The Meridian UAS: Detailed Design Review


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# Table of Revisions

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Executive Summary

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# Table of Contents

Nomenclature .................................................................................................................. vii
Abbreviations ................................................................................................................. viii
1 Introduction............................................................................................................... 1
2 A Note on Nomenclature .......................................................................................... 1
3 Summary of Current Design.................................................................................... 1
4 Requirements Summary........................................................................................... 5
5 Aircraft Structural Design ....................................................................................... 6
  5.1 Pertinent Requirements ....................................................................................... 7
  5.2 Wing .................................................................................................................... 8
    5.2.1 Preliminary Trade Studies........................................................................... 9
    5.2.2 Final Structural Arrangement ................................................................. 10
    5.2.3 Detailed Structural Analysis ................................................................... 13
    5.2.4 Material Selections and Manufacturing Plan............................................ 20
  5.3 Fuselage Structural Design ............................................................................... 23
    5.3.1 Preliminary Trade Studies......................................................................... 23
    5.3.2 Final Structural Arrangement ................................................................... 24
    5.3.3 Strength Analysis of the Center Fuselage Skins....................................... 26
    5.3.4 Strength Analysis of the Firewall ............................................................. 29
    5.3.5 Strength Analysis of the Payload Hatch ................................................... 30
    5.3.6 Structural Design of the V-tail................................................................. 32
    5.3.7 Dynamic Analysis of the Whole Aircraft ................................................. 32
    5.3.8 Material Selections and Manufacturing Plan............................................ 33
    5.3.9 Fuselage Manufacturing Costs................................................................. 36
  5.4 Control Surfaces................................................................................................ 38
    5.4.1 Ailerons ..................................................................................................... 38
    5.4.2 Flaps .......................................................................................................... 39
    5.4.3 Ruddervators............................................................................................. 40
6 Flight Control System............................................................................................. 40
  6.1 Pertinent Requirements ....................................................................................... 40
  6.2 Ground Station Design........................................................................................ 40
    6.2.1 Ground Station Unit .................................................................................. 40
    6.2.2 Operator Interface PC ............................................................................... 41
    6.2.3 Futaba Console .......................................................................................... 42
    6.2.4 Operator Software Interface ...................................................................... 42
  6.3 Onboard System................................................................................................ 43
    6.3.1 Indicated Airspeed Control ....................................................................... 43
    6.3.2 Altitude Control ........................................................................................ 43
    6.3.3 Turn Rate Control ..................................................................................... 44
    6.3.4 Pitch Damper Control ............................................................................... 44
    6.3.5 Yaw Damper Control ............................................................................... 44
    6.3.6 Turn Compensator .................................................................................... 44
    6.3.7 Heading Tracker ........................................................................................ 44
    6.3.8 Ruddervator Mixing .................................................................................. 44
  6.4 Navigation System............................................................................................... 45
6.4.1 Waypoints Navigation ................................................................. 45
6.5 Actuator Sizing ................................................................................. 46
7 Communications System ........................................................................ 48
  7.1 Pertinent Requirements ................................................................. 48
  7.2 System Description ................................................................... Error! Bookmark not defined.
  7.3 Airborne Communications .......................................................... 48
    7.3.1 Power and Dimensions for Avionics suit .............................. 52
  7.4 Onboard Communications ......................................................... Error! Bookmark not defined.
  7.5 Ground Station ............................................................................. 53
    7.5.1 Piccolo Ground Station ......................................................... 54
    7.5.2 ACRA KAM-500 Ground Station .......................................... 54
8 Propulsion System ............................................................................... 55
  8.1 Pertinent Requirements ................................................................. 55
  8.2 Engine Selection ............................................................................ 56
  8.3 Structural Integration .................................................................... 57
  8.4 Propeller Selection ....................................................................... 57
  8.5 Intake and Exhaust Design ............................................................ 57
  8.6 Autopilot Interface ........................................................................ 58
  8.7 Fuel System ................................................................................... 58
9 Landing Gear System ........................................................................... 58
  9.1 Structural Integration .................................................................... 59
10 Anti-Icing System ............................................................................. 61
11 Electrical Power Distribution System ............................................... 61
  11.1 Onboard Power System ............................................................ Error! Bookmark not defined.
  11.2 Ground Operations ................................................................... Error! Bookmark not defined.
12 Payload System Integration ............................................................... 63
  12.1 Electrical Interface ..................................................................... 63
  12.2 Structural Integration ................................................................ 63
    12.2.1 Primary Payload ............................................................... 64
    12.2.2 Secondary Payload ............................................................ 64
    12.2.3 Magnetometer ................................................................. Error! Bookmark not defined.
    12.2.4 Antennas .......................................................................... 65
13 Weight and Balance ............................................................................ 66
14 Project Schedule ................................................................................ 66
15 Conclusions and Future Work ........................................................... 66
16 References .......................................................................................... 67
Appendix A. Aircraft Tail Loads ............................................................. A-1
List of Figures

Figure 3.1 – Comparison of the Meridian with Innodyn and Thielert Engines .................. 3
Figure 3.2 – Isometric View of the Meridian UAV ............................................................. 3
Figure 3.3 – Threeview of the Meridian UAS .................................................................... 4
Figure 4.1 – Design Mission Profile ................................................................................... 6
Figure 5.1 – Aircraft V-n Diagram [2] ................................................................................ 8
Figure 5.2 – Proposed Wing Spar Configurations [xxx] ..................................................... 9
Figure 5.3 – Meridian Wing Structural Arrangement ....................................................... 10
Figure 5.4 – Wing Inboard Geometry .............................................................................. 11
Figure 5.5 – Wing Outboard Geometry ........................................................................... 11
Figure 5.6 – Wing Outboard Geometry (Left View) ....................................................... 12
Figure 5.7 – Wing Outboard Spar Detail ........................................................................... 12
Figure 5.8 – Wing Splice Geometry .................................................................................. 13
Figure 5.9 – Control Surface Integration ......................................................................... 13
Figure 5.10 – Maximum Deflection and Stress of the Wing [xxx] .................................. 14
Figure 5.11 – Wing Finite Element Geometry .................................................................. 15
Figure 5.12. Geometry and finite element model of in-board wing spar caps ................. 16
Figure 5.13. Stresses of in-board wing spar caps .............................................................. 17
Figure 5.14. First wing buckling mode ............................................................................. 18
Figure 5.15. Compression stresses in the in-board wing spars ......................................... 19
Figure 5.16. The first bending mode and the first torsion mode of the wing .................... 20
Figure 5.17 – Wing Spars ................................................................................................. 21
Figure 5.18 – Wing Rib Geometry .................................................................................... 21
Figure 5.19 – Wing Skin Geometry .................................................................................. 22
Figure 5.20 – Fuselage Finite Element Model .................................................................. 23
Figure 5.21. Schematic of aft fuselage skin panel ............................................................ 24
Figure 5.22 – Fuselage Geometry ..................................................................................... 25
Figure 5.23 – Fuselage Structural Arrangement (Isometric View). ................................. 26
Figure 5.24 – Fuselage Structural Arrangement (Right View) ....................................... 26
Figure 5.25. Geometry and finite element model of the center fuselage skin ................. 27
Figure 5.26. Stresses of center fuselage skin of ply 1. maximum principal stress .......... 29
Figure 5.27. Geometry and finite element model of the firewall ...................................... 30
Figure 5.28. Stresses of the firewall .................................................................................. 30
Figure 5.29 – Stresses for the Hatch for Critical Wing Loads .......................................... 31
Figure 5.30. Stresses of the hatch for V-tail secondary dive loads ..................................... 31
Figure 5.31. The first ten modes of the whole aircraft ..................................................... 33
Figure 5.32 – Fuselage Manufacturing Breakdown ............................................................ 35
Figure 5.33 – Lower Fuselage Skin Tooling ..................................................................... 36
Figure 5.34 – Manufacturing Plan for the Fuselage .......................................................... 37
Figure 5.35 – Aileron Structural Layout .......................................................................... 39
Figure 5.36 – Flap Track Geometry .................................................................................. 39
Figure 6.1 Ground Station Setup [1] ................................................................................. 40
Figure 6.2 Ground Station Unit Front Panel [1] ............................................................... 41
Figure 6.3 Telemetry Page of the Operator Interface [1] .................................................. 43
Figure 6.4 – Ruddervator Mixing [1] ................................................................. 44
Figure 6.5 – Operator Interface Navigation Page [1] ........................................... 45
Figure 7.1: Piccolo II Autopilot (left) and Iridium A3LA-D Modem .................. 48
Figure 7.2: AMPRO Ready Board 800 Single Board Computer ......................... 49
Figure 7.3: Nav-420 Inertial Measurement Unit .................................................. 49
Figure 7.4: ACRA Controls KAM-500 Data Acquisition System (left), E/S Band
Airborne Telemetry 5W Transmitter (middle), and E/S Band Telemetry Receiver (right)
................................................................................................................................... 50
Figure 7.5: Diagram of Complete Avionics System ............................................. 51
Figure 7.6: T2000UAV Transponder ................................................................... 51
Figure 7.7: EBC 102A Emergency Locator Transmitter ...................................... 52
Figure 7.8: Lumi Star Portable Ground Station for the KAM-500 ......................... 54

Figure 8.1 Innodyn 165TE Turbo-propeller  Figure 8.2 Centurion 1.7 Diesel
Engine ....................................................................................................................... 56
Figure 8.3 – Thielert Centurion 2.0 Firewall Forward Kit ...................................... 57
Figure 8.4 – Fuel System Schematic ..................................................................... 58
Figure 9.1 – Propeller Clearance ........................................................................... 59
Figure 9.2 – Landing Gear Structural Integration .................................................. 60
Figure 9.3 – Landing Gear Pod Geometry ............................................................... 60
Figure 12.1 – Payload Bay Definitions ................................................................... 63
Figure 12.2 – Payload Structural Integration .......................................................... 64
Figure 12.3 – Secondary Payload Bay Geometry .................................................... 65
Figure 12.4 – Antenna Lateral Clearance ............................................................... 66

