공학석사학위논문

품질기능전개방법을 이용한 회전익기 개념설계

Quality Function Deployment as a Rotorcraft Conceptual Design Tool

2019년 8월

서울대학교 대학원

기계항공공학부

강 세 권

품질기능전개방법을 이용한
회전익기 개념설계

**Quality Function Deployment as a
Rotorcraft Conceptual Design Tool**

지도교수 이 관 중

이 논문을 공학석사 학위논문으로 제출함

2019 년 7 월

서울대학교 대학원

기계항공공학부

강 세 권

강세권의 공학석사 학위논문을 인준함

2019 년 6 월

위 원 장 이 복 직 (인)

부위원장 이 관 중 (인)

위　　원 김 규 홍 (인)

ABSTRACT

Quality Function Deployment as a Rotorcraft Conceptual Design Tool

Sekwon Kang
Mechanical and Aerospace Engineering
The Graduate School
Seoul National University

The engineering design of a rotorcraft requires multi-disciplinary decision-making process, often times having to work with incomplete requirements and mission objectives of the complex aerospace systems. In addition, the traditional serial-design approach requires information from one discipline to be passed down for rigorous design iteration. Such design iterations for extremely complicated rotorcraft design demand for excessive resources especially with the absence of formal design methodology. This thesis focuses on the development and integration of a multi-attribute rotorcraft conceptual design framework and the Quality Function Deployment (QFD), a system-engineering-based requirement analysis tool. Rotorcraft design requires complex Multidisciplinary Design Optimization (MDO) having significant effects on the rotorcraft performances simply manipulating interdependent sizing variables. The House of Quality (HOQ) is adopted as part of the QFD process to transform user demands into design quality and prioritize array of design characteristics that impact customer attributes. The proposed design framework also adopts parallel-design approach with the inclusion of higher fidelity analysis in the conceptual design phase. This framework considers various technical aspects including the aerodynamic, structure, propulsion, transmission design, weight and balance, stability and control, noise analysis, and economic analysis.

As a system integration of the QFD process, this thesis outlines the proposed conceptual design framework to design a high-altitude mountain rescue vehicle. By employing morphological alternative matrix, and carrying out HOQ analysis, four wing-mounted propellers winged-helicopter configuration with hybridized propulsion system was designed. This result was obtained in conjunction with the HOQ analysis for which, critical design variables identified, were specifically studied to obtain the optimal solution. By adopting the design process, rotorcraft that can hover at 29,100 ft and attain a cruise speed of 185 knots was designed. The proposed design framework provides a central collaborative repository to design aerospace vehicles and provide essential information to initiate preliminary design by the integration of the QFD process.

Keywords: Quality Function Deployment, Conceptual Design, Rotorcraft, High-altitude Mountain Rescue Vehicle

Student Number: 2017-20158

Table of Contents

List of Figures

List of Tables

I. Introduction

The engineering design of a rotorcraft requires multi-disciplinary decision-making process, often times having to work with incomplete requirements and mission objectives. These challenges become more prominent when designing a rotorcraft, one of the most complex flying vehicle. The rotorcraft's particular characteristic of being a runway independent vehicle has made it one of the most desired aeronautical vehicle. Ever since the development of a reliable rotorcraft, rotorcraft has become the indispensable asset in various application fields including commercial sector, military, surveillance, firefighting operation, police operation, and medical evacuation. In addition, rotorcraft has recently received scrutiny in light of urban air-taxi operation. To meet various needs and requirements from all sectors, rotorcraft has been designed in several different configurations including winged, coaxial, tandem, side-by-side, multicopter and other convertible forms. Furthermore, sustainability goals set forth by the NASA N+3 goals together with the recent innovative but rather disruptive technologies have challenged the rotorcraft community to propose various forms of aeronautical vehicles. Aside from the main rationale of rotorcraft's hover capability, improved forward flight capability has become a recent demand to expand the mission envelop of rotorcraft [1].

The conceptual design phase, including the pre-design process, are the most essential process that affects the overall design and performance of the rotorcraft. During the conceptual design phase, design requirement analysis, feasibility study, and subsystem definition are defined. As with the engineering design process, there exists no single answer fulfilling all requirements of the design process. This makes it even difficult for the engineer to design especially with the absence of formal rotorcraft design methodology. Efficient design process for rotorcraft is therefore required to provide systematical and well-established processes for the engineers to make logical and objective decisions for the design problem.

Through literature study surveys, efficient design process methodologies that can be applied have been consistently presented in the field of industrial engineering and system engineering. However, only a few developments and applications of the conceptual design methodologies are presented in the aerospace engineering field and almost trivial developments of rotorcraft design methodology.

Kirby [2] has established the TIES method, which is a mathematical formulation of providing tradeoff impacts, and enable decision-maker to easily assess sophisticated alternatives. By employing the design process in Figure 1, the transformation of qualitative to quantitative justification can be brought into the earlier phases of the design. However, TIES method is proposed for designing and exploration of various configuration of general aerospace vehicles. Rotorcraft, being one of the most complex vehicles, requires detailed numerical analysis to capture dynamical aerodynamics interaction of the rotor coupled with other disciplines. It has been observed that tradeoff impacts of rotorcraft demand for complex calculation and multidisciplinary studies, requiring design process specifically adopted for conceptual rotorcraft design.

Similarly, Schrage [3] has established Integrated Product-Process Development (IPPD) based approach which synthesizes design optimization for rotorcraft. This process has successfully established an objective and efficient design process such as decision-making model considering airworthiness certification regulations, alternatives configurations, and assessment processes. With current computing power, IPPD enables rapid conceptual design by formulating and integrating various disciplines involved in rotorcraft design. Preliminary advantage includes automation that reduces considerable design cycle time and its capability to run intensive optimization and computation for Design of Experiments [4]. However, the IPPD design process has been observed to purely incorporated detailed design analysis of rotorcraft's transmission design.

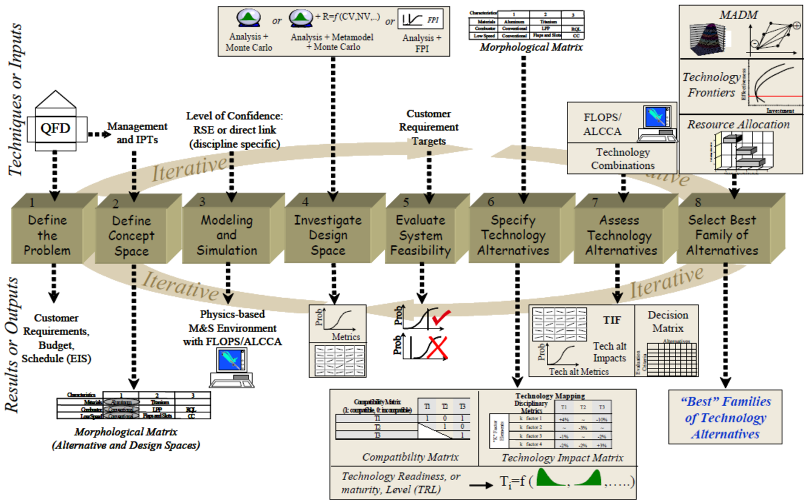


Figure 1 Forecasting environment of TIES method [2]

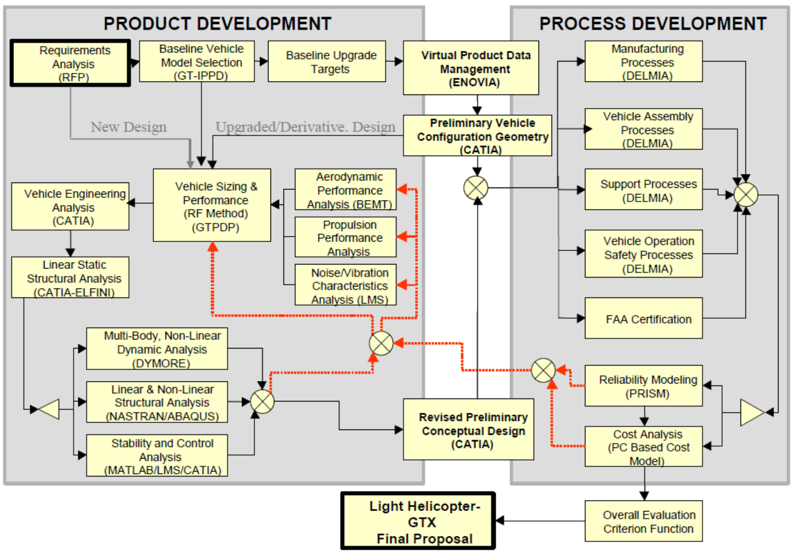


Figure 2 IPPD Rotorcraft Design Methodology [3]

Problem Definition

With the availability of current computing power and various design tools, multidisciplinary analysis of the rotating elements, structures, powertrain, and stability can be carried out. However, there still exists a large gap in integrating multidisciplinary aerospace systems especially during the conceptual design phase, where many design decisions are made. This is mainly due to the fact that each analysis tools are developed independently from each other. These tools are therefore incompatible with other advanced analysis tools of various disciplines and are almost impossible to obtain output data for coupling. Therefore, engineers will have to analyze the data obtained from various disciplines to make various design decision. Unfortunately, these decisions are oftentimes made without sufficient tradeoffs and are overlooked due to the limited time and monetary resources. For this reason, a methodology specifically adopted for rotorcraft design is needed to enable designers to make high-level design decision, and introduce complex and detailed analyses in the conceptual design process.

Purpose of Thesis

The purpose of this thesis is to develop a systematical approach in incorporating system-engineering based QFD analysis with multidisciplinary optimization process to design complex rotorcraft system. This thesis will provide a broad picture of how customer attributes can be translated into engineering attribute, and how various disciplinary analysis tools are incorporated in the conceptual design of a rotorcraft. The proposed design framework will enable rapid prototyping that is not only technically feasible and technologically advanced but also meet the requirement set forth by the customer by providing detailed information that is essential for the designers to make appropriate decisions. Through effective design iterations, the proposed design process aims to bridge the gap between different discipline and formulate an optimal viable solution that is fully customizable to accommodate new designs.

At the end of this research, a formulation of the design methodology is introduced while covering various disciplines involved in the design of rotorcraft. In addition, a sample problem for proof of concept is solved as a verification of the entire process that goes into the conceptual rotorcraft design. This includes a collaborative integration of structures, aerodynamics, stability and control, avionics, and other economic studies that bridge down from QFD integrated multidisciplinary optimization results.

This thesis is organized as follows:

- Chapter II presents a literature review on the QFD process and how it could be adopted.
- Chapter III introduces and describes the proposed design process methodology for conceptual rotorcraft design.
- Chapter IV verifies the design process by demonstrating the design methodology application by carrying out the conceptual design of extreme altitude mountain rescue vehicle. Detailed optimizations and multidisciplinary analyses are carried out to demonstrate and verify the effectiveness of the proposed design process.
- Chapter V provides concluding remarks about the overall effort in designing rotorcraft.
- APPENDIX provide detailed technical information for the proof of concept

II. Literature Review

A brief background study on Quality Function Deployment (QFD) is presented in this chapter. The QFD process was first introduced in 1966 by Dr. Yoji Akao [8]. The QFD is derived from the Japanese characters: “Hinshitsu” (quality); “Kino” (function); “Tenkai” (deployment). As stated by Cohen [10], it essentially displays the customer’s wants and needs, and its impact on technical responses to meet those needs. Throughout extensive surveys, the QFD process is commonly reported to be an effective planning and guidance tool that attempts to meet customer requirements.

With regards to the development of a new product, the customer’s voice must be systematically incorporated in the conceptual design phase where the design freedom is at the highest. The historical trend and development process of a product has shown that significant cost is incurred even with simple design change at the latter phase. This is clearly depicted in Figure 3 where the cost is inversely proportionate to the design freedom whereby redesign at the latter phase would cost much more compared to the conceptual design phase. Therefore, the designer must concentrate its resources and adopt all foreseeable problems in the early design phase to remain competitive and effective.

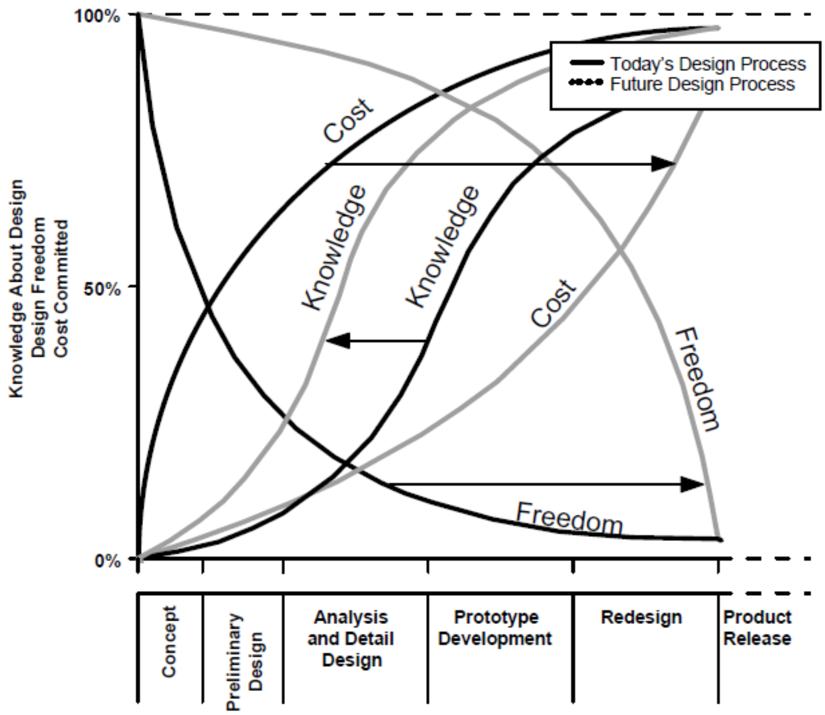


Figure 3 Design Freedom, Knowledge and cost Relationship [4]

QFD literature studies also revealed wide applications including automobile [14], aircraft design [11], and even typical applications that do not fit the model of product development [10]. In order to achieve the underlying flexibility applications of the QFD, variety of planning and decision making tools are drawn from "Seven Management Planning Tools" [10], which is based on the Total Quality Management (TQM).

The benefits of the QFD is explicitly summarized by Bossert [19] and listed for convenience.

Customer Driven Attributes

- Creates focus on customer requirements
- Uses competitive information effectively
- Prioritizes resources
- Identifies items that can be acted upon
- Structures resident experience/ information

Reduces Implementation Time

- Decreases midstream design change
- Limits post-introduction problems
- Avoids future unwanted redundancies
- Identifies future application opportunities

Promotes Teamwork

- Consensus-based
- Creates communications at design interfaces
- Identifies actions at design interfaces
- Creates a global view out of details

Provides Documentation

- Documents the rationale for the design
- Is easy to assimilate
- Adds structure to the information
- Adapts to changes, a living document
- Provides a framework for sensitivity analysis

Decision-Planning Tools in QFD

QFD adopts various mathematical and psychological tools which are extremely powerful for structuring and organizing qualitative information. These mainstay tools of QFD include: *affinity diagram, tree diagram, matrix diagram,* and *prioritization matrix*. Other helpful tools often included in the Seven Management and Planning Tools are *interrelationship diagram, process decision program chart, matrix data analysis,* and *arrow diagram* [10].

Affinity Diagram

The affinity diagram provides hierarchical structuring of ideas by organizing qualitative information from ascending order. Beginning with brainstorming with the design team, this tool is used to organize gathered customer attributes to establish categories of information.

Tree Diagram

Similar to the affinity diagram, a tree diagram is also a form of a hierarchical structure of ideas. It usually starts with an existing structure such as hierarchy created through affinity process. By examining and analyzing, it can effectively be used to enhance data that comes from the brainstorming of the development team [14].

Matrix Diagram

The matrix diagram is exploited extensively throughout the QFD process where rows of comparable subject are associated with columns of subject. This method utilizes the psychological nature of human thinking whereby human tend to provide a better choice between two subjects than three or more. Several binary relationships are represented by a symbol depicting the relationships of the subject on the row and subject on the column.

Prioritization Matrix

The prioritization matrix is an extension to the aforementioned matrix diagram indicating the degree of relationships between two subjects instead of a binary relationship. For explicit representation, the scale of 0, 1, 3, and 9 are used to represent aggregated relationships. Through prioritization process, ideas can be prioritized based on the relationships for further deployment into the QFD process.

House of Quality

The House of Quality (HOQ) consists of several complex matrices attached covering customer's wants, technical responses, technical correlations, planning matrix, benchmarks, targets and relationships. These rigorous planning matrixes demand for high-level product goal based on the design team's interpretation of the customer's wants and market research.

The HOQ shown in Figure 4 would begin with the customer's attributes from interviews. The technical responses are filled with the design team's engineering responses to meet the customer attributes for high-level analysis and obtain target values of the technical responses. Planning matrix in the HOQ consists of quantitative market search data ranked based on the customer attributes relationships.

Voice of Customers (WHATs)

This procedure is the most crucial part of the QFD during product planning as it often leads to important misinterpretation and misunderstandings of the mission objectives. This process in the QFD is usually obtained or provided in the Request for Proposal (RFP) and interviewing customers.

Voice of Engineer (HOWs)

Technical responses, also known as the voice of engineers (VOE), are usually obtained in a form of quantitative performance measurement. It is vital that these formulation of the technical responses are solution-independent to provide more opportunities in bridging the gap between customer wants and needs.

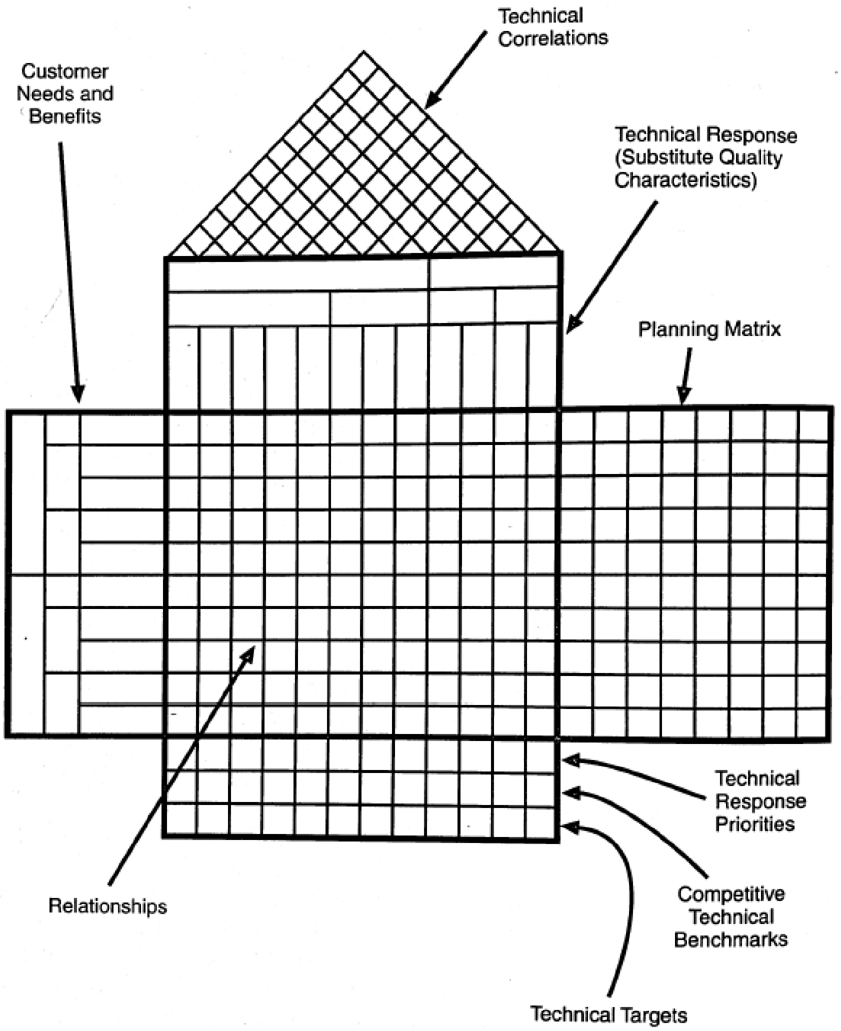


Figure 4 The QFD House of Quality [10]

III. Design Process Methodology

The main objective of this chapter is to outline the proposed conceptual design methodology process that has been formulated for the multidisciplinary design optimization of rotorcraft. The literature review discussed in Chapter II is incorporated into the design methodology of the rotorcraft.

The schematics of the conceptual design process of rotorcraft is depicted in Figure 5. Integration of QFD process, MDO, and detailed analysis in the conceptual design phase provide a well-defined approach in designing a rotorcraft.

Quality Function Deployment (QFD)

Establishing the Need

The conceptual design begins with establishing the need. The need for a new rotorcraft is usually presented in the form of "Request for Proposal (RFP)". A well-defined RFP would provide all the necessary information on mission requirements. Therefore, the first step of any design should be requirement analysis by breaking down the RFP to obtain all the necessary information. This process is the most crucial steps in determining the vehicle attributes, technological refinements, and other constraints the designer should consider.

Mission Objective

Through requirement analysis, the proposed design process extracts top-level mission objectives. The analytical hierarchy process (AHP) is implemented to decompose and systematically rank the order of importance using pairwise comparison. The AHP is a psychological and mathematical decision-making process developed by Satty in the 1970s based on the fact that the brain uses a phased or hierarchical analytic process to make a decision [11]. The analyzed qualitative requirements are converted into numerical priority values.

Gathering Voice of Customers

Based on the defined mission objectives and mission requirements from the RFP, further breakdown of the mission objectives using the affinity diagram is executed. The affinity diagram promotes brainstorming for new ideas, needs, and possible solutions to the problem systematically. The main group generated from the affinity diagram consists of the voice of customer and voice of engineer. To this end, it is extremely useful to gather customers, clients, and designers' opinions through a survey if resources permit.

Quantitative Conversion

Subsequently, HOQ is deployed for requirement analysis to help designer articulate and interpret the needs of the customer. Necessary steps required to obtain a target solution for each of the technical responses develops a quantitative solution from qualitative standards of the voice of customer. This process is obtained from the initial comprehensive sizing code that utilizes MDO described by Lee [10].

Generation of feasible design

As part of the planning matrix, useful techniques for the exploration of other novel configuration using the morphological alternative matrix method is executed. Morphological analysis, developed by Fritz Zwicky in 1967 enables exploration of all possible solution to a multi-dimensional and non-quantified complex problem. This enables the designer to explore combinations of solutions to achieve the mission objectives set forth by the RFP. For rotorcraft design, Table 4 has been specifically generated for rotorcraft morphological analysis.

