Solid State Aircraft
Phase II Project NAS5-03110

Final Report

Prepared for:
Robert A. Cassanova, Director
NASA Institute for Advanced Concepts

May 31, 2005
# Table of Contents

List of Tables .............................................................................................................. iii  
List of Figures .............................................................................................................. v  
List of Contributors .................................................................................................. xv  
Executive Summary ................................................................................................ xvi  

## Chapter 1.0 Introduction ......................................................................................... 1 
  1.1 SSA Concept Description .................................................................................... 1  
  1.2 Flapping Wing Flight  
    1.2.1 Insect Flight ................................................................................................. 5  
    1.2.2 Bird, Mammal and Dinosaur Flight ............................................................. 8  
    1.2.3 Wing Shape ................................................................................................. 12  
  1.3 Mission Capabilities  
    1.3.1 Operation on Earth .................................................................................... 16  
    1.3.2 Operation on Venus ................................................................................... 23  
    1.3.3 Operation on Mars  
      1.3.3.1 Imagery ................................................................................................. 29  
      1.3.3.2 Magnetic Field Mapping and Investigation ............................................. 31  
    1.3.4 Near Infrared and Neutron Spectroscopy ................................................... 33  
    1.3.5 Radar Sounding ........................................................................................... 34  
    1.3.6 In-situ Atmospheric Science ....................................................................... 34  

## Chapter 2.0 SSA Configuration and Operation ....................................................... 37  
  2.1 Wing and Fuselage Material Development  
    2.1.1 IPMC Development Background .................................................................. 39  
    2.1.2 General Considerations on IPMC as SSA Wing Material ......................... 40  
    2.1.3 IPMC Manufacturing Techniques .................................................................. 42  
    2.1.4 Three-dimensional Fabrication of SSA Wing Material  
      2.1.4.1 Imagery .................................................................................................. 47  
      2.1.4.2 Magnetic Field Mapping and Investigation ............................................. 47  
    2.1.5 Electrical Performance ................................................................................ 51  
    2.1.6 Thermodynamic Efficiency ........................................................................ 53  
    2.1.7 Force Density for SSA Wing Materials ...................................................... 55  
    2.1.8 Cold Temperature Properties of IPMC For High Altitude and Planetary Applications  
      2.1.8.1 Near Infrared and Neutron Spectroscopy ............................................... 56  
      2.1.9 Modeling and Simulation ......................................................................... 63  
  2.2 Thin Film Batteries ............................................................................................ 77  
  2.3 Thin Film Array  
    2.3.1 Introduction ................................................................................................. 81  
    2.3.2 How a Solar Cell Works ............................................................................... 81  
    2.3.3 State-of-the-art Thin Film Solar Cells ......................................................... 83  
    2.3.4 Near Term Advancements for Thin film Solar Cell Technology ............... 84  
    2.3.5 Long-term Advancements for Thin Film Solar Cell Technology ............... 85  

## Chapter 3.0 Operational Environments ................................................................... 87  
  3.1 Environmental Conditions for Flight on Venus ................................................. 87  
  3.2 Environmental Conditions for Flight on Earth .................................................. 95  
    3.2.1 East Coast Wind Profiles ............................................................................ 104  
    3.2.2 West Coast Wind Profiles .......................................................................... 104  
  3.3 Environmental Conditions for Flight on Mars ................................................... 107  
    3.3.1 Physical Properties .................................................................................... 109  
    3.3.2 Atmospheric Conditions ........................................................................... 110  
    3.3.3 Dust Storms and Wind .............................................................................. 113  

## Chapter 4.0 SSA Design and Feasibility Analysis .................................................... 117  
  4.1 Aerodynamic Design and Analysis  
    4.1.1 Previous Work on Flapping Flight Aerodynamics ....................................... 117  
    4.1.2 Two-dimensional Airfoil Analysis ................................................................ 121
<table>
<thead>
<tr>
<th>Appendix A: List of References</th>
<th>A-1</th>
</tr>
</thead>
<tbody>
<tr>
<td>Appendix B: Presentations, Media Exposure, and Future Development Interest</td>
<td>B-1</td>
</tr>
<tr>
<td>• Future Development Interest</td>
<td>2</td>
</tr>
<tr>
<td>Appendix C: Planetary Atmosphere Data</td>
<td>C-1</td>
</tr>
<tr>
<td>Appendix D: WPI Student Report</td>
<td>D-1</td>
</tr>
</tbody>
</table>

Phase II Final Report
List of Tables

Table 1-1: Potential SSA mission capabilities and equipment .................................................. 19
Table 2-1: Current Capabilities of IPMC Materials ................................................................. 62
Table 2-2: Rechargeable battery characteristics ....................................................................... 80
Table 3-1: Physical and orbital properties of Venus ................................................................. 88
Table 3-2: Venus atmospheric composition .............................................................................. 94
Table 3-3: Earth’s physical properties ....................................................................................... 96
Table 3-4: Turbidity factor equation coefficients for East Coast U.S. ...................................... 97
Table 3-5: Turbidity factor equation coefficients for East Coast U.S. ...................................... 97
Table 3-6: Major gas components of Earth’s atmosphere ....................................................... 101
Table 3-7: Raw Wind Data Sites ............................................................................................. 103
Table 3-8: Input parameter ranges for wind equations ............................................................ 105
Table 3-9: Physical properties of Mars .................................................................................... 109
Table 3-10: Mars atmosphere composition ............................................................................. 110
Table 4-1: Curved tip flat plate lift coefficient for various angles of attack .................... 135
Table 4-2: WIND and XFOIL generated performance results for the Eppler 378 airfoil ....... 138
Table 4-3: Validation results for the Selig 1223 airfoil ........................................................... 140
Table 4-4: Re and CL for different types of flapping .............................................................. 163
Table 4-5: Parameters used in strain calculations ................................................................. 165
Table 4-6: Strain gage calibration numbers ............................................................................. 165
Table 4-7: Analysis baseline assumptions ............................................................................. 237
Table 4-8: Baseline operational conditions for Earth flight .................................................... 238
Table 4-9: Flight results for the baseline SSA operating on Mars .......................................... 264
Table 4-10: Performance increase required for flight on Mars ................................................. 265
Table 4-11: Baseline performance values for development evaluation ................................. 266
# List of Figures

<table>
<thead>
<tr>
<th>Figure</th>
<th>Description</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>1-1</td>
<td>Artist’s drawing of the Solid State Aircraft concept</td>
<td>1</td>
</tr>
<tr>
<td>1-2</td>
<td>Component Layout of the SSA</td>
<td>2</td>
</tr>
<tr>
<td>1-3</td>
<td>Flap-Glide cycle flight profile</td>
<td>3</td>
</tr>
<tr>
<td>1-4</td>
<td>Artist’s image of a pteradon in flight</td>
<td>3</td>
</tr>
<tr>
<td>1-5</td>
<td>Range of sizes for flying creatures</td>
<td>5</td>
</tr>
<tr>
<td>1-6</td>
<td>Conventional airfoil and insect wing lift generation mechanisms</td>
<td>7</td>
</tr>
<tr>
<td>1-7</td>
<td>Flapping insect wing leading edge vortex formation</td>
<td>7</td>
</tr>
<tr>
<td>1-8</td>
<td>Insect wing lift generation profile</td>
<td>8</td>
</tr>
<tr>
<td>1-9</td>
<td>Diagram of lift generation due to flow over an airfoil</td>
<td>8</td>
</tr>
<tr>
<td>1-10</td>
<td>Wing motion and force vectors during flapping flight</td>
<td>10</td>
</tr>
<tr>
<td>1-11</td>
<td>Illustration of wing motion</td>
<td>11</td>
</tr>
<tr>
<td>1-12</td>
<td>Mechanical wing ontihopter in flight</td>
<td>11</td>
</tr>
<tr>
<td>1-13</td>
<td>Wing shapes from various creatures and their aspect ratios</td>
<td>13</td>
</tr>
<tr>
<td>1-14</td>
<td>Artist’s drawing of a pteranodon</td>
<td>14</td>
</tr>
<tr>
<td>1-15</td>
<td>Selig S1227 airfoil cross-section</td>
<td>14</td>
</tr>
<tr>
<td>1-16</td>
<td>Helios and Pathfinder solar-powered aircraft</td>
<td>16</td>
</tr>
<tr>
<td>1-17</td>
<td>Balloon launch with 2 stowed SSA</td>
<td>17</td>
</tr>
<tr>
<td>1-18</td>
<td>Deployment sequence for the SSA</td>
<td>18</td>
</tr>
<tr>
<td>1-19</td>
<td>SSA observation and surveillance imaging mission capability</td>
<td>20</td>
</tr>
<tr>
<td>1-20</td>
<td>E-Wasp UAV and micro-camera</td>
<td>20</td>
</tr>
<tr>
<td>1-21</td>
<td>Communication links to and from the SSA</td>
<td>21</td>
</tr>
<tr>
<td>1-22</td>
<td>Automotive thin film antenna</td>
<td>21</td>
</tr>
<tr>
<td>1-23</td>
<td>JPL mini-mass spectrometer</td>
<td>22</td>
</tr>
<tr>
<td>1-24</td>
<td>Aeroshell packaging and deployment of a conventional aircraft</td>
<td>23</td>
</tr>
<tr>
<td>1-25</td>
<td>Venus entry and deployment sequence for the SSA</td>
<td>24</td>
</tr>
<tr>
<td>1-26</td>
<td>Venus atmosphere sampling</td>
<td>25</td>
</tr>
<tr>
<td>1-27</td>
<td>Atmospheric sounds of the Venus atmosphere</td>
<td>26</td>
</tr>
<tr>
<td>1-28</td>
<td>Communications relay function for the SSA</td>
<td>27</td>
</tr>
<tr>
<td>1-29</td>
<td>Illustration of a rover aircraft communications link on Venus</td>
<td>28</td>
</tr>
<tr>
<td>1-30</td>
<td>High resolution image of the Mars surface taken by Pathfinder lander</td>
<td>30</td>
</tr>
<tr>
<td>1-31</td>
<td>Example of a wide-angle context camera type image</td>
<td>31</td>
</tr>
<tr>
<td>1-32</td>
<td>Magnetic field shape on Earth and Mars</td>
<td>33</td>
</tr>
<tr>
<td>1-33</td>
<td>Radar sounding for investigating surface and subsurface features</td>
<td>34</td>
</tr>
<tr>
<td>1-34</td>
<td>Atmospheric sampling and data collection over grid</td>
<td>35</td>
</tr>
<tr>
<td>2-1</td>
<td>Smaller version of the SSA with its driving electronics</td>
<td>37</td>
</tr>
<tr>
<td>2-2</td>
<td>Flapping configurations of the smaller version of the SSA</td>
<td>38</td>
</tr>
<tr>
<td>2-3</td>
<td>Larger version of the SSA</td>
<td>38</td>
</tr>
<tr>
<td>2-4</td>
<td>General architecture of the SSA fuselage with SSA wings attached and two pairs of electrodes on each wing</td>
<td>39</td>
</tr>
<tr>
<td>2-5</td>
<td>Various configurations of the IPMC wings activated by the pairs of electrodes on each wing</td>
<td>40</td>
</tr>
<tr>
<td>2-6</td>
<td>Four-fingered robotic grippers made with plastic muscles</td>
<td>41</td>
</tr>
<tr>
<td>2-7</td>
<td>Successive photographs of an IPMC strip showing very large deformation (sample in a and b are 1cmx8cmx0.34mm under 4 volts). Note that t=0.5 sec between a, b</td>
<td>41</td>
</tr>
<tr>
<td>Figure</td>
<td>Description</td>
<td>Page</td>
</tr>
<tr>
<td>--------</td>
<td>-----------------------------------------------------------------------------</td>
<td>------</td>
</tr>
<tr>
<td>2-37</td>
<td>Force improvement by chemical tweaking showing the effect of changing cations</td>
<td>65</td>
</tr>
<tr>
<td></td>
<td>from H+ to Na+ to Li+</td>
<td></td>
</tr>
<tr>
<td>2-38</td>
<td>Experimental evidence for the effect of different ions and their hydration</td>
<td>66</td>
</tr>
<tr>
<td></td>
<td>numbers on the tip force and thus deformation of an IPMC strip</td>
<td></td>
</tr>
<tr>
<td>2-39</td>
<td>General structures of an IPMC or IPCC film with near boundary functionally</td>
<td>68</td>
</tr>
<tr>
<td></td>
<td>graded electrodes and surface electrodes</td>
<td></td>
</tr>
<tr>
<td>2-40</td>
<td>Actuation under a low frequency electric field</td>
<td>69</td>
</tr>
<tr>
<td>2-41</td>
<td>Step response displacement characteristics of IPMC samples (d: arc length,</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>Lo: effective beam length), Lo = 1.0 inch (left) and Lo = 1.5 inch (right)</td>
<td></td>
</tr>
<tr>
<td>2-42</td>
<td>Experimental determination of Onsager coefficient L using three different</td>
<td>71</td>
</tr>
<tr>
<td></td>
<td>samples</td>
<td></td>
</tr>
<tr>
<td>2-43</td>
<td>Variation of curvature versus cross capacitance Cg and time t</td>
<td>73</td>
</tr>
<tr>
<td>2-44</td>
<td>Variation of curvature versus cross resistance Rg and time t</td>
<td>74</td>
</tr>
<tr>
<td>2-45</td>
<td>Variation of curvature versus cross electric field E and time t</td>
<td>74</td>
</tr>
<tr>
<td>2-46</td>
<td>Variation of maximum tip deflection versus average cross capacitance, and</td>
<td>75</td>
</tr>
<tr>
<td></td>
<td>time t</td>
<td></td>
</tr>
<tr>
<td>2-47</td>
<td>Variation of maximum tip deflection versus the average electric field, and</td>
<td>76</td>
</tr>
<tr>
<td></td>
<td>time t</td>
<td></td>
</tr>
<tr>
<td>2-48</td>
<td>Variation of maximum tip deflection versus average cross resistance, and</td>
<td>76</td>
</tr>
<tr>
<td></td>
<td>time t</td>
<td></td>
</tr>
<tr>
<td>2-49</td>
<td>Examples of thin film batteries</td>
<td>77</td>
</tr>
<tr>
<td>2-50</td>
<td>Thin film battery cross-section</td>
<td>78</td>
</tr>
<tr>
<td>2-51</td>
<td>Excellatron lithium battery capacity as a function of charge/discharge</td>
<td>79</td>
</tr>
<tr>
<td></td>
<td>cycles</td>
<td></td>
</tr>
<tr>
<td>2-52</td>
<td>Output voltage for a 1 cm² flexible lithium thin film battery</td>
<td>80</td>
</tr>
<tr>
<td>2-53</td>
<td>Schematic of solar cell operation</td>
<td>82</td>
</tr>
<tr>
<td>2-54</td>
<td>The solar spectrum is split into three spectral bands illustrating spectrum</td>
<td>83</td>
</tr>
<tr>
<td></td>
<td>splitting InGa(A1)P/GaAs/Ge triple junction solar cell</td>
<td></td>
</tr>
<tr>
<td>2-55</td>
<td>Flexible amorphous silicon solar cell on 1-mil Kapton substrate</td>
<td>84</td>
</tr>
<tr>
<td>3-1</td>
<td>Venus from space</td>
<td>87</td>
</tr>
<tr>
<td>3-2</td>
<td>Temperature profile of Venus’s atmosphere</td>
<td>89</td>
</tr>
<tr>
<td>3-3</td>
<td>Temperature profile within Venus’s atmosphere</td>
<td>90</td>
</tr>
<tr>
<td>3-4</td>
<td>Atmosphere density as a function of altitude</td>
<td>91</td>
</tr>
<tr>
<td>3-5</td>
<td>Atmospheric solar attenuation as a function of altitude at 720 nm wavelength</td>
<td>92</td>
</tr>
<tr>
<td>3-6</td>
<td>Average wind speed versus altitude within the Venus atmosphere</td>
<td>93</td>
</tr>
<tr>
<td>3-7</td>
<td>Speed of sound as a function of altitude within the Venus atmosphere</td>
<td>94</td>
</tr>
<tr>
<td>3-8</td>
<td>Earth from space</td>
<td>95</td>
</tr>
<tr>
<td>3-9</td>
<td>Profile of Earth’s Atmosphere</td>
<td>99</td>
</tr>
<tr>
<td>3-10</td>
<td>Troposphere region of the atmosphere</td>
<td>100</td>
</tr>
<tr>
<td>3-11</td>
<td>Mean wind speeds for Albuquerque, New Mexico</td>
<td>102</td>
</tr>
<tr>
<td>3-12</td>
<td>Mean wind speeds for Cape Kennedy, Florida</td>
<td>102</td>
</tr>
<tr>
<td>3-13</td>
<td>Winter mean wind speed profile for the East Coast</td>
<td>105</td>
</tr>
<tr>
<td>3-14</td>
<td>Mean east coast wind speed for an altitude of 21.5 km</td>
<td>106</td>
</tr>
<tr>
<td>3-15</td>
<td>Mean west coast wind speed for an altitude of 21.5 km</td>
<td>106</td>
</tr>
<tr>
<td>3-16</td>
<td>Mars from Space</td>
<td>107</td>
</tr>
<tr>
<td>3-17</td>
<td>Image of Mars atmosphere taken from Pathfinder lander</td>
<td>108</td>
</tr>
<tr>
<td>3-18</td>
<td>Orbital image of surface features on Mars</td>
<td>108</td>
</tr>
<tr>
<td>3-19</td>
<td>Image of Mars surface from Pathfinder lander</td>
<td>109</td>
</tr>
</tbody>
</table>
Solid State Aircraft

Figure 3-20: Daily pressure variation (Pathfinder data) .......................................................... 112
Figure 3-21: Pressure variation over a one-month period (Pathfinder data) ...................... 112
Figure 3-22: Atmosphere temperature variation throughout a day (Pathfinder data) ....... 113
Figure 3-23: Wind direction throughout the day (Pathfinder data) ....................................... 114
Figure 3-24: Measurements taken during a dust devil (Pathfinder data) ........................... 115
Figure 4-1: The coordinate system used to describe wing motion. The pitch axis, y, sweeps in the
XY plane of the inertial XYZ coordinate system, while the wing pitches about the y-axis of the wing-fixed xyz coordinate system ..................................................... 119
Figure 4-2: Lift vs. drag coefficient validation data for XFOIL .............................................. 122
Figure 4-3: Lift coefficient vs. angle of attack validation data for XFOIL ............................. 123
Figure 4-4: Moment coefficient vs. angle of attach validation data for XFOIL ...................... 123
Figure 4-5: XFOIL generated lift vs. drag coefficient for various airfoils ............................ 124
Figure 4-6: XFOIL generated lift coefficient vs. angle of attack for various airfoils ............ 124
Figure 4-7: XFOIL generated pitching moment coefficient vs. angle of attack for various airfoils
125
Figure 4-8: Candidate thin airfoil sections .............................................................................. 126
Figure 4-9: Mach number contours around the Selig 1091 airfoil at Reynolds number 100,000;
Mach number 0.19; and 6 degrees angle of attack ............................................................ 127
Figure 4-10: Static pressure contours around the Selig 1091 airfoil at Reynolds number 100,000;
Mach number 0.19; and 6 degrees angle of attack ............................................................ 128
Figure 4-11: Lift coefficient vs. angle of attack for WIND and XFOIL computations ........ 128
Figure 4-12: 299 x 100 grid used for the CFD calculations .................................................... 129
Figure 4-13: Streamlines around the Selig 1091 airfoil at Reynolds number 100,000; Mach num-
ber 0.19; and 18 degrees of angle of attack ........................................................................ 130
Figure 4-14: Drag coefficient vs. angle of attack for WIND and XFOIL computations ...... 130
Figure 4-15: 0\degree angle of attack static pressure distribution .......................................... 131
Figure 4-16: 0\degree angle of attack velocity vectors .............................................................. 131
Figure 4-17: 0\degree angle of attack modified turbulent viscosity ........................................ 132
Figure 4-18: 0\degree angle of attack lift coefficient .............................................................. 132
Figure 4-19: 15\degree angle of attack static pressure variation .................................................. 133
Figure 4-20: 15\degree angle of attack modified turbulence intensity (nt) .............................. 133
Figure 4-21: 15\degree angle of attack velocity vectors .............................................................. 134
Figure 4-22: Lift coefficient at 15\degree angle of attack (steady state value is around 1.15) ...... 134
Figure 4-23: Lift coefficient for 10\degree angle of attack .......................................................... 135
Figure 4-24: Eppler 378 airfoil CFD calculation grid ............................................................. 136
Figure 4-25: Mach number contours for E378 airfoil at 4\degree angle of attack .................... 137
Figure 4-26: Mach number contours for E378 airfoil at 12\degree angle of attack .................... 137
Figure 4-27: Eppler 378 airfoil CFD calculation grid ............................................................. 138
Figure 4-28: Selig 1223 airfoil CFD calculation grid .............................................................. 139
Figure 4-29: Mach number contours for S1223 airfoil at 6\degree angle of attack ................. 140
Figure 4-30: Hybrid mesh used in the present simulation. The axis of the elliptical outer boundary is inclined to capture as much of the wake as possible and make the outflow boundary condition more accurate. Note the large size of the domain compared to the airfoil .............................................................................................................. 141
Figure 4-31: Part of the grid close to the airfoil surface .......................................................... 141
Figure 4-32: A magnified view of the hybrid grid at the airfoil leading edge ....................... 142
Figure 4-33: Velocity vectors; the LEV is still attached to the top surface .................................. 142
Figure 4-34: Static pressure distribution; the leading edge vortex can be clearly seen ............. 143
Figure 4-35: Lift variation, each cycle represents a series of events consisting of leading edge and trailing edge formation, transport and shedding ................................................. 143
Figure 4-36: Drag variation; note the nearly 180 deg. phase difference between the lift and the drag peaks .......................................................... 144
Figure 4-37: Grid for turbulent simulations ..................................................................................... 145
Figure 4-38: Static pressure contours, $\alpha = 5^\circ$, Re = 100,000 .............................................. 145
Figure 4-39: Velocity vectors, $\alpha = 5^\circ$, Re = 100,000 .......................................................... 146
Figure 4-40: Turbulent viscosity, $\alpha = 5^\circ$, Re = 100,000 ..................................................... 146
Figure 4-41: Pressure contours, $\alpha = 10^\circ$, Re = 100,000. Note that the LEV is larger compared to the small angle-of-attack case. The LEV is just detaching from the surface and the TEV is forming at this instant. ............................................................................ 147
Figure 4-42: Turbulent viscosity, $\alpha = 10^\circ$, Re = 100,000. The vortex cores show high turbulence levels. ..................................................................................................................... 147
Figure 4-43: Velocity vectors, $\alpha = 10^\circ$, Re = 100,000. The vortex structure is clearly visible. The vortex structure is characterized by several secondary vortices. These also manifest as high frequency components in the lift-curve. ............................................ 148
Figure 4-44: Static pressure contours, $\alpha = 15^\circ$, Re = 100,000. ........................................ 148
Figure 4-45: Velocity vectors, $\alpha = 15^\circ$, Re = 100,000. ......................................................... 149
Figure 4-46: Turbulent viscosity, $\alpha = 15^\circ$, Re = 100,000. .................................................... 149
Figure 4-47: Lift variation, $\alpha = 10^\circ$, Re = 100,000. ............................................................ 150
Figure 4-48: Pitch schedule during half-cycle. A sinusoidal profile (top) has been used in the preliminary simulations. The top hat profile may have advantages over the sinusoidal profile. .................................................... 153
Figure 4-49: Starting mesh ........................................................................................................... 154
Figure 4-50: Mesh at the end of one cycle .................................................................................... 154
Figure 4-51: Static pressure contours at the end of one cycle .................................................. 155
Figure 4-52: Velocity vectors at the end of one cycle .............................................................. 155
Figure 4-53: Mesh at phase angle $\pi/2$ ...................................................................................... 156
Figure 4-54: Static pressure contours at phase angle $= \pi/2$ ............................................................ 156
Figure 4-55: Velocity vectors at phase angle $\pi/2$ .................................................................... 157
Figure 4-56: Schematic of the experimental setup ...................................................................... 158
Figure 4-57: Flapping wing mechanism ...................................................................................... 159
Figure 4-58: Illustration of stroke angle, $\Psi$. ........................................................................ 159
Figure 4-59: Side view of motor, encoder, and wheel ................................................................. 160
Figure 4-60: Photograph of wing with strain gauge ................................................................. 160
Figure 4-61: Pin angular position .............................................................................................. 161
Figure 4-62: Timing of trigger, flapping, and pitching with respect to $\dot\alpha$ .............................. 161
Figure 4-63: Wing position controlled by servo ......................................................................... 162
Figure 4-64: Calibration of strain indicator ............................................................................... 166
Figure 4-65: Lab view calibration diagram .................................................................................. 167
Figure 4-66: Normal force at mid-stroke for flapping-only motion ........................................... 167
Figure 4-67: Normal force at mid-stroke for pitching flapping .................................................. 168
Figure 4-68: Camera positioning for flow visualization .............................................................. 169
Figure 4-69: Flow visualization image plane ................................................................................ 169
Figure 4-70: Preliminary photograph of the wake vortex system created by a flapping wing viewed from the wing tip. Hydrogen bubbles are used as tracer particles. The bright streak on the left is the wing tip. The image resolution has been degraded while importing from AVI format to jpg format .............................................................. 170