List of Tables

Table 3.1 – Meridian UAV Geometry and Weights Summary ............................... 5
Table 4.1 – High-Level Requirements Summary ................................................. 6
Table 5.1 – Material selection of out-board wing skin ........................................... 10
Table 5.2 – Description of in-board wing spars ..................................................... 16
Table 5.3. Summary of in-board wing spar buckling analysis ................................ 19
Table 5.4 – Trade study of fuselage aft skin ........................................................... 24
Table 5.5. Description of the center fuselage skins .............................................. 26
Table 5.6 – Description of the firewall ................................................................. 29
Table 5.7 – Material Costs for the Fuselage Structure ......................................... 38
Table 7.1: Power Budget, Weight, and Dimensions for the Avionics System ....... 53
Table 8.1 – Published Engine Data [xxx] ............................................................... 55
## Nomenclature

<table>
<thead>
<tr>
<th>Symbol</th>
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# Abbreviations

<table>
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<tr>
<td>UAV</td>
<td>Unmanned Aerial Vehicle</td>
</tr>
<tr>
<td>UAS</td>
<td>Unmanned Aircraft System</td>
</tr>
<tr>
<td>s.f.c.</td>
<td>Specific Fuel Consumption</td>
</tr>
<tr>
<td>CReSIS</td>
<td>Center for Remote Sensing of Ice Sheets</td>
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1 Introduction

2 A Note on Nomenclature

There are several terms in the current vernacular for autonomous aircraft: UAV, UAS, UA, RPV, etc. In addition there are many definitions for these acronyms: such as Unmanned Air Vehicle, Uninhabited Air Vehicle, Uncrewed Air Vehicle, etc. The Department of Defense makes note of this problem in the DoD UAS Roadmap [xxx]. The latest iteration of buzzwords is UAS for Unmanned Aircraft System and UA for Unmanned Aircraft. This change is meant to signify the importance of system level design. The convention used throughout this document, though it may vary slightly, will be:

- UAS – Uninhabited Aircraft System
- UAV – Uninhabited Air Vehicle

This is a mix of the old and new nomenclature, but this is done as the term UA is not well recognized yet. The term UAS has been adopted to indicate the importance of system-level requirements and design consideration to the Meridian development. Also, the term Uninhabited is used over others such as unmanned or uncrewed as these give the appearance of an unmonitored aircraft.

3 Summary of Current Design

The Meridian Unmanned Aircraft System (UAS) has been under development over the last year (The requirements development and mission planning for this aircraft have been going on for over three years.) and has gone through both preliminary [1] and detailed [2] design reviews. The primary mission of the Meridian is to carry a variety of scientific payloads including an ice penetrating radar and a magnetometer in support of the Center for Remote Sensing of Ice Sheets (CReSIS). The Meridian will primarily be operated in polar regions, the first of which will be near Jakobshavn, Greenland and Thwaites Glacier in Antarctica. The operational requirements have impacted the design of the system in terms of engine selection, material selection, and landing gear design (among other aspects of the design). These design decisions are documented in [1] and [2].

The following key changes have been made to the Meridian since the critical design review:
• The primary engine has been changed from the Innodyn 165TE to the Thielert Centurion 2.0
• The structural layout and manufacturing plan have changed significantly and will be discussed
• The wing and V-tail anti-icing solutions have been changed from engine bleed air to one of two options: electro-expulsive or weeping wing (See Section 10)

The most significant change was that of moving to the Thielert diesel engine over the Innodyn turboprop engine. This decision was made based on unreliability in the testing of the Innodyn engine. This engine is a new and therefore susceptible to bugs. A series of hot starts and hot spots in the engine have delayed the engine testing, which is critical as the specific fuel consumption of the Innodyn has not been fully characterized. The Thielert engine, which was the back-up engine at the time of the CDR, has become the primary engine for the Meridian. The testing of the Innodyn will continue as planned. The Innodyn will be considered for future growth of the Meridian.

The decision to change the engine had several impacts on the design in terms of geometry, weight and balance, and performance. The net effect, however, was minimal due to the key differences in the engines. The Thielert engine weighs 330 lbs where the Innodyn weighs only 188 lbs. This, however is offset by the fact that the Innodyn has nearly double the s.f.c. of the Thielert (0.90 lbs/hp-hr and 0.36 lbs/hp-hr) respectively. The fuel required for the Meridian with the Innodyn is 295 lbs. The fuel required for the Meridian with the Thielert engine is 120 lbs, resulting in a weight savings of 175 lbs. This almost directly offsets the difference in engine weight.

The difference in engine size resulted in a significant change in fuselage geometry as depicted in Figure 3.1. This resulted in an increase in the fuselage zero-lift drag coefficient from 0.0032 (Innodyn) to 0.0039 (Thielert). The weight and balance difference between the two engines was also a manageable problem for several reasons:

• The weight of the Innodyn was overestimated at 295 lbs in the CDR to account for engine accessories, or any variation from the manufacturer’s specification as the engine is new.
• The moment arm of the Innodyn (from firewall to engine c.g.) is larger for the Innodyn than the Thielert: 28” and 21” respectively.
The current design of the Meridian is shown in Figure 3.2 and Figure 3.3. The aircraft parameters are given in Table 3.1.
Figure 3.3 – Threeview of the Meridian UAS
Table 3.1 – Meridian UAV Geometry and Weights Summary

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<tr>
<td>Power</td>
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4 Requirements Summary

The high level requirements for the Meridian UAS are shown in Table 4.1 and the design mission profile is shown in Figure 4.1. These represent the high-level requirements for the aircraft preliminary design. These requirements are further broken into more detailed derived and desired requirements for each subsystem design. These derived requirements will be discussed throughout this report.
Table 4.1 – High-Level Requirements Summary

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<td>Endurance</td>
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<td>Cruise Speed</td>
<td>100-120 kts (~180-220 km/hr)</td>
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<td>Maximum Ceiling</td>
<td>15,000 ft (~4,500 m)</td>
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<tr>
<td>Rate of Climb</td>
<td>1,600 ft/min (~490 m/min)</td>
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<tr>
<td>Takeoff Distance</td>
<td>1,500 ft (~450 m)</td>
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</tr>
<tr>
<td>Landing Distance</td>
<td>1,500 ft (~450 m)</td>
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</tr>
<tr>
<td>Payload Volume</td>
<td>20&quot; x 20&quot; x 8&quot; (~0.5 x 0.5 x 0.2 m)</td>
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<tr>
<td>Payload Weight</td>
<td>120 lbs (~55 kg)</td>
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<td>Maneuvering Requirements</td>
<td>FAR 23, where applicable</td>
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<td>Aircraft Wingspan</td>
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<tr>
<td>Aircraft Length</td>
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**Figure 4.1 – Design Mission Profile**

1. Warmup 6. Data Acquisition (120 kts @ 5,000 ft AGL)
2. Taxi 7. Cruise Return (Optimum Alt. and Speed)
3. Takeoff 8. Descent (No Range Credit)
4. Climb (No Range Credit) 9. Land/Taxi
5. Cruise Out (Optimum Alt. and Speed)

5 Aircraft Structural Design

The purpose of this section is to describe the structural design of the Meridian UAV. This includes the general structural layout, finite element analysis results, and manufacturing plans. The results of the structural sizing are:

**Strength Analysis**

Under ultimate load, all strength margins of safety are positive. As expected, stress concentrations appear when stiffness/geometry experience drastic changes, e.g., the wing-firewall joint and payload cutouts.
Stiffness Analysis

Under ultimate load, the deflection of the wing tip, calculated as the average of leading edge and trailing edge nodes, is 0.255 m (10.0 in), as shown in Error! Reference source not found., yielding a deflection angle along the wing span of 3.62°. Under ultimate load, the wing tip twisting is -0.11°. At cruise, the wing span deflection is 0.043 m, which is 0.61° span-wise, and the wing tip twisting angle is -0.02°, which is deemed acceptable.

Stability Analysis

Buckling analysis is performed on the wing, fuselage and V-tail. The wing upper skins are free of buckling up to 2.14 times cruise load, which is deemed acceptable. The aft fuselage and the V-tail are free of buckling under ultimate load.

Flutter Analysis

Flutter calculation is carried out, which gives a wing divergence speed of 444 km/hr. Meridian’s design maximum dive speed is 350 km/hr, which is less than the predicted divergence speed, even after multiplied by a factor of 1.2.

Fatigue Considerations

At cruise, the most critical stress level is 12.3 KSI in the firewall, close to its joint with the wing. It is below the fatigue limit stress of all materials used.

5.1 Pertinent Requirements

The most critical requirements for the Meridian Structural Design are:

- Strength/Stiffness. The worst case stress seen in service should not exceed any of the yield stress allowables of the materials selected. The wing tip deflection angle should not exceed 4° under ultimate load.
- Elastic and aerodynamic stability. There should be no buckling or crippling at limit load.
- Weight, cost and manufacturing time should be minimized.
- Shipping requirement. This UAV should fit in standard 20’ containers. Thus, a wing splice is required at BL 43.25”.
- Landing gear integration into the wing
- Fuel system integration
- 4 antennas (~6 lb each) on each side of aircraft. Attach points for incorporation of 2 more antennas (one near the fuselage, the other near the wing tip) are desired too.
The aircraft flight envelope is shown in Figure 5.1. The V-n diagram was used throughout the structural design process. Figure 5.1 was developed in [2].

