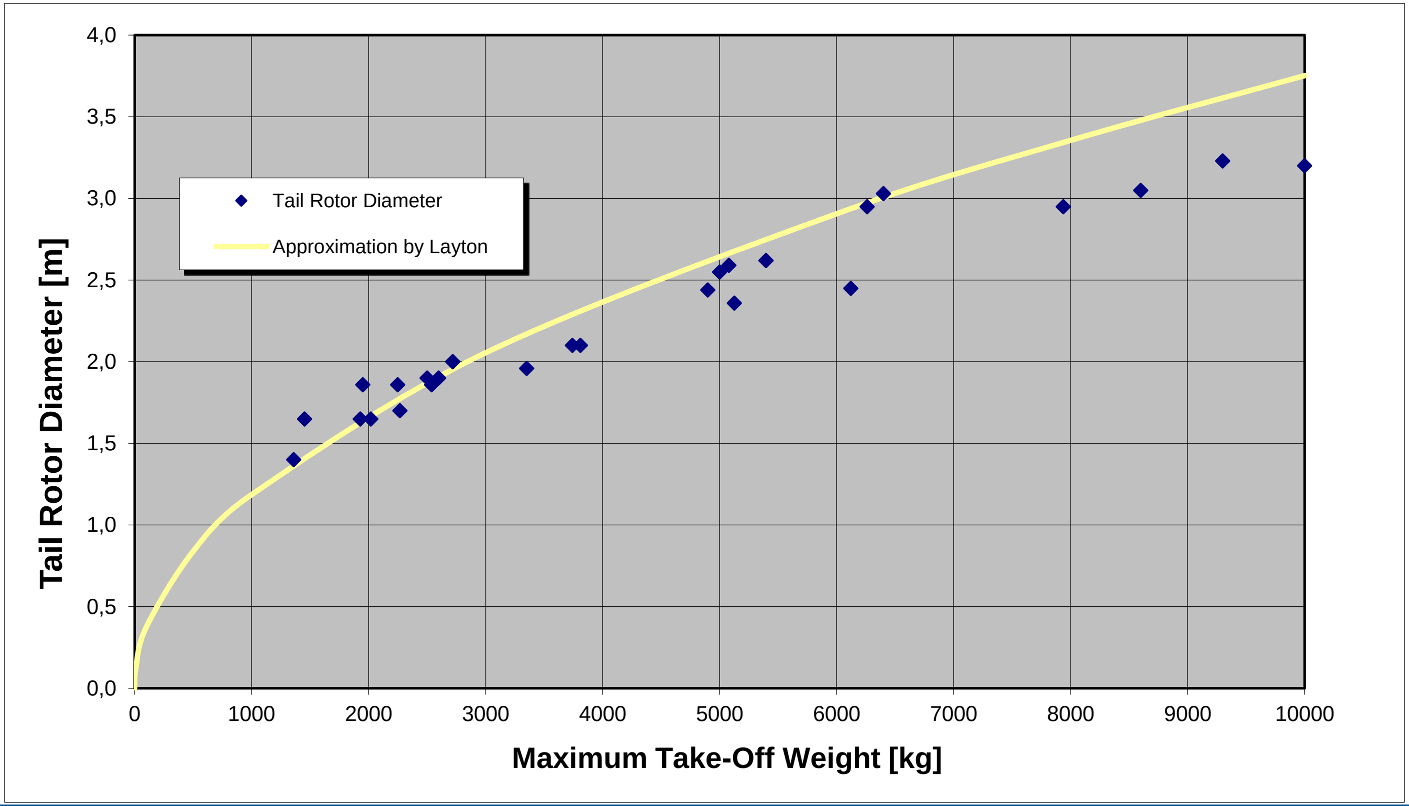


Diagram according to Ch. Keßler

3.4. Tail rotor design

Tail rotor diameter – trends

$$\frac{Tail\ rotor\ radius}{Main\ rotor\ radius} = \frac{R_{TR}}{R_{MR}} = 0{,}15 \ldots 0{,}25$$

Increasing with the DL of the main rotor:

- At higher disc loading the main rotor requires so much power that a large and efficient tail rotor gets attractive

- A low DL is leads to a large main rotor radius and a long tail boom. A small tail rotor facilitates the compliance with a favourable centre of gravity position close to the main rotor axis

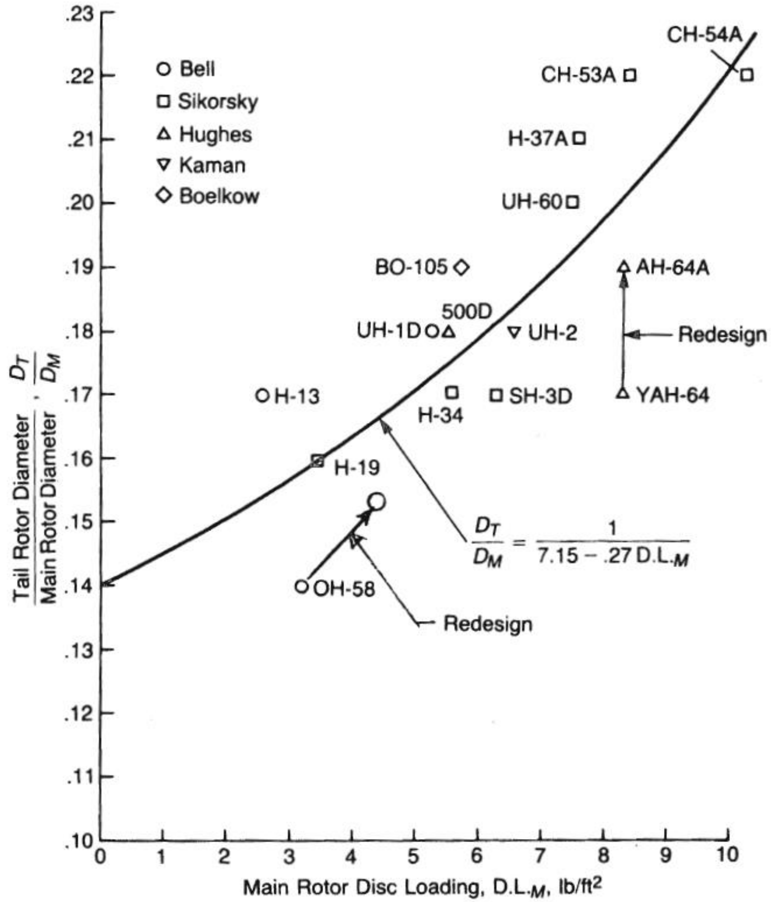


Diagram from Prouty: *Helicopter Stability, Performance and Control*

3 Sizing

3.4. Tail rotor design

- ✓ Position
- ✓ *Pusher/Tractor*
- ✓ Direction of rotation
- ✓ Radius
- ➔ Rotational speed
- Solidity
- Airfoils
- Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

For the **rotational speed of the tail rotor** and the thereby related **tip speed** apply similar limits as for the main rotor.

High rotational speed

- Allows a low weight of the tail rotor
- Increased *stall* margin
- Due to lower torques a lighter drive system required

Low rotational speed

- Decreases noise; due to higher frequencies the tail rotor has a pronouced noise signature compared to the main rotor
- Delays mach number effects at the advancing blade

3.4. Tail rotor design

The tip speeds of the tail rotor are in average at similar values as V_{TIP} of the main rotors.

Correspondent to the smaller radius of the tail rotor, the rotational speed Ω_{TR} of the tail rotor is higher.

$$\Omega_{TR} \approx \frac{R}{R_{TR}} \cdot \Omega$$

According to Layton:

$$\Omega_{TR} \approx 4{,}5 \cdot \Omega$$

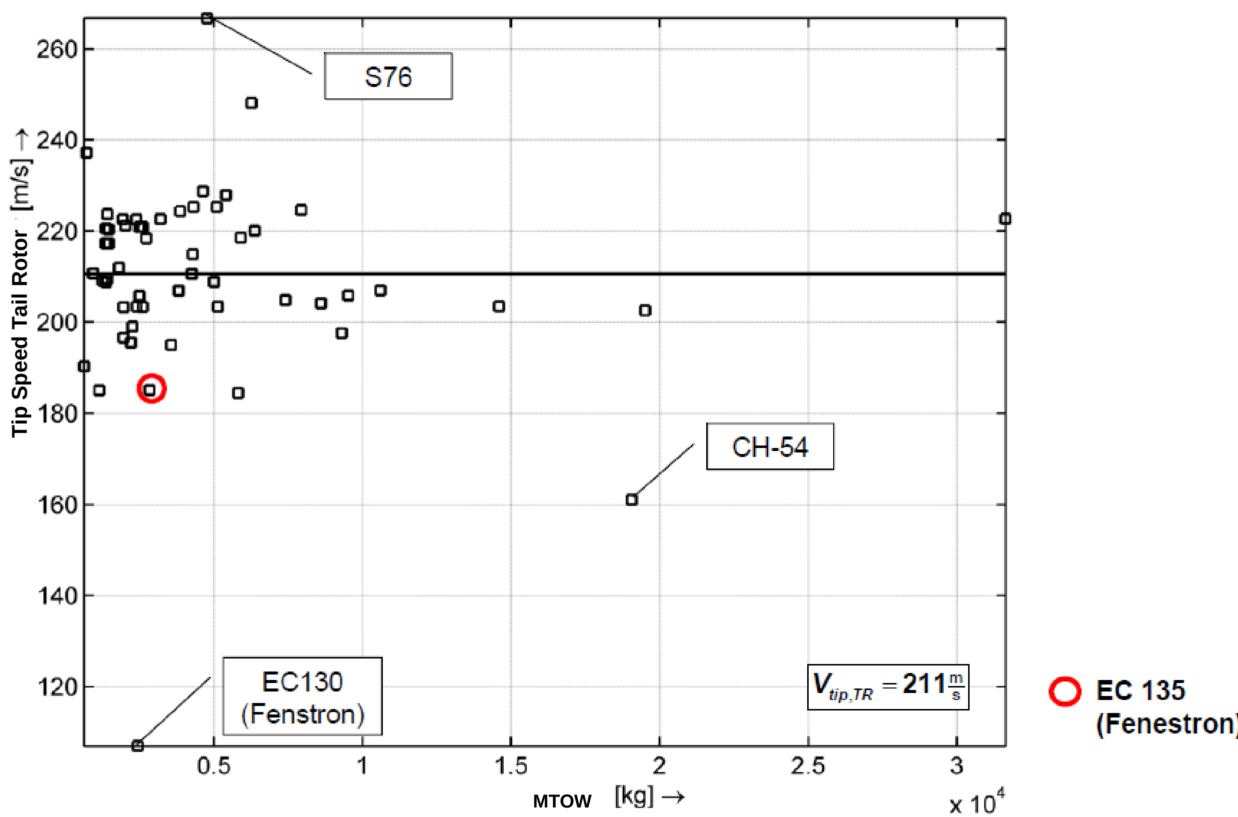


Figure: DLR, Ch. Keßler

3 Sizing

3.4. Tail rotor design

- ✓ Position
- ✓ *Pusher/Tractor*
- ✓ Direction of rotation
- ✓ Radius
- ✓ Rotational speed
- ➔ Solidity
- • Airfoils
- • Number of blades/blade geometry
- • Alternatives to conventional tail rotor

3.4. Tail rotor design

Solidity of tail rotor

The solidity of the tail rotor needs to be chosen that under all flight conditions enough thrust is available to compensate the main rotor torque. For manoeuvring and as a safety margin an additional thrust margin to the stall of the tail rotor needs to be provided (>10%).

As criteria mostly the vertical climb at full power or the sideward flight up to a certain requested velocity is used.

These can be critical in different conditions:

- Low altitude:
  High available engine output, high torque to compensate

- High altitude:
  Low air density for thrust generation

3.4. Tail rotor design

Thrust requirement of tail rotor

Example: Vertical climb with maximum power

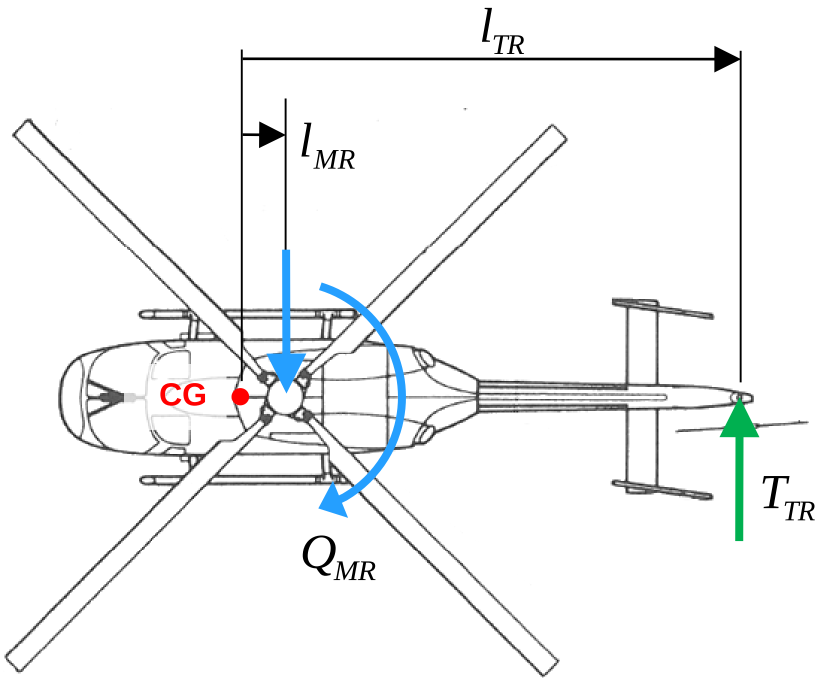
Tail rotor thrust T_{TR} dependent on main rotor drive torque Q_{MR}

$$0 \overset{!}{=} T_{TR} \cdot l_{TR} - T_{TR} \cdot l_{MR} - Q_{MR}$$

$$T_{TR} = \frac{Q_{MR}}{l_{TR} - l_{MR}}$$

$$C_{T_{TR}} = \frac{T_{TR}}{\rho \cdot A_{TR} \cdot V_{TIP,TR}^2}$$

$$\sigma_{TR_{\min}} = \frac{C_{T_{TR,\max}}}{\left(\frac{C_T}{\sigma}\right)_{\max}}$$



3.4. Tail rotor design

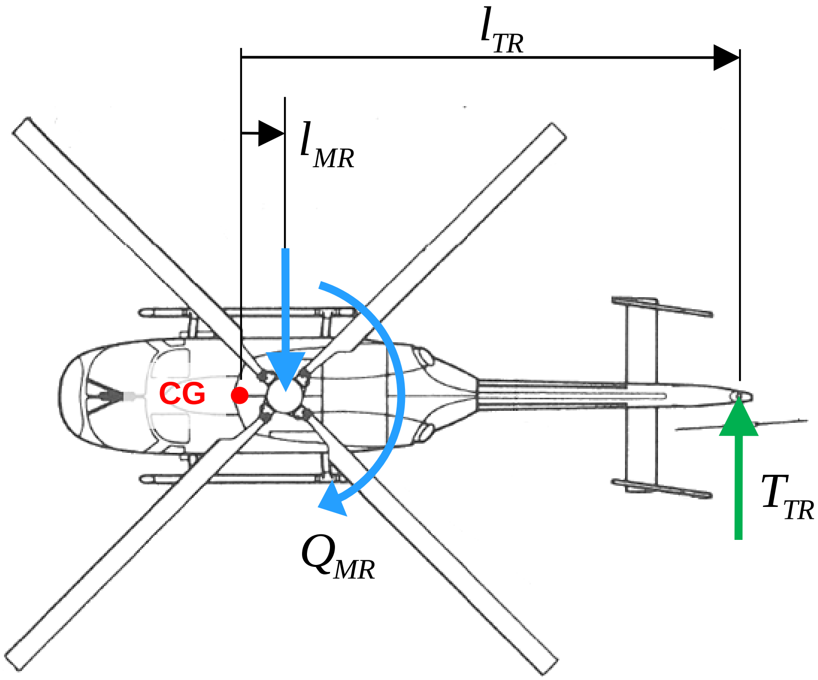
Thrust requirement of tail rotor

Example: Vertical climb with maximum power

$$\sigma_{TR_{\min}} = \left(\frac{Q_{MR}}{\rho}\right)_{\max} \cdot \frac{1}{A_{TR} \cdot V_{TIP,TR}^2 \cdot \left(\frac{C_T}{\sigma}\right)_{\max} \cdot \left(l_{TR} - l_{MR}\right)}$$

$$\frac{C_T}{\sigma} \overset{\text{Hover}}{\overset{\downarrow}{=}} \frac{\overline{C}_L}{6} \qquad \left(\frac{C_T}{\sigma}\right)_{\max} \approx \frac{C_{L_{\max}}}{6}$$

$$\sigma_{TR_{\min}} \approx \left(\frac{Q_{MR}}{\rho}\right)_{\max} \cdot \frac{6}{A_{TR} \cdot V_{TIP,TR}^2 \cdot C_{L_{\max}} \cdot \left(l_{TR} - l_{MR}\right)}$$



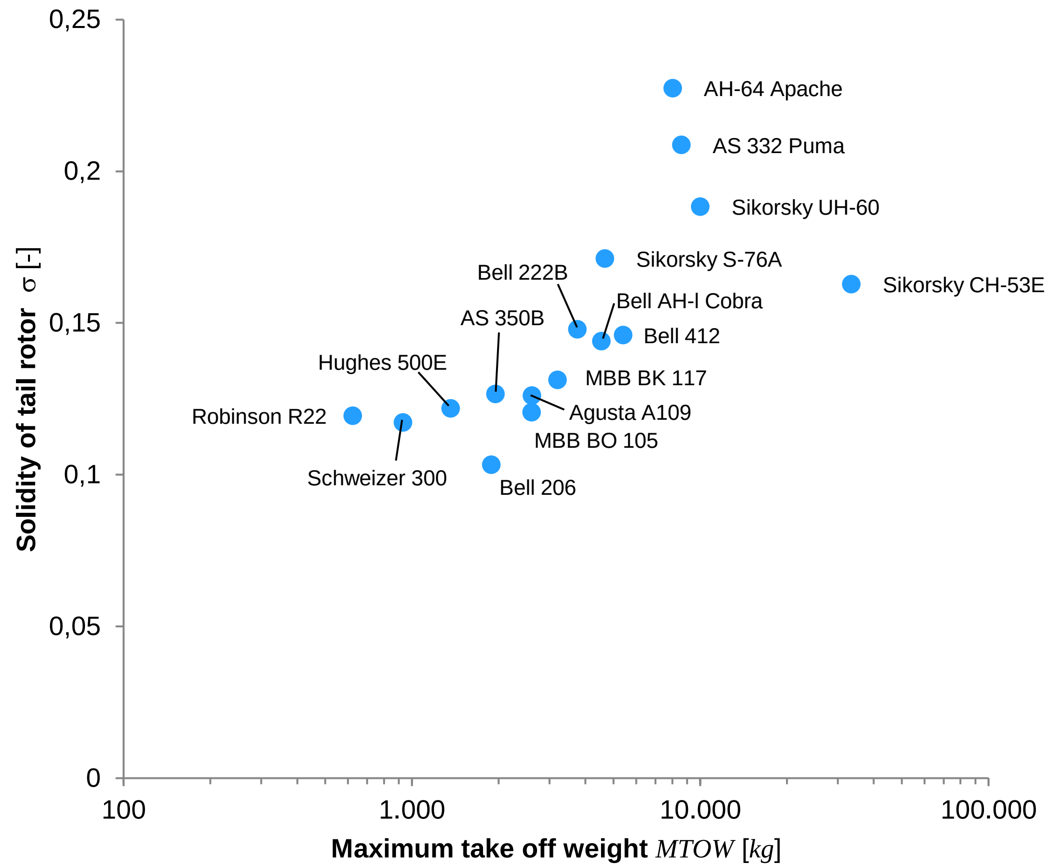
3.4. Tail rotor design

Required solidity of tail rotor

$$\sigma_{TR_{\min}} \sim \left(\frac{Q_{MR}}{\rho}\right)_{\max}$$

As $\left(\frac{Q_{MR}}{\rho}\right)_{\max}$ is still unknown,

first of all a comparison with existing helicopters with similar requirements is used to determine $\sigma_{TR_{\min}}$.



Data from Leishman: *Principles of Helicopter Aerodynamics*

3 Sizing

3.4. Tail rotor design

- ✓ Position
- ✓ *Pusher/Tractor*
- ✓ Direction of rotation
- ✓ Radius
- ✓ Rotational speed
- ✓ Solidity
- ➔ Airfoils
- Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

Profiling

- Short, torsional rigid blades allow more strongly cambered airfoils than at the main rotor, however at higher control loads

- Camber can be unfavourable at low (negative) angles of attack

- Compressibility effects have lower impact on total power consumption, thicker airfoils possible

➔ Compared to main rotor airfoils, a higher maximum lift coefficient is possible in favour of a lower solidity (weight...), however with higher control loads

3 Sizing

3.4. Tail rotor design

- ✓ Position
- ✓ *Pusher/Tractor*
- ✓ Direction of rotation
- ✓ Radius
- ✓ Rotational speed
- ✓ Solidity
- ✓ Airfoils
- ➔ Number of blades/blade geometry
- Alternatives to conventional tail rotor

3.4. Tail rotor design

Number of blades

- Low number of blades simple to manufacture and to maintain
- High number of blades has lower tip losses

➔ Mostly chosen so that the *aspect ratio* of the blades amounts to

$$AR = \frac{\text{radius}}{\text{chord length}} = 4 \ldots 9$$

Twist and taper

- Favourable in a design point, reduction of efficiency at lower or reversed flow
- Twist unfavourable when passing vortex ring state
- Higher production costs

➔ No or low twist (>-10°)
Mostly untapered

Photo: U.S. Army

3 Sizing

3.4. Tail rotor design

- ✓ Position
- ✓ *Pusher/Tractor*
- ✓ Direction of rotation
- ✓ Radius
- ✓ Rotational speed
- ✓ Solidity
- ✓ Airfoils
- ✓ Number of blades/blade geometry
- ➔ Alternatives to conventional tail rotor

3.4. Tail rotor design

The ducted tail rotor (*Fenestron*)

- Rotor is protected from damage by contact with objects
- Protection of people, no free spinning blades
- Duct always has efficiency advantage
  - Reduction of induced and profile power due to stream expansion
  - Accelerated stream on round inlet provides thrust contribution
  - Reduced tip losses
  - Transformation of spin in thrust due stator row
  - No blockage by vertical tail
  - Less drag in forward flight
- Noise shielding due to housing

Photo: Eurocopter, Olivier Balmain

3.4. Tail rotor design

The ducted tail rotor (*Fenestron*)

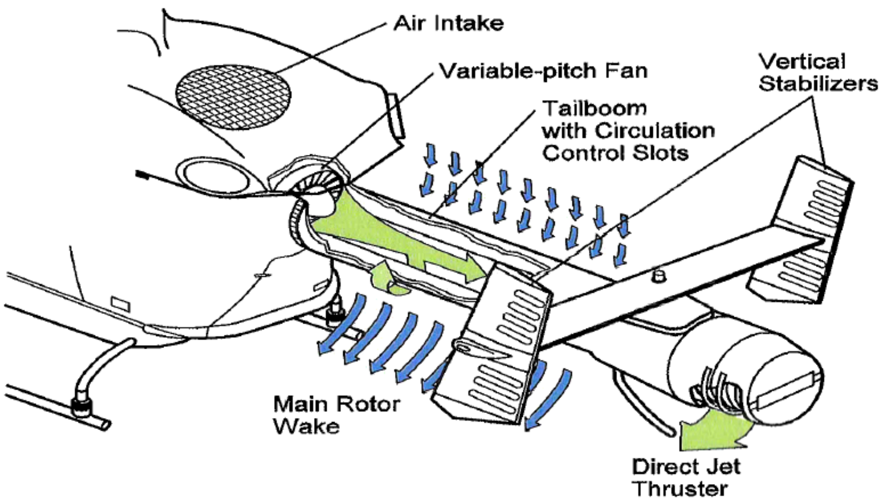
- Higher weight, especially in tail unfavourable for centre of gravity
- Dimensions limited due to housing, smaller than conventional tail rotors
- Thick fin necessary (~20% of diameter)
  - High drag in forward flight
  - Rotor inflow perturbations possible in forward flight due to housing

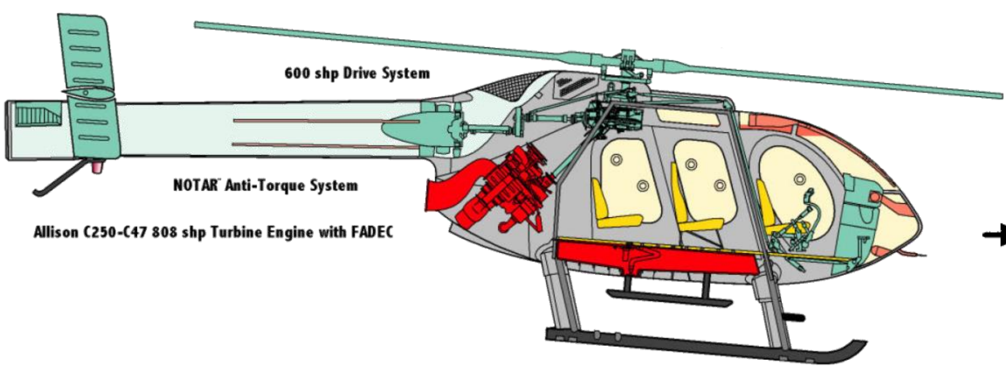
Photo: Flug Revue,Grägel

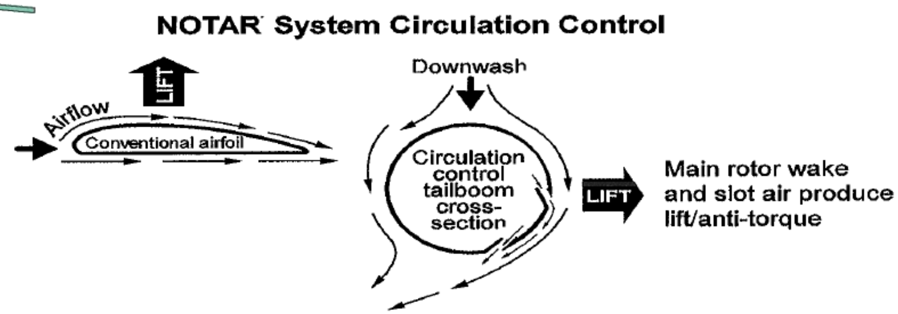
3.4. Tail rotor design

NOTAR (No Tail Rotor)

- Fan in fuselage
- Slots along the tail boom
  provide torque compensation due to
  lateral force in rotor downwash
- Rotatable steering thruster
  at the end of tail boom







Figures: MD Helicopters

3.4. Tail rotor design

NOTAR (No Tail Rotor)

- No damage due to objects
- Protection of people, no openly spinning parts
- Fuselage shields outside noise of fan
- Cabin noise level increased by fan

Photo: MD Helicopters

3. Sizing

3.5. Fuselage and Stabilisers

➔ Fuselage and fittings

- Stabilisers

3.5. Fuselage and fittings

Besides the connection of all individual components of the helicopter and the absorption of flight loads, further demands on the fuselage can be made:

- Low structure weight
- Simple accessibility of cargo hold and cabin
- Suitable placement of payload (rows of seats, level cargo area…)
- Low drag in forward flight
- Low download in hover
- Possibility to hold external loads (rescue hoist, additional tanks, weapons carrier)
- Crash safety, bullet resistance
- Maintenance-friendly installation of components (engines, gear box..)
- Simple design, low production expenditure, low manufacturing costs
- …

3.5. Fuselage and fittings

As the design objectives generally pose a conflict, the single aspects need to be assessed correspondent to the requirements of the helicopter.

Depending on the focus of the mission there are different designs:

Photos: Dan Marsh; Kaman Helicopters

3.5. Fuselage and fittings

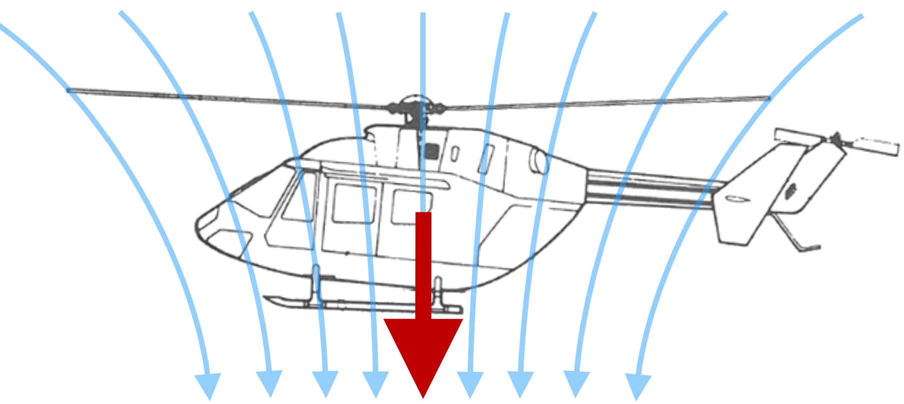
Depending on the focus of the mission there are different designs:

Photo: Eurocopter, Eric Raz; AgustaWestland; Eurocopter

3.5. Fuselage and fittings

Besides the structure mass, the fuselage design influences the flight performances of the helicopter essentially:

- Increased thrust requirement in hover/slow flight as a consequence of *download* in the *downwash.*



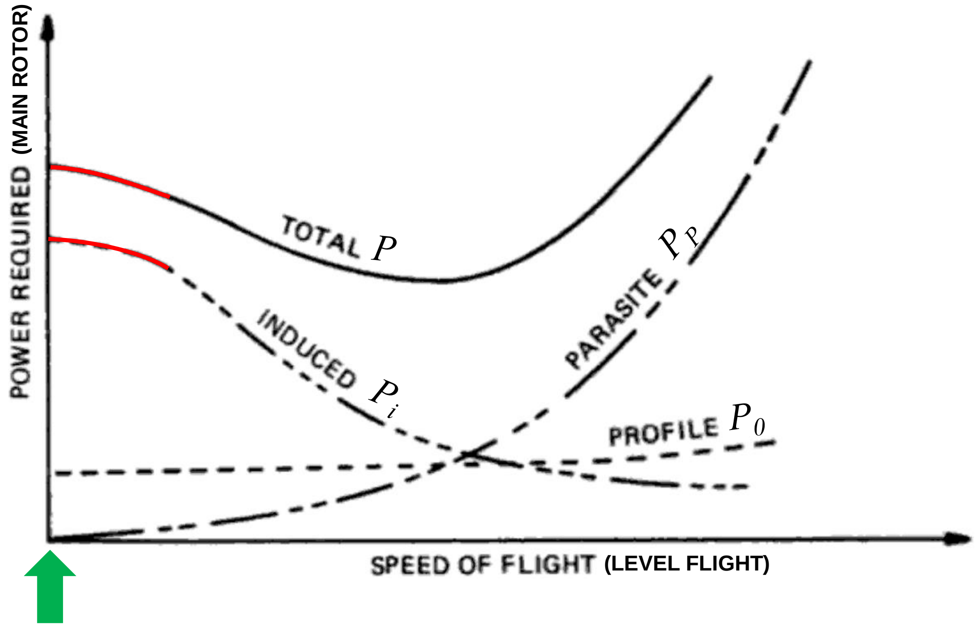


Figure from Stepniewski: *Rotary-Wing Aerodynamics*

3.5. Fuselage and fittings

Increased thrust requirement in hover/slow flight arises as a consequence of *download* in the *downwash*.

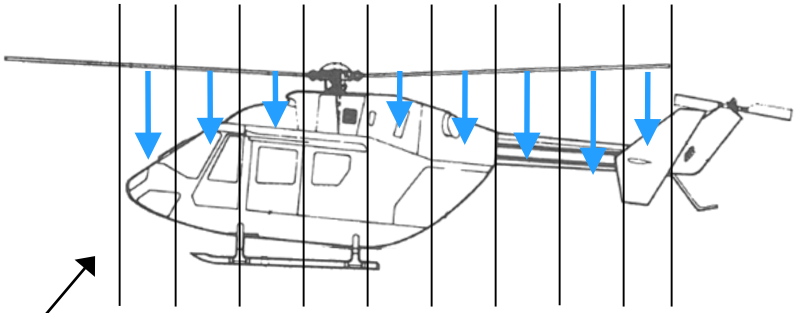
At the calculation of the required hover power this can be considered by adding an additional thrust component D_V (*vertical drag*) besides the weight force.

$$T = W + D_V$$

As the fuselage download in hover on the other hand depends on the rotor thrust, D_V can be assumed as part of the total thrust via a scale factor k_{DL} .

Thereby applies $T = W + k_{DL} \cdot T$

and $T = \dfrac{W}{1 - k_{DL}}$



with $k_{DL} \approx 2...9\,\%$
(to determine from *CFD* calculations, fuselage element method…)

3.5. Fuselage and fittings

Besides the structure mass, the fuselage design influences the flight performances of the helicopter essentially:

- Due to high fuselage drag, the parasite power P_P increases. It dominates the power consumption in forward flight.

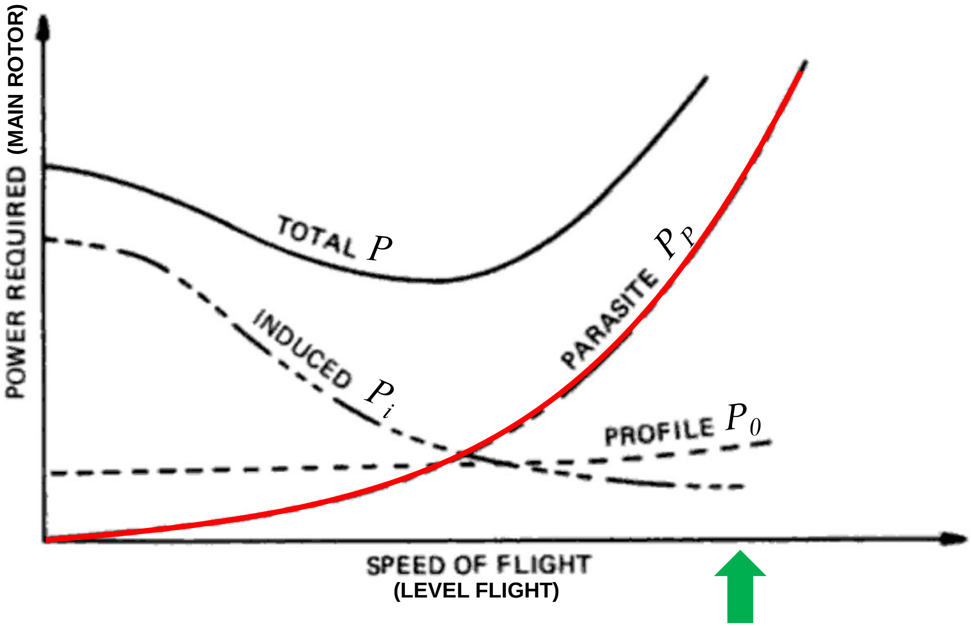


Figure from Stepniewski: *Rotary-Wing Aerodynamics*

3.5. Fuselage and fittings

Due to high fuselage drag, the parasite power increases P_P.
It dominates the power consumption in forward flight.

$$P_P = V_\infty \cdot D$$

$$P_P = V_\infty \cdot q \cdot C_D S = \frac{1}{2}\rho \cdot V_\infty^3 \cdot C_D S$$

$$D = \frac{1}{2}\rho V_\infty^2 \cdot C_D S = q \cdot C_D S$$

← Drag area $C_D S \approx \text{const.}$ neglecting the

- small variation of flow direction
- Reynolds number
- mach number

Dynamic pressure, variable with density and speed of flight

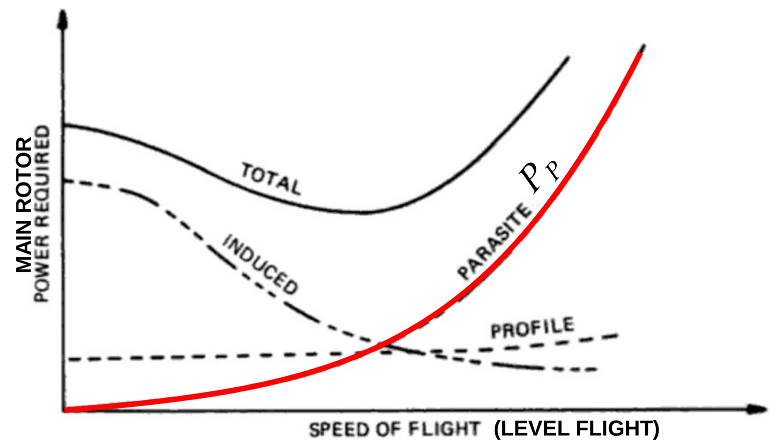


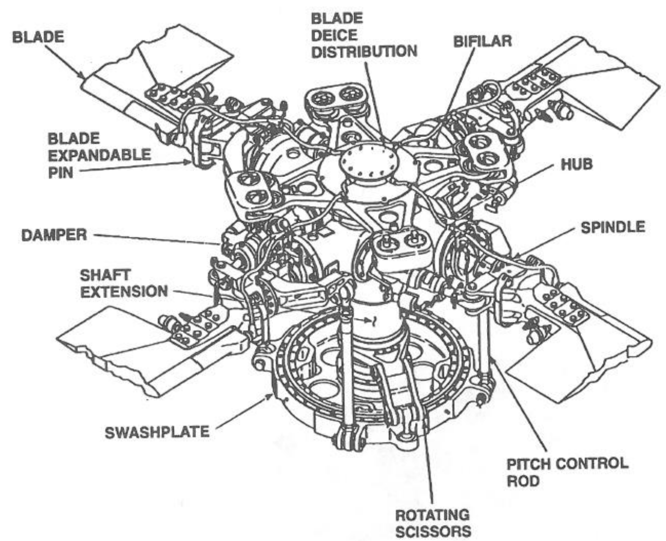
Figure from Stepniewski: *Rotary-Wing Aerodynamics*

3.5. Fuselage and fittings

For determination of the total drag area it is common to sum up and estimate all single components (fuselage, empennage, landing gear, armament…).

Typical amounts (example) of single components for total *drag area* (*Equivalent Flat Plate Area)* $C_D S$:

Component	% of Total
Fuselage	30
Nacelles	6
Rotor hub & shaft	35
Tail rotor hub	4
Main landing gear	6
Tail landing gear	4
Horizontal tail	1
Vertical tail	1
Rotor/fuselage interference	7
Exhaust system	3
Miscellaneous	3
Total	100



Numerous functional requirements complicate the aerodynamic optimisation of the rotor hub

Table from Leishman: *Principles of Helicopter Aerodynamics;* Figure from Leoni: *Black Hawk: The Story of a World Class Helicopter*

3.5. Fuselage and fittings

Fuselage drag – trends

Assuming that the density of parts are similar at different helicopters, mass and fuselage volume behave proportionally

$$MTOW \sim V$$

$MTOW \sim V \sim l^3$ with a characteristic length l .

Generally the air drag is proportional to a characteristic area S

$D = \frac{1}{2}\rho V_\infty^2 C_D \cdot S$ so $D \sim C_D S \sim l^2$

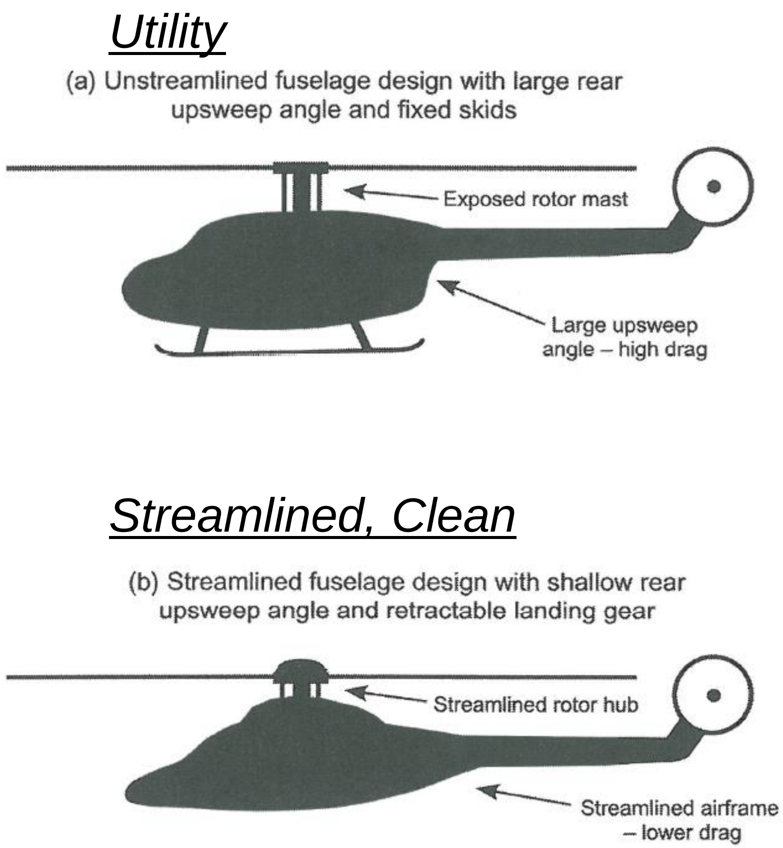
consequently is estimated

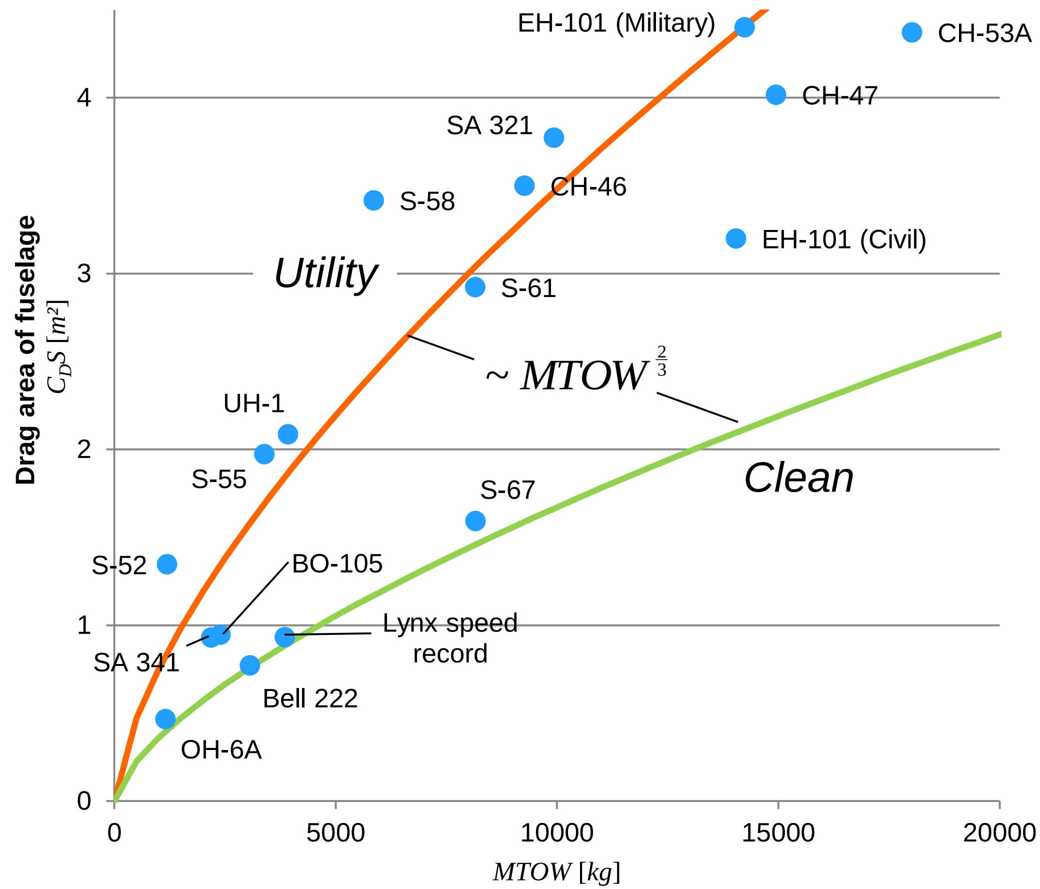
$$D \sim C_D S \sim MTOW^{\frac{2}{3}}$$

Photo from Leishman: *Principles of Helicopter Aerodynamics*

3.5. Fuselage and fittings

3.5. Fuselage drag – trends

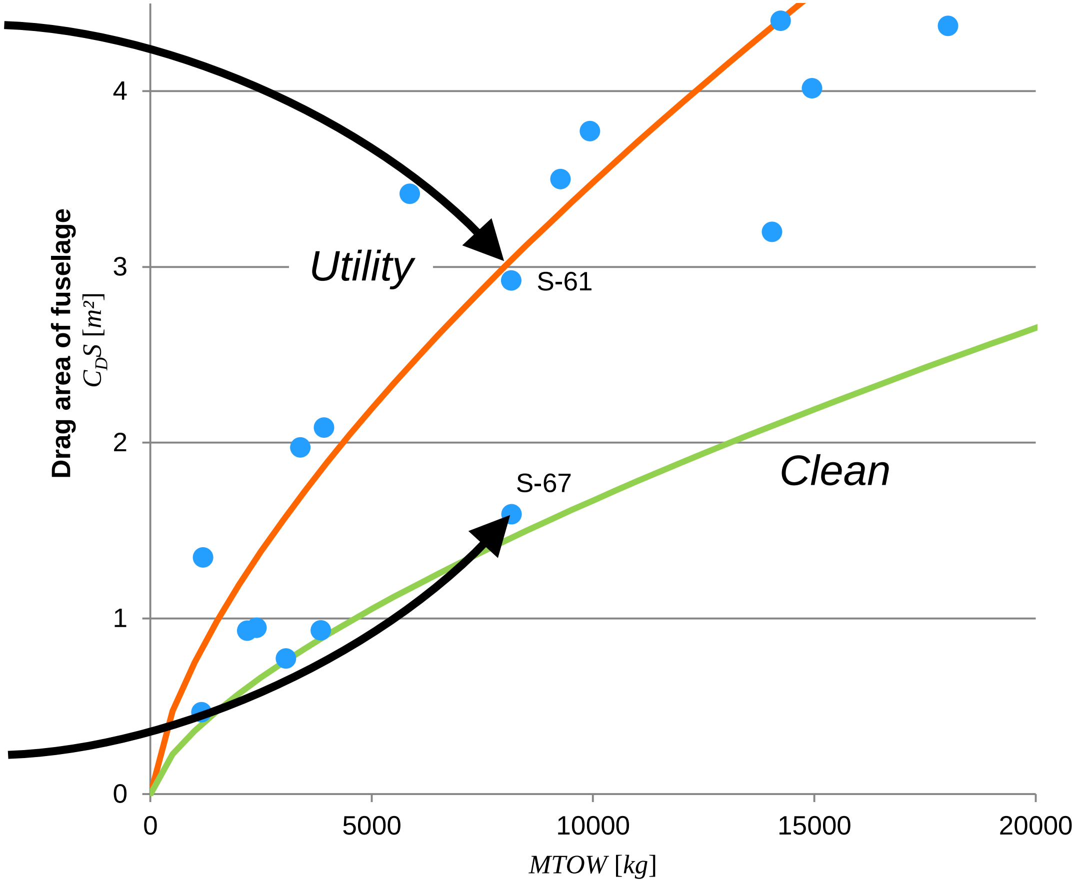




Figures und Data from Leishman: *Principles of Helicopter Aerodynamics*

3.5. Fuselage and fittings

3.5. Fuselage drag – trends



Data from Leishman: *Principles of Helicopter Aerodynamics;* Photos: Steven Bennett; U.S. Army

3. Sizing

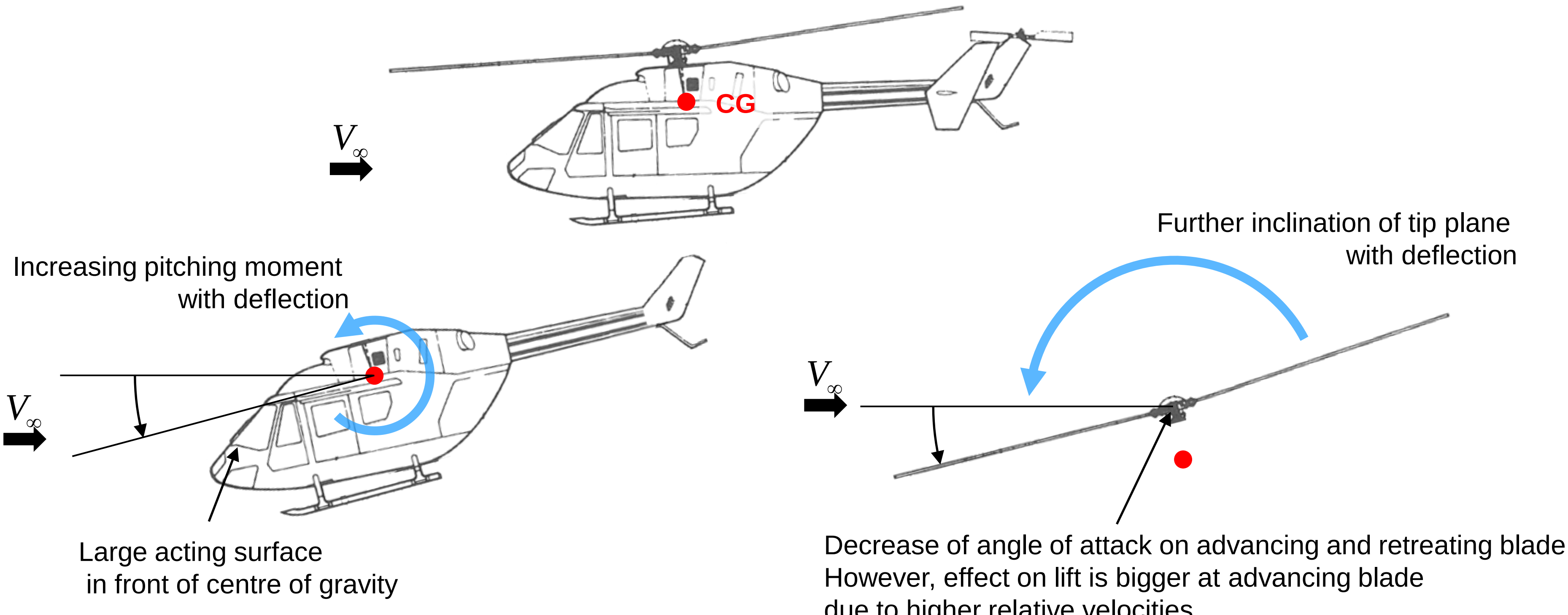
3.5. Fuselage and empennage

- ✓ Fuselage and fittings
- ➔ Empennage

3.5. Horizontal stabiliser

The horizontal stabiliser provides pitch damping and angle of attack stability in forward flight. It causes consequently an improvement of the *handling qualities*.

- Generally the fuselage and rotor without stabiliser act statically destabilising around the pitch axis (no angle of attack stability):



3.5. Horizontal stabiliser

- Helicopters without horizontal stabiliser are possible, but the required flight control actions lead to a high pilot work load.

- A sufficiently large horizontal stabiliser acts against an unwanted change of pitch position and relieves the pilot in forward flight.

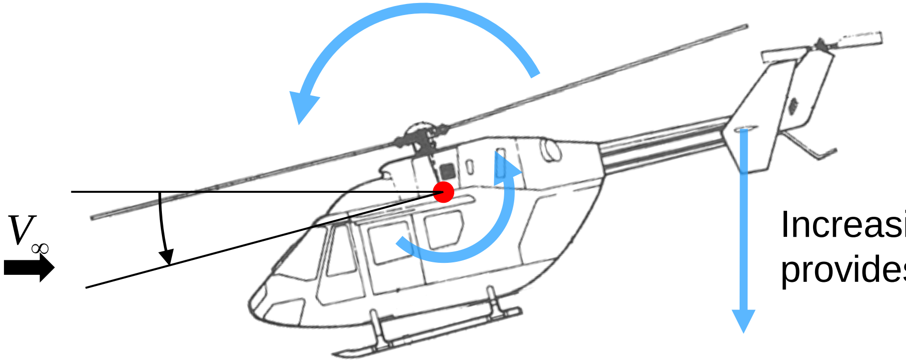
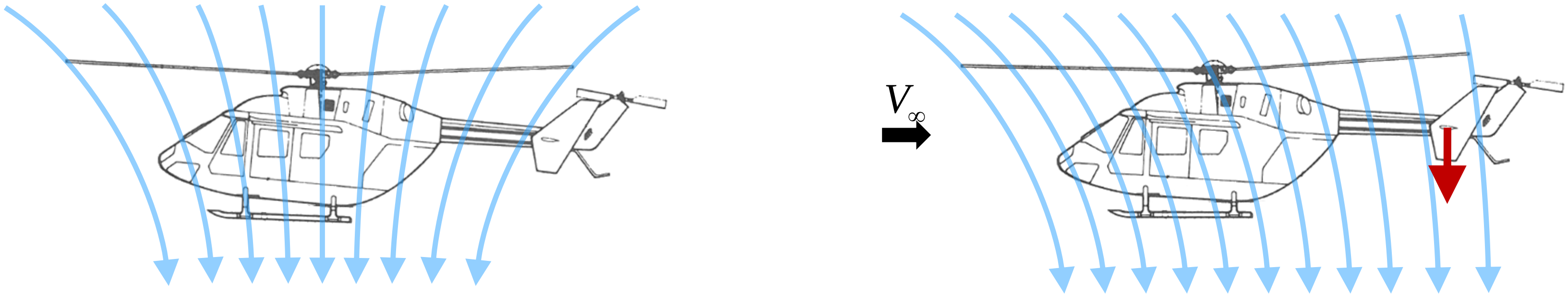


Photo: U.S.Air Force Museum

3.5. Horizontal stabiliser

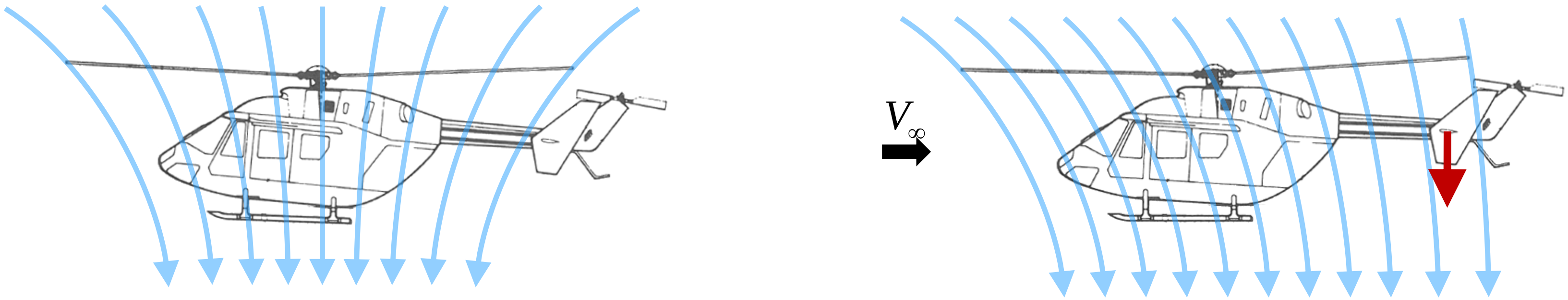
The convenient effect of the horizontal stabiliser in forward flight can be problematic due to interaction with the rotor downwash.



At the transition of hover into forward flight, the horizontal stabiliser enters the rotor downwash. Since the induced velocities at low forward speeds are high, the stabiliser is exposed to high download.

Inversely the stabiliser is suddenly unloaded, if it moves out of the rotor stream at the transition into hover.

3.5. Horizontal stabiliser



The sudden load change is unwanted because of following reasons:

- Work load of pilot:
  Fast reaction required to control pitch position
- Strong load changes promote fatigue of structure

This applies particularly for

- Large horizontal stabiliser surface with fixed angle of attack
- High disc loading, high induced downwash
- Helicopters with low control power

3.5. Horizontal stabiliser

Possibilities of horizontal stabiliser configuration:

At the end of tail boom

- Protects people of tail rotor
- Load changes because of rotor downwash

In the centre of tail boom

- Also at hover in rotor downwash, no load changes
- Large surface necessary due to small distance to centre of gravity
- Download in hover

Photos: Wikipedia, Ahunt; Adrian Pingstone

3.5. Horizontal stabiliser

Possibilities of horizontal stabiliser configuration:

<u>At the top of the fin</u>

- Low influence of rotor stream
- Load application at tip of fin structurally unfavourable

Photos: Adrian Pingstone; MD Helicopters

3.5. Horizontal stabiliser

At some helicopters the angle of attack of the horizontal stabilisers is variable:

- Coupled with the longitudinal control unit to influence the pitch position in level flight (*Bell*-Modelle)
- Unlocking of horizontal stabiliser in slow flight (*floating*), to reduce download due to the rotor stream
- With actively controlled angle dependent on speed of flight, collective pilot input, pitch rate. Download is reduced, pitch attitude influenced and pitch damping selectively increased due to pitch rate feedback.

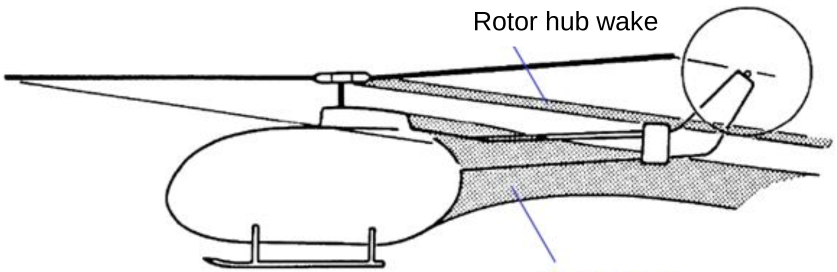
However, the advantages of this solution face additional constructional expenditure, increased costs and higher weight.

Photos:MD Helicopters; U.S. Coast Guard

3.5. Horizontal stabiliser

Due to numerous aerodynamic interactions, a prediction of the effectiveness of horizontal stabilisers is difficult. Changes of the stabiliser in the process of flight testing can be required also with new models.

Eurocopter Tiger Prototypes



Interference
- Fuselage-Empennage
- Fuselage-Tail Rotor
- Rotor-Fuselage
- Rotor-Empennage
- Rotor-Tail Rotor

Figure: DLR, Dr. Keßler; Photos: Flug Revue

3.5. Vertical stabiliser

The fin at the helicopter is in standard configuration not mandatory,

as the tail rotor provides yaw damping and wind vane stability.

Nevertheless most modern helicopters feature one to comply with further requirements:

- Streamlined support of tail rotor
- Complementary directional stability and yaw damping
- Mount for a high installed horizontal stabiliser
- Flight stabilisation in case of tail rotor loss/malfunction
- Unloading of tail rotor in forward flight by incidence angle or camber, primarily in order to reduce fatigue loads on tail rotor. However, this means increased main rotor power due to parasite and induced drag.

Photo: Bell Helicopter Textron

3.5. Vertical stabiliser

Disadvantages of a large vertical stabiliser area:

- Stronger blockage of tail rotor stream leads to thrust loss
- Additional weight at far rear, very unfavourable for centre of gravity position
- Increased parasite drag
- Unfavourable in sideward flight due to wind vane stability

Photo: Eurocopter, Jérome Deulin

3.5. Vertical stabiliser

Instead of an increased vertical stabiliser area, the directional stability can be also achieved in similar way by end plates.

- Mostly outside of tail rotor stream
- Rise of effectiveness of horizontal stabiliser
- Clean flow outside the wake of fuselage and hub

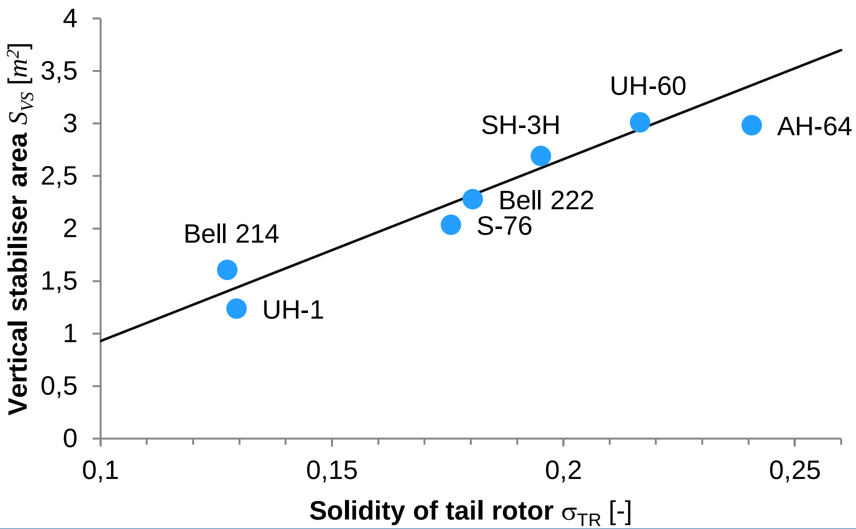
However, the structure gets heavier due to the mounting at the end of the horizontal stabiliser than an area increase of the middle vertical stabiliser.

Photo: Eurocopter, Claes Axtral

3.5. Vertical stabiliser

Design suggestion according to Layton:

- Tail rotor level with main rotor
- Sweep of ¼-chord line approx. 45° , ➔ centre of gravity to front
- Taper to 1/3 of chord length at the root
- Fin area dependent on tail rotor solidity
- Incidence angle/profiling, so that tail rotor is completely unloaded at speed of flight of 80 kts



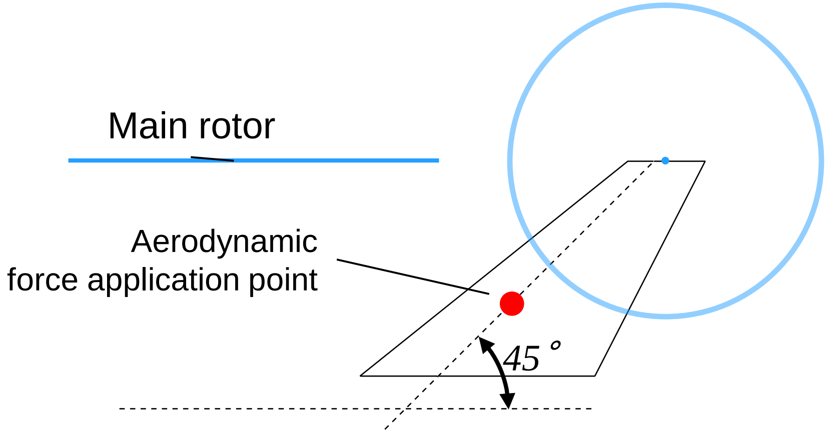


Diagram according to Layton: *Introduction to Helicopter Conceptual Design*

3.5. Vertical stabiliser

Also the vertical stabilisers can be designed controllable.

Example: *MD Helicopters Vertical Stabilizer Control System* (*VSCS*).

Automatic adjustment of fins dependent on

- Forward speed, collective control input:
  torque compensation, autorotation
- *Yaw Stability Augmentation System* (*YSAS*): increase of yaw damping
- Lateral acceleration: support of coordinated turning flight

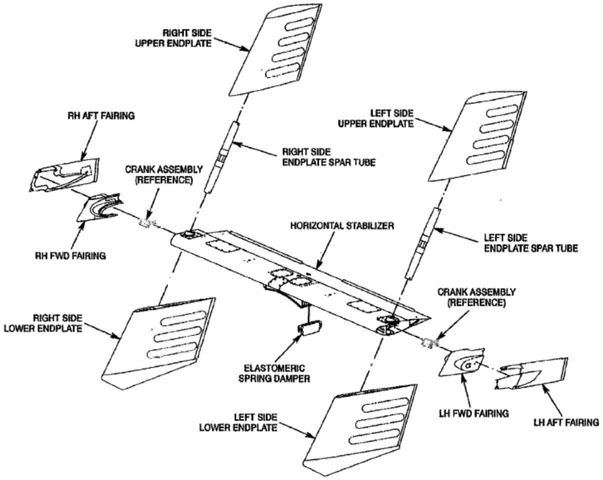


Figure: MD Helicopters; Photo : Jürgen Lehle (albspotter.eu)

3 Sizing

3.5 Refined performance calculation

3 Sizing

3.5. Refined performance calculation

➔ Composition of total power requirement

- Standard atmosphere
- Determination of required engine power
- Selection of engine and gearbox
- Refined mass estimation

3.5. Refined performance calculation

Composition of total power requirement P

$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

Consisting of

- P_i Induced power
- P_0 Overcoming of profile drag of rotor blades (*profile power*)
- P_p Propulsion power against air drag of fuselage and accessories (*parasite power*)
- P_C Climb power
- P_{TR} Tail rotor power
- P_a Power to drive the accessories
- P_{tl} Transmission losses

3.5. Refined performance calculation

Composition of total power requirement P

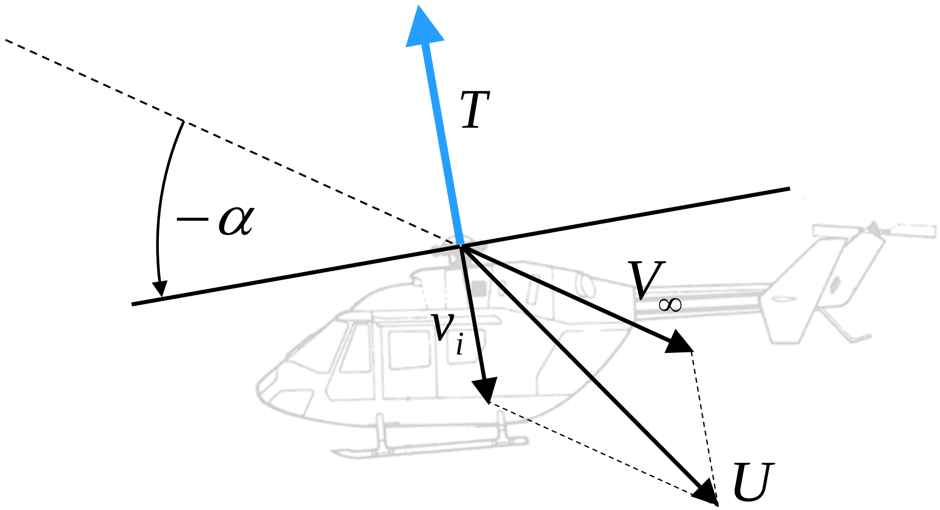
$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

<u>Induced power P_i</u>

Generally $P_i = \kappa \cdot T \cdot v_i$ with:

- Rotor thrust T
- Induced velocities v_i
- Correction factor κ for non-ideal thrust generation

$$\kappa \approx \underset{\text{Hover}}{\underset{\uparrow}{1{,}15}} \ldots \underset{\text{Highspeed flight}}{\underset{\uparrow}{1{,}4}}$$



3.5. Refined performance calculation

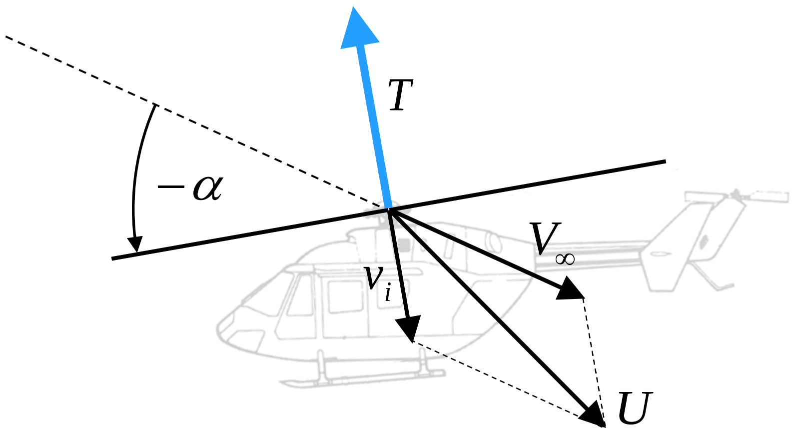
Composition of total power requirement P

$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

<u>Induced power</u> $P_i = \kappa \cdot T \cdot v_i$

Determination of induced velocity v_i for translatory motion (not valid for low sink rate in axial flight)

$$\boxed{v_i} = \frac{T}{2\rho AU} = \frac{T}{2\rho A\sqrt{V_\infty^2 - 2V_\infty \boxed{v_i} \sin\alpha + \boxed{v_i}^2}}$$



The solution for v_i in flight mechanics codes occurs typically iterative

3.5. Refined performance calculation

Induced power $P_i = \kappa \cdot T \cdot v_i$

The calculation of v_i simplifies for particular cases

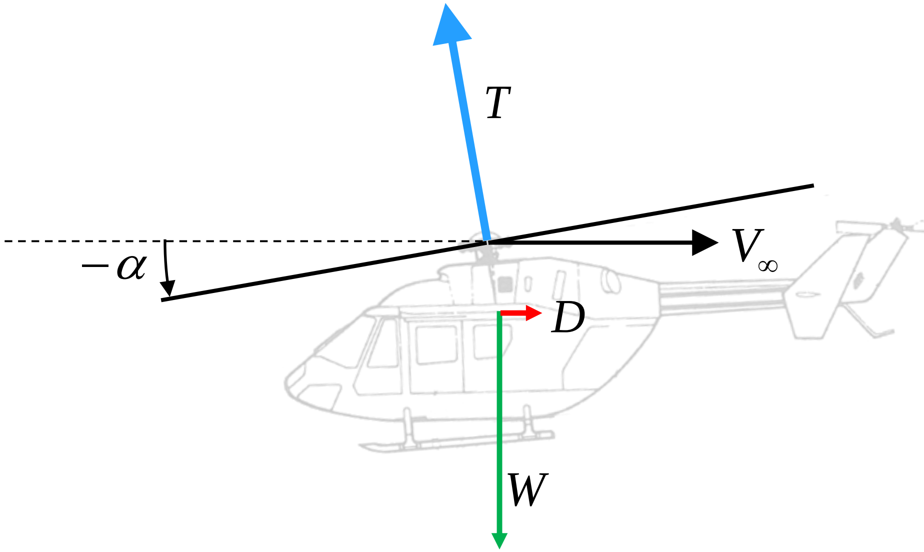
- Hover $v_h = \sqrt{\dfrac{T}{2\rho A}}$

- Approximation for high velocities

  $v_i = \dfrac{T}{2\rho A V_\infty}$ for $\mu \gtrsim 0{,}1$

- Solution for $\alpha = 0$
  e.g. level flight with $W >> D$

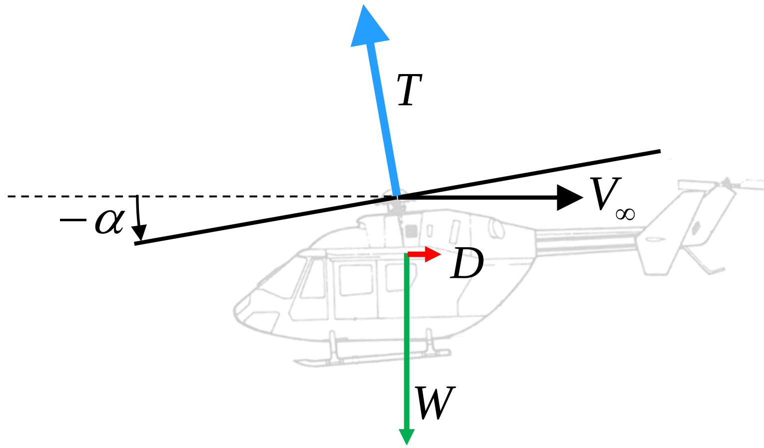
$$v_i = \sqrt{-\frac{V_\infty^2}{2} + \sqrt{v_h^4 + \frac{V_\infty^4}{4}}}$$



3.5. Refined performance calculation

Ideal induced Power

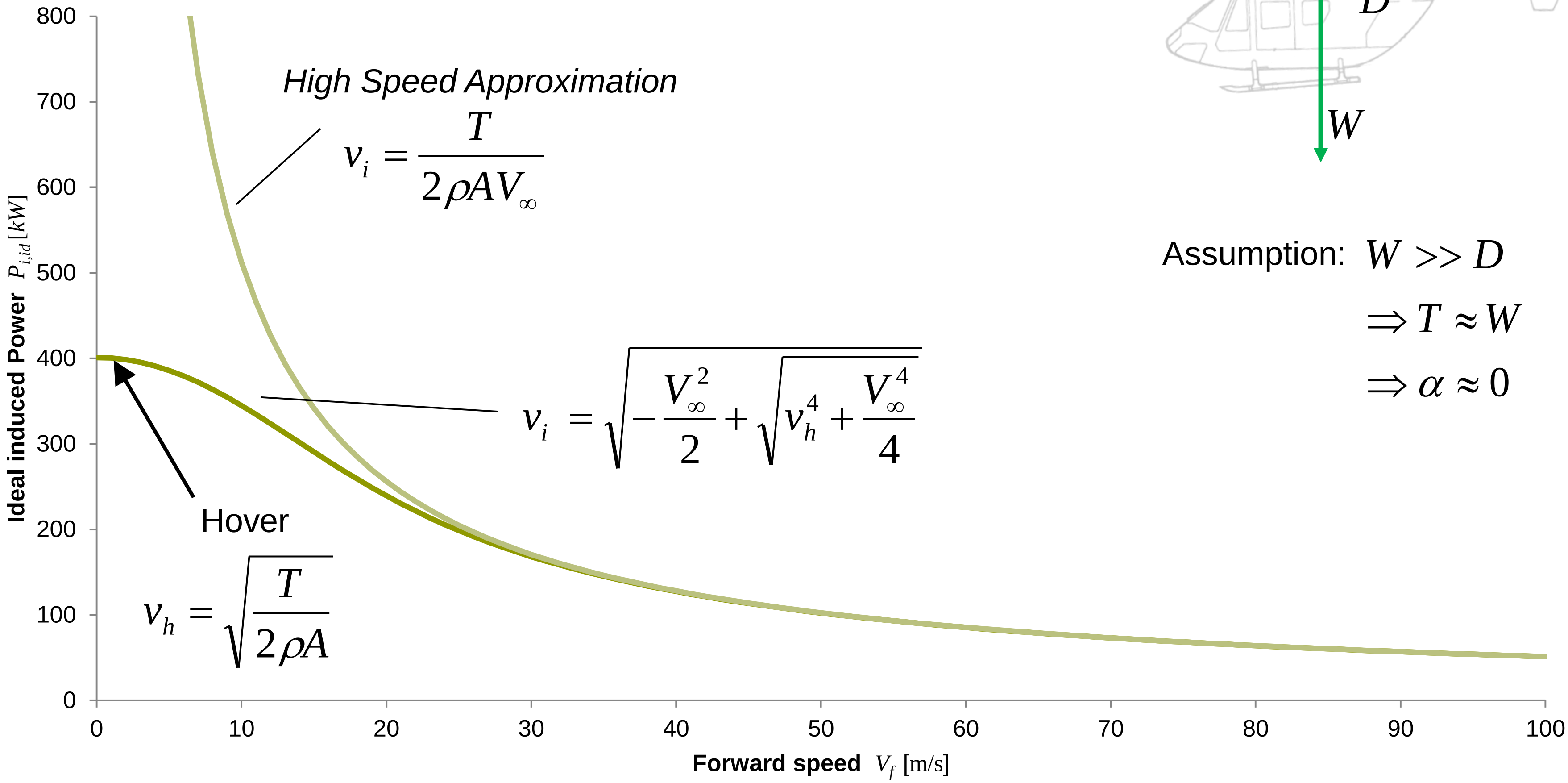
$$P_{i_{id}} = T \cdot v_i$$



Assumption: $W >> D$

$$\Rightarrow T \approx W$$

$$\Rightarrow \alpha \approx 0$$



3.5. Refined performance calculation

Composition of total power requirement P

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Consisting of

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- P_{TR} Tail rotor power
- P_a Power to drive the accessories
- P_{tl} Transmission losses

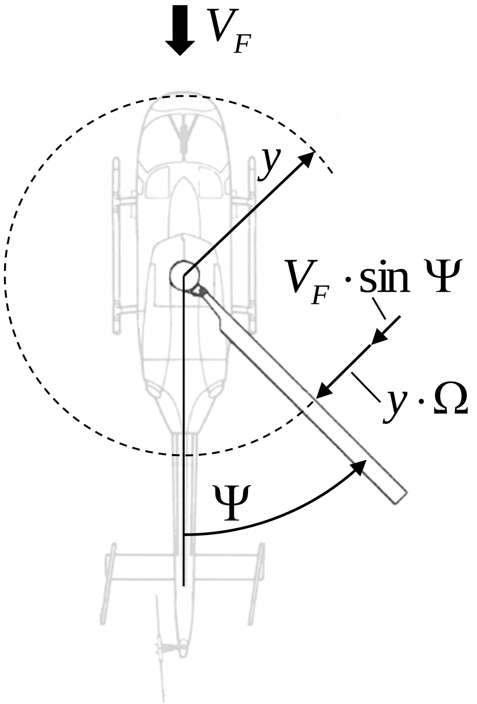
3.5. Refined performance calculation

Composition of total power requirement P

$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

Profile power P_0

Application to forward flight involves the power, which is required for rotary and forward motion of the profile cross sections.



