garra

Final Proposal

Undergraduate Design Team

University of Maryland – College Park

38th Annual Vertical Flight Society
Student Design Competition

Sponsored by: The Boeing Company & Altair

Alfred Gessow Rotorcraft Center
Department of Aerospace Engineering
University of Maryland
College Park, MD, 20740

To the Vertical Flight Society:

The members of the University of Maryland Undergraduate Student Design Team hereby grant full permission to distribute the enclosed Executive Summary, Final Proposal, Safety Analysis, and Weight Optimization Paper for the 38th Annual Design Competition as they see fit.

Best Regards,

The UMD Undergraduate Design Team

Alfred Gessow Rotorcraft Center
Department of Aerospace Engineering
University of Maryland
College Park, MD, 20740

Derek A. Safieh
Undergraduate Student (Team Lead)
dsafieh@terpmail.umd.edu

Andrew Lent
Undergraduate Student (Aerodynamics)
Andrew.lent99@gmail.com

Joseph Cohen
Undergraduate Student (CAD)
jsc6121@terpmail.umd.edu

Tim Young
Undergraduate Student (Propulsion)
Tim.j.young04@gmail.com

Emmanuel Azadze
Undergraduate Student (Propulsion)
eazadze35@gmail.com

Rashaun Williams
Undergraduate Student (Avionics)
rwilliamsofficial01@gmail.com

Noam Kaplan
Undergraduate Student (Aerodynamics)
c.Kaplan.noam@gmail.com

Ariq Zufar
Undergraduate Student (Structures)
ariqzufar23@gmail.com

Jordan Schmidt
Undergraduate Student (Safety)
officialjordanschmidt@gmail.com

Dr. Vengalattore Nagaraj
Senior Research Scientist
vnagaraj@umd.edu

Dr. James Baeder
Professor
baeder@umd.com

Dr. Inderjit Chopra
AGRC Director and Professor
chopra@umd.edu

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- Nicholas Rehm - Aerospace Engineering Masters Student AGRC
- Brandyn Phillips - Aerospace Engineering Masters Student AGRC
- Fred Tsai - Aerospace Engineering Masters Student AGRC
- Shashank Maurya - Aerospace Engineering Masters Student AGRC

External

- Natalia Alvarez
- Vikram Hrishikeshavan

Contents

I Introduction 1
- I.A RFP Analysis . . . 1
  - I.A.1 Mission Environment . . . 1
  - I.A.2 Package Description . . . 3
  - I.A.3 Mission 1: Delivery to end user . . . 3
  - I.A.4 Mission 2: Logistics center delivery . . . 4

II Con guration Selection 5
- II.A Identi cation of Design Drivers using an Analytical Hierarchy Process (AHP) . . . 5
  - II.A.1 Design Driver Formulation . . . 5
  - II.A.2 AHP Matrix . . . 5
- II.B Con guration selection using a Pugh Matrix Analysis . . . 6
  - II.B.1 Tilt-Wing . . . 7
  - II.B.2 Tilt-Rotor . . . 7
  - II.B.3 Co-axial Helicopter . . . 8
  - II.B.4 Single Main Rotor (SMR) . . . 8
- II.C Single Main Rotor Compounding . . . 8

III Vehicle Sizing 9
- III.A Methodology . . . 9
- III.B Code Validation . . . 10
- III.C Package Deployment Time Trade Study . . . 10
- III.D Main Rotor Trade Studies . . . 10
  - III.D.1 Main Rotor Trade Study: Varying Disk Loading . . . 10
  - III.D.2 Main Rotor Trade Study: Varying Aspect Ratio . . . 10

IV Main Rotor Design 11
- IV.A Main Rotor Aerodynamic Design . . . 11
  - IV.A.1 Pareto Plot Code Setup . . . 11
- IV.B Main Rotor Blade Design . . . 12
- IV.C Main Rotor Dynamics . . . 13
- IV.D Main Rotor Hub . . . 14

V Tail Rotor, Pusher Propeller and Empennage Design 14
- V.A Empennage Con guration Selection . . . 14
  - V.A.1 NOTAR with Pusher Propeller . . . 14
  - V.A.2 Ducted Pusher Propeller with Variable Guide Vanes . . . 15
  - V.A.3 Swiveling Tail Rotor . . . 15
  - V.A.4 Fenestron with Pusher Propeller . . . 15
  - V.A.5 Open Tail Rotor and Pusher Propeller . . . 15
- V.B Tail Rotor Design . . . 15
  - V.B.1 Tail Rotor: Radius . . . 16
  - V.B.2 Tail Rotor: Aspect Ratio and Tip Speed . . . 16
  - V.B.3 Tail Rotor: Airfoil and Twist . . . 17
  - V.B.4 Tail Rotor: Placement, Type, and Direction . . . 17
- V.C Tail Rotor Swashplate . . . 17
- V.D Vertical Stabilizer Design . . . 18
- V.E Horizontal Stabilizer Design . . . 19
  - V.E.1 Horizontal Stabilizer Sizing . . . 19
  - V.E.2 Horizontal Stabilizer Location and Sweep . . . 19
- V.F Pusher Propeller Design . . . 19
  - V.F.1 Propeller Trade Studies . . . 19
  - V.F.2 Propeller Aerodynamic Design . . . 20

V.F.3 Need for a Rear Gear Box . . . 20
V.G Pusher Propeller Variable Pitch Mechanism . . . 21

VI Drag Estimation 21
VI.A Flat Plate Area Estimation . . . 22
VI.B Hover Download Estimation . . . 22

VII Vehicle Performance 23
VII.A Vehicle Power Usage . . . 23
VII.B Segment By Segment Performance . . . 23
VII.C Performance Metric Calculation . . . 23
VII.D Range and Endurance . . . 24

VIII Propulsion 24
VIII.A Power Plant Selection . . . 24
VIII.B Engine Selection . . . 25
VIII.C Tail Con guration . . . 27
VIII.D Transmission . . . 27
VIII.E Tail Shaft . . . 27
VIII.F Oil and Housing . . . 28

IX Airframe Structural Design 29
IX.A Description . . . 29
IX.B Fuselage Structure . . . 29
IX.B.1 Fuselage Structural Enhancement . . . 29
IX.C Empennage Structure . . . 31
IX.D Cowling Structure . . . 32
IX.E Load Paths . . . 32
IX.F Material Selection . . . 32
IX.G Landing Gear . . . 32
IX.G.1 Static Stability Angles . . . 32

X Avionics 33
X.A Mission Requirements . . . 33
X.A.1 Takeo and Landing . . . 34
X.A.2 Cruise . . . 34
X.A.3 Collision Avoidance . . . 34
X.A.4 Supporting Equipment . . . 34
X.B Sensor Suite . . . 34
X.B.1 Autonomy . . . 34
X.B.2 Collision Avoidance . . . 34
X.B.3 Flight Navigation . . . 35
X.B.4 Takeo and Landing . . . 36
X.B.5 Pilot in the Loop . . . 36
X.B.6 Supporting Equipment . . . 36

XI Concept of Operations 37
XI.A Mission Requirements . . . 37
XI.B Center of Gravity Envelope . . . 39
XI.C Package Deployment System Overview . . . 39
XI.C.1 Releasing Claw Mechanism . . . 39
XI.C.2 Winch Mechanism . . . 41
XI.C.3 Deployment Door Mechanism . . . 41
XI.D Payload Deployment Concept of Operations . . . 41
XI.E Payload Deployment Failure Modes and Solutions . . . 42

XI.F Vehicle Preparation 42
XI.F.1 Pre- ight Inspection and Maintenance 42
XI.F.2 Vehicle Turn-Around 45
XI.F.3 Access Panels 45
XI.G Ground Transportation 45

XII Aircraft Acoustics 45
XII.A Noise Requirements 45
XII.B Acoustics Analysis 46

XIII Safety 46
XIII..1 Safety during Package Delivery 46
XIII..2 Con guration Redundancy 47
XIII..3 Autonomous System Safety 47
XIII.A Top Technologies to Improve Safety 47
XIII.B Safety Throughout the Whole Mission 47

XIV Cost 48

XV Weight Breakdown 49

XVI Summary 50

List of Figures

1 Visual Flight Rules (VFR) Minimums . . . 1
2 Package Size in Perspective . . . 3
3 AHP Matrix . . . 6
4 Pugh Matrix . . . 6
5 Possible con gurations for the RFP . . . 7
6 Other Proven Concepts like The Garra . . . 9
8 Flowchart of Sizing Code . . . 9
7 Validation of Sizing Code Scripts . . . 9
9 Main Rotor Trade Studies . . . 11
10 Main Rotor Pareto Plot, over 30,000 Con gurations . . . 12
11 Main Rotor Blade Geometry . . . 13
12 Blade Structural Composition . . . 13
13 Fan Plot . . . 13
14 Main rotor swash plate . . . 14
15 Tail Rotor Trade Studies . . . 16
16 Tail Rotor Placement . . . 17
18 Tail Rotor Swashplate . . . 17
17 Tail Rotor Blade Geometry . . . 18
19 Empennage Structure . . . 19
20 Pusher Propeller Trade Studies . . . 20
21 Pusher Propeller Blade Geometry . . . 21
22 Pusher Propeller Swashplate . . . 21
23 Garra Power Curves . . . 23
24 Range Endurance Plots . . . 25
25 Main Drive and Tail Drive . . . 28
26 Main Bay Diagram . . . 29
27 Study of the E ects of Supports on the Open Bottom Fuselage . . . 31
28 Landing Gear Pugh Matrix . . . 32
29 Static Stability Angles . . . 33
30 Avionics control diagram . . . 33
31 Avionics Suite . . . 35
32 Stereo Camera and Scanning Lidar FOV . . . 37
33 Footprint of Garra . . . 37
34 Center of Gravity Envelope . . . 39
35 Package Deployment System Overview . . . 39
36 Deployment System Overview . . . 40
37 Claw Mechanism Overview . . . 40
38 In-House Built Winch System . . . 41
39 Door Mechanism Overview . . . 42
40 Vehicle Access Panels . . . 45
41 Garra Pickup Truck Transportation . . . 46
42 Garra Acoustics Analysis . . . 46
43 Vehicle Flies into the Sunset . . . 50

List of Tables

1 RFP Compliance . . . 3
2 Delivery Mission Steps . . . 4
3 Logistics Mission Steps . . . 4
4 Design Drivers . . . 5
5 Design Drivers . . . 6
6 Modi ed momentum theory power required for 200 knot cruise . . . 8
7 Delivery Mission Package Deployment Trade Study . . . 10
8 Geometries Varied in Main Rotor Design . . . 11
9 Main Rotor Design Details, V_{cruise} = 69.4 m/s (135 kts) . . . 12
10 Design Drivers For Rotor Hub . . . 14
11 Tail Rotor Radius vs Crosswind Rejection Trade Study . . . 15
12 Tail Rotor Design Details . . . 18
13 Empennage Geometry . . . 18
14 Pusher Propeller Design Details . . . 21
15 Flat Plate Area Drag Estimation . . . 22
16 Hover Download Trade Study . . . 23
17 Segment by Segment Vehicle Performance, Logistics Mission . . . 24
18 Segment by Segment Vehicle Performance, Delivery Mission . . . 24
19 Vehicle Performance . . . 26
20 Power Plant Selection Pugh Matrix . . . 26
21 Engine Comparison . . . 26
22 Tail Con guration Weight Comparison . . . 27
23 Gear Material Comparison and Gear Ratio . . . 27
24 Tail Shaft Comparison . . . 28
25 Avionics Selections, Power, and Weight . . . 38
26 Pre-Flight Inspection . . . 44
27 Turn Around Procedure Between Flights . . . 45
28 Values Used in Cost Analysis . . . 48
29 Garra Weight Breakdown . . . 49