Figure 5.1 – Aircraft V-n Diagram [2]

### 5.2 Wing

The wing of the Meridian has several design features that affect the structural arrangement. First, the landing gear is mounted to the wing. This is not uncommon, but adds a level of complexity to the wing design that will be discussed. The second design feature requires up to ten hardpoints along the span of the wing for mounting antennas. The design of these antennas is currently underway and will change throughout the life of this aircraft. For this reason some amount of robustness must be integrated into the structural design such that future modifications can be made to the type and geometry of the antennas.

The third design feature of the wing is the fact that the wing must be removable after each flight. This requirement is based on the limited facilities in both Antarctica and Greenland as well as the shipping requirements. The aircraft must be able to fit in a standard 20 foot container [xxx] for shipping. The wing splice occurs just outboard of the landing gear attachment (BL 43.25°), which complicates the structural integration at this point. This will be discussed further in this section.
5.2.1 Preliminary Trade Studies

Several preliminary trade studies were performed on the wing structural arrangement. These included concepts utilizing different materials, manufacturing strategies, and arrangements. The decision to utilize a two spar structural arrangement was made prior to Critical Design Review and is documented in [2]. The trade studies discussed in this report refer to various two-spar wing concepts including:

- Aluminum tube spars with machined ribs and aluminum skins
- Aluminum rectangular spars with aluminum ribs and skins
- Built-up sheet metal spars with aluminum ribs and skins
- Composite structure with Carbon/Epoxy (C/E) spars and skins
- Aluminum spars and ribs with non-structural skins

These as well as several other configurations were considered, but eventually narrowed to the three configurations shown in Figure 5.2. These are aluminum tube spars, aluminum rectangular spars, and built up sheet metal spars.

The circular tube spar is the most economical configuration and the easiest to assembly, but it results in an approximate wing weight of 156.6 lb, which is higher than the other two concepts. The rectangular tube spar is a little more weight efficient than the circular tube spar, yielding a wing weight of 149.5 lb. At last, a built-up C-beam configuration is analyzed, giving a wing weight of 130 lb. The built up C-channel spar configuration is chosen as it is the most structurally efficient. A Carbon Fiber Reinforced Polymer (CFRP) spar concept is explored too. Further weight saving can be achieved (about 20 lb) but resulting concerns, e.g., wing splice difficulty and tooling cost, override this option.
**Wing Skins**

For the in-board wing skin, aluminum alloy material (2024-T3 sheet) is chosen, over carbon/epoxy based on considerations of wing-fuselage and wing-landing gear integration.

For the out-board wing skin, possible weight-saving might be achieved by using carbon/epoxy wing skins. So the possibility is explored, as shown in Error! Reference source not found.. The weight-saving is about 12 lb. However, aluminum skins provide ease of antenna integration, ease of wing splice and damage tolerance at the wing splice. Combining all considerations, aluminum alloy is chosen for out-board wing skins.

<table>
<thead>
<tr>
<th>Material selection of out-board wing skin</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aluminum out-board wing skin</td>
</tr>
<tr>
<td>Carbon/Epoxy out-board wing skin</td>
</tr>
</tbody>
</table>

**5.2.2 Final Structural Arrangement**

The final wing structural arrangement is shown in Figure 5.3. The wing is comprised of an inboard and outboard section.
Figure 5.4 – Wing Inboard Geometry

Figure 5.5 – Wing Outboard Geometry
Figure 5.6 – Wing Outboard Geometry (Left View)

Figure 5.7 – Wing Outboard Spar Detail
5.2.3 Detailed Structural Analysis

Under ultimate load, the deflection of the wing tip, calculated as the average of leading edge and trailing edge nodes, is 0.255 m (10.0 in), as shown in Figure 5.10, yielding a deflection angle along the wing span of 3.62°. Under ultimate load, the wing tip twisting is -0.11°. At cruise (1g), the wing span deflection angle is 0.61° and the wing tip twisting angle is -0.02°, which is deemed
acceptable. At cruise, the wing vertical deflection is 0.14" (0.043 m), which is 0.61° span-wise, and the wing tip twisting angle is -0.02°, which is deemed acceptable.

Figure 5.10 – Maximum Deflection and Stress of the Wing
Ribs are positioned at:

- Structural attach points, e.g., antenna and landing gear attachments.
- Wing end closures.

Ribs will be hammer-formed aluminum alloy sheets. 5052-H34 sheets are used due to its good formability, except at highly-loaded regions, e.g., rib #3 (wing-fuselage integration), rib #4 and #6 (landing-gear integration) and rib #7 (wing splice). These more highly-loaded ribs will be machined from 7075-T651 plate. Ribs are positioned at 20" intervals when possible and at structural attach points when needed, as shown in Figure 5.11.

**Wing-Landing Gear Integration and Wing Splice**

The landing gear is integrated with the wing by two machined ribs, rib #4 and #6, as stated previously. For ease of attachment and detachment of the outer wing, only spars will be left to transfer all out-board loads to the in-board wing section at the splice. Thus, skins and stringers are all discontinued, as shown in Error! Reference source not found.. This makes the splice a highly-loaded part of the wing. Clevises will be machined at the ends of in-board and out-board spars, then pinned together. Also shown in Error! Reference source not found., in-board of the splice, all leading-edge and trailing-edge structures, including skins and ribs, are taken off for ease of assembly. Fairings will be added instead for aerodynamic purposes.
Various stress types and locations are checked and the most critical one is plotted.

**Table 5.2. Description of in-board wing spars**

<table>
<thead>
<tr>
<th>Material</th>
<th>Thickness (in)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Forward upper spar cap</td>
<td>7075-T651 plate</td>
</tr>
<tr>
<td>Forward lower spar cap</td>
<td>7075-T651 plate</td>
</tr>
<tr>
<td>Aft upper spar cap</td>
<td>7075-T651 plate</td>
</tr>
<tr>
<td>Aft lower spar cap</td>
<td>7075-T651 plate</td>
</tr>
</tbody>
</table>

For spar caps, the geometry and finite element model are shown in Figure 5.12. It can be seen that upper spar caps are basically in compression and lower ones in tension. The most critical stress, which is the minimum principal stress at Z2 as shown in Figure 5.13, is \(-4.29 \times 10^8\) Pascal (62.2 KSI) at the joint of element 65518 and 65505, node 55584, where the in-board forward upper wing spar meets the firewall. The Margin of safety is the same for the spar caps and webs:

\[
MS = \frac{\sigma_{allowable}}{\sigma_{critical}} - 1 = \frac{76}{62.2} - 1 = 0.22
\]
5.2.3.1 Buckling Analysis of the Wing

5.2.3.1.1 Buckling Analysis of the Wing Skins

The wing buckling is always a concern due to its bad effect on aerodynamic efficiency. The upper skin buckling is the most critical case, because the upper skin is usually under compression during most of the time of cruise. It is checked using the finite element model by NASTRAN buckling solution. Limited by memory size, the model is simplified by removing structures that do not connect directly to the wing. From previous strength analysis, the most direct load path is kept.

The buckling calculation result is shown in Figure 5.14. The first buckling mode of the in-board upper skin happens in between the stiffeners at 0.34 times of the ultimate load, which means that there will be no wing buckling at any load that is lower than 2.01 times the cruise condition. This is deemed acceptable.
5.2.3.1.2 Buckling Analysis of the Wing Spars

Because the spar caps are restricted by skins and webs, they are unlikely to have primary buckling. Secondary buckling calculations are carried out (Please see Appendix B for detailed documentation) under ultimate load, with both positive and negative load factors (+3.95/-1.52) to check the compression of both upper and lower spar caps. The in-board wing spars have the greatest compression loads and do not have structural leading edge and trailing edge skins to ease the loading. So they are chosen as the most critical component.
It is observed that all four spar caps do not have buckling at ultimate loads. For the two forward spar caps, cut-off stress is reached.

Table 5.3. Summary of in-board wing spar buckling analysis

<table>
<thead>
<tr>
<th></th>
<th>Averaged compression stress</th>
<th>Secondary buckling allowable stress</th>
<th>Margin of safety</th>
</tr>
</thead>
<tbody>
<tr>
<td>Forward upper spar</td>
<td>$1.210 \times 10^8$</td>
<td>$4.892 \times 10^8$</td>
<td>3.04</td>
</tr>
<tr>
<td>Aft upper spar</td>
<td>$0.836 \times 10^8$</td>
<td>$4.727 \times 10^8$</td>
<td>4.65</td>
</tr>
<tr>
<td>Forward lower spar</td>
<td>$1.020 \times 10^8$</td>
<td>$4.892 \times 10^8$</td>
<td>3.80</td>
</tr>
<tr>
<td>Aft upper spar</td>
<td>$0.689 \times 10^8$</td>
<td>$4.727 \times 10^8$</td>
<td>5.86</td>
</tr>
</tbody>
</table>

5.2.3.2 Flutter Analysis of the Wing

Three things are checked to ensure that Meridian will not have flutter problems:

- The commonly used criterion [3] is that the aircraft should be flutter free up to 1.2 times the design dive speed. Meridian’s design dive speed is 190kts (350 km/h), giving design critical speed of 230 kts (420 km/hr). The divergence speed of Meridian is calculated as:
\[ V_D = \sqrt{\frac{\pi^2 J}{2 \alpha c L^2}} \sqrt{\frac{G}{\rho}} = 444 \text{ km/hr (240 kts)} \] [4]

- Where \( J \) is the torsional rigidity 100525 N-m. Thus, our predicted divergence speed exceeds 1.2 times the design dive speed.

- The center of gravity of the wing is forward of the wing elastic axis by 2”.

- The first wing bending mode is at 8.5 Hz; the first wing torsion mode is at 23.8 Hz, as shown in Figure 5.16. The frequency ratio of the bending mode to the torsion mode is 2.8, compared to 2.33 of Slick 360, a manned aircraft of similar size, that have flown flutter free up to 386 km/hr.

Each of these analyses supports a flutter free prediction.