Figure 4-71: Preliminary velocity vectors from PIV image interrogation of the water flow created by a flapping wing model conducted in the UMR Fluid Dynamics Lab. .......... 170

Figure 4-72: The photograph shows a wing fabricated in our lab by mounting a flexible membrane material on a frame made of thin stainless steel sheet. This wing was fabricated to study dynamic camber variation. .............................................................. 171

Figure 4-73: Plunging (instead of flapping) and pitching motion schematic. Wing translates in the x-direction in oscillatory mode (also known as plunging or heaving) and undergoes pitching about the y-axis. The DC motor, encoder and wheel arrangement is shown below. The wheel position is provided by the encoder. Trigger signal from the encoder is used to actuate the servomotor. The wing pitches in one direction when the signal turns high and pitches in the opposite direction when the signal turns low. .. 172

Figure 4-74: Normal Force, Lift, and Drag for various Fixed Angle of Attack (flapping frequency, \( f = 0.242 \) Hz) ....................................................................................................... 173

Figure 4-75: Fixed angle of attack and symmetric pitching (\( f = 0.232 \) Hz). ....................... 175

Figure 4-76: Vector diagrams for wing motion during upstroke ............................................. 178

Figure 4-77: View of wing shape from observer looking in flight direction ............................ 179

Figure 4-78: Lift coefficient validation data for a cambered plate, 1.93% thickness, 4% camber, leading edge radius of .6, Re= 140,000 ............................................................... 181

Figure 4-79: Drag coefficient validation data for a cambered plate, 1.93% thickness, 4% camber, leading edge radius of .6, Re= 140,000 ............................................................... 181

Figure 4-80: Lift-drag ratio versus angle of attack for a cambered plate: 1.93% thickness, 4% camber, leading edge radius of .6, Re=140,000 .............................................. 182

Figure 4-81: Lift coefficient validation data for a flat plate: 1.96% thickness, 0% camber, leading edge radius of .6, Re=80,000 ............................................................... 182

Figure 4-82: Drag coefficient validation data for a flat plate: 1.96% thickness, 0% camber, leading edge radius of .6, Re=80,000 .................................................. 183

Figure 4-83: Lift-to-drag ratio versus effective angle of attack for a flat plate: 1.96% thickness, 0% camber, leading edge radius of .6, Re=80,000 .............................................. 183

Figure 4-84: Final optimized airfoil geometry: 2% thickness and 50% location, 5% camber at 20% location, .15% leading edge radius, and Re=100,000 ................................. 184

Figure 4-85: Lift and drag coefficients for optimum airfoil: 2% thickness and 50% location, 5% camber at 20% location, .15% leading edge radius, and Re=100,000 .............. 184

Figure 4-86: Environmental factors that affect the SSA’s power production capabilities ..... 186

Figure 4-87: Wing shape and geometry ................................................................. 187

Figure 4-88: Solar array incident flux geometry .......................................................... 188

Figure 4-89: Wing geometry ................................................................................. 190

Figure 4-90: Wing selection chord distribution .......................................................... 191

Figure 4-91: Effect of temperature on efficiency of various types of solar cells ................. 192

Figure 4-92: Light spectrum and absorption characteristics of a thin film solar array .......... 193

Figure 4-93: Venus: available power throughout a day at various latitudes ..................... 194

Figure 4-94: Earth: available power throughout the day at 0o latitude ............................ 194
Figure 4-95: Earth: available power throughout the day at 20° latitude ........................................ 195
Figure 4-96: Earth: available power throughout the day at 40° latitude ........................................ 195
Figure 4-97: Earth: available power throughout the day at 60° latitude ........................................ 196
Figure 4-98: Earth: available power throughout the day at 80° latitude ........................................ 196
Figure 4-99: Mars: available power throughout the day at 0° Latitude ......................................... 197
Figure 4-100: Mars: available power throughout the day at 20° latitude ....................................... 197
Figure 4-101: Mars: available power throughout the day at 40° latitude ....................................... 198
Figure 4-102: Mars: available power throughout the day at 60° latitude ....................................... 198
Figure 4-103: Mars: available power throughout the day at 80° latitude ....................................... 199
Figure 4-104: Available power at 0o latitude for various attitudes and wing angles ..................... 200
Figure 4-105: Available power: June 21st, 30o latitude for various attitudes and wing angles ........... 200
Figure 4-106: Available power: June 21st, 60o latitude for various attitudes and wing angles ........... 201
Figure 4-107: Available power: Dec. 21st, 30o latitude for various attitudes & wing angles ............. 202
Figure 4-108: Available power: Dec. 21st, 60o latitude for various attitudes & wing angles ............. 202
Figure 4-109: Output power variation along the wing on June 21st, 0o latitude, 0o attitude .......... 203
Figure 4-110: Output power variation along the wing on June 21st, 45o latitude, 0o attitude .......... 204
Figure 4-111: Output power variation along the wing on June 21st, 45o latitude, 90o attitude ......... 204
Figure 4-112: Output power variation along the wing on Dec. 21st, 45o latitude, 90o attitude ........... 205
Figure 4-113: Analysis Flow Diagram ..................................................................................... 206
Figure 4-114: Cruise flight speed as a function of atmospheric density ......................................... 207
Figure 4-115: Cruise Reynolds number per chord length versus atmospheric density ............... 208
Figure 4-116: Lift coefficient vs. angle of attack for a curved flat plate at various low Reynolds numbers ................................................................. 208
Figure 4-117: Lift coefficient vs. drag coefficient for a curved flat plate at various low Reynolds numbers ........................................................................................................... 209
Figure 4-118: Eppler E377 airfoil cross section ........................................................................... 210
Figure 4-119: Eppler 377 airfoil lift & drag coefficients as a function of angle of attack .............. 210
Figure 4-120: Power consumption mechanisms for the SSA ..................................................... 211
Figure 4-121: Examples of potential flap/glide combinations ...................................................... 212
Figure 4-122: Wing bending motion and geometry ..................................................................... 213
Figure 4-123: Force and mass distribution along a wing section (for a 5 m wingspan) ................. 214
Figure 4-124: Maximum segment angle and acceleration along the wing section .................... 215
Figure 4-125: Diagram of acceleration profile throughout the flap cycle .................................. 216
Figure 4-126: Force distribution along the wing section at various operating points .................. 217
Figure 4-127: Acceleration force diagram .................................................................................. 218
Figure 4-128: Force along the wing as a function of distance traveled by each segment of the wing section .................................................................................................................. 219
Figure 4-129: Drag components acting on the SSA ................................................................. 220
Figure 4-130: Wing segment velocity due to motion for 60o maximum flap angle, 4s flap period & 6m wingspan vehicle ........................................................................................................ 222
Figure 4-131: Wing segment velocity due to motion for 45o maximum flap angle, 4s flap period
Figure 4-132: Wing segment velocity due to motion for 60° maximum flap angle. 2s flap period & 6 m wingspan vehicle ................................................................. 222
Figure 4-133: Wing segment velocity due to motion for 60° maximum flap angle. 4s flap period & 10 m wingspan vehicle ........................................................................ 223
Figure 4-134: Lift generation .................................................................................. 224
Figure 4-135: Velocity vector angle of attack throughout the flap cycle for a horizontal wing .... 225
Figure 4-136: Lift force generation due to flapping ..................................................... 226
Figure 4-137: Wing angle of attack throughout the flap cycle at various stations .............. 227
Figure 4-138: Wing twist profile at various stations throughout the flap cycle ................. 228
Figure 4-139: Induced drag coefficient along the wing & through the flap cycle .............. 229
Figure 4-140: Lift coefficient along the wing & through the flap cycle ......................... 230
Figure 4-141: Incremental lift at various wing stations throughout the flap cycle ............ 231
Figure 4-142: Incremental drag at various wing stations throughout the flap cycle ........... 231
Figure 4-143: Total lift and drag generated throughout the flap cycle ............................ 232
Figure 4-144: Vertical lift and thrust generation throughout the flap cycle ...................... 233
Figure 4-145: Vertical lift at various stations along with wing & throughout the flap cycle .. 234
Figure 4-146: Thrust at various stations along the wing & throughout the flap cycle ........ 234
Figure 4-147: Lift/glide cycle for the SSA operation ................................................... 235
Figure 4-148: Iterative diagram for glide/flap analysis ............................................... 236
Figure 4-149: Required and available energy as a function of flap duration for the baseline operational conditions ......................................................... 238
Figure 4-150: Required and available thrust as a function of flap duration for the baseline operational conditions ..................................................... 239
Figure 4-151: Magnified view of the required thrust and power crossover ....................... 240
Figure 4-152: Required and available energy for base configuration at 14.5 km altitude ..... 241
Figure 4-153: Delta energy for various latitudes on 3/21 (10 m wingspan, 10 km altitude) ... 242
Figure 4-154: Delta energy for various latitudes on 6/21 (10 m wingspan, 10 km altitude) .. 243
Figure 4-155: Delta energy for various latitudes on 9/21 (10 m wingspan, 10 km altitude) ... 244
Figure 4-156: Delta energy for various latitudes on 12/21 (10 m wingspan, 10 km altitude) . 245
Figure 4-157: Delta energy at various altitudes for a 10 m wingspan aircraft .................. 246
Figure 4-158: Delta thrust at various altitude for a 10 m wingspan aircraft ....................... 246
Figure 4-159: Delta energy at various altitude for a 15 m wingspan aircraft .................... 247
Figure 4-160: Delta thrust at various altitudes for a 15 m wingspan aircraft ..................... 247
Figure 4-161: Delta energy at various altitudes for a 20 m wingspan aircraft .................... 248
Figure 4-162: Delta thrust at various altitudes for a 20 m wingspan aircraft ..................... 249
Figure 4-163: Delta energy at various altitudes for a 30 m wingspan aircraft .................... 249
Figure 4-164: Delta thrust at various altitudes for a 30 m wingspan aircraft ..................... 250
Figure 4-165: Delta energy at various altitudes for a 50 m wingspan aircraft .................... 250
Figure 4-166: Delta thrust at various altitudes for a 50 m wingspan aircraft ..................... 251
Figure 4-167: Maximum achievable altitude for 0° latitude throughout the year .............. 252
Figure 4-168: Maximum achievable altitude for 15° latitude throughout the year .............. 252
Figure 4-169: Maximum achievable altitude for 30° latitude throughout the year .............. 253
Figure 4-170: Maximum achievable altitude for 45° latitude throughout the year .............. 253
Figure 4-171: Maximum achievable altitude for 60° latitude throughout the year .............. 254
Figure 4-172: Maximum achievable altitude for 75° latitude throughout the year ............... 254
Figure 4-173: Flap & glide durations for maximum altitude operation at 0° latitude .......... 256
Figure 4-174: Flap & glide durations for maximum altitude operation at 15° latitude .......... 256
Figure 4-175: Flap & glide durations for maximum altitude operation at 30° latitude .......... 257
Figure 4-176: Flap & glide durations for maximum altitude operation at 45° latitude .......... 257
Figure 4-177: Flap & glide durations for maximum altitude operation at 60° latitude .......... 258
Figure 4-178: Flap & glide durations for maximum altitude operation at 75° latitude .......... 258
Figure 4-179: Maximum and minimum altitude range for various latitudes .................... 261
Figure 4-180: Flap and glide duration at the maximum altitude for various latitudes ........ 262
Figure 4-181: Flap and glide duration at the minimum altitude for various latitudes ........ 263
Figure 4-182: Effect on SSA size of variations in critical performance parameters .......... 267
Figure 4-183: Percent change in wingspan as a function of percent change in parameter performance ................................................................. 268
List of Contributors

OAI/Northland Scientific
   Mr. Anthony Colozza (PI)
   Ohio Aerospace Institute/Northland Scientific, Inc.
   21000 Brookpark Road
   Mail Stop 309-1
   Cleveland, Ohio 44135
   Email: Anthony.Colozza@grc.nasa.gov
   Phone: 216-433-5293

OAI
   Mr. Phillip Jenkins
   Mr. Curtis Smith
   Ms. Terri Deacey

Environmental Robots, Inc.
   Dr. Mohsen Shahinpoor

University of Missouri-Rolla
   Dr. Kakkattukuzhy Isaac

Worcester Polytechnic Institute
   Dr. David Olinger
   Mr. Andrew Day
   Ms. Lindsey Robbins

Independent Consultant
   Mr. Teryn DalBello
Executive Summary

A revolutionary type of unmanned aircraft may now be feasible, due to recent advances in polymers, photovoltaics, and batteries. This is a "solid-state" aircraft (SSA), with no conventional mechanical moving parts. Airfoil, propulsion, energy production, energy storage and control are combined in an integrated structure.

The work performed on this concept was focused on determining the feasibility and applicability of this type of solid-state aircraft. Because the aircraft is solar powered, Earth, Venus and Mars were identified as potential locations where the aircraft might be capable of operating. Environmental models of these three planets were produced. These models define the key aspects of the atmospheres as functions of altitude and are utilized in determining the flight envelope for the SSA. Mission applications were also identified for each operational location. These applications took into consideration the potential flight envelope of the aircraft and its the payload carrying capacity. The unique structure of the aircraft provides a number of operational benefits over conventional solar powered aircraft. The aircraft structure is very durable and flexible, unlike the lightweight and fragile structure of a conventional solar powered aircraft. This enables the SSA to be deployed under any weather condition and enables the aircraft to be compactly stowed prior to launch. It is envisioned that for Earth operation the SSA would be carried to altitude on a balloon. Since the aircraft is flexible it could be wrapped around a gondola and then released once the balloon reaches altitude. This flexible stowage capability provides a significant benefit for planetary applications. It enables the SSA to be easily placed within an aeroshell by wrapping it around a central support. The deployment of the SSA is accomplished by simply unwrapping from the support and dropping out of the aeroshell.

The key material of this concept is an ionic polymeric-metal composite (IPMC) that provides the source of control and propulsion. This material has the unique capability of deforming in an electric field like an artificial muscle, and returning to its original shape when the field is removed. Combining the IPMC with emerging thin-film batteries and thin-film photovoltaics provides both energy source and storage in the same structure. The aircraft is constructed by layering the thin film battery and thin film array onto the IPMC material. An electrode grid is etched onto the IPMC to provide the desired electric field. By varying the electric field shape and strength the deformation of the IPMC material and therefore wing can be controlled. A computer control system would be utilized to provide the correct voltage to the electrodes thereby generating the desired field. It is envisioned that the aircraft would have thousands or even tens of thousands of electrodes. The greater the number of electrodes the finer the possible wing motion would be. This fine wing motion is needed in order to produce efficient flapping flight and to be able to truly reproduce the wing motion seen in birds and other flying creatures.

The IPMC material is a fairly new material that is somewhat still within the development stage. Operational samples of the material exist and have been utilized to demonstrate its capabilities and potential. For this program the largest known sheets of the material have been produced (these sheets measured 46 cm X 21 cm). The sheets were produced to evaluate the manufacturability of the IPMC material as well as produce a demonstration wing section. In addition to its ability to move within the presence of an electric field, the IPMC has another property that is critical to the operation and control of the SSA. If bent or deformed the IPMC material will pro-
duce a voltage. This voltage can be read by the same electrode grid that is utilized to generate the wing motion. This characteristic produces an inherent feedback system. The operation of the electrode grid and the feedback is very similar to the operation of a nervous system in biological creatures.

The power for the aircraft to operate comes from the solar array. The solar array collects energy from sunlight. Thin film solar arrays are light-weight and highly flexible. This type of array technology is a perfect fit for the SSA application. The output of the solar array will fluctuate continuously during the flapping motion of the wings. Therefore in order to produce continuous power for the electric field generation and other aircraft systems a thin film lithium battery is utilized. The energy collected from the solar array is stored in this battery until it is needed to flap the wings. By incorporating an energy storage system, the total energy collected by the solar array during the time when the aircraft is both gliding and flapping can be utilized. This ability to collect and utilize energy throughout the complete flap and glide cycle is a key aspect to the energy balance characteristics of the design.

Both the array and battery are thin film devices. That is they are constructed through the deposition of various materials onto a substrate. The ultimate goal for the integration of these three component materials would be to utilize the IPMC as the substrate for both the thin film array and the thin film battery. In this approach all of the active layer of those components would be deposited directly onto the IPMC providing a truly integrated and composite structure. The integration of the IPMC, thin film array and thin film battery would enable the wings to flap thereby providing propulsion and control for the aircraft. With a flight profile similar to a hawk or eagle, the Solid State Aircraft will be able to soar for periods of time and utilize flapping to regain lost altitude.

The actual required motion of the wing will be dictated by the wing aerodynamics. The aerodynamics of a flapping wing is very complicated. The twisting and bending of the wing produces a number of three-dimensional effects that require extensive computer capabilities to model. Under this program the goal of our aerodynamic analysis was to provide airfoil and wing performance data that could then be utilized to produce an analytical model of the wing motion. This analytical model would subsequently be utilized to determine the basic performance of the wing, estimating the lift and drag throughout the flapping motion. This would in turn be used to determine the flight capabilities of the aircraft and its power requirements.

To provide aerodynamic data, a computer fluid dynamic (CFD) model of various two-dimensional airfoils was generated utilizing the flight conditions that would be experienced by the flapping wing. This analysis provided lift coefficient and drag coefficient data on each candidate airfoil over a range of airfoil angle of attack values and operational Reynolds numbers. The airfoil selection was restricted to thin cambered airfoils including curved flat plates. Based on the data generated the Eppler 378 airfoil was for the subsequent feasibility analysis. In addition to the steady state two-dimensional analysis a variable flow field two-dimensional aerodynamic analysis was also performed. This analysis provided some insight into the aerodynamic effects of a flapping wing and set the ground-work for a full three-dimensional CFD analysis of a flapping wing.
Executive Summary

Using the basic configuration of the aircraft, its aerodynamic characteristics and the environmental models that were produced a feasibility analysis was performed. The objective of the analysis was to determine the flight capabilities of the SSA at the three potential planets of operation (Earth, Venus and Mars). This analysis was based on an energy balance over a complete flap/glide cycle. The energy balance determines if there is sufficient power produced to meet the power consumption requirements of the aircraft over the complete flap/glide cycle. If the energy balance works then that specific aircraft configuration would be capable of operating under the specified conditions. To evaluate this the power produced and power consumed by the aircraft over a flap/glide cycle were calculated.

The available power from the solar array was modeled. This model took into account the motion of the wings and orientation of the aircraft as well as the environmental and operational conditions such as time of day, time of year, latitude and altitude. The power required by the aircraft was based on the power needed to move the wings, overcome the drag produced and operate the specified aircraft systems. By varying the glide duration, flap duration, wing length and wing motion of travel a number of design configurations were produced to enable flight over a range of latitudes and times of the year on Earth and Venus. The results for Earth operation, using a number of baseline assumptions, produce aircraft that could fly from near the surface up to an altitude of just over 20 km. For flight on Venus there was an upper and lower altitude limit. The upper limit was just over 70 km where as the lower limit was around 32 km. The lower limit is due to the thick cloud cover on the planet and the significant increase in temperature with decreasing altitude. On Mars there were no solutions found utilizing the baseline assumptions. However, by varying these assumptions and increasing items such as IPMC material efficiency or photovoltaic array efficiency, aircraft configurations were produced that would be capable of flight on Mars. The results produced by varying the baseline assumptions also provide insight into what variables have the greatest effects in increasing the SSA's flight envelope and performance for all planets of operation. This analysis indicated that the IPMC mass followed by array efficiency were the two main factors in increasing the aircraft’s performance capability.
Chapter 1.0 Introduction

1.1 SSA Concept Description

Due to the recent advancements in photovoltaics, batteries, and polymer materials, a unique type of unmanned aircraft may be feasible. This is a "solid state" aircraft (SSA) with no moving parts. An artist rendering of the concept is shown in Figure 1-1.

Figure 1-1: Artist’s drawing of the Solid State Aircraft concept

The unique structure combines aerodynamic lift, propulsion, energy collection, energy storage and control. Thin-film solar arrays are used to collect sunlight and produce power that is stored in a thin-film lithium battery. This power is used to fly the aircraft by setting up an electromagnetic field (EMF) along the wing of the vehicle. The wing, made with ionic polymeric-metal composite (IPMC) synthetic muscles, bends in the presence of this EM field producing the desired flapping motion. This layering of the various component materials is shown in Figure 1-2. This aircraft would fly in a similar fashion to a hawk or an eagle. It would glide for long distances and flap infrequently to regain altitude. The solid-state nature of the aircraft allows it to be very robust, extremely lightweight and capable of flight unlike any other present day air vehicle.
This unique material composition of the aircraft is what enables the flapping motion of the wing to be utilized as the main means of propulsion, thereby eliminating the need for a more conventional propulsion system. As mentioned, the aircraft flight motion will consist of an intermittent flapping and periods of gliding. During the flapping portion of the flight, the aircraft will gain altitude. Then, during the gliding portion, it will glide back down to the starting altitude. This cycle is shown in Figure 1-3.

Due to the estimated low wing loading of the aircraft, it will be able to soar for periods of time and utilize the wing flapping to regain lost altitude. The ratio of the flap time to the gliding time will depend on the available power, power consumption rate and flight conditions. The flap to glide ratio is a critical aspect of the vehicle optimization. Various combinations of glide times to flap times can be utilized. During gliding, the wing shape can be altered to enable steering and control of the aircraft. This control mechanism is similar to that of gliding birds, changing the angle of attack and/or wing shape to produce directional lift on a given wing. This variation in shape can be achieved by utilizing a grid of electrodes that are computer controlled and incorporated into the wing structure. The voltage potential can be varied over the grid thereby tailoring the electric field generated to produce a non-uniform bending in the wing. The variation in lift between the wings can be used to steer and control the aircraft. The force vectors generated by the wing are shown in Figure 1-10 for the upstroke and downstroke.
Chapter 1.0 Introduction
1.1 SSA Concept Description

This type of air vehicle has a number of potential applications as a research platform on both Earth and other planetary bodies. Because of its projected relatively small mass and flexibility, the aircraft is ideal for planetary exploration. These characteristics allow the aircraft to be easily stowed and launched at a minimal cost. Potentially, a fleet of these aircraft could be deployed within a planet's atmosphere and used for comprehensive scientific data gathering/observation or as communications platforms. A complete planetary science gathering or communications/navigation architecture can be built around these lightweight, easily deployable, robust aircraft.

