2009 AHS Student Design Competition: Graduate Category

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Georgia Institute of Technology
Graduate Design Report
“PEREGRINE”

ALTERNATIVE DRIVE ROTOR SYSTEM

DEPARTMENT OF AEROSPACE ENGINEERING

GEORGIA INSTITUTE OF TECHNOLOGY

ATLANTA, GA 30332

AND

LIVERPOOL UNIVERSITY

IN RESPONSE TO THE 26TH ANNUAL AMERICAN HELICOPTER SOCIETY STUDENT DESIGN COMPETITION – GRADUATE CATEGORY

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Didier Contis
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<th>Peregrine</th>
<th>Section</th>
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<td>Non-conventional Rotor/Drive System</td>
<td>Yes</td>
<td>No</td>
<td>Yes</td>
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<td>Rotorcraft Flight Characteristics</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
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<td>150</td>
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<td>Main Rotor Tip Speed (ft/s)</td>
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2009 AHS Student Design Competition: Graduate Category
**Peregrine Specifications**

Empty Weight 7572 (3435 kg)
Max GW 14700 lbs (8165 kg)
GTOW 12360 lbs (5606 kg)
Payload 4500 lbs (2041 kg)
Fuel 2272 (1031 kg)
Number of Seats: 8

Max Range
420 nm
(581.5 km)

Max Airspeed (IRP)
249.8 knots
(462.6 km/h)

Cruise Speed (MCP)
222.5 knots
(412.1 km/h)

Max Range
120 knots
(222.2 km/h)

Max Endurance
2.53 hours

IRP 3100 shp
MCP 2880 shp

Rotor Radius
21 ft (6.4m)

Flat Plate Area
14.49 sq ft (1.35 sq m)

Tip Speed (Fast)
650 ft/s (198 m/s)

Tip Speed (Slow)
420 ft/s (128 m/s)

Disk Loading
9.07 lb/sq ft (434 N/sq m)

Max Service Ceiling
15,813 ft (4820 m)
1. Introduction

Historically, rotary wing flight has been a trade between hover performance and forward flight speed. The search for an alternative drive that is capable of bridging this gap is one of the next frontiers of current rotorcraft aviation research. The high disk loading of current configurations would allow for high forward speed, but this also limits hover efficiency, low speed controllability, and hover endurance while creating a high downwash. These hover characteristics are desirable attributes as well. Additionally, operational costs required to increase power loading are often the limiting factor in designs. The design goal was to fulfill the Request For Proposal (RFP) of this design competition and bridge the gap of high speed forward flight with hover efficiency. Previous attempts to build fast helicopters have resulted in significant degradations in at least one common design parameter whether noise, hover performance, controllability, or operational cost.

This report represents the completion of the three iteration loops from the Product Development side of the Integrated Product and Process Development (IPPD) Methodology. The first two iterations of the Conceptual Design Loop were conducted simultaneously in Systems Evaluation and Design Analysis. The AgustaWestland Super Lynx 300 was used as the baseline helicopter because it meets the initial requirements of the RFP. The initial Rf method\(^1\) was used to conduct sizing analysis. Once the most efficient design was found in terms of horsepower and gross weight, each component of the aircraft was optimized in the Preliminary Design Iteration Loop. With this optimized aircraft, a last iteration of optimization was done to maximize performance of the aircraft in the Initial Product Data Management Loop.

The Peregrine was selected after examining the concepts that were available during the first conceptual analysis iteration. On a parallel timeline, while the conceptual selection was being completed, the group began to size the aircraft using the basic methods in the initial Rf method and the Georgia Tech Preliminary Design Program. These

![Diagram of Product Development and Process Development loops](image)

Figure 1.1: Georgia Tech Integrated Product and Process Development (IPPD) Methodology\(^2\)

The Peregrine was selected after examining the concepts that were available during the first conceptual analysis iteration. On a parallel timeline, while the conceptual selection was being completed, the group began to size the aircraft using the basic methods in the initial Rf method and the Georgia Tech Preliminary Design Program. These
two steps were vital for the development of the alternative drive system. Once the definition of what was considered qualifying and alternative, the process of helicopter design continued. With a baseline helicopter, an alternative concept was selected along with student-produced baseline targets; the design process was moved from the Conceptual Design Iteration Loop in Figure 1.1 to the Preliminary Design Iteration Loop. During the next two iterations, the focus was to clearly model the alternative drive system using CATIA for visual representation and aerodynamically validate the alternative drive concept. As greater understanding of the design process was developed, conceptual checks were made to ensure the design sizing and the alternative drive system would coalesce into a feasible design.

To accurately reflect the design process of each concept, this report covers the initial concept selection in Chapter 2. With the baseline aircraft, mission, selection criteria, and initial target values selected, the initial concept was modeled from the initial Rf method to develop the basic size and weight criteria needed to design the subcomponents. This initial Rf method is documented in Chapter 3. Chapter 4 completes the documentation of the concept selection process. To validate the final design, the same sizing program was used with the most current data available on the aircraft. Following the concept selection process, the focus was on the alternative drive system itself and the performance parameters that were derived from testing. Chapter 5 is the heart of the alternative drive system. This chapter also includes the performance characteristics of the transmission. The performance of the aircraft validates the design and overall concept selection. The performance of the aircraft was measured in the performance of its subcomponents that constitute the remaining parts of the drive system: the rotor, the ducted pusher propeller, and the engine. The selection of the rotor characteristics and features are listed in Chapter 6. The dynamic characteristics of the rotor and its hub are listed in Chapter 7. The rotor acoustic performance is listed in Chapter 8. The next major alternative characteristic is the ducted pusher propeller. The design and performance characteristics of this last main alternative feature are discussed in Chapter 9. A conventional engine was chosen because the development of a non-turbine, non-reciprocating engine was determined to be beyond the scope of this analysis. Chapter 10 discusses the performance characteristics of the engines chosen for the Peregrine. Chapter 11 discusses the Peregrine’s fuel control system. The weight of the fuselage was vital to ensuring the payload would remain on target without sacrificing safety and structural integrity of the newly designed airframe which is discussed in Chapter 12. The baseline aircraft, the AgustaWestland Super Lynx, has a very blunt nose and front fuselage area. To reduce the drag and aerodynamic moment of the fuselage, significant airframe improvements were made. The analysis of these aerodynamic improvements is detailed in Chapter 13. A FLIGHTLAB\textsuperscript{3} model was created to conduct further analysis of the aircraft. The details of the analysis are listed in Chapter 14. From the FLIGHTLAB results, a simulation model of the aircraft was created to develop the flight controls and stability augmentation system required for the Peregrine. The details of the analysis and results of the simulation model are listed in Chapter 15. No design can be considered feasible without a cost estimate. In the Peregrine’s case, the aircraft cost analysis is listed in Chapter 16. Finally, a basic safety analysis and an estimated flight certification timeline are listed in Chapter 17.

2. Vehicle Configuration and Selection Methods

Due to the lack of concise requirements in the RFP, particular attention was paid to developing student designated performance requirements and engineering targets to validate the design. Significant increases in performance while maintaining noise, safety, and cost was chosen as the most important design parameters. Several configurations of drive systems and overall aircraft designs were looked at since the baseline helicopter must be a current production aircraft. The 2009 AHS student design Request for Proposal (RFP) requires the design of a new, non-conventional rotor-drive system for a helicopter, using an existing design in terms of size, weight, and performance as a starting point.

2.1. Helicopter Mission Configuration

The primary role selected for the helicopter was a military utility mission. Since the baseline helicopter is the Agusta Westland Super Lynx 300\textsuperscript{4}, the baseline helicopter also has to have the capability to perform search and rescue missions, as well as para-military and other military missions. These capabilities for diverse mission capability were maintained while focusing primarily on the design mission. Within this military utility mission, an operational radius of 210 NM was chosen as a target. Fuel calculations include a five minute run up time, two minute take off time, four minutes for climb and descent each, an additional two minutes for the landing period and
a twenty minute reserve. Particular missions within the military utility role such as fast rope, rescue hoist, and communications will not be covered though the aircraft will be configurable for these profiles.

2.2. Overall Design Trade Study Approach

A trade study was used for selection of the overall concept, engines, transmission, pusher propeller, rotor and hub configuration, rotor blades, rotor controls, and flight control configuration. In each case, analysis of the subcomponent was considered independently of other subcomponents and their analysis. In this section, the Quality Function Deployment (QFD) house of quality approach was used to develop an Overall Evaluation Criterion (OEC) to meet the student derived design parameters. With the established OEC, each subcomponent was analyzed within its category. When the OEC did not fit the particular component, an analysis specific to the category was used to differentiate performance characteristics.

2.3. Quality Function Deployment Analysis

The Quality Function Deployment (QFD) matrix quantifies which engineering characteristics are more heavily favored in the overall design. Having customer requirements allowed the translation of the broad set of design requirements into the student created engineering requirements generated for the design. In this case, the roof of the QFD and the central matrix of the QFD were used. These two rooms, along with the relative risk and absolute risk weightings, help form the decision of weights within the OEC. The OEC allowed quantitative comparison of various designs and how well they meet the most important design factors.

An important consideration in the house of quality and central matrix is that all the rankings are relative. The QFD gives a method of determining rank and a numerical solution but is not an absolute ranking. Since the RFP requires improvement in performance overall, performance characteristics placed highest in the relative ranking scale while noise and safety, though still important, did not rank as high. Additionally, creating an engineering target for an alternative drive system was not quantifiable, though it qualified as a means of measurement. Because of its lack of quantifiable measures, the alternative drive was used as screening criteria instead of evaluation criteria. These results confirm the analysis of the RFP and are depicted in the QFD.

![Figure 2.1: Peregrine Quality Function Deployment](image-url)
2.4. Overall Evaluation Criterion

The Overall Evaluation Criteria (OEC) is the evaluation method used to assess the overall performance of various design concepts through a series of rankings of all of the engineering requirements specified in the QFD from Section 2.3. Each value in the OEC is a ranking of a given design concept with respect to a target value. The OEC is presented as a ratio of benefits to costs with the goal of any design to have a score greater than one. The OEC for the non-conventional drive system project is shown in Equation 2.1.

\[
OEC = \frac{0.620(MCI) + 0.202(SI) + 0.178(NCI)}{\text{Cost Index}}
\]  

(2.1)

MCI: Mission Capability Index, SI: Safety Index, NCI: Noise Comfort Index. The coefficients for each of these indices were derived from the QFD and are shown in Equation 2.1. These coefficients are the total of the relative importance values from each of the engineering requirements within each evaluation criterion divided by the sum of the relative importance values from all of the engineering requirements within the OEC.

2.4.1. Mission Capability Index

The mission capability index (MCI) evaluates all of the engineering requirements that reflect how well the rotorcraft can perform a given mission. The MCI is defined by Equation 2.2.

\[
MCI = 0.357 \left( \frac{v_c}{215 \text{ kts}} \right) + 0.3 \left( \frac{v_d}{250 \text{ kts}} \right) + 0.2 \left( \frac{v_e}{100 \text{ kts}} \right) + 0.309 \left( \frac{R_{\text{action}}}{210 \text{ nm}} \right) + 0.256 \left( \frac{\text{Payload}}{4500 \text{ lbs}} \right) + ...
\]  

(2.2)

\[v_c: \text{cruise speed (target 215 kts)}, \quad v_d: \text{dash speed (target 250 kts)}, \quad v_e: \text{max endurance speed (target 100 kts)}, \quad R_{\text{action}}: \text{radius of action (target 210 nm)}, \quad \text{Payload: max payload of vehicle (target 4,500 lb)}, \quad \text{MTBF: mean time between failure (22.8 hours)}, \quad \text{MTTR: mean time to repair (1.2 hours)}, \quad S/C: \text{service ceiling (target 20,000 ft)}.

The coefficients in front of each term in Equation 2.2 are the relative importance of each of the engineering requirements within the MCI, and the sum of these coefficients is 1.0.

The radius of action is defined as the one way distance traveled during the mission. For instance, if the mission was to take off from a ship and fly to a destination out at sea, that destination would be 210 nm away, meaning the entire mission would be 420 nm of total travel.

The Mean Time Between Failure (MTBF) is defined as any failure in any part of the rotorcraft that keeps it from performing its mission. Any component or subcomponent that breaks and keeps the aircraft from flying safely constitutes a failure. The goal of this term in Equation 2.2 is to keep the mean time to repair (MTTR) as low as possible and the mean time between failures (MTBF) as high as possible. A combination of these two positive influences on mission capability will drive the values in Equation 4.2 higher.

2.4.2. Safety Evaluation Criterion

The Safety Index (SI) expresses how safely a given design concept will perform a given mission. It is defined by Equation 4.3.

\[
SI = 0.841(E_{pg}) + 0.086 \left( \frac{AI}{25} \right) + 0.073(Surv)
\]  

(2.3)
EPR: Excess power ratio (target 1), AI: Autorotative index (target 25), Surv: Survivability (target = 1). The coefficients of Equation 2.3 are derived the same way they are in Equation 4.2.

The excess power ratio is the ratio of the 30-second one engine inoperative (OEI) power available to the hover out of ground effect (HOGE) power required at sea level on a standard day. If and when a rotorcraft loses an engine during any point during flight, the remaining engine spools up to its OEI rating, a 30-second rating that is double the current power required. An aircraft with a large amount of OEI power in its engines is likely to be able to safely continue flight in the event of an engine failure.

The autorotative index is a ratio that includes the most important factors that influence the autorotative performance of a helicopter, such as kinetic energy stored in the rotor, weight of the aircraft, and the rotor disc area. Any rotorcraft that experiences a total loss in power at any point during flight should be capable of performing an autorotation landing. The autorotation is an energy management maneuver where the descent rate and forward speed of the rotorcraft cause the lift vector to tilt forward, driving the rotor, and maintaining the rotor speed with rotational inertial energy. Once the aircraft glides down close enough to its landing surface, the pilot converts much of the kinetic energy stored in the rotor to thrust, thus slowing the descent rate of the vehicle and allowing for a safe landing. The autorotative index used in this project is shown in Equation 2.4.

\[ AI = \frac{I_r \Omega^2}{2WDL} \]  \hspace{1cm} (2.4)

\( I_r \): main rotor inertia, \( \Omega \): main rotor rotational speed, \( W \): gross weight of the vehicle, \( DL \): disc loading (ratio of gross weight to main rotor disc area). High inertia and RPM, combined with low weight and disc loading (large main rotor diameter), provide the optimum combination for autorotative performance.

The survivability term is essentially the probability that a passenger will survive any given flight, and it is defined by Equation 2.5.

\[ Surv = (1 - P_f)(1 - P_c)(P_{srv}) \]  \hspace{1cm} (2.5)

\( P_f \): probability of an in-flight failure, \( P_c \): probability that the failure is catastrophic, \( P_{srv} \): probability of surviving a catastrophic crash. The first term in Equation 2.5 is the probability of a successful flight with no failures; the second term is the probability that the failure is not catastrophic (i.e. does not lead to a crash); and the third term is the probability that a passenger will survive a crash. The product of the three terms is the probability of a successful flight, and one minus the survivability term is the probability that there will be an in-flight engine failure that causes a crash in which the crew does not survive.

2.4.3. Noise Evaluation Criterion

The Noise Comfort Index (NCI) is a measure of internal and external noise as well as the vibration level experienced by passengers and is defined by Equation 2.6.

\[ NCI = .461 \left[ .35 \left( 1 - \frac{N_{fo} - 90}{100} \right) + .35 \left( 1 - \frac{N_{60i} - 85}{100} \right) + .3 \left( 1 - \frac{N_{IP} - 70}{100} \right) \right] + .274(1 - 10F) + .265 \left( \frac{50}{NI} \right) \]  \hspace{1cm} (2.6)

\( N_{fo} \): Noise measured when aircraft flies over at cruise speed 500 ft above ground level (AGL) (target 90 dB), \( N_{60i} \): Noise measured from 60 ft away while hovering (target 85 dB), \( N_{IP} \): Noise in-plane at cruise from 1 mile away (target 70 dB), \( F \): passenger vibration throughout the flight regime (target 0.004f+0.01), \( NI \): interior noise index measured in cruise flight (target = 50 dB). Once again, the coefficients in Equation 4.6 are derived using the same methods as Equations 2.2 and 2.3.

2009 AHS Student Design Competition: Graduate Category 7
The target F value of 0.004f+0.01 is due to the fact that this expression is the vibration limit at each frequency level (f/rev) (i.e. 1/rev vibration limit is 0.014, 4/rev limit is 0.026, etc). Equation 2.6 represents a ranking of the comfort level of not only the passengers and crew inside the aircraft, but also to everyone within the environment where the rotorcraft operates. Each of the target values attempt to minimize interior and exterior noise as well as vibration levels to the crew and passengers.

2.4.4. Cost Evaluation Criterion

The cost index (CI) is the denominator of the OEC in Equation 2.1. It is a numeric value that includes the effects of production costs, fuel costs, and operations costs. It is defined by Equation 2.7.

\[
CI = 0.502 \left[ 0.4 \left( \frac{prod}{\$2 mil} \right) + 0.6 \left( \frac{RDTE}{\$50 mil} \right) + 0.274 \left( \frac{SFC}{.34} \right) + 0.224 \left( \frac{C_{ops}}{400} \right) \right]
\] (2.7)

Prod: production cost (target $2 million per aircraft), Research, Development, Testing and Evaluation (RDTE) cost (target $50 million), SFC: specific fuel consumption (target .34 lb/hp/hr), C_{ops}: cost of operations (target $400/hr).

The goal of each of these parameters is obviously to keep them as small as possible, thus driving the cost index down and the OEC up.

Values for the CI inputs (in 2009 dollars), completed with the Bell PC Model as discussed in Section 15, are Average unit production cost: $6.18 million, RDTE: $231 million, SFC: 0.49 lb/hp/hr, C_{ops}: $1,093/hr. These values yield a Cost Index of 3.02.

2.5. Initial Concept Selection

Multiple alternative rotor designs were initially considered. Because the RFP requires the ability to hover and autorotate, several designs were quickly eliminated. Within the category of desirable but non-qualifying concepts, various ducted fan configurations were briefly considered but rejected because of their lack of autorotative capability. Tilt configurations of wings, ducts, and rotors, as well as vectored thrust and deflected slipstream, were all rejected for this same lack of autorotative capability. With this initial screening, tip jet, compound, coaxial, and standard configurations of production helicopters were the remaining concepts available.

2.6. Hub Design Selection

A prioritization matrix of hub qualities was conducted to determine those most important to this rotorcraft. Each quality was scored pairwise against each of the other qualities. Two scales, one using an aggressive scoring system of 1 (much less important), 3, 9, 10 (much more important) to narrow the few most important traits, as well as a more conservative scale of 1 (much less important), 2, 3, 6 (much more important), were used. The scores from the pairwise comparisons were totaled and then used to determine the respective weightings of importance given to each quality in the final selection process of the rotor and hub configuration.

![Figure 2.2: Prioritization Ranking, Aggressive Scale](image)
Safety, measured in mean time between failure (MTBF), is the most important characteristic. Vibrations and drag receive higher scores in the case of the Peregrine than would be expected for a conventional helicopter because they both heavily influence the high speed cruise condition of flight. The conservatively scored weightings were then applied to three candidate rotor concepts. The conservative scale was used for the final concept choice in order to not totally eliminate Direct Operating Cost as a factor in the decision, since its relative weighting according to the aggressive scale is less than 1% of the total.

![Figure 2.3: Prioritization Ranking: Conservative Scale](image)

**Concept 1: 3-Blade, Articulated Rotor, Swashplate Control**

This is the standard configuration for a coaxial rotor. It has been widely produced and used in helicopters such as the Kamov Ka-25 and Ka-32. For this hub selection process it will be considered the baseline against which more advanced rotor concepts will be compared. Its positive aspects are the well-known inherent advantages of a coaxial rotor such as reduced disc area and anti-torque without a tail rotor. Its primary disadvantages are increased hub drag due to the large shaft between the rotors and the many exposed parts of the two articulated rotors themselves as well as mechanical complexity. A 2-Blade Coaxial Teetering configuration would reduce the drag due to exposed hinges and linkages, but was eliminated from initial consideration since using fewer than 3 blades would sacrifice the gains in reduced disc area and tip speed due to the coaxial rotor.

![Figure 2.4: 3-Blade, Articulated Rotor](image)

**Concept 2: 3-Blade, Bearingless Rotor, Ind. Blade Control by HMA**

Controlling the blades individually with hydro-mechanical actuators (HMA) provides weight savings and improved control response. Using a bearingless rotor instead of an articulated hub also reduces hub drag and operation cost.

![Figure 2.5: XH-59A 3-Blade Rotor with fairing and enclosed control mechanisms](image)

**Concept 3: 4-Blade, Bearingless Rotor, Ind. Blade Control by HMA**

Combining the advantages of the bearingless hub and HMA control with a 4-Blade rotor results in the best, but most expensive candidate rotor concept. Hover and autorotation performance will improve due to the 4 blade configuration, which will also allow a lower tip speed in forward flight and reduce noise in all flight regimes.
Concept 3 is a combination of the concepts displayed in Figure 2.6 and Figure 2.7. The grades for each hub quality shown in Figure 2.8 are weighted and totaled in Figure 2.9. The 4-blade bearingless rotor receives the highest scores across nearly every quality, but makes its biggest gains over the other two candidates in noise and autorotative index by using more blades. Hub drag will also be significantly less than the articulated hub due to fewer exposed parts.

Figure 2.6: 4-Blade Coaxial Rotor Used by X2 Demonstrator

Figure 2.7: Individual Blade Controls enclosed in fairing

Figure 2.8: Rotor Concept Scores

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<tr>
<th></th>
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**TOTAL**

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<td>0.150735</td>
<td>0.090453307</td>
<td>0.150735</td>
</tr>
<tr>
<td>Maneuverability</td>
<td>0.150735</td>
<td>0.090453307</td>
<td>0.150735</td>
</tr>
<tr>
<td>Drag</td>
<td>0.129952</td>
<td>0.02599233</td>
<td>0.129952</td>
</tr>
<tr>
<td>Weight</td>
<td>0.087941</td>
<td>0.026382214</td>
<td>0.087941</td>
</tr>
<tr>
<td>Price</td>
<td>0.02604</td>
<td>0.018227712</td>
<td>0.02604</td>
</tr>
<tr>
<td>Noise</td>
<td>0.021031</td>
<td>0.012850867</td>
<td>0.021031</td>
</tr>
<tr>
<td>Direct Operating Cost</td>
<td>0.016646</td>
<td>0.001664592</td>
<td>0.016646</td>
</tr>
</tbody>
</table>

Figure 2.9: Rotor Concept Trade Study Final Results

The initial rotor trade study results in a 4-Blade bearingless rotor with moderate twist and separate tip speeds, 650 ft/s for low speed flight, and 420 ft/s for high speed compound flight receiving the highest score, indicated in green. The four blade bearingless configuration receives a 38% more favorable score than the three blade articulated configuration and a 7% more favorable score than the three blade bearingless configuration.
### 3. Concept Selection, Sizing, and Performance

In the conceptual design phase, the configuration of rotorcraft is tested and selected by using first level analysis. This allows the optimum geometry to be defined for a given set of requirements so that each component can be further optimized. Georgia Tech Preliminary Design Program (GTPDP)\(^6\) and Requirements-Driven Fuselage Design Program (RDFD)\(^7\) were employed to analyze the baseline model and candidates. Because these tools are coupled, the automated analysis environment was built in the ModelCenter\(^8\) framework.