Nomenclature

English Symbols

"'	Area of the Main Rotor
3	Airfoil Drag Coe cient
0	Zero Lift Drag Coe cient on Fuselage
E	Vertical Drag Coe cient
5	Skin Friction Coe cient
;	Airfoil Lift Coe cient
<	Airfoil Moment Coe cient
?	Power Coe cient
?8	Induced Power Coe cient
?>	Pro le Power Coe cient
??	Parasitic Power Coe cient
)	Thrust Coe cient
) %A>?	Propeller Thrust Coe cient
5	E ective Flat Plate Area
	Form Factor
"	Figure of Merit
	Interference Factor
	Propeller Advance Ratio
! • 4	Lift over E ective Drag Ratio
"	Mach Number
" 33	Drag Divergence Mach Number
" C8?	Rotor Mach Tip Speed
=2A8C	n criterion
%'	Main Rotor Power
%'	Tail Rotor Power
@	Dynamic Pressure
'	Rotor Radius
(A45	Reference Area
(F4C	Wetted Area
E 2AD8B4	Cruise Velocity
E C8?	Rotor Tip Speed
F	Width
, ?0H;>03	Payload Weight

Greek Symbols

[?	Propeller E ciency
\ 75	Geometric Blade Twist at 75% Radius
`	Rotor Advance Ratio
f	Rotor Solidity
Z	Performance Metric from the RFP

Abbreviations

'	Aspect Ratio
>	Angle of Attack
")	Blade Element Momentum Theory
	Computational Fluid Dynamics
!	Above Ground Level
$+	Field of View
)$,	Gross Takeo Weight
	Hover in Ground E ect
$	Hover out of Ground E ect

(	International Standard Atmosphere
$ (%!	Overall Sound Pressure Level
' %	Request for Proposal
(	Speci c Fuel Capacity
+'(	Vortex Ring State

I. Introduction

The Vertical Flight Society's 38th Annual Student Design Competition, sponsored by Boeing in 2020-2021, asks teams to design a fully autonomous cargo vehicle capable of aiding in a future pandemic or natural disaster. According to the speci cations, the package will be designed to carry sensitive materials such as vaccines or aid. Due to the rapidly evolving climate situation and health concerns, the RFP states that this vehicle has to be operational by the year 2025 only using technology readily available today.

In response to this year's RFP, the undergraduate design team from the University of Maryland has proposed the "Garra". Garra is a fully autonomous thrust compounding single main rotor helicopter (see 3-D drawing in following pullout page) capable of delivering the desired package at high speeds and within the speci ed mission envelope. The Garra is named after the Garra Rufa sh, also known as the doctor sh, which is known to be able to sprint at high speeds underwater. The team found the name tting as this sh can rush to aid whenever it is needed. Through the design process, the team consulted with industry leaders from Sikorsky, Boeing, and Joby Aviation as well as with experienced commercial helicopter pilots. Based on their input, the Garra is on the forefront of autonomous aviation and will be capable to provide change in the event of a future natural disaster or pandemic in rural settings.

Some of the most notable capabilities of the Garra are its ability to achieve high speeds at a zero-degree shaft tilt 250.02 km/hr (135 knots), package deployment versatility, and redundant safety measures to ensure mission and vehicle integrity. Through its thrust compounded design, the Garra is able to keep the package from any unwanted attitudes during all stages of ight: a key safety feature of the design. The vehicle is powered by a twin four-cylinder, two-cycle engine combination, providing 74 kW (99 HP) of available power throughout the missions. This power surplus provides the design with incredible versatility and maneuverability. In addition, the twin engine con guration provides an extra safety feature for the Garra as the chance of a total engine failure is reduced. The vehicle is also designed to complete both mission pro les (with a time penalty) with only one engine operative.

The Garra is designed to achieve an unprecedented task, so the design and operation combines traits of classical cargo helicopters like the Sikorsky Skycrane with the versatility of stealth vehicles like the Lockheed Chayenne and the Boeing Sheibel Camcopter.

A. RFP Analysis

The RFP issued by the Boeing Company had a recurring topic of safety. The safety aspects that the RFP highlights are those of the mission, operators, bystanders, and of the package. The RFP did not state many restrictions on the design, giving the team space to be creative in the con guration, propulsion, and deployment systems. The main restrictions are outlined on table 1.

1. Mission Environment

The RFP states that the vehicle must be able to complete two di erent missions without recon guration. It is important to note that the vehicle may refuel between completed missions and the fuel capacity can be varied depending on the mission to enhance performance. This subsection will outline the di erent missions required by the RFP. The RFP indicates that the vehicle will operate in a very speci c set of conditions during all times. There were three main environmental constraints: launch altitude, ight altitude, and temperature. The rst states that the vehicle will initially takeo from a location that is 1200 m (3937 ft) mean sea level (MSL). The second states that the vehicle shall maintain the ight at an altitude of 150 m (492 ft) above ground level (AGL), this would put the vehicle at an overall ight altitude of 1350 m (4429 ft) MSL. The temperature of the operation was selected by the RFP committee to be ISA + 20C (68 F) and the temperature would be held during all operations, with this restriction the vehicle will be e ectively ying at an outside temperature of 35 C (95 F). In summary, the atmospheric conditions outlined by the RFP show that the vehicle will y in a hot day at a high altitude. In addition to the atmospheric conditions, the RFP states that the vehicle will only y in Visual Flight Rules (VFR) conditions. Therefore, the vehicle will not y inside clouds or poor visibility. Figure 1 shows the VFR minimums presented by the FAA for any operations within the United States and its territories. The Garra is designed to operate on these minimums to maintain safety as a top priority.

Fig. 1 Visual Flight Rules (VFR) Minimums

2. Package Description

One of the main constraints of the RFP is the sizing of the payloads that are being carried. The RFP outlines two possible package con gurations: a cube and a rectangular prism. These have 0.7 m x 0.7 m x 0.7 m (2.3 ft x 2.3 ft x 2.3 ft) and 1.4 m x 0.5 m x 0.5 m (4.6 ft x 1.6 ft x 1.6 ft) respectively. It is important to note that the vehicle only has to be able to carry one package per mission. Figure 2 shows both packages with respect to an average American male for size perspective. The safety of the package is of utmost importance for the mission because it is designed to carry sensitive materials intended to make a di erence during the relief of a pandemic or a natural disaster. The team de ned that these contents could be medicines, vaccines, surgical equipment, etc. These possible payloads led the team to take extra safety measures during the mission design, the most important being that the package must be level at all times. As the current COVID-19 climate has shown us, payloads like vaccines are very sensitive, which reassures the assumption made that the payload can not be tilted. This interpretation requires a new approach for loading and unloading of the package.

Fig. 2 Package Size in Perspective

The RFP also states that the team has full control over the packages design and attachment points. It can be assumed that it will be manufactured with the desired components. Its important to note that the package is self-contained and it is out of the scope of the design for this project.

Table 1 RFP Compliance

RFP Restrictions	Solution	Chapter
Vehicle must be an unmanned vertical lift concept that can deliver packages at high speeds in a rural setting	Vehicle is a fully autonomous aircraft with great cruise performance	VII, X
Without recon guration, the vehicle must be able to deliver payloads to end-user customer sites up to 50 km (27nm) radius and to logistics centers up to 200 km away (108 nm).	The vehicle has excellent range, and the ability to transport and deliver various package sizes.	VII, XI
The design must include only current year (2020) technologies.	All systems on the Garra are o the shelf certi ed products	X
The vehicle must t in 4.6 m (15 ft) by 4.6 m (15 ft) footprint.	The vehicle is sized to t in a 4.6 m (15 ft) by 4.6 m (15 ft) footprint.	III
The vehicle only should carry one of the designated packages at a time.	The package bay is designed to t either of the RFP packages	XI
Refueling is permitted between completed missions	Garra has a fuel tank large enough to complete the longest mission. Fuel level will be indicated prior to each mission	VIII
System must be able to perform each mission autonomously	Comprehensive avionics suite designed for complete autonomy with pilot-in-the-loop capability	X
Overall Operational Safety	Garra includes safety in every aspect o the design, including package, operator, vehicle and bystander safety	all

3. Mission 1: Delivery to end user

The rst mission pro le described by the RFP is a round-trip delivery mission. During this mission, the vehicle must carry the package to the location, autonomously deliver it, and then y back without a package. It is important to note

that the delivery at this location can be both from the air or from land, and this is determined by the individual teams. The steps of this operation are found on Table 2, it is important to note that the package is delivered through a hoisting mechanism and the vehicle does not land at the location. The reason for which the Garra does not land as a primary way of delivering the package is because during a natural disaster it can be hard to ensure a clear landing site. Debris on the sites can pose a serious threat to both the package, the vehicle, and other operators. By hoisting the package down, it is assured that the Garra will be able to complete the mission. The RFP mentions that the maximum radius of operation for this mission is 50 km (27 nm) and an allocated block time of 28 minutes for the rst leg. For this mission the RFP outlines that the vehicle should be able to operate for 20 minutes after reaching the location at best endurance speed.

Table 2 Delivery Mission Steps

Action	Time (min)	Description
Load Package	5	Package is loaded into the vehicle
Warm-up	5	Engine and systems warm-up
Takeo (HOGE)	2	Takeo out of ground e ect and hover test
Climb-Cruise-Descent	13	Climb is performed at best climb speed and cruise speed is trimmed to achieve the mission in time
HOGE	1	Hover out of ground e ect
Package Deployment	2	Hoist the package from the vehicle to the ground
Takeo (HOGE)	1	Takeo out of ground e ect and prepare for the return leg
Climb-Cruise-Descent	13	Climb is performed at best climb speed and cruise speed is trimmed to achieve the mission in time
Land (HOGE)	1	Land out of ground e ect maneuver and shutdown
Total Allocated Mission time	43 mins	

4. Mission 2: Logistics center delivery

For this mission, the RFP only requires the vehicle to travel one way to a logistics center to deliver the intended package. In this case, the vehicle does not have to return to its original location after delivering the package. The deployment in this case can be done autonomously or manually. The method of delivery can be determined by the operator prior to the beginning of the mission. For the autonomous delivery, the package would be hoisted down to the location and then the vehicle would land in a designated area at the logistics center. For the manual delivery, the vehicle would land in the designated area and completely shut down before the operator approaches the vehicle to retrieve the package. The methodology will be explored in a further chapter of the report. The step-by-step operation for this mission is outlined in table 3. This mission has a maximum range of 200 km (108 nm) and an allocated 75 minutes of block time which includes all the warm up and ight procedures, leaving only 60 minuted for ight. For this mission, the RFP outlines that the vehicle should be able to operate for 20 minutes after reaching the location at the best endurance speed. The Garra is designed to exceed these requirements in both speed and range.