Like all flapping wing flyers in nature the SSA will operate within a low Reynolds number flight regime. This is due mainly to its low wing loading and the potential for high altitude operation where the air density is low. Selecting the correct wing geometry and airfoil properties is a key aspect in providing good aerodynamic performance and minimizing power consumption. To minimize mass as well as be compatible with the IPMC material properties the wing will have a thin cross-section. The overall shape of the wing will be optimized to the flapping motion of the wing. The initial wing design point is based on that of a pteranodon. The Pteranodon, shown in Figure 1-4, was the largest flying creature that ever lived on Earth [34]. Its thin membrane wing and its estimated wing loading are similar to that of the SSA. Because of

Figure 1-3: Flap-Glide cycle flight profile

Figure 1-4: Artist’s image of a pteradon in flight
this and the realization that nature has a way of finding the optimal design configuration, the Pteranodon is a good starting point for the SSA wing design.

Because the sun is the main power source the SSA will need to operate at locations that have sufficient available solar radiation. However, utilizing the sun to produce power means that oxygen is not required for the operation of this aircraft. This is a large benefit, compared to conventional powered aircraft, in applying this concept to atmospheres outside of Earth. Potentially, the inner planets of the solar system with atmospheres (Venus, Earth and Mars) would be places where this type of aircraft could be utilized.

There are a number of potential applications for the SSA from planetary exploration to a quickly deployable Earth observation and communication system. By integrating a thin film antenna to the underside of the SSA, communications between the aircraft and the surface can be achieved. Also transparent metallic antenna technology would allow for the installation of an antenna on the upper surface of the aircraft on top of the solar array. This would enable the SSA to communicate to a satellite. An illustration of this capability is shown in Figure 1-21. These antenna can also be used for science data gathering by providing a means of sounding the atmosphere. Other potential science and data gathering capabilities include, high resolution and context camera imagery, atmospheric measurements, magnetic field measurements, communications relay transmitter / receiver, atmospheric sounding, beacon.

The technology to produce this type of aircraft is presently available. There have been great advances in recent years in each of the three main components areas that make up the aircraft (thin-film photovoltaic arrays, thin-film batteries, and polymer composites). Because of these advances this type of aircraft may now be possible.

1.2 Flapping Wing Flight

"The bird is a machine that operates according to mathematical law."

- Leonardo da Vinci

Flapping wing flight has fascinated humanity for centuries, as can be inferred from Leonardo da Vinci's efforts to build some of the earlier models. Such flying machines are called ornithopters. Birds and machines are subject to the same rules, which we call the aerodynamic laws of nature. Even though nature is far ahead in many areas, technology evolves much faster than plants and animals. We'll fly as birds do, and then we'll do it better than birds. That's one of the things that make it so interesting to work on a project such as the solid state aircraft, with active materials such as ionic polymeric artificial muscles and fly like birds or bats. It is because these materials can deform correctly when the flapping is upward and stretch out to flap correctly when the flapping is downward to create a net lift. Historically there was a dinosaur (Pteranodon) that flew by doing exactly that. Early failed attempts at flapping wing flight, in the 1800s, convinced many people that humans could not fly by flapping wings. However, all it really proved was that they didn't yet have the technology to succeed at this difficult task. More recently, manned ornithopters such as Vladimir Toporov's ornithopter Giordano have made successful, controlled flights with a pilot onboard.
Chapter 1.0 Introduction

1.2 Flapping Wing Flight

The main goal in our field is to mimic bird or insect flight more closely at its own scale. Current challenges include improved flight efficiency and learning to take advantage of the potential maneuverability that flapping wings can offer. Thus, the right wing material is a key and our NIAC/SSA team has selected a polymeric artificial muscle to provide the accurate wing motion necessary to mimic flapping flight.

![Figure 1-5: Range of sizes for flying creatures](image)

Flapping flight is nature's means of producing creatures that can fly. This type of flying has been perfected over the millennia and has been adapted by numerous types and sizes of creatures. The most striking variation in the creatures that can fly is in their size. Sizes range from tiny insects, less than a millimeter across, to the giant flying dinosaurs that had wingspans of many meters as shown in Figure 1-5.

The key to flapping flight is in the motion of the wing. For a bird, bat or the large dinosaurs this wing motion has evolved to produce lift and thrust throughout the flap cycle while minimizing drag thereby conserving energy. The wing motion of these creatures is very complex. It consists of bending, twisting and deformation in the wing shape (chord-wise to adjust wing chamber). Precisely mimicking this motion has been a very difficult task. These creatures have evolved to being able to produce this complex motion through a combination of joints, muscles and in the case of birds, feathers that enable them to produce the wing shape and motion that enables efficient flapping flight.

These flapping wing creatures can be broken into two main categories. The first category is the insects and the second is the birds, mammals and dinosaurs. The main difference between these categories, for the purposes of modeling and understanding flapping wing flight for the SSA application, is in how they generate lift.

### 1.2.1 Insect Flight

The aerodynamics of insect flight are rooted in their small size. Because of their size insects fly in a very low flight Reynolds number regime less than a thousand and 100,000. Conventional aerodynamics utilized by aircraft and birds would be severely limited operating within this very low Reynolds number regime. The main issue is laminar separation of the boundary layer. This separation can cause loss of lift resulting in an abrupt and catastrophic loss of lift. To avoid this
flow separation the boundary layer must be transitioned from laminar to turbulent. Within low Reynolds number flow it is very difficult (if possible at all) to transition to a turbulent boundary layer. This flow restriction is a major factor that severely limits the flight envelope and capabilities of a conventional flight within this low Reynolds number flight regime.

Although conventional flight may be difficult under such low Reynolds numbers insects have succeeded in efficiently exploiting the low Reynolds number flight regime. Although not completely understood, the mechanisms in insect flight are significantly different of conventional aircraft. First investigated in 1994 by Charles Ellington at the University of Cambridge, the main mechanism for lift generation on an insect wing was determined to be vortex interaction caused by the flapping motion. This interaction is dependent on Reynolds number. As the Reynolds number increases this lift producing mechanism diminishes. Experiments have shown that flow on an insect wing at Reynolds numbers greater than 10^6 there is a crisis of flow over the wing caused by early boundary layer separation. As the Reynolds number decreases to around 10^4, this crisis is greatly reduced and the flow displays a smoother shape. At Reynolds numbers of 10 to 10^3, flow separation is absent. As the Reynolds number decreases other lift producing mechanisms such as differential velocity and drag and other boundary layer effects may come into play. These Reynolds number effects are a main reason for the difference in the flight characteristics between birds and insects. A diagram of this vortex generation is shown in Figure 1-6 [13]. This vortex generation is not completely explained by present theory. However, it is believed to be caused by the separation of flow over the leading edge of the insect wing. A diagram of the vortex formation is shown in Figure 1-7, [95, 47].

Flapping alone is not sufficient to generate the maximum vortex circulation possible for achieving maximum lift. This limit on reaching the maximum circulation levels is due to the flapping rate of the wings and the time delay required for the growth of the vortex circulation. It is believed that insects overcome this issue by the interaction of the insect wing with the vortex as it is shed. Unlike with conventional airfoils there is no dramatic reduction in lift after the wing achieves super critical angles of attack. This suggest that flow separation prior to the vortex formation does not occur. It is believed that this resistance to flow separation during vortex formation is due to the low flight Reynolds number and the high wing flap rate of 10^{-1} to 10^{-2} seconds.
An additional lift producing mechanism which insects take advantage of is the magnus force. This is the force generated due to the rotational motion of the wing during each flap. This force is most widely know for its effect in producing a "curveball" in baseball. Insect flight control is achieved by controlling these lift producing mechanisms form wing to wing. Based on these
mechanisms insects are capable of achieving lift coefficients on the order of 5. This high lift coefficient and the forces that are used to generate it is what allows them to fly in a manner that is different from conventional aircraft or birds. It also gives them the ability to hover, rise vertically and change direction instantly. A diagram of the lift produced through a stroke of the insects wings is shown in Figure 1-8 [95, 47, 13].

Because of the estimated size of the SSA the characteristics of insect flight would not be applicable. Although they have superior lift generating capabilities the aerodynamics of how they generate lift cannot be utilized by the SSA and therefore the wing designs and motion of an insect would not apply to the SSA.

1.2.2 Bird, Mammal and Dinosaur Flight

The flight of larger creatures such as birds, mammals and the dinosaurs is based on more conventional aerodynamics. Although these creatures also generate lift by flapping their wings, the lift generating mechanisms and aerodynamics are similar to those used by aircraft. The lift is generated by the pressure difference produced due to the airflow over the wing. This conventional method for generating lift over an airfoil shaped surface is shown in Figure 1-9 [79].
For the flight conditions encountered in flapping flight, an airfoil derives the majority of its lift due to the pressure on the upper surface being less than that on the lower surface. The pressure difference is produced by the nonsymmetrical shape of the airfoil and the angle it is at with respect to the oncoming air stream. These two factors cause the air moving over the upper surface to accelerate compared to the air moving over the lower surface. This difference in flow velocity, based on the Bernoulli effect, will produce a difference in pressure between the air on the upper surface and the air on the lower surface of the airfoil. It is this difference in pressure that generates the lifting force generated by the airfoil. However, it must be understood that this is a simplified description of how lift is generated by an airfoil. In reality, the generation of lift is a complex process that encompasses the conservation of mass, momentum, and energy and the effects of viscosity of the flow around the airfoil. This complete set of physical phenomena is described by the Navier-Stokes equations, which form the basis for fluid dynamics [76, 79].

The amount of lift generated is highly dependent on a number of factors related to the airfoil and wing geometry and flow field. These factors include:

- The airfoil shape
- Cross Sectional Shape
- Camber
- Chord Length
- The Wing Shape
- Aspect Ratio
- Chord Length Distribution
- Wingspan
- The Airflow Velocity
- The Angle of Attack

Unlike with an aircraft where many of these factors are constant or vary slightly during operation, in flapping flight most of these parameters can vary considerably throughout the flapping motion. Because the wing is moving, the velocity and flow direction relative to the wing are constantly changing. Therefore, to generate sufficient lift and thrust needed to maintain flight and to minimize drag, the wing must twist and deform while flapping to maintain an optimal airfoil shape and angle of attack. The wing motion that is employed by various flapping flight creatures can vary significantly. This variation is dependent on the type of creature, mainly its size, and the type of flight that is being performed, gliding, high-speed flight, low-speed flight, takeoff, landing etc. A diagram showing an example of the motion and force vectors on an airfoil on the inner and outer sections of a wing during flapping flight is given in Figure 1-10 [135]. The wing motion illustrated in Figure 1-10 is not universal but it does show the extent to which the wing will twist and deform under flapping flight conditions.

During the flapping motion, as the wing moves in the downward direction, the angle of attack it makes with the incoming air stream generates a positive lifting force as well as a forward thrust force. On the upstroke, the wing is twisted to minimize the negative thrust (drag) that can be generated. Lift is also produced on the upstroke. Figure 1-10 also demonstrates that the forces generated throughout the flapping motion are not uniform along the wing length. This is due to the speed of motion along the wing varies from the root to the tip and that the wing twisting also
As a bird flaps its wings there will be a twisting and a camber change along the wing throughout the flapping motion.

In general the sections of the wing accomplishes different tasks as you move from the root, near the body, to the tip. The inner portion of the wing is used mainly for generating vertical lift to keep the creature in the air, whereas the outer portion of the wing is used to generate thrust to overcome drag and maintain flight velocity. The specifics of how the wing will move will also be significantly affected by the flight conditions. For example if the flight velocity is high the angle of attack on the inner portion of the wing can be reduced because the increased flight velocity will produce increased lift at lower angles of attack. To maintain this higher velocity, however, greater thrust will need to be generated by the outer portion of the wing. During slow speed flight the wing must now generate significantly more lift and does not require as much thrust. This requires a greater angle of attack along the whole length of the wing and minimizing drag and negative thrust production on the upstroke. The upstroke drag can be minimized by pulling the wing in, closer to the body, thereby reducing its effective area. Also the chamber of the airfoil can be modified so that positive thrust on the upstroke is produced.

To achieve the desired lift and trust throughout the flap the wing motion can become fairly complex. This is illustrated by Figure 1-11. This figure shows a motion trace of the wing tip and the leading edge of the wing near the body. This trace shows how variable the motion along the wing can be. Being able to generate this type of motion requires a very flexible wing with a large amount of freedom of motion. Conventional materials and actuators with mechanical links and joints have a difficult time mimicking this type of motion. There are two significant factors that a mechanical wing has to overcome in order to fly with the capabilities and efficiency of a bird, bat or other flapping wing creature.
Chapter 1.0 Introduction

1.2 Flapping Wing Flight

There have been a number of attempts at mimicking flapping flight through the analysis, design, construction and operation of mechanical wings. These mechanical winged vehicles called ornithopters have met with mixed results. The most successful has been developed at the University of Toronto [28, 29, 25, 27, 26]. A flight sequence from a test flight of their vehicle is shown in Figure 1-12 [134]. This is one of the few mechanical flapping wing based vehicles that has been able to achieve true powered flight. The vehicle operation is fully mechanical. A transmission system is utilized to convert the rotary motion of the drive engine to the flapping motion of the wings. The overall configuration is very similar to a conventional aircraft. It has a fixed tail that is utilized for stability and control, a main body fuselage which housed the propulsion system and the main wings that are of similar shape to the wings of a higher aspect ratio sail plane. The wings are hinged slightly outboard from the fuselage and rise and fall to produce the flapping motion that generates the lift. The wings twist and bend during the flapping motion at a fixed rate to providing enhanced efficiency.

Figure 1-11: Illustration of wing motion

Figure 1-12: Mechanical wing ornithopter in flight
Another area where mechanical flapping wings have received significant interest is in the area of micro air vehicles [77]. These are remotely piloted vehicles used for surveillance and data gathering. Their main mission is military however there are some civilian applications such as search and rescue within buildings. These vehicles are for limited duration applications and could carry visual, acoustic, chemical or biological sensors. Their operational Reynolds number is very low and their flight mechanics and wing motions are modeled more after insects than larger flying creatures like birds.

Although there have been successes in producing mechanical wing flapping flight vehicles, to truly mimic the motion of a bird or other creature's wing would require a different design approach. The main issue with constructing a wing that can truly mimic those in nature is in the ability to generate the fine motion and control of the wing surface that is accomplished by creatures that have perfected flapping wing flight.

The fine control of the shape, twist and motion of the wing not only provides the ability to adjust the flapping motion to accommodate different flight speeds and conditions but to provide subtle control to adjust to unforeseen factors such as wind gusts or abrupt changes in direction. The skeletal, muscle and feather structure of a bird's wing provides the ability to produce these fine adjustments as well as move in a complex fashion to maximize efficiency. The combination of bending, twisting and surface changes is presently unmatched by any mechanical device [45].

To better approximate this wing motion new materials can be utilized that may enable a wing to be constructed that can have the same range of motion as a birds or bats wing without the need for extensive mechanical linkages and drive motors. The IPMC proposed for use in the SSA can potentially provide this capability. The characteristics of the IPMC are very similar to those of muscles in living creatures. This material can be fully articulated. Since the whole surface of the material is active it provides the possibility of very fine motion control as well as the ability to produce complex motions. A wing constructed from the IPMC material would behave very similarly to the muscle driven wing of present and past flying creatures.

1.2.3 Wing Shape

In addition to the wing motion the wing shape plays a critical role in determining the capabilities of the flight creature. Depending on the flight regime, creature size, flight Reynolds number, flight speed etc. the wing shape can vary significantly. Figure 1-13 [76, 131] shows wing shapes for various types of flying creatures. The shape of these wings dictates how and what type of flight the bird or other creature will be capable of. For example long, narrow pointed wings are utilized on larger creatures and indicate a soaring type of flight. These types of wings are highly efficient for gliding over long distances such as over oceans and into wings where minimal flapping is required. These high aspect ratio wings shown in Figure 1-13 for the Gull and Pterasaur generate significant lift and reduce drag. However, because of their configuration and size they are not effective at low speed higher frequency flapping. This type of flight is more applicable to the pheasant wing that enables quick takeoffs from the ground and enhances the maneuverability during flight.

The SSA is concept is an aircraft that will have a very low wing loading. Therefore it will be capable of gliding for extended periods of time. Also consistent rapid wing movements will
probably not be possible due to the high energy demand this would require. Therefore for the SSA flight regime, the wing shape will need to be similar to the gliding and soaring wings shown in Figure 1-13. These wings are long and slender with a high aspect ratio to maximize efficiency. Also the composite material makeup of the wing, consisting of the IPMC, thin film array material and battery material, will produce a wing structure similar to wings that utilize a membrane covering. Based on these similarities the wing shape of the flying dinosaurs is a good starting point for designing a wing for the SSA.

The pterosaurs of the dinosaur era were the next class of creatures to evolve the capability of flight after the insects. They existed in a range of sizes from small creatures with wingspans of around 15 cm, the size of some present day birds, to the largest creatures that have ever flown, with wingspans over 10 meters. These animals have characteristics that are more similar to the SSA in terms of their wing-spans, Reynolds numbers, aspect ratios, and flight profiles. Archaeological finding for one of the largest pterosaur, the pteranodon shown in Figure 1-14, indicate flapping frequency of 1.2 hertz, flapping angle of 35-40 deg., a pitch angle of flapping axis with respect to freestream of 7.5 deg, and a dynamic twist of 4.2-5.0 deg/ft. The pterosaur’s long wing has excellent glide characteristics with an L/D of approximately 28:1. Wing camber is similar to Selig's S1223, shown in Figure 1-15 [132], but with a thinner airfoil section. Zero lift occurred at about -8 to -10.5 degrees, with a quarter chord pitching moment of about -0.3 [88].
Based on a pteranodon wing planform which does not have a substantial aerodynamic tail, a factor that will drive the SSA layout will be the need to stabilize the wing pitching moment. According to von Mises: Theory of Flight [138], a cambered, reflexed wing without a tail can be designed to null out the nose-down pitching moment. Due to the high flexibility of this highly adaptive wing structure that the SSA will incorporate, a properly shaped reflexed wing adapted to the local flow conditions is a possibility. The NASA Pathfinder-Plus and Helios aircraft, shown in Figure 1-16, demonstrates that non-swept wing aircraft can be flown without a tail (although it has some kind of elevator on the trailing edge of the wing). The yaw and roll modes would be controlled by changing the wing shape evenly across the entire span. This is direct contrast to the more traditional methods of producing asymmetric lift via ailerons that act only
over the outer portions of the wing and modify the lift distribution only over the span over which they are connected. The other option would be to incorporate a short coupled horizontal tail, much like a hawk or airplane in which the tail generates negative lift to compensate for the pitching moment. Or possibly have a combination of the two (a small elevator and wing twist).

Any insight that can be generated on the flight characteristics and control mechanisms of the Pteranodon or other similarly sized flying creatures can be very useful in designing the SSA. Based on the makeup of the SSA design, it will be the first vehicle capable of truly duplicating flapping flight that occurs in nature. Because of this significant benefits in the design, control and operation of the SSA can be had by understanding how creatures in nature fly and applying this knowledge to the design of the SSA.

1.3 Mission Capabilities

The unique characteristics of the SSA open up a large potential for flight applications and missions. The SSA is a very flexible robust solar powered aircraft. This is unlike any other type of solar powered aircraft that has proposed. In order to operate on the available power from the sun the wing loading for a solar powered aircraft has to be very low. To construct a conventional aircraft with low wing loading requires a very light-weight structure. The downside to this type of construction is that the vehicle becomes very fragile. This can be seen in the Pathfinder and Helios solar powered aircraft. These aircraft have been the only truly successful large-scale solar aircraft to have ever flown. However, significant limitations on their flight conditions are required to insure that they are not destroyed during takeoff. In fact the Pathfinder aircraft was severely damaged on the ground due to a gust of wind blowing through a hanger where it was being stored [68] and the Helios prototype aircraft was destroyed during a test flight due to a structural failure caused by oscillations from a control problem [6]. As ground-breaking as these aircraft are, the inherent fragileness of their structures due to the need for very low wing loadings, significantly limits their operational capability.[3]
Figure 1-16: Helios and Pathfinder solar-powered aircraft

The SSA brings a new approach to producing a robust solar powered aircraft. Although the wing loading on the SSA is also low its structure and construction would be very durable. This is because the aircraft structure is essentially solid and the key material (the IPMC) is a flexible plastic. The other component materials are all thin film based which enables them to flex and move with the aircraft without any structural damage. This lightweight, flexible structure provides some significant mission benefits over more conventional rigid structure vehicles. The Pathfinder and Helios aircraft could only be launched under very specific conditions. For example the Helios aircraft cannot takeoff if the wind speeds exceed 3.6 m/s near the surface. Also it has to be a clear day with little or no vertical wind shear (turbulence) since this can severely damage the aircraft [32]. By comparison the SSA would be capable of being launched under almost any weather condition.

1.3.1 Operation on Earth

To deploy the SSA it is envisioned that it would be balloon launched, for Earth based applications. A balloon, similar to a weather balloon, would carry the SSA to the desired flight altitude. Because of its flexible nature the SSA can be easily stowed in a compact fashion on the balloon while ascending. Once at altitude the SSA would be deployed from its stowed configuration and begin its flight. This launch and deployment scheme is shown in Figures 1-17 and 1-18.
As shown in Figure 1-17 the balloon will carry the SSA vehicles above the cloud layer. In this scheme two SSA are wrapped around a stowage gondola. This provides a means for compact stowage. The gondola can also carry communication and other equipment internally. It is also envisioned that this gondola can also provide power to the SSA while they are stowed and fully charge the SSA batteries prior to release from the gondola.

Once the desired flight altitude is reached the aircraft will be individually released and begin their flight mission. The deployment sequence is begun when the brackets that secure the aircraft to the gondola are released. Once released the aircraft unwraps from the cylindrical gondola and begins flight.
This method for initiating the flight of the SSA enables the SSA to be deployed under any weather condition in which the balloon can be launched. Also since the balloon provides the lift to get to the flight altitude, cloud cover is not that much of a concern as long as the desired altitude is above the cloud layer. The ability to compactly stow the SSA on the balloon enables conventional balloon designs and controls to be utilized. No special handling of the aircraft would be required by the balloon system.