Four total candidate aircraft were compared to the baseline SuperLynx 300. At first, the baseline model is calibrated with regard to actual data and a coaxial version is added. Two candidates are developed from each base model; compound rotorcraft and compound rotorcraft with slowed rotor. Therefore, a total of four candidates were modeled and compared. Based on the gross weight, ceiling and required horse power, the coaxial compound rotorcraft with slowed rotor is selected. Finally, the parametric study and optimization of the preliminary candidate were performed and the results become the fundamental data for detailed design of each subcomponent. The brief procedure of conceptual design is depicted in Figure 3.1.\(^9\)

![Baseline: Super Lynx 300](image)

**Figure 3.1: Conceptual Design Procedure**

#### 3.1. Georgia Tech Preliminary Design Program (GTPDP)

GTPDP is a preliminary helicopter design code to obtain a rough configuration for a user's specifications in a few seconds. It has the capability to give an optimized design for the given configuration as well as a performance analysis. GTPDP presently can perform vehicle and component sizing, mission analysis, performance estimation, cost analysis, design optimization, and trade-off studies. Other extensions include a weight and balance routine, direct operating costs and coaxial evaluations.

Results of the program have proven to be highly accurate when compared to flight test data. GTPDP uses RF sizing method for configuration analysis. The optimization section design starts with a user provided initial point. The program then makes use of an optimization routine and an internal constraint check to determine an optimal solution. The configuration analysis employs an iteration process to produce a performance analysis of a helicopter. An extremely wide range of performance options are available.
3.2. Requirements Driven Fuselage Design Program (RDFD)

RDFD can estimate the empty weight and flat plate drag area to satisfy the requirements of the mission. RDFD is based upon a set of regressions from historical data, and semi-empirical physics-based equations for geometry calculations. It estimates the major component weights depending on fundamental design parameters and also estimates the geometric scale of the vehicle depending on payload requirements and required fuel amount. The geometric scale of the vehicle refers to the external surface area and the internal volume of each vehicle component. This geometry is used to predict the aerodynamic flat plate drag area of each component, and finally to apply the component-build-up method for the vehicle’s total flat plate drag and downloading value.

The method for the helicopter empty weight estimation has been adopted from the method of Boeing-Vertol, RTL. These formulas are statistically sophisticated semi-empirical equations dependent on the vehicle’s physical geometric and/or non-geometric characteristics. The method has shown to be an accurate prediction of empty weight. To estimate the component-build-up drag, streamlined component estimations are performed based on calculated surface area, and the estimation for non-streamlined components is based on regression to historical gross-weight dependent trends. For streamlined components, a more refined geometric description of the sub-components is required. To facilitate these estimated parameters, a statistical method and genetic algorithm have been used to build the Equivalent Airframe Model (EAM). EAM creates sub-component models based upon the original Rapid Helicopter Model (RHM) mesh, and then calculates the flat plate drag of each sub-component, treating it as a simple shape with known characteristic drag models. Once the component drags have been accumulated, the flat plate drag area of the entire aircraft, with variation of attack angle is predicted.

3.3. Integration by Model Center

GTPDP is able to perform better analyses for compound configurations when involving information such as empty weight and flat plate drag area scaling by RDFD. Simultaneously, RDFD can execute more accurate analysis when gross weight, fuel weight, engine size and vehicle geometry are given by GTPDP. Because of the coupling variables, the analysis requires the environment which can integrate two tools and guarantee the convergence. The ModelCenter provides a proper environment, which is interactive and automated. In addition, it has the tool ‘converger’ which can solve the matter of coupling variables. Figure 3.2 shows the environment for the preliminary design of the compound helicopter.

3.4. Calibration of Baseline Model

As a first step, the baseline helicopter, the Augusta Westland Super Lynx 300, was sized and calibrated in the Model Center environment. From the graph of Payload and range (Figure 3.3), the calibration point is selected as 800 kg (1763 lb) and 210 NM. The specification at this point is described in the Table 3-1.

<table>
<thead>
<tr>
<th>Category</th>
<th>Values</th>
</tr>
</thead>
<tbody>
<tr>
<td>Empty Weight</td>
<td>6312 lb</td>
</tr>
<tr>
<td>Payload</td>
<td>1763 lb</td>
</tr>
<tr>
<td>Crew Weight</td>
<td>400 lb</td>
</tr>
<tr>
<td>Fuel Weight</td>
<td>1735 lb</td>
</tr>
<tr>
<td>Gross Weight</td>
<td>10210 lb</td>
</tr>
<tr>
<td>Disk Loading</td>
<td>7.370 lb/ft²</td>
</tr>
</tbody>
</table>
The mission profile, which includes 210NM flight with 1763 lb payload, can be determined from Figure 3.4.

By using GTPDP and RDFD, the conceptual model of Super Lynx 300 was developed. The results showed little error and proved the model was accurately calibrated. Table 3-2 shows the calibrated results and actual data. From this, it is clear that the modeling of the aircraft using these methods is accurate and even the maximum speed of Super Lynx 300, including the safety margin, is relatively accurate.

Table 3-2: Calibration Result of Baseline

<table>
<thead>
<tr>
<th></th>
<th>SL 300</th>
<th>GTPDP</th>
<th>Error (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>IRP (HP)</td>
<td>810</td>
<td>809.5</td>
<td>0.1</td>
</tr>
<tr>
<td>MCP (HP)</td>
<td>633.5</td>
<td>633.1</td>
<td>0.1</td>
</tr>
<tr>
<td>Gross Weight (LB)</td>
<td>10210</td>
<td>10208.8</td>
<td>0.0</td>
</tr>
<tr>
<td>Max. Endurance (Hr)</td>
<td>3</td>
<td>2.951</td>
<td>1.7</td>
</tr>
<tr>
<td>Max. Speed (kts)</td>
<td>150</td>
<td>156</td>
<td>3.8</td>
</tr>
</tbody>
</table>

Figure 3.5 shows the graph of power required versus forward speed of Super Lynx 300. MCP, IRP and maximum speed are almost same as those of actual data.
From the baseline model, the design goals dictated that the target should have 20% more payload, 20% more range, 250 knots flight speed, and 20,000 ft service ceiling. The aforementioned four candidates are developed and evaluated to confirm the advantages and disadvantages of each candidate. A trade study was then performed for these candidates based on: empty weight, maximum dash speed, maximum cruise speed, and available engine power. For initial design selection, gross weight is assumed to be reduced by 15% because of composite material use.

3.5.1. Empty Weight Trade Study

First, the performance was compared in the case that the weights of four candidates were limited to 10,000 lbs. Here candidate 4 shows the best flight speed and endurance. In Figure 3.7, the performance of four candidates are compared and Figure 3.6 shows the graphs of power required versus forward speed.
3.5.2. Cruise Speed Trade Study

The configuration and performance are compared in the case that the cruise speed of four candidates is required as 215 knots. Candidate 4 shows the lightest gross weight, the best flight speed and endurance despite the slightly higher drag. In Figure 3.9, the performance of the four candidates is compared and Figure 3.8 shows the graphs of power required versus forward speed. The coaxial rotorcraft with compounding and slowed rotor shows the lowest empty weight for the same cruise speed.

Figure 3.8: Power Required vs. Forward Speed Graph of Candidates
3.5.3. Dash Speed Trade Study

The configuration and performance are compared in the case that the dash speed of four candidates is required as 250 knots. The result shows that the rotorcraft can not reach 250 knots without slowed rotor. The candidate 4 shows the lightest gross weight, the best flight speed and endurance. In Figure 3.11, the performance of four candidates was compared and Figure 3.10 shows the graphs of power required versus forward speed. The graphs of candidate 1 and 3 are not illustrated because the result shows that the rotorcraft can not reach 250 knots without slowed rotor. Coaxial rotorcraft with compounding and slowed rotor shows the lowest empty weight for the same dash speed.
3.5.4. Limited Engine Power Trade Study

The configuration and performance are compared in the case that the maximum engine power of four candidates is limited as 3441 HP. Likewise, the candidate 4 shows the lightest gross weight, the best flight speed and endurance instead of higher drag. In Figure 3.13, the performance of four candidates are compared and Figure 3.12 shows the graphs of power required versus forward speed. Coaxial rotorcraft with compounding and slowed rotor shows the highest flight speed with the same engine power.

---

**Figure 3.12: Power Required vs. Forward Speed Graph of Candidates**

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**Figure 3.13: Comparison of Performance**

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3.6. Optimization

The coaxial compound with a slowed rotor was selected because of the advantages: light gross weight, higher speed, less required horse power, less fuel consumption, slower rotor tip speed, and longer endurance time. This design was then further optimized and subject to the engineering judgment and historical trend based constraints listed below:

**Objective:** Minimize Gross Weight

**Constraints:**

\[ 6.5 \leq \text{Disk Loading} \leq 9.5 \]
\[ \text{Required horse power} \leq 3441 \]

In addition, to consider the new technology, three assumptions were employed. First, the empty weight can be reduced by 15% by using composite materials. Second, fuselage drag can be reduced by 25% from boundary layer control as explained in this section. Last, specific fuel consumption was increased by 10% using the newly selected engine.

The optimal solution was found to the point with higher disk loading. As a result, the gross weight is reduced to 7572 lb where power required is 3441 shp. In Table 3-3, the new model is compared to baseline and the normalized values are shown in Figure 3.14 and Figure 3.15 shows the graphs of power required versus forward speed of the new model.

<table>
<thead>
<tr>
<th></th>
<th>Super Lynx 300</th>
<th>New Model</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Empty weight (lb)</strong></td>
<td>6312</td>
<td>7572</td>
</tr>
<tr>
<td><strong>Drag – Fuselage (ft(^2))</strong></td>
<td>7.17</td>
<td>5.95</td>
</tr>
<tr>
<td><strong>Drag - Landing gear (ft(^2))</strong></td>
<td>4.4</td>
<td>3.4</td>
</tr>
<tr>
<td><strong>Drag - Hub (ft(^2))</strong></td>
<td>4.03</td>
<td>6.03</td>
</tr>
<tr>
<td><strong>IRP (shp)</strong></td>
<td>1620</td>
<td>3100</td>
</tr>
<tr>
<td><strong>MCP (shp)</strong></td>
<td>1267</td>
<td>2552</td>
</tr>
<tr>
<td><strong>Disk Loading</strong></td>
<td>7.37</td>
<td>9.07</td>
</tr>
<tr>
<td><strong>Rotor tip speed</strong></td>
<td>650</td>
<td>330</td>
</tr>
<tr>
<td><strong>Payload (lb)</strong></td>
<td>1763</td>
<td>2115.6</td>
</tr>
<tr>
<td><strong>Range (NM)</strong></td>
<td>210</td>
<td>252</td>
</tr>
<tr>
<td><strong>Max speed (kts)</strong></td>
<td>150</td>
<td>252</td>
</tr>
<tr>
<td><strong>Cruise speed (kts)</strong></td>
<td>132</td>
<td>226</td>
</tr>
<tr>
<td><strong>Fuel Required (lbs)</strong></td>
<td>1732.3</td>
<td>1948.1</td>
</tr>
<tr>
<td><strong>Service ceiling (ft)</strong></td>
<td>10750</td>
<td>21000</td>
</tr>
</tbody>
</table>

![Figure 3.14: Comparison of Baseline and New Model](image-url)
The result of conceptual design phase is not an exact value. However, the candidate can be selected with reduced time and cost factors. In addition, if the assumed values are provided from high fidelity analysis later, the accuracy becomes higher and the result can reach the best answer during the iterative process.

4. Conceptual Design Validation

At the beginning of design stage, the conceptual design was performed by GTPDP and RDFD. These programs use historical data and physics-based simplified equations. Therefore, the design needs to be confirmed with high fidelity analysis as well. This high fidelity analysis is compiled in Section 14. Table 4-1 describes a part of the assumed and analyzed values. Except the data in Table 4-1, tail propfan performance and the aerodynamic properties at each Mach number were updated to reflect the design configuration of the Peregrine.

<table>
<thead>
<tr>
<th></th>
<th>Assumed values</th>
<th>Analyzed values</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flat plate drag area (ft$^2$)</td>
<td>11.98</td>
<td>13.93</td>
</tr>
<tr>
<td>Transmission efficiency</td>
<td>0.95</td>
<td>0.98</td>
</tr>
<tr>
<td>IRP (HP)</td>
<td>3442</td>
<td>3958</td>
</tr>
<tr>
<td>MCP (HP)</td>
<td>2552</td>
<td>2880</td>
</tr>
</tbody>
</table>

The drag of blade is lower and available engine power is higher than assumed value. However, because the flat plate drag area is calculated higher, overall performance is reasonably similar to the prediction at the conceptual design stage.

![Mission Profile of Peregrine](image_url)
The mission profile for Rf method is shown in Figure 4.1. As specified in the design goals, this mission profile was based on 20% more payload and mission range than those of baseline helicopter Super Lynx 300. Required fuel for the mission was estimated as 2272 lb. Service ceiling was calculated to be 15813 ft at standard air condition. Maximum range and endurance are 314 NM and 2.53 hour, respectively, at sea level standard condition. Figure 4.2 shows the graph of required power versus forward speed. Maximum dash speed at intermediate engine power is 249.8 knots, and the cruise speed at maximum continuous power is 222.5 knots at the sea level standard condition. Maximum dash speed at intermediate engine power is 225.3 knots, and the cruise speed at maximum continuous power is 185.6 knots at the 5000 ft height and 95 °F. For initial testing of the Peregrine model, the rotor speed was assumed to be changed at 100 knots. This transition of rotor speed allowed the Peregrine to have the advantages of higher flight speed without sacrificing the performance in the low flight speed regime. As further modeling occurred during the design process, it was determined that the transition of the rotor speed would occur at the maximum endurance airspeed where required power would be minimized throughout the transition process.

a) Sea Level Standard Condition (0ft 59°F)

Figure 4.2: Required and Available Power .vs. Forward Speed

Most of the performance indices were improved from the baseline except for maximum endurance. Most notably, the flight speed was increased by approximately 1.6 times over the baseline. In the case of endurance time, the
Peregrine’s additional propulsion system for high flight speed does not add to the endurance flight time. The comparison of main performance between the baseline and Peregrine is shown in Figure 4.3.

Figure 4.3: Performance Comparison of the Baseline and the new model

To confirm the improvement of performance with the alternative drive system, the performance of Peregrine was compared to that of the baseline rotorcraft by sizing only. This sizing of derivative rotorcraft was executed based on the same mission profile until it had the same empty weight and engine performance to allow for an accurate side-by-side comparison. This comparison of specification of the sized rotorcraft and Peregrine is shown in Figure 4.4.

Figure 4.4: Performance Comparison of the Sized Rotorcraft and the new model

The trend of the results is the same as the comparison of the baseline and the Peregrine. Because Peregrine has almost 1.3 times higher flight speed, the required time for the mission is reduced by 9.7%. In addition, significantly less fuel is consumed during the mission. A standard helicopter must tilt its thrust vector forward to increase its forward speed. This forward tilt also tilts the fuselage forward, increasing the parasitic drag of the body. Because Peregrine can keep the same equivalent flat plate drag area in all flight speed regime, the decreased parasite drag causes the 34% reduction of fuel consumption. This decreased required power can be confirmed in Figure 4.5.
5. **Transmission**

5.1. **Transmission Requirements**

The main objective of the transmission design was to obtain a feasible design power and operate the main rotor at two distinct speeds as well as the pusher propeller.

5.2. **Concept Availability**

In order to satisfy these requirements, multiple designs were considered (both conventional and non-conventional) including:

1. 2-Speed Planetary System—Conventional dual planetary system with clutch system to change output member.
2. Pericyclic Continuously Variable Transmission (CVT) — A non-conventional transmission unit capable of varying the gear ratio using pericyclic motion.
3. Ball CVT — A traction based continuously variable transmission.
4. Planetary CVT — Planetary system with controlled variable ring speed system.
5. Cone CVT — A belt / chain driven gear system using two geometrically varying conical gears.

Two-speed designs are less complex compared to variable speed designs but possess inherent power interruptions during speed ratio transition. Rotorcraft application requires positive and continuous power transfer and variable speed. The major portions of a mission are hover and cruise. Transition between the above operation points is a minor portion of the flight mission but very important to the handling qualities of the aircraft. Two-speed designs can be adaptable to be quasi-variable through variable transition assist, which may be either external powered (controller) or internally take-off driven (variator - traction drive or power electronics motor-generator system). Speed range changes for CVTs need to be computer controlled sensing both transmission and engine speed/power.

5.3. **Concept Selection**

TOPSIS (Technique for Ordered Preference by Similarity to Ideal Solution) method of multi criteria decision-making was used to identify and select the best concept. The TOPSIS used specific criteria to evaluate these concepts:
1. Weight – overall weight of the system
2. Reliability
3. Non-conventionality
4. System and sub-system feasibility
5. Down-shifting performance
6. Overall efficiency of the system
7. Life-cycle cost
8. Size – total volume of the system
9. Part Count

Table 5-1: TOPSIS Weights

<table>
<thead>
<tr>
<th>Criteria</th>
<th>Weightings</th>
<th>Normalized weights</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight</td>
<td>3</td>
<td>0.18</td>
</tr>
<tr>
<td>Reliability</td>
<td>9</td>
<td>0.55</td>
</tr>
<tr>
<td>Non-conventionality</td>
<td>9</td>
<td>0.55</td>
</tr>
<tr>
<td>Feasibility</td>
<td>6</td>
<td>0.36</td>
</tr>
<tr>
<td>Down-shifting performance</td>
<td>3</td>
<td>0.18</td>
</tr>
<tr>
<td>Efficiency</td>
<td>6</td>
<td>0.36</td>
</tr>
<tr>
<td>Cost</td>
<td>3</td>
<td>0.18</td>
</tr>
<tr>
<td>Size (volume)</td>
<td>3</td>
<td>0.18</td>
</tr>
<tr>
<td>Part Count</td>
<td>1</td>
<td>0.06</td>
</tr>
</tbody>
</table>

Scores were then generated for the concepts for each criterion. From the criteria scores, the analysis of these results follows after Figure 5.1 in Section 5.4.

Table 5-2: Concepts initial criteria scores

<table>
<thead>
<tr>
<th>Concepts</th>
<th>Weight</th>
<th>Reliability</th>
<th>Non-conventionality</th>
<th>Feasibility</th>
<th>Down-shifting performance</th>
<th>Efficiency</th>
<th>Cost</th>
<th>Size (volume)</th>
<th>Part Count</th>
</tr>
</thead>
<tbody>
<tr>
<td>2-speed</td>
<td>3</td>
<td>6</td>
<td>1</td>
<td>6</td>
<td>1</td>
<td>9</td>
<td>6</td>
<td>9</td>
<td>6</td>
</tr>
<tr>
<td>P-CVT</td>
<td>6</td>
<td>6</td>
<td>9</td>
<td>5</td>
<td>6</td>
<td>6</td>
<td>1</td>
<td>6</td>
<td>6</td>
</tr>
<tr>
<td>Ball CVT</td>
<td>3</td>
<td>1</td>
<td>6</td>
<td>6</td>
<td>3</td>
<td>6</td>
<td>6</td>
<td>6</td>
<td>1</td>
</tr>
<tr>
<td>Cone CVT</td>
<td>1</td>
<td>3</td>
<td>6</td>
<td>3</td>
<td>9</td>
<td>6</td>
<td>3</td>
<td>1</td>
<td>6</td>
</tr>
<tr>
<td>Planetary-CVT</td>
<td>6</td>
<td>9</td>
<td>6</td>
<td>6</td>
<td>9</td>
<td>6</td>
<td>3</td>
<td>3</td>
<td>3</td>
</tr>
</tbody>
</table>

Normalizing the above and multiplying by the weights:

Table 5-3: Concepts final criteria scores

<table>
<thead>
<tr>
<th>Concepts</th>
<th>Weight</th>
<th>Reliability</th>
<th>Non-conventionality</th>
<th>Feasibility</th>
<th>Down-shifting performance</th>
<th>Efficiency</th>
<th>Cost</th>
<th>Size (volume)</th>
<th>Part Count</th>
</tr>
</thead>
<tbody>
<tr>
<td>2-speed</td>
<td>0.057</td>
<td>0.257</td>
<td>0.040</td>
<td>0.211</td>
<td>0.013</td>
<td>0.210</td>
<td>0.101</td>
<td>0.128</td>
<td>0.034</td>
</tr>
<tr>
<td>P-CVT</td>
<td>0.115</td>
<td>0.257</td>
<td>0.357</td>
<td>0.176</td>
<td>0.079</td>
<td>0.140</td>
<td>0.017</td>
<td>0.086</td>
<td>0.034</td>
</tr>
<tr>
<td>Ball CVT</td>
<td>0.057</td>
<td>0.043</td>
<td>0.238</td>
<td>0.035</td>
<td>0.079</td>
<td>0.070</td>
<td>0.101</td>
<td>0.086</td>
<td>0.006</td>
</tr>
<tr>
<td>Cone CVT</td>
<td>0.019</td>
<td>0.128</td>
<td>0.238</td>
<td>0.106</td>
<td>0.115</td>
<td>0.140</td>
<td>0.050</td>
<td>0.014</td>
<td>0.034</td>
</tr>
<tr>
<td>Planetary-CVT</td>
<td>0.115</td>
<td>0.385</td>
<td>0.238</td>
<td>0.211</td>
<td>0.079</td>
<td>0.210</td>
<td>0.101</td>
<td>0.043</td>
<td>0.017</td>
</tr>
<tr>
<td>Ideal Solution</td>
<td>0.115</td>
<td>0.385</td>
<td>0.357</td>
<td>0.211</td>
<td>0.119</td>
<td>0.210</td>
<td>0.101</td>
<td>0.128</td>
<td>0.034</td>
</tr>
</tbody>
</table>

Figure 5.1: TOPSIS Results
Figure 5.1 shows that the planetary Continuously Variable Transmission (CVT) is the ideal solution. It was closely followed by the Pericyclic – CVT but the planetary system was chosen because of its combination of smaller size, lower part count, lower cost, and greater reliability.

5.4. Planetary CVT Operations

The Differential Planetary Drive, shown in Figure 5.2, capitalizes on the output variability of a dual-input to single-output planetary differential using one input to serve as a controller. Primary power is input to the sun gear, output power is transferred thru the carrier, and speed variation is achieved by varying the speed of a special ring gear from zero speed to full speed with a variable speed controller device/system.

The ring gear is unique and nonconventional in that it has both an internal pitch diameter and an external pitch diameter contained within an integral ring. As depicted, ring gear speed is varied from zero to full RPM by a speed controller driving the external pitch diameter. The controller ratio may be varied in design permitting selection of the optimal power and speed range. The controller rotates in the opposite direction of the primary input. The speed controller may be a variety of possible devices either externally powered and controlled or take-off driven from the transmission power input shaft. The power take-off may be a continuously variable speed device as suggested for the other configurations. The main disadvantage is the power loss to spin/control the ring speed.