![Figure 5.16. The first bending mode and the first torsion mode of the wing](image)

### 5.2.4 Material Selections and Manufacturing Plan

#### 5.2.4.1 Wing Spars

The wing spars (outboard and inboard) will be manufactured out of 7075-T651 aluminum. The spars will be machined either by utilizing the KU machining equipment or by an outside party. The KU machining equipment cannot machine the entire spar at once as it is larger than the machine table size. This will require additional set up time to reposition the part throughout the machining process. For this reason, the spars will most likely be manufactured by an outside vendor.

The decision to machine the spars was made based on comparing the cost of the machining to the labor costs associated with manufacturing and assembling the spars out of built-up sheet metal [xxx]. The spars are shown in Figure 5.17. The spar cross sectional area tapers from root to tip in steps. The cross sectional geometry is constant in each of the pockets. The thickness of the spar caps is reduced at each of the vertical web stiffeners.
5.2.4.2 Wing Ribs

The main wing ribs are 5052-H32 aluminum. This decision was made based on the manufacturing method as the ribs are relatively lightly loaded. The ribs will be laser cut then hammer formed. This method has been utilized on the prototype wing and produced excellent results. The rib spacing is 20 inches in the outboard wing section based on the antenna spacing requirement. This spacing is satisfactory in terms of skin buckling (See detailed stress report – Appendix B).

There are various modifications to this general wing geometry for the wing closeout ribs for example.
5.2.4.3 Wing Skins

The wing skins are 2024-T3 Aluminum sheet. The upper and lower skins are split in two places: at the wing splice, and midway along the outboard wing span. The inboard skins are 0.04” thick, while the outboard skins are 0.025” thick. The wing skins comprise approximately 42 lbs of the total wing weight.

![Figure 5.19 – Wing Skin Geometry](image)

5.2.4.4 Wing Leading Edge

The wing leading edge is integrated with the anti-icing system. Currently two manufacturers of leading edge anti-icing systems are being investigated:

- CAV Aerospace Ltd.
  - Weeping wing anti-icing system
- Ice Management Systems Inc.
  - Electro-expulsive deicing system

The primary manufacturer is currently CAV Aerospace; however both of these companies will provide the anti-icing solution in the form of a leading complete leading edge. This relieves the internal manufacturing team from being required to produce the leading edge skin. Integrating the anti-icing with the leading edge skin also results in a more structurally synergistic and therefore more efficient solution.
5.3 **Fuselage Structural Design**

5.3.1 Preliminary Trade Studies

The center fuselage skins cover many loads that run through major load carrying members such as the firewall caps, payload hatch door and engine mount longerons. Monolithic composite skins are chosen for ease of manufacturing and assembly. As a result, they need to be supported by aluminum alloy substructures (frames and longerons).

On the contrary, the aft fuselage skin does not need to cover major load carrying members. So a comparison is made to choose between monolithic composite skins and honeycomb composite skins. The honeycomb composite skins have the advantage on buckling resistance. Moreover, less parts and tooling are needed for honeycomb composite skins, since it does not require substructures, e.g., frames and longerons. Three configurations are explored, as shown in Error! Reference source not found..

Based on the trade study, it is observed that the stresses remain almost unchanged when switching from monolithic composite of 4 layers of cloth to honeycomb composite with 2 layers of cloth on each side. But the buckling performance is greatly improved. But it is noticed that 2 layers of cloth will yield a facesheet thickness as 0.028. Out of damage tolerance considerations, facesheets made of 3 layers of cloth are studied, and the weight penalty of adding one layer on each side is checked. The total performance of honeycomb composite with 3 layer facesheets is considered desirable, so it is chosen as the fuselage aft skin.

The facesheets both have a configuration of [45, 0, 0], so the 0° cloth, together with the honeycomb, can be stopped to form a [45, 0, 0, 45] monolithic composite skin, when the aft fuselage meets its neighboring skins, as shown in Figure 5.21.
Table 5.4 – Trade study of fuselage aft skin

<table>
<thead>
<tr>
<th>Configurations</th>
<th>Critical stress (KSI)</th>
<th>Buckling</th>
<th>Resulting aircraft weight (lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Monolithic composite skins (4 layers of C/E cloth), plus aluminum substructures</td>
<td>22.9</td>
<td>At 0.76 times ultimate load</td>
<td>306.81</td>
</tr>
<tr>
<td>Honeycomb composite skin (2 layers of C/E cloth on each face) without substructure</td>
<td>22.8</td>
<td>No buckling at ultimate load</td>
<td>305.31</td>
</tr>
<tr>
<td>Honeycomb composite skin (3 layers of C/E cloth on each face) without substructure</td>
<td>16.7</td>
<td>No buckling at ultimate load</td>
<td>311.46</td>
</tr>
</tbody>
</table>

Figure 5.21. Schematic of aft fuselage skin panel

5.3.2 Final Structural Arrangement

The fuselage is composed of carbon/epoxy skins with aluminum sub-structure. There is one primary bulkhead (the firewall) where both the engine and wing attach. There are two frames in the payload section of the fuselage: one to for the rear spar attachment and one to close out the payload bay doors. The section between the payload area and the v-tail will utilize honeycomb core carbon/epoxy skins. The V-tail utilizes two aluminum spars that attach to two machined aluminum frames. These frames also serve as the structural attachment points for the tail gear.
Figure 5.22 – Fuselage Geometry
5.3.3 Strength Analysis of the Center Fuselage Skins

The center fuselage skin is not a highly-loaded component, but it experiences sizable loads under both wing-critical and V-tail critical loads, because it is connected to both structures. The center fuselage skin is basically made of 4-layer composites, but locally thickened to 5 layers where it meets cutouts and frames/longerons.

<table>
<thead>
<tr>
<th>Material</th>
<th>Layup</th>
</tr>
</thead>
<tbody>
<tr>
<td>4 layer composite skin</td>
<td>AS4/3501-6 Cloth [45, 0, 0, 45]</td>
</tr>
<tr>
<td>5 layer composite skin</td>
<td>AS4/3501-6 Cloth [45, 0, 45, 0, 45]</td>
</tr>
</tbody>
</table>
The maximum stress theory [5] is applied, ply by ply, to AS4/3501-6 cloth composites:

\[
\sigma_L = \sigma_x \cos^2(\theta) + \sigma_y \sin^2(\theta) + 2\sigma_{xy} \cos(\theta) \sin(\theta) < \sigma_{LU} = \begin{cases} 45 \text{ (tension)} \\ 51 \text{ (compression)} \end{cases} \text{ KSI}
\]

\[
\sigma_T = \sigma_x \sin^2(\theta) + \sigma_y \cos^2(\theta) - 2\sigma_{xy} \cos(\theta) \sin(\theta) < \sigma_{TU} = \begin{cases} 45 \text{ (tension)} \\ 51 \text{ (compression)} \end{cases} \text{ KSI}
\]

\[
\sigma_{LT} = -\sigma_x \sin(\theta) \cos(\theta) + \sigma_y \sin(\theta) \cos(\theta) + \sigma_{xy} [\cos^2(\theta) - \sin^2(\theta)] < \sigma_{LTV} = 10 \text{ KSI}
\]

where \( \theta \) is the angle between the \( x \) stress and the ply principal axis, which is 0 degree in the following calculations since Patran has the functionality of converting stresses into material principal coordinates; \( L \) and \( T \) are longitudinal and transverse directions defined in material principal coordinates. The material principal coordinate 1 is plotted as a straight line in magenta using the Laminate Modeler function of Patran Tools.

**At wing critical loads**

Various stress distributions, including maximum principal, minimum principal and max shear stresses, have been checked to find the most critical location.

For ply 1, the maximum principal and maximum shear stress indicate the same critical location, at the joint of element 69707 and 69326, node 53993, as shown in

Figure 5.26. The stress tensor at this node is plotted, together with the material principal coordinate 1 (shown in magenta). So at this node, the critical stresses in ply 1’s material principal coordinates are:

- \( \sigma_L = 7.86 \times 10^7 \text{ Pascal (11.4 KSI)} \).
- \( \sigma_T = -2.36 \times 10^7 \text{ Pascal (-3.4 KSI)} \).
- \( \sigma_{LT} = -1.65 \times 10^7 \text{ Pascal (2.4 KSI)} \).
The resulting margins of safety in L, T and LT directions are:

\[ MS_L = \frac{\sigma_{LU}}{\sigma_L} - 1 = \frac{45 \text{ KSI}}{11.4 \text{ KSI}} - 1 = 2.95 \]
\[ MS_T = \frac{\sigma_{TU}}{\sigma_T} - 1 = \frac{51 \text{ KSI}}{3.4 \text{ KSI}} - 1 = 14.0 \]
\[ MS_{LT} = \frac{\sigma_{LTU}}{\sigma_{LT}} - 1 = \frac{10 \text{ KSI}}{2.4 \text{ KSI}} - 1 = 3.17 \]

For ply 1, the minimum principal stress indicates that the critical location is at the joint of element 65504 and 65507, node 54855 as shown in Figure 5.26. The stress tensor at this node is plotted, together with the material principal coordinate 1 (shown in magenta). So at this node, the critical stresses in ply 1’s material principal coordinates are:

- \( \sigma_L = -2.14 \times 10^7 \text{ Pascal (-3.1 KSI)} \).
- \( \sigma_T = 5.19 \times 10^7 \text{ Pascal (7.5 KSI)} \).
- \( \sigma_{LT} = 2.05 \times 10^7 \text{ Pascal (3.0 KSI)} \).