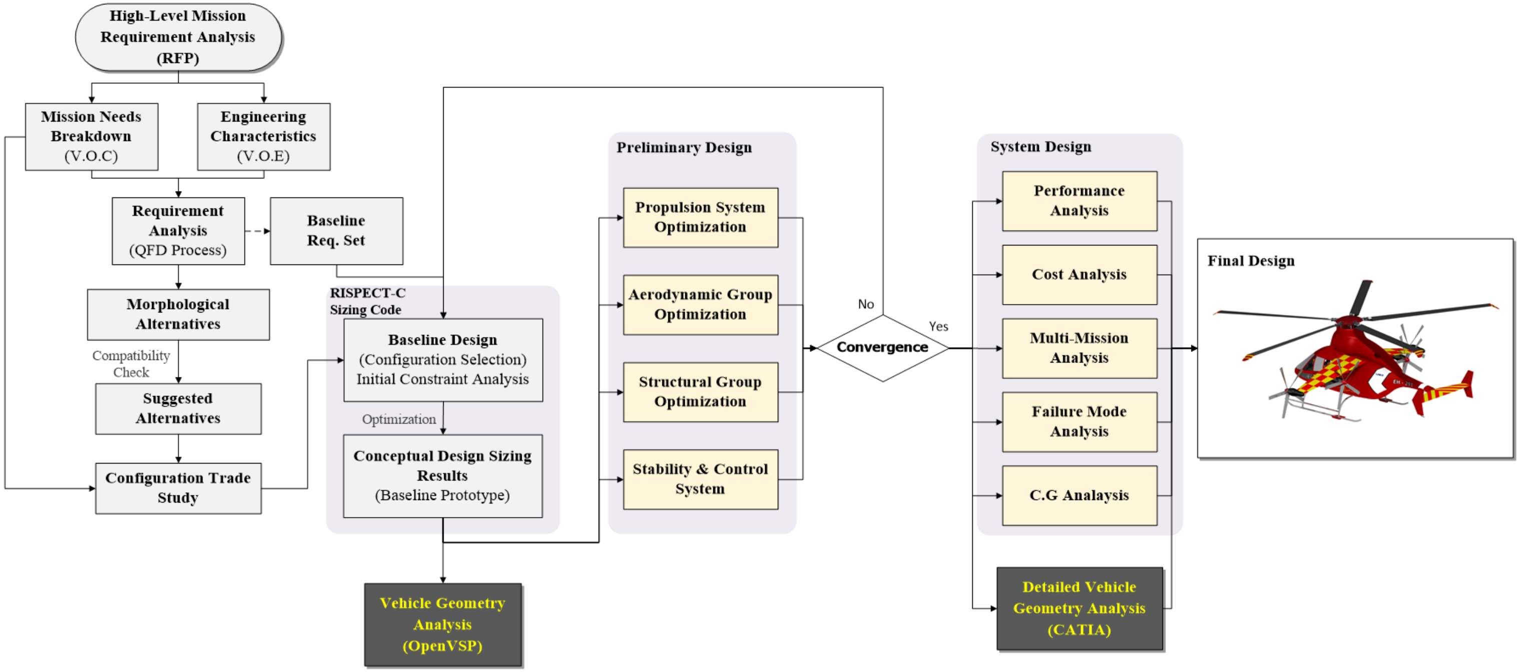


Figure 5 QFD Incorporated Conceptual Design Framework

Configuration Selection

For the final configuration selection, a comprehensive multidisciplinary sizing code is also used. From the QFD analysis planning matrix described in Section II, rotorcraft concepts that were likely meet the mission objectives are distinguished. These configurations that received significant importance are sized using the sizing code for physics-based constraint analysis. Also, subsystem optimizations were conducted on the initial sizing of the selected configurations to determine an optimal solution for the given mission. Based on the final configuration selection, each of the rotorcraft components of the baseline prototype is sub-divided for detailed analysis and optimization (APPENDIX).

Parallel Design Optimization

After obtaining a baseline prototype, subsystem optimization using high-fidelity tools are utilized according to the design parameters ranked in the HOQ process. Each system would apply a necessary design solution to meet individual target value determined from the HOQ. For example, blade planform optimization, propeller optimization, structural optimization and etc. These processes are elaborated thoroughly in the next chapter.

Closing Conceptual Design loop

At the end of the rigorous optimization process, optimized baseline prototype with considerable upgrades that meet the target has been designed. Sufficient information of the rotorcraft should have been identified from the initial sizing. The performance analysis of the vehicle is carried out using the output to ensure that the vehicle satisfies the mission requirements.

IV. Implementation and Results

The proposed design methodology was imposed for the design of the 36th Annual Student Design Competition (SDC) 2019 RFP published by the Vertical Flight Society (VFS). 36th Vertical Flight Society's RFP specified the need for a rotorcraft capable of performing a mountain rescue operation at the highest peak of the planet. The need for Search and Rescue (SAR) operable vehicle is indisputable, and major suppliers have already developed a various configuration of rescue vehicles capable of operating in the unpredictable weather conditions. Regardless, no rotorcraft currently exists capable of reaching the highest peak of the planet and performing mountain rescue operation of stranded mountaineers. This chapter entails the proposed design process framework applied for the implementation and verification of the design process.

The RFP requires that the rotorcraft completes three particular leg profiles as shown in Figure 6. This mission consists of a transfer flight from the international airport to a smaller airport nearby for refueling with 3 crew members and 150kg EMS equipment onboard. The vehicle must take off hover at 1,402m (4,600ft) for 2-minutes, and then climb to 3,780m (12,400ft) for a 135km (73 nautical miles) level cruise. Upon arrival at the smaller airport, it must land for 20-minute refueling. During leg 2 of the mission, the vehicle takeoffs from the smaller airport and climb to an altitude of 8,870m (29,100ft) for a 28km (15 nautical miles) level cruise flight. Once the vehicle has reached the destinations, it must hover for 30-minutes with an additional 170kg (374.786lb) PAX. Then, the vehicle descends down to 3,780m (12,400ft) and level cruise for 28km (15nautical miles). When the vehicle returns to the smaller airport, it must land with a fuel margin of 10%. For the leg 3 returning flight to the larger airport mission, the vehicle must transport the 2 PAX, and fly through the same path back to the international airport.

In summary, the rotorcraft shall be able to accomplish a mountain rescue mission starting from a larger international airport, with 20 minutes refuel stopover at a smaller airport close to the mountain peak, in less than 3 hours, including 30 minutes hovering at the peak during the search and rescue.

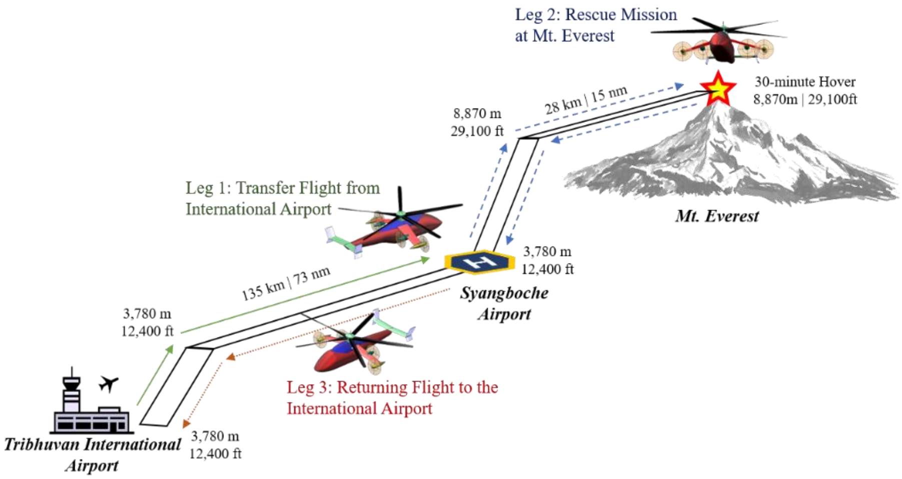


Figure 6 Mission Profile of the 2019 VFS's SDC

Quality Function Deployment

Establishing the Need

Prior to defining and designing a rotorcraft specifically for the extreme altitude mission, a process of establishing design requirements using the QFD process was adopted as part of the proposed design process. In order to evaluate various rotorcraft configuration to capture key objectives and specific requirements set forth by the RFP, the design team has adopted physics-based constraint sizing analysis of various rotorcraft configurations for the final design configuration selection. Furthermore, rigorous iteration process of coupling the comprehensive initial sizing code with numerical analysis was initiated early in the design phase for the best attainable estimates.

Mission Objective

To obtain a better understanding of the mission before narrowing down the design space, brainstorming and detailed analysis of the given RFP identified key objectives and considerations that the rotorcraft specifically designed for high altitude mountain rescue should exhibit. Based on the RFP, top-level mission objectives were identified as follows:

- ✓ **High Altitude Operation**
- ✓ **Flight Safety**
- ✓ **Search & Rescue**
- ✓ **Multi-mission capability**
- ✓ **Emergency Deployment**

When presented with a range of choice of qualitative objectives, it is important to establish systematic evaluation criteria for the final configuration selection. The five top-level mission objectives were established by carrying out a detailed analysis of the given RFP, surveys, and brainstorming. These key objectives were analytically analyzed to prioritize design driving elements using the Analytical Hierarchical Process (AHP). The mission objectives defined earlier were compared to each other to determine their relative importance as shown in Table 1.

Table 1 Extreme Altitude AHP Relative Importance

	Search & Rescue	Emergency Deployment	Flight Safety	High-altitude Operation	Multi-mission Capability	Weight
Search & Rescue	0.221	0.289	0.261	0.138	0.223	0.226
Emergency Deployment	0.121	0.159	0.130	0.274	0.272	0.191
Flight Safety	0.275	0.399	0.326	0.321	0.252	0.315
High-altitude Operation	0.324	0.118	0.206	0.203	0.193	0.209
Multi-mission Capability	0.059	0.035	0.077	0.063	0.060	0.059

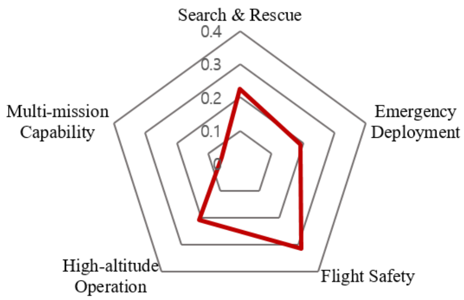


Figure 7 Spider Diagram of Mission Objectives

A spider diagram of the AHP results was generated to better visualize the ranking of the criteria. The outward represents a favorable trait on the spider diagram. It should be noted that the presented outcome does not, in any way, reflect one of these mission objectives are unimportant. As clearly shown in Figure 7, the relative importance of mission goals was as follows:

Flight Safety — As with the design of any rotorcraft, the safety of the vehicle is the utmost priority that every engineer must consider. This mission is a variation of normal operating condition to extreme operating condition. The design drivers that directly affects the safety must be addressed in the conceptual design phase to minimize critical design adjustments in the midst of the detailed design phase. To meet this objective, a redundant system should be available, and no compromise should be made if it directly affects the safety of the rotorcraft in case one fails. Nevertheless, having unlimited safety system has an adverse effect on the overall performance of the vehicle. Therefore, a systematic method of determining key design drivers relating to safety must be carried out in addition to the overall performance of the vehicle before establishing a SAR protocol of stranded mountaineers at extreme altitude.

Search & Rescue — SAR mission in mountainous terrain is considered as one of the most demanding rotorcraft mission. Technical difficulties of the terrains, skills needed by crews and pilot, and the ability to locate and rescue the victims must be addressed. Major design consideration of SAR vehicle includes fuselage volume for 3 crew, 2 medical litters with medical equipment and the necessary supply of oxygen tank for high altitude operation. Moreover, rotorcraft must be equipped with essential equipment such as external/internal hoist system, and Helicopter Terrain Awareness and Warning System (HTAWS) to operate effectively without compromising the safety of the crew. These systems will enable not only the effectiveness of the SAR operation by reducing the pilot workload but also increases the chances of survival of the victims

High Altitude Operation — The physical limitation due to the atmospheric condition has made almost impossible for a conventional rotorcraft to reach the summit of the Everest. SAR operations involve safety risks and require additional considerations for the rotorcraft to operate in extreme conditions. Some of the major conditions that affect the ability to operate at extreme altitude include thin air and extreme weather conditions. Rotor blade stall must be avoided in all scenarios while operating at almost 1/3 of sea level air density. In addition, all air-breathing engines' performance is greatly affected as these engines rely on available atmospheric oxygen for combustion. Operating at high altitude also requires additional consideration for extreme temperature that increase the chance of unwanted icing to occur on the major components. These requirements also directly affect safety of the operation. Hence, requirements related to the high altitude operation must be thoroughly studied, and apply innovative solutions to meet the requirements

Emergency Deployment — For SAR mission, the rotorcraft will need to have the capability to return the injured person(-s) within the "Golden Hour". Aside from performing a systematical SAR mission, the vehicle must be able to quickly be deployed without delays and bring back the litters as soon as possible for medical treatment. This places emphasis on vehicle requiring high cruise speed and low maintenance for emergency deployment.

Multi-role Capability — Rapid reconfigurability of the vehicle and integration of various equipment are also identified as a key design consideration. An economically viable solution for multi-purpose capability is also relevant topics to be identified in the early phase of the design. This directly relates to the cost of manufacturing and development cost of the vehicle. For this, a simple design for low operating cost, configurability, and easy manufacturing will be the design consideration.

These qualitative outcomes from the AHP analysis have provided essential information to identify the relative importance of the mission objectives. The quantitative requirements need to be further broken down for objective decision making in the rotorcraft configuration selection. Therefore, through the evaluation of the mission objectives, QFD was adopted to establish logical, quantitative standards from the presented qualitative objectives.

Gathering Attributes

While identifying the Voice of Customer (VOC), it is important that the engineer's views be excluded as much as possible. A brainstorming with Korea Navy UH-60 pilot and a detailed breakdown of the RFP were conducted. The Voice of Engineer (VOE), namely the quantitative standards, was also established. Rotorcraft performance is divided into 4 disciplines: aerodynamics, propulsion, equipment, and other sub-components. These disciplines were used to determine VOE accordingly as shown in Figure 8 and Figure 9.

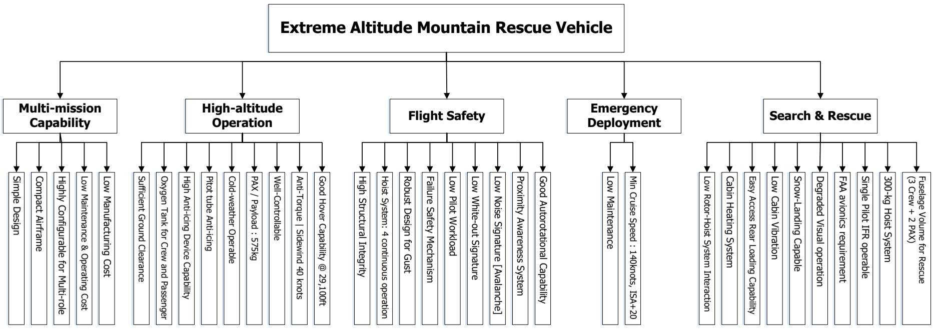


Figure 8 Tree Diagram of VOC

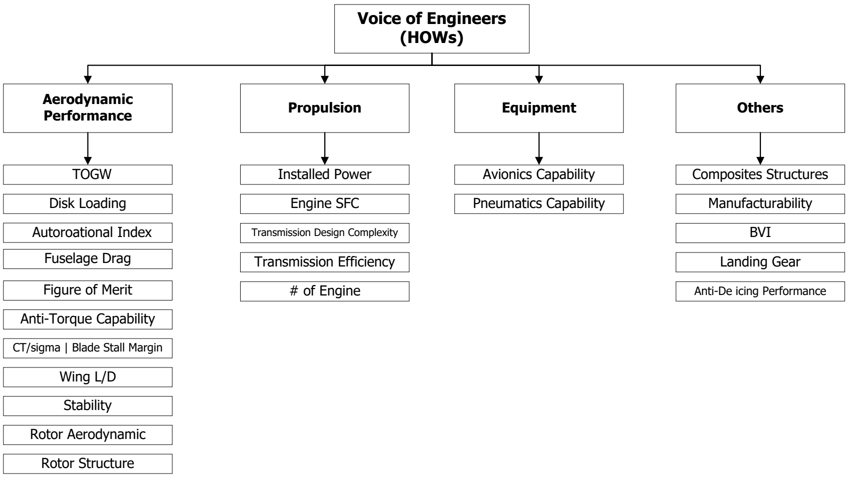


Figure 9 Tree Diagram of VOE

Quantitative Assessments of Mission Requirements

For any multi-attribute problem, the selection of the "best" alternative is inherently subjective with no single answer that fulfills all requirements of the design space. Therefore, QFD using HOQ was adopted to transform user demands into design quality and prioritize array of design characteristics that impact customer attributes, minimizing the gap of information by focusing on customer's needs. The HOQ was executed to identify key engineering aspects and critical parameters that affect the overall design. The most important questions addressed were: What are the customer's needs and how do an engineer meet those needs with necessary design choices.

The customer's needs were reflected on the VOE to rank and quantify engineering requirements from QFD analysis. The weighted importance from the AHP analysis was also incorporated as "AHP normalized weights" in the HOQ. The HOQ shown in Figure 10 identified design parameters such as gross weight, rotor blade aerodynamic, installed power, and other parameters shown in Table 2 to be the most important aspects that would influence the overall design.

				Voice of Engineers (HOWs)																					
				Aerodynamic Performance											Propulsion					Equipment			Others		
													Blade Design												
Voice of Customers (WHATs)		Absolute Weights (1-5)	AHP Normalized Weights	TOGW	Disk Loading	Autoroational Index	Fuselage Drag	Figure of Merit	Anti-torque Capability	CT/sigma | Blade Stall Margin	Wing L/D	Stability	Aerodynamic	Structural	Installed Power	Engine SFC	Transmission Design Complexity	Transmission Efficiency	# of Engine	Avionics Capability	Pneumatics Capability		Manufacturability	BVI	Anti/De-Icing Performance
	Direction of Improvements			↓	↓	↑	↓	↑	↑	↑	↑	↑	↑	↑	↓	↓	↓	↑	↑	↑	↑		↑	↓	↑
High Altitude Operation	Good Hover Capability @ 29,100ft	5	1.044049	9	9	1	0	9	9	3	0	0	9	3	9	9	9	3	3	0	1		0	1	1
	Anti-Torque | Sidewind 40 knots	3	0.626429	3	3	0	0	0	9	1	1	0	1	0	3	1	3	3	3	0	3		0	1	1
	Well-Controllable	4	0.835239	1	3	1	1	0	3	3	1	9	3	1	3	1	1	1	3	3	9		0	1	0
	PAX / Payload : 575kg	5	1.044049	9	3	1	0	3	0	1	1	1	9	3	9	3	1	1	1	0	0		0	0	1
	Cold-weather Operable	5	1.044049	1	1	0	1	3	3	1	3	0	3	1	9	3	1	1	3	3	1		1	0	9
	High Anti-icing Device Capability	5	1.044049	1	0	0	1	0	0	0	0	0	0	0	3	1	0	0	0	1	0		1	0	9
	Oxygen Tank for Crew and Passenger	5	1.044049	3	0	0	0	0	0	0	0	0	0	0	1	1	0	0	0	0	0		0	0	0
	Sufficient Ground Clearance	3	0.626429	0	0	0	0	0	0	1	0	0	0	3	0	0	1	0	0	0	0		3	0	0
Search & Rescue	Fuselage Volume for Rescue (3 Crew + 2 PAX)	3	0.679292	9	3	0	1	3	1	0	0	0	1	0	1	3	1	1	9	3	3		3	3	0
	300-kg Hoist System	5	1.132153	9	1	0	0	1	0	0	0	1	1	1	1	3	0	0	1	0	1		1	0	0
	Single Pilot IFR operable	5	1.132153	0	0	0	0	0	0	0	0	0	1	0	0	0	0	0	1	9	0		0	1	3
	FAA avionics requirement	5	1.132153	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	3	3	0		0	1	3
	Degraded Visual operation	4	0.905722	1	1	0	0	0	0	0	0	0	1	0	1	0	0	0	0	9	1		0	0	3
	Snow-Landing Capable	2	0.452861	1	0	0	0	1	0	1	0	1	3	1	3	0	0	0	0	1	0		0	3	3
	Low Cabin Vibration	3	0.679292	9	3	1	3	1	3	0	0	1	3	3	1	1	3	3	1	0	0		1	9	0
	Easy Access Rear Loading Capability	3	0.679292	1	1	0	0	0	0	0	9	1	1	1	0	0	0	0	0	0	0		1	1	0
	Cabin Heating System	2	0.452861	1	0	0	0	0	0	0	0	0	0	0	3	3	0	0	1	0	1		1	0	3
	Low Rotor-Hoist System Interaction	1	0.226431	1	3	0	1	1	0	1	1	1	9	1	1	1	0	0	0	0	0		0	1	0
Safety Operation	Good Autorotational Capability	5	1.573498	9	9	9	0	1	1	3	3	3	9	3	1	0	1	3	0	0	0		0	9	3
	Proximity Awareness System	2	0.629399	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	3	0		0	0	0
	Low Noise Signature [Avalanche]	3	0.944099	9	3	3	1	3	3	1	1	1	9	3	1	0	1	0	0	0	0		0	9	1
	Low White-out Signature	3	0.944099	3	3	3	0	3	0	0	0	0	3	0	0	0	0	0	0	1	0		0	0	1
	Low Pilot Workload	3	0.944099	1	1	1	0	1	3	3	1	3	3	1	3	1	1	3	1	3	1		0	1	3
	Failure Safety Mechanism	4	1.258799	3	1	9	0	1	3	3	9	3	1	3	3	0	1	3	9	1	1		0	0	3
	Robust Design for Gust	5	1.573498	3	3	3	1	3	3	3	3	3	3	3	3	0	1	0	9	0	3		0	0	0
	Hoist System: 4 continuous operation	4	1.258799	3	1	0	0	3	0	0	0	1	0	0	1	0	0	0	0	0	0		1	0	0
	High Structural Integrity	2	0.629399	3	0	3	1	1	0	0	0	0	0	9	1	0	3	1	1	0	0		1	0	0
Emergency Deployment	Min Cruise Speed : 140knots, ISA+20	5	0.956342	9	9	1	9	9	3	9	9	3	9	3	9	3	3	1	3	1	1		1	0	0
	Low Maintenance	3	0.573805	9	3	0	3	9	1	0	0	0	0	1	1	0	9	9	9	1	3		9	0	1
Multi-mission Capability	Low Manufacturing Cost	2	0.117583	9	9	1	1	1	1	0	3	0	1	1	3	1	3	3	3	3	3		3	0	1
	Low Maintenance & Operating Cost	3	0.176375	9	9	1	9	9	1	0	3	0	3	1	1	9	1	1	1	0	1		0	0	1
	Highly Configurable for Multi-role	1	0.058792	1	1	1	0	1	1	1	3	1	3	3	3	3	3	3	1	1	1		1	3	1
	Compact Airframe	3	0.176375	9	1	0	9	3	1	0	1	1	0	0	1	1	3	1	1	0	1		1	0	0
	Simple Design	5	0.293959	9	1	3	9	1	3	0	0	0	1	0	0	0	3	9	3	0	0		1	0	0
	Weighted Importance			114.93	66.941	44.502	25.28	55.115	43.978	35.318	44.478	33.088	79.046	41.123	68.888	32.78	35.875	32.259	61.343	39.775	26.379		17.84	38.965	47.879
	Normalized Importance			0.1166	0.0679	0.0451	0.0256	0.0559	0.0446	0.0358	0.0451	0.0336	0.0802	0.0417	0.0699	0.0333	0.0364	0.0327	0.0622	0.0403	0.0268		0.0181	0.0395	0.0486
	Rank			1	4	8	20	6	10	15	9	16	2	11	3	17	14	18	5	12	19		21	13	7

Figure 10 House of Quality

Upon quantifying the requirements in Table 2, various parameters such as gross weight and engine power can be explicitly adapted into the sizing optimization process as the main objectives. These quantifications were obtained through extensive brainstorming and through simple mathematical approach such as parametric studies. However, parameters such as the aerodynamics of the main rotor and disk loading must be extensively studied in order to achieve desirable performance at the low-density environment with sufficient stall margin. Likewise, minimized rotor power was desired for the air-breathing engines (if used) because of the effect of the lapse rate. However, requirement such as disk loading was rather difficult to capture without physics-based tradeoff studies. These requirements and various design parameters that require further analysis will be studied through detailed numerical analysis throughout the design process. To meet the target value, various design considerations were also explored. These considerations consider for early adoption of various technologies in the conceptual design phase where design freedom is maximized.