Table 3 Logistics Mission Steps

Action	Time (min)	Description
Load Package	5	Package is loaded into the vehicle
Warm-up	5	Engine and systems warm-up
Takeo (HIGE)	2	Takeo In Ground E ect
Climb-Cruise-Descent	60	Climb is performed at best climb speed and cruise speed is trimmed to achieve the mission in time
Package Deployment	2	Hoist the package from the vehicle to the ground
Land (HOGE)	1	Land out of ground e ect
Total Allocated Mission time	75 mins	

II. Con guration Selection

The Garra is a single main rotor thrust compounded aircraft designed for safety, forward ight capability, and extreme controllability. The RFP gives a lot of space for creativity when it came to the design, so careful attention was given to the restrictions and the voice of the customer. A rigorous process was followed to take all design requirements and identify all possible con gurations. From these, a systematic approach was followed to choose the best con guration to ensure the safety and completion of the mission.

The rst step in the selection process was to translate what the customer was looking for into quanti able design drivers. Once these drivers were selected from the RFP information, an analytical hierarchy process was used to rank each driver based on their relative importance to the design. After ranking the drivers, these were applied to a Pugh matrix to compare the major con gurations of one another. Through this matrix, some of the most proven con gurations were studied and ranked based on the desired drivers. From this matrix, a newly narrowed list was created for further investigation.

This process was followed because it opened the horizons to new con gurations with the best qualities of each vehicle. In the process, the team was able to analyze what qualities would make the perfect con guration match the voice of the customer at the highest degree. To accomplish this task, the team conducted an extensive literature review using the University of Maryland's past designs, the VFS helicopter directory, and historical designs. After this was conducted, a list of four main types was identi ed: single main rotor, coaxial, VTOL with transition, and distributed electric aircraft.

A. Identi cation of Design Drivers using an Analytical Hierarchy Process (AHP)

An AHP analysis was utilized to categorize the design drivers from the RFP to be able to use them in further Pugh matrix analysis. This method was rst ideated by Thomas L. Saaty in the early 1970s with the goal of organizing complex traits using a combination of math and customer psychology. In today's world, this is the most used method to quantify and rank non-quanti able features.

1. Design Driver Formulation

This year's RFP did not present hardline design drivers or strict requirements, giving way to the team's creative ideas. In light of this, the team took a holistic approach to de ne the design drivers for the design. Through this process, each team member of the University of Maryland's undergraduate team performed its own independent analysis of the RFP. Later, a brainstorming session was done to share individual opinions and gave the opportunity to discuss di ering view points. Through this process, every team member brought their own perspective to the table and ensured that none of the desires of the customer were missed from the RFP interpretation. This yielded a set of twenty distinct attributes that then were grouped and reduced to eight top design drivers. These design drivers and de nitions are summarized

Table 4 Design Drivers

Design Driver	De nition
Safety	Minimizing risk to life or property during in- ight emergency
Cruise Performance	High cruise speed, maximizing range/endurance, and minimizing the power required
Hover Performance	IGE/OGE Performance, stability, and minimizing power required
Mechanical Simplic-ity	Keeping the design of the rotorcraft with the fewest parts
Maneuverability	The agility of the helicopter in complicated situations to ensure safety of the operation
Payload Capacity	Maximization of payload capacity including the added weight and space of deployment systems
Compactness	The RFP states that the vehicle ideally ts in a 15' x 15' footprint
Noise	Low noise footprint to ensure ability to operate near homes and other establishments.

2. AHP Matrix

With the design drivers well de ned and approved by all team members, the AHP matrix, found on gure 3, was created to rank the drivers. This process took into account the opinions of every team member and was later revised by the advisor group. To ensure that there were not outliers in the process and to nd di erences of opinion easily, the team calculated a consistency factor: this was below the 0.1 threshold.

Fig. 3 AHP Matrix

The normalized AHP matrix shown in gure 3 produced the priority scale shown in Table 5. These values are the baseline for the weighting system of the Pugh Matrix that will be described in a later section.

Table 5 Design Drivers

Design Driver	Score
Safety	29.86%
Cruise Performance	16.64%
Hover Performance	14.90%
Mechanical Simplicity	8.62%
Maneuverability	8.55%
Payload Capacity	8.50%
Compactness	6.76%
Noise	6.17%
Total	100%

Taking a closer look at the design drivers, we may highlight a few. The most important driver for the design is safety; considering that this vehicle will provide aid to areas hit by natural disasters, ensuring the safety of the mission, outsiders, the package, and the vehicle is of utmost importance. Our next top driver is the cruise performance of the vehicle. The shape of the package is unique in the sense that it has a large frontal area, a trait that is not ideal for high-speed ight. Designing the vehicle for its cruise performance would de ne whether or not the mission could be completed. Due to the team decision to deploy the package from a hover, obtaining an e cient hover performance is essential. In addition, the package shape poses a challenge when reducing the downwash of the rotor on the fuselage. Finding a balance between both cruise and hover performance is key to the success of the Garra, a consideration that is shown by its close ranking on the AHP analysis. The other design drivers were still taken into account for the con guration process and were key in nding the correct con guration for the Garra.

B. Con guration selection using a Pugh Matrix Analysis

Fig. 4 Pugh Matrix

Once the design drivers were properly selected and ranked through the AHP analysis described in the section above, a Pugh Matrix method was used to compare the the various con gurations. Through this method, a large number of con gurations were able to be compared with the design drivers in a weighted scale. The weighted scale method was implemented to ensure the proper in uence that each design driver would have on each con guration's score.

One of the key features of the Pugh Matrix is the selection of a control group to have a relative score for the selection. This con guration receives a neutral score (0) in all categories as well as in the overall score. Each of the other con gurations is compared directly to the control group, a positive score shows that the con guration is better than

the control, and a negative score shows that it is worse. The overall score of each con guration is calculated through the weighted average system put in place. The ones with the best positive scores were selected for further analysis.

The scoring for the Pugh Matrix was de ned within a range from -3 to 3 by only selecting integers. In this scale, 3 means that the compared con guration is absolutely better than the control and -3 means that it is absolutely worse. The values seen on gure 28 do not represent an integer as it represents the average of all scores given by each team member. Similarly to the AHP analysis, a coe cient of variance was calculated to ensure that there were not any outliers in the data set.

From the Pugh matrix, it was de ned that four di erent con gurations would t the needs to comply with the RFP. These con gurations were found to be the SMR, Tilt-Rotor, Co-Axial, and Tilt-Wing con gurations.

(a) Tilt-Wing (b) Tilt-Rotor (c) Co-axial (d) SMR

Fig. 5 Possible con gurations for the RFP

1. Tilt-Wing

A tilt-wing is a novel type of VTOL vehicle that is now commonly used for urban air mobility applications. The most famous quasi-commercial vehicle with this con guration is the Lilium vehicle. Most of these designs use distributed electric propulsion to meet the power demands for both hover and forward ight. This design provides a lot of safety features. The most important one is the redundancy of the electric motors on board. In addition, the con guration has the gliding capabilities of a xed wing aircraft while providing a reliable landing on a runway. Even though the con guration provides with great traits, there are three main traits that disqualify it from being useful for this RFP. The rst is the footprint, because it relies solely on wings for forward ight, the footprint of the vehicle tends to grow exponentially with load. This provides a considerable problem when tting the RFP constraints. The second is the readiness of the technology. The RFP states that the vehicle must be tted with today's technology to be operational in 2025 (4 years from now). Sadly, the electric technology needed for the operation of this con guration is not up to aviation industry standards yet. The nal reason is that these vehicles tend to be very mechanically complex. The mechanism to tilt the wing is extremely complicated and prone to failure, especially when it is stored for long periods of time (as the vehicle required for the RFP will be). The con guration was discarded due to its lack of safety record, complex systems, and footprint.

2. Tilt-Rotor

The tilt-rotor is a type of VTOL or STOL vehicle in which the nacelles rotate 90 degrees to provide vertical thrust in hover and forward thrust in forward ight. This technology has been in the spotlight and has been perfected over the past years. The most known type of aircraft of this type is the Bell-Boeing V-22 Ospray Tilt-rotor. This type of vehicle provides the versatility of a helicopter with the capabilities of a xed wing aircraft in forward ight. This con guration provides great advantages that could prove useful for this mission. The rst thing is safety, the vehicle may glide like an airplane during an in- ight emergency and land at an airport in an airplane con guration. Another added plus is the proven technology and existing FAA approval for operation. However, this vehicle also has drawbacks that disqualify it from being the correct con guration for this RFP. The rst one is the compactness, similar to the tilt-wing, this vehicle requires a large wingspan to provide the desired lift in forward ight. The second is the lack of optimization for either stage of ight; while prop-rotors are a fantastic feat of engineering, they fall short as a propeller or as a rotor. As this mission is dominated by one of the modes, not being able to optimize for one of them makes the design fall short of expectations. The third short coming is its mechanical complexity, tilt-rotors are notorious for its development and operational complexity lifting major safety concerns. This was shown during the development of the V-22 Osprey. This complex mechanism also provides more failure points during operation and during storage of the vehicle. The last major issue with the con guration for this mission is the tendency to have high disk loading. Due to its smaller rotors, the disk loading on these vehicles is immense: a very dangerous factor for operators on the ground. This can cause problems to the navigation and orientation systems of landing on a dusty environment and presents a hazard to any animal or human

near the helicopter during operation.

3. Co-axial Helicopter

The coaxial helicopter is a concept that has been tested over the years by multiple players around the world. Co-axial helicopters are best known for its great lifting capabilities and the eradication of the need for a tail rotor. This con guration combines two conventional rotors stacked on top of each other for enhanced hover performance. These rotors are counter rotating, therefore the rotary system produces a net of zero torque on the vehicle, e ectively eliminating the tail rotor. For this call for design of the RFP, the co-axial helicopter provides with great features. The most important one to highlight is its increased lifting capability as well as its enhanced hover performance. If desired, this con guration could also be thrust compounded so that the helicopter could achieve high speeds. An additional bene t of this design is the safety due to the lack of a tail rotor. With this design, the rotary system could be more compact and pose less risk to outsiders. The team initially had selected this con guration but decided to change early in the process. The reasons for this change revolve around the complexity of the system and the unnecessary lifting capability. The coaxial design has an extraordinarily complex hub system that not only provides more failure points but it also increases the drag of the vehicle exponentially. This increase of drag leads to the requirement of a pusher propeller which presents the same safety risks as the tail rotor. Overall, the coaxial is a good choice but it provides unnecessary capability and complexity for the design.

4. Single Main Rotor (SMR)

The single main rotor is the most proven design in the world of helicopters. The concept has been around since the birth of rotary wing vehicles and has been approved for ight even at the autonomous level. The Boeing Sheibel Camcopter is one of the rst of its kind by being the rst and only autonomous helicopter to receive authorisation by the EAA. The single main rotor presents many advantages for this RFP. The SMR design is the most mechanically simple out of all other con gurations with the least moving parts and simple rotor hub. In addition, pilots and aviation mechanics do not have to pursue more rigorous training to be able to operate the vehicle. This is an essential attribute when working in an environment where the personnel might be very limited, as it would be during a pandemic or natural disaster. In the safety aspect, this con guration also provides features that are attractive for the design. First its autorotative capability in case of an in- ight emergency. The second is the low level noise produced by the rotor during hover and in cruise, providing acoustic safety to those around the vehicle. And lastly its its tendency for a lower disk loading, this provides the vehicle with the versatility to operate in all terrains. The most important feature is the proven concept, the team believes that it will ease the path of certi cation and meet the 2025 deadline for operation.

