Rapid Design and Virtual Testing of UAV
Within the DEE Framework

MASTER OF SCIENCE THESIS

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Abstract

Unmanned aerial vehicles are used in both civilian and military applications predominantly for aerial survey. In the current context aerial survey refers to search operations, monitoring of forest fires, geographic survey, disaster monitoring, traffic monitoring and border patrol. The major advantages in using unmanned aerial vehicles for commercial applications are lower cost (manufacturing/operational/maintenance), suitability for wide range of applications and quick response time. The UAVs are classified based on their size, endurance and range. The UAVs with a maximum take-off weight below 150 Kg are termed as small UAVs. Most electric powered UAVs belong to this category and offer ample choices for design to the designer.

Electric powered UAVs are gaining popularity with rapid improvements in battery technology. In traditional aircraft design, the major components are scaled based on a reference or an empirical estimation is used. It becomes essential to develop methodologies that can utilize the system engineering approach, where scaling of components is not done but a selection is performed from an available database. The trend in development of technology indicates the growth in areas of computation, high fidelity analysis techniques and tools, material technology, production and manufacturing processes. However, the issue related with weight estimation remains unsolved. The concept of ‘snow ball effect’ is described in many publications. The challenge in the design process is that existing conceptual and preliminary design frameworks depend on data collected from existing aircraft. A physics based framework is required for UAVs in order to make the design process independent of empirical data. This is essential since only limited empirical data is available for this class of aircraft and current UAVs are not necessarily the optimal configuration.

The focus of the current research is on the design of the fuselage and the selection of the on-board systems and propulsion system for electric powered UAVs. In general, the battery pack has the highest weight fraction. The estimation of the mass of the battery pack is considered based on empirical relations. A detailed approach is provided for the estimation of the weight of the battery pack. The optimal combination of components such as the propeller and motor is selected automatically with a routine from the database of available components, based on the overall mission requirements for the aircraft.

A physics based design framework is developed, which uses top level requirements as inputs. The fuselage shape design was also performed in order to display the capacity of the framework to integrate with a multi model generator and to allow seamless integration with aerodynamic analysis tools in the future. Furthermore, the new concept of multi-domain interaction was demonstrated in the framework. This concept enables the designer to identify possible unwanted interactions.
between components and to use this knowledge in the selection of the location. An example of multi-domain interaction is that a sensor cannot be placed too close to the battery due to the heat constraints. The multi domain interactions considered in the current framework are interactions in the spatial domain and in radiative heat transfer.
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Chapter 1

1. Introduction

1.1. Unmanned aerial vehicles

Unmanned aerial vehicles are used in both civilian and military applications predominantly for aerial survey. In the current context aerial survey refers to search operations, monitoring of forest fires, geographic survey, disaster monitoring, traffic monitoring and border patrol. The major advantages in using unmanned aerial vehicles for commercial applications are lower cost (manufacturing/operational/maintenance), suitability for wide range of applications and quick response time. The UAVs are classified based on their size, endurance and range. The UAVs with a maximum take-off weight below 150 Kg are termed as small UAVs. Most electric powered UAVs belong to this category. The enormous scope for design and development of new technologies is contributing to the rapid development of UAVs.

This category of aircraft is exempted from the certification specification [1] which reduces the regulatory requirements, provided the cruise velocity is below 70kts [2]. Hence ample choices for design are available to the designer. Due to the large number of design variables, the designer is forced to make many assumptions during the conceptual design phase. The deviation in the values between the assumed and the actual values contribute to the snow-ball effect, where the deviation in the value of the base variable contributes to the increase in value of the dependent variable.

Electric powered UAVs are gaining popularity with rapid improvements in battery technology. The existing aircraft design paradigm is based on using a combustion engine as power source. The existing design process is highly dependent on empirical data. The definition of aircraft states it as an object heavier than air capable of having sustained motion in air. The primary focus is suspension in air followed by sustained motion. Suspension in air is affected by mass and gravity. Therefore weight is the considered as the fundamental parameter. For the development of the design framework, if weight is fixed as the fundamental parameter, then high fidelity data for weight of the components at the earliest stage for the design process can simplify the process.
However, considering the developments of electric powered UAVs, systems can be identified apriori and therefore high fidelity data with respect to weight can be obtained. In traditional aircraft design, the major components are scaled based on a reference or empirical estimation is used. It becomes essential to develop methodologies that can utilize the system engineering approach, where scaling of components is not done but a selection is performed from database available.

The trend in development of technology indicates the growth in areas of computation, high fidelity analysis techniques and tools, material technology, production and manufacturing processes. However, the issue related with weight estimation remains unsolved. The concept of ‘snow ball effect’ is described in many publications, with reference to increase in weight due to changes effected at different stages of aircraft design. In order to eliminate the recursive nature of the problem, it is essential to fix the parameter ‘weight’. In this case, weight is the central parameter that must be updated at every instance. For the same reason, automation is highly desired in order to implement the design process. Another requirement is that it is essential to give the end user the flexibility to vary the configuration and to suit different mission profiles.

Considering the design of electric powered UAVs, the battery pack has the highest weight fraction [3]. The estimation of the mass of the battery pack is considered based on empirical relations. A detailed approach is required for the estimation of the weight of the battery pack. In aircraft design, the design of propulsion system is focussed on thrust and power requirements. The energy supplied by the energy source is considered proportional to the power required. When the energy is electricity and the storage medium is battery, the power delivered and energy delivered is inversely proportional. The relationship between the energy available and power delivered is given by the Ragone plot. Figure 1 shows the zones where the Ragone plot will be located for different energy storage devices. The lines marked 100s, 1s and 0.01s shows the discharge times. It could be observed that batteries have the highest energy density which makes them the most efficient energy storage devices.

![Ragone plot](image)

Figure 1: Ragone plot (energy available in storage devices for fixed power) [4]
When the effect of the Ragone plot is considered during the design phase based on the power required and the energy available (stored in battery), improved weight estimation can be performed. Therefore it becomes essential to consider thrust, power and energy for the design of electric powered UAV.

The advantages offered by electric powered UAVs [1] are 1) High reliability, 2) Easy to start (even in flight restart is possible) 3) Precise power management 4) Low noise/vibrations 5) Reduced heat emissions 6) Elimination of volatile fuels onboard.

The multi-domain interaction (MDI), concept introduced by Foeken [5] was developed for detecting interaction between system components in mechatronic design. The approach combines parametric CAD with object oriented programming and knowledge based engineering principles. As per Foeken[5], using modelling primitives representing system components and containing both geometric and behavioural information, automated reasoning taken into account component attributes such as geometric and material properties applied to derive spatial and energy interaction. Most CAD systems can detect the interference however the challenge lies in detecting the interaction in domains other that the spatial. Most CAD systems are difficult to programme or to incorporate user defined concepts. In case of a KBE system the ease of programming allows the designer to incorporate interactions in multiple domains. In the current work, the MDI concept is used to detect collision in physical domain using simple bounding box approach, thermal radiation problem was solved using analytical method using a novel KBE technique which allows detection of objects in the line of sight.

1.2. Design challenges
The various design challenges faced during the design of small UAVs are 1) lack of data 2) changes in technology 3) introduction of novel concepts.

Empirical data is not available for small UAVs, unlike the case of commercial aircraft. The data available for existing UAVs cannot be extrapolated as the design conditions vary too much in each case. Changes in technology refer to improvement in battery technology, motor technology and miniaturization of onboard systems. Due to the unavailability of empirical data, the empty weight fraction cannot be determined by extrapolation method.

The development of novel concepts for the unmanned aerial vehicle design and evaluation of the design at different phases of mission is crucial with technologies which we never tried before. During the design phase, changes in configuration and inclusion of different systems need to be facilitated. In case of design of small UAVs, various mission phase need to be considered as it is difficult to identify which mission phase imposes severe restriction on the design.

In design of aircraft powered by piston engines or gas turbine engines, the fuel fraction is determined by using the Breguet range equation. For design of electric powered UAVs, a modified range equation was introduced by Trips [6]. The capacity of the battery required can be determined using this method. The weight of the battery is determined from the plot between weight of the batteries and capacity. The contribution of the battery to the weight fraction is high and therefore an improved estimate of the weight of the battery is desired.
In traditional aircraft design, the design of propulsion unit is performed considering the engines that are available and scale them to meet the requirements. In case of small UAV design, engines are not scaled but selection is performed. The performance of the motor varies with respect to the manufacturer. A selection procedure is required rather than performing a selection based on the power output.

The challenge in the design process is that existing design frameworks depend on the data collected from other aircraft. A physics based framework is required in order to make the design process independent of empirical data. The geometry of the aircraft must be parameterized. Complex shapes must be represented and must be reproducible. Positioning of the components is desired in order to determine the physical properties of the aircraft. The detection of interaction between the different components at an earlier stage is desired in order to prevent design changes at a later stage.

The challenge involved in the development of a design framework is to maintain simplicity and to have scope for development. The need for using right fidelity analysis was stated by Wayne Johnson [7]. The increase in fidelity of data at every stage of the design would ensure that the design iteration would converge faster. Modifications in design must be facilitated and also facilitate integration of the design framework with other analysis tools. The automation of the creation of the geometry and the ability to vary the geometry during runtime are the key issues addressed in this project.

1.3. System engineering approach

A system engineering approach is adopted for design of UAVs [8]. The system engineering approach can be considered with respect to the product, process and programme. The system engineering approach is suitable for the development of the product, considering the UAV platform as the product. The propulsion system, avionics, payload and structural components can be considered as individual systems. The approach allows the designer to validate different configurations. The approach allows the designer to configure the UAV for different missions. The focus here is on the components and analyses are performed on the individual components. The component selection based on existing subsystems rather than scaled versions. A wider choice is available to the designer in selecting the systems and components. The development of the aerial platform is performed considering the technology which is currently available as the development time is shorter, unlike the case of commercial aircraft.

The system engineering approach is considered for the design process. The process is captured in a modular format and the modules can remain independent of each other. The analysis modules performing analysis with different levels of fidelity can be added and right fidelity analysis could be performed. Simple selection procedure can be replaced with optimization routine. The different modules can be easily replaced with analysis modules with high fidelity analysis. The SE approach allows the user to analyse different components individually as well as to detect the interaction.

System engineering approach is considered for the development of the environment. The focus is on using right fidelity analysis which can ensure balance in usage of computational resources. The advantages of development of such environment id describe in the work of Price, Raghunathan and
Curran [8]. The modular approach ensures that the different modules can interface between each other without any dependency.

Object oriented paradigm is used for the development of the framework. The approach distinguishes the data from the methods, which allows multiple methods to operate using the same data. The implementation of the system engineering approach is thus facilitated.

1.4. Design and engineering engine (DEE Concept)

The DEE concept introduced by La Rocca [9], is a modular computational design system to support distributed multidisciplinary design and optimization of aircraft. The concept supports capturing the design process and automation of the design process. The concept of the multi-model generator enables geometry to be captured in a neutral format and provides multiple views of the model for different analysis tools. The concept is aimed at incorporation of various analysis tools, the analysis tools can be in-house or commercial off the shelf (COTS).

The design of aircraft is performed as parametric design considering the conceptual and preliminary design phases. The DEE concept allows the aircraft design process which is essentially parametric, to be coupled with the geometry modelling.

In order to enable automation of the design process, the automation of geometry generation is desired. The generation of the geometry during runtime facilitates seamless interaction between the design process and the KBE application. Multidisciplinary design is captured in the DEE framework and the optimization is considered at the discipline level. One to one mapping of design parameters in DEE to the KBE application acting as the geometry kernel allows the development in the KBE application to be reflected in the parametric design process.

The incorporation of the analysis tools or performing analytical computation inside the KBE application reduces the number of stages in the design process as conversion between different formats for the geometries and input file formats conversion. Geometry creation during real time reduces the requirement to store or retrieve geometry. The parametric information is stored and retrieved using ASCII file format.

The DEE concept is envisaged for incorporating both high fidelity and low fidelity analysis tools. Data is stored in neutral file format which enables the DEE to provide views for the different analysis tool. The automation of repetitive tasks reduces the iteration time.

Although the fidelity of the analysis is considered as low with respect to the particular domain considered, the availability of the data in a design framework supporting optimization should be considered as high fidelity data for the conceptual and preliminary design process.

1.5. Objectives

The objective of the thesis work is the development of a physics based integrated design framework for electric powered UAVs in the below 150kg class. The aim is to facilitate rapid comparison of different configurations. Changes in the design process needs to be implemented.
The objective of the design framework is to determine the optimal system based on a set of predetermined criteria. The mission performance is computed from the mission analysis and the energy and power required for different mission segments are determined. The motors considered for the design were selected from the database of motors which are available. The batteries considered for the design are to be selected from the commercially available batteries. The framework can to allow select commercially available propellers.

The next objective is the estimation of the power-loading and wing-loading. The framework must have flexibility in incorporating analysis tools and other modules. The objective is to estimate the mission thrust, power and energy requirements. The mission performance needs to be computed from the modified Breguet range equation.

The next objective is to provide user interface for design of the fuselage shape. Fuselage shape design is required for housing the components. Representation of fuselage in terms of KBE application and in a parametric form needs to be emulated. Another requirement is detection of interaction between different systems in physical domain, detection of presence of objects in line of sight and radiative heat transfer.

1.6. Deliverables

The deliverables for the thesis work are physics based design framework, capable of integrating with the MMG of the DEE. The design framework is built based on the OOP concept. The OOP concept is convenient in representing the data and functions together, called as an object which is instantiated from a class. This enables the design framework to have one to one mapping between the physical system and the modules in the framework itself.

The framework contains fuselage shape representation module in the KBE platform with capability to interact with the design framework. Graphical user interface for depiction of fuselage shape representation in different analysis tools. Modelling of components as wireframe representation including representation of physical properties, with the capability to instantiate components during runtime.

KBE based routine for detecting system interaction, the domains considered for the interaction are based on the concept proposed by Foeken [5]. The routine capable of detecting direct interaction based on geometric interference. Detection of radiative heat transfer via radiation using analytical method. To detect the presence of the obstacles in the line of sight. The content of this report are as described below.

Chapter 2 describes the design framework and its functionalities. Chapter 3 describes the design case, mission considered, system identification and cloud point representation of the fuselage. Chapter 4 describes the preliminary sizing where linearized equations are using for programming. Chapter 5 describes the propulsion system selection based on the mission segments considered. Commercially available components are considered. Chapter 6 describes the concept of multi domain interaction by considering interaction in spatial and thermal domains. Parametric fuselage shape is depicted using class shape transformation. Chapter 7 describes the results and discussions.
2. Design framework

2.1. Introduction

In this chapter, the design framework adopted for the proposed research and the general design and engineering engine design [9] framework are presented. The different modules which are present in the design framework, their functionality and inputs/outputs of the design framework are explained in this unit.

In order to design an aircraft, the designer is dependent on empirical data. The empirical data is obtained from statistical compilation of parameters from existing aircraft. However, in case of UAV design, empirical data is only available to a limited extent and due to rapid advancements in the technology related to UAVs, the data collected can quickly become outdated. Also, the available data need not correspond to that of an optimal design. The current practice is to extrapolate existing data which were obtained for the design of manned aircraft. The data available is noisy and scattered. This means that the data need not accurately represent the real aircraft and the values might not be coherent. The challenge lies in the determining the initial parameters for the design and very often, the selections are far from the optimum. Therefore, there is a need to have an intelligent physics based design framework which can help the designer in making right decisions and which can provide considerable saving in resources.

The scope will be to develop a physics-based framework capable of performing preliminary sizing. The objective is also to determine the optimal combination of systems [e.g., motor, battery, and propeller] for a small EPUAV configured for different requirements. It must also determine the effect of different configuration for the systems considering interaction in multiple domains using knowledge-based engineering approach. Also, to design the shape of the fuselage using a methodology which can provide intricate shapes and parameterized with the least number of
control parameters which could be used for shape optimization in future. The scope is also to determine the position of components and their effect on the flight performance.

The different modules in the DEE are as below:
1. requirement specifications
2. initiator
3. multi-model generator
4. disciplinary analysis tools
5. converger and evaluator

The modules in the current design framework are a preliminary sizing module, a propulsion system selection module, a configuration module and a fuselage shape design module. The framework was developed in Matlab® and GDL was used for the development of MMG.

A simple surveillance mission profile is considered for the system selection for the design of the test case unmanned aerial vehicle. ‘UAV Challenge 2010 – search and rescue challenge’[10] is considered for the surveillance mission. The UAV needs to fly at an altitude of 200 m and detect the heat signature produced by the lamp placed near the human dummy.

The framework enables the designer to make decisions and to perform quantitative assessment of the vehicle. The architecture of the framework is depicted in figure below.

The figure 1 shows the architecture of the design framework for the multi-domain interaction. The framework obtains the inputs directly from the user. Database created using the data from the web based catalogue are also used to provide the input. The output from the design framework can be provided to the flight mechanics model. The multi-model generator is developed which can provide multiple views for the multi domain interaction module. The multi-domain interaction module detects the interference of components in 3-D space. The module also determines the temperature of the participating surfaces using radiative heat transfer calculation.