By coupling the comprehensive multidisciplinary sizing tool coupled with multi-objective optimization evolutionary algorithm, a single optimal solution can be obtained. For this thesis, RFP specifically specified for minimized gross weight and power, necessary steps were taken to obtain the solution that meets various requirements quantified from the QFD analysis.

Table 2 Quantification data from House of Quality

Rank	V.O.E	Interpretation	Design Consideration	Target Value
1	TOGW	Directly relates to all aspects of rotorcraft performances, requiring intense studies to reduce the overall weight of the rotorcraft	1. Optimal solution for all variables during initial sizing 2. Composite materials 3. Simple design 4. Low installed power → Less fuel required	3629 kg 8000 lb
2	Rotor Blade Aerodynamic Design	Affects the overall performance and has direct influence to the safe operation of the rotorcraft	1. Optimal airfoil selection 2. Planform shape optimization	Hover Performance @ 8,870m (29,100 ft)
3	Installed Power	All Internal Combustion Engines[ICE] are affected by the lapse rate as they rely on oxygen available in the atmosphere.	1. Electric power source → not affected by the lapse rate 2. Hybridized engine for simple design → reduce weight 3. Additional compressor	2500 hp
4	Disk Loading	Low disk loading is required for good hover capability but high disk loading required for controllability of the rotorcraft. [Optimization required for tradeoff analysis]		19.53kg/m^2 4 lb/ft^2
5	# of Engines	For safety operation of the rotorcraft, redundant system is required in case one engine fails.	1. More than one engine required 2. Emergency power system [auxiliary engine or energy storage system]	2
6	CT/sigma Blade Stall Margin	At high altitude where density is low, collective pitch must be sufficient to generate thrust for hover. For safety of the rotorcraft, sufficient blade stall margin must be secured in case of unexpected gusts	1. Cambered airfoil 2. Low Tip speed 3. Low Disk Loading	Blade stall margin 3°
7	Figure of Merit [Hover]	Direct measure of hover efficiency. Best hover efficiency is desired at 29,100 ft	1. High L/D airfoil at given atmospheric condition 2. High twist → tradeoffs required	0.8
8	Anti-de icing Performance	Since operational condition for current RFP will cause icing to occur due to the extreme atmospheric conditions, all major components that has direct influence on safety, must be fully identified and equipped with necessary equipment.	1. Rotor anti-de icing mechanism analysis 2. Avionics equipped to perform well below freezing point	FAR Requirements
9	Avionics Capability	As the operational conditions at the Everest are unpredictable, all unforeseen situations must be thoroughly identified and avionics must provide sufficient data to the pilots for the safe operation of the rotorcraft.	1. Terrain Awareness and warning system 2. Infrared camera 3. Redundant fly-by-wire system	FAR Requirements
10	Autorotative Index	In case all redundant system and safety features fail, final maneuver that pilot can make is autorotation and it is defined as $AI = \frac{\Omega^2 I_R}{2WDL}$	1. Low Disk loading → large radius or low GW 2. High Vtip 3. Heavy rotor (undesirable)	> 10
11	Empennage Aerodynamics	During cruise flight, highly efficient wing(if applicable) is desired for the operation of the aircraft. In addition, empennage should provide sufficient or even high performance empennage for the controllability in unpredictable weather condition	1. Large wing span 2. Airfoil → high L/D 3. Multi-purpose flaps → increase CL & reduce download due to wake of the rotorcraft 4. Vertical tail → provide all the required yaw at certain speed for safety	FAR Stability Requirements
12	Landing Gear	For various landing capability such as snow landing, uneven terrain landing, and even slope hover without compromising the performance of the vehicle, landing gear should be properly designed and analyze for efficient and safe operation of the vehicle	1. Settling protectors 2. Crampons/spikes 3. Composite landing gear/ski	Skid Type
13	Blade Structural Design	Rotor blade should be designed such that structural safety margin is secured without compromising the weight of the rotorcraft. Blade structural design can also affect the fatigue and vibratory characteristics and therefore require careful analysis of the rotor design.	1. Composite materials 2. Active controls (flaps, piezoelectric devices)	Partial Rigid Rotor
14	BVI	Blade generated vortex interactions could affect the comfort for the crew and passengers. In addition, noise induced or vortex/wake generated by the rotor must not cause catastrophic disaster such as avalanches	1. Low tip speed 2. Blade planform design	FAR Requirements
15	Composite Structures	Composite structure provides light and reliable structural support especially when making the rotor blade and part of the fuselage. If well applied, significant weight can be saved increasing the performance of the vehicles. However, cost considerations must be taken into account.	1. Rotor blade 2. Fuselage structure (skid, skin, empennage and etc)	Maximize

Configuration Selection

VOC was also used to reflect the various feasible designs against each other to narrow down the design space for configuration selection. Existing configurations such as single rotor, coaxial, tandem helicopters, and other compound helicopters were reflected on the VOC. In conjunction with the existing configurations, new configurations of rotorcraft were also explored using a systematic and structured method of morphological analysis as shown in Table 4. This analysis generates and identifies alternative configurations of the system by exploring possible combinations of the components. Through this exploration, and filtering illogical solution combinations, 3 non-existing configurations shown in Table 3 were identified to be reflected on the VOC for final configuration selection. However novel and exciting these new concepts may be for the rotorcraft community, the design team had to consider for the low technology level that could hinder the development of these rotorcrafts.

Table 3 Filtered Morphological Alternatives Analysis

	Configuration 1	**Configuration 2**	**Configuration 3**
Description	Single Rotor Thrust Tail	Quad-Single Rotor	Tandem-Tricopter
Components	Main rotor, Thrust Tail	Main rotor + 4 Props	3-Rotor (tandem + side-by-side)
Anti-torque Mechanism	Tail rotor	Tilting Props	Side-by-side
Engine Type	Turboshaft	Parallel Hybrid	Turboshaft
Control Mechanism	Pitch Control	Pitch & RPM Control	Pitch Control
Forward Thrust	Thrust Props	Tilting Props	Main rotors
Conceptualization			

Table 4 Morphological Alternatives Chart for Rotorcraft Design

Characteristics \ Alternatives	1	2	3	4	5
Aircraft Types	Fixed-wing	Rotorcraft			
# of Rotors	1	2	3	4	5
Rotor Control	Pitch	RPM	Flaps		
Hub Configuration	Fixed	Teetering	Articulated	Bearingless	Hingeless
# of Blade	1	2	3	4	5
Rotor Overlap	Yes	No	Completely		
Wing	Yes	No			
Propulsion Type	Shaft driven	Tip-Driven	Rotor-Driven		
Anti-torque	Yes	No			
Thrust	Main rotor	Propeller	Rocket	Jets	
Rotor Configuration	Conventional	Side-by-side	Convertible	Synchropter	Multi-rotor
Landing Gear	Skid	Skid crampons	Ski type	Wheeled	
Engine Type	Turboshaft	Diesel	Hybrid	Electric	Nuclear
Engine #	1	2	3	4	
Vertical Tail	Single Tail	H-tail	V-tail		
Tail wing	Single	T-tail	V-tail	Boom tail	
Wing type	Conventional	Canard	Boxed wing	BWB	
Tail rotor	Tractor	Pusher	Fan-in-win	NOTAR	Winged Prop
Fuselage	Utility round	Streamlined	Others		
Ducted	Yes	No			
Cabin Door	None	Sliding	Passenger Ramp		

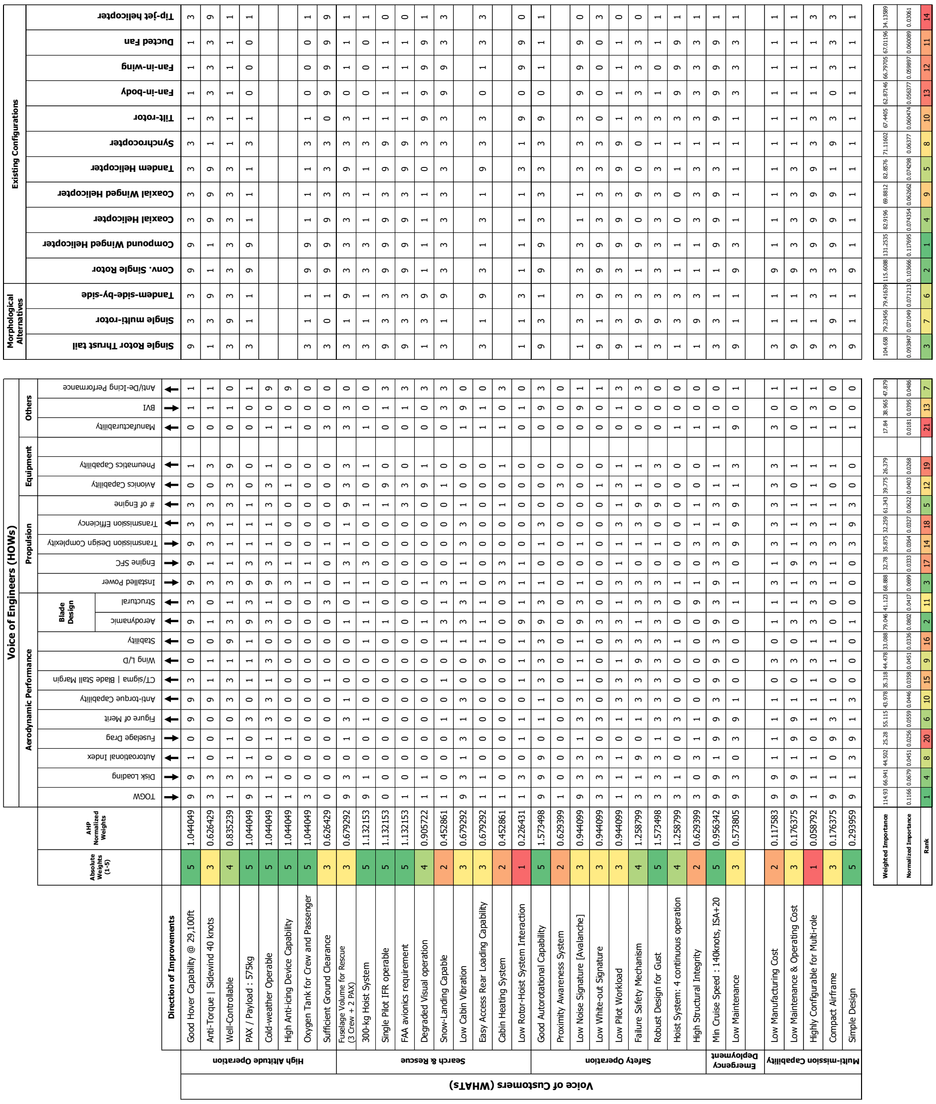


Figure 11 Complete House of Quality

By reviewing various concepts on the VOC, explicit requirement such as hover capability at 8,870m (29,100ft) eliminated concepts like the tilt-rotor, fan-in body and ducted fan concepts due to their high disk loading. Air density at the peak would be extremely low and rotorcraft concepts would require excessively large rotor without stalling the rotor. The interpretations in Table 2 were used in conjunction with the VOC to reflect how each concept meets the explicit requirement set forth by the RFP. In addition, RFP specifically specified that particular attention must be paid to design a compact airframe/rotor configuration. These requirements provided significant guidance on qualitative comparisons of various concepts and narrowed down the design space for physics-based sizing analysis in Table 6.

Morphological Alternatives			Existing Configurations										
Single Rotor Thrust tail	Single multi-rotor	Tandem-side-by-side	Conv. Single Rotor	Compound Winged Helicopter	Coaxial Helicopter	Coaxial Winged Helicopter	Tandem Helicopter	Synchrocopter	Tilt-rotor	Fan-in-body	Fan-in-wing	Ducted Fan	Tip-jet helicopter
104.658	79.23456	79.41639	115.6088	131.2535	82.9196	69.8812	82.8576	71.11602	67.4405	62.87146	66.79705	67.01196	34.13589
0.093847	0.071049	0.071213	0.103666	0.117695	0.074354	0.062662	0.074298	0.06377	0.060474	0.056377	0.059897	0.060089	0.03061
3	7	6	2	1	4	9	5	8	10	13	12	11	14

Figure 12 Rotorcraft viable concepts for Extreme Altitude Operation

Through this analysis, rotorcraft concepts that were likely meet the mission objectives were distinguished, and most viable concepts were ranked as follows:

Compound Winged Helicopter — Compound Winged helicopter demonstrated to be the superior solution that meets the objective set forth by the RFP. Lift and propulsion augmented design similar to the Eurocopter X3 was considered for compound winged helicopter concept. Winged helicopter uses wings to offload the rotor and its rotor is slowed down to avoid drag divergence at higher airspeeds. In addition to propulsive force from the main rotor, propellers on the wing provide propulsive force and necessary anti-torque. Like single rotor helicopter, winged helicopter exhibit high hovering efficiency due to its low disk loading relative to other concepts. However, transmission complexity to operate the propellers and rotor-propeller-ground clearance must be addressed to ensure safe operation of the vehicle. Although no winged helicopters are currently in full production, the demonstrators X3 has provided sufficient examination of the potential of this configuration to meet the requirement set forth by the RFP.

Figure 13 Winged Helicopter, Eurocopter X^3 [5]

Single Rotor Helicopter/thrust tail — Conventional single rotor helicopter and single rotor helicopter with thrust tail were considered to exhibit similar characteristics aside from augmented thrust produced by the thrust tail. The single rotor offers various advantages such as low empty weight fraction, high reliability, good hover capability, and relatively low production and maintenance costs. However, single rotor helicopters have relatively poor range and physically limited forward speed due to various effects such as blade stall in the retreating side, compressibility effect in the advancing side, and lift imbalance. Since the RFP only states the minimum cruise speed of 140 knots which is a typical maximum cruise airspeed of the single rotor, this concept will still be considered for physics-based sizing analysis for final configuration selection.

Figure 14 Single Rotor EC-145 [6]

Coaxial Helicopter — The coaxial rotor such as Sikorsky X2 with thrust compounding has significantly high cruise speed, and provides compact fuselage by eliminating the need of tail rotor by having counter-rotating rotors counteracting the torque generated. The main advantage of this configuration is that the retreating side of each rotor is offloaded during forward flight. However, the rigid rotor must be used to prevent the danger of counter-rotating upper rotor and lower rotor to strike each other during unpredictable flapping. This rigid rotor significantly increases the overall weight of the aircraft in addition to its bulky and complex counter-rotating transmission/hub. Despite these drawbacks, the intrinsic anti-torque system and high disk loading for controllability have made this concept a viable option for extreme altitude SAR mission

Figure 15 Coaxial Rotor X2 [7]

Tandem Helicopter — Tandem rotor has already proven its capability through military operations and it is undeniable that a tandem rotor is a viable option for SAR mission. Tandem rotor, typically used for carrying large payloads, is a configuration with one rotor situated at the front and the other at the rear of the helicopter enabling a wider range of center of gravity travel and is typically used for carrying large payloads. With a large amount of open cabin space and low disk loading, tandem rotor seems a viable option for SAR mission as it can carry a high payload. However, due to its large rotors and airframe causing relatively high parasitic drag, this configuration has poorer fuel economy despite being able to cruise at relatively high airspeed. In addition, relatively high empty gross weight, cost of production are other weak points of this configuration. Most importantly, bulky airframe/rotor configuration is directly opposite to the requirement of the RFP mandating for compact airframe, which makes the tandem rotor excluded for the constraint sizing analysis in Table 6.

Final Selection

The RFP defined a mountain rescue mission starting from a larger international airport, with possible refuel stopover at a smaller airport close to the mountain peak with three crew and 150 kg of EMS equipment. The 150 kg EMS equipment consists of EMS interior with one cabin seat and two stretchers, medical floor, oxygen system, and medical kit. Upon refueling at the smaller airport, the vehicle will need to climb and perform 30 min hover at 8,870m (29,100 ft) with an additional 2 PAX onboard. Due to strong winds at 8,870 m (29,100 ft), the RFP also specified that the vehicle control system must be able to maintain its heading in hover with wind from any azimuth up to 74 km/h (40 knots). Moreover, the rotorcraft must include an internal or external hoist system rated for a 300 kg load. Upon returning and refueling at the nearby smaller airport, the vehicle must takeoff and cruise descent to the international airport for medical treatments. These are the mission profiles that the final design should perform on a regular basis.

Table 5 Detailed Mission Breakdown

	Mission Description	Time [hour]	VROC	Speed	Altitude
Leg 1	Taxing	0.033	0 km/h (0 knots)	0 km/h (0 knots)	1,402m (4,600ft)
	Takeoff Hover	0.033	0 km/h (0 knots)	0 km/h (0 knots)	1,402m (4,600ft)
	Cruise Climb	0.160	14.8 km/h (8 knots)	305.6 km/h (165 knots)	1,402m (4,600ft)
	Outbound Cruise	0.251	0 km/h (0 knots)	342.6 km/h (185 knots)	3,780m (12,400ft)
	Landing Hover	0.033	0 km/h (0 knots)	0 km/h (0 knots)	3,780m (12,400ft)
	Refueling	0.333			
Leg 2	Taxing	0.033	0 km/h (0 knots)	0 km/h (0 knots)	3,780m (12,400ft)
	Takeoff Hover	0.033	0 km/h (0 knots)	0 km/h (0 knots)	3,780m (12,400ft)
	Cruise Climb	0.275	18.5 km/h (10 knots)	185.2 km/h (100 knots)	3,780m (12,400ft)
	Outbound Cruise	0.025	0 km/h (0 knots)	185.2 km/h (100 knots)	8,870m (29,100ft)
	Hover + Payload(170kg)	0.5	0 km/h (0 knots)	0 km/h (0 knots)	8,870m (29,100ft)
	Cruise Descent	0.275	-18.5 km/h (-10 knots)	185.2 km/h (100 knots)	8,870m (29,100ft)
	Inbound Cruise	0.025	0 km/h (0 knots)	185.2 km/h (100 knots)	3,780m (12,400ft)
	Landing Hover	0.033	0 km/h (0 knots)	0 km/h (0 knots)	3,780m (12,400ft)
	Taxing	0.033	0 km/h (0 knots)	0 km/h (0 knots)	3,780m (12,400ft)
Leg 3	**Refueling**	0.333			
	Taxing	0.033	0 km/h (0 knots)	0 km/h (0 knots)	3,780m (12,400ft)
	Takeoff Hover	0.033	0 km/h (0 knots)	0 km/h (0 knots)	3,780m (12,400ft)
	Cruise Descent	0.160	-14.8 km/h (-8 knots)	305.6 km/h (165 knots)	3,780m (12,400ft)
	Inbound Cruise	0.251	0 km/h (0 knots)	342.6 km/h (185 knots)	1,402m (4600ft)
	Landing Hover	0.033	0 km/h (0 knots)	0 km/h (0 knots)	1,402m (4600ft)
	Taxing	0	0 km/h (0 knots)	0 km/h (0 knots)	1,402m (4600ft)
	Total	**2.89**			

As clearly shown in Table 5, to meet the mission requirements of performing the mission within 3 hours, minimum outbound cruise speed should be 185 knots or 342.62 km/h (212.9mph). Minimum cruise speed of 342.62 km/h (212.9mph) is a key design driver that will impact the configuration of the final design due to the physical limits of the rotorcraft. Previous QFD analysis was able to capture the qualitative requirements but was insufficient to determine quantitative requirements that the rotorcraft must comply. Breakdowns of the mission identified two of the harshest mission, which was the outbound cruise for the transfer flight (leg 1) and OGE hover at 8,870m(29,100ft).

For the final configuration selection, a comprehensive sizing code described in the APPENDIX were used for constraint analysis and comparison of the aforementioned rotorcraft configurations. Also, subsystem optimizations were conducted on the initial sizing of the winged, coaxial, and single rotor helicopters based on the mission profile shown in Table 5. These baseline results served as an important representation for the final design and enabled the design team to compare and decide which configuration best suits for a rescue vehicle in the extreme altitude mountain. The single rotor and the thrust augmented single rotor was unable to meet the cruise speed requirement due to the physical limits of the main rotor blade. Based on the sizing results, a comparison of the thrust augmented winged helicopter, and the thrust augmented coaxial helicopter was carried out.