Assumptions for the first approximation:

- Constant profile drag coefficient C_{d0}
- Constant chord length c from axis to blade tip
- Disregarding the radial blade flow

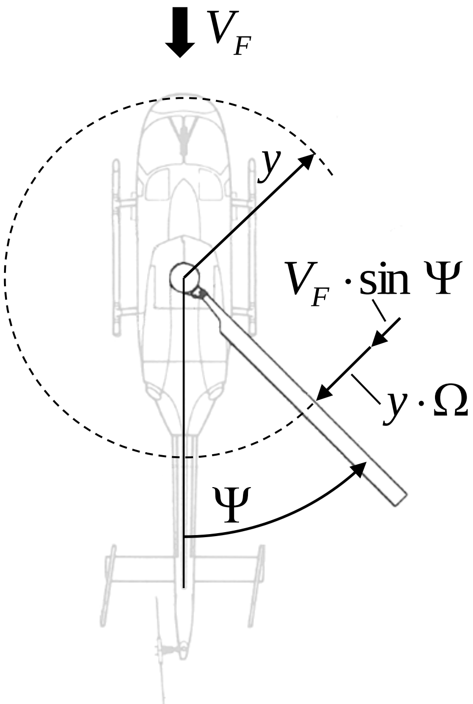
3.5. Refined performance calculation

Profile power P_0

By integration over the blade revolution and the radius we obtain

$$P_0 = \frac{N_b}{2\pi} \cdot \int_0^{2\pi}\int_0^{R} \frac{1}{2}\rho \cdot c \cdot C_{d0}\left(y \cdot \Omega + V_F \cdot \sin\Psi\right)^3 dy\, d\Psi$$

$$= \frac{1}{8}\rho \cdot A \cdot V_{TIP}^3 \cdot \sigma \cdot C_{d0}\left(1 + 3\mu^2\right)$$



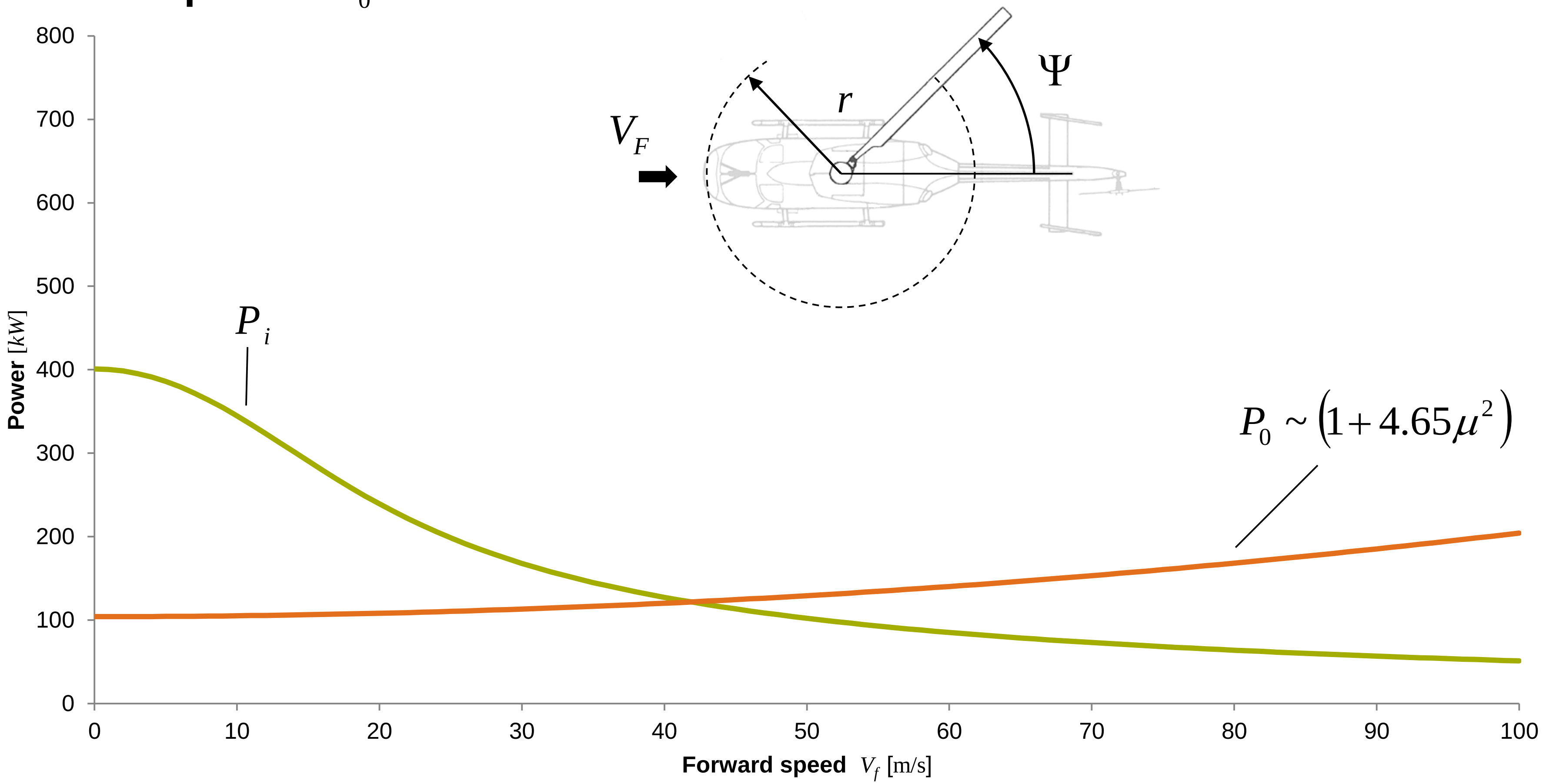
In approximation of the effects of reverse flow, radial velocity component and rise of C_d due to angular flow, the influence of the advance ratio μ increases:

$$P_0 = \frac{1}{8}\rho \cdot A \cdot V_{TIP}^3 \cdot \sigma \cdot C_{d0}\left(1 + 4.65\mu^2\right)$$

A further increase of the profile power is generated at high velocities due to compressibility effects. Corrections dependent on the mach number can be included, but are disregarded here.

3.5. Refined performance calculation

Profile power P_0



3.5. Refined performance calculation

Composition of total power requirement P

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P_{TR}	Tail rotor power
P_a	Power to drive the accessories
P_{tl}	Transmission losses

3.5. Refined performance calculation

Composition of total power requirement P

$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

<u>Parasite power P_p</u>

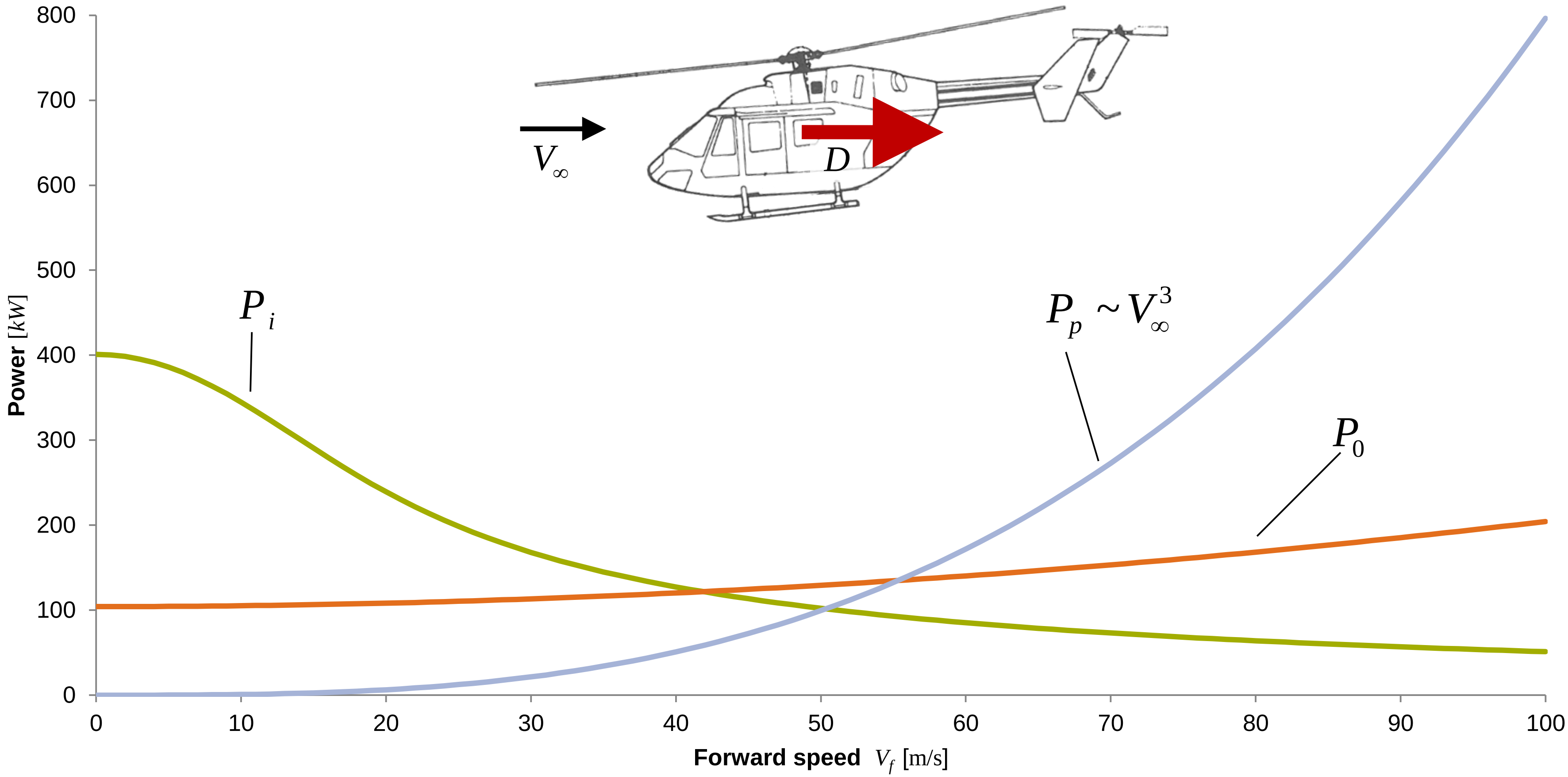
$$P_P = V_\infty \cdot D$$

$$P_P = V_\infty \cdot q \cdot C_D S$$
$$= \frac{1}{2} \rho \cdot V_\infty^3 \cdot C_D S$$

Drag area $C_D S \approx \text{const.}$
disregarding the
- Variation of flow direction
- Reynolds number
- Mach number

3.5. Refined performance calculation

Parasite power P_p



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P_{tl}	Transmission losses

3.5. Refined performance calculation

Composition of total power requirement P

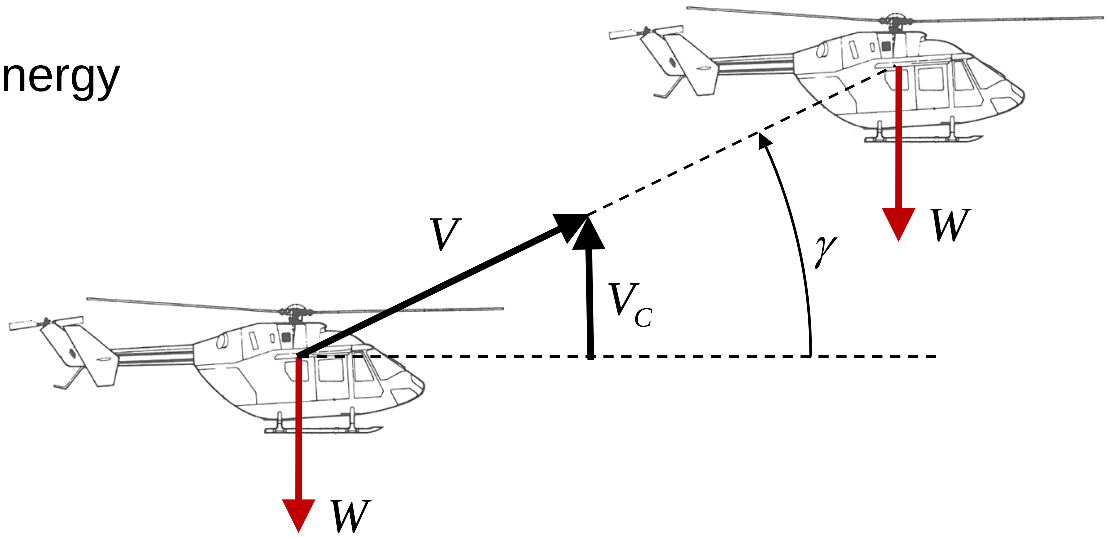
$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

<u>Climb power P_C</u>

Temporal change of potential energy

$$P_C = W \cdot V_C$$

Large angles of climb γ in steady flight lead to low fuselage attack angles.



Therefore a consideration of a vertical drag can be required for the calculation of the parasite power P_p (neglected here)

3.5. Refined performance calculation

Composition of total power requirement P

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3.5. Refined performance calculation

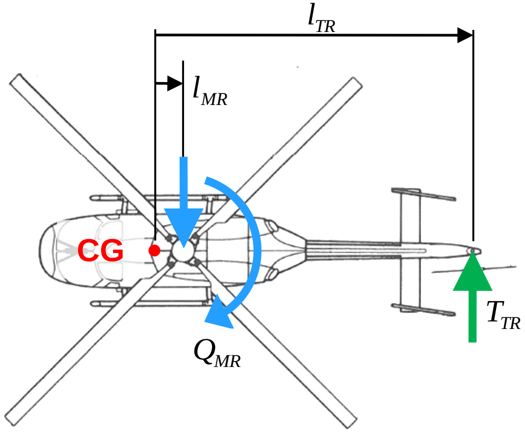
Composition of total power requirement P

$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

<u>Tail rotor power P_{TR}</u>

The power percentages discussed so far are implemented by the main rotor.

They form the main rotor power P_{MR}, which is transmitted via the rotary motion of the rotor shaft:



$$P_{MR} = P_i + P_0 + P_p + P_C$$

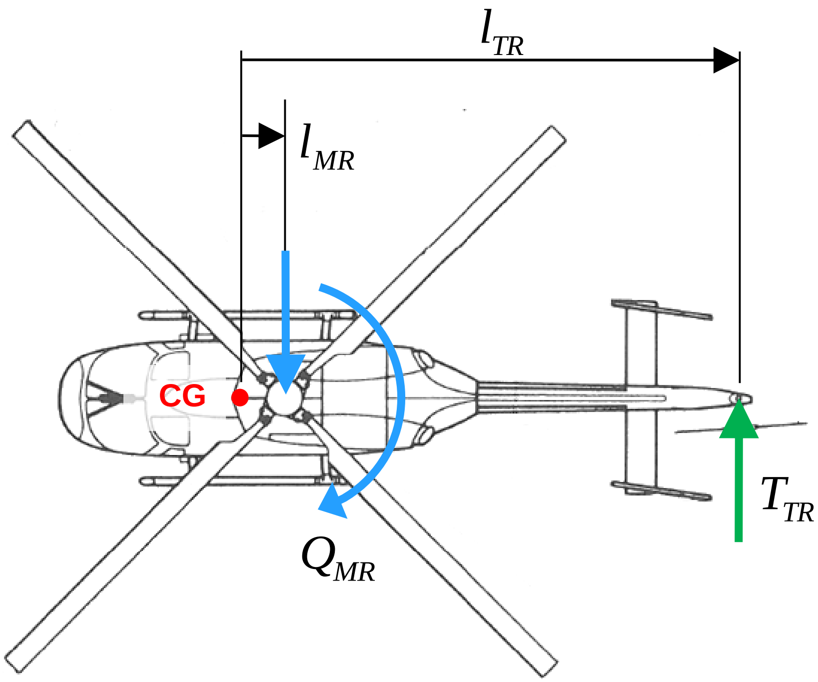
The drive torque $Q_{MR} = \dfrac{P_{MR}}{\Omega}$ needs to be compensated in steady flight by the tail rotor or the vertical stabiliser.

3.5. Refined performance calculation

Tail rotor power P_{TR}

Assuming that the main rotor torque is only compensated by the tail rotor, for the required tail rotor thrust applies

$$T_{TR} = \frac{Q_{MR}}{l_{TR} - l_{MR}} = \frac{P_{MR}}{\Omega \cdot (l_{TR} - l_{MR})} = \frac{P_i + P_0 + P_C + P_P}{\Omega \cdot (l_{TR} - l_{MR})}$$

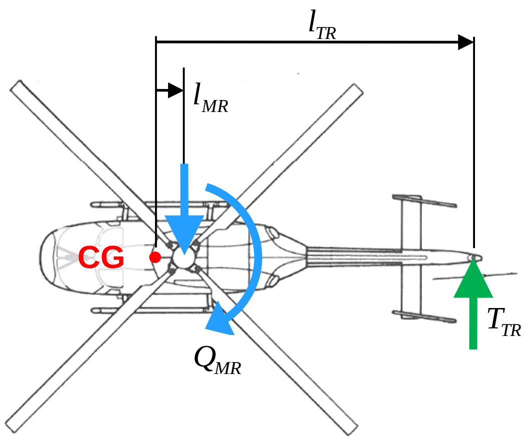


The required main rotor thrust increases due to the occurrent lateral force component. However, this effect is neglectable in view of the large helicopter weight W.

3.5. Refined performance calculation

Tail rotor power P_{TR}

Analogue to the main rotor, the induced power and the profile power of the tail rotor can be determined with the calculated tail rotor thrust:



$$P_{i_{TR}} = \kappa_{TR} \cdot T_{TR} \cdot v_{i_{TR}}$$

$$P_{0_{TR}} = \frac{1}{8} \rho \cdot A_{TR} \cdot V_{TIP_{TR}}^{3} \cdot \sigma_{TR} \cdot C_{d0_{TR}} \left(1 + 4.65 \mu_{TR}{}^{2}\right)$$

The outcome of this is the tail rotor power

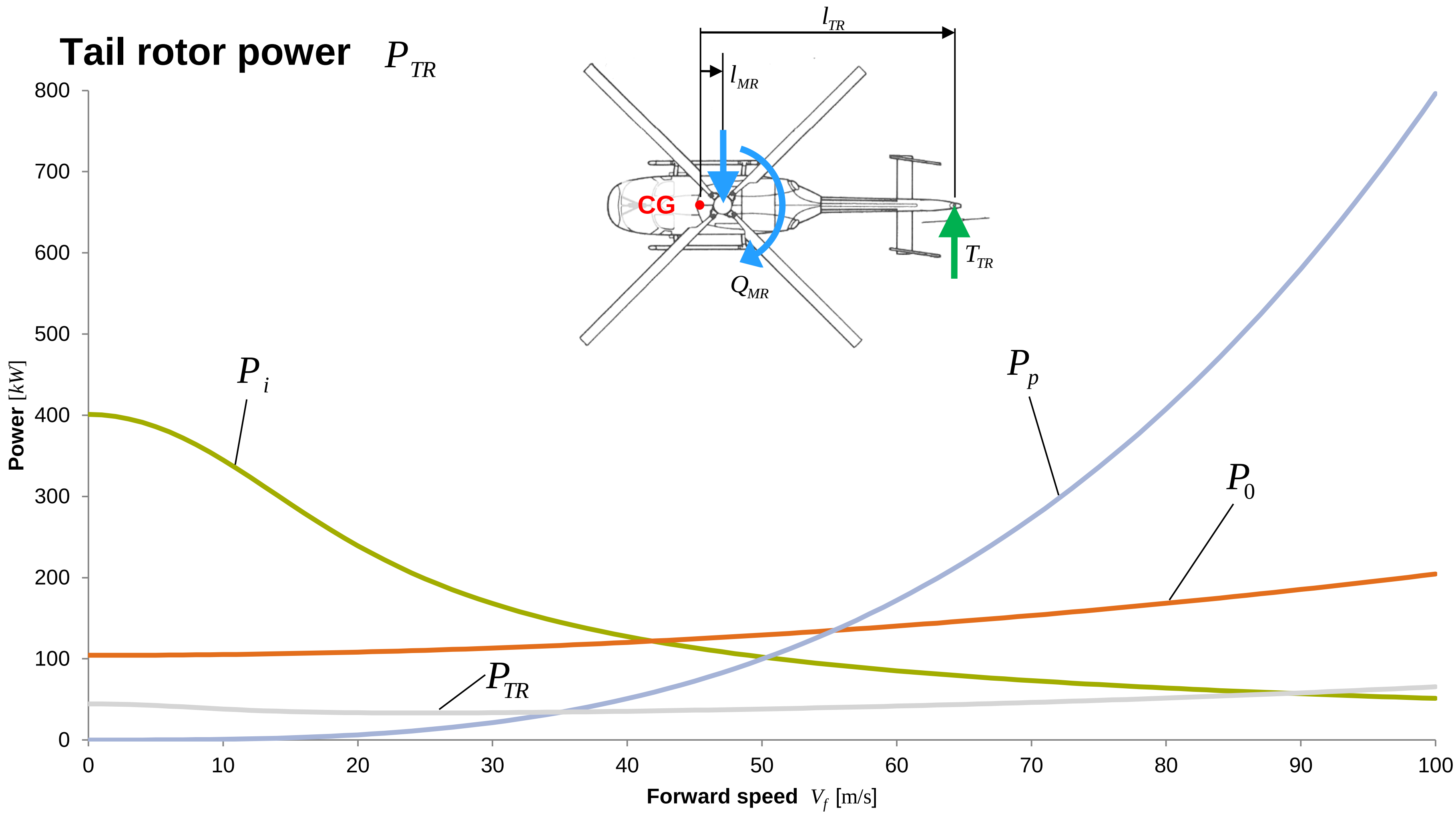
$$P_{TR} = P_{i_{TR}} + P_{0_{TR}}$$

As the power contribution of the tail rotor is comparatively low, in first approximation it can be also described as percentage of the main rotor thrust, e.g.

$$P_{TR} = 0{,}05 \cdot P_{MR} = 0{,}05 \cdot \left(P_i + P_0 + P_p + P_C\right)$$

3.5. Refined performance calculation

Tail rotor power P_{TR}



3.5. Refined performance calculation

Composition of total power requirement P

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P_{TR}	Tail rotor power
P_a	Power to drive the accessories
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3.5. Refined performance calculation

Composition of total power requirement P

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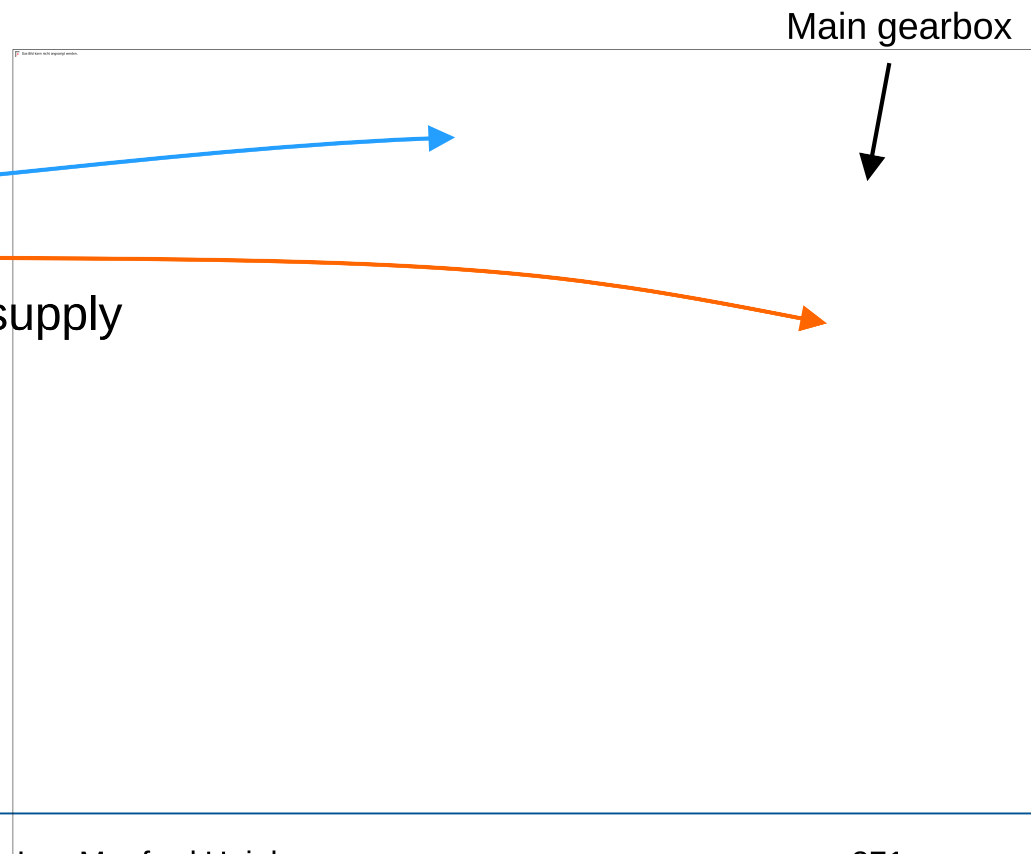

Power to drive the accessories P_a:

For example:

- Oil cooler
- Hydraulic pump
- Generators for on-board power supply

P_a can be initially assumed to be constant by helicopter type:

	$MTOW\,[kg]$	$P_a\,[kW]$
Light Utility	< 1400	11
Medium Utility		48
Heavy Utility	> 11000	92
Light Attack	< 1400	62
Medium Attack		165
Heavy Transport	> 11000	118

Data from Layton: *Introduction to Helicopter Conceptual Design*; Figure from Bittner: *Flugmechanik der Hubschrauber*

3.5. Refined performance calculation

Composition of total power requirement P

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3.5. Refined performance calculation

Composition of total power requirement P

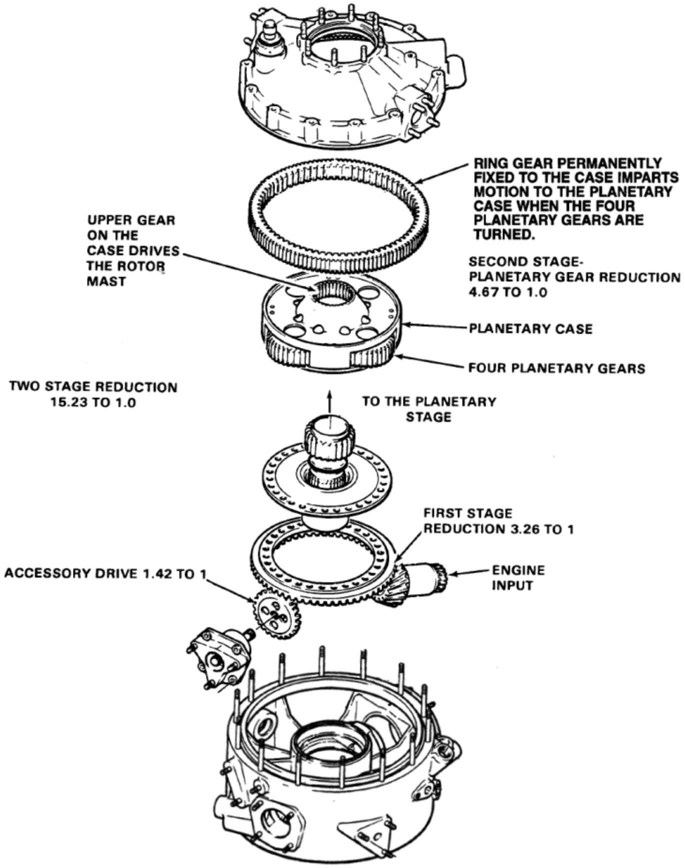
$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

<u>Transmission losses P_{tl}</u>

The efficiency η_t of the drive system can be estimated on the basis of the used gear stages.
Neglecting the tail rotor drive, the additionally required engine power adds up to

$$P_{tl} = \left(\frac{1}{\eta_t} - 1\right) \cdot P_{MR} = \left(\frac{1}{\eta_t} - 1\right) \cdot \left(P_i + P_0 + P_p + P_C\right)$$

Assumption here: $\eta_t \approx 0{,}98$

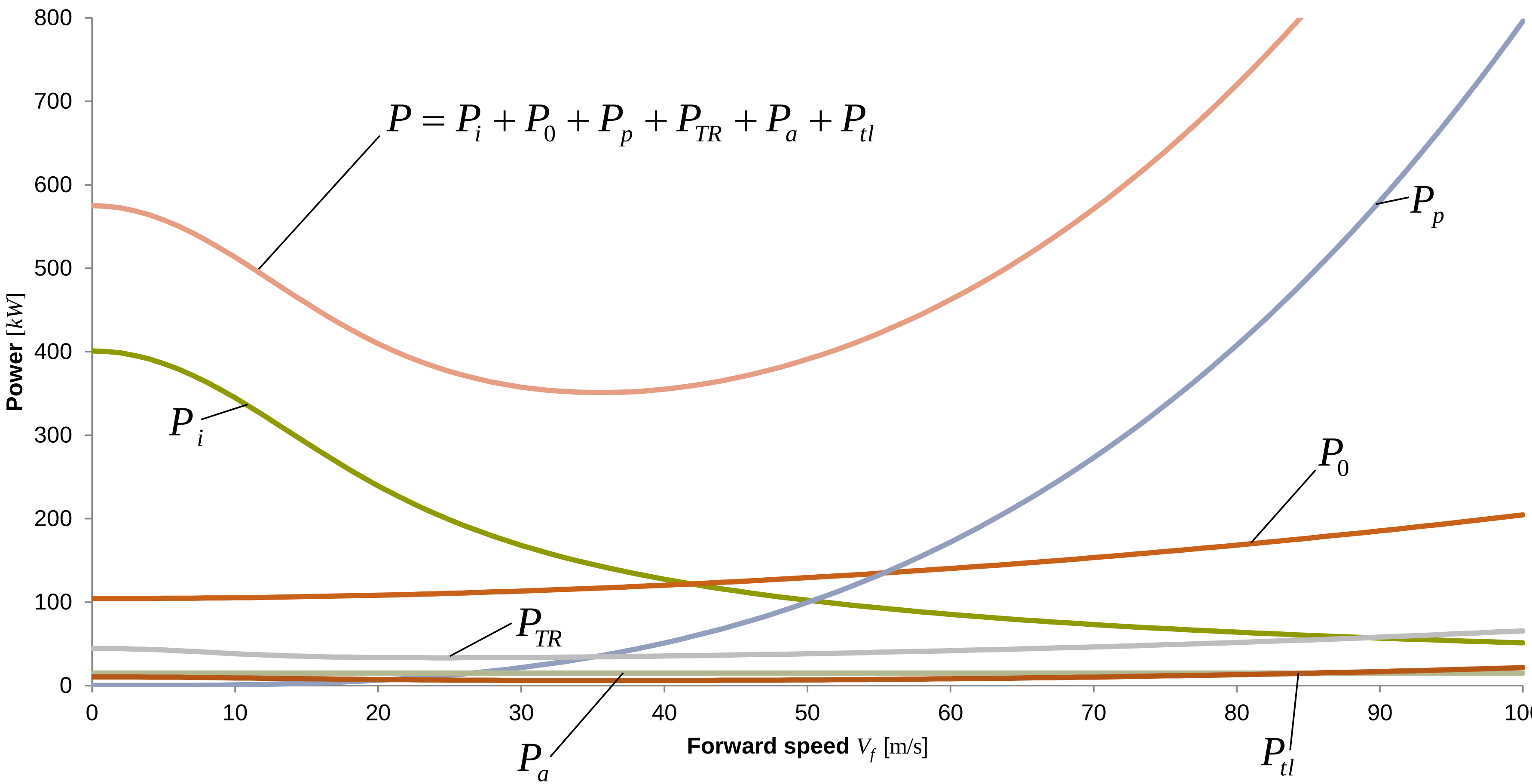


Main rotor gear box Bell 206BIII

Figure: Bell Helicopter Textron

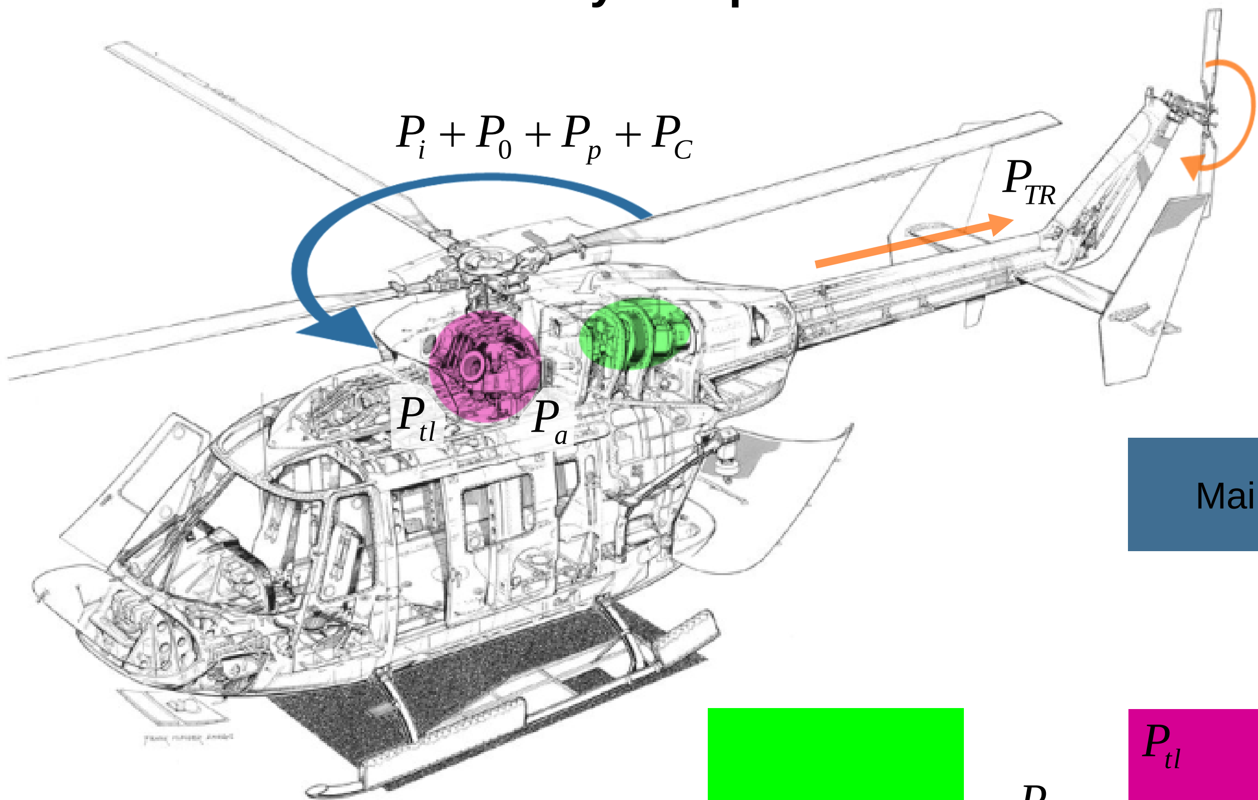
3.5. Refined performance calculation

Total power in level flight



3.5. Refined performance calculation

Power distribution by components



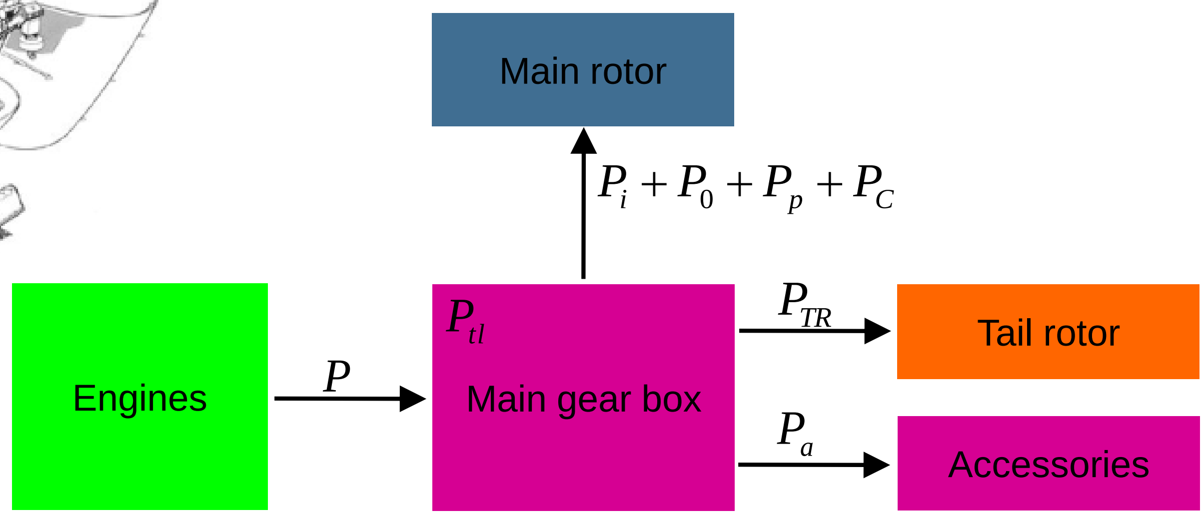


Figure : *Flight International*

3 Sizing

3.5. Refined performance calculation

- ✓ Composition of total power requirement
- ➔ Standard atmosphere
- Determination of required engine power
- Selection of engine and gearbox
- Refined mass estimation

3.5. Refined performance calculation

Standard atmosphere

Different flight conditions with the required flight performances are now calculated to determine the required installed engine power.

In this respect, besides the mass of the helicopter also atmospheric conditions are important. Especially low air densities, as they appear at flights in high altitudes and high temperatures (*hot&high),* can be critical due to the influence on the induced power.

Usually the flight performances are therefore postulated at a defined altitude and temperature in standard atmosphere (*International Standard Atmosphere, ISA*).

Figure : *Wikipedia, Thuringius*

3.5. Refined performance calculation

Standard atmosphere (*International Standard Atmosphere, ISA*)

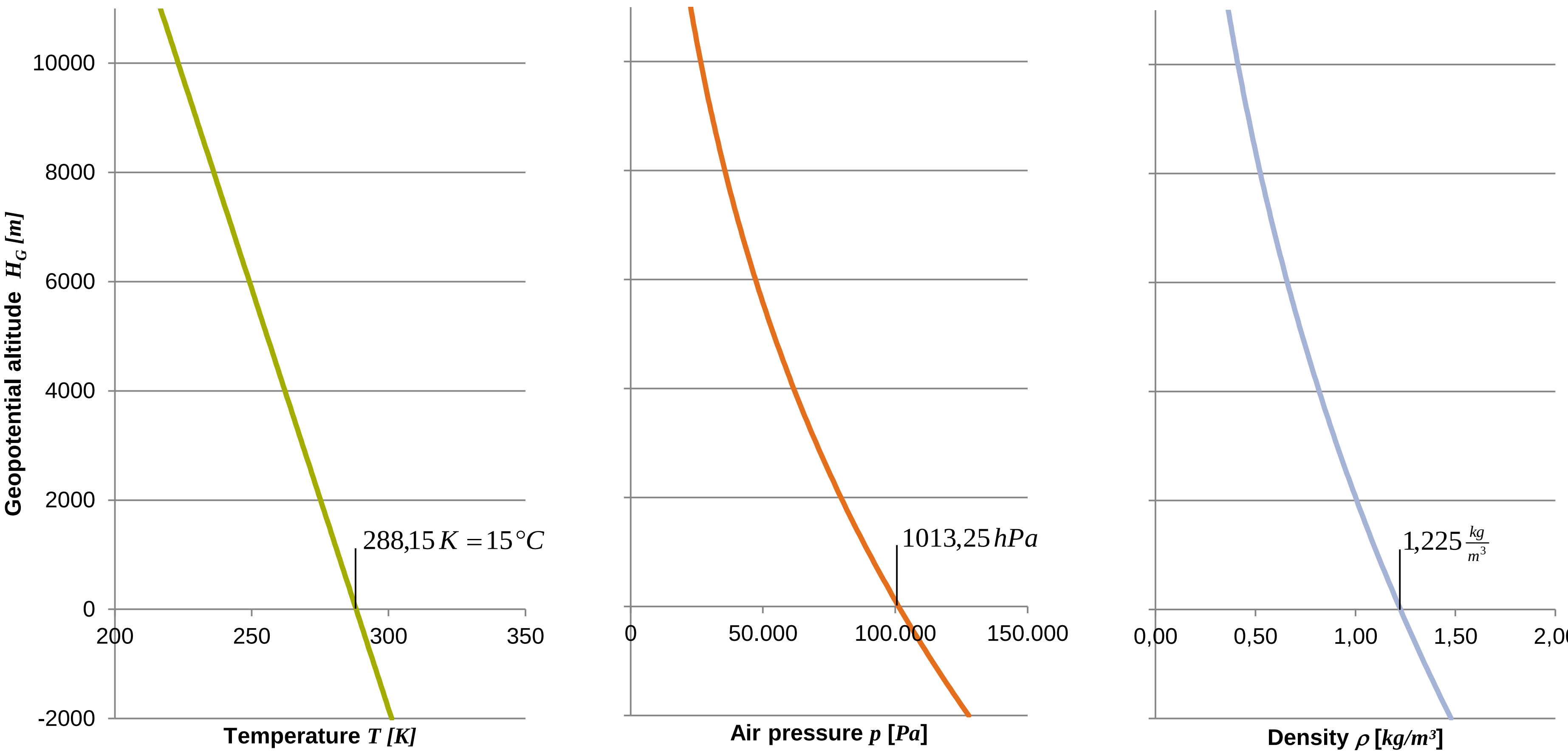
Geopotential altitude H_G above mean sea level (*MSL*)
from -$2000\ m$ to $11000\ m$ (Troposphere)

- Pressure $p = p_S \cdot \left(1 - \frac{n_{Tr}-1}{n_{Tr}} \frac{g_S}{R_s \cdot T_S} \cdot H_G\right)^{\frac{n_{Tr}}{n_{Tr}-1}}$
- Temperature $T = T_S + \gamma_{Tr} \cdot H_G$
- Density $\rho = \rho_S \cdot \left(1 - \frac{n_{Tr}-1}{n_{Tr}} \frac{g_S}{R_s \cdot T_S} \cdot H_G\right)^{\frac{1}{n_{Tr}-1}}$

$$p_S = 101325 \tfrac{N}{m^2}$$
$$\rho_S = 1{,}225 \tfrac{kg}{m^3}$$
$$T_S = 288{,}15\,K$$
$$g_S = 9{,}80665 \tfrac{m}{s^2}$$
$$n_{Tr} = 1{,}235$$
$$R_s = 287{,}05 \tfrac{J}{kg \cdot K}$$
$$\gamma_{Tr} = -0{,}0065 \tfrac{K}{m}$$

3.5. Refined performance calculation

Standard atmosphere (*International Standard Atmosphere*, *ISA*)



3.5. Refined performance calculation

Standard atmosphere (*International Standard Atmosphere, ISA*)

At an air temperture deviating from the standard atmosphere with ΔT_{ISA} , the density arises from the ideal gas law.

- Pressure $p = p_S \cdot \left(1 - \frac{n_{Tr} - 1}{n_{Tr}} \frac{g_S}{R_s \cdot T_S} \cdot H_G\right)^{\frac{n_{Tr}}{n_{Tr}-1}}$
- Temperature $T = T_S + \gamma_{Tr} \cdot H_G + \Delta T_{ISA}$
- Density $\rho = \frac{p}{R_s T}$

$$p_S = 101325 \, \tfrac{N}{m^2}$$
$$\rho_S = 1{,}225 \, \tfrac{kg}{m^3}$$
$$T_S = 288{,}15 \, K$$
$$g_S = 9{,}80665 \, \tfrac{m}{s^2}$$
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3 Sizing

3.5. Refined performance calculation

- ✓ Composition of total power requirement
- ✓ Standard atmosphere
- ➔ Determination of required engine power
- Selection of engine and gearbox
- Refined mass estimation

3.5. Refined performance calculation

Determination of the required engine power

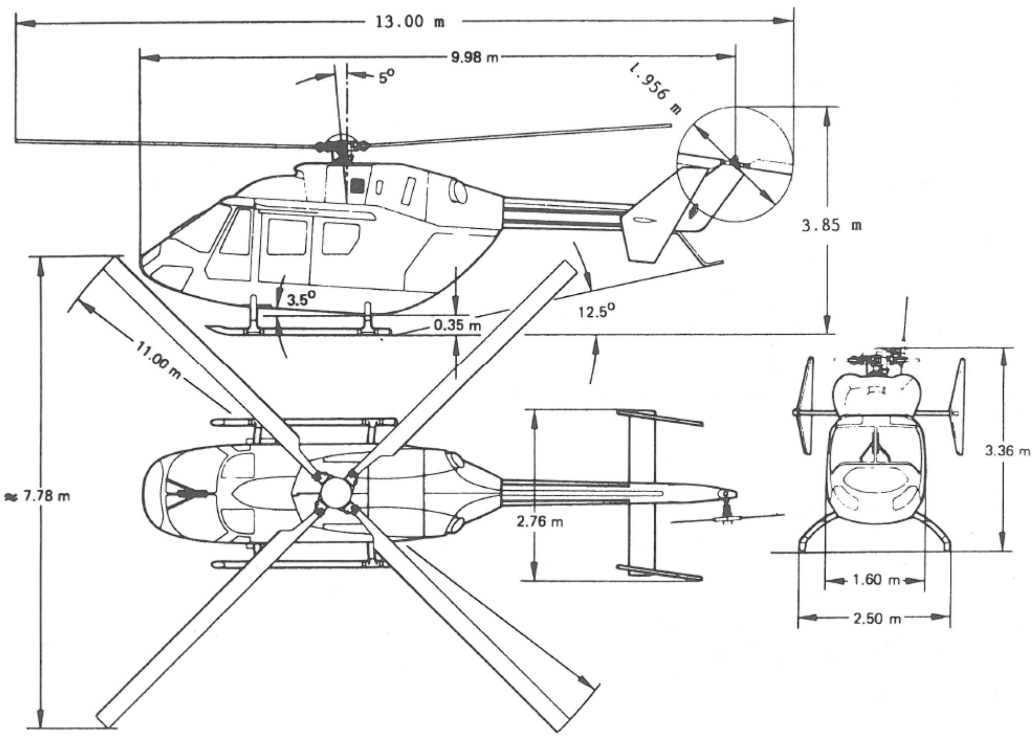
For the chosen configuration, the power requirement is now estimated for defined flight conditions in order to be able to make a selection for engine and drive system on the basis of the required power.

Initially the following flight cases are analysed:

- Hover ceiling out of ground effect
- Hover ceiling in ground effect
- Maximum forward speed in level flight
- Required climb power

3.5. Refined performance calculation

Example: MBB/Kawasaki BK117



Tip speeds

$V_{TIP} = 221 \frac{m}{s}$ $\qquad$ $V_{TIP_{TR}} = 221 \frac{m}{s}$

Solidities

$\sigma = 0.074$ $\qquad$ $\sigma_{TR} = 0.131$

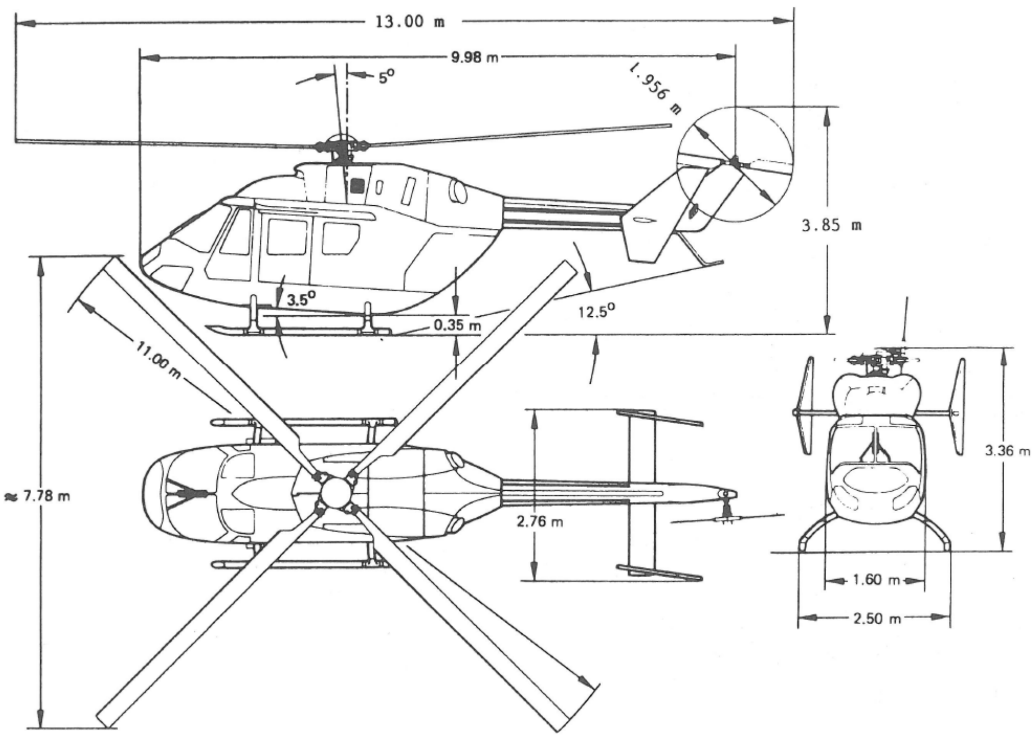
Maximum take off weight

$MTOW = 3200\, kg$

Data from Prouty: *Helicopter Performance, Stability and Control*; Leishman: *Principles of Helicopter Aerodynamics*

3.5. Refined performance calculation

Example: MBB/Kawasaki BK117



Assumptions:

Empirical correction factor

$\kappa = 1{,}15$ $\kappa_{TR} = 1{,}3$

Airfoil drag coefficient

$C_{d0} = 0{,}011$ $C_{d0_{TR}} = 0{,}011$

Total drag area

$C_D S = 1{,}25\,m^2$

Fuselage download factor in hover

$k_{DL} = 4\%$

Efficiency of main gear box

$\eta_t = 98\%$

Power for accessories

$P_a = 15\,kW$

Figure from Prouty: *Helicopter Performance, Stability and Control*

3.5. Refined performance calculation

Hover power out of ground effect (OGE)

$$P = P_i + P_0 + P_{TR} + P_a + P_{tl}$$

<u>Example requirements:</u>

- HOGE in 1000 m at ISA + 25K
- Temperature $\Delta T_{ISA} = +25\,\mathrm{K}$ ➔ $T = 33{,}5\,°\mathrm{C}$

➔ Air density $\rho = 1{,}02\,\frac{kg}{m^3}$

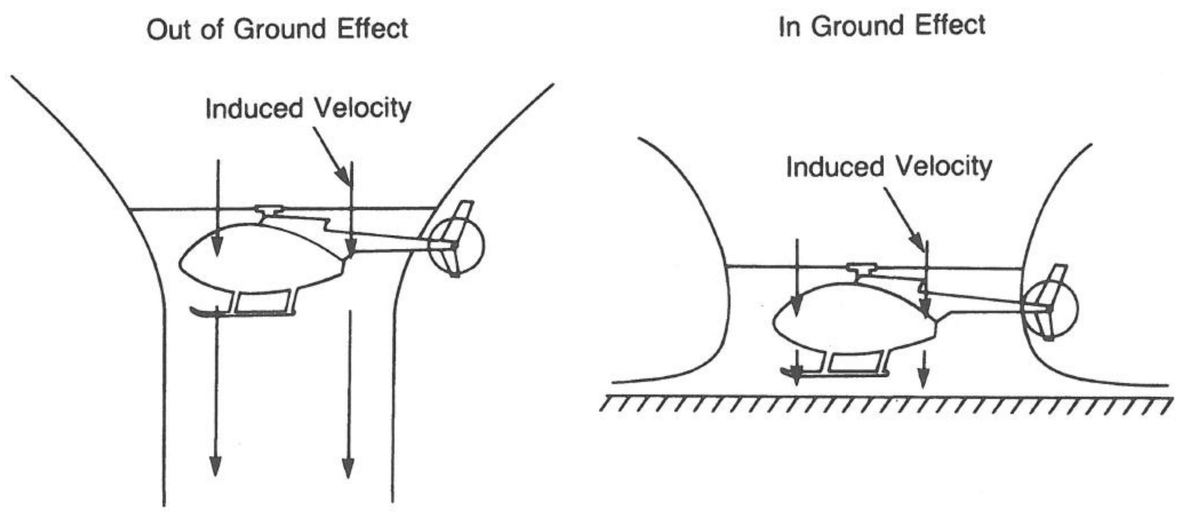
- Mass M=3200kg

➔ Induced power	$P_i = 488\,\mathrm{kW}$
➔ Profile power	$P_0 = 107\,\mathrm{kW}$
➔ Tail rotor power	$P_{TR} = 63\,\mathrm{kW}$
➔ Accessories	$P_a = 15\,\mathrm{kW}$
➔ Transmission loss	$P_{tl} = 12\,\mathrm{kW}$
➔ Total power	$P = 685\,\mathrm{kW}$

Photo: Wikipedia, *Arcturus*

3.5. Refined performance calculation

Required hover power in ground effect



The downwash of the rotor gets interfered when hovering at low altitude above the ground. The reduction of the induced flow leads to a lower induced power consumption.

Figure from Prouty: *Helicopter Performance, Stability and Control*

3.5. Refined performance calculation

Required hover power in ground effect

The power consumption in ground effect (*IGE*) indicates the power where the helicopter is just able to take off vertically.

That practically means that a helicopter with not enough power for hovering out of ground effect (*OGE*) is able to hover in ground effect.

When accelerated at ground level and so the induced power consumption gets further reduced, it can be proceeded with the climb.

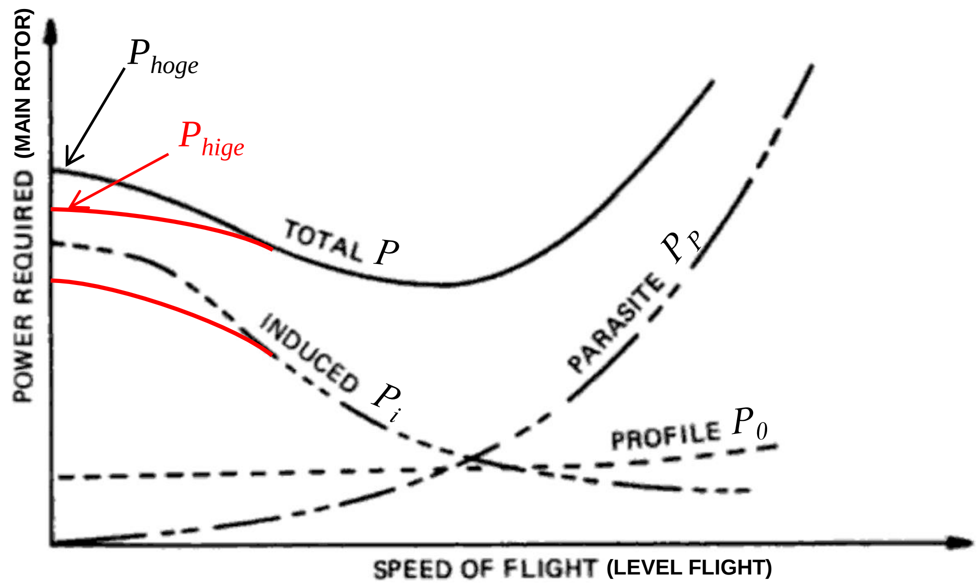


Figure from Stepniewski: *Rotary-Wing Aerodynamics*

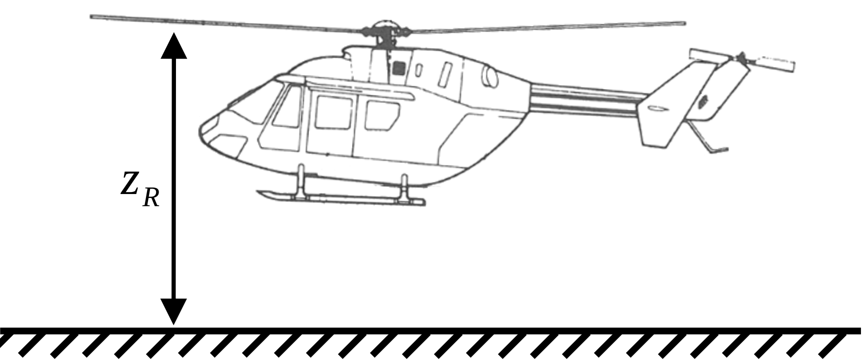
3.5. Refined performance calculation

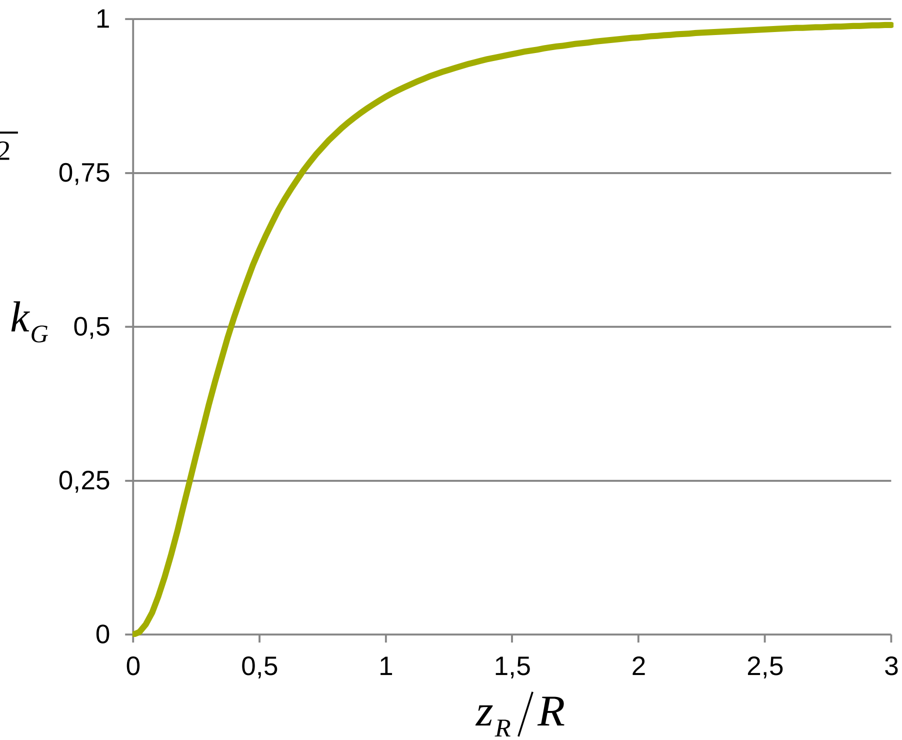
Required hover power in ground effect

The reduction of the induced power P_i in ground effect can be estimated via empirical relations:

In hover according to Hayden:

$$P_{i_{ige}} = k_G \cdot P_{i_{oge}} \qquad k_G = \frac{1}{0{,}9926 + 0{,}0379 \cdot \left(2\frac{R}{z_R}\right)^2}$$





3.5. Refined performance calculation

Hover power in ground effect (IGE)

$$P = P_i + P_0 + P_{TR} + P_a + P_{tl}$$

<u>Example requirement:</u>

Departure at *El Alto International*, Bolivia

- Take off elevation (*HIGE*) 4075 m
- Outside temperature in ISA: $T = -11{,}5\,°\mathrm{C}$
- Mass M=3200kg

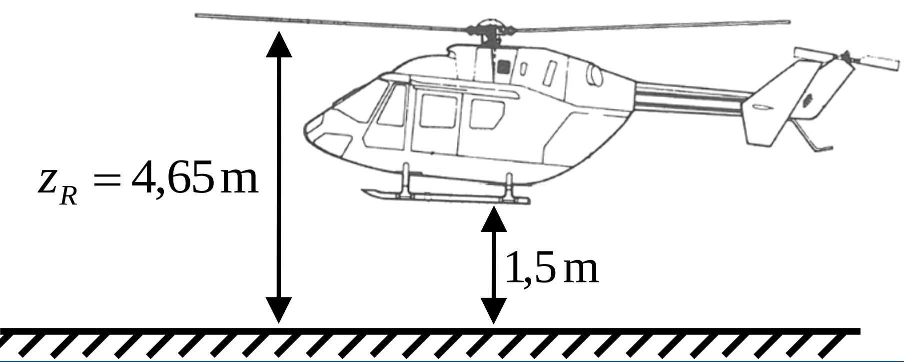


Photo: Wikipedia, Maurihamm

3.5. Refined performance calculation

Hover power in ground effect (IGE)

$$P = P_i + P_0 + P_{TR} + P_a + P_{tl}$$

<u>Example requirement:</u>

Departure at *El Alto International*, Bolivia

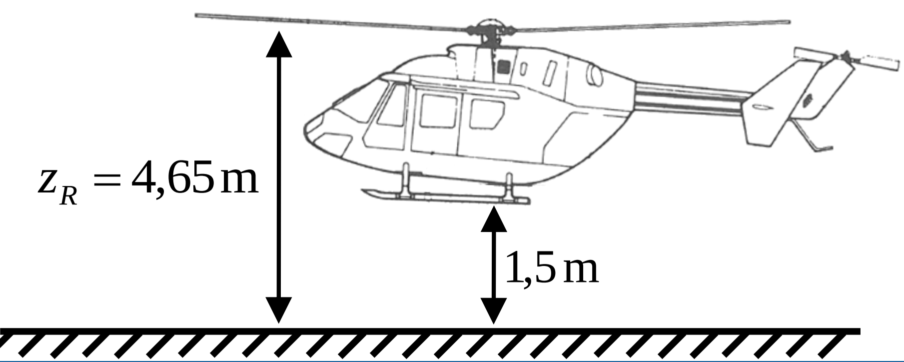
- Take off elevation (*HIGE*) 4075 m
- Outside temperature in ISA: $T = -11{,}5\,°\mathrm{C}$

➔ Air density $\rho = 0{,}81 \frac{kg}{m^3}$

- Mass M=3200kg

$$P_i = k_G \cdot P_{i_{oge}} = k_G \cdot 547\,\mathrm{kW}$$

$$\frac{z_R}{R} = \frac{4{,}65\,\mathrm{m}}{5{,}5\,\mathrm{m}} = 0{,}85 \Rightarrow k_G = 0{,}83$$



➔ Induced power $P_i = 454\,\mathrm{kW}$

➔ Profile power $P_0 = 85\,\mathrm{kW}$

➔ Tail rotor power $P_{TR} = 60\,\mathrm{kW}$

➔ Accessories $P_a = 15\,\mathrm{kW}$

➔ Transmission loss $P_{tl} = 11\,\mathrm{kW}$

➔ Total power $P = 625\,\mathrm{kW}$

3.5. Refined performance calculation

Required forward speed in level flight

$$P = P_i + P_0 + P_p + P_{TR} + P_a + P_{tl}$$

Example requirement:

- Altitude 3000m
- Temperature ISA: $T = -4{,}5\,°\mathrm{C}$

➔ Air density $\rho = 0{,}91\,\frac{kg}{m^3}$

- Mass M=3200kg
- Speed of flight

$$V_\infty = 80\,\frac{m}{s} = 156\,\mathrm{kt} = 288\,\frac{km}{h}$$

➔ Parasite power $P_p = 291\,\mathrm{kW}$

➔ Profile power $P_0 = 153\,\mathrm{kW}$

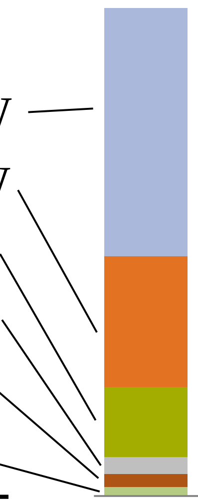
➔ Induced power $P_i = 82\,\mathrm{kW}$

➔ Tail rotor power $P_{TR} = 20\,\mathrm{kW}$

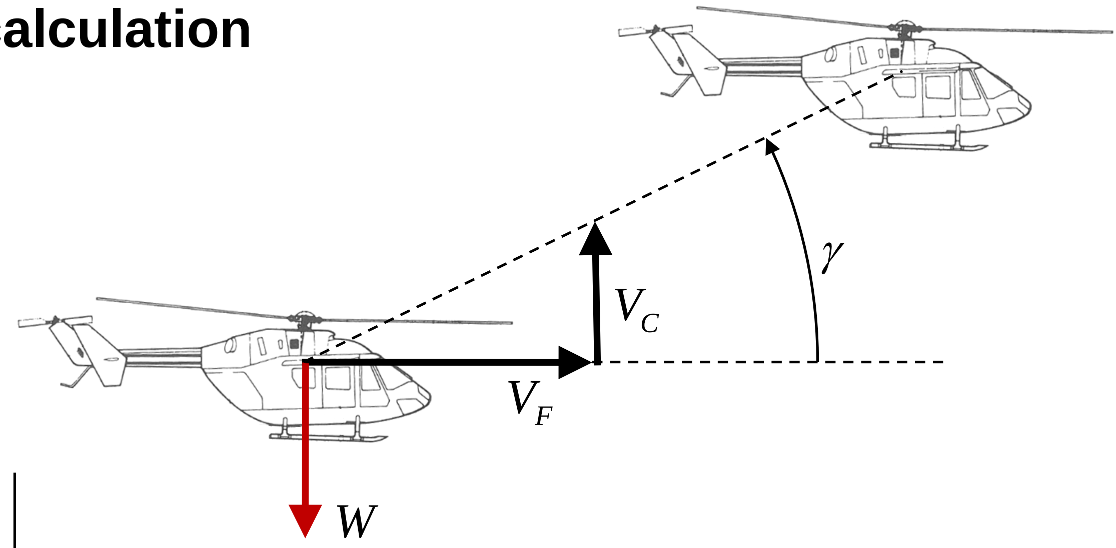
➔ Accessories $P_a = 15\,\mathrm{kW}$

➔ Transmission loss $P_{tl} = 11\,\mathrm{kW}$

➔ Total power $P = 572\,\mathrm{kW}$



3.5. Refined performance calculation



Climb power in forward flight

$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

Example requirement:

- Altitude 3000m
- Temperature
  $\Delta T_{ISA} = +20\,\mathrm{K}$ ➔ $T = 15{,}5\,°\mathrm{C}$

➔ Air density $\rho = 0{,}85\,\frac{kg}{m^3}$

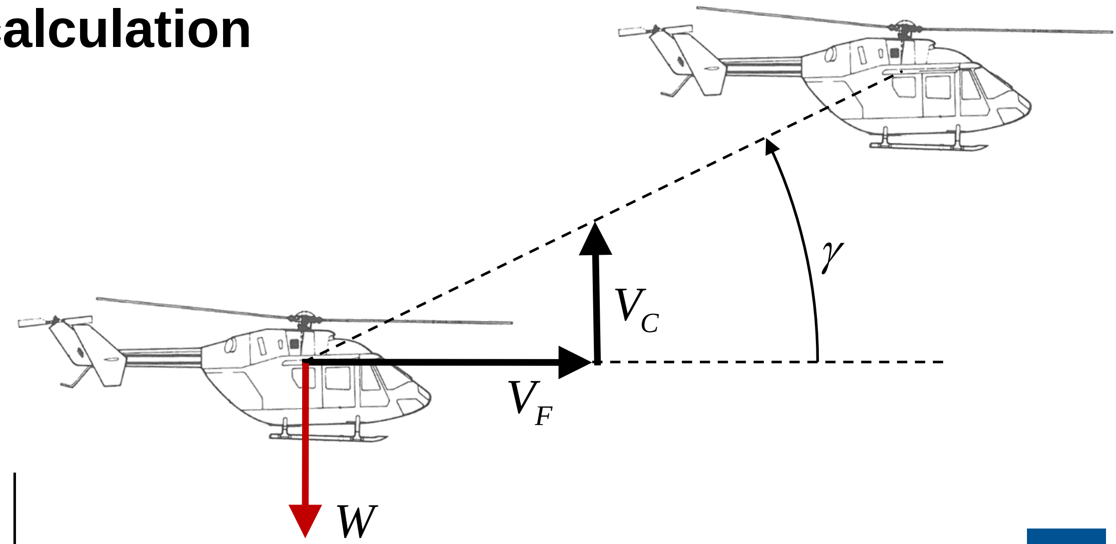
- Mass M=3200kg
- Forward speed
  $V_F = 50\,\frac{m}{s} = 97\,\mathrm{kt} = 180\,\frac{km}{h}$
- Climb rate
  $V_C = 5\,\frac{\mathrm{m}}{\mathrm{s}} = 948\,\frac{\mathrm{ft}}{\mathrm{min}}$

The influence of V_C on the remaining power components is neglected due to the low angle of climb γ.