The fuselage shape is modelled here. The fuselage shape design module can be further enhanced to estimate the skin friction drag. The shape could also be used for aerodynamic optimization as the shape is developed in the parametric form. The components include the propulsion system components as well for the multi-domain interaction.
2.2. Functions

The function of the design framework is to explore the design space by performing right fidelity analysis. Since the framework deals at conceptual and preliminary design high fidelity data is not available. Using high fidelity analysis would prevent the designer from exploring the design space. Analytical tools capable of performing analysis at various fidelity levels are required to be integrated into same framework. Therefore it becomes essential to perform right fidelity analysis. The function of the MMG was to act as a CAD tool. GDL is used to develop multi-model generator (MMG) which performs the function of the CAD tool.

The function of the design framework is to perform parametric design. The first function of the initial sizing module is to perform weight estimation, wing loading-power loading and drag estimation. Development of the drag estimation module might be an extension of the current research work. The function of the propulsion design module is to perform the mission analysis for different mission segments and to determine the thrust and power required for each segment. The second function is to determine the energy requirement for the different mission profiles. The function of the propeller sub-module is to analyse a propeller defined by the user or to use available experimental data. The
function of the propulsion module is to determine the propeller and motor combination. The other function is to determine the number of batteries required for the propulsion system design.

The function of the fuselage shape design module is to provide interface for the definition of the fuselage shape parameters, representation of the fuselage in the KBE application. The function of the multi domain interaction module is to determine the interference in the physical domain, determine the presence of the obstacle in line of sight and to determine the radiation heat transfer. The weight, centre of gravity and moment of inertia are to be determined by the configuration module. The availability of such parameters during the initial design phase is required for flight mechanics analysis.

### 2.3. Requirements

The requirements can be classified as requirements with respect to the product and requirements with respect to the process. The requirements for the propulsion system selection process are databases of commercially available motors, propellers and batteries. The geometric parameters of the components are to be represented using the multi-model generator. The requirement on the process is seamless integration between the KBE application and the design framework.

The framework must operate with right fidelity data. Fidelity refers to how accurately the mathematical model can depict the physical model [7]. In most implementations of DEE, the focus was primarily to have a high fidelity data. Here the requirement is on right fidelity data and analysis. The idea is to have a design framework for exploring different concepts, experimenting with different configurations. The framework must provide a basis for performing design optimizations later on. However design optimization itself is not the area of focus for the current research work. The framework must support multidisciplinary design, analysis and optimization. In such a case the information provided to different disciplines must be with right fidelity.

The different modules must be capable of communicating with other modules present in different platforms and coded with different programming languages. The information passed between different modules must be managed and there should not be any data loss. The information must be coherent and must be safe.

### 2.4. Inputs

The development is considered as a standalone framework. This means that the inputs are not obtained from other programs. The inputs are provided by the designer. A graphical user interface is provided for the designer to interact with the design framework.

The database for the propellers is obtained from the web catalogue [11]. Web catalogue refers to an online collection of data pertaining to a specific system, such as the motor, battery, propeller etc. The static thrust curves and power curves are provided in the web catalogue. The thrust and power curves for different velocities are not available in the framework.

The geometric information of the components considered for modelling can be obtained from the web catalogue. The geometric information is programmed using GDL to instantiate the components.
Design framework

in the multi model generator. The initial positions of different components are to be provided as inputs. The mass of the components are to be provided. The orientation of the components must be supplied.

The frames required for the fuselage lofting are provided by the user utilizing a graphical user input window. The shape can be modified from within the design framework latter on, in case complete automation is desired. The frames are generated using class shape transformation. The location and position of the frames need to be specified. Here location refers to the orientation in the lateral direction and position refers to the placement in the longitudinal direction.

2.5. Design variables

Some of the design variables are described in the table given below

<table>
<thead>
<tr>
<th>Component</th>
<th>Variable</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>UAV</td>
<td>W</td>
<td>Maximum take-off weight</td>
</tr>
<tr>
<td></td>
<td>( V_s )</td>
<td>Stall velocity</td>
</tr>
<tr>
<td></td>
<td>( C_{t_{\text{max}}} )</td>
<td>Maximum lift coefficient</td>
</tr>
<tr>
<td>Battery</td>
<td>( n_{p_s} )</td>
<td>Number of cells in series</td>
</tr>
<tr>
<td></td>
<td>( n_{p_p} )</td>
<td>Number of cells in parallel</td>
</tr>
<tr>
<td></td>
<td>( C_{h} )</td>
<td>Discharge rating</td>
</tr>
<tr>
<td></td>
<td>( L_{b} )</td>
<td>Length - battery</td>
</tr>
<tr>
<td></td>
<td>( B_{b} )</td>
<td>Breadth - battery</td>
</tr>
<tr>
<td></td>
<td>( H_{b} )</td>
<td>Height - battery</td>
</tr>
<tr>
<td></td>
<td>( W_{b} )</td>
<td>Weight - battery</td>
</tr>
<tr>
<td>Propeller</td>
<td>( D_{p} )</td>
<td>Propeller diameter</td>
</tr>
<tr>
<td></td>
<td>( p_{p} )</td>
<td>Propeller pitch</td>
</tr>
<tr>
<td>Component</td>
<td>( x_{cg} )</td>
<td>CG Position in x</td>
</tr>
<tr>
<td></td>
<td>( y_{cg} )</td>
<td>CG Position in y</td>
</tr>
<tr>
<td></td>
<td>( z_{cg} )</td>
<td>CG Position in z</td>
</tr>
<tr>
<td>Motor</td>
<td>( W_{m} )</td>
<td>Weight of motor</td>
</tr>
<tr>
<td></td>
<td>( H_{m} )</td>
<td>Height of motor</td>
</tr>
<tr>
<td></td>
<td>( D_{m} )</td>
<td>Diameter of motor</td>
</tr>
</tbody>
</table>

Table 1: Design variables for the design framework

2.6. Constraints

Computations for mission analysis are performed using linearized equations. The scope of the design framework is to determine single point design. The number of design variables is large and hence it is feasible to perform a single point design.

The propellers and motors considered are commercially available components for aero model building. The unmanned aerial vehicle considered must have maximum takeoff weight below 150 Kg. The constraint arises from the fact that this category is exempt from certification specification.

Time based simulations are not considered. The design framework focuses on the conceptual and preliminary design. The framework works independent to that of flight mechanics module.
2.7. Outputs
Motor, battery, propeller selected for performing the simple surveillance mission. These components are selected from the web based catalogue. The selection was performed on a reduced set of data, only to depict the working of the design framework.

Centre of gravity and inertia for the configuration module suitable for providing it to the flight mechanics model. The components can be selected by the user during the run time. The motor, battery and propeller are also considered in this computation. It is worthwhile to note that the battery contributes to about 50% - 60% of the maximum take-off weight of the unmanned aerial vehicle under consideration for design.

Fuselage shape design is obtained from the fuselage shape design module. The parametric definitions of the trunks and the adapters are obtained from the module. About ten frame definitions are available as predefined sections. The module allows the user to define custom frames for the trunk/adapter definitions.

Interaction between components in spatial and energy domains and the collision between the components are determined based on the bounding box approach. The surface temperatures are determined based on radiative heat transfer.

2.8. Design and engineering engine (DEE)
The DEE framework is shown in figure 4. The main program is executed in a programming language of designer’s choice. The multi-model generator is essentially a graphics kernel supporting object oriented programming concept and the geometry is stored and retrieved in a neutral file format capable of providing multiple views for different analysis in various disciplines.

Although the fidelity of the analysis is considered as low with respect to the particular domain considered, the availability of the data in a design framework supporting optimization should be considered as high fidelity data for the design process.

The initial development of DEE used ICAD for the development of multi-model generator. Current versions use GDL for the development of multi-model generator. The MMG must be efficient and simple. The MMG must be capable of providing low fidelity analysis and high fidelity views. In other words, MMG must provide right fidelity data to the disciplinary analysis.

It is important to have seamless transition between various stages. The different stages are developed in different platforms. Efficient methods are required at every step in order to reduce the computational effort. Computational effort refers to the amount of memory utilized, CPU time required, number of platforms/software utilized etc.

The seamless transition helps in performing design iterations without human intervention. Only when the program is void of human intervention the design framework could be automated. Right fidelity analysis being performed with right fidelity data ensures that program run time per iteration is optimal.
High level primitives (HLP) are sections which could be considered as building blocks for complex and complete configuration of aircraft components [12]. High level primitives could be defined as an object. A HLP completely defines the system component completely inheriting all major properties.
Report writers block post process the results from the different analysis tools. The report writers depend on the nature of the analysis tool. Knowledge about the different type of analysis to be performed can greatly reduce the effort spent on the development of the MMG.

2.9. System engineering approach
A system engineering approach is followed for the development of the design framework. The system engineering approach allows the user to focus on individual modules rather than focusing on the complete design process. The system engineering approach is better suited for the design process as low fidelity analysis tools and high fidelity analysis tools can be used within the design framework.

The focus on using right fidelity analysis is stated by Johnson [7]. The usage of right fidelity analysis tools improves the fidelity of the design framework as a whole.

The suitability of system engineering approach for the design framework can be discussed based on two ways.

1. With respect to process
2. With respect to product

2.9.1. With respect to process
The aircraft design process implemented in SE approach allows the designer to modify an individual analysis module without affecting the rest. The SE approach also allows the framework to switch between modules during the program execution. In this context ‘switch’ refers to changing execution of one module to another.

2.9.2. With respect to product
In traditional aircraft design, the individual systems are not considered but mass of the systems are only considered. The mass of the components are not obtained directly but scaled based on the reference aircraft.

The approach is particularly suitable for UAV design as the components can be selected directly from the database. The systems/components are commercially available, thus the properties of the systems are known a priori. It must be noted that the scaling of the components might not be appropriate as the properties can vary substantially between different vendors. The approach also provides choice to the designer.

The components are selected based on existing subsystems. Due to the availability of large number of components, a wide choice is available for the designer. The technology is mature unlike the design of commercial aircraft where the components are developed only after the design of the aircraft is conceptualized.

The technology changes rapidly and the designer must be able to incorporate new systems and subsystems.
2.10. **Object oriented design paradigm**

The design process is captured using object oriented analysis and design (OOAD) model. The capturing of the model in OOAD allows the designer to capture the knowledge and convert it to the design framework. The ability to switch from high fidelity to low fidelity analysis is facilitated by the framework. The approach provides flexibility to the framework as the data and the methods are decoupled. The object oriented approach facilitates the framework development as new tools can be incorporated. The changes in one module are restricted to the particular module and the modules are decoupled.

The implementation of the framework using the object oriented paradigm reduces the complexity in integrating new modules and analytical tools.

The figure below depicts the class diagram for mission analysis. The Mission class is a composition of all other mission segments. The mission segments representing classes are Takeoff, Climb, Cruise, Maneuuvre, Loiter and Landing. The class condition stores the atmospheric condition at the particular altitude.

![Class diagram for mission analysis](image)

The figure below depicts the class diagram for the propulsion system selection. The class diagram shows the composition of the Propulsion_unit class. This class consists of instances of all other classes such as the Motor class, Propeller1 class and Battery_pack class. The Motor class consists of
frames stored in the database. Frame refers to a set of values for an instance of the class. The propeller can be of user defined type or commercially available. For the test case under consideration only the commercially available propellers are considered. Battery_pack is an aggregation of the Battery class. The Battery class consists of multiple frames made available from the web based catalogue.

Figure 6: Class Diagram for propulsion system selection
3. Design Case

3.1. Introduction
In order to design an aircraft, the type of mission it has to perform must be elicited. In this case the requirements are based on the requirement of an unmanned aerial vehicle to take up ‘Outback challenge’[10] is considered. The mission considered for the UAV design is based on the ‘Search and Rescue Challenge’ [10]. First, the mission considered for the design case is described. The second part describes the baseline model considered for the design case. A commercially available model aircraft is used as a baseline model. The next part describes the systems identified for the mission considered. The subsequent part describes the cloud point model used for describing the fuselage shape. Finally the assumptions for the design case are described.

3.2. Mission definition
The test case was designed based on the rules for the 'UAV Challenge 2010 - search and rescue challenge'. The objective was to search for an outback traveller in a designated search area and to provide emergency medical supplies. The medical supply consists of a 50 ml (liquid pack) which has a weight 50 g. The objective of the development of framework is to demonstrate physics based design environment capable of detecting multi domain interaction between various systems. The framework is developed considering single engine aircraft and the database contains model aircraft engines with power up to 1 kW. Therefore it is essential to consider reduced maximum take-off weight, so that the payload also suits the dimensions of the baseline aircraft.

The UAV must be able to identify the location of a person in a designated area. The system should locate the person by flying in the defined corridor of approximately 1 nautical mile (1.852 km) in length and 0.2 nautical miles (0.3704 km) in width. The mission corridor is located within a mission boundary of 2 x 3 nautical miles. The GPS coordinates for the search area and the mission boundary are specified. A dummy is placed in the search area and a heat signature is generated by a lamp. The payload for drop is considered as 50 g medication with dimensions 50mm*40mm*30mm (assumed). The total flight time is considered to be 60 minutes. The UAV must be capable of autonomous flight with the possibility to operate as radio controlled model.
Assuming search area can be traversed in 2 passes (1 forward and 1 return), the required range for the aircraft is 3.704 km. The cruising altitude is designated as 200m. The search area is shown in appendix C.

![Figure 7: Mission definition for design case](image)

<table>
<thead>
<tr>
<th>Segments</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>1-2-3</td>
<td>Take-off</td>
</tr>
<tr>
<td>3-4</td>
<td>Climb</td>
</tr>
<tr>
<td>4-5</td>
<td>Cruise/loiter</td>
</tr>
<tr>
<td>5-6</td>
<td>Descent</td>
</tr>
<tr>
<td>6-7-8</td>
<td>Landing</td>
</tr>
</tbody>
</table>

The figure above shows the mission profile. The mission segment 4-5 is of interest here. The range and the required endurance are mentioned above.

### 3.3. Baseline UAV model

The Multiplex Ezstar model is considered as baseline model for the design. The UAV is made of foam however only the outer mould shape is considered for the fuselage shape representation. Only the fuselage shape is considered and wing / tail plane are not considered. The reference model can be purchased off the shelf.

![Figure 8: Multiplex Ezstar](image)

<table>
<thead>
<tr>
<th>Specification</th>
<th>Details</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wingspan</td>
<td>54&quot; (1370mm)</td>
</tr>
<tr>
<td>Wing Area</td>
<td>372 sq in (24 sq dm)</td>
</tr>
<tr>
<td>Weight</td>
<td>24oz. (680 g)</td>
</tr>
</tbody>
</table>
### 3.3.1. Aircraft systems components

Based on the mission to be performed, the systems are identified. The UAV needs to acquire and transmit live video. It must also be capable of acquiring thermal images. The thermal imaging could be still photograph or video but it must be transmitted in real time. The systems identified are presented in appendix B. The system/components that are required are: GPS sensor, IR sensor, radio (Command link 900 MHz), video camera / video transmitter (video link 5.8 GHz) and radio control (2.4 GHz).

The release mechanism for the payload drop is not considered and is beyond the scope of this work. The payload considered is considered as 50g as specified in the rules in order to match the capacity of motors and the characteristics of the propellers available in the database.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Specification</th>
</tr>
</thead>
<tbody>
<tr>
<td>Length</td>
<td>34” (870mm)</td>
</tr>
<tr>
<td>Airfoil</td>
<td>Flat bottom, high wing</td>
</tr>
<tr>
<td>Center of Gravity</td>
<td>3.07” (78mm) Back from the wing’s leading edge at the fuselage sides.</td>
</tr>
<tr>
<td>Control Throws- Elevator</td>
<td>Up &amp; Down 3/16” (5mm), Rudder: Right &amp; Left 3/8” (10mm)</td>
</tr>
</tbody>
</table>

Table 3: Major dimensions of EzStar

### 3.3.2. Fuselage cloud point representation

Multiplex EzStar representation as a cloud point model is considered for extracting the fuselage shape. The cloud point model provides the external mould shape of the aircraft under consideration. The cloud point model was originally obtained from a 3-D scanner. The current model is used for representing the aircraft in a graphical format in a flight simulator [13]. Although the
model represents the entire aircraft model, only fuselage is of relevance for the development of the design framework being described.

3.4 Assumptions

Framework stores the details considering it as a single point design. Only the outer mould shape is considered for modelling. The mass and dimensions of the payload is adjusted to suit the baseline model. Wireframe model is represented as a light weight representation of the component. Light weight refers to minimal consumption of resources for computation and display. Only the major dimensions are represented for the design of components. Only a single mission profile was considered for the design case. Rectangular surfaces are considered for the radiative heat transfer problem.

Propeller chart is considered with only static thrust case. Complete propeller chart is unavailable. Three trunks and two adapters are the high level primitives used for defining the fuselage model.
4. Preliminary sizing

4.1. Introduction

The set of systems onboard the aircraft is identified based on the mission requirements as described in previous chapter. The component and its properties are stored as specific instances in the programme. The framework design is based on KBS (knowledge based system), however for storing and retrieving the components FBS (frame based system) is preferred. Here frames are used to define specific instances of the given class. New components are added based on user requirement or based on input during design iteration.

The systems/components are available as predefined models and can also be instantiated during the runtime of the design framework.