Upon release the flight mission would begin. The mission would depend on the type of equipment carried on board the SSA. Typical potential missions or capabilities that could be performed by the SSA for Earth based flight are summarized in Table 1-1. These missions can be categorized into three main areas, observations, communications and sampling or sensing. Each type of data gathering or operations will require specialized sensors or equipment. The ability to carry multiple types of equipment will depend on the mass and volume of the devices. The lighter and more compact the sensor or device, the more the aircraft will be capable of carrying. Therefore, the development of lightweight, low power consumption sensors and equipment (such as communications transmitters and receivers) will greatly increase the mission capability of the SSA platform.
Imagery is probably the single greatest application that can be performed by the SSA. The observation and surveillance of the surface from a high altitude platform that is capable of controlled flight and can be placed into position fairly quickly has a number of applications both civilian and military. An example of this type of mission is illustrated in Figure 1-19. The type of imagery will depend on the observation objective. High and low resolution visual imagery can be performed or cameras that image other parts of the spectrum (such as infrared or ultraviolet) can be utilized. The ability to perform a specific type of imagery will depend on the mass and power demand of the camera. There have been significant advances in the miniaturization of cameras for UAV applications. This miniaturization lends itself directly for use on the SSA. There has recently been significant development of miniature cameras. A main source of this development has been micro-UAV applications. These types of vehicles are small, light-weight aircraft that can potentially be utilized for surveillance. The military is the prime force behind the majority of the development for these types of vehicles. An example of this type of vehicle, shown in Figure 1-20, would be the E-Wasp under development by Aerovironment for the Department of Defense [129]. Because of there size and limited power production capability the sensors and cameras utilized by these types of vehicles need to be small, lightweight and consume little. These are all characteristics that fit will with the SSA payload requirements. Therefore, much of the sensor and camera technology that is being developed for these micro-UAVs should be directly applicable to the SSA. This development includes not only visible light cameras (as shown in Figure 1-20) but also Inferred and ultraviolet cameras as well as imaging systems for specific parts of the EM spectrum.
Communications to and from the aircraft are also a critical capability. Getting data collected off the aircraft and sending commands to the aircraft will be required for almost all potential missions. Also the communications itself can be a potential mission for the SSA by allowing the aircraft to act as a quickly deployable communications relay. The communications system will consist of a transmitter, receiver and antenna. As with the camera system the SSA can benefit
from the significant development that has and is taking place for UAVs and in remote communication capability in general (such as cellular phones). The SSA will need to be capable of communicating to and from the surface as well as to other aircraft and to satellites. Examples of these potential communication paths is shown in Figure 1-21.

The antenna for the communications system on the SSA will need to be integrated into the aircraft structure. One potential option is to utilize the recently developed thin film antenna technology. Thin film antennas (shown in Figure 1-22) are presently utilized in the automobile industry as a means of placing a flexible antenna on a vehicle with little impact on the design. The antennas are clear and flexible and are usually applied to the windshield or rear window.

Another potential antenna technology from the automobile industry that could be applied to the SSA is the transparent metallic antenna. This antenna is actually a transparent metallic coating that can be applied to the windshield of an automobile and utilized as the antenna for AM and
FM radio reception. Because of its size this antenna reduces many of the auxiliary electronic amplification components that are necessary with other types of antennas. Since this type of antenna is transparent to visible light it could be applied over the top of the solar cells with minimal effect on their performance [97]. By applying this technology to the SSA it would enable an antenna to be placed over the lower and upper surface of the wing. This enables communications both to and from the surface as well as to and from satellites or other vehicle flying above the SSA.

The other main parts of the communications system consist of the transmitter and receiver. The developments in the cell phone industry have successfully miniaturized these components to where they would be applicable to the SSA. Texas Instruments has recently integrated most of the functionality of a cell phone onto a single silicon chip [44].

Other possible application of the SSA would be for science data gathering. This would include items such as sampling the atmosphere and looking for specific gasses or particles and monitoring the metrological atmospheric conditions, such as temperature, pressure and humidity. For sampling the atmosphere and looking at chemical composition a mass spectrometer would be the ideal tool. This device can identify chemical composition of a substance by determining its molecular weight. However, these are usually large devices that would not be applicable to the SSA. There has is ongoing work being performed to reduce the size and power requirements of mass spectrometers for space applications. Some of this work is being performed at JPL in an effort to produce a small lightweight mass spectrometer for use on the international space station [53]. They have succeeded in reducing the mass spectrometer device to a unit approximately 5 cm in length, shown in Figure 1-23. Additional electronics are also needed for the device to operate but further efforts on electronics miniaturization could significantly reduce the electronics size requirement.

The remaining types of scientific sensors that have been identified in Table 1-1, are all small devices that can be easily integrated into the wing structure. The magnetometer sensors would be placed at the wing tips to try and get as far a separation as possible between them. The meteorological sensors such as the temperature (thermocouple) and pressure can be incorporated at almost any location on the aircraft.
The aircraft capabilities listed in Table 1-1 are not only applicable to Earth based missions but can also be utilized for planetary exploration. The SSA has the potential of operating within any atmosphere in which there is sufficient solar energy available to meet the flight and payload needs of the aircraft. As mentioned previously Venus and Mars are the other planetary locations within our solar system that can meet these requirements. Even though the atmospheres of these planets are composed of mainly carbon dioxide, the lack of oxygen is not a factor for the SSA. Unlike conventional aircraft power plants which require oxygen for combustion, the SSA's power comes completely from the solar array and therefore is not dependent on the oxygen content of the atmosphere.

1.3.2 Operation on Venus

For operation on another planet the SSA will need to be deployed upon entry into the planet's atmosphere. To package a conventional fixed wing aircraft into an aeroshell requires the aircraft to be folded in a number of locations. The wings are usually hinged in one or more locations and the tail boom is usually also hinged. An example of this packaging scheme for a Venus aircraft concept is shown in Figure 1-24 [18, 66]. The unfolding and flight initiation of an aircraft from this initial stowed configuration can be a risky maneuver.

However, because of the flexible nature of the SSA it can be easily stowed into an aeroshell. This is a significant advantage over conventional aircraft configurations. It enables the SSA to be in its flight configuration instantly on deployment, reducing the risk of transitioning to flight from the stowed configuration. The deployment sequence for the SSA is shown in Figure 1-25. This deployment concept shows how the SSA can be compactly stowed within the aeroshell and enter full flight configuration upon release.
Once released from the aeroshell the SSA can begin its science and exploration mission. The tasks listed in Table 1-1 provide a base of capabilities that can be applied to planetary exploration.

Contrary to the popular image of Venus as being a hot inhospitable environment, Venus may actually house the closest environment to Earth's anywhere in the solar system. At an altitude above 50 km, where the SSA can potentially fly, the Venus environment is very Earth like. The pressure is similar to Earth surface atmospheric pressure, it is cool (less than 50° C) and it contains all of the compounds needed for life (Carbon, Hydrogen, Oxygen and Nitrogen) [65]. And with the abundant solar energy available may be one of the more likely places in our solar system to find life. To perform this type of astrobiology mission, atmosphere sampling would be an invaluable tool. The sampling would search for biogenic gasses, such as CH₄, NH₃ or H₂S). These gasses could exist in low concentrations that could not be detected without direct sampling. But if there, these gasses would indicate that microbial life possibly exists within the upper layers of Venus's atmosphere. Also sampling the atmosphere can provide data on the chemical makeup of Venus's atmosphere as well as the aerosol content within the atmosphere. This type of composition and astrobiology investigation of the atmosphere could potentially be performed by the SSA. Atmospheric sampling, as shown in Figure 1-26, can provide significant insight into the characteristics and operation of the planets atmosphere as well as search for the
telltale signs of life. It also could be utilized to search for volcanic emissions to infer active volcanism on the planet.

![Venus atmosphere sampling](image)

*Figure 1-26: Venus atmosphere sampling*

In addition to sampling the atmosphere to determine composition and potentially search for life, the SSA would be capable of performing a number of other missions within Venus's atmosphere. These would include the evaluation of the atmospheric dynamics and structure. This can be performed by measuring the various meteorological properties over time and at different altitudes within the atmosphere and by sounding the atmosphere, as shown in Figure 1-27, to determine density and cloud layer height. By sending a sonar or radar signal into the atmosphere these and other atmospheric properties can be determined. Also direct imagery of the atmosphere can provide insight into its structure and dynamics. Images of the atmosphere can be taken in various spectrums, such as visible light or infrared. This imagery can provide significant information on the dynamics and structure of the atmosphere. Because the aircraft is within the atmosphere this type of imagery can provide a prospective not achievable from satellites. For example, by imaging horizontally through the atmosphere in the infrared, temperature distributions and variations can be determined. This would provide information on the structure and mixing within the atmosphere. Also similar imaging within the above cloud layer could provide significant information of the dynamics and motion within the atmosphere.
In addition to the imaging and sensing that can be performed by the aircraft, it can also act as a communications platform. Communications are necessary for retrieving the data collected and to communicate with the aircraft and direct it along its mission. The main communications path would be from the aircraft to a satellite that would then relay the information back to earth. If more than one aircraft is in flight they can communicate between each other. This can be utilized as a means of relative navigation or as a mechanism for relaying data to the satellite if one of the vehicles is out of its range. These relay and communication capabilities are illustrated in Figure 1-28.
In addition to communicating with other aircraft, it could be possible to utilize the SSA as a relay and control center for a surface-based vehicle. This concept is illustrated in Figure 1-29 [66]. Because of the high temperature and pressure at the surface of Venus, operating a surface vehicle on the surface for extended periods of time is difficult. Any electronics and computer systems on-board the surface vehicle need to be housed in a cooled chamber for them to operate. Providing a cooling chamber requires significant power, which in turn increases the size and complexity of the vehicle. To minimize this cooling requirement, it may be possible to house the controls and computing capability for the vehicle in an aircraft that is operating high up in the atmosphere where it is much cooler. If the aircraft can circle above the ground vehicle, it would be capable of being in constant communications with the ground rover and provide continuous instructions. This remote operation significantly simplifies the equipment necessary on the ground vehicle and makes longer-term operation much more feasible.
Figure 1-29: Illustration of a rover aircraft communications link on Venus

An aircraft within the Venus atmosphere is the only means in which this type of continuous communications and control can be achieved. Because of the slow rotation rate of Venus a geostationary satellite would be much too far from the planet's surface to be useful as a communications link. The transmitting power of the ground vehicle and its receiving antenna would need to be very large for this type of communications link to successfully operate. Also the time delay in the communications path would impact the surface rovers performance and require navigation capabilities resident on the rover, which would defeat the purpose of the remote operational system. Lower orbiting satellites would not be capable of continuous communications that again would significantly affect the rovers performance or require on-board on the rover intelligence. A constellation or low orbiting satellites would be required to successfully accomplish the remote operation of the surface vehicle. However, this type of infrastructure is very costly and
would take considerable time to put into place. An air vehicle solves these problems by being able to provide continuous coverage above the rover, by circling over the area it is operating and being close enough so that the communications transmission delay time is negligible.

### 1.3.3 Operation on Mars

As with flight on Earth and Venus, there are a number of potential science applications that can be performed by the SSA if it is capable of flight on Mars. The items shown in Table 1-1 are also applicable to the exploration of Mars. Because of the low atmospheric density of the Martian atmosphere, the SSA would need to fly near the surface of the planet. Unlike flying on Venus in which the aircraft would be flying a great distance from the surface, flying on Mars would require the aircraft to be very near the surface. This closeness to the surface has its own set of benefits and risks different from those for flight on Venus or even Earth. Being close to the surface would enable the ability to perform some unique science missions, as well as having a very desirable vantage point for high-resolution imagery. Entry and deployment of the SSA into the Martian atmosphere would be similar to that shown for Venus. Once deployed the aircraft would begin its flight mission.

Mars, with regard to the other planets within our solar system, has been the main target of scientific exploration for more than twenty-five years. Most of this exploration has taken place using orbiting spacecraft or landers. Orbiters offer the ability to image large areas over an extended period of time, but are limited in their resolution. Landers can handle surface and atmospheric sampling, but are limited to the immediate landing site. Mobility is the key to expanding the scientific knowledge of Mars. The Pathfinder/Sojourner and MER missions offered a new opportunity to scientists in that it was the first time that an autonomous mobile platform could be used for exploration. This allowed scientists the freedom to explore the surrounding terrain, maneuver to scientifically interesting sites, and perform an analysis of soil and rock composition over a broader area. In short, it offered many more options to the scientific community. However, the terrain limits the rover: large rocks and canyons are obstacles that are difficult for a surface rover to overcome. [119,120]

An airborne platform such as the SSA can achieve science objectives difficult to achieve from orbit or from surface rovers. They can cover much larger distances in a single mission than a rover and are not limited by the terrain, therefore they could more easily provide imaging of very rocky or steep terrain. Airborne platforms can return images of more than a magnitude higher resolution than state-of-the-art orbiting spacecraft. Near infrared spectrometry, crucial to detecting mineralogy on the planet, and high spatial resolution magnetometry, which may provide clues as to the origin of high crustal magnetism seen from orbit, require moving platforms. Being close to the surface also increases the resolution and sensitivity of these instruments. Finally, atmospheric sampling can study variations over a much greater area. [75,23,71]

#### 1.3.3.1 Imagery

The SSA can provide ultra-high resolution imaging (on the order of 10s of centimeters resolution) over extensive areas. Similar to the type of images acquired by landers or rovers. An example of a high-resolution surface image is shown in Figure 1-30. This type of data can make possible recognition of individual rocks and specific land features. It can also aid the identifica-
tion of areas for further examination by a rover. High-resolution imaging is also highly valuable for interpreting the geologic history of a region and examining such processes as Aeolian, hydrothermal, aqueous, volcanic, cratering, and tectonic based on their geomorphology. High-resolution imagery can also be valuable in examining layers in crater walls and hydrothermal system associated with volcanoes and impact craters. In addition data collected by other science instruments such as magnetic and neutron observations can be correlated with local geologic features.

![Figure 1-30: High resolution image of the Mars surface taken by Pathfinder lander](image)

The SSA can also potentially provide high-resolution compositional information on surface rocks using infrared spectroscopy and other techniques that can take advantage of observation elevations from a few hundred meters to a few kilometers and collect measurements at the meter spatial resolution. Mineralogy is directly related to the formation environment of rocks. Thus locating key mineral deposits is central to locating sites that may have allowed life on early Mars to thrive and to understanding the chemical evolution of the Martian surface and atmosphere.

In past science missions, imaging has provided the most beneficial planetary science and has contributed to the most planetary science discoveries that have been made. Based on previous results of exploration missions it can be inferred that the higher the resolution of the imagery the greater the science value and discoveries that are made. The SSA is ideally suited for producing high-resolution imagery. Its flight speed is relatively slow compared to other more conventional aircraft and it would be capable of flying near the surface enabling very high resolution and very good perspective of the terrain.

The Mars Orbiter Camera on the Mars Global Surveyor spacecraft can achieve 1.5 m/pixel images of the surface. This compares to the Viking Orbiter resolution of 200 m/pixel. This 100 times increase in resolution has greatly increased our understanding of Mars. With a flight vehicle such as the SSA we can achieve resolutions on the order of 0.01 m/pixel, a 100 times improvement over the Global Surveyor. This type of resolution will allow the study of weather-
The complete range of imaging of the surface can be achieved with two separate cameras:

- A camera can take images of the surface at high resolution
- A lower resolution wide-angle camera to provide context for these images so that they can be related to observations from orbiters. An example of this type of context image is shown in Figure 1-31.

![Figure 1-31: Example of a wide-angle context camera type image](image)

The purpose of the low resolution wide angle camera (context camera) is to provide a context for which low resolution orbital imagery can identify where the aircraft is taking data and pictures. Also the picture has to be high enough resolution so that the high-resolution camera images can be found within the picture. This staging of picture resolutions from orbital to wide angle context imagery to high-resolution imagery enables the detailed high-resolution images to be referenced to a global view of the terrain. Orbital imagery at Mars can presently achieve about 1 m to 3 m per pixel. Based on this the context camera with a resolution of 0.15 m/pixel (6 to 1 ratio) should be sufficient to place the high-resolution imagery (at .01 m/pixel) within the context of the orbital spacecraft pictures. For very close-up imagery (such as imaging a specific rock, or the strata on a cliff) the ability to place the high-resolution image into the global imagery may not be possible. However the context camera can at a minimum place the detailed image into the general area in which it was taken. This ability to reference the imagery to a global scale is vital to having the data aid in the overall understanding of the planet.

### 1.3.3.2 Magnetic Field Mapping and Investigation

Strong remnant magnetic fields have been identified from orbital observations of the Martian surface. Aircraft such as the SSA are an ideal platform to investigate these fields. Because of its flight altitude on the order of 100s of meters, the SSA can provide both the spatial resolution and the signal strength for detailed magnetic field mapping. The information gathered, by studying the magnetic signatures of impact craters, can lead to a greater understanding of the early
history of magnetism on Mars. Also, information relevant to the thermal evolution of the planet can be obtained by identifying young craters in older terrain and mapping the magnetic fields around these craters. The history of Mars' magnetic field may be an important link for understanding the radiation environment due to the early Sun's solar wind.

From previous scientific investigation it is know that the magnetism of Mars varies greatly over the planets surface. It is very highly magnetized at certain locations (an order of magnitude greater than the magnetic field strength on Earth, and at altitudes an order of magnitude higher than on Earth, greater than 1500 nT at 100 km altitude and greater than 250 nT at 400 km altitude) and weakly magnetized at others. The locations of the strong magnetic fields correspond with some of the older and highly cratered areas of Mars.

Presently there is no active mechanism within the planet for forming a uniform magnetic field (as on Earth). Because of this lack of a planetary magnetic field it is much easier to measure the crustal magnetization directly. This crustal magnetization can exceed several Gauss (200,000 nT). The magnetic landscape of Mars was first discovered by the Mars Global Surveyor spacecraft. This spacecraft measured many large-scale, highly magnetic locations on the surface of Mars, many of which extended over hundreds of kilometers. However due to the distance from the surface it was not able to discern any small scale variations in these magnetic fields, which are believed to exist. The SSA can be used to investigate these magnetic field regions and provide high resolution data on the magnetic field strength variations within these regions. An example of the differences between the magnetic fields on Mars and Earth are shown in Figure 1-32.

The magnetic field mapping done by the SSA would enable a greater understanding of the crustal magnetism on Mars. For this to be accomplished it would require that measurements of the magnetic field be performed with enough spatial resolution to relate magnetism to specific geologic features and structures. With the SSA the spatial resolution on these measurements would be less than 1m, orders of magnitude greater than what is achievable from orbit.

The magnetic field mapping requires a three-axis magnetometer to be utilized. The magnetic field sensors would need to be mounted in a location that minimizes magnetic field contamination from other systems or instruments on the SSA, such as near the wing tips. To minimize the magnetic signature of the SSA it should have as little magnetic materials within it as possible. Also any magnetic field inducing devices (such as from the power generation system) will have to be shielded and or properly grounded to minimize the magnetic field effects. The mass and power of this type of device is on the order of 0.2 kg and 150 mW respectively.
1.3.4 Near Infrared and Neutron Spectroscopy

The distribution of water (ice or liquid) is vital to the search for life. Neutron spectroscopy is a powerful technique for detecting an excess of hydrogen to a depth of about a meter. Such a technique can be implemented from orbit but has a resolution of several hundred kilometers. From a platform within the atmosphere, such as the SSA, the spatial resolution is several orders of magnitude better and so the potential exists for locating kilometer-sized bodies or much smaller.

Mineralogy is a key tool for investigating the formation and geologic history of Mars. Near infrared spectroscopy can be used to provide data on the mineralogy of Mars. This includes measuring the pH, abundance and phase of water, atmospheric chemistry, temperature, and surface pressure. It can also be useful in examining the geologic processes of the planet such as sedimentation, volcanism, and hydrothermal alteration.

Mineral makeup can be determined through near infrared absorption and spectroscopic evaluation. This technique has been widely used in the past both on Earth and for planetary exploration (this technique was used on Phobos to determine surface mineral composition). Near infrared spectroscopy (at the wavelengths of between 0.7 m and 2.5 m) can provide information on soil makeup and identify materials such as iron oxides, iron oxyhydroxides, carbonates, clays, olivines and pyroxenes. As well as establish their degree of crystallinity. This type of science will allow the determination of presence of these minerals and their abundances within the soil.

The objective of any near infrared spectroscopy investigation should be to link the mineralogy with specific geologic formations on the planet (imagery) thereby providing a more detailed understanding of the geologic processes of the planet.

The ability to perform imaging spectroscopy from the SSA will probably not be possible due to the size of the instruments required (unless there is significant advances in sensor technology).
Therefore non-imaging spectroscopy would be the applicable choice for this type of data collection. However to get useful data from a non-imaging spectroscopy system it would need to be closely integrated to the camera imaging to provide useful information.

### 1.3.5 Radar Sounding

Radar sounding can investigate subsurface structure and search for buried ground ice and subsurface water. This type of exploration has been proposed from orbiting spacecraft but by performing this from a few 100 meters or so above the surface increased spatial resolution and greater depth resolution can be achieved. Aerial radar sounders have a proven capability to detect subsurface water beneath glacial ice at a depth of up to 4 km with more than 100 subglacial lakes identified in Antarctica. Significant miniaturization will be needed for radar sounders to be compatible with the SSA, however because of its close proximity to the surface, compared to a satellite, the power required by this device will be minimized. This type of sounding is similar to what is used with the European Space Agencies Mars Express orbiter and its use is illustrated in Figure 1-33.

![Figure 1-33: Radar sounding for investigating surface and subsurface features](image)

### 1.3.6 In-situ Atmospheric Science

The SSA can potentially take atmospheric samples over a range of altitudes from near the surface up to more than a kilometer. It can also take these samples in a controlled grid fashion providing a comprehensive view of the atmosphere near the surface. These samples can be used to validate global remote sensing data from orbiting spacecraft, that sound the Mars atmosphere. This analysis could include ultra-sensitive compositional observations using mass spectrometric and tunable diode laser techniques developed for stratospheric research. In addition to sampling the atmosphere, basic atmospheric meteorology can be performed which would include wind velocity, temperature and pressure measurements. The meteorological data would be taken at different vertical altitudes and at various points above the surface. This data can provide a comprehensive survey of the atmospheric conditions with altitude over a large surface region. This
type of grid sampling is demonstrated in Figure 1-34. Each node or line intersection would be a data collection point.

![Figure 1-34: Atmospheric sampling and data collection over grid](image)

The dust on Mars is one of the unique features of its atmosphere. Since there is no rain any dust particles lifted into the atmosphere tend to remain within the atmosphere for extended periods of time. This causes the optical depth of the planet to remain above 0.5 (based on Viking lander data). When dust storms occur, they can be global in size and last for months before dying down. Therefore the SSA can expect to fly with an optical depth of between 0.5 and 1.0 throughout its mission.

This dust is a major influence on the transfer of heat to and from the planets surface. Presently only particle size and optical properties are known about the dust on Mars. The SSA, flying above the surface, can sample the long-lived airborne dust. Key science objectives in understanding the effects of the dust on the Mars environment include direct measurements of the radiation field, direct determination of the size distribution of the airborne dust, and determination of the electrostatic charging of the dust.

The photochemistry and trace gases within the Martian atmosphere are not well understood. The chemistry of prime interest is the photodissociation of H$_2$O, O$_2$, and CO$_2$ which can result in the production of a variety of reactive oxidizing species such as O$_3$, H$_2$O$_2$, O, H, OH, HO$_2$ and possibly others. The concentration of these species can tell us about atmospheric photochemistry as well as provide insight into the nature of the oxidative processes that are responsible for the absence of organics in the Martian soil, which may be a key piece of evidence in looking for life on Mars. As with Venus, the search for trace gases such as CH$_4$, H$_2$S, NH$_3$, N$_2$O, C$_2$H$_6$, etc., which are reducing agents, would be of particular interest if detected on Mars. The presence of any of these reduced gases in the oxidizing environment of Mars would indicate the possibility of life on the surface or subsurface.
Chapter 2.0 SSA Configuration and Operation

2.1 Wing and Fuselage Material Development

The material for the solid state aircraft flapping and morphing wings are ionic polymer metal composites (IPMCs). These materials are multi-functional materials that are biomimetic distributed nanosensors, nanoactuators, nanotransducers, and artificial muscles. The ERI team conducted a thorough investigation on manufacturing the best possible material for the SSA project and on integrating such material with the SSA fuselage as well as thin sheet solar arrays and appropriate electronics. This final report further presents briefly the breadth and the depth of all aspects of chemistry, manufacturing, characterization, electronic driver, modeling and simulation and applications of IPMCs as biomimetic robotic distributed nanosensors, nanoactuators, nanotransducers and artificial/synthetic muscles for the SSA applications. One of the pictures of the small version of the SSA built by the SSA team is shown in Figures 2-1 and 2-2.