Compared to level flight, the climb power is added:

$$P_C = V_C \cdot W$$

3.5. Refined performance calculation



Climb power in forward flight

$$P = P_i + P_0 + P_p + P_C + P_{TR} + P_a + P_{tl}$$

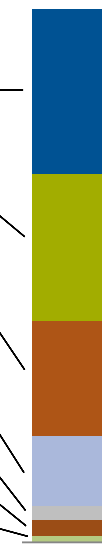
<u>Example requirement:</u>

- Altitude 3000m
- Temperature
  $\Delta T_{ISA} = +20\,\text{K}$ ➔ $T = 15{,}5\,°\text{C}$

➔ Air density $\rho = 0{,}85\,\frac{kg}{m^3}$

- Mass M=3200kg
- Forward speed
  $V_F = 50\,\frac{m}{s} = 97\,\text{kt} = 180\,\frac{km}{h}$
- Climb rate
  $V_C = 5\,\frac{\text{m}}{\text{s}} = 948\,\frac{\text{ft}}{\text{min}}$

➔ Climb power	$P_C = 157\,\text{kW}$
➔ Induced power	$P_i = 141\,\text{kW}$
➔ Profile power	$P_0 = 109\,\text{kW}$
➔ Parasite power	$P_p = 66\,\text{kW}$
➔ Tail rotor power	$P_{TR} = 14\,\text{kW}$
➔ Accessories	$P_a = 15\,\text{kW}$
➔ Transmission loss	$P_{tl} = 10\,\text{kW}$
➔ Total power	$P = 511\,\text{kW}$



3.5. Refined performance calculation

Summary of the power requirements

	H_G $[\mathrm{m}]$	ΔT_{ISA} $[\mathrm{K}]$	P $[\mathrm{kW}]$
HOGE	1000	25	685
HIGE	4075	0	625
Level flight	3000	0	572
Climb flight	3000	20	511

Engines need to be selected which are able to deliver the required power P in the corresponding environmental conditions in order to guarantee the completion of the required flight tasks.

3 Sizing

3.5. Refined performance calculation

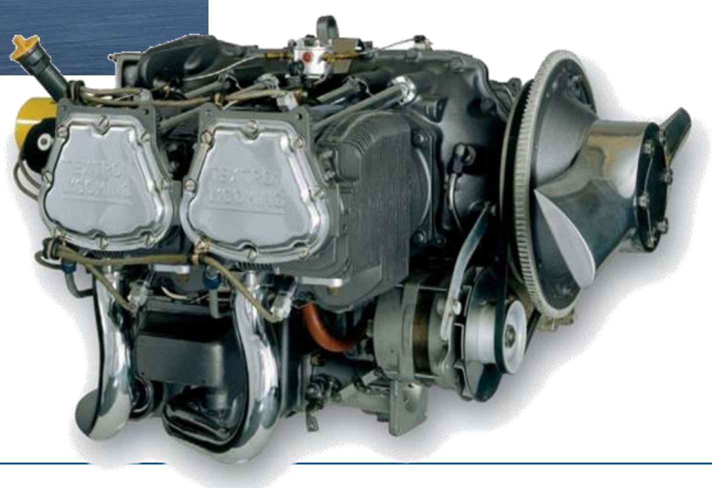
- ✓ Composition of total power requirement
- ✓ Standard atmosphere
- ✓ Determination of required engine power
- ➔ Selection of engine and gearbox
- • Refined mass estimation

3.5. Refined performance calculation

Selection of engine

<u>Piston engine</u>

Low costs, nowadays only used for light helicopters, favourable fuel consumption also in part-load operational range



<u>Turboshaft engine</u>

High power density

Photos: Robinson Helicopter Company; Lycoming Engines; Eurocopter, Patrick Penna; Eurocopter

3.5. Refined performance calculation

Turboshaft engine

The available engine power depends on density and temperature of the inlet air as well as installation losses.

Limiting for the power conversion in the combustion chamber is the turbine entry temperature T_4 which is given by the thermal durability of the blade material.

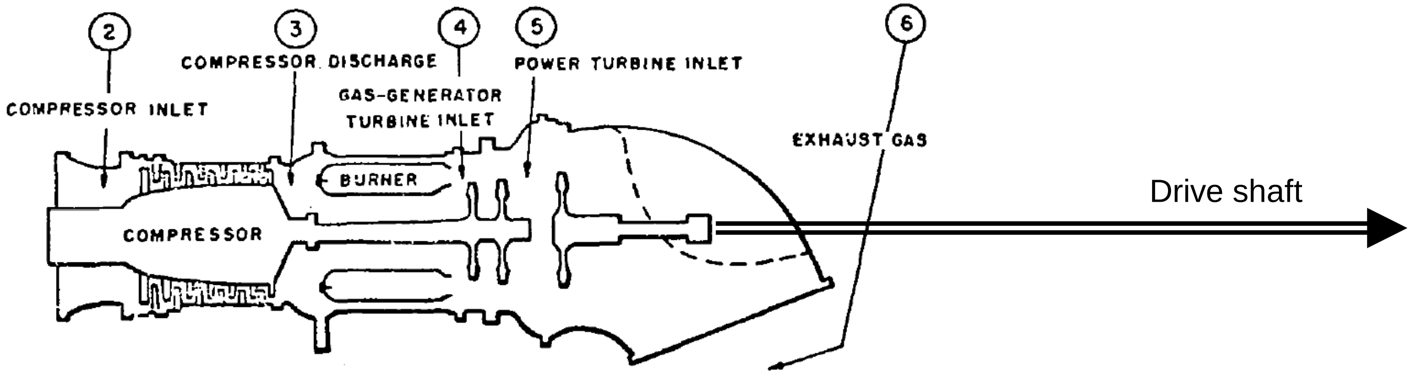


Figure 1.5 Typical gas flow diagram showing stations used in performance analysis

Figure from Stepniewski: *Rotary-Wing Aerodynamics*

3.5. Refined performance calculation

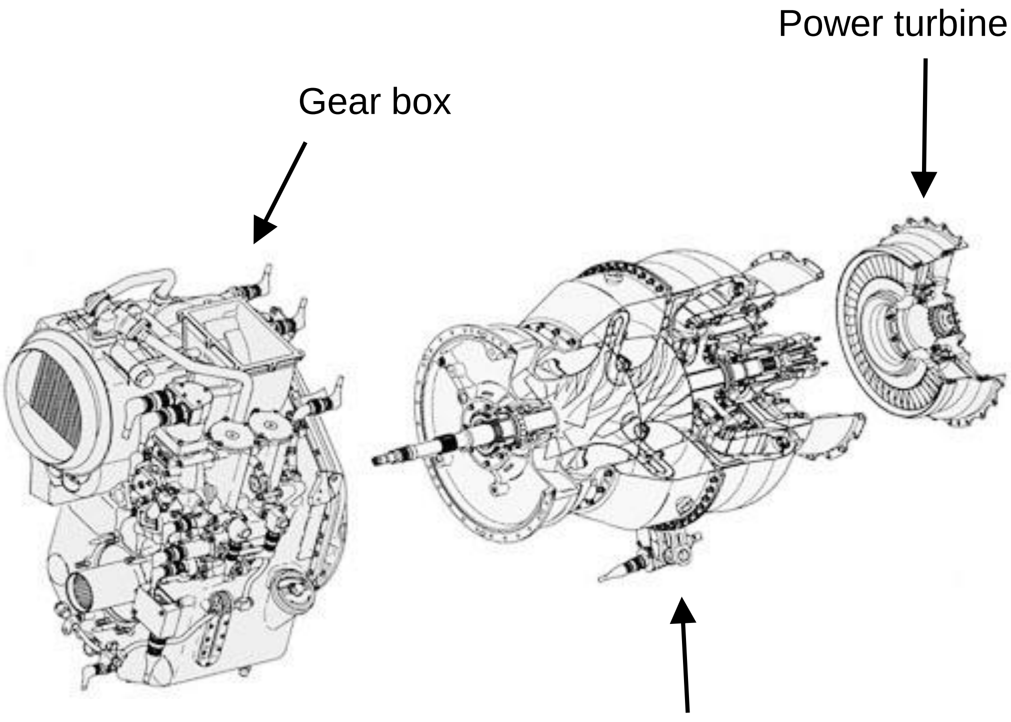
Turboshaft engine

The provided power of a turboshaft engine P_{TS} can be approximated for different environmental conditions:

$$P_{TS} = P_{TS,S} \cdot \frac{\delta}{\sqrt{\theta}}$$

with

- Power at 0m ISA: $P_{TS,S}$
- Temperature ratio $\theta = \frac{T}{T_s}$
- Pressure ratio $\delta = \frac{p}{p_s}$



Engine *MTR390* (*EC Tiger*)

Figure : MTU Turbomeca Rolls-Royce GmbH

3.5. Refined performance calculation

Turboshaft engine

The provided power of a turboshaft engine P_{TS} can be approximated for different environmental conditions:

$$P_{TS} = P_{TS,S} \cdot \frac{\delta}{\sqrt{\theta}}$$

with

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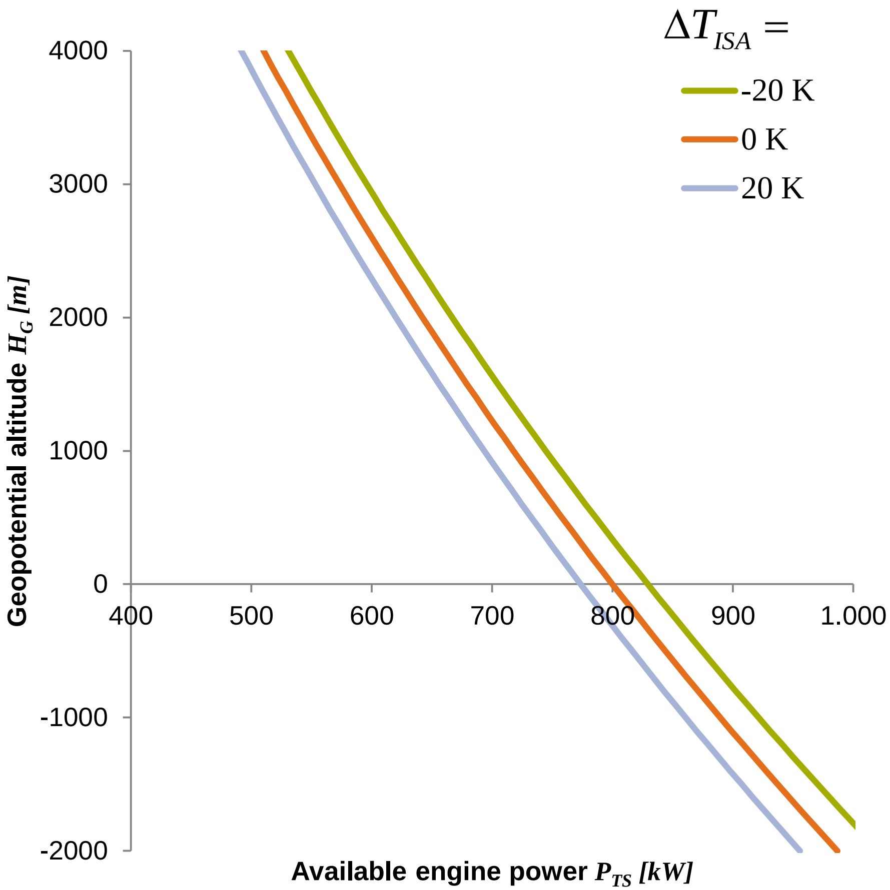
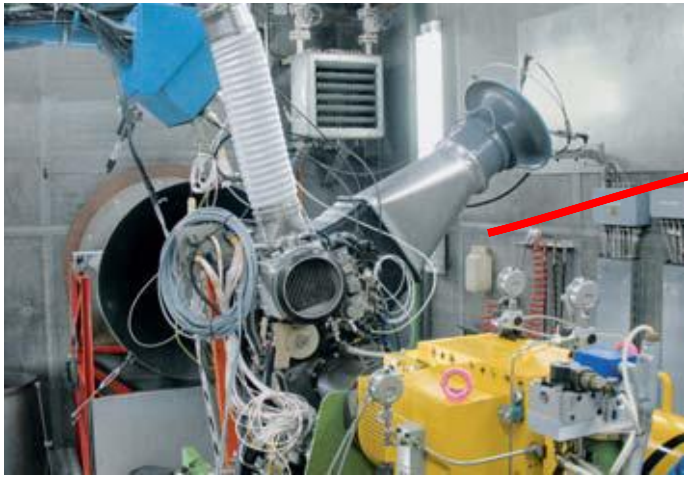


3.5. Refined performance calculation

Installation losses

Compared to the operation on the test bench, an engine mounted in the helicopter provides a lower power due to installation losses.



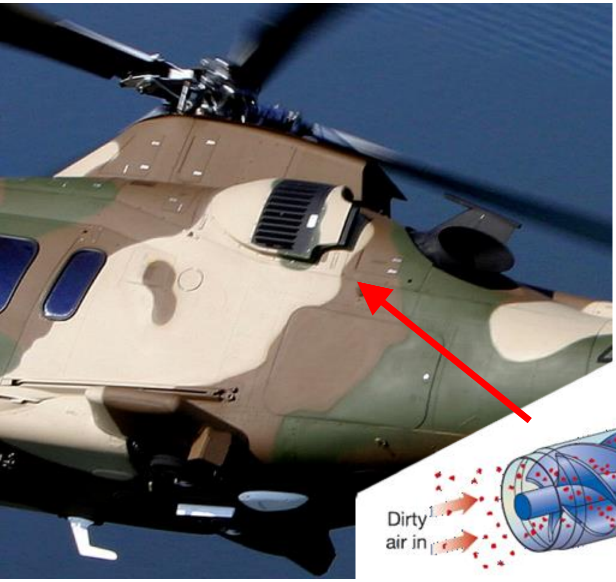
Photos: MTU Aero Engines; Eurocopter, Eric Raz

3.5. Refined performance calculation

Installation losses

Intake losses

- Pressure loss due to wall friction
- Temperature increase due to recirculation of exhaust in hover
- Temperature increase due to gear box in inlet air stream
- Pressure losses due to debris seperator or filter



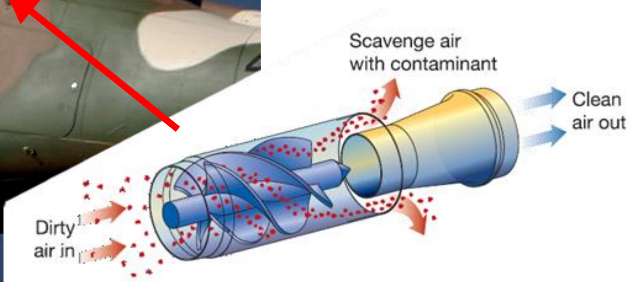
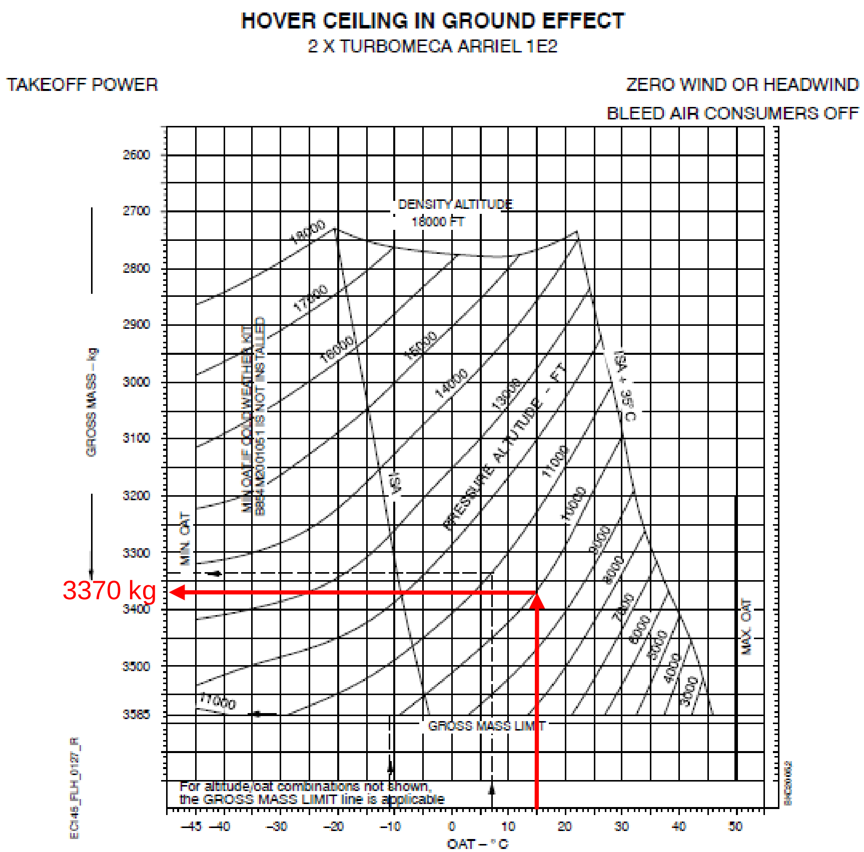


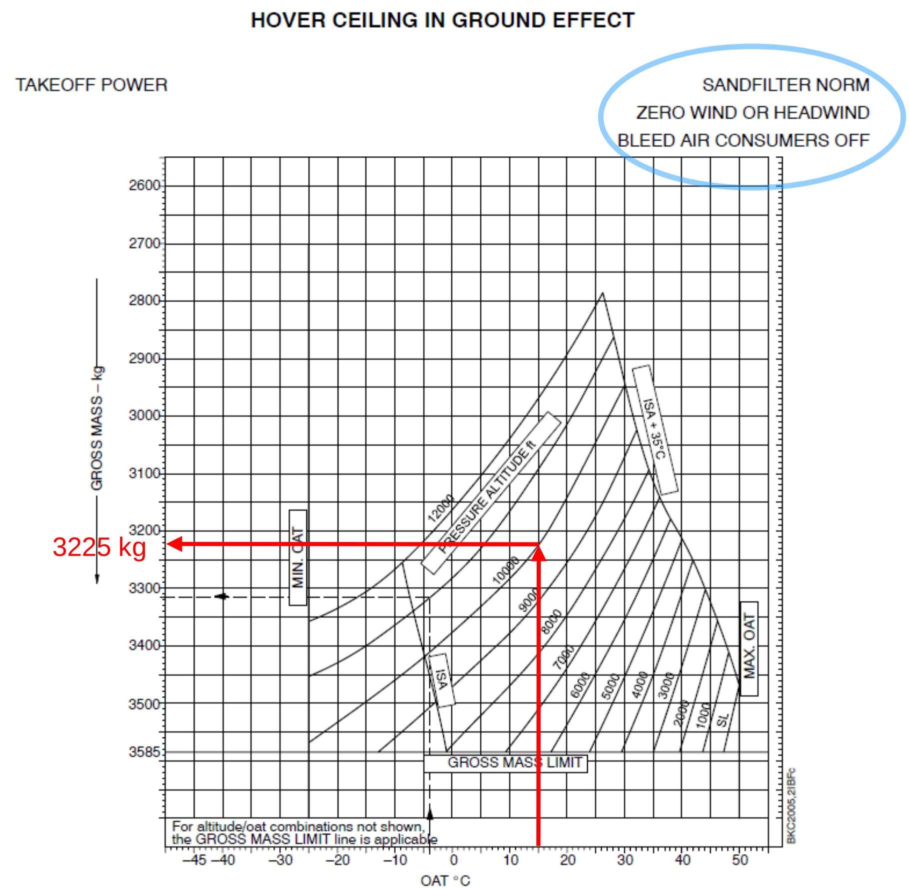
Figure: Pall Corporation; Photos: AgustaWestland; Eurocopter USA

3.5. Refined performance calculation

Installation losses

Example: EC145 *Sand Filter*





Figures: *Flight Manual BK 117 C-2*

3.5. Refined performance calculation

Installation losses

Outlet losses

- Flow losses in the outlet diffusor, especially due to the infrared suppressor

Bleed air losses

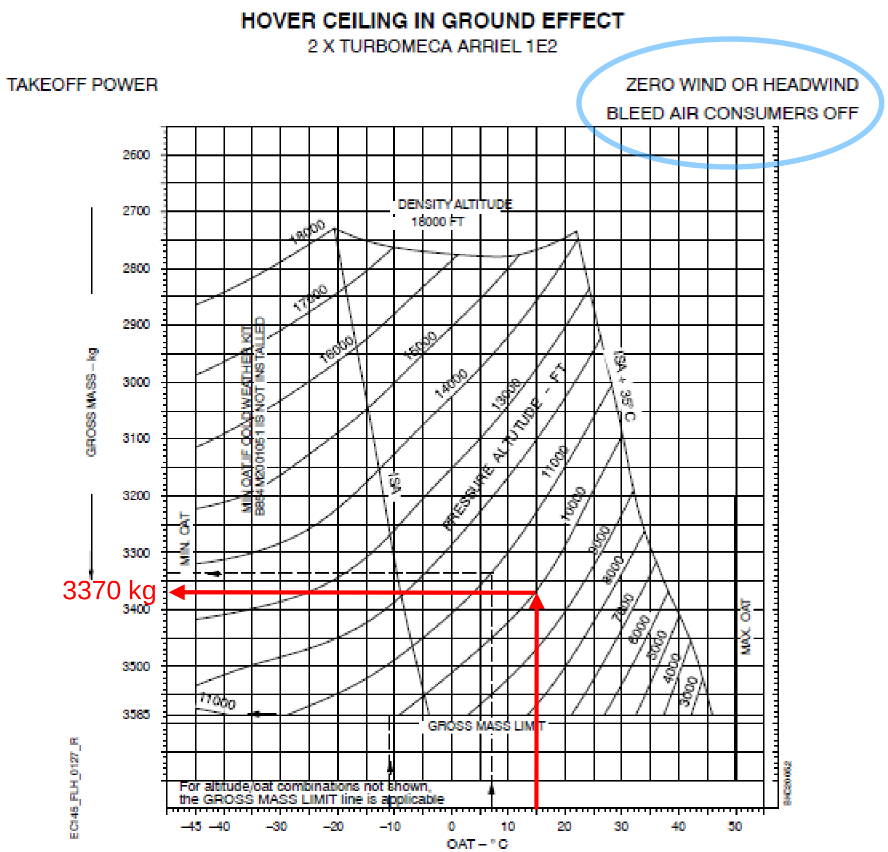
- Warm air from the compressor for de-icing of engine inlet
- Bleed air for air conditioning in cabin

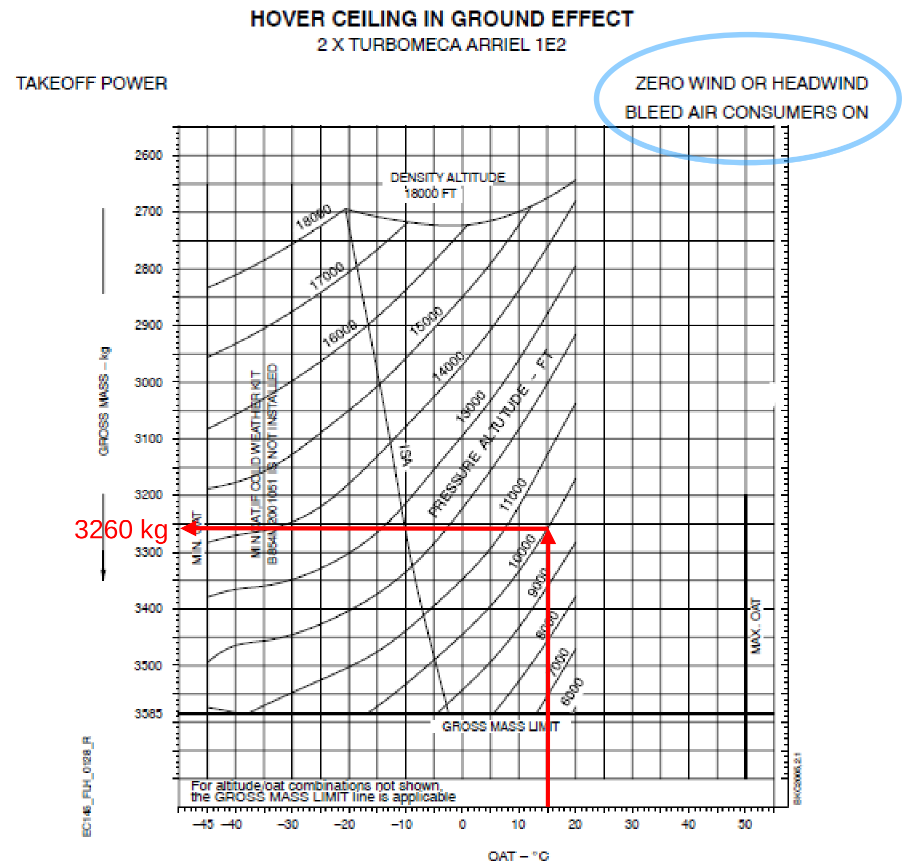
Photos: Eurocopter, Frédéric Lert

3.5. Refined performance calculation

Installation losses

Example: EC145 *Bleed Air*





Figures: *Flight Manual BK 117 C-2*

3.5. Refined performance calculation

Power ratings

For engines and gear boxes apply performance limits which are depending on the flight situation and are nowadays generally monitored automatically:

- At the turboshaft engine defined by maximum permissible turbine entry temperature T_4 respectively the associated turbine outlet temperature T_5
- At the gear box determined by the maximum transmissible torque

For short periods of time higher values can be allowed compared to the permitted *maximum continuous power*. Depending on the *rating* strategy, different power ratings can be introduced, for example:

- *Take-off power, contingency* (2 to 10 min)
- *Intermediate, military* (30 min)
- *One engine inoperative (OEI) ratings* for increased limits on remaining engines

3.5. Refined performance calculation

Power ratings

Example: EC145, 2x Turbomeca Arriel 1E2, Sealevel ISA

		AEO	OEI
Engine (single)	2.5 min		574
	TOP (5 min)	550	
	MCP	516	550
Gear box	2.5 min		551
	TOP (5 min)	2 x 388	
	MCP	2 x 316	404

Engine: Power available dependent on air density and temperature

Gear box: Determined by transmissible torque, independent from atmospheric conditions

AEO: all engines operating

OEI: one engine inoperative

TOP: take-off power

MCP: maximum continuous power

Data: Eurocopter

3.5. Refined performance calculation

Power ratings

EC135 First Limit Indicator

Advises the pilot of limits:

- Torque
- Turbine outlet temperature (*TOT*)
- Turbine speed
- Mast moment

Figure: *Flight Manual EC 135 P2+*

3.5. Refined performance calculation

Deterioration of engines: Regular checks of power available (Example EC145)

5.1.4.1 Power check procedures

Two different engine power check procedures are provided:

a) Ground power check:

This procedure shall be exercised on ground to make certain that the engine power available is within the limits established for legal use of the flight manual performance charts.

b) Inflight power check:

This procedure is provided to check the engine power levels in cruising flight to make certain that the engine power available is within the limits established for legal use of the flight manual performance charts. It is no alternative to the ground power check when a power check is required **before flight** by operational rules.

The power check diagrams (figures 5-1 to 5-6) show:

– the maximum allowable N_1 as a function of adjusted torque or

– the minimum percent torque as a function of adjusted N_1 .

NOTE Observe power check procedures according to FMS 9.2–22, if SANDFILTER SYSTEM is installed.

5.1.4.2 Power check intervals

Either ground or inflight power check shall be accomplished

– at intervals not exceeding 100 flying hours for Category B operation.

– whenever abnormal engine function is suspected.

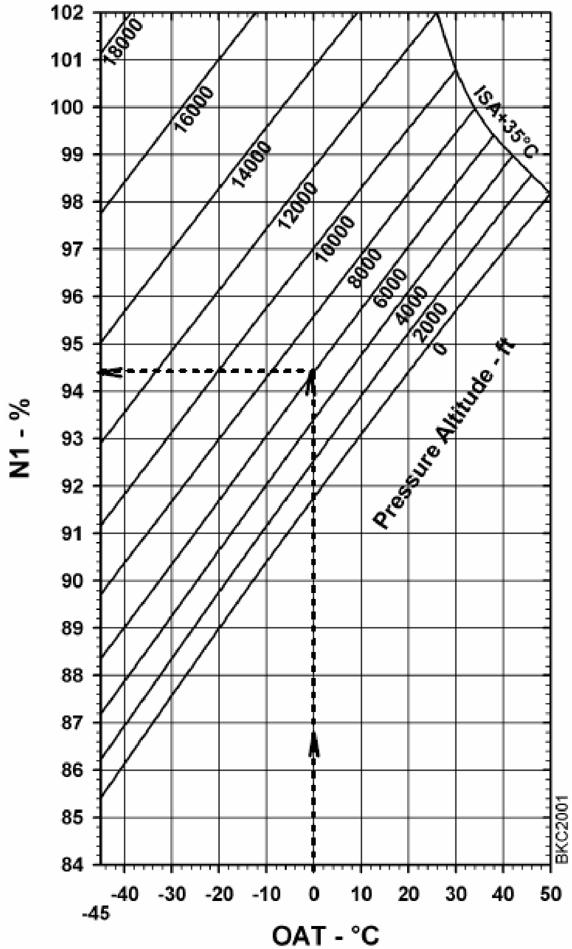


Figure: *Flight Manual BK 117 C-2*

3.5. Refined performance calculation

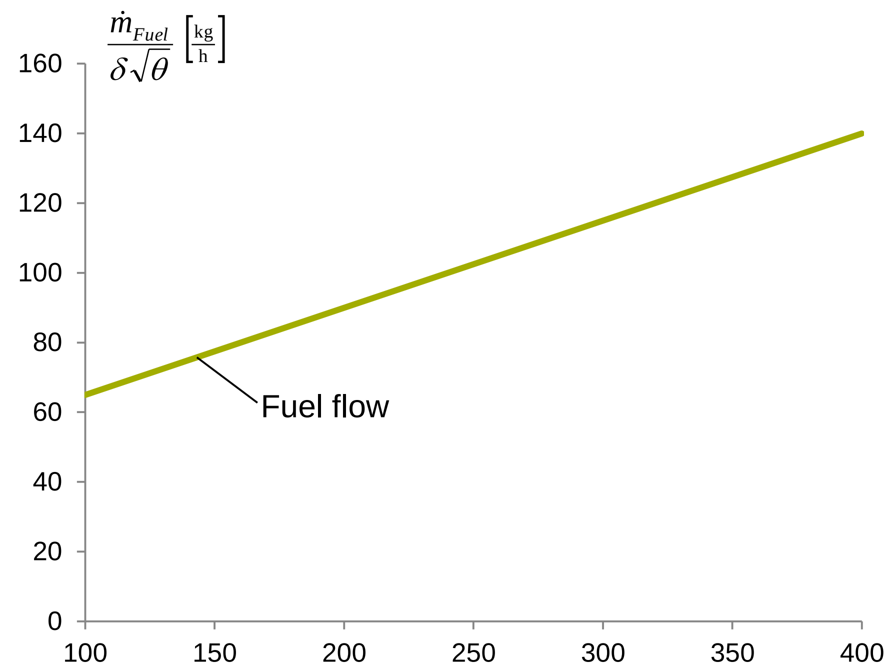
Fuel flow

By approximation the fuel flow $\dot{m}_{Fuel}$ of a turboshaft engine can be plotted as a linear function of the power output P:

$$\frac{\dot{m}_{Fuel}}{\delta \cdot \sqrt{\theta}} = A_E + B_E \cdot \frac{P}{\delta \cdot \sqrt{\theta}}$$

with

- Engine-dependent parameters A_E, B_E
- Temperature ratio $\theta = \frac{T}{T_s}$
- Pressure ratio $\delta = \frac{p}{p_s}$



3.5. Refined performance calculation

Specific fuel consumption

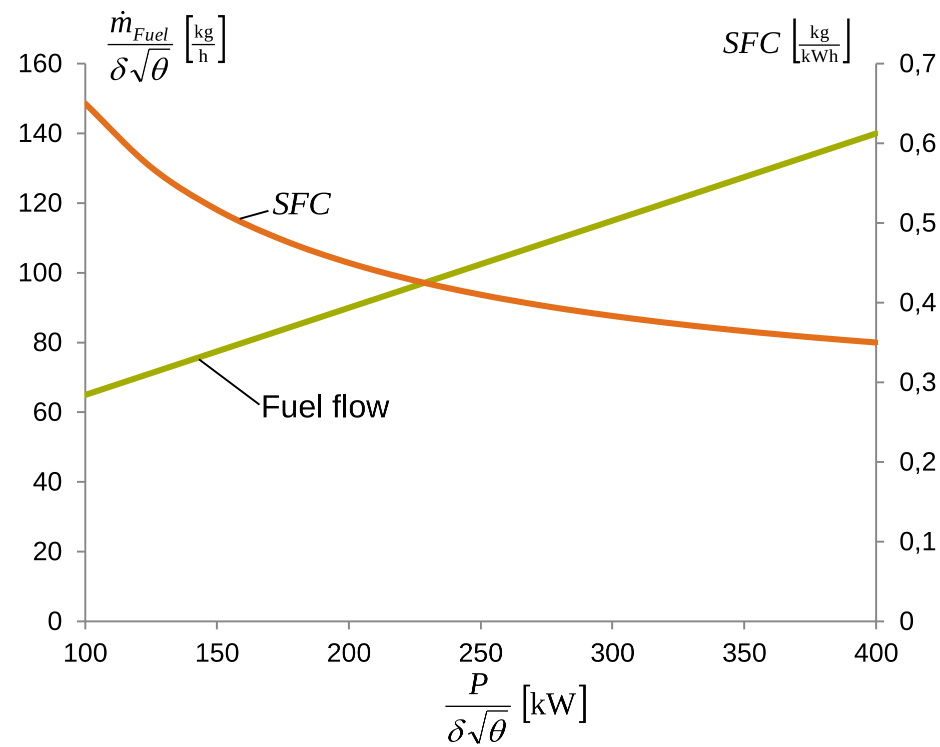
If we relate the fuel flow $\dot{m}_{Fuel}$ to the power output P, we get the specific fuel consumption:

$$SFC = \frac{\dot{m}_{Fuel}}{P}$$

With the introduced approximation of the fuel flow $\dot{m}_{Fuel}$ for a turboshaft engine, it follows

$$SFC = A_E \cdot \frac{\delta \cdot \sqrt{\theta}}{P} + B_E$$

➔ Efficiency of power conversion increases with delivered power P



3.5. Refined performance calculation

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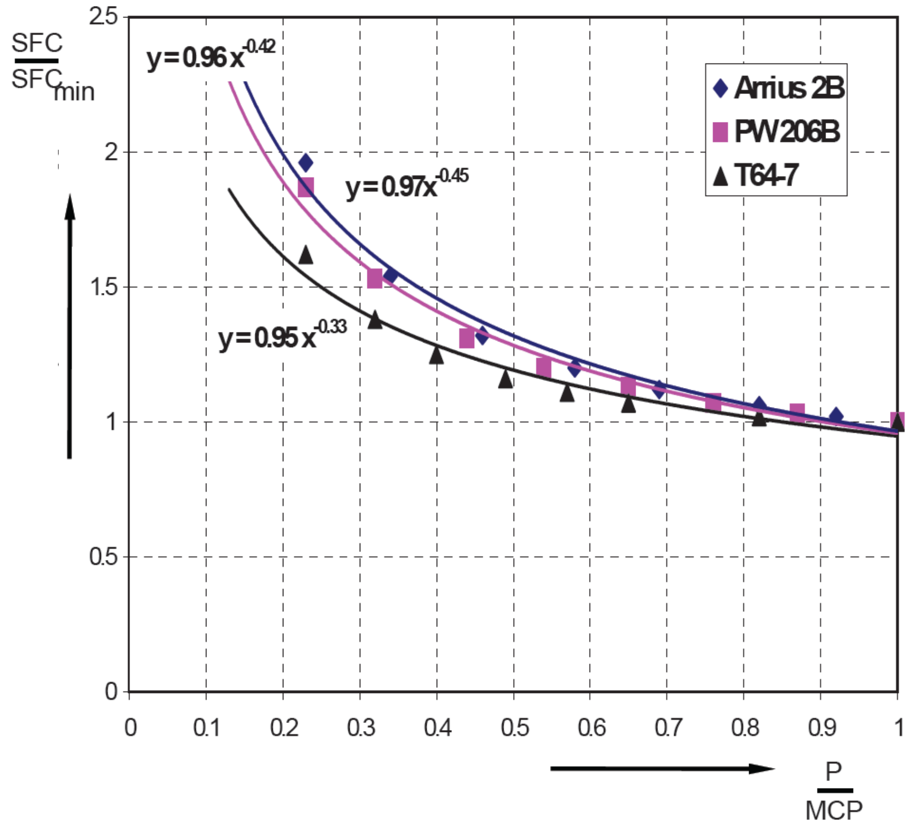


Diagram: Eurocopter, Koletzko

3.5. Refined performance calculation

Specific fuel consumption

Generally the specific fuel consumption decreases with the power potential of the engines.

This can be attributed to proportionally lower losses, e.g. due to blade tip gaps.

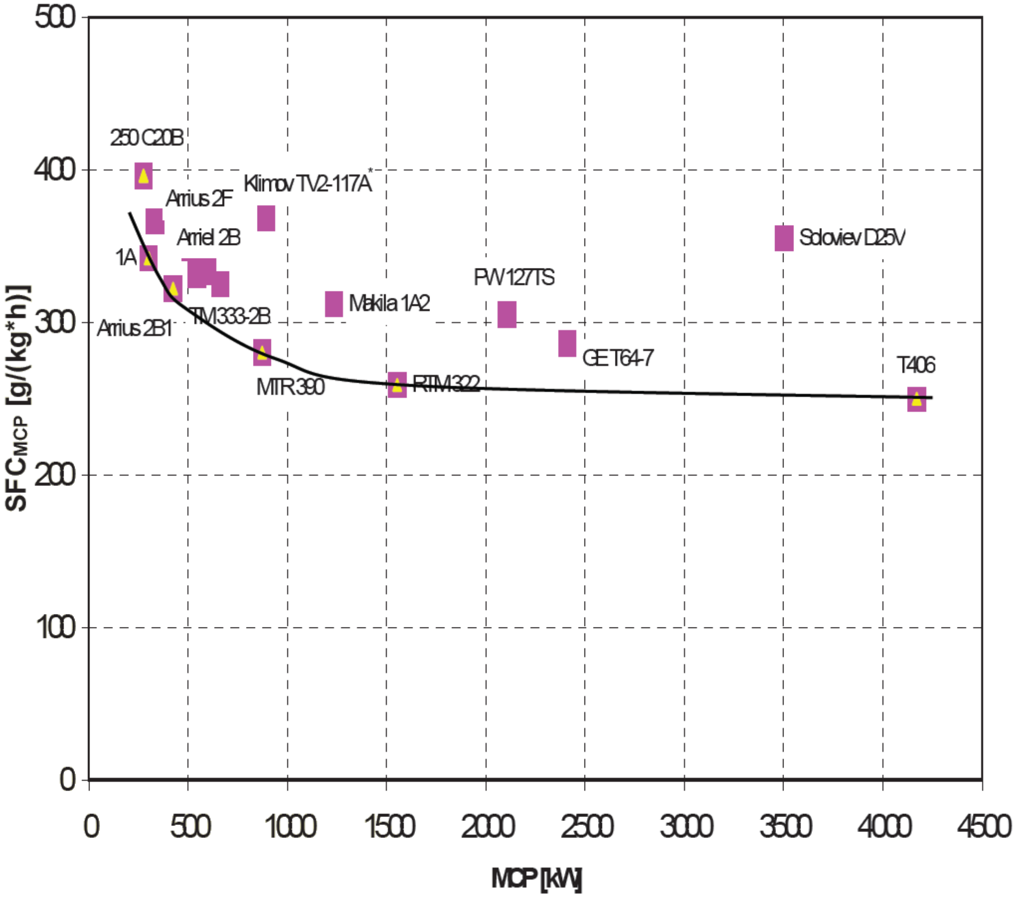


Diagram: Eurocopter, Koletzko

3.5. Refined performance calculation

Selection of engine – *Turboshaft Engine*

Power density

$$\frac{\text{Power available}}{\text{Engine mass}} = \frac{P_{av}}{m_{Engine}}$$

The power density of an engine increases with available engine power.

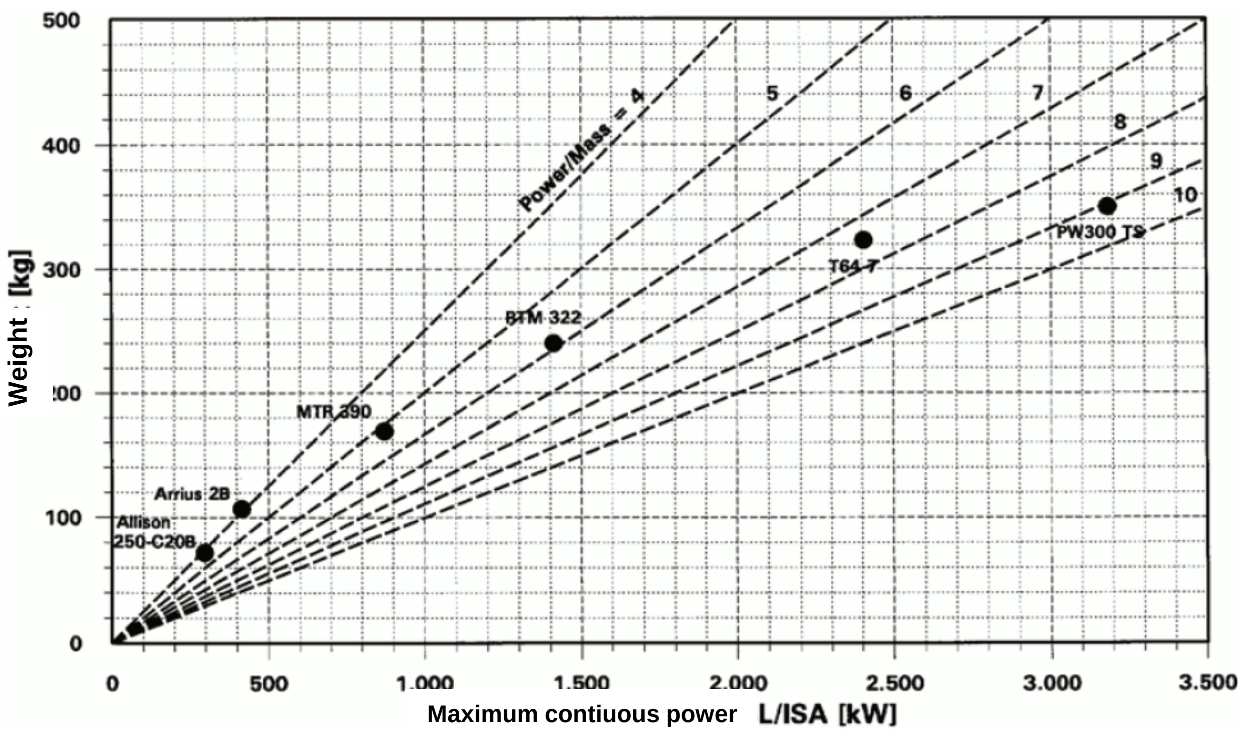


Diagram: Eurocopter, Koletzko

3.5. Refined performance calculation

Selection of engine

Different approaches can be chosen for the selection of an engine :

- Rubber engine (early design stage)
  Hypothetical engine which provides exactly the required power. Weight and fuel consumption are determined from empirical trends of actual engines.

- Available engine types
  Actual available engines in the market. The warranted performances and characteristics are in this process provided from the engine manufacturer in specifications.
  Due to numerous influencing factors (atmosphere, installation losses, abrasion, bleed air…), a simulation programme from the engine manufacturer can be delivered which reproduces the characteristics of the engine (*engine deck*).

3.5. Refined performance calculation

Number of engines

- Several engines provide an increased safety due to residual power in case of an engine failure (*OEI*)
- At least two engines required for *CAT-A* certification

- Low number of engines for low costs (acquisition, maintenance…)
- Low consumption due to few large engines (lower SFC)
- Lower weight with low number of engines (higher power density)

➔ Mostly 1-2 engines,
rarely also 3-4 (heavy helicopters)


AS350, 1x Turbomeca Arriel 2B1, 632kW TOP

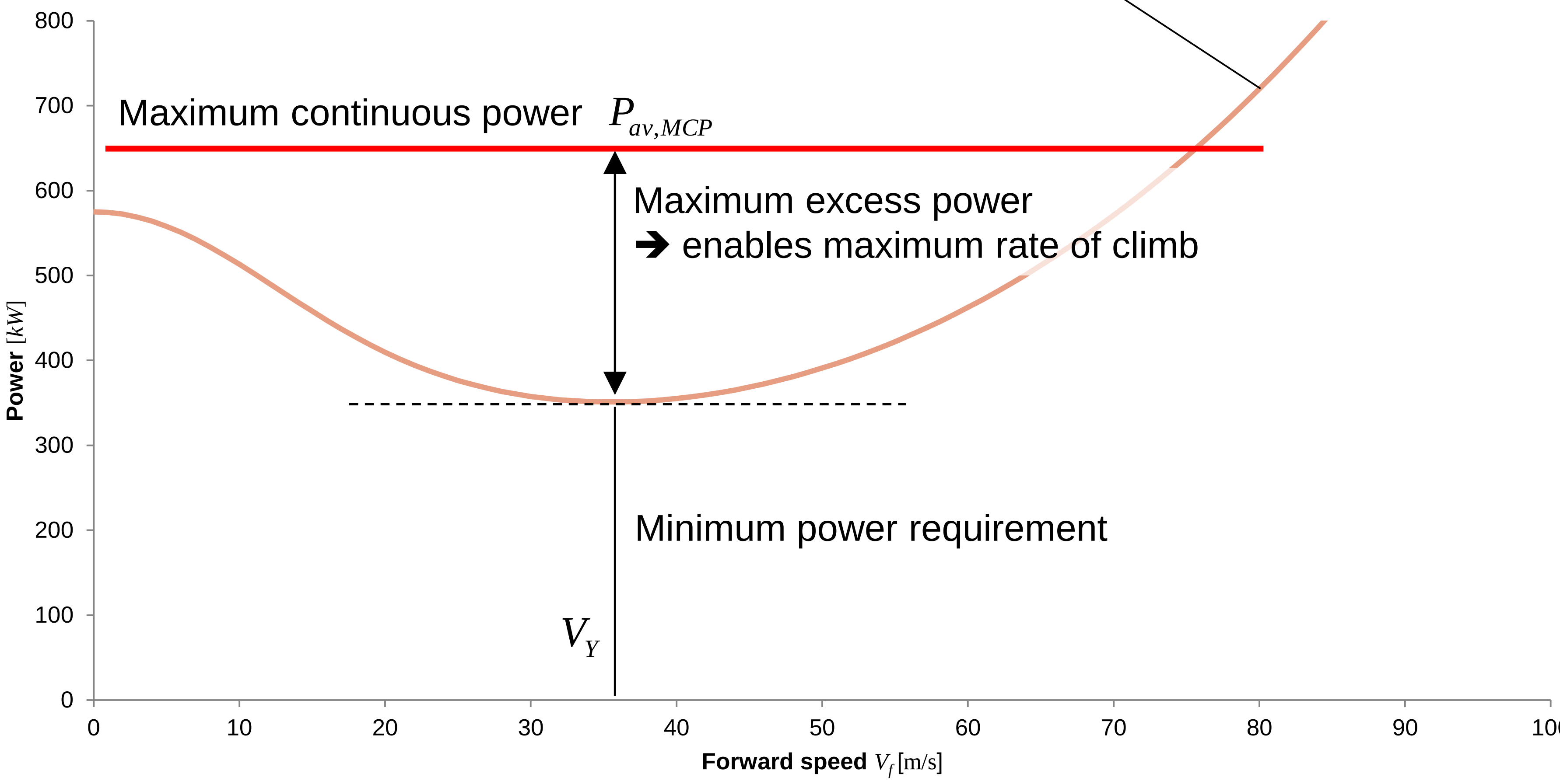
AS355, 2 x Turbomeca Arrius 1A1 à 343kW TOP

Photos: Eurocopter, Alexandre Dubath, Christophe Guibbaud

3.5. Refined performance calculation

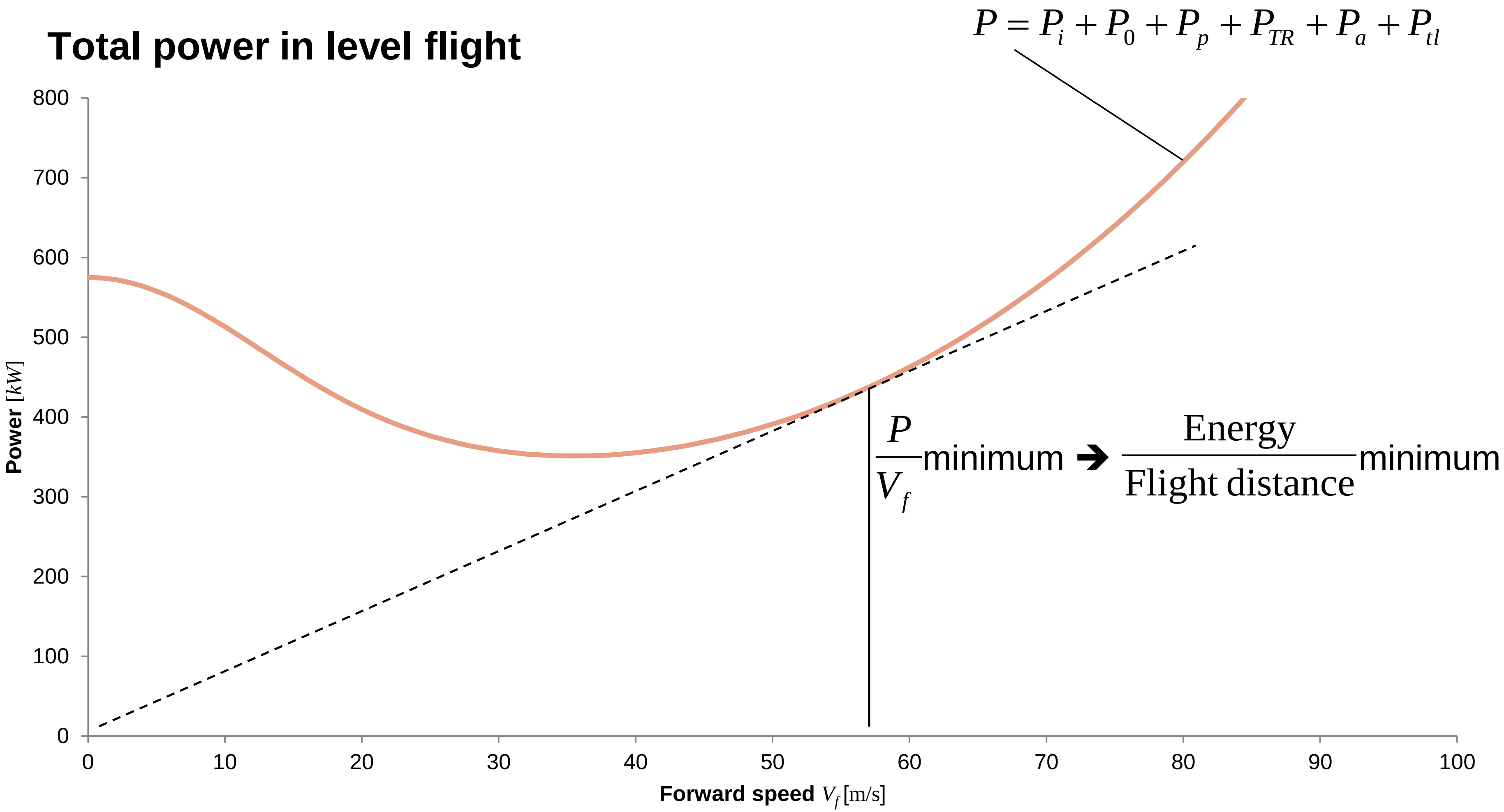
Total power in level flight

$$P = P_i + P_0 + P_p + P_{TR} + P_a + P_{tl}$$



3.5. Refined performance calculation

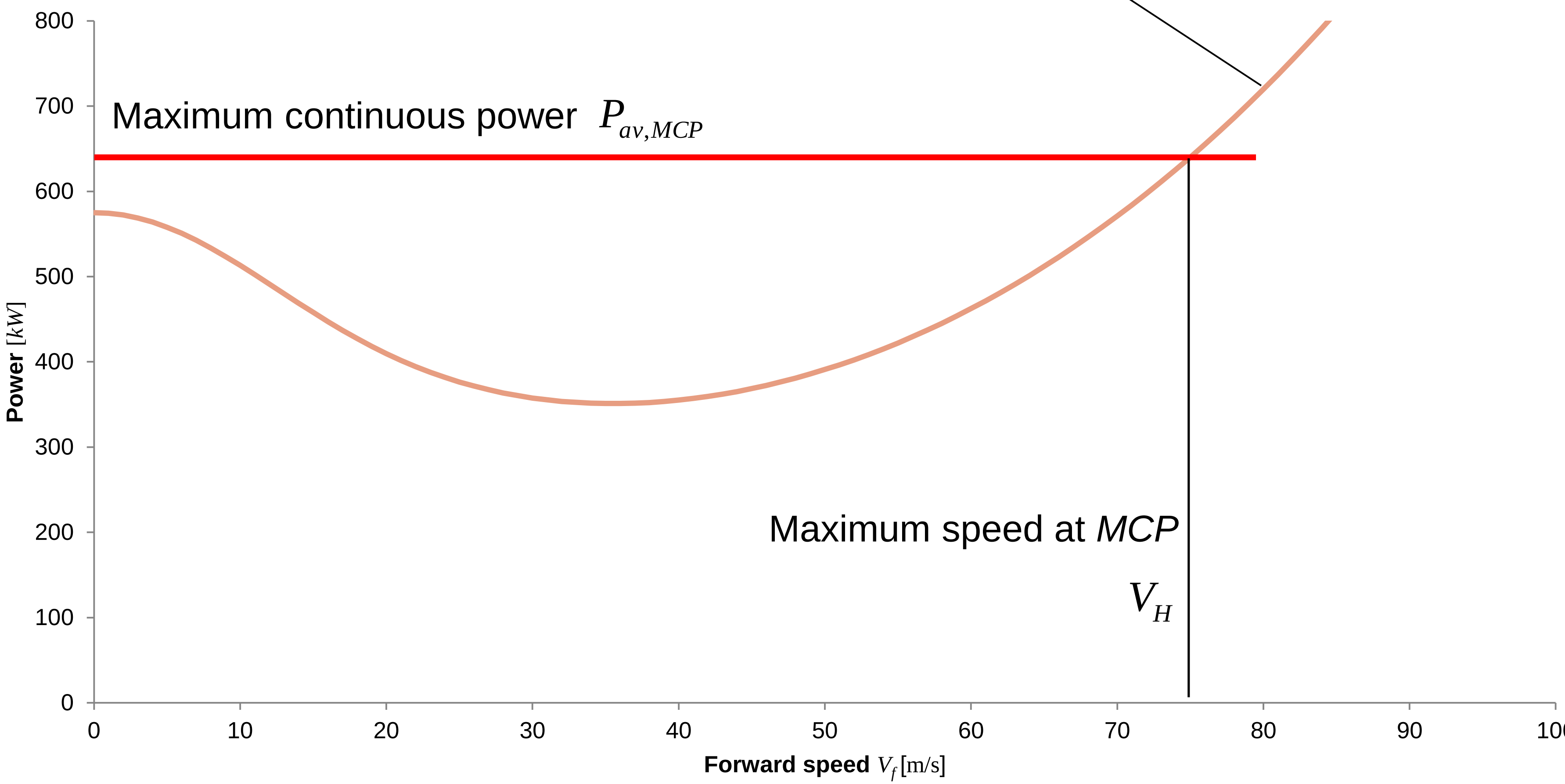
Total power in level flight



3.5. Refined performance calculation

Total power in level flight

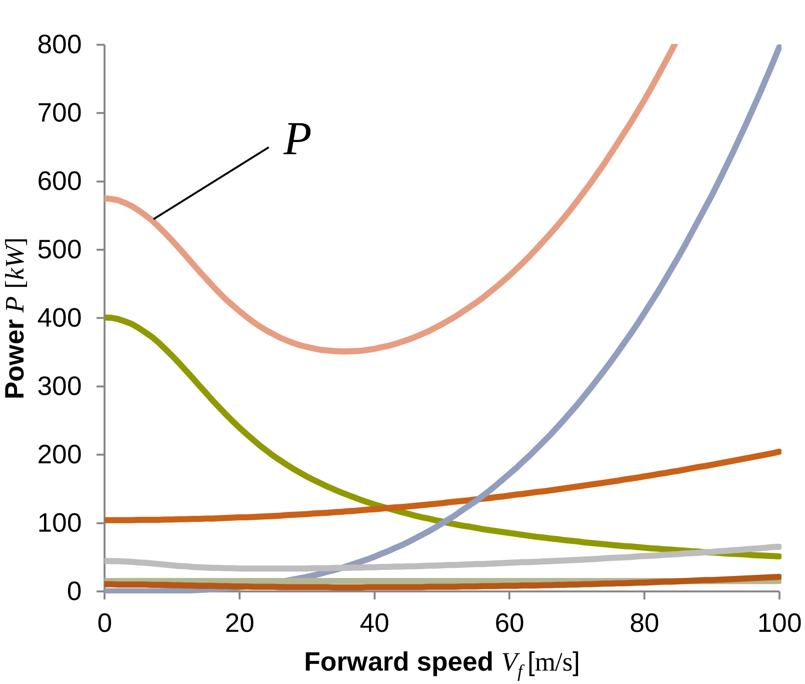
$$P = P_i + P_0 + P_p + P_{TR} + P_a + P_{tl}$$

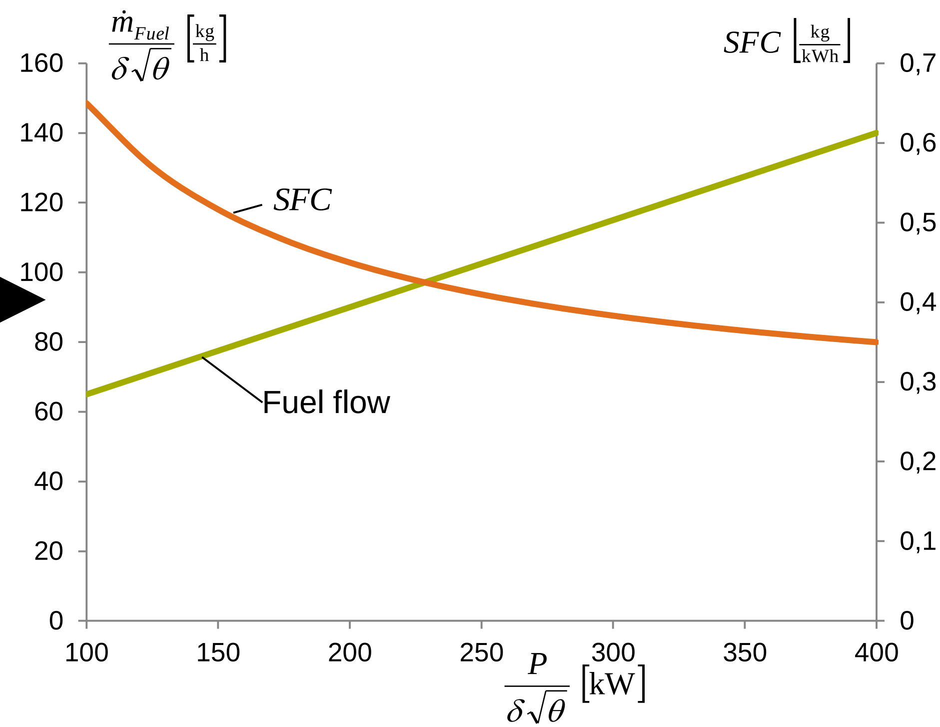


3.5. Refined performance calculation

Fuel flow in level flight

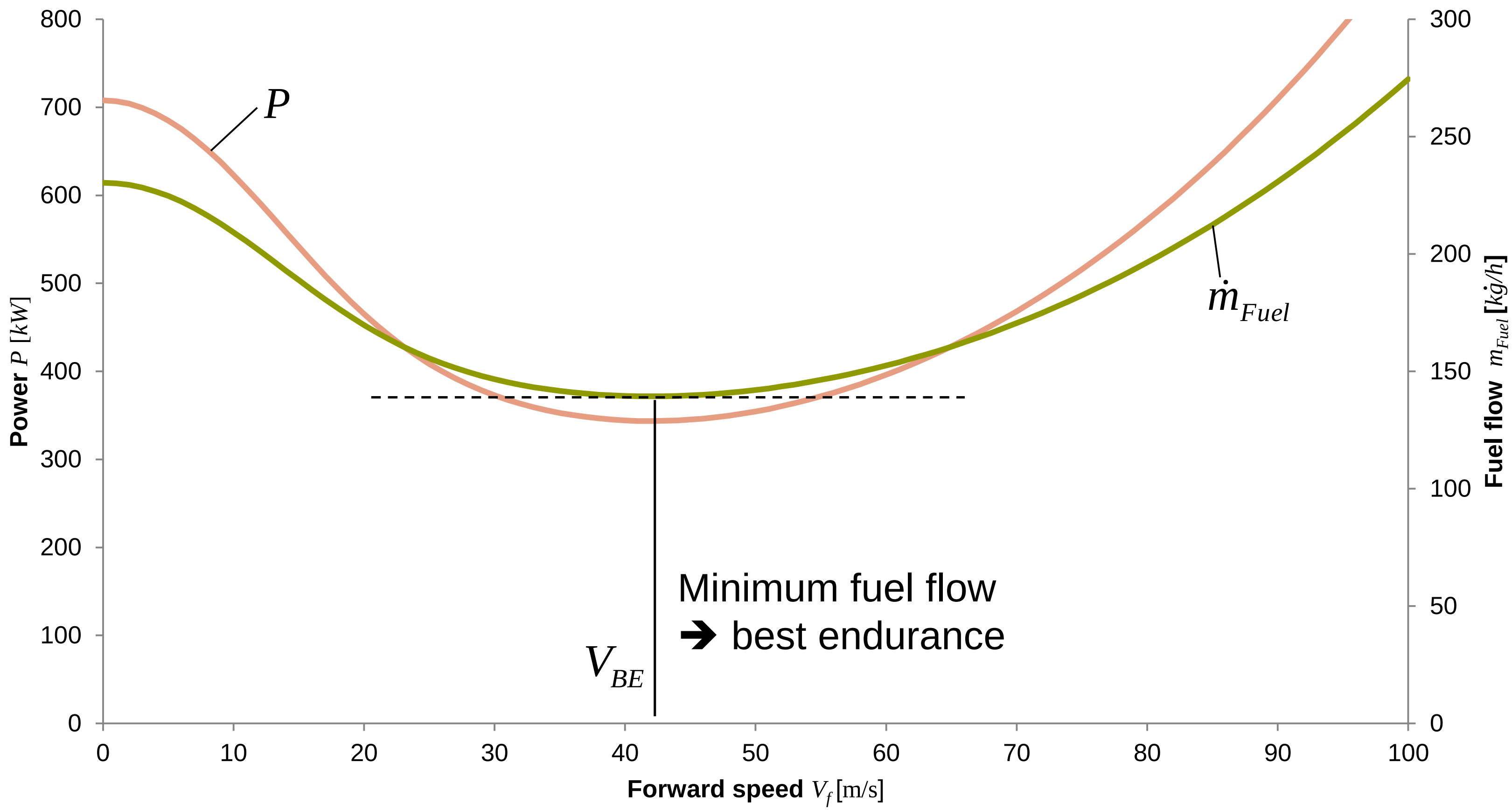
For given environmental conditions (ρ, T), the required power P can be related with the consumption characteristic of the engine, in order to determine the fuel consumption dependent on the flight speed.