4.2. Initial weight estimation

The system identification is described in the previous chapter.

\[ W = W_{st} + W_{sys} + W_p + W_{pu} \]  

The empty weight fraction is traditionally obtained by interpolating with respect to the existing aircraft. The empty weight fraction is available only up to a limited extent for UAVs since they are configured for different mission requirements. Furthermore, the available data need not represent an optimised configuration as in the case of transport aircraft. In equation 1, the maximum take-off weight is the sum of structural weight, system weight, payload weight and propulsion unit weight.

The weight fraction contribution from the propulsion unit is considered. In traditional aircraft design, the engine could be scaled to meet the power/thrust requirements. In the case of electric powered aircraft, the motor can be chosen from commercially available motors. The system engineering approach is suitable for the design process as accurate data about the product is available.

The weight of the battery constitutes for 50% to 60% [3] of the maximum take-off weight of an UAV. It can be observed that the battery weight constitutes for a greater fraction of the weight and therefore it requires greater attention while designing.
The weight of the UAV is assumed to remain constant throughout the mission process. The source of the energy is considered as lithium polymer (LiPo) battery. The reason for choosing LiPo battery is that they have high energy density when compared with other battery types. A brief comparison is given in chapter 5. The maximum take-off weight is often specified as user requirement in the case of the design of unmanned aerial vehicle.

The relationship is provided by Mueller[3], the relationship was specified is for the MAV. For the EPUAV, (upto 150 Kg MTOW) the empirical relationship is defined as following:

\[ E \propto \frac{W_b/W_1}{(1 + W_b/W_1)^{3/2}} \]

The relationship can be tabulated as shown below.

<table>
<thead>
<tr>
<th>( W_b/W_1 )</th>
<th>0.25</th>
<th>0.5</th>
<th>1.0</th>
<th>1.5</th>
</tr>
</thead>
<tbody>
<tr>
<td>Endurance</td>
<td>50%</td>
<td>71%</td>
<td>92%</td>
<td>max</td>
</tr>
</tbody>
</table>

Table 4: Weight fraction

For the design of UAV, \( W_{t_o} \) is provided as a constraint unlike conventional aircraft design where \( W_{t_o} \) is to be determined and the weight of the payload is specified.

The linearized equations depicting wing loading – power loading relationship are formulated as minimax problem. The \texttt{fminimax} algorithm in Matlab\textsuperscript{*} minimizes the worst-case (largest) value of a set of multivariable functions, starting at an initial estimate.

\[
\min \max_{x \ i} F_i(x) \text{ such that } \begin{cases} 
  c(x) & \leq 0 \\
  ceq(x) & = 0 \\
  A \cdot x & \leq b \\
  Aeq \cdot x = beq \\
  lb & \leq x \leq ub
\end{cases}
\]
Where $x, b, b_{eq}, lb$ and $ub$ are vectors. $A$ and $A_{eq}$ are matrices and $c(x), ceq(x)$ and $F(X)$ are functions that return vectors. $F(X), c(x)$ and $ceq(x)$ are nonlinear functions [14].

For the current work, the weight of the motor is considered separately from the empty weight as the aim is to select the motor from commercially available motors and to have minimum weight for the propulsion system. In traditional aircraft design, a combustion engine is selected based on the power requirement and the engine is scaled up or scaled down. In our case, it is not possible to scale the DC motor but due to the availability in wide ranges, a direct selection of a suitable motor will be performed.

The empty weight will be considered which is the actual empty weight minus the weight of the motor. The weight of the UAV remains constant as batteries are the source of energy and no liquid fuel is carried on-board.

4.3. Wing loading – power loading

The wing loading – power loading diagram is formulated as multi-objective optimization problem. For the current problem the, the linearized equations are used as the focus is on the design rather than on performance analysis.

Sizing for stall speed [15], [16] the wing loading can be expressed as:

$$\frac{W}{S} = \frac{1}{2} \rho V^2 \frac{C_{L_{max clean}}}{\rho}$$

Sizing for take-off is described in subsequent sections:

The take-off parameter (TOP) cannot be defined due to lack of data for conventional take-off of UAVs in the desired class. Takeoff parameter is an empirical value obtained by relating the wing loading and power loading of existing aircrafts belonging to same category. The requirement on clearing obstacle is considered with obstacle height ($h_{ob}$) considered as 50ft (Considering along with military aircraft).

Considering a conventional take-off the forces acting on the aircraft can be formulated as described below. This represents the equation of motion.

$$m \frac{dV_{so}}{dx} = T - D - \mu_r (W - L)$$

$$C_D = C_{D,0} + \Delta C_{D,0} + (k_1 + Gk_3)C_L^2$$

Where $C_{D,0}$ is the zero-lift drag coefficient, $\Delta C_{D,0}$ is the increment in the zero-lift wave drag and zero lift parasite drag coefficients, $(k_1 + Gk_3)$ contains the effect of wave drag due to lift [15]. G considers the ground effect.

Combining the equations given above, the acceleration of the UAV during ground run becomes.

$$\frac{dV_{so}}{dx} = g \left( \frac{T}{W} - \mu_r - \frac{\rho_{so}}{2(W/S)} \left[ C_{D,0} + \Delta C_{D,0} + \left( k_1 + \frac{G}{\text{REAR}} \right) C_L^2 - \mu_r C_L \right] V^2_{so} \right)$$

At this stage no approximations were made. The take-off distance is equal to the sum of ground roll distance and airborne distance, which is denoted as.
The ground roll distance is determined by considering the equations of motion for ground roll and is approximated to be:

\[ s_g \approx \frac{1.21 \left( \frac{W}{S} \right)}{g \rho_\infty C_{l_{\text{max}}}} \left( \frac{T}{W} \right) \]

The equation gives the relationship between wing loading and thrust loading. The radius for flare is determined to be:

\[ R = \frac{6.96 (V_{\text{stall}}^2)}{g} \]

Here \( n \) is considered to be 1.19 and \( V_\infty = 1.15 V_{\text{stall}} \).

The angle subtended by the obstacle at the position where rotation is performed is determined to be:

\[ \theta_{OB} = \cos^{-1} \left( 1 - \frac{h_{OB}}{R} \right) \]

The airborne distance to clear the obstacle height is given by:

\[ s_a = R \sin \theta_{OB} \]

The power loading during take-off is formulated as given below:

\[ W/P = \left[ s_{to} - \frac{6.96 V_s^2}{g} \sin^{-1} \left( 1 - \frac{h_{ob} \rho C_{l_{\text{max}}}}{6.96 V_s^2} \right) \right] \frac{g \rho_\infty C_{l_{\text{max}}}}{1.21 (W/S)(1.1 V_s)} \]
For catapulted take-off, the $V_{lo} = 1.1V_s$ \cite{16}, $V_{lo} = V_{end} + \Delta V_{thrust}$ (The catapult is assumed to be ground based and the wind speed is assumed to be zero.) $V_{end}$ is the velocity when leaving the catapult. $\Delta V_{thrust}$ is the velocity added due to the thrust generated by the propeller.

Now, sizing for landing is considered. The distance required in the case of conventional landing is considered as given below.

\[ s_l = s_a + s_f + s_g \]

\[ V_{\infty} = V_{TD} \]

\[ V_{\infty} = V_0 \]

\[ V_{\infty} = V_{TD} \]

\[ V_{\infty} = 0 \]

\[ R \]

\[ \theta_f \]

\[ V_{\infty} \]

\[ s_a \]

\[ s_f \]

\[ s_{fr} \]

\[ s_g \]

\[ \theta_a \]

\[ h_f \]

\[ 50 \text{ ft} \]

\[ \text{Approach distance} \]

\[ \text{Flare distance} \]

\[ \text{Ground roll} \]

\[ \text{Total landing distance} \]

\[ \text{Figure 13: Landing distance}[15] \]

The distance for landing is given as sum of approach distance after clearing obstacle height, flare distance and ground roll distance. The equations below give the formulation for the distances mentioned.

\[ s_a = \frac{15.24 - h_f}{\tan \theta_a} \]

The approach angle is given by $\theta_a$, the height at which the UAV starts to flare is given as $h_f$.

\[ s_f = R \sin \theta_a \]

\[ R = \frac{V_f^2}{0.2g} \]

\[ h_f = R (1 - \cos \theta_a) \]

\[ \theta_a = 2.5^0 - 3^0 \]
The most common value for the approach angle is given above.

\[ s_a = \frac{h_{ob} - \frac{1.2V^2}{0.2g} (1 - \cos \theta_a)}{\tan \theta_a} \]

\[ s_g = jN \sqrt{\frac{2W}{\rho_\infty S (C_L)_{max} L}} + \frac{j^2 W}{g \rho_\infty (C_L)_{max} [D/W + \mu_r (1 - \frac{L}{W})]} \]

The above equation is approximated to:

\[ s_g = jN \sqrt{\frac{2W}{\rho_\infty S (C_L)_{max} L}} + \frac{j^2 W}{g \rho_\infty (C_L)_{max} \mu_r} \]

The values for some variables are described below. \( N=3s; \mu = 0.4; j = 1.1 \); considering the UAV to be in the category of military aircraft. \( N \) is the time increment for free roll immediately after touch down, before the brakes are applied.

Now considering the sizing for climb, (drag polar estimate)

The climb rate and climb gradient needs to be specified. The UAV considered is below 150Kg category which is far below the requirement of less than 2721Kg. The climb gradient must be at least 8.3% \((c/V)\) considering the UAV to be land based small aircraft.

\[ W/P = \frac{\eta_P}{\sqrt{W/S}} \left[ \frac{2}{\sqrt{\rho}} + \frac{C_{L}^{3/2}}{C_D} \right] \]

From the aircraft performance theory, the relationship between the wing loading and power loading is obtained.

\[ \frac{C_{L}^{3/2}}{C_D} = 1.345 \left( \frac{(Ae)^{3/4}}{C_{D,0}^{1.4}} \right) \]

Here the climb rate is considered for the formulation.

\[ \frac{W}{P} = \frac{\eta_P}{\sqrt{W/S}} \left[ \frac{2}{\rho(C_{L,\text{max, clean}} - 0.2)} \left( \frac{c}{V} + \frac{C_D}{(C_{L,\text{max, clean}} - 0.2)} \right) \right] \]

Now sizing for manoeuvres is considered.

\[ \frac{W}{P} = \frac{\eta_P}{0.5C_D \rho V^3 + \frac{n_{\text{max}}^2 W}{W/S + 0.5\pi Ae \rho V}} \]
Initial drag polar estimation is performed as given below:

\[ C_D = C_{D,0} + \Delta C_{D,0} + (k_1 + Gk_3)C_L^2 \]

\[ G = \frac{16h}{b^2} \]

G considers the ground effect (in ground effect drag co-efficient/ out –of-ground effect drag coefficient)[15].

\[ \Delta C_{D,0} = \frac{W}{S}K_{wet}m^{-0.215} \]

The wetted surface area and the f are to be determined, where \( f = S/C_{frole} \) (skin friction coefficient)

\[ C_{D,0} = \frac{S_{wet}}{f} = 0.02 - 0.04 \]

The empirical value is provided considering the aircraft to be in the homebuilt category with fixed landing gear.

The value of the \( \Delta C_{D,0} \) is given in the table below:

<table>
<thead>
<tr>
<th></th>
<th>min</th>
<th>max</th>
</tr>
</thead>
<tbody>
<tr>
<td>Clean</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>T-O</td>
<td>0.01</td>
<td>0.02</td>
</tr>
<tr>
<td>Landing</td>
<td>0.055</td>
<td>0.075</td>
</tr>
<tr>
<td>U/C</td>
<td>0.015</td>
<td>0.025</td>
</tr>
</tbody>
</table>

Table 5: Drag coefficient values

In order to determine the induced drag, the Oswald factor needs to be estimated. For the clean configuration the value of the Oswald factor is approximated between 0.65 and 0.75. A lower value will decrease the efficiency of the wing and higher value will be difficult to manufacture wing with elliptical lift distribution. For the take-off and landing configuration the value is approximated between 0.55 and 0.65. The position of the undercarriage does not have any effect on the Oswald efficiency factor.

The design of the fuselage needs to be considered [15] for the estimation of the value of \( C_{D,0} \).

<table>
<thead>
<tr>
<th>Variable</th>
<th>units</th>
<th>Description</th>
<th>Constraints</th>
</tr>
</thead>
<tbody>
<tr>
<td>s_{to}</td>
<td>m</td>
<td>take off distance</td>
<td>To be determined</td>
</tr>
<tr>
<td>h_{ob}</td>
<td>m</td>
<td>obstacle height, considered as military aircraft</td>
<td>15.24m</td>
</tr>
<tr>
<td>c</td>
<td>m/s</td>
<td>climb velocity</td>
<td>5 m/s</td>
</tr>
<tr>
<td>c/v</td>
<td>-</td>
<td>climb gradient (CS 23.65)</td>
<td>0.083</td>
</tr>
<tr>
<td>n_{max}</td>
<td>-</td>
<td>maximum load factor</td>
<td>3</td>
</tr>
</tbody>
</table>

Table 6: Parameters denoting performance

The design variables are given in the table below.

<table>
<thead>
<tr>
<th>Variable</th>
<th>Units</th>
<th>Description</th>
<th>Constraints</th>
</tr>
</thead>
<tbody>
<tr>
<td>V_s</td>
<td>m/s</td>
<td>stall velocity</td>
<td>0 - 31.4</td>
</tr>
</tbody>
</table>
### Table 7: Design variables

| A | - | Aspect ratio | 5 – 15 |
| W/S | N/m² | Wing loading | see below |
| $C_{l_{\text{max, clean}}}$ | - | clean configuration | 1.0 - 1.8 |
| $C_{l_{\text{max, TO}}}$ | - | take off configuration | 1.0 - 1.8 |
| $C_{l_{\text{max, L}}}$ | - | landing configuration | 1.0 - 2.0 |

The equations relating the power loading and the wing loading for different configurations are formulated as equations below.

\[
W/P = \left[ s_{\text{to}} - \frac{V_S^2}{g} \sin \cos^{-1} \left( 1 - \frac{h_{\text{ob}} g}{6.96V_S^2} \right) \right] \frac{g \rho \omega C_{l_{\text{max}}}}{1.21(W/S)(1.1V_S)}; \quad (\text{takeoff consideration}) \quad 30
\]

\[
W/P = \frac{\eta_p}{\sqrt{W/S} \sqrt{2/\rho}} \frac{c}{1.345 \left( \frac{Ae}{C_l^{1/4}} \right)^{1/4}} \quad (\text{climb consideration}) \quad 31
\]

\[
W/P = \eta_p \left[ \frac{2}{\rho (C_{l_{\text{max, clean}}} - 0.2)} \left( \frac{c}{V} + \frac{C_P}{C_{l_{\text{max, clean}}} - 0.2} \right) \right] \quad (\text{climb rate consideration}) \quad 32
\]

\[
W/P = \frac{\eta_p}{0.5 C_{l_{\text{max}}} \rho \omega (W/S) + \frac{n_{\text{max}}}{0.5 \pi A e \rho V}} \quad (\text{manoeuvre consideration}) \quad 33
\]

The constraints for solving the equations above are given below.

\[
W/S \leq \frac{1}{2} \rho V_S^2 C_{l_{\text{max, clean}}} \quad (\text{stall landing configuration}) \quad 34
\]

\[
0 \leq V_s \leq 31.4 \quad (\text{from regulation for single engine propeller aircraft}) \quad 35
\]

\[
0.25 \leq W_b/(W - W_b) \leq 1.5 \quad (\text{based on Muller equation}) \quad 36
\]

\[
s_l = s_a + s_f + s_g \quad (\text{landing consideration}) \quad 37
\]

\[
5 \leq A \leq 15 \quad 38
\]

\[
1.0 \leq C_{l_{\text{max, clean}}} \leq C_{l_{\text{max, TO}}} \leq C_{l_{\text{max, L}}} \leq 2.0 \quad 39
\]

The problem is formulated as minimax problem and solved to get the power loading and wing loading.
4.4. Initial drag polar estimation
Determination of initial drag polar estimation is beyond the scope of this work. The values are assumed for using it in the computation. Initial estimation of drag polar needs feedback from other modules. Assumed Cd value will be used here. The ground effect is considered for the drag polar estimation for take-off and landing manoeuvres. Using the MMG containing the shape of the fuselage, the skin friction drag could be estimated for the fuselage.

4.5. Results and discussion
The graph below shows the convergence of different parameters over multiple iterations. It could be seen that some design variables such as the different lift coefficients and aspect ratio taking up the higher value for their limits specified. This is mainly due to absence of drag prediction module which is capable of setting constraints on the parameters.
It is assumed that plan form of the wing is considered as that of the baseline aircraft. The lift and drag coefficients are considered to be fixed. The wing loading – power loading diagram is obtained through optimizer routine.

$$W/S = 72.45 \text{ N/m}^2$$
$$W/P = 0.1865 \text{ N/W}$$
$$S = 0.1349 \text{ m}^2$$
$$P = 52.39 \text{ w}$$

### 4.6. Summary

In this chapter it is described how the thrust loading power loading diagram is obtained without the use of empirical data. It is not implied that engineering assumptions were excluded from this process. It only means that the data collected from observing other aircrafts were not used. The existing module can be upgraded if drag polar could be obtained from the MMG. The weight estimation was performed without use of empirical data.
5. Propulsion system selection

5.1. Introduction

Electric powered UAVs are considered for design using the design framework. Fixed wing small sized UAVs which are propeller driven are the most energy efficient vehicles, when there is no hover requirement. The advantages offered by electric powered UAVs [1] are listed below.