Table 5-4 Planetary CVT Drive Settings

<table>
<thead>
<tr>
<th>Mode</th>
<th>Tip Speed</th>
<th>Input</th>
<th>Output</th>
<th>Ring Gear Speed</th>
</tr>
</thead>
<tbody>
<tr>
<td>High Speed</td>
<td>650 ft/s</td>
<td>Sun Gear</td>
<td>Planet Carrier</td>
<td>0 – Locked</td>
</tr>
<tr>
<td>Low Speed</td>
<td>400 ft/s</td>
<td>Sun Gear</td>
<td>Planet Carrier</td>
<td>391 RPM</td>
</tr>
</tbody>
</table>

5.5. Drive Train Design

The design of the gearbox followed the recommended procedure by the American Gear Manufactures Association (AGMA) as outlined in Andrew Bellochio’s Design Thesis,¹¹ and in Robert Norton’s Machine Design Textbook.¹² 30° Helical gears with 25° full tooth loading were selected for use in the transmission due to their quieter running noise and better capability for high load and high speed applications than standard transmission configurations. This is because the helical curve of the gear tooth ensures a gradual loading on the tooth. In the sizing discussion, a naming convention is used for ease of reference. This convention refers to the gears as numbers 1-12 and system, or sets of gears, as A-E according to Figure 5.3.
5.6. Drive Train Sizing Parameters

Even with the helical and loading angles selected the transmission design outlined above still had 25 independent variables. The gear ratio and diametrical pitch (Pd) applies to a gear set (A-E) and the material selection is done independently for each of the 12 gears. These variables are listed below:

- **Pd** = diametric pitch, number of teeth per inch, for each gear
  - Pd = [1, 1.25, 1.5, 1.75, 2, 2.5, 3, 4, 5, 6, 8, 10, 12, 14, 16, 18]

- **Gear Ratios**
  - Engine Gear Box: 1 – 3
  - System A: 1-10
  - System B: 1-10
  - System C: 1-4
  - System D: 0.2-5
  - System E: 0.2-5

- **Number of Planets in System B**: [2, 3, 4]
- **Material for each gear**: [AISI 9310, VASCO X2M, Pyro-wear 53]

These parameters were selected based on knowledge of other gear designs. The diametrical pitch (Pd) set listed is the most commonly manufactured Pd’s for course gears that are used in aircraft transmissions. The gear ratio ranges were selected based on the type of gears used in the system. Spur gears typically can withstand ratios up to 10:1 and bevel gear systems can withstand rations up to 1:5. System C was limited to 1:4 due to the high loads it would see since it was the last stage of reduction to the rotor. The three material selections are common gear materials used in the main transmission. All three materials are steel alloys and are case-hardened and carburized. Their material properties are shown in Table 5-5.

For sizing, the selection process used this design criteria:

- Engine operated at 3100 HP (3442 Contingency Power), 23,000RPM (plus a 20% factor of safety)
- Rotor tip speed: 680 ft/s in High Mode, 400 ft/s in Low Mode.
- Transmission component life: 3,500 flight hours

5.7. Drive Train Sizing with Genetic Algorithm

A Matlab program was written to optimize the weight of the gear train given a certain set of input parameters as outlined above. The equations and gear factors used followed the recommend practices of the AGMA as outlined in Robert Norton’s book. The program’s general design was object oriented in nature. Each gear was sized in its own program and the parameters passed to the main program. In order to ensure the rotor system and propeller each spun at the required RPM the gear ratio of system B and E were calculated given the input gear ratio of the rest of the gears. This left 23 independent variables.

Within each gear sizing program, the general flow was to step through the number of teeth of the pinion, calculate the number of teeth of the gear, and then for each tooth set, increased the face width until the bending and contact stress were less than the allowable stress. The program then estimated the weight for that gear set. This was done for every tooth combination and the lowest weight gear’s parameters (teeth of the pinion and gear, face width, diameters of each gear, and stresses of each gear) were passed back to the main program which then added up the weight of the entire system. Thus, the Matlab program was able to optimize the numbered teeth diameter and face width for the lowest overall weight. However, there were 23 independent variables which also had to be optimized.
In order to optimize these input parameters a Genetic Algorithm was set up in Model Center. A genetic algorithm is a stochastic search method that is specifically designed for discrete variables and because it is stochastic it has an increased probability of finding a global optimum in the presence of many local optimums. This was very fitting for the transmission sizing since, given the number of variables there was a very high likelihood of many local optimums and 18 of the 23 variables were discrete by their very nature (Pd of each system, material of each gear, and number of planets in system b.) When the continuous variables were discretized to two decimal places there were over $4.598 \times 10^{34}$ possible combinations. If a full factorial design was conducted and a computer was capable of running 1000 designs a second, it would still take $1.46 \times 10^{24}$ years to test all possible combinations. Thus, the genetic algorithm sizing was a substantial technology improvement in and of itself in that the transmission design could be optimized when it might otherwise not have been able to do so. A general procedure of how a typical genetic algorithm is run is shown in Figure 5.5. The genetic algorithm that was run to size the transmission had several variations from this general procedure in order to help it converge faster.
The transmission algorithm used a “multiple elitist” strategy. This ensured that the top 1/3 of the designs were automatically promoted to the next generation. The other 2/3 were generated as detailed above but also used a tournament selection method. With this method two parent candidates were selected based off the normalized fitness but only the best candidate from those two was promoted to become a parent. This helps to ensure that the parent population is made up of the best individuals. The other model parameters for the actual transmission genetic algorithm are summarized in Table 5-6. Another benefit to using the multiple elitist strategy is that the cross over probability can be 100%. This means that the bottom 2/3 of the designs are guaranteed to change but the best designs are preserved in the top 1/3 elite. This also helps to speed up the convergence of the program and uses an exploitive versus exploratory type method to find the best optimum while ignoring other local optimums.

Table 5-6: Transmission Genetic Algorithm Parameters

<table>
<thead>
<tr>
<th>General Parameters</th>
<th>Probability of Cross Over</th>
<th>Probability of Mutation</th>
</tr>
</thead>
<tbody>
<tr>
<td>Population Size</td>
<td>Probability of Cross Over</td>
<td>Probability of Mutation</td>
</tr>
<tr>
<td>Continuous</td>
<td>Discrete</td>
<td></td>
</tr>
<tr>
<td>100</td>
<td>100</td>
<td>0.1</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Constraints</th>
<th>Gear</th>
<th>Constraint</th>
<th>Reason</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gear</td>
<td>5 – Ring Gear</td>
<td>Max Diameter = 22 in</td>
<td>Engine Air intake obstruction</td>
</tr>
<tr>
<td>Gear</td>
<td>7+8 – Rotor Bevel Gears</td>
<td>Max Diameter = 30 in</td>
<td>Fuselage width – Drag Reduction</td>
</tr>
<tr>
<td>Gear</td>
<td>6 – Input to rotor bevel gears</td>
<td>Max Diameter = 10 in</td>
<td>Fuselage height – Drag reduction</td>
</tr>
</tbody>
</table>

The results of the genetic algorithm produced an optimum gear set for the above parameters. The program took 113 generations to converge and built approximately 6.24 million transmissions in the process. The convergence history is shown at the right for the final sizing run. The parameters for this gear set are shown in Table 5-7. As can be seen from these parameters the constraints in gears 6, 7, and 8 are active and gears systems A and C were sized primarily on contact stress while the rest were sized based on bending stress. Also, note that there is an inherent 20% factor of safety in all actual stress estimates.

Table 5-7: Optimized Gear Parameters

<table>
<thead>
<tr>
<th>System Parameters</th>
<th>Gear Parameters</th>
<th>Helical Angle (deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Gears</td>
<td>Material</td>
</tr>
<tr>
<td>Gear ID</td>
<td>System ID</td>
<td>Mach Adv</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Mixing Gear System</td>
<td>A</td>
<td>0.5048</td>
</tr>
<tr>
<td>North Planetary System</td>
<td>5</td>
<td>0.2035</td>
</tr>
<tr>
<td>Torsion Bevel Gear System</td>
<td>C</td>
<td>0.005</td>
</tr>
<tr>
<td>South Bevel 1</td>
<td>D</td>
<td>0.0958</td>
</tr>
<tr>
<td>South Bevel 2</td>
<td>E</td>
<td>0.2322</td>
</tr>
</tbody>
</table>

Figure 5.6: Genetic Algorithm Convergence History
Table 5-8: Gear Bending and Contact Stress

<table>
<thead>
<tr>
<th>System Parameters</th>
<th>Gear Parameters</th>
<th>Contact Stress (psi)</th>
<th>Bending Stress (psi)</th>
</tr>
</thead>
<tbody>
<tr>
<td>System</td>
<td>Gear ID</td>
<td>RPM</td>
<td>Actual</td>
</tr>
<tr>
<td>Mixing Spur Gears</td>
<td>1</td>
<td>11128</td>
<td>198.640</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>3534.6</td>
<td>100.190</td>
</tr>
<tr>
<td></td>
<td>3</td>
<td>3534.6</td>
<td>197.490</td>
</tr>
<tr>
<td></td>
<td>4</td>
<td>3525.9</td>
<td>192.490</td>
</tr>
<tr>
<td></td>
<td>5</td>
<td>0</td>
<td>65.590</td>
</tr>
<tr>
<td>North Planetary System</td>
<td>B</td>
<td>6</td>
<td>885.36</td>
</tr>
<tr>
<td></td>
<td>7</td>
<td>295.57</td>
<td>211.110</td>
</tr>
<tr>
<td></td>
<td>8</td>
<td>295.57</td>
<td>211.110</td>
</tr>
<tr>
<td>Torque Bevel Gear System</td>
<td>C</td>
<td>9</td>
<td>3534.6</td>
</tr>
<tr>
<td></td>
<td>10</td>
<td>3204.6</td>
<td>198.240</td>
</tr>
<tr>
<td></td>
<td>11</td>
<td>3304.6</td>
<td>207.700</td>
</tr>
<tr>
<td></td>
<td>12</td>
<td>2500</td>
<td>207.700</td>
</tr>
</tbody>
</table>

5.8. Final Transmission Design

![Transmission Schematic](image)

Figure 5.7: Transmission Schematic
Figure 5.8: Complete Transmission CATIA Model

Figure 5.9: Planetary CVT CATIA Model
5.9. **Clutch System**

The clutch system is required to allow the ring gear to operate stationary at 100% rotor speed and to rotate in order to reduce the tip speed during high speed flight. The main transmission housing which houses the planetary gearbox has four clutch bands. Two electric actuators control each clutch band. The actuators have a spring mechanism, which engage the clutches in case of failure. Each clutch is spring mounted to the ring gear and actuator power is required to disengage the clutch. This helps to ensure that a failure will result in high tip speed operations.

5.10. **Power Electronics Module (PEM)**

The PEM is the heart of the transmission control system. It controls the sequence of clutch engagement and gradual slowing of drive shaft rpm. It translates signals into precisely timed voltages, telling the motor to respond with the proper speed, direction of rotation and torque.

The PEM uses contactless optical speed encoders. The optical speed encoder shines a beam of light from a transmitter across a small space and detects it with a receiver at the other end. If a disc is placed in the space (spur gear with no helical gears), which has slots cut into it, then the signal will only be picked up when a slot is between...
the transmitter and receiver. If a reflector patch is mounted on the drive shaft, the transmitter/emitter will calculate the rpm based on the signal received.

The PEM monitors the voltage delivered by the Helicopter Electrical system the speed of rotation of the motor and final speed of the drive shaft and the temperatures of the motor and power electronics.

![Figure 5.12: Power Electronics Module](image)

Table 5-9: Motor Specifications

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Max. Power</strong></td>
<td>124</td>
<td>hp</td>
</tr>
<tr>
<td><strong>Max. Speed</strong></td>
<td>14,000</td>
<td>rpm</td>
</tr>
<tr>
<td><strong>Weight</strong></td>
<td>70</td>
<td>lbs</td>
</tr>
</tbody>
</table>

*3-phase, 4-pole, AC induction air-cooled, 375 volt, variable frequency drive electric motor

5.11. **Gear Stress Analysis with Simulia**

As a second stress analysis each gear was built in Catia and then tested in Simulia. The test was done at max power plus an additional 20% factor of safety. In this analysis the entire gear was able to be tested including the shaft connection points as opposed to the face and tooth stress analysis done above. These results showed that all gears were within allowable stress tolerances designated by AMGA. Figure 5.13 is an example of two of these analyses.

![Figure 5.13: Gear Stress Analysis Simulation](image)
6. Main Rotor Blade and Hub Design

In the design of the Peregrine rotor blades, emphasis was made on achieving a forward speed of 250 knots. The rotors are responsible for the thrust and the propulsion until the maximum endurance speed of 100 knots after which the ducted pusher propeller provides the forward thrust for the aircraft.

The Super Lynx has a BERP Tip rotor of radius 21 feet rotating at a speed of 650 ft/sec. The effective chord of the blade capable of lifting target mission weight at 420 ft/sec is 1.7 feet at the inboard section.

A Blade Element Theory was used in the analysis. A uniform inflow model as well as a linear inflow model was considered. The Super Lynx uses a blade whose root cut-out is about 25% of the rotor radius. The main disadvantage in using a co-axial rotor is the separation between the rotors results in high hub drag. To alleviate this, it was necessary to keep the distance between the two rotors at a minimum and thus, use stiff composite blades. While modeling the blade in Blade Element Theory, the elasticity of the blade was accounted by using a vertical hinge until such a length whose flapping frequency would be near that of the stiff blade.

![Mach No - Advancing Side](image1)

![Mach No - Retreating Side](image2)

Figure 6.1: Advancing and Retreating Blade Mach Numbers at 250 knots, SLS

Due to the reduction in tip speed, the rotor encounters a Mach number distribution on the advancing side that is seen on most rotorcraft which operate at lower speeds. However, at the retreating side, the entire blade experiences reverse flow. The reverse flow region leads to high drag and produces negative lift.

6.1. Initial Airfoil Selection

Modern rotor blades use multiple airfoil sections as well as non-linear twist in some sections of the blade planform. At the initial stage, a simple survey of the airfoils available in GTPDP was conducted while assuming a rectangular planform and single airfoil. A fixed configuration of 18000 lbs gross weight, 30 ft rotor radius, -10 degrees twist, and 0.09 solidity were used since the final configuration was not chosen at this point. The Boeing VR8 airfoil is clearly shown to be the airfoil that provides the best performance in hover, low speed, and high speed flight as it is the lowest power required curve over the entire range of the graph in Figure 6.2.

![Airfoil Trade Study](image3)

Figure 6.2: Airfoil Trade Study
6.2. Final Airfoil Selection

Trade studies using GTPDP indicated that the VR7 and VR8 airfoils give the best performance in terms of lift versus drag. Hence these airfoils were more carefully considered. For a forward speed of 250 knots, the Cl/Cd ratio was computed for both the airfoils along the blade at a blade azimuth position of 90°. Until about 0.8 of the radius, the thicker VR7 airfoil exhibits a higher Cl/Cd ratio. After that, all the way to the tip, the Cl/Cd of the VR8 is higher in magnitude mainly due to its thinner profile. It is noted that the Cl of the airfoils under consideration are roughly the same and the difference in magnitude of the ratios is due to the lower Cd of the VR8. Thus, it was decided that the blade was to have the VR7 planform until 70% of the rotor radius and the VR8 for the remaining blade section. A transition is between the two airfoil section occurs between 0.70 to 0.75 R.

Figure 6.3: Lift/Drag Ratios of Selected Airfoils

Figure 6.3 shows the Lift/Drag ratios of the airfoils separately and of the combination that is used.

6.3. Tip Speed Selection

Blade stall is a major limiting factor if the rotor is to be powered at high speed flight. Figure 6.4 shows the power predicted by GTPDP required to overcome blade stall at various tip speeds. GTPDP calculates the stall power empirically based on a set of coefficients supplied by the user which were found in the model calibration procedure described in Section 3. The power consumed by stall alone quickly becomes more than is feasible for the helicopter to supply even at speeds below 100 knots.

Stall power can be nearly eliminated by using a tip speed of 650 ft/s in the low speed, pure helicopter, non-compound range of operation. Figure 6.5 shows the power saved by using 650 ft/s and clearly shows the advantage in using two tip speeds.

Figure 6.4: Power Required Due to Blade Stall.

Figure 6.5: Comparison of Total Horsepower Required
The difference in power required becomes even more apparent when the power limits are plotted, assuming a maximum power available of 4000 hp at sea level. The speed limit in normal helicopter mode at sea level is 50 knots greater for 650 ft/s tip speed.

6.4. Solidity Selection

Blade radius and solidity are tradeoffs between figure of merit in hover and profile drag in forward flight. Higher solidity relieves blade loading and results in lower tip losses, but also increases profile power required. The horsepower requirement for different rotor configurations was compared at two cruise conditions, the Super Lynx 300’s standard 132 knot cruise speed, and the Future Lynx 250 knot cruise speed. The results show an improvement in power required to meet the cruise condition for increases in solidity as expected, so the optimized values of solidity and radius found in the vehicle sizing study and detailed rotor design will be used.

6.5. Final Blade Planform Design

The blade planform selected for the Peregrine is shown in Figure 6.8. From the blade root to 0.4 R, it has a constant chord of 1.3 ft. From 0.5 R to the tip, it is elliptical with the center at 0.7 R. The chord length at this section is 1.85 ft. From 0.4 R to 0.5 R, the blade transits from the rectangular planform to the elliptical shape.

In the paper by Ashish Bagai, a similar planform was selected over the tapered planform implemented on the X59A. At high speeds on the retreating side, higher drag is observed in the inboard section of the latter and hence to alleviate this, the chord near the root is reduced in the former. However, to maintain the same thrust weighted solidity at high forward speeds as the tapered blade, the X2TD blade is given an elliptical shape at the outboard sections.

A comparison between Peregrine blade planform and the Super Lynx blade is shown in Table 6-1. As the speed of the blade is slowed down to 420 ft/sec, the high amount of sweep given to the BERP tip blade becomes necessary. The BERP tip is specifically designed to perform as a swept tip at high Mach numbers and low angles of attack, though it is also designed to operate at high angles of attack without stalling. Since the Peregrine uses a co-axial rotor, the loss of lift on the retreating side of one rotor is countered by the advancing blade of the other rotor. Hence, the BERP tip is not considered in the calculations.
The power required by the rotors for the above mentioned planform is plotted in Figure 6.9. It can be seen that for the same thrust weighted solidity, the blade planform has a lower value. As the retreating side on the blade will fully be in the reversed flow region, high amount of drag is experienced at the 270 azimuth position. The drag on the inboard section is lower than that of the rectangular blade planform. The negative lift produced at this azimuth is reduced for the elliptical planform significantly for the elliptical planform. This will also help in reducing main rotor vibrations. Another advantage of the elliptical planform is the improved auto-rotational index since the blade mass is distributed more out-board, thereby increasing its moment of inertia about the rotor axis.

![Figure 6.9: Sectional Drag at $\psi = 270$ deg](image)

**Table 6-1: Blade Properties**

<table>
<thead>
<tr>
<th>Condition</th>
<th>Elliptical Blade</th>
<th>Super Lynx Blade</th>
</tr>
</thead>
<tbody>
<tr>
<td>Forward Flight, 250 knots</td>
<td>-511</td>
<td>-446</td>
</tr>
<tr>
<td>Lift/Blade, lbf at $\psi=270$</td>
<td>791</td>
<td>848</td>
</tr>
<tr>
<td>Hover</td>
<td>0.81</td>
<td>0.8</td>
</tr>
</tbody>
</table>

**Table 6-2: Section Drag Characteristics**

<table>
<thead>
<tr>
<th>Condition</th>
<th>At Hover</th>
<th>At 250 knots</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lift/Blade, lbf</td>
<td>Elliptical</td>
<td>Tapered</td>
</tr>
<tr>
<td>Figure of Merit</td>
<td>4638</td>
<td>3112</td>
</tr>
<tr>
<td>Drag at $\psi=270$</td>
<td>0.82</td>
<td>0.81</td>
</tr>
</tbody>
</table>

It is observed that although the sectional drag near the inboard is reduced, the total drag on the blade does not change much. However, the negative lift produced at $\psi = 270$ deg is reduced for the elliptical planform significantly.

At hover, the thrust produced along with the figure of merit is calculated for both the blade planforms. While the figure of merit does not significantly change, the thrust produced by the blades is significantly higher in the elliptical blade.

**7. Rotor Dynamics**

Rotor dynamics including structural design has two important criteria for coaxial rotorcraft. The first is a vibration problem like all other rotorcraft. The rotor of a helicopter in forward flight is under harmonic aerodynamic force because the blade of advancing side makes more lift than that of retreating side. In addition, because rotor has significantly large aspect ratio, the resonance of blade can cause malfunction and even break blades by severe vibration. Therefore, the frequency of rotor system should be examined by rotor dynamics.17

The second is a deformation problem by aerodynamic force. When the coaxial rotorcraft is in forward flight, the advancing side of lower rotor is under the retreating side of upper rotor. The structural deformation by aerodynamic force reduces the distance between two rotor systems. Therefore, the clearance is required in order to prevent the collision and this significantly increases the hub drag of coaxial rotorcraft and thus the parasite drag. The parasite drag increases with the cube of the velocity so reducing the parasitic drag and hub drag is a key component to a high speed helicopter.

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The analysis of rotor dynamics is highly coupled with structural analysis and aerodynamics. Aerodynamic force causes structural deformation and the vibration of blades. The centrifugal force of blades also effect on structural deformation. Conversely, the structural deformation changes the angle of attack of blades which is important characteristic in aerodynamics and the center of pressure which can have an effect on dynamic responses. Therefore, these three disciplines of rotor analysis need to be considered at the same time. Figure 7.1 shows the schematic diagram to simultaneously analyze aerodynamics, structural analysis and multi-body dynamics of rotor system.

The general procedure of rotor structural dynamics consists of three steps. First, each blade section models was built in ANSYS. The graphic user interface of ANSYS provides geometrical design and meshing tool that are needed in the process. By using script in ANSYS, the input files for VABS are prepared with geometric shape and material properties. Figure 7.2 shows one example of blade sections by ANSYS.

The rotor system of Peregrine mainly consists of three parts; Hingeless hub, VR-7 airfoil and VR-8 airfoil. To reduce the flapping, the hub supports the blade through bearings. This system can have a proper torsional stiffness to control and high bending stiffness for less clearance distance. Figure 7.3 shows the whole rotor system sectional design by the GUI of ANSYS. This rotor system is designed as stiff in plane for higher bending stiffness. Therefore, most of materials in the blade and hub are stiffer than usual blade of other rotorcraft. The materials of each part is described in Table 7-1.
Table 7-1: Material Properties in the Rotor System

<table>
<thead>
<tr>
<th>Material</th>
<th>Density (slug/ft³)</th>
<th>Young’s modulus (lbf/ft²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hub</td>
<td>Tungsten carbide</td>
<td>30.7</td>
</tr>
<tr>
<td>Bearing</td>
<td>Aluminum oxide</td>
<td>7.66</td>
</tr>
<tr>
<td>Spar</td>
<td>Graphite fiber</td>
<td>3.01</td>
</tr>
<tr>
<td>Filling</td>
<td>Aluminium hexweb</td>
<td>0.68</td>
</tr>
<tr>
<td>cover</td>
<td>Graphite fiber</td>
<td>3.01</td>
</tr>
</tbody>
</table>

Second, Variational Asymptotical Beam Sectional Analysis (VABS) is employed to calculate the cross-sectional properties including structural properties and inertia properties such as torsional/bending/shearing stiffness, shear center, center of mass, mass per unit span, and mass moments of inertia. These structural properties are the part of input file for DYMORE.

The last step is static analysis. DYMORE is selected to estimate structural deformation and dynamic response such as displacements, strain, forces, and velocity. DYMORE is a finite element based tool for the analysis of nonlinear flexible multibody systems. To predict the motion of rotor, DYMORE needs the data of aerodynamic forces, structural properties, rotor motion, and kinematical constraints. This analysis provides the distance between upper and lower rotor systems. In addition, the natural frequency of blade system can be analyzed by Eigen values. From this data, a fan plot was made to predict the vibration characteristics of the rotor.