Figure 2-1: Smaller version of the SSA with its driving electronics
A larger version of the SSA was also constructed but proved difficult to flap due to excessive weight of the longer muscles with almost three feet of wingspan, as shown in Figure 2-3.
2.1.1 IPMC Development Background

The main task of the research team at ERI was to develop the optimum electroactive polymeric material (ionic polymer metal composites, IPMCs) for the SSA wings. The material had to be robust enough to flap under low electrical fields of a few 10's of volts per millimeter thickness of the wing material and also to be friendly enough to be integrated with thin film batteries and solar cells. In that endeavor the ERI team has been able to develop the right material for the fabrication of SSA wings. However, the size became an important factor.

The wings had to be a pair each having the dimensions of 18 inches (~46 cm) long and eight inches (~21 cm) wide with some thickness and width variations from the point of attachment to the body of the SSA or the fuselage of the SSA. Figure 2-4 depicts the general architecture of the SSA and the wings with just two pairs of electrodes attached to each wing.

Note in Figure 2-4 that the pairs of electrodes can be wired in an opposite manner so that when one pair is energizing and moving the wing in one direction the other pair does the opposite and moves the wing in the opposite direction. In this manner the wings can undulate as shown in Figure 2-5 to create a net lift and maneuvering capability.

In this endeavor work was performed with other team members through telecons and other means of communication to initiate the activities in connection with different tasks to be completed in various phases of activities.

A list of materials and chemicals for the initial manufacturing of deformable electroactive IPMC wing material was prepared and purchased and the ERI research team started the optimization of electromechanical properties of the wing material through the manufacturing process of wing materials for the SSA project.
Figure 2-5: Various configurations of the IPMC wings activated by the pairs of electrodes on each wing

2.1.2 General Considerations on IPMC as SSA Wing Material

Ionic polymeric materials suitably made into a functionally-graded composite with a conductor such as a metal, graphite or synthetic metal such as conductive polymers that act as a distributed electrode can exhibit large dynamic deformation if placed in a time-varying electric field (see Figures 2-6 and 2-7) [104, 107, 2].

A recent book by Shahinpoor, Kim and Mojarrad [114] and four fundamental review papers by Shahinpoor and Kim [112, 58, 105, 106] present a thorough coverage of the existing knowledge in connection with ionic polymeric conductor composites (IPCCs) including ionic polymeric metal composites (IPMCs) as biomimetic distributed nanosensors, nanoactuators and artificial muscles and electrically controllable polymeric network structures.
Figure 2-6: Four-fingered robotic grippers made with plastic muscles

Figure 2-7: Successive photographs of an IPMC strip showing very large deformation (sample in a and b are 1cmx8cmx0.34mm under 4 volts). Note that t=0.5 sec between a, b

Furthermore, in reference [114], methods of fabrication of several electrically and chemically active ionic polymeric gel muscles such as polyacrylonitrile (PAN), poly(2-acrylamido-2-methyl-1-propane sulfonic) acid (PAMPS), and polyacrylic-acid-bis-acrylamide (PAAM) as well as a new class of electrically active composite muscle such as Ionic Polymeric Conductor Composites (IPCC's) or Ionic Polymer Metal Composites (IPMCs) made with perfluorinated sulfonic or carboxylic ionic membranes (chlor-alkali family) are introduced and investigated that have resulted in seven US patents regarding their fabrication and application capabilities as distributed biomimetic nanoactuators, nanotransducers, nanorobots and nanosensors.

Theories and numerical simulations associated with ionic polymer gels electrodynamics and chemodynamics are also discussed, analyzed and modeled for the manufactured material.
In this final report we concentrate on perfluorinated sulfonic ionic membranes and ionic poly-acrylonitrile materials only, as potentially powerful ionic polymers for biomimetic distributed nanosensing, nanoactuation, nanorobotics, nanotransducers for power conversion and harvesting, as well as artificial muscles for medical and industrial applications.

It must be noted that widespread electrochemical processes and devices utilize poly (perfluorosulfonic acid) ionic polymers. These materials exhibit [112, 58, 105, 106, 57], good chemical stability, remarkable mechanical strength, good thermal stability, and high electrical conductivity when sufficiently hydrated and made into a composite with a conductive phase such as metals, conductive polymers or graphite.

As described in [106] a number of physical models have been developed to understand the mechanisms of water and ion transport in ionic polymers and membranes. Morphological features influence transport of ions in ionic polymers.

These features have been studied by a host of experimental techniques including: small and wide-angle x-ray scattering, dielectric relaxation, and a number of microscopic and spectroscopic studies.

The emerging picture of the morphology of chlor-alkali ionic polymers is that of a two-phase system made up of a polar fluid (water)-containing ion cluster network surrounded by a hydrophobic polytetrafluoroethylene (PTFE) medium. The integrity and structural stability of the membrane is provided by the PTFE backbones and the hydrophilic clusters facilitate the transport of ions and water in the ionic polymer.

These nanoclusters have been conceptually described as containing an interfacial region of hydrated, sulfonate-terminated perfluoroether side chains surrounding a central region of polar fluids. Counterions such as Na+ or Li+ are to be found in the vicinity of the sulfonates. It must be noted that the length of the side chains has a direct bearing on the separation between ionic domains, where the majority of the polar fluids resides, and the nonpolar domains. High resolution NMR of some perfluorionomer shows an unusual combination of a non-polar, Teflon-like backbone, with polar and ionic side branches. It has also been well established [105, 106] that anions are tethered to the polymer backbone and cations (H+, Na+, Li+) are mobile and solvated by polar or ionic liquids within the nanoclusters of size 3-5 nanometers.

### 2.1.3 IPMC Manufacturing Techniques

Manufacturing an IPMC begins with selection of an appropriate ionic polymeric material. Often, ionic polymeric materials are manufactured from polymers that consist of fixed covalent ionic groups. The currently available ionic polymeric materials that are convenient to be used as IPMC's are:

1. Perfluorinated alkenes with short side-chains terminated by ionic groups [typically sulfonate or carboxylate (SO3− or COO−) for cation exchange or ammonium cations for anion exchange (see Figure 2-8)]. The large polymer backbones determine their mechanical strength. Short side-chains provide ionic groups that interact with water and the passage of appropriate ions.
2. Styrene/divinylbenzene-based polymers in which the ionic groups have been substituted from the phenyl rings where the nitrogen atom is fixed to an ionic group. These polymers are highly crosslinked and are rigid.

\[
- \frac{(CF_2CF_2)_n}{CF_2} - \frac{CFO(CF_2 - CFO)_m}{CF_3} CF_2 CF_2 SO_3^- \cdots Na^+
\]

or

\[
- \frac{(CF_2CF_2)_x}{CF_2} - \frac{CFO(CF_2 - CFO)_m}{CF_3} (CF_2)_n SO_3^- \cdots Na^+
\]

*Figure 2-8: Perfluorinated sulfonic acid polymers*

In Figure 2-8, \( n \) is such that \( 5 < n < 11 \) and \( m \sim 1 \), and \( M^+ \) is the counter ion (\( H^+ \), \( Li^+ \) or \( Na^+ \)). This material is capable of absorbing large amounts of polar solvents, i.e. water. Metal ions, which are dispersed throughout the hydrophilic regions of the polymer, are subsequently reduced to the corresponding metal atoms. This results in the formation of a dendritic and fractal type electrodes penetrating into macromolecular network.

In perfluorinated sulfonic acid polymers there are relatively few fixed ionic groups. They are located at the end of side-chains so as to position themselves in their preferred orientation to some extent. Therefore, they can create hydrophilic nanochannels, so called *cluster networks*. Such configurations are drastically different in other polymers such as styrene/divinylbenzene families that limit, primarily by crosslinking, the ability of the ionic polymers to expand (due to their hydrophilic nature).

The preparation of ionic polymer-metal composites (IPMC's) requires extensive laboratory work including manufacturing polymer-metal composites by means of chemical oxidation-reduction (REDOX) operation. State-of-the-art IPMC manufacturing techniques\[114, 57\] incorporate two distinct preparation processes: first initial oxidation process and then final reduction process. Different preparation processes result in morphologies of precipitated platinum that are significantly different. Figure 2-9 shows illustrative schematics of two different preparation processes (top-left and bottom-left) and two top-view SEM micrographs for the platinum surface-electrode (top-right and bottom-right).

Note in Figure 2-9 that (top-left) is a schematic showing the initial process of making the ionic polymer metal composite, (top-right) shows its top-view SEM micrograph, while (bottom-left) shows a schematic of the process of depositing surface electrodes on the ionic polymer and the (bottom-right) shows its top-view SEM micrograph where platinum deposited predominately on top of the initial Pt layer.
The initial process of making the functionally-graded composite (IPMC's) requires an appropriate platinum salt such as Pt(NH$_3$)$_4$HCl or Pd(NH$_3$)$_4$HCl in the context of chemical reduction processes.

The principle of the process of making a functionally graded composite is to metallize the inner surface of the material by a chemical-reduction means such as LiBH$_4$ or NaBH$_4$.

The ionic polymeric material is soaked in a salt solution to allow platinum-containing cations to diffuse through via the ion-exchange process (oxidation). Later, a reducing agent such as LiBH$_4$ or NaBH$_4$ is introduced to platinize the material (reduction). As can be seen in Figure 2-10, the metallic platinum particles are not homogeneously formed across the membrane but concentrate predominantly near the interface boundaries.

It has been experimentally observed that the platinum particulate layer is buried microns deep (typically 1-20 µm) within the IPMC surface and is highly dispersed.

The fabricated IPMC's can be optimized to produce a maximum force density by changing multiple process parameters. These parameters include time-dependent concentrations of the salt and the reducing agents. The primary reaction is,

$$LiBH_4 + 4[Pt(NH_3)_4]^+ + 8OH^- \rightarrow 4Pt^0 + 16NH_3 + LiBO_2 + 6H_2O$$

Equation 2-1

![Figure 2-9: Two schematic diagrams showing different preparation processes](image)
Figure 2-10: Two SEM micrographs showing the cross section (left) and close-up (right) of a typical IPMC

In the subsequent placement of surface electrodes, multiple reducing agents are introduced (under optimized concentrations) to carry out the reducing reaction similar to Equation 2-1, in addition to the platinum layer formed by the initial process of making the composite.

This is clearly shown in Figure 2-9 (bottom-right), where the roughened surface disappears. In general, the majority of platinum salts stays in the solution and precedes the reducing reactions and production of platinum metal. Other metals (or conductive media), which are also successfully used, include palladium, silver, gold, carbon, graphite, and carbon nanotubes.

To characterize the surface morphology of the IPMC, Atomic Force Microscopy (AFM) can be used. Its capability to directly image the surface of the IPMC can provide detailed information with a resolution of a few nano-meters.

In Figure 2-11, a number of representative AFM images (its surface analysis) reveal the surface morphology of the IPMC's.

As can be seen, depending on the initial surface roughening, the surface is characterized by the granular appearance of platinum metal with a peak/valley depth of approximately 50 nm. This granular nano-roughness is responsible for producing a high level of electric resistance, yet provides a porous layer that allows polar liquid movement in and out of the ionic polymer.

During the AFM study, it was also found that platinum particles are dense and, to some extent, possess coagulated shapes. Therefore, the study was extended to utilize TEM (transmission electron microscopy), to determine the size of the deposited platinum particles.
In Figure 2-11, the scanned area is 1 µm². The brighter/darker area corresponds to a peak/valley depth of 50 nm. The surface analysis image has a view angle set at 22 degrees.

Figure 2-12 shows a TEM image on the penetrating edge of the IPMC.

The sample was carefully prepared in the form of a small size and was ion-beam treated. The average particle size was found to be around 47 nm.
2.1.4 Three-dimensional Fabrication of SSA Wing Material

It was important to make the SSA wings with certain three-dimensional geometry. The previous findings of Kim and Shahinpoor [57] were utilized for this purpose. Kim and Shahinpoor [57] had reported fabrication methods that can scale-up or down the IPMC artificial muscles in a strip size of micro-to-centimeter thickness for various soft robotic applications, using liquid form of perfluorinated ionic polymers. By meticulously evaporating the solvent (isopropyl alcohol) out of the solution, recast ionic polymer can be obtained [57]. The SSA team then decided to look into making a mold for the SSA wing using stereolithography or selected laser sintering via rapid prototyping approach. ERI had to revert to an alternate approach (plan B) for the manufacturing of the larger wings because it was decided that it would take too much time to manufacture a mold using the rapid prototyping facilities at NASA Glenn using selected laser sintering to generate the required STL (stereo lithography) slice files. Plan B for the manufacturing of the larger wing material was based on lamination of melted pieces into the right shape for the larger wing.

Plans were initiated and carried out to manufacture larger wings from ionic polymeric materials robust enough to flap under low electrical fields of a few 10's of volts per millimeter thickness of the wing material and also to be compatible enough to be integrated with thin film solar arrays. For this demonstration the available thin film solar arrays proved to be too stiff for the flexibility required by the wing materials and it was decided to integrate the photovoltaic solar arrays with the body of the SSA rather than SSA wing materials.

The ERI research team subsequently manufactured four more large sheets (21cmx46 cm by 0.3mm thickness) of ionic polymeric metal composite wing materials. The goal was to cut these to the right shape for the wings and to laminate them into right thickness variation, from the point of attachment to the body of the SSA bird or the fuselage of the SSA to the tip of the wings, for the final fabrication of the SSA wings. In this endeavor work was performed with other team members through telecons and other means of communication to initiate the activities in connection with different tasks to be completed.

The first stationary flapping wing prototype model of the SSA was fabricated using a one gallon plastic soda bottle the fuselage with attached electrode arrays (Figure 2-13) which also act as supporting structures for the larger wings stretching some 41 cm on each side, making the span about 1 meter. Drooping is currently an issue, which we are dealing with. However, the drooping seen by this ground model will not be an issue for an actual flight vehicle. This is because when the SSA is in flight it the aerodynamic forces on the wing can be utilized to maintain the wing in a horizontal position.
Figure 2-13: The larger SSA fuselage with attached electrode arrays (which also act as supporting structures for the larger wings stretching some 41 cm on each side, making the span about 1 meter), as well as the flapping tail.

Figures 2-14 and 2-15 depict some of the larger wings manufactured.

Figure 2-14: Manufacturing of initial larger SSA wings.
Chapter 2.0 SSA Configuration and Operation
2.1 Wing and Fuselage Material Development

Figure 2-15: The larger SSA wings made with IPMCs and the anticipated model SSA wing span

We also designed and built two smaller IPMC material sections in the shape of the SSA planform with the electronic drivers. Figure 2-16 depicts the smaller IPMC section and the associated electronic board and the driver. Figure 2-17 depicts the bending motion achieved with a single centrally located electrode.

Figure 2-16: Smaller versions of the SSA and the electronic driver board demonstrated at the SPIE Electroactive Conference
The electronic circuit board and drivers for the two smaller demonstration IPMC segments are shown in Figure 2-18. These were designed for the optimum operation of the smaller IPMC sections. Note that the bigger version of the SSA is designed to have thin solar arrays integrated with fuselage of the SSA.
2.1.5 Electrical Performance

In order to assess the electrical properties of IPMC, the standard AC impedance method that can reveal the equivalent electric circuit has been adopted. A typical measured impedance plot, provided in Figure 2-19, shows the frequency dependency of impedance of the IPMC. An impedance analyzer was purchased and detailed experimental tests were performed in connection with the SSA flight performance. It is interesting to note that the IPMC is nearly resistive (> 50\(\Omega\)) in the high frequency range and fairly capacitive (> 100 \(\mu F\)) in the low frequency range.

*Figure 2-19: The measured AC impedance spectra (magnitude) of an IPMC sample. The IPMC sample has a dimension of 5 mm width, 20 mm length, and 0.2 mm thickness*

Based upon the above findings, we considered a simplified equivalent electric circuit of the typical IPMC such as the one shown in Figure 2-20.

*Figure 2-20: A possible equivalent electric circuit of typical IPMC*
In this approach, each single unit-circuit (i) was assumed to be connected in a series of arbitrary surface-resistance ($R_{ss}$) in the surface. This approach was based upon the experimental observation of the considerable surface-electrode resistance.

We assumed that there are four components to each single unit-circuit: the surface-electrode resistance ($R_s$), the polymer resistance ($R_p$), the capacitance related to the ionic polymer and the double layer at the surface-electrode/electrolyte interface ($C_d$) and an impedance ($Z_w$) due to a charge transfer resistance near the surface electrode. For the typical IPMC, the importance of $R_{ss}$ relative to $R_s$ may be interpreted from $\Sigma R_{ss} / R_s \approx L / t >> 1$, where notations $L$ and $t$ are the length and thickness of the electrode, respectively. It becomes a two dimensional electrode. In order to increase the surface conductivity, a thin layer of a highly conductive metal (such as gold) is deposited on top of the platinum surface-electrode [113].

Figure 2-21 depicts measured cyclic current/voltage responses of a typical IPMC (the scan rate of 100 mV/sec).

As can be seen, a rather simple behavior with a small hysteresis is obtained. Note that the reactivity of the IPMC is mild such that it does not show any distinct reduction or re-oxidation peaks within +/- 4 volts, except for a decomposition behavior at $\sim \pm 1.5$ V where the extra current consumption is apparent due to electrolysis. Overall behavior of the IPMC shows a simple trend of ionic motions caused under an imposed electric field.

Note that the scan rate is 100 mV/sec. A simple behavior with a small hysteresis can be seen. It does not show any distinct reduction or re-oxidation peaks within +/- 4 volts, except for a decomposition behavior at $\sim \pm 1.5$ V where the extra current consumption is apparent due to electrolysis.

For theoretical calculation the following experimentally measured values were used;
1. $L_{12} = L_{21} = 2 \times 10^{-8}$ \{cross coefficient, (m/s)/(V/m)\},

2. $k = 1.8 \times 10^{-18}$ \{hydraulic permeability, m$^2$\}, and iii) $\bar{E} = E_0 / h$ where $h = 200$ µm \{membrane thickness\}.

### 2.1.6 Thermodynamic Efficiency

The bending force of the IPMC is generated by the effective redistribution of hydrated ions. This is an ion-induced actuation phenomenon. Typically, such a bending force is electric field-dependently distributed along the length of the IPMC strip. The IPMC SSA wing strip bends due to this ion migration-induced hydraulic actuation and redistribution.

The bending force of the IPMC wing strip is exerted by the effectively strained IPMC due to hydrated ions transport. Typically, such force is field-dependently distributed along the length of the IPMC strip. The IPMC wing strip bends due to this force. The total bending force, $F_t$, can be approximated as,

$$F_t \approx \int_0^L f dS$$  \hspace{1cm} \text{Equation 2-2}

where $f$ is the force density per unit arc length $S$ and $L$ is the effective beam length of the IPMC strip. Assuming a uniformly distributed load over the length of the IPMC, then, the mechanical power produced by the IPMC strip can be obtained from,

$$P_{out} = \frac{1}{2} \int_0^L f v dS$$ \hspace{1cm} \text{Equation 2-3}

Where $v$ is the local velocity of the IPMC in motion. Note that $v$ is a function of $S$ and can be assumed to linearly vary, such that $v = (v_{tip}/L)S$, $0 \leq S \leq L$. Finally, the thermodynamic efficiency, $E_{\text{ff,em}}$, can be obtained as,

$$E_{\text{ff,em}} (\%) = \frac{P_{out}}{P_{in}} \times 100$$ \hspace{1cm} \text{Equation 2-4}

where $P_{in}$ is the electrical power input to the IPMC, i.e., $P_{in} = V(t)I(t)$, where $V$ and $I$ are the applied voltage and current, respectively.

Based on Equation 2-4, one can construct a graph (see Figure 2-22) which depicts the thermodynamic efficiency of the IPMC as a function of frequency. Note that this graph presents the experimental results for the conventional IPMC and the improved additive-treated IPMC.
It is of note that the optimum efficiencies occur at near 5-10 Hz for these IPMCs. The optimum values of these IPMC’s are approximately 2.5-3.0%. At low frequencies, the water leakage out of the surface electrode seems to cost the efficiency significantly. However, the additive-treated IPMC shows a dramatic improvement in efficiency.

The important sources of energy consumption for the IPMC actuation could be from,

- the necessary mechanical energy needed to cause the positive/negative strains for the IPMC strip,
- the I/V hysteresis due to the diffusional water transport within the IPMC,
- the thermal losses-Joule heating (see Figure 2-26),
- the decomposition due to water electrolysis, and
- the water leakage out of the electrode surfaces.

In Figure 2-22 the samples have a dimension of a 20 mm length and a 5 mm width and 0.2 mm thickness, the applied potential is 1-volt step, lines are least square fits and resonant efficiencies are not included in the figure.

Figure 2-23 displays IR thermographs taken for an IPMC in action (the sample size of 1.2 x 7.0 cm). They show spectacular multi-species mass/heat transfer in a sample of IPMC under an oscillatory step voltage of 3 and frequency of 0.1 Hz. The temperature difference is more than 10 degrees C. In general, the hot spot starts from the electrode and propagates toward the tip of the IPMC strip (left to right).
Figure 2-23: IR thermographs of an IPMC inaction. The hot spot starts from the electrode and propagates toward the tip of the IPMC strip. The electrode is positioned in the left side of the IPMC. The temperature difference is more than 10 degrees C when a DC voltage of 3 was applied for the IPMC sample size of 1.2 x 7.0 cm.

The thermal propagation is simultaneously conjugated with the mass transfer along with the possible electrochemical reactions.

It clearly shows the significance of water transport within the IPMC. These coupled transport phenomena are currently under investigation.

2.1.7 Force Density for SSA Wing Materials

Figure 2-24 depicts the force density experiments with IPMC SSA wing material. As depicted in these figures the IPMC can lift more that 40 times its own weight and thus is highly suitable for the SSA wing material.

Figure 2-24: IPMC SSA wing material lifting 20 times its own weight (left) and 40 times its own weight (right)
2.1.8 Cold Temperature Properties of IPMC For High Altitude and Planetary Applications

In order to determine the cold temperature operating characteristics of the SSA wing made with IPMC sensors and actuators in connection with the harsh conditions on Mars, Venus and at high altitudes on Earth, various samples of IPMC had been tested previously in a cryochamber under very low pressures of down to 2 Torrs and temperatures of down to -150 degrees Celsius.

*Figure 2-25: Deflection of the IPMC strip as function of voltage and temperature*

This was done to simulate these harsh cold low-pressure environments. The results are depicted in the following Figures 2-23 through 2-28.