1) High reliability
2) Easy to start (even inflight restart is possible)
3) Precise power management
4) Low noise/vibrations
5) Reduced heat emissions and
6) Elimination of volatile fuels on-board.

Therefore electric propulsion was chosen for the type of the propulsion system for design using the framework.

The electric propulsion unit consists of a propeller, an electric motor, a speed controller and batteries. For the current research the focus will be on fixed pitch propellers. Only brushless DC motors are considered for the design. Lithium polymer batteries are considered as they offer highest energy density. A detailed discussion will be provided further in the chapter.

5.2. Objective

The objective of the design framework is to determine the combination of an optimal system based on predetermined criteria. The criteria are described below in the next section. The motors considered for the design are selected from the database of commercially available motors. In the current research, it is envisaged not to consider any possibility of designing or customizing the motor, thus motors are considered at systems level.
The batteries considered for the design will also be selected from a catalogue of commercially available batteries. The framework is proposed to allow the designer to select commercially available propellers.

5.3. Requirements
The requirements that must be fulfilled by the selection are

1. Minimum weight for the overall system. It was shown earlier that the batteries provide greater contribution towards the maximum take-off weight.
2. Highest efficiency for climb at maximum throttle or specific excess power required for climb. The requirement on climb arises from the certification specification/customer requirement specification.
3. Maximum endurance/maximum range settings. The unmanned aerial vehicle needs to perform the mission for the maximum endurance and for range requirements at partial throttle setting.
4. Optimal configuration. The configuration of components must be such that they could be packed in the least amount of space. This allows the fuselage to take a shape with minimum drag. Proper CG position can be achieved by optimal placement of the components.

5.4. Mission analysis
In the design framework, the mission to be performed by the UAV is defined. A generic mission profile is considered, consisting of take-off, climb, cruise, loiter, descent and landing. Conventional take-off and landing is considered in the current development of the design framework. No reserve power is considered for the design of the unmanned aerial vehicle. For conventional aircraft, the requirement on the reserve fuel is specified by the Certification Specification. For small UAVs (<150kg), the airworthiness regulations are not specified [1]. However, the reserve can be considered by increasing loiter time. Since batteries are used as power source and there are no expendable components and thus the weight of the UAV remains constant during the mission.

In traditional aircraft design, the initial estimate of the weight fraction is based on statistical data and the Breguet range and endurance equations. Most often during the initial design, the battery weight is assumed to be 50-60% of the total aircraft weight. Here the attempt is to use a more refined method for obtaining the weight of the battery weight required for the UAV. Since the climb, cruise and loiter missions phases are most important, only these missions will be considered here.

The electrical power required for the controls and the other subsystems will be defined by the user. L/D is considered as fixed within the mission analysis module. The wing loading is obtained from the preliminary sizing module.

![Figure 16: Mission definition](image-url)
The mission profile for the mission considered is shown above. The altitude is considered as 300ft. The mission time is considered as 60 minutes. Climb (3-4), cruise/loiter (4-5) and descent (5-6) are considered for further analysis.

5.4.1. Climb

In traditional aircraft design, the weight fraction for the climb segment is obtained from a historical trend. For electric powered UAV, the climb performance is important as the demand for power is high when compared with the power required for the cruise flight. When the power output from the battery is high then the specific energy gets reduced as described in the Ragone plot. Therefore performance calculation for climb must be performed and incorporated in design calculation. The figure 2 shows the ragone plot for few batteries. The lines originating from the origin are the constant discharge rate lines. The 360mah cell has a ragone plot with a gentle slope than 640 mah cell. This implies that when the power supplied in increased the total energy supplied by the cell decreases rapidly for the 640 mah cell. Therefore the total energy supplied by the battery will be much lower than the specified charge capacity.

![Ragone plot](image)

Figure 17: Ragone plot[3]

The objective is to determine the power required and energy required for the climb, in order to select the propulsion system. Since we are in initial phase of the design process, it is necessary to make suitable assumptions.

The achievable rate of climb desired at MSL (mean sea level) is specified by the designer. The power available is determined at a velocity, which need not necessarily correspond to the minimum power required. The electrical power is determined by considering the mechanical efficiency of the motor. It is assumed that the electrical power available is constant throughout the climb. Also the rate of climb is assumed to be constant. Since the focus of this work is only on small UAVs (<150kg) operating in low altitude, these assumptions are valid.

\[
c = \frac{P_a - P_r}{W}
\]
Climb rate is determined from the specific excess power.

\[
\frac{C_L^{3/2}}{C_D} = 1.345 \left( \frac{Ae}{c_D^{1/4}} \right)^{3/4}
\]

The lift and drag ratio corresponds to minimum power required. The specific excess power is computed at this point even though this might not correspond to the maximum specific excess power.

\[
P_{elec\_climb} = \frac{1}{\eta_m \eta_p} W \left( c + \frac{W^2}{\frac{1}{S} \rho (C_L/C_D)} \right)
\]

\[
E_{elec\_climb} = P_{elec} \frac{h_2 - h_1}{c}
\]

Note that in order to determine the maximum rate of climb, the power available is not considered as constant. The climb does not correspond to the minimum power required, and the \(C_L^3/C_D^2\) value is not the maximum value but only correspond to the maximum climb rate.

### 5.4.2. Cruise

The requirement for having best performance during cruise is to have minimum drag and \(C_L/C_D\) should be maximum. In the figure below, it is assumed that the power available is constant.
The range equation [2] for the electric powered UAV is derived as explained below. This represents the modified version of the Breguet’s equation for range and endurance.

The integral is computed between the initial and final time instance of the cruise flight.

\[ R = \int_{t_1}^{t_2} V_{cr} dt \]  

Electric current is the rate of discharge of the electric charge.

\[ I_{elec} = \frac{dC}{dt} \Rightarrow dt = \frac{dC}{I_{el}} \]
Propulsion system selection

\[ R = \int_{C_1}^{C_2} \frac{V_{cr}}{I_{elec}} dC = \int_{C_2}^{C_1} \frac{V_{cr}}{I_{elec}} dC \]

Now, the power required to overcome the drag at the velocity for cruise is computed,

\[ P_r = DV_{cr} \]

\[ P_a = \eta_{\text{tot}} P_{\text{elec}} \]

Equating the power available and the power required,

\[ P_a = P_r \]

\[ P_{\text{elec}} = \frac{DV_{cr}}{\eta_{\text{tot}}} \]

\[ P_{\text{elec}} = U_{\text{elec}} I_{\text{elec}} \]

\[ \frac{V_{cr}}{I_{el}} = \frac{\eta_{\text{tot}} U_{el}}{D} \]

\[ D = \frac{C_D}{C_L} W \]

The weight remains constant throughout the cruise and thus the angle of attack and altitude can remain constants throughout the cruise phase.

\[ R = \eta_{\text{tot}} \frac{C_L}{C_D} \frac{1}{W} U_{\text{elec}} \int_{C_2}^{C_1} dC \]

\[ R = 3600. \eta_{\text{tot}} \frac{C_L}{C_D} \frac{1}{W} U_{el}(C_1 - C_2) \]

The range is computed using the equation given above. C denotes the capacity of the battery used for the cruise.

\[ E_{\text{elec}cr} = 3600 U_{el}(C_1 - C_2) = \frac{R}{\eta_m \eta_p \frac{C_L}{C_D} \frac{1}{W}} \] (Energy)

The energy is computed using the equation given above.

\[ P_{\text{elec}cr} = \frac{3600 U_{el}(C_1 - C_2)}{t} = \frac{V_{cr}}{\eta_m \eta_p \frac{C_L}{C_D} \frac{1}{W}} \] (Power)

\[ P_{\text{elec}cr} = \frac{3600 U_{el}(C_1 - C_2)}{t} = \frac{1}{\eta_m \eta_p \sqrt{\frac{S \rho}{2 W}} \frac{C_L}{C_D} \frac{1}{W}} \] (Power)

The energy required is also determined along with the power required and range.
5.4.3. Loiter
The requirement for having best performance during loiter is to have minimum power required. The endurance, energy required and power will be derived in the following section.

\[ E = \int_{t_1}^{t_2} dt \]

\[ I_{elec} = \frac{dC}{dt} \Rightarrow dt = \frac{dC}{I_{elec}} \]

\[ E = \int_{c_1}^{c_2} \frac{1}{I_{elec}} dC = \int_{c_1}^{c_2} \frac{1}{I_{elec}} dC \]

\[ P_r = DV_{loiter} \]

\[ P_a = \eta_{tot} P_{elec} \]

The power available and the power available are equated.

\[ P_a = P_r \]

\[ P_{elec} = \frac{DV_{loiter}}{\eta_{tot}} \]

\[ P_{elec} = U_{elec} I_{elec} \]

\[ \frac{1}{I_{elec}} = \frac{\eta_{tot} U_{el}}{DV_{loiter}} \]

\[ D = \frac{C_D}{C_L} W \]

\[ V_{loiter} = \sqrt{\frac{W 2 1}{S \rho C_L}} \]

\[ E = \eta_{tot} \left( \frac{1}{W} \frac{C_L^3}{C_D^2} \frac{S \rho}{2W} U_{elec} \right) \int_{c_1}^{c_2} dC \]

\[ E = 3600 \left( \frac{C_L^3}{C_D^2} \frac{S \rho}{2W^3} \eta_{tot} \eta_{elec} (C_1 - C_2) \right) \]

The endurance of the electric powered UAV is given by the equation above.

\[ E_{elec_{loiter}} = 3600 U_{el} (C_1 - C_2) = \frac{E}{\eta_m \eta_p \sqrt{S \rho \frac{C_L^3}{C_D^2} 1}} \]

The energy consumed during the loiter is given in the equation above.
It can be observed that the equations for power, for loiter and cruise are same however it must be noted that \( V_c, C_L \) and \( C_D \) are different in the two cases. \( C_L/C_D \) is maximum for the range performance and \( C_L^2/C_D^2 \) is maximum for the loiter performance.

5.5. Electric propulsion system components

The propulsion unit consists of the propeller, the electric motor, the speed controller and batteries. For the current research the focus will be on fixed pitch propellers. Only brushless DC motors are considered for the design. Lithium polymer batteries are considered as they offer the highest energy density.

5.5.1. Propeller

The actuator disk model is considered for the analysis of propellers. The assumptions are that the flow is incompressible, continuous (except at the disk), irrotational and isentropic. Inflow factor (a) is the ratio of increase in the velocity at the disc to the free stream velocity. Slipstream factor (b) is the ratio between the increases in velocity at the fully developed flow to the free stream velocity.

\[
b = 2a
\]
\[
\eta_f = \frac{1}{1+a}
\]

The power required to turn the propeller is given by the cubic power law [3]. The exponent typically varies between 2.9 and 3.1.

\[
P = K_{prop} \times N^3
\]

In the current case, the coefficients are obtained from the web catalogue. Most often only the coefficient and exponent data are provided for commercially available propellers. The geometry of the propeller (propellers for radio controlled models) is not generally provided by the manufacturer. The power curve is obtained from measurements, which can be well represented using the formula mentioned above.

The thrust and torque coefficients are determined by the following equations.

\[
\eta_p = \frac{T \times V}{P_{out}}
\]
\[
\eta_p = \frac{T \times V}{Q \times \Omega}
\]

\[
T_c = \frac{T}{\rho V^2 D^2}
\]

The thrust coefficient is obtained by,
Propulsion system selection

\[ C_T = \frac{T}{\rho n^2 D^5} \]

The torque coefficient is obtained by,

\[ C_Q = \frac{Q}{\rho n^2 D^5} \]

The power coefficient is obtained by,

\[ C_P = \frac{P}{\rho n^2 D^4} \]

The advance ratio is given by,

\[ J = \frac{V}{nD} \]

\[ \eta_p = J \frac{C_T}{C_P} \]

\[ a = -1 \pm \sqrt{1 + \frac{8T_C}{\pi}} \]

\[ \sigma_r = \frac{BC}{\pi r} \]

\[ \Omega = 2\pi n \]

The advance angle is given by,

\[ \tan \phi = \frac{V(1 + a)}{r\Omega} \]

5.5.2. Battery

The battery types which are commercially available are Nickel Cadmium (NiCd), Nickel Metal Hydride (NiMH), Lithium Polymer (LiPo) and Lithium ion (Li ion). Only rechargeable batteries are considered here as they can be reused, are cost effective and are eco-friendly. Lithium polymer batteries are considered as they have the highest energy to weight ratio. They are compact and have high discharge efficiency. The nominal cell voltage is 3.7V. Lithium polymer batteries have a high risk of explosion if they are not properly used. However the advantages outweigh the disadvantages.

The batteries contribution to the weight fraction is high, therefore it becomes necessary to provide adequate treatment. High energy density and high power density are desired for the energy source. The power requirement for the payload and controls can be assumed to be 2 - 10W for small UAVs.

<table>
<thead>
<tr>
<th>Type</th>
<th>Advantage</th>
<th>Disadvantage</th>
<th>Cell Voltage (V)</th>
<th>Energy Density by weight (Whr/Kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>NiCd</td>
<td>Cheapest, Fast charging, not very sensitive to overcharging,</td>
<td>Heavy, memory problem, low capacity,</td>
<td>1.2</td>
<td>40-80</td>
</tr>
</tbody>
</table>
Propulsion system selection

<table>
<thead>
<tr>
<th>Batteries</th>
<th>Features</th>
<th>Capacity</th>
<th>Voltage Range</th>
</tr>
</thead>
<tbody>
<tr>
<td>NiMH</td>
<td>No memory problems, Medium cost, Heavy, sensitive to overcharging</td>
<td>1.2</td>
<td>60-100</td>
</tr>
<tr>
<td>LiPo</td>
<td>Light weight, Highest capacity and no memory problems, 3.7 volts/cell</td>
<td>3.7</td>
<td>130+</td>
</tr>
<tr>
<td>Li ion</td>
<td>No memory effect, high energy density</td>
<td>3.6</td>
<td>100-160</td>
</tr>
</tbody>
</table>

**Table 9: Comparison between different types of batteries**

The LiPo battery is usually arranged in series of 1-8 cells providing 3.5V - 30V, for typical UAV applications. The sizing of the battery pack depends on the capacity (mAh) and voltage (V) which in turn depends on the operating voltage range of the motor. A battery pack with higher voltage will discharge a lower current. The resistance loss is proportional to $I^2$ and hence a higher voltage is advantageous, however a battery pack with large number of cells gets a lower energy density. Therefore, a trade-off is required between choosing large number of cells with smaller capacity to less cells with higher capacity. The cut-off limit affects the battery life and the cut-off limit is usually set as 3.0v for LiPo and 0.7V-0.9V for the NiCd batteries.

The discharge profile of the battery (figure 4) shows the characteristic of the battery. When the battery is operating with no load the voltage at the terminals is 4.2, once a load is applied the voltage suddenly drops and stabilizes at around 3.7v. As the discharge rate C is increased, the discharge time and the average voltage is reduced. The nominal capacity is indicated by C rating. Lipo with capacity rating n can be discharged at constant current of n/5 in 5 hours for the voltage to drop from full voltage to 3.0 volt.

\[
\text{Capacity (delivered)} = I \times t \text{ amp} - \text{min} \\
\text{Energy (delivered)} = I \times t \times v \text{ watt} - \text{min} \\
\text{Specific energy} = I \times t \times v/W \text{ watt} - \text{min/g} \\
\text{Specific power} = I \times v/W \text{ watt/g}
\]

$IxR$ losses (current times resistance, heating loss) lead to internal heating of the battery which is also considered for the minimization. The discharge curves are converted into Ragone plot by integrating the area under the curve. The discharge curve is obtained from the manufacturer.

![Figure 19: Discharge curve of a LiPo battery](image-url)
The Ragone plot for different capacity cells are shown. The batteries considered belong to the same manufacturer and it could be observed that the lines have same slopes. The natural tendency will be to pick the battery with higher specific energy. Difference in slopes for batteries with different discharge rating is hardly noticeable.

![Figure 20: Ragone plot for batteries in the database](image)

The relation between the energy and power density is the Ragone plot in which a collection of data points are plotted with specific energy density (Wh/g) on the y-axis and power density (W/kg) on the x-axis. Each point on the Ragone plot is the result of a constant power discharge experiment [6]. A linear relationship is described for the Ragone plot for various battery systems.

The plot between specific energy $E_e$ and specific power $E_p$ is known as Ragone plot.  
1) The design point must be farthest from the origin  
2) If two curves intersect, the highest performance is given by flattest curve for short discharge times and steepest curve for large discharge times.  
3) For a specified discharge time, the cell with highest $E_e$ gives better performance.  
4) When the C rating (discharge rating)is higher, flatter is the curve.  
5) When C rating is higher, heavier is the battery.