Because each rotor system has four blades, 1P and 4P should be avoided and design emphasis was placed on reducing potential harmonics in this region. In the case of 1P, it does not cause any problem at all, because the first flapping mode has 1.3 times higher frequency than 1P in the all rotational speed regime. However, Peregrine is a rotorcraft with variable rotor speed main rotor. During transition from a rotor speed of 650 ft/s to 420 ft/s, the second flapping mode and the first lead-lag mode pass through the regime of 4P. Furthermore, the frequencies of second flapping and first lead-lag mode are very similar in the low speed mode. This can cause the coupling vibration. Both of these situations can result in instabilities in the rotor system. Therefore, the blade was stiffened by adjusting the position and shape of spar. Figure 7.4 shows the fan plot for the improved blade. In this design, the frequencies of second flapping and first lead-lag mode are always higher than 4P. In addition, the coupling vibration was also solved by separating the frequencies during all normal speed regimes.

![Figure 7.4: Fan Plot of Final Blade Design](image-url)
As mentioned before, the deformation of blade is also acquired as a function of time. To avoid potential blade contact, the hubs needed precisely calculated clearance distance to ensure the rotor tips have clearance. The simulation is performed under the sea level standard condition and 250 knots level flight. The maximum upward deformation of lower blade is 0.278 ft and the maximum downward deformation of upper blade is 0.101 ft. As a result, the clearance distance is decided as 1.2 ft which is about three times of maximum deformation, because the gust or sudden change of flight operation can make higher aerodynamic load.

8. **Acoustics**

Wop-Wop is a software program developed by PSU to calculate the various noise levels due to the blade. Using the data entered, it solves the Ffowcs Williams Hawkings Equations using Farassats formulations. A Fast Fourier Transform is applied to post process the data. The noise produced by the main rotors of the Peregrine is analyzed using Wop-Wop. Like any other computational analysis software, it can be divided into 3 stages.

I. Pre-processing or the stage where the rotor geometry is created, the aerodynamic forces acting on the blade are input and the observer grid is generated.

II. Processing or the actual computational analysis done by Wop-Wop to find the noise at the observer grid points.

III. Post-processing or the presentation of results in a form that can be easily analyzed by the analyst.

8.1. **Pre-Processing**

PSU Wop-Wop is object oriented. Certain features of Fortran 95 are used to define an object hierarchy that helps to build large, complex objects.

The core of Wop-Wop is a patch object. This is a surface which provides a grid to store input data and over which the aero-acoustic integration is carried out. Each patch can store different types of data like chord-wise loading vectors, surface loading vectors, surface pressure, permeable/impermeable surface data. The grid on the patch can either be structured (from CFD calculations) or unstructured.

Various patches are combined to form other complex objects which are again combined to form even more complex objects. For example a blade is composed of at least 3 patches. Different numbers of blades are used to create a rotor which in turn constitutes an aircraft.

Observers are the microphones or the listening positions over which the data calculated by the patches is post-processed. Wop-Wop provides the options of a single observer point or multiple observer grid points. These grid points are either rectangular or spherical in shape.

8.2. **Input Conditions**

All basic parameters such as rotor radius, velocity, advance ratio, blade geometry, and environmental constants are input. The blade is further meshed into a grid depending on the accuracy. Airloads and lift coefficients were calculated using Blade Element Theory or higher fidelity CFD analysis and applied to each surface or node of the grid generated. For the Peregrine, the airloads were calculated using a BET mentioned above.

Figure 7.5: Blade Deformation
Input files are used to define each object in the object hierarchy. Case files contain the path and filename of the particular case that is to be analyzed. The Namelist file contains the main definitions of the objects mentioned above that are to be used. Change of base files define the motion of various objects. Patch files are used to create surface patches.

For the Peregrine, simple patch files using an impermeable surface with aerodynamic forces as the loading vector is used for the blade. The elliptical planform of the blade is created using a geometry file created in Fortran 90. The blade is divided span-wise into 30 sections. At each section, the loading file inputs the lift and drag forces calculated earlier as periodic functions of the azimuth angle.

The observer grid is chosen to be a spherical grid so as to obtain noise levels that are required to calculate the Noise Index.

The initial conditions on the rotorcraft that were input included:

- Rotor Radius: 21 feet
- Tip Speed: 650 ft/sec at hover, 420 ft/sec at max dash speed
- Blade Type: Elastic
- No of Rotors: 2 (Co-axial)
- No of Blades: 4 per rotor
- Blade Elements: 30 per blade
- Blade Airfoils: VR7/VR8
- Blade Loading: Normal aerodynamic loads applied at the centroid of each element calculated earlier as functions of azimuth.

8.3. Processing

The Ffowcs Williams-Hawkings Equation is solved to find the acoustic pressure or the sound level at the observer grid points. This is done by using a time-domain integral formulation developed by Farassat using a Retarded Time Algorithm or the Source Time Algorithm. For the flexible blade case in the Peregrine, Farassat’s Formulation 1A using a Source Time Algorithm is implemented. In this algorithm, the source time or the time at which the noise is produced at the blades is used as a starting point and observer time is calculated from this. The main advantage of this algorithm is that it is computationally less intensive.

8.4. Post Processing:

An observer grid is generated in a sphere in front of the helicopter. The rotorcraft is assumed to be hovering/flying above 100 feet.

The Fast Fourier Transform is used to transform data from the time domain to the frequency domain. The FFT library that Wop-Wop uses is a package called Fast Fourier Transform in the West. This library is used to calculate the complex pressure at various frequency bins. The real and imaginary parts of the pressure are used to calculate the mean square pressure and the phase in a particular frequency bin.

8.5. Output files

The output may be in terms of the acoustic pressure or the sound level in decibels calculated at the previously defined observer grid points. These are Plot 3D structured binary files and are read using Fieldview or Tecplot.

8.6. Results

Figure 8.1 represents the total sound level of the helicopter for the hover case and forward flight case. The forward flight case shows low level of sound mainly due to the reduced tip speed. However, the sound levels are for an 8 bladed coplanar rotor.
The thickness noise, the loading noise and the total noise levels are calculated at the observer grid points. Figure 8.1 shows a variation of the total noise level at various tip speeds starting from 420 ft/sec to 650 ft/sec keeping the speed of the aircraft to be 250 knots.

![Figure 8.1: Total Noise Levels](image)

The noise levels are calculated at a point 500 ft away from the helicopter at a point 30 deg below the hub plane. We can see that as we increase the tip speed, the intensity of the overall noise increases. Thus, the reduction of the tip speed of the Peregrine from 650 ft/sec, which would have been in the case of the Super Lynx, to 420 ft/sec, albeit done mainly to reduce the high drag at the blades, also helps in reducing the noise produced by the rotorcraft.

Figure 8.2 shows the overall noise distribution in a sphere with a radius of 500 ft and 60 degree below tip path plane.

![Figure 8.2: Acoustic Profile at 250 knots, 500 ft AGL](image)
The noise level is loudest in the region 30 degree below tip path plane. In a conventional helicopter, there is a high noise region on the advancing side. As the Peregrine is a co-axial rotor, a more symmetrical distribution of the noise level is observed. Though loud enough to be categorized in a ‘noisy environment’ it still matches the target of being less than 85 decibels as required to calculate the noise index. Figure 8.3 illustrates the the overall noise level at hover about 60 ft. away. Again, the target noise level at this distance is nearly matched.

Figure 8.3: Overall Noise at a Hover, 60 feet away

9. **Ducted Pusher Propeller**

The focus on this concept, for the purpose of this design, stems from the basis that high speed rotorcraft flight is not possible with a current single rotor design with traditional tail rotor due to transonic tip speed, retreating blade stall, and reducing parasite drag. Historical and current attempts at high speed rotorcraft flight have successfully utilized pusher propeller configurations, although most vehicles remained experimental. In order to improve on the current field of pusher-type vehicles, this design will attempt to successfully integrate theories proposed by F.R. Goldschmied regarding static pressure thrust. This fuselage “self-propulsion” could contribute substantially to the AHS RFP requirement of a non-conventional drive system.

9.1. **Design Theory and Approach**

The fundamental claims and findings of Goldschmied, built on the earlier work of several others, are concisely summarized as follows. A detailed description of his form thrust theory and test data regarding wake regeneration can found in from the American Institute of Aeronautics and Astronautics meetings in 1986 and 1987.\(^{18,19}\)

- Possible reduction in propulsive power of up to 50%
- Experimental finding of propeller efficiency of 122% for advance ratios yielding zero net thrust (equilibrium flight of body).
Propulsive efficiency of 103% at equilibrium because the propeller-induced flow field over the body causes additional drag. Despite the "thrust-deduction," the power gain of the wake propeller was 41% over the free-stream propeller. The two major systems comprising the pusher propeller are the duct/housing and the propulsor.

In the 1980s, F.R. Goldschmied addressed the great potential of aerodynamic static-pressure thrust (also known as form thrust or negative pressure drag or negative form drag) for fuselage self-propulsion. Using theories proposed in the previous 50-150 years, he conducted a series of tests to determine the best configuration to reduce pressure drag in aerodynamic bodies. The test results yielded that an axis-symmetric body with a suction/pusher-ducted—enclosed-tail fan could achieve a reduction in propulsive power of up to 50%. Similar gains should be expected for rotorcraft tailored for high-speed flight.

In 1865, Froude originated the concept of body wake regeneration for propulsion based on the overall momentum balance of a moving vehicle. In steady motion through a single fluid, as with an aircraft, the established Rankin drag/thrust concept is misleading in its apparent simplicity and it invariably results in the adoption of reduced performance targets. For instance, when a fuselage is said to have a certain drag at a given speed, it is implied that the fuselage wake's momentum is condemned to useless dissipation, without possible recourse of any kind; it is implied that the drag can only be balanced by an equal propeller thrust, according to the Rankin concept. Still today, general aviation aircraft are viewed essentially as powered gliders, with the thruster (propeller or jet unit) installed in a manner not conducive to efficient wake regeneration. Even fuselage-mounted pusher propellers are too large and are not tailored to the specific wake.

A NASA wind tunnel test was performed on a 50-inch diameter axis-symmetric fuselage (essentially a full-scale fuselage for general aviation aircraft) with a wake-propeller. The wake-propeller had four blades with a 24" diameter (less than half body diameter) where the diameter was matched to the measured diameter of the wake without the propeller. The seemingly impossible result is that propeller efficiency is up to 122% for advance ratios yielding zero net thrust (equilibrium flight of body): this is conclusive proof that the wake-propeller does indeed operate within an area of lower velocities and that the efficiency does not have to decrease. However, the propulsive efficiency is down to a mere 103% at equilibrium because the propeller-induced flow field over the body causes additional drag, as compared to the bare body, or a so-called "thrust-deduction" of the propeller thrust. The same propeller, mounted in a conventional free-stream installation, achieved 73% propeller and propulsive efficiencies. Thus despite the "thrust-deduction," the power gain of the wake propeller was an impressive 41% over the free-stream propeller. This illustrates the power that can be extracted from the fuselage wake's kinetic energy and which is going to waste today in the general aviation aircraft. It can be added that the "thrust-deduction" is not a necessary evil: it can be eliminated by designing the body shape through a complex iterative process that includes the propeller's effect on the body pressure distribution and boundary-layer development. The final body shape is such that maximum pressure recovery is achieved on the fuselage's aftbody with the propeller, while the aftbody flow would be fully separated without the propeller's effect. This maximum pressure recovery is assumed to be that given by the Goldschmied turbulent separation criterion.

In conclusion, since the fuselage represents at least 50% of the total aircraft drag, fuselage self-propulsion by static-pressure thrust should yield at least 25% total power reduction. A preliminary design study by F. R. Goldschmied for 200 MPH cruise speed indicates a potential 60% power saving for 2-seat aircraft and 40% power saving for 4-seat aircraft, as compared to specific current designs. A similar study was presented by the same author for general aviation aircraft based on the McLemore body/wake-propeller configuration. Pressure thrust is the most efficient form of fuselage self-propulsion.

9.2. Duct Design

In order to integrate the Goldschmied static pressure thrust concept, the pusher propeller must be mounted closely to the aftbody of the fuselage to properly ingest the fuselage wake. As a primary safety concern, due to the pusher prop’s closeness to the fuselage and subsequent proximity to flight crew and ground personnel, it is critical that the propeller be enclosed in a shroud. Additionally, by ducting the propulsor, a significant increase in thrust and efficiency can be obtained, particularly at static, or low forward flight velocities (sub-transonic). Previous experimental results indicate that the addition of the duct results in a 30% increase in thrust and a 25% increase in power. This confirms theoretical momentum theory analysis typically applied to Fenestron tail rotors for helicopters. Ducted propellers also have advantages related to community noise levels and enhanced public acceptance.
9.3. **Duct Sizing**

To prevent an increase in the overall flat plate drag of the vehicle, the duct’s outer diameter was constrained to fit within the existing fuselage profile. This limitation affected the selection of the propulsor configuration and operating speed in order to produce the maximum thrust, and is also supported by Goldschmied’s observation that most pusher propellers are oversized, preventing effective wake regeneration. The minimum inner diameter, occurring at the blade axes, was established at 1.22 m (4.0 ft) in order to allow for structural rigidity for housing both the propulsor and horizontal and vertical stabilizers. This allows a 1 cm blade tip clearance, which is 0.82% of the duct’s inner diameter. The design length is 1.7 m (5.58 ft) to allow proper clearance between the counter-rotating propfans and the stabilizers to minimize interference. The forward portion of the duct is expanded as it overlaps the fuselage in order to maintain a fixed effective inlet area of 1.12 m², preventing an increase in inflow airflow velocity. This forward placement of the diffuser inlet along the fuselage should prevent boundary layer separation along the aft fuselage, generating static pressure thrust and allowing this otherwise turbulent wake to be ingested by the pusher prop. Additionally, the nominal thickness of the shroud is 2.4 cm and the max diameter (outer) is 1.65 m with a radius of 0.824 m.

![Figure 9.1: Ducted Pusher Propeller](image)

9.4. **Diffuser**

The aft portion of the duct is designed with a 10 degree half-angle to serve as a diffuser to prevent wake contraction while maintaining the desired length and proportion of the duct. This results in a wake contraction parameter of 1.27, which should allow for reduced wake velocity enabling a 35% decrease in power required to an unducted system.

9.5. **Stabilizers**

In order to increase the moment arm of the vertical and horizontal stabilizers while preventing additional empennage weight and drag, the stabilizers are positioned aft of the propfans, mounted internal to the duct. This also reduces the download penalty associated with horizontal control surfaces acted upon by rotor downwash at a hover. Since the airflow influencing these control surfaces is first acted upon by the propfan, the required surface area for equivalent forces is able to be reduced. The vertical stabilizer, with a span of 1.24 m (4.1 ft) and an area of 0.27 m² (2.86 ft²), uses a symmetric NACA64A010 airfoil with zero degree offset. It is divided into two fixed sections by a V-shaped notch to allow movement of the horizontal stabilator. The traditional offset design of a vertical fin to offload the main rotor at forward airspeeds is not required because of the vehicle’s coaxial rotor. An anticipated requirement for directional control supplementation during the initial design phase resulted in a design which would allow the vertical fin to be adjustable up to +/- 90 degrees left or right for thrust vectoring. This requirement has since been deemed unnecessary, essentially making the vertical fin a stator vane.

The horizontal stabilator utilizes an inverted asymmetric NACA 2410 airfoil to help overcome an observed pitch-down moment at higher forward airspeed. With a span of 1.24 m (4.1 ft), an area of 0.32 m² (3.44 ft²), and an aspect ratio of 4.5, it should provide up to 1,390 lbf of downward thrust at 250 knots. For control of this pitch trim supplementation at higher speed, the horizontal stabilizer is designed for an adjustable angle of attack of +/- 20 degrees pitch (evolving it from a stabilizer into a stabilator). The pitch angle is set by an electrically controlled screw-actuator mounted internal to the propfan hub assembly and rear spinner.

After Flightlab analysis and simulation, it was determined that the existing internal horizontal stabilator was not enough to overcome a large pitch-down moment of the fuselage generated at high speed. An additional fixed horizontal stabilizer was designed to be added external to the duct to produce the additional 2,250 lbf of downthrust required. The new external horizontal stabilizer is a fixed (immovable) inverted NACA 2410 with an offset of -4 degrees. It is a total of 2.04 m² (22 ft²), with the "wing" split on each side of the duct. If this tapered planform...
"wing" was modeled singly (without being divided because of the duct) it would have an aspect ratio of 4.55, with a span of 3.05 m (10 ft) and a taper ratio of 1/3. This equates to a root chord of 1.0 m (3.3 ft) and a tip chord of .34 m (1.1 ft). This should produce up to 2,298 lbf of down-thrust at 250 knots, acting 4.3 m (14 ft) behind the CG.

9.6. Propulsor Design

A relative comparison of propeller configurations was performed using safety, vibration, responsiveness, direct operating cost, weight, price, thrust, and noise as grading criteria. This comparison supported the use of a ducted, constant-speed, counter-rotating, composite propeller (or propfan). The pusher-prop will be powered by the two main turbines to maximize the use of the primary engine power that will be available as the main rotor is slowed in forward flight and no longer required to produce forward thrust. The transition will occur as the aircraft accelerates above its maximum endurance airspeed, where the power required for the slowed main rotor intersects the power required for the high-speed rotor. Utilizing the main powerplants will prevent the additional weight and cost of adding a separate power source which would be underutilized at low airspeeds.

A variable-pitch, constant-speed propeller best suits the need for rapid thrust response, as well as to provide maximum propeller efficiency throughout a highly variable flight profile. Counter-rotating propfans improve efficiency by removing the swirl created by single-rotation propfans, and also provide increased power over a single-rotating propfan of the same tip diameter. Additionally, a counter-rotating prop will prevent a roll-wise torque from being placed on the fuselage due to the unique placement of the pusher-prop. It will be noted here that the effects of rotorwash on the pusher prop performance is expected to be negligible during its primary use in high speed forward flight.

9.7. Design Attributes

Basic design parameters were established early from simple analysis of the expected operating envelope. Since the diameter of the propfan is constrained to 1.2m (3.94ft) by the duct, a maximum operating speed of 2,500rpm was determined for a maximum effective tip speed (accounting for induced flow at 250 knots) of 0.612 Mach. This is below appreciable compressibility effects and should minimize both generated noise and the need for highly swept blades. Experimental data on contra-rotating propfans using the SR-3 thru SR-7 family of airfoils was examined, however, the high inner solidity (greater than 1) was undesirable. Additionally, since these airfoils were designed for transonic/supersonic operation, the added design complexity and resulting manufacturing cost were further support to find an alternative design. Anticipating the possible use of the propulsor at low airspeeds for required directional control supplementation, airspeeds from 25 knots to 250 knots were included in the design. The resulting Prop Advance Ratio (J) ranges from 0.257 to 2.57. Examination of previously successful propeller airfoil data resulted in the selection of a NACA 4415 which demonstrates a relatively shallow decline of the Lift Coefficient as the blade enters stall.

PHYSICAL ATTRIBUTES:
- Two 10-bladed counter-rotating fans
- Fan Blade Airfoil: NACA 4415
- Variable Pitch from -10° to +55°
- Operating speed: 2,500 rpm
- Fan Diameter: 1.2m = 3' 11.25”
- Hub Diameter: 0.24m = 9.45”
- Max chord (at tip): 0.17m = 6.69”
- Min chord (at root): 0.065m = 2.56”
- Twist: linear -35° from root to tip
- Leading-edge sweep angle: +3.13°
- Solidity: 0.519
- Inter-prop spacing: 0.13m = 5.12”

Figure 9.2: Propfan Attributes
With these parameters established, a MATLAB program was created primarily using Blade Element Theory, modeling a single disk, to calculate the Thrust produced and Power required based on user input physical and aerodynamic characteristics. Lift, Drag, Thrust, Torque, and Power required were calculated at each blade element at radial stations (r) from the hub to the tip, using a table lookup for Lift and Drag Coefficients (C_L, C_D) at the calculated effective angle of attack for each blade element. A blade element schematic is presented in Figure 9.3, and key equations are represented in Figure 9.4.

Figure 9.3: Blade Element

Optimization was performed primarily for the intended cruise airspeed of 215 knots, while considering performance at the maximum dash speed of 250 knots. Numerous optimization cycles were performed to define the blade number, hub size, maximum chord, chord distribution, twist, and blade pitch. Additionally, the earlier selection of the NACA4415 was confirmed, showing a 10-20% increase in thrust per horsepower over a baseline Clark-Y airfoil. Final results of the geometric optimization are summarized in Figure 9.2

The chord distribution was selected from a final optimization with maximum Lift Coefficient versus Power Coefficient as the primary factor. This optimization sought to minimize the root chord and maximize the tip chord. The root of the blade was constrained at 6.5cm (2.6in) to ensure there would be no inter-blade contact for zero pitch and negative pitch operations, and to ensure the blades could be reasonably produced with the required structural properties. The maximum chord was maintained at the tip since tip losses will be minimized in the ducted design. Calculated efficiency of a single prop with Integral Momentum Theory and Actuator Disk Theory yielded 73-89% at higher airspeed in the range of expected angle of attack (AoA). Performance data was then calculated from 25 to 250 knots and -10 to +20 degree effective AoA at the blade tip. Mechanical pitch control, from -10 to 55 degrees, will be by means of an electrically-actuated servo operating a pitch-change control shaft routed internal to the main propfan driveshaft. Both the fore and aft propeller will maintain equal pitch (zero degree offset). All internal components to the hub affecting propeller drive and pitch are of conventional design. The Propfan Axis of rotation is centered laterally and acts through the vehicle’s Center of Gravity.
9.8. Performance

Since the MATLAB program used to optimize the propulsor was for a single disk, overall performance of the ducted two disk system was estimated based on literature of experimental data. Factors considered when estimating overall efficiency adjustments were tip losses, swirl, interference effects, installment, thrust contribution by the duct itself, effect of the duct diffuser, and the Goldschmied static pressure thrust effect. As a note, the Goldschmied effect was minimally applied here since it is also applied as a factor to the current estimation of effective flat plate area of the fuselage. Due the small blade tip clearance with the shroud, tip losses were considered negligible. With the inter-prop spacing equal to 80% of the blade chord, swirl was assumed to be negligible and the change in inflow momentum to the aft disk was also negated. Since input power to the propfan is a limiting constraint, all performance gain estimations were applied only to the resulting thrust. Therefore, power requirements for the dual-prop system were simply calculated as twice the single prop results. Thrust for the dual-prop system was taken to be 1.8 times the single prop performance. This estimation yields maximum Thrust to Power Ratios presented in Table 9-1 which is slightly optimistic but in general agreement with comparable data.