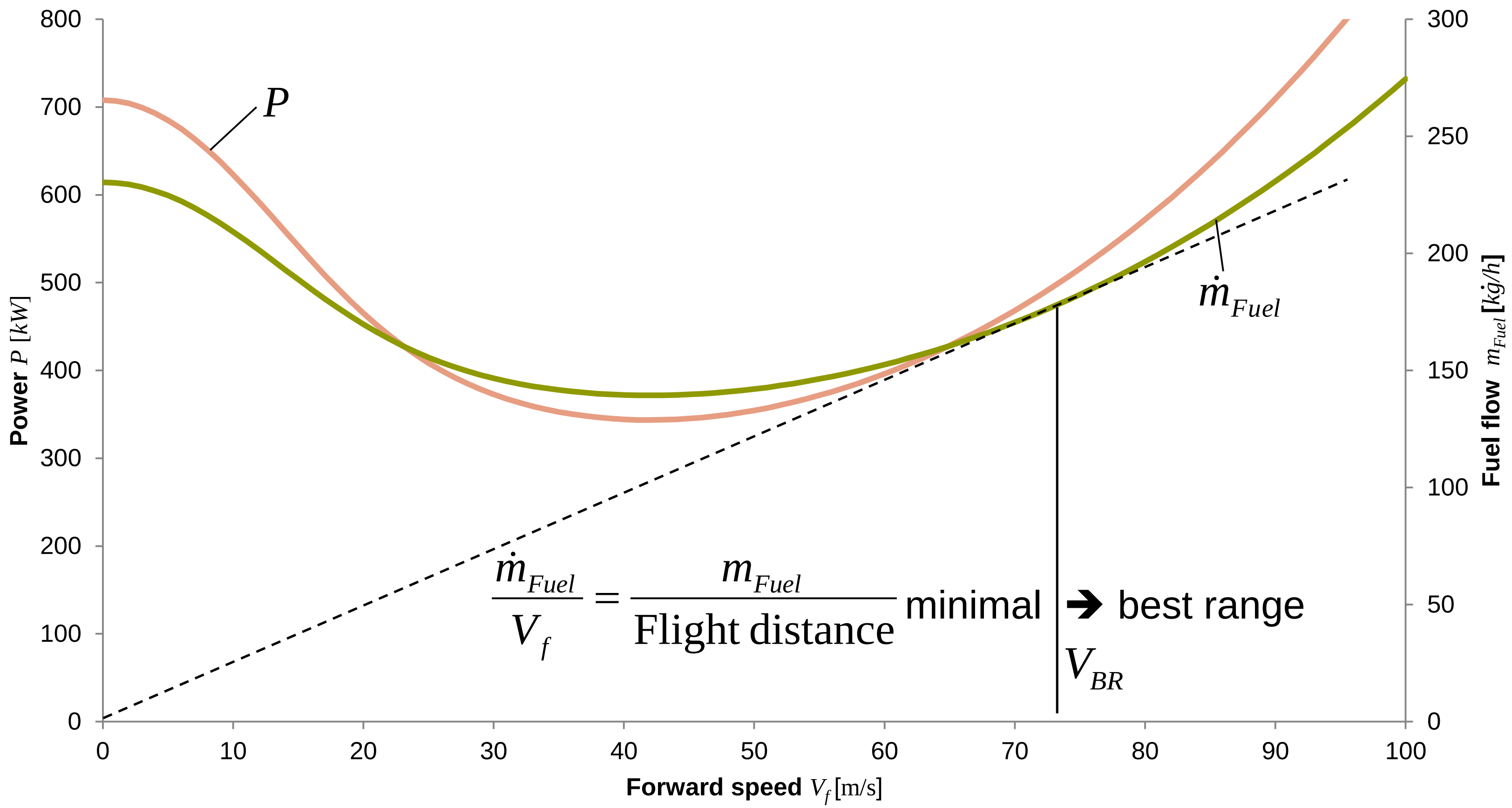
3.5. Refined performance calculation

Fuel flow in level flight



3.5. Refined performance calculation

Fuel flow in level flight

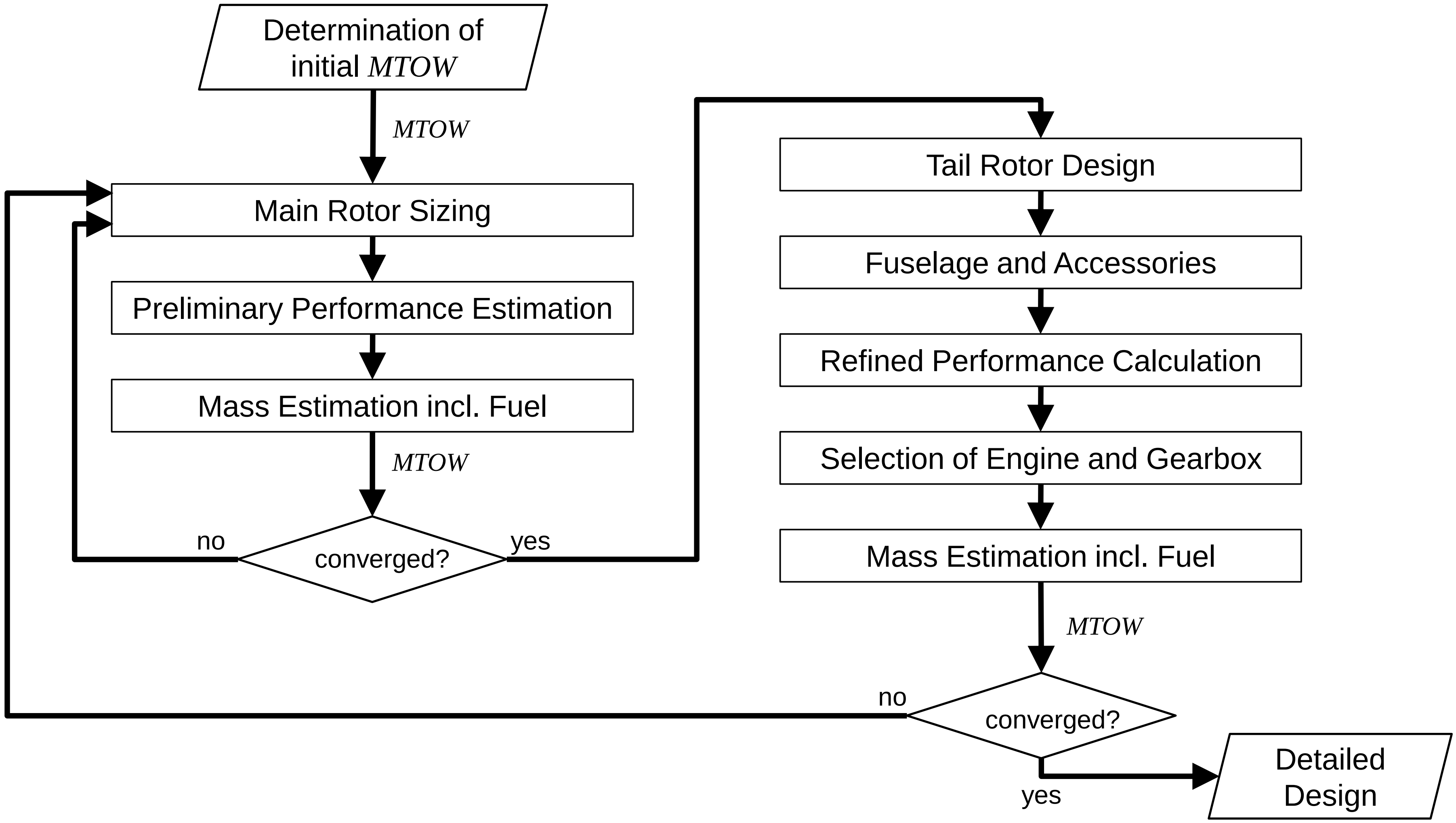


3 Sizing

3.5. Refined performance calculation

- ✓ Composition of total power requirement
- ✓ Standard atmosphere
- ✓ Determination of required engine power
- ✓ Selection of engine and gearbox
- ➔ Refined mass estimation

3 Sizing - Overview

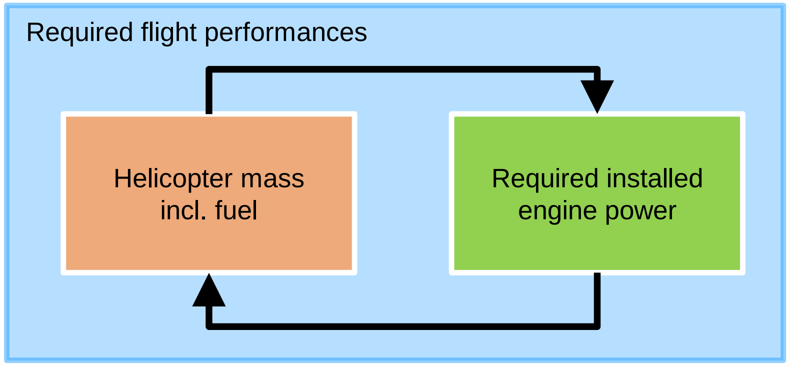


3.5. Refined performance calculation

Refined mass estimation

Since the first rough estimation of the maximum take off weight $\boldsymbol{MTOW}$**,** more accurate knowledge about dimensioning, loads and mass contributions of the single components has been gained:

- Engine
- Gear box
- Tail rotor
- Fuselage
- Empennage



Thus a refined mass estimation of the empty weight $\boldsymbol{EW}$ becomes possible.

3.5. Refined performance calculation

Refined mass estimation – Mission calculation

With the deduced relations for the required total drive power $\boldsymbol{P}$ and the associated fuel consumption, it is possible now to calculate specifically missions for the design.

Thereby we obtain details about the mass contributions to the *useful load* $\boldsymbol{UL}$ *:*

- Crew with equipment
- Total fuel requirement
- Payload, mission-specific equipment

$\left.\right\}$ *useful load* $\boldsymbol{UL}$

Furthermore the required tank capacity and thereby the contribution of the tank to the empty weight $\boldsymbol{EW}$ can be estimated via the required fuel amount.

3.5. Refined performance calculation

Example: EC145

EFFECT OF THE EQUIPMENT ON THE LEVEL FLIGHT PERFORMANCE

AEO PERFORMANCE

Equipment installed	FLM/ FMS	Max. horizontal speed			Economic cruise speed			Max. endurance	
		Speed KTAS	Fuel cons.	Range	Speed KTAS	Fuel cons.	Range	Fuel cons.	Endurance
Bleed air heating	FLM	–	+2%	–2%	–	+2%	–2%	+2%	–2%
Sandfilter sytem (filter mode)	9.2–22	Chart	+1.5%	–1.5%	–	+1.5%	–1.5%	+1%	–1%
Sandfilter system (bypass mode)	9.2–22	Chart	+1%	–1%	–	+1%	–1%	+0.5%	–0.5%
Sandfilter (IBF–System) (Sandfilter–NORM)	9.2–50	Chart	+3.5%	–3.5%	–	+2.5%	–2.5%	+1.5%	–1.5%
Sandfilter (IBF–System) (Sandfilter–OFF)	9.2–50	Chart	+2.5%	–2.5%	–	+2%	–2%	+1.5%	–1.5%
Cargo hook mirror	9.2–4	–3.5	–	–3%	–3.5	–	–3%	+1%	–1%
Emergency float. system	9.2–9	–4	–	–3%	–4	–	–3%	+1%	–1%
External hoist system	9.2–11	–2	–	–2%	–2	–	–2%	+0.5%	–0.5%
External loudspeakers	9.2–12	–3.5	–	–3%	–3.5	–	–3%	+1%	–1%
FLIR Ultraforce II	9.2–35	–2.5	–	–2%	–2.5	–	–2%	+0.5%	–0.5%
Searchlight SX–16(IR) LH	9.2–23	–6	–	–4.5%	–5.5	–	–4.5%	+1.5%	–1.5%
Open/removed doors	9.1–2	–5.5	+5.5% [1]	–5.5%	–	–	–	+1.5%	–1.5%

[1] increase of fuel consumption: only at V_{NE} limit for operation with open doors = 100 KIAS (Power required lower then AEO MCP)

NOTE All other optional equipment, presented in chapter 9.2, but not included in this lists, has no or negligible influence on the level flight performance.

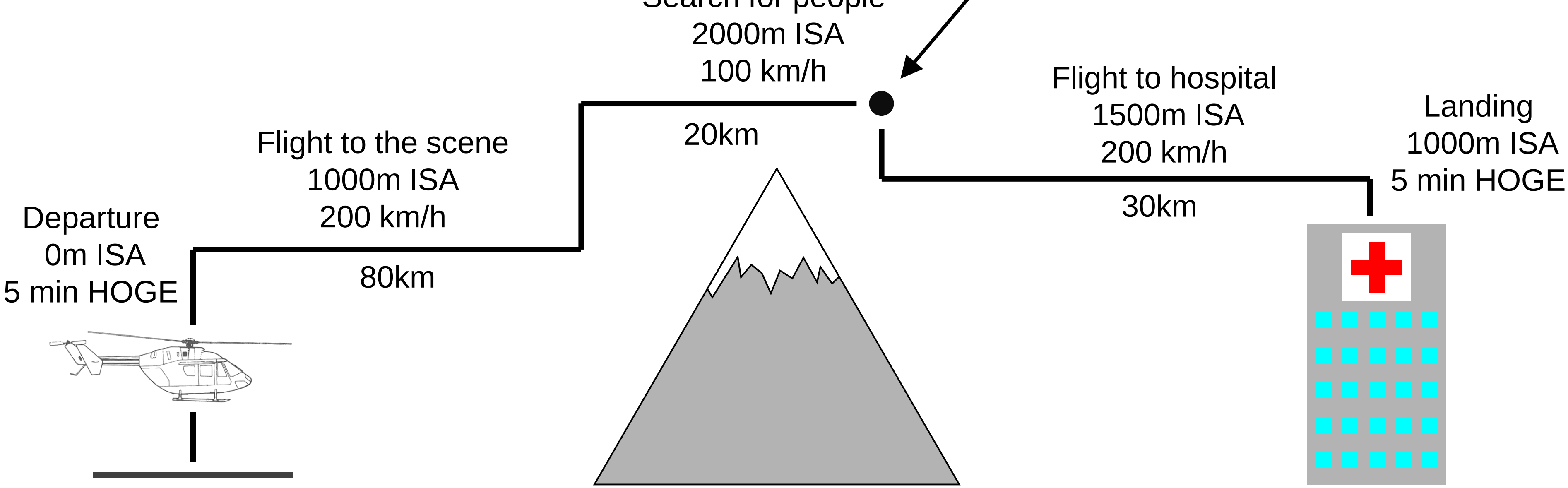
Figure: *Flight Manual BK 117 C-2*

3.5. Refined performance calculation

Mission example: Mountain rescue

Crew + Payload:

- 2 people paramedics
- An injured person is accommodated
- Medical equipment
- Rescue hoist
- 1 Pilot

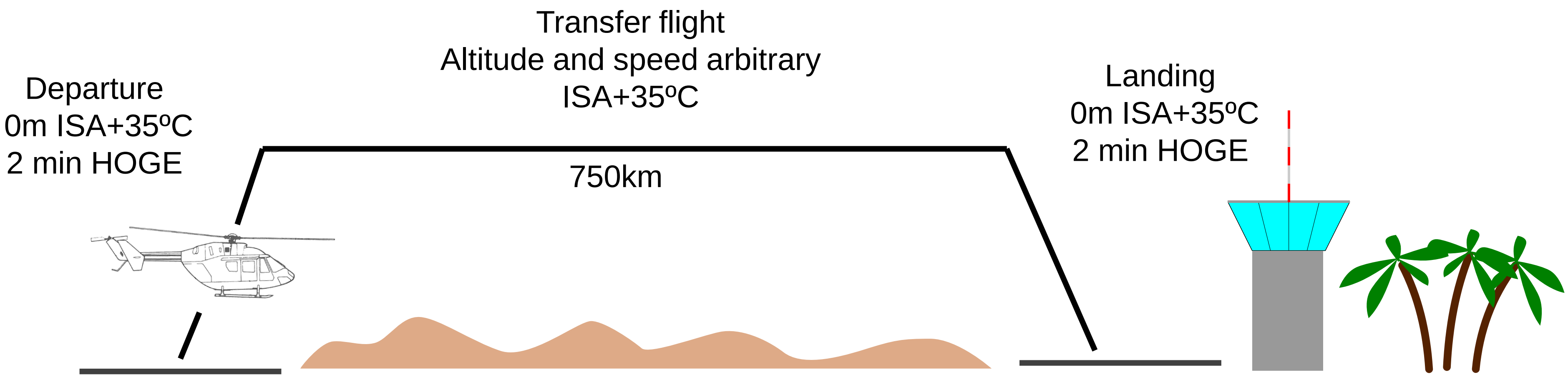


3.5. Refined performance calculation

Mission example: Transfer to desert region

Crew + Payload:

- 2 Pilots

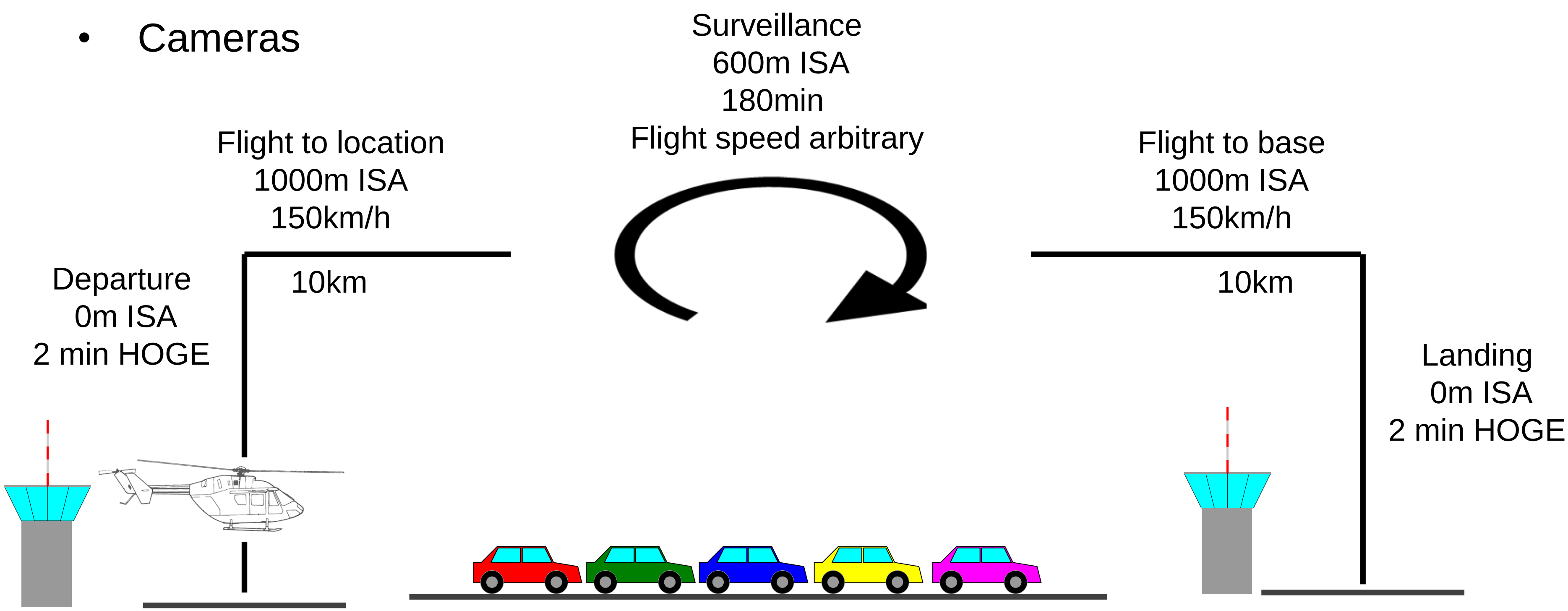


3.5. Refined performance calculation

Mission example: traffic supervision

<u>*Crew+Payload*:</u>

- 1 Pilot
- 1 Observer
- Cameras



Mission example: Operational Maneuver From the Sea (OMFTS)

Sikorsky CH-53K

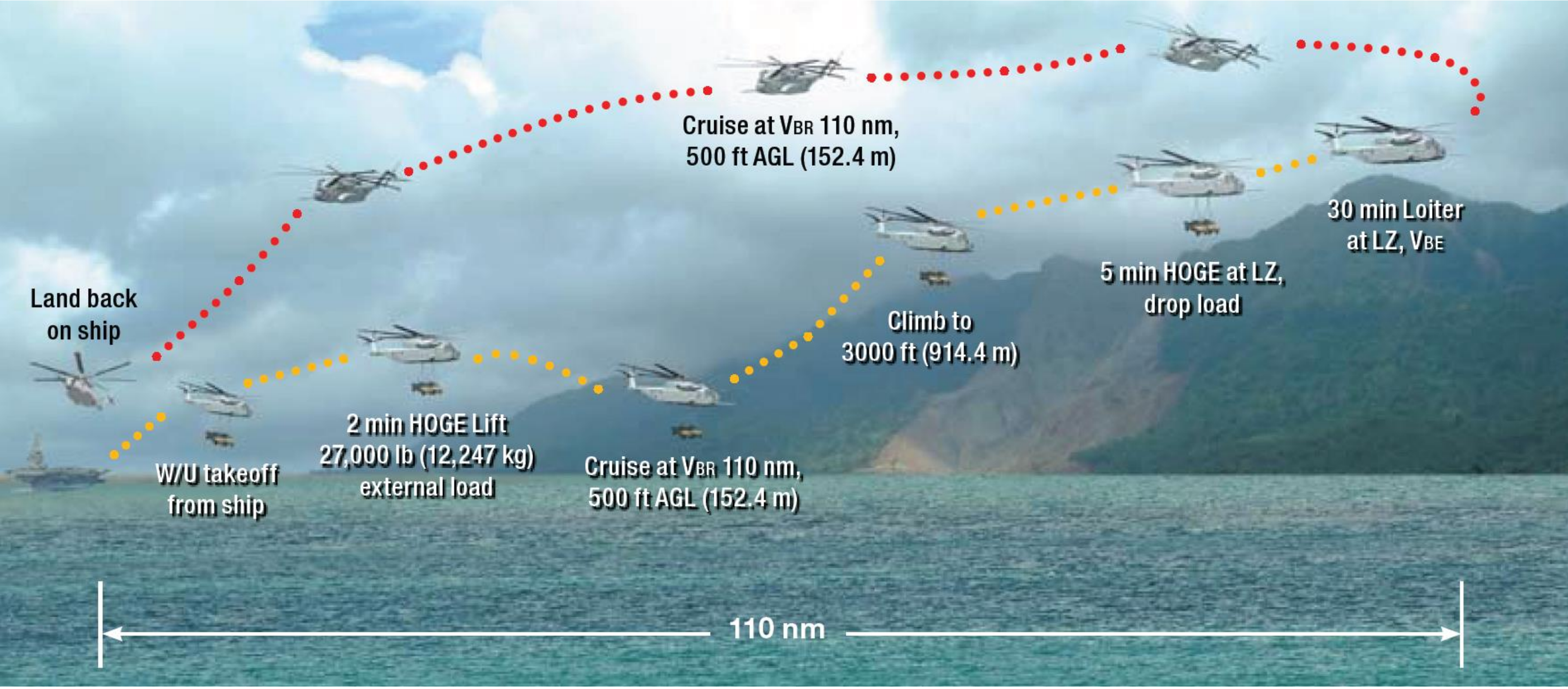


Figure: Sikorsky Aircraft Corporation

3.5. Refined performance calculation

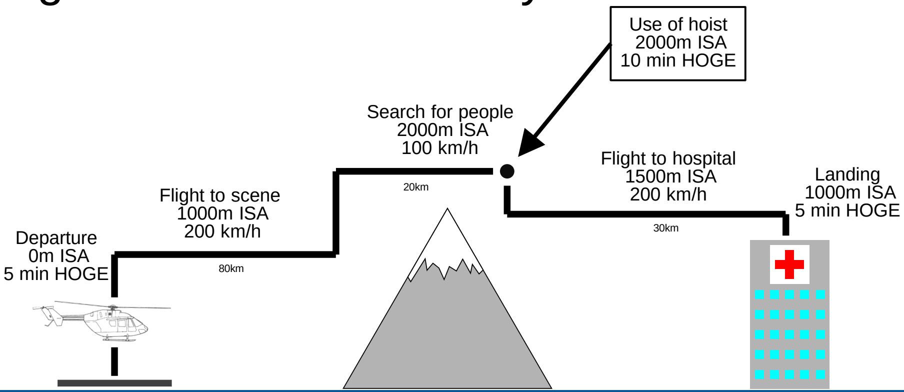
Refined mass estimation – Mission calculation

At the calculation of the single mission points, it needs to be considered that the required drive power and the fuel consumption depend on atmospheric conditions.

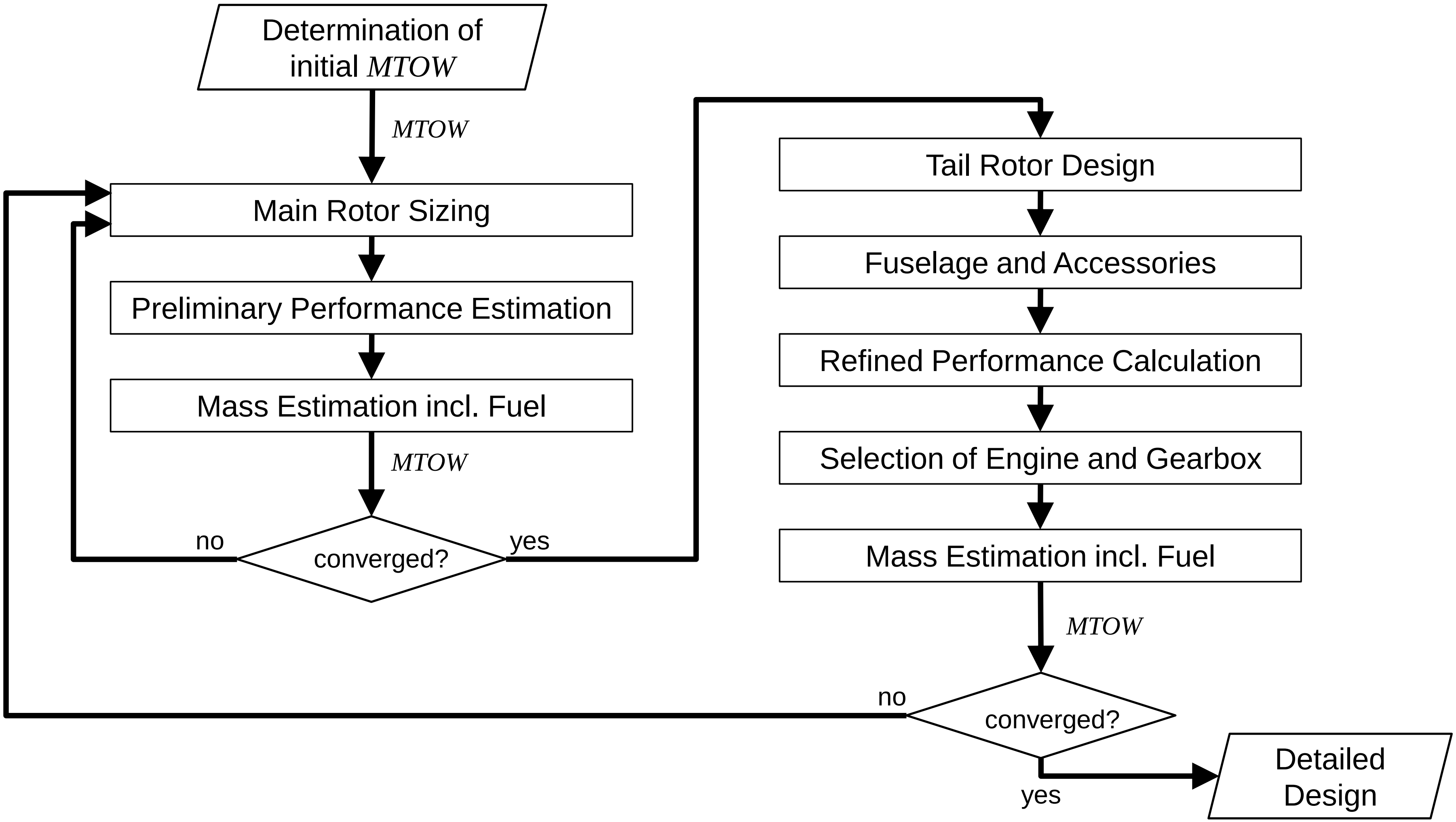
Furthermore the carried fuel mass changes permanently and thereby the total mass of the helicopter. This not only leads to a continuous change of the power requirement at a given flight condition, but also the speeds for best range and maximum endurance vary.

For the design a new maximum take off weight $MTOW$ is eventually determined from the single missions .

In case a converged solution was achieved, it can be moved on to the detailed design.



3 Sizing - Overview



4 Flight Performance

4 Flight Performance

After determining the design parameters along with the choice of engines and transmission, the flight performance characteristics of the design can be examined in greater detail.

The following chapters discuss in more detail the relationships:

4.1 Required drive power P in horizontal flight

4.2 Flight performance at available drive power P_{av}

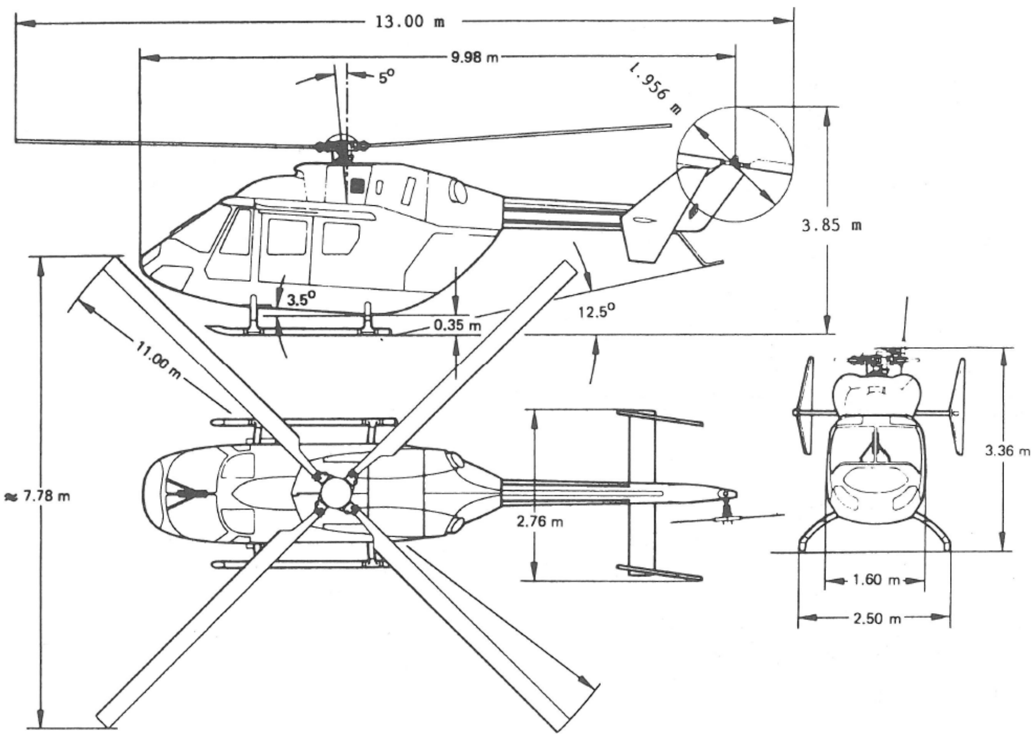
4.3 Endurance, range and load capacity

4.4 Carpet Plot: Design of a Personal Air Vehicle

A helicopter base model similar to the MBB/Kawasaki BK117 was chosen as basis for the discussion.

4 Flight Performance

Example: MBB/Kawasaki BK117



Tip speed (main and tail rotor)

$$V_{TIP} = 221\tfrac{m}{s} \qquad V_{TIP_{TR}} = 221\tfrac{m}{s}$$

Solidity (main and tail rotor)

$$\sigma = 0.074 \qquad \sigma_{TR} = 0.131$$

Gross weight

$$GW = 3200\,kg$$

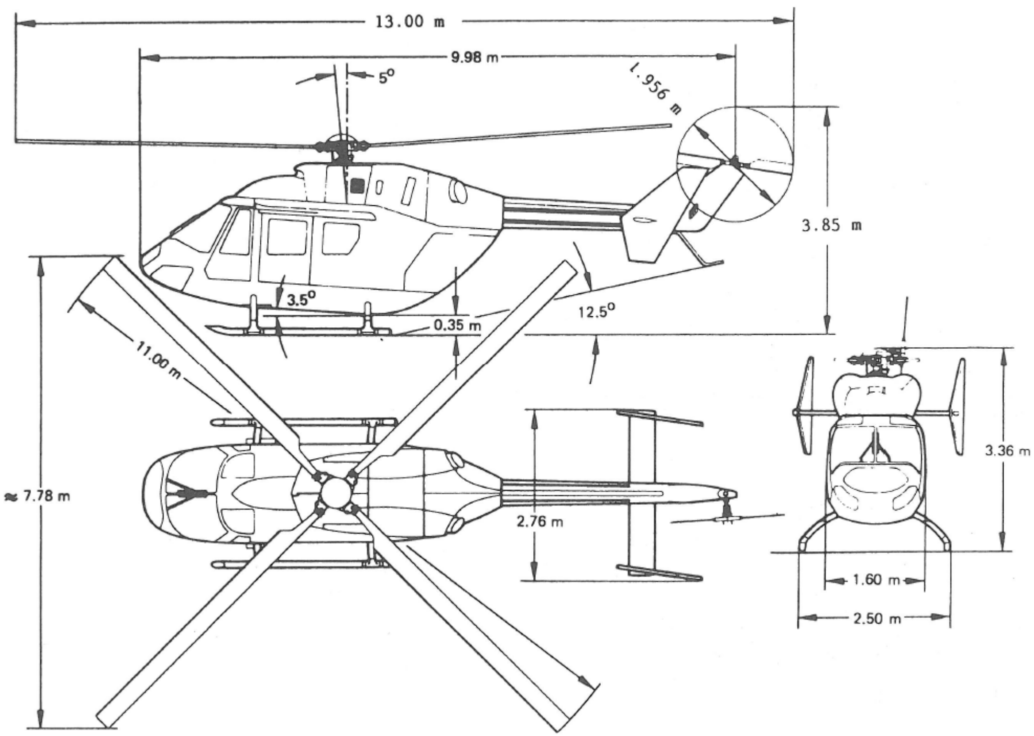
Atmosphere

$$0m\ ISA$$

Data from Prouty: *Helicopter Performance, Stability and Control*; Leishman: *Principles of Helicopter Aerodynamics*

4 Flight Performance

Example: MBB/Kawasaki BK117



Assumptions:

Empirical correction factors

$\kappa = 1{,}15$ $\kappa_{TR} = 1{,}3$

Zero-lift drag coefficient

$C_{d0} = 0{,}011$ $C_{d0_{TR}} = 0{,}011$

Total drag area

$C_D S = 1{,}25\, m^2$

Fuselage download factor

$k_{DL} = 4\%$

Main rotor transmission efficiency

$\eta_t = 98\%$

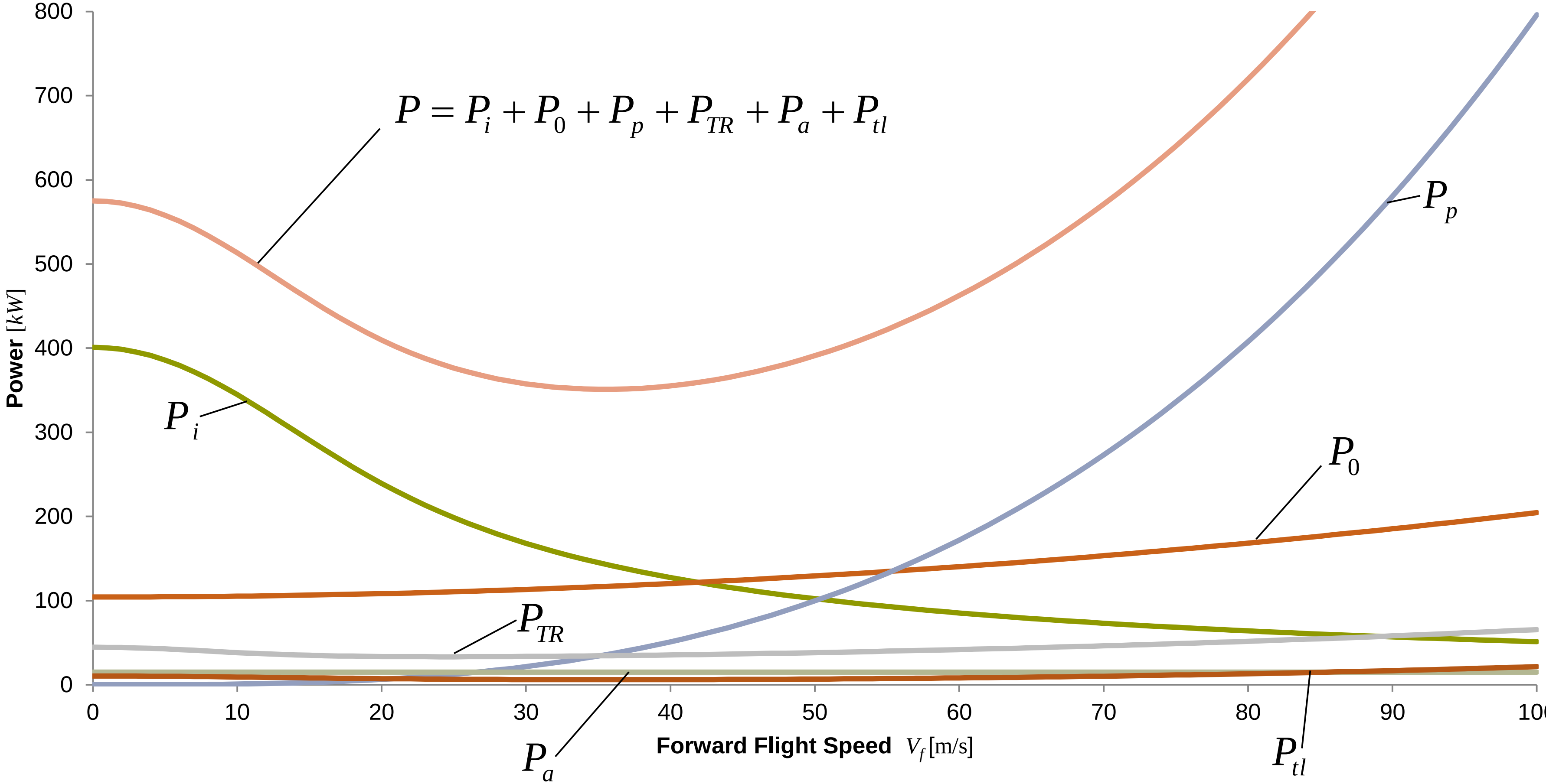
Accessory power

$P_a = 15\, kW$

Figure from Prouty: *Helicopter Performance, Stability and Control*

4 Flight Performance

Total power in horizontal flight



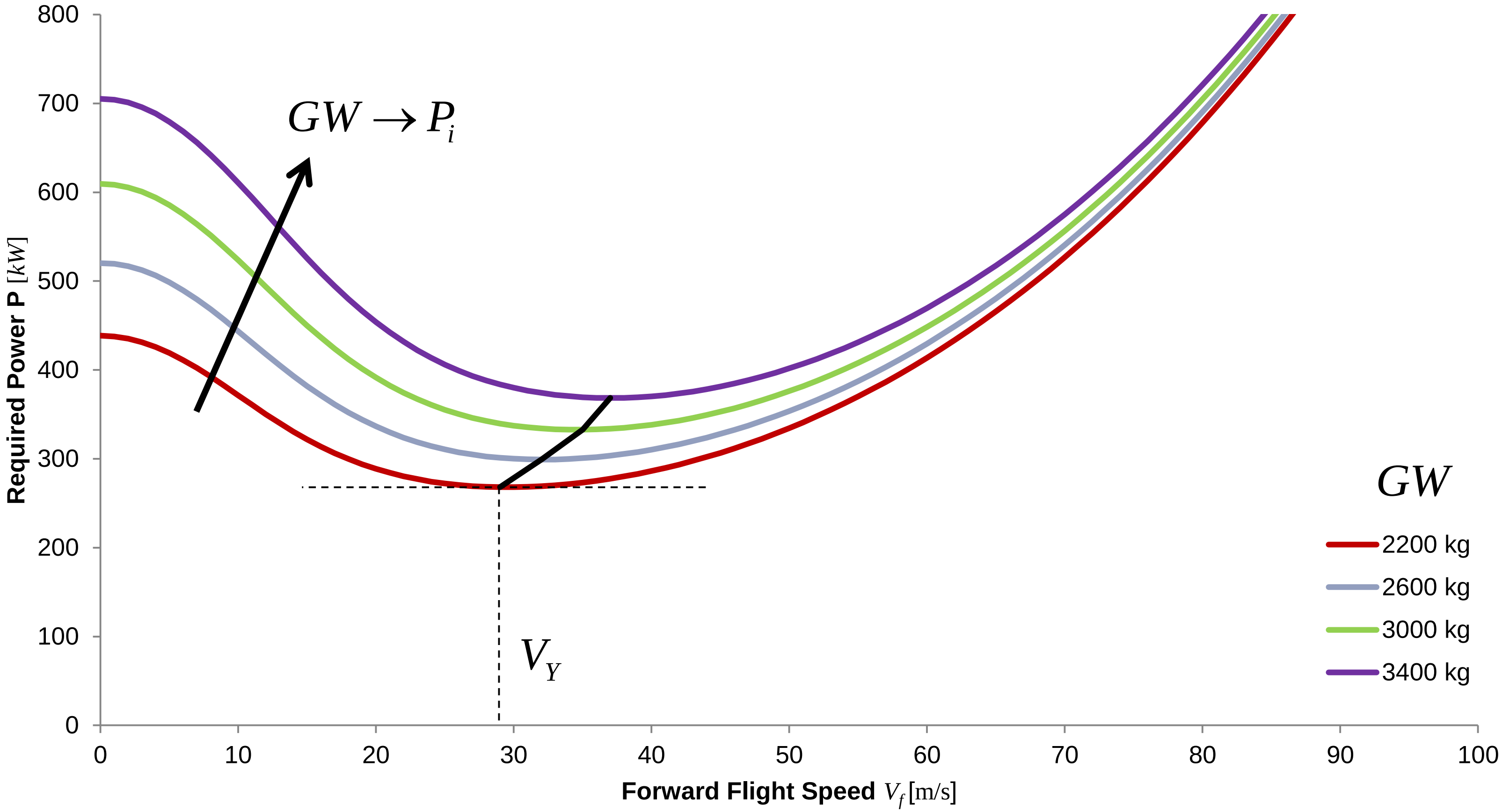
4 Flight Performance

4.1 Required drive power P in horizontal flight

- ➔ Influence of gross weight GW
- Influence of air density
- Influence of parasitic drag
- Flight manual diagram

4.1 Required drive power P in horizontal flight

Influence from gross weight GW



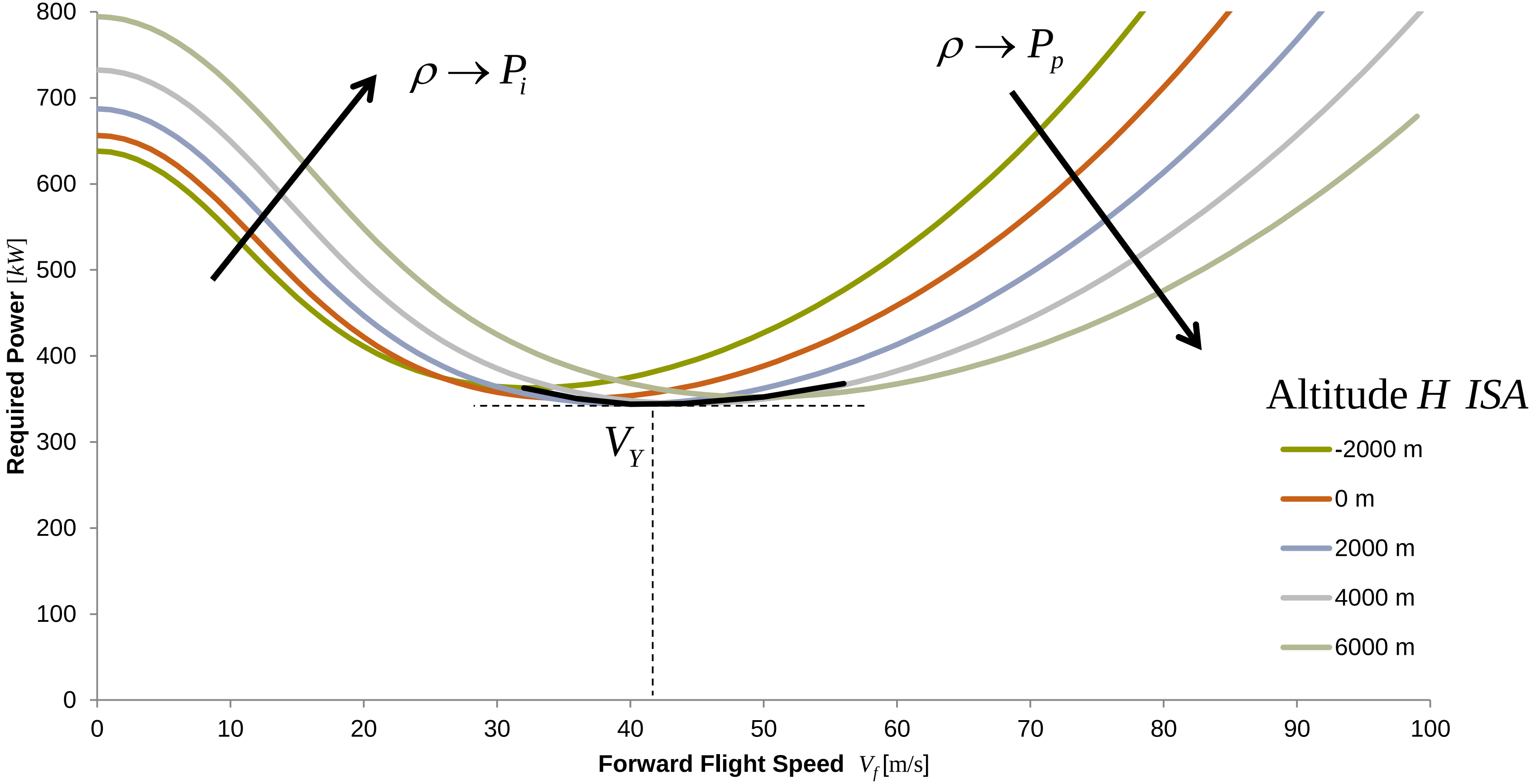
4 Flight Performance

4.1 Required drive power P in horizontal flight

- ✓ Influence of gross weight GW
- ➔ Influence of air density
- Influence of parasitic drag
- Flight manual diagram

4.1 Required drive power P in horizontal flight

Influence of air density ρ



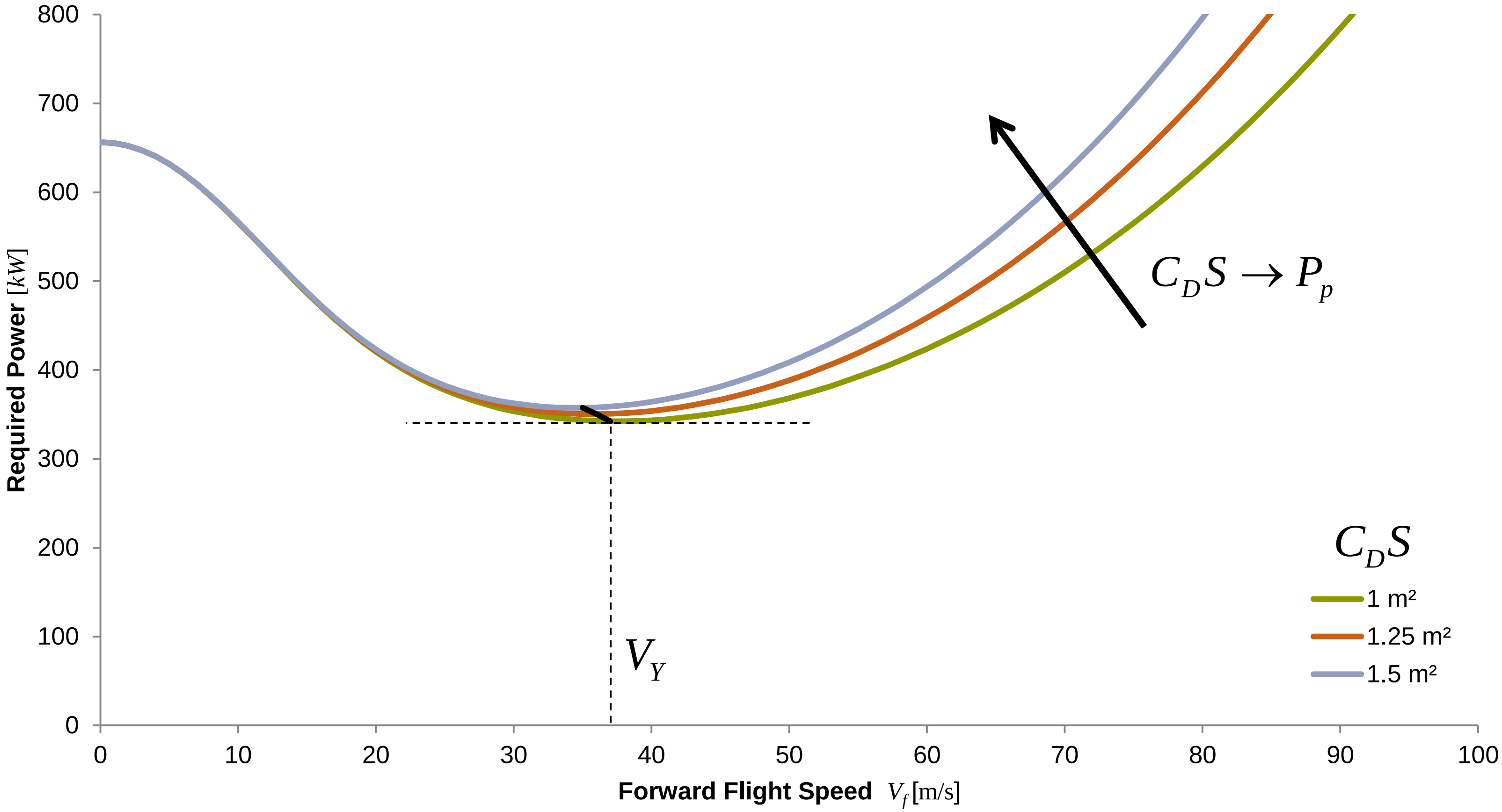
4 Flight Performance

4.1 Required drive power P in horizontal flight

- ✓ Influence of gross weight GW
- ✓ Influence of air density
- ➔ Influence of parasitic drag
- Flight manual diagram

4.1 Required drive power P in horizontal flight

Influence of parasitic drag area $C_D S$



4 Flight Performance

4.1 Required drive power P in horizontal flight

- ✓ Influence of gross weight GW
- ✓ Influence of air density
- ✓ Influence of parasitic drag
- ➔ Flight manual diagram

4.1 Required drive power *P* in horizontal flight

Flight manual diagram

Showing the required engine torque depending on:

- Gross mass
- Flight speed
- Ambient temperature
- Pressure altitude

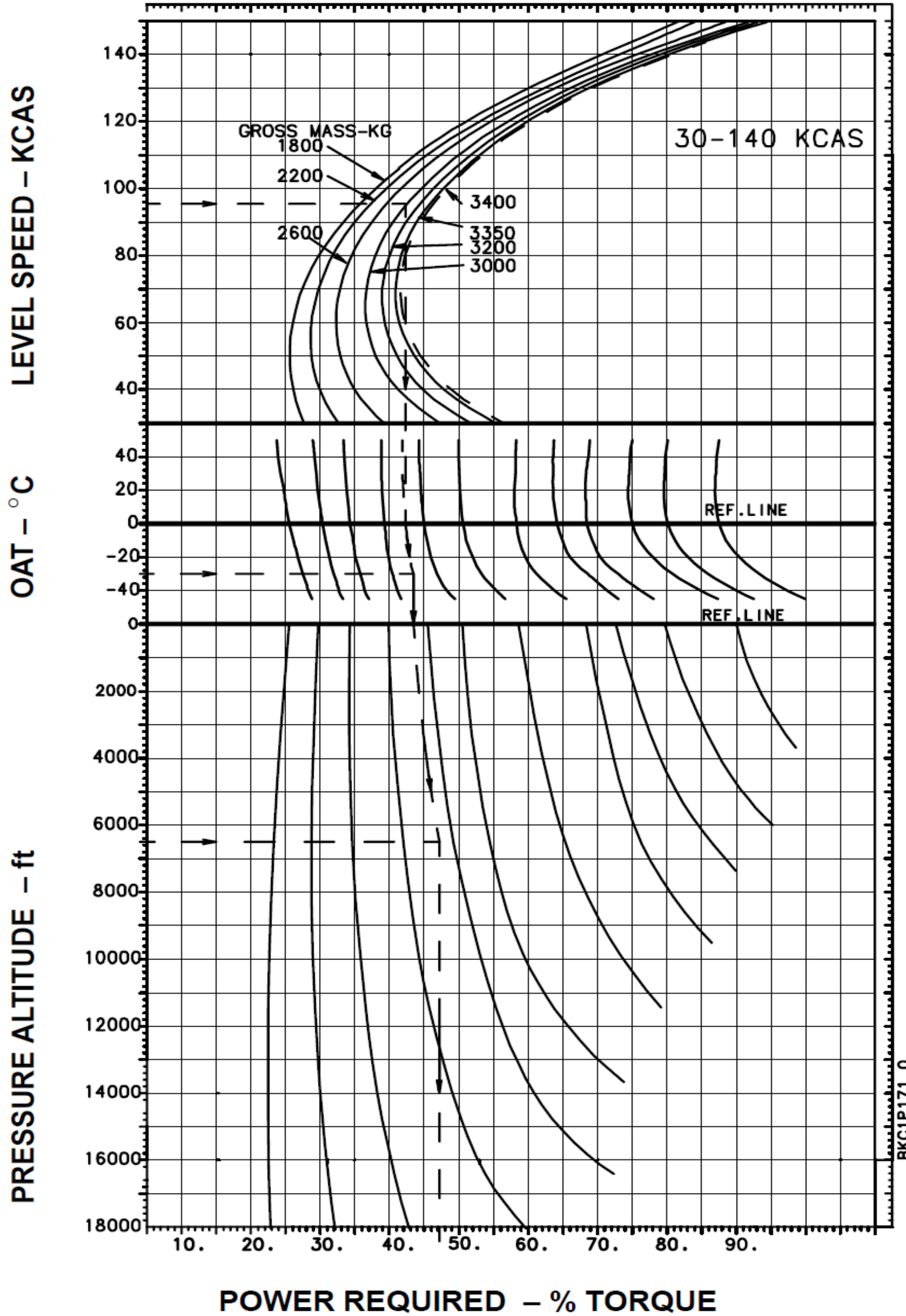


Abbildung aus *Eurocopter Flight Manual BK117 C-1*

4 Flight Performance

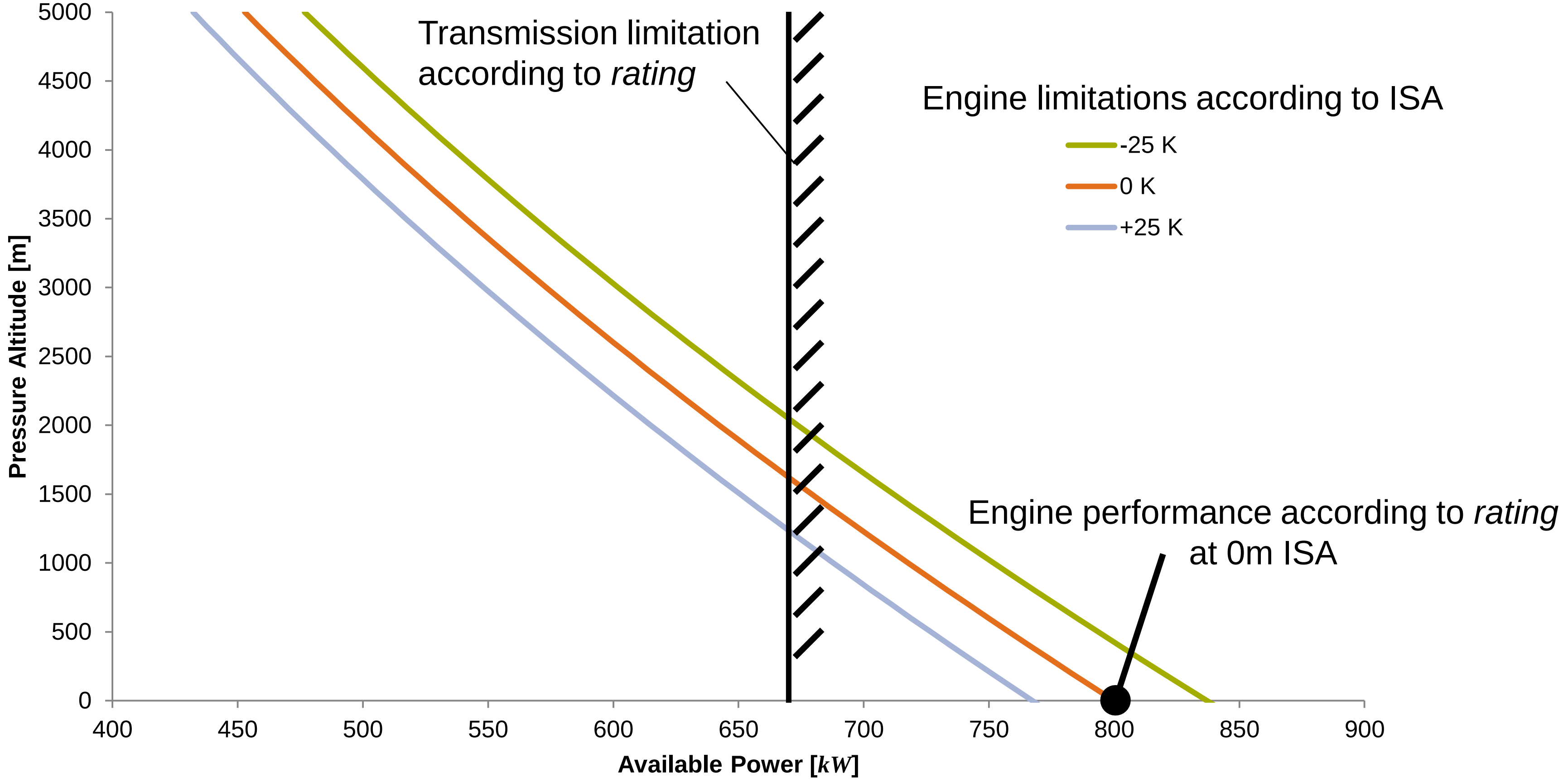
4.2 Flight performance at available drive power P_{av}

➔ P_{av} depending on environmental conditions

- Hover ceiling
- Climb performance
- Ceiling
- Top speed in horizontal flight

4.2 Flight performance at available drive power P_{av}

P_{av} depending on environmental conditions



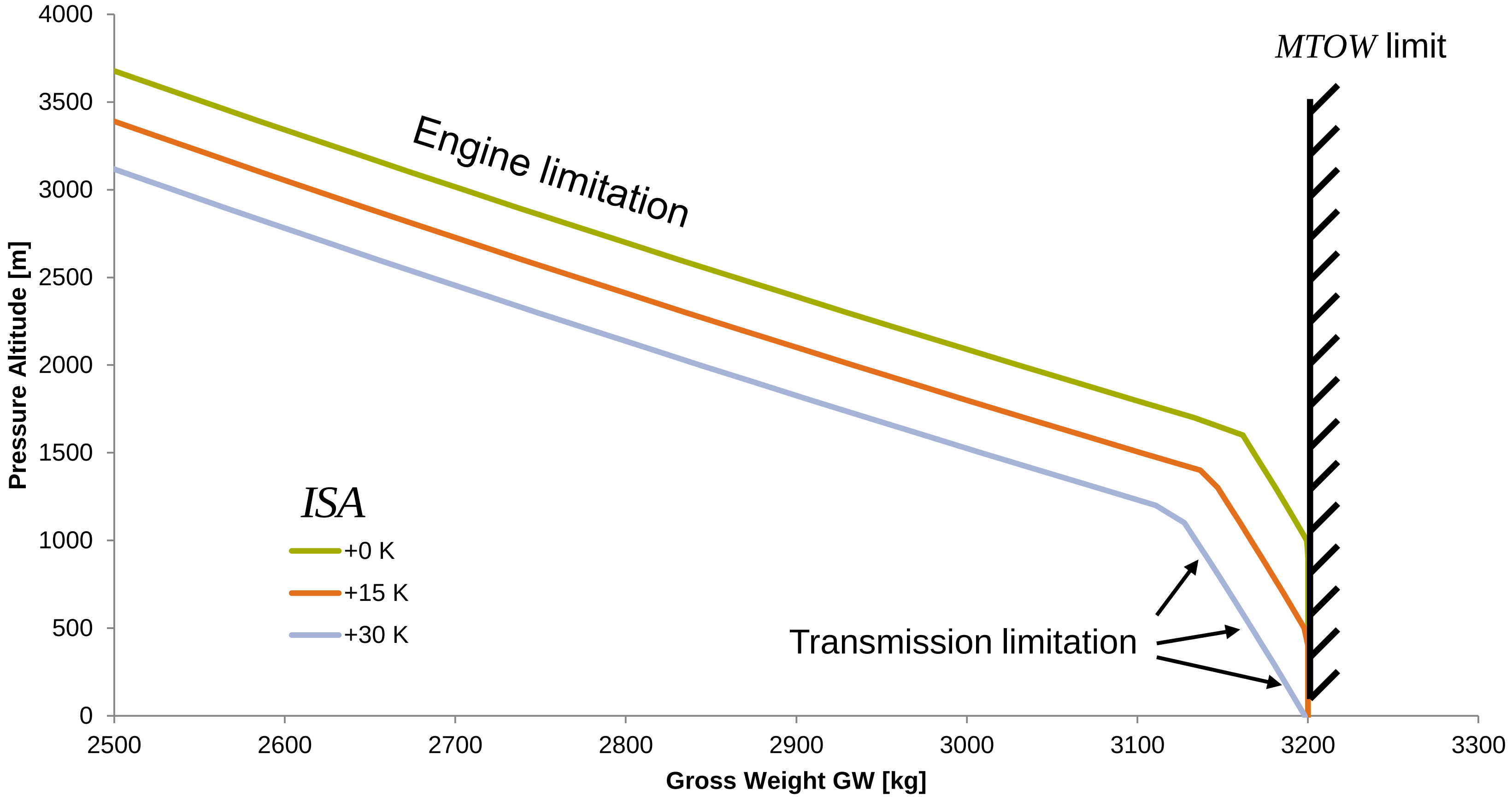
4 Flight Performance

4.2 Flight performance at available drive power P_{av}

- ✓ P_{av} depending on environmental conditions
- ➔ Hover ceiling
- Climb performance
- Ceiling
- Top speed in horizontal flight

4.2 Flight performance at available drive power P_{av}

Hover ceiling



4.2 Flight performance at available drive power P_{av}

Hover ceiling

Discontinuity due to variations of rotor rotational speed
➔ Varying available Transmission capacity at constant torque

Pressure Altitude -ft-	OAT - °C –45	–35	–30	–20	–10	0	+10	+20	+30	+40	+50
0	98.5	98.5	98.5	98.5	98.5	98.5	98.5	98.5	98.5	98.5	98.5
2000	98.5	98.5	98.5	98.5	98.5	98.5	98.5	98.5	98.5	99.5	100.0
4000	98.5	98.5	98.5	98.5	98.5	98.5	99.0	99.5	100.0	101.0	
6000	98.5	98.5	98.5	98.5	99.0	99.5	100.0	101.0	101.5	101.5	
8000	98.5	98.5	99.0	99.5	100.0	101.0	101.5	101.5	101.5		
10000	99.0	100.0	100.5	101.0	101.5	101.5	101.5	101.5	101.5		
12000	100.5	101.5	101.5	101.5	101.5	101.5	101.5	101.5	101.5		
14000	101.5	101.5	101.5	101.5	101.5	101.5	101.5	101.5			
18000	101.5	101.5	101.5	101.5	101.5	101.5	101.5	101.5			

N_{RO} - %

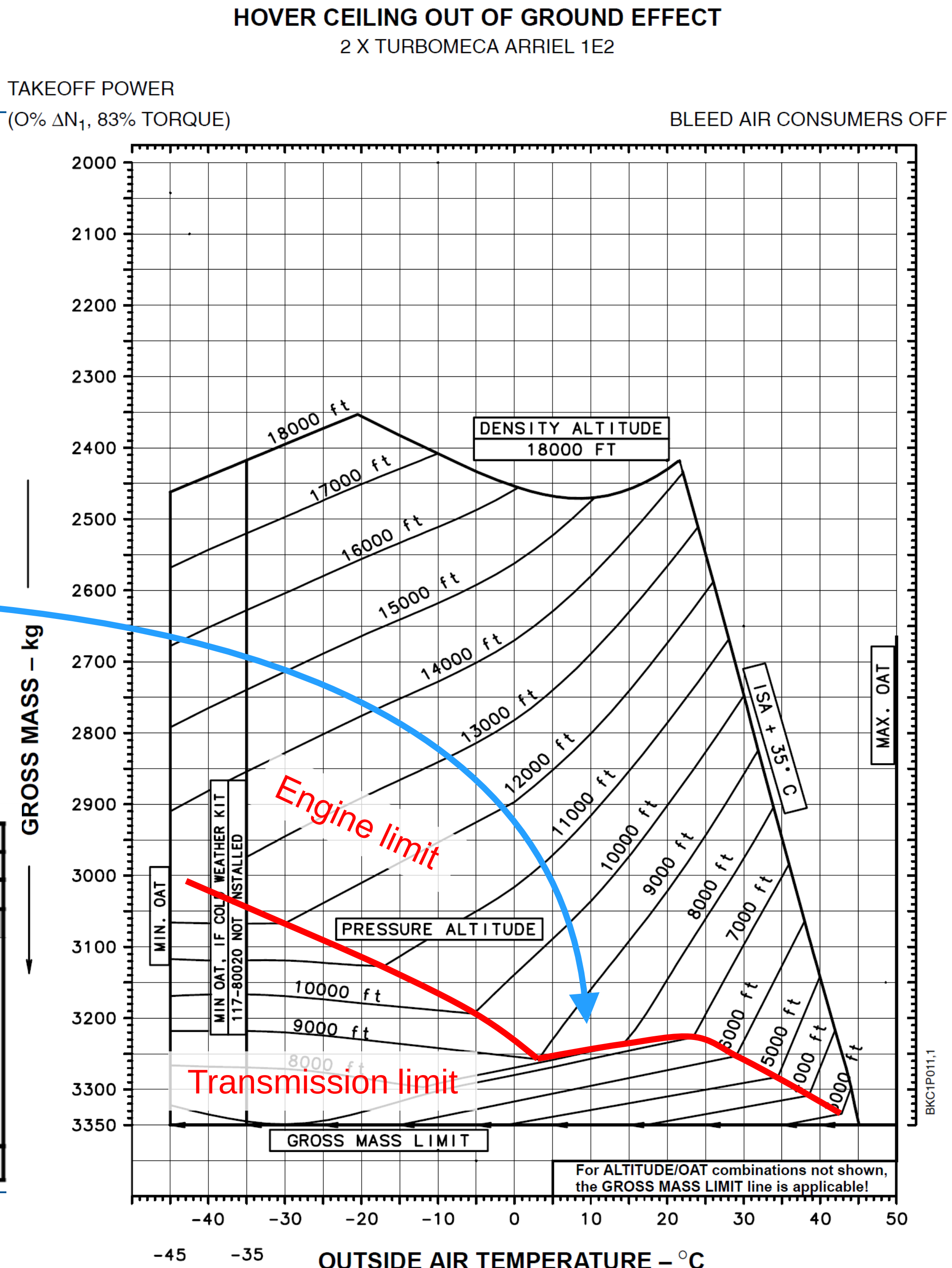


Abbildung aus *Eurocopter Flight Manual BK117 C-1*

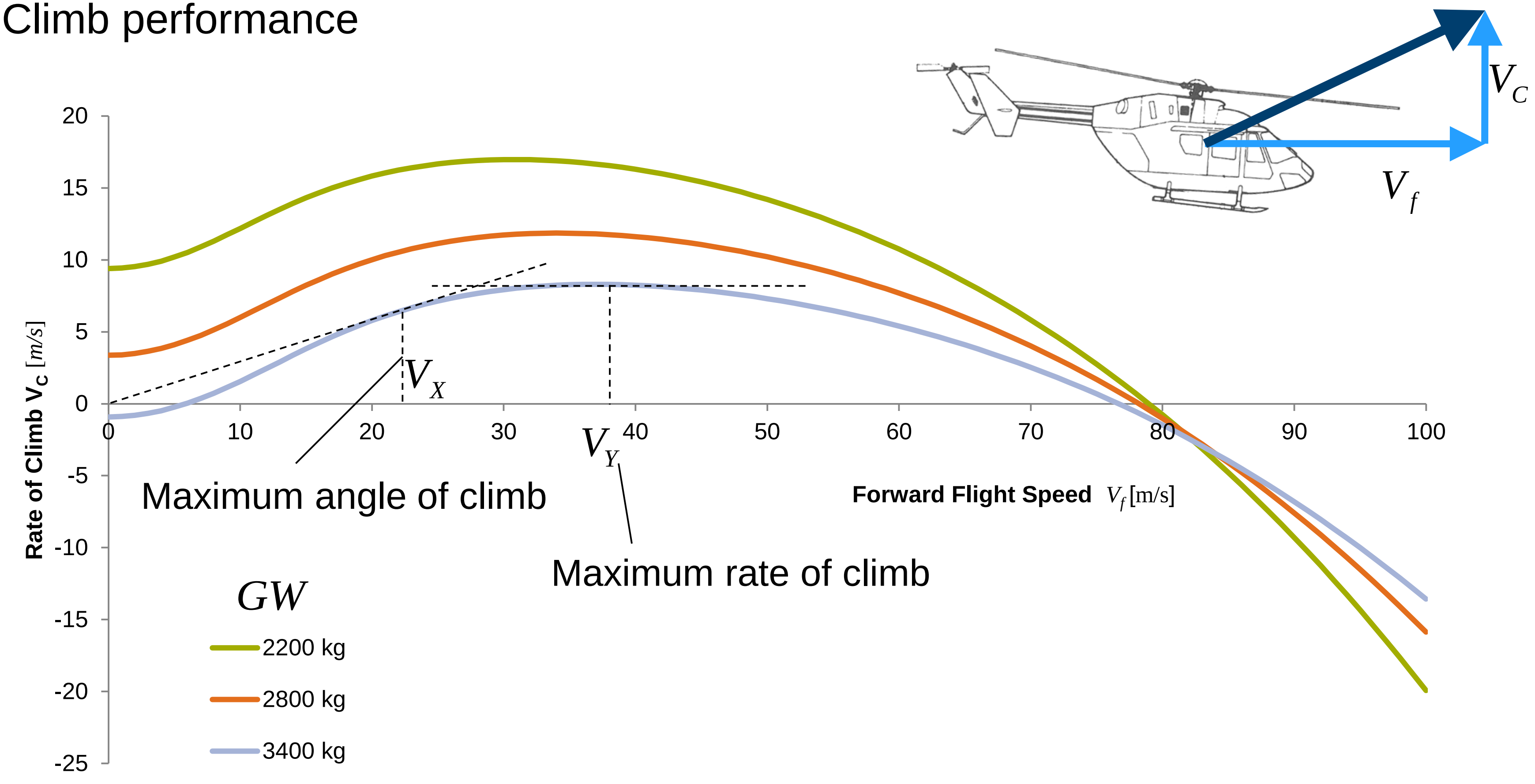
4 Flight Performance

4.2 Flight performance at available drive power P_{av}

- ✓ P_{av} depending on environmental conditions
- ✓ Hover ceiling
- ➔ Climb performance
- Ceiling
- Top speed in horizontal flight

4.2 Flight performance at available drive power P_{av}

Climb performance



4.2 Flight performance at available drive power P_{av}

Flight manual diagram

Showing maximal rate of climb at V_Y depending on:

- Gross mass
- Ambient temperature
- Pressure altitude

RATE OF CLIMB

2 X TURBOMECA ARRIEL 1E2

MAXIMUM CONTINUOUS POWER (-1.7% ΔN_1, 71% TORQUE)

V_Y = 65 KIAS

BLEED AIR CONSUMERS OFF OR ON

NOTE FOR OPERATION BELOW -35 °C OAT THE COLD WEATHER KIT (BK117-80020) SHALL BE INSTALLED AND OPERATIONAL (SEE ALSO SECTION 2)