C. Single Main Rotor Compounding

The selection of the conventional SMR con guration led the team to study possible compounding capabilities. The con gurations that were studied were non-compounding, wing on retreating side, wing on retreating side and pusher prop, wing on both sides of the fuselage and pusher prop, and pusher propeller only. For initial validation, a test using modi ed momentum theory was done using a 4 bladed 4,545 kg (10,000 lb) SMR with a rotor radius of 7.62 m (25 ft) as a baseline. To this design, the di erent compounding was added.

Table 6 Modi ed momentum theory power required for 200 knot cruise

Concept	Power Required	Shaft tilt (deg)
Baseline	623 kW (835 Hp)	6.52
Wing on retreating side	881 kW (1182.5 Hp)	25.68
Wing on retreating side and pusher prop	893 kW (1198 Hp)	0
Wing and pusher prop	1118 kW (1500 Hp)	0
Pusher prop	556 kW (746 Hp)	0

The results from Table 6 show that adding a thrust compounding option could provide a reduction of power required during cruise and a 0 degree shaft tilt. In addition, it permits the vehicle achieve higher speeds with the same amount of power because the power curve gets extended. Lift compounding was not a viable option because the advance ratio was not high enough leading to its addition to just be a drag increment.

Adding the thrust compounding pusher propeller provides a competitive advantage to the Garra from a performance and safety perspective. It will be shown in later sections that the Garra is able to accomplish the desired mission and

beyond by having the surplus of speed. Other helicopters have used this method in the past successfully, some to note are the Lockheed Chayenne and Piasecki X-49. Boeing is also adapting this con guration for the FARA competition with the Boeing Scout FARA entry. See gure 6 for visual on the con gurations.

(a) Lockheed Cheyenne (b) Piasecki X-29 (c) Boeing FARA Scout Concept

Fig. 6 Other Proven Concepts like The Garra

III. Vehicle Sizing

A. Methodology

Fig. 8 Flowchart of Sizing Code

The RFP lists two missions as reviewed in Sections I.A.3 and I.A.4 that the aircraft must complete: the Delivery and Logistics missions. For the design, it was necessary to determine the mission which would determine the size of the aircraft. To accomplish this, a sizing code was developed in house (see gure 7 for ow chart) in which each phase of both missions: hover, climb, cruise, and descent was modeled using modi ed momentum theory modi ed to determine the power required to complete each phase [14]. The phase of the mission that required the most power became the sizing segment of the mission and determined the power required that the aircraft must carry on board. The weights of the sub-components were estimated using the Aero Flight Dynamics Directorate (AFDD) equations, which take the power required and calculate the weights for each subgroup of the aircraft using statistical approximations [15]. The program then uses xed point iteration to converge on a Gross Takeo Weight (GTOW) and power

Fig. 7 Validation of Sizing Code Scripts

required for the aircraft. A mechanical design was assumed to start and fuel burn was tracked for each segment of both missions and incorporated into the GTOW prediction.

B. Code Validation

Since the code was developed in house by the team, it needed to be validated before any trade studies could be conducted. The hover and cruise scripts were validated against ight test data from the Robinson R-66 helicopter at a blade loading) •f = 0.102. The validation is depicted in Fig. 7. The output of the code showed good agreement with the R-66 ight test data with an over prediction of 8% near hover and 6% at higher advance ratios. However, in the interest of conservatism early in the design process, these deviations were deemed acceptable.

C. Package Deployment Time Trade Study

Through modi ed momentum theory calculations, it was revealed that the cruise phase would determine the power required to comply with the RFP missions. In addition, by calculating the block time requirements from the RFP, it quickly became clear that the aircraft would need to y faster for the delivery mission and the cruise speed would be a function of the time required to deploy the payload. From these two constraints, the delivery mission was used to size the vehicle, and if the aircraft was designed to complete the delivery mission, it would have the power and capabilities to complete the logistics mission. Furthermore, it necessitated careful drag estimation and design considerations to ensure high-speed ight as well as designing a payload deployment system that would operate quickly and safely. First, a preliminary trade study was conducted with the sizing code to determine how the power required and cruise speed varied with time required to deploy the payload. The results of this study are summarized in Table 7.

Table 7 Delivery Mission Package Deployment Trade Study

Minutes to Deploy the Payload	V_{cruise}	Delivery Mission Power Required
0.5	60.3 m/s (117 kts)	57.36 kW (76.9 Hp)
1	62.5 m/s (121 kts)	61.6 kW (82.6 Hp)
2	67.5 m/s (131 kts)	73.6 kW (98.7 Hp)
3	73.6 m/s (143 kts)	90.8 kW (121.8 Hp)
4	80.6 m/s (157 kts)	115.0 kW (154.2 Hp)

D. Main Rotor Trade Studies

The sizing code developed in Section III was also used for the initial trade studies of the main rotor which have been summarized in Fig. 9(a-b).

1. Main Rotor Trade Study: Varying Disk Loading

First, the diameter was varied with each run of the sizing script and plotted against the nal converged GTOW. The main rotor diameter was varied between 2.0 m (6.56 ft) and 3.4 m (11.16 ft). 3.4 m was set as the max main rotor diameter for the initial sizing studies as this was a conservative estimate for how large it could grow to t within the allotted 4.57 m x 4.57 m (15 ft x 15 ft) footprint. This assumed a main rotor to tail rotor diameter ratio of 4.7, allotting approximately 15.2 cm (6.0 in) for clearance between the rotors, and providing 30 cm (11.8 in) for the pusher propeller. As the vehicle is intended for operation during the aftermath of a natural disaster or during a future pandemic, the team wanted to design so that the vehicle could also deploy the payload safely on unprepared surfaces. Therefore, there was an interest in minimizing the downwash from the main rotor to ease ground operations. Since downwash is directly proportional to disk loading, the variable rotor diameter in this trade study was represented in terms of disk loading in Fig. 9 (a). In the same gure, the sizing code was carried out for two through six blade rotor con gurations. The study revealed little increase in GTOW as disk loading decreased from 431 N/m^2 (9 lb/ft^2) to 239 N/m^2 (5 lb/ft^2) for each con guration, this was a desirable feature for both safety and hover performance.

2. Main Rotor Trade Study: Varying Aspect Ratio

A second trade study was conducted in which the aspect ratio was varied with a xed main rotor diameter of 3.4 m (11.2 ft). In this study, varying aspect ratio was represented as blade loading to understand how GTOW could be minimized while maintaining a comfortable main rotor stall margin below) = 0.12. This is because varying the aspect ratio is equivalent to changing the rotor solidity as $\frac{\#_1}{c}$. The number of blades was also varied over separate

(a) Main Rotor Trade Study 1: Vary Disk Loading (b) Main Rotor Trade Study 2: Vary AR

Fig. 9 Main Rotor Trade Studies

runs. Solidity was not held constant given the same aspect ratio and a di erent number of blades. The results of this study can be seen in Fig. 9(b) where the dotted lines are isolines of constant aspect ratio. The study showed that while two blades at an aspect ratio of 12 to 14 could work for the design, three blades at an higher aspect ratio of 16 to 18 seemed more desirable. Given that the cruise segment dominates the sizing of the aircraft, a lower aspect ratio produces less drag on the main rotor in forward ight. Three blades over two blades would result in lower vibratory hub forces, which is bene cial for the protection and safety of the payload. Therefore, the main rotor was designed around a three bladed con guration with an aspect ratio of 16.

For these initial studies, v_{tip} was xed at 213 m/s (700 ft/s). Initial studies showed that reducing $\Omega_b v$ could result in a lower power required. Figure of Merit was set to 0.75, and e ective at plate area to 0.25 m^2 (2.69 ft^2) which was critical to the maximum forward ight speed. Section VI describes how the drag was calculated for the vehicle.

Table 8 Geometries Varied in Main Rotor Design

Geometry Type	Values
Airfoil Combinations	NACA0012 VR12/VR15, OA212/OA209 SSCA1012r8/SSCA09 RC4-10/RC3-8
Airfoil Transition	0.3R to 0.8R
Inboard Taper	1:1 to 2.5:1
Outboard Taper	1:1 to 2.5:1
Taper Transition	0.3R to 0.8R
Inboard Twist	-2^o/span to -20^o/span
Outboard Twist	-2^o/span to -10^o/span
Twist Transition	0.3R to 0.8R

IV. Main Rotor Design

A. Main Rotor Aerodynamic Design

There are a lot of geometric variables that comprise the main rotor of a helicopter. In lieu of trying to vary one at a time, the team decided to use a multi-design objective approach and generate a Pareto plot which depicts how di erent designs perform in hover vs forward ight," vs ! • $_4$.

1. Pareto Plot Code Setup

To generate this Pareto plot, over 30,000 rotor con gurations were analyzed. To accomplish this, an in-house blade element momentum theory (BEMT) script was developed for both hover and in forward ight. The BEMT hover case accounted for nonuniform in ow while the forward ight case implemented Drees' Linear In ow model [2]. The script was validated against the same R-66 ight test data used in Section III. Both cases used airfoil tables which were generated over a range of Reynolds numbers from 250,000 to 2,000,000 using XFoil at $\pm_{2A8C} = 9$. Interpolation was used to calculate the appropriate airfoil $_3$, and $_<$ at a given Reynolds number. If the Reynolds number at a certain point on the disk was outside of the range 250,000 to 2,000,000, the airfoil data at the closest bound was used. The script also took main rotor trim into account and converged to a collective value to ensure an appropriate number.

To generate the rotor con gurations, airfoils, taper ratio, and twist and their respective transition regions were considered. All con gurations used two airfoils except for a baseline design using only the NACA0012. Additionally,

all designs took advantage of a bilinear twist and taper. To maximize aerodynamic e ciency, the inboard airfoils were selected to have high c_{l_3} values at Mach Numbers 0.1-0.4 as well as $c_{d_0} \leq 0.02$ over the angle of attack (AoA's) of interest. Higher lift rotorcraft airfoils with higher coe cient of moments were not considered for this aircraft in the interest of geometric simplicity as well as ease-of-maintenance as such a choice reduces loading on the swashplate. Outboard airfoils were selected to be within the same family as the inboard airfoil and were chosen for their higher M_{dd} - greater than 0.80. Mixing airfoil families for the inboard and outboard airfoils was not considered in this design to improve blending of the inboard and outboard sections and for manufacturing simplicity. The airfoils considered as well as the range over which the geometric parameters were varied may be found summarized in Table 8.

The results of the Pareto plot are in Fig. 10. The main rotor airfoil con guration prioritized M_{dd} without taking a large penalty in Figure of Merit. Ultimately, a point was chosen around c/R_4 of 4.5 as higher ratios resulted a large decrease in hover performance with a marginal increase in forward ight performance. The speci c main rotor geometry at this point was selected and is shown in Fig. 11. The geometry and performance parameters may be found in Fig 9. The nal E_{C8} of 219 m/s (720 ft/s) was selected to keep M_{C8} below 0.85 and M_{dd} below 0.37 at a sprint speed of 79.7 m/s (155 kts). Once the geometry was selected, several tip variations were considered. Ultimately, a tip sweep of 20° was used over the nal 5% of the blade to keep M_{C8} below 0.80 at cruise speed and allow margin for a sprint speed without a severe drag divergence penalty. Using a thinner airfoil like the OA206 for the tip was considered, however it was ultimately deemed unnecessary as drag from the fuselage limited forward ight speed where the higher M_{33} that the OA206 provided was not needed and would only complicate the manufacturing of the blade. A 20° anhedral was added as [3] showed an improvement of 1 dB in noise of the aircraft in hover at a C_T value of 0.011. [3] showed insigni cant improvement in greater droop angles so 20° was chosen to prevent further complications in manufacturing. The reduction in noise due to the anhedral presented an enhanced safety feature for those operating near the vehicle. A 20% root cutout was selected to provide room for the hub as well as mitigate unproductive sections of the blade in cruise.