The resulting margins of safety in L, T and LT directions are:

\[ MS_L = \frac{\sigma_{LU}}{\sigma_L} - 1 = \frac{51 \text{ KSI}}{3.1 \text{ KSI}} - 1 = 15.45 \]
\[ MS_T = \frac{\sigma_{TU}}{\sigma_T} - 1 = \frac{45 \text{ KSI}}{7.5 \text{ KSI}} - 1 = 5.0 \]
\[ MS_{LT} = \frac{\sigma_{LTU}}{\sigma_{LT}} - 1 = \frac{10 \text{ KSI}}{2.4 \text{ KSI}} - 1 = 2.33 \]

In summary, ply 1 is critical in LT direction at node 54855 with a margin of safety 2.33.
5.3.4 Strength Analysis of the Firewall

The firewall is sized under wing critical loads. It transfers two major loads: wing loads and engine mounting loads. So it is machined from a 7075-T651 plate. At the joint of the wing and the firewall, the firewall web is thickened locally to 0.177” to reduce the stress concentration caused by drastic change of stiffness, as shown in orange in Figure 5.27.

Table 5.6. Description of the firewall

<table>
<thead>
<tr>
<th>Material</th>
<th>Thickness (in)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Firewall cap</td>
<td>7075-T651 plate</td>
</tr>
<tr>
<td>Firewall web</td>
<td>7075-T651 plate</td>
</tr>
</tbody>
</table>

Various stress types and locations are checked and the most critical one is plotted. The geometry and finite element model of stringers are shown in Figure 5.27. The most critical stress, which is the minimum principal at Z2 as shown in Figure 5.28, is \(-5.02 \times 10^8\) Pascal (72.8 KSI) at the joint of element 65995 and 65996, node 55167, where the firewall meets the in-board forward upper spar cap. The margin of safety is calculated as:

\[
MS = \frac{\sigma_{\text{allowable}}}{\sigma_{\text{critical}}} - 1 = \frac{76 \text{ KSI}}{72.8 \text{ KSI}} - 1 = 0.04
\]
5.3.5 Strength Analysis of the Payload Hatch

The payload hatch on top of the fuselage cuts load paths of the wing loads, as well as the V-tail loads. Plus, it has a very complex 3-D shape. Out of the above considerations, the hatch is made of machined 7075-T651 plates, then connected by flanges made by bent 2024-T3 sheets.

Stress results under wing critical and V-tail secondary dive load cases are both plotted. The stress levels are $-3.94 \times 10^7$ Pascal (5.71 KSI) and $-4.21 \times 10^7$ Pascal (6.11 KSI), which are deemed safe.
Figure 5.29 – Stresses for the Hatch for Critical Wing Loads

Figure 5.30. Stresses of the hatch for V-tail secondary dive loads
5.3.6 Structural Design of the V-tail

The V-tail is sized under V-tail secondary dive and dive load cases. The V-tail spars will be machined out of 7075-T651 plate because it is one of the critical load carrying members and it has complex shapes. The skins are monolithic composites, for ease of manufacturing complex contours. Ribs are hammer-formed 5052-H34 aluminum alloy sheets. Please see Appendix B for detailed documentation.

5.3.7 Dynamic Analysis of the Whole Aircraft

Normal modes analysis is carried out to check the natural frequencies and mode shapes of the whole aircraft. Besides the previously mentioned wing bending and twisting modes, landing gear/wing coupling modes and antenna modes are observed, as shown in Figure 5.31.
5.3.8 Material Selections and Manufacturing Plan

The fuselage geometry is composed of large, sweeping polyconic surfaces with complex curvature, the shape of which is driven by aerodynamic considerations. Composite structures are well suited to this type of geometry.

The fuselage skins will be made of AS4/3501-6 carbon fiber cloth with Nomex honeycomb core in the aft fuselage section. The tooling for the fuselage skins will be machined from medium density polyurethane foam as shown in Figure 5.33. The overall tool dimensions are larger than...
the machining center tables at KU so the tool will be manufactured in pieces and assembled. This method has been utilized on several projects and works quite well.

The fuselage sub-structure consists of one primary bulkhead, two payload frames, one intermediate ring-frame, and two V-tail spars/frames. In addition, there are longerons that frame both payload hatches and the upper surface of the wing intersection.
**Figure 5.32 – Fuselage Manufacturing Breakdown**

<table>
<thead>
<tr>
<th>#</th>
<th>Part Name</th>
<th>Material</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>firewall</td>
<td>7075-T651</td>
</tr>
<tr>
<td>2</td>
<td>frame 110</td>
<td>7075-T652</td>
</tr>
<tr>
<td>3</td>
<td>frame 120</td>
<td>7075-T653</td>
</tr>
<tr>
<td>4</td>
<td>fuse skin lwr</td>
<td>Carbon/Epoxy</td>
</tr>
<tr>
<td>5</td>
<td>fuse skin upr</td>
<td>Carbon/Epoxy</td>
</tr>
<tr>
<td>6</td>
<td>payload hatch</td>
<td>Carbon/Epoxy</td>
</tr>
<tr>
<td>7</td>
<td>v tail lead edge</td>
<td>Alum. TBD</td>
</tr>
<tr>
<td>8</td>
<td>ruddervators</td>
<td>Fiberglass/Foam</td>
</tr>
<tr>
<td>9</td>
<td>v tail fwd spar</td>
<td>7075-T651</td>
</tr>
<tr>
<td>10</td>
<td>v tail aft spar</td>
<td>7075-T651</td>
</tr>
</tbody>
</table>
5.3.9  Fuselage Manufacturing Costs

5.3.9.1  Tooling Costs

The primary cost in the fuselage manufacturing will be the tooling. The tooling cost is determined using two methods. The first method used is to use material costs from vendors and estimates for machining time. The results of this method are:

- **Materials:** $12,500
  - 25 Sheets of FR-3700 Foam (96” x 48” x 6”) @ $500/sheet
- **Machining:** $10,000
  - 200 hours @ $50/hr
- **Total:** $22,500

The second method was to request a quote from a low cost tooling vendor. The cost for medium density foam tools based on vendor information is $140/ft². The fuselage surface area is approximately 95ft². Using a factor of 1.5 to account for excess material, the resultant cost of outsourced tooling is approximately:

- **Outsourced Tooling:** $19,950

The two cost estimation methods produced very similar results. These two methods represent two manufacturing concepts: In-house tooling and outsourced tooling. The results indicate that either option is viable.

![Figure 5.33 – Lower Fuselage Skin Tooling](image-url)
Figure 5.34 – Manufacturing Plan for the Fuselage
5.3.9.2 Material Costs

The material costs are estimated primarily based on weight. The costs per lbs of the materials in the fuselage are:

- Aluminum: $10/lbs
- Carbon Fiber: $100/lbs

The resultant material cost estimates for the fuselage are shown in Table 5.7

Table 5.7 – Material Costs for the Fuselage Structure

<table>
<thead>
<tr>
<th>Part Name</th>
<th>Material</th>
<th>Blank Weight</th>
<th>Material Cost</th>
</tr>
</thead>
<tbody>
<tr>
<td>firewall</td>
<td>7075-T651</td>
<td>122</td>
<td>$1,215.52</td>
</tr>
<tr>
<td>frame 110</td>
<td>7075-T651</td>
<td>73</td>
<td>$729.04</td>
</tr>
<tr>
<td>frame 120</td>
<td>7075-T651</td>
<td>64</td>
<td>$641.72</td>
</tr>
<tr>
<td>frame 200</td>
<td>7075-T651</td>
<td>7</td>
<td>$73.37</td>
</tr>
<tr>
<td>fuse skin lwr</td>
<td>Carbon/Epoxy</td>
<td>44</td>
<td>$4,400.00</td>
</tr>
<tr>
<td>fuse skin upr</td>
<td>Carbon/Epoxy</td>
<td>43</td>
<td>$4,300.00</td>
</tr>
<tr>
<td>payload hatch</td>
<td>Carbon/Epoxy</td>
<td>6</td>
<td>$612.31</td>
</tr>
<tr>
<td>v tail F spar</td>
<td>7075-T651</td>
<td>8</td>
<td>$84.37</td>
</tr>
<tr>
<td>v tail F spar</td>
<td>7075-T651</td>
<td>8</td>
<td>$78.55</td>
</tr>
<tr>
<td>Total</td>
<td></td>
<td></td>
<td>$12,134.88</td>
</tr>
</tbody>
</table>

5.4 Control Surfaces

5.4.1 Ailerons

The Meridian will utilize a simple aileron that is mass and aerodynamically balanced without the use of a horn. The hinge was located approximately 30% along the aileron chord length. The aileron will be composed of built up aluminum parts bonded together. The aileron skins and spars will be 0.025” thick 2024-T3 aluminum sheet. The ribs will be 0.025” thick 5052-H32 aluminum. The leading edge will be fiberglass with tungsten inserts for mass balancing. The hinge brackets will be machined 2024-T3 aluminum with steel bushing inserts. The hinges are collocated with the wing ribs to reduce local deflections.
5.4.2 Flaps

The flap structural layout is very similar to the aileron except that the simple hinges are replaced by flap tracks as the flap is a single slotted fowler flap. The flap tracks are shown in Figure 5.36. These tracks are very similar to those found on a Cessna 182 [www.cessna.com]. The maximum flap deflection is 45°.
5.4.3 Ruddervators

The ruddervators will be made using wood ribs, foam core, and fiberglass skins. The plywood ribs will be laser cut, then used as templates to hot-wire cut the foam core. The ribs and core will be bonded together then wrapped in fiberglass cloth. This method is often used in both remote control and homebuilt aircraft manufacturing.

6 Flight Control System

6.1 Ground Station Design

The whole ground station is constructed by the following features:

- Ground station unit, supplied by Cloud Cap Technology
- Ground station UHF 890-960 MHz antenna
- Operator Interface PC
- Futaba pilot console for manual pilot control

Figure 6.1 shows basic setup for Piccolo ground station system.

![Ground Station Setup](image)

Figure 6.1 Ground Station Setup [1]

6.1.1 Ground Station Unit

The main piccolo ground station unit manages the communications to the avionics, interfaces with the pilot console and streams telemetry and command and control data to and from the operator interface PC. The ground station interface connects to the operator interface using a 9-pin serial connection.
To communicate with the avionics the ground station interface utilizes a 1 Watt 910 MHz or 2.4 GHz transceiver. This gives a RF communication range of approximately 10 miles. Additionally a GPS unit is also used for differential GPS correction relays to the avionics on board the Meridian.