Table 6 Configuration initial sizing results comparison

	Conventional Helicopter	Winged Helicopter	Coaxial Helicopter
Gross Weight	-	3026.8kg | 6673 lb	3648.7kg | 8044 lb
Eng. Power	-	1706.9 kW | 2289 hp	2948.5 kW | 3954 hp
MR Radius	-	4.73 m | 15.52 ft	4.46 m | 14.62 ft
MR Chord	-	0.3 m | 0.97 ft	0.43 m | 1.412 ft
	Unable to meet cruise speed criteria		Requires complex coaxial transmission design with thrust propellers

Clearly from the physics-based sizing analysis in Table 6, thrust augmented winged helicopter was selected due to its exceptional capabilities in all aspects including safety, SAR, high altitude operation, emergency deployment, and multi-mission capability. The placement of wings to carry load presents an opportunity for efficient, simple, and most importantly, safe operation of the vehicle. This configuration, despite having the possibility of whirl flutter and complex transmission design, Eurocopter has successfully demonstrated its capability without stability augmentation system.

Apart from the sizing results in Table 6, thrust augmented winged helicopter exhibits explicitly stated RFP design requirements that placed emphasis on designing a compact, simple, light, and efficient rotorcraft for what was to become the *CRANE*.

Summary of Baseline Prototype

Through the rigorous sizing process, a total of 3 baseline prototypes were designed with major systems analyzed. The main purpose of the initial sizing was to provide the most realistic design through detailed analysis, attempting to cover all aspects of aerodynamics, structure, propulsion system, weight, and stability controls. By adopting the QFD analysis and detailed analysis, a compound winged helicopter with distributed wing-mounted propulsion system was proposed. A single rotor together with the wing generates lift while four propellers mounted on the wing provide thrust required to overcome drag and main rotor torque. The initial conceptualization of the proposed concept using OpenVSP with 4 wing-mounted propellers is shown in Figure 16.

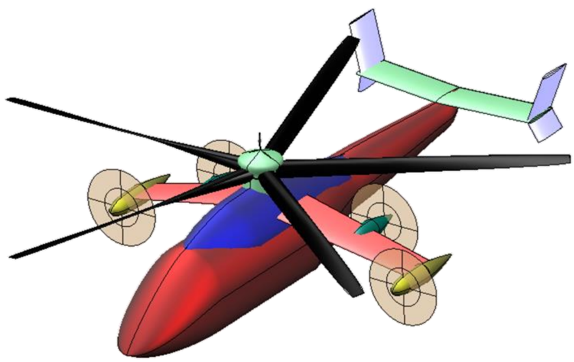


Figure 16 Initial Conceptualization of the Baseline Prototype

Parallel Design Optimization

After obtaining a baseline prototype, subsystem optimization using high-fidelity tools are utilized according to the design parameters ranked in the HOQ process. Beginning with the rotor group design where QFD analysis showed to have the most influence on meeting the needs of the customer, detailed optimization is carried out to achieve the target goal.

Rotor Group Design

Designing a rotor blade planform requires multi-disciplinary analysis as the parameter attributes of a rotor efficiency in hover and rotor in forward flight are contradictory. These contradictory design requirements often result in a design that is efficient in neither hover nor forward flight. However, previous study on initial sizing showed that hovering rotor at 8,870m (29,100ft) requires the most focus as it was determined to be the harshest mission. Design of blade planform was carried out using the in-house hybrid CFD tool described in the APPENDIX. By following up from the mission analysis where hovering at 8,870m (29,100ft) was essentially the harshest mission for the engines, main rotor, and the wing-mounted propellers, the main rotor was specifically designed for maximized hover efficiency with minimal forward flight performance degradation.

Rotor Planform Parametric Study

As prescribed from the QFD analysis where the rotor blade requires significant design studies for safety, the detailed and rigorous design process has achieved significant improvements to the aerodynamic performance of the rotor blade.

In the aerodynamic design of the rotor blade, the tip shape and the airfoil are known to influence most on the overall performance of a rotorcraft. As such, a tradeoff study of the existing rotor blade tip shape, followed by the airfoil shapes were carried out. Through parametric studies and optimization of the main rotor blade for maximum aerodynamic efficiency such as figure of merit of 0.8 and stall margin of 3˚, final blade was optimized as shown in Figure 17.

Figure 17 shows the initial baseline rectangular rotor with NACA0012 versus the final rotor blade optimized for maximum hover efficiency. The final rotor blade configuration with NASA RC(4)-10 airfoil, parabolic planform shape, 30˚ anhedral angle, and modified NASA RC(4)-10 airfoil with thickness 7% was shown to be the superior design for high altitude hover. Through the optimization, significant hover efficiency and stall margin were achieved, enhancing the overall safety of the vehicle.

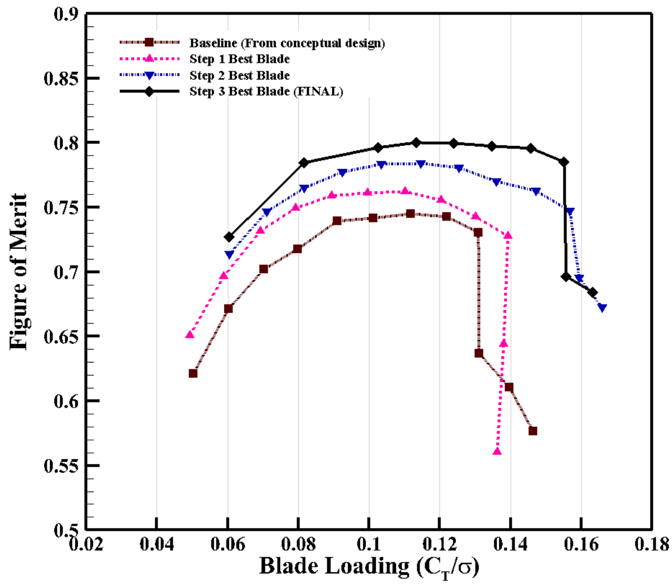


Figure 17 Final Blade Planform Optimization

Rotor Structural Design

Furthermore, the structural design of the rotor blade was designed to ensure a sufficient margin of safety. The design process of the KSEC-2D cross-section is depicted in the APPENDIX. Through this process, a design satisfying the design space was obtained. The T300 graphite/epoxy that has high stiffness, ultimate strength, and lightweight was selected for the main material for the composition of the rectangular spar and skin on which most loads are exerted on. The layer pattern of the rectangular spar consisted of ±45° for the torsional stiffness of the outer and the uni-directional plies for the flap stiffness. Layers with 90˚ plies were laminated with [±45°, ±45°, 0°, 90°, 0°, 0°, 0°, 90°, 0°] layer patterns respectively to prevent delamination of the composite structure. The accumulation of ±45° layers was applied to the skin for high torsional stiffness. The leading edge was composed of titanium cap for anti-erosion, and tungsten-tuning mass was placed at 25% c.g location. Uni-directional S-glass was used as the leading edge fill to enhance the stiffness of the closed section, and Rohacell foam was used to fill the remaining empty volume. Likewise, Rohacell foam also fills the rectangular spar while the Kevlar honeycomb core constitutes the remaining volume from the spar aft to the trailing edge. An overview of the internal structure is depicted in Figure 18.

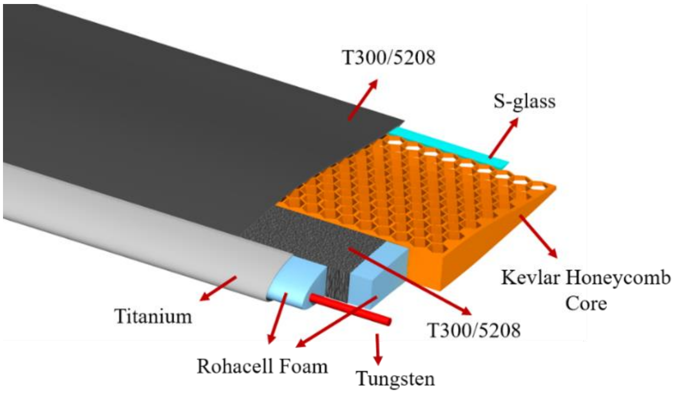


Figure 18 Rotor Blade Internal Structure Design

Rotor Hub Design

The *CRANE* employed an inter-blade damper elastomeric bearing hub due to its availability and lower drag profile shown in Figure 19. The inter-blade hub, compared with a conventional configuration such as articulated which dampers are interposed between each blade and the rotor hub, increases the lever arm between the dampers and drag axes. This also serves to cause two dampers to act on each blade and thereby reducing the ground resonance. The uniqueness of the rotor hub is therefore favorable for combating the ground resonance. Furthermore, *CRANE*'s blade has a relatively compact root cutout of 0.15R implying less room for pitch horns, damper and necessary flap/lead-lag mechanism which led to the final inter-blade rotor hub selection.

The inter-blade hub employs unique elastomeric bearing for the flap, lead-lag, pitch motion to provide necessary rotor dynamics making the inter-blade hub simple [15]. The elastomeric bearing is shown in Figure 19 consists of alternating layers of metal plate and elastomer for the flap, pitch, and lead-lag motion. The torsional moment and blade shear reactions are transferred to the elastomeric package. The top surface of the elastomeric bearing is mounted on the main rotor mount while the bottom surface is mounted to the rotor hub mounting flange, as shown in Figure 19. The conical bearings are always under compression due to the rotating rotor reacting to the dynamics for the main rotor.

Each inter-blade damper is fastened to two blades by a ball joint centered on the pitch axis. Pitch link is connected to the pitch horn located in between the elastomeric bearing mount and the hub for pitch control. Aerodynamic cowling was designed to reduce drag during forward flight and protect the hub from an external substance. As a result, a considerable level of inter-blade oscillation such as lead-lag movements can be reduced, making it possible to increase the lifetime and performance of the dampers.

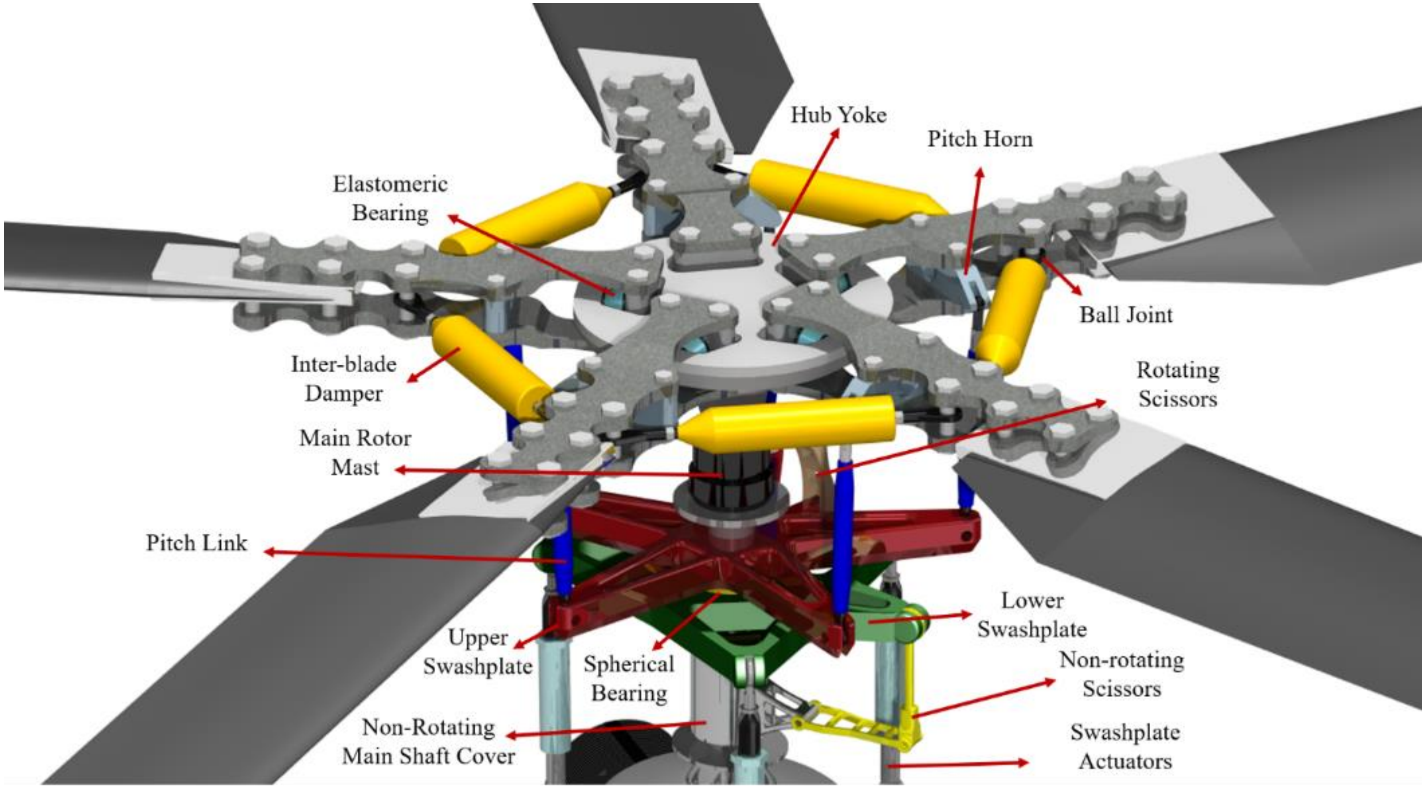


Figure 19 Interblade Rotor Hub Design

In addition to the design of the main rotor, a novel design of variable pitch propeller propelled by the three separate motors connected in parallel was designed. By adopting the variable pitch propeller, constant propeller efficiency was achieved throughout the mission profile. The propeller blade twist was optimized for hovering at 8,870m (29,100ft). Together with the main rotor blade performance, the *CRANE* was designed to meet the requirement of the RFP and serve as the safest platform for extreme attitude mountain SAR mission.

Propeller Design

The optimization of the propeller blade was also carried out in conjunction with the design of the propeller hub. Because all rotating elements are required to provide thrust during forward flight and hover, the blade must be specifically designed to perform in the extreme altitude. The given design point of the propeller was also similar to the design of the main rotor where detailed analysis using high fidelity tool was required in the conceptual design phase for reliable sizing of the propeller.

Upon detailed tradeoff analysis and brainstorming, variable pitch propeller was chosen over the variable RPM propeller because of the responsiveness of thrust generation which directly affects the safety and controllability of the vehicle. Furthermore, a fixed pitch propeller was unable to provide the required thrust during forward flight while the variable pitch propeller can be optimized to provide necessary torque almost instantly. To this end, a novel design of the wing mounted propeller, and an optimized planform of the propeller blades was proposed for the design of the *CRANE*.

As discussed thoroughly, the anti-torque requirement at the extreme altitude has led to the novel design of the *CRANE* adopting four wing-mounted propellers. These propellers would undergo from normal operating condition to extreme condition. During hover, these propellers must provide the necessary thrust to counteract the torque generated by the main rotor. These propellers must also provide the necessary thrust to enable *CRANE* to cruise at 185knots in order to operate within the “Golden Hour”. The mission analysis deduced that the harshest mission for both the engine and the rotating elements was hovering at 8,870m (29,100ft). With variable pitch, the operable space of the *CRANE* was significantly enhanced.

For the design of the propeller blade, the ADM method introduced in the initial sizing phase was used for the parametric study. The initial ADM result has shown that significant consideration should be accounted for the twist angle of the propeller. This is mainly due to the high thrust that the propellers must generate during high-speed cruise flight and at extreme altitude, securing significant stall margin was important for the safe operation of the vehicle. The airfoil used was the NACA0012 airfoil mainly due to the reverse thrust that the propeller had to generate during hover and transition. Based on the hover design point, the final propeller for the *CRANE* had a twist of -39.1˚ acquiring 3˚ stall margin as shown in Figure 20. During hover at the 8,870m (29,100ft), the required collective pitch angle (measured at the root) was 35.5˚.

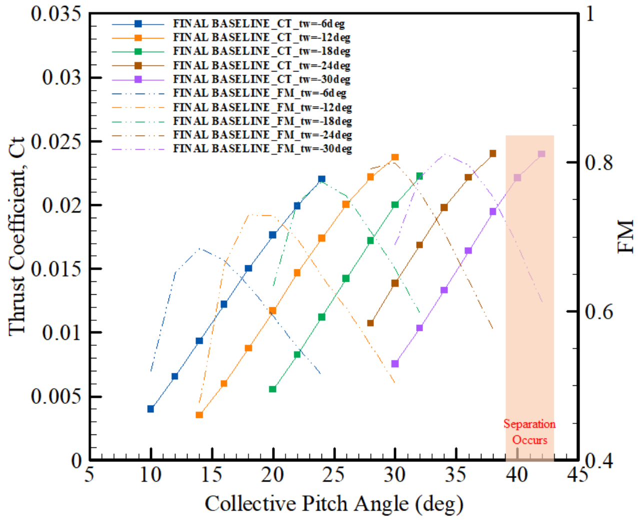


Figure 20 Wing-mounted Propeller ADM Optimized Performance

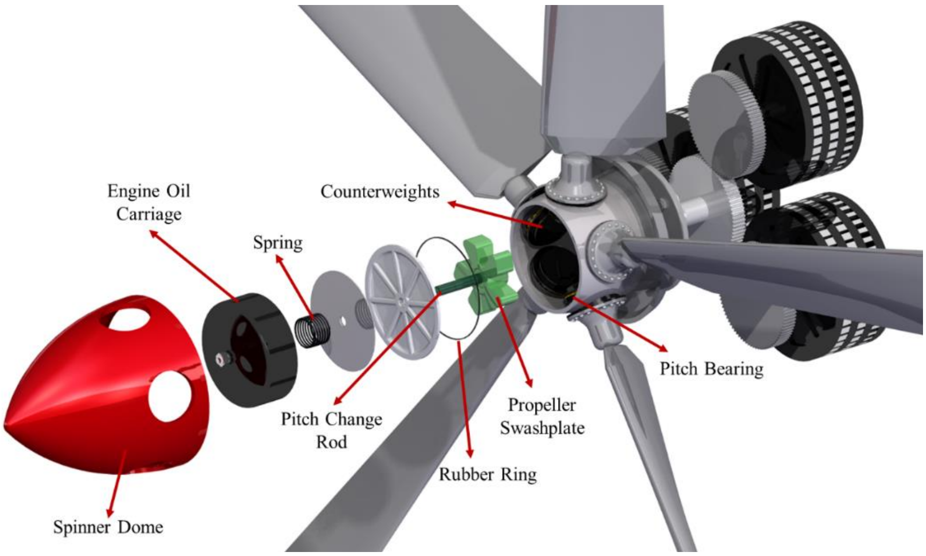


Figure 21 Wing-mounted Propeller Hub Design

Propulsion Group Design

The precedent winged helicopter such as the Eurocopter X3 employed a variable rotor speed. This variable rotor speed design adds complexity to the drivetrain of the propulsion system but essential due to the physical limits of the rotor to avoid drag divergence at the advancing side for high-speed maneuver. The propellers that provide thrust and required anti-torque are driven by a shaft connected to the main gearbox. However, the initial sizing of the *CRANE* design required four propellers mounted on the wings to meet the anti-torque requirement. The expected complexity by this propulsion mechanism in addition to the engine's lapse rate effect has triggered the design team to explore and suggest a unique drivetrain design.

Drivetrain Requirement

It was a significant challenge for the gas turbine engine to provide sufficient power to the rotor at 8,870 m (29,100 ft) at ISA+20. Based on the initial sizing, the required shaft power to operate the rotors and the propeller during 8,870m (29,100ft) hover was 1051hp. To provide such power would place a significant challenge to the turboshaft engine at extreme altitude. In addition, a highly efficient transmission system capable of reducing the rotor speed without affecting the propeller speed must be explored.

Propulsion System Architectures

Because this vehicle was designed to act as a reliable and safe platform for SAR mission, turboshaft engine was selected for its reliability and high power-to-weight ratio. In addition, the turboshaft engine generally has higher maintenance span of 1,000 hrs or so which are relatively higher than the piston engines. Initial sizing sized a turboshaft engine that had a Mean Sea Level (MSL) Maximum Continuous Power (MCP) of 1230.6hp and Intermediate

Rated Power (IRP) power of 1458 hp. However, it was decided that running this configuration solely from the turboshaft engine was incompatible due to the complexity of the propeller gearboxes. Furthermore, a significant reduction of the output power of the engine at extreme altitude has resulted in an oversizing of the turboshaft engine. Through investigation of current engine technology, it was identified that engine manufacturers have already gone through a rigorous process of optimizing the engine compressors and turbines to maximize efficiency while providing reliable and safe operation. In the process of determining the best propulsion system layout for extreme altitude atmosphere, two possible alternatives aside from the conventional turboshaft architecture were considered.

Fully Electrified: Fully-electrified concept was particularly considered for the alternative propulsion as the battery does not rely on the atmospheric air and hence has a great advantage on the extreme altitude where oxygen content is severely low. The battery trend was extrapolated based on the NASA Advanced Battery Technologies Forum [16] which shows a rough estimate of battery specific energy of 200 Wh/kg in 2019. Unfortunately, the state-of-the-art battery was unable to meet the requirements due to its low specific energy, increasing the overall empty weight of the vehicle.

Serial-Partial-Hybrid: Among many hybrid architectures, Serial-Partial-Hybrid shown in Figure 22 was considered as the second alternative propulsion system. This architecture uses both the battery and the power generated from the engine to augment power for the propellers during high power loads such as hovering at 8,870m (29,100ft) and high-speed cruise flight. Since only the most essential battery for the propellers is used for this concept, battery weight is far less than other hybridized concept.

The fully electric propulsion system was discarded after tradeoff studies and carrying out sizing analysis using the sizing code. The sizing code was unable to converge showing that fully electrified concept will not work at the current technology level. Until there are significant battery advancements to have much higher specific-power of more than 1000Wh/kg using new material such as Li-S, fully electrified propulsion design will not be attainable. The hybrid-electric propulsion system, on the other hand, has proven to be achievable through initial sizing process. Since this propulsion system does not solely rely on the battery, a converged solution was obtained even with current battery technology. Adopting Serial-Partial-Hybrid propulsion system, overall transmission architecture could also be simplified over the shaft driven concept. The electric power generated from the generator and the battery is transmitted to the propellers through electric wires. Such architecture enables distribution of power without the need for gearboxes and shafts, which could greatly reduce the weight and complexity of the vehicle drivetrain.