Fig. 10 Main Rotor Pareto Plot, over 30,000 Con gurations

Table 9 Main Rotor Design Details, v_{cruise} = 69.4 m/s (135 kts)

Geometry Parameter	Value	Performance Parameter	Value
No. Blades	3	Disk Loading	277 N/m^2 (5.78 lb/ft^2)
Aspect Ratio	16	Power Loading	46.6 N/kW (7.76 lb/hp)
Radius	1.70 m (5.42 ft)	$I_Z \cdot 2$	0.112
V_{tip}	219 m/s (720 ft/s)	FM	0.807
Solidity, 2	0.0597	S_{tip} @ v_{cruise}	0.816
Shaft Tilt	0°	$R \cdot J_e$ @ v_{cruise}	4.54
Main Rotor RPM	1230 rpm	- @ v_{cruise}	0.317

B. Main Rotor Blade Design

The design of the blade was done with safety and reliability in mind using a D-spar made of S-Glass to resist the inertial forces and bending loads. A tungsten weight was placed at the leading edge to locate the center of gravity of the blade near the quarter-chord and eliminate the potential for aeroelastic utter. Rohacell 51 foam was chosen to ll the D-spar. The foam both constrains the tungsten weight chord-wise and maintains the shape of the front quarter of the blade. This foam was chosen as it is light and inexpensive. Rohacell 31 was chosen to maintain the shape of the aft three-quarters of the blade and was selected as it is also light and inexpensive. A less dense foam was chosen for the aft portion of the blade to promote a forward center of gravity.

Fig. 11 Main Rotor Blade Geometry

Fig. 12 Blade Structural Composition

Four 45^{o} plies of T300 graphite infused with epoxy make up the skin and majority of the torsional sti ness of the blade. The same material is used at the trailing edge to form a wedge and maintain the shape of the airfoil. While the vehicle will only y in a VFR environment according to the RFP, there is always the chance of a lightning strike on the tarmac. Lightning strikes can damage the skin of the blade. To mitigate this, the blade is covered in a thin copper mesh to ground the electrical load. Finally, a stainless steel strip is placed on the leading edge to prevent erosion of the front of the blade. A deicing strip was considered, but since the RFP speci cally asks for an aircraft that will y in ISA +20 and in a good visual environment, it was deemed an unnecessary component.

C. Main Rotor Dynamics

The Fan Plot is shown in Fig. 13. The lines show the 1st through 4th apping frequencies for RPM values from zero to 120% of the operating RPM. The apping frequencies were calculated by solving for the eigenvalues of the mass-sti ness system matrix using 20 nodes evenly spaced along the blades and assuming uniform sti ness. The rst ap frequency is 1.08/rev at the operating RPM and it can be seen that the second and third ap frequencies are well spaced from the n/rev frequencies. The rotor structural design is in the early stages therfore there is little information on the torsional and in-plane sti ness of the rotor, due to this the fan plot does not present the torsional and lag modes.

Fig. 13 Fan Plot

D. Main Rotor Hub

Fig. 14 Main rotor swash plate

The main rotor hub selection was of utmost importance due to the high speed forward ight for this size of aircraft. A process similar to the con guration selection was done to pick the type. For this process, the Pugh matrix analysis was made to compare four types of hubs: Articulated, Teetering, Hingeless, and Bearingless. For this study, the articulated rotor was picked as the control group and the same -3 to 3 integer scale was used for the matrix. The results of this are shown in table 10. This analysis shows that a hingeless rotor could provide the best control authority, stability, and external disturbance rejection. The hingeless rotor also provides a mechanically simple and compact design, this provides an enhanced safety feature as it shows fewer points of failure. In addition, the hub integrates a lag damper to avoid ground resonance (not shown).

Table 10 Design Drivers For Rotor Hub

Design Driver	Articulated	Teetering	Hingeless	Bearingless
Control	0	-1	2	1
Stability	0	0	1	1
Gust rejection	0	0	1	1
Vibrations	0	1	0	0
Complexity	0	1	1	0
Sum	0	1	5	3

V. Tail Rotor, Pusher Propeller and Empennage Design

This section details the design of the empennage of the aircraft as well as the aft rotors. First, several empennage con gurations are considered. Once one is selected, this design choice as well as the need for a low drag design drives the speci cs of the rear rotors and the empennage geometry.

A. Empennage Con guration Selection

Given the need of a pusher propeller section to achieve the speeds required to complete the delivery mission as well as a tail rotor to account for anti-torque in hover, several empennage designs were considered.

1. NOTAR with Pusher Propeller

A NOTAR tail con guration is one such that no tail rotor is used to provide the anti-torque for the the aircraft in hover. Instead, an engine-driven variable pitch fan controls air ow through the tail boom, which is then blown out the side to provide a moment arm which counteracts the main rotor torque. This was initially a very attractive design for the Garra because of the emphasis placed on safety in the RFP. The use of a NOTAR design meant no rotating tail rotor in

the rear of the aircraft which represents a hazard especially if the aircraft is to deploy a payload in an unprepared or uncontrolled environment. This seemed feasible when a hybrid design was being considered where the pusher propeller could be electric. However, the propulsion team determined that a hybrid propulsion system would be too heavy due to the generator required to operate the pusher propeller. Therefore, it was determined that using a compressor to drive the NOTAR con guration and a shaft to drive an aft pusher propeller would undermine the mechanical simplicity of our design.

2. Ducted Pusher Propeller with Variable Guide Vanes

A ducted pusher propeller design with variable guide vanes is one in which a rotor is placed in a duct at the rear of the aircraft and guide vanes may be used to vector its thrust to get a balance of anti torque and forward thrust. This design was employed by Piasecki's X-49. The design appeared attractive because it could eliminate one of the two rotors which saves weight and mechanical complexity. It also was attractive from a safety perspective as it would shroud the rear rotor. However, the guide vanes added their own source of complexity and it was found to be too di cult to trim the aircraft to achieve a stable hover, which is of utmost importance during package deployment.

3. Swiveling Tail Rotor

A swiveling tail rotor is one in which the tail rotor can be rotated to function as either a tail rotor or a pusher propeller or both depending on what is needed in a speci c ight condition. This design was highly considered when the option for a hybrid aircraft existed where the rear rotors could be electric. However, once a fully mechanical design was determined by the propulsion sub-team, the mechanical complexity of this design increased signi cantly. It was found that the trade o required to design a rotor that would function well both as a tail rotor and as a propeller was too high to make this design work. The propeller had to be highly twisted and optimized to t the power and size constraints of the aircraft which would perform poorly as a tail rotor with a crosswind from the 270° azimuth. From a safety point of view, this design also provided poor controllability when changing ight modes rapidly.

4. Fenestron with Pusher Propeller

A fenestron is a design in which the tail rotor is shrouded in a small duct. This gives the rotor additional aerodynamic e ciency and gets points for safety over the traditional open tail rotor, but it comes at the cost of weight due to the shroud and large vertical stabilizer required. This was desirable at rst when a hybrid aircraft was being considered. However, once the decision to go entirely mechanical was made, this design proved too heavy structurally.

5. Open Tail Rotor and Pusher Propeller

An open tail rotor and pusher propeller design is one in which the tail rotor is a xed to the tail boom or one of the rear stabilizers and the propeller is placed at the back, typically in line with the tail rotor shaft. Since there are two separate rotors, there are two required swashplates, and thus this design is not as mechanically simple as some of the other designs. However, given the aircraft was to be entirely mechanical, this design did result in the lightest con guration, and it was a proven design on the Lockheed Cheyenne which was reassuring considering how quickly the Garra needed to be elded. The design could be deemed less safe than some of the other designs as none of the rotors are shrouded. However, considering the vertical stabilizer could be designed to prevent a rotor ground strike, package loading would occur with rotors stopped, and package delivery would occur from the air, the lack of a rear rotor shroud was deemed acceptable.

Considering all of these points, an open tail rotor and pusher propeller con guration were chosen for the Garra due to its simplistic design and adaptability to the design needs.

B. Tail Rotor Design

The primary design drivers for the tail rotor of the Garra were: (1) low drag in forward ight, (2) good gust rejection for a stable hover during package deployment, and (3) system safety. A trade study was conducted to determine the crosswind rejection from the 270° azimuth. This is where a tail rotor can be most susceptible to losing control as a wind from this direction simulates decent and can cause the rotor to get into Vortex

Table 11 Tail Rotor Radius vs Crosswind Rejection Trade Study

Tail Rotor Radius	Crosswind at which Yaw Authority is Lost
0.45 m (1.48 ft)	9.0 m/s (17.5 kts)
0.40 m (1.31 ft)	10.2 m/s (19.8 kts)
0.35 m (1.15 ft)	11.6 m/s (22.6 kts)
0.30 m (0.98 ft)	13.5 m/s (26.3 kts)
0.25 m (0.82 ft)	16.3 m/s (31.6 kts)
0.20 m (0.66 ft)	20.3 m/s (39.5 kts)

Ring State (VRS). Table 11 details the crosswind at which a tail rotor will exhibit VRS based on its radius given the Garra's main rotor torque and tail boom geometry. Next, a series of trade studies were conducted to determine how varying radius, aspect ratio, twist, and airfoil affected the power of the tail rotor. The results of these trade studies are summarized in Fig. 15(a-d). It should be noted that Fig. 15(a-c) used modified momentum theory while Fig. 15(d) use the BEMT hover code used in Section IV.A

(a) Tail Rotor Trade Study 1: Vary Radius

(b) Tail Rotor Trade Study 2: Vary AR

(c) Tail Rotor Trade Study 3: Vary

(d) Tail Rotor Trade Study 4: Vary Twist and Airfoil

Fig. 15 Tail Rotor Trade Studies

1. Tail Rotor: Radius

Fig. 15(a) illustrates the variation of the power of the tail rotor relative to the main rotor as a function of the tail rotor radius and the number of blades. Despite there being a power minimum at 0.35 m (1.15 ft), a tail rotor radius of 0.3 m (0.98 ft) was chosen for a crosswind rejection improvement of 16.3% and a slight reduction of drag in forward flight. This resulted in a main rotor diameter to tail rotor diameter ratio of 5.67. This is well within the typical range of values for single main rotor helicopters [4].

2. Tail Rotor: Aspect Ratio and Tip Speed

Fig. 15(b) depicts how solidity varies with aspect ratio. An aspect ratio of 4.5 was selected for the two bladed configuration to balance low drag in forward flight as well as falling within the typical region of tail rotor solidities from 0.14 to 0.19 [4]. Fig. 15(c) shows how varies with Convention says that a designer should try to match the of the tail rotor to the main rotor. However, that would have put the of the tail rotor at 0.83, well above the of a 10-12% thickness airfoil. Therefore, of 187 m/s (613 ft/s) was chosen to get an of 0.75 at A slight leading edge tip sweep was later added to reduce to 0.71 to allow for margin for the aircraft to reach its sprint speed without compressibility penalties at the tip of the tail rotor.