The battery is the heaviest component in an electric power UAV system however only approximate methods are deployed. Gur et al. [17], derived a relation between the energy capacity and mass of the battery. A similar relation is derived based on the data available in the database.

$$EmAh = 43.22 \times m_B(g) - 32.99$$

The line depicting the equation is shown in the figure below.

$$C_b A_h = E_b / v$$
Propulsion system selection

Figure 21: Relationship between mass and capacity of batteries

Current database consists of lipo battery data pertaining to Enerland polyquest and the number of cells considered is 25.

In this approach the influence of discharge capacity of the battery, internal resistance, and the output voltage are not considered. The possible configurations involved are also considered. A more refined approach can be applied. The performance of the battery is better represented by a Ragone plot. Muller [1] provides a method for using the Ragone plot.

The weight of the battery also depends on the discharge capacity, higher the discharge capacity higher is the mass of the battery. The above relationship does not depict that relationship. For eg,

Muller[1] reports that the endurance of the electric powered UAV is proportional to

\[ E \propto \frac{W_B/W_1}{(1 + W_B/W_1)^{3/2}} \]

Endurance equation derived earlier in the chapter,

\[ E = 3600 \frac{C_l^3}{C_D^2} \frac{S \rho}{2W_1^2} \eta_{tot} U_{el}(C_1 - C_2) \]

The plot relating the endurance to the ratio of battery weight to the aircraft mass (minus batteries) is given below. \( T_{max} \) represents the maximum possible endurance.
Mueller [1] provided the relationship between the endurance and the weights. In this relationship the influence of other design parameter are not shown.

In figure above, Trips endurance equation is used keeping all variables constant except for the weight. This is done because, Muller’s equation shows the relationship between endurance and weights only. The relationship appears to be empirical. In Trips equation, the influence of battery weight on the capacity, lift coefficient, drag coefficient and voltage are not shown. This proves that there exists a relationship between weight and the other parameters. Since battery weight is the highest contributor to the weight, its influence on other parameters cannot be ignored. The objective of the work [3], was aerodynamic optimization and the objective function for endurance (considering a finite wing) is given by:
\[
\frac{\sqrt{(Ae)^3}}{(C_{D,0})^2}
\]

The aspect ratio \(A\), also influences the power loading which needs consideration.

Figure 24: Mueller/ Trips equations considering \(C_l, C_d\) variation

In figure above, since the \(W_b\) accounts for significant portion of the total aircraft weight, variation in \(W_b\) changes total aircraft weight significantly and in cruise \(L=W\), therefore \(C_l\) needs to be modified which affects \(C_d\) as well. (Here \(S\) is not varied). This is done because \(C_l\) (which in turn affects \(C_d\), are the aerodynamic parameters which) will be affected when the change in total aircraft weight is considered.

Figure 25: Mueller/ Trips relationships with capacity included (linear variation)
In figure above, the relationship between battery weight and battery capacity is included in the Trips model. A linear relationship proposed by Ohad [3] is used. A hypothetical case where the entire battery is used up during cruise is considered. Mathematically, \( C_2 = 0 \) to \( C_2 = C_{b1} \).

![Figure 26: Mueller/ Trips relationships with capacity included (second order variation)](image)

In the figure above, a more accurate second order relationship determined statistically from the commercially available battery [3] is used. It can be observed that the variation between the two models [3, 6] is minimal.

The relationship between the capacity and battery mass does not consider the influence of the discharge capacity on the battery mass. It is observed that the higher the capacity, higher is the mass especially for smaller battery packs.

![Figure 27: Battery capacity vs battery mass (C – Discharge rating)](image)
Figure 12 shows the variation between the mass and capacity of the battery. It can be observed that considering a battery delivering 4000mAh (3.7V), the battery mass is 83.1g (18C) and 99.4g (30C) which is a 19.4% increase. The requirement is to remove dependency on empirical data. The data obtained from the database represents the current level of technology and the database can be updated to ensure the validity of the database.

Once the capacity is determined from the propeller and motor analysis, then an initial estimate of battery pack mass is obtained. The number of cells in parallel and series are defined before the selection of the motor and propeller. In order to determine the mass of the battery pack, the mass indicated for the capacity specified must be obtained. The mass of the cells arranged in parallel are already accounted. The mass indicated must be multiplied by number of cells in series in order to obtain the mass of the battery pack.

After the selection of the motor and propeller, the parameters for different missions (current, voltage, rpm, time are known) the total capacity required can thus be determined. The capacity is the total value of the battery pack. If the capacity cannot be provided by a single cell or multiple cells in series, then a parallel configuration must be utilized. Even otherwise multiple smaller cells can be used but this is not beneficial as the increasing the number of cells in series

5.5.2.1. Methodology

1. Select the motor/propeller and determine the current drawn and discharge time.
2. Calculate the capacity required.
3. Obtain the mass of the battery from the mass-capacity relationship. When the cells are added in series, then voltage is increased and the capacity remains constant. There for the mass obtained must be multiplied by the number of cells in series to get the mass of the battery pack.
4. Check whether capacity can be provided by a battery pack in 1P configuration. If not, the number of cells in parallel is set as 2 (3,4 etc). Thus the number of cells in parallel is determined.
5. Select the candidate cells and obtain the Ragone plot for the individual candidate cells.

The database used in the work [2] contains data from various sources and contains batteries from different manufacturers. In the work [2] only the mass of the battery was considered as propulsion system optimization was considered. Battery database from Drive calculator [11] contains an extensive list of batteries but the discharge profile is not available. Obtaining discharge data for all the batteries available in the drive calculator database is not feasible due to fact that a large number of batteries are available. Most often the data is not published in the web catalog of the battery manufacturer.

For the research work - discharge profiles, mass and also the dimensions are required. Due to limited availability of data on discharge curves only batteries manufactured by Enerland ltd are only considered as extensive data is available and batteries with different discharge rating are available.

The drive calculator database provides the following data: \( I_{\text{max}}, I_{\text{peak}} \), capacity, weight, internal resistance and cell voltage. Motocalc database provides capacity, weight, impedance, cell voltage, C Rating.
Ragone plots are not directly available from the manufacturer. Some manufacturers provide the discharge curves. The Ragone plot can be constructed either by performing a discharge test. For this, the battery must be procured in order to perform the test. It is not feasible to obtain all battery packs and perform discharge test due to time/budget constraints. Due to these reasons the Ragone plot will be constructed from the discharge curve obtained from the web catalog. The discharge curve provides the capacity vs. the measured cell voltage. The area under the graph gives the energy provided by the battery under specific discharge rate being considered. The advantage of using a Ragone plot is that different cells can be compared using a single plot.

The discharge data provided by the manufacturer contains the voltage vs. capacity plotted for three different discharge ratings. A sample curve is shown in figure 2 for Enerland PQ300XP lipo battery. It can be clearly seen that the when the cell is discharged at 20 C, the energy obtained from the battery is greatly reduced. From this plot it is possible to obtain three points for the Ragone plot. When the current drained is increased, the voltage decreases and the capacity is lowered due to the presence of internal resistance.

The procedure followed for obtaining the Ragone plot is described below.

1) Obtain area under the graph for different discharge rating. The area represents the total energy supplied by the battery pack. The graph is available in raster format which must be converted into vector format in order to compute the area.

2) Compute the specific energy by dividing the total energy by the mass of the cell.

3) In a Ragone plot, the constant discharge lines originate from the origin. Determine the constant discharge lines in Ragone plot corresponding to the discharge rating specified in the discharge plot obtained from the manufacturer.

4) Plot the points on the specific constant discharge line corresponding to the specific energy.

5) Connect the three points using a spline.
5.5.2.2. Battery selection

Select the battery based on Ragone plot such that the both power requirement and energy requirements are met and battery weight is minimum.

5.5.3. Motor

In brushless motors the windings are present in the casing. The magnets are present in the rotor. The motors with brushes create electrical noise as they make and break contact. They need suppression using a capacitor. The brushless motor does not have these problems. The in-runner brushless motor has stationary coils which surround the rotating magnet at the centre. An out-runner brushless motor has stationary coils at the centre, and the rotating magnet on the outside. Out-runner motors have lower Kv ratings, in other words they operate at a lower speed, but with more torque. The high torque of the out-runner enables to direct drive larger propellers without any need for a gearbox.

DC electric motors are attractive as they have high efficiency with 50-75% for small 1-20W motors to upto 90% for larger motors. The internal combustion engines in this range provide a very low efficiency. The electric motor can be cored, coreless and brushless. Brushless motors offers highest specific power, has the greatest durability and highest efficiency [1].

Electronic speed controller efficiency needs consideration as the brushless motor cannot operate without the ESC. ESC vary the average voltage applied to the motor by pulse width modulation. The output power is regulated by adjusting the duty cycle length.
Propulsion system selection

The data available in the database shows considerable variation in the no load current therefore the effect of this variation will be considered. Measurement data also includes the temperature and altitude, however these are not considered currently.

\[ I_0 = \frac{V}{R} \]
\[ N_0 = K_v \times V \]

Shaft power: \[ P(W) = \frac{N \times T_M (g \cdot cm)}{97.500} \]

Efficiency: \[ \eta_M = \frac{P}{V} \times I \]

Motor heating: \[ H(W) = V \times I \times (1 - \eta_M) \]

Maximum power \( P_{\text{max}} \) occurs at \( T_M = \frac{T_{MS}}{2} \)

Maximum efficiency occurs at \( P<P_{\text{max}}, \) at current \( I = \sqrt{I_0 I_S} \)

Is - Stalled motor current

5.5.3.1. Operating Principles for DC motors

1) Maximum power occurs midway between no load and stall load

2) Maximum efficiency occurs below maximum power

3) Increasing motor loads with higher pitch props will increase current and motor heating and will not necessarily increase current and motor heating; will not necessarily increase power

4) Reasonable operating loads, current and rpm are always between maximum power and maximum efficiency
5) Motor heating limits maximum useful power.

6) At some point below maximum power, a best compromise between power and efficiency exists.

7) If best operating RPM is too high, gearing down motor can provide equal power, torque, and efficiency at lower rpm (scaled by gearing ratio), though motor efficiency is reduced by gearbox efficiency $\eta_g$ such that new $\eta'_M = \eta_M \times \eta_g$.

Power to turn a propeller at different speeds can be approximated by $P = K_{prop} \times N^3$. $K_{prop}$ depends primarily on diameter and least on the blade profile. This is particularly suitable for thin airfoil model propellers 13 - 30 cm (5-12 in) diameters with pitches 7.6 - 30 cm (3-12 in). The exponent value 3 is changed to 2.9 to 3.1 with change in hub/blade/pitch. $K_{prop}$ is obtained from one or two known P and rpm points. $K_{prop}$ sometimes made available in catalog.

1) Stall torque and no load speed are proportional to supply voltage.

2) Current is independent of the supply voltage, depends only on torque.

### 5.5.3.2. Modelling

1) Power factor is equal to unit. (Applicable to small brushless permanent magnet motors).

2) Magnetic losses (Eddy current/P/ hysteresis) are neglected.

3) Constant power to mass ratio $B_{p-m}$ considered.

$$P_{out-max} = B_{p-m} m_M$$  \hspace{1cm} (104)

4) Torque constant is assumed to be equal to speed constant

$$K_Q = K_V$$  \hspace{1cm} (105)

5) First order model is considered

$$\Omega = (V_{in} - I_{in} R_a) \cdot K_V$$ \hspace{1cm} (106)

$$i = \left(V - \frac{a}{K_V} R_a\right)$$ \hspace{1cm} (107)

Driver input power $P_{in}$ and output power $P_{out}$ are related by

$$P_{in} = V_{in} \cdot I_{in}$$ \hspace{1cm} (108)

$$P_{out} = (I_{in} - I_0) \cdot (V_{in} - I_{in} R_a)$$ \hspace{1cm} (109)

In: Driver input current, $V_{in}$: Driver input voltage, $I_0$: Current at no load. $R_a$: Motor resistance. The electric system efficiency is given by:

$$\eta_s = \eta_D \left(1 - \frac{I_{in} R_a}{V_{in}}\right) \left(1 - \frac{I_0}{I_{in}}\right)$$ \hspace{1cm} (110)

$\eta_D$ is considered as constant (=0.95)
5.6. Selection procedure

5.6.1. Case 1

Flight speed $V$ and applied motor voltage $V_0$ are specified and are kept constant for case 1.
5.6.2. Case 2
Flight speed V and thrust required T are specified for case 2.

Match propeller torque Q at V to the motor torque Qm at Vo, determine angular velocity Ω.

Determine thrust T from the propeller analysis performed at Ω for V.

Determine propeller efficiency η_p for the specified V and Ω.

Determine the mechanical efficiency η_m for V and Ω.

a) η_m > 75%, η_p > 75% - basic criteria; b) η_m lies right of the peak, η_p lies left of the peak.

Determine the angular velocity Ω for specified thrust T.

Determine the propeller efficiency η_p for the Ω.

Determine Vo: a) Obtain the propeller torque Q at specified V and Ω determined.
b) Determine Vo for which Qm=Q at the specific Ω.

Determine the mechanical efficiency η_m for the Ω.

a) η_m > 75%, η_p > 75% - basic criteria; b) η_m lies right of the peak, η_p lies left of the peak.

\[ V_o = (K_v Q + I_0) R + \frac{\Omega}{K_v} \]  \hspace{1cm} 118

Battery capacity determination based on mission- power and energy requirements. Impact of Ragone plot on the determining the capacity of the battery

5.7. Results and discussion
Mission analysis results considering the test case mission. A selection detail for the battery, propeller and motor is performed. The values for the efficiencies and rpm are given for the mission segments – climb, cruise and loiter.

<table>
<thead>
<tr>
<th>Motor</th>
<th>KD A20-22L</th>
</tr>
</thead>
<tbody>
<tr>
<td>Propeller</td>
<td>Graupner speed – 6.50x6.50</td>
</tr>
</tbody>
</table>
5.8. Summary

In this chapter, the selection of propulsion system is described. An extensive discussion on Ragone plot and its application in the selection of propulsion system is provided. The importance of considering the weight of the battery is emphasized.

<table>
<thead>
<tr>
<th>Battery</th>
<th>PQ 4350XP</th>
</tr>
</thead>
<tbody>
<tr>
<td>( n_{b, S} )</td>
<td>2</td>
</tr>
<tr>
<td>( n_{b, P} )</td>
<td>1</td>
</tr>
<tr>
<td>( \eta_p )</td>
<td>[0.4999, 0.2385, 0.2385]</td>
</tr>
<tr>
<td>( \eta_m )</td>
<td>[0.5476, 0.8151, 0.8151]</td>
</tr>
<tr>
<td>( \eta_{\text{tot}} )</td>
<td>[0.2733, 0.1944, 0.1944]</td>
</tr>
<tr>
<td>Rpm</td>
<td>[8262, 9323, 9323]</td>
</tr>
<tr>
<td>( T_{ci} )</td>
<td>1.8344 N</td>
</tr>
<tr>
<td>( V_{cr} )</td>
<td>8.79 m/s</td>
</tr>
</tbody>
</table>

Table 10: Selection
6. Multi-domain interaction

6.1. Introduction

Knowledge-Based Engineering is described by La Rocca as an expert system involving parametric CAD, object oriented programming (OOP) and rule-based design [9]. The KBE approach is better suited for engineering design and analysis when prior knowledge about the product is available. The product knowledge implies not only the geometric information but also properties of the components which includes physical, thermal, electrical and other properties. The focus of the CAD system is based on the geometry. Traditional engineering analysis application focuses on capturing the geometry or simplified representation of the component. The KBE environment can be used to capture the geometry, the associated physical properties and thus the product knowledge is accumulated.

In knowledge based engineering, the focus is on a mathematical model of the geometry. The shape of the object is known apriori unlike the CAD based approach where the focus is on the process of generating the desired geometrical model. The properties of the systems considered and the domains in which they interact are also known apriori. The CAD based approach focuses on a single point design whereas the KBE approach can be used to generalize a family of products. In KBE, the product knowledge and the process are captured. It allows reusability of the product knowledge as well as the process knowledge captured at one phase of the development. Reusability, availability within a design framework and suitability for optimization are the requirements of a design framework.

Foeken[5] focuses on detecting the multi-domain interaction in the early phase of the design in order to reduce the cost incurred due to redesign. The anomalous behaviour of components when integrated into a system, which otherwise function normally is substantiated by Foeken [5].

In the research, the interaction is detected using analytical method and evaluation in performed in real time facilitating the design iteration for conceptual aircraft design. The object oriented design
and the associated data is translated to automate design process. The products and interacting domains are defined such that the integration at higher level is performed seamlessly.

Spatial interaction is based on spatial proximity which was detected based on geometric interference [5]. In the research, detection is performed based on the bounding box approach. This approach not only reduces the order of complexity involved but provides right fidelity data for the aircraft design process. In the conceptual design phase, the configuration for the systems is used to determine the inertia and the centre of gravity. The availability of the data in automated design process enables the designer to evaluate multiple configurations.

In the previous chapter, the contribution due to the weight fraction of the battery is described. Since the weight fraction of the battery has a greater contribution and it can influence the position of centre of gravity which in turn influences the static margin. However the computation is beyond the scope of the current work.

The computation of the view factor by various methods such as the analytical method, hemi cube method, was considered. The computational domain is considered to be in 2-d surface, which are continuous and smooth. If curved surfaces are considered then Gauss Legendre integration method might be required.