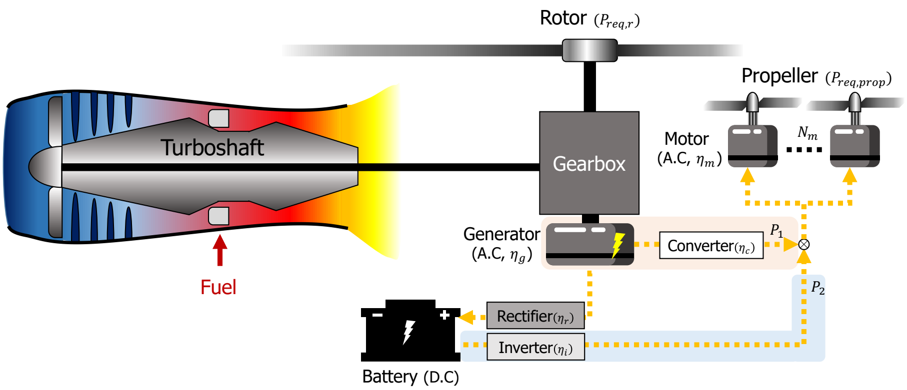


Figure 22 Serial-Partial Hybrid Drivetrain Architecture

The main rotor is driven by the turboshaft engine connected to the variable speed reduction gearbox. The generator shafts are also connected to the reduction gearbox which is powered by the turboshaft engine to generate electrical energy for the battery or the propeller motors accordingly. Dual-speed transmission design is assumed to be adapted to enable variable rotating speed of the main rotor rpm while generating a constant power for the generator.

Adopting the hybrid architecture required additional consideration and how the engine should be sized. Detailed studies and brainstorming showed that one of the most effective methods was to determine the Degree of Hybridization (DOH), which is the ratio of electric power divided by total power.

$$\mathrm{DOH} = \frac{\mathrm{P_{EM}}}{P_{EM}+P_{turb}} \tag{1}$$

By implementing equation (1), the engine could be sized to operate at its MCP throughout the mission as much as possible. In general, the turboshaft engine has better power-to-weight ratio when operating at its MCP. As depicted in Figure 23, the deficient power shown in red, after implementing a partial-hybrid parameter, is provided by the battery. Since this propulsion architecture does not fully rely on the battery while implementing its advantages of being able to produce its full power, the performance of the vehicle was enhanced. Therefore, it was decided that a hybrid electric propulsion system was indeed an essential and outstanding solution for the final propulsion architecture.

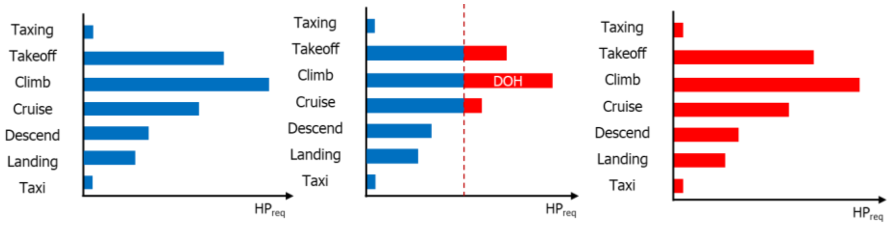


Figure 23 Hybrid-Electric Degree of Hybridization [DOH]

To accomplish the requirement of obtaining the variable speed rotor while generating constant power for the propellers, a few transmission designs were analyzed to identify the best option for the variable speed rotor. It was decided that manageable accommodation for dual-speed transmission design could be adopted by placing two planetary gear sets stacked on top of each other that is controlled by a clutch controller. This design enables a smooth transition without compromising the safety of the vehicle. Assuming constant output RPM of the turboshaft engine, the final shaft RPM was calculated to be 238 and 220 for the main rotor. After the overall layout of the transmission was decided, the gear ratio for the generator was calculated. It was assumed that the rotating speed for the best performance of the Halbach Array motor was 3200 RPM.

The drivetrain starts at the GE-T700 "rubberized" engine which has 21945 RPM at MCP condition. The overrunning clutch is attached to the engine to allow the engine to drive the gear but does not allow the gear to drive the engine during an emergency maneuver such as autorotation. A pinion shaft gear with a reduction ratio of 0 is placed to lower the engine input location and shift the engine forward to balance the center of gravity.

A reduction of the engine output RPM starts at Stage C by a simple pinion shaft gear with a reduction ratio of 0.364. The Stage D central gear and Stage E spiral gear have a helix angle of 30˚. This was intended to distribute high stress applied by both engines' torque directly and reduce vibrations associated with the gearbox. Using helical gears generates an axial load, but careful placement of the thrust bearings can balance the load without significant structural weight addition.

The *CRANE* is designed to reduce the RPM of the main rotor by roughly 9% when the airspeed reaches 110 knots. Stage F and G are stacked on top of each other for the variable rotor speed. During normal operation, the clutch controller engages stage F planetary gears while holding the Stage G reduction ratio of 1. When the airspeed reaches above 110 knots, the clutch controller disengages the Stage F planetary gears and engage Stage G gears for reduction. All of these operations are carried out without affecting the RPM of the generators that are attached to the Stage E spiral bevel gear for constant power generation, as shown in Figure 24. Gear layout is also given in Table 7.

The shafting is hollow to reduce the weight while ensuring that the shaft can withstand the torsional stress and various loads within the transmission. The gearbox casing is to be manufactured from aluminum by an aluminum 3D printer for weight savings and accuracy in manufacturing. Aluminum has a lower thermal expansion rating than magnesium, allowing better alignment of the transmission at extreme temperature. Static masts are mounted on the transmission casing to the keel beams and bulkheads of the airframe structure. The overall weight of the variable transmission was 534.18 lb.

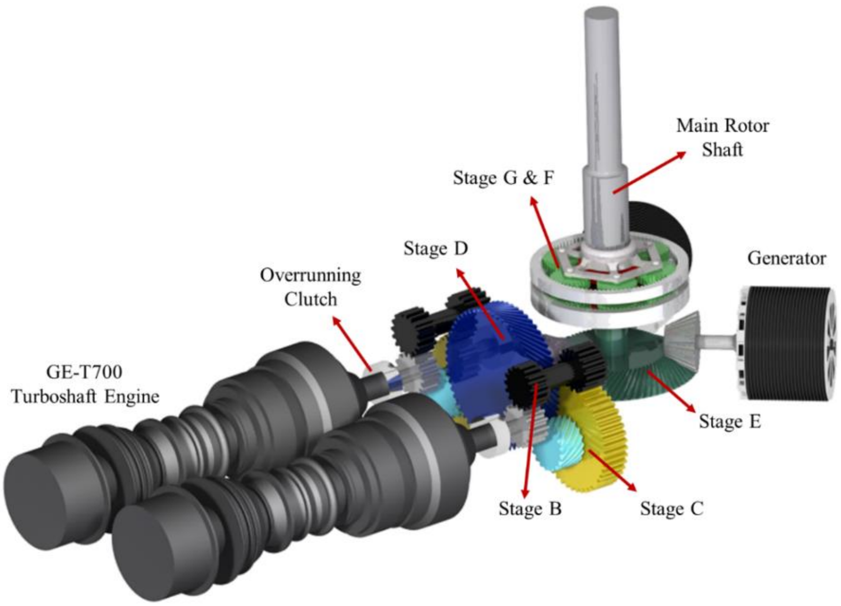


Figure 24 CAD overview of Drivetrain Architecture

Table 7 Transmission Layout Design Overview

<table>
<tr><th>Stage</th><th>Gear Type</th><th></th><th>Teeth #</th><th>Reduction Ratio</th><th>RPM</th><th>Material</th></tr>
<tr><td>A</td><td>Engine</td><td>-</td><td>-</td><td>-</td><td rowspan="3">21945</td><td>VASCOX2M</td></tr>
<tr><td>B</td><td>Pinion Shaft</td><td>-</td><td>-</td><td>1</td><td>VASCOX2M</td></tr>
<tr><td rowspan="2">C</td><td rowspan="2">Pinion Shaft</td><td>Pinion</td><td>16</td><td rowspan="2">0.364</td><td rowspan="2">VASCOX2M</td></tr>
<tr><td>Gear</td><td>44</td><td rowspan="2">7980</td></tr>
<tr><td rowspan="2">D</td><td rowspan="2">Double Helical Spur</td><td>Pinion</td><td>19</td><td rowspan="2">0.322</td><td rowspan="2">VASCOX2M</td></tr>
<tr><td>Gear</td><td>59</td><td rowspan="2">2570</td></tr>
<tr><td rowspan="2">E</td><td rowspan="2">Spiral Bevel</td><td>Pinion</td><td>20</td><td rowspan="2">0.274</td><td rowspan="2">VASCOX2M</td></tr>
<tr><td>Gear</td><td>73</td><td rowspan="2">704</td></tr>
<tr><td rowspan="3">F</td><td rowspan="3">Planetary 1 (Normal)</td><td>Sun</td><td>46</td><td rowspan="3">0.338</td><td>AISI9310</td></tr>
<tr><td>Planet</td><td>136</td><td>238</td><td>VASCOX2M</td></tr>
<tr><td>Ring</td><td>90</td><td>0</td><td>VASCOX2M</td></tr>
<tr><td rowspan="3">G</td><td rowspan="3">Planetary 2 (Slowed down)</td><td>Sun</td><td>43</td><td rowspan="3">0.314</td><td>704</td><td>AISI9310</td></tr>
<tr><td>Planet</td><td>137</td><td>220</td><td>VASCOX2M</td></tr>
<tr><td>Ring</td><td>94</td><td>0</td><td>VASCOX2M</td></tr>
<tr><td rowspan="2">H</td><td rowspan="2">Bevel (Generator)</td><td>Pinion</td><td>73</td><td rowspan="2">4.563</td><td>704</td><td rowspan="2">VASCOX2M</td></tr>
<tr><td>Gear</td><td>16</td><td>3200</td></tr>
</table>

As described thoroughly, a series-partial-hybrid propulsion system with variable rotor speed transmission architecture was designed to meet and exceed all the RFP requirements in terms of safety, performance, and reliability. The inboard twin turboshaft engines based on GE-T700 rubberized engine was used to operate the main rotor and generate electrical energy for the propellers. The battery onboard provides deficient power supply mainly for the wing-mounted propellers. In an emergency, it can also be used to operate the generator to provide power for the main rotor in the uttermost failure. The novel design of the transmission of allowing variable speed without affecting the generators enables constant power supply while keeping the weight down and transmission complexity low.

Airframe Design

The *CRANE* was designed specifically for the extreme altitude mountain rescue vehicle. Many of the characteristics and components were also designed to make highly reconfigurable for multi-mission capability. As part of the specific requirement set forth by the RFP, fuselage must have an internal cabin volume for a pilot, two crews, and two litters. The arrangement of the cabin layout was explored prior to determining the Outer Mold Line (OML) of the fuselage. At the beginning of the QFD analysis, a non-pressurized cabin for weight savings was decided. In order to provide minimum center of gravity (CG) change during rescue and minimize the degradation of the aerodynamic performance of the vehicle, an internal hoist system was chosen. This decision was in conjunction with the safety of the operation where the external hoist system and the doors situated close to the propeller on the wing tip was inexpedient. Furthermore, for ease of unloading the stretcher/litter out of the vehicle, the aft-loading ramp was desired to be on the vehicle.

Internal Layout

The dimension of litter and sitting arrangement based on actual products were identified prior to the design of fuselage OML. The standard litter kits available from the United Rotorcraft that received FAA approval for various rotorcraft were used. Since internal hoist system was selected, the fuselage must also provide sufficient internal openings to access the litter at an angle during the rescue. Adopting these requirements, design of the OML and the structure of the fuselage were carefully constructed. The cockpit of the *CRANE* is designed to fit two pilots with avionics satisfying minimum requirement for single pilot operations.

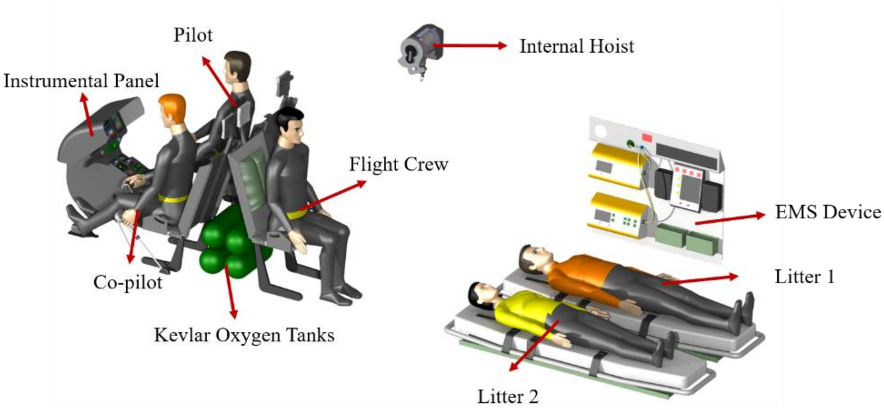


Figure 25 Internal Layout Design

With the decision of the unpressurized cabin, the vehicle is equipped with oxygen tanks made out of Kevlar fiber [18] for each flight personnel. Pressurizing the cabin adds significant weight. It was more desirable to simply supply the patient and flight personnel with supplemental oxygen than pressurizing the cabin. As specified in Part 25-20, exposure to cabin altitudes in excess of 25,000ft without supplemental oxygen could cause permanent physiological damage. Each flight personnel and PAX are supplied with individual oxygen masks capable of providing sufficient supply of oxygen at extreme altitude. It must be noted that all flight crew members receive altitude awareness and hypoxia training as prescribed in 14 CFR § 61.31.

Internal Hoist System

A 300kg rated hoist system was mandated in the RFP. However, upon research, only the 600lb or 273.16kg lifting load hoist system was commercially available. The design team selected GOODRICH's Pegasus rescue hoist system for its superior performance, reliability as well as low cost of ownership. The hoist system is positioned in the fuselage, close to the CG of the rotorcraft. The Pegasus hoist system has a maximum weight of 48kg with a cable length of 91m (300 ft). It is operated by the power generated from the generator during the rescue of stranded mountaineers. In the event of an emergency, the main pilot can overwrite the input of the crew and the pilot-initiated spring-loaded cable cutter can be employed to allow an emergency release of the payloads.

Figure 26 Kevlar O_2 Tank [left][18] Pegasus Hoist [right][13]

Fuselage Design

Adopting the RFP requirements, the design team constructed the OML using the CATIA, based on the selected cabin layout shown in Figure 25. *CRANE*'s airframe is a semi-monocoque made out of aluminum-lithium frames. The main structure of the airframe consists of 5 main bulkheads and frames, keel beams, longeron and stringer, which are carefully located considering the internal layout, door openings, and primary load-paths. The overall height of the fuselage structure is 7.4 ft tall, and the length was initially proposed to be 30.4 feet. However, a detailed analysis of the rotor wake showed that the empennages are directly in the rotor wake, which would cause a detrimental effect on the stability and the performance of the rotorcraft. Therefore, an adjustment was adopted considering the rotor wake, lengthening the fuselage tail by 3.5 feet. With a width of 5.9 feet, it provides sufficient room for two litters side-by-side and a spacious room for the crew to perform medical procedures. Overall specifications are summarized in Table 11.

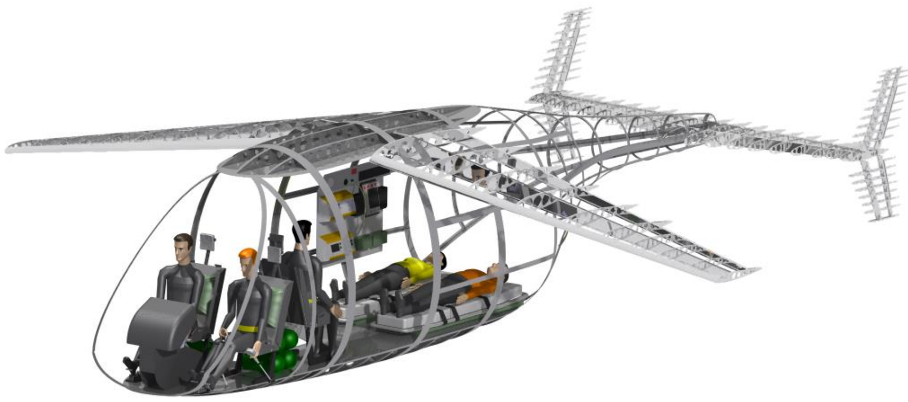


Figure 27 Airframe Structural Design

Wing/ Empennage Design

Since SAR vehicle are required to secure sufficient internal volume as discussed, a high-wing with 9˚ anhedral was designed to eliminate the risk of the rotor blade striking the wing-mounted 4.28 ft-radius propellers. The wing on the *CRANE* provides a portion of lift required and control inputs in airplane mode. For the given mission, the wing has to embrace all the flight conditions defined in the RFP. The primary load paths of the wing are the lift and propulsive force produced by the propellers. These wing loads are directly transferred to the main fuselage bulkheads and keel beams. The main forward and rear spar of the wing transmit wing torsion as well as the bending loads which are attached to the 3^{rd} and 4^{th} bulkheads.

The HOQ also identified that the required sizing of the empennage of a rotorcraft was a crucial factor having a significant impact on the performance and stability of the rotorcraft. Thorough research has identified that some rotorcraft such as the UH-60 has a cambered airfoil to provide a portion of yaw during forward flight. The cambered airfoil of the vertical tail could also prevent the uncontrolled spinning of the aircraft with the weathercock stability induced from the vertical tail. Therefore, careful sizing of the empennage was required to obtain adequate handling qualities and static/dynamic stability of the rotorcraft. To this end, a careful selection and sizing of the empennage were initiated. Several tail configurations were considered during the initial design phase. Initially, a conventional tail design was selected for its lightweight and simplicity. However, to meet the requirement of weathercock stability, a significantly large vertical tail was sized. As discussed earlier, *CRANE*'s empennage is located within the rotor radius but outside the influence of the rotor wake during hover. This implies that significantly large vertical tail increases the risks of the rotor striking the vertical tail. Therefore, H-tail was selected over other configurations since it provides a larger surface area for a lower span, decreasing the possibility of a tail-strike. Literature studies also showed that H-tail configuration minimizes the wake impact. This tradeoff studies resulted in the selection of the H-tail configuration.

The sizing of the empennage was conducted based on Raymer tail volume coefficient method [20] satisfying static margin of the aircraft. Prior studies on the fuselage have determined the overall length and crude location of the major components. In addition, the vertical tail was given a specific requirement of counteracting the torque produced by the main rotor during the cruise. A Clark Y airfoil was chosen for the vertical tail due to its high L/D ratio with a fixed attachment angle of 5° while NACA2412 for the horizontal tail. The outcome of the sizing was that a total vertical tail span of 4.14ft and a horizontal tail span of 12 ft were required for *CRANE*. The vertical tail was lowered to reduce the risk of a tail strike and sweep for aesthetics. A detailed assessment of the stability analysis of the rotorcraft is described in the APPENDIX. Table 8 summarizes the sizing of the wing and empennage.

Table 8 Empennage Design Overview

	Area	Span	Chord	Airfoil
Main Wing	190.86ft^2	29.65ft	6.44ft	NACA23012
Vertical Stabilizer	36.11ft^2	4.14 ft	2.18ft	Clark Y
Horizontal Stabilizer	68.57ft^2	12.00ft	5.71ft	NACA2412

Landing Gear

During the QFD analysis, it was recognized that skid-type landing gear was essential for the rescue vehicle. The SAR vehicle covers various mission including an uneven ground landing and slope landing.

A novel design of retractable landing skid based on Eurocopter patent [21] with fixed cross-tubes is adopted on the *CRANE*. On each side of the vehicle, two rollers are fastened to the keel beams to carry loads. These rollers, together with electric motors create a mechanism to retract the landing gear. Fully-retractable landing skid was initially proposed. A lightweight settling protector is also attached to the landing skid to increase surface area. The settling protectors help to guard against sinking into the snow and soft ground. Anti-slip crampons were excluded for the helicopter skid to enable advanced flight maneuvers such as running takeoff and landing.

The landing gear has a 5.03m (16.5ft) long track which extends down to a maximum height of 0.87m (2.86ft) to provide sufficient ground clearance for the wing-mounted propellers. The 14 CFR 23.925 mandates that at least seven inches between the propeller and the ground be secured. *CRANE*'s propeller adopted with high retractable landing skid has secured 0.76m (2.5ft) of ground clearance when the landing gear is fully deployed. To assist with entry and exit to the cabin, cabin/crew step was incorporated in addition to landing skid structural support for toe-in hover.

Ice Protection System Design

The QFD analysis identified icing to be vital parameters for designing extreme altitude mountain rescue vehicle. Aircraft icing is a phenomenon in which a super-cooled droplet created in a cold and humid environment strikes the surface of an aircraft and then freezes. It is commonly known that icing occurs mainly in a cold humid environment, but it can occur on an aircraft even when an aircraft passes through a cloud layer of sub-zero altitude in summer. Freezing is a meteorological phenomenon that can occur anywhere on the structure exposed to the outside, which is a serious threat to flight safety as well as the performance of aircraft. The fuselage and external components of an aircraft exposed to freezing cause problems such as malfunction, performance degradation, and severe vibration posing a serious threat to the safety of aircraft operations. Due to the nature of the mountainous climatic conditions, unpredictable and extreme weather conditions at the summit of Everest place the *CRANE* in the inevitable freezing conditions. Therefore, it is necessary to take all necessary precautions by installing anti/de-icing devices to prevent malfunction or performance deterioration caused by freezing on the fuselage or external components. Moreover, the purposes of this analysis were to apply the requirement, adopt the Ice Protection System (IPS) for every component exposed to freezing and obtain certification in the icing environment.

Icing Analysis

During forward flight, the rotor is unloaded by the wing enabling the *CRANE* to cruise at 185 knots. High cruise speed of the *CRANE* leads to fuselage freezing phenomenon similar to the fixed wing aircraft. During hover, downwash from the main rotor generally causes freezing phenomenon to occur at the top surface of the fuselage. This analysis was vital for the winged helicopter that has an additional wing surface for the icing to occur.

During forward flight, the pilot has the authority to avoid possible weather conditions. However, rescue hover mission at the 8,870m (29,100ft) may not give alternatives for the pilot to avoid, which led to the detailed numerical analysis of icing was carried out for the design of anti/de-icing devices on the *CRANE*.