*Figure 2-26: Deflection characteristics of IPMC as a function of time and temperature*
Figure 2-27: Power consumption of the IPMC strip bending actuator as a function of activation voltage
Figure 2-28: Deflection versus current drawn (top) and power input (bottom) at a high pressure of 850 Torrs and a low pressure of 0.4-1 Torrs.
2.1 Wing and Fuselage Material Development

Shahinpoor has presented a review on sensing and transduction properties of ionic polymer conductor composites [111]. Shahinpoor also reported [108, 109] that IPMCs by themselves and not in a hydrogen pressure electrochemical cells as reported by Sadeghipour, Salomom and Naogi [96] can generate electrical power like an electromechanical battery if flexed, bent or squeezed. Shahinpoor [108, 109] reported the discovery of a new effect in ionic polymeric ionic polymeric

---

**Figure 2-29: IPMC strip static (V/I) and dynamic (V/I resistance at various temperatures**

**Figure 2-30: The relation between voltage and current for an IPMC strip that was exposed to Room T=20 °C and to -100 °C**
gels, namely the ionic flexogelectric effect in which flexing, compression or loading of IPMC strips in air created an output voltage like a dynamic sensor or a transducer converting mechanical energy to electrical energy. Keshavarzi, Shahinpoor, Kim and Lantz [56] applied the transduction capability of IPMC to the measurement of blood pressure, pulse rate, and rhythm measurement using thin sheets of IPMC’s. Motivated by the idea of measuring pressure in the human spine, [40] applied pressure across the thickness of an IPMC strip while measuring the output voltage. Typically, flexing of such material in a cantilever form sets them into a damped vibration mode that can generate a similar damped signal in the form of electrical power (voltage or current) as shown in Figures 2-31 and 2-32.

**Figure 2-31: A typical sensor configuration and voltage response of an IPMC**

**Figure 2-32: A typical voltage response of an IPMC**
The experimental results showed that almost a linear relationship exists between the voltage output and the imposed displacement of the tip of the IMPC sensor (Figure 2-31). IPMC sheets can also generate power under normal pressure. Thin sheets of IPMC were stacked and subjected to normal pressure and normal impacts and were observed to generate large output voltage. Endo-ionic motion within IPMC thin sheet batteries produced an induced voltage across the thickness of these sheets when a normal or shear load was applied. A material testing system (MTS) was used to apply consistent pure compressive loads of 200N and 350N across the surface of an IPMC 2x2cm sheet. The output pressure response for the 200N load (73 psi) was 80mV in amplitude and for the 350N (127 psi) it was 108 mV. This type of power generation may be useful in the heels of boots and shoes or places where there are a lot of foot or car traffic.

Figure 2-34 depicts the output voltage of the thin sheet IPMC batteries under 200N normal load. The output voltage is generally about 2 mV/cm length of the IPMC sheet.

This capability to produce a voltage when bent provides a key benefit to the SSA concept. It enables the wing material to act as a feedback sensor providing motion and perturbation information to the control system. By having this built-in sensor network, the wings of the SSA will provide continuous feedback enabling quick response to any perturbation experienced by the aircraft. This inherent feedback capability is similar in function and capability to a living creature’s nervous system.
Figure 2-34: Out voltage due to normal impact of 200N load on a 2cmx2cmx0.2mm IPMC sample

Table 2-1 depicts the general properties of ionic polymeric muscles for soft robotic applications.

**Table 2-1: Current Capabilities of IPMC Materials**

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Young's Modulus, E</td>
<td>Up to 2 GPa</td>
</tr>
<tr>
<td>Shear Modulus, G</td>
<td>Up to 1 GPa</td>
</tr>
<tr>
<td>Poisson's ratio,</td>
<td>Typical: 0.3-0.4</td>
</tr>
<tr>
<td>Power density (W/volume))</td>
<td>Up to 200 mW/cc</td>
</tr>
<tr>
<td>Max force density (Cantilever Mode)</td>
<td>Up to 50 Kgf/Kg</td>
</tr>
<tr>
<td>Max displacement/strain</td>
<td>Up to 4% linear strain</td>
</tr>
<tr>
<td>Bandwidth (speed)</td>
<td>Over kHz in a cantilever vibratory mode for soft robotic actuation</td>
</tr>
<tr>
<td></td>
<td>Over MHz for soft sensing in cantilever mode</td>
</tr>
<tr>
<td>Resolution (force and displacement control)</td>
<td>Displacement accuracy down to 1 micron Fire resolution down to 1 mg</td>
</tr>
<tr>
<td>Efficiency (electromechanical)</td>
<td>More than 25% (frequency dependent) for actuation</td>
</tr>
<tr>
<td></td>
<td>More than 90% for sensing</td>
</tr>
<tr>
<td>Density</td>
<td>Down to 1.8 g/cm3</td>
</tr>
</tbody>
</table>
2.1.9 Modeling and Simulation

De Gennes, Okumura, Shahinpoor and Kim [24] presented the first phenomenological theory for sensing and actuation in ionic polymer metal composites. Asaka and Oguro [7] discussed the bending of polyelectrolyte membrane-platinum composites by electric stimuli and presented a theory on actuation mechanisms in IPMC by considering the electro-osmotic drag term in transport equations. Nemat-Nasser and Li [81] presented a modeling on the electromechanical response of ionic polymer-metal composites based on electrostatic attraction/repulsion forces in IPMC’s. Later, Nemat-Nasser [80] presented a revised version of their earlier paper and stressed the role of hydrated cation transport within the clusters and polymeric networks in IPMC’s. Nemat-Nasser and Wu [82] have presented a discussion on the role of backbone ionic polymer and in particular sulfonic versus carboxylic ionic polymers, as well as, the effect of different cations such as K+, Na+, Li+, Cs+ and some organometallic cations on the actuation and sensing performance of IPMC’s. Tadokoro [126], Tadokoro, Yamagami, Takamori and Oguro [128] and Tadokoro, Takamori and Oguro [105] have presented an actuator model of IPMC for robotic applications on the basis of physico-chemical phenomena. A recent comprehensive review by Shahinpoor and Kim [105] on modeling and simulation of ionic polymeric artificial muscles discusses the various modeling approaches for the understanding of the mechanisms of sensing and actuation of ionic polymers. The most plausible mode of actuation and sensing is expected to be the following: under the field $E$, the cations drift, and they carry with them a certain number of water molecules depending on the hydration number $n$ of the cation $M^{(+)}$. When these hydrated molecules of water carried by cations pile up near the cathode, they create a local over-pressure (see Figures 2-35 and 2-36), which tends to deform the material. A thin membrane, which was originally flat, then tends to acquire a certain spontaneous curvature $C(E) = 1/\rho_c$ where $\rho_c$ is the radius of curvature.
Figure 2-35: Bending of an ionic gel strip due to an imposed electric field gradient
These observations clearly establish that the main mechanism of actuation is polar plasticizer migration in the form of cationic hydrated water as well as and the Coulombic type ionic redistribution and forces may play a minor role. Figure 2-37 depicts the effect of changing cations on the force capabilities of SSA wing material.
Figure 2-38 depicts experimental evidence for the effect of different ions and their hydration numbers on the tip force and thus deformation of an IPMC strip.

![Figure 2-38: Experimental evidence for the effect of different ions and their hydration numbers on the tip force and thus deformation of an IPMC strip](image)

Balance of forces on individual cations hydrated with $n$ molecules of water inside the molecular network, clusters and channels based on the diffusion-drift model of ionic media due to Nernst and Plank (Nernst-Plank equation) can be stated as:

$$Ne\rho_{M^+}E_x(x,y,z,t) = N(\rho_{M^+}M_{M^+} + n\rho_wM_w)(\frac{dv_x}{dt}) + N\rho_{M^+}\eta v_x + Ne\rho_{M^+}kT\left(\frac{\partial \ln(\rho_{M^+} + n\rho_w)}{\partial x} + \frac{\partial P}{\partial x}\right)$$  \hspace{1cm} \textbf{Equation 2-5}

$$Ne\rho_{M^+}E_y(x,y,z,t) = N(\rho_{M^+}M_{M^+} + n\rho_wM_w)(\frac{dv_y}{dt}) + N\rho_{M^+}\eta v_y + Ne\rho_{M^+}kT\left(\frac{\partial \ln(\rho_{M^+} + n\rho_w)}{\partial y} + \frac{\partial P}{\partial y}\right)$$  \hspace{1cm} \textbf{Equation 2-6}

$$Ne\rho_{M^+}E_z(x,y,z,t) = N(\rho_{M^+}M_{M^+} + n\rho_wM_w)(\frac{dv_z}{dt}) + N\rho_{M^+}\eta v_z + Ne\rho_{M^+}kT\left(\frac{\partial \ln(\rho_{M^+} + n\rho_w)}{\partial z} + \frac{\partial P}{\partial z}\right)$$  \hspace{1cm} \textbf{Equation 2-7}

where, $M_{M^+}$ and $M_w$ are the molecular weight of cations and water, respectively, $P$ is the local osmotic fluid pressure $P_f$ minus the local swelling pressure or stress $\sigma^*$, such that $P=P_f-\sigma^*$, and:

$$e^{\psi_0}(x,y,z,t) = \left[eE(x,y,z,t)_{x}, eE(x,y,z,t)_{y}, eE(x,y,z,t)_{z}\right]^T$$  \hspace{1cm} \textbf{Equation 2-8}
Chapter 2.0 SSA Configuration and Operation

2.1 Wing and Fuselage Material Development

is the force vector on an individual cation due to electro-osmotic motion of an ion in an electric field, \( k \) is the Boltzmann's constant and:

\[
\mathbf{v}(x,y,z,t) = [v_x(x,y,z,t), v_y(x,y,z,t), v_z(x,y,z,t)]^T
\]

Equation 2-9

is the velocity vector of the hydrated cations, and

\[
\eta \mathbf{v}(x,y,z,t) = [\eta v_x(x,y,z,t), \eta v_y(x,y,z,t), \eta v_z(x,y,z,t)]^T
\]

Equation 2-10

is the force vector of the viscous resistance to the motion of individual hydrated cations in the presence of a viscous fluid medium with a viscosity of \( \eta \), and

\[
kT \nabla [\ln(\rho_{M+}(x,y,z,t) + n\rho_w(x,y,z,t))] \]

Equation 2-11

is the force vector due to diffusion of individual cations and accompanying molecules of hydrated water in the polymer network with the following x, y, and z components, respectively:

\[
kT \left( \frac{\partial \ln[\rho_{M+}(x,y,z,t) + n\rho_w(x,y,z,t)]}{\partial x} \right)
\]

Equation 2-12

\[
kT \left( \frac{\partial \ln[\rho_{M+}(x,y,z,t) + n\rho_w(x,y,z,t)]}{\partial y} \right)
\]

Equation 2-13

\[
kT \left( \frac{\partial \ln[\rho_{M+}(x,y,z,t) + n\rho_w(x,y,z,t)]}{\partial z} \right)
\]

Equation 2-14

and the force vector due to inertial effects on an individual hydrated cation is:

\[
(M_{M+} + nM_w) \left( \frac{d\mathbf{v}}{dt} \right)
\]

Equation 2-15

such that in a compact vectorial form the force balance equation reads:

\[
N e \rho_{M+} \mathbf{E} = N(\rho_{M+}M_{M+} + n\rho_wM_w)(\frac{d\mathbf{v}}{dt}) + N \rho_{M+}\mathbf{v} + N \rho_{M+}kT \nabla [\ln(\rho_{M+} + n\rho_w)] + \nabla P_f - \nabla \cdot \mathbf{\sigma}^* \]

Equation 2-16

where the stress tensor \( \mathbf{\sigma}^* \) can be expressed in terms of the deformation gradients in a non-linear manner such as in Neo-Hookean or Mooney-Rivlin type constitutive equations as suggested by Segalman, Adolf, Witkowski and Shahinpoor, [103, 101, 102, 99, 100]. The flux of hydrated cations is given by:

\[
Q = [\rho_{M+}(x,y,z,t) + n\rho_w(x,y,z,t)] \mathbf{v}(x,y,z,t)
\]

Equation 2-17

Such that the equation of continuity becomes:

\[
\left( \frac{\partial [\rho_{M+}(x,y,z,t) + n\rho_w(x,y,z,t)]}{\partial t} \right) = -\nabla \cdot Q
\]

Equation 2-18

Equations 2-2 through 2-18 are the governing equations for the dynamics of ionic polymer metal composites as biomimetic distributed nanosensors, nanoactuators, soft nanorobots and artificial muscles.
Here below, a linear steady state version of the formulation is presented to obtain some preliminary understanding of the complex ionic diffusion and drift in these electronic materials. Figures 37(a) and 37(b) depict the general structure of the IPMCs after chemical plating and compose manufacturing. The structure bends towards the anode. The nature of water and hydrated ions transport within the IPMC can affect the moduli at different frequencies.

Let us now summarize the underlying principle of the IPMC’s actuation and sensing capabilities, which can be described by the standard Onsager formulation using linear irreversible thermodynamics.

![General structures of an IPMC or IPCC film with near boundary functionally graded electrodes and surface electrodes](image)

**Figure 2-39: General structures of an IPMC or IPCC film with near boundary functionally graded electrodes and surface electrodes**

When static conditions are imposed, a simple description of mechanolectric effect is possible based upon two forms of transport: ion transport (with a current density, $J$, normal to the material) and solvent transport (with a flux, $Q$, we can assume that this term is water flux). The conjugate forces include the electric field, $E$, and the pressure gradient, $-\nabla p$. The resulting equation has the concise form of,

$$J(x, y, z, t) = \sigma E(x, y, z, t) - L_{12} \nabla p(x, y, z, t)$$  \hspace{1cm} \textbf{Equation 2-19}

$$Q(x, y, z, t) = L_{21} E(x, y, z, t) - K \nabla p(x, y, z, t)$$  \hspace{1cm} \textbf{Equation 2-20}

where $\sigma$ and $K$ are the material electric conductance and the Darcy permeability, respectively. A cross coefficient is usually $L = L_{12} = L_{21}$. The simplicity of the above equations provides a compact view of the underlying principles of both actuation, transduction and sensing of the IPMCs as also shown in Figure 2-13.

When we measure the direct effect (actuation mode, Figure 2-35), we work (ideally) with electrodes which are impermeable to water, and thus we have $Q = 0$. This gives:

$$\nabla p(x, y, z, t) = \frac{L}{K} E(x, y, z, t)$$  \hspace{1cm} \textbf{Equation 2-21}

This $\nabla p(x, y, z, t)$ will, in turn, induce a curvature $\kappa$ proportional to $\nabla p(x, y, z, t)$. The relationships between the curvature $\kappa$ and pressure gradient $\nabla p(x, y, z, t)$ are fully derived and
described in Reference [24]. Let us just mention that \( (1/\rho_c)=M(E)/YI \), where \( M(E) \) is the local induced bending moment and is a function of the imposed electric field \( E \), \( Y \) is the Young’s modulus (elastic stiffness) of the strip which is a function of the hydration \( H \) of the IPMC and \( I \) is the moment of inertia of the strip. Note that locally \( M(E) \) is related to the pressure gradient such that in a simplified scalar format:

\[
\nabla p(x, y, z, t) = (2P/t^*) = (M/I) = Y/\rho_c = Y \kappa .
\]

Equation 2-22

Now from Equation 2-22 it is clear that the vectorial form of curvature \( \kappa_E \) is related to the imposed electric field \( E \) by:

\[
\kappa_E = (L/KY) E
\]

Equation 2-23

Based on this simplified model the tip bending deflection \( \delta_{\text{max}} \) of an IPMC strip of length \( l_g \) should be almost linearly related to the imposed electric field due to the fact that:

\[
\kappa_E \cong \left[ 2\delta_{\text{max}} / (l_g^2 + \delta_{\text{max}}^2) \right] \cong 2\delta_{\text{max}} / l_g^2 \cong (L/KY)
\]

Equation 2-24

The experimental deformation characteristics depicted in Figure 2-40.

**Figure 2-40: Actuation under a low frequency electric field**

Note that the results depicted in Figure 2-40 are clearly consistent with the above predictions obtained by the above linear irreversible thermodynamics formulation which is also consistent with Equations 2-23 and 2-24 in the steady state conditions and has been used to estimate the value of the Onsager coefficient \( L \) to be of the order of \( 10^{-8} \text{ m}^2/\text{V-s} \). Other parameters have been experimentally measured to be \( K \sim 10^{-18} \text{ m}^2/\text{CP}, \sigma \sim 1 \text{ A/mV} \text{ or S/m}. \) Figure 2-41 depicts the
experimental results of IPMC deflection under a step voltage with and without a dispersant poly-vinyl pyrrolidone (PVP).

Figure 2-41: Step response displacement characteristics of IPMC samples ($\delta$: arc length, $L_o$: effective beam length), $L_o = 1.0$ inch (left) and $L_o = 1.5$ inch (right)

The experimental deformation characteristics depicted in Figures 2-40 and 2-41 are clearly consistent with the above predictions obtained by the above linear irreversible thermodynamics formulation which is also consistent with Equations 2-23 and 2-24 in the steady state conditions and has been used to estimate the value of the Onsager coefficient $L$ to be of the order of $10^{-8}$ m$^2$/V-s.

Here, we have used a low frequency electric field in order to minimize the effect of loose water back diffusion under a step voltage or a DC electric field.

Other parameters have been experimentally measured to be $K \sim 10^{-18}$ m$^2$/CP, $\sigma \sim 1$ A/mV or S/m.

Figure 2-42 depicts a more detailed set of data pertaining to Onsager coefficient $L$. 
Chapter 2.0 SSA Configuration and Operation
2.1 Wing and Fuselage Material Development

Figure 2-42: Experimental determination of Onsager coefficient $L$ using three different samples

When we study the inverse effect (transduction mode), we apply a bending moment $M$ to the cantilever membrane, and we impose two conditions:

1. no current is produced ($J = 0$)
2. the strip stays bent with a curvature $\kappa_E$.

Then, as fully derived and described in Reference [24], the water pressure gradient $\nabla p(x, y, z, t)$ turns out to be proportional to $M$.

The condition $J = 0$ gives from Equation (6.127):

$$\bar{E}(x, y, z, t) = \frac{L}{\sigma} \nabla p(x, y, z, t)$$

and since

$$\nabla p(x, y, z, t) = (M / I)$$

the electric field $E(x, y, z, t)$ generated is thus proportional to $M$ according to:

$$E(x, y, z, t) = M \left( \frac{L}{\sigma I} \right) = \kappa_E \left( \frac{LY}{\sigma} \right)$$

It clearly establishes that for any imposed curvature $\kappa_E$ by the application of a bending moment $M$ an electric field $E(x, y, z, t)$ is generated.

It must, however, be noted that although the value of $L/\sigma$ is considerably smaller, by an almost two orders of magnitude, than the corresponding one in the actuation mode in the presence of an imposed electric field. This is due to the fact that in actuation mode, the imposition of an exter-
nal electric field changes the values of both $L$ and $\sigma$. Thus, imposing the same deformation creates an electric field almost two orders of magnitude smaller than the electric field necessary to generate the same deformation. The reader is referred to References [112, 58, 57] for further discussion on these issues.

For other formulations of micromechanics of IPMC actuations and sensing the reader is referred to References [7, 81, 126].

Based on the above equations the equation for the total curvature of ionic polymer strip becomes:

**Equation 2-28**

$$\kappa_E = \left(\frac{1}{2}YC^*\right)\left\{RT\left[2C_gV\left[1 - \exp\left((-t / R_gC_g)\right)\right]\right]/(n*eA_v(l_gw_gt_g)) + 2\Pi_{hw}\right\}$$

Note that units in Equation 2–28 are consistent, namely that $2Q/(n*eA_v(l_gw_gt_g))$ has the unit of number of gm-moles per m$^2$ and thus, that times RT gives the units of pressure in Pa which is then multiplied by $(1/2YC^*)$ that leaves the units of (1/m) because $Y$ also has the units of Pa. Note that the electric field $E$ is given by $E = V/2C^*$ and thus Equation 2-28 further simplifies to:

**Equation 2-29**

$$\kappa_E = E\left(2C_gRT/n*eA_v(l_gw_gt_g)Y\right)\left[1 - \exp\left((-t / R_gC_g)\right)\right] + (\Pi_{hw}/YC^*)$$

If the curvature at time $t=0$ is denoted by $\kappa_0$, then Equation 2-29 is further generalized to:

$$\kappa_E - \kappa_0 = E\left(2C_gRT/n*eA_v(l_gw_gt_g)Y\right)\left[1 - \exp\left((-t / R_gC_g)\right)\right] + (\Pi_{hw}/YC^*)$$

**Equation 2-30**

Equations 2-29 and 2-30 are simple expressions for the time-dependent curvature of the strip in an average and approximate fashion and clearly indicate that the induced curvature is directly proportional to the imposed electric field $E$ is a non-linear function of the capacitance of the gel strip $C_g$ and cross-resistance $R_g$, and inversely proportional to the strip volume and the Young modulus of elasticity of the strip $Y$.

These observations are in complete harmony with experimental results on the bending of IPMC strips. Figures 41, 42 and 43 display a number of simulations for curvature $\kappa_E$ versus capacitance $C_g$, electric field $E$, resistance $R_g$ and time $t$. 
Note in Figure 2-43 and subsequently in Figures 2-44 and 2-45 that the following values were used for the parameters:

- $E=20000 \text{ Volt/m}=J/C-m$ 
- $C_g=1000E-6 \text{ capacitance, C/V, F=C/V, 200 microfarads}$
- $R=8.314 \text{ gas constant, Pa-m}^3/\text{gmole-K}$
- $T=300 \text{ absolute temperature, K}$
- $V_g=(l_g w_g t_g) =2E-8 \text{ sample volume, m}^3$, $l_g=2\text{cm}$, $w_g=0.5\text{cm}$, $t_g=0.2\text{mm}$,
- $Y=100E6 \text{ modulus of the sample, Pa}$
- $n=+1 \text{ valance charge}$
- $e_1=1.602192E-19 \text{ an electron charge, C}$
- $Av=6.022E23 \text{ Avogadro's number, 1/mole}$
- $Rg=100 \text{ resistance, 100 Ohm=V/A, also C=A*sec}$

**Figure 2-43: Variation of curvature versus cross capacitance $C_g$ and time $t$.**
Figure 2-44: Variation of curvature versus cross resistance $R_g$ and time $t$.

Figure 2-45: Variation of curvature versus cross electric field $E$ and time $t$. 
In a cantilever mode the maximum tip deflection $\delta_{\text{max}}$ can be shown to be approximately related to the absolute value of the curvature $|\kappa_E|$ by:

$$|\kappa_E| = \frac{2\delta_{\text{max}}}{l^2 + \delta_{\text{max}}^2}$$  \hspace{1cm} \text{Equation 2-31}$$

which when combined with Equation 2-29 results in the following equation relating the maximum tip deflection of an IPMC bending strip to the imposed electric field, the time $t$, the strip cross capacitance, the strip cross resistance, the strip volume and the strip modulus of elasticity $Y$:

$$\frac{2\delta_{\text{max}}}{l^2 + \delta_{\text{max}}^2} = E \left( \frac{2C_gRT}{n^*eA}\left(\frac{l}{w}\right)Y\right) \left[1 - \exp\left(\frac{-t}{R_s C_g}\right)\right] + \left(\Pi_{\text{hw}} / YC^*\right)$$  \hspace{1cm} \text{Equation 2-32}$$

Figures 2-46, 2-47 and 2-48 depict the variations of maximum deflection versus the electric field $E$, the average cross capacitance $C_g$ and the average cross resistance $R_s$ for the same values of the parameters used in Figures 2-43, 2-44 and 2-45.
Figure 2-47: Variation of maximum tip deflection versus the average electric field, and time \( t \)

Figure 2-48: Variation of maximum tip deflection versus average cross resistance, and time \( t \)
2.2 Thin Film Batteries

A main component of the aircraft is the thin film battery or capacitor. Lightweight energy storage is critical for the operation of the SSA. Due to the movement of the wings the output power will vary significantly during each flap cycle. Therefore energy storage must be utilized to level off the variation and provide continuous power to the wings. Also by having energy storage available the power produced by the solar array while the SSA is gliding can be stored and utilized during the next flap cycle. Super-capacitors can provide the low capacity, high impulse power required to manage the loads associated with alternately gliding and flapping the wings. These devices have a very high peak-power capability and can withstand the millions of charge discharge cycles needed for a flapping wing.