In order to implement the concept of multi-domain interaction, it is important to have seamless transition between the multi-model generator, various disciplinary tools and the convergence evaluator step. The computational effort must be minimal such that multiple configurations can be evaluated.
The knowledge-based approach to multi-domain interaction detection [5] is implemented and validated in the context of conceptual design of UAV. The detection of interaction between components is performed using knowledge based engineering principles and object oriented programming. The geometric and behavioural information is used to detect the interaction in spatial and energy domains.

6.2. Parametric fuselage shape representation

Lofting is considered as a special case of a more general surface construction referred to as a directrix-generator representation. This is a representation that describes a surface in terms of sweeping a generator curve along a set of guide lines [1].

The three ingredients for such a representation are
(1) a set of longitudinal curves called directrices (also called meridians in mathematics),
(2) a correspondence rule that relates each point of a directrix with a unique point on every other directrix, and
(3) a generator rule that defines a curve through all the points on the directrices that are related by the correspondence rule.

In traditional aircraft design, cross sections primarily used for fuselage design are conic sections. The major requirement is ease of reproducibility. In KBE, the fuselage is segmented and represented as a trunks and adapters. The major requirement for representing in KBE is reusability.

The Design and Engineering Engine framework [9] (DEE) framework proposed by La Rocca et al. for aircraft design processes uses GDL® for the development of multi-model generator. The DEE is capable of performing variable fidelity analysis in real time and performing design iterations which require a shape parameterization technique which uses the least number of parameters for fuselage shape representation. The requirement is on ease of representation, data handling and ability to represent complex shapes.

A typical CAD based model focus on accurate representation of the shape but there is no requirement on the simplicity to represent shapes. Extracting geometry data is also inconvenient. GDL® is used for development of the MMG for fuselage shape representation. The requirement imposed by the tool is ease of code maintenance. A GUI is developed in Matlab® for representing the fuselage. The directrices representing the top, bottom and side is user defined. The point locations can be modified by the user. The MMG defined in GDL® for fuselage shape representation is independent of the frame definition defined using the GUI. The Matlab tool allows the user to develop complex fuselage shapes.

The cross sectional shape of the trunk is defined at locations 0, 1/3, 2/3 and 1 computed from the nose to the end. The trunks are represented using line represented by cubic polynomial. The cross sectional shape of the trunk is defined at locations 0 and 1 computed from beginning to end. Ten cross sectional shapes are predefined. Cross sectional shapes can be added or removed by the user. The cross sectional shapes are defined using Bernstein polynomials in the GUI and frame definition files are generated. The GUI defined in Matlab® can generate fuselage with three trunks and two adapters. Fuselage definition with three trunks and two adapters is sufficient for representing complex and novel fuselage shapes. However the number of trunks and adapters for the definition of the fuselage shape can be increased or decreased. The trunk and the adapter shapes can also be generated using the Matlab.

The cross sectional shapes are scaled and stacked at respective positions. A lofted surface is created using the cross sectional shapes in GDL®. The frame is scaled such that the top, bottom and sides coincide with the extremes of the frame shape definition. The cross section definitions are provided in GDL using first order curve and therefore sharp edges can be captured, unlike in a traditional CAD package. The shape must be easily reproducible. The surface of the trunk is represented using a third order surface. The surface of the adapter is represented using first order surface.

An improved mathematical modelling of the trunk surface can be developed in order to ensure continuity between two adjoining fuselage trunks. The fuselage shape definition files can be generated from Matlab® and hence shape definition can be controlled in a Matlab program. Intersection between the components and the fuselage shape is not considered for the present
development. The upper section and the lower section are defined separately and symmetry is
assumed about the plane x-z plane. For further development, the intersection between the
components and fuselage shape can be implemented.

It is a traditional conic lofting example of an airplane fuselage. The directrices are conics and the
generator consists of two conics, one above the maximum width line and one below. The
correspondence rule relates points with equal x-values. Many surfaces can be defined by means of a
directrix-generator construction. The directrices for the ruled surface are the bounding curves and
the generators are the straight-line segments that connect the corresponding points on these
curves. A surface of revolution has a single directrix, namely, the curve being rotated, and the
generators are circles. The idea of a directrix-generator representation can be generalized to
allow directrix to act as control points rather than being interpolated. The traditional lofting
methods involve the description of the frames using conic sections and the directrices.

The KBE model described by La Rocca describes the representation of the components as an
aggregation of the high level primitives (HLP). The requirement is placed on the least number of
control parameters. The advantage is that complex shapes can be generated with minimal control. It
allows the designer to evaluate multiple configurations

The implementation is performed considering a two-step approach. The steps are decoupled and
they can function independently. The trunks are implemented as objects in GDL and the definition
files are provided for section definition. A graphical user interface is developed in Matlab which can
generate the definition files for the GDL model creation module.

Class-shape transformation (CST) method[18]

\[
\zeta(\eta) = 2\eta^{0.5}(1 - \eta)^{0.5}
\]

The shape function is considered as a constant with value 2.

Where:

\[
\eta = y/w \text{ and } \zeta = z/h
\]

\[
Su(\eta) = \frac{\zeta u(\eta)}{\eta^{NC1}(1 - \eta)^{NC2}} \text{ (shape function for upper lobe)}
\]

The shape function is kept as 2 and the class function exponents (NC1 & NC2) are set as 0.5.

\[
C(\eta) = \eta^{Nu}(1 - \eta)^{Nu}(\text{class function}) \text{ (upper section)}
\]

\[
C(\eta) = \eta^{Ni}(1 - \eta)^{Ni}(\text{class function}) \text{ (lower section)}
\]

\[
z(\eta) = Su(\eta)C(\eta)
\]

When symmetry about the vertical line is considered,

\[
C(\zeta) = \zeta^{NC1}(1 - \zeta)^{NC2}(\text{class function})
\]

The right and left lobe shapes are given as described above.

\[
\eta(\zeta) = Su(\zeta)C(\zeta)
\]
The CST method is used to develop the cross sectional frame of the fuselages. The shapes which are generated in Matlab and input file is generated (see appendix D) for the GDL program. The input file is read by the GDL program to denote the trunk.

![Figure 30: Fuselage frames defined using CST method](image)

The KBE concepts for the fuselage representation including trunks and adapters are used for the representing the fuselage shape in GDL. The lofted-surface class is used for representing the trunk or the adapter concept. The order for representation of the curve considered as first order, in order to consider the sharp curves.

![Figure 31: Fuselage shapes from GDL](image)

A graphical user interface is developed in Matlab to facilitate the generation of fuselage definition files for the GDL based multi-model generator. The GUI allows the user to define a fuselage shape with three trunks and two adapters.
The GUI allows the designer to define the side view represented by three directrices, named as top, bottom and side. The top and bottom directrices are 2 dimensional, considered being located on the XZ plane. The side directrix is a 3 dimensional curve, and the side view provides the projection of the curve in the XZ. The top view provides the projection of the side directrix on to the XY plane.

The definition of the top directrix is used to define start and end positions of the trunks and adapters. The GUI allows for definition of three trunks and two adapters. The fuselage trunk is defined using four cross sectional frames as the third order polynomial is used in lofting direction. The fuselage adapter is defined with two cross sectional frames. For the current development, the end section of the first trunk and the first section of the subsequent trunk is used for the definition of the adapter. The points picked interactively and curve is generated by fitted using a cubic polynomial for the trunk. The directrix for the adapter is defined as a first order curve.

The adapter can be defined such that the segment of the directrix defining the top, side and the bottom are tangential to the directrix segments in the adjacent trunks at the terminal location. The top directrix is defined as multiple segments whereas the side and the bottom directrices are defined as continuously. The beginning and the end points of the trunks and adapters are defined using the end points of the top directrix. Ten cross sectional shapes are predefined and cross sectional shapes can be also be defined by the user.

The frame is defined in unit square defined as $-0.5 \leq x \leq 0.5$ and $-0.5 \leq y \leq 0.5$. The top, side and the bottom directrices are used to scale the frame. The upper and lower directrices are used to scale the frame in the y direction. The side directrix is used to scale the frame in the x direction. The fuselage is symmetric about the xz plane.

The focus is on the intuitive graphical user interface which facilitates the rapid design of the fuselage. The graphical user interface and the GDL program are decoupled. The shape definition in
GDL is void of limitation in number of trunks and adapters. The frame defined in the GUI is based on the CST technique. The frame definition for the GDL program is independent and any frame file that can be specified.

### 6.3. Representation of systems

The components are described in wireframe representation for the visual display of the components. The wireframe representation provides a lightweight implementation. The components are considered as rigid bodies and the physical properties are associated with the centre of mass.

An object hereby denoted as **node** is defined as a point mass with co-ordinate system representing X-Y-Z axis colour coded with R-G-B (red – green – blue). The mass and inertia are represented by the node. The inertia is represented as inertia tensor.

The location and orientation can be set relative to the level from which it is accessed. For e.g., the node defined within a component with location L and orientation N will retain them relative to the location and orientation of the component.

The origin of a component is indicated by positioning a node with zero mass and inertia tensor set to a null matrix. The node object is reused to denote centre of gravity location by defining the position, mass and the inertia tensor set to the user provided value. The CG node location is specified with respect to the origin of the component.

The CG node of every component is accessed to compute the position of the centre of gravity for a specified configuration. The computation is performed with respect to a specified global origin and a node is created at the location of centre of gravity for the entire configuration. The inertia for the configuration is computed by accessing the inertia tensors defined and subsequently transforming with respect to the global axis system.
The inertia for components with simple geometries, like the battery and the motor are computed during runtime as the mass and dimensions are based on the user selected values. Uniform density is assumed in both cases. The other components are initially modelled as solid part in a commercial CAD package (SolidWorks®). The centre of gravity and the inertia tensor are obtained after adjusting the density to obtain the specified mass for the components.

The translation and orientation in GDL are performed using direct representation rather than relative representation. The approach allows alterations to be made from a higher level.

\[
\text{(center } #(0 0 0)) \quad \text{representation of the center of the component}
\]

\[
\text{(orientation } #2A((1 0 0) (0 1 0) (0 0 1))) \quad \text{representation of the orientation of the component}
\]

The rotation of the component is achieved using manipulation using transformation angle

\[
\{E_1\} = [\psi]\{E_0\}
\]

Where \([\psi]\) is the as rotation transformation matrix denoted as the orientation matrix in GDL.

The inertia tensor is represented as:

\[
[J] = \begin{bmatrix}
I_x & -C_{xy} & -C_{xz} \\
-C_{xy} & I_y & -C_{yz} \\
-C_{xz} & -C_{yz} & I_z
\end{bmatrix}
\]

When the component is rotated, the change in representation of inertia tensor is effected by

\[
[J_1] = [\psi][J_0][\psi]^T
\]

The computation is performed at the centre of mass position computed for the entire configuration.

### 6.4 Interference detection based on the bounding box approach

The physical interference detection based on the bounding objects is described by Agoston [19].

Definition[19]: *Bounding object for an object A is any object B that contains A. The desirable properties are 1) Ease of computation, 2) Scope for determining intersection and 3) Provides a close fit.*

A box in \(R^n\) is any subset of the form

\[
[a_1, b_1] \times [a_2, b_2] \times \ldots \times [a_n, b_n]
\]

where \(a_i, b_i \in R\).

A bounding box for an object is a box that contains the object. GDL® by default evaluates the bounding box for components defined as wireframe/solids and the value is accessed with computed slot bounding-box.

The Minimax test [19] described below is used for detecting the spatial interference of the bounding boxes. The boxes X and Y, defined as
Multi-domain interaction

\[ X = [a_1, b_1] \times [a_2, b_2] \times \ldots \times [a_n, b_n]; \ Y = [c_1, d_1] \times [c_2, d_2] \times \ldots \times [c_n, d_n] \]

will intersect if and only if

\[ lb_i = \max(\min(a_i, b_i), \min(c_i, d_i)) \leq ub_i = \min(\max(a_i, b_i), \max(c_i, d_i)) \]

for all \( i \). If they intersect, then

\[ X \cap Y = [lb_1, ub_1] \times [lb_2, ub_2] \times \ldots \times [lb_n, ub_n] \]

This approach detects the presence of intersection and provides the value for the \( n \)-dimensional object representing the zone of intersection.

### 6.5. Interaction in thermal domain

The form factor can be defined as the projection of area of the one surface onto another surface. The form factor calculation is based on the geometry. The form factor is also known as the view factor, configuration factor, angle factor or shape factor. The form factor computation is based on the geometric details and the integral is computed based only on the geometry.

Computation of the integral using Newton quadrature method should be adopted for surfaces involving curvature. The method reduces the computational effort but increase in accuracy was not observed. The direct method is used for solving the matrix, namely Gauss elimination. When the size of the matrix is increased, iterative solver like the Gauss Siedal method is required and the use of iterative method lead to increase in error during computation. The surface considered for the current work is rectangular surface therefore direct summation is performed. When oblique/curved surface in 2D is considered the integral can be computed using the trapezoidal rule or Simpson’s rule. The surfaces are considered as black and there is no reflected radiation. Energy balance on a surface element can be considered such that the radiative heat transfer calculation can be performed.

The view factor computation is performed by analytical method. The view factor for each cell is independent of view factor computed for the adjacent cell. Since the view factor of a cell is independent of value computed at an adjacent cell, differencing scheme is not required.

![Figure 34: A square cell (trimetric view - left, top view - right)](image_url)
The cell is defined as an object and the physical properties are defined as the attributes of the object. The cell is used for computation as well as post processing/visual display. GDL can represent the colour coding, VRML display can be invoked from GDL for visual display. The cells are created and destroyed dynamically, which implies that the objects are instantiated in the memory during the runtime of the KBE application.

The concept of cell can be identified with definition of 2-d elements in object-oriented finite element methods [20, 21], however the later focuses on the application of the object-oriented approach to the finite element method and here the focus is towards object-oriented approach towards the physical component and its associated properties. The term cell is coined to disassociate the concept from the term ‘element’ used in finite element methods. The cell, in the present work refers to a 2-d rectangular surface wrapped to the periphery of the physical component.

The cell is defined as object therefore additional data members can be added. The possibility to have nodes at the corners of the cells can be used to generate input files for other applications directly without the requirement to segment the surface to obtain the corner points.

In the present work, the computation is performed at the centroid of the cell, therefore a node is created at the centroid. The approach eliminates the need for generation of files containing the mesh details as in the case of finite element methods. The computation will be performed at the centroid of the cell which is suitable for the domain currently under consideration.

6.5.1. Grid

The grid is considered as an arrangement of cells. In the current context, the grid is considered as arrangement in rectangular pattern. This approach can be used as the mathematical definition for the surface to be formed is known in prior. The grid is defined during the definition of the component as an object. In this current work, arrangement in rectangular pattern is utilized.

![Grid representation](image)

A node is placed as a reference and the cells are arranged with respect to the cell. The node acts as a handle for orienting the grid with respect to the surface of the component. The term ‘Grid’ is coined to represent the arrangement of cells in the KBE technique which is analogous to the mesh defined in the FEM method. The former is generated using the mathematical function denoting the geometry, whereas the latter is generated by discretizing the geometry. The grid is generated
dynamically and the mesh need not be stored. The grid is generated dynamically therefore only the mathematical model is stored or encoded in the application. The grid and its associated cells are not stored. For the current work, rectangular grid is generated.

The grid is generated and is attached to the wireframe geometry of the component using a ‘handle’. The handle is represented using a node. The number of grids present in the object is known apriori. The location of each grid is also determined in priori.

In KBE technique, the shape of the object is known in prior. The shape is represented using a mathematical function therefore the mathematical model for development of the grids is known in prior. The cells in the grids are accessed directly for analytical computation. The properties of the grid are not used directly, only the cells are accessed for computation. The grids are represented as a sequence.

### 6.5.2. View factor

The view factor represents the fraction of energy which is emitted by one surface and intercepted by another. The view factor represents the relative location and orientation of the surfaces of the analysed domain.

The view factor $[\] \text{between two infinitesimal surface elements } dA_i \text{ and } dA_j \text{is defined as}$

$$dF_{dA_i \rightarrow dA_j} = \frac{\text{diffuse energy leaving } dA_i \text{ directly toward and intercepted by } dA_j}{\text{total diffuse energy leaving } dA_i}$$  \hspace{1cm} (130)

The total energy leaving $dA_i$ and intercepted by $dA_j$ is $F_{dA_i \rightarrow dA_j} dA_i$, can be viewed as energy leaving $dA_j$ and intercepted by $dA_i$, that is, $F_{dA_j \rightarrow dA_i} dA_j$. The reciprocity law states that

$$dA_i F_{dA_i \rightarrow dA_j} = dA_j F_{dA_j \rightarrow dA_i}$$  \hspace{1cm} (131)

For given surface elements $i$ and $j$, the vector $s_{ij}$ is defined as the vector joining the centre nodes of $i$ and $j$. The vectors $\hat{n}_i$ and $\hat{n}_j$ are defined as the normals to $i$ and $j$ respectively.