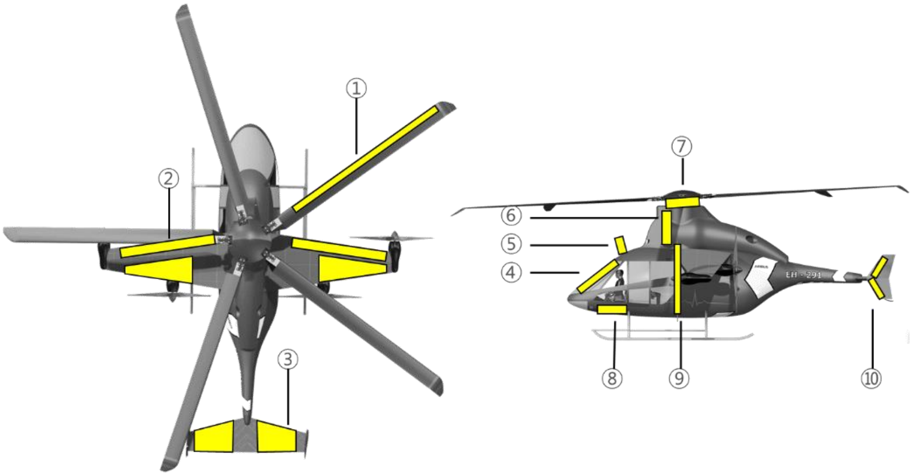


Figure 28 Ice Protection System (IPS) Location

Table 9 IPS Types and Location

Position ID	Location	Anti/de	Type	Description
1	Rotor Blade	De	Electro-thermal	
2	Main Wing	Anti	Air-bleed	Forward Flight
	Control Surface	De	Electro-thermal	
3	Horizontal Tail	Anti	Electro-thermal	Forward Flight
4	Wind Shield	Anti	Electro-thermal	
5	Wire Cutter	Anti	Electro-thermal	
6	Engine Intake	Anti	Air-bleed	
7	Rotor Mast	Anti	Electro-thermal	Forward Flight
8	Pitot Tube & Sensors	Anti	Electro-thermal	
9	Propeller	De	Electro-thermal	
10	Vertical Tail	Anti	Electro-thermal	Forward Flight

Weight Estimates

The weight estimates of the final design presented in Table 10 are based on the sizing code. These estimates are performed by using the most critical leg 2 mission as explained. The longitudinal CG is referred from the nose to tail, and the vertical location referenced from the bottom of the fuselage. The positioning of each component is based on the analysis described in various sections. The values in Figure 29 are for the leg 2 mission with fully loaded fuel with 2 PAX onboard.

Center of Gravity Analysis

Table 10 shows the CG shift of the entire aircraft. The longitudinal and vertical CG are located 3.6m (11.8ft) from the nose, and 1.89m (6.21 ft) from the bottom of fuselage respectively. As described, the *CRANE* is designed to maintain minimal CG location shifts as much as possible during the rescue. The longitudinal position of CG is iteratively adjusted to balance and place CG nominally under the main rotor. This also led to an aft positioning of the fuel tanks to balance and minimize shifting of CG due to the additional PAX loaded at the peak of Everest. This is due to the fact that substantiate the amount of fuel is burnt when the additional payloads are loaded, subsequently shifting the CG fore while payloads are loaded at the aft of the aircraft. The longitudinal shifts of CG were also ensured that it lies fore of aerodynamic center of the wing. These changes do not substantially affect the dynamics and stability of the aircraft throughout the SAR mission.

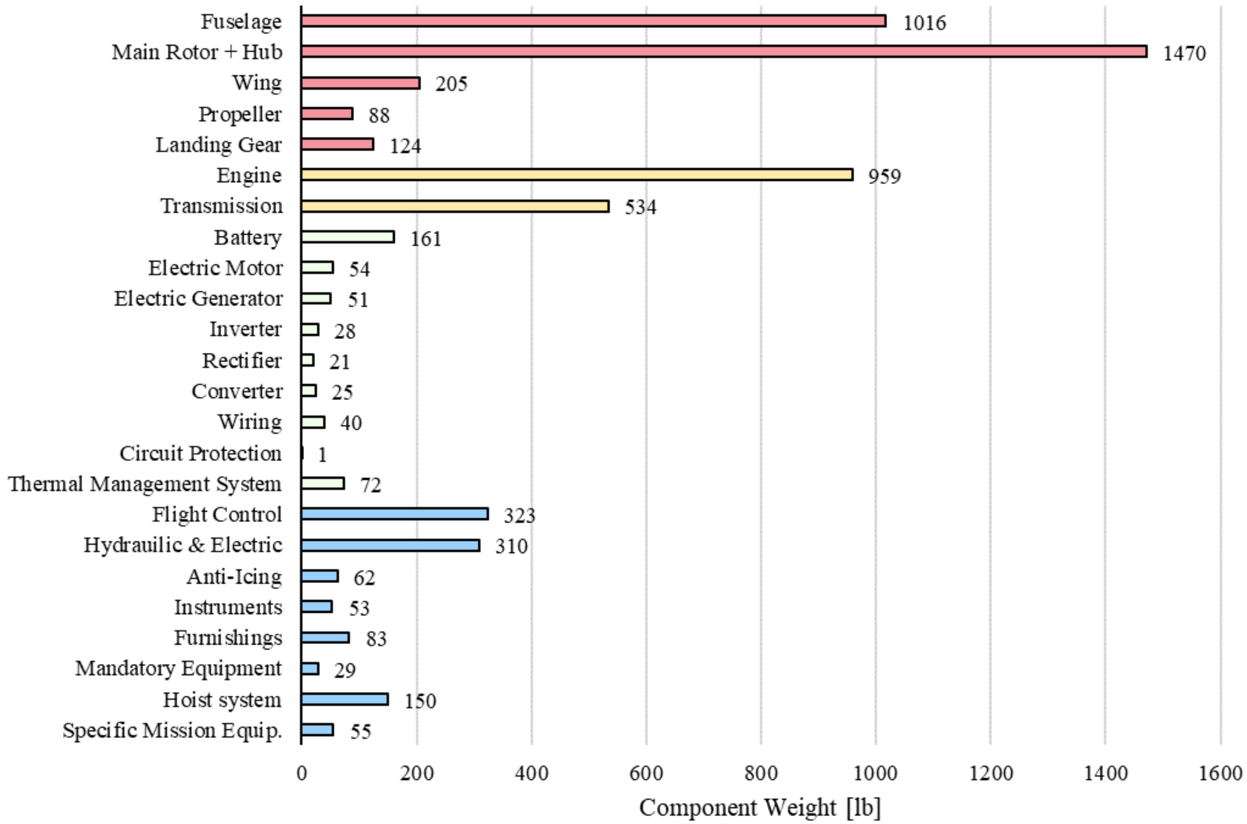


Figure 29 Weight Breakdown Overview

Table 10 Component Weight

Component Description	Weight		% Empty Weight	$X_{c.g}$ (m)	$Y_{c.g}$ (m)	$X_{c.g}$ (ft)	$Y_{c.g}$ (ft)
	kg	lb					
Fuselage	461.23	1016.85	13.10	3.66	1.26	12.01	4.13
Main Rotor + Hub	667.10	1470.706	18.94	3.49	3.68	11.47	12.08
Main Wing	93.43	205.986	2.65	3.65	2.04	11.99	6.69
Tail Horizontal Wing	45.86	101.101	1.30	9.20	1.29	30.18	4.22
Tail Vertical Wing	1.478	3.258	0.04	9.20	1.29	30.18	4.22
Engine	435.08	959.183	12.35	4.57	2.52	15.00	8.27
Transmission	242.30	534.18	6.88	3.53	2.55	11.60	8.38
Landing Gear	56.36	124.246	1.60	2.95	0.23	9.67	0.76
Flight Control	146.70	323.426	4.17	0.88	0.71	2.88	2.32
Hydraulic, Pneumatic Electrical Sys	140.54	309.841	3.99	1.98	0.46	6.51	1.49
Air Conditioning Anti-Icing System	28.18	62.129	0.80	2.38	2.18	7.81	7.16
Instruments	24.05	53.018	0.68	0.89	0.71	2.92	2.32
Furnishings and Equipment	37.62	82.931	1.07	1.31	0.49	4.31	1.62
Wing-mounted Propellers	40.01	88.214	1.14	4.08	1.43	13.37	4.68
Battery	72.95	160.835	2.07	4.60	0.24	15.08	0.80
Electric motor	24.30	53.562	0.69	4.08	1.43	13.37	4.68
Electric Generator	23.17	51.084	0.66	3.12	2.55	10.23	8.38
Inverter	12.67	27.936	0.36	3.16	2.45	10.36	8.03
Rectifier	10.08	22.23	0.29	4.08	1.43	13.37	4.68
Converter	11.42	25.194	0.32	4.08	1.43	13.37	4.68
Wiring	18.21	40.156	0.52	2.49	0.32	8.16	1.06
Circuit Protection	1.113	2.453	0.03	3.20	2.32	10.49	7.60
Thermal Management System	32.86	72.436	0.93	3.12	2.55	10.23	8.38
Mandatory Equipment	13.15	29	0.37	2.21	0.30	7.25	0.97
Hoist system	68.04	150	1.93	3.18	2.05	10.43	6.73
Specific Mission Equip.	24.95	55	0.71	2.22	0.31	7.29	1.02
Medical Equipment	150	330.693	4.25	4.60	1.14	15.08	3.74
EMPTY WEIGHT	2882.59	**6355.65**	**81.86**				
Crew x 1	85	187.393	2.41	2.53	1.10	8.29	3.61
PAX x 2	170	374.786	4.83	4.57	0.64	15.00	2.10
Fuel Tank	213.95	471.679	6.08	5.18	1.83	17.00	6.02
Pilot x 2	170	374.786	4.83	1.63	0.82	5.34	2.71
TOTAL WEIGHT	**3521.51**	**7766.15**	**100**	**3.60**	**1.89**	**11.80**	**6.21**

Final Design Overview

Through the design process of the *CRANE*, extensive sizing optimization using the Evolutionary Algorithm was adopted during the initial design process. Furthermore, a high-fidelity analysis was incorporated into the conceptual design process to obtain the best attainable sizing results. Through the optimization of the rotor blade planform, weight, and propulsion system, final design parameters are summarized in the tables below.

Table 11 Final Design Weight Specification

	Value
Empty Weight	2882.9kg | 6355.6lb
Gross Weight	3521.5kg | 7766.2lb
Fuel Weight (leg 1)	120.2kg | 265.1lb
Fuel Weight (leg 2)	214.1kg | 471.9lb
Fuel Weight (leg 3)	100.9kg | 222.5lb

Table 12 Final Design Engine Specification

	Value
Engine Type	GE T-700 Turboshaft
# of Engine	2
MCP per engine	917.7kW | 1230.6hp
IRP per engine	1087.8kW | 1458.7hp
CRP per engine	1220.5kW | 1636.7hp

Table 13 Final Design Rotating Element Specification

	Radius	**Chord**	**Airfoil**	**# of Blade**	**Twist (r/R)**
Main Rotor	24.898ft	6.44ft	NASA RC (4)-10	5	-12.36˚
Propeller	4.28ft	2.18ft	NACA0012	5	-39.1˚

Table 14 Final Design Empennage Specification

	Span	Chord	Airfoil
Main Wing	9.04m | 29.65ft	1.96m | 6.44ft	NACA23012
Vertical Stabilizer	1.26m | 4.14ft	0.66m | 2.18ft	Clark Y
Horizontal Stabilizer	3.66m | 12.0ft	1.74m | 5.71ft	NACA2412

Table 15 Final Design other Specification

Parameter	Value
Fuselage Capacity	3 crew + 2 Litter
Rotor Solidity	0.099472
Rotor AR	16
Shaft Tilt Angle	2˚
Wing Attachment angle	10.25˚
Main rotor RPM	241 Normal Operating 220 slowed down
Propeller RPM	1500
# of Generator	2
# of Motor	12
# of Battery Packs	6

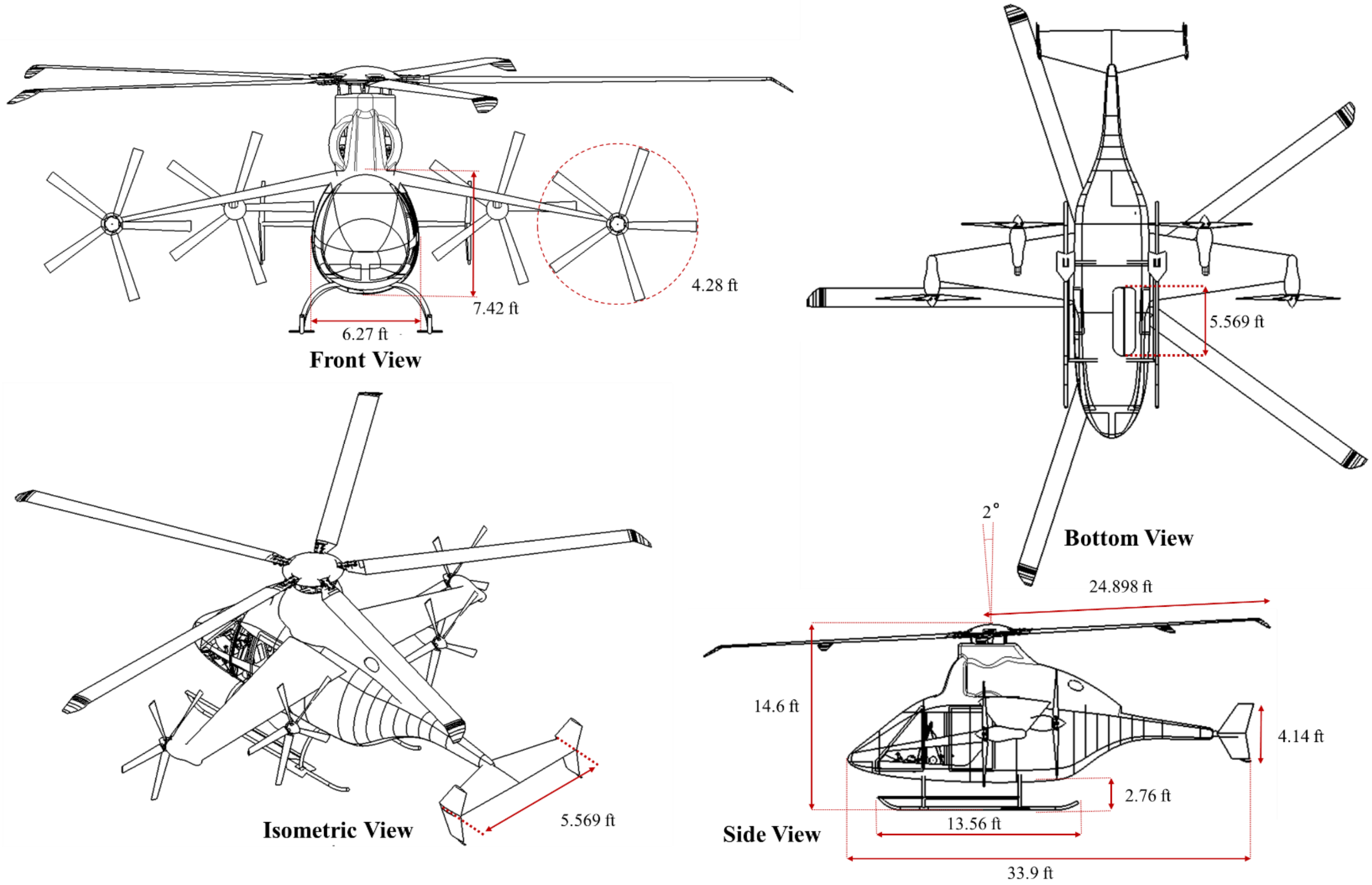


Figure 30 Final Design Overview

V. Concluding Remarks

The conceptual design methodology of rotorcraft is discussed incorporating systematical QFD process, multi-disciplinary design optimization, and high-fidelity analysis. The main purpose of the thesis was to provide an efficient and effective design methodology instead of the traditional trial and error method and engineering intuition in the conceptual design phase. Various mathematical tools are also incorporated to obtain mission objectives and other attributes relating to the performance of the rotorcraft. The proposed design methodology has been applied to design rotorcraft capable of performing mountain rescue at the highest peak of the planet. Performance and mission objectives are prioritized through the QFD process with the customers' attributes obtained mainly from the RFP and interviewing various groups.

The House of Quality (HOQ) is adopted as part of the QFD process to transform user demands into design quality and prioritize array of design characteristics that impact customer attributes. The proposed design framework also adopts parallel-design approach with the inclusion of higher fidelity analysis in the conceptual design phase. This framework considers various technical aspects including the aerodynamic, structure, propulsion, transmission design, weight and balance, stability and control, noise analysis, and economic analysis.

As a system integration of the QFD process, this thesis has also outlined the proposed conceptual design framework to design a high-altitude mountain rescue vehicle. By employing morphological alternative matrix, and carrying out HOQ analysis, four wing-mounted propellers winged-helicopter configuration with hybridized propulsion system was designed. This result was obtained in conjunction with the HOQ analysis for which, critical design variables identified, were specifically studied to obtain the optimal solution. By adopting the design process and detailed numerical analysis, the winged helicopter that can hover at 29,100 ft and attain a cruise speed of 185 knots was designed.

The proposed design framework aims to provide a central collaborative repository to design aerospace vehicles and provide essential information to initiate preliminary design by the integration of the QFD process. In essence, the adoption of the proposed design methodology attempts to provide a means to systematically design rotorcraft while maximizing the resources in obtaining valuable information.

APPENDIX

This section provides various information with regards to the implementation of the proposed design framework. Information available in the appendix are provided by the design team consisting of Hyeongseok Kim, Dounguk Lee, Daejin Lim, Dawoon Lee, Taekeun Yoon, Yoonpyo Hong, and Soomin Jeong.

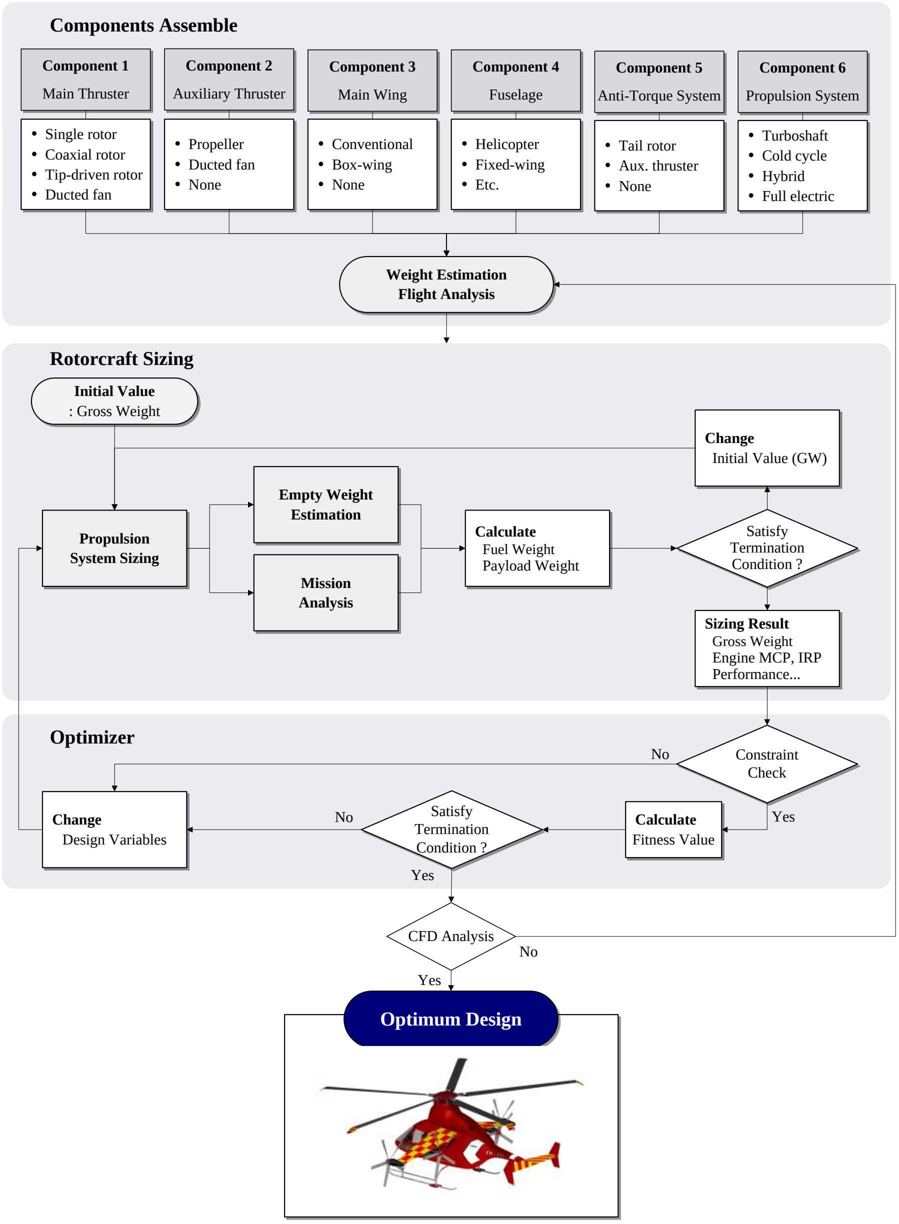
Sizing Algorithm

Winged helicopter design framework was developed based on the preliminary design methodology study [9]. The schematic of the algorithm in the sizing code shown below performs an iterative process to provide sizing result that meets the required payload and the specified mission.

Design optimization using the Evolutionary Algorithm (EA) was coupled with the sizing code to manipulate design variables to meet the design objectives. Since the QFD analysis showed that takeoff gross weight parameter requires the most focus, the optimization design objective was set to minimize gross weight of the vehicle that carries out the mission profiles stated in the RFP. Through 1) ~ 7) iterative procedures, initial sizing for a winged helicopter was conducted based on the identified mission requirements:

1. Design user inputs (variables and design parameters) are used to calculate basic design geometries of the winged helicopter (disk area, solidity, etc.).
2. Using the gross weight initial value and the lift sharing factor, wing sizing is sized to provide sufficient lift during the mission specified. Then, the wing position satisfying the static margin of the design parameter is determined.
3. Engine sizing is carried out based on the rubber engine methodology introduced in the SSP program [9].
4. Empty weight is estimated based on the weight estimation formula.
5. Fuel weight required to carry out the mission is calculated within the mission analysis module.

6. Using the calculated empty weight and the fuel weight, the payload is obtained. An iterative calculation is performed, correcting the TOGW until the calculated empty weight is within 3% error attaining the targeted payload.
7. Until the termination condition is met, design variables are manipulated to obtain an optimized result.



Initial Sizing Code (RISPECT-C) Schematic

Common elements of output results of the sizing code were component weight, empty weight, engine power, and fuel weight at each mission segments. Since the QFD analysis identified rotor related parameters and engine related parameters to be two most important requirements, design process iteration that relates to the rotor and engines will be explicitly shown in the next section for better vehicle sizing estimates.

Optimization

The Evolutionary Algorithm (EA) that was used throughout the design process is an adaptive search technique which enables quick computation of approximate solutions and an appropriate fitness function. Since EA works with a distribution of solutions to obtain the most desirable solution that meets the design objective(s), such characteristics make it an ideal optimization algorithm to solve multi-objective optimization problems. A total of 30,000 iterations were carried out to obtain less than 25% feasible design solution due to the inherent difficulty of the given mission. The final optimum design solution is selected through a rigorous CFD iteration process.