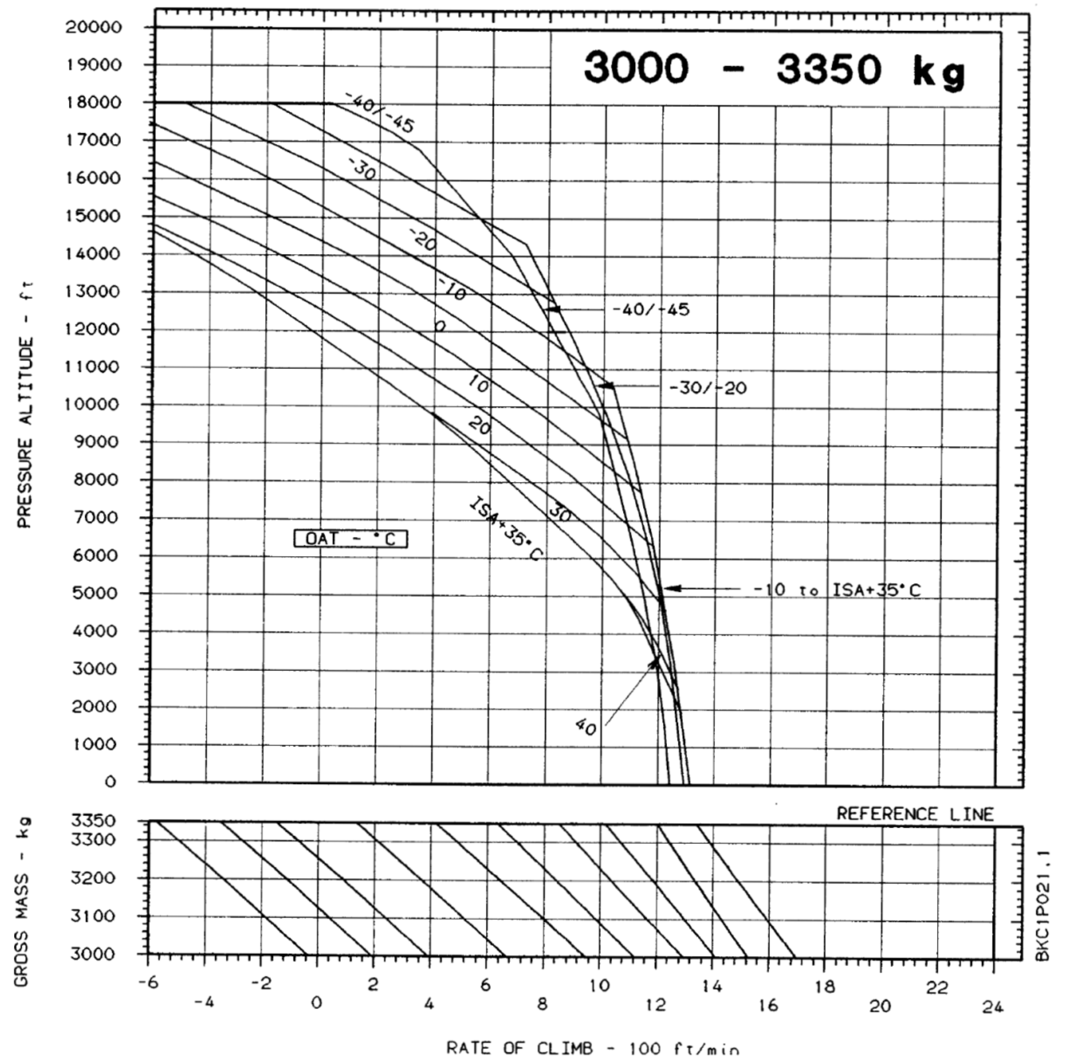


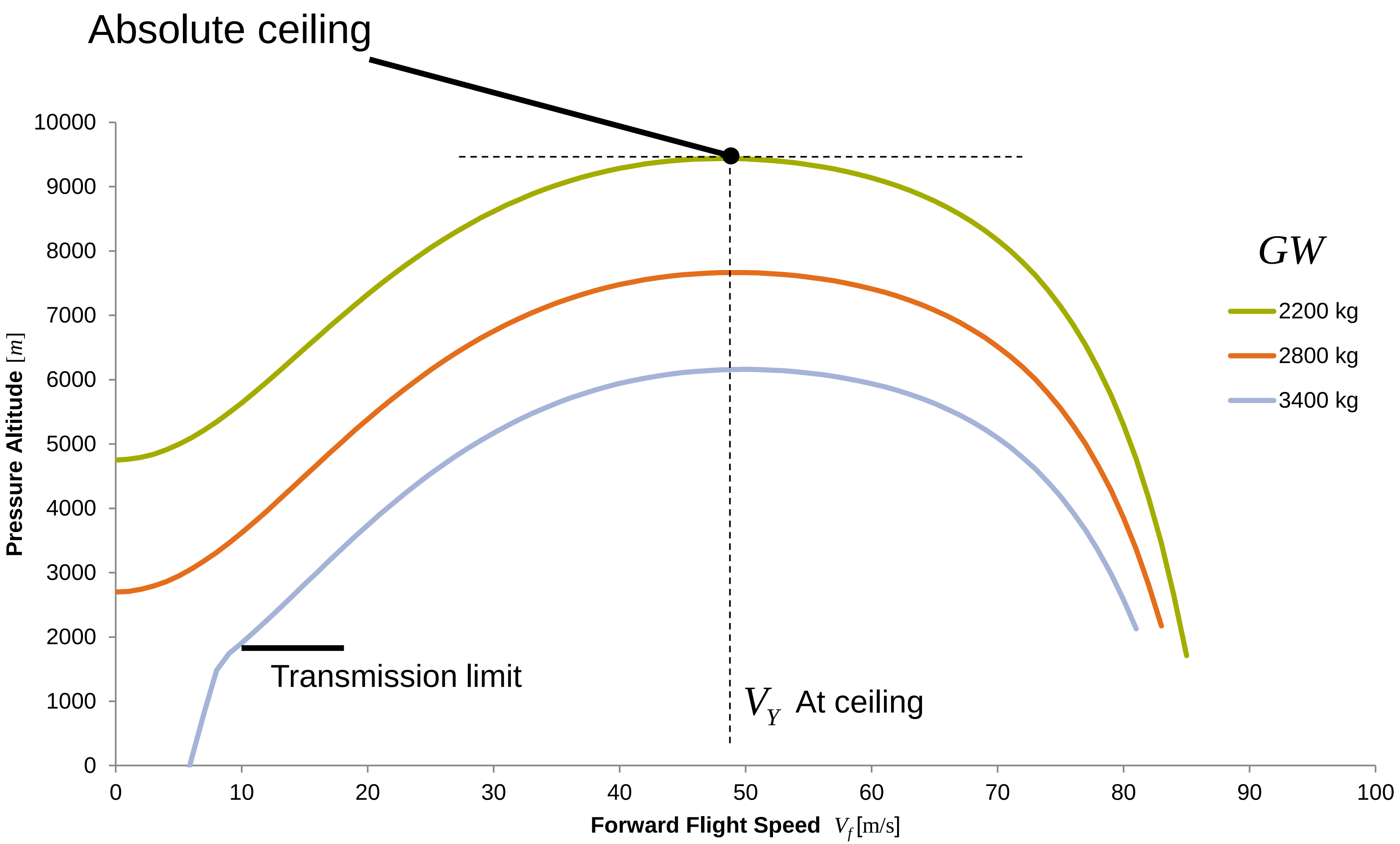
Abbildung aus *Eurocopter Flight Manual BK117 C-1*

4 Flight Performance

4.2 Flight performance at available drive power P_{av}

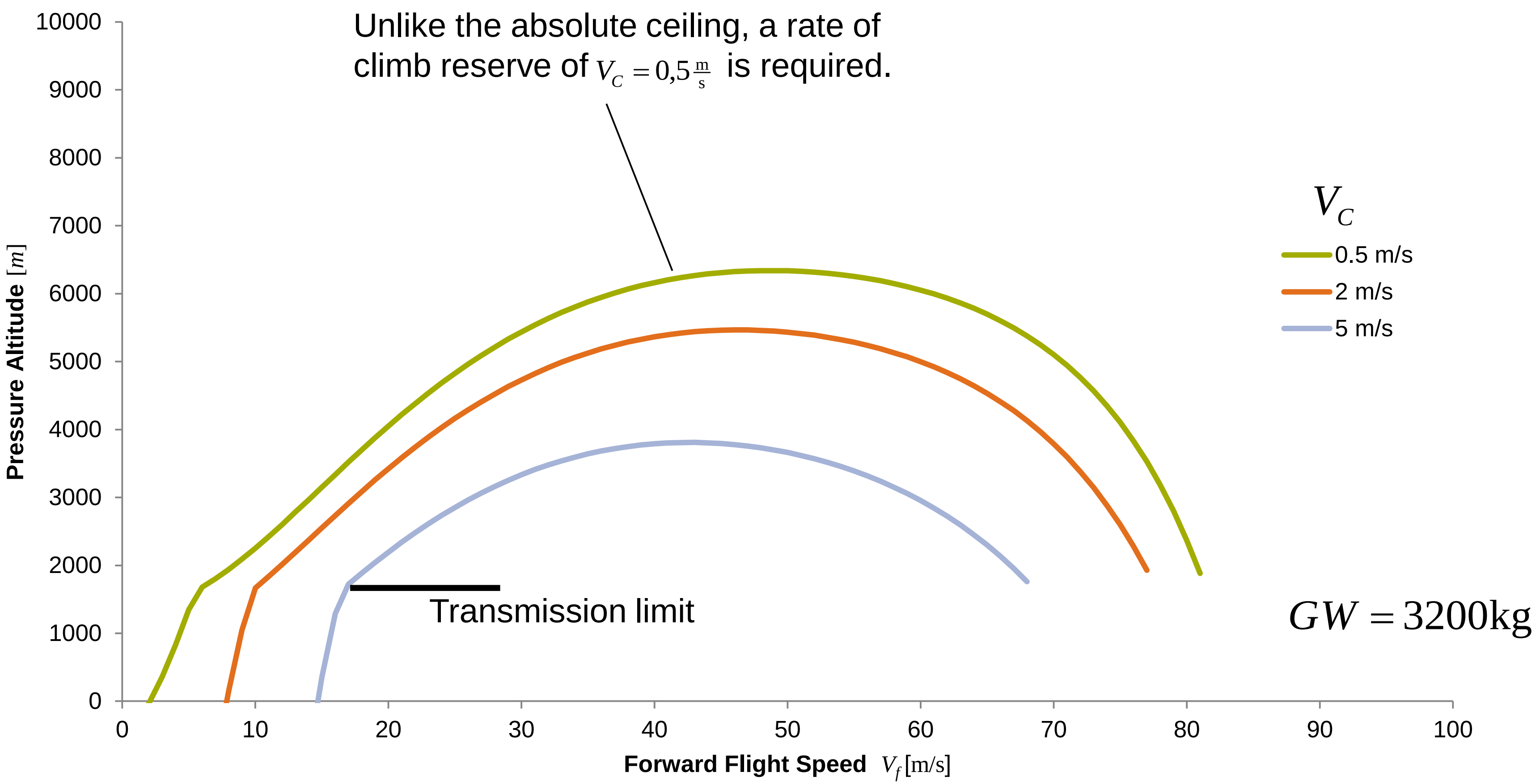
- ✓ P_{av} depending on environmental conditions
- ✓ Hover ceiling
- ✓ Climb performance
- ➔ Ceiling
- • Top speed in horizontal flight

4.2 Flight performance at available drive power P_{av}



4.2 Flight performance at available drive power P_{av}

Service ceiling



4 Flight Performance

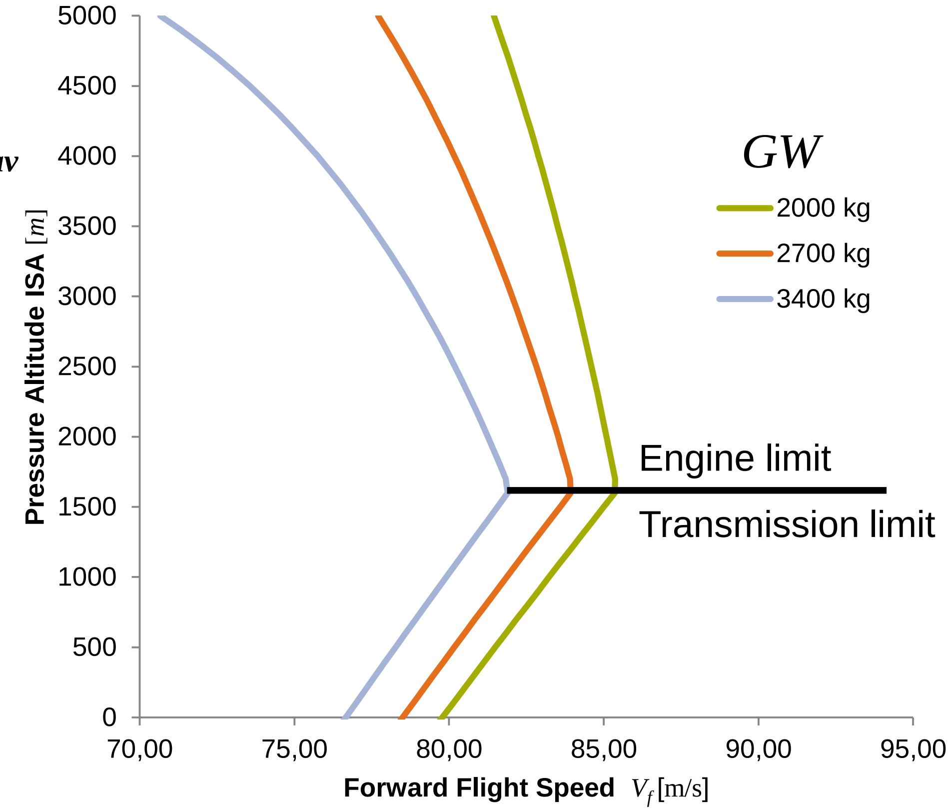
4.2 Flight performance at available drive power P_{av}

- ✓ P_{av} depending on environmental conditions
- ✓ Hover ceiling
- ✓ Climb performance
- ✓ Ceiling
- ➔ Top speed in horizontal flight

4.2 Flight performance at available drive power P_{av}

Top speed in horizontal flight at P_{av}

A simple depiction without considering compressibility and stall effects at the rotor.



4.2 Flight performance at available drive power P_{av}

Flight manual diagram

Showing top cruising speed at ISA conditions and *MCP* depending on:

- Gross mass
- Pressure altitude

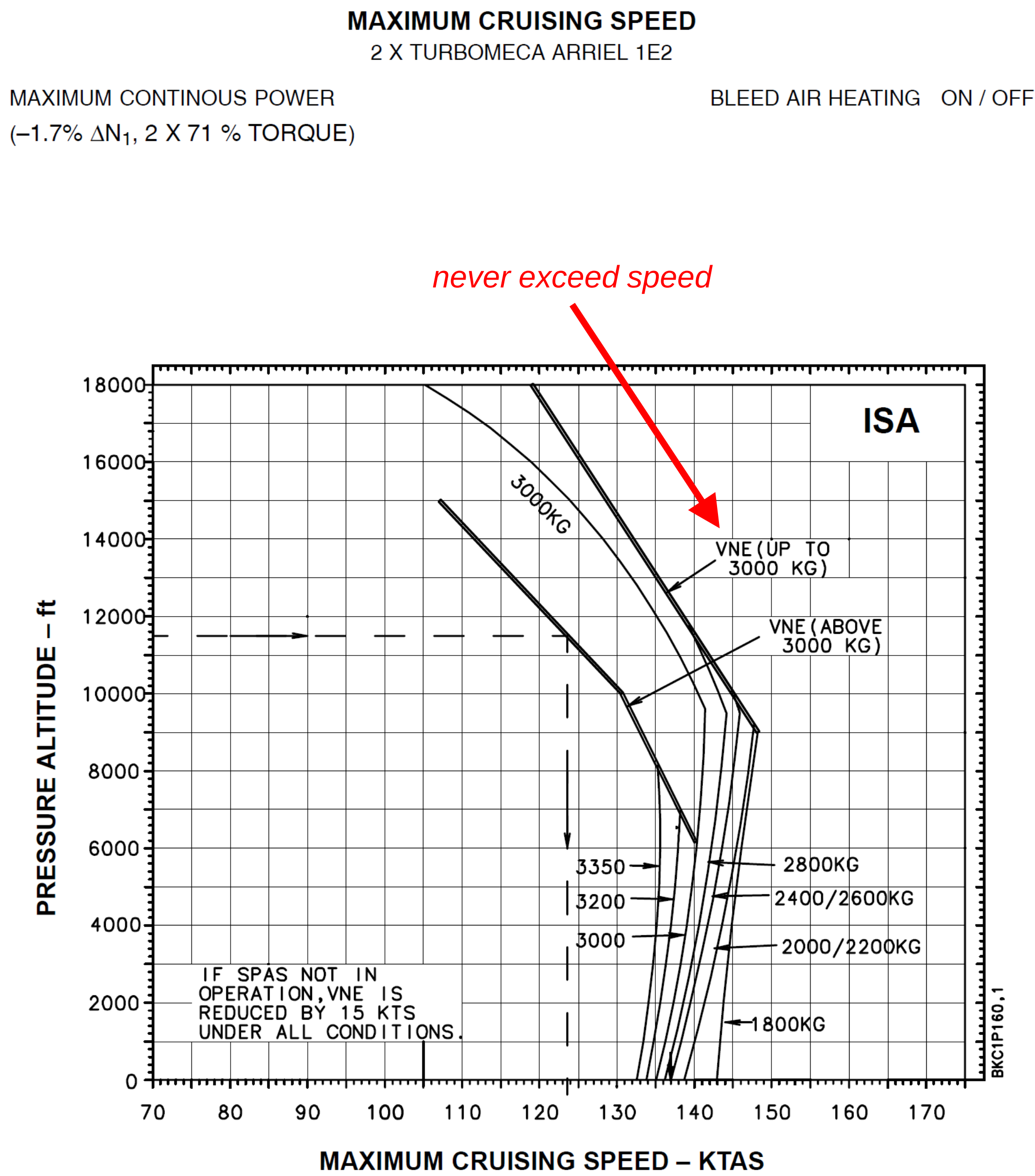


Abbildung aus *Eurocopter Flight Manual BK117 C-1*

4 Flight Performance

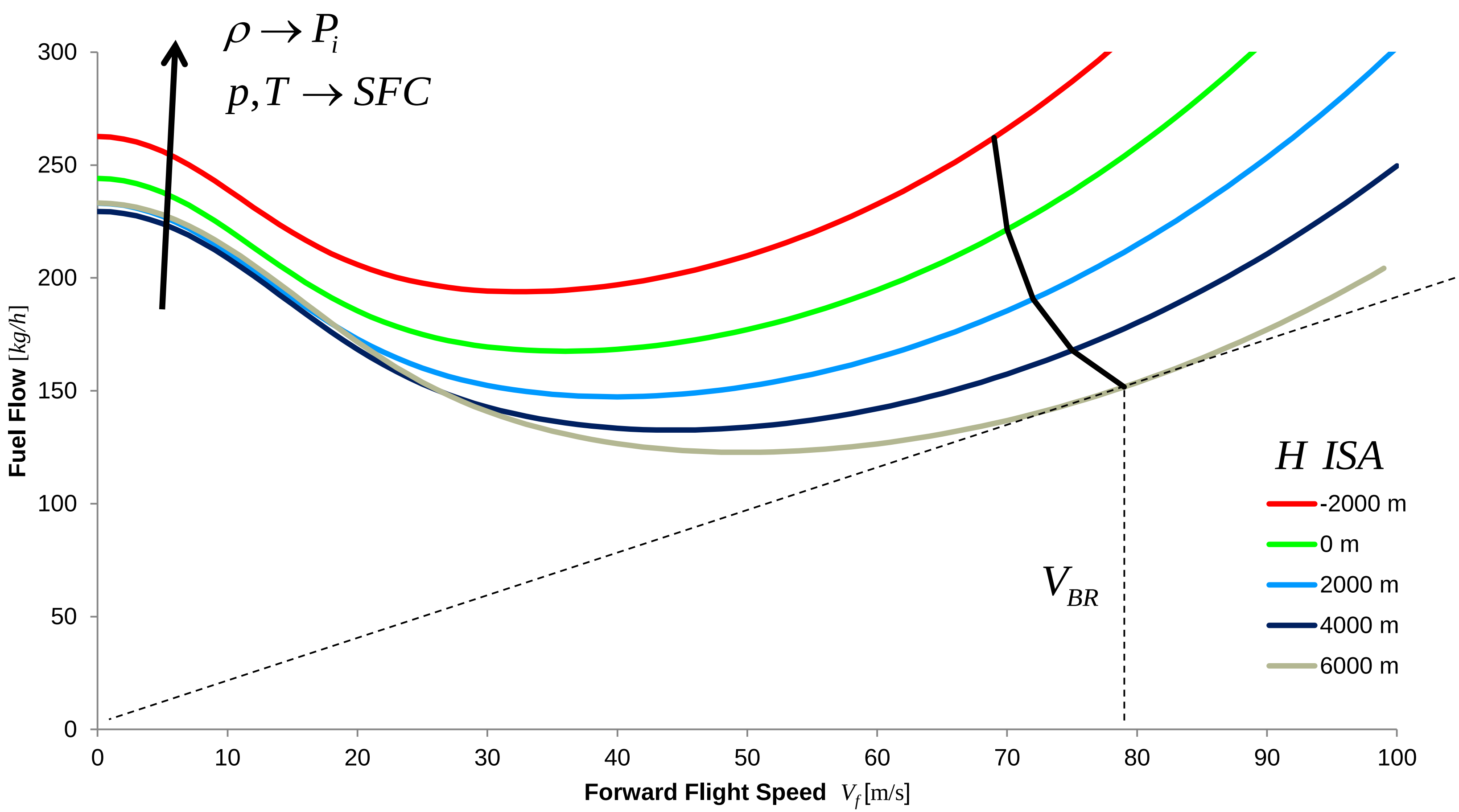
4.3 Endurance, range and load capacity

➔ Fuel consumption in horizontal flight

- Range
- Endurance

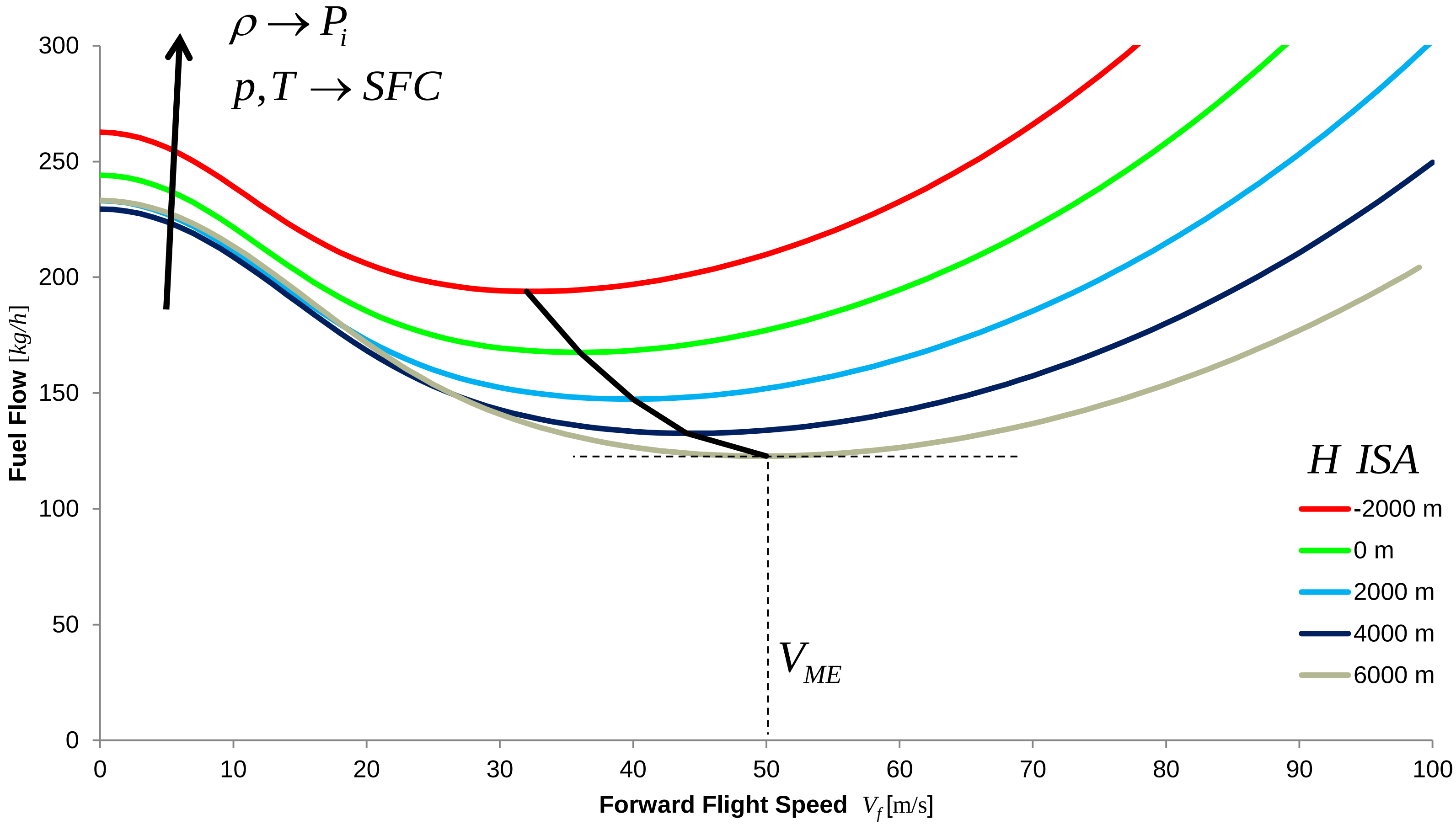
4.3 Endurance, range and load capacity

Horizontal flight: Influence of altitude H on fuel consumption



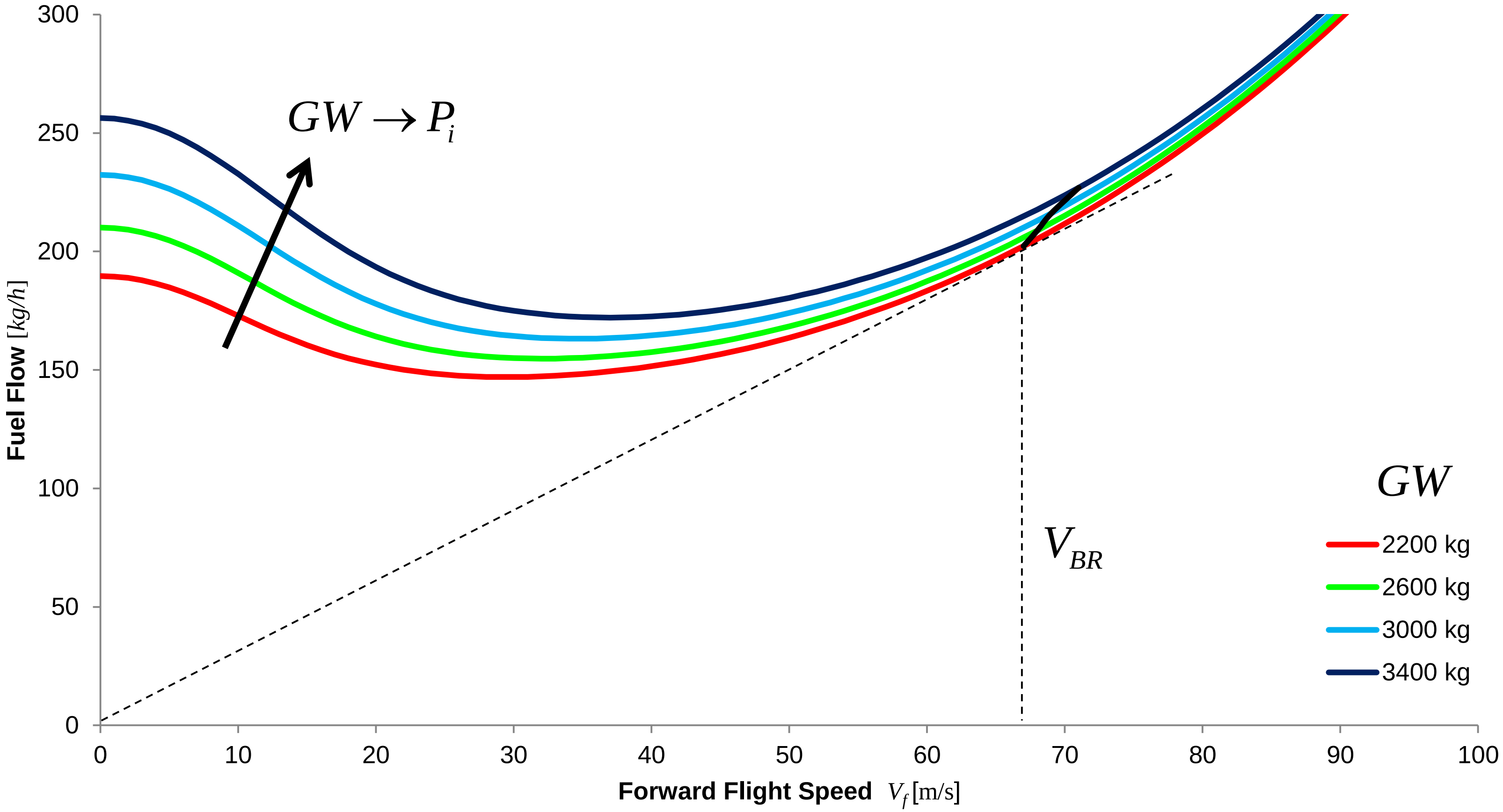
4.3 Endurance, range and load capacity

Horizontal flight: Influence of altitude H on fuel consumption



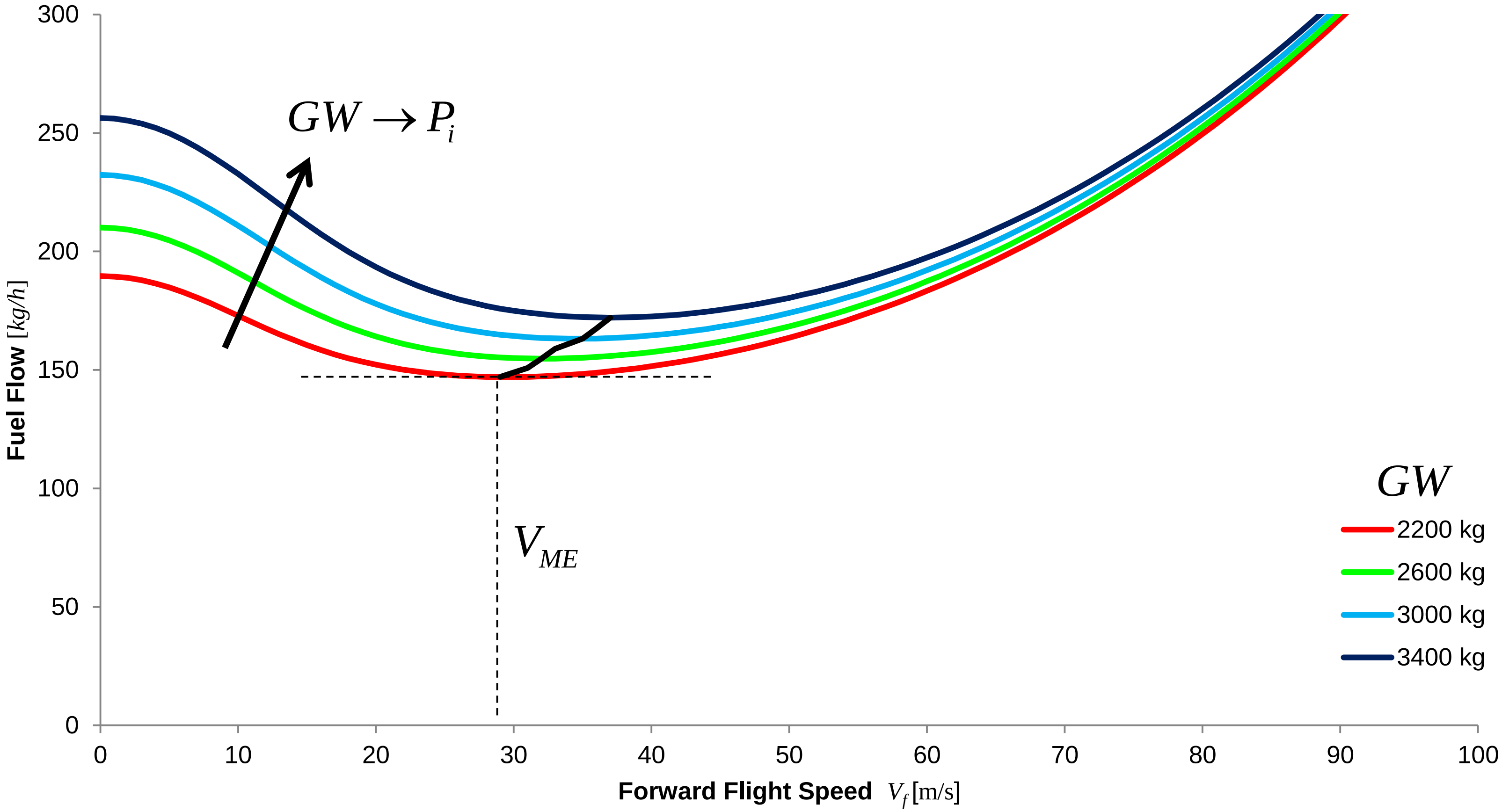
4.3 Endurance, range and load capacity

<u>Horizontal flight:</u> Influence of gross weight ***GW*** on fuel consumption



4.3 Endurance, range and load capacity

<u>Horizontal flight:</u> Influence of gross weight GW on fuel consumption



4.3 Endurance, range and load capacity

Flight manual diagram

shows fuel consumption depending on:

- Required torque
  ➔ Required Power
- Ambient temperature
- Pressure altitude

Engine specific characteristic:
Here: compressor air bleed at low power output
➔ Efficiency drops

FUEL CONSUMPTION
2 X TURBOMECA ARRIEL 1E2

POWER AS REQUIRED — BLEED AIR HEATING OFF

NOTE WITH BLEED AIR HEATING ON, THE SPEC. FUEL CONSUMPTION IS 2.0 % HIGHER.

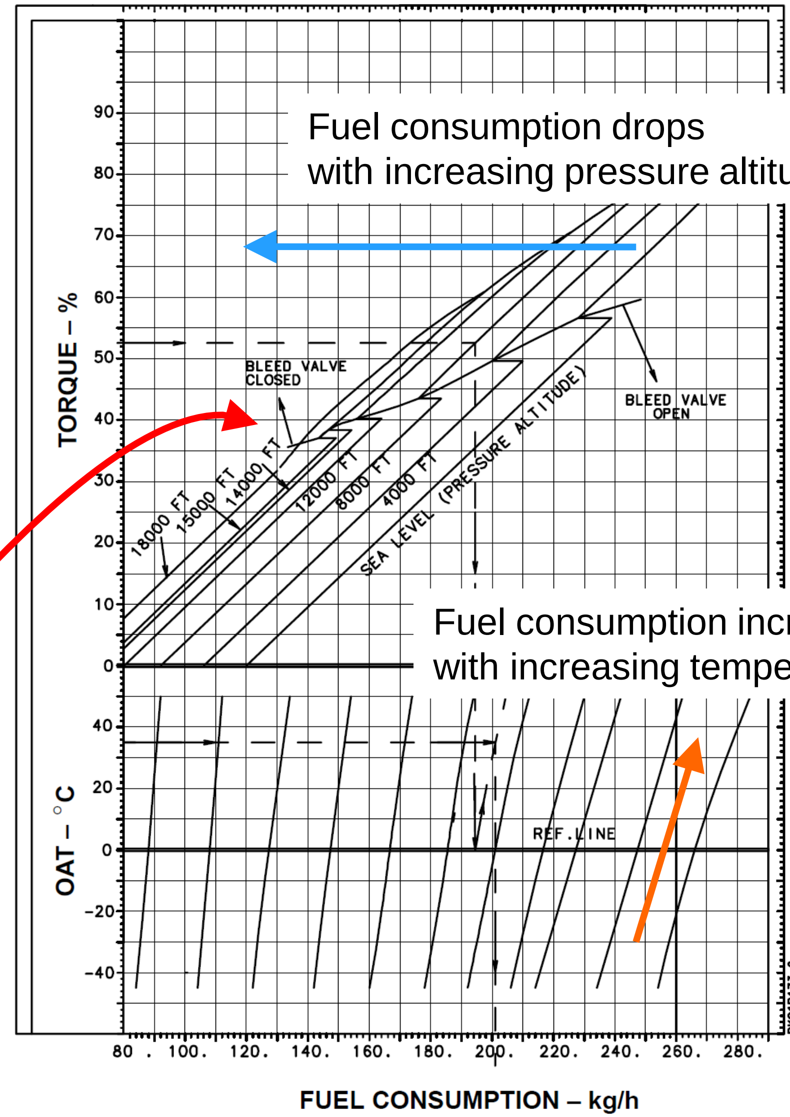


Abbildung aus *Eurocopter Flight Manual BK117 C-1*

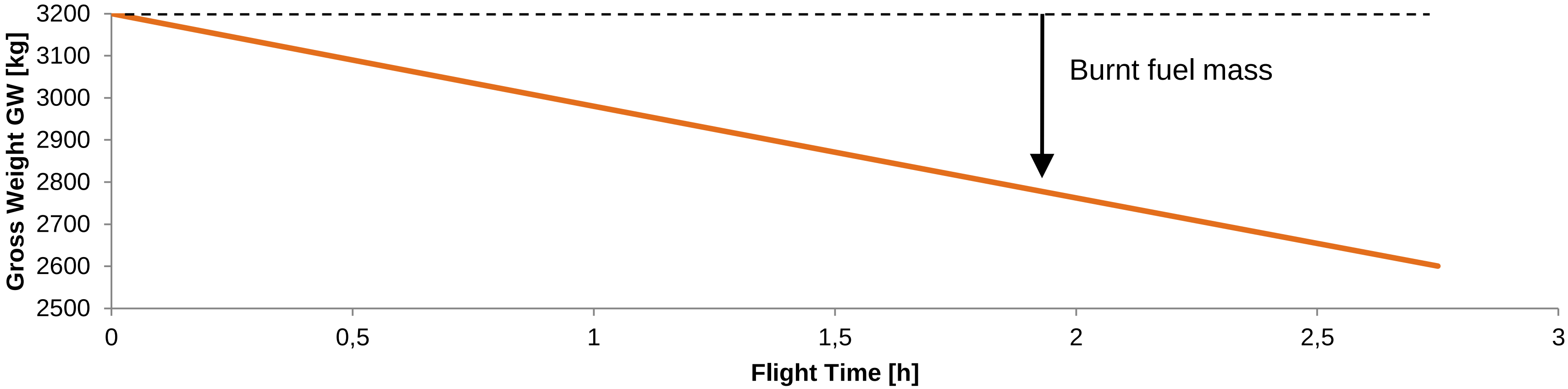
4 Flight Performance

4.3 Endurance, range and load capacity

- ✓ Fuel consumption in horizontal flight
- ➔ Range
- • Endurance

4.3 Endurance, range and load capacity

<u>Range</u>

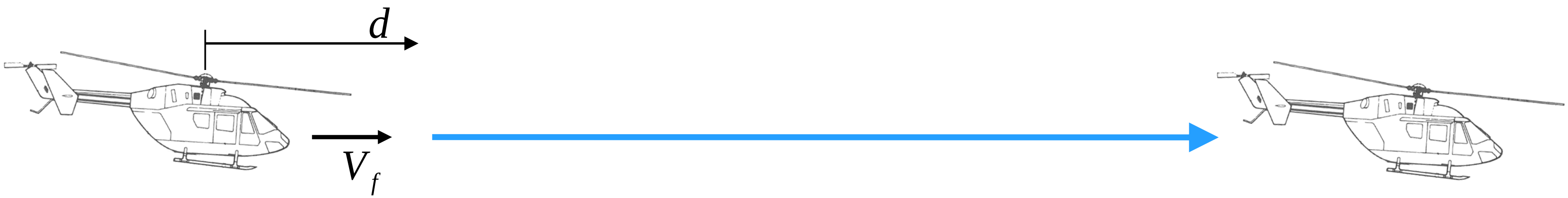


Gross weight GW changes continuously

➔ Power consumption P as well as speeds for maximum range V_{BR} and maximum endurance V_{ME} vary.

4.3 Endurance, range and load capacity

Range



While in flight the distance covered d can be expressed dependant on the specific range $\frac{V_f}{\dot{m}_{Fuel}}$:

$$\mathrm{d}d = -\frac{V_f}{\dot{m}_{Fuel}} \cdot \mathrm{d}m_{Fuel}$$

The covered distance can therefore be calculated based on the difference in fuel mass Δm_{Fuel} :

$$d = -\int_0^{\Delta m_{Fuel}} \frac{V_f}{\dot{m}_{Fuel}} \cdot \mathrm{d}m_{Fuel}$$

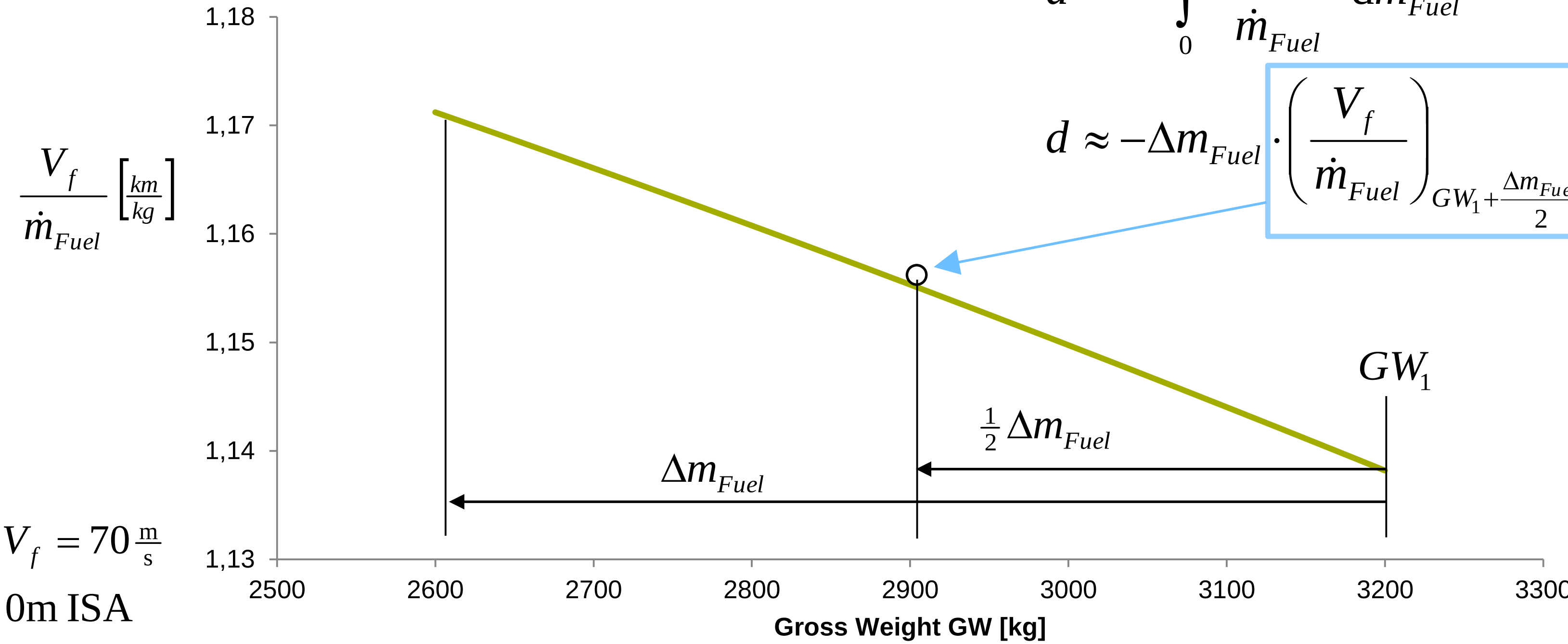
4.3 Endurance, range and load capacity

Range in horizontal flight at constant flight speed

Assuming that the specific range is linear the total range d can be calculated using the mean gross weight $GW_1 + \frac{\Delta m_{Fuel}}{2}$:

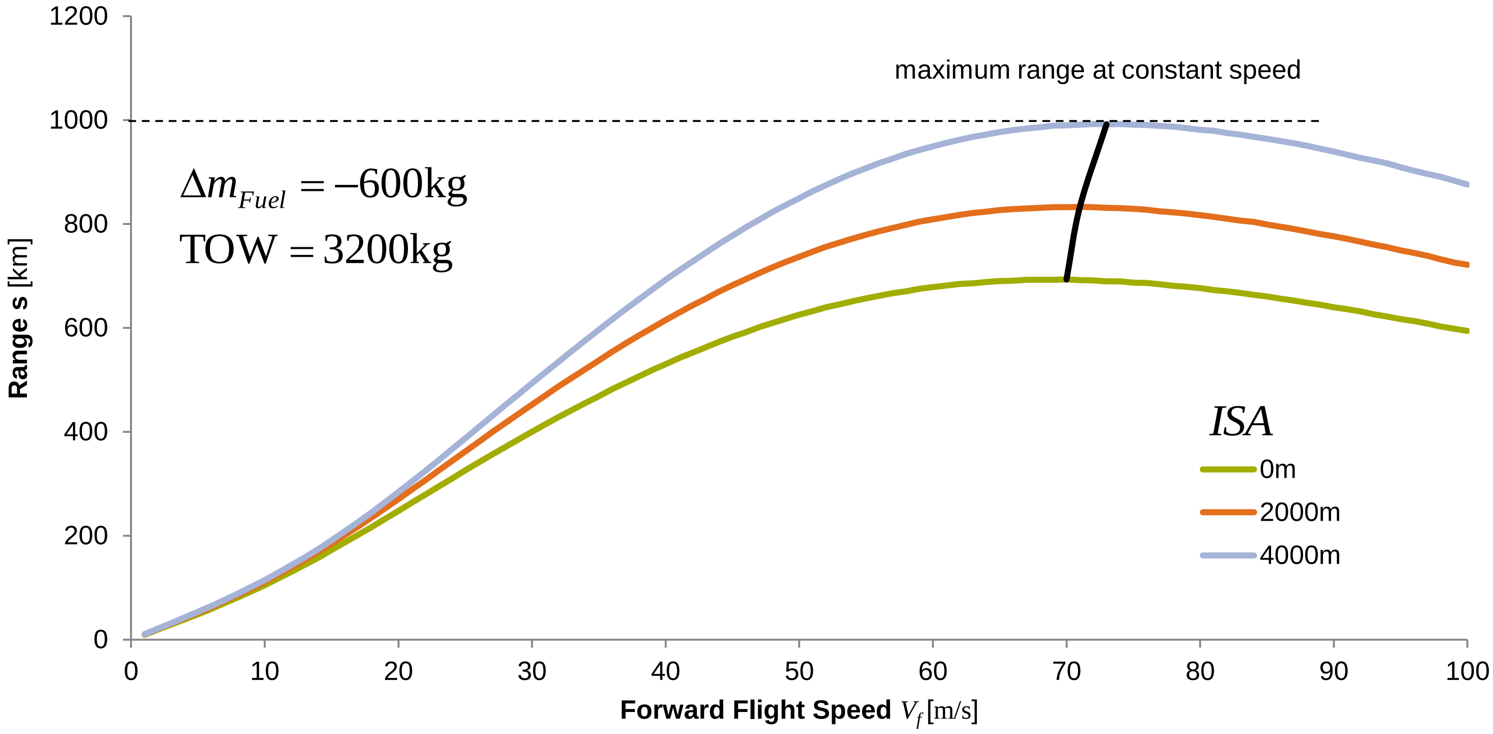
$$d = -\int_0^{\Delta m_{Fuel}} \frac{V_f}{\dot{m}_{Fuel}} \cdot \mathrm{d}m_{Fuel}$$

$$d \approx -\Delta m_{Fuel} \cdot \boxed{\left(\frac{V_f}{\dot{m}_{Fuel}}\right)_{GW_1 + \frac{\Delta m_{Fuel}}{2}}}$$



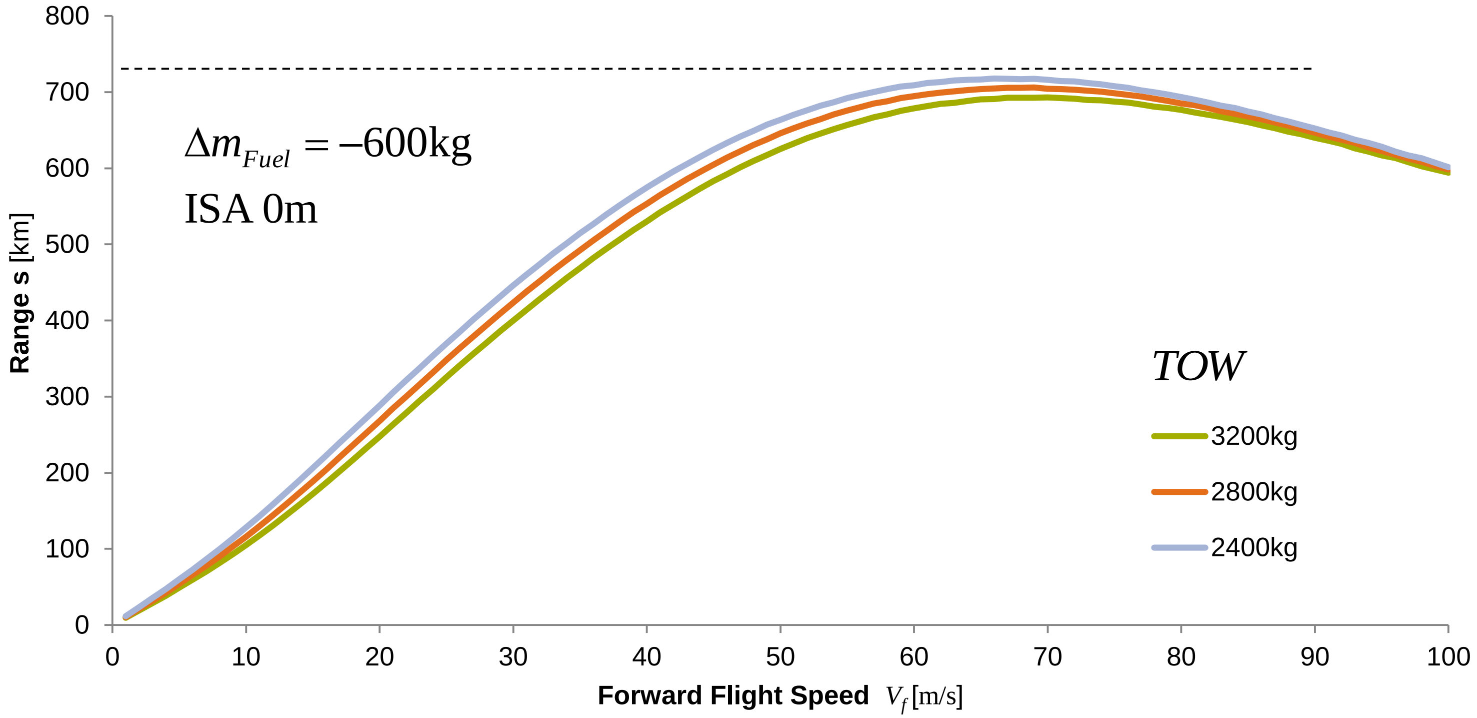
4.3 Endurance, range and load capacity

Range in horizontal flight at constant flight speed
(without considering power limitations)



4.3 Endurance, range and load capacity

<u>Range</u> in horizontal flight at constant flight speed
(without considering power limitations)



4.3 Endurance, range and load capacity

USABLE FUEL 558 KG
POWER AS REQUIRED
BLEED AIR HEATING OFF

NOTE WITH BLEED AIR HEATING ON, THE MAXIMUM RANGE IS 2.0 % LESS.

Flight manual diagram:

Showing max. range in horizontal flight at constant speed depending on:

- Flight speed
- Take-off weight
- Environmental conditions

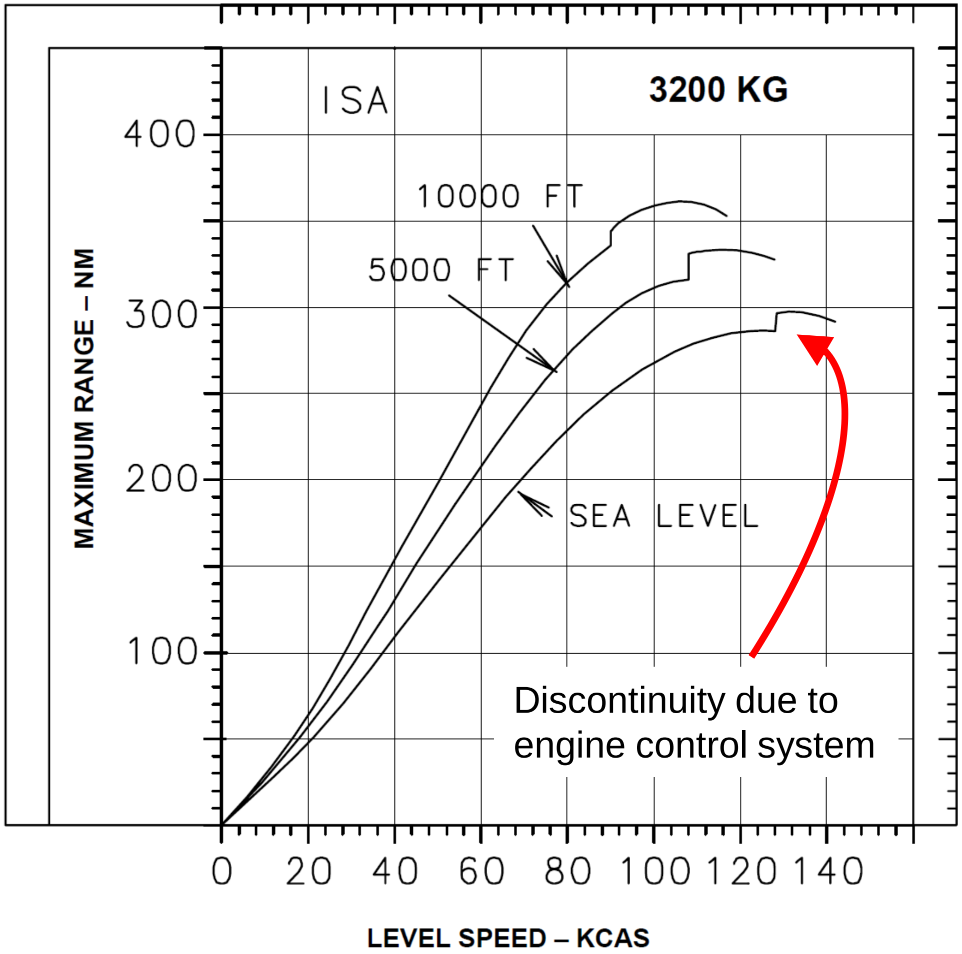
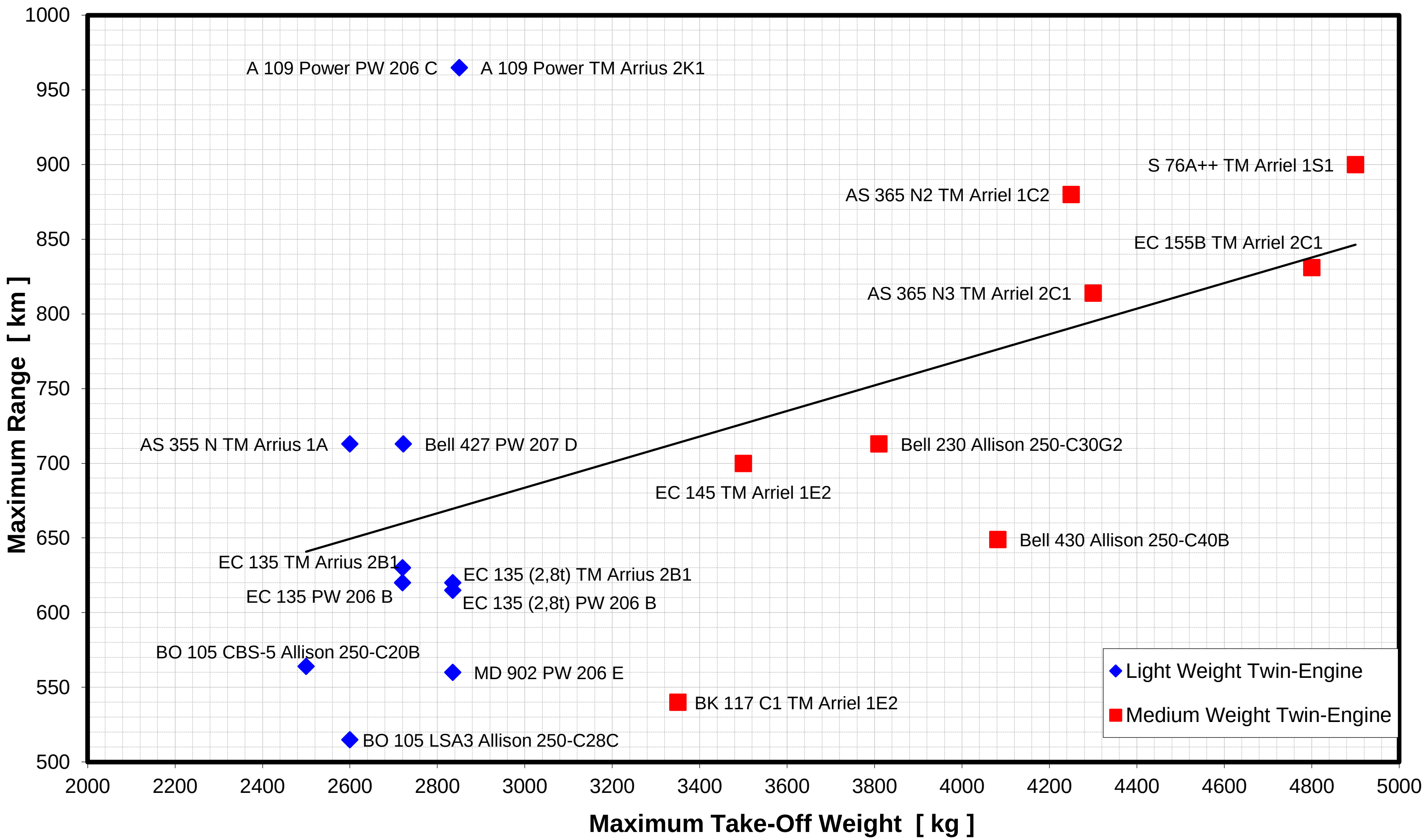


Abbildung aus *Eurocopter Flight Manual BK117 C-1*

4.3 Endurance, range and load capacity

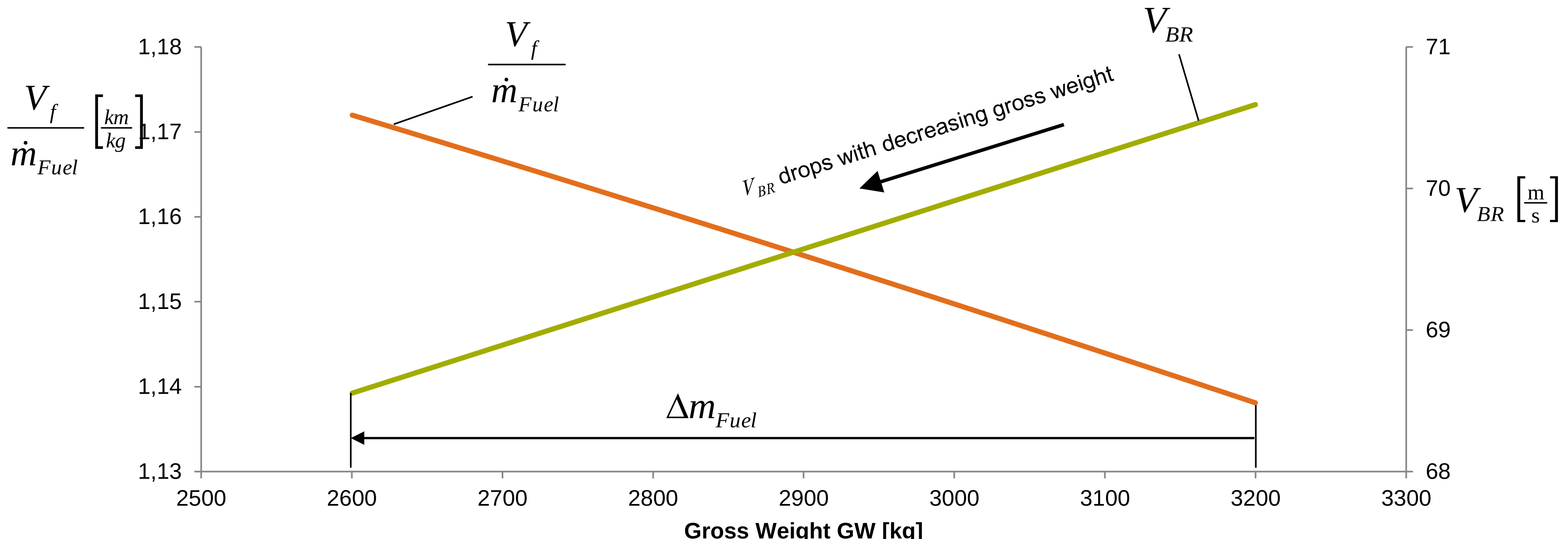


Daten: Eurocopter

4.3 Endurance, range and load capacity

<u>Maximum range</u> in horizontal flight at arbitrary speed

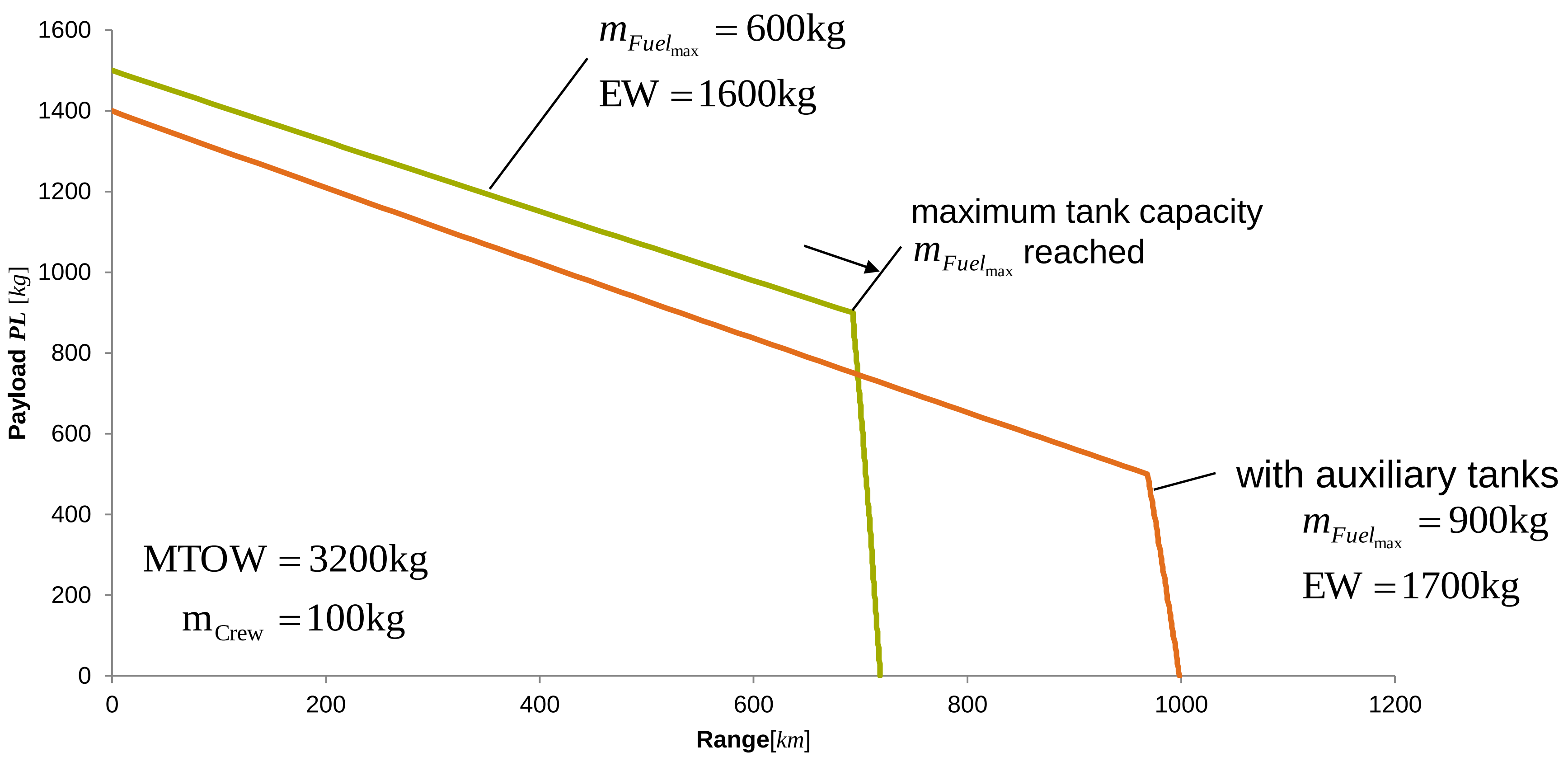
The covered flight distance for an available fuel mass becomes maximal when the forward flight speed always equals $V_f = V_{BR}$



ISA 0m

4.3 Endurance, range and load capacity

<u>Range</u> dependent on payload $\boldsymbol{PL}$ (*payload-range* diagram)



4 Flight Performance

4.3 Endurance, range and load capacity

- ✓ Fuel consumption in horizontal flight
- ✓ Range
- ➔ Endurance

4.3 Endurance, range and load capacity

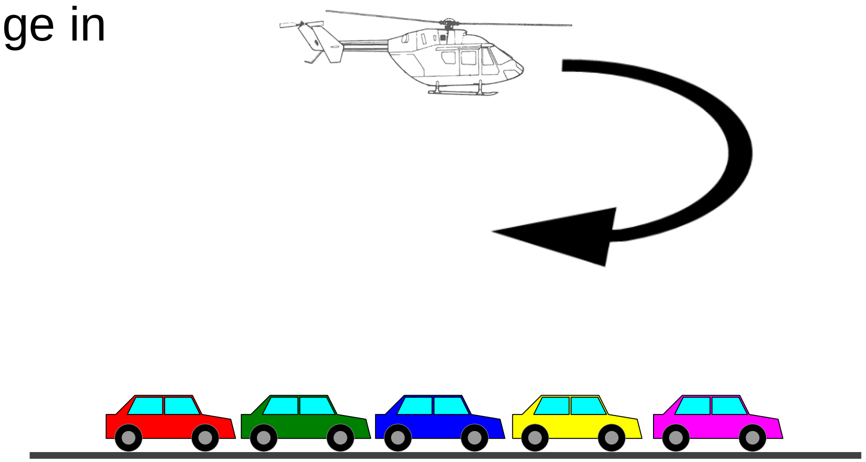
Endurance

Analogously to range calculation the endurance E can be determined using the specific endurance: $\frac{1}{\dot{m}_{Fuel}}$

$$\mathrm{d}E = -\frac{1}{\dot{m}_{Fuel}} \cdot \mathrm{d}m_{Fuel}$$

Therefore the endurance for a given change in fuel mass: Δm_{Fuel}

$$E = -\int_0^{\Delta m_{Fuel}} \frac{1}{\dot{m}_{Fuel}} \cdot \mathrm{d}m_{Fuel}$$



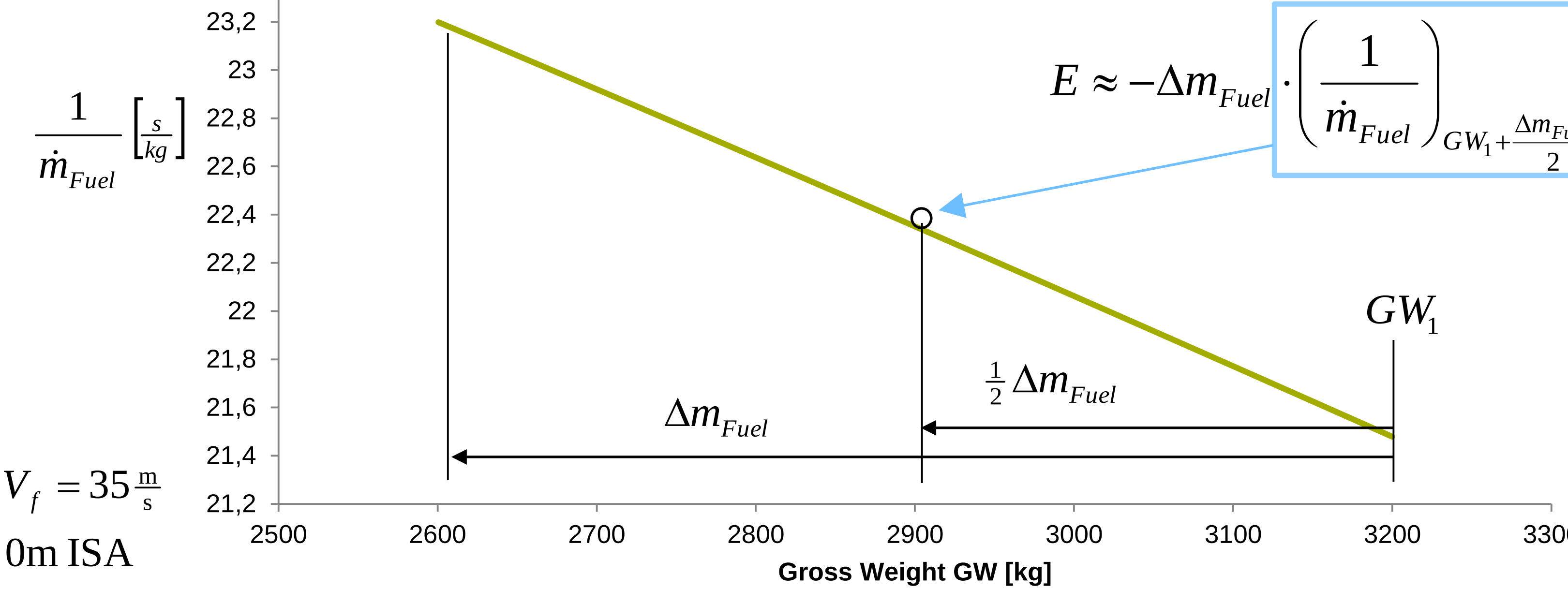
4.3 Endurance, range and load capacity

Endurance in horizontal flight at constant flight speed

Also for estimating the endurance a constant specific endurance can be assumed. The specific endurance used is its value at: $GW_1 + \frac{\Delta m_{Fuel}}{2}$