The view factor is given as

$$F_{dA_i \rightarrow dA_j} = \frac{(\hat{n}_i \cdot s_{ij})(\hat{n}_j \cdot s_{ij})}{\pi(s_{ij} \cdot s_{ij})^2} dA_j \hspace{1cm} (132)$$

Integrating to determine the view factor between two finite areas $i$ and $j$,

$$F_{A_i \rightarrow A_j} = \frac{1}{A_i} \int_{A_i} \int_{A_j} \frac{\cos \theta_i \cos \theta_j}{\pi s_{ij}^2} dA_i dA_j$$  \hspace{1cm} (133)

It is assumed that the entire surface is isothermal.

Integration technique such as quadrature method can be used to reduce the computational effort while considering curved surfaces. In the current work, only rectangular surfaces are considered. Therefore the problem is formulated using direct summation.
The integration is performed with respect to the \( dA \) as the integral does not contain any physics based terms which are functions of \( dA \). It is sufficient to represent the domain considered. The integral is transformed to a summation. It must be noted that the other terms depend on the positions, say in x and y direction and \( \rightarrow dxdy \). The effectiveness of this approach is justified by the mesh convergence results.

**Mesh convergence:**

The mesh convergence was displayed using two surfaces with cell dimensions of 3x3, and two surfaces with 5x5. The percentage error observed is low as the surfaces considered are rectangular in shape.

**Justification for this approach**

The grazing angle is defined as the angle subtended between the normal to the surface under consideration and direction of the emitted radiation.

The grazing angle is the angle subtended between \( \hat{n}_i \) and \( s_{ij} \); \( \hat{n}_j \) and \( s_{ij} \).

\[
F_{A_i \rightarrow A_j} = \frac{1}{A_i} \sum_{i=1}^{N} \sum_{j=1}^{N} \frac{\cos \theta_i \cos \theta_j}{\pi s_{ij}^2} dA_i dA_j
\]

The maximum value for grazing angle is set as 90 degrees for the computations reported in this work. However based on experimental evidence, the surfaces participating in radiative heat transfer do not contribute considerably beyond grazing angles with a value of 60\(^{\circ}\) [22].

![Figure 36: Directional variation of surface emittances for non-metals (top) and metals (bottom) [22]](image-url)
Since multiple components and henceforth multiple surfaces are considered for interaction. The grazing angle is considered for each cell individually. Form factor for two participating grids will be considered only if grazing angle computed is greater than the set value. In order to increase the participation from multiple grids, the grazing angle for the current approach is set as 90 degrees for demonstration.

The components are considered as blackbodies and therefore $\varepsilon = 1$.

**Line of sight algorithm**

The obstruction in the line of sight is determined by ray intersection with the bounding box of the component. The components are classified as participants and non-participants. The participants are the components which are involved in radiative heat transfer problem.

The participating components are instantiated twice, in order to simplify the implementation. The view factors are computed for the surfaces on the participating components with respect to another surface. The grids in the participant surfaces are used in the computation. The grids in the same participant components are excluded, considering the components as convex.

The vector connecting the line joining the centres of the cells given by $i$ and $j$ are given by $PQ$,

$$PQ = LC_j - LC_i = x_i \hat{i} + y_i \hat{j} + z_i \hat{k}$$

Let $O$ be a point on the plane with the normal as $\hat{n}$ is considered, the point of intersection is given by,

$$t_x = \frac{PO \cdot \hat{n}}{x_1}$$

$$I_x = LC + t_x \cdot PQ$$

The sides of the bounding box are oriented with the global axis system, therefore $\hat{n} = \hat{i}, \hat{j}$ and $\hat{k}$. The sides are represented with plane and the limits are described. The intersection of the line with the sides of the bounding box is considered.

![Figure 37: Grids having view (a) and no view (b)](image)
A two-step approach is considered, one for detecting the presence of the other surface within the domain and another for determining the collision with any object obstructing the way. Any obstruction will exclude the surface for which the form factor is computed. The receiving surfaces are considered as black body, implying that the incident radiation is completely absorbed.

The GDL environment provides the view factors, $F$ in the form of a matrix

$$A = \begin{pmatrix}
1 - F_{1-1} & -F_{1-2} & \ldots & -F_{1-n} \\
-F_{2-1} & 1 - F_{2-2} & \ldots & -F_{2-n} \\
\vdots & \vdots & \ddots & \vdots \\
-F_{n-1} & -F_{n-2} & \ldots & 1 - F_{n-1}
\end{pmatrix}$$

The local heat flux $q$ is given as,

$$q_i = F_{i-2}(E_{b1} - E_{b2})$$

$$A.e_b = b$$

$i = 1,2..n$ denotes the surfaces interacting for radiative heat transfer

$$e_b = \begin{pmatrix} E_{b1} \\ E_{b2} \\ \vdots \\ E_{bn} \end{pmatrix}$$

Here $H_o$ is the external irradiation term.

$$b = \begin{pmatrix}
q_1 + H_{o1} + \sum_{j=n+1}^{N} F_{1-j}E_{bj} \\
q_2 + H_{o2} + \sum_{j=n+1}^{N} F_{2-j}E_{bj} \\
\vdots \\
q_n + H_{on} + \sum_{j=n+1}^{N} F_{n-j}E_{bj}
\end{pmatrix}$$

Matrix $A$ is inverted and thus the temperature is determined,

$$e_b = A^{-1}.b$$

### 6.6. Results and discussion

The form factor is obtained from the MMG in the form of a matrix, a sample output is shown in the table given below.

<table>
<thead>
<tr>
<th>Number of surfaces</th>
<th>8</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>0</td>
</tr>
<tr>
<td>0.302</td>
<td>0.0026</td>
</tr>
<tr>
<td>0.0036</td>
<td>0.0074</td>
</tr>
</tbody>
</table>
Table 11: GDL output (form factor)

<table>
<thead>
<tr>
<th>Value 1</th>
<th>Value 2</th>
<th>Value 3</th>
<th>Value 4</th>
<th>Value 5</th>
<th>Value 6</th>
<th>Value 7</th>
<th>Value 8</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0.0087</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>0.0045</td>
<td>0.0213</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>0</td>
<td>0.0032</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>0.0034</td>
<td>0.011</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>0</td>
<td>0.0015</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>

The error observed is low. The integral for the view factor consists only the geometry. The geometry considered in the sample cases consists of rectangular surfaces. Oblique surfaces are not considered and curvatures are not considered here.

6.7. Summary

The collision detection based on bounding box provides the right fidelity data for conceptual design process. The space to be occupied by the structure and wirings are not considered at this stage. Therefore the use of the bounding box approach is justified for the parametric design.

High fidelity collision detection based on surface interaction can be performed by coupling the GDL application to a high end CAD package where the implementation is available. The approach would ensure balance between accuracy and usage of computational resources.

The cell can be defined with computation based on the vertices or the cell edges for considering other domains in case of conduction emulating the finite element method or convection emulating finite volume method. Complex patterns for the grid can be formed either by mapping the indices directly to the position and orientation of the cell by means of a mathematical function or through a linked-list representation.

The grid object can also be replaced by a more conventional meshing algorithm, the computation from the top layer performing the computation directly access the cells and is independent of the grid layer. The grazing angle selection can be performed based on the material type of the component under consideration.

The radiosity method could be considered for the analysis in thermal domain and the properties. The requirement on the computational resources must be considered in a development of framework. The integration scheme for evaluation of the form factor should be converted to Gauss-Legendre if curved surfaces are to be considered in order to reduce the computational effort. \[\] reported that the computational effort is reduced and error is not reduced.
Chapter 7

7. Results and discussion

7.1. Introduction

The UAV fuselage and systems design framework was developed in Matlab® and GDL was used for the development of a knowledge based engineering engine. The framework based was developed using object oriented programming in matlab which consequently reduced the effort for translating the objects and shapes to the KBE application.

The system components were selected a priori using the system engineering approach for aircraft design process. The components were selected based on the requirements for the ‘Search and rescue challenge’ mission. The details of the components are provided in the appendix B. The components were modelled in the KBE application as wireframe objects. Wireframes were chosen to reduce the computational effort.

Upgrading the functions and incorporating new methodologies were also simplified by using the object oriented programming. OOP simplified the representation of the design framework to a user.

The figure below shows the depiction of a class definition in Matlab.
The previous figure shows the class representation. The variables belonging to the class are given within the ‘properties’ block. The functions that belong to the class are specified in the ‘methods’ block. The generic functions which access the variables defined for the class are given in the ‘methods (static)’ block.

The activity diagram shows the various events involved in the preliminary sizing. Aerodynamic analysis was not performed in the design framework. Therefore, the wing shape was not considered in the framework. Drag polar was determined using only an empirical relationship.

The selection of the propulsion system was performed by considering the propeller as actuator disc. This is due to the fact analysis could not be performed at low Reynolds number. Unavailability of propeller charts for the commercially available model propellers was also causing a bottle neck.

Figure 39: Activity diagram depicting initial sizing

The analysis of the wing shape could be considered for future developments. The framework was conceptualized to work along with current development in the design and engineering engine. The wing loading – power loading selection is performed by computing the equations as a minimax problem. The fuselage shape was obtained from class-shape transformation. Drag prediction could be considered for further developments.
The class diagram is depicted below shows the classes invoked for the development of the framework. The classes are defined in Matlab. The motor, battery and propeller were adopted from the commercial available components. The class – condition describes the atmospheric properties at the specified altitude. The method MDI described in the Component class computes the surface temperature on the participating components. The propeller was considered as actuator disc even though an attempt was made to depict the profile shapes and to perform aerodynamic analysis.

The result of the preliminary sizing is given in the figure below. The range was 2 nautical miles and the endurance is 10 minutes. The framework computes the aspect ratio to its upper limit 10, as the module lacks aerodynamic inputs during the runtime. Also, the weights are not updated during the preliminary sizing of the unmanned aerial vehicle.

![Class diagram for the framework](image)

**Table 1**

<table>
<thead>
<tr>
<th>Name</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>U1</td>
<td></td>
</tr>
<tr>
<td>W</td>
<td>9.7706</td>
</tr>
<tr>
<td>W_st</td>
<td>5.8506</td>
</tr>
<tr>
<td>W_sys</td>
<td>3.92</td>
</tr>
<tr>
<td>AR</td>
<td>10</td>
</tr>
<tr>
<td>E</td>
<td>0.85</td>
</tr>
<tr>
<td>Cd_0</td>
<td>0.03</td>
</tr>
<tr>
<td>CL_max</td>
<td>1.1828</td>
</tr>
<tr>
<td>Vcr</td>
<td>15</td>
</tr>
<tr>
<td>C</td>
<td>3</td>
</tr>
<tr>
<td>E</td>
<td>0.1667</td>
</tr>
<tr>
<td>R</td>
<td>3704</td>
</tr>
<tr>
<td>Vs</td>
<td>10</td>
</tr>
</tbody>
</table>
Results and discussion

Table 12: Results - Preliminary sizing

<table>
<thead>
<tr>
<th></th>
<th>Value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>$W_{by_S}$</td>
<td>72.449</td>
<td>N/m²</td>
</tr>
<tr>
<td>$W_{by_P}$</td>
<td>0.1865</td>
<td>N/W</td>
</tr>
<tr>
<td>$S$</td>
<td>0.1349</td>
<td>m²</td>
</tr>
<tr>
<td>$P$</td>
<td>52.3983</td>
<td>W</td>
</tr>
</tbody>
</table>

Figure 41: minimax formulation for the wing loading - power loading

Figure 5 shows the graph for the minimax formulation of the wing loading – power loading problem. It must be noted that the variation of drag and weight of the structure due to increase in aspect ratio is not considered in the design framework. The aspect ratio reaches the maximum limit specified for the aspect ratio. It could be seen that the power loading increases between iterations 20 to 40, when the aspect ratio is increased to the maximum value. The value of the power loading decreases once the wing loading is increased in order to reduce the power loading for the climb phase.
<table>
<thead>
<tr>
<th>Mission</th>
<th>Power (W)</th>
<th>Energy (J)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Climb</td>
<td>80.7</td>
<td>2.69e03</td>
</tr>
<tr>
<td>Cruise</td>
<td>42.43</td>
<td>6.43e03</td>
</tr>
<tr>
<td>Loiter</td>
<td>42.43</td>
<td>2.54e03</td>
</tr>
</tbody>
</table>

Table 13: Mission requirements

The power and energy requirements for the climb cruise and loiter mission segments are provided in the table above. In general, only thrust and power are considered for the aircraft design. However for the case of electric powered unmanned aerial vehicle design, it is essential to consider the energy required as the equivalent energy must be supplied by the battery.

Batteries from the manufacturer – ‘Polyquest’ was considered for the design selection. It could be seen that LP series with discharge capacity 16C having a higher ordinate value for the given abscissa. However PQ-4350XP was selected for the mission having a discharge capacity of 25C.

The aerial platform considered for test case was Multiplex Ezstar. The fuselage was generated in GDL–MMG. The fuselage is segmented into adapters and trunks. The three trunk regions of the fuselage are shown in the diagram below. The trunks are created by the user using the Matlab® GUI. The adapters are not shown in the diagram.
Results and discussion

Figure 43: Trunks for the design of fuselage

Figure 44: Fuselage shape (left), fuselage with components housed (right)

Figure 45: Components housed inside fuselage
The wireframe structure of the fuselage and the components housed within are depicted in the figures above. The VRML view of the fuselage is shown on the top left corner.

<table>
<thead>
<tr>
<th>Components</th>
<th>Total_mass</th>
<th>Cm-x</th>
<th>Cm-y</th>
<th>Cm-z</th>
</tr>
</thead>
<tbody>
<tr>
<td>16</td>
<td>574.64</td>
<td>-</td>
<td>-</td>
<td></td>
</tr>
</tbody>
</table>

Table 14: Component positions

<table>
<thead>
<tr>
<th>Components</th>
<th>l-xx</th>
<th>l-xy</th>
<th>l-xz</th>
<th>l-yy</th>
<th>l-yz</th>
<th>l-zx</th>
<th>l-zy</th>
<th>l-zz</th>
</tr>
</thead>
<tbody>
<tr>
<td>16</td>
<td>2.14E+07</td>
<td>212740.3</td>
<td>2530581</td>
<td>212740.3</td>
<td>363317.5</td>
<td>-250691</td>
<td>2530581</td>
<td>-250691</td>
</tr>
</tbody>
</table>

Table 15: Component Inertias
Results and discussion

<table>
<thead>
<tr>
<th>Number-of-collisions</th>
<th>1</th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Component1</td>
<td>Component2</td>
<td>lb-x</td>
<td>lb-y</td>
<td>lb-z</td>
<td>ub-x</td>
<td>ub-y</td>
</tr>
<tr>
<td>Motor</td>
<td>PropDisc</td>
<td>-1087</td>
<td>-14</td>
<td>86</td>
<td>-1087</td>
<td>14</td>
</tr>
</tbody>
</table>

**Table 16: Collision detection**

The tables above shows the physical properties and location of the components. The report is generated by multi model generator developed in GDL. The data is processed by the framework. This forms the input required for the flight mechanics model.

![Figure 46: Participating components and surfaces](image)

The figure above shows the components interacting in the thermal domain. The view factors are formed into a matrix which is depicted in the table below.

No-of-surfaces: 6

<table>
<thead>
<tr>
<th></th>
<th>0</th>
<th>0.0024</th>
<th>0</th>
<th>0</th>
<th>0</th>
<th>0</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0.0042</td>
<td>0.0129</td>
<td>0.0049</td>
<td>0.0089</td>
<td></td>
</tr>
<tr>
<td>0.0188</td>
<td>0.0074</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>0</td>
<td>0.0055</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>0</td>
<td>0.0087</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>0</td>
<td>0.0038</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td></td>
</tr>
</tbody>
</table>

**Figure 47: Form factors**

7.2. Evaluation of tools

Matlab® and Genworks® GDL were used for the development of the framework. The algorithm for the benchmark is shown in appendix E. The benchmark test is described in [ref] generates the coefficients for a 100 term polynomial and evaluates the 100 term polynomial and iterated for 500000 times. The same test is performed by evaluating the polynomial using a function call.

<table>
<thead>
<tr>
<th>Language</th>
<th>Single body (s)</th>
<th>With call (s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Matlab*</td>
<td>1.507*</td>
<td>154.063**</td>
</tr>
<tr>
<td>GDL*</td>
<td>20.229**</td>
<td>20.742**</td>
</tr>
</tbody>
</table>

*Averaged for 10 runs*
** Average for 3 runs

The overhead involved in generating the workspace in Matlab is not considered. The time for computing increased from 1.507 seconds to 38.228 seconds for single body computation in Matlab, when the output is echoed to the command window.

The benchmarking was performed in order to get the report written in order to get this done in a simple format rather than as a tougher measure.

7.3. Recommendations

The multi-model generator can be extended to depict the wing and the tail structures. The deflection of the primary and secondary control surfaces can be generated as components can be moved using Eulerian rotation matrices.

A comprehensive wind tunnel testing of propellers for small uavs and development of propeller charts for the propellers are highly desirable for the selection of the propellers for the design. Evaluation of the drag polar accurately can increase the overall fidelity of the design framework.

The depiction of the fuselage surface in the AC3D format can enable the shape to be represented in the flight mechanics model.

Larger and complex radiative heat transfer problems could be solved by subsequent enhancement of the MDI module.