Rotor Blade Aerodynamics

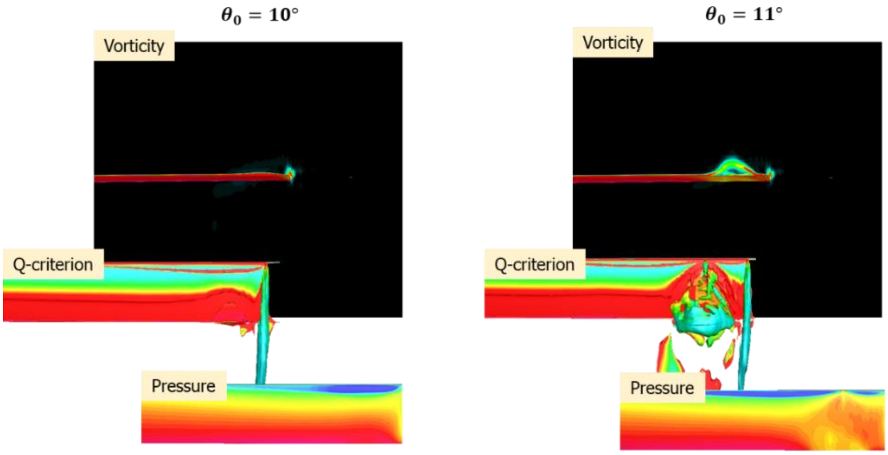
The rotor aerodynamic analysis of RISPECT-C is based on the Blade Element Theory (BET) and Blade Element Momentum Theory (BEMT). During mission analysis, the atmospheric condition of ISA +20 was assumed for the sizing process. This comprehensive in-house code has been validated extensively and was used to compare with the CFD results and Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics (CAMRAD II) trimmed results. Although RISPECT-C was validated extensively, unpredictable physical effects that may occur in the extreme atmosphere were captured by coupling the Hybrid CFD code to attain the most achievable, realistic and safe design solution.

The design of the blade requires iterative calculation and coupling the CFD analysis with the sizing tool would demand excessive computational resources. Furthermore, the flow field near the rotor is extremely sensitive to the tip shape, necessitating significant accuracy of CFD solver. To meet these two requirements demanding for a relatively low computational power while providing high accuracy, in-house hybrid CFD tool was used for numerical analysis of the main rotor blade and blade planform design in Section 0. The in-house CFD tool used is a well-validated CFD solver based on Reynolds-Averaged Navier-Stokes equation with Splalart-Allmaras Detached Eddy Simulation as a turbulence model. Table below shows free wake model variables used, which is based on the Vatistas vortex core model and Bhagwat-Leishman core growth model.

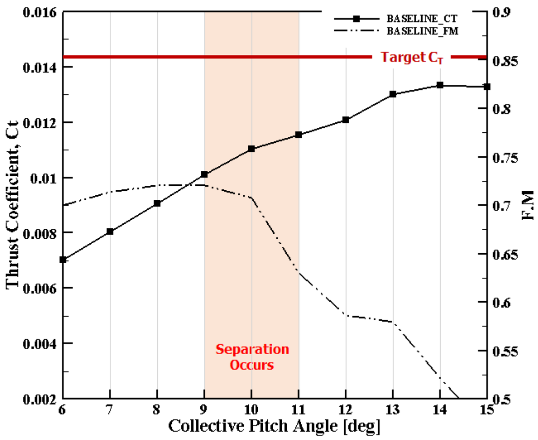
Free Wake Model Variables

Parameters	**Value**
Artificial Viscous Parameter	1.2×10^{-4}
Initial Vortex Core Size	0.15c (c=chord length)
Wake Update Frequency	every 6°
Number of Inboard Wake Trailers	10
Number of Wake Revolutions	Hovering: 15 / Forward Flight: 4

Baseline sizing results from the RISPECT-C were then compared with the CFD results. Mission analysis has shown that hovering at 29,100ft was the most critical and challenging mission that is the most hazardous for the flight personnel. A total of 3 baseline sizing was carried out with the feedback adopted from the CFD analysis. To begin with, with the main rotor radius of 4.73m (15.52ft), CFD result showed that the rotor blade stall is inevitable when the collective pitch is increased to meet target thrust during hover. This is clearly shown in Figure below where the shock-induced separation between $\theta_0 = 10^{\circ}$ and 11° occurs, not being able to meet the mission requirement of the RFP. Other attempts such as increasing the tip speed to bring down the target C_T during hover were initially attempted, which were later concluded that increasing the tip speed during hover would lower the target C_T but would simultaneously cause the rotor blade stall at a lower angle. Such effect was deemed unwanted especially for the rotor blade that undergoes dynamic stalls, and having low stall angle would be detrimental for SAR vehicle at extreme altitude. Through the CFD analysis of the rotor blade, a resizing of the vehicle was necessitated to obtain sufficient stall margin for safety.



Baseline Rotor Blade Flowfield

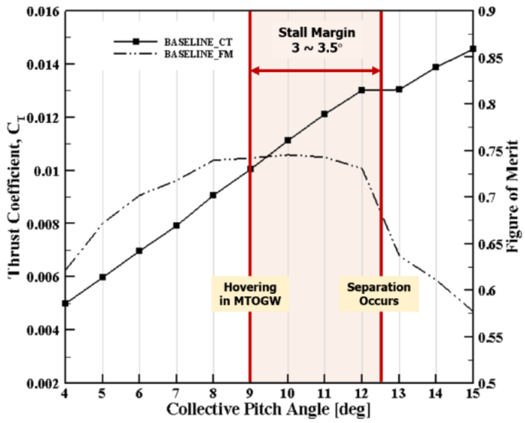


Performance Analysis Results of the Baseline Blade

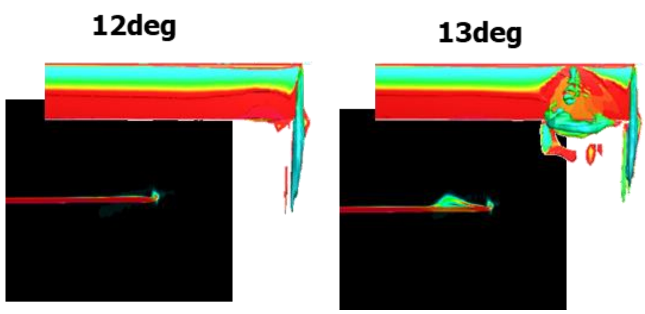
After undergoing several rigorous iterations of going back-and-forth, the second baseline sizing of the rotor was completed. Previous attempt to secure sufficient stall margin has eventually led to increasing the radius of the rotor blade thereby reducing the disk loading for hover capability at 29,100ft. Adopting the specification, CFD analysis of a rotor blade with NACA0012 as a baseline airfoil was carried out. Required C_T at Maximum Takeoff and Gross Weight (MTOGW) for the new baseline was 0.0101. The result of CFD analysis of the main rotor showed that sufficient stall margin was secured while operating at MTOGW.

Baseline Prototypes Rotor Specifications

	Initial Baseline	Second Baseline
Radius	4.73m | 15.52ft	7.59m | 24.898ft
# of Blade	5	5
AR	15.8	16
Twist	-12	-12.36
Target C_T	0.0143	0.0101
$Re\#_{Tip}$	2,486,000	2,903,000
M_{Tip}	0.65	0.588
Airfoil	NACA0012	NACA0012
Planform Shape	Rectangular	Rectangular



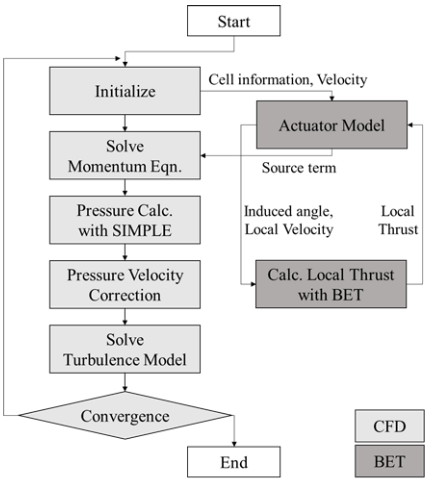
Performance Analysis Results of the Second Baseline Blade



Flowfield of Second Baseline Blade before and after stall

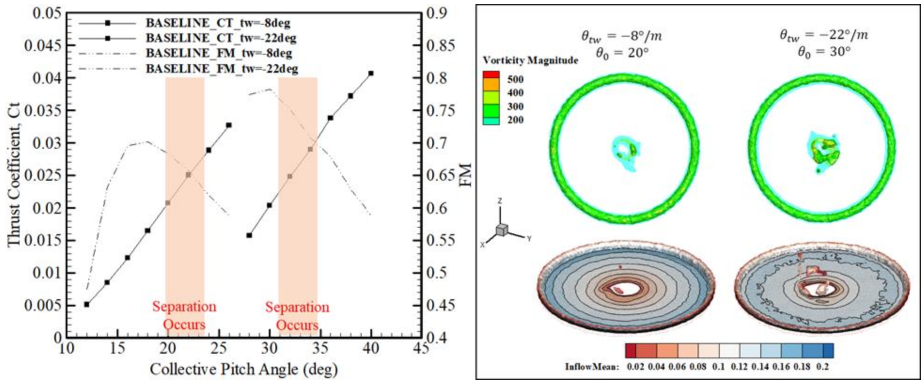
Wing-Mounted Propeller Blade Operability

The third baseline sizing was initiated by the anti-torque requirement at 8,870m (29,100ft). The performance of the wing-mounted propellers is also directly related to the safety of the operation. As described earlier, wing-mounted propellers are in charge of counteracting the main rotor torque. The requirement set forth by the RFP, the anti-torque system must be capable of maintaining heading in hover with wind from any azimuth up to 74km/h (40knots) at 8,870m (29,100ft). It was vital that the sizing of the propeller should provide sufficient anti-torque and forward thrust to successfully carry out the extreme altitude mountain rescue mission. Due to the difficulty of capturing physical effects of rotor blade at extreme altitude using the sizing code, Actuator Disk Model (ADM) analysis [12] was carried out for the propeller.



Actuator Disk Model Flowchart

Based on the specification from the sizing code, the performance of the propeller was computed at a varying pitch. This analysis showed that dual wing-mounted propellers configuration was insufficient to counteract the main rotor torque during hover. Increasing the radius rotor could be part of the solution to generate sufficient thrust. However, due to the rotor blade-propeller and ground clearance constraining the size of the propeller, a novel design of the distributed propulsion system for the wing-mounted propeller was proposed for the final design.



Initial Baseline Propeller Performance Results

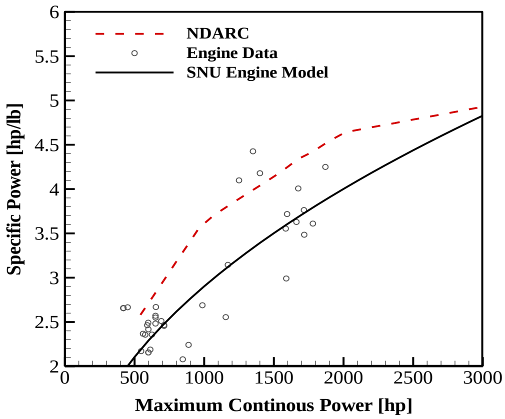
For effective integration of the wing-mounted propellers, the design team proposed a concept that two propellers located at the wing tip while two other propellers are mounted on the wing location where aerodynamic interference by the wing-tip-mounted propeller could be minimized. This led to a winged helicopter design that has four wing-mounted propellers providing the necessary thrust to meet the RFP requirements. Moreover, this smaller-sized propeller system not only secured the propeller clearances but also enhanced the safety of the vehicle. With a minor adjustment to the comprehensive sizing code and substituting the main rotor variables, a final baseline anti-torque system from the initial sizing was obtained as shown in the table below.

Design Specifications of Propeller

	Initial Baseline	Final Baseline
# of propeller	2	4
# of Blades	8	5
Planform	Rectangular	Rectangular
Radius	4.95ft	4.28ft
Airfoil	NACA0012	NACA0012
RPM	1250	1500

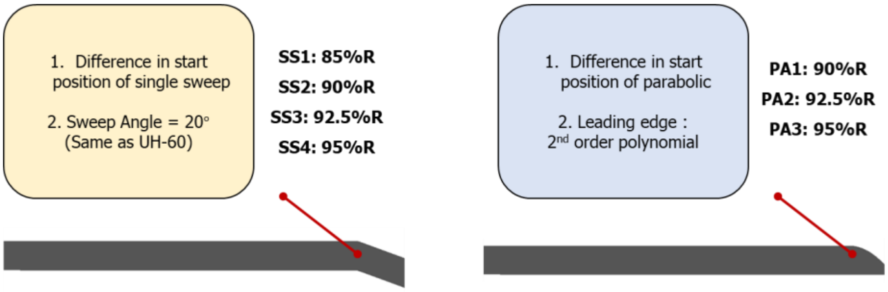
Propulsion System Sizing

Since the RFP mandated that existing engine technology must be assumed for all performance estimations, the "rubber" engine performance calculation was conducted based on the GE-T700 series engine with consideration of the lapse rate. Further refinement was made for the weight of the turboshaft engine since rubber engine module was used throughout the design process. A correlation of the engine data available from the European Union Aviation Safety Agency (EASA) was adopted. Comparing the engine weight correlation to the NDARC showed that SNU engine model provides a conservative analysis of the "rubber" engine model.

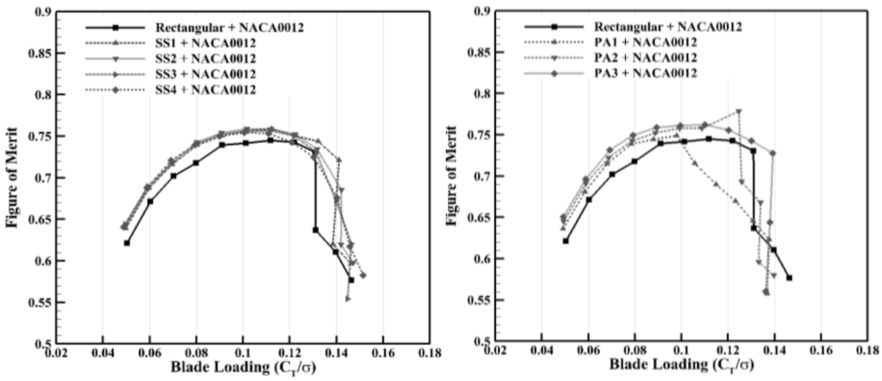


SNU's Engine Model

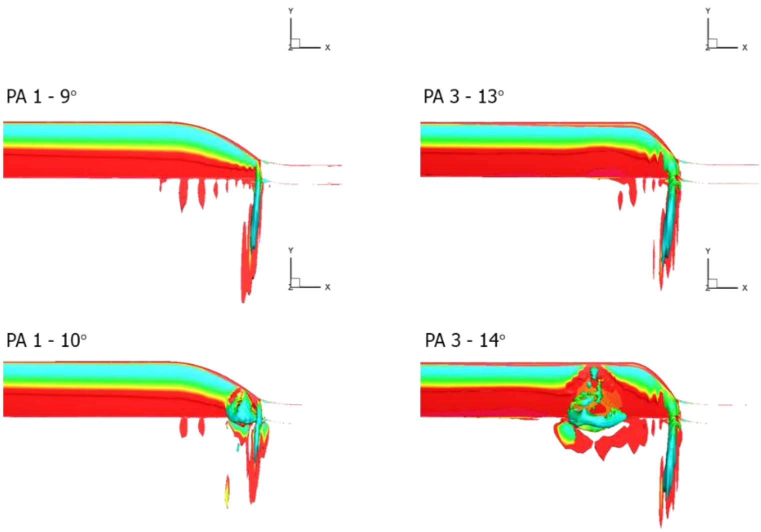
Four main tip shapes were selected for initial tradeoff study: single sweep (UH-60), Parabolic (ONERA7AD), BERP, and Blue Edge planform shape. These blade shapes are known for their distinctive characteristics and superior performance. The single sweep and parabolic shapes are relatively simple and structurally robust that can greatly reduce wave drag. On the other hand, the double sweep designs like the BERP and Blue Edge design have an aerodynamically superior design with a higher range of stall margins. In addition, the noise performance of the rotor was enhanced by generating a counter-vortex near the notch. However, as a SAR rescue vehicle for which structurally robust design in addition to the simple design requirement of the RFP is prescribed for safe operation especially at extreme altitude, single sweep or parabolic shape were chosen for detailed parametric study of *CRANE*'s rotor planform design.



Parametric study of the blade shapes

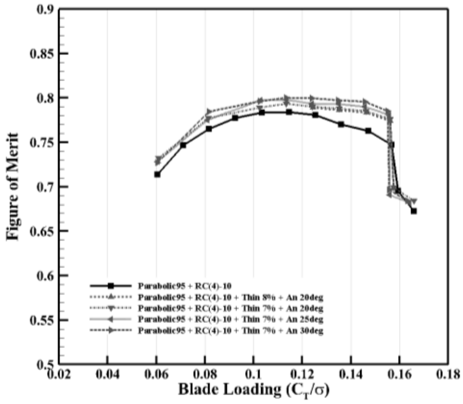


Performance Comparison of Single Sweep and Parabolic Planform Designs



Main Rotor Hovering Flowfield

Ultimately, comparing the aforementioned parameters have led to the final rotor blade configuration with NASA RC(4)-10 airfoil, parabolic planform shape, 30˚ anhedral angle, and modified NASA RC(4)-10 airfoil with thickness 7% which was shown to be superior design for high altitude hover.



Final Design Rotor Hover Performance

Forward Flight

Based on the parametric studies on the rotor blade design, a forward flight performance analysis was carried out to ensure that the *CRANE* is capable of not only hover at 8,870m (29,100ft) but also high-speed flight as identified from the breakdown of mission profile. Trimmed results obtained from CAMRAD II were used.

Flight Condition of Forward Flight

	100kts @ 29100ft	**185kts @ 12400ft**
$Re\#_{TIP}$	2.913 x 10^6	3.992 x 10^6
M_{TIP}	0.604	0.523
μ (advance ratio)	0.271	0.543
Target C_T	0.00947	0.0015
θ_0	8.44	3.27
θ_{1c}	0.483	-1.10
θ_{1s}	-6.96	-5.31
β_0	0.0	0.0
α_{shaft}	8.99	5.95

A comparison of forward flight performance of the baseline blade (rectangular NACA0012) and the final blade design is depicted in the table below. Despite the rotor blade being designed solely for the hover performance at extreme altitude, an appropriate design consideration that the design team has taken led to an increased forward performance of the final blade design.

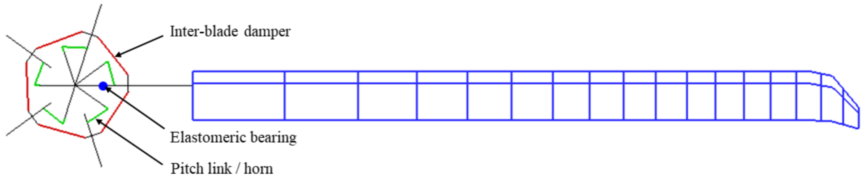
Performance (C_Q) of Forward Flight

C_Q (Torque Coefficient)	100kts	185kts
Baseline blade	0.0008686	0.000409
Final blade	0.0008131 (6.4% decrease)	0.0003947 (3.5% decrease)

Structural Dynamic Analysis of the Main Rotor

Comprehensive Analytical Model of Rotorcraft Aerodynamics and Dynamics II (CAMRAD II) was used throughout the structural design process. This comprehensive tool enables all the required analysis such as aerodynamics, structural dynamics, and vibration for the design of a rotor blade.

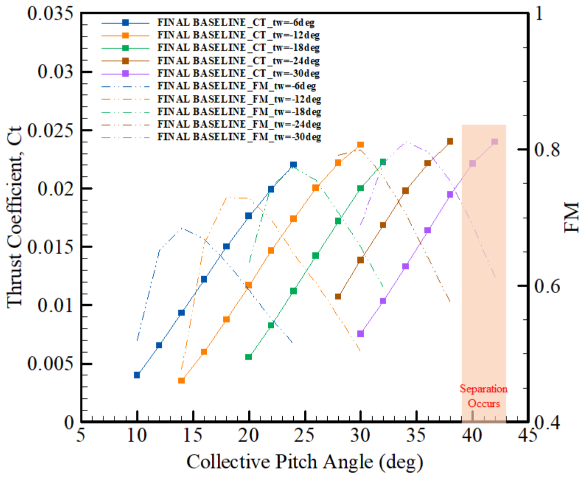
KSEC-2D, in-house code developed by Konkuk University, was used to perform finite element cross-sectional analysis on the non-homogeneous and non-uniform distribution of material properties. The weighted modulus approach is adopted to find the properties of the laminated composite material. The shear center is calculated using the Trefftz' definition, and the torsion constant is obtained using the St. Venant torsion theory. Throughout the design process, the CAMRAD II model for the inter-blade hub was used as shown.



CAMRAD II Hub and Rotor Modeling

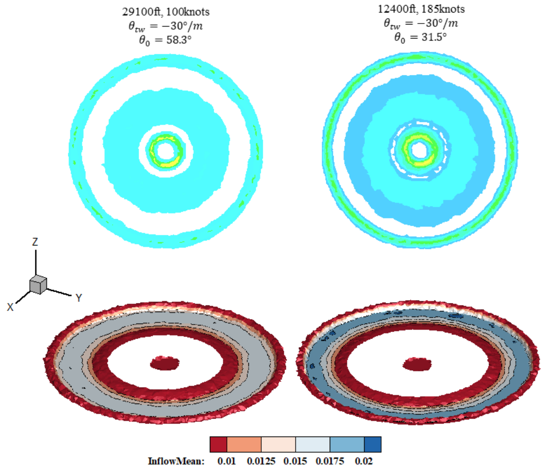
Propeller Design

The optimization of the propeller blade was carried out in conjunction with the design of the propeller hub. For the design of the propeller blade, the ADM method introduced. The radius of the propeller was obtained from the sizing code, and the parameters were used. The main variables for the parametric studies were the twist of the propeller. The tip sweep was found to have minimal effect on the performance. The initial ADM result has shown that significant consideration should be accounted for the twist angle of the propeller. This is mainly due to the high thrust that the propellers must generate during high-speed cruise flight and at extreme altitude, securing significant stall margin was important for the safe operation of the vehicle. The airfoil used was the NACA0012 airfoil mainly due to the reverse thrust that the propeller had to generate during hover and transition. Based on the hover design point, the final propeller for the *CRANE* had a twist of -39.1˚ acquiring 3˚ stall margin. During hover at the 8,870m (29,100ft), the required collective pitch angle (measured at the root) was 35.5˚.