3. Tail Rotor: Airfoil and Twist

Tail rotor airfoil and twist were determined using the in-house BEMT hover code described in Section IV.A. While a higher twist would increase the Figure of Merit of the rotor, allowing it to perform more e ciently in hover, a higher twist is punished in forward ight where the tail rotor is being dragged through the air rather than providing the anti torque. Fig 15(d) demonstrates how Figure of Merit varied with twist and airfoil selection. From this plot, the RC410 at a -9^{o}/span was chosen for the tail rotor.

4. Tail Rotor: Placement, Type, and Direction

The tail rotor blockage proved to be an important consideration for this design and a driver for the type of tail rotor and its placement. Fig. 16 depicts the three positions considered for the tail rotor.

The tail boom could not be the thin shaft that is typical of single main rotor helicopters as an aerodynamic tail was needed to help mitigate pressure drag in forward ight. The tail boom instead took the shape of a smooth curve from the main fuselage to the pusher prop. Due to this, placing the tail rotor at position 1 in Fig. 16 which would be a conventional distance from the main rotor would result in high tail rotor blockage and unnecessary power losses. Also, it would get in the way of loading the longer package with the tail rotor right at an operator's chest. However, the tail rotor could not be placed too far aft (to the right of position 2 in Fig. 16) on the aircraft where the tail boom was thinner as there was a propeller there that needed room for apping. To provide room for the propeller and the vertical stabilizer, the tail boom would block large portions of the tail rotor disk. The solution to this problem to prevent unnecessary power losses was to select a pusher tail rotor that is located at the tip of the horizontal stabilizer [4]. This was used to balance distance the tail rotor needs to be from the fuselage to keep the tail rotor blockage factor below 1.15.

Fig. 16 Tail Rotor Placement

An aft-at-the-top direction was chosen for the tail rotor as this direction provides greater yaw control in variable hover conditions [4]. This direction worked well with the gear design in that the direction of the pusher propeller was in such as way that the torque generated by it could be mitigated partially by the vertical stabilizer which doubled as counter-torque for the main rotor in forward ight.

The team considered placing the tail rotor above the tail boom in position 3 in Fig. 16 as is common in a single main rotor helicopter. However, in the Garra's con guration, that would unnecessarily increase tail rotor blockage. Also, it would prevent the vertical stabilizer from doubling as a prop guard to prevent a ground strike of the pusher propeller, requiring an additional structure beneath the tail boom for this safety feature. The nal tail rotor geometry and performance details can be found in Table 12 and Fig. 17.

C. Tail Rotor Swashplate

The Garra's tail rotor swashplate was designed with the compactness and mechanical simplicity design as the main drivers. The goal was to increase safety by reducing the number of failure points and selecting an industry proven design. The system consists of a single hub plate with a bearing and a torque tube attachment for each rotor following a hingeless design. The torque tube contains a pitch horn that connects to the slider for pitch control. This mechanism permits the blades to achieve positive and negative collective angles e ectively increasing the Garra's maneuverability in hover and in forward ight. Figure 18 shows the detailed design for the hub.

Fig. 18 Tail Rotor Swashplate

Table 12 Tail Rotor Design Details

Geometry Parameter	Value	Performance Parameter	Value
No. Blades	2	P_{TR}/P_{MR} in Hover	8.2%
Aspect Ratio	4.5	FM in Hover	0.768
Radius	0.30 m (0.98 ft)	$I_Z \cdot 2$ in Hover	0.0814
V_{tip}	187 m/s (613 ft/s)	S_{tip} @ v_{cruise}	0.727
Solidity, 2	0.141	Crosswind Rejection	13.5 m/s (26 kts)
Airfoil	RC410		
Tail Rotor RPM	5952 rpm		

Fig. 17 Tail Rotor Blade Geometry

D. Vertical Stabilizer Design

The vertical stabilizer was sized with two purposes in mind: provide the anti-torque in forward flight and to prevent a prop strike of the pusher propeller during a flare in landing for an improved safety trait. Given these goals, the span was determined such that it would prevent a prop strike in landings up to a flare angle of 15^o. Since there was no gearbox that needed to be in the tail rotor, a NACA 4412 airfoil was chosen for the vertical stabilizer. The mean chord was then determined based on this airfoil at an angle of incidence of 0^o such that enough lift was generated at v_{cruise} to provide the anti-torque for the aircraft. The placement of the vertical stabilizer on the bottom also aids in counteracting the moment created by the pusher propeller on the aircraft due to the torque required to drive it. Details of the vertical stabilizer design are summarized in Table 13. The placement of the vertical stabilizer was chosen to provide a clearance of 15^o for flapping of the pusher propeller blade and to remain out of the wake of the tail rotor in hover. A sweep of 30^o was added to the vertical stabilizer to reduce drag in forward flight.

Table 13 Empennage Geometry

Parameter	Vertical Stabilizer	Horizontal Stabilizer
Airfoil	NACA 4412	NACA 4412
Area	0.082 m^2 (0.883 ft^2)	0.288 m^2 (3.10 ft^2)
Span	0.373 m (1.06 ft)	0.686 m (2.25 ft)
Mean Chord	0.22 m (0.72 ft)	0.420 m (1.38 ft)
Sweep Angle	30^o	0^o
Taper Ratio	1.5	1
Angle of Incidence	0^o	0^o

E. Horizontal Stabilizer Design

1. Horizontal Stabilizer Sizing

The horizontal stabilizer's primary design driver was to act as a high lift-over-drag airfoil that would counteract the moment on the fuselage in forward ight without a large drag penalty. Since only a tail rotor shaft needed to be housed in the horizontal stabilizer and not a gearbox, the NACA 4412 was also the airfoil chosen. The span of the horizontal stabilizer was determined to keep the wake of the tail rotor out of the tip path plane of the pusher propeller in forward ight to improve the e ciency of the pusher propeller. Using CFD cases run on the fuselage in forward ight, the pitching moment that the horizontal stabilizer was required to counteract was determined to be 127 N-m (93.7 lb-ft). Based on this value, the airfoil and chord were determined to achieve the surface area required to generate the negative lift needed to counteract the pitching moment. Details of the vertical stabilizer design are summarized in Table 13.

Fig. 19 Empennage Structure

2. Horizontal Stabilizer Location and Sweep

The location of the horizontal stabilizer was entirely dependent on the tail rotor location. Because the tail boom needed to be more aerodynamic than the traditional helicopter to reduce drag in forward ight, the tail rotor needed to be moved away from the surface of the tail boom. The horizontal stabilizer was used to accomplish this to prevent unnecessary power losses, similar to the Lockheed Cheyenne. A sweep was considered being added to the horizontal stabilizer to reduce drag in forward ight, but the mechanical and structural weight required to achieve this - two additional gearboxes and an angled tail rotor shaft - outweighed the bene ts of the drag reduction. The tail rotor shaft will be supported by bearings embedded in the ribs of the horizontal stabilizer to prevent dynamic issues as shown in Fig. 19.

F. Pusher Propeller Design

The main goal for the design pusher propeller was an e cient geometry that could deliver the required thrust needed in forward ight. To accomplish this, rst, a BEMT code was developed in-house to calculate the performance characteristics of a propeller of a given geometry. The code was validated against results from [5].

1. Propeller Trade Studies

Once validated, the code was used to conduct several trade studies to understand how varying non-dimensional parameters of a propeller a ected the thrust coe cient $)_{?A>?}$, power coe cient %, and the e ciency[?. These trade studies are summarized in the plots in Fig. 20(a-c). It was found that $)_{?A>?}$ and needed to be carefully balanced to get a twist that would give an optimal[? of greater than 0.80. Choosing an optimal[? also kept % down to improve propeller performance. Later, the code was adapted to calculate collective trim as well, as the required pusher propeller would need to be variable pitch to trim out thrust during the package deployment.

(a) Pusher Propeller Trade Study 1 vs

(b) Pusher Propeller Trade Study 2 vs

(c) Pusher Propeller Trade Study 3 vs

Fig. 20 Pusher Propeller Trade Studies

2. Propeller Aerodynamic Design

The Clark-Y airfoil was chosen for the pusher propeller design. The Clark-Y is an industry standard propeller airfoil with a high maximum Lift-to-Drag of 115. This airfoil has a relatively high moment coefficient, but since the inflow does not vary around the propeller disk as much as it does on a helicopter main rotor and because propeller blades are shorter and more rigid than a main rotor, this was deemed acceptable and safe. From here, was the first value that was chosen. It was known that a relatively large amount of thrust would be required of the propeller from modified theory given the drag on the aircraft, therefore to keep down, was maximized such that at was at 0.75, just below the of the Clark-Y airfoil. This resulted in a of 242 m/s (794 ft/s). From there, using the required thrust which resulted from the flat plate area calculations in Section VI, the diameter was chosen according to Fig. 20(a-c) to get an advance ratio, thrust coefficient, and twist that gave a reasonable efficiency greater than 0.80. Once this occurred, aspect ratio and number of blades were varied until an appropriate and was found. Then, diameter was changed again to its final value of 0.9 m (2.96 ft) so that the propeller did not exceed the maximum torque available from the engines. A flower petal-like taper was added to the propeller to increase chord and thus lift in the driving section of the blade and decrease chord and thus drag in the driven section of the blade. The final geometry and performance details of the propeller can be found summarized in Table 14 and visualized in Fig. 21.

3. Need for a Rear Gear Box

Attempts were made to match the rotations per minute of both the tail rotor and the pusher propeller so that a rear gear box would not be required. However, lowering the of the tail rotor any further could result in stall in some crosswind cases from the 90 azimuth, and raising the diameter resulted in worse crosswind rejection from the 270 azimuth. Similar was true about the propeller in that raising the of the pusher propeller resulted in drag due to compressibility effects at the tip, and lowering the diameter resulted in too high a power at .

Table 14 Pusher Propeller Design Details

Geometry Parameter	Value	Performance Parameter	Value
No. Blades	4	Power	54.0 kW (72.4 Hp)
Aspect Ratio	6	Torque @ v_{cruise}	100.4 N-m (74.0 lb-ft)
Radius	0.45 m (1.48 ft)	$I_{Z_{Vrop}}$	0.137
V_{tip}	242 m/s (794 ft/s)	P @ v_{cruise}	0.883
Solidity, 2	0.212	($_p$ @ v_{cruise}	0.803
Airfoil	Clark Y	S_{tip} @ v_{cruise}	0.758
Propeller RPM	5135 rpm	$_{75}$ @ v_{cruise}	31.5^o

Fig. 21 Pusher Propeller Blade Geometry

G. Pusher Propeller Variable Pitch Mechanism

The pusher propeller utilizes a sliding variable pitch mechanism (shown in gure??) to achieve the pitch desired during operation. This mechanism is actuated using an electro-hydrostatic actuator. The slider permits for a uniform change in pitch across all the blades of the construction. In addition, this provides a very mechanically simple way of providing variable pitch, the mechanism is easy to maintain and similar to the one seen on xed wing aircraft. This feature makes it accessible to any aviation mechanic, a key feature when the capacity of workers is limited.

VI. Drag Estimation

Since Garra's missions are dominated by forward ight and it is to be evaluated by the performance metric de ned in Eq. 1, an accurate estimate of how fast the aircraft could y was of the utmost importance.