Figure 6.2 and Figure 6-2 provides basic description of Piccolo ground station unit.

![Figure 6.2 Ground Station Unit Front Panel](image1)

![Figure 6-2 Ground Station Unit Back Panel](image2)

### 6.1.2 Operator Interface PC

The command and control interface for the meridian will function from a PC using the Piccolo operator interface. Currently this computer is an Itronix GOBook III rugged notebook PC that contains the following features:

- 1.8 GHz Intel Centrino Pentium M 745
- 1 GB SDRAM
- Integrated GPS and Wireless Communication
- Fully-Rugged to Mil-STD 810F
- Cold Tested to -30 °C
- Heated Hard Drive
- Water-Proofed Keyboard
6.1.3 Futaba Console

Pilot commands are relayed to the Meridian when it is in manual mode using a Futaba T9CAP super console. The console is connected to the ground station interface unit using the Futaba compatible buddy-box 6 pin DIN connector. Using the pilot console the Meridian pilot will be able to activate or deactivate the autopilot and have full control authority over the aircraft.

6.1.4 Operator Software Interface

The operator interface environment software allows for total monitoring of the aircraft health status, including attitudes, rates, accelerations, airspeed, ground speed, GPS location, winds aloft and communication signal strength. Commands are also issued to the aircraft through the operator interface. The limits of the control surfaces as well as airspeed are inputted into the aircraft through this system as well.

In addition the position of the aircraft can be monitored real-time on the navigation page, where a map can be used for geographic referencing to the GPS coordinates (Figure 6.3). As mentioned previously flight plans will be displayed on the maps and can be updated at any time from the operator interface.
6.2 **Onboard System**

The Piccolo II system uses the PID controller technique to form the flight control system architecture. It allows control of the following features:

6.2.1 **Indicated Airspeed Control**

The Piccolo uses a Proportional-Integral-Derivative (PID) controller to control the indicated airspeed using dynamic pressure feedback to the elevator.

6.2.2 **Altitude Control**

Piccolo makes use of two different feedback loops to control altitude:
• Altitude to Throttle
  o A PID w/rate limiter control scheme is used to maintain a commanded altitude using the 115 KPa Barometric altitude sensor to determine the current density altitude

• Altitude to Elevator
  o A PD scheme is used to maintain a commanded altitude, also using the 115 KPa Barometric altitude sensor to determine the current density altitude

6.2.3 Turn Rate Control
Piccolo II uses a PID scheme to control turn rate using roll to aileron feedback.

6.2.4 Pitch Damper Control
A PD scheme is used by the Piccolo II to damp out pitch oscillations using pitch angle to elevator feedback

6.2.5 Yaw Damper Control
A proportional control scheme is used by the Piccolo II to damp out oscillations in yaw using yaw rate feedback to the rudder.

6.2.6 Turn Compensator
A proportional scheme of bank angle feedback to throttle and bank angle feedback to elevator is used by the Piccolo II to maintain altitude during turns.

6.2.7 Heading Tracker
A PID scheme is used to track the desired heading generated by the waypoints. Heading error is feedback to the turn rate command to get the desired tracking.

6.2.8 Ruddervator Mixing
The ruddervator mixing in the Piccolo II is done by a simple algorithm, given as follows:

![Figure 6.4 – Ruddervator Mixing [1]](image)
6.3 Navigation System

The Piccolo uses GPS defined waypoints for the navigational outer loops. Flight plans are created at the operator interface and can be sent to the Piccolo at any time, including while in flight. The system allows for a great level of flexibility for changing flight situations (i.e. weather, air traffic, etc.).

6.3.1 Waypoints Navigation

The Piccolo II can store up to 100 waypoints, with each connecting waypoint forming the connected flight plan. Each waypoint includes a latitude, longitude and altitude coordinate. The commanded altitude to the aircraft is set at to the waypoint towards which the aircraft is flying.

A pre-turn algorithm is encoded in the Piccolo that determines when to begin the turn to the next waypoint to waypoint flight plan so as not to overshoot. However, if it is required that waypoint flyover occurs this feature can be disabled by the operator.

An orbiting waypoint may also be set, with the aircraft flying in a clockwise orbit at a specified radius for a specified amount of time.

Figure 6.5 – Operator Interface Navigation Page [xxx]
6.4 Actuator Sizing

The actuator sizing was performed using hingemoment derivatives from the Advanced Aircraft Analysis Software. The hingemoment derivatives are shown in xxx along with the required inputs for the servo torque calculation. The value \( d_s \) refers to the servo arm length, while \( d_{cs} \) refers to the length of the control surface horn length.

The hinge locations of the ailerons and ruddervators were placed such that a ‘reasonable’ servo torque was achieved. This reasonable torque value is based on the available COTS servos. Typical servos for UAVs of this size range from 50-200 in-lbs of torque. In addition, there is a desired requirement to use the same servo on all control surfaces to minimize cost.

The hingemoment analysis resulted in a required servo torque of 22 in-lbs, which was driven by the aileron sizing. Note: standard linear actuators will be used for the flaps. The Kearfott K-2000 servos have been selected for the ailerons, ruddervators, and tailwheel actuators.
Table 6.1 – Servo Sizing Inputs

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Units</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>q</td>
<td>psf</td>
<td>41.5</td>
</tr>
<tr>
<td>$S_a$</td>
<td>ft$^2$</td>
<td>3.74</td>
</tr>
<tr>
<td>$c_a$</td>
<td>ft</td>
<td>0.66</td>
</tr>
<tr>
<td>$S_w$</td>
<td>ft$^2$</td>
<td>1.96</td>
</tr>
<tr>
<td>$c_w$</td>
<td>ft</td>
<td>0.41</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>rad$^{-1}$</td>
<td>0.012</td>
</tr>
<tr>
<td>$\delta_a$</td>
<td>rad$^{-1}$</td>
<td>-0.043</td>
</tr>
<tr>
<td>$\beta$</td>
<td>rad$^{-1}$</td>
<td>-0.0657</td>
</tr>
<tr>
<td>$\delta_w$</td>
<td>rad$^{-1}$</td>
<td>-0.355</td>
</tr>
</tbody>
</table>

$\delta_{cs}$ | 1 in |

$\delta_s$ | 1 in |

Figure 6.6 – Kearfott K-2000 Servo (www.kearfott.com)

Figure 6.7 – Kearfott Servo Data (www.kearfott.com)
7 Communications System

7.1 Pertinent Requirements

7.2 Airborne Communications

There are many possibilities for systems, which can fulfill the requirements listed above; however, a system that is user friendly is a necessity. Also, looking into the future and developing a system which is highly adaptable should be the ultimate goal.

The first thought in the process is to develop a system, which uses the capabilities of the Piccolo (see Figure 7.1) and its operator interface. The piccolo has a radio frequency (RF) transceiver as well as the capability to connect to an Iridium modem (satellite communications). The RF transceiver is only capable of line of sight communications and has a range of only 20 miles. The Iridium modem (see Figure 7.1) uses satellites as a relay for over the horizon communications. Unfortunately, testing shows that the Piccolo to Iridium connection is inconsistent at best. It is nearly impossible to make this simple connection; therefore, a modification to the original design has been made.

The new design calls for a Piccolo to act as the solo autopilot with supporting roles for all other components. Piccolo contains the communications system for line of sight only. Therefore, it uses the RF transceiver in a similar fashion to an RC aircraft. This communications system is used during take off, landing, and line of sight operations. When the autopilot is switched on, Iridium communications will take over as the main telemetry and command and control system.

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1 Cloud Cap Technology http://www.cloudcaptech.com/piccolo_II.htm

2006
However, instead of Iridium being connected to Piccolo, it is connected to the single board computer (SBC) on board. The SBC is the Ampro Ready Board 800 (seen in Figure 7.2). Piccolo connects to the SBC through two serial connections, the program port and the payload port. The program port sends the telemetry information which exactly mirrors the information sent over the RF transceiver to the ground station. The program port receives this information and relays it to Iridium which sends it to the ground station. The payload port plays a pivotal role in the command and control part of the process. The ground station sends commands to Iridium on board the aircraft. The SBC takes these commands and sends them to Piccolo over the payload port.

![Figure 7.2: AMPRO Ready Board 800\textsuperscript{3} Single Board Computer](image)

Now that the issue over command and control is solved, the other supporting parts can now be discussed. To have a secondary source of data to confirm the truth of the Piccolo system, the Nav-420 (seen in Figure 7.3), an inertial measurement unit (IMU), will be included in the avionics system. The IMU will provide basically the same information as the Piccolo; however, this makes the system more robust and adaptable. With the Nav-420 on board, a system could later be developed to replace Piccolo.

![Figure 7.3: Nav-420\textsuperscript{4} Inertial Measurement Unit](image)


For flight-tests it is beneficial to have as much data as possible. The downside of Piccolo and the Nav-420 is that they are both limited to the type of sensors being used. Therefore, a third source of telemetry confirms that the data Piccolo and the Nav-420 are collecting is true. Acra’s Kam-500 (seen in Figure 7.4) system is the stand alone telemetry system, which supplies a third set of telemetry to the ground station which can include additional sensors. The Kam-500 uses 2.4 GHz transmitters (seen in Figure 7.4) and a sensitive receiver (seen in Figure 7.4) to relay their data to the ground. This system along with a high gain antenna (12dB gain) on the ground will allow a communications range (line of sight) of 60 miles.