Super-capacitors with energy densities of nearly 10 kW/kg are commercially available. A potential candidate for this type of super-capacitor is a newly developed thin film capacitor based on a dielectric polymer film, Ployvinylidenefluoride.

Another option is to utilize the emerging technology of thin film lithium-ion or lithium-polymer batteries. These types of batteries, an example of which is shown in Figure 2-49 [52, 38], are rechargeable, lightweight and flexible. They can be rapidly charged and discharged, have a long shelf life, operate over a wide temperature range and can be utilized over 10,000s of charge / discharge cycles with little loss in capacity. The thin film battery is produced by depositing the various material layers, cathode, electrolyte, anode and current collector, onto a substrate. This thin film layer is usually less than 5 micrometers thick. Variations in the cathode material (magnesium oxides, cobalt oxides or yttrium oxides) are the main difference in different types of Lithium-Ion batteries. These types of batteries are presently not commercially available. However significant development is being performed at various companies and research laboratories to bring this type of battery to the commercial market.

![Figure 2-49: Examples of thin film batteries](image)

Potential commercial markets for these types of batteries are significant and can include; implantable medical devices such as hearing aids or pacemakers, integration into component structures, combining with a solar array to provide a flexible power source and in microwireelectronics such as smart cards and computer chips [52].
The typical structure of a thin film battery, shown by the illustration in figure 2, is similar to that of a conventional battery and consists of the cathode, anode, current collectors and an ionically conducting electrolyte. The layers are deposited through various sputtering techniques. A brief description of the sputtering techniques for the Excellatron thin film lithium battery are given below.

“The cathode and anode current collectors (approximately 0.3m thick) are deposited by sputtering of the appropriate metal in argon (Ar) atmosphere. Cathode films of LiCoO₂ or LiMn₂O₄ are deposited by RF magnetron sputtering of sintered targets of the respective compounds in Ar + O₂ while films of V₂O₅ are deposited by reactive sputtering of V in Ar + O₂. A sputtered LiPON electrolyte film covers the cathode and a portion of the substrate up to the anode current collector in order to insulate the substrate from direct contact with the anode. For a thin film lithium battery, a thin layer of lithium metal is thermally evaporated on LiPON as the anode. For a thin film lithium ion battery, a thin layer of Sn₃N₄ (deposited by sputtering of Sn target in N₂ environment) is used as the anode. Finally, the battery is sealed.” [38]

Utilizing these sputtering techniques the potential exists for replacing the battery substrate with the IPMC material and depositing the battery layers directly onto the IPMC material. The deposition onto a polymer foil has been demonstrated. The main issue with depositing the materials onto a substrate is that the deposition processes are usually performed at high temperatures. This becomes an issue when utilizing materials such as polymers that cannot survive the high temperatures needed for the deposition. Recently there have been processes developed that utilize lower temperature deposition, on the order of 350°C, that is compatible with some polymers enabling them to be utilized as a substrate for the battery. Once the battery material is applied to the substrate it must be sealed. This is presently accomplished by utilizing protective coatings [38].

In addition to being flexible and having the potential to be integrated directly onto the IPMC layer, solid state, Lithium thin film batteries have additional characteristics that meet the needs of the SSA.

- This type of battery is in a solid state that is there are no fluids internal to its structure. Therefore there are no issues associated with leakage or sealing the battery.
Chapter 2.0 SSA Configuration and Operation

2.2 Thin Film Batteries

• The long charge / discharge cycle life makes it ideally suited to the flapping motion of the SSA wings. This cycle capacity is illustrated in Figure 2-51 [38] for an Excellatron thin film lithium battery.

• The operational temperature range is wide. The solid-state materials the batteries are composed of enable them to survive and operate at much higher temperature than most conventional batteries. The operating temperature range is from approximately –45°C to 100°C and the safe storage temperature range is from –55°C to 300°C. [52, 38].

• The battery output voltage is fairly flat and stable over the discharge period. This trend is shown in figure 4 for various discharge currents.

• The battery is capable of high current rate discharge and charge. This is a critical feature since it enables the batteries to be discharged quickly which is required during the wing flapping motion. A continuous discharge rate of 60 °C has been demonstrated and much higher discharge rates are possible for pulse operation [38].

![Image](image_url)  
*Figure 2-51: Excellatron lithium battery capacity as a function of charge/discharge cycles*
Thin film lithium batteries are presently not commercially available but their potential and characteristics match well with the requirements of the SSA. A comparison of the performance specifications of various types of rechargeable batteries is shown in Table 2-2. From this comparison it can be seen that thin film lithium batteries provide the critical capabilities of high cycle life and high specific energy and power needed by the SSA. Also these types of batteries are flexible and solid-state in design enabling them to be truly integrated into the SSA architecture.

Table 2-2: Rechargeable battery characteristics

<table>
<thead>
<tr>
<th>Battery Type</th>
<th>Specific Energy (Wh/kg)</th>
<th>Energy Density</th>
<th>Specific Power (W/kg)</th>
<th>Cycle Life</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nickel Cadmium</td>
<td>40</td>
<td>100</td>
<td>400</td>
<td>400</td>
</tr>
<tr>
<td>Nickel Metal Hydride</td>
<td>90</td>
<td>245</td>
<td>180</td>
<td>600</td>
</tr>
<tr>
<td>Lithium Ion (liquid electrolyte)</td>
<td>155</td>
<td>410</td>
<td>300</td>
<td>500</td>
</tr>
<tr>
<td>Lithium Polymer</td>
<td>180</td>
<td>380</td>
<td>360</td>
<td>500</td>
</tr>
</tbody>
</table>
2.3 Thin Film Array

2.3.1 Introduction
The optimum solar cell technology for the SSA will have a high power-to-weight ratio (specific power), high efficiency and easily integrated into a SSA wing structure. One potential structure for the SSA wing section will be layered and consist of a solar array, battery, control electronics and IPMC layers. Ideally, each of these layers will be manufactured in a roll-to-roll process where one layer is deposited upon the next. At this stage in SSA development, it is most important to develop the efficacy of each layer (or sub-system) with an eye towards eventually addressing how these layers will be integrated. In this section we will present a technology roadmap for achieving the best solar cell technology for SSA.

2.3.2 How a Solar Cell Works
A solar cell is effectively a large area p/n junction diode. At the p/n junction, an electric field is formed. Light entering the solar cell, is absorbed and creates electron-hole (e-h) pairs. These e-h pairs are then separated by the electric field and collected by contacts on the top and bottom. This process is shown in Figure 2-53.
The bandgap of the semiconductor material determines what portion of the sunlight is absorbed and ultimately what maximum efficiency can be achieved. Conventional solar cells made from silicon, have a theoretical efficiency of approximately 23% under Air Mass zero (AM0) conditions. In practice, silicon solar cells have reached 20% efficiency. Efficiency can be improved by splitting the solar spectrum into several spectral bands, and “tuning” the bandgap of several solar cells to match each of the spectral bands as illustrated in Figure 2-54. This spectrum splitting is accomplished by arranging the cells with the highest band gap cell on top, followed by cells with decreasing bandgap. Each cell absorbs light above its bandgap energy and transmits the remaining light to the cells below. Today, the highest efficiency system is a multi-junction (MJ) solar cell made from indium gallium phosphide (GaInP), gallium arsenide (GaAs) and germanium (Ge) junctions. The efficiency of these cells is currently about 28% in production volume and may reach 30%-35% in the near future.
The current technology of high efficiency solar cells is not generally applicable to the SSA. The high efficiency solar cells described above are fabricated on ridge substrates. Flexible thin film solar cells are a relatively new technology. The efficiency of these cells range between around 5%-12% in commercial production.

### 2.3.3 State-of-the-art Thin Film Solar Cells

There is a considerable effort in the space PV community to develop thin film PV technology. The requirements for the SSA are not significantly different than those of space satellites. The SSA can easily take advantage of the current development efforts for space thin film PV. The most mature thin film technology is triple-junction amorphous silicon (a:Si) cells produced on 0.0005” stainless steel substrates with efficiency of approximately 12%. Cells of a similar design have also been produced on 0.001” thick Kapton® films and have demonstrated specific power of greater than 1000W/Kg and efficiencies of 8-10% at AM0. Figure 2-55 shows a laboratory prototype flexible thin film solar cell on 0.001” Kapton® made by UniSolar Corp. The efficiency of a:Si on Kapton will likely reach that of a:Si on thicker metallic substrates which are approximately 12% at AM0. Other solar cell materials such as the II-VI materials, CuInGaSe
and CdTe have promise in the long term, for high efficiency thin film solar cells, but in spite of decades of research, these technologies still lag aSi performance at the production level.

![Flexible amorphous silicon solar cell on 1-mil Kapton substrate.](Photo courtesy of Uni-Solar Corp.)

**2.3.4 Near Term Advancements for Thin film Solar Cell Technology**

High efficiency photovoltaic devices (~ 30%) are fabricated exclusively from crystalline III-V semiconductor materials. These materials, such as GaAs and InGaP possess nearly optimum electrical and optical properties. Unfortunately, the growth of these high efficiency devices has been restricted to lattice matched, crystalline substrates such as GaAs and germanium. Germanium has been the substrate material of choice for commercial multi-junction III-V devices because it offers the opportunity to form a bottom photovoltaic junction from the germanium as well as being mechanically more robust than GaAs, thereby allowing the use of a thinner substrate, resulting in a lower mass solar cell. Unfortunately, crystalline Ge offers no capability to integrate the finished photovoltaic device into a flexible module or a pathway to achieve the high mass specific powers required for the SSA.

There is currently work underway at the NASA Glenn Research Center and the Ohio State University, to fabricate poly-crystalline III-V MJ solar cells on flexible substrates. The development of these devices on metal foil substrates initially and eventually on hybrid (metal/polymer) or all polymer substrates will greatly advance the state of the art. The ability to accomplish this integration is based upon two separate laboratory demonstrations. First, >20% AM1.5 polycrystalline GaAs cells were demonstrated several years ago [89]. These devices were produced on polycrystalline Ge substrates, believed to offer a path to low cost, high efficiency solar cells. The second development involves the recently demonstrated ability to re-crystallize thin amorphous Ge films deposited on ceramic plates and metal foils [72]. The grain sizes achieved in the re-crystallized Ge films was ~ 1mm², and is perfectly suitable for the integration of high efficiency III-V devices. By combining these demonstrated technologies and extending them to include MJ III-V devices, a pathway to high efficiency (>20%) flexible thin film solar arrays is readily apparent.
2.3.5 Long-term Advancements for Thin Film Solar Cell Technology

Basic research being conducted in solar cells, so-called “third generation” photovoltaics, is trying to exploit the unique properties of nano-sized particles and layers. Particles only a few nanometers in size exhibit quantum confinement effects and introduce the possibility of solar cell efficiencies as high as 71% [11]. In principal, a solar cell containing an array of quantum wells (2-dimensional quantum confinement structures) or quantum dots (3-dimensional quantum confinement structures), can be engineered so that its performance is effectively like a solar cell with a nearly continuous array of graded bandgap junctions. Much like the triple-junction cell has much greater efficiency over its single junction counterpart, the quantum-structured solar cell has the potential of raising solar cells efficiency close to the thermodynamic limit. These new structures are many years away from having a practical device, and likely years further before such devices are fabricated on thin film substrates.
Chapter 3.0 Operational Environments

3.1 Environmental Conditions for Flight on Venus

The environment in which the aircraft will operate has a large influence on its performance. This influence, which is greater than that for most conventional aircraft, is due to mainly to the fact that the aircraft receives all of its operating power from the sun. Therefore environmental conditions that affect the power available from the sun have a significant impact on the aircraft performance. The aircraft can potentially operate at any location that has sufficient solar intensity and atmospheric density. Therefore the aircraft can potentially operate on Venus, Earth and Mars. The environmental conditions on each of these planets are drastically different and must be characterized in order to determine what effect they would have on the aircraft's operations. The characteristics of the environments that are of interest include the physical characteristics of the planet, the atmospheric composition and conditions at various altitudes and the solar power available at different altitudes, latitudes and times of the year.

3.1 Environmental Conditions for Flight on Venus

Venus, shown in Figure 3-1, is the third planet from the sun and has a number of unique characteristics that makes its environment both interesting and challenging for flight. The basic physical and orbital properties of Venus are given in Table 3-1 [118]. Venus is very similar in size to Earth. However this is where the similarities end. The environmental conditions on Venus are very unique and unlike those on any other planet or moon. The planet has a very thick atmosphere with cloud cover over the entire planet. However, strange is this may seem Venus may be an ideal place to fly the solar powered solid-state aircraft. That is because the cloud cover only
Solid State Aircraft
extends from approximately 45 km above the surface to approximately 64 km above the surface. At the top of the cloud layer the atmospheric pressure is around 0.1 bar. Within this altitude range the atmospheric temperature form 80°C to -35°C. The structure of the Venus atmosphere is shown in Figure 3-2 and the temperature profile on Venus is shown Figure 3-3 [143]. Operation within this temperature range of the upper atmosphere will not be an issue. In fact operation at the colder end of the range will increase the solar array performance. The top of the cloud layer corresponds to a pressure altitude on Earth of 16 km (52,500 ft). Although high, this altitude is well within the range of modern aircraft and flight aerodynamics within this regime are well understood.[64] The properties of the Venus atmosphere from the surface to 100 km are listed in Appendix C.

**Table 3-1: Physical and orbital properties of Venus**

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum Inclination of Equator to Orbit (δ&lt;sub&gt;max&lt;/sub&gt;)</td>
<td>3.39°</td>
</tr>
<tr>
<td>Orbital Eccentricity (e)</td>
<td>0.0067</td>
</tr>
<tr>
<td>Mean Radius of Orbit (r&lt;sub&gt;m&lt;/sub&gt;)</td>
<td>108E6 km</td>
</tr>
<tr>
<td>Day Period</td>
<td>243 (Earth Days)</td>
</tr>
<tr>
<td>Solar Radiation Intensity</td>
<td>Mean: 2613.9 W/m²</td>
</tr>
<tr>
<td></td>
<td>Parihelion: 2649 W/m²</td>
</tr>
<tr>
<td></td>
<td>Aphehelion: 2579 W/m²</td>
</tr>
<tr>
<td>Albedo</td>
<td>0.65</td>
</tr>
<tr>
<td>Gravitational Constant (g)</td>
<td>8.87 m/s²</td>
</tr>
<tr>
<td>Sidereal Year</td>
<td>224 (Earth Days)</td>
</tr>
<tr>
<td>Surface Temperature</td>
<td>737 °K</td>
</tr>
<tr>
<td>Diameter</td>
<td>12,104 km</td>
</tr>
</tbody>
</table>

Because of the thick atmosphere the pressure and density throughout most of the atmosphere is much greater than that on Earth. The atmospheric pressure and density we experience near the surface of Earth occurs at just over an altitude of 50 km on Venus. For a flight vehicle this means that flying at 50 km on Venus is similar aerodynamically to flying near the surface on Earth. The atmospheric density (ρ) within the Venus atmosphere can be represented by Equation 3-1 as a function of altitude (h) in kilometers. This equation is plotted in Figure 3-4. The temperature profile within the Venus atmosphere is given by Equation 3-2.

\[ \rho = 64.85 - 3.3257h + 0.067373h^2 - 0.00066981h^3 + 3.224E - 6h^4 - 5.6694E - 9h^5 - 1.8971E - 12h^6 \]  \hspace{1cm} \textit{Equation 3-1}

\[ T = 738.26 - 9.1909h + 0.17429h^2 - 0.007965h^3 + 0.001518h^4 - 1.2336E - 6h^5 + 3.7325E - 9h^6 \]  \hspace{1cm} \textit{Equation 3-2}
The viscosity of Venus as a function of altitude is given by Equation 3-3. This equation represents a curve fit of the viscosity (µ) data given in Appendix A.

\[
\mu = 3.5827E - 5 - 5.906E - 6h + 8.8642E - 7h^2 - 5.9485E - 8h^3 \\
+ 2.1929E - 9h^4 - 4.8611E - 11h^5 + 6.6513E - 13h^6 - 5.5001E \\
- 15h^7 + 2.5199E - 17h^8 - 4.9099E - 20h^9
\]

\textit{Equation 3-3}

*Figure 3-2: Temperature profile of Venus's atmosphere*
Figure 3-3: Temperature profile within Venus’s atmosphere
Chapter 3.0 Operational Environments
3.1 Environmental Conditions for Flight on Venus

Above this cloud layer there is an abundant amount of solar energy. The solar flux at the orbit of Venus is 2600 W/m² which is much greater than the 1360 W/m² available at Earth orbit. This nearly 100% increase in solar flux can double the performance of the aircraft. Even within the cloud layer there may be sufficient solar energy to operate aircraft. At the bottom of the cloud layer (45 km altitude) the solar intensity is between 520 W/m² and 1300 W/m² depending on the wavelength of the radiation being collected. This is comparable to the solar intensity at Mars or Earth. Therefore, even within the cloud layer, the ability to fly under solar power on Venus will be no worse than it is to fly on Earth or Mars. And flying above the cloud layer will produce a much more capable aircraft than flying at similar pressure altitudes on Earth or Mars.[64]

The solar intensity as a function of altitude (h) in kilometers can be represented by the following equations. These equations represent a curve fit of attenuation [64] ($I/I_o$ which is the ratio of the intensity at the selected altitude, $I$, to the solar intensity above the atmosphere, $I_o$) for the mid-spectrum wavelength of 0.72 µm.

From 0 to 49 km altitude:

$$\frac{I}{I_o} = 0.10306 + 0.017383h - 7.99E - 4h^2 + 2.752E - 5h^3$$

$$-5.2011E - 7h^4 + 3.874E - 9h^5$$

Equation 3-4

Figure 3-4: Atmosphere density as a function of altitude
From 50 km to 65 km altitude:

\[ \frac{I}{I_o} = -1.3639 + 0.036023h \]  

\textit{Equation 3-5}

Above 65 km there is effectively no attenuation and therefore \( \tau \) has a value of 1. A graph of the solar attenuation as a function of altitude is shown in Figure 3-5.

\[ 0 \quad 0.1 \quad 0.2 \quad 0.3 \quad 0.4 \quad 0.5 \quad 0.6 \quad 0.7 \quad 0.8 \quad 0.9 \quad 1 \]

\[ 0 \quad 10 \quad 20 \quad 30 \quad 40 \quad 50 \quad 60 \quad 70 \quad 80 \quad 90 \quad 100 \]

\textit{Figure 3-5: Atmospheric solar attenuation as a function of altitude at 720 nm wavelength}

Another unique aspect of Venus is that the day length is longer than the year. Due to this slow rotational rate the speed to remain over a point on the surface is very low, approximately 13.4 km/hr. Therefore it is conceivable that a solar powered aircraft could remain within the sunlit portion of the planet indefinitely.

Overcoming the wind will be the key to maintaining the air vehicle's position within the sunlit portion of the planet. The winds within the atmosphere blow fairly consistently in the same direction as the planetary rotation (East to West) over all latitudes and altitudes up to 100 km. Above 100 km the winds shift to blow from the day side of the planet to the night side. The wind speeds decrease as a function of altitude from \( \sim 100 \) m/s at the cloud tops (60 km) to \( \sim 0.5 \) m/s at the surface. These high wind speeds and the slow rotation of the planet produce a super rotation of the atmosphere (nearly 60 times faster than the surface). A curve fit of mean wind speed (\( V \)) in meters per second versus altitude (\( h \)) in kilometers was produced. This curve is given in Equations 3-6 through 3-8 for the specified altitude ranges. A graph of the mean wind speed is also shown in Figure 3-6.
Chapter 3.0 Operational Environments

3.1 Environmental Conditions for Flight on Venus

From the surface to 57 km altitude:
\[ V = 0.89941 - 0.11201h - 0.017082h^2 + 0.0040604h^3 + 0.0010345h^4 - 9.96E - 5h^5 + 3.28e - 6h^6 - 4.7E - 8h^7 + 2.495E - 10h^8 \]  \[ \text{Equation 3-6} \]

From 58 km to 65 km altitude:
\[ V = 21498 - 1087.9h + 18.31h^2 - 0.10214h^3 \]  \[ \text{Equation 3-7} \]

From 66 km to 100 km altitude:
\[ V = -3860.1 + 637.42h - 32.206h^2 + 0.76199h^3 - 0.009357h^4 + 5.783E - 5h^5 - 1.42E - 7h^6 \]  \[ \text{Equation 3-8} \]

The gravitational acceleration on Venus (8.87 m/s\(^2\)) is slightly less than that on Earth which aids somewhat in the lifting capability of the aircraft. The atmospheric composition on Venus can also pose problems for the aircraft. The atmosphere is composed mostly of CO\(_2\) but also has trace amounts of corrosive compounds such as hydrochloric, hydrofluoric and sulfuric acids. [118] The atmospheric composition is given in Table 3-2 [83]. Because of this composition the speed of sound within the atmosphere is generally less than it is within Earth’s atmosphere. The speed of sound (a) in meters per second as a function of altitude (h) above the surface in kilometers can be represented by Equation 3-9 and is shown in Figure 3-7 as a function of altitude.

\[ a = 410.15 - 2.1102h + 0.008751h^2 - 0.00072086h^3 + 1.0136E - 5h^4 - 3.6825E - 8h^5 \]  \[ \text{Equation 3-9} \]

Figure 3-6: Average wind speed versus altitude within the Venus atmosphere
Figure 3-7: Speed of sound as a function of altitude within the Venus atmosphere

Table 3-2: Venus atmospheric composition

<table>
<thead>
<tr>
<th>Gas</th>
<th>Percent Volume</th>
</tr>
</thead>
<tbody>
<tr>
<td>Carbon Dioxide (CO₂)</td>
<td>96.5</td>
</tr>
<tr>
<td>Nitrogen (N₂)</td>
<td>3.5</td>
</tr>
<tr>
<td>Sulfur Dioxide (SO₂)</td>
<td>150 ppm</td>
</tr>
<tr>
<td>Carbon Monoxide (CO)</td>
<td>17 ppm</td>
</tr>
<tr>
<td>Water Vapor (H₂O)</td>
<td>20 ppm</td>
</tr>
<tr>
<td>Neon (Ne)</td>
<td>7 ppm</td>
</tr>
<tr>
<td>Argon (Ar)</td>
<td>70 ppm</td>
</tr>
<tr>
<td>Helium (He)</td>
<td>17 ppm</td>
</tr>
</tbody>
</table>
3.2 Environmental Conditions for Flight on Earth

The atmosphere of our planet is a very dynamic environment (Figure 3-8) with great fluctuations in temperature, density, pressure, wind speeds and solar intensity. This environment in which the solid state aircraft will operate has a large influence on its performance, design and capabilities. The main physical properties of the Earth's environment are given in Table 3-3 and the structure of the atmosphere is shown in Figure 3-9. The influence of the environment is greater for a solar powered aircraft than it is for most conventional aircraft flying under the same conditions. This is due mainly to the fact that the aircraft receives all of its operating power from the sun. Because of this characteristic the SSA design is very sensitive to the available incident solar radiation. This in turn will determine where and when the aircraft can fly. To enable controlled flight and not be moved off station, the SSA must be capable of generating sufficient power to overcome the wind speed at its flight location. Therefore understanding the wind environment is also key in determining the feasibility of the SSA and its potential range of operations. The SSA can potentially operate at any location that has sufficient solar intensity to generate the required power to keep the aircraft in flight. Daily solar intensity profiles will vary only with the time of year and latitude whereas the statistical mean and 99th percentile wind speeds will vary with the time of year, latitude, longitude and altitude.