<table>
<thead>
<tr>
<th>Lift</th>
<th>$L = \frac{1}{2} \rho \cdot V e^2 \cdot C_L \cdot A$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Drag</td>
<td>$D = \frac{1}{2} \rho \cdot V e^2 \cdot C_D \cdot A$</td>
</tr>
<tr>
<td>Thrust</td>
<td>$T = L \cdot \cos \phi - D \cdot \sin \phi$</td>
</tr>
<tr>
<td>Torque</td>
<td>$F_Q = r \cdot (L \cdot \sin \phi + D \cdot \cos \phi)$</td>
</tr>
<tr>
<td>Power</td>
<td>$P = \omega \cdot F_Q = 2\pi \cdot n \cdot F_Q$</td>
</tr>
<tr>
<td>Advance Ratio</td>
<td>$J = \frac{v}{n \cdot D}$</td>
</tr>
</tbody>
</table>

Where:
- $v$ velocity m/s
- $D$ diameter m
- $n$ rotations per second 1/s
- $\rho$ density of air kg/m³
- $P$ power W
- $T$ thrust N

Table 9-1: Propfan Maximum Thrust/Horsepower

<table>
<thead>
<tr>
<th>Airspeed</th>
<th>Advance Ratio</th>
<th>Thrust/Power (maximum)</th>
<th>Tip AoA (effective)</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>1.03</td>
<td>2.86</td>
<td>1° to 4°</td>
</tr>
<tr>
<td>150</td>
<td>1.54</td>
<td>1.92</td>
<td>3° to 4°</td>
</tr>
<tr>
<td>200</td>
<td>2.06</td>
<td>1.44</td>
<td>1° to 8°</td>
</tr>
<tr>
<td>250</td>
<td>2.57</td>
<td>1.15</td>
<td>0° to 11°</td>
</tr>
</tbody>
</table>

The propfan’s operational schedule has four major characteristics, and is depicted graphically in Figure 9.5.

0-50 knots: unpowered, blades feathered

50-100 knots: 2,500 rpm, minimum power, zero thrust (prevent drag effects)

100 knots: 2,500 rpm, pitch as required to provide total forward thrust component.

90 knots (and decelerating): return to zero thrust as main rotor speed increases.
The maximum design power load will occur at 200 knots, 44 degrees mechanical pitch (at tip), resulting in an effective AoA of 11 degrees, requiring 2,504 hp to produce 3,587 lbf of thrust. This point is highlighted in Figure 9.6 by a red circle. Stall effects become visible at AoAs higher than 11-12 degrees. For structural analysis, a worst-case load was used which is not expected to be seen during operation since it would require too much engine power. At 250 knots and 51 degree mechanical pitch (at the blade tip), 4,027 lbf of thrust would be produced if 3,520 hp were delivered to the pusher prop, resulting in a disk loading of 330 lbf/sqf. Since the propfan is constant-speed, specific performance throughout the flight profile is highly varied, as a function of the angle of attack input to the prop blades.

**10. Engine Performance Requirements**

Preliminary design studies and sizing initially indicated the power requirements to obtain a 250kt capability combined with a range 20% greater than the Super lynx 300 at standard day conditions (sea level, 59 deg F):

- Total Maximum Continuous Horsepower (MCP): 2552 shp
- Total Intermediate Rated Horsepower (IRP): 3100 shp

**10.1. Approach**
First, a number of engines in the required power range were selected. The engines were selected such that the total number of engines required to power the vehicle would number two or three in order to allow for redundancy but not add to excessive weight, fuel consumption or overall volume. Next, factors deemed important in the ultimate selection of a single engine were chosen. After careful consideration the factors chosen were: Weight, Power, SFC, Maintainability, Reliability and Safety. The reason these factors were chosen were as follows: the weight, power and SFC directly impact the performance of the vehicle and as such the requirement is that the weight of the engine be as low as possible while its outputs the maximum possible power simultaneously with a specific fuel consumption as low as possible.

Maintainability was considered as an important factor since the time between overhauls (TBO)’s of the different engines differed vastly. While most American and Canadian made engines have a TBO of between 2500 and 3000 hrs the older French made engines have TBO’s of 3500 hrs. However the newer French engines have no specific TBO but an hourly replacements cycle requiring no TBO i.e. parts are replaced after a few number of hours periodically e.g. the oil filters are replaced every 80 hrs on the RTM 322 and similarly for the other parts as well. The underlying point is that while such an engine is more reliable due to the fact that parts may not be old and worn out and hence have a lower chance of failure, it does make the maintenance a hassle since rather than every 3000 hours or so the vehicle requires periodic maintenance every few hundred hours.

All of these factors were then ranked against each other to determine the weights for each. The rankings and the final weights are given in Table 10-1:

<table>
<thead>
<tr>
<th>Criterion</th>
<th>Maintainability</th>
<th>Reliability</th>
<th>Safety</th>
<th>Weight</th>
<th>Power</th>
<th>sfc</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maintainability</td>
<td>0.2</td>
<td>0.111111111</td>
<td>0.2</td>
<td>0.25</td>
<td>0.125</td>
<td></td>
</tr>
<tr>
<td>Reliability</td>
<td>5</td>
<td>0.5</td>
<td>1</td>
<td>3</td>
<td>1</td>
<td></td>
</tr>
<tr>
<td>Safety</td>
<td>9</td>
<td>2</td>
<td>9</td>
<td>2</td>
<td>9</td>
<td></td>
</tr>
<tr>
<td>Weight</td>
<td>5</td>
<td>1</td>
<td>0.111111111</td>
<td>4</td>
<td>0.5</td>
<td></td>
</tr>
<tr>
<td>Power</td>
<td>4</td>
<td>0.333333333</td>
<td>0.5</td>
<td>0.25</td>
<td>0.5</td>
<td></td>
</tr>
<tr>
<td>Sfc</td>
<td>8</td>
<td>1</td>
<td>0.111111111</td>
<td>2</td>
<td>2</td>
<td></td>
</tr>
</tbody>
</table>

The preferences were scaled according to this ranking system:

- **Verbal Judgment of Preference**       **Numerical Rating**
  - Extremely Preferred             9
  - Very strong to extremely        8
  - Very strongly preferred        7
  - Strongly to very strongly       6
  - Strongly preferred              5
  - Moderately to strongly          4
  - Moderately preferred            3
  - Equally to moderately           2
  - Equally preferred               1

The preferences were scaled according to this ranking system:

- **Verbal Judgment of Preference**       **Numerical Rating**
  - Extremely Preferred             9
  - Very strong to extremely        8
  - Very strongly preferred        7
  - Strongly to very strongly       6
  - Strongly preferred              5
  - Moderately to strongly          4
  - Moderately preferred            3
  - Equally to moderately           2
  - Equally preferred               1

![Figure 10.1: Weights of Tradeoff Criteria](image)

It may be observed that safety is the criterion that is the most important while maintainability is the least.
10.2. Choice of Engines

After considering various engines including:

1. Pratt & Whitney Canada PT6T-6 Twin Pac
2. Rolls Royce Turbomeca RTM322
3. Turbomeca Makila 2A
4. General Electric CT7-6A
5. Turbomeca Makila 1A1
6. MTU Turbomeca Rolls Royce MTR 390-2C
7. Turbomeca Ardiden 1A
8. LHTEC CTS800-5

The engines were then also rated with a subjective rank value for their maintainability, reliability and safety. Engines that had the longest TBO received the best rating for maintainability while and engine that required frequent maintenance ranked low. Reliability was based on the parts count of the engines; a larger parts count would mean lesser reliability since there are more components that can fail. As for safety a larger number of engines would mean greater safety. The ratings for each engine based on these attributes are given in Table 10-2:

<table>
<thead>
<tr>
<th>Engine</th>
<th>Weight</th>
<th>Power</th>
<th>sfc IRP</th>
<th>sfc MCP</th>
<th>Maintainability</th>
<th>Reliability</th>
<th>Safety</th>
</tr>
</thead>
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<tr>
<td>PT6T-6</td>
<td>1320</td>
<td>1970</td>
<td>1745</td>
<td>0.591</td>
<td>3</td>
<td>1</td>
<td>9</td>
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<tr>
<td>RTM322</td>
<td>1104</td>
<td>2101</td>
<td>1842</td>
<td>0.454</td>
<td>1</td>
<td>9</td>
<td>3</td>
</tr>
<tr>
<td>Makila 2A</td>
<td>1036</td>
<td>2101</td>
<td>1870</td>
<td>0.469</td>
<td>9</td>
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<td>3</td>
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<tr>
<td>CT7-6A</td>
<td>986</td>
<td>2000</td>
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<td>0.454</td>
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<td>3</td>
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</tr>
<tr>
<td>Makila 1A1</td>
<td>1036</td>
<td>1820</td>
<td>1589</td>
<td>0.548</td>
<td>9</td>
<td>3</td>
<td>3</td>
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<tr>
<td>MTR 390-2C</td>
<td>1119</td>
<td>1303</td>
<td>1187</td>
<td>0.454</td>
<td>9</td>
<td>3</td>
<td>3</td>
</tr>
<tr>
<td>Ardiden 1A</td>
<td>1701</td>
<td>1430</td>
<td>1220</td>
<td>0.46</td>
<td>3</td>
<td>3</td>
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<tr>
<td>CTS800-5</td>
<td>1020</td>
<td>1550</td>
<td>1226</td>
<td>0.49</td>
<td>2</td>
<td>3</td>
<td>9</td>
</tr>
</tbody>
</table>

Using the values in Table 10-2 a TOPSIS was executed to make the best choice for the engines. The final ranking for the engines is shown in Figure 10.2.

![Figure 10.2: TOPSIS Scores of Engines](image-url)
The MTR390-2C, the CTS800-5, and the Ardiden 1A are close in terms of scores. However the LHTC CTS800-5 is deemed to be the best primarily because it has a lower SFC and a lower weight than the other two. The CTS800-5 is also a newer engine which entered service only in 2006 hence making it more advanced than the other two.

The LHTEC CTS 800-5\textsuperscript{53} was chosen primarily for three main reasons:

i. Commonality with the current Super Lynx
ii. Reduced sfc of 0.49 compared to most engines with sfc’s around 0.52
iii. Appropriate Power rating

A total of two CTS 800 engines will be used to power the peregrine. Two were chosen in order to obtain enough power to meet certain requirements and also for reasons of safety. Some of the performance characteristics of the LHTEC CTS 800-5 are shown in Table 10-3 and Figure 10.4.

<table>
<thead>
<tr>
<th>Performance</th>
<th>Minimum thermodynamic shaft horse power</th>
<th>Sfc lb/shp-hr (max)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Contingency</td>
<td>1721</td>
<td>0.469</td>
</tr>
<tr>
<td>Maximum</td>
<td>1681</td>
<td>0.470</td>
</tr>
<tr>
<td>Intermediate</td>
<td>1550</td>
<td>0.473</td>
</tr>
<tr>
<td>Max continuous</td>
<td>1276</td>
<td>0.490</td>
</tr>
<tr>
<td>4000 feet, 95°F, static Contingency</td>
<td>1269</td>
<td>0.488</td>
</tr>
<tr>
<td>Maximum</td>
<td>1232</td>
<td>0.491</td>
</tr>
<tr>
<td>Intermediate</td>
<td>1110</td>
<td>0.502</td>
</tr>
<tr>
<td>Max continuous</td>
<td>905</td>
<td>0.523</td>
</tr>
</tbody>
</table>

10.3. All Engines On (AEO) Performance

The power required curves were obtained from the GTPDP sizing. On these the power available curves were overlaid to perform the performance analysis.

From Figure 10.4 it can be observed that the Maximum Continuous Power (MCP) available on a standard day is 2934 shp and on a hot day at 5000 ft it is 1600 shp. The Intermediate Rated Power (IRP) available on a standard day
is 3958 shp and on a hot day at 5000 ft it is 2600 shp. Lastly, the minimum power required occurs along the power curve where the tip speed is 420 ft/s.

Regarding excess power, it can be observed that on a standard day around 450 shp of excess power is available for a vertical climb at 0 knots forward speed. Also, on a hot day around 1000 shp of excess power is available for a vertical climb at 0 knots forward speed. Finally, maximum excess power to the order of around 2250 shp is available at a forward speed of 105 knots on a standard day and around 1700 shp at 105 knots on a hot day. This massive amount of excess power allows for climb rates of up to 3100 ft/min.

Figure 10.5 it can be observed that the maximum (dash) forward speed at service ceiling on a hot day is 215 knots. Additionally, from the intersection of the power required and power available curves it may be seen that the dash speed at SLS 249.8 knots. Finally, the best velocity for endurance is 105 knots while best cruise speed at service ceiling is 120 knots. It may be observed that with all engines operative there is enough available power for hover at both sea level standard and at 5000ft on a 95 degree day.

10.4. One Engine Inoperative (OEI) Performance

With One Engine Inoperative (OEI) it can be observed that with OEI the helicopter loses hover capability except when the engine is running at emergency power or IRP at sea level. In flight, climb performance is restricted to less than 300 ft/min at 100 knots. Meanwhile, maximum forward speed is restricted to 190 knots on a standard day at SLS and 160 knots on a hot day at 5000ft.
10.5. Climb Performance and Service Ceiling

The rate of climb was calculated using the excess power available. The rate of climb represented here is purely the vertical climb at a forward speed of 0 knots. At sea level at best climb speed of 100 knots the maximum climb rate was calculated to be 5683ft/min. However with a zero forward speed and AEO the maximum climb rate on a standard day at SLS was found as 3139ft/min. On a hot day the maximum rate of climb was found to reduce drastically to under 500ft/min.

The excess power also allows for an absolute ceiling of above 18000 ft and a service ceiling of 17500ft. However on a hot day (ISA +20C) the absolute ceiling is reduced to 4100ft and the service ceiling to 3200ft. 3000ft was chosen as a reasonable service ceiling since this altitude is comfortable for passengers with respect to pressure and temperatures in case of an unpressurized cabin.

10.6. Range Performance

At maximum gross weight the range was found to be 420 Nm while cruising at 120 knots on a standard day. This range however reduces to 395.5 Nm on a hot day.
11. Full Authority Digital Engine Control (FADEC)

A Full Authority Digital Engine Control system provides engine control and governs engine operations. The FADEC can be seamlessly integrated into the fly-by-wire flight control system on the Peregrine. The main purpose of the FADEC is to reduce pilot workload. In a helicopter this places more onus on the FADEC since more functions need to be performed with respect to a FADEC on a fixed wing aircraft. Normally a FADEC would be used to perform primary functions on any engine, such as:

- Power up Function Check
- Auto start and Auto relight
- Automatic fuel flow
- Over speed System protection
- Surge/Blowout protection
- Fault Monitoring

However a rotorcraft FADEC also needs to accomplish two important features:

- Torque spike elimination
- Power dip elimination

Both of these are a phenomenon related to collective control. Once the collective is increased, the rotor experiences increased drag and hence requires increased torque and power until it can reach its steady state. A rotorcraft FADEC should have an ‘anticipatory system’ which based on collective inputs can either speed up or slow down the engine to match the torque and power required by the rotors. Similarly when the collective is lowered the rotor drag reduces and momentarily the power and torque on the rotor are more than required. Again it is up to the FADEC to use collective inputs and the ‘anticipatory system’ to make sure that the torque spike does not occur and that the engine is slowed down to match the torque required by the rotor.

11.1. Functional Development

In order to develop a suitable FADEC architecture, first the functions of the FADEC were decomposed\(^{36}\) to gain a better understanding of the functions to be performed.

![Figure 11.1: Functional Development Level 1 and 2](image-url)
Once the FADEC’s functions were broken down it could be seen that both torque and rpm of the engine can be governed by fuel flow. The rate of fuel flow itself is governed by parameters such as: turbine temperature, combustor temperature, speed of gas generator governor, speed of compressor and the pilot power lever angle (PLA). Using this information the FADEC architecture in Figure 11.3 was developed.  

Figure 11.2: Functional Development Level 1 and 3

Figure 11.3: FADEC Architecture

From the control architecture, the component schematic in Figure 11.4 was developed:
The ECU takes in flight conditions and sensor inputs from the engine and uses them to maintain the desired power output as required by the rotors. In order to do this effectively the ‘anticipatory system’ is used. The collective inputs are used by the anticipatory system to anticipate any changes in power requirement. This system then commands the fuel metering. However the over speed and surge protection system can over ride the anticipatory system in case a surge or over speed is imminent. Both the over speed and surge protection systems can do so by monitoring the gas generator turbine speed and the gas generator exit temperature.

Features of the FADEC

- Power up Function Check
  - The FADEC performs voltage and current draw check on all solenoids and compares them to pre-established limits.
- Auto start and Auto relight
  - Flameout is relayed to other engine/FCS by ARINC link. This effectively overrides any protection on the operative engine so as to allow time for the inoperative engine to restart. Relight procedure same as start up procedure. However on failure to relight at up to 40% of max governor speed the engine is deemed in operative.
- Over speed System protection
  - Allows over speeding of the engine by up to 12% compared to normal operation- Required for OEI conditions.
  - Monitors turbine speed to reduce fuel flow when turbine speed exceeds 15% of speed with respect to MCP.
  - The rate of increase in NG is also limited in order to avoid over speed by over fueling.
- Surge/Blowout protection
  - Monitors combustor governor speed and combustor temperature to match to predetermined tables.
  - In case the values are lower fuel flow is increased.
  - Else flow is decreased to avoid surge and then increased slowly according to predetermined rate.

Figure 11.4: Concept for FADEC Schematic Diagram
12. Structural Analysis

The Peregrine airframe is designed to reduce the weight and lifecycle cost of the vehicle compared to the baseline helicopter. Thus, the design objective for the derivative version of the current Super Lynx 300 structure is to reduce the part count, and to save weight so as to get more freedom concerning the rotor drive system design. Besides, this corresponds to the policy of AgustaWestland which launched a re-airframing program to support its customers and to extend the longevity of their Lynx helicopters by twenty years according to the firm’s officials.

In addition to that, the Finmeccanica’s company started the production of 70 Future Lynx helicopters which are to enter frontline service with the British Army in 2014. The Future Lynx airframe design is based on a wide use of composite materials, an extensive use of monolithic machine components, and three-dimensional digital modeling.

The starting point for a trade-off study was defined according to technical papers about the EH101 Merlin airframe and its material breakdown, as well as the Westland Advanced Technology Fuselage (ATF) research program.

12.1. Trade Study

A trade study was conducted with three candidates to determine the best configuration: a conventional light alloy structure, a hybrid metal and composite airframe, and a fully composite fuselage design. This study addresses the airframe structural layout in order to locate the bulkheads and the ribs as efficiently as possible along the fuselage and to minimize the weight while providing both primary and secondary load paths so as to ensure the crashworthiness and the resistance to fatigue of the rotorcraft. Additionally, several configurations were considered for the skin, subfloor, and energy absorbers.

12.1.1. Bulkheads and Ribs

The center section fuselage consists of three main bulkheads located respectively just behind the cockpit, at the main rotor axis, and at the back of the cabin. There are fore and aft beams to support the engine deck and distribute the loads from the rotor drive train system. This structure constitutes the primary load path.

In addition to that, ribs linking the bulkheads will provide more stiffness, more survivability through geodetic design, and hard points to fix the skin panels. Smaller bulkheads will be added so as to offer secondary load paths, mainly around the cockpit and under the engines.

Based on the Advanced Technology Fuselage research program, three materials have been considered for the frames: a conventional Lithium-Aluminum light alloy (AA 8090), a Titanium-Aluminum alloy (Ti-6Al-4V), and carbon. The titanium-framed structure employs I-section electron-beam welded frames composed of five parts each.

The carbon-fiber framed structure has a similar configuration to that just described. Side members are of top-hat section while roof members consist of back-to-back channels separated by a foam spacer. These components are bonded together using a 120°C curing epoxy adhesive.
The Titanium alloy offers many advantages over the conventional aluminum alloy in terms of tensile strength, fatigue resistance and corrosion. Nevertheless, Ti-6Al-4V is more expensive and difficult to shape because of its high proof/ultimate ratio making cold forming difficult. Cost and weight savings are estimated respectively at 5 per cent and 10 percent for the Titanium-framed fuselage. The relatively low cost saving with the Titanium alloy can actually be attributed to the metal’s high cost. On the other hand, carbon has high specific properties and would allow cutting costs in the manufacturing process by automating production methods for component manufacture, assembly and inspection. Cost and weight savings with carbon-fiber can be expected to be respectively 18 per cent and 16 per cent.

The cost and weight studies completed to date in the industry, and especially at Westland, indicate that significant savings can be obtained from a carbon-framed structure. However, a quick weight analysis of the CAD model will provide additional information to decide which option best suits the design concept. In addition to that, a static load analysis will allow us optimizing the structural layout and particularly the location of the bulkheads. This would ideally include two cases: flight weight freight condition (gearbox and rotor head vertical loads with a distributed floor loading) and crash loading (gearbox and rotor head vertical loads in a 15g crash with a distributed ditching load applied to the underside of the structure). Further comparisons using DELMIA might help assessing the manufacturing cost reductions that can be expected according to the selected material and airframe.

The first iteration of the analysis using AFC showed that the structure would withstand static loads from the rotor hub, engine, transmission, and Pusher propeller. Further dynamic analysis and optimization of the structure is required as the design progress. Structural optimization is going to guarantee we satisfy the weight and strength requirements. Therefore, another software is included to corroborate any result from AFC, RdM Le Mans, which performs static analysis in 2-D for simplicity, including beam profiles, dimensions and materials selection.

The test of the stick model showed that with loads applied at similar locations as the 3D model in AFC, similar results were obtained that correlated to other results and that confidence allowed for continued frequency analysis.

In addition, a static load analysis will allow us optimizing the structural layout and particularly the location of the bulkheads. This would ideally include two cases: flight weight freight condition (gearbox and rotor head vertical loads with a distributed floor loading) and crash loading (gearbox and rotor head vertical loads in a 15g crash with a distributed ditching load applied to the underside of the structure).

12.1.2. Fuselage Skin

Two materials are contending in the skin panel’s application: Lithium-Aluminum alloy and four-ply carbon-fiber skinned Nomex honeycomb sandwich. The composite panels were selected for their weight advantages, as well as for their resistance to corrosion, and the ease of manufacture. A curing adhesive allows bonding the panels to the frames.

12.1.3. Subfloor and Energy Absorbers

Weight reduction is a critical part of the design since the Peregrine has to carry greater loads. Therefore, the structure of the aircraft had to be as light as possible. However, there is a important aspect that are as important as weight saving which are the impact forces transmitted to the occupants and to maintain the structural integrity of the fuselage to ensure a minimum safe occupant volume. However, this new concept is going to be analyzed using finite element analysis to determine if the design would satisfy the requirements. The energy absorbing subfloor will be designed to dissipate kinetic energy through stable crushing, the rigid structural floor will be designed to react the loads generated by crushing of the subfloor and to provide stable platform for seat and restrain attachment.

12.1.4. Trade Study Conclusions

More data was required, especially from the CAD model, to complete the airframe trade study concerning the concept, but the trends seemed to lead to a full composite fuselage, including structure, skin, and subfloor, while some minor components may remain metallic. In addition to the cost and weight savings, composite materials have slow crack propagation rates and fail-safe characteristics. Composites also show advantages in fatigue and their generally anisotropic nature allows the designers to tailor stiffness and weight distributions independently from each
other to the specific application. With this composite fuselage finally selected, its architecture was closely derived from the EH101 Merlin shown in Figure 12.2.

Improved ‘damage tolerance’ configurations might be considered since carbon-fiber reinforced plastic (CFRP) is vulnerable to impacts that cause undetected internal damage and subsequent delamination and failure. Thus, thermoplastic composites such as the ICI’s APC2 can be employed. In this material, Hercules AS4 carbon fibers are embedded in a matrix of Polyether Ether Ketone (Peek) instead of the conventional epoxy resin.

In this case, curing is not required; the material is simply heated in an oven to around 380°C then quickly transferred to a hydraulic press, where the component is formed under pressure. Unlike autoclave curing, this is a reversible process. The same strength properties can be expected but when damage does occur, it is at the surface, visible, and it does not proliferate.