Fig. 22 Pusher Propeller Swashplate

$$Z = \frac{E_{2AD8B4\ ?0H;>03}}{)\$,} \tag{1}$$

A. Flat Plate Area Estimation

To calculate the equivalent at plate area of the fuselage, an in-house developed code was used. The fuselage was discretized into a sections that are modeled as cylinders. From there, the following equations are used to calculate the Form Factor, and Skin Friction Coe cient, of each section.

$$= \frac{-F4C}{} = 1 \cdot 1•513•;^{\frac{3}{2}} \cdot 7^{1}3•;^{3} \quad (2)$$

$$\bar{c} \frac{1•328}{\sqrt{'4}}{}^{1}1 \quad ?^{} \cdot \frac{0•455}{{}^{1}\log_{10}'4 \; {}^{1}2•58{}^{1}1 \cdot 0•144'' \; 2o0•65}{}^{1}?^{} \quad (3)$$

Here, 3•; is the neness ratio of the fuselage is the percentage of turbulent ow over the aircraft's skin is assumed to be 0.65 because a well designed composite aircraft may have laminar ow up to 35% of the surface of the aircraft [6]. Flat plate area for each section is then calculated using

$$= \; {}_{0}(\; = \sum_{=1} \; \; (\quad (4)$$

where(is the wetted area of an individual section obtained from CAD, and is an interference factor that accounts for drag caused by attachment points of components to the main airframe. All three of the above equations are obtained from Hoerner's Fluid Dynamic Drag [7]. The results of this estimation may be found in Table 15.

The e ective at plate area to main rotor area, is quite high for this design, 0.309, compared to other helicopters. This is due to the footprint that the aircraft had to t in relative to the size of the payload. Given that it is relatively high, a study on the hover download is conducted in the next section.

Table 15 Flat Plate Area Drag Estimation

Component	f (m^2)	f (ft^2)	f/A_{MR}	% of Total
Fuselage	0.143	1.54	0.0158	57.6
Rotor Hub and Cowling	0.067	0.72	0.00738	27.0
Skid Landing Gear	0.019	0.20	0.00209	7.7
Horizontal Stabilizer	0.0058	0.062	0.00064	2.4
Vertical Stabilizer	0.0025	0.027	0.00027	1.0
Tail Rotor	0.0061	0.066	0.00067	2.4
Exhaust	0.0015	0.016	0.00017	0.6
Miscellaneous	0.0033	0.035	0.00036	1.3
Total	0.248	2.67	0.0273	100
Additional 10%	0.273	2.94	0.0301	

B. Hover Download Estimation

Given that the main rotor is small relative to the size of the payload and thus the fuselage compared to the average helicopter, a study was also done to determine the download on the fuselage in hover to ensure adequate power margin.

$$C= \; {}^{1} \; \; \;

would cause in cruise. Therefore, a main rotor diameter of 3.4 m (11.2 ft) was kept. The relatively high hover download of 14.7% of the GTOW was accounted for in the design and the performance plots.

Table 16 Hover Download Trade Study

Rotor Diameter	Hover Download [% GTOW]	R• J e	I Z • 2
3.2 m (10.5 ft)	16.6%	5.09	0.125
3.4 m (11.2 ft)	14.7%	4.54	0.112
3.6 m (11.8 ft)	13.1%	3.98	0.099
3.8 m (12.4 ft)	11.8%	3.57	0.088

VII. Vehicle Performance

A. Vehicle Power Usage

Figure 23 shows a summary of the vehicle performance metrics. The Garra can takeo at a weight of up to 313 kg (690 lb) even though to complete the missions and carry the fuel and payload it must be loaded to 255.7 kg (565.9 lb) for the logistics mission (25.7 kg (56.7 lb) fuel and 50 kg (110.23 lb) payload) and 247.3 kg (545.41 lb) for the delivery mission (17.3 kg (38.1 lb) of fuel and 50 kg (110.23 lb) payload).

(a) Vehicle Power Curve (b) Vehicle Power Curve at Di erent Flight Conditions

Fig. 23 Garra Power Curves

B. Segment By Segment Performance

C. Performance Metric Calculation

The performance metric is de ned to be the productivity of the vehicle which can be calculated as [Payload*blockSpeed]/GTOW.Th performance metric was calculated for both missions (table 18 and table 17). With a cruise speed of 135 knots, a climb-descent speed of 59.6 knots, and taking into account time for hover and package load-unload the block time for the logistics mission is 63.4 minutes (3802 s) and the block time for the delivery mission is 27.4 minutes (1642 s). The gross takeo weight of the vehicle for these two missions at 135 knots are 255.7 kg (565.9 lb) and 247.3 kg (545.2 lb) respectively. Therefore, the productivity for the missions at 135 knots are:

$$Z_{I>68BC82B} = {}^{1}50{:}6 \quad 200{:}< \bullet 3802B \quad 1000< \bullet :< {}^{o}\bullet 255\bullet 7{:}6 = 10\bullet 3< \bullet B^{1}20\bullet 0{:}CB \tag{6}$$

$$Z_{4;8E4AH} = {}^{1}50{:}6 \quad 50{:}< \bullet 1642B \quad 1000< \bullet :< {}^{o}\bullet 247\bullet 3{:}6 = 6\bullet 16< \bullet B^{1}12\bullet 0{:}CB \tag{7}$$

Table 17 Segment by Segment Vehicle Performance, Logistics Mission

Logistics Mission	Flight Speed	Power Required	Energy Consumed	Time
Load Packages	N/A	0 W (0 Hp)	0 kW-hr (0 BTU)	5 min
Warm Up	N/A	4.50 kW (6.03 Hp)	0.38 kW-hr (1280 BTU)	5 min
Take O HIGE	N/A	53.9 kW (72.3 Hp)	1.80 kW-hr (6130 BTU)	2 min
Climb to 1350m ASL	30.7 m/s (59.6 kts)	33.7 kW (45.2 Hp)	0.19 kW-hr (633 BTU)	0.33 min
Level Cruise	69.5 m/s (135 kts)	73.5 kW (98.6 Hp)	58.4 kW-hr (199 kBTU)	47.7 min
Descend to 1200m ASL	30.7 m/s (59.6 kts)	31.5 kW (42.2 Hp)	0.173 kW-hr (591 BTU)	0.33 min
Land HOGE	N/A	49.0 kW (65.7 Hp)	0.82 kW-hr (2786 BTU)	1 min
Unload Payload	N/A	48.7 kW (65.3 Hp)	1.62 kW-hr (5540 BTU)	2 min
			Block Total:	63.36 min

Table 18 Segment by Segment Vehicle Performance, Delivery Mission

Delivery Mission	Flight Speed	Power Required	Energy Consumed	Time
LEG 1				
Load Package	N/A	0 W (0 Hp)	0 kW-hr (0 BTU)	5 min
Warm Up	N/A	4.50 kW (6.03 Hp)	0.38 kW-hr (1280 BTU)	5 min
Take O HIGE	N/A	51.2 kW (68.7 Hp)	1.71 kW-hr (5827 BTU)	2 min
Climb to 1350m ASL	30.7 m/s (59.6 kts)	33.1 kW (44.4 Hp)	0.182 kW-hr (621 BTU)	0.33 min
Level Cruise	69.5 m/s (135 kts)	73.3 kW (98.3 Hp)	14.3 kW-hr (48.8 kBTU)	11.7 min
Descend to 1200m ASL	30.7 m/s (59.6 kts)	32.3 kW (43.2 Hp)	0.18 kW-hr (605 BTU)	0.33 min
Land HOGE	N/A	49.8 kW (66.8 Hp)	0.83 kW-hr (2833 BTU)	1 min
Deliver Package	N/A	49.7 kW (66.7 Hp)	1.66 kW-hr (5657 BTU)	2 min
LEG 2				
Takeo HOGE	N/A	38.0 kW (51.0 Hp)	0.63 kW-hr (2163 BTU)	1 min
Climb to 1350m ASL	30.7 m/s (59.6 kts)	24.1 kW (32.3 Hp)	0.13 kW-hr (452 BTU)	0.33 min
Level Cruise	135 kts	70.6 kW (94.7 Hp)	13.8 kW-hr (47.0 kBTU)	11.7 min
Descend to 1200m ASL	30.7 m/s (59.6 kts)	23.6 kW (31.6 Hp)	0.13 kW-hr (442 BTU)	0.33 min
Land HOGE	N/A	37.0 kW (49.6 Hp)	0.62 kW (2103 BTU)	1 min
			Block Total:	27.36 min

D. Range and Endurance

The Garra was designed to be fueled to be able to y the logistics mission requirement of 200 km (108 nm) at 250 km/hr (135 kts) when fully fueled with 20 minutes of reserve fuel as shown in Fig. 24(a). When fully fueled the vehicle can carry a total payload of 107 kg before rotor stalling. This is signi cantly more than our required payload of 50 kg. See 19 for the performance metrics.

VIII. Propulsion

A. Power Plant Selection

To nd the best propulsion system for this design a Pugh matrix was created with the key design drivers weighted with their relative importance (see Table 20). The types of power plant systems evaluated were: (i) IC (internal combustion engine with fully mechanical transmission to the main rotor and tail system), (ii) battery (lithium-ion battery power plant with fully electric transmission to the main rotor and tail system), (iii) IC + electric anti-torque (internal combustion engine with mechanical transmission to main rotor and pusher propeller, and electric transmission to anti-torque rotor), (iv) IC + fully electric tail (internal combustion engine with mechanical transmission to the main rotor, and electric

(a) Range vs. Velocity

(b) Endurance vs. Velocity

(c) Range vs. Fuel Mass

(d) Endurance vs. Fuel Mass

Fig. 24 Range Endurance Plots

transmission to tail system), (v) IC as generator (internal combustion engine to charge batteries with electric transmission to main rotor and tail system), and (vi) hydrogen electric (hydrogen fuel cell power plant to charge batteries with electric transmission to main rotor and tail system). Each type of propulsion system is scored in each category from -3 to 3 with internal combustion engine being the baseline. The top two categories in this Pugh matrix are safety and bulkiness. The bulkiness category considers the size and weight of the system. Bulkiness scored the second highest because of the need for weight minimization. The result of the analysis showed that an internal combustion engine was the best choice.

B. Engine Selection

A comparison between diesel, piston, and turboshaft engines was carried out to evaluate the system that best meets the requirements of the design. Turboshaft engines that have the power outputs required for our design are not available. A study was carried out to compare power to weight ratios for various engines including internal combustion using gasoline and diesel. Table 21 shows a side-by-side comparison of ten internal combustion engines. Based on the required power and an emphasis on low weight the Limbach L550 EF was chosen. A twin-engine installation was selected to increase the safety of the overall design because one engine is able to provide the 20 minute OFI ight after a failure as speci ed in the RFP. Each engine is connected via a sprung clutch allowing for smooth power transfer and declutching during autorotation. Each engine will be cooled by a separate fan located in front of the engine. The engine layout in the main bay is shown in Figure 26.