![Figure 7.4: ACRA Controls KAM-500 Data Acquisition System](http://www.acracontrol.com/airborne_data/documents/KAM-500ProductGuide2006.pdf), 2005

The system also has to send data to the radar system in order to correctly calculate the image the radar is collecting at any given time. The SBC decodes the data from Piccolo and the Nav-420 and sends it over a RS-232 serial cable to the radar processor for the post processing. Also, the radar’s processor will relay images to the ground through the processor and Iridium for the confirmation that the system is working properly.

The system will be connected as seen in Figure 7.5. The processor acts mainly as a decoder, encoder, and relay for the IMUs, radar’s processor, and ground station. The KAM-500 will only be on board during flight tests. After the flight tests finish, the need for the third truth source will be unnecessary, and just add weight to the aircraft. Therefore, the KAM-500 can act strictly as a standalone system.

---

Other avionics systems must be on board to safely and legally fly an aircraft. One such device is the transponder, which acts as aids radar in tracking the aircraft. Since the UAV is limited in size and weight, the smallest possible transponder benefits the aircraft. Therefore, the T2000UAV (see Figure 7.6), developed by Microair, is the best transponder for this application. Its weight is only 1.1 pounds and most importantly has the ability to change squawk codes in the air. This transponder is made specifically for UAVs, so it has remote capabilities.

Another safety device is the Emergency Locator Transmitter (ELT). In case of a crash, this device transmits a signal so the aircraft can be located. Again, a smaller, lighter device is the best choice for these operations. The EBC 102A (see Figure 7.7) developed by Emergency

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8 Microair Avionics Pty Ltd.
November, 2006
Beacon Corporation is only 1.2 pounds and can sustain temperatures down to -20°F. The beacon will transmit for up to 72 hours and is automatically activated upon impact.

7.2.1 Power and Dimensions for Avionics suit

For proper design of the power plant needed on board, there needs to be sufficient knowledge of how much power is used. Also, sizing and placement of certain items needs to be considered for weight and balance, as well as optimizing space on board the vehicle. Table 7.1 shows the dimensions, weight, and power for each component in the avionics system. The Acra Controls components uses are limited to flight test and are not intended to be flown for actual missions.

---

Table 7.1: Power Budget, Weight, and Dimensions for the Avionics System

<table>
<thead>
<tr>
<th>System</th>
<th>Subsystem</th>
<th>Manufacturer</th>
<th>Model</th>
<th>Volt</th>
<th>Amps</th>
<th>Power</th>
<th>Weight</th>
<th>Volume</th>
</tr>
</thead>
<tbody>
<tr>
<td>Avionics</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>Watts</td>
<td>lbs</td>
<td>in x in x in</td>
</tr>
<tr>
<td>Auto Pilot</td>
<td>Iridium</td>
<td>NAL</td>
<td>Picollo 2.4 GHz</td>
<td>12</td>
<td>0.3</td>
<td>3.6</td>
<td>0.47</td>
<td>4.8x2.4x1.5</td>
</tr>
<tr>
<td>Modem</td>
<td></td>
<td>NAL Research</td>
<td>A3LA-S</td>
<td>4.4</td>
<td>1</td>
<td>4.4</td>
<td>1</td>
<td>7.7x3.3x1.5</td>
</tr>
<tr>
<td>Antenna</td>
<td>Data Acquisition</td>
<td>Research</td>
<td>SAF7352-IG</td>
<td>3</td>
<td>0.015</td>
<td>0.045</td>
<td>0.1</td>
<td>2.2x1.3x0.5</td>
</tr>
<tr>
<td>Onboard Processor</td>
<td></td>
<td>Crossbow Ready Board</td>
<td>NAV 420</td>
<td>12</td>
<td>0.35</td>
<td>4.2</td>
<td>1.3</td>
<td>3 x 3.8 x 3</td>
</tr>
<tr>
<td>Transponder</td>
<td></td>
<td>AMPRO</td>
<td>800</td>
<td>5</td>
<td>3.7</td>
<td>18.5</td>
<td>0.25</td>
<td>4.5x6.5x0.6</td>
</tr>
<tr>
<td>Chassis</td>
<td></td>
<td>ACRA Controls</td>
<td>KAM-500 Chassis</td>
<td>28</td>
<td>0.11</td>
<td>18</td>
<td>4.6</td>
<td>3.2x3.7x8.8</td>
</tr>
<tr>
<td>Analog Sensors</td>
<td>Controls</td>
<td>ACRA</td>
<td>KAD/ADC/xxx</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>PCM Decoder</td>
<td>Controls</td>
<td>ACRA</td>
<td>KAD/DEC/003</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>PCM Encoder</td>
<td>Controls</td>
<td>ACRA</td>
<td>KAD/BCU/101</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>Digital Sensors</td>
<td>Communications</td>
<td>ACRA Controls</td>
<td>KAD/UAR/102</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
<td>--</td>
</tr>
<tr>
<td>Receiver</td>
<td>Controls</td>
<td>ACRA</td>
<td>Airborne receiver</td>
<td>12</td>
<td>0.23</td>
<td>2.76</td>
<td>0.992</td>
<td>4.7x3.1x1.6</td>
</tr>
<tr>
<td>Low Noise Amplifier</td>
<td>Controls</td>
<td>ACRA</td>
<td>RF Low Noise Amplifier</td>
<td>12</td>
<td>0.084</td>
<td>1.008</td>
<td>0.1</td>
<td>--</td>
</tr>
<tr>
<td>Transmitter</td>
<td>Controls</td>
<td>ACRA</td>
<td>Airborne transmitter 5W</td>
<td>12</td>
<td>1.5</td>
<td>18</td>
<td>0.727</td>
<td>1.18x3x5.6</td>
</tr>
<tr>
<td>Antenna</td>
<td>Controls</td>
<td>ACRA</td>
<td>S-band Antenna</td>
<td>0</td>
<td>0</td>
<td>0.198</td>
<td>3.4x2.6x1.3</td>
<td></td>
</tr>
<tr>
<td>Emergency Locator</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Transmitter</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

7.3 Ground Station

The ground station is vital to the safety involved in flying a UAV. The ground station will give the pilot the ability to visualize the aircraft without being able to see the aircraft. The telemetry that streams data down from the aircraft to the ground station gives the pilot the ability to fly the aircraft similar to Instrument Flight Rules (IFR). Since there is a significant delay (approximately 2 seconds) from the signal of the aircraft to the ground station, the pilot is not able to manually fly the aircraft, but the pilot would be able to change many of the pre-programmed flight plans. These changes include: waypoints, throttle settings, and altitude.
7.3.1 Piccolo Ground Station

The ground station is made up of two distinct parts. The main ground station that controls the aircraft’s autopilot will be handled with two GoBook laptops. These laptops have the Piccolo operator interface loaded onto them. This allows for the monitoring of the telemetry sent by Piccolo. They also stream science data, as well as, telemetry from the Nav-420.

The communications part of the Piccolo ground station includes an RF transceiver and the Iridium modem. The RF transceiver is used for line of sight communications and Iridium is used for over the horizon communications.

7.3.2 ACRA KAM-500 Ground Station

The ground station supplied with the stand-alone KAM-500 data acquisition system is contracted to Lumi Star for financial reasons. Lumi Star offers a portable ground station (see Figure 7.8), which includes; a rugged portable computer, S-band receiver, bit synchronizer, multi function decomutator, RS-422 board, keyboard, display and software. This ground station communicates only with the KAM-500 and displays data of all the sensors connected to the KAM-500.

Figure 7.8: Lumi Star Portable Ground Station for the KAM-500\textsuperscript{10}
8 Propulsion System

The current design configuration of the Meridian is a single engine aircraft with a propeller in a puller configuration. The basis of the choice was to keep the propulsion system simple while minimizing fuel consumption. At the time of the CDR, there were two possible engine candidates for the proposed UAV; the Innodyn 165TE turbo-propeller and the Thielert Centurion 2.0 Diesel engines. Pictures of these engines are shown in Figure 8.1 and Figure 8.2 respectively. A third engine, the Rotax 914 piston engine has also been considered, but there are no plans for its acquisition and testing at this time. The Innodyn engine was acquired in the fall of 2006 and has been undergoing testing with disappointing results. For this reason the primary engine has been changed to the Thielert Centurion.

The engine testing of the Innodyn will continue as this is still a viable option for future versions of the Meridian aircraft. The engines will be tested at the Mal Harned Propulsion Laboratory located at Lawrence airport. The performance of the propulsion system depends not only on the engine performance but also the characteristics of the propeller, air inlet and the exhaust nozzle used.

8.1 Pertinent Requirements

The propulsion system for the Meridian has five major requirements; power, weight, specific fuel consumption (SFC), cold weather performance, and cost. Published data for the engines being considered are shown in table 7-1.

<table>
<thead>
<tr>
<th>Engine</th>
<th>Innodyn 165TE</th>
<th>Centurion 2.0</th>
<th>Rotax 914</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuel Type</td>
<td>Jet A, JP4, JP5</td>
<td>Diesel, Jet A</td>
<td>Avgas 100LL</td>
</tr>
<tr>
<td>Cooling</td>
<td>Air</td>
<td>Liquid</td>
<td>Liquid/Air</td>
</tr>
<tr>
<td>Power (HP)</td>
<td>165@100%</td>
<td>135 @ takeoff</td>
<td>115 @ 100%</td>
</tr>
<tr>
<td>SFC (lb/hp.hr)</td>
<td>0.65 @ cruise</td>
<td>0.36 @ BEP</td>
<td>0.48 @ MCP</td>
</tr>
<tr>
<td>Dry Weight (lb)</td>
<td>183</td>
<td>300</td>
<td>180</td>
</tr>
</tbody>
</table>

The Meridian requires an engine with a minimum power of 100 HP. The weight of the engine affects the overall weight of the aircraft, not only as an individual component weight, but also by the weight of the extra supporting structure that is required to integrate the engine into the aircraft.
aircraft. The specific fuel consumption affects the range and weight of the vehicle. Since the location for the UAV missions are in Greenland and Antarctica, performance of the engines in the expected cold weather regime of these areas will be considered. The cost of the turbo-propeller and diesel engines are approximately the same.