**Figure 12.2: EH101 Merlin Material Breakdown**

12.1.5. **Strength Requirements**

As mentioned earlier and required by the RFP, the structure of the design aircraft has to withstand certain load limits and ultimate loads which are just the limit loads multiplied by a prescribed safety factor. As require by 14 CFR PART 29, the factor of safety is 1.5 times the load limit.

12.2. **Peregrine Structural Design**

The Peregrine showcases a full composite crashworthy airframe designed to accommodate the non-conventional drive train of the rotorcraft as well as to save weight since new features were added to the baseline AW159 helicopter.

Thus the frames were made of carbon fiber and the fuselage skin of composite panels which offers a quasi immunity to corrosion and is therefore highly desirable for a naval helicopter. Disposition of the crew and equipment inherits directly from the baseline vehicle while the fuselage cross-sectional shape was updated and optimized so as to take the benefit from the static differential pressure reduction provided by the pusher prop. The fuel tank location was chosen in order to adjust the CG position.

12.3. **Design Evolution**

A Lithium-Aluminum structure based on the AW159 airframe layout and on current technologies applied to the Anglo-Italian EH101 Merlin, such as the Health and Usage Monitoring System (HUMS)\textsuperscript{38}, served as starting point. Next, so as to provide the Peregrine with an advantage in terms of lifecycle cost and longevity\textsuperscript{39}, the composite solution was retained concerning both the fuselage skin and the airframe. For the future, the fundamental trend developing in the civil and military rotorcraft markets is indeed the longevity of helicopter lives with new designs being in service for up to 40 or perhaps even 50 years.\textsuperscript{40} Moreover, the weight savings that could be expected allowed for more freedom in the drive-train design. Finally, all structural properties and allowable load factors are dictated by the FAR part 29 Subpart C (14 CFR): Strength Requirements.\textsuperscript{41}
12.4. Structural Sections

The airframe design (Figure 12.3) is based on the Advanced Technology Fuselage Research Program (ATF)\textsuperscript{42}, and uses, as a consequence, three primary load bearing carbon fiber bulkheads as well as several secondary frames so as to help maintain structural shape and to support auxiliary equipment like a hoist or some weapons for instance.

![Figure 12.3: Airframe Layout Overview](image)

The first primary bulkhead connects the cockpit to the center fuselage and forms the front support for the engine and transmission deck. The second primary bulkhead forms an intermediate support for the transmission deck. The third primary bulkhead supports the main landing gear as well as the engine mounts.

In addition to that, secondary bulkheads are located in the nose, between the first and second primary bulkheads, and aft of the fuselage. Three stages of stringers provide external shape continuity as well as support for door openings and for the pusher prop. Finally, a composite subfloor provide the crashworthy structure, the fuel tank housing, and ensure thanks to its monolithic construction no water ingress during an emergency water landing.

Large side doors offer rapid deployment capability and allow easy litter handling. On the other hand, the aft part of the fuselage is streamlined so as to ensure a laminar flow around its body.

Each of the primary bulkheads has been designed to provide proper load paths for the loads arising from landing, flight, take-off, ground handling and rotor operations. Landing loads are an important issue for a naval helicopter that has to cope with the unsteady motion of a ship deck, especially in case of extreme weather conditions for SAR missions.

The helicopter is designed according to FAR 29.337 for a maximum maneuvering load factor ranging from a positive limit of 3.5 to a negative limit of -1.0; or any positive limit maneuvering load factor not less than 2.0 and any negative limit maneuvering load factor of not less than -0.5.

![Figure 12.4: n-V Diagram](image)
The wind gust issue is also very important, especially in the case of a naval helicopter. In addition to the FAR 29.341 requirement, the rotorcraft has to remain safe in the event of a 50 knots horizontal or vertical gust at each critical airspeed.\textsuperscript{43}

12.5. Crashworthiness

As a military utility naval helicopter, the Peregrine may be subject to a large variety of loads, weather conditions, and has also to be able to operate through unfriendly airspace in its anti-tank or troop transport versions. Thus, the airframe is robust and designed with the goal of versatility in mind. Its geometry is such that bulkheads and stringers will collapse progressively in case of overloading so as to maximize energy absorption and dissipation. The objective is to minimize spinal loads for the passengers. Besides, protection of human occupants is achieved through the use of BAE Systems Mobility & Protection Systems S Series crashworthy seats.

The lightweight S7000 crew seat qualified based on MIL-STD 58095A and can accommodate a passenger whose weight ranges from the 5\textsuperscript{th}-percentile female to 95\textsuperscript{th}-percentile male thanks to variable load energy-absorber. It features a patented monolithic armored bucket which provides 30-cal. AP ballistic protection and a 5-point SCHROTH restraint system. The assembly weighs about 86 lbs.\textsuperscript{44}

The S3000 lightweight foldable troop seat is certified for a forward, aft, and side-facing mounting (TSO-C127 and TSO-C39b).\textsuperscript{45} It can accommodate a passenger whose weight ranges from the 5\textsuperscript{th}-percentile female to 95\textsuperscript{th}-percentile male. It features a 4-point SCHROTH restraint system and no tools are required for its installation so that the cabin arrangement can be quickly modified in response to an unexpected situation. This seat weighs 21.47 lbs and the protection of the passenger is ensured through a “tube and die” interface frame which weighs approximately 30 lbs, fixed to the fuselage of the helicopter. On the other hand, self-sealing crashworthy fuel tanks are also used to prevent ignition in case of a crash.

Heavy mass items such as the rotor hub or the drive-train have to be restrained so as to make sure that they won’t injure any occupant or intrude into the cabin in the event of a high-load crash. Ultimate inertial load factors (Table 12-1) are specified by the FAR part 29 Subpart C (14 CFR). Also included is the British Military requirement referred to as AVP 970.\textsuperscript{46}

<table>
<thead>
<tr>
<th>Direction</th>
<th>FAR 29</th>
<th>AVP 970</th>
</tr>
</thead>
<tbody>
<tr>
<td>Upward</td>
<td>4 g</td>
<td>7.5 g</td>
</tr>
<tr>
<td>Forward</td>
<td>16 g</td>
<td>15 g</td>
</tr>
<tr>
<td>Sideward</td>
<td>8 g</td>
<td>9 g</td>
</tr>
<tr>
<td>Downward</td>
<td>20 g</td>
<td>24 g</td>
</tr>
<tr>
<td>Rearward</td>
<td>1.5 g</td>
<td>15 g</td>
</tr>
<tr>
<td>Max. Resultant</td>
<td>20 g</td>
<td>24 g</td>
</tr>
</tbody>
</table>

Table 12-1: Ultimate Inertial Load Factors for each occupant and item of mass inside the cabin
On the other hand, sizing of the structure accounts for a 1.5 factor of safety. Moreover, doors have been designed so as to remain operable after a high load event and allow passengers to exit the vehicle as quickly as possible.

### 12.6. Fatigue Requirements

The Peregrine features a non-conventional drive system which generates vibrations that necessarily affect the whole airframe. Nevertheless, the structure was designed so as to avoid any catastrophic failure due to fatigue. A Fail-Safe approach was considered which means that the structure withstands design limit loads without failure after a partial failure and within its operational life.

Moreover, the extensive use of composite materials is expected to increase the lifespan of the primary airframe up to 12,000 hours as opposed to 10,000 hours for current naval helicopter designs.

### 12.7. Structural Details

#### 12.7.1. Engine and Transmission Deck

The engine and transmission deck supports the heart of the helicopter and isolates dangerous components from the cabin. It is to be made from titanium-aluminum plates (Ti-6Al-4V) bonded to the structure so as to ensure a good resistance to heat and fire and thus to protect the crew. This material has also better properties than composite panels in terms of vibrations and noise.

#### 12.7.2. Tail Section

The tail boom was deleted so as to use a pusher prop whose duct consists of a thermoplastic composite envelope including 4-ply (0, 90, 90, 0) of carbon fibers embedded in a matrix of polyether ether keytone (Peek). A Nomex honeycomb core is used to ensure light weight, and Kevlar-reinforced polyether imide (PEI) is used along the leading edge.

#### 12.7.3. Material Considerations

As it was mentioned previously, material selection was mainly driven by weight and part count considerations. Besides, criteria of importance included corrosion resistance, availability, productivity, and fatigue life. Thus, frame members are made from I-section carbon fiber beams composed of six different parts bonded together using a 120°C curing epoxy adhesive. They provide a higher stiffness-to-weight ratio and allow therefore reducing the number of structural members. The skin consists in four-ply carbon fiber skinned Nomex honeycomb sandwich panels. They were selected for their weight advantage, as well as for their resistance to corrosion, and the ease of manufacture. A curing adhesive allows bonding the panels to the frames.

However, in spite of the benefits listed above, using light-weight stiff composite panels is likely to affect the acoustic qualities of the fuselage (tests showed an increase of noise transmission in the cabin), while the resistivity of carbon may be of importance in case of a lightning strike for instance.

Even if composite material can be twice as much expensive in terms of acquisition cost than aluminum for instance, the cost savings achieved through manufacturing and maintenance due to a smaller part count and a better resistance...
to the environment can be expected to reach 18 percent for a carbon-framed fuselage. Moreover, a 16 percent weight reduction compared to a traditional metallic airframe seems to be a reasonable target.

Titanium-Aluminum beams were also considered but the benefits of this technology in terms of weight and cost savings were slightly less important.

12.7.4. Manufacturing Construction

Once the virtual prototype is complete, a simulation of the assembly would ratify the design feasibility. Since the structure is one of the most important components of the aircraft and it is the one that will support the drive system, the simulation of assembly focuses in this area. The manufacturing process is first layout in Process Engineer (software from Dassault Systems) which is transferable to DELMIA (software from Dassault Systems). This process helps define the path to follow in the manufacturing/assembly process. The simulation is divided in three different categories to ensure every aspect of the process is review:

- Digital Process for Manufacturing (DPM): This tool would corroborate the design manufacturability. It shows if CATIA models are too optimistic, something very important since it is a non-conventional design, once these defects are encounter, CATIA can be modified.
- Ergonomics design and analysis: It confirms that the aircraft has a realistic design. This step is very important since the pilots and crew member are set in place and their comfort is analyzed. An example of this is placing a mannequin as a pilot and checking the view from the window as shown in Figure 12.6. It helps the designer narrow the range of options of placing subsystems and visualized the limitations as well.

![Figure 12.6: Vision Window and Sight View of Pilot](image)

- Robotics: This Workbench helps optimized the manufacturing-assembly line if possible. A trade study is performed to analyze the feasibility of implementing robotics in the assembly line to reduce production cost reducing the life cycle cost of the aircraft. Figure 12.7 shows an example of robotics used in the assembly line.
12.7.5. Landing Gear

For better ground handling characteristics, a wheeled landing gear was selected. It is heavier than skids but allowed using deck lock devices embedded in the undercarriage track. So as to save weight though, a fixed landing gear similar to the AW159 was designed.

The deck lock is a claw-like hydraulic system featured by the naval version of the AW159 which enables to fix the helicopter on the gridded deck of a ship in case of poor weather conditions.

<table>
<thead>
<tr>
<th>Ship Type</th>
<th>Heading</th>
<th>Pitch Angle</th>
<th>Roll Angle</th>
<th>Vertical Acc’n</th>
<th>Lateral Acc’n</th>
</tr>
</thead>
<tbody>
<tr>
<td>Large ship</td>
<td>120°</td>
<td>1.0°</td>
<td>1.2°</td>
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</tr>
<tr>
<td></td>
<td>60°</td>
<td>0.6°</td>
<td>5.1°</td>
<td>0.03g</td>
<td>0.05g</td>
</tr>
<tr>
<td>Supply ship</td>
<td>130°</td>
<td>1.3°</td>
<td>1.3°</td>
<td>0.2g</td>
<td>0.05g</td>
</tr>
<tr>
<td></td>
<td>60°</td>
<td>0.8°</td>
<td>6.6°</td>
<td>0.04g</td>
<td>0.06g</td>
</tr>
<tr>
<td>Destroyer (stabilised)</td>
<td>120°</td>
<td>1.7°</td>
<td>1.5°</td>
<td>0.1g</td>
<td>0.08g</td>
</tr>
<tr>
<td></td>
<td>60°</td>
<td>1.0°</td>
<td>5.1°</td>
<td>0.03g</td>
<td>0.05g</td>
</tr>
</tbody>
</table>

Landing gear crashworthiness was defined according to FAR part 29 Subpart C (14 CFR) and the Engineering Design Handbook. The helicopter is able to operate from surfaces with fifteen degrees slope; its landing gear shall be able of fully decelerating the fully loaded vehicle from an impact velocity of 20 fps with only minor structural damage. Each wheel must be able to withstand the static ground reaction corresponding to the maximum gross weight accounting for a 25% allowance in gross weight growth and a critical center of gravity location.

12.7.6. Tire Sizing

Type III 12.50-16 tires were selected. Each of them has 14 plies and an inflation pressure of 90 psi. Type III tires are classified as low-speed, low-pressure tires, and are used where good flotation capabilities are desired. Each wheel can independently support a maximum static load of 15,000 lbs. Tire selection was based on MIL-PRF-5041K and FAR part 29 Subpart C (14 CFR) specifications.

12.7.7. Shock Absorbers Sizing

Air-oil strut sizing was accomplished according to MIL-L-8552 and taking into account that the landing gear must withstand a 12 inch drop test to meet the requirements of FAR part 29.727. With that, the Engineering Design Handbook provides a formula for oleo sizing:

\[ l = 1.25 \times \text{piston diameter} + \text{stroke} + \text{clearances} \]
Given an air-oil strut efficiency between 0.8 and 0.93\(^{56}\), a 20 fps descent velocity results in an oleo stroke of about\(^{57}\) 14.2 inches. Assuming a pressure of 3000 psi\(^{58}\) at the compressed position, the oleo inner diameter was found to be 2.75 inches. As a result, with 4 inches for clearances, the shock absorber has a length of 21.63 inches.

12.7.8. Emergency Floatation Gear

As the baseline rotorcraft from which derives the Peregrine is a naval helicopter, an emergency flotation capability is provided so as to enable the occupants to exit the vehicle in the event of a water crash or a forced landing. Thus, a four bag emergency system can be automatically deployed thanks to submersion switches. Flotation bags are located on both sides of the cockpit as well as in the landing gear fairings.

12.7.9. Doors

A pair of jettisonable sliding doors allows fast access to both the cockpit area and the troop seats. It also provides the cabin with a space where specific equipment can be set up so as to adapt the vehicle to a variety of missions, from special covert operations to SAR.

12.7.10. Survivability

In addition to a high dash speed, survivability is mainly achieved through the use of armored crew seats and subfloor. On the other hand, infrared signature is reduced by placing the engine exhaust in the rotor wake and using IR suppressors.

12.8. Finite Element Model (FEM)

As the design phase gets to the final iteration, a finite element model is developed in Abaqus for Catia to verify if the structure is going to be able to perform certain maneuvers and satisfy the requirements from 14 CFR – Part 29 (subpart c). The finite element model should be able to withstand any critical condition with the proper factor of safety, which is defined as the yield of the material over the maximum stress. A factor of safety of 1.5 is set as a constraint to optimize the structure. Based on the accuracy from first assumptions constructing the CATIA model, the number of iterations would vary from results of the FEM. However, since the CATIA model is parametric, these updates are performed almost instantly. Static loads are applied in different areas as seen in Figure 12.9, the main loads are applied from the hub and blades, transmission, engines and rear propfan since they will influence the primary structure. Loads applied were 18,512N for the hub and blades, 7,879N for the transmission, 3,332N for both engines and 3,400N for the rear propfan.
The structure was optimized constraining the maximum stress without reaching any plastic deformation for any metal and tensile and compression strength in case of the composite structure, the relation used between Von Mises, yield strength, and factor of safety was: $\sigma_{VM} \leq (\sigma_Y / FS)$

12.9. Cabin Layout

The cockpit provides seating for the pilot and his navigator. They both have access to a full set of flight controls and instruments.

The cabin was designed to accommodate up to eight fully equipped soldiers (270 lbs each) on foldable crashworthy seats arranged back to back so as to enhance the troop deployment capability. Its layout also reflects the multiple missions that can be assured by the helicopter. As a consequence, the troop seats and their supporting frames can be easily removed without using any specific tool so that the vehicle can be equipped with litters and accommodate a search and rescue team as well as up to five patients.
Given this arrangement and the technical choices that were made during the design process, the CG travel diagram shows that the center of gravity is clearly located forward of the fuselage which might be an issue in terms of handling qualities.

Figure 12.11: CG Longitudinal Travel

13. **Fuselage Aerodynamics**

High speed forward flight is one of the featured capabilities around which the Peregrine is designed. Confident prediction of the vehicle’s fuselage aerodynamics, drag in particular, is therefore extremely important. To this end, a computational fluid dynamics analysis of the Peregrine airframe was conducted in conjunction with empirical and spreadsheet-based drag buildup methods to provide the most accurate estimate of fuselage aerodynamic forces. CFD analysis of the airframe was conducted in Fluent 6.2 using grids generated in Gambit. Drag due to the external landing gear was estimated using Hoerner’s expressions for wheel and strut components. An analysis was also conducted to determine if the fuselage drag could be reduced by using suction from the ducted fan to direct the flow over the rear portion of the fuselage.

13.1. **Aerodynamic Simulation Setup**

An initial analysis of an approximate AgustaWestland Lynx fuselage geometry was conducted to validate the CFD model as well as provide a baseline set of lift, drag, and pitching moment data to use in the aerodynamic design of the Peregrine. Figure 13.1 shows the baseline Lynx fuselage with a centerline cut of the unstructured tetrahedral mesh generated around it in Gambit. Runs were conducted over +/- 10° sweeps of angle of attack and sideslip using the approximate fuselage length and frontal profile area as reference dimensions. Additionally, two runs were conducted at 30 m/s and 100 m/s freestream speed to confirm speed independence in the drag calculation. Table 13-1 summarizes the assumptions and models used in the CFD analysis of the fuselage.

Table 13-1: CFD Model Summary

<table>
<thead>
<tr>
<th>Grid</th>
<th>Unstructured Tetrahedral</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flow Models</td>
<td>Incompressible Flow K-ε Turbulence</td>
</tr>
<tr>
<td>( V_\infty )</td>
<td>40 m/s and 100 m/s</td>
</tr>
<tr>
<td>Sref</td>
<td>15 m²</td>
</tr>
<tr>
<td>Lref</td>
<td>15 m</td>
</tr>
<tr>
<td>( \alpha ) Range</td>
<td>+/- 10°</td>
</tr>
<tr>
<td>( \beta ) Range</td>
<td>+/- 10°</td>
</tr>
</tbody>
</table>

Figure 13.1: Baseline Lynx Fuselage
13.2. CFD Model Verification

The results of the baseline CFD runs were compared to polynomial expressions for lift, drag, and side forces developed from flight testing. All forces were normalized the reference quantities given in Table 13-2. The flat plate drag of the Lynx fuselage found with Fluent was compared to the flight test polynomial expression for drag at $\alpha = 0^\circ$ and $V_{\infty} = 100$ m/s with an estimated fuselage drag component of 30% of the total drag based on Prouty’s approximate drag buildup for conventional single main rotor helicopters\textsuperscript{59}. Table compares the fuselage flat plate drag area of the CFD runs at two freestream velocities compared to the polynomial curve fit of flight test data. Together with the comparison of lift, drag, and side forces presented in figures, the CFD model can be considered sufficiently accurate for use within the range of small angle approximations. The assumption of incompressible flow and speed independence are also justified based on the similarity of the drag results at the two freestream speeds covering nearly the full range of operation of the Peregrine in forward flight. The Peregrine is not expected to fly at speeds exceeding $M = 0.35$ in forward flight.

![Figure 13.2: Coefficient of Power (Cp) Distribution over Lynx fuselage at \(\alpha=0\)](image)

Figure 13.2: Coefficient of Power (Cp) Distribution over Lynx fuselage at $\alpha=0$

![Figure 13.3: Baseline Lynx Lift Coefficient](image)

Figure 13.3: Baseline Lynx Lift Coefficient

![Figure 13.4: Baseline Lynx Side Force Coefficient](image)

Figure 13.4: Baseline Lynx Side Force Coefficient

![Figure 13.5: Baseline Lynx Side Force Coefficient](image)

Figure 13.5: Baseline Lynx Side Force Coefficient

Table 13-2: Equivalent drag areas found using CFD and flight test data

<table>
<thead>
<tr>
<th>CF</th>
<th>Padfield</th>
</tr>
</thead>
<tbody>
<tr>
<td>$f$</td>
<td>$6.58 \text{ ft}^2$</td>
</tr>
</tbody>
</table>
13.3. Fuselage Design and Aeroloads

The goal of the fuselage design was to find the most favorable set of aeroloads – that is the lowest drag along with minimal adverse forces to be counteracted by control surfaces and stability augmentation. Without compromising weight and performance. Forces and moments about all 3 body frame axes were collected and used in the generation of a Flightlab model in addition to their consideration in the fuselage design process. X, Y, Z forces and L, M, and N moments in the body frame as well as the lift and drag forces in the wind frame were examined for each fuselage geometry considered.

The aerodynamic design of the fuselage consisted of three geometry iterations. Immediately after the selection of a compound coaxial rotorcraft in the configuration trade study, modifications were begun on the baseline Lynx fuselage toward realizing the high speed ducted fan concept. Fuselage Iteration 1 is simply the Lynx fuselage with the tailboom removed to make way for the eventual design of the ducted fan pusher propeller. Iteration 1 represents the very earliest stage of aerodynamic analysis which occurred before the fan duct design was completed. Figure 13.9 to Figure 13.12 detail the aerodynamic loads of Iteration 1. The most prominent effect of removing the tailboom was the high nose down pitching moment observed in Figure 13.11.