11. Motocalc.


13. FMS.


Weight | $W$ | Maximum takeoff weight  
| $W_p$ | Battery weight  
| $W_m$ | Motor weight  
| $W_p$ | Payload weight  
| $W_{eb}$ | Empty weight  
| $W_{st}$ | Structural weight  
| $W_{sys}$ | Systems weight  
| $W_{pu}$ | Propulsion weight  

Endurance | $E$  

Density | $\rho$  

Stall velocity | $V_s$  

Maximum lift coefficient | $C_{l_{max}}$  

Wing loading – stall | $(W/S)_{stall}$  

Takeoff distance | $s_{to}$  

Ground roll distance | $s_{to}$  

Take off distance | $s_{to}$  

Wing loading - takeoff | $(W/S)_{to}$  

Power loading – takeoff | $(W/P)_{to}$  

Acceleration due to gravity | $g$  

Takeoff distance |  

Distance – obstacle to flare | $s_{fl}$  

Free roll distance | $s_{fl}$  

Ground roll | $s_{fl}$  

Wing loading - landing | $(W/S)_{l}$  

Power loading – landing | $(W/P)_{l}$  

Thrust | $T$  

Power | $P$  

Rolling friction | $\mu_r$  

Velocity -approach | $V_a$  

Approach angle | $\theta_a$  

Landing distance | $s_l$  

Drag coefficient | $C_{D}$  

Drag coefficient – Zero lift | $C_{D_{0}}$
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\delta C_{D,0}$</td>
<td>Increment in base drag</td>
</tr>
<tr>
<td>$C_L$</td>
<td>Lift coefficient</td>
</tr>
<tr>
<td>$k$</td>
<td></td>
</tr>
<tr>
<td>$A$</td>
<td>Aspect ration</td>
</tr>
<tr>
<td>$e$</td>
<td>Ostwald efficiency</td>
</tr>
<tr>
<td>$c$</td>
<td>Climb</td>
</tr>
<tr>
<td>$c/N$</td>
<td>Climb gradient</td>
</tr>
<tr>
<td>$\eta_p$</td>
<td>Propulsive efficiency</td>
</tr>
<tr>
<td>$(W/S)_{climb}$</td>
<td>Wing loading during climb</td>
</tr>
<tr>
<td>$(W/S)_{climb \ rate}$</td>
<td>Wing loading specified climb rate</td>
</tr>
<tr>
<td>$C_{L,climb}$</td>
<td>Lift coefficient climb</td>
</tr>
<tr>
<td>$C_{D,climb}$</td>
<td>Drag coefficient</td>
</tr>
<tr>
<td>$(W/P)_{climb}$</td>
<td>Power loading</td>
</tr>
<tr>
<td>$(W/P)_{climb \ rate}$</td>
<td>Power loading specified climb rate</td>
</tr>
<tr>
<td>$P$</td>
<td>Power</td>
</tr>
<tr>
<td>$S$</td>
<td>Wing area</td>
</tr>
<tr>
<td>$n_{b_s}$</td>
<td>Number of batteries – series</td>
</tr>
<tr>
<td>$n_{b_p}$</td>
<td>Number of batteries - parallel</td>
</tr>
<tr>
<td>$c_b$</td>
<td></td>
</tr>
<tr>
<td>$C_b$</td>
<td></td>
</tr>
<tr>
<td>$I_{b_i}$</td>
<td></td>
</tr>
<tr>
<td>$L_b$</td>
<td>Length - battery</td>
</tr>
<tr>
<td>$B_b$</td>
<td>Breadth - battery</td>
</tr>
<tr>
<td>$H_b$</td>
<td>Height - battery</td>
</tr>
<tr>
<td>$W_b$</td>
<td>Weight - battery</td>
</tr>
<tr>
<td>$v_b$</td>
<td></td>
</tr>
<tr>
<td>$r_b$</td>
<td></td>
</tr>
<tr>
<td>$v_{b_t}$</td>
<td></td>
</tr>
<tr>
<td>$c_{b_t}$</td>
<td></td>
</tr>
<tr>
<td>$t_{b_i}$</td>
<td></td>
</tr>
<tr>
<td>$W_{b_t}$</td>
<td>Weight</td>
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<td>$I_{m_o}$</td>
<td>Current</td>
</tr>
<tr>
<td>$v_m$</td>
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<tr>
<td>$r_m$</td>
<td>Resistance</td>
</tr>
<tr>
<td>$K_{\nu_m}$</td>
<td>Speed constant</td>
</tr>
<tr>
<td>RPM$_m$</td>
<td>Revolutions per minute</td>
</tr>
<tr>
<td>$\eta_m$</td>
<td>Motor efficiency</td>
</tr>
<tr>
<td>$I_m$</td>
<td></td>
</tr>
<tr>
<td>$W_m$</td>
<td>Weight of motor</td>
</tr>
<tr>
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<td></td>
</tr>
<tr>
<td>$H_m$</td>
<td>Height of motor</td>
</tr>
<tr>
<td>$D_m$</td>
<td>Diameter of motor</td>
</tr>
<tr>
<td>$D_p$</td>
<td>Propeller diameter</td>
</tr>
<tr>
<td>$p_p$</td>
<td>Propeller pitch</td>
</tr>
<tr>
<td>$v_{des}$</td>
<td>Velocity design</td>
</tr>
<tr>
<td>Symbol</td>
<td>Description</td>
</tr>
<tr>
<td>---------</td>
<td>------------------------------</td>
</tr>
<tr>
<td>( n_p )</td>
<td>Number of blades</td>
</tr>
<tr>
<td>B</td>
<td></td>
</tr>
<tr>
<td>T</td>
<td>Thrust</td>
</tr>
<tr>
<td>( \eta_p )</td>
<td>Propeller efficiency</td>
</tr>
<tr>
<td>( C_q )</td>
<td>Torque coefficient</td>
</tr>
<tr>
<td>( C_t )</td>
<td>Thrust coefficient</td>
</tr>
<tr>
<td>( C_p )</td>
<td>Power coefficient</td>
</tr>
<tr>
<td>J</td>
<td>Advance ratio</td>
</tr>
<tr>
<td>a</td>
<td>Inflow factor</td>
</tr>
<tr>
<td>( T_c )</td>
<td>Trust coefficient</td>
</tr>
<tr>
<td>( \Omega )</td>
<td></td>
</tr>
<tr>
<td>( \beta_i )</td>
<td></td>
</tr>
<tr>
<td>( \alpha_i )</td>
<td></td>
</tr>
<tr>
<td>( \phi_i )</td>
<td></td>
</tr>
<tr>
<td>( c_i )</td>
<td></td>
</tr>
<tr>
<td>( r_i )</td>
<td></td>
</tr>
<tr>
<td>( a_i )</td>
<td></td>
</tr>
<tr>
<td>( c_{l_i} )</td>
<td></td>
</tr>
<tr>
<td>( c_{d_i} )</td>
<td></td>
</tr>
<tr>
<td>( P )</td>
<td>Power</td>
</tr>
</tbody>
</table>

**Component**

<table>
<thead>
<tr>
<th>Component</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>( x_{cg} )</td>
<td>CG Position in x</td>
</tr>
<tr>
<td>( y_{cg} )</td>
<td>CG Position in y</td>
</tr>
<tr>
<td>( z_{cg} )</td>
<td>CG Position in z</td>
</tr>
</tbody>
</table>
## Appendix B

<table>
<thead>
<tr>
<th>System</th>
<th>Model</th>
<th>Dimension</th>
<th>Mass</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Autopilot</td>
<td>ArduPilot</td>
<td>47 mm x 30mm x 17mm</td>
<td>25 g (assumed)</td>
<td>Sensors: Thermopiles, GPS, Airspeed, voltage Pressure sensor: 10 - 60 degree temperature compensated</td>
</tr>
<tr>
<td>Thermopile</td>
<td>Autopilot XYZ Horizon Sensor</td>
<td>Dimensions available</td>
<td>20+10 g (assumed)</td>
<td>Doc attached</td>
</tr>
<tr>
<td>Radio control</td>
<td>Futaba - R149DP/Futaba - R6008HS 8-Channel Receiver</td>
<td>55.58 mm x 33.02 mm x 20.32 mm</td>
<td>45 mm x 22 mm x 13 mm</td>
<td>38.8 g</td>
</tr>
<tr>
<td>---------------------</td>
<td>-----------------------------------------------------</td>
<td>---------------------------------</td>
<td>-----------------------</td>
<td>--------</td>
</tr>
<tr>
<td>IR camera</td>
<td>Indigo systems Tiny IR</td>
<td>34.3 mm x 36.8 mm x 48.3 mm</td>
<td>113 g</td>
<td></td>
</tr>
<tr>
<td>Radio</td>
<td>XBee Pro</td>
<td>available</td>
<td>25 g (assumed)</td>
<td>900 MHz, 6 mi range (based on requirement)</td>
</tr>
<tr>
<td>---------------</td>
<td>----------</td>
<td>-----------</td>
<td>----------------</td>
<td>------------------------------------------</td>
</tr>
<tr>
<td>video transmitter</td>
<td>FPV 5.8 GHz</td>
<td>assumed</td>
<td>50g (assumed)</td>
<td>5.8GHz (based on requirement)</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>----------</td>
<td>------------------</td>
<td>-----</td>
<td>-----</td>
<td></td>
</tr>
<tr>
<td>video camera</td>
<td>FPV CCD</td>
<td>20 x 20</td>
<td>22 g</td>
<td></td>
</tr>
<tr>
<td>Motor speed controller</td>
<td>Multicont X16</td>
<td>27 mm x 20 mm x 8 mm</td>
<td>27 g</td>
<td>2 Lipo, Imax = 16</td>
</tr>
<tr>
<td>Component</td>
<td>Model</td>
<td>Dimensions</td>
<td>Weight</td>
<td>Remarks</td>
</tr>
<tr>
<td>------------</td>
<td>--------------------</td>
<td>--------------------</td>
<td>--------------</td>
<td>-----------------------------</td>
</tr>
<tr>
<td>GPS</td>
<td>EM 406 GPS 1HZ</td>
<td>30 mm x 30 mm x 10.5 mm</td>
<td>20g (assumed)</td>
<td>44 mA - power consumption,</td>
</tr>
<tr>
<td>Servo</td>
<td>Multiplex Tiny S</td>
<td>30 mm x 12 mm x 30 mm x 33 mm</td>
<td>17g</td>
<td>(existing)</td>
</tr>
<tr>
<td></td>
<td>Multiplex micro</td>
<td></td>
<td>28.1 g</td>
<td>(proposed, digital)</td>
</tr>
<tr>
<td></td>
<td>speed mc/v2</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Payload</td>
<td></td>
<td>50 mm x 40 mm x 50 g</td>
<td>settable</td>
<td></td>
</tr>
<tr>
<td>30mm</td>
<td>Motor</td>
<td>To be obtained from database</td>
<td>To be obtained from database</td>
<td>To be determined from database</td>
</tr>
<tr>
<td>-----------------------</td>
<td>-----------------------------</td>
<td>------------------------------</td>
<td>------------------------------</td>
<td>-------------------------------</td>
</tr>
<tr>
<td>Battery</td>
<td>To be obtained from database</td>
<td>To be obtained from database</td>
<td>To be determined from database</td>
<td></td>
</tr>
<tr>
<td>Propeller</td>
<td>To be obtained from database</td>
<td>To be obtained from database</td>
<td>To be determined from database</td>
<td></td>
</tr>
</tbody>
</table>
Appendix C

Figure 48: Mission area
<table>
<thead>
<tr>
<th>Boundary Point</th>
<th>South (WGS 84 Degrees:Minutes:Seconds)</th>
<th>East (WGS 84 Degrees:Minutes:Seconds)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mission Boundary</td>
<td></td>
<td></td>
</tr>
<tr>
<td>MB-1</td>
<td>26° 34' 10.4&quot;</td>
<td>151° 50' 14.5&quot;</td>
</tr>
<tr>
<td>MB-2</td>
<td>26° 34' 11.8&quot;</td>
<td>151° 50' 21.9&quot;</td>
</tr>
<tr>
<td>MB-3</td>
<td>26° 34' 36.6&quot;</td>
<td>151° 50' 28.1&quot;</td>
</tr>
<tr>
<td>MB-4</td>
<td>26° 34' 38.3&quot;</td>
<td>151° 50' 25.2&quot;</td>
</tr>
<tr>
<td>MB-5</td>
<td>26° 34' 53.4&quot;</td>
<td>151° 50' 31.0&quot;</td>
</tr>
<tr>
<td>MB-6</td>
<td>26° 34' 42.5&quot;</td>
<td>151° 50' 50.9&quot;</td>
</tr>
<tr>
<td>MB-7</td>
<td>26° 35' 58.2&quot;</td>
<td>151° 51' 10.4&quot;</td>
</tr>
<tr>
<td>MB-8</td>
<td>26° 36' 33.2&quot;</td>
<td>151° 52' 20.1&quot;</td>
</tr>
<tr>
<td>MB-9</td>
<td>26° 38' 43.7&quot;</td>
<td>151° 52' 55.4&quot;</td>
</tr>
<tr>
<td>MB-10</td>
<td>26° 38' 36.9&quot;</td>
<td>151° 52' 49.3&quot;</td>
</tr>
<tr>
<td>MB-11</td>
<td>26° 36' 15.4&quot;</td>
<td>151° 50' 03.9&quot;</td>
</tr>
<tr>
<td>Search Area</td>
<td></td>
<td></td>
</tr>
<tr>
<td>SA-1</td>
<td>26° 37' 02.8&quot;</td>
<td>151° 50' 35.5&quot;</td>
</tr>
<tr>
<td>SA-2</td>
<td>26° 37' 06.0&quot;</td>
<td>151° 51' 20.1&quot;</td>
</tr>
<tr>
<td>SA-3</td>
<td>26° 38' 16.2&quot;</td>
<td>151° 51' 05.7&quot;</td>
</tr>
<tr>
<td>SA-4</td>
<td>26° 36' 19.8&quot;</td>
<td>151° 50' 56.8&quot;</td>
</tr>
<tr>
<td>SA-5</td>
<td>26° 28' 14.6&quot;</td>
<td>151° 50' 21.8&quot;</td>
</tr>
<tr>
<td>Entry/ Exit Lanes</td>
<td></td>
<td></td>
</tr>
<tr>
<td>EL-1</td>
<td>26° 36' 26.0&quot;</td>
<td>151° 50' 43.4&quot;</td>
</tr>
<tr>
<td>EL-2</td>
<td>26° 37' 03.0&quot;</td>
<td>151° 50' 36.8&quot;</td>
</tr>
<tr>
<td>EL-3</td>
<td>26° 37' 06.1&quot;</td>
<td>151° 50' 59.2&quot;</td>
</tr>
<tr>
<td>EL-4</td>
<td>26° 36' 29.1&quot;</td>
<td>151° 51' 05.9&quot;</td>
</tr>
<tr>
<td>Loss of Data Link</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Airfield Home</td>
<td>26° 35' 05.2&quot;</td>
<td>151° 50' 32.4&quot;</td>
</tr>
<tr>
<td>Comm Hold</td>
<td>26° 36' 26.0&quot;</td>
<td>151° 50' 43.4&quot;</td>
</tr>
</tbody>
</table>

**Figure 49: Mission area - GPS Co-ordinates**
Sample – Trunk definition file

section-list
("frame1.dat" "frame1.dat" "frame1.dat" "frame1.dat")

frame-position-list
(18.207300 184.407100 350.606900 516.806700)

Dt-list
(5.970154 42.553287 59.644618 65.671641)

Db-list
(-16.719031 -38.146337 -53.840360 -64.248449)

Dsx-list
(5.681820 25.133497 40.605972 52.186668)

Dsy-list
(-5.970150 -14.745563 -20.438143 -23.371531)
Matlab

% Benchmarking for matlab based on the algorithm used in
% http://dan.corlan.net/bench.html

function testpol2
tic
n = 500000;
x = 0.2;
pu = 10.0;
%global mu;
mu = 10.0;

for i =1:n
    [mu,su] = dopoly(x,mu);
    pu = pu + su;
end

pu
toc
end

function [mu,su] = dopoly(x,mu)

for j =1:100
    mu = (mu + 2.0) / 2.0;
    pol(j) = mu;
end

su=0.0;

for j=1:100
    su = x *su +pol(j);
end
end

GDL

(in-package :gdl-user)

(define-object testpol (base-object)
    :input-slots
((su 0.0)
 (mu 10.0)
 (x 0.2)
 (n 500000))

:computed-slots

((pu (the (eval-pol 500000 0.2)))
 (t (time (the (eval-pol 500000 0.2)))))
)

:objects

()

:functions

((eval-pol (n x) (do ((i 1) (val 0) (pol (make-array 100)) (mu 10.0) (su 0.0))
  ((> i n) val pol mu su)
  (do ((j 0))
      ((> j 99))
      (setq mu (/ (+ mu 2.0) 2.0))
      (setf (aref pol j) mu)
      (setq j (+ j 1)))
   ; (setq su 0)
   (do ((j 0))
       ((> j 99))
       (setq su (+ (aref pol j) (* su x)))
       (setq j (+ j 1)))
   (setq val (+ su val))
   (setq i (+ i 1))))
)