In addition to these two main environmental factors, there are other environmental issues that can come into play and influence the aircraft design. These factors include UV radiation, temperature, electrical discharges and turbulence within the atmosphere.

![Figure 3-8: Earth from space](image-url)
The average solar intensity, given in Table 3-3 [1], is that which occurs above Earth’s atmosphere. The atmosphere attenuates the incoming solar radiation so the actual usable solar flux will be less than these orbital values. Determining the actual flux on the SSA will depend on its location (latitude), time of year and altitude. Also this flux will vary throughout the day because at lower elevation angles the sunlight must travel through a greater amount of atmosphere, thereby reducing the flux incident on the SSA.

The attenuation due to the atmosphere ($I/I_o$) is given by Equation 3-10 [116]. The attenuation depends on several factors that will vary depending on location time of year and altitude. These factors include the Linke atmospheric turbidity factor ($T_l$), the relative optical air mass ($R_{am}$) and the Rayleigh optical thickness ($R_{ot}$).

$$\frac{I}{I_o} = e^{-0.8662 T_l R_{am} R_{ot}} \quad \text{Equation 3-10}$$

The Linke atmospheric turbidity factor varies depending on the location and time of year. Using monthly turbidity factor data for the east and west coast of the United States [115], curve fits were produced that provide the turbidity factor as a function of latitude ($\phi$) in degrees for these two regions. Using the coefficients listed in Tables 3-4 and 3-5 for the east and west coast respectively, the turbidity factor can be calculated with Equation 3-11. The coefficients listed in Table 3-4 are valid for a latitude range of 28° N to 44° N along the east coast and those listed in Table 3-5 are valid for a latitude range of 35° N to 48° N along the west coast.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inclination of Equator to Orbit</td>
<td>23.45°</td>
</tr>
<tr>
<td>Orbital Eccentricity</td>
<td>0.01673</td>
</tr>
<tr>
<td>Day Period</td>
<td>23h 57.8 m</td>
</tr>
<tr>
<td>Mean Orbital Radius</td>
<td>149.6E6 km</td>
</tr>
<tr>
<td>Solar Radiation Intensity</td>
<td>Mean: 1352 W/m² Perihelion: 1399 W/m² Aphelion: 1307 W/m²</td>
</tr>
<tr>
<td>Albedo</td>
<td>0.37</td>
</tr>
<tr>
<td>Gravitational Constant</td>
<td>9.81 m/s²</td>
</tr>
<tr>
<td>Sidereal Year</td>
<td>365.26 (Earth Days)</td>
</tr>
<tr>
<td>Surface Temperature Extremes</td>
<td>130°K to 300°K</td>
</tr>
<tr>
<td>Earth Diameter/Radius ($r_e$)</td>
<td>12,756 km/6,378 km</td>
</tr>
</tbody>
</table>
Chapter 3.0 Operational Environments
3.2 Environmental Conditions for Flight on Earth

\[ T_i = M_0 + M_1 \phi + M_2 \phi^2 + M_3 \phi^3 \]  \hspace{1cm} \text{Equation 3-11}

<table>
<thead>
<tr>
<th>Month</th>
<th>( M_0 )</th>
<th>( M_1 )</th>
<th>( M_2 )</th>
<th>( M_3 )</th>
</tr>
</thead>
<tbody>
<tr>
<td>January</td>
<td>32.538</td>
<td>-2.2064</td>
<td>0.052895</td>
<td>-0.000421</td>
</tr>
<tr>
<td>February</td>
<td>-2.9015</td>
<td>0.75604</td>
<td>-0.028274</td>
<td>0.0003157</td>
</tr>
<tr>
<td>March</td>
<td>-38.364</td>
<td>3.963</td>
<td>-0.12216</td>
<td>0.00121</td>
</tr>
<tr>
<td>April</td>
<td>-40.955</td>
<td>3.9986</td>
<td>-0.117022</td>
<td>0.0011048</td>
</tr>
<tr>
<td>May</td>
<td>-5.3106</td>
<td>1.1074</td>
<td>-0.039367</td>
<td>0.0004209</td>
</tr>
<tr>
<td>June</td>
<td>38.886</td>
<td>-2.79922</td>
<td>0.075216</td>
<td>-0.000684</td>
</tr>
<tr>
<td>July</td>
<td>34.273</td>
<td>-2.2024</td>
<td>0.054789</td>
<td>-0.000473</td>
</tr>
<tr>
<td>August</td>
<td>77.508</td>
<td>-6.03772</td>
<td>0.16585</td>
<td>-0.001526</td>
</tr>
<tr>
<td>September</td>
<td>-15.864</td>
<td>2.1556</td>
<td>-0.071699</td>
<td>0.0007365</td>
</tr>
<tr>
<td>October</td>
<td>-46.402</td>
<td>4.8748</td>
<td>-0.1526</td>
<td>0.0015257</td>
</tr>
<tr>
<td>November</td>
<td>-10.076</td>
<td>1.6095</td>
<td>-0.0579</td>
<td>0.0006313</td>
</tr>
<tr>
<td>December</td>
<td>8.5379</td>
<td>-0.2264</td>
<td>-0.000541</td>
<td>0.0000526</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Month</th>
<th>( M_0 )</th>
<th>( M_1 )</th>
<th>( M_2 )</th>
<th>( M_3 )</th>
</tr>
</thead>
<tbody>
<tr>
<td>January</td>
<td>-206.18</td>
<td>15.188</td>
<td>-0.36495</td>
<td>0.0028988</td>
</tr>
<tr>
<td>February</td>
<td>-294.7</td>
<td>21.881</td>
<td>-0.53287</td>
<td>0.0042967</td>
</tr>
<tr>
<td>March</td>
<td>-13.962</td>
<td>1.3727</td>
<td>-0.035963</td>
<td>0.000307</td>
</tr>
<tr>
<td>April</td>
<td>1.3442</td>
<td>0.71808</td>
<td>-0.031336</td>
<td>0.0003602</td>
</tr>
<tr>
<td>May</td>
<td>-194.55</td>
<td>14.386</td>
<td>-0.34652</td>
<td>0.0027667</td>
</tr>
<tr>
<td>June</td>
<td>-164.12</td>
<td>12.3942</td>
<td>-0.30471</td>
<td>0.0024837</td>
</tr>
<tr>
<td>July</td>
<td>-148.06</td>
<td>10.914</td>
<td>-0.26126</td>
<td>0.0020754</td>
</tr>
<tr>
<td>August</td>
<td>-79.504</td>
<td>5.6651</td>
<td>-0.1293</td>
<td>0.0009846</td>
</tr>
<tr>
<td>September</td>
<td>-83.864</td>
<td>5.9748</td>
<td>-0.13658</td>
<td>0.0010377</td>
</tr>
<tr>
<td>October</td>
<td>3</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>
Optical air mass represents the comparison between the length of the path the sunlight takes through the atmosphere compared to the length of the vertical path (which is the shortest distance) through the atmosphere. Optical air mass is expressed by Equation 3-12 [54] and is dependent on the altitude (h) in meters and the adjusted solar elevation angle (\( \theta_a \)) in degrees which is given by Equation 3-13. The corrected elevation angle is derived from the solar elevation angle (\( \theta \)) in degrees.

\[
R_{um} = e^{-h/6434.5} \left( \frac{1}{\sin(\theta_a) + 0.50572(\theta_a + 6.07995)^{-1.6364}} \right)
\]

\[
\theta_a = \theta + \frac{0.061359(0.1594 + 1.123\theta + 0.065656\theta^2)}{1 + 28.9344\theta + 277.3971\theta^2}
\]

The last term needed to calculate the attenuation is the Rayleigh optical thickness. This factor is dependent on the optical air mass and is given by Equations 3-14 and 3-15.

For optical air mass values less than or equal to 20:

\[
R_{ot} = \frac{1}{6.6296 + 1.7513R_{um} - 0.1202R_{um}^2 + 0.0065R_{um}^3 - 0.0013R_{um}^4}
\]

For optical air mass values greater than 20:

\[
R_{ot} = \frac{1}{10.4 + 0.718R_{um}}
\]
Chapter 3.0 Operational Environments
3.2 Environmental Conditions for Flight on Earth

Earth’s atmosphere is broken down into layers, as shown in Figure 3-9 [83]. The layers of interest for the SSA are the troposphere (from the surface to approximately 14 km), the tropopause (from ~14 km to ~18 km) and the lower 1/3 portion of the stratosphere (~18 km to ~30 km). The troposphere is the region in the atmosphere where all the active weather occurs. Air rises and falls due to heating and ground effects causing winds and weather patterns. The majority of cloud formations and the atmosphere as we see it occurs in the troposphere, Figure 3-10. There is a gradual change from troposphere to the stratosphere that begins at approximately 14 km in altitude. This is called the Tropopause. The temperature in the lower stratosphere is extremely stable and cold at ~57 °C. Here, strong winds occur as part of defined circulation patterns and are mostly horizontal with little mixing. Above the tropopause is the stratosphere, high cirrus clouds sometimes form in the stratosphere, but for the most part there are no significant weather patterns in the stratosphere [140].

The Earth standard atmosphere table was used as the source of atmospheric properties for this analysis. It is based on idealized year round conditions at 45° N latitude. From this table the ideal gas constant for the Earth’s atmosphere is 8.31432 J/mol°K. The viscosity (µ) and thermal conductivity (k) of the atmosphere are based on temperature (T) and can be approximated by Equations 3-16 and 3-17 [48].
Viscosity in (kg/m s)

\[ \mu = \frac{(1.458E - 6)T^{1.5}}{T + 110.4} \]  

Equation 3-16

Conductivity in (kg/m s)

\[ k = \frac{(6.325E - 7)T^{1.5}}{T + 245.4 \times 10^9 \left(\frac{12}{T}\right)} \]  

Equation 3-17

The temperature and corresponding density profiles, within the atmosphere, vary with the different layers of the atmosphere that are shown in Figure 3-9. The atmospheric temperature (T) and density (ρ) for the regions within the atmosphere based on altitude are given by Equations 3-18 through 3-25. The geopotential altitude (hg) used in the calculation of temperature and density is given by Equation 3-25.

For altitude (h) up to 11 km:

\[ T = 288.15 - 0.0065h_g \]  

Equation 3-18

\[ \rho = 1.225 \left(\frac{288.15}{T}\right)^{-4.256} \]  

Equation 3-19

For altitudes (h) from 11 km up to 20 km:

\[ T = 216.65 \]  

Equation 3-20

\[ \rho = 0.364e^{\left(\frac{1000-h}{6341.62}\right)} \]  

Equation 3-21
Chapter 3.0 Operational Environments

3.2 Environmental Conditions for Flight on Earth

For altitudes (h) from 20 km up to 32 km:

\[ T = 216.65 + \frac{(h - 20000)}{1000} \]

\[ \rho = 0.088 \left( \frac{216.69}{T} \right)^{35.16} \]

For altitudes (h) from 32 km up to 47 km:

\[ T = 228.65 + 2.8 \frac{(h - 32000)}{1000} \]

\[ \rho = 0.013 \left( \frac{228.65}{T} \right)^{13.2014} \]

\[ h_g = \frac{r_e h}{(r_e + h)} \]

The atmospheric composition is given in Table 3-4 [48] and the standard atmospheric profile is given in Appendix C.

Table 3-6: Major gas components of Earth’s atmosphere

<table>
<thead>
<tr>
<th>Gas</th>
<th>Percent Volume</th>
</tr>
</thead>
<tbody>
<tr>
<td>N\textsubscript{2}</td>
<td>78.084</td>
</tr>
<tr>
<td>O\textsubscript{2}</td>
<td>20.947</td>
</tr>
<tr>
<td>Ar</td>
<td>0.934</td>
</tr>
<tr>
<td>CO\textsubscript{2}</td>
<td>0.0314</td>
</tr>
<tr>
<td>Ne</td>
<td>0.00181</td>
</tr>
<tr>
<td>He</td>
<td>0.000524</td>
</tr>
</tbody>
</table>

The wind speed on Earth is highly variable. It is dependent on the location, time of year and altitude. Mean wind profiles are shown for two locations (Albuquerque, New Mexico and Cape Kennedy, Florida) throughout the year in Figures 3-11 and 3-12 [122]. Although the absolute value of the wind speed will vary the trends with altitude are similar for most locations.
**Figure 3-11: Mean wind speeds for Albuquerque, New Mexico**

**Figure 3-12: Mean wind speeds for Cape Kennedy, Florida**
The wind speed at the flight altitude and latitude throughout the desired period of operation are critical factors in the aircraft’s design and capabilities. To accurately evaluate the feasibility of the SSA concept a detailed model of the wind environment is needed. The wind model is used to provide mean winds along an area of interest for various times of the year, latitude and altitude. To start the evaluation as well as bound the area being considered to a manageable region the mean wind environment along the East and West coast of the U.S. was considered.

The wind data for the derivation of the modeling equations was based on the data provided in Reference [122]. This data originated from the National Climatic Center and represented all available wind data at the specified locations from the surface up to 10 mb (~31km) in altitude. The data was presented seasonally and included mean, mean +5%, mean −5%, 95th percentile and 99th percentile winds. Examples for mean winds are shown in Figures 3-11 and 3-12. The locations along or near the coast where the wind data was available are listed in Table 3-7.

### Table 3-7: Raw Wind Data Sites

<table>
<thead>
<tr>
<th>Location</th>
<th>City</th>
<th>Latitude</th>
<th>Longitude</th>
<th>Data Collection Period</th>
<th>Total Profiles Sampled</th>
</tr>
</thead>
<tbody>
<tr>
<td>East Coast</td>
<td>Portland, Maine</td>
<td>44°N</td>
<td>71°W</td>
<td>January 1948 to September 1972</td>
<td>10,483</td>
</tr>
<tr>
<td></td>
<td>Washington D.C.</td>
<td>38°N</td>
<td>76°W</td>
<td>October 1960 to December 1971</td>
<td>13,439</td>
</tr>
<tr>
<td></td>
<td>Charleston, S. Carolina</td>
<td>32°N</td>
<td>80°W</td>
<td>January 1948 to December 1971</td>
<td>11,795</td>
</tr>
<tr>
<td></td>
<td>Cape Kennedy, Florida</td>
<td>28°N</td>
<td>81°W</td>
<td>February 1950 to May 1970</td>
<td>16,424</td>
</tr>
<tr>
<td>West Coast</td>
<td>Spokane, Washington</td>
<td>48°N</td>
<td>118°W</td>
<td>January 1950 to December 1970</td>
<td>11,144</td>
</tr>
<tr>
<td></td>
<td>Oakland, California</td>
<td>37°N</td>
<td>122°W</td>
<td>January 1950 to December 1970</td>
<td>10,771</td>
</tr>
<tr>
<td></td>
<td>Vandenberg, California</td>
<td>34°N</td>
<td>121°W</td>
<td>July 1958 to May 1970</td>
<td>11,933</td>
</tr>
</tbody>
</table>

The wind profile data from the sites listed in Table 3-5 was used to generate equations that would provide mean wind speeds for a specified altitude and latitude along the East and West coasts of the United States. The equations generated represent mathematical surfaces of wind speed versus altitude versus latitude. Since the wind data for each of these locations was separated into seasons, there were four wind speed equations generated for each coast. These equations were used in the subsequent analysis to provide the wind speed information and greatly simplified the analysis process. An example of the output of one of the wind surface equations is given in Figure 3-13. This figure was generated using the equation for mean wind speed along the East coast in the winter. The eight wind speed profiles, which encompass mean wind speed...
for both coasts and each season, are given in Equations 3-27 through 3-34. For these equations latitude (φ) is in degrees, Altitude (h) is given by atmospheric pressure (P, in millibars) and the wind velocity (V) is in m/s. The conversion of atmospheric pressure (P) in millibars to kilometers is given in Equation 3-35.

### 3.2.1 East Coast Wind Profiles

Mean wind velocity for spring:

\[
V = (-6266.7606 + 639.33071 \phi - 21.334456 \phi^2 + 0.17361051 \phi^3 + 3.59335E-3 \phi^4 - 5.341E-5 \phi^5 + 0.18089073P + 0.1.10336E-3P^2 - 5.966E-6P^3 + 8.329E-9P^4 - 3.644E-12P^5)^{0.5144}
\]

Mean wind velocity for summer:

\[
V = (-9156.146 + 909.857053 \phi - 28.318281 \phi^2 + 0.13604351 \phi^3 + 7.33643E-3 \phi^4 - 9.155E-5 \phi^5 - 0.3459049P + 3.2912E-3P^2 - 1.003E-5P^3 + 1.1961E-8P^4 - 4.893E-12P^5)^{0.5144}
\]

Mean wind velocity for autumn:

\[
V = (-8905.7603 + 865.059882 \phi - 25.524994 \phi^2 + 0.05131931 \phi^3 + 8.60907E-3 \phi^4 - 9.918E-5 \phi^5 - 0.0759347P + 2.39655E-3P^2 - 8.876E-6P^3 + 1.1346E-8P^4 - 4.794E-12P^5)^{0.5144}
\]

Mean wind velocity for winter:

\[
V = (-10056.977 + 985.214629 \phi - 30.129253 \phi^2 + 0.12888569 \phi^3 + 8.12247E-3 \phi^4 - 9.918E-5 \phi^5 + 0.05001086P + 2.31194E-3P^2 - 9.575E-6P^3 + 1.2515E-8P^4 - 5.313E-12P^5)^{0.5144}
\]

### 3.2.2 West Coast Wind Profiles

Mean wind velocity for spring:

\[
V = (504339.36 - 61362.878 \phi + 2976.05397 \phi^2 - 71.916641 \phi^3 + 0.8659332 \phi^4 - 4.1564E-3 \phi^5 - 0.1008834P + 2.47988E-3P^2 - 8.736E-6P^3 + 1.0845E-8P^4 - 4.493E-12P^5)^{0.5144}
\]

Mean wind velocity for summer:

\[
V = (611887.222 - 74493.194 \phi + 3614.89414 \phi^2 - 87.400052 \phi^3 + 1.05288402 \phi^4 - 5.0561E-3 \phi^5 - 0.2639922P + 3.1923E-3P^2 - 1.03E-5P^3 + 1.2493E-8P^4 - 5.137E-12P^5)^{0.5144}
\]

Mean wind velocity for autumn:

\[
V = (437807.731 - 53265.846 \phi + 2583.24068 \phi^2 - 62.42262 \phi^3 + 0.75161779 \phi^4 - 3.6078E-3 \phi^5 - 0.0208545P + 1.94285E-3P^2 - 7.45E-6P^3 + 9.5675E-9P^4 - 4.044E-12P^5)^{0.5144}
\]

Mean wind velocity for winter:

\[
V = (571292.993 - 69505.987 \phi + 3370.79711 \phi^2 - 81.449701 \phi^3 + 0.98064803 \phi^4 - 4.7067E-3 \phi^5 - 0.1381809P + 2.82793E-3P^2 - 9.826E-6P^3 + 1.2162E-8P^4 - 5.039E-12P^5)^{0.5144}
\]
Chapter 3.0 Operational Environments
3.2 Environmental Conditions for Flight on Earth

Figure 3-13: Winter mean wind speed profile for the East Coast

The curve fits given in Equations 3-6 through 3-21 are very useful for providing wind speed information quickly and in a format that is easily integrated into an analysis tool. However, limitations to the equations based on the range of the data from which they were derived. The equations should not be used to predict wind speed outside of limits of the original data. The range limits for these equations are given in Table 3-8.

Table 3-8: Input parameter ranges for wind equations

<table>
<thead>
<tr>
<th>Location</th>
<th>Latitude Range</th>
<th>Altitude Range</th>
</tr>
</thead>
<tbody>
<tr>
<td>East Coast</td>
<td>28°N to 44°N</td>
<td>Surface to 28 km</td>
</tr>
<tr>
<td>West Coast</td>
<td>35°N to 48°N</td>
<td>Surface to 28 km</td>
</tr>
</tbody>
</table>

Using the equations given above, example plots were generated to show the variation in wind speed throughout the year over the latitude range of interest along both the east and west coasts. The plots, shown in Figures 3-14 and 3-15, represent wind speeds at the operational altitude of 21.5 km.
Figure 3-14: Mean east coast wind speed for an altitude of 21.5 km

Figure 3-15: Mean west coast wind speed for an altitude of 21.5 km
3.3 Environmental Conditions for Flight on Mars

The Mars environment is very different from that here on Earth. Therefore there are issues and concerns associated with operating a vehicle in this environment that are not encountered on Earth. Mars has an atmosphere (Figure 3-17) but it is very thin. Near the surface on Mars the atmospheric density is similar to the density of Earth's atmosphere at 30 km. The atmosphere is made up almost entirely of carbon dioxide. The temperature on Mars is on average much colder than on Earth. Although at certain times of the year and locations the temperature will rise above freezing, most of the time temperatures are well below the freezing point of water. The gravitational force on Mars is about one third what it is on Earth. Therefore the lift that the solid state aircraft would need to generate to carry a given amount of mass is only one third what it would need to be on Earth. This reduced gravity (and hence lower lift requirement) is a large benefit for operating the aircraft. However the environmental conditions on Mars are not all beneficial. The low atmospheric density means that the SSA will be flying in a very low Reynolds number flight regime. There are a number of aerodynamics concerns with flight at very low Reynolds numbers. Most of these are based on flow separation that can affect the aircraft’s stability, control and lift generation.
Other environmental characteristics that are important to the system design include the surface temperature, atmospheric dust, solar intensity, soil and atmospheric composition and terrain characteristics (Figures 3-18 and 3-19). These factors influence just about every aspect of the solid-state aircraft's operation and potential mission capabilities. Some of these operational issues include what latitudes can the vehicle fly at and at what times during the year, can the vehicle land and be operational on the surface, the type and capabilities of the communications system, the type and approach for the navigation and control system, the construction materials used in the vehicle among others.
Chapter 3.0 Operational Environments
3.3 Environmental Conditions for Flight on Mars

As a basis for investigating the solid-state aircraft's applicability and capabilities, a concise summary of the Mars environmental conditions was assembled. Some of the more recent Mars science missions (particularly the Pathfinder mission) have provided detailed information on various aspects of the Mars environment. However, many aspects of the Mars environment are still not well understood. Therefore, the information provided in the following tables and figures represents the present state of knowledge of the Mars environment [118]. This information and its influence on the solid-state aircraft design may be subject to change as our understanding of the environment of Mars increases.

3.3.1 Physical Properties

Table 3-9: Physical properties of Mars

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inclination of Equator to Orbit</td>
<td>25.2°</td>
</tr>
<tr>
<td>Orbital Eccentricity</td>
<td>0.093</td>
</tr>
<tr>
<td>Day Period</td>
<td>24h 39 m</td>
</tr>
<tr>
<td>Mean Orbital Radius</td>
<td>227.9E6 km</td>
</tr>
</tbody>
</table>
| Solar Radiation Intensity          | Mean: 590 W/m²
|                                    | Parihelion: 718 W/m²
|                                    | Apehelion: 493 W/m²       |
| Albedo                             | 0.15                      |
| Gravitational Constant             | 3.73 m/s²                 |
| Sidereal Year                      | 687 days (Martian)        |