---

**Figure 13.5: Baseline Lynx Drag Coefficient**

**Figure 13.6: Vehicle Body Frame**

**Figure 13.7: Vehicle Wind Frame**
Iteration 2 incorporated more details into the fuselage geometry such as a rotor pylon and sponsons for landing gear housings. Iteration 2 was also the first fuselage design to include a duct at the fuselage trailing edge. Based on the high nose down pitching moment seen in Iteration 1, the nose and windshield were redesigned in Iteration 2 with a gentler slope upward. This was done due to the buildup of very high pressure on the windshield of Iteration 1 which can be seen as the red and orange regions in Figure 13.9.
The nose and windshield modifications made in Iteration 2 clearly mitigate the high pressure region seen in Iteration 1. Figure 13.14 shows a much more moderate pressure buildup on the leading edge of the fuselage. These design aspects were thus kept for Iteration 3. Iteration 3 is the final fuselage geometry settled upon for the Peregrine after the previous two cycles of design and analysis. It includes the same features of the nose cone, windshield, and duct which were developed based on the CFD runs conducted in the previous iterations. The duct was increased in size in Iteration 3 in an attempt to reduce drag by making it conform more to the shape of the rear fuselage. The larger duct was also designed to support the horizontal tail which was sized based on the pitching moment generated by the fuselage. The rotor pylon was also integrated more smoothly with a lower profile into the rest of the body to reduce drag.
Figure 13.18: Final Fuselage

Figure 13.19: Final Fuselage \( C_p \) Distribution at \( \alpha = 0 \)

Figure 13.20: Final Fuselage Body Forces

Figure 13.21: Final Fuselage Lift Coefficient

Figure 13.22: Final Fuselage Pitching Moment

Figure 13.23: Final Fuselage Lateral Moments versus \( \beta \)
13.4. Aerodynamic Design Results

The lift, drag, pitching moment, and equivalent flat plate drag area of all 3 fuselage iterations at the trim targeted trim attitude of zero degrees pitch and zero degrees sideslip are summarized in Table 13-3. Plots of drag and pitching moment across the full angle of attack sweep are given in figures. The iterative aerodynamic design process resulted in a reshaping of the nose, windshield, pylon, and duct to reduce drag and nose down pitching moment. In spite of the measures taken to reduce drag, the equivalent flat plate area is shown to increase with each progressive step due to the addition of components such as sponsons and landing gear housings to the fuselage. Drag reduction will be further discussed in the following section. The efforts to reduce adverse pitching moment also proved to be somewhat ineffective, as CM stays relatively unchanged for each geometry. It was thus concluded that a horizontal tail should be added to the fuselage design. The horizontal tail was sized according to the following formula, where $I_{stabil}$ is the distance from the fuselage center of gravity to the aerodynamic center of the tail.

$$S_{stabil} = \frac{-C_M S_{ref} L_{ref}}{C_{L,stab} I_{stabil}}$$

Table 13-3: Fuselage Aeroloads at $\alpha=0$

<table>
<thead>
<tr>
<th></th>
<th>It. 1</th>
<th>It. 2</th>
<th>It. 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>CL</td>
<td>-0.02108</td>
<td>0.03202</td>
<td>0.04568</td>
</tr>
<tr>
<td>CD</td>
<td>0.04294</td>
<td>0.05300</td>
<td>0.07037</td>
</tr>
<tr>
<td>CM</td>
<td>-0.01197</td>
<td>-0.01234</td>
<td>-0.01241</td>
</tr>
<tr>
<td>$\ell$</td>
<td>7.28 ft²</td>
<td>8.56 ft²</td>
<td>11.36 ft²</td>
</tr>
</tbody>
</table>
13.5. Drag Analysis

Accurate prediction of the fuselage drag is critical to the Peregrine’s high speed flight capability. A hybrid method combining CFD drag estimation for the fuselage and most of its components with Hoerner’s drag buildup method for the undercarriage components of the landing gear wheels and struts was used. According to the literature, this combined CFD and semi-empirical calculation is commonly used for fuselage design in the absence of a wind tunnel model for drag estimation. The ducted pusher propeller thrust was simulated in Fluent to see if the effect of the propeller suction would reduce the drag due to the fuselage.

13.5.1. Fuselage Drag

Based on the polynomial expression for body forces, the DRA research Lynx Mk 7 has a total equivalent flat plate drag area of 21.03 ft², of which 6.31 ft² is due to the fuselage. The Peregrine fuselage contributes 11.36 ft² to the total flat plate drag area of the vehicle. The larger disproportionate contribution of the Peregrine fuselage is an immediate indication that some effort should be made to reduce its drag.

13.5.2. Component Drag

An empirical baseline Lynx drag buildup was conducted based on the total experimentally verified drag area of 21.03 ft² with the percentage contribution of each component assigned according to Prouty’s representative breakdown of component drag for a conventional single main rotor helicopter. For the Peregrine drag buildup, Hoerner’s method was used for the landing gear drag based on a C₀=0.31 for landing gear with hollow wheels and a reference area equal to the tire width times the diameter. The landing gear struts were considered simple cylinders. The other external components were run in Fluent separately and then attached to the fuselage to calculate not only the component drag, but also the installation drag due to interference. An important reduction in the drag due to the hub was achieved through the use of HMA blade control which drastically reduces the number of exposed parts. An additional reduction in hub drag is achieved by trimming the Peregrine to fly at α = 0° instead of a nose down trim angle. Table 13-4 shows the results of the runs conducted on both the isolated rotor hub and the rotor-fuselage combination to determine the hub’s installation drag. Table 13-4 lists the drag buildup by component for both the Lynx and Peregrine. The initial total flat plate drag area of the Peregrine is 15.96 ft². Section 13.5.3 will describe the efforts undertaken to improve the Peregrine’s forward flight performance by drag reduction.

Table 13-4: Baseline Lynx Drag Buildup

<table>
<thead>
<tr>
<th>Component</th>
<th>f (ft²)</th>
<th>%</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>6.31</td>
<td>30</td>
</tr>
<tr>
<td>Main Rotor Hub</td>
<td>7.36</td>
<td>35</td>
</tr>
<tr>
<td>Landing Gear</td>
<td>2.10</td>
<td>10</td>
</tr>
<tr>
<td>Interference</td>
<td>1.47</td>
<td>7</td>
</tr>
<tr>
<td>Tail Rotor Hub</td>
<td>0.84</td>
<td>4</td>
</tr>
<tr>
<td>Empennage</td>
<td>0.42</td>
<td>2</td>
</tr>
<tr>
<td>Misc. Components</td>
<td>2.52</td>
<td>12</td>
</tr>
<tr>
<td>TOTAL</td>
<td>21.03 ft²</td>
<td>100</td>
</tr>
</tbody>
</table>

For the Peregrine:

<table>
<thead>
<tr>
<th>Component</th>
<th>f (ft²)</th>
<th>%</th>
</tr>
</thead>
<tbody>
<tr>
<td>Airframe</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fuselage</td>
<td>10.36</td>
<td>64.91</td>
</tr>
<tr>
<td>Fan Duct</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Exhaust</td>
<td>1.00</td>
<td>6.27</td>
</tr>
<tr>
<td>Pylon</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Sponsos</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Landing Gear</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wheels</td>
<td>0.72</td>
<td>4.51</td>
</tr>
<tr>
<td>Struts</td>
<td>0.39</td>
<td>2.44</td>
</tr>
<tr>
<td>Hub</td>
<td>2.70</td>
<td>16.92</td>
</tr>
<tr>
<td>Hub Installation</td>
<td>0.79</td>
<td>4.95</td>
</tr>
<tr>
<td>TOTAL</td>
<td>15.96 ft²</td>
<td>100</td>
</tr>
</tbody>
</table>
13.5.3. Drag Reduction

The Peregrine’s fuselage contributes about 65% of the vehicle’s total drag in forward flight. Considerable performance gains could be made if this figure were to be even marginally reduced. A brief investigation was conducted on boundary layer control over the aft portion of the fuselage. The shape of the fuselage and propeller duct employ what may be called aerodynamic static pressure thrust or negative form drag to prevent the boundary layer near the fuselage trailing edge from displacing the flow from the fuselage surface. The power provided to the pusher propeller system is thus used not only for thrust, but also for drag reduction. A survey of this concept was conducted by Goldschmied with specific investigations of the use of ducted fans done by Fanucci and McLemore. The previous experimental studies have yielded promising results as high as 40 - 50 % propulsive power reduction for axisymmetric bodies. Even a modest reduction in flat plate area drag would result in substantial performance improvement at all cruise conditions. An actuator disk was created inside the duct halfway between the axial locations of the counterrotating fans. The fuselage drag was simulated at 100 m/s while prescribing the pusher propeller system’s maximum thrust of 3587 lbf using pressure inlet and pressure outlet boundary conditions with a target mass flow assigned across the disc for continuity. The actuator disc was considered separate from the fuselage wall boundaries in Fluent in order to isolate the actual thrust generated by the propeller and determine only the change in axial force due to the change in flow pattern. Table 13-5 details the reduction in airframe drag.

<table>
<thead>
<tr>
<th>f</th>
<th>C_D</th>
</tr>
</thead>
<tbody>
<tr>
<td>No Prop</td>
<td>0.064</td>
</tr>
<tr>
<td>Max Thrust</td>
<td>0.059</td>
</tr>
<tr>
<td>Δf</td>
<td>1.47 ft$^2$</td>
</tr>
<tr>
<td>% Reduction</td>
<td>12.9%</td>
</tr>
</tbody>
</table>

Table 13-5: Drag Reduction Due to Propeller at $V_\infty = 100$ m/s

Figure 13.28: $V_\infty = 100$ m/s, propeller not engaged
Figure 13.29: $V_\infty = 100$ m/s, propeller engaged

Figure 13.28 and Figure 13.29 show the effect of the propeller suction on the trailing edge fuselage flow. The flow is clearly more attached. Rather than avoiding an adverse pressure gradient region leading into the duct seen in, the flow is sucked into the duct while following the shape of the aft fuselage much more closely. The final result of the simulated thrust analysis was a 12.9% reduction in fuselage drag. Table 13-6 lists the final drag buildup for the Peregrine with the fan operating in forward flight.

<table>
<thead>
<tr>
<th>Component</th>
<th>$f_{\text{fan off}}$</th>
<th>$f_{\text{fan engaged}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Airframe</td>
<td>10.36</td>
<td>9.23</td>
</tr>
<tr>
<td>Fuselage</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fan Duct</td>
<td>1.00</td>
<td>0.66</td>
</tr>
<tr>
<td>Exhaust</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Pylon</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Sponsons</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Landing Gear</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wheels</td>
<td>0.72</td>
<td>0.72</td>
</tr>
<tr>
<td>Struts</td>
<td>0.39</td>
<td>0.39</td>
</tr>
<tr>
<td>Hub</td>
<td>2.70</td>
<td>2.70</td>
</tr>
<tr>
<td>Hub Installation</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Drag</td>
<td>0.79</td>
<td>0.79</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td><strong>15.96 ft^2</strong></td>
<td><strong>14.49 ft^2</strong></td>
</tr>
</tbody>
</table>

Table 13-6: Final Drag Buildup (Propeller operating at $T = 3587$ lbf)

14. FLIGHTLAB Model

GTPDP is based on rather simple analytical and empirical formula and is widely used to generate performance data quickly and thus to compare various configurations in early design stages. However, even if it is validated with several helicopter designs, it is still not suitable to generate detailed engineering data which are needed beyond conceptual and preliminary design processes. At this phase, a simulation and analysis tool with high fidelity can be used effectively along with wind tunnel tests. This design team used FLIGHTLAB, which is widely used in both industries and academia, to build a detail simulation model for detailed engineering analysis that will follow this preliminary design. This chapter describes a FLIGHTLAB model of the Peregrine and analysis results.

14.1. Introduction to FLIGHTLAB

FLIGHTLAB is a comprehensive modeling, simulation, and analysis tool for rotorcraft. The program consists of two parts, flme for rotorcraft modeling and xanalysis for simulation and analysis. The primary simulation engine of FLIGHTLAB is called Scope. The Scope is a MATLAB-like interpretive language and it has various kinetic, aerodynamic, structural, and control component classes that can be used to model and simulate varieties of dynamic systems, not just rotorcraft. FLIGHTLAB features a rotorcraft modeling tool named flme which has a template for a
rotorcraft of a two rotor system. (Figure 14.1). Once a rotorcraft model is built with flme, it can be analyzed with xanalysis. (Figure 14.2) The two most important functions of xanalysis are trim and simulation. User can generate trim settings of the rotorcraft at various test conditions automatically.

Figure 14.1: flme Sample Screen

Figure 14.2: xanalysis sample screen
14.2. FLIGHTLAB model description

The FLIGHTLAB model provides users flexibility in terms of modeling fidelity. High fidelity models of finite element and free wake components are complex enough to be used for vibration analysis. However, this needs significant amount of computation resources. Performance analysis usually does not require such a detailed model. A rotor model of rigid blade elements and a dynamic inflow model may be sufficient for this purpose.

14.3. Rotor Model

This model consists of coaxial rotor and propeller. Table 14-1 shows the rotor configuration data used in the FLIGHTLAB model. This FLIGHTLAB model includes flapping and lead-lag hinges. The Peters-He 3-state inflow model is used for both rotors. The 3-state interference model is used to account for the interference between the two rotors.

<table>
<thead>
<tr>
<th>Table 14-1: Rotor Configuration</th>
</tr>
</thead>
<tbody>
<tr>
<td>Rotor type</td>
</tr>
<tr>
<td>Blade motion</td>
</tr>
<tr>
<td>Rotational direction</td>
</tr>
<tr>
<td>Rotational speed (nominal)</td>
</tr>
<tr>
<td>Rotor radius</td>
</tr>
<tr>
<td>Num. of blade per rotor</td>
</tr>
<tr>
<td>Num. of segment per blade</td>
</tr>
<tr>
<td>Blade twist</td>
</tr>
<tr>
<td>Airload</td>
</tr>
<tr>
<td>Airfoil</td>
</tr>
<tr>
<td>Inflow</td>
</tr>
<tr>
<td>Interference</td>
</tr>
</tbody>
</table>

14.4. Fuselage

Fuselage is assumed to be rigid. Detail data of fuselage model is as follows.

- Total weight: 14,772 [lbf]
- Reference area: 161.5 [ft^2]
- Reference length: 49.2 [ft]

The aerodynamic coefficient for fuselage and ducted fan appeared in the previous sections.

14.5. Control Riggings

Since the purpose of current analysis is analyzing basic performance and acquiring trim control settings, a simple mechanical rigging is used to represent the control system. Control system is separated into four channels- roll, pitch, yaw, and heave. Figure 14.3 shows the structure of longitudinal channel. B1S1out and B1S2out are swashplate longitudinal cyclic angle of upper rotor and lower rotor, respectively. These angles are of the same values.
Figure 14.3: Linkage of Longitudinal Control Input

Figure 14.4 shows the structure of lateral channel. Basically, this structure is similar as longitudinal one. However, values for the upper and lower rotors have opposite signs.

Figure 14.4: Linkage of Lateral Control Input

Figure 14.5 depicts the collective and pedal control linkages that are coupled. Collective input affects the pitch angles of the upper and lower rotors in the same direction. Pedal input affects the pitch angles in different ways.

Figure 14.5: Linkage of Collective and Pedal Control Input

14.6. Horizontal Stabilator

Initial analysis showed the fuselage generates significant amount of pitch-down moment along with the two rotors. This resulted in large power required and small control margin. So, it was proposed to attach a horizontal stabilizer at the end of tail. The stabilizer is modeled as follows:

Airfoil: NACA2410 (inverted)
Area: 2 x 11.0 [ft²]
Distance from CG: 14.0 ft
Incidence: -4.0 degrees

To augment control moment, vertical and horizontal movable fins are placed in the propeller downstream. Basic configuration parameters are as follows:

Airfoil: NACA 2410
Area: 3.44 [ft²]
Distance from CG: 15.0 ft

The action of control fins should be modeled as an integral part of the pusher because they are under influence of propeller downwash. The propeller thrust and power are computed using 2D tables which are functions of speed and tip angle of attack. The thrust is used to compute the inflow. In this case a formula for axial flight is used. This
inflow determines the free stream condition for the stabilators. Using the flow condition and the angle setting, lift and drag are computed for each stabilator. The lift is used as the control force. The resultant thrust of the pusher is computed by subtracting the drag from the propeller thrust. The scheme is represented in Figure 14.6.

![Figure 14.6: Pusher and control fin model](image)

14.7. Analysis results

Trim settings are computed at various forward speeds. The primary purpose of this analysis is to find trim control settings and performance data.

Figure 14.7 shows the force component along the body axis x-direction. During the trim, the propeller is set to compensate the fuselage drag automatically. This figure confirms this configuration.

Figure 14.8 shows the z-direction force. The horizontal stabilator generates downward z-force as pre-programed. The z-component of propeller thrust also increases as the speed increases. This is necessary to compensate the pitching moment. As the downward forces from horizontal stabilator and propeller increase, the rotor thrust also increases to keep the balance in z-direction.

Figure 14.9 shows the pitching moment components. Myx means resultant pitch moment at CG due to Fx, and Myz due to Fz. The pitch down moments from the main rotor and fuselage increases with the speed. To compensate this, propeller generates an increasing downward force which generates pitch up moment. The stabilator almost exactly cancels out the pitch-down moment from fuselage.

Figure 14.10 shows trim power settings. The power required by the propeller is used to compensate fuselage drag and also to generate downward force for pitching moment. As speed increases, this propeller power dominates and becomes even bigger than the power consumed by the two rotors. Since the rotor does not generate the forward component, contrary to the conventional rotorcraft, its power does not increase so rapidly as the propeller power does. The difference between the amounts of power required by the upper and lower rotors is not that significant even if an inflow model is utilized. One of the primary reasons is believed to be the short separation distance.

Figure 14.11 shows trim control settings. All the control settings show conventional behaviors. The vehicle shows clearly positive longitudinal static stability, while should alleviate pilot loads.

The maximum speed is limited by the thrust force. Majority of the thrust force is used to generate pitch up moment not to compensate the fuselage drag.
Figure 14.7: Trim Settings – Body x-dir Forces

Figure 14.8: Trim Settings – Body z-dir Forces
Figure 14.9: Trim Settings – Body Pitching Moment

Figure 14.10: Trim Power Settings
15. Handling Qualities Improvement and Piloted Simulation

Evaluation of the handling qualities of the Peregrine configuration was performed using the nonlinear FLIGHTLAB model described in section 14. The evaluation has been subdivided into two primary tasks:

1. Assessment of predicted levels of handling qualities based upon published performance standards such as ADS-33E-PRF65
2. Assessment of assigned handling qualities through real-time, piloted simulation of the model

The results from this analysis of the unaugmented aircraft were used as a basis for the development of a feedback controller to improve the handling qualities and reduce the pilot’s workload.

15.1. Handling Qualities of the Unaugmented aircraft

The ADS-33 handling qualities philosophy is that the performance criteria can be broken down depending upon the required amplitude and frequency of a pilot’s control inputs. This approach gives us the dynamo construct, as seen in Figure 15.1.
The dynamo construct is subdivided into four primary areas:

1) High frequency, small amplitude inputs – assessed through bandwidth analysis
2) Low to medium frequency, small amplitude inputs/disturbances – assessed through modal stability analysis
3) Moderate amplitude inputs – assessed through the attitude quickness
4) Large amplitude inputs – assessed through control power

In addition to these criteria, assessed in each axis of response, we can add the strength of couplings between the axes.

Analysis of the Peregrine model revealed several areas in which deficiencies were present and improvements were required. The first of these was extreme sensitivity to control inputs, and was experienced in all axes. This was manifested by very high figures for control power, quickness and bandwidth. An example of the very high quickness is shown in Figure 15.2 where the results fall to the upper right-hand corner of the chart.

Although analysis against the ADS-33 parameters generally gave Level 1 results for these agility criteria, sensitivity to the level exhibited here can be detrimental to the overall performance of an aircraft. This is especially so when the high sensitivity is accompanied by poor stability. In these circumstances, the poor stability would mean that the pilot would want to interact with the controls on a regular basis to manually stabilize the response, but the high sensitivity would restrict his ability to do this, as a pilot would generally want to minimize the magnitude and frequency of control inputs here to eliminate the possibility of over-controlling the aircraft.
In the case of Peregrine, stability is indeed poor for both lateral-directional and longitudinal responses. Stability variation across the flight envelope is illustrated in Figure 15.3. The Dutch Roll and Phugoid modes are both neutrally stable in the hover. In the case of the Dutch Roll, stability remains neutral across the speed range, due to the lack of vertical surface area or a tail rotor at the rear of the aircraft to provide stabilization. As the airspeed increases, the natural frequency of the Dutch Roll mode increases, moving into the Level 2 region. The Phugoid mode becomes progressively more unstable as the airspeed increases, although in this case the frequency decreases. The Phugoid mode is still predicted to result in Level 2 handling qualities, however. The Short Period mode is not present in the hover and at very low speeds, the associated states giving decoupled pitch and heave subsidence modes instead. As the airspeed increases, these modes couple together to form the Short Period oscillation, which is well damped, giving predicted Level 1 handling qualities. As the airspeed increases, the frequency of the mode, which had been increasing, begins to decrease, leading to the pitch and heave states becoming once again decoupled as the airspeed increases beyond 150 knots. Of note is that the pitch mode becomes progressively less stable as the airspeed continues to increase, leading to a significant instability at the very highest speeds – in the example shown at 210 knots, the time to double amplitude of this pitch divergence mode is as little as 0.8s.

The coaxial configuration of the aircraft means that the strength of cross couplings is minimal, with Level 1 results for all ADS-33 criteria. The primary goals for handling qualities augmentation therefore, are improvements to the natural stability of the aircraft and reduced sensitivity of response following control inputs.

15.2. Control Augmentation

In order to address the poor modal stability, damping was added into each channel through feedback of the appropriate angular rate, this being combined with the pilot’s control inputs and fed through a proportional, integral and derivative (PID) controller to deliver a rate command response. However, whilst the additional damping would reduce the sensitivity of the response slightly, it would not be sufficient to place the predicted handling qualities at the appropriate places on the ADS-33 charts. An additional element was therefore added in to the augmentation scheme – a set of transfer functions that encapsulate the handling qualities that are desired. This is an approach that has previously been applied successfully to tilt rotor aircraft66. The response of the transfer function was selected to give good Level 1 handling qualities across the flight envelope.

The structure of the controller is shown in Figure 15.4. The three gains (Kp, Ki and Kd) were tuned at a variety of airspeeds to optimize the response and the match the desired response specified by the transfer function as well as possible. The resulting set of gains was then scheduled against the airspeed to deliver an optimized response across the flight envelope.
An example of the improvement in response offered by the handling qualities upgrade is shown in Figure 15.5. The applied input in each case was of the doublet type, and the amplitude at the stick was the same for both runs. The reduced sensitivity is clear. Uncommanded oscillations have been significantly reduced, and roll rate tracking has been improved. Similar improvements in response have been made in the other axes and at other airspeeds.

How does this improved response fare when analysed against the ADS-33 performance criteria? Figure 15.6 shows the roll quickness, assessed with the control augmentation active, with the aircraft in the hover. It is apparent that the quickness is lower than that of the unaugmented aircraft, seen in Figure 15.2. However, the results show that performance should still be suitable for the most demanding tasks.
Results are similar in the pitch axis. One further example of the improved handling qualities is shown in Figure 15.7, where the pitch axis bandwidth is illustrated. The augmented aircraft exhibits a bandwidth that falls on the Level 1/Level 2 boundary for the most aggressive tasks. This is in fact a reduction from the unaugmented aircraft. However, the high bandwidth result prior to the handling qualities upgrade was due to the very high response sensitivity, and so the reduction could well be beneficial to the overall handling qualities of the aircraft.

Results have been presented here for the hover condition only. However, the nature of the controller is such that handling qualities will remain constant across the flight envelope.

15.3. Piloted Simulation

Piloted, real-time simulation of both Peregrine configurations – unaugmented and augmented – has been performed using the University of Liverpool’s Heliflight motion flight simulator. Heliflight is a reconfigurable, PC-based system based upon the high-fidelity FLIGHTLAB modeling environment, and is capable of providing a realistic simulation environment for both rotary- and fixed-wing aircraft. Heliflight is illustrated in Figure 15.8.

The piloted simulation was conducted with the assistance of an experienced helicopter pilot. Each sortie consisted of a general familiarization with each configuration, followed by a series of tasks that may typically be considered to form part of the overall mission profile required of Peregrine.

The first configuration to be assessed was the bare airframe, unaugmented aircraft. The pilot immediately commented on the very sensitive controls that forced him to significantly limit the aggression with which corrective
control inputs could be applied, with subsequent consequences on the accuracy of flight that could be achieved. As the aircraft was accelerated away from the hover, the lack of directional stability became apparent, with high workload being required in the yaw axis to suppress undesired sideslip. This deficiency became progressively more evident as the airspeed increased, and ultimately, when combined with the unstable pitch mode, produced a workload that was intolerable – control could not be reliably retained for extended periods of time.