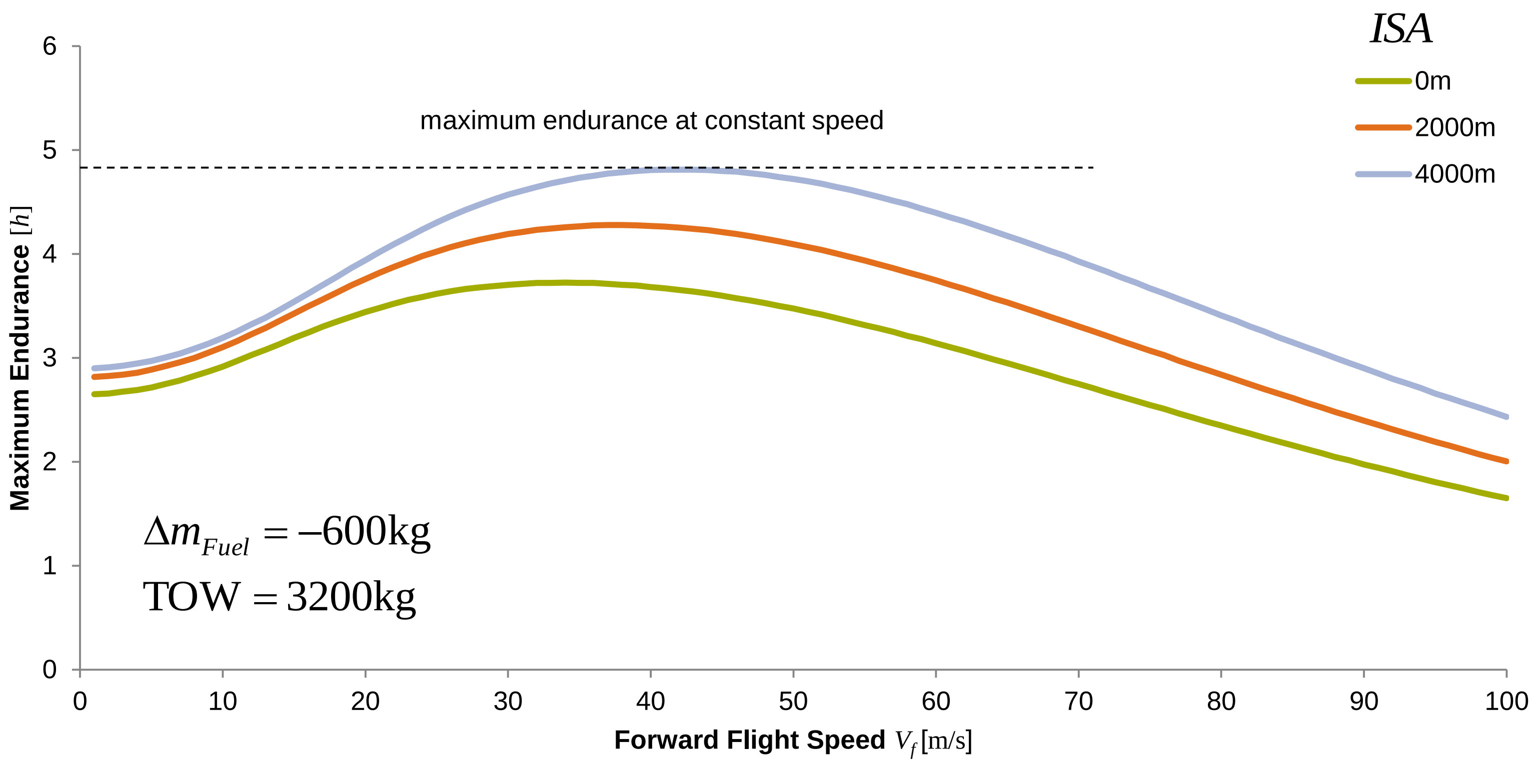
$$E = -\int_0^{\Delta m_{Fuel}} \frac{1}{\dot{m}_{Fuel}} \cdot \mathrm{d}m_{Fuel}$$

$$E \approx -\Delta m_{Fuel} \cdot \boxed{\left(\frac{1}{\dot{m}_{Fuel}}\right)_{GW_1 + \frac{\Delta m_{Fuel}}{2}}}$$



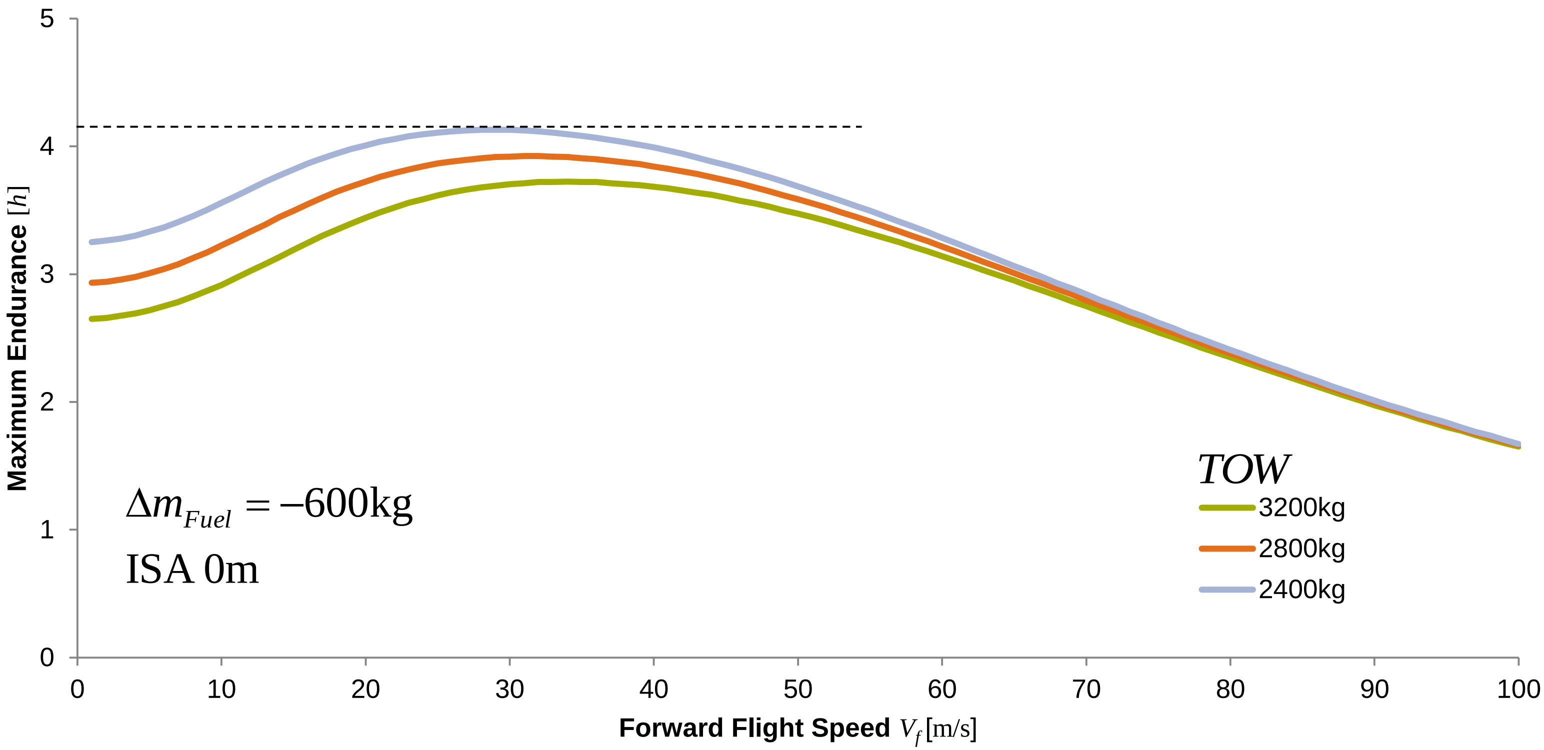
4.3 Endurance, range and load capacity

<u>Endurance</u> in horizontal flight at constant flight speed
(without considering power limitations)



4.3 Endurance, range and load capacity

Endurance in horizontal flight at constant flight speed
(without considering power limitations)



4.3 Endurance, range and load capacity

Flight manual diagram:

Maximum endurance in horizontal flight at constant speed dependent on:

- Flight speed
- Take-off weight
- Environmental conditions

USABLE FUEL 558 KG
POWER AS REQUIRED
BLEED AIR HEATING OFF

NOTE WITH BLEED AIR HEATING ON, THE MAXIMUM RANGE IS 2.0 % LESS.

influence of engine control system

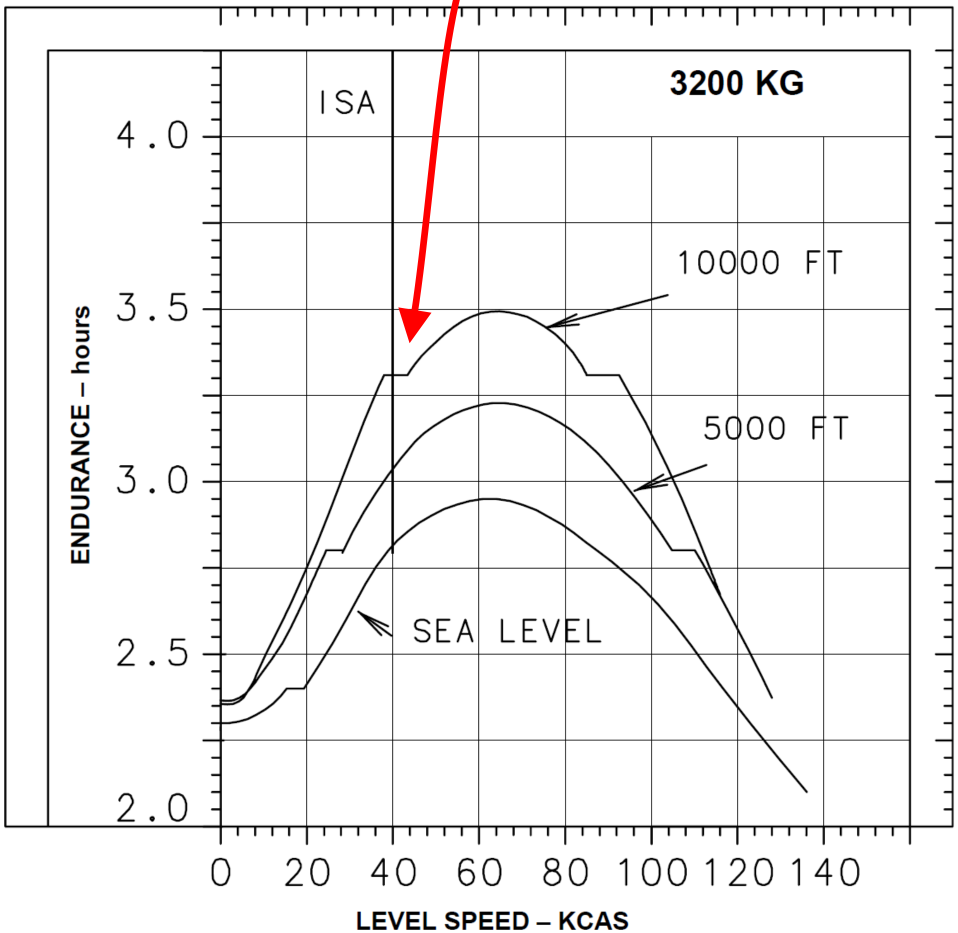
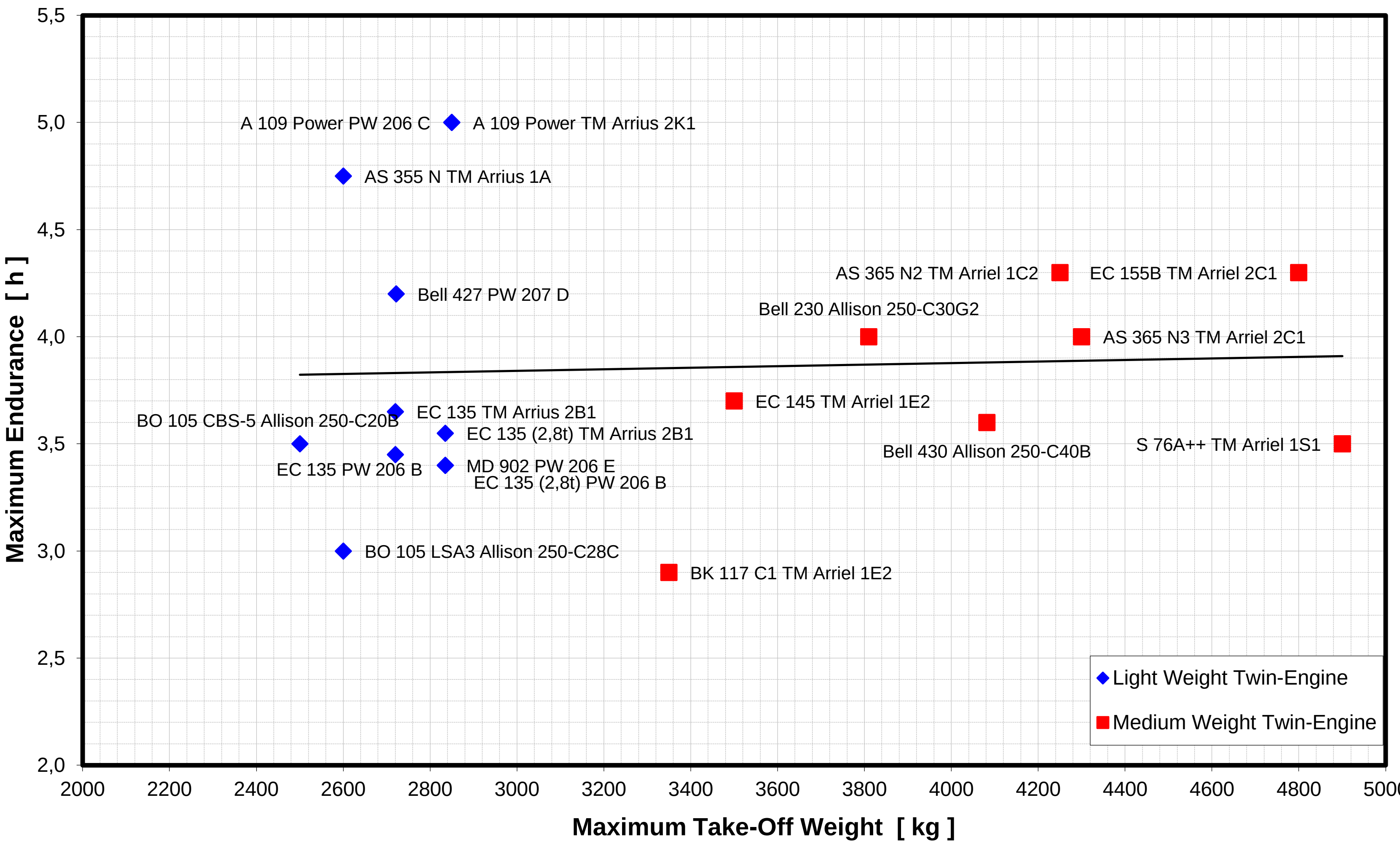


Abbildung aus *Eurocopter Flight Manual BK117 C-1*

4.3 Endurance, range and load capacity

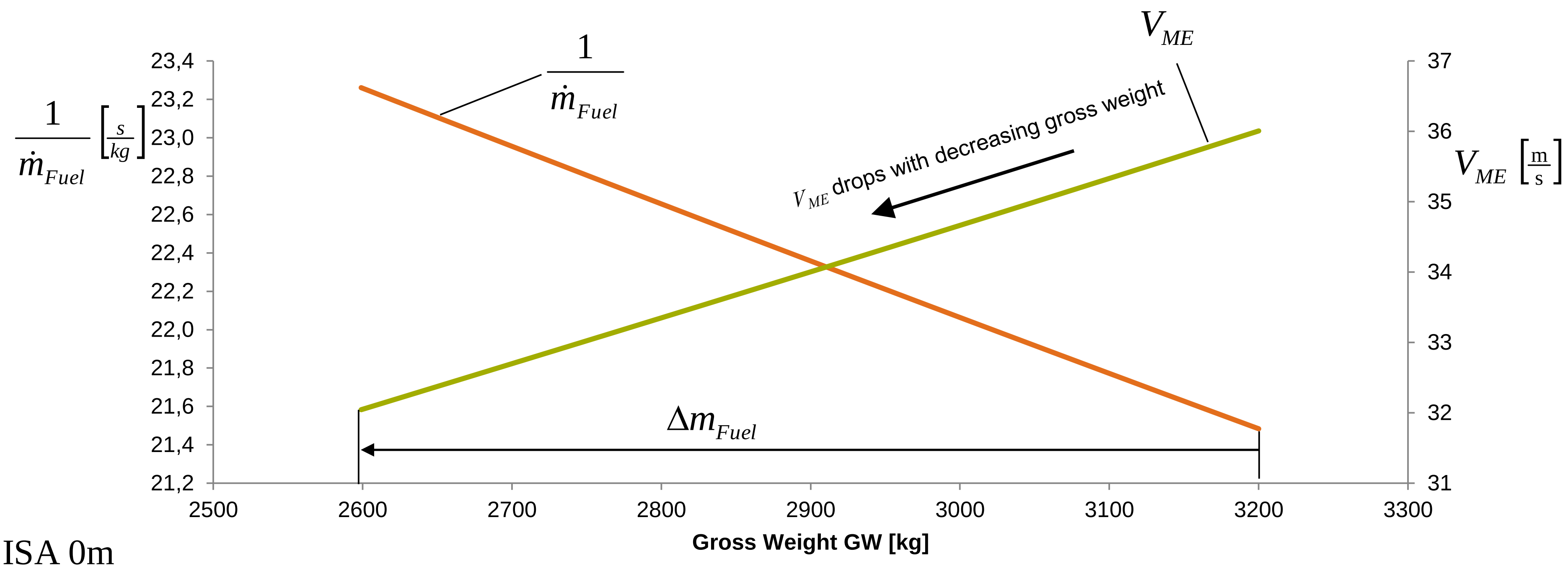


Daten: Eurocopter

4.3 Endurance, range and load capacity

<u>Maximum Endurance</u> in horizontal flight at arbitrary speed

The total endurance for an available fuel mass becomes maximal when the forward flight speed always equals $V_f = V_{ME}$



ISA 0m

4 Flight Performance

Example EC145 (BK117-C2)

The specified hover flight mass at 11000ft ISA couldn't be achieved by a small margin in the former configuration. (Power required approx. 500kW)

Measures:

- Increasing the specified engine power by 2%

➔ +11kW / +45kg at 11000ft

- Insulation of the oil cooler

➔ +9kW / +36kg at 11000ft

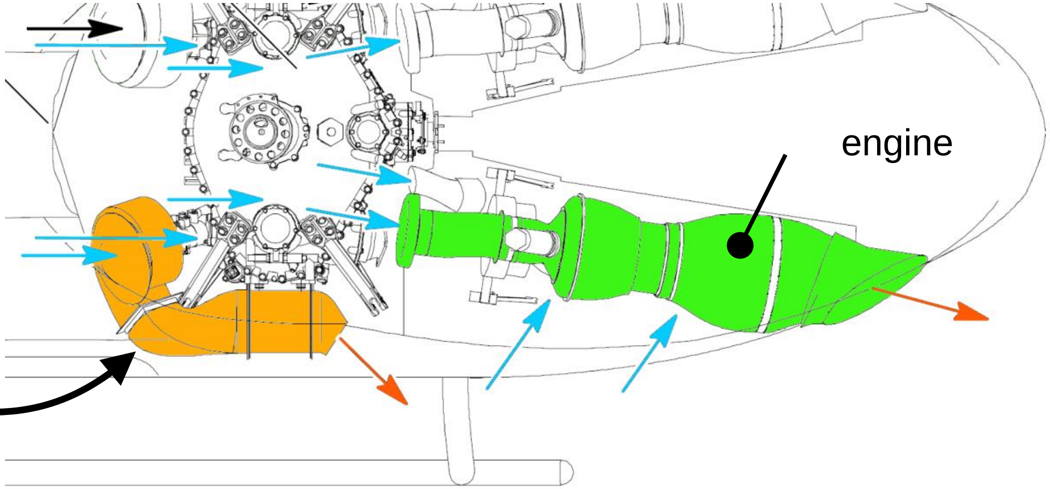


Foto: Eurocopter, Rémy Michelin; Abbildung: Eurocopter

4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Carpet Plot

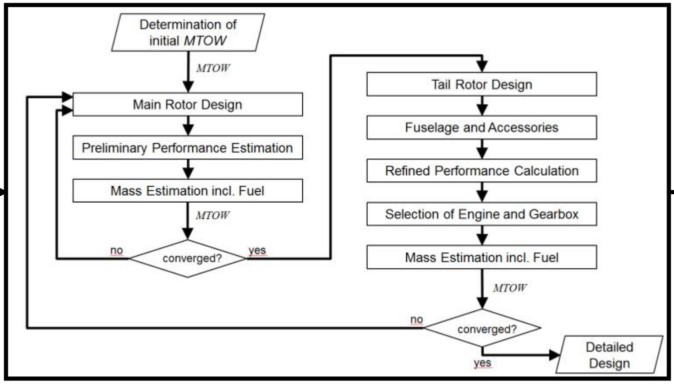
=> suitable tool for post-processing and evaluating a helicopter design

- visualize uncertainties and sensitivities
- identification of *design drivers*
- quantify the design: take-off weight, dimensions, …
- visual interpolation of the results (for this reason more suitable than surfaces and isoline-plots)

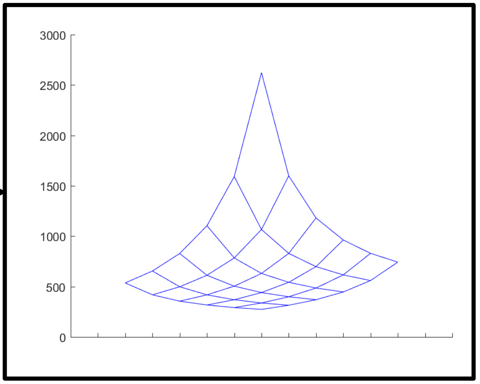
Szenario

design
mission
&
design
variables

Design Methodology



Evaluation



4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Szenario 1

Design Mission:
payload: 100 kg
range: 50km
time amount of hover: 10%
altitude: 500m ISA +5K
=> density: 1.1471 kg/m³

A — V_{BR} — B

5% of overall mission time each:
- HOGE @ A
- HOGE @ B

Configuration:
combustion engine:
- DL = 300 N/m²
- FM = 0,65
- (L/D) = 3.5

TOW = EW + m_{pay} + m_{fuel}

4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Design Methodology 1

TOW_{init}

$$\mathrm{PL} = \frac{FM}{\sqrt{DL}} * \sqrt{2\rho}$$

$$\dot{m}_{fuel,hover} = SFC * GW/PL$$

$$\dot{m}_{fuel,cruise} = m_{tot} * \left(1 - {1}\Big/{e^{\frac{SFC*range*g}{V_F*\left({L}/{D}\right)}}}\right)$$

$$m_{fuel} = \sum_j \dot{m}_{fuel,j} * t_j$$

design variable: V_F

determine: $\left(\frac{\dot{m}_{fuel,cruise}}{V_F}\right)_{min}$

Calculation of $\mathbf{V_{BR}}$

calculation of power

$$TOW = g * \frac{m_{payload} + m_{fuel}}{1 - \left({EW}/{TOW}\right)}$$

calculation of total weight

iterate until TOW has converged

$\vec{z}_{solution}$

Breguet range equation

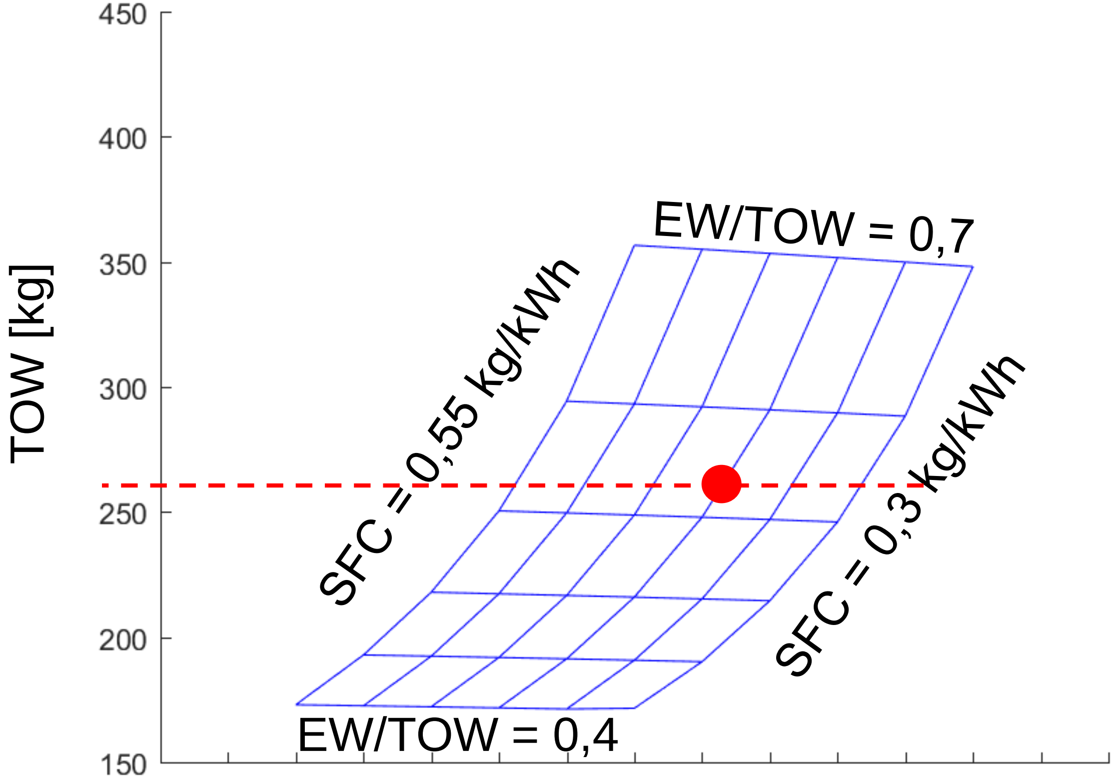
j: Index of mission segment

4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Evaluation 1

Take-Off Weight depending on **SFC** and **EW/TOW**

- generalized design methodology; no main rotor sizing
- empty weight ratio has major influence on take-off weight
- SFC has an influence on TOW esp. at high empty weight ratio
- example:
  SFC = 0,4
  EW/TOW = 0,6
  => 260 kg



4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Szenario 2

Design Mission:
payload: 100 kg
range: 50km
time amount of hover: 10%
altitude: 500m ISA +5K
=> density: 1.1471 kg/m³

A V_{BR} B

5% of overall mission time each:
- HOGE @ A
- HOGE @ B

Configuration:
combustion engine:
- Vtip = 200 m/s
- Cd0 = 0.01
- $C_T/\sigma = 0.1$

$TOW = EW + m_{pay} + m_{fuel}$

4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Design Methodology 2

$$TOW_{init}$$

empirical approaches

$$\sigma = f(TOW, \sigma_0)$$

$$(C_D S) = f(TOW, (C_D S)_0)$$

momentum theory

$$P_{hover} = f(A, TOW, \sigma, C_D S)$$

$$P_{cruise} = f(A, TOW, \sigma, C_D S, V_F)$$

$$\dot{m}_{fuel,j} = SFC * P_j$$

$$m_{fuel} = \sum_j \dot{m}_{fuel,j} * t_j$$

Calculation of V_{BR}

constraint:

$$\frac{C_T}{\sigma} = \left(\frac{C_T}{\sigma}\right)_{ref}$$

design variables: (V_F, A)

determine: $\left(\frac{\dot{m}_{fuel,cruise}}{V_F}\right)_{min}$

Main rotor sizing & calculation of power

$$TOW = g * \frac{m_{payload} + m_{fuel}}{1 - \left({EW}/{TOW}\right)}$$

calculation of total weight

iterate until TOW has converged

$$\vec{z}_{solution}$$

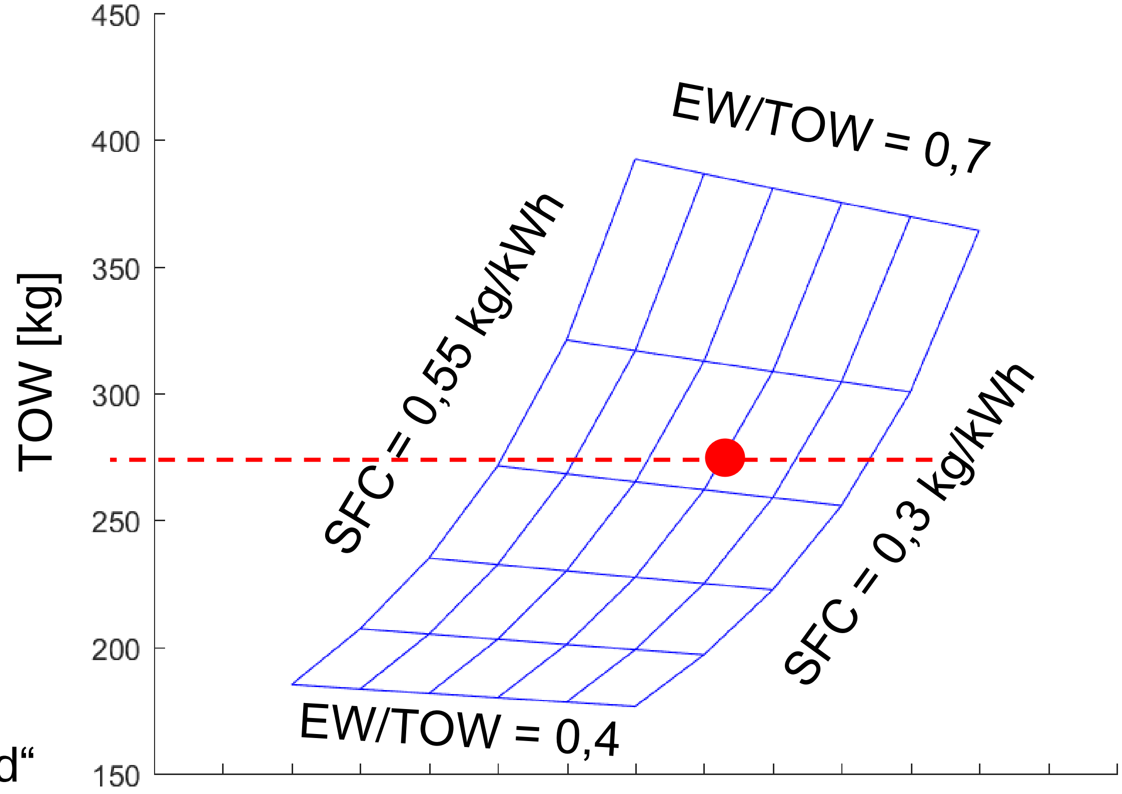
j: Index Missionssegment

4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Evaluation 2

Take-Off Weight depending on **SFC** and **EW/TOW**

- empty weight ratio has major influence on take-off weight
- SFC has an influence on TOW esp. at high empty weight ratio
- example:
  SFC = 0,4
  EW/TOW = 0,6
  => 275 kg
- 15kg difference in TOW compared to „generalized" design methodology

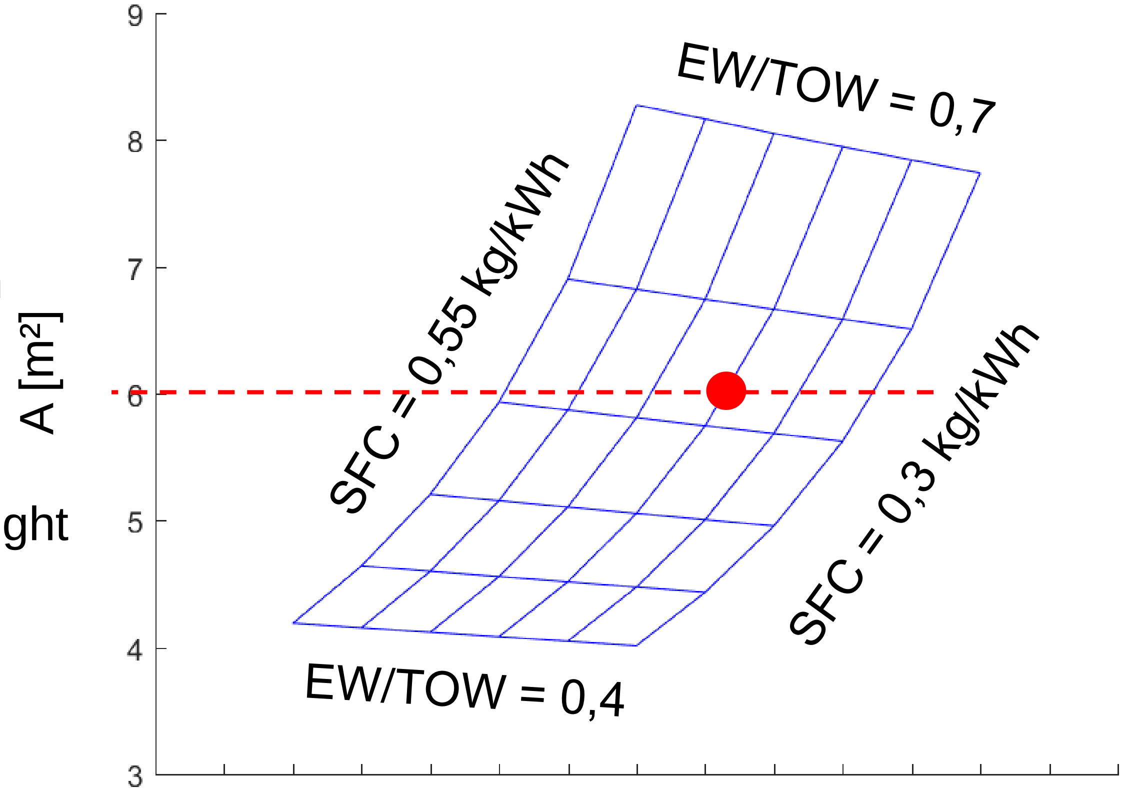


4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Evaluation 2

Main rotor disc area depending on **SFC** and **EW/TOW**

- empty weight ratio has major influence on take-off weight
- SFC has an influence on TOW esp. at high empty weight ratio
- Main rotor disc area correlates with gross weight
- example:
  SFC = 0,4
  EW/TOW = 0,6
  => 6 m²
  => DL = 450 N/m² (> 300 N/m²)



4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Szenario 3

Design Mission:
payload: 100 kg
range: 50km
time amount of hover: 10%
altitude: 500m ISA +5K
=> density: 1.1471 kg/m³

A V_{BR} B

5% of overall mission time each:

- HOGE @ A
- HOGE @ B

Configuration:
electrically powered multicopter:

- Vtip = 200 m/s
- Cd0 = 0.01
- $C_T/\sigma = 0.1$

$TOW = EW + m_{pay} + m_{fuel}$

4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Design Methodology 3

$$TOW_{init}$$

Main rotor sizing & calculation of power

empirical approaches

$$\sigma = f(TOW, \sigma_0)$$

$$(C_D S) = f(TOW, (C_D S)_0)$$

Calculation of V_{BR}

momentum theory

$$P_{hover} = f(A, TOW, \sigma, C_D S)$$

$$P_{cruise} = f(A, TOW, \sigma, C_D S, V_F)$$

$$P_{equivalent} = P_{cruise} + \eta_{hover} P_{hover}$$

$$\left(\frac{P}{V}\right) = \frac{P_{equivalent}}{V_F}$$

constraint: $\frac{C_T}{\sigma} = \left(\frac{C_T}{\sigma}\right)_{ref}$

design variables: (V_F, A)

determine: $\left(\frac{P}{V}\right)_{min}$

calculation of total weight

$$t_{cruise} = \frac{s_{range}}{V_F}$$

$$E_{mission} = P_{equivalent} * t_{cruise}$$

$$m_{cell} = {E_{mission}}/{ED}$$

$$TOW = g * \frac{m_{payload} + m_{cell}}{1 - \left({EW}/{TOW}\right)}$$

iterate until TOW has converged

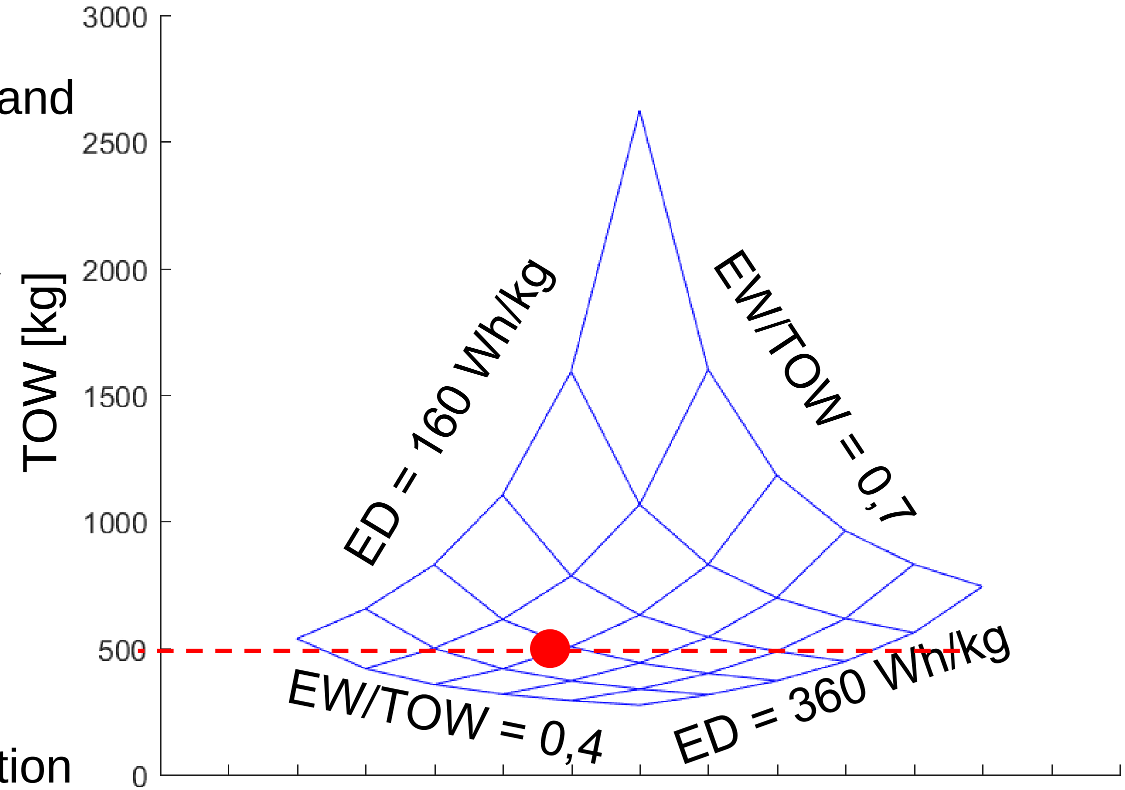
$$\vec{z}_{solution}$$

4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Evaluation 3

Take-Off Weight depending on **SFC** and **EW/TOW**

- Both empty weight ratio and specific energy density have a dominant impact
- No gearbox: Less empty weight ratio possible
- example:
  EW/TOW = 0.5,
  ED = 240
  => TOW = 500kg
- An electrically powered configuration is much heavier then a configuration powered by a combustion engine

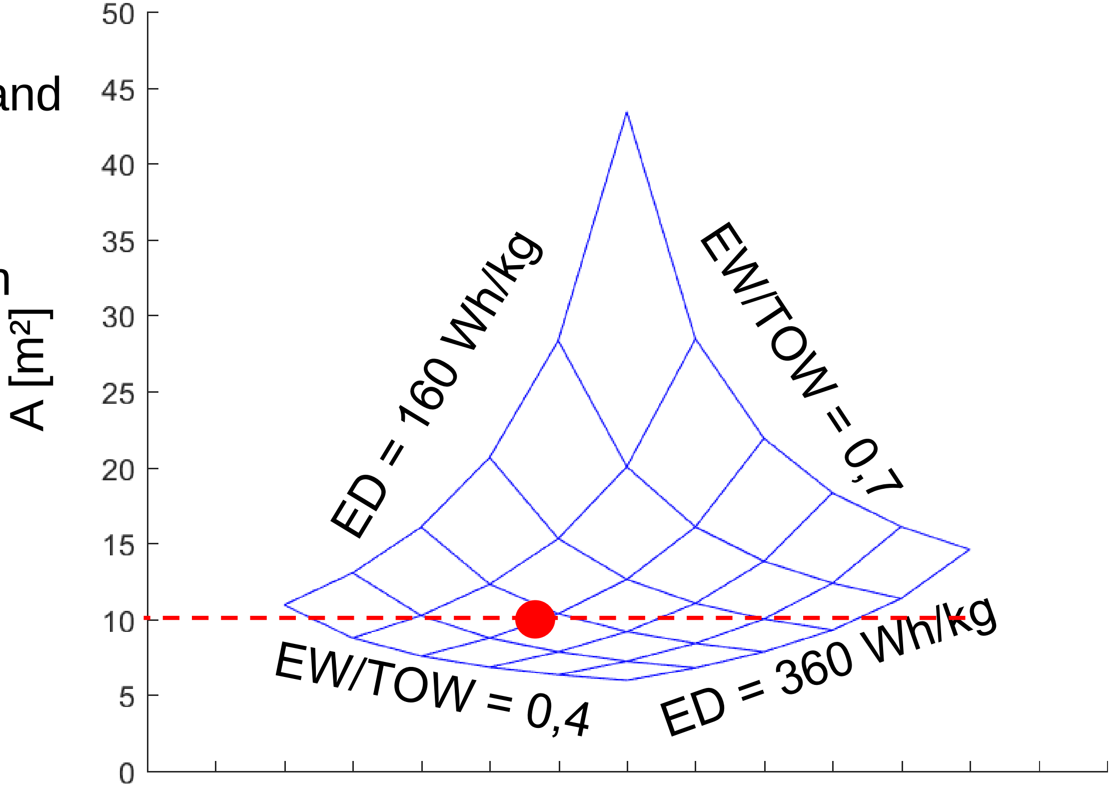


4.4 Carpet Plot: Design of a Personal Air Vehicle (PAV)

Evaluation 3

Main rotor disc area depending on **SFC** and **EW/TOW**

- Both empty weight ratio and specific energy density have a dominant impact
- since much heavier, main rotor disc area is much larger compared to combustion engine powered helicopters



5 Cost and Mass Estimation

5 Cost and Mass Estimation

5.1 Introduction

5.2 Mass Estimation and Optimization

- 5.2.1 Estimation of Mass Parts
- 5.2.2 Determination of Center of Gravity Position
- 5.2.3 Mass Optimization

5.3 Cost Estimation and Optimization

- 5.3.1 Cost Incurrence and Composition
- 5.3.2 Cost Estimation and Optimization

5.4 Criteria for Component Selection and Development

5.1 Introduction

5.1 Introduction

- This chapter will provide first estimates for component masses based on the determined design parameters.
- The component masses can be used for preliminarily determining the CG position of the helicopter.
- Furthermore they can be used as a basis for component selection and mass optimization. However the component masses will require continuous updating in the iterative design process.
- Also the expected costs of procurement/development and the running costs of the components can be estimated. The cost as well as the mass are important criteria for component selection.
- There are however many more selection criteria, the nature and extent of which vary depending on component type. Because existing components or alternative technologies provide mostly discrete solutions, finding a common “optimum” of all selection criteria is only possible by means of a trade off and experience.

5.1 Introduction

Note:

- The aforementioned variables are decisive factors in the early project stage of a preliminary design concerning the development project feasibility, planning transparency but also the competitiveness of the future product.
- Naturally at the beginning of the project only rough technical values are available for an estimation and optimization problem which stabilize as the development project progresses.
- Especially optimizations should be performed only once the configuration is fixed to the greatest possible extent and the design parameters are subject to only small variations.

5.2 Mass Estimation and Optimization

5.2 Mass Estimation and Optimization

- An extensive experience as well as profound knowledge of design trends are necessary for successful mass estimation.
- The primary tools of a so called Weight Engineer are approximate equations for individual components, that were derived from the masses of current helicopters by means of multiple linear regressions. This approach enables weighting of all parameters that influence component masses.
- The equations have to be adjusted with new helicopters emerging on the market and the progressing level of detail over the course of the development.
- The structure and especially the parameters of such equations are closely guarded corporate secrets of helicopter manufacturers.

5.2 Mass Estimation and Optimization

Design parameters established during preliminary design:

MTOW	Maximum Take-off Weight	[kg]
R	Main Rotor Radius	[m²]
$\mathbf{N_b}$	Number of (Main) Rotor Blades	[-]
C	(Main) Rotor Blade Chord	[m]
$\mathbf{P_{h,OGE}}$	Hover Flight Power	[kW]
$\mathbf{m_{fuel}}$	Fuel Mass	[kg]

Further parameters for the component mass estimation:

$\mathbf{S_{f,wet}}$	Wetted Fuselage Surface	[m²]
$\mathbf{S_{t,wet}}$	Wetted Tail Surface	[m²]
$\mathbf{n_{pax+crew}}$	Number of Passengers (Passengers & Crew)	[-]
$\mathbf{m_{special}}$	Special Equipment (Avionics) Mass	[kg]
EW	Empty Weight	[kg]
$\mathbf{n_{legs,wlg}}$	Number of legs in case of wheel landing gear	[-]

5.2 Mass Estimation and Optimization

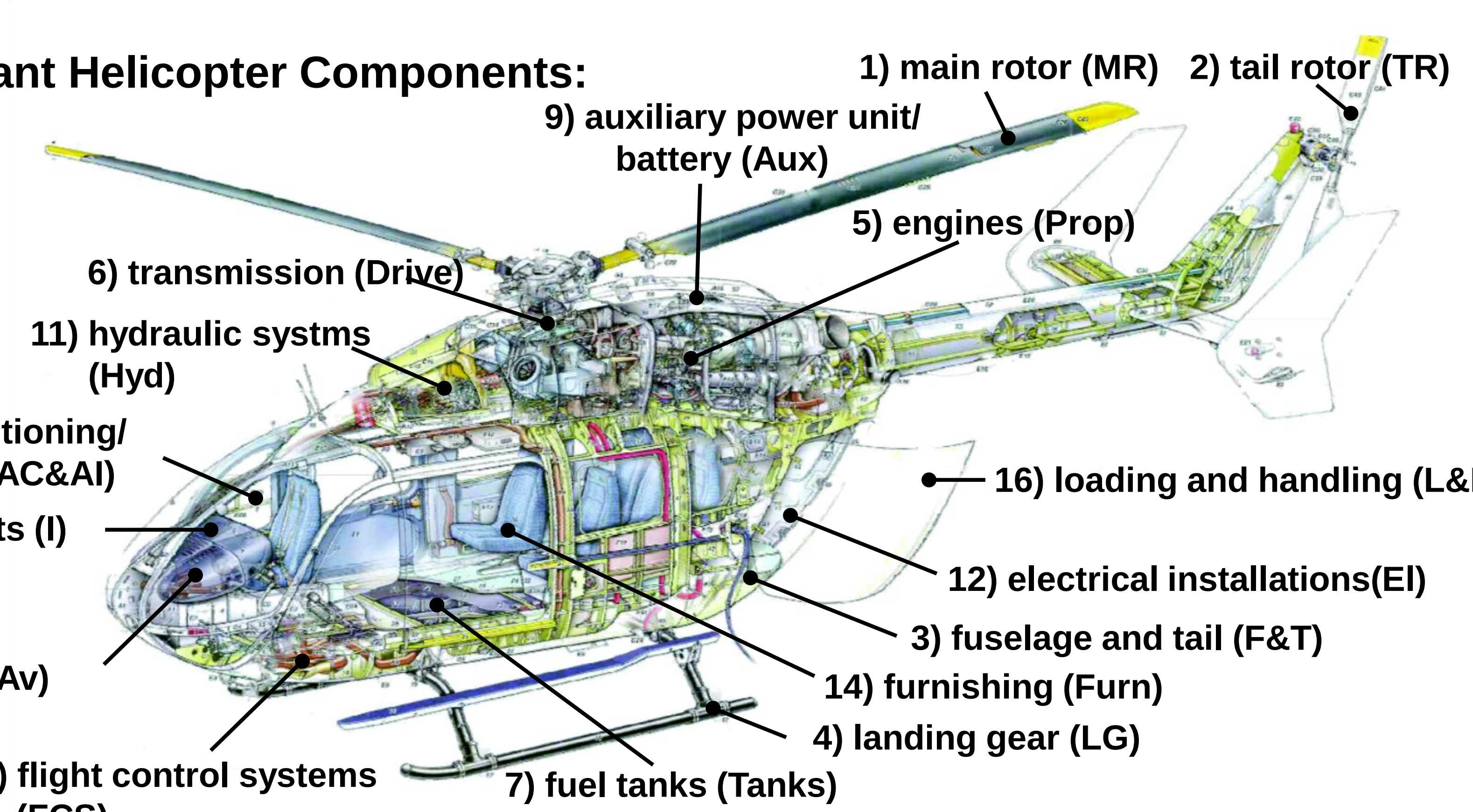
Weight Class Definition [1]:

Light Helicopter	MTOW < 1360 kg
Medium Helicopter	1360 kg < MTOW < 11340 kg
Heavy Helicopter	MTOW > 11340 kg

[1]: only applicable to mass equations

5.2.1 Estimation of Mass Parts

Important Helicopter Components:



Empty Weight

$$EW = m_{MR} + m_{TR} + m_{F\&T} + m_{LG} + m_{Prop} + m_{Drive} + m_{Tanks} + m_{FCS} + m_{Aux} + m_{I} + \\ + m_{Hyd} + m_{El} + m_{Av} + m_{Furn} + m_{AC\&AI} + m_{L\&H}$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters:

Wetted tail surface:

$$S_{t,wet} = 0{,}004684 \cdot \frac{P_{h,OGE}}{kW} m^2 - 0{,}75307 m^2$$

Wetted fuselage surface:

$$S_{f,wet} = 59{,}09386 \cdot e^{0{,}0000194463 \frac{MTOW}{kg}} m^2$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

1) **Main rotor (m_{MR})**

$$m_{MR,med} = \frac{33 \cdot R \cdot c \cdot N_b \cdot kg}{m^2} + 16kg$$

2) **Tail rotor (m_{TR})**

$$m_{TR} = 0{,}003942 \cdot MTOW + 5{,}66kg$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

3) Fuselage and tail ($m_{F\&T}$)

$$m_{F\&T} = 0{,}11907 \cdot MTOW - 66{,}666 kg$$

4) Landing gear (m_{LG})

$$m_{LG} = \begin{cases} 0{,}011113004 \cdot \left(0{,}9 \dfrac{MTOW}{0{,}453592kg}\right)^{0{,}8606} \cdot N_b^{\;0{,}8046} \cdot kg & \text{skids} \\ 0{,}187333496 \cdot \left(0{,}9 \dfrac{MTOW}{0{,}453592kg}\right)^{0{,}6662} \cdot 1^{0{,}1198} \cdot n_{legs,wlg}^{\;0{,}536} \cdot kg & \text{wheels, rigid} \\ 0{,}187333496 \cdot \left(0{,}9 \dfrac{MTOW}{0{,}453592kg}\right)^{0{,}6662} \cdot 2^{0{,}1198} \cdot n_{legs,wlg}^{\;0{,}536} \cdot kg & \text{wheels, retractable} \end{cases}$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

5) Engines (m_{Prop})

$$m_{\mathrm{Prop}} = \begin{cases} N_{Engine} \cdot m_{Engine} & \text{engine conf. known} \\ 1{,}83 \cdot \left(133{,}8 + 0{,}1156 \cdot \dfrac{P_{h,OGE}}{kW} \right) \cdot kg & \text{engine conf. unknown} \end{cases}$$

6) Transmission (m_{Drive})

$$m_{Drive} = 0{,}00000166 \cdot \left(0{,}9 \cdot \frac{MTOW}{kg} \right)^2 \cdot kg + 0{,}087780096 \cdot MTOW - 113{,}81241656 kg$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

7) Fuel tanks (m_{Tanks})

$$m_{Tanks} = 164{,}751 \cdot \ln\left(\frac{m_{Fuel}}{2{,}948\,kg}\right) kg - 751{,}33\,kg$$

8) Flight control systems (m_{FCS})

$$m_{FCS} = 95{,}6368 \cdot e^{0{,}000111114 \frac{MTOW}{kg}}\,kg$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

9) Auxiliary power unit (m_{Aux})

$$m_{Aux} = 0{,}0 kg$$

10) Instruments (m_I)

$$m_I = 25{,}444 \cdot \ln\left(\frac{P_{h,OGE}}{0{,}7457 kW}\right) kg - 141{,}62 kg$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

11) Hydraulic systems (m_{Hyd})

$$m_{Hyd} = 0{,}003258 \cdot MTOW + 5{,}24kg$$

12) Electrical systems (m_{El})

$$m_{El} = 218{,}496 \cdot \ln\left(\frac{S_{f,wet}}{0{,}092903m^2}\right)kg - 1267{,}49kg$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

13) Avionics (m_{Av})

$$m_{Av} = 113{,}4 kg + m_{Special}$$

14) Furnishing (m_{Furn})

$$m_{Furn} = 0{,}854 \cdot S_{f,wet} \cdot \frac{kg}{m^2} + 9{,}98 \cdot n_{Pax+Crew} \cdot kg - 4{,}54 kg$$

5.2.1 Estimation of Mass Parts

Example – Component masses of medium helicopters :

15) Air conditioning/ anti ice ($m_{AC\&AI}$)

$$m_{AC\&Ai} = 55{,}542 \cdot \ln\left(10{,}7369 \cdot \frac{S_{f,wet}}{m^2}\right) \cdot kg - 331.21kg$$

16) Loading and handling ($m_{L\&H}$) $m_{L\&H} = 38kg$

5.2.1 Estimation of Mass Parts

MTOW updated with component masses and useful load:

$$\boxed{\begin{aligned} MTOW &= EW + UL = EW + m_{Payload} + m_{Fuel} + m_{Crew} = \\ &= m_{MR} + m_{TR} + m_{F\&T} + m_{LG} + m_{Prop} + m_{Drive} + m_{Tanks} + m_{FCS} + m_{Aux} + m_{I} + \\ &+ m_{Hyd} + m_{El} + m_{Av} + m_{Furn} + m_{AC\&AI} + m_{L\&H} + m_{Payload} + m_{Fuel} + m_{Crew} \end{aligned}}$$

$\Rightarrow$ Based on this updated result for the MTOW the component masses can be updated.

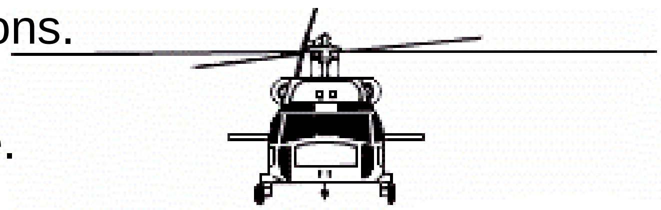
$\Rightarrow$ Better and better results can be achieved iteratively until the deviation between the change in MTOW per iteration falls below a predetermined threshold ε.

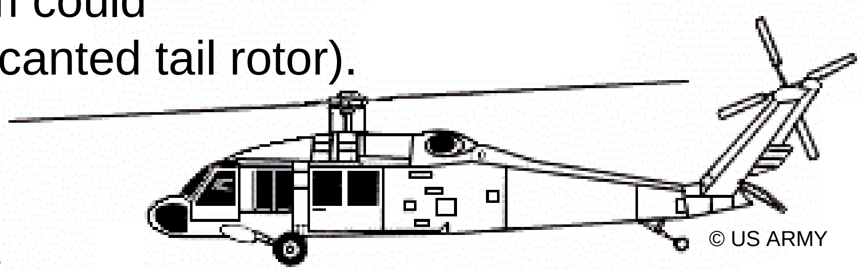
$\Rightarrow$ In case the selected disc-loading cannot be achieved with an updated MTOW it might be necessary to increase the main rotor radius.

5.2.2 Determination of Center of Gravity Position

Optimal center of gravity position:

- The optimal center of gravity position along the longitudinal axis in a helicopter is just before the main rotor. Also in the hands of experienced designers, the CG “slides” somewhat behind the main rotor in the process of preliminary design.
- **Example:** During the development of the UH-60A, the total length of the helicopter was a limiting factor since it had to fit into a C-130 Hercules without having to undergo major modifications. Because the main rotor radius was already fixed the fuselage had to have a short nose. Also the fuel tanks situated behind the passenger cabin led to the CG shifting backwards. The resulting trim problem could only be solved by tilting the tail rotor (canted tail rotor).





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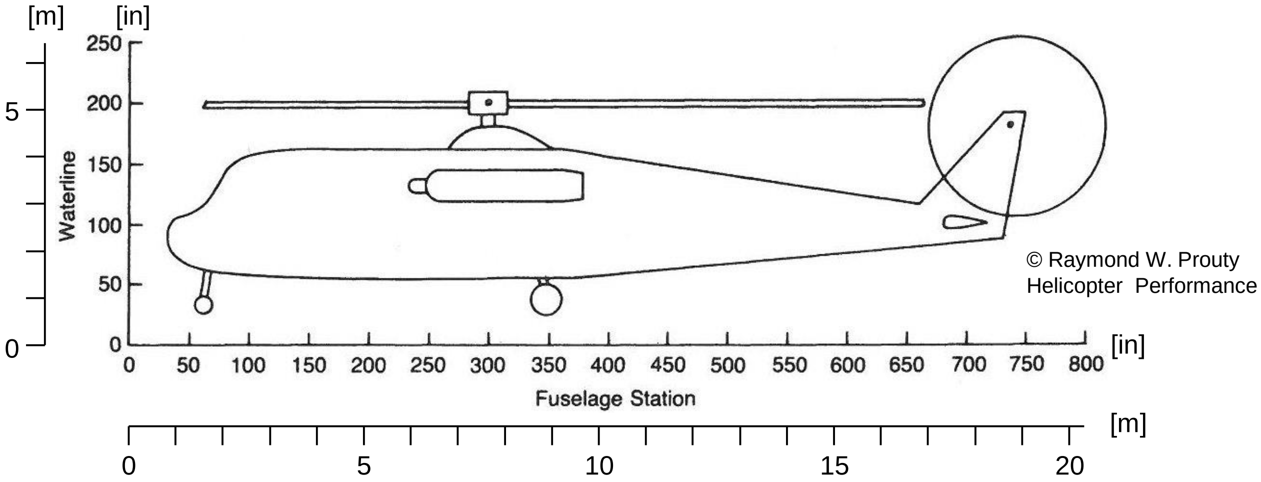
5.2.2 Determination of Center of Gravity Position

Determining the design coordinate system:

- The CG position along the longitudinal axis of an empty helicopter can be determined from the equilibrium of static moments caused by the mass parts around an arbitrary point. This point should be in front of and underneath the helicopter nose so as to produce positive values for all the moments:



5.2.2 Determination of Center of Gravity Position

CG position of empty helicopter:

The center of gravity position along the longitudinal axis of the helicopter can be determined with the already established mass parts and their longitudinal distance from the coordinate system origin :

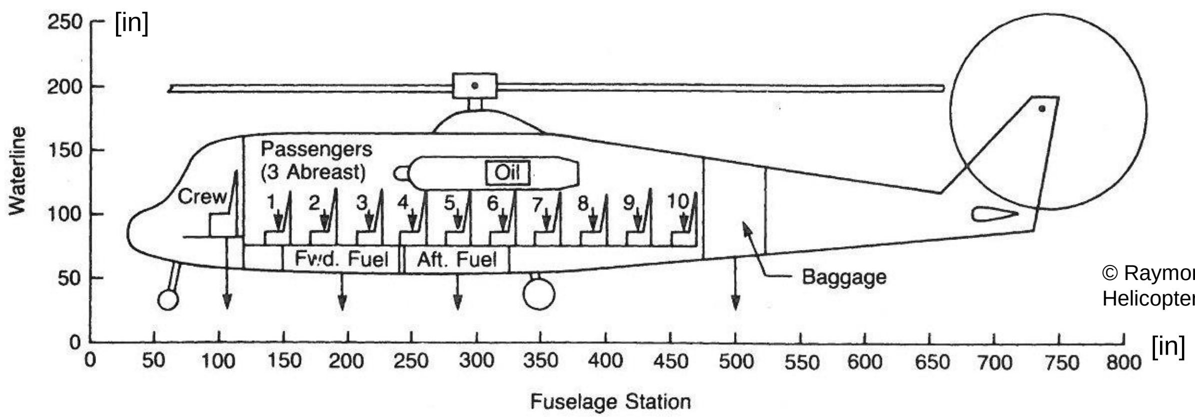
$$\boxed{CG_{long,empty} = \frac{\sum M_i}{EW}} \quad [m]$$

$\Rightarrow$ The result will get more precise as the design process progresses, since more and more detailed information about individual components come together as well as their position within the helicopter.

5.2.2 Determination of Center of Gravity Position

CG position of loaded helicopter:

- For determining the center of gravity position of a loaded helicopter all elements of the **useful load** that cause a CG shift have to be taken into account (crew, fuel, lubricants, passengers and luggage i.e. **load items)**.



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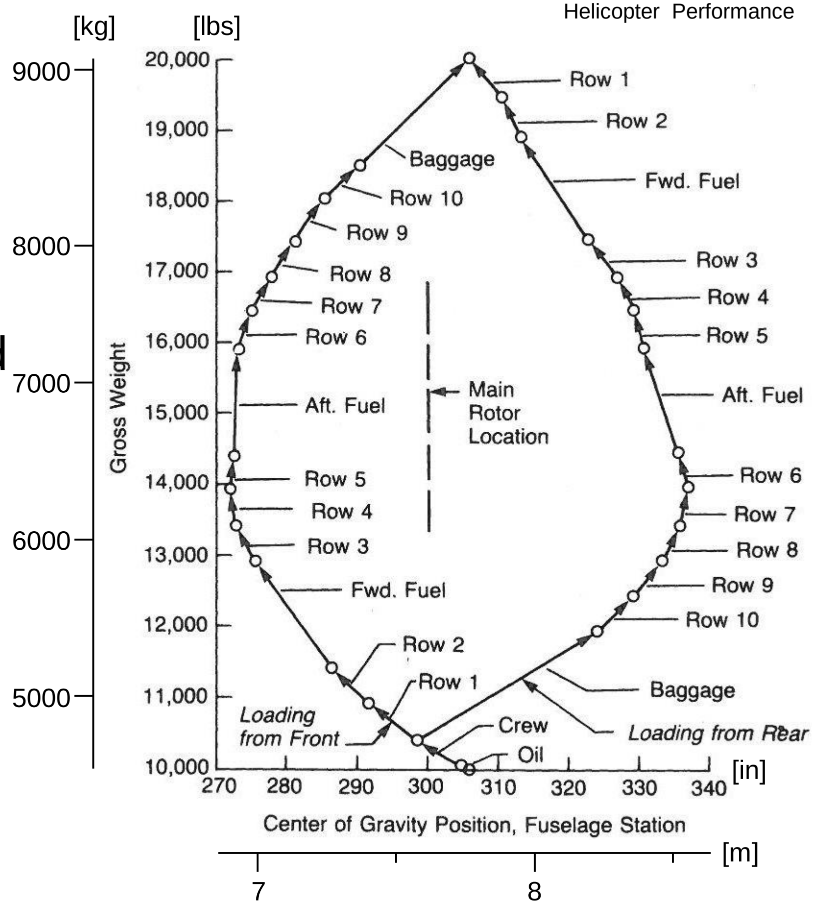
$$\boxed{CG_{long,loaded} = \frac{M_{ges} + \sum_{j=1}^{n} M_{loaditem,j}}{EW + \sum_{j=1}^{n} W_{loaditem,j}}}$$ [m]

5.2.2 Determination of Center of Gravity Position

CG position of loaded helicopter :

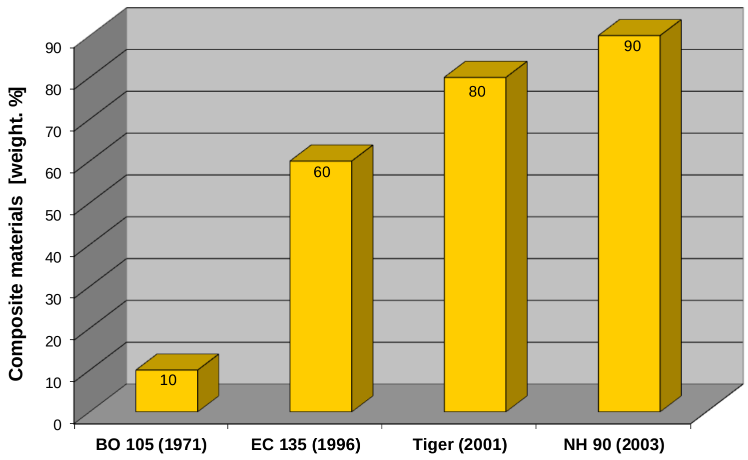
- The loading process from the front/back shows the respective individual curve of the CG shifting in a diagram.
- When loading in a different order than shown, the momentary CG and the new curve will always be between the displayed curves in the diagram.
- All curves meet at the exact same point when the helicopter is fully laden.



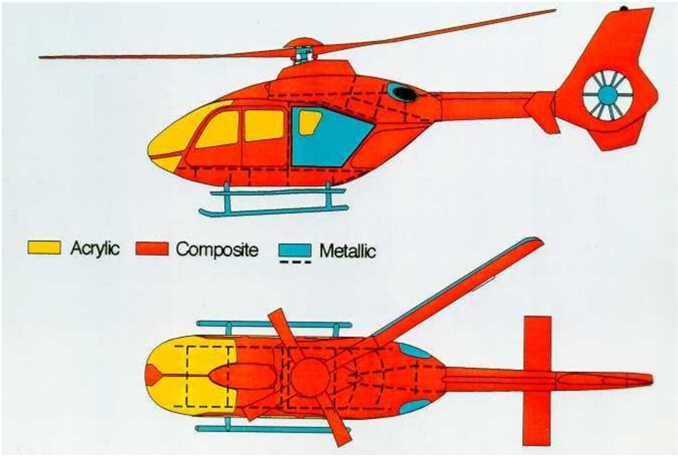
5.2.3 Mass Optimization

- The helicopter mass optimization is carried out through reduction of component masses with the help of light weight materials and structures.
- Any saved component mass can also be used to optimize the CG position, to increase climb performance, maximum payload, fuel capacity or range.

Increasing proportion of composite materials



Materials distribution EC135



5.2.3 Mass Optimization

Examples of lightweight materials:

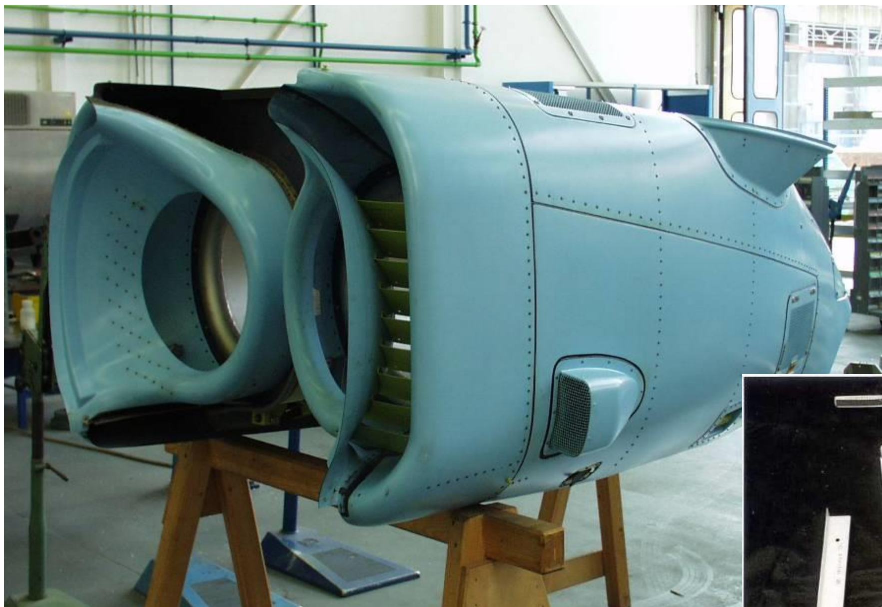
- Fiber Reinforced Composites: Carbon Fiber Reinforced Polymer(CFRP), Glass Fiber Reinforced Polymer (GFRP), Aramid Fiber Reinforced Polymer, Hybrid Composites(CFRP/GFRP)
- Metals: Titan, Aluminium, Steel
- Polymers
- Textiles
- …

Examples of lightweight structures:

- Forming, stretch forming of metals
- Composite structures
- Prepreg laminates
- …

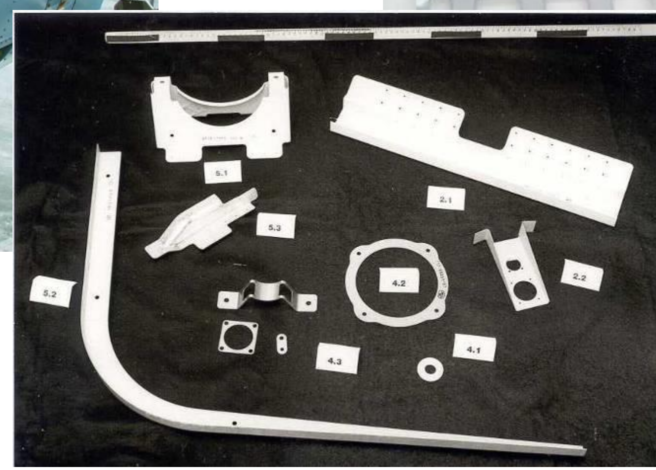
5.2.3 Mass Optimization

Examples of lightweight structures – non-cutting manufacturing (forming, stretch forming) of aluminium, titan and steel:



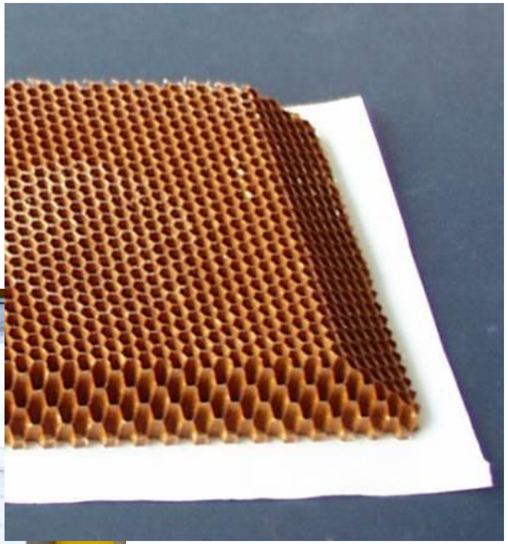
IR Suppressor: Tiger

Cargo door panel



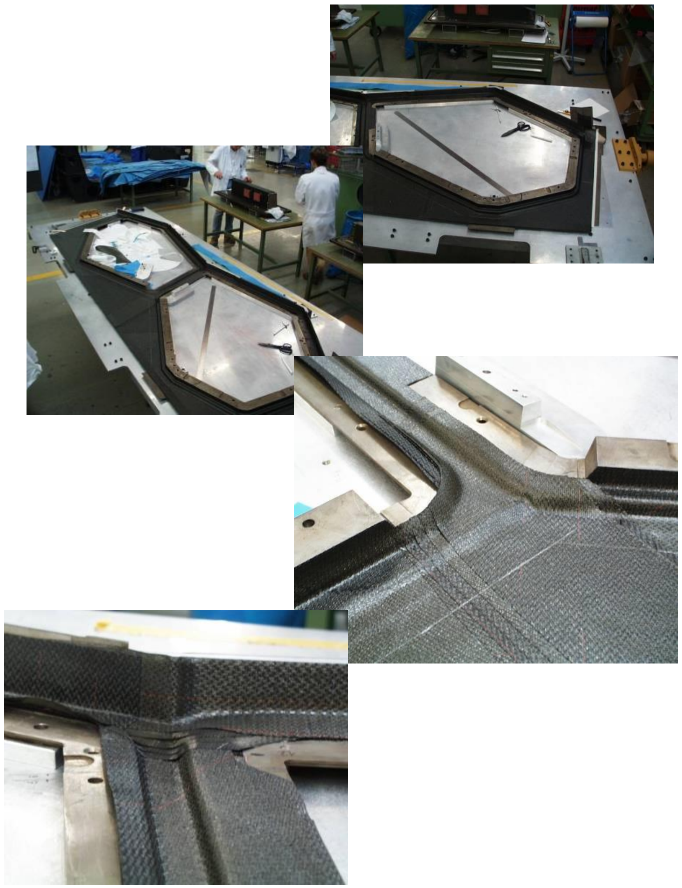
5.2.3 Mass Optimization

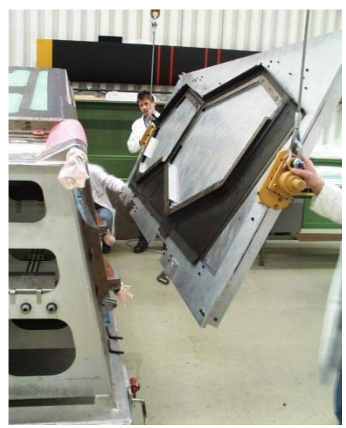
Examples of lightweight structures – Composite – structures (CFRP,GFRP, epoxide, honeycomb, foams, …):



5.2.3 Mass Optimization

Examples of lightweight structures – Prepreg - laminates:





Tiger- cabin structure

5.2.3 Mass Optimization

- The different approaches of mass optimization through application of lightweight design will not be addressed in detail in this lecture.
- Using lightweight design usually leads to higher material and production costs.
- This results in a conflict of objective: the optimal mass configuration has to be confronted with a cost estimation and optimization.