Table 19 Vehicle Performance

Parameter	Value
Empty Weight	180 kg (397 lb)
Max GTOW	313 kg (689 lb)
Design GTOW, 50 kg payload, full fuel	256 kg (563 lb)
Fuel, Logistics, 110 kts	22.3 kg (49.1 lb)
Fuel, Logistics, 135 kts	25.7 kg (56.7 lb)
Fuel, Delivery, 135 kts	17.3 kg (38.1 lb)
Cruise Velocity, ISA+20	69.5 m/s (135 kts)
Max Range, 50 kg Payload	252 km (136 nm)
Max Endurance, 50 kg Payload	111 min
Performance Metric, Delivery	6.16 m/s (12.0 kts)
Performance Metric, Logistics	10.28 m/s (20.0 kts)
Velocity Best Range,$+_{1A}$	30.7 m/s (59.6 kts)
Velocity Best Endurance,$+_{14}$	45.9 m/s (89.2 kts)

Table 20 Power Plant Selection Pugh Matrix

Category	Weight	IC	Battery	IC + Electric Anti-torque	IC + Fully Electric Tail	IC as Generator	Hydrogen Electric
Safety	27.03%	0	-0.5	0	-0.5	-1	-0.5
Reliability/Readiness	18.92%	0	-1	0	0	0	-2
Responsiveness	10.81%	0	2	1	2	2	2
Thermal E ciency	13.51%	0	2	0	-1	-2	1
Noise	5.41%	0	2	0	-1	-2	1
Bulkiness	24.32%	0	-3	-1	-1	-2	-1
Weighted Sum		0	-0.4595	-0.1351	-0.2432	-0.7568	-0.2973
Relative Ranking		1	5	2	3	6	4

Table 21 Engine Comparison

Engine	Power	Weight	Output Shaft RPM	Cooling	Power-to-Weight Ratio (W/g)
Austro Wankel AE50R	41 kW (55 hp)	28 kg (61 lb)	7750	Liquid	1.472
Rotax 582UL Inline WC Twin	48 kW (65 hp)	47 kg (104 lb)	6500	Water	1.025
Hirth F-33 2 Stroke	45 kW (60 hp)	32 kg (70 lb)	6500	Air	1.406
Hirth F-23 Opposed Twin	37 kW (49.5 hp)	32 kg (71 lb)	6150	Air	1.155
Hirth 3202 Inline Twin	41 kW (55 hp)	37 kg (82 lb)	5500	Air	1.100
Hirth 3502 Inline WC Twin	45 kW (60 hp)	36 kg (79 lb)	5000	Water	1.246
Hirth 3203 Inline Twin	48 kW (65 hp)	37 kg (82 lb)	6300	Air	1.300
Hirth 3503 Inline WC Twin	52 kW (70 hp)	35 kg (79 lb)	6500	Liquid	1.454
McCulloch 4318A	54 kW (72 hp)	36 kg (80 lb)	3400	Air	1.476
Limbach L 550 EF	37 kW (49.5 hp)	15 kg (33 lb)	7500	Air	2.467

C. Tail Con guration

Three tail rotor con gurations were considered: (i) an electric anti-torque rotor with electric pusher propeller, (ii) an electric anti-torque rotor with mechanical pusher propeller, and (iii) a mechanical anti-torque rotor with a mechanical pusher propeller. All three con gurations provided safety for the mission. A comparison of the weight is shown in Table 22. The fully mechanical tail system was selected because of its low complexity and weight. The higher weights of the electric anti-torque, electric pusher prop and electric anti-torque and mechanical pusher propeller were attributed to the high weights of the battery and generator.

Table 22 Tail Con guration Weight Comparison

Component	Electric Anti-torque + Electric Pusher Prop	Electric Anti-torque + Mechanical Pusher Prop	Mechanical Anti-torque + Mechanical Pusher Prop
Cables, Generators, Controllers, Motors	31.7 kg (69.9 lb)	20.5 kg (45.2 lb)	0 kg (0 lb)
Battery (10 minutes of power)	8.8 kg (19.4 lb)	1.2 kg (2.6 lb)	0 kg (0 lb)
Transmission Shaft, Gearboxes	0 kg (0 lb)	11 kg (24.3 lb)	5.3 kg (11.7 lb)
Total Weight	40.5 kg (89.1 lb)	32.7 kg (72.1 lb)	5.3 kg (11.7 lb)

D. Transmission

Table 23 Gear Material Comparison and Gear Ratio

Gear Number	Ferrium C61TM	Vasco X-2M Steel	9310 Carburized Steel	Ratio	RPM
1	0.212 kg (0.467 lb)	0.206 kg (0.454 lb)	0.210 kg (0.462 lb)	-	7500
2	0.331 kg (0.730 lb)	0.322 kg (0.710 lb)	0.328 kg (0.723 lb)	1.250	6000
3	8.109 kg (17.877 lb)	7.875 kg (17.321 lb)	8.028 kg (17.698 lb)	4.878	1230
4	0.476 kg (1.049 lb)	0.462 kg (1.018 lb)	0.471 kg (1.038 lb)	0.246	5008
5	1.065 kg (2.348 lb)	1.034 kg (2.280 lb)	1.054 kg (2.324 lb)	-	5008
6	0.851 kg (1.876 lb)	0.827 kg (1.823 lb)	0.843 kg (1.858 lb)	0.894	5602
7	0.746 kg (1.645 lb)	0.724 kg (1.596 lb)	0.738 kg (1.627 lb)	0.308	4000
Total Weight	11.79 kg (25.99 lb)	11.450 kg (25.24 lb)	11.672 kg (25.73 lb)		

The objective of the transmission is to provide mechanical power to the main rotor, pusher propeller, anti-torque rotor, and the generator which charges the batteries to power the avionics, winch system, and bomber doors. Straight bevel gears were selected because of their ability to change direction and their improved safety due to lower contact stress compared to face gears of similar weight. Several materials including ATI Allvac Vasco X-2M Steel, AISI 9310 Carburized Steel, and Ferrium C61TM type steel were considered for the gear material. They all meet the industry standards for gear materials and after carrying out an analysis for weight it was found that ATI Allvac Vasco X-2M Steel was the best option for this design (see Table 23). Figure 25 shows the arrangement of the drive system. The 7500 rpm inputs from the engines are reduced to 6000 rpm by the rst stage and further to the main rotor rpm of 1230 using stage 2. Gear 4 is a step up from gear 3 to achieve the 5008 rpm required for the pusher prop. The output rpm at gear 4 was selected to be the same as the pusher propeller rpm to avoid an additional gear reduction. Gear 6 is a step up to achieve the 5602 rpm required for the anti-torque rotor. Gear 7 will operate at 4000 rpm and will drive the generator to charge the batteries to power the onboard avionics, winch system, and bomber doors. The gear ratio and output rpm for each gear are listed in Table 23.

E. Tail Shaft

The tail shaft will operate at 5008 rpm with a maximum torque of 85 N-m. For the construction of the tail shaft, aluminum and graphite/epoxy composite were considered. The shaft has an outer diameter of 2.4 cm and a wall thickness

(a) Main Drive (b) Tail Drive

Fig. 25 Main Drive and Tail Drive

of 1.0 mm. The shaft has one mid-span bearing and one ex coupling to reduce instability and improve repairability. Quantitative comparison was made between the shaft constructed from each material in Table 24. Aluminum was selected because it is well established and extensively used. In addition, the aluminum shaft has better life traceability. It will show signs of wear well before failure. The aluminum shaft will operate at 83 Hz and will have a critical shaft speed of 192 Hz making it sub-critical. A sub-critical shaft improves safety and eliminates the need for dampeners.

Table 24 Tail Shaft Comparison

Property	Aluminum	Graphite/Epoxy
Outer Diameter	2.4 cm (0.94 in)	2.4 cm (0.94 in)
Wall Thickness	1.0 mm (0.039 in)	1.0 mm (0.039 in)
Overall Shaft Length	2.45 m (96.5 in)	2.45 m (96.5 in)
Shaft Weight	0.496 kg (1.09 lb)	0.269 kg (0.593 lb)
Operating Speed	83 Hz	83 Hz
Critical Speed	192 Hz	317 Hz
Torsional Sti ness	106 N-m (78.2 lb-ft)	278 N-m (205 lb-ft)
Operating Power	27.0 kW.0 (36.2 hp)	27.0 kW (36.2 hp)
No. of Mid Span Bearings	1	1
No. of Mid span Flex Couplings	1	1
Bearing Mass	0.132 kg (0.291 lb)	0.132 kg (0.291 lb)
Flex Coupling Mass	0.014 kg (0.031 lb)	0.014 kg (0.031 lb)
Total Weight	0.643 kg (1.42 lb)	0.416 kg (0.217 lb)

F. Oil and Housing

The housing is made of AZ91D magnesium alloy to minimize weight without reducing structural integrity. The housing will support the gears of the transmission ensuring that there is no unintended movement during operation. The top structure of the housing is secured to the bulkheads. This distributes the forces from the main rotor to the airframe structure. The lubrication oil to be used is the MIL-PRF-23699. It was chosen because of its ability to tolerate extreme conditions and allow for quick cold starts. There will be six di erent oil injectors to ensure that all gear connections are covered, and a magnetic oil lter to ensure that any debris from the gears is collected. The full transmission can be seen in gure 26.

Fig. 26 Main Bay Diagram

IX. Airframe Structural Design

A. Description

The Garra's structure was designed with safety for the vehicle and the package in mind. The main structure of the vehicle is constructed from bulkheads in a rib cage fashion to provide space and security for the package. The fuselage features a novel open bottom design with enhanced torsional sti ness achieved by carbon ber supports. The structure can be divided into three main sections: the fuselage, the empennage, and the cowling. A detailed review of all components can be found in Garra's Structure Pullout Chart.

B. Fuselage Structure

The fuselage structure is divided into two main sections: the main bay and the package bay. The main bay is designed to take on most part of the loads while the package section is designed to mostly provide and aerodynamic fearing for the package. The main bay is built from ve aluminum closed box beam bulkheads that feature an enhanced safety feature due to the exible nature of the aluminum. In this bay, the propulsion and hoisting systems are housed. The top beam design was proven safe and reliable by Sikorsky through the certi cation of the Skycrane helicopter. The history of this structure combined with the structural analysis prove the safety and the ease of future certi cation of the design.

The package bay on the other hand is designed as an aerodynamic fearing. This section of the fuselage is not designed to take the main loads from the rotary systems. The package bay features an open bottom design that gives the vehicle its hoisting package delivery capability. As some of the bulkheads are "open" the torsional sti ness of the fuselage was decreased. To address this, a secondary carbon ber truss-like structure was added to the sides of the package bay, leading to a 70% increase in torsional sti ness without any major weight increases.

The main structure of the fuselage consists of aluminum 2024- T3 box beam members that have a 15 mm (0.6 in) x15 mm (0.6 in) cross-section with the factor of safety of 1.6. As an added safety feature, the Garra's structure features strain gauges located in the most stressed parts of the aircraft to ensure that they never exceed safe limits. This will also provide operators with information on the life of their product and reduce the risk of an accident.

1. Fuselage Structural Enhancement

Due to the open bottom design, the initial fuselage required thick bulkheads to prevent twisting. To reduce the mass of the bulkhead, two supporting structures were added on the sides of the fuselage that connect the two main bulkheads. The support structures are made of carbon ber/epoxy which is lighter and better in compression/tension than aluminum alloy 2024-T3.

A comparison study was done to prove that the two carbon ber supports could maintain the fuselage shape under a torsional load. A torque twice as much as the vehicle is experiencing during cruise was applied during this test on both bulkheads with reacting directions. The performance of the structure is then compared under two cases: thin bulkheads with no support, and thin bulkheads with support. The test was performed with Altair SimSolid software, a more detailed description is found in the structural optimization paper. The initial hypothesis is the supports will