Turning to the augmented configuration, the improved stability was immediately obvious, with the pilot able to achieve an accurate hover with a relatively low workload. As the aircraft was taken towards the cruise condition, the pilot commented that, although the overall workload had been significantly reduced, the yaw axis was still somewhat problematic. This was because, although the yaw controller was working as designed, this still did not provide any directional stability – the pilot had to close the loop between heading and sideslip manually. The incorporation of a turn coordination system would alleviate this deficiency. Overall, the pilot expressed satisfaction at the improvement to the aircraft, commenting that the full mission profile could now be accomplished with low workload and in without risk of loss of control.

Turning to the specific tasks, the first was a hover turn – a heading change whilst remaining above the same point on the ground. In Figure 15.9 the extremely small movement on the pedals required to generate the yaw rate with the unaugmented aircraft is evident. Station keeping above the hover point is rather poor here also. With the augmented aircraft, a more conventional scaling between pedal deflection and yaw rate can be seen, and position keeping was much more accurate. However, there does appear to be room for improvement in the yaw response, as some corrective inputs can be observed in the pedals as the pilot attempts to capture his final heading. The comment from the pilot was that this was primarily due to the sensitivity of the response still being somewhat higher than he would find comfortable.

![Figure 15.9: Hover Trim Maneuver](image)

The second test was a departure away from the hover, following a steady acceleration profile. This maneuver is shown in Figure 15.10. The high workload with the unaugmented aircraft is immediately clear. This workload consists primarily of high frequency corrective inputs, of the sort that demands a continuous high level of concentration from a pilot, and is thus rather tiring. The large, irregular pedal inputs required to overcome the lack of directional stability can be seen. In contrast, the augmented aircraft requires a much lower level of compensation from the pilot, especially so once the acceleration profile has been established.
Finally, steady cruise at 2000ft was examined. The results from this test are shown in Figure 15.11. The lack of directional stability results in large sideslip excursions. In addition, the pitch attitude and bank angle are by no means steady. This lead to a very inconsistent speed profile, and ultimately, a loss of control, as the oscillations in all axes developed to a point at which the pilot was no longer able to apply the required corrective inputs at the necessary frequency.
With the augmented aircraft, performance was much better. Although some disturbances can be seen, they are of a much smaller magnitude than those seen with the unaugmented aircraft. In addition, the improved stability conferred by the controller gave the pilot confidence that he could make rapid corrections to suppress the disturbances, leading to the desired flight path being recaptured quickly and with low workload. The pilot was able to command a relatively smooth acceleration up to the cruise speed, and, once there, maintain the desired speed accurately.

16. Cost Analysis

Accurate cost estimation is critical during the design process. Since this project is based on modifications of an existing baseline, costs are examined both as a stand-alone value and as a percent change to the baseline. These values are then applied to the OEC for the Cost Index in Section 2.4.4.

16.1. Overview of Life Cycle Cost

Life cycle cost is the total cost of ownership of machinery and equipment, including its cost of acquisition, operation, maintenance, conversion, and/or decommission. The objective of LCC analysis is to choose the most cost effective approach from a series of alternatives in order to achieve the lowest long-term cost of ownership.

![Life Cycle Cost Concept](image)

The initial step of the design should consider the development cost, comprised of the design phase, flight testing and systems management. The manufacturing phase along with the dynamic system, engines, and avionics are considered in the recurring production cost. Lastly, the operating costs are the summation of the flight crew cost, fuel, airframe maintenance, and engine maintenance.

Since this design project is in response to an RFP focusing on the development of a non-conventional drive system, a standard LCC approach may yield limited insight due to lack of a comparable customer base and historical data. The primary consideration for all LCC results is the impact those values will have on the OEC.

The Bell PC Model was used for the calculations in this section. Some assumptions were made in order to obtain a reliable model, and the non-conventional configuration was approximated in the Bell PC Development Inputs. For example, although the drive system is an alternative idea, it will be developed from modifications to existing technologies and components, therefore a 75 percent new design is used. For the rest of the helicopter, reasonable assumptions have to be considered as well. The values generated by the program, in 2001 dollars, were adjusted to 2009 dollars by using a CPI adjustment of 23% increase. Two specific systems, the drive system and rotor blade, were examined more closely with alternative pricing methods to better model recurring costs.
16.2. Engine Cost Model

The Bell PC Cost model allocates $1.01 million per unit for powerplant cost, with an estimated $242/FH as a DOC for powerplant overhaul and maintenance. Although the design uses an off-the-shelf available engine, the LHTEC CTS800-5N, no cost information was available for comparison.

16.3. Transmission Cost Model

The Bell PC Cost model allocates $169,000 (2009$) per unit for the cost of the main transmission and mast, with a weight of 824 lb (374 kg). Additional gearboxes and shafting comprise the rest of the overall $397,000 (2009$) allocated for the unit drive system, totaling 1048 lbs (476 kg). These values are the average costs for the first 100 units with a learning curve applied. As compared to the baseline unit weight and cost for the Superlynx of 884 lb (401 kg) and $336,000, this is an 18% increase in both weight and average cost. Research and development costs associated with the drive system are estimated to take 277,000 engineering, design, and testing man-hours and total $19.3 million (2009$). Further analysis was performed to estimate the average cost of 400 units by comparing three methods.

16.3.1. Method 1

The first method was simply adjusting the Bell PC Cost model program to calculate the average cost of the drive system if helicopter production was increased to 400 units.

16.3.2. Method 2

The second method involved extracting the Cost Estimating Relationships (CERs) used within the Bell PC Cost model and applying them to the results of the gear sizing optimization performed for the Peregrine. Although the gears were sized precisely, the housings and cases were estimated. Since this initial estimation was very close to the original weight allocated by Bell PC Cost model, the housing estimation factor was increased by 20% to have the total system weight be equal for all methods (1,048 lbs). These CERs were first applied to calculate the theoretical first unit cost, and a learning curve was applied (88% for labor and 93% for material and subcontract costs).

16.3.3. Method 3

Using CERs from a Production Cost Trade-Off Tool based on a RAND report from the early 1990s, manufacturing estimates for Quality Assurance and Manufacturing man-hours, and for material costs were calculated.

Results

By increasing the production quantity from 100 to 400, the average cost per unit for 400 drive system assemblies is reduced to $328,000 (2009$) and will take approximately 2,500 man-hours to complete.
16.4. Main Rotor Blade Cost Model

Similar cost methods as described for the drive system above were also applied to the production of the main rotor blade, which facilitated a trade study for selecting materials and manufacturing processes. The starting point for this comparison was once again the basic output from the Bell PC Cost Model. These results are shown in Table 16-1 (listed in 2001$).

Table 16-1: Initial Bell PC Production Costs – 2001$

<table>
<thead>
<tr>
<th>Category</th>
<th>Composite</th>
<th>Metal</th>
</tr>
</thead>
<tbody>
<tr>
<td>Blade weight</td>
<td>61.6 lb</td>
<td>62.5 lb</td>
</tr>
<tr>
<td>1st Production Unit</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Labor – hours</td>
<td>627.6 hrs</td>
<td>542.5 hrs</td>
</tr>
<tr>
<td>Labor - cost</td>
<td>$25,105</td>
<td>$21,700</td>
</tr>
<tr>
<td>Material cost</td>
<td>$13,369</td>
<td>$5,549</td>
</tr>
<tr>
<td>Subcontract cost</td>
<td>$9,733</td>
<td>$32,686</td>
</tr>
<tr>
<td>TOTAL</td>
<td>$48,207</td>
<td>$32,686</td>
</tr>
<tr>
<td>Average for 100 Units</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Labor – hours</td>
<td>405.8 hrs</td>
<td>350.8 hrs</td>
</tr>
<tr>
<td>Labor - cost</td>
<td>$16,228</td>
<td>$14,029</td>
</tr>
<tr>
<td>Material cost</td>
<td>$11,738</td>
<td>$4,872</td>
</tr>
<tr>
<td>Subcontract cost</td>
<td>$8,546</td>
<td>$4,774</td>
</tr>
<tr>
<td>TOTAL</td>
<td>$36,512</td>
<td>$23,675</td>
</tr>
</tbody>
</table>

From this start-point, production costs for combinations of materials for the Spar, Core, and Skin of the rotor blade were calculated. Taking the total blade volume from CATIA (0.078m$^3$, 2.75ft$^3$), the percent volume of the spar, core, and skin were then used to calculate the component weight using each material’s density. This allowed for easy switching between various materials of different densities. The materials, combinations, and volumes are presented in Table 16-2. It is of note here that metal was not considered for use in the skin due to the planform of the selected blade. Additionally, only materials with high stiffness properties were considered due to the desired hub spacing.

Table 16-2: Material Combinations
Overall, five methods were used to compare manufacturing labor, material costs, and manufacturing processes, although only the three costing methods outlined above were validated enough to be reliable. Figure 16.3 compares the four different blade material combinations, and demonstrates how much manufacturing contributes to total cost.

Results showed that the best material combination for the blade is for a composite spar and skin, but with an Aluminum honeycomb core. Since a large portion of the manufacturing process for composites is layup and curing, an alternative to the traditional autoclave was sought for cost savings. Autoclaves, used to apply both heat and pressure to the component for an extended period of time, rely on inert gases as the heat transfer medium and temperature change rates are limited to 1-3 °C/min. A relatively new curing method, called Quickstep (from Quickstep Technologies\textsuperscript{69}) uses liquid Heat Transfer Fluid to heat and cool the component during curing, able to increase heating/cooling rates to 6-8 °C/min. In addition to the time and energy benefits listed in Figure 16.4, this process promises better curing due to increased viscosity within the composites resulting in better permeation and adhesion of the matrix. This process also does not require waiting on the batching of parts for curing. Based on the advertised savings of this process, the initial cost results (still for unit T1) were adjusted by reducing manufacturing man-hours and material costs. These results are shown in Table 16-3.
Benefits of Quickstep Process:
- Materials cost reduced by 21%
- Tooling and Capital investment lowered by 62%
- Time Savings of 43%
- Layup, bagging and curing labor lowered 25%
- Energy cost reduction of 80%

Figure 16.4: Rotor Blade Cost

Table 16-3: Average Blade Cost Comparison – 2001$

<table>
<thead>
<tr>
<th></th>
<th>Cost TO Tool</th>
<th>Bell PC 1</th>
<th>Bell PC 2</th>
<th>Average</th>
<th>Quickstep</th>
</tr>
</thead>
<tbody>
<tr>
<td>SPAR</td>
<td>$37,197.08</td>
<td>$24,948.47</td>
<td>$31,072.77</td>
<td>$33,088.27</td>
<td></td>
</tr>
<tr>
<td></td>
<td>$18,396.37</td>
<td>$19,345.47</td>
<td>$18,870.92</td>
<td>$16,259.35</td>
<td></td>
</tr>
<tr>
<td>CORE</td>
<td>$6,044.43</td>
<td>$14,257.97</td>
<td>$10,151.20</td>
<td>$5,442.64</td>
<td></td>
</tr>
<tr>
<td></td>
<td>$4,039.43</td>
<td>$7,704.35</td>
<td>$5,871.89</td>
<td>$3,570.19</td>
<td></td>
</tr>
<tr>
<td>SKIN</td>
<td>$51,268.56</td>
<td>$46,640.71</td>
<td>$48,954.64</td>
<td>$45,312.94</td>
<td></td>
</tr>
<tr>
<td>TOTAL BLADE</td>
<td>$90,190.19</td>
<td>$74,833.48</td>
<td>$78,773.35</td>
<td>$81,265.67</td>
<td>$79,954.06</td>
</tr>
<tr>
<td></td>
<td>$94,510.07</td>
<td>$85,847.15</td>
<td>$88,292.82</td>
<td>$83,843.85</td>
<td></td>
</tr>
<tr>
<td></td>
<td>$71,389.48</td>
<td>$69,230.48</td>
<td>$73,091.47</td>
<td>$75,336.14</td>
<td>$65,142.49</td>
</tr>
<tr>
<td></td>
<td>$73,704.37</td>
<td>$73,890.53</td>
<td>$76,613.52</td>
<td>$83,125.15</td>
<td></td>
</tr>
</tbody>
</table>

A learning curve was then applied to this T1 Quickstep process for an average unit cost for 1,000 units of $41,125 (2001$), taking almost 380 man-hours per blade. This is comparable to the initial Bell PC estimate, despite the design rotorblade weighing over twice as much as the Bell PC estimate.
Learning Curve Applied to Quickstep Method

For Unit 1:
- 750 Man hours
- $63,125 / blade

For Unit 1,000:
- 327 Man hours
- $37,862 / blade

Average for 1-1,000:
- 376 Man hours
- $41,125 / blade

Figure 16.5: Quickstep Method

16.5. Helicopter Cost Model

16.5.1. Research, Development, Testing, and Evaluation (RDTE) Cost

RDTE cost for the Peregrine was calculated using the Bell PC Cost Model, with certain approximations or estimations fitted where the exact configuration of the non-conventional drive aircraft was unable to be specifically entered. Figure 16.6 depicts that engineering will be the most costly development factor, followed by manufacturing (three prototypes), tooling, general administration, ROM adjustments, logistics, and manufacturing engineering. This was calculated using three prototypes (with a majority of 50% new components by weight), one ground test vehicle, one static test article and one fatigue test article. Total RDTE Cost equals $231 million (2009$), almost twice the desired OEC value.

Figure 16.6: Development Cost

16.5.2. Recurring Production Cost
The recurring production cost for this design was determined using the average unit production cost for 100 production units, produced over 5 years (20 units per year). Cost for the Total Vehicle (defined as a military transport at 7,572 lbs) is:

- Recurring production cost = $6.18 million (2009$)
- Average unit cost with amortized non-recurring cost = $7.22 million (2009$)
- Unit “Price” including 12% profit = $8.08 million (2009$)

The systems which contribute most to this cost (a combined 88%) are the power plant, fuselage, flight controls, rotor, drive system, electrical, nacelles, and landing gear. Figure 16.7 displays the costs for these systems in relation to the baseline Superlynx estimation.

Key changes in the development inputs relate to the increase in rotor blades, upgraded engines, a primarily composite fuselage, and fly-by-wire addition. The pusher propeller was estimated as a nacelle tail rotor, expecting the weights of both systems to be similar for CG requirements. The non-conventional drive system impacts were primarily seen in the RDTE, not production.

16.5.3. Direct Operating Cost

The final part of LCC to be analyzed is the Operations and Support Cost (O&S), comprised of Direct Operating Cost (DOC) and Indirect Operating Cost (IOC). DOC refer to the resources immediately associated with the system or its operating unit, mainly fuel (and lubricants), airframe maintenance, and engine overhaul and maintenance. IOC involves a broader category of cost attributable to the system, including any cost that would not occur if the system did not exist. Examples of this are flight crew labor, insurance, and facilities. Depreciation of the unit is also considered as IOC. Figure 16.8 provides a comparison of O&S Costs between the baseline and new design. As a note, although Total Operating Cost was examined, only the Direct Operating Cost ($1,093/hr) is considered as part of the OEC.
17. Safety and Certification

17.1. Functional Analysis

The purpose of functional analysis is to “transform the functional, performance, interface, and other requirements that were identified through requirements analysis into a coherent description of system functions.” The mission profile was decomposed into flight segments. These segments were then further decomposed through the use of a functional flow block diagram (FFBD). This analysis tool defines task sequences and relationships – identifying functional interactions within the system. Figure 17.1 shows a three-level functional decomposition for the Peregrine. An autorotation phase was also included in the mission analysis to complete the study helicopter in all possible phases. For simplicity only the top two levels are shown in Figure 17.1 for the transmission and drive system; however, the actual analysis went to three levels.
17.2. Functional Hazard Assessment

The goal in conducting the Functional Hazard Assessment (FHA) is to clearly identify the circumstances and severity of each failure condition along with the rationale for its classification. This predictive technique attempts to explore the effects of functional failures on parts of a system, ensuring compliance with airworthiness requirements. The FHA is one of the preliminary activities in the safety assessment process outlined by ARP 4754.  Figure 17.2 lists the FHA for the entire aircraft and a separate FHA for the speed controller assembly in the transmission.

<table>
<thead>
<tr>
<th>Function</th>
<th>Failure Condition</th>
<th>Flight Regime</th>
<th>Action</th>
<th>Classification</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fire protection</td>
<td>Loss of fire protection</td>
<td>Start-up</td>
<td>Immediate shut-down abort mission</td>
<td>Hazardous – Extremely Remote</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All other flight modes</td>
<td>Land immediately</td>
<td>Catastrophic – Extremely Remote</td>
</tr>
<tr>
<td>Control power generation/regulated engine RPM (MV, rotor load)</td>
<td>Loss of power generation</td>
<td>Start-up</td>
<td>Abort mission</td>
<td>No effect</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All other flight modes</td>
<td>Autostabilitate restart if able</td>
<td>Major – Remote</td>
</tr>
<tr>
<td>Separate engine from transmission</td>
<td>Engine torque cannot be separated</td>
<td>Autostability</td>
<td>Unable to establish autostability</td>
<td>Hazardous – Remote</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All other flight modes</td>
<td>Autostability attempt to restart if able</td>
<td>Hazardous – Remote</td>
</tr>
<tr>
<td>Autoregulatory trim</td>
<td>Unable to control rate of descent</td>
<td>Autostability</td>
<td>Unable to establish autostability</td>
<td>Hazardous – Remote</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All other flight modes</td>
<td>Autostability attempt to restart if able</td>
<td>Hazardous – Remote</td>
</tr>
<tr>
<td>Reengage engine with transmission</td>
<td>Unable to reengage engine with transmission</td>
<td>Autostability</td>
<td>Establish autostabilized flight profile</td>
<td>Hazardous – Remote</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All other flight modes</td>
<td>Autostability attempt to restart if able</td>
<td>Hazardous – Remote</td>
</tr>
<tr>
<td>Communicate with ground</td>
<td>Voice communication and/or navigation system failure</td>
<td>All modes</td>
<td>Abort/return to ground as necessary</td>
<td>Minor – Remote</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All modes</td>
<td>Abort/return to ground as necessary</td>
<td>Minor – Remote</td>
</tr>
<tr>
<td>Provide spatial orientation</td>
<td>Instrumentation failure</td>
<td>All modes</td>
<td>Abort/return to ground as necessary</td>
<td>Minor – Remote</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All modes</td>
<td>Abort/return to ground as necessary</td>
<td>Minor – Remote</td>
</tr>
<tr>
<td>Loss of Thrust Control</td>
<td>Inability to perform high-speed flight</td>
<td>All modes</td>
<td>Abort/return to ground as necessary</td>
<td>Minor – Remote</td>
</tr>
<tr>
<td></td>
<td></td>
<td>All modes</td>
<td>Abort/return to ground as necessary</td>
<td>Minor – Remote</td>
</tr>
<tr>
<td>Produce electrical power</td>
<td>Loss of electrical power</td>
<td>All modes</td>
<td>Abort/return to ground</td>
<td>Catastrophic – Extremely Remote</td>
</tr>
</tbody>
</table>

Figure 17.2: Aircraft Level Functional Hazard Assessment
17.3. Certification

The Peregrine certification plan provides the necessary documentation of the Federal Aviation Administration (FAA) requirements to certify a new aircraft. The typical certification process involves five phases: conceptual design, requirements definition, compliance planning, implementation, and post certification. The Partnership for Safety Plan (PSP) is common throughout the entire process. The PSP represents an umbrella agreement between the applicant and the FAA aimed at establishing the standard operating procedures and expectations of the certification process. In the first three stages of the process, the applicant formulates a Project Specific Certification Plan (PSCP) that follows the guidelines of the PSP in order to address the unique certification characteristics of a particular new design. In the implementation and post certification phases, the required tests and evaluations are completed – providing the basis for continued airworthiness activities.\textsuperscript{72}

The development of a complete PSCP is beyond the scope of this project. An initial proposal for certification was created in Figure 17.4 to identify critical issues that may arise during the design and actual certification of the Peregrine. The main area of the design must comply with FAR Part 29. This is an example list that provides some of the major rotorcraft tests required for certification:\textsuperscript{73}

- FAR Section 29.1 – Applicability
- FAR Section 29.2 - Special retroactive requirements
- FAR Section 29.21 - Proof of compliance
- FAR Section 29.25 - Weight limits
- FAR Section 29.27 - Center of gravity limits
- FAR Section 29.143 - Controllability and maneuverability
- FAR Section 29.71 - Helicopter angle of glide: Category B
- FAR Section 29.87 - Height-velocity envelope
- FAR Section 29.301 – Loads
- FAR Section 29.601 – Design
- FAR Section 29.691 - Autorotation control mechanism
- FAR Part 29 Appendix A - Instructions for Continued Airworthiness
- FAR Part 29 Appendix B - Airworthiness Criteria for Helicopter Instrument Flight

Figure 17.3: Drive System Functional Hazard Assessment
18. Conclusion

The Peregrine represents an alternative to today’s standard drive helicopter. It is capable of high performance flight in terms of increased speed, range, payload, endurance, and noise signature as well as fuel consumption by minimizing drag throughout the helicopter. The focus of the student created design parameters was the high speed flight performance while committing to alternative systems in terms of the transmission, rotor and pusher propeller.

- At the heart of the aircraft, the variable speed transmission was the focus of the design project. The combination of the engine and transmission were the initial emphasis of the alternative drive train. The variable speed allows the rotor increase the advance ratio without creating the problematic characteristics of compressibility and increasing power requirements.

- The rotor configuration represents an alternative to the standard configuration of commercial helicopters. The coaxial rotor with a hingeless hub and Individual Blade Control with Higher Harmonic Control aides the aircraft in its pursuit of high speed performance while simultaneously reducing vibrations.

- The pusher propeller provides the necessary thrust to off load the main rotor in forward flight. This dramatically lowers power requirements of the main rotor and additionally reduces fuselage static pressure drag. These features of the pusher propeller are just as vital to the success of the design and represent a departure from standard configuration helicopters.

- In addition, the improvements of the Flight Control Architecture, improved engines, composite fuselage, low life cycle cost and upgraded avionics represent dramatic improvements to the baseline Agusta Westland Super Lynx 300.

Ultimately, the integrated approach of the concurrent product and process development translated into a unique alternative design by simultaneously increasing design quality and reducing anticipated design costs while maintaining focus on high performance flight. Every aspect of the Peregrine design process was intended to achieve the goal of a high performance alternative drive rotor system.
19. References

1. Introduction

1 Schrage, Daniel P., “Extension of RF Method to VTOL Aircraft Conceptual and Preliminary Design”, A6333 Rotorcraft Design I Course Notes, Georgia Institute of Technology, Fall Semester, 2008.


3 http://www.flightlab.com/

2. Vehicle Configuration and Selection Methods


3. Concept Selection, Sizing, and Performance

6 Schrage, Daniel P., “Georgia Tech Preliminary Design Program (GTPDP)”, AE6333 Rotorcraft Design I Course Notes, Georgia Institute of Technology, Fall Semester, 2008.


4. Conceptual Design

See Section #3

5. Transmission


6. Main Rotor Blade and Hub Design

7. Rotor Dynamics


8. Acoustics

9. Ducted Pusher Propeller


10. Engine Performance Requirements

35 CTS 800-5 Product Brochure http://www.t800ets.com/cts800_5.pdf


11. Full Authority Digital Engine Control (FADEC)

12. Structural Analysis


41 FAR Part 29 (14 CFR), Subpart C.


47 FAR Part 29 (14 CFR), Subpart C.


13. Fuselage Aerodynamics


14. FLIGHTLAB Model

15. Handling Qualities Improvement and Piloted Simulation


16. Cost Analysis

68 Designer’s Production Cost Trade-off Tool, App II-III. GaTech AE4803 course reference material.


17. Safety and Certification


73 http://www.flightsimaviation.com/data/FARS/part_29.html