5.3 Cost Estimation and Optimization

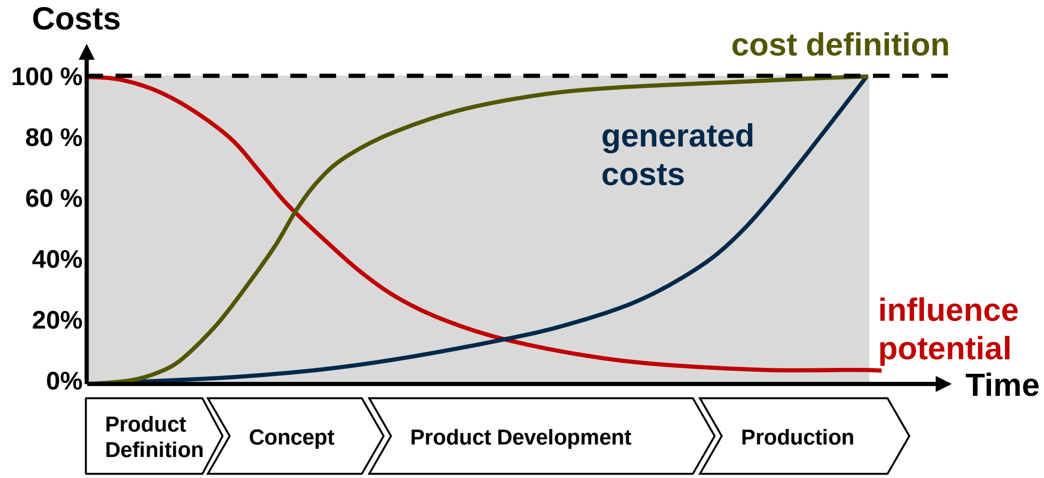
5.3.1 Cost Incurrence and Composition

- Contrary to technical data (i.e. power, mass) there is much less statistical data available for cost estimation.
- Especially research and development costs from competitors are rarely available. Therefore every company makes an effort to establish a consistent cost monitoring system.
- The scaling factors for helicopter components can be mass, applied technology, complexity (e.g. number of parts), …
- The cost estimation conclusions serve as a measure for the product's success on the market (operational costs, product performance). Development and production costs along with the possible market price and sales forecast serve as key factors in the developer's/manufacturer's profitability calculation.
- It takes approx. 5 years after certification and rollout of the product for the development program to start making profit.

5.3.1 Cost Incurrence and Composition

Influence potential on costs:

- In order to make a cost estimation and optimization all life cycle costs have to be taken into account and examined in detail. The potential to have any influence on generated costs drops as the development life cycle progresses to next phases. Generated costs however increase.



- Therefore it is of utmost importance to determine the costs generated by the selected components early on in the development and to compare them with alternative configurations.

5.3.1 Cost Incurrence and Composition

Lifecycle cost composition of a component:

To determine which costs are incurred by the component development all lifecycle costs LC of the component have to be established:

- **Research and Development** $RD \; [€]$
- **Initial Purchase** $IP \; [€]$
- **Operating Costs** $OC = YO \cdot SL \; [€]$
- **Maintenance Costs** $MC = YM \cdot SL \; [€]$
- **Replacement Costs** $RC = RP - SV \; [€]$

$YO \; [€/a]$	yearly operation cost	$RP \; [€]$	replacement price
$SL \; [a]$	service life	$SV \; [€]$	salvage value
$YM \; [€/a]$	yearly maintenance cost		

Another important factor for the lifecycle cost is the:

- **Mean Time Between Replacement** $MTBR \; [h]$

5.3.2 Cost Estimation and Optimization

For the calculation of operational and maintenance cost the following is needed

- **Yearly Operational Costs** $$YO = HO \cdot AOH \cdot n_{component} \quad [€/a]$$
- **Yearly Maintenance Costs** $$YM = HM \cdot AOH \cdot n_{component} \quad [€/a]$$
- **Number of required replacements** $$N_{RPL} = \underbrace{\frac{MTBR_{helicopter}}{MTBR_{component}}}_{\text{Round up to an integer!}} - 1 \quad [-]$$

$HO\ [€/h]$ hourly operational costs

$AOH\ [h/a]$ yearly operation hours

$HM\ [€/h]$ hourly maintenance costs

$n_{component}\ [-]$ number of installed components of the same type

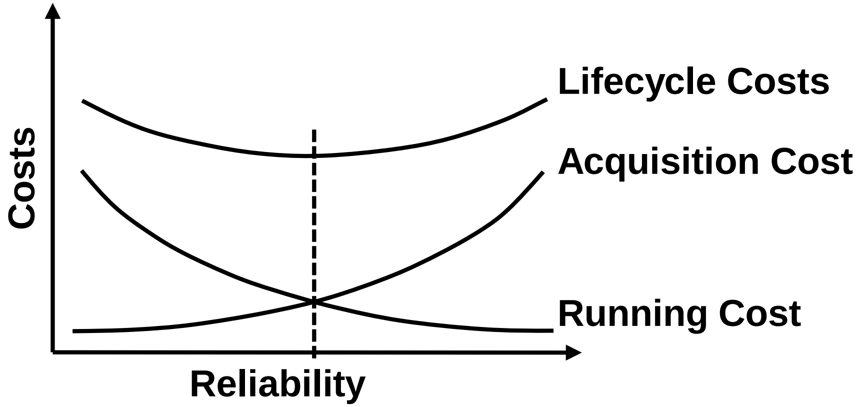
- With these results the **Lifecycle Costs** can be determined:

$$\boxed{LC = n \cdot \left[RD + IP + SL \cdot (YO + YM) + N_{RPL} \cdot (RP - SV)\right] \quad [€]}$$

5.3.2 Cost Estimation and Optimization

Lifecycle cost minimum:

The lifecycle costs of component development is at its minimum (a good compromise between costs and reliability) when the **sum** of **acquisition cost** (research, development and procurement costs) and **running cost** (operational, maintenance, and replacement costs) is **minimal**:



⇒ From a cost-cutting point of view helicopter components should be selected with the requirement of having minimal lifecycle cost.

5.3.2 Cost Estimation and Optimization

Final conclusions:

A number of additional aspects have to be accounted for when selecting helicopter components, e.g.:

- safety regulations ($\Rightarrow$component redundancy)
- mass
- availability
- maintainability
- reliability
- performance

$\Rightarrow$ Optimal points of the selection criteria seldom coincide.

$\Rightarrow$ Furthermore the criteria can be subject to different weighting.

$\Rightarrow$ An “optimal” component selection can only be achieved when **all** relevant selection criteria have been considered.

5.4 Criteria for Component Selection and Development

5.4 Criteria for Component Selection and Development

Introduction:

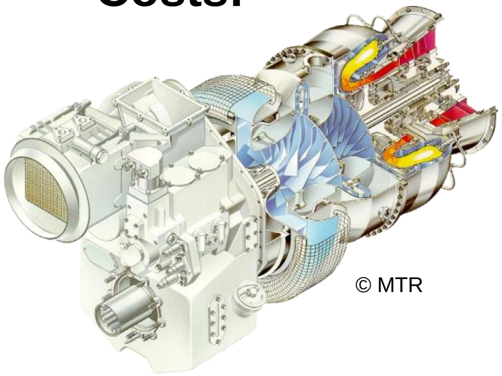
- The selection of a component is influenced by a large number of component specific target figures, which interact with each other. The target figures can vary greatly depending on component type.
- The multitude of available alternatives often consists of discrete solutions, therefore analytical or numerical optimization methods are suitable only to a limited extent for selecting components.
- The selection or development of a specific component may be based on a generalized model, comprised of the target figures: time, quality (fulfilling all requirements) and costs.
- The fulfillment of these targets ought to be carefully monitored during the component development process with the help of suitable project management tools.

5.4 Criteria for Component Selection and Development

Example of engine selection criteria:

Third-party production/development ⇨ Based on the limited range of available engines, the assortment is highly **discrete**

- **Power:** - 1 / 2 – engine power
  - fulfilling requirement (≤100% or >100%?)
- **Consumption:** - technology influences the price
  - specific fuel consumption influences MTOW
- **Mass:** - absolute
  - kg/kW
- **Costs:** - procurement (producer bears development costs)
  - maintenance costs ((special) tooling,…)
  - operational costs (depreciation, insurance, fuel consumption,…)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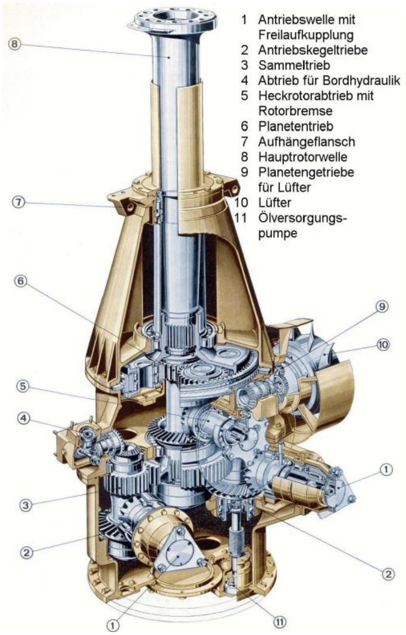
5.4 Criteria for Component Selection and Development



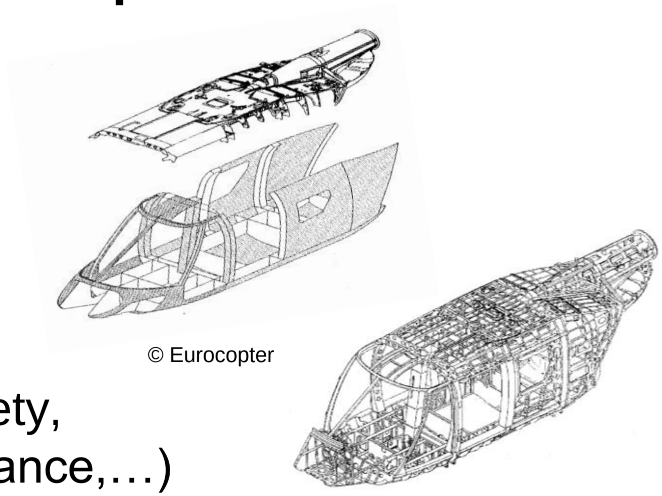
Example of transmission selection criteria :

In-house <u>and</u> third-party production/development
(i.e. BO105 ⇨ ZF, Dauphin ⇨ Fiat Avio)

- **Power Throughput:** - fulfilling requirement (≤100% or >100%?)
  - Design-to-spec or off-the-shelf?
- **Mass:** - max. transmissible torque
  - OEI-performance
  - number of stages
  - gearing technology / type
- **Costs:** - one-time costs (i.e. development, …)
  - recurring costs (i.e. production/ procurement)
  - maintenance costs

⇨ Also here there are **discrete** parameters, such as number of stages, gearing technology and type,…

5.4 Criteria for Component Selection and Development


© Eurocopter

Example of structure selection criteria :

In-house and third-party production/development

- **Mass:**
  - material/design
  - design appropriate for the material?
  - requirements(crash safety, fire and corrosion resistance,…)
- **Costs:**
  - material costs
  - processing costs
  - initial investment in tools and technology

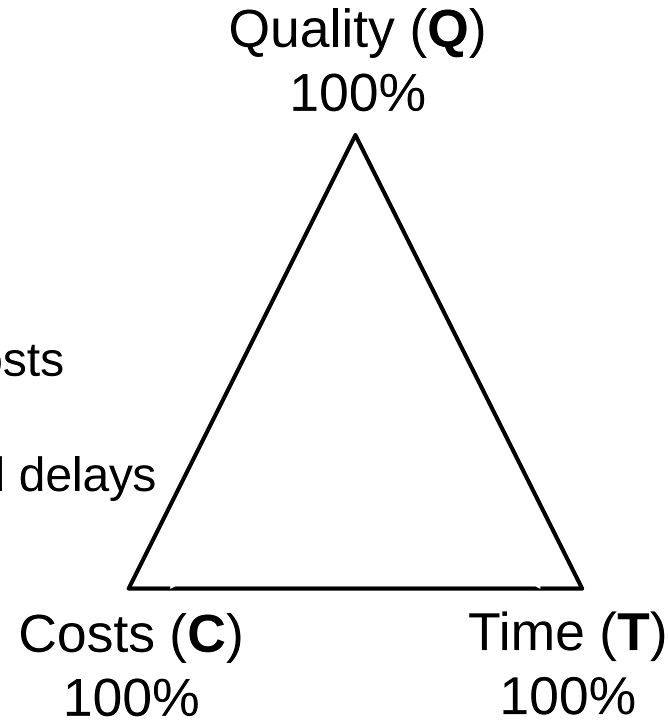
⇨ Again **discrete** parameters emerge, such as material, design,…

5.4 Criteria for Component Selection and Development

General goal criteria for component selection/development:

- **ideal component selection/development:**
  simultaneous fulfillment of quality (**Q**),
  costs (**C**) and time (**T**) at 100 %

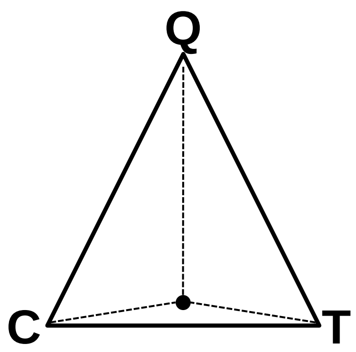
- **real-life component selection/development :**
  - improvement of quality mostly leads to higher costs
  - reduction in time mostly leads to higher costs
  - cost reduction mostly leads to loss of quality and delays



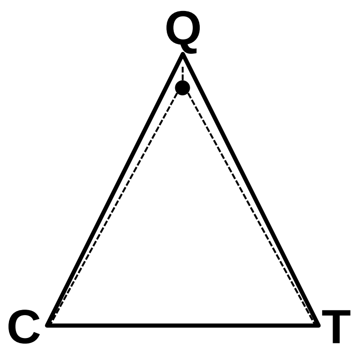
5.4 Criteria for Component Selection and Development

Emphasis of goal criteria in practice:

Fulfillment of cost and time frame but failure to fulfill requirements:



Fulfillment of requirements but at the same time exceeding the budget and time frame:



⇒The QTC-triangle is an indispensable tool for weighting the general goal criteria during the selection and development processes.

⇒The fulfillment of goals requires constant monitoring during the development process with the help of project management.

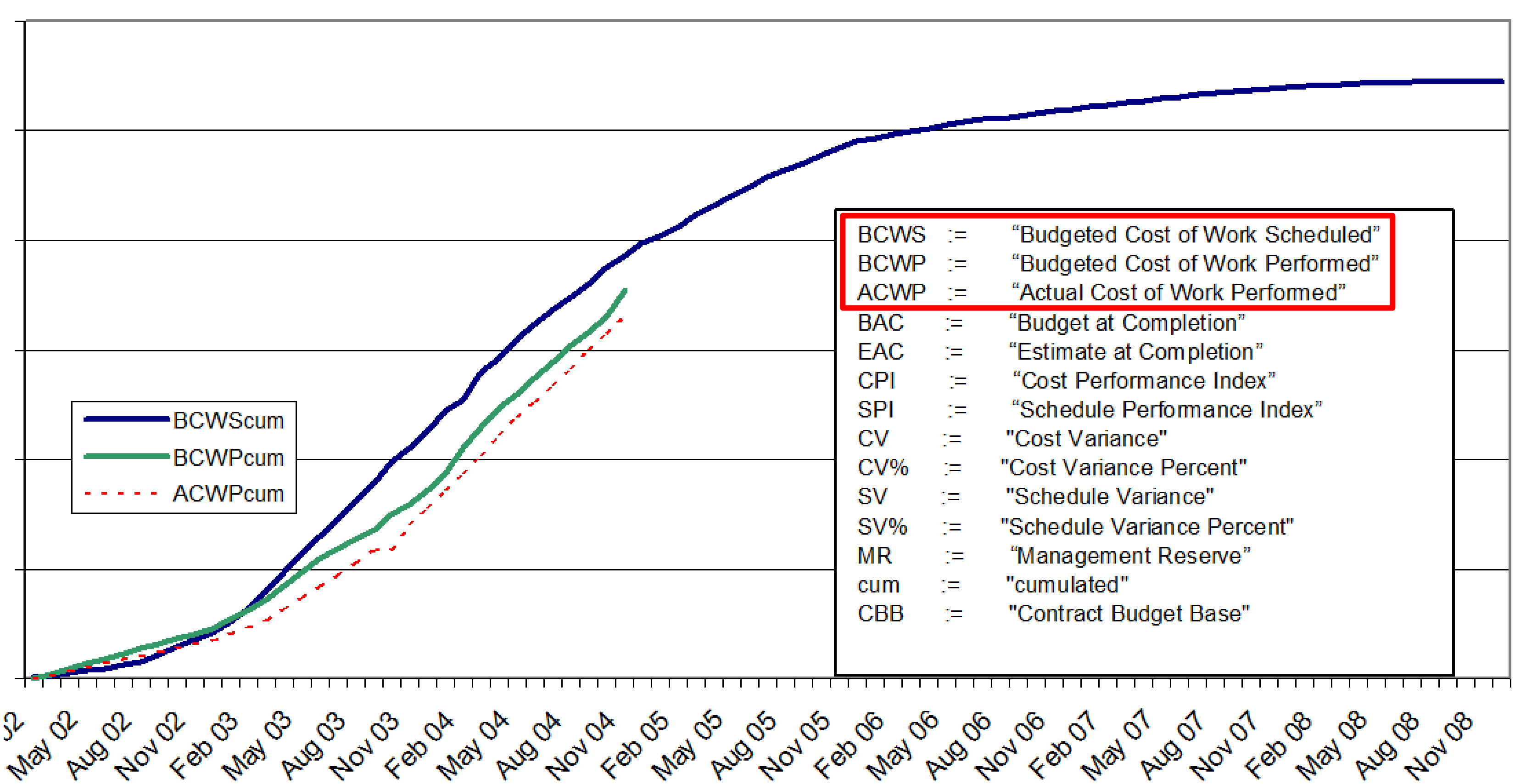
5.4 Criteria for Component Selection and Development

Monitoring the goal fulfillment of development projects using Earned Value Management; EVM:

- progress evaluation of projects
- presentation of current schedule and cost situation using **figures**:
  - **budgeted cost of work planned**
  - **budgeted cost of work performed**
  - **actual cost of work performed**

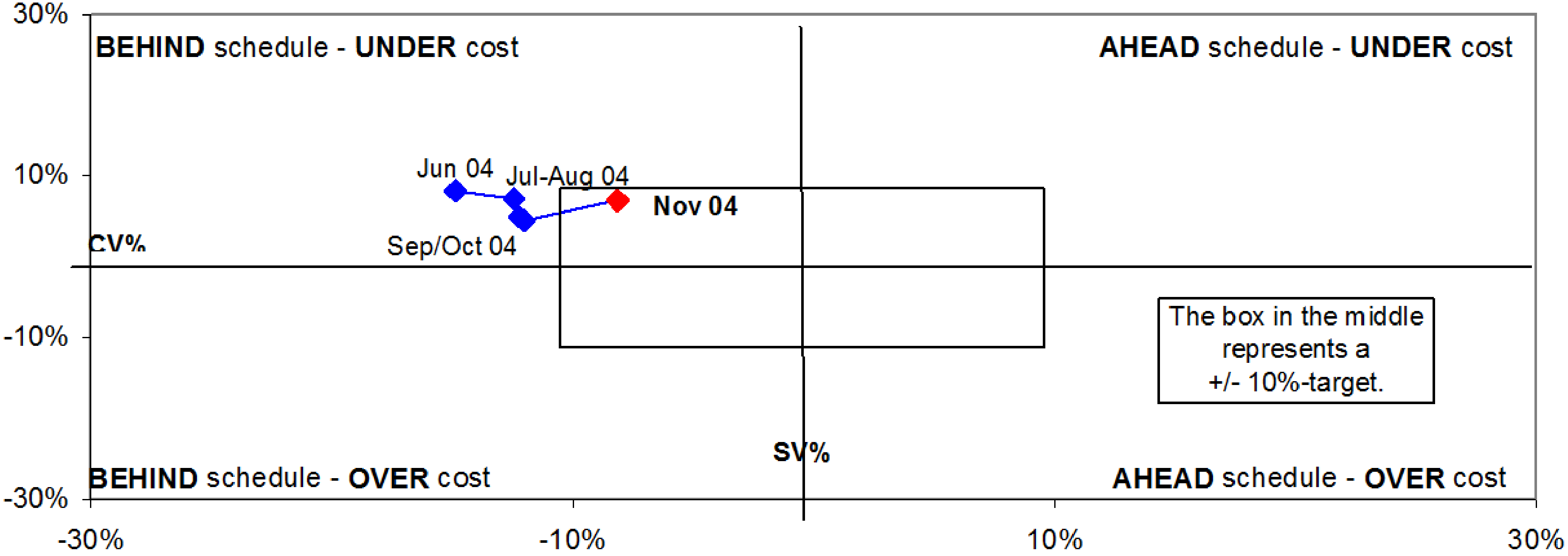
5.4 Criteria for Component Selection and Development

Monitoring the goal fulfillment of development projects using Earned Value Management; EVM:



5.4 Criteria for Component Selection and Development

Monitoring the goal fulfillment of development projects using Earned Value Management; EVM:



6 Architecture and Component Design

6 Architecture and Component Design

6.1 Product Development Process
6.2 Rotor Hub Types
6.3 Rotor Blades
6.4 Fuselage Design
6.5 Example: Requirement – Transport of a Black Hawk inside of a C-130

6.1 Product Development Process

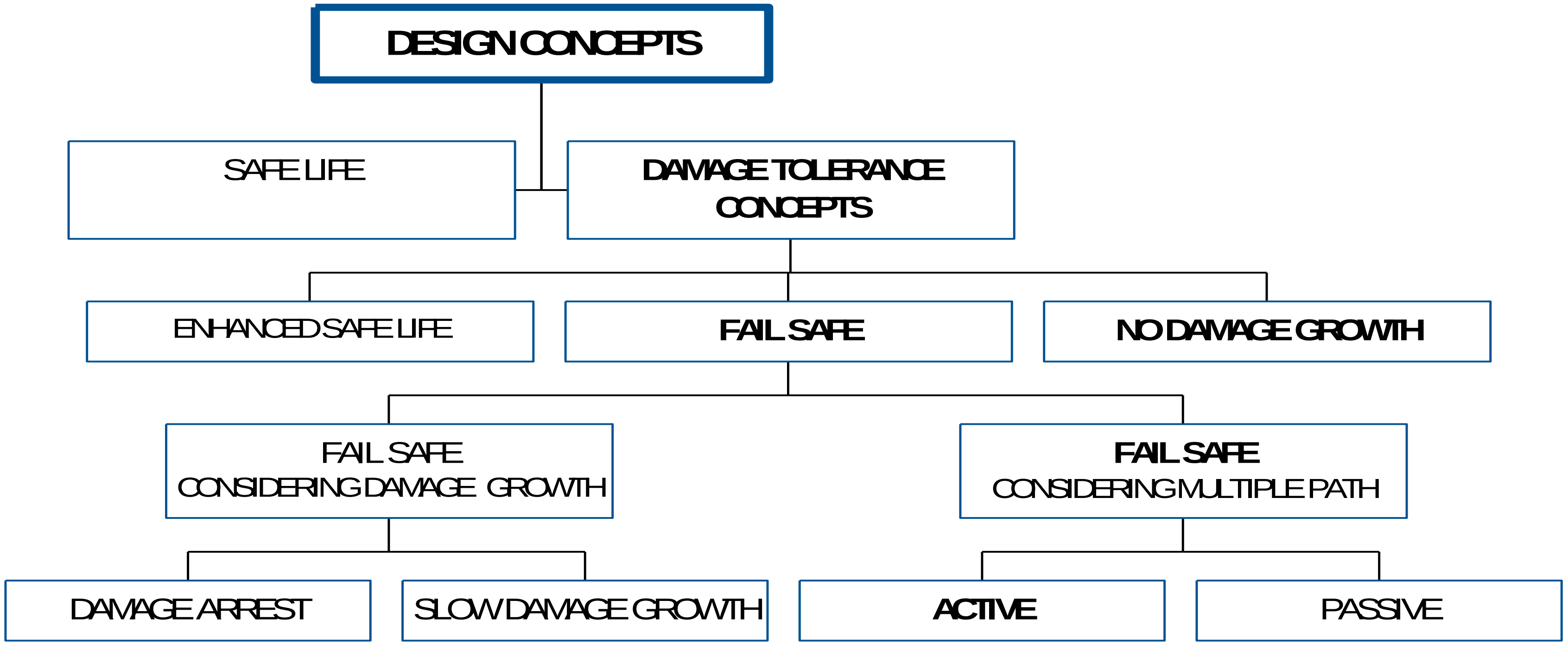
Using the data defined during the **preliminary design** such as **rotor characteristics, mass distribution** and **aerodynamic surface areas** the structural loads resulting from flight and landing maneuvers are calculated with the help of mathematical models. These calculated loads serve as **input variables for the structural design.** This is an **iterative process** since the initial assumptions have to be adjusted to actual data generated during development.

Input variables for the design:

- Design loads for flight, landing and ground operating conditions
- Detailed geometrical data (3D solid-models for FE-meshes)
- Mass distribution of the helicopter
- Requirements and safety factors from design regulations
- Fatigue behavior requirements
- Stiffness requirements

6.1 Product Development Process

Design Concepts Structure:

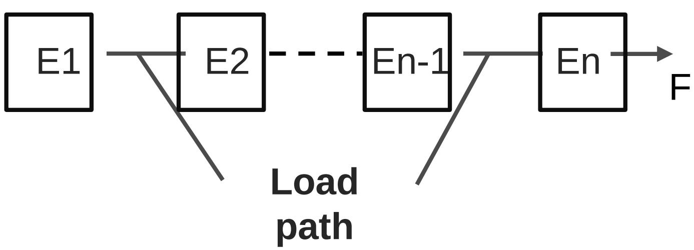


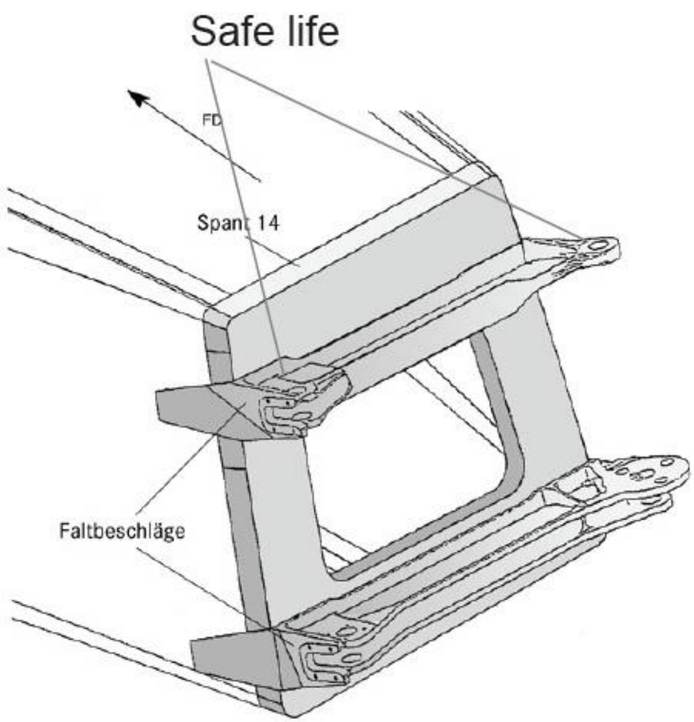
6.1 Product Development Process

Safe Life – only one load path:
Failure of one element leads to failure of overall system

- Reliability and quantity of elements are decisive for the overall system

- All elements have to comply with high safety requirements

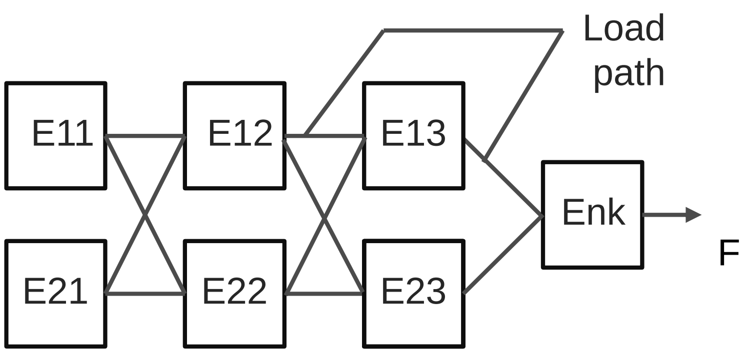




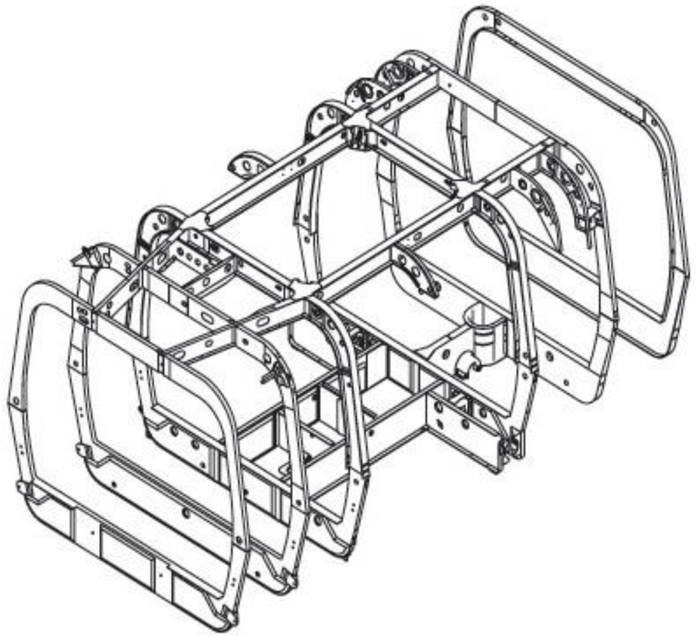
6.1 Product Development Process

Fail Safe – multiple load paths:

- Failure of one element does not lead to overall system failure
- Damage detection is to be ensured through inspection



Fail Safe und Damage Tolerant



6 Architecture and Component Design

6.1 Product Development Process

6.2 Rotor Hub Types

6.3 Rotor Blades

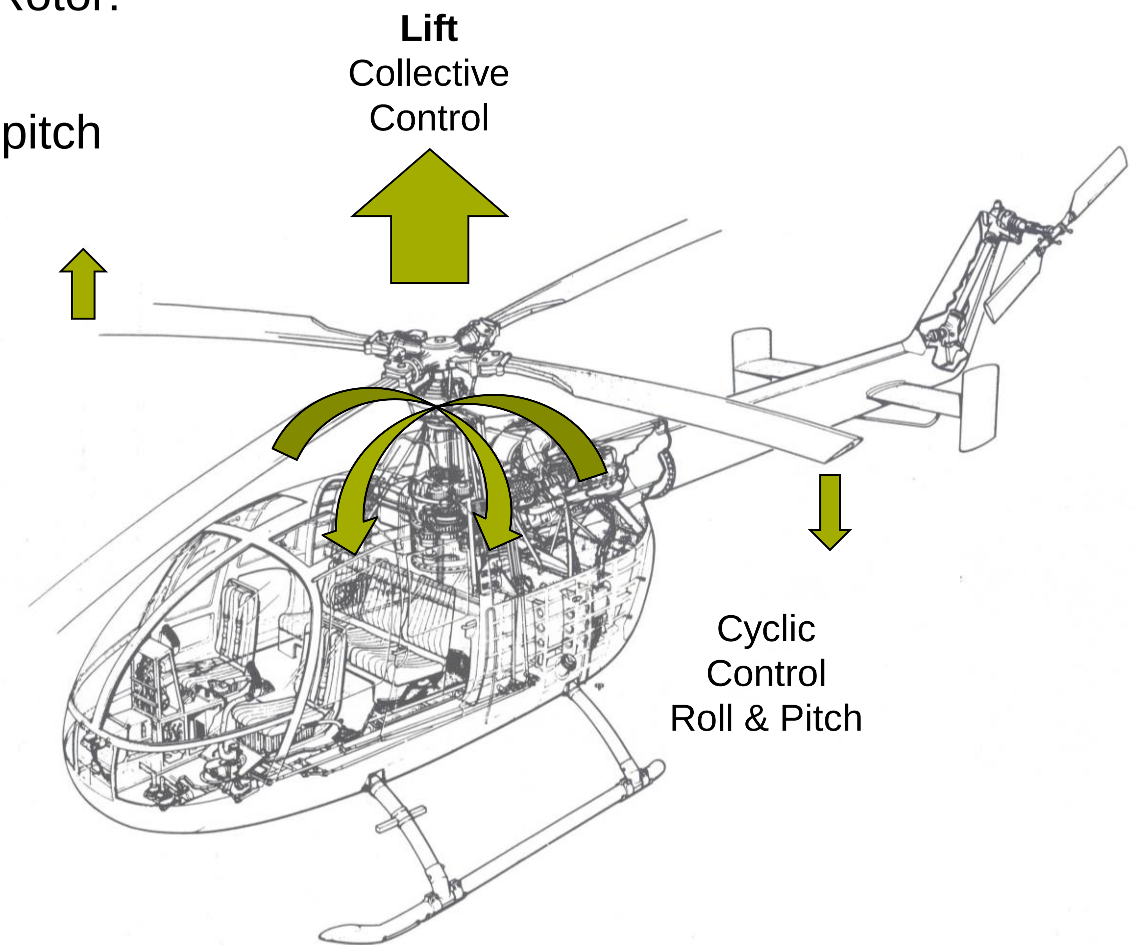
6.4 Fuselage Design

6.5 Example: Requirement – Transport of a Black Hawk inside of a C-130

6.2 Rotor Hub Types

Functions of the Main Rotor:

- Lift
- Control around the pitch and roll axis

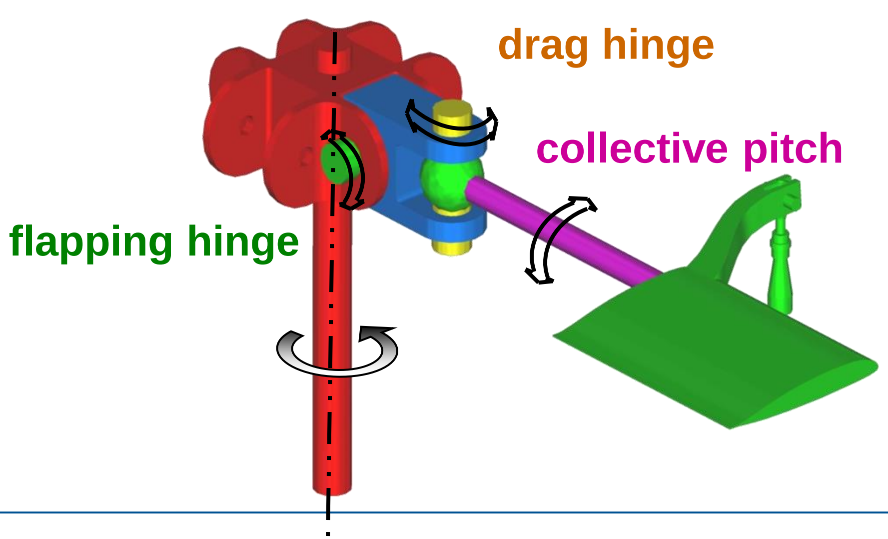


BO 105

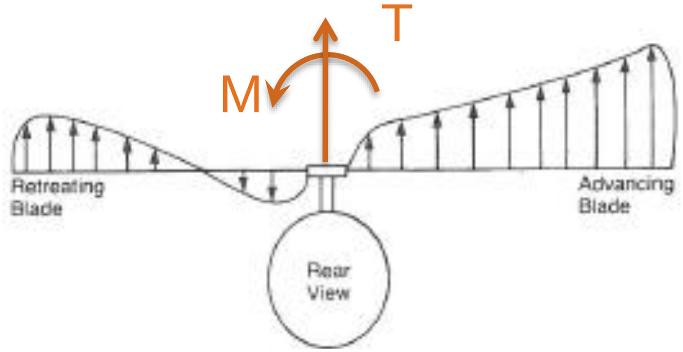
6.2 Rotor Hub Types

Necessity of Hinges:

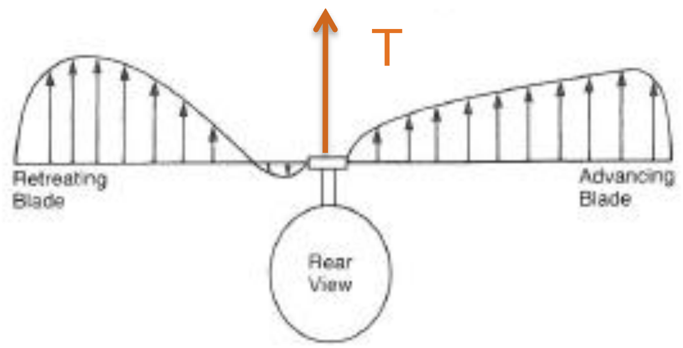
- **Collective Pitch** around the blade longitudinal axis to change angle of attack
- **Flapping Hinge** due to the uneven lift distribution in forward flight
- **Drag hinge** to compensate the back-and-forth acceleration due to the Coriolis effect.



Forward flight:



Lift Distribution on a Rigid Rotor



Lift Distribution on a Hinged Rotor

Abb.:Prouty – Helicopter Performance, Stability and Control

6.2 Rotor Hub Types

Rotor Head Design Factors:

- Low mass
- Low aerodynamic drag
- Low cost
- High durability
- Easy to maintain
- As few individual components as possible
- Appropriate control forces have to be transmitted

Conflict of objectives for the rotor head design:

- On the one hand rotor mass inertia should be high to ensure good autorotation characteristics: minimum of 1sec of hover after engine failure, so the pilot can react – mass has to be placed towards the blade tip;
- On the other hand the mass should be kept small to reduce the rotor head mass as a whole.

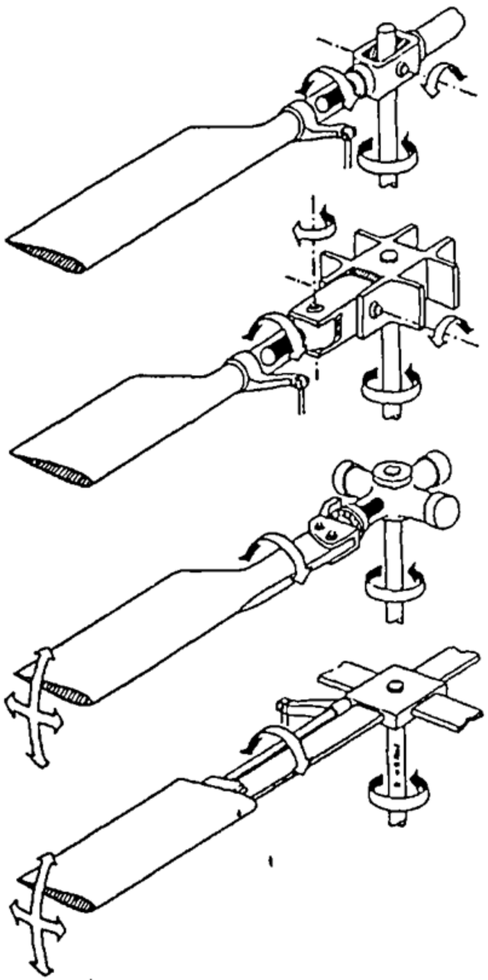
6.2 Rotor Hub Types

Parasitic Aerodynamic Drag:

Component	% of total drag
fuselage	36,5 - 26,5
main rotor head	21,0 - 24,0
tail rotor head	4,5 - 5,5
rotor mast	6,2 - 7,2
landing gear	11,4 - 14,4
vertical fin	2,7 - 3,2
horizontal fin	5,5 - 6,4
rest	12,2 - 12,8
Σ	100

Source: Dr. Kessler - DLR

6.2 Rotor Hub Types: Principles



Teetering	1 feathering bearing 1 central teetering hinge	• Special case: only for two blade rotors
Hinged	3 hinges	• Fully articulated, effective hinge offset
Hingeless	1 feathering bearing	• Flapping and drag hinges are replaced by elastic blade
Bearingless	no hinges or bearings	• Flapping and drag hinges as well as feathering bearing replaced by elastic blade

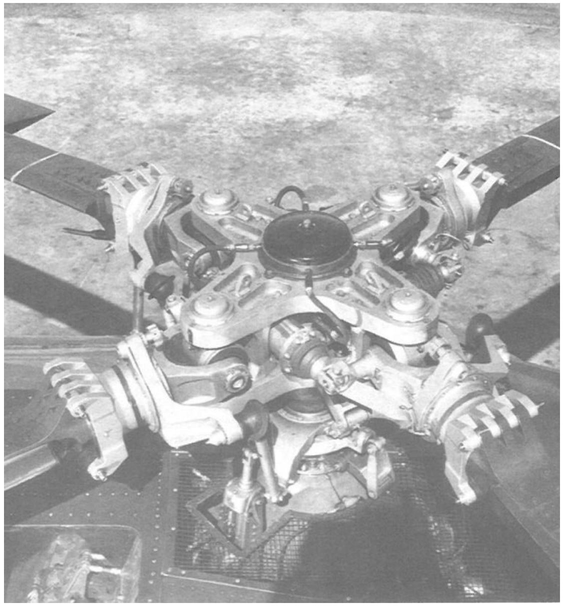
6.2 Rotor Hub Types: Fully articulated

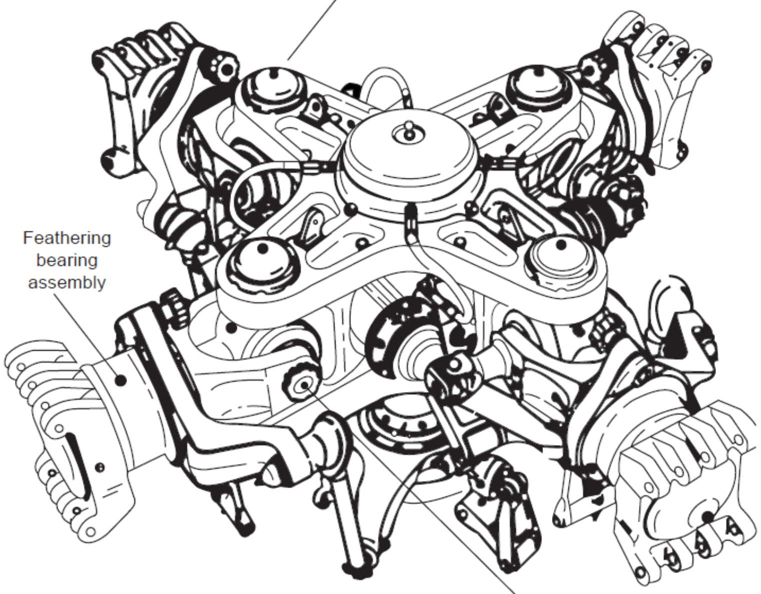
Rotor Head of a Westland Wessex Helicopter

(Source: Bramwell's Helicopter Dynamics)

- High mass
- High amount of components
  → high maintenance effort
- Turbulence

  → Drag

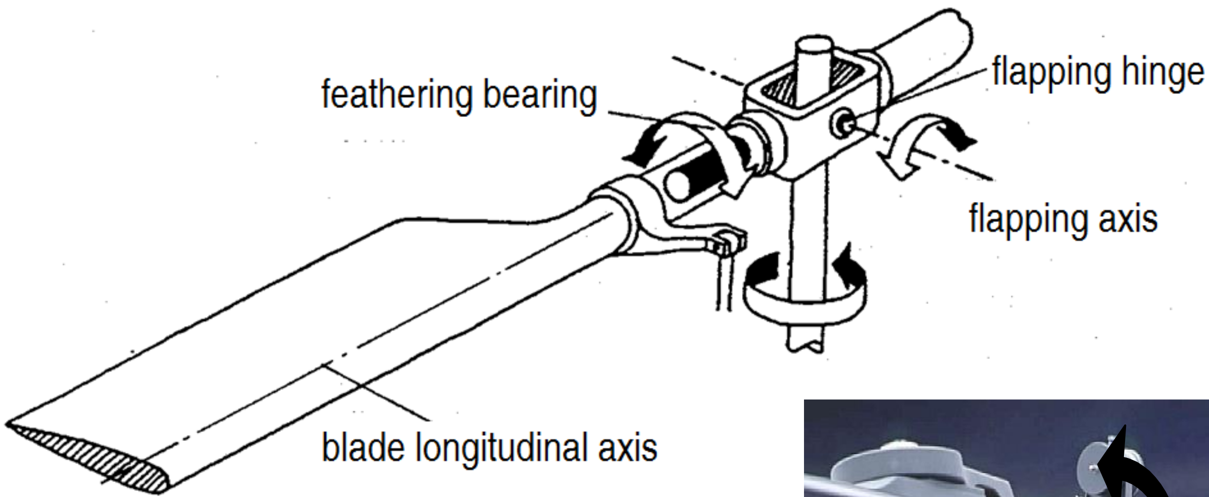




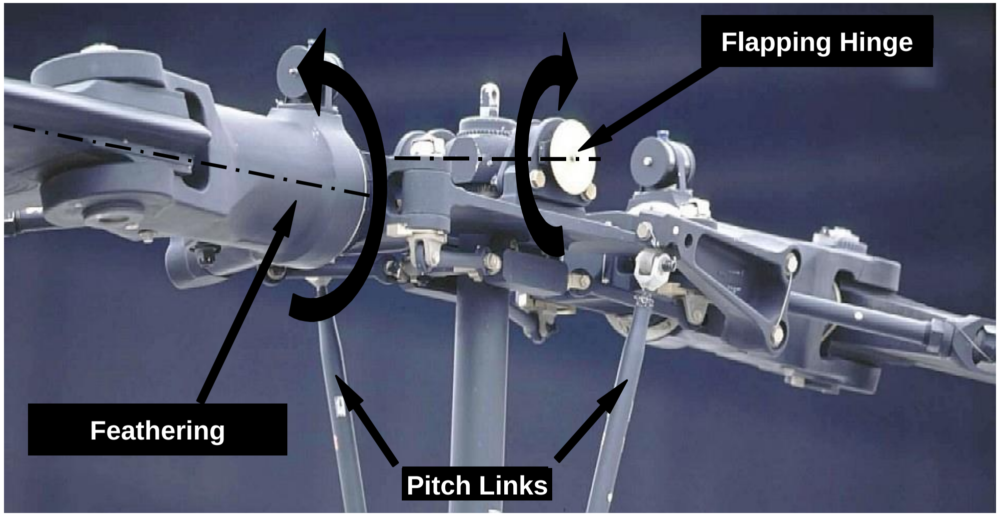
Pic.:Prouty – Helicopter Performance, Stability and Control

6.2 Rotor Hub Types: Semi-rigid

See-Saw Rotor: Bell 206



- Simple rotor
- No drag hinges
- Only possible with two blades

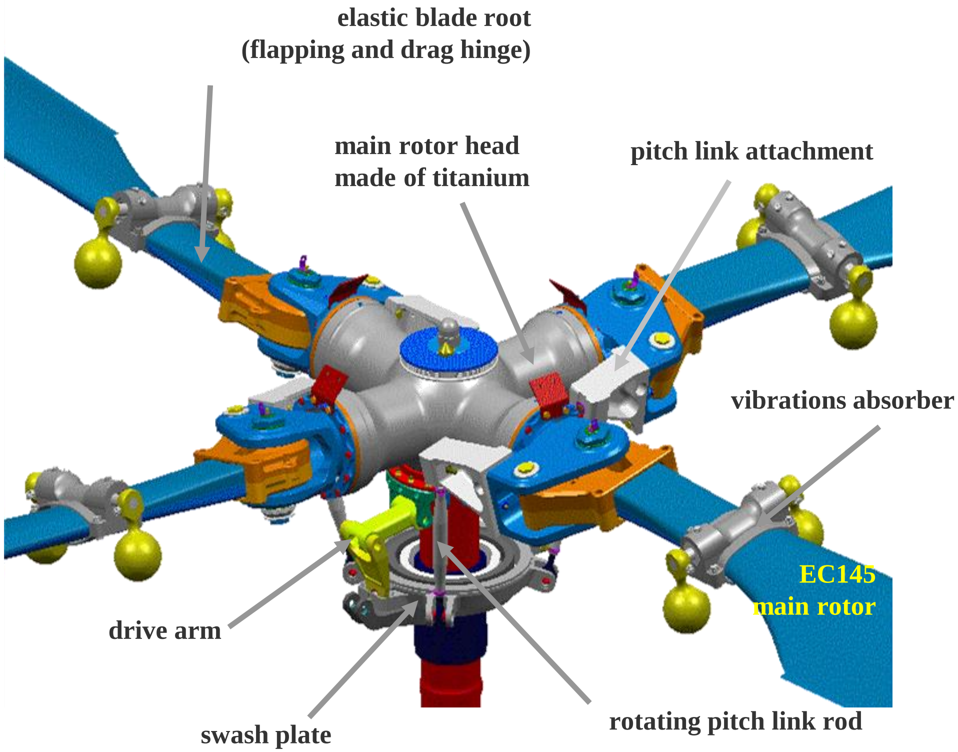


Source: Kuntze-Fechner-EC-HS-Seminar-2004-
"Konstruktion / Bauweisen Rotorsysteme"

6.2 Rotor Hub Types: Hingeless

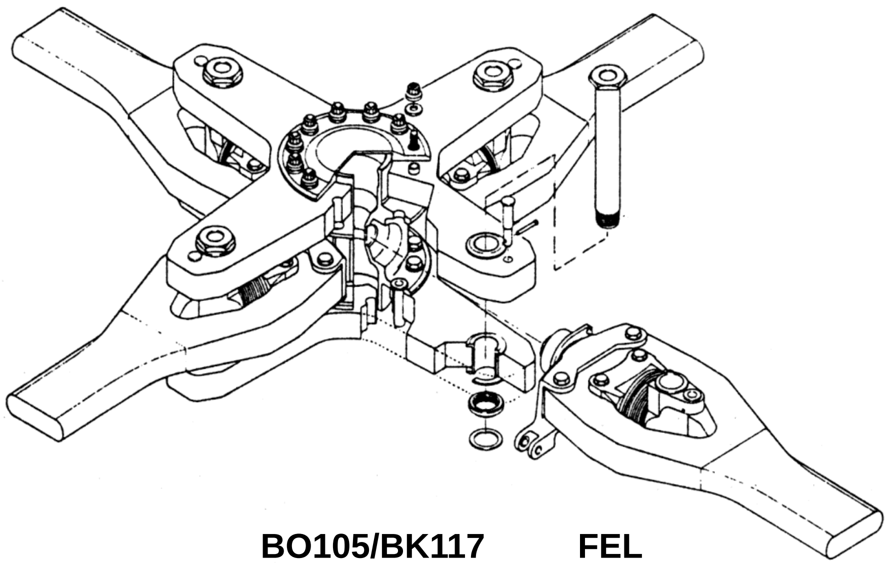
BO 105 / BK 117:

- Rotor without mechanical flapping and drag hinges
- Hinges implemented in fiber composite structure
- Significant simplification of rotor head and mass reduction

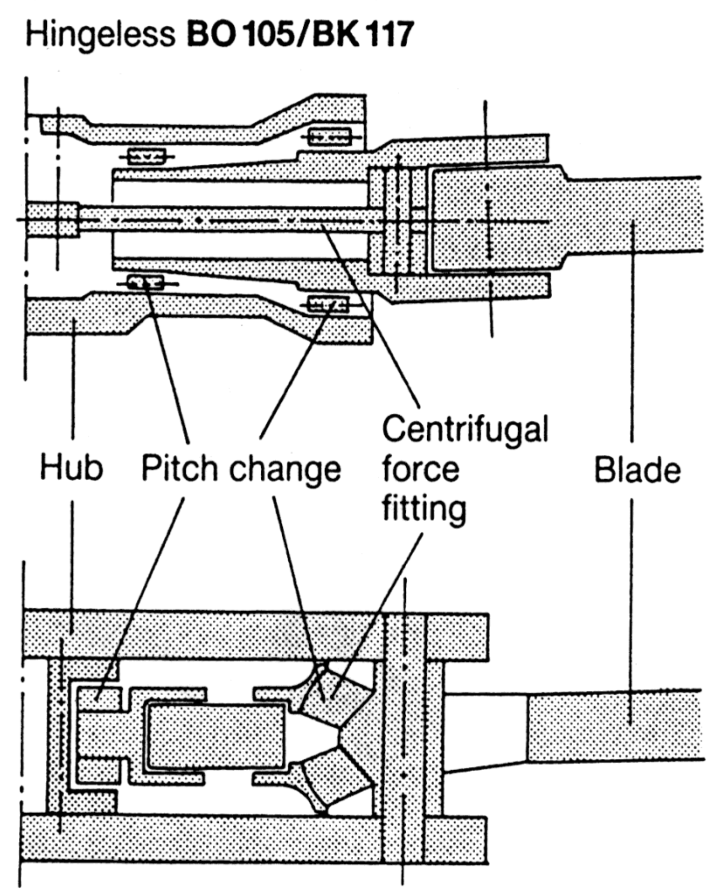


Source: Kuntze-Fechner-EC-HS-Seminar-2004-
"Konstruktion / Bauweisen Rotorsysteme"

6.2 Rotor Hub Types: Fiber ELastomeric Rotor (FEL)



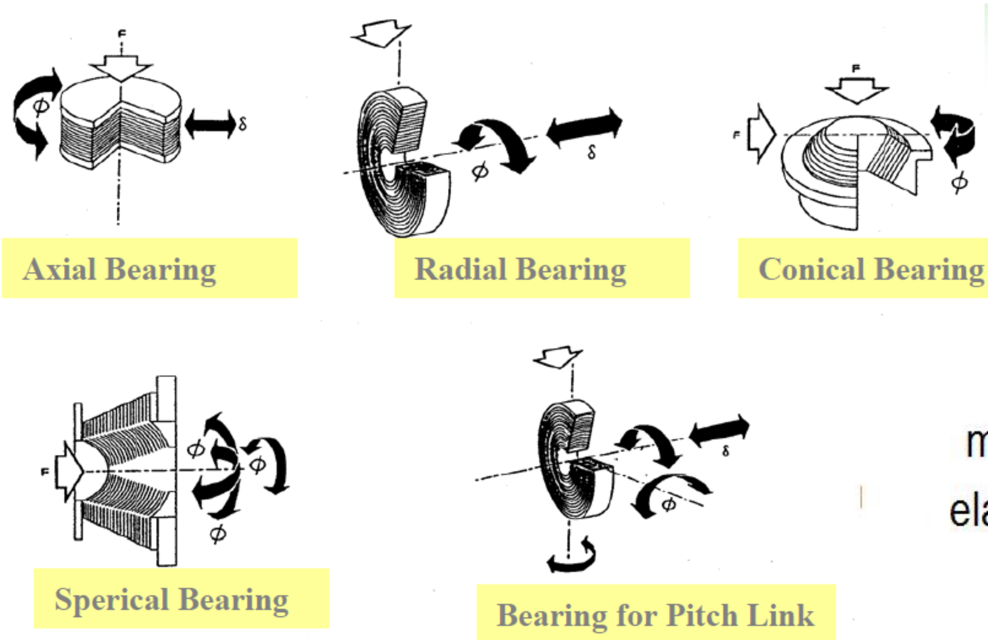
	BO105/BK117	FEL
Hub	Titanium	Fiber composite material plates
Blade pitch bearing	2 roller bearings oil-lubricated	2 elastomeric bearings maintenance-free
Centrifugal force fitting	Bendix element (tie bar)	Conical elastomeric bearing
Blade	Fiber composite flexible	Fiber composite flexible

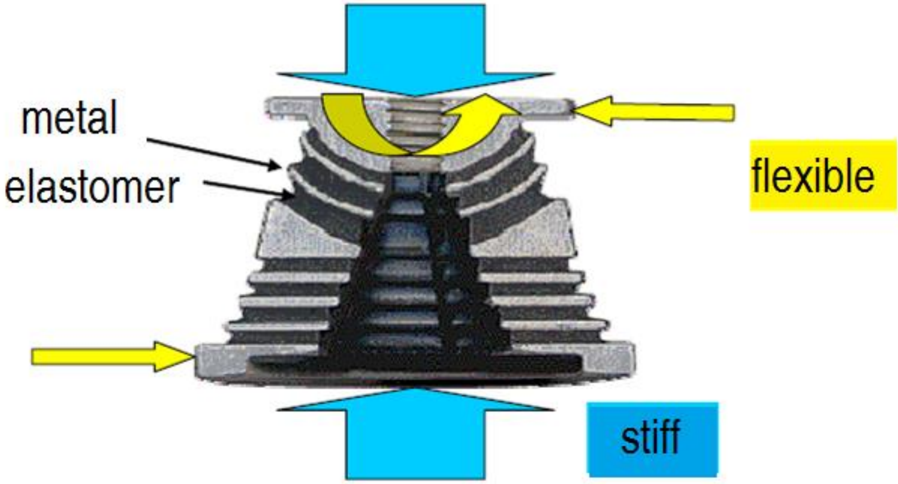


FEL - Tiger

6.2 Rotor Hub Types: Fiber ELastomeric Rotor (FEL)

Elastomeric bearings consist of alternating thin meta and elastomer layers





Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme"

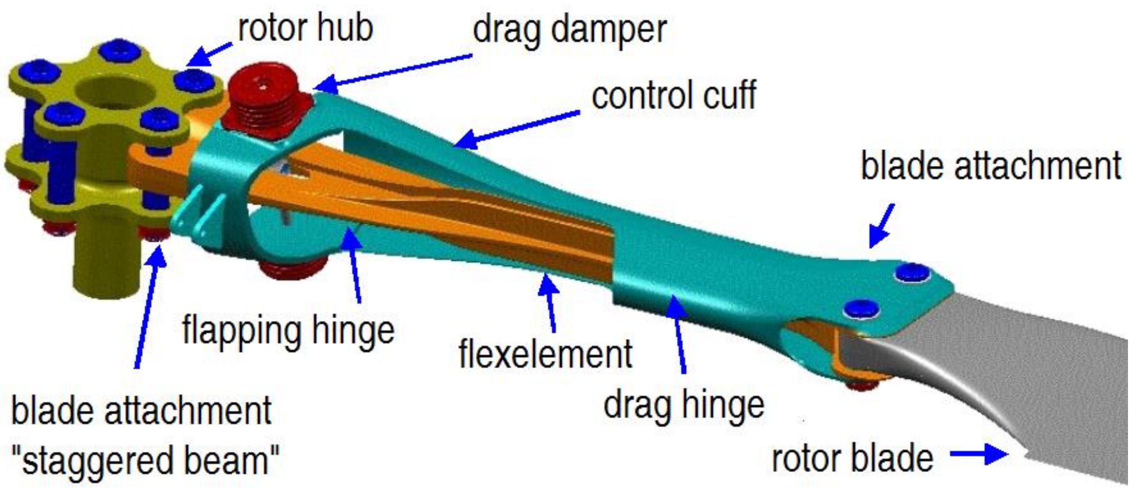
6.2 Rotor Hub Types: Fiber ELastomeric Rotor (FEL)

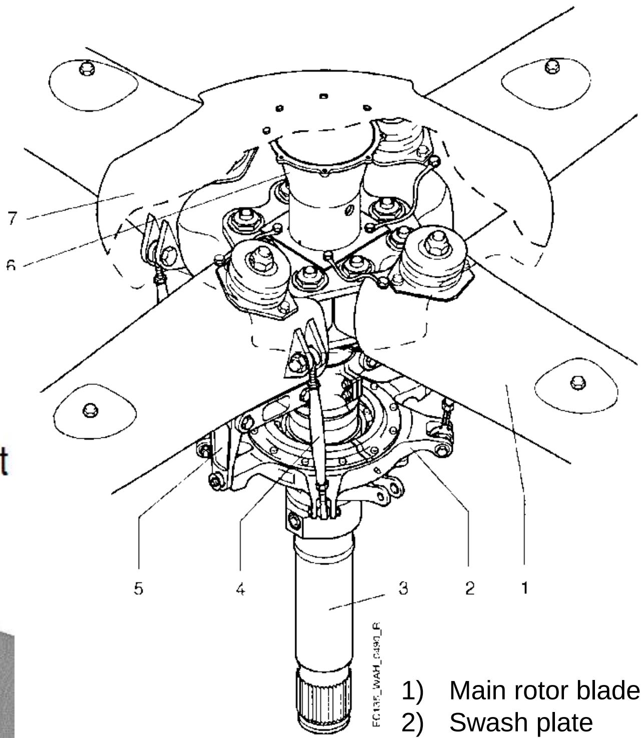
HAL ALH Dhruv (Hindustan Aeronautics Limited - Advanced Light Helicopter)

IDS – **I**ntegrated **D**ynamic **S**ystem

6.2 Rotor Hub Types: Bearingless Blade Attachment EC 135

- Flapping, dragging, pitch as well as dragging dampers implemented with elastic components
- Few, lightweight components
- Low maintenance effort
- Better aerodynamic properties



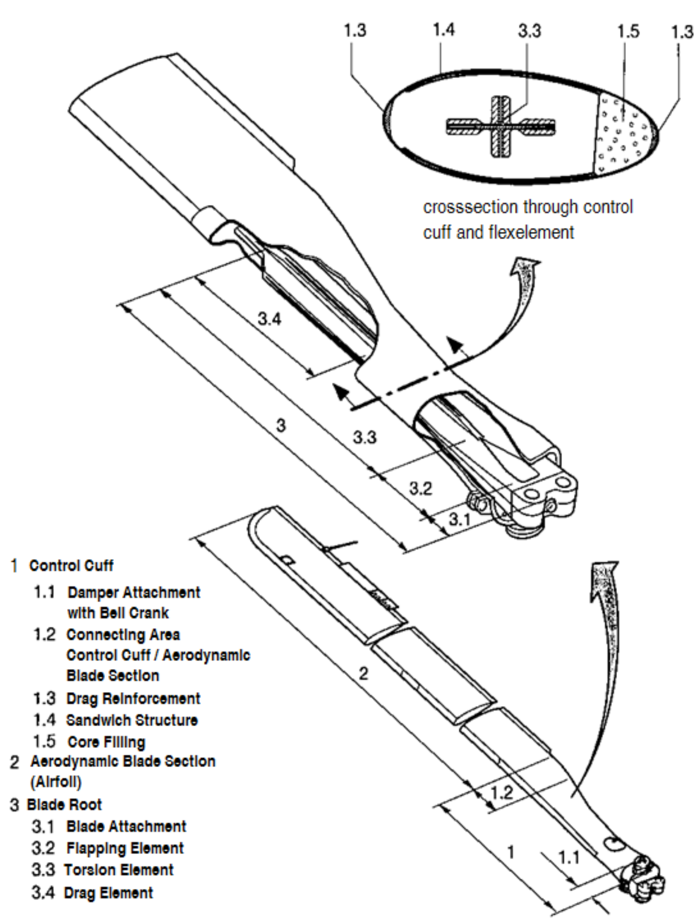


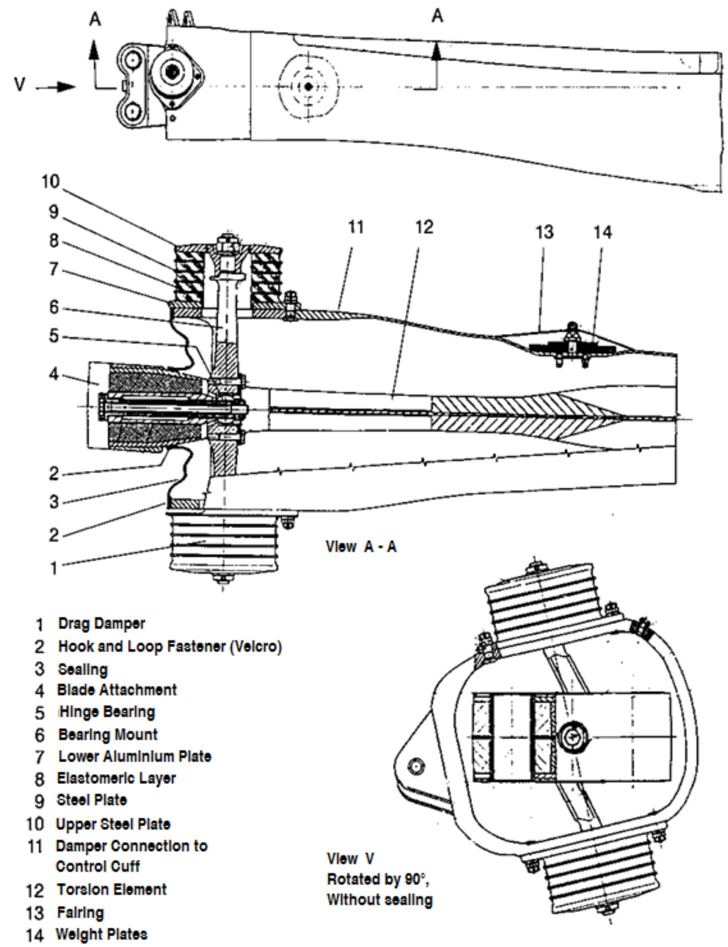
1) Main rotor blade
2) Swash plate
3) Main rotor mast
4) Pitch link
5) Drive arm
6) Distance tube
7) Hub cap

Functionality just like fully articulated blade attachment!