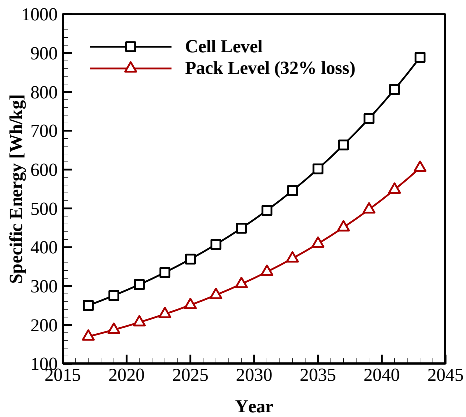


Final design propeller performance results

The propeller capability to provide thrust at extreme altitude must be met to successfully meet the requirements of the RFP to maintain heading at a given side wind. Furthermore, the anti-torque capability is directly related to the safety of the whole system which was the most important objective described in the QFD analysis. During forward flight, *CRANE*’s propellers were able to meet the cruising requirement. Upon reviewing the vorticity magnitude, the designed propeller could exhibit superior performance in both hover and high-speed flight.



Final design propeller vorticity contour



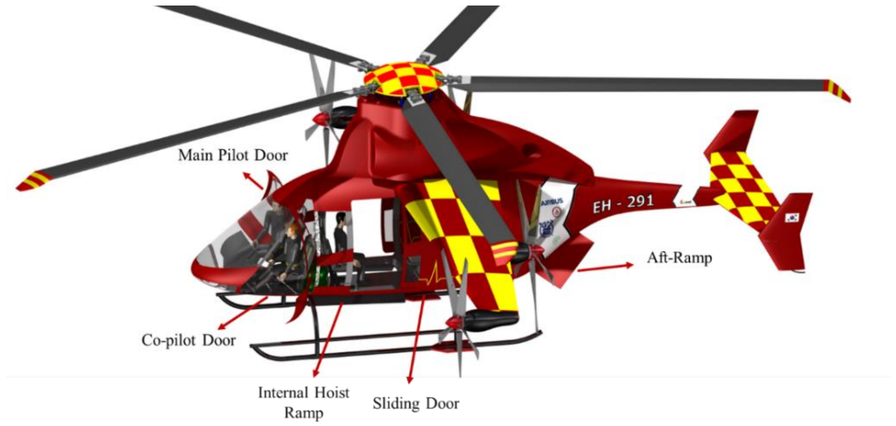
NASA Advanced Battery Technologies Expected Trend [16]

Battery System

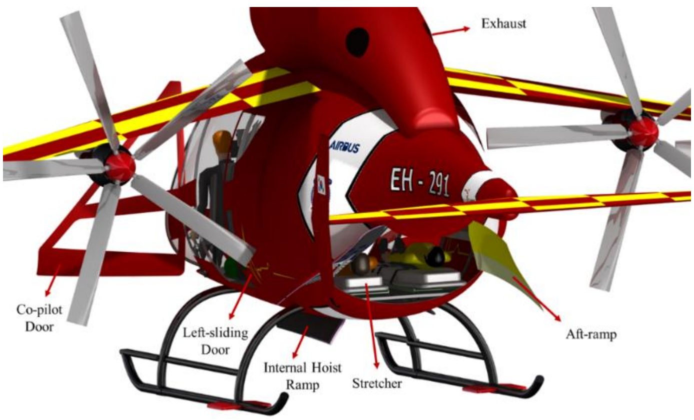
The battery pack used in the aerospace application. A typical battery pack contains more than 300 cells. Converting battery cell to pack level causes a significant decrease in specific energy. Such battery pack is essential as it contains various wirings and heat management system to prevent over-heating. The team initially planned on having no consideration on the thermal system for optimal sizing as the rotorcraft would operate in an extremely low-temperature environment. It was decided thermal systems will be onboard for conservative sizing considering the multi-mission capability. Based on 2019 technology level, battery sizing was carried out for final design. Since the main power supply comes from the turboshaft engine, the overall weight of the battery was only 72.94kg (160.8lb).

Accessibility

As a rescue vehicle, the unloading and loading of the litter similar to a typical ambulance were accommodated by aft-ramp. As depicted below, the *CRANE* has a total of 6 door opening for the crew to access; 2 x pilot doors, 2 x side sliding doors, aft-ramp, and hoist ramp.



Door Openings of the Final Design



Aft-Ramp

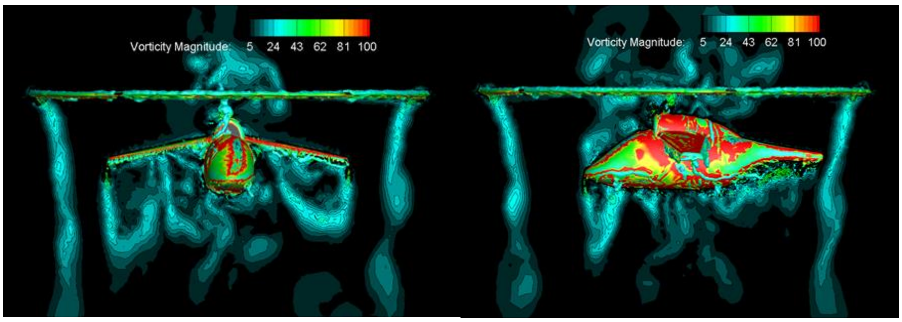
Icing Analysis

The icing analysis was performed using the Ice Shape Evaluation and Performance Analysis Code (ISEPAC) developed by Son et al. [23] in the following order: flow field analysis, droplet field analysis, and thermodynamic analysis. For the flow field analysis, Actuator Surface Method (ASM) was used. ASM is a method developed by Kim [24] to simulate a blade by assigning a source term to a virtual blade surface and generate tip vortex and wake of each blade. The droplet field analysis is based on the Eulerian droplet equation. Since the temperature assumed was a standard atmosphere of -28˚C, it can be assumed that all the attached droplets freeze. Therefore, the thermodynamic interpretation is omitted. Detailed methodology and verification can be found in [23].

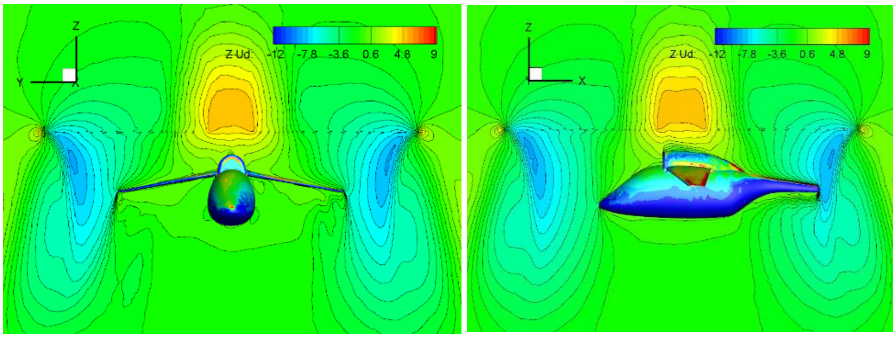
The analytical conditions used were standard atmospheric conditions at the altitude for the hovering mission, and Liquid Water Content (LWC) and Median Volume Diameter (MVD) were set according to the temperature given in 14 CFR parts 25 and 29 appendix C continuous maximum atmospheric icing conditions.

Atmospheric Icing Condition [4 CFR parts 25 and 19 appendix C]

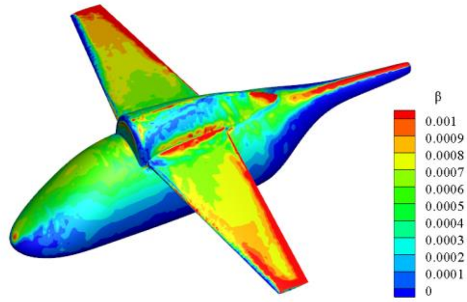
Parameter	Value
Pressure [kPa]	31.36
Temperature [K]	245.4
Density [kg/m^3]	0.5164
LWC [g/m^3]	0.0001
MVD [µm]	27.5



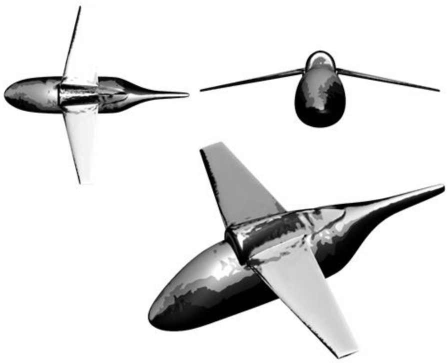
Instance Vorticity Contour [left: front view] [right: side view]



Z-Direction Droplet Velocity Contour (Surface-Collection Efficiency)



Collection Efficiency



Ice Shape

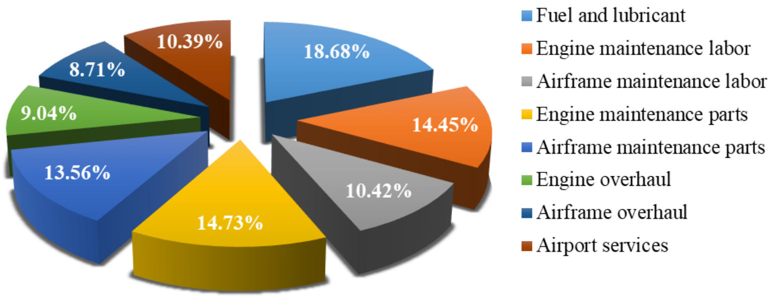
Cost Analysis

The design of a winged helicopter has already been well validated by a demonstrators X^3 which provided sufficient examination of the potential of this configuration. All components presented and adopted in the *CRANE* were incorporated based on the existing technologies. However, element such as the hybridized propulsion system requires intensive validation and FAA approval through the development of the rotorcraft. With the excellent ability for growth, the novel design of the *CRANE* would provide significant advancement to the rotorcraft community as well as recent popularization of air-taxi operation, and ultimately significantly reducing the overall cost of the rotorcraft.

The cost of the *CRANE* categorized into a compound helicopter is divided into four major parts: development cost, production cost, operational cost and end of life cost. The sum of these categories produces a total life-cycle cost per unit of a vehicle. Although developers and designers make much of these cost categories, the customer is mainly concerned with the cost in a different point of view: acquisition cost, operational cost, and end of life cost of a unit. Therefore, in this report, the description of cost analysis is provided in the sequence of 1. Acquisition cost, 2. Operational cost, 3. End of life cost, and 4. Life cycle cost. The all estimated cost values are presented in 2018 USD according to the averaged consumer price index (CPI) [30].

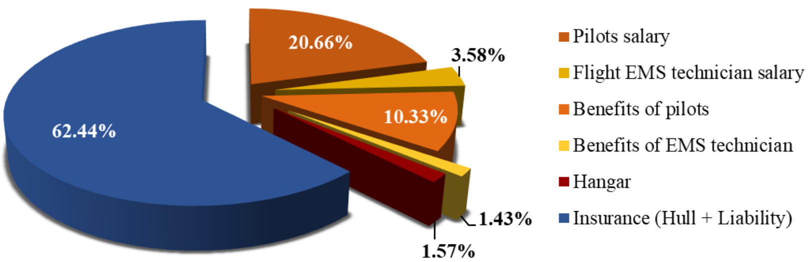
The *CRANE* DOC Per Flight Hours

Direct Operating Costs	USD / FH
Fuel and lubricant oil	145.55
Engine maintenance labor	112.56
Airframe and system maintenance labor	81.17
Engine maintenance parts	114.78
Airframe and system maintenance parts	105.67
Engine overhaul	70.46
Airframe and system overhaul	67.89
Airport services	80.98



The *CRANE* Annual IOC

Indirect Operating Costs	**USD / year**
Two pilots' salary	327,100
Flight EMS technician's salary	56,640
Benefits of pilots	163,550
Benefits of EMS technician	22,656
Hangar	24,840
Insurance (Hull + Liability)	988,667



The total life-cycle cost per unit of the *CRANE* is the sum of the above three components, corresponding to \$60 million/unit cost with the assumption that a lifetime operation is 20 years.

Reference

[1] Germanowski, P. J., Stille, B. L., and Strauss, M. P. "Technology Assessment for Large Vertical Lift Transport Tiltrotor." NASA/CR–2010-216384, May 2010.

[2] Michelle R. Kirby, 2001, “A Methodology for Technology Identification, Evaluation, and Selection in conceptual and preliminary aircraft design”, Ph.D. Dissertation, Georgia Institute of Technology

[3] Mavris, D., Baker, A., Schrage, D., “IPPD through robust design simulation for an affordable short haul civil tiltrotor”, Proceedings of the American Helicopter Society 53rd Annual Forum, Virginia Beach, VA, April 29-May 1 1997

[4] Sylvester Vikram. Ashok, 2013, “An Integrated Product-Process Development (IPPD) based approach for rotorcraft drive system sizing, synthesis and design optimization”, Ph.D. Dissertation, Georgia Institute of Technology

[5] Rawlings, D., “Record Breaking X3 Rotorcraft,” Redback Aviation Available: http://www.redbackaviation.com/record-breaking-x3-rotorcraft/.

[6] “Metro Aviation sells first EC145 e-lite,” Helicopter InvestorAvailable: https://www.helicopterinvestor.com/articles/metro-aviation-sells-first-ec145-e-lite-reduces-price-on-next-five-789/.

[7] Leishman, J. G., *Principles of Helicopter Aerodynamics*, 2nd Ed, Cambridge University Press, Cambridge, New York, 2006.

[8] Akao, Yoji, “QFD-integrating Customer Requirements into Product Design,” Productivity Press, Cambridge, Massachusetts, 1990

[9] Schwartzberg, M. A., Smith, R. L., Means, J. L., Law, H. Y. H., and Chappell, D. P., “Single-Rotor Helicopter Design and Performance Estimation Programs, Volume 1, Methodology,” U.S. Army Air Mobility R&D Laboratory, SRIO Report Number 77-1, 1977.

[10] Cohen, Lou,” Quality Functional Deployment—How to Make QFD Work for Your,” Addison-Wesley Publishing Company. Inc., Reading, Massachusetts, 1995

[11] Bae, Bo-Young, et al. “Process of Establishing Design Requirements and Selecting Alternative Configurations for Conceptual Design of a VLA.” Chinese Journal of Aeronautics, vol. 30, no. 2, 2017, pp. 738–751., doi: 10.1016/j.cja.2017.02.018.

[12] Kim, T., Oh, S., and Yee, K., “Improved Actuator Surface Method for Wind Turbine Application,” Renewable Energy, Vol. 76, Nov. 2014, pp. 16–26.

[13] ʺPegasus® Rescue Hoist,” Goodrich Rescue Systems Available: http://www.goodrichhoistandwinch.com/products/detail/pegasus/.

[14] Rendell Kheng Wah, Tan, 2000, “Quality functional deployment as a conceptual aircraft design tool”, M.S Thesis, Monterey, California. Naval Postgraduate School

[15] Smith, E. C., Govindswamy, K., Beale, M. R., Vascsinec, M., and Lesieutre, G. A., “Aeroelastic Response & Stability of Helicopters with Elastomeric Lag Dampers,” Journal of the American Helicopter Society, vol. 41, 1996, pp. 257–266.

[16] Misra, A., “Summary of 2017 NASA Workshop on Assessment of Advanced Battery Technologies for Aerospace Applications,” SciTech Forum, AIAA, 2018.

[17] Ricci, M., “High Efficiency, High Power Density Electric Motors," Product Presentation, 2010.

[18] Cylinders Kevlar - Mountain High Aviation Oxygen Supply Available: http://www.mhoxygen.com/index.php/cylinders/kevlar.

[19] Bosset, James L., “QFD—A Practitioner’s Approach”. ASQC Quality Press, Milwaukee, Wisconsin, 1991

[20] Raymer, D. P., Aircraft Design: A Conceptual Approach, Education Series, AIAA, Washington, DC, 1992

[21] ″US8561945B2 - Skid-type landing gear for a helicopter,” Google Patents Available: https://patents.google.com/patent/US8561945.

[22] Paur, J., “Sikorsky X2 Breaks Helicopter Speed Record,” Wired Available: https://www.wired.com/2010/08/sikorsky-x2-breaks-helicopter-speed-record/.

[23] Son, Chankyu, Sejong Oh, and Kwanjung Yee. "Ice accretion on helicopter fuselage considering rotor-wake effects." Journal of Aircraft 54.2 (2016): 500-518.

[24] Kim, T., Oh, S., and Yee, K., “Improved Actuator Surface Method for Wind Turbine Application,” Renewable Energy, Vol. 76, April 2015, pp. 16–26.

[25] FAA AIR-120. AC 20-73A, Aircraft Ice Protection, Aug. 16, 2006.

[26] Fink, R., “USAF Stability and Control DATCOM,” AFWAL-TR-83-3048, 1978

[27] Gareth D. Padfield, “Helicopter Flight Dynamics: The Theory and Application of Flying Qualities and Simulation Modeling,” Blackwell Science Ltd, 1996.

[28] ″Nodin Aviation AS,” Copybook Available: https://www.copybook.com/companies/nodin-aviation-as/articles/fast-mounting-stretchers-for-patient-transportation.

[29] Dailymail.com, M. P. F., “Uber working with US Army to develop a 'flying taxi' service to get soldiers to the front lines,” Daily Mail Online Available: https://www.dailymail.co.uk/sciencetech/article-6091809/Uber-working-Army-develop-flying-taxi-service-soldiers-lines.html.

[30] “U.S. Department of Labor, Consumer Price Index, Bureau of Labor Statistics,” https://www.bls.gov/cpi/home.htm.

[31] Harris, F. D. and Scully, M. P., “Helicopters Cost Too Much,” American Helicopter Society 53rd Annual Forum, Virginia Beach, VA, April 1997.

[32] Roskam, J., Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing and Operating, Design, Analysis, Research corporation, Lawrence, KS, 2015.

[33] “U.S. Department of Labor, Aircraft Mechanics and Service Technicians, Bureau of Labor Statistics,” https://www.bls.gov/oes/2017/may/oes493011.htm.

[34] NASA, “Maintenance Cost Study of Rotary Wing Aircraft,” NASA CR 152003, Jun 1977.

[35] “Tribhuvan International Airport, Airport Regulations,” https://www.tiairport.com.np/regulations.

[36] Zhao, X., Verhagen, W.J., Curran, R., Borsato, M., Wognum, N., Peruzzini, M. and Stjepandić, J., “Aircraft Disposal and Recycle Cost Estimation,” Advances in Transdisciplinary Engineering, Vol. 4, 2016.

[37] "Safety in Alpine Helicopter Rescue Operations—Minimal Requirements of Alpine Skills for Rescue Personnel," The Annals of Occupational Hygiene, 2013.

[38] Prouty, R. W., "Helicopter Performance, Stability, and Control," PWS Publishers, 1986.

[39] Roskam, J., Airplane Design | Part II: Preliminary Configuration Design and Integration of the Propulsion System, Design, Analysis and Research Corporation, Lawrence, KS, 2004.

[40] "HurleyIR Falcon Spectral II Thermal Imaging System," HurleyIR Falcon Spectral II Aerial Gimbal for Bell 407Available: https://www.aero-access.com/products/HurleyIR-Falcon-Spectral-II-Thermal-Imaging-System.html.

[41] Guide, A., "Aircraft Windshield Frost, Fog and Portable Water Tank Ice Control Systems, Water and Waste Tank Ice Prevention - Aircraft Ice Protection," Aircraft Systems Available: https://www.aircraftsystemstech.com/2017/05/aircraft-windshield-frost-fog-and-ice.html.

[42] "The T700 Engine," GE Aviation Available: https://www.geaviation.com/military/engines/t700-engine.

초 록

회전익기 설계에는 여러 분야의 의사 결정 프로세스가 필요하며 복잡한 항공 우주 시스템의 불완전한 요구 사항 및 임무 목표를 가지고 설계를 수행한다. 또한 기존의 직렬 설계 방식은, 한 분야의 정보를 전달하여 엄격한 설계 반복을 거쳐야 한다. 하지만 공식적인 설계 방법론이 없고 극도로 복잡한 회전익기 설계에 대한 설계 반복성은 과도한 자원을 요구하기에 이 논문은 시스템 엔지니어링 기반 요구 사항 분석 도구 인 QFD (Quality Function Deployment), 즉 품질기능전개방법을 통한 개념적 설계 프레임워크 개발 및 통합 설계에 중점을 둔다. 회전익기 설계는 복잡한 MDO (Multidisciplinary Design Optimization)를 요구하여 상호 의존적 인 사이징 변수로 인한 성능 변화에 상당한 영향을 미친다. 이를 위해 House of Quality (HOQ)는 사용자 요구를 설계 품질로 변환하고 고객 특성에 영향을 미치는 일련의 설계 특성의 우선 순위를 정하여 초기 설계 단계에서 최대한 자원을 사용하여 설계를 진행한다. 이를 위해 QFD 프로세스로 채택되었고 제시 된 설계 프레임 워크는 개념 설계 단계에서 다양한 분석을 포함하는 병렬 설계 접근법을 채택하여 회전익기를 설계하는 방법을 제시하였다. 또한, 제시된 프레임 워크는 공기 역학, 구조, 추진력, 변속기 설계, 중량 및 균형, 안정성 및 제어, 소음 분석 및 경제적 분석을 비롯한 다양한 기술적 측면을 고려하여 설계 반복과정으로 이어진다.

이 논문은 QFD 과정의 시스템 통합적 검증을 위해 고고도의 구조 헬기를 설계하여 제시 된 개념 설계 프레임 워크를 적용하였다. Morphological Analysis와 다양한 QFD 분석을 수행함으로써, 날개에 장착된 4개의 프로펠러와 하이브리드 추진 시스템을 장착한 winged-helicopter가 설계되었다. 이 결과는 중요한 설계 변수가 확인 된 HOQ 분석과 최적의 솔루션을 얻기 위해 구체적으로 설계 된 결과이다. 제시된 디자인 프로세스를 채택함으로써 29,100ft에서 OGE 호버링이 가능하면서 185knots 의 순항 속도를 달성 할 수 있는 로터 크래프트가 설계되었습니다. 제안 된 설계 프레임 워크는 QFD 프로세스의 통합으로 항공우주비행체를 설계하고 예비 설계를 시작하는 데 필수적인 정보를 제공하는 더욱 효율적이고 효과적인 설계방법으로 적용될 수 있다.

주요어 : 품질기능전개방법, Quality Function Deployment, 회전익기, 개념설계 기법, 최적 설계, 고고도 구조 헬기

학 번 : 2017-20158