![Innodyn 165TE Turbo-propeller](image1.png)

**Figure 8.1 Innodyn 165TE Turbo-propeller**

![Centurion 1.7 Diesel Engine](image2.png)

**Figure 8.2 Centurion 1.7 Diesel Engine**

### 8.2 Engine Selection

The Innodyn 165TE is the first engine to be tested for this project and is in the preliminary stages of testing. At the time of this report results for the Innodyn engine are inconclusive so the Innodyn is being considered only as a possible future modification to the design. Testing for the Centurion 2.0 will begin in summer 2007. The Centurion 2.0 is a certified engine, which means the testing of it is much less critical.

There are also three areas of concern that impact the logistics of the mission rather than the aircraft requirements and it concerns the type of fuel used. The first item is to determine what
contaminates will be left behind from the engine exhaust. Part of the static test plan, is the measurement of un-burnt hydrocarbons in the exhaust of these engines. The second item is to determine whether the fuel or fuel/additive combination is suitable for the expected cold weather operation.

8.3 Structural Integration

The Thielert engine is being purchased as a retrofit for a Piper PA-28 aircraft [xxx]. This will include all necessary engine accessories as well as the engine mount. The Meridian fuselage geometry at the firewall is designed to match that of the PA-28 such that the cowl from the Piper aircraft can be used or copied.

![Figure 8.3 – Thielert Centurion 2.0 Firewall Forward Kit](image)

8.4 Propeller Selection

The propeller selection will depend heavily on the engine selected. The engine manufacturer usually provides the information on the recommended propellers that have been used on an engine to produce the published data. Currently, the Innodyn 165TE uses a NSI CAP200 propeller and the Centurion 2.0 uses an MT-6-a/187-129 propeller.

8.5 Intake and Exhaust Design

The intake and exhaust of the Thielert engine will be the same for the Meridian as they are for the PA-28, so very little design work is required.
8.6 Autopilot Interface

The Thielert Centurion engine utilizes a Full Authority Digital Engine Control (FADEC) unit that requires only one input from a pilot – throttle setting. This is ideal in terms of autopilot integration. The Piccolo will output a PWM signal which will be converted into the signal input required by the FADEC.

8.7 Fuel System

The fuel system for the Meridian consists of four separate fuel tanks as shown in Figure 8.4. The fuel is stored entirely in the inboard wing torque box. Fuel bladders will be utilized to minimize the possibility of fuel leaks. The detailed design of the fuel bladders is currently underway with Aerotech Labs Inc. ATL manufacturers custom fuel cells for a variety of UAVs.

9 Landing Gear System

The landing gear of the Meridian is another implementation of COTS hardware. The main landing gear for the Meridian are nose gear struts for a Lancair Legacy aircraft. These struts are approximately $4,500 per strut and can handle up to a 2,200 lbs aircraft. Lancair has been reluctant to offer any specific dimensions of their landing gear, which has resulted in a delay in locking in the detail geometry of the drag brace attachment as is evident in Figure 9.2. An understanding is currently being reached with Lancair, which will result in finalized main gear attachment geometry shortly.
As the landing gear system is critical to the overall success of the project, a secondary vendor has been contacted to manufacture the landing gear in the event that Lancair is unwilling to sell a landing gear for a non-lancair related purpose: Infinity Aerospace [www.infinityaerospace.com]. Infinity Aerospace manufacturers landing gear for the Long EZ, Cozy III, as well as other homebuilt aircraft. This option would require a longer lead time than the Lancair solution, but this is not expected to create a project delay as this will not affect the manufacturing or assembly of the rest of the aircraft.

Figure 9.1 shows the propeller clearance based on the Lancair landing gear. The lateral clearance of the antennas is shown in Figure 12.4. The lateral and longitudinal tipover requirements have been met as documented in [2].

![Figure 9.1 – Propeller Clearance](image)

### 9.1 Structural Integration

The landing gear is mounted to two machined ribs that hang below the wing as shown in Figure 9.2. The details of the landing gear integration such as the drag brace mount location have not been specified as the exact dimensions of the landing gear are currently unknown. The landing gear is being purchased off-the-shelf from Lancair.
The landing gear structure is covered in an aerodynamic fairing as shown in Figure 9.3. The nose of the fairing is hinged such that access to the wing attachment is available. Also, this allows the user to access the wiring connections from the inboard to the outboard wing. The wing removal procedure is to open the landing gear pod, remove the wing bolts, and disconnect the antenna and servo wires.
10 Anti-Icing System

Several options have been considered for the anti/de-icing of the Meridian aircraft structure including:

- Engine Bleed Air
- Electrical Heating Elements
- Weeping Wing
- Electro-Expulsive

The requirements for the anti-icing system are:

- The anti/de-icing system must be able to autonomously sense ice accretion on the critical surfaces and respond accordingly
- The system must not interfere with the payload or flight control system
- The aircraft must be able to fly in known icing conditions for at least 1.5 hours continuously
- The system should weigh no more than 40 lbs and require less than 300 W of power

The anti-ice system options have been narrowed to two based on the derived requirements discussed:

- Weeping Wing
- Electro-Expulsive

Two separate vendors have been contacted for these parts and both offer similar solutions. These vendors will be responsible for manufacturing both the wing and v-tail leading edge with the anti-icing system integrated. This means that the leading edge will not need to be manufactured at KU, and the anti-ice system can be changed at a later date by simply replacing the leading edge. At this time, both systems are being pursued.

11 Electrical Power Distribution System

Derived Requirements:

1. provide power to the starter
2. provide heater power for the payload, avionics and battery
3. provide 300W of switched power to the payload
4. provide switched power to the servos (control surfaces, flaps and landing gear)
5. provide appropriately filtered and switched power to the avionics
**System Selection:**

The primary decisions for the power system were the selection of a voltage and the level of redundancy of components. A 28 volt system was selected primarily due to the fact that all acceptable servos require at least 28 volts.

The primary components of a basic on-board power system include an alternator/regulator, a battery and a power distribution unit. Some consideration was given to multiple alternators and batteries, but such a system would require a relatively complex and expensive power distribution unit. Therefore, only one alternator/regulator will be used.

A 28 volt battery assembly will be used, and will have provisions for using both 12 and 6 volt power takeoffs. The battery will be recharged by the alternator/regulator whenever the engine is operating. The battery will be enclosed in an insulated enclosure and will be heated throughout the duty cycle with an electric blanket on which the battery will be positioned. A thermostat will control the enclosure temperature to 60°F. The on-board battery will provide all on-board electrical needs throughout the mission, except the power for the starter. The starter will be powered by a 28 volt on-ground battery assembly on a sled. This same battery will be attached in parallel to the on-board battery from the time the aircraft is switched off a 28-volt power supply in the hangar until it is needed for the engine start, thus providing uninterrupted power and heat to the aircraft.

The on-ground battery assembly will be installed in a Styrofoam box on an American Flyer hand-pulled sled. It will be heated by an electric blanket on which it is positioned. A thermostat will control the box temperature to 60°F.

The power distribution system will include a 300W power relay for the payload, with actuation provided by either the Piccolo or the alternate processor. The rest of the aircraft will be powered by a Blue Mountain Avionics Power Board 2. The Power Board, which will handle up to 750 W, is a popular unit for home-built aircraft in the same weight class as the Meridian. Figure xxx is a sketch of the power system connectivity. Note that there is a master switch, six separately-switched circuits and an avionics bus with filtered power. There are also separate circuits for the flaps and trim servos (which are not included on the current design).
12 Payload System Integration

12.1 Electrical Interface

The payload will include data storage capabilities for recording airborne telemetry throughout the flight. This will include vehicle accelerations, angular rates, as well as the Kalman filtered state complete state estimate. This data will need to be synchronized with the GPS unit included in the payload package, which is separate from the unit used by the autopilot. This will be accomplished utilizing an external avionics package – the NAV 420 (See Section Error! Reference source not found. of this report). These electrical interfaces utilize standard RS-232 interconnects.

12.2 Structural Integration

There are three locations for the vehicle payload indicated in xxx. The primary payload is the radar system which is located in the primary payload bay. The secondary payload is yet undefined, but may include a magnetometer system, a camera, a dropsonde release system, among other possibilities. The third payload area is designed to contain the spatial magnetometer. This piece of equipment is rather sensitive to EMI so the sensitive portion of the equipment will be located in the tail of the aircraft, as far from the avionics and payload system as possible.

Figure 12.1 – Payload Bay Definitions
12.2.1 Primary Payload

The primary payload bay consists of two frames and the firewall, which will be utilized as the primary mounting structure. The frames will be pre-drilled with hole patterns that utilize the same vertical spacing as a standard 19" rack [xxx]. Cross beams will be manufactured from aluminum sheets with matching hole patterns. This will allow for integration of a wide variety of payload packages and flexibility in the positioning of the payload and avionics equipment. Figure 12.2 shows a generic payload integration. The final layout will be developed as the

12.2.2 Secondary Payload

The secondary payload bay is located in the belly of the fuselage directly behind the aft spar frame. There is a cutout in the lower fuselage skin, which is framed by two ring frames and longerons. The cutout was sized to support a wide variety of payloads including a camera, a laser altimeter, and a dropsonde release system.
12.2.3 Antennas

The primary sensor for the Meridian is the ice penetrating radar, which consists of the radar system stored in the payload bay and a number of wing mounted antennas. The antennas are shown in Figure 12.4.
13 Weight and Balance

xxx – Bill/Russ

14 Project Schedule

See Appendix C.

15 Conclusions and Future Work
16 References


Appendix A. Aircraft Tail Loads

See Attached.
Appendix B. Detailed Stress Report

See Attached
Appendix C. Project Schedule

See Attached