6.2 Rotor Hub Types: Bearingless Blade Attachment EC 135

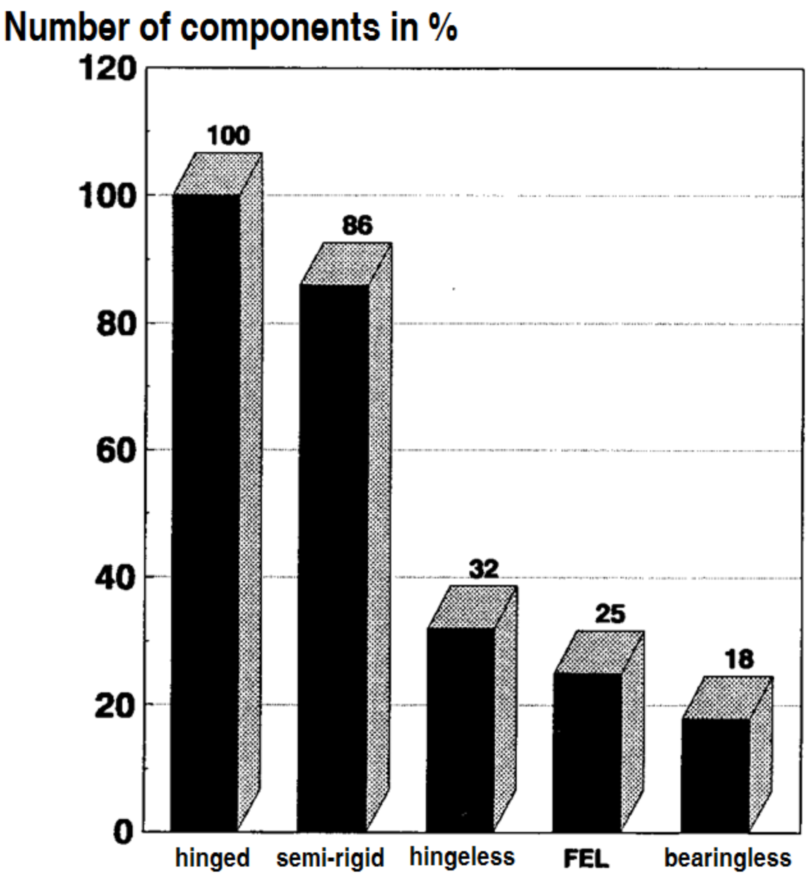
EC 135 Details

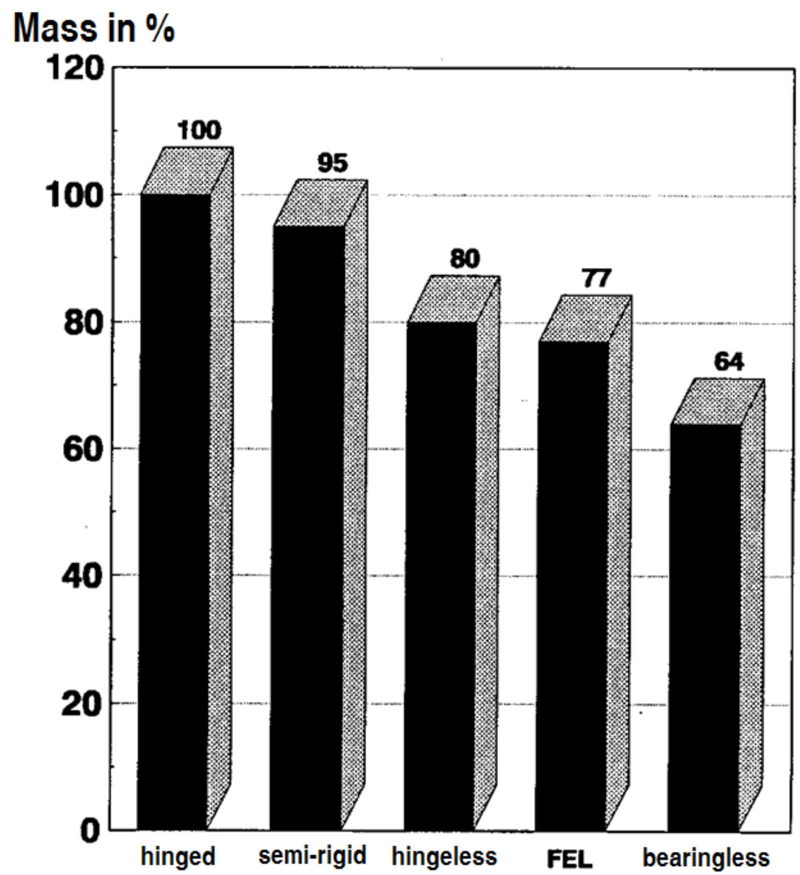




6.2 Rotor Hub Types

Comparison of Rotor Concepts

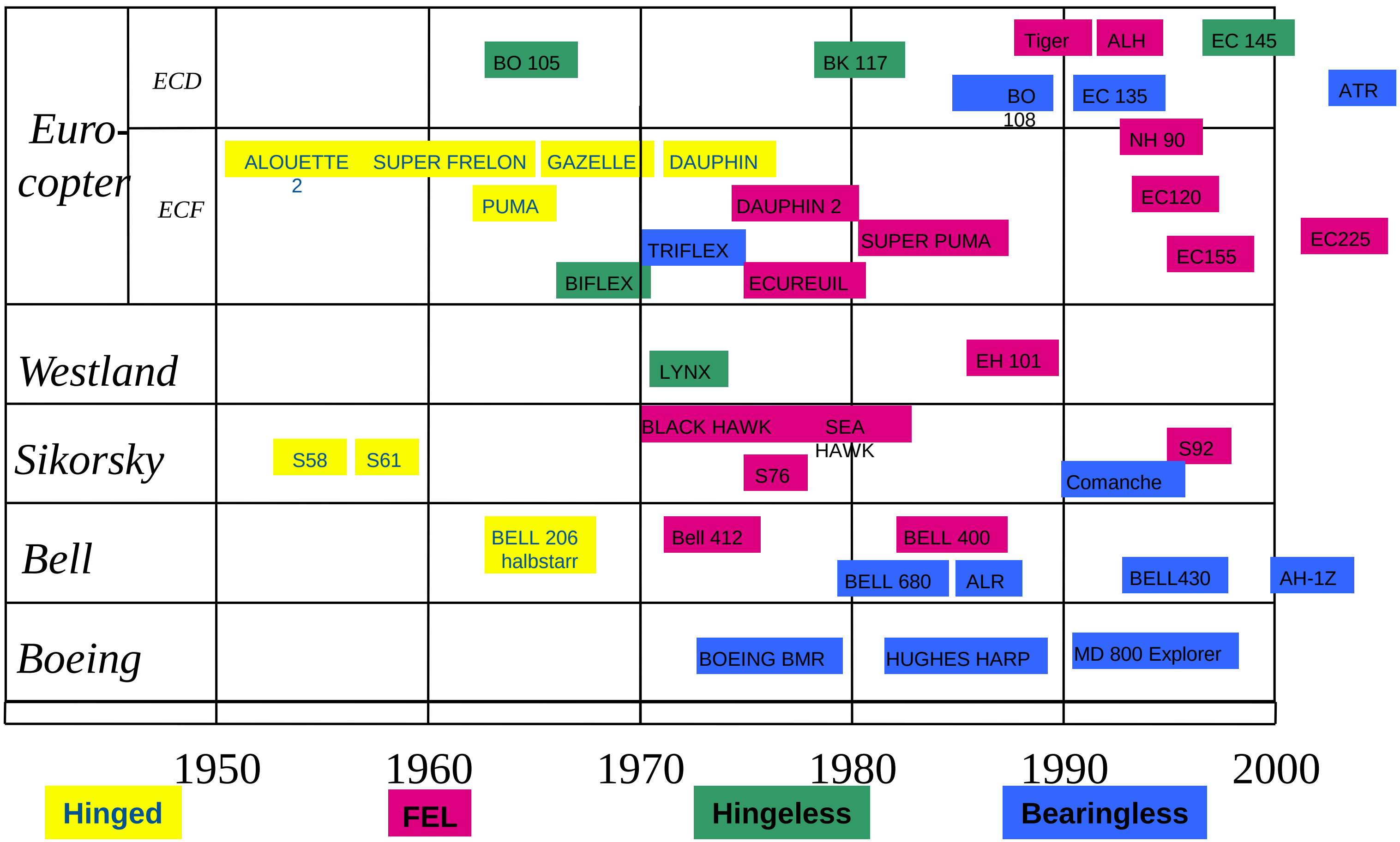




Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme”

6.2 Rotor Hub Types

Temporal Overview of Various Rotor Concepts



Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme"

6 Architecture and Component Design

6.1 Product Development Process

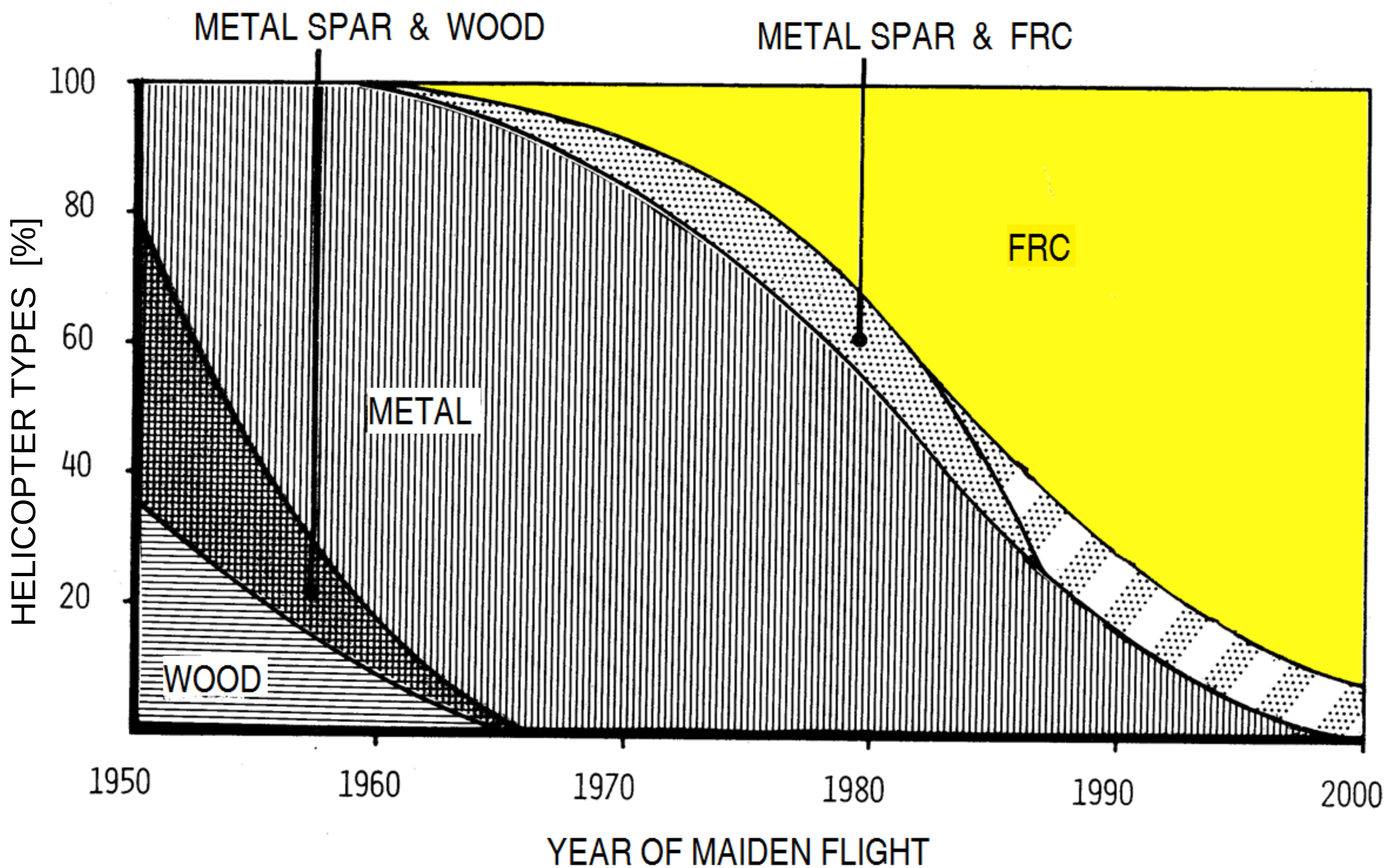
6.2 Rotor Hub Types

6.3 Rotor Blades

6.4 Fuselage Design

6.5 Example: Requirement – Transport of a Black Hawk inside of a C-130

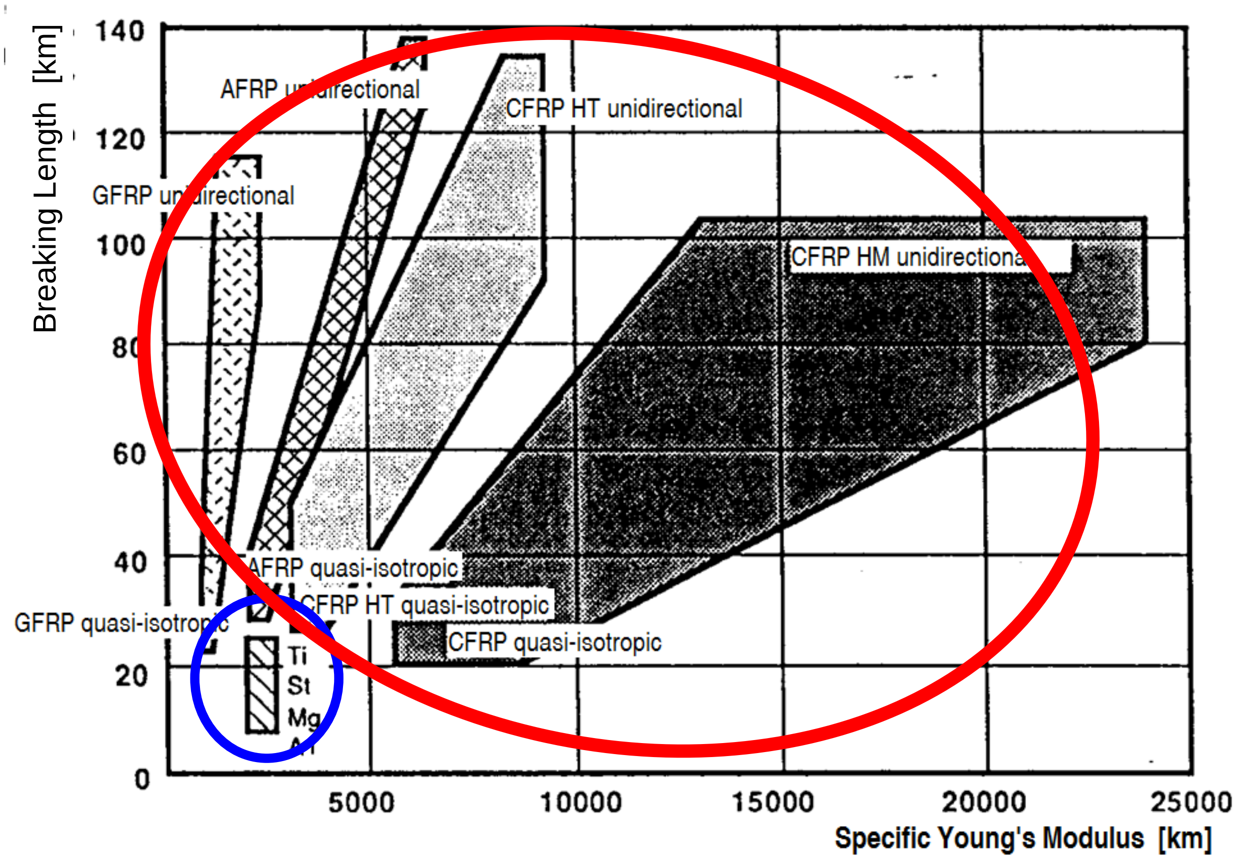
6.3 Rotor Blades - Materials



Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme"

6.3 Rotor Blades - Materials

Strength of Fiber Reinforced Composites compared to Metallic Materials:



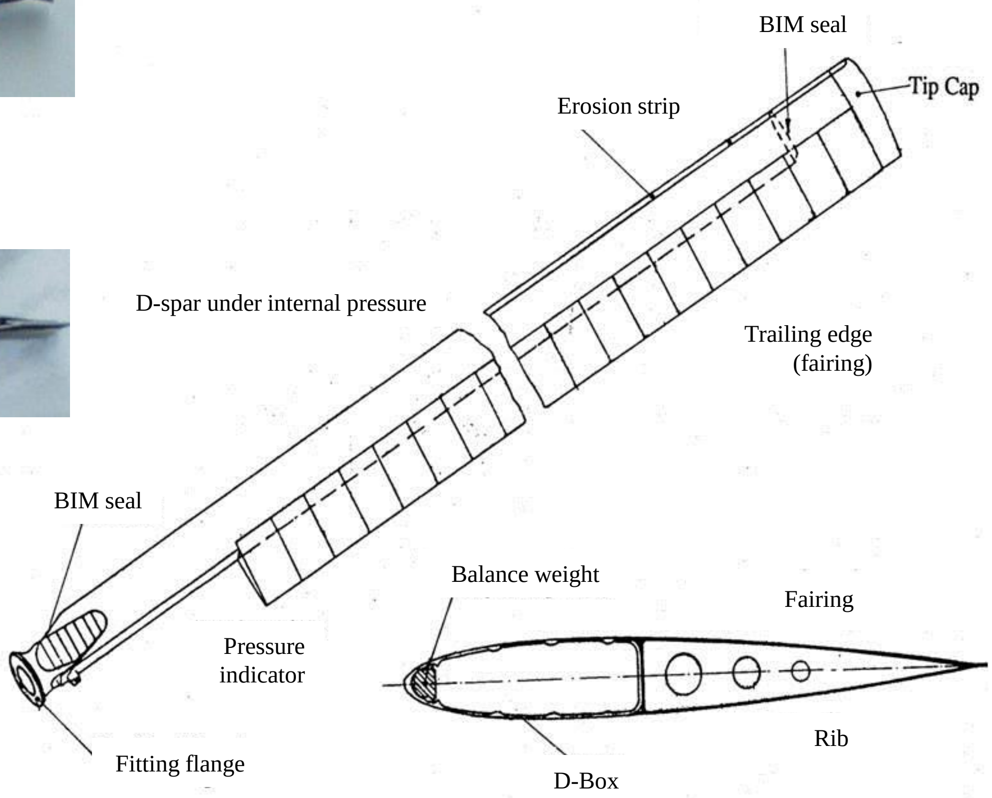
Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme"

6.3 Rotor Blades

Metal Designs:

CH 53G – Main Rotor Blade

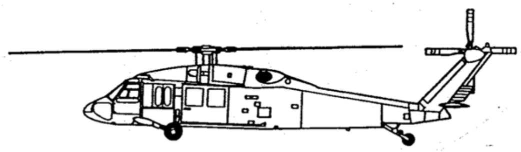
Boeing Chinook – Main Rotor Blade



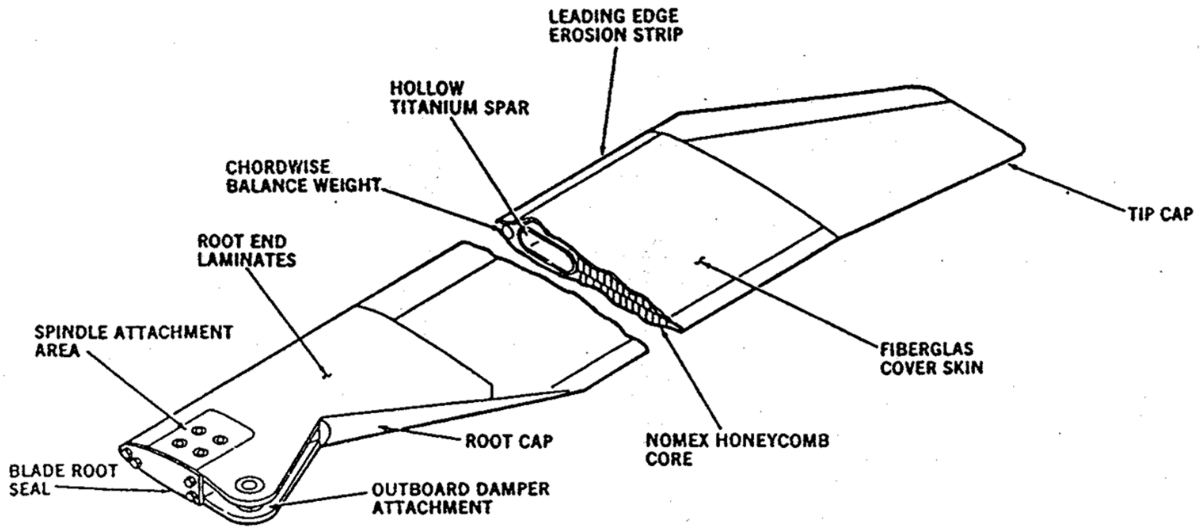
Source: Kuntze-Fechner-EC-HS-Seminar-2004-
"Konstruktion / Bauweisen Rotorsysteme”

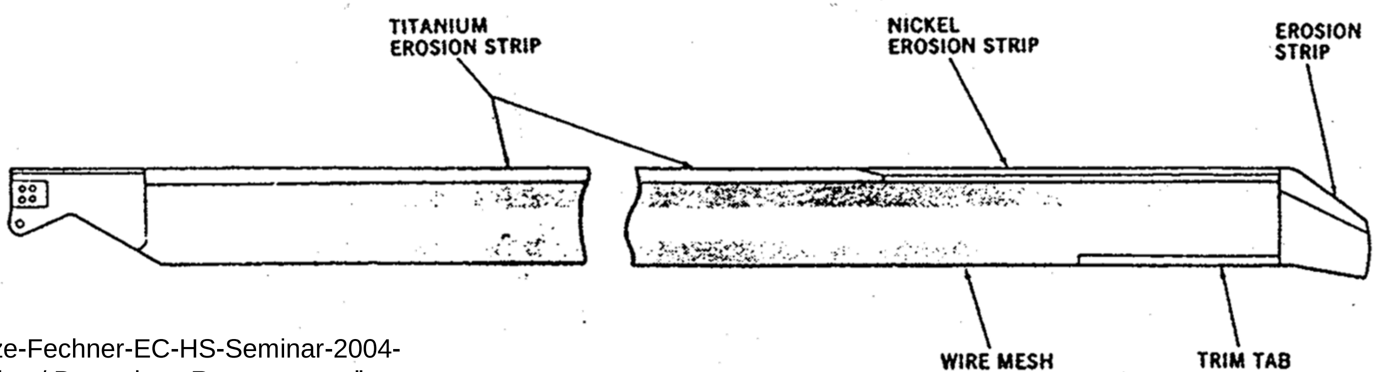
6.3 Rotor Blades

Mixed Designs:



Sikorsky UH-60A Black Hawk

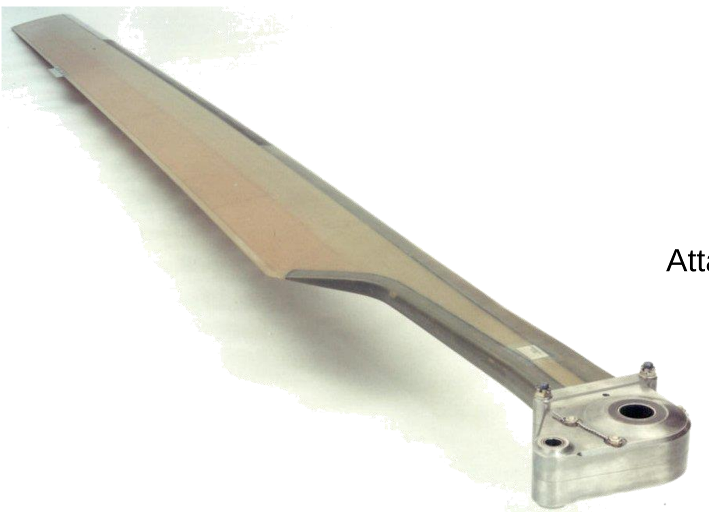


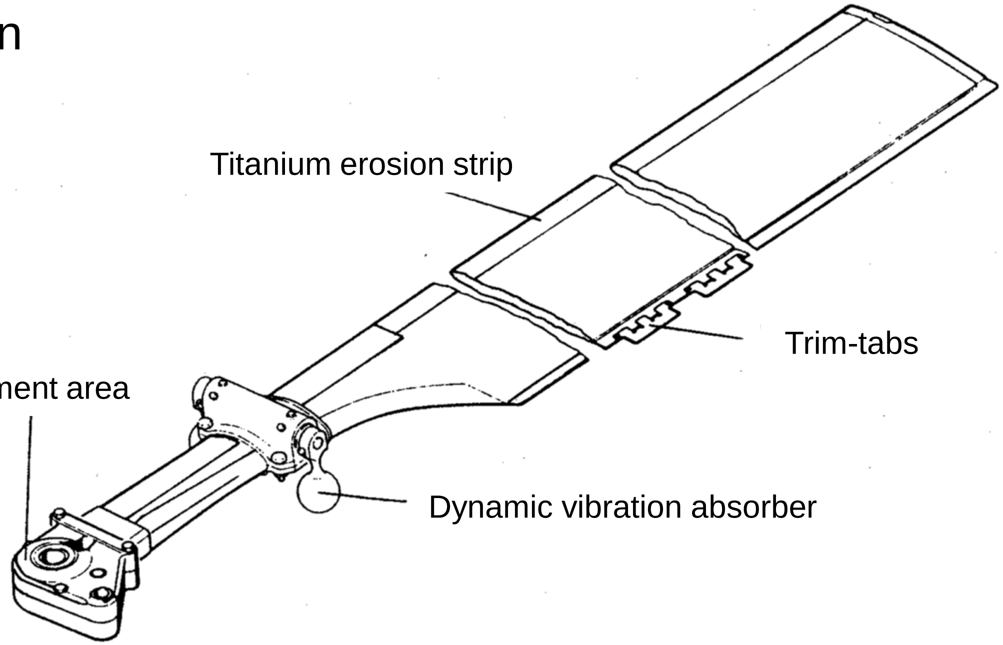


Source: Kuntze-Fechner-EC-HS-Seminar-2004-
"Konstruktion / Bauweisen Rotorsysteme”

6.3 Rotor Blades

Fiber Reinforced Composite Design

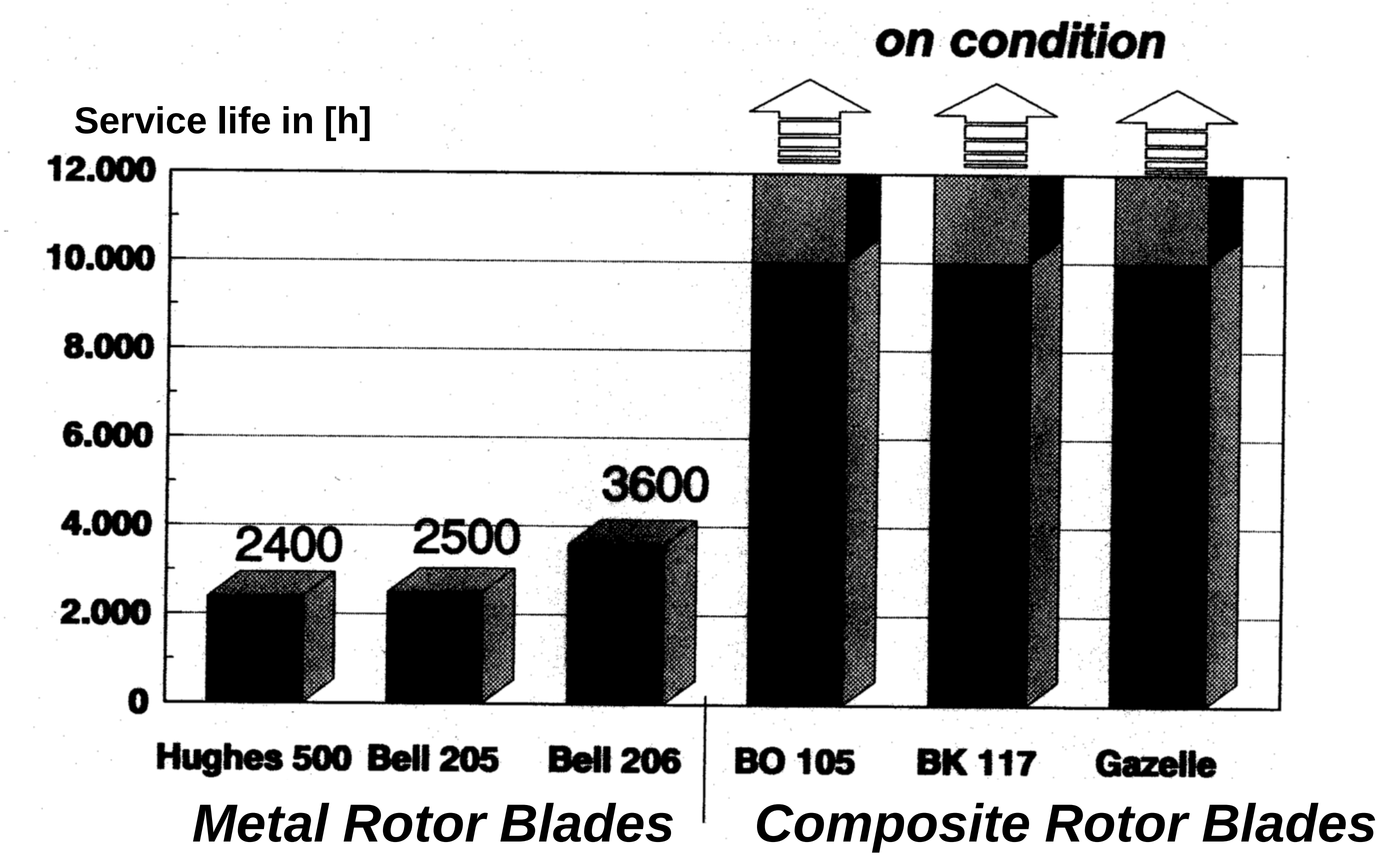




Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme”

6.3 Rotor Blades

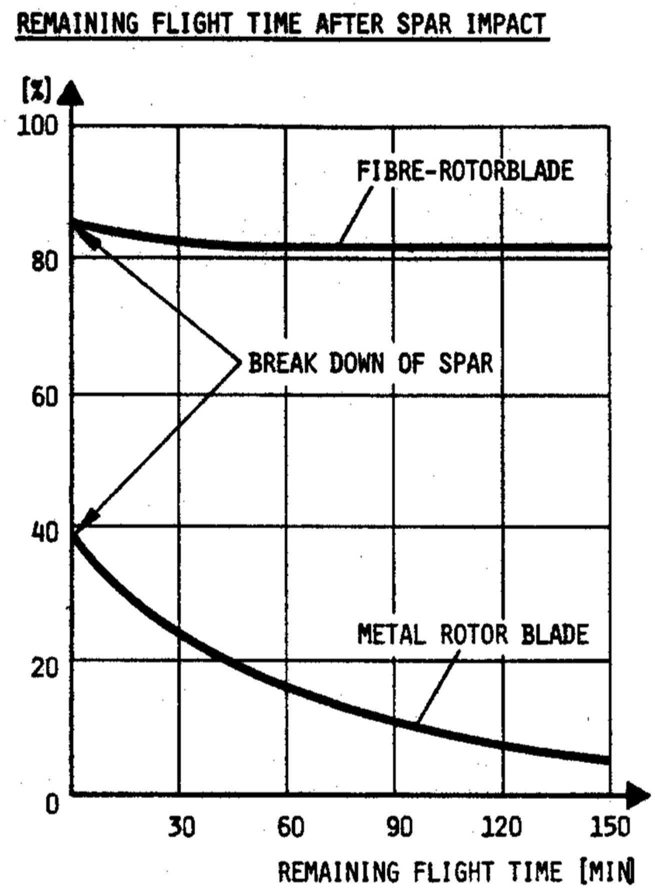
Rotor Blade Service Life



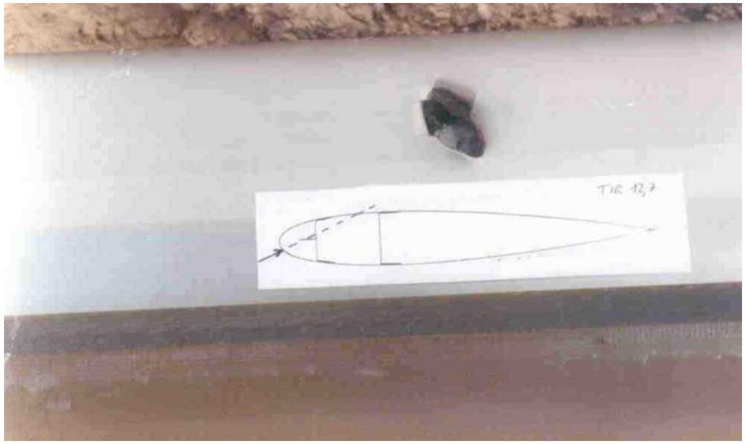
Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme"

6.3 Rotor Blades

Damage Tolerance:



Damage taken by Rotor Blade after being subjected to ballistic projectile



Entry

Exit

Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme"

6.3 Rotor Blades - Manufacturing

Laying of blade skin (Carbon ->high torsion stiffness)

Laying of blade root skin (Glass -> elastic blade root)

Laying of glass rovings (Glass UD -> centrifugal forces, bending)

Laying of blade completed (core made of PMI foam)

The epoxi pre impregnated woven fabrics and rovings are heated gradually and fully cured at 135° C within 75min. The entire process takes about 6 hours.

The mirror-inverted upper mold is positioned on top of the bottom mold using a lifting and turning device.

Source: Kuntze-Fechner-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Rotorsysteme”

6 Architecture and Component Design

6.1 Product Development Process

6.2 Rotor Hub Types

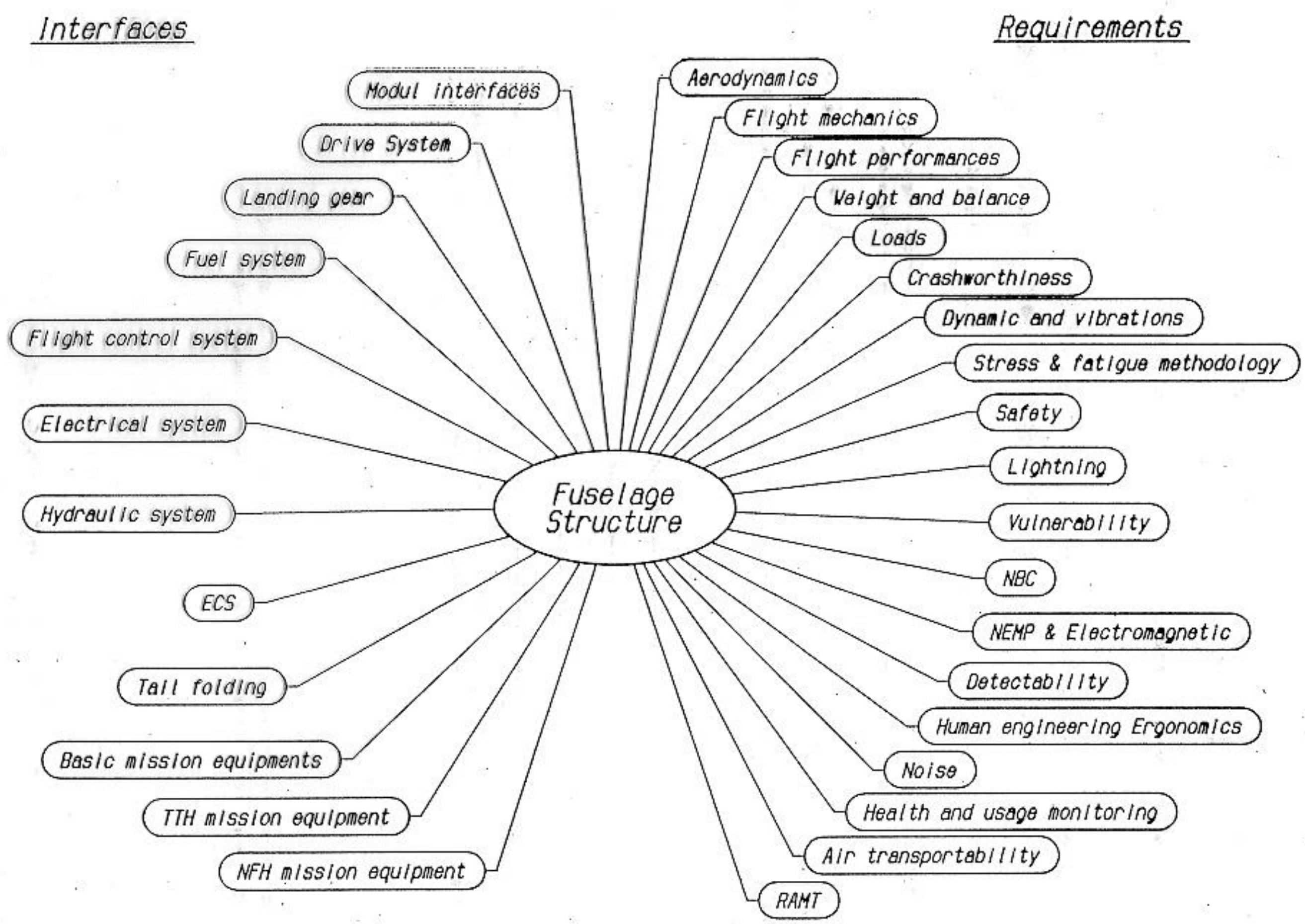
6.3 Rotor Blades

6.4 Fuselage Design

6.5 Example: Requirement – Transport of a Black Hawk inside of a C-130

6.4 Fuselage Design

Fuselage Structure Requirements
NH90 as Example



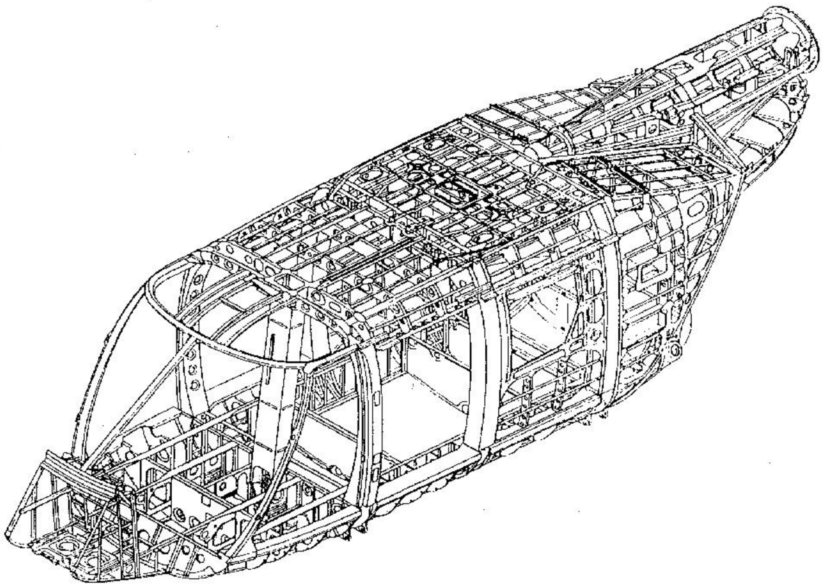
Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Constructive Design:

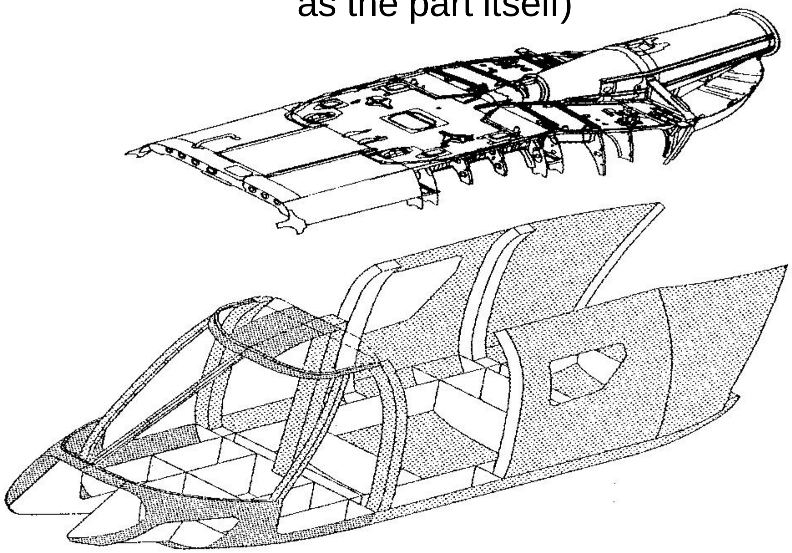
Differential Design

(Very common among metal structures)



Integral Design

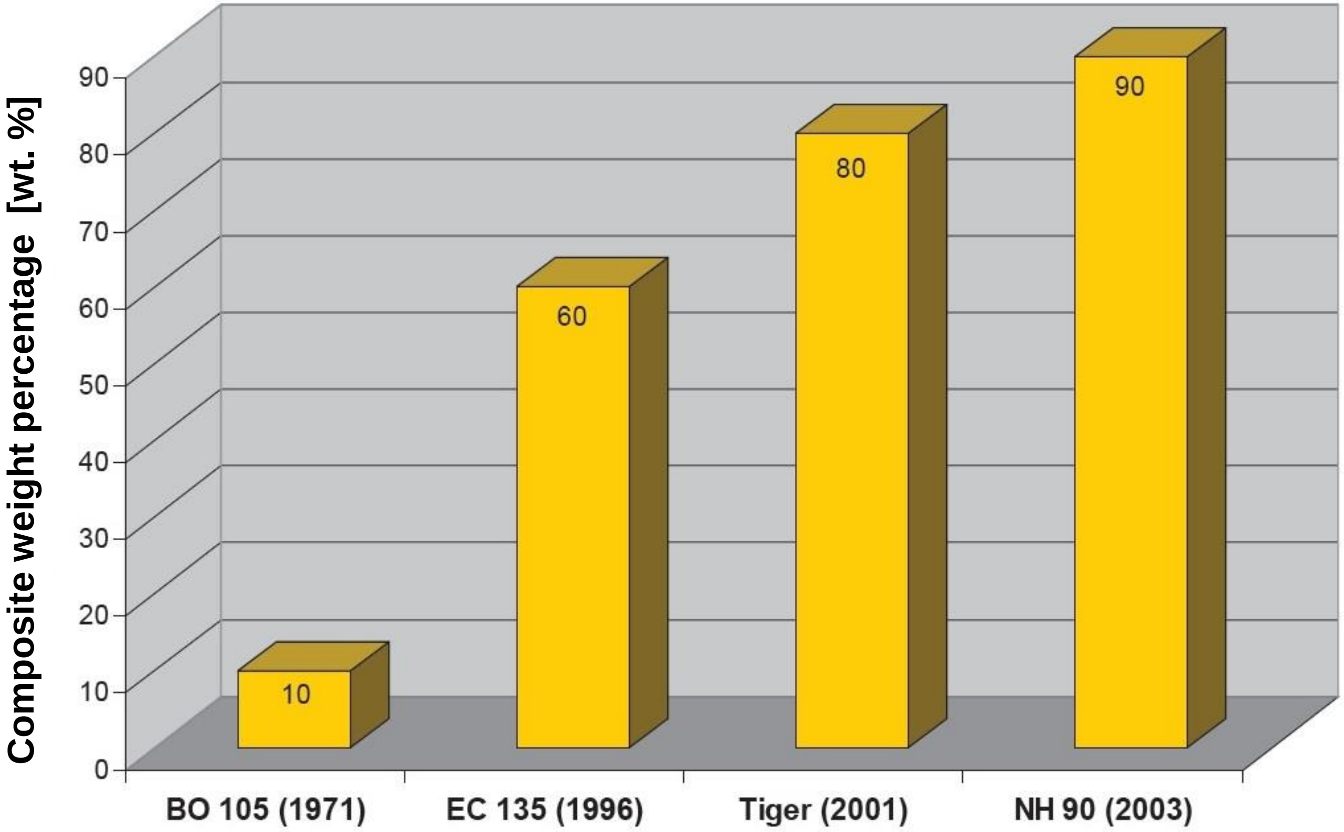
(Preferred method using composites; the raw material is created at the same time as the part itself)



A technical / economical trade-off must be found for the manufacturing and maintenance.

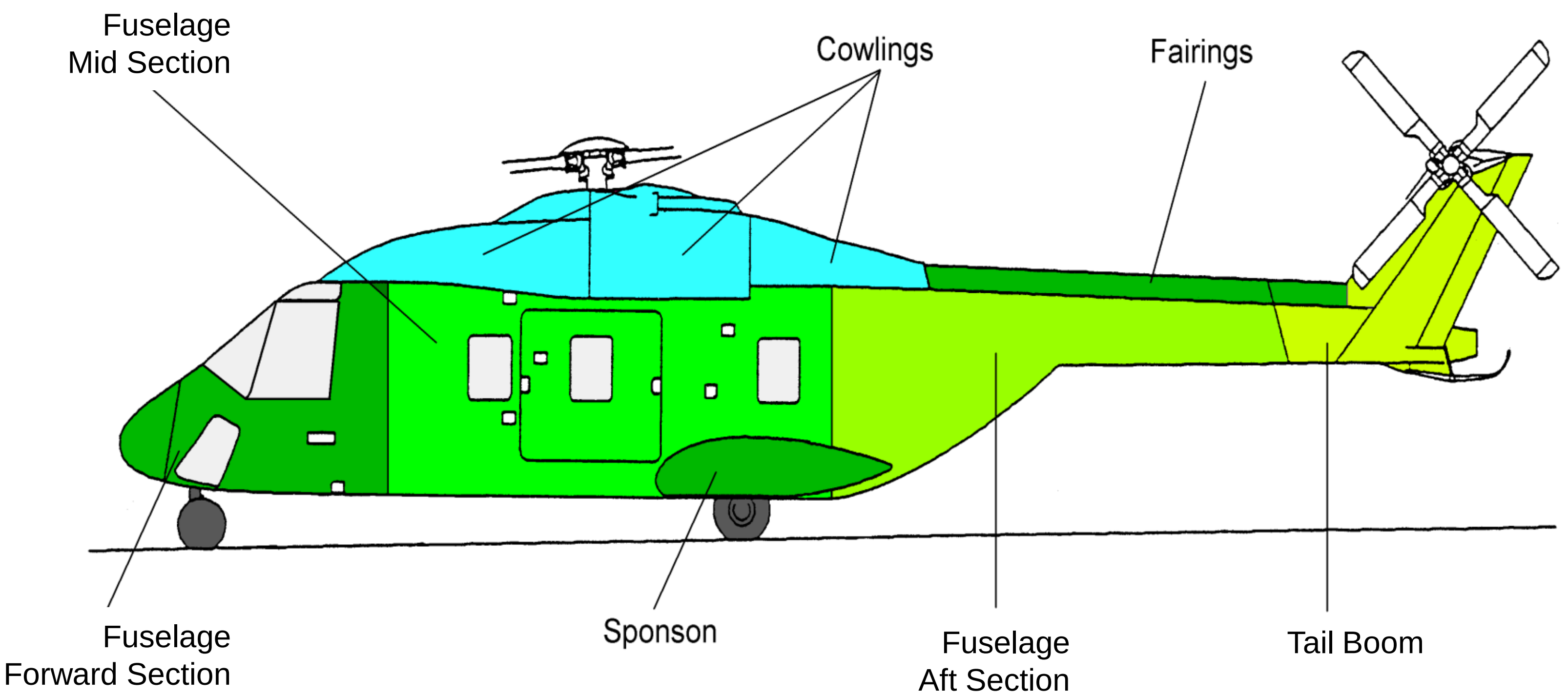
6.4 Fuselage Design

The increasing application of fiber reinforced composites can be clearly observed in fuselage designs as well:



6.4 Fuselage Design

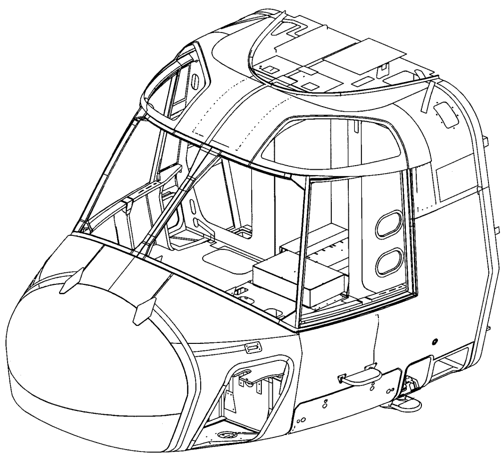
Example NH90 – Fuselage Classification

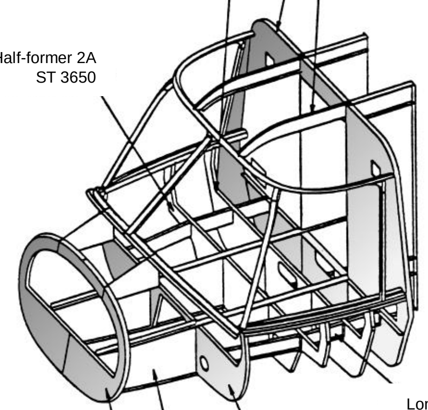


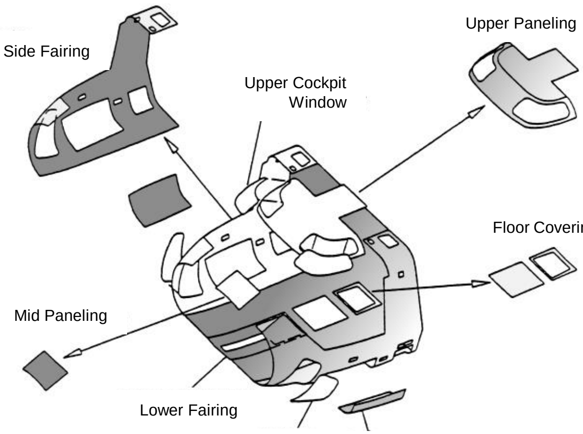
Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Example of NH90 Structure Design



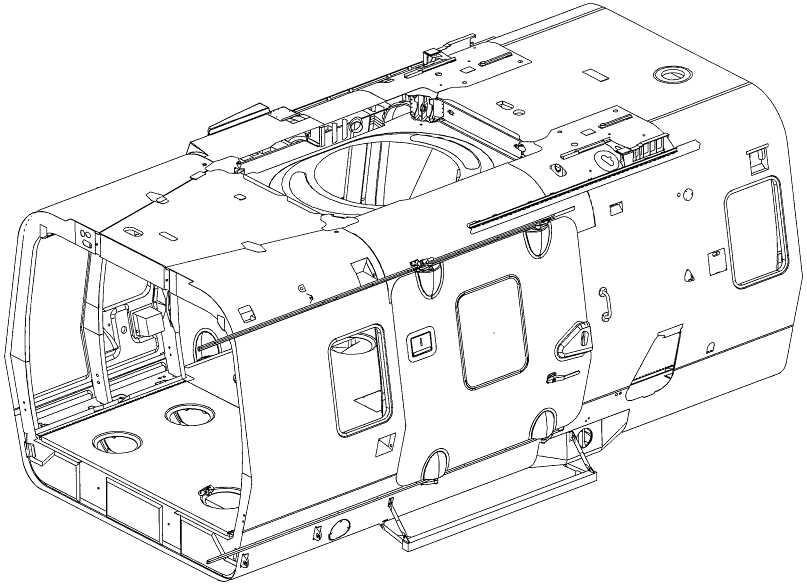


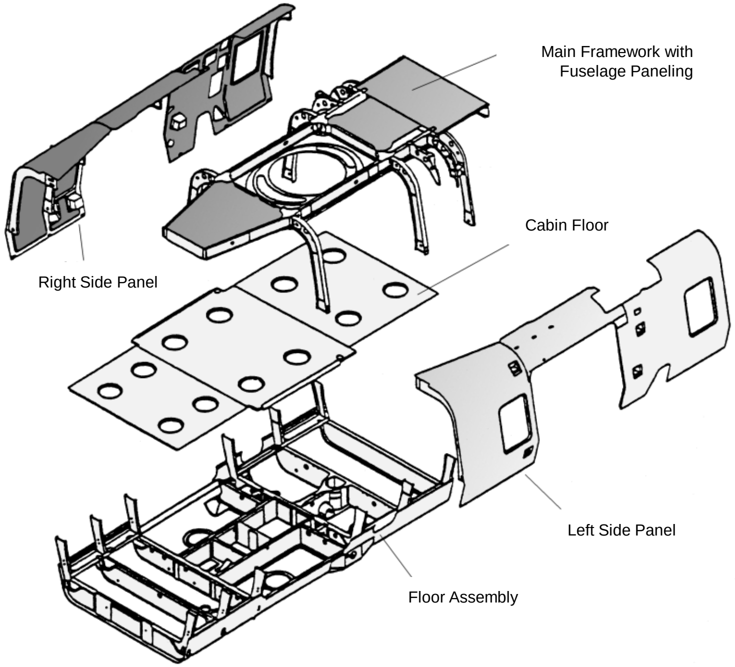


Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Example of NH90 Structure Design

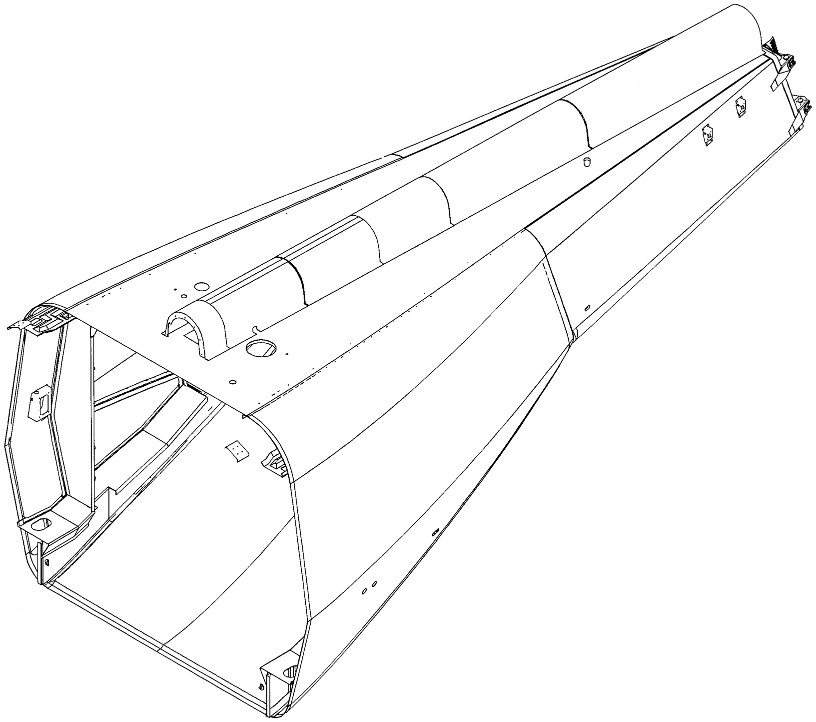


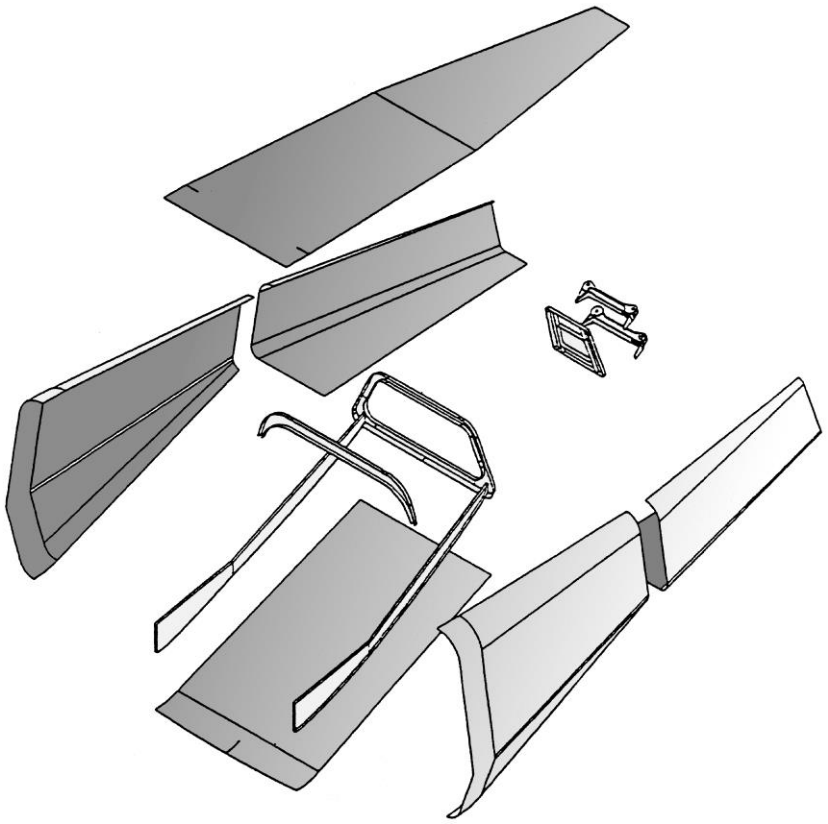


Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Example of NH90 Structure Design

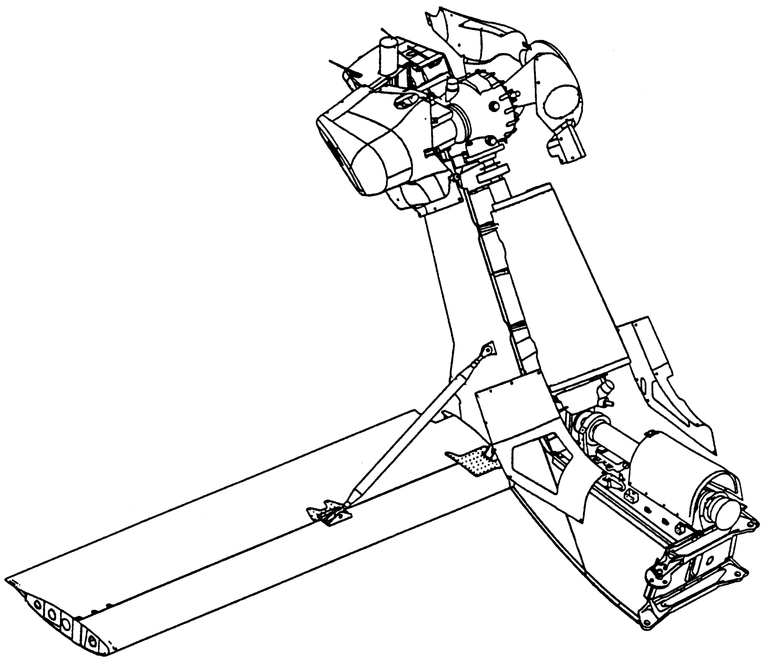


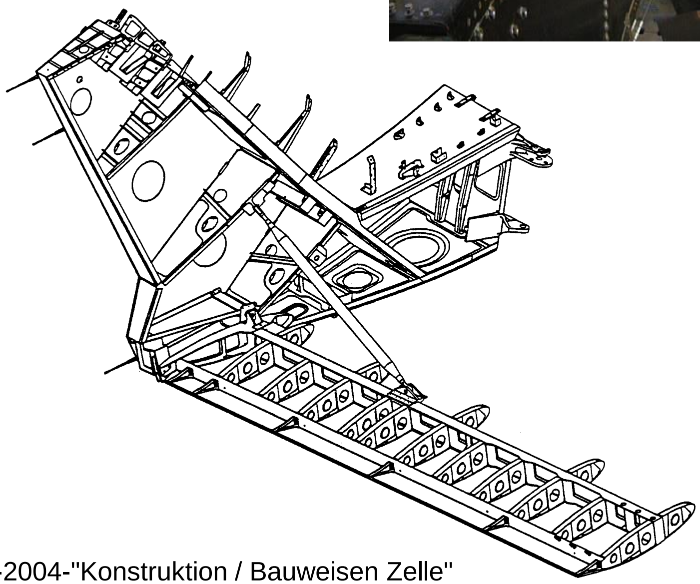


Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Example of NH90 Structure Design – Tail Boom



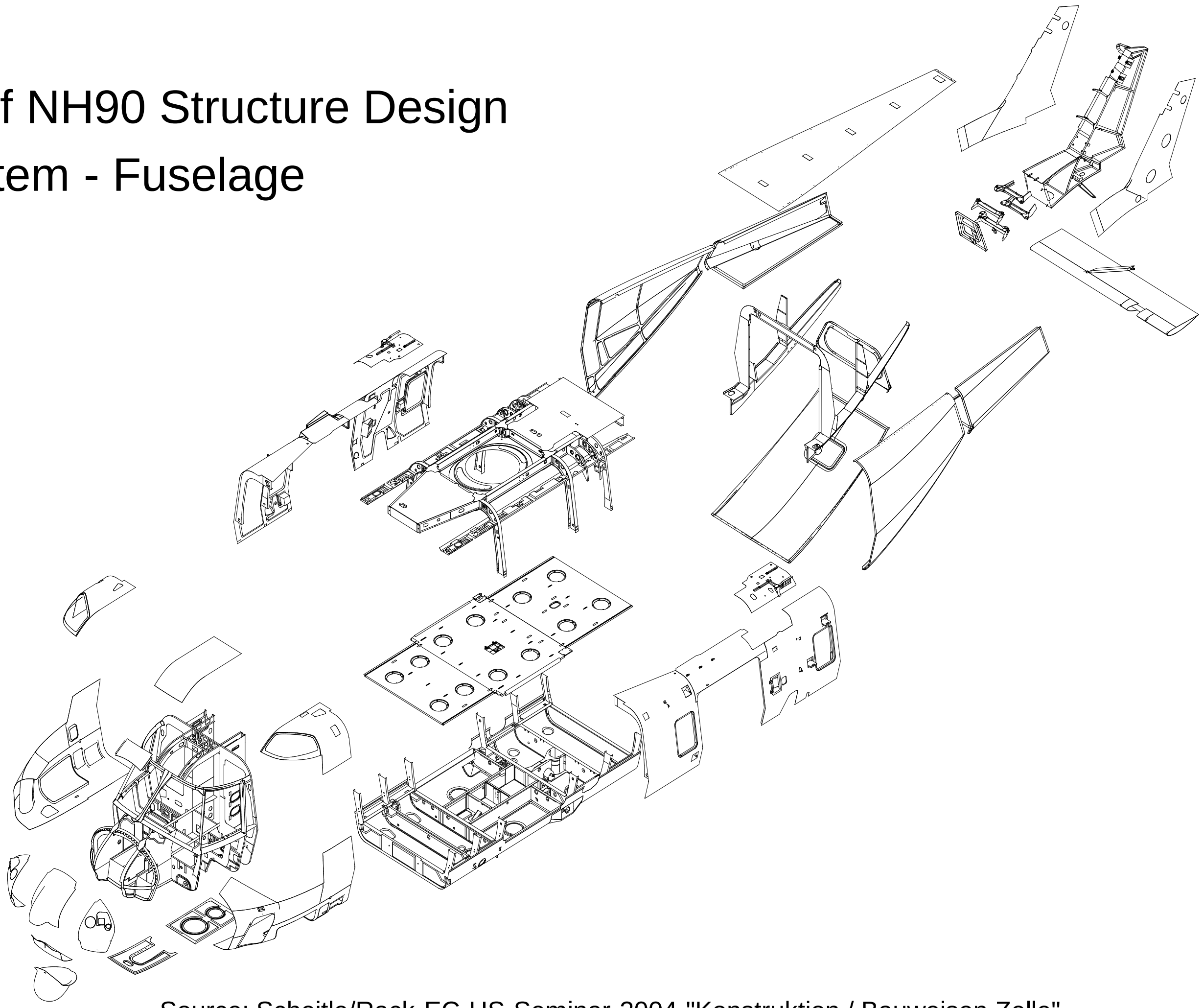


Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Example of NH90 Structure Design

Entire System - Fuselage



Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Example of NH90 Structure Design

Source: Scheitle/Rack-EC-HS-Seminar-2004-"Konstruktion / Bauweisen Zelle"

6.4 Fuselage Design

Assembling the Subassemblies:

- Adhesive: permanent connection (integral design)
  Preferred method using composites
- Riveting: connection of main assemblies
  Repair techniques usually for sheet metal design
- Screwing: for detachable connections

On the whole a combination of all three methods is used.

6.4 Fuselage Design - Joints

Rivet Joints:

Advantages:

- Low staff training requirements
- Easily done
- No temperature or moisture influence
- Low surface treatment requirements
- Easy to inspect
- Suitable for field repair with limited means

Disadvantages:

- Heavier than using adhesive
- Weakening of structure through holes for the rivets
- Stress concentration at rivet holes

6.4 Fuselage Design - Joints

Adhesive Joints:

The adhesive joint is the most adequate connection technique for fiber reinforced composites with polymer matrix systems. The curing pressure and temperature for manufacturing an adhesive joint should be well below the curing pressure and temperature of the involved structural parts that are being joined together. The resin systems of the involved parts and the adhesive must be compatible.

Advantages:

- High efficiency (high specific strength of the joint)
- Suitable for thin-walled part repair
- Force transmission occurs over a wider area
- Fewer parts involved
- Minimizing danger of corrosion (when joining different material types)
- No weakening effect on the base laminate (no holes)

Disadvantages:

- Highly trained staff necessary
- Higher requirements on necessary specialized tooling
- Time consuming
- Quality inspection difficult (especially non destructive testing)
- Ambient temperature and moisture absorption have a negative impact on the strength of the joint
- Special surface treatment is necessary before applying adhesive

6 Architecture and Component Design

6.1 Product Development Process

6.2 Rotor Hub Types

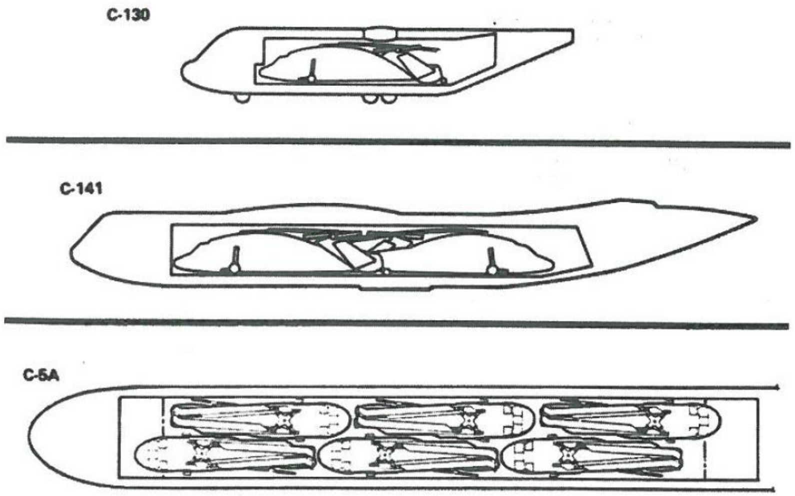
6.3 Rotor Blades

6.4 Fuselage Design

6.5 Example: Requirement – Transport of a Black Hawk inside of a C-130

6.5 Example:
Requirement – Transport of a Black Hawk inside of a C-130

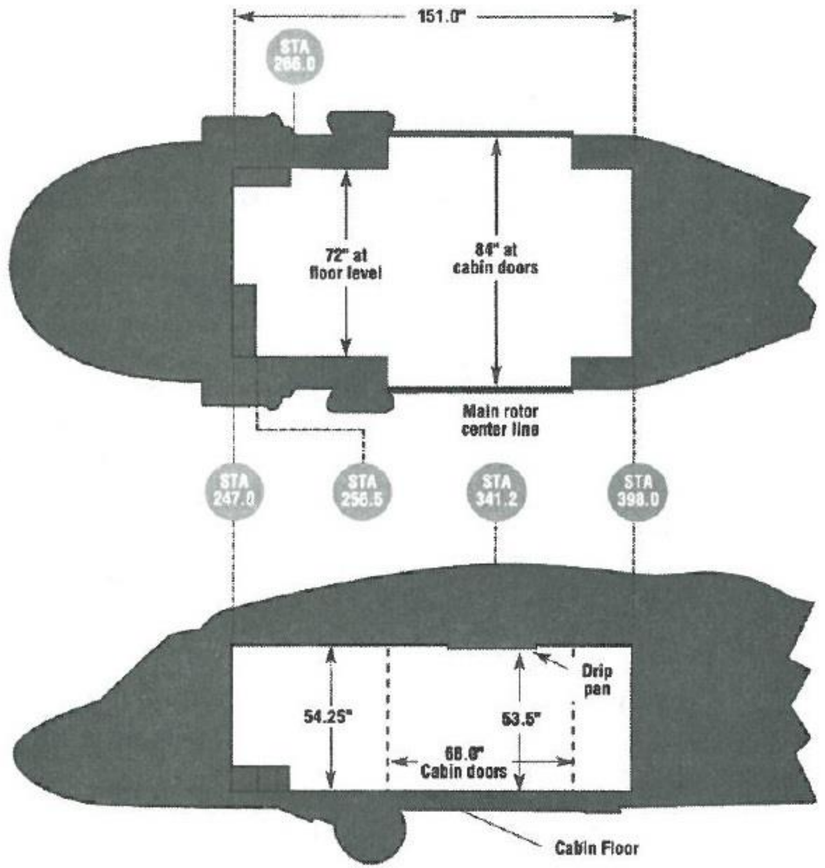
What has to be done in order to get a Black Hawk inside of a C-130 for transport?



Source: „Black Hawk“, Ray D. Leoni

6.5 Example:
Requirement – Transport of a Black Hawk inside of a C-130

The biggest challenge of the design was the minimum cabin height of 54 inch due to the limited cargo bay size of the C-130.



Source: „Black Hawk“, Ray D. Leoni

6.5 Example:
Requirement – Transport of a Black Hawk inside of a C-130

Since the specifications included a fixed loading time it was imperative that as few parts as possible had to be dismantled.

➡ Thus the rotor blades and tail boom are foldable.

Source: „Black Hawk“, Ray D. Leoni

6.5 Example:
Requirement – Transport of a Black Hawk inside of a C-130

Other technical design aspects that were necessary to meet the requirements

Canted Tail Rotor:

- Generates additional Lift thanks to which the main rotor radius could be smaller while not changing the hovering performance;
- The center of gravity could be placed further towards the back thanks to which the cabin could be shortened;

Source: „Black Hawk“, Ray D. Leoni

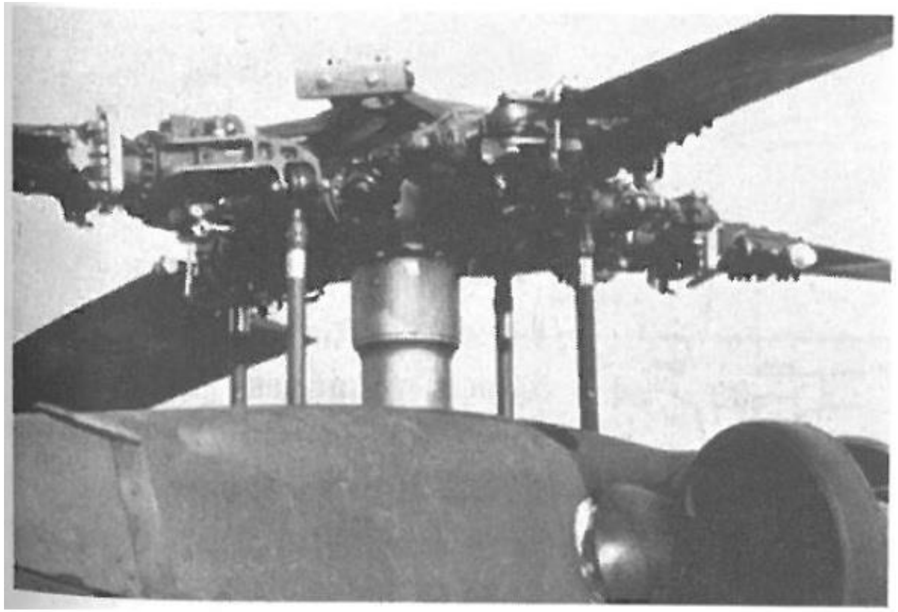
Problem: The tilted rotor causes a rotation coupling effect of yaw and pitch control.

6.5 Example:
Requirement – Transport of a Black Hawk inside of a C-130

Two different rotor positions:

- A very short main rotor mast causes strong undesired vibrations due to rotor-fuselage interactions;
- Dismantling the main rotor and transmission would take too much time;

- **Solution:** Rotor mast elongation by 15 inch for flight



Source: „Black Hawk“, Ray D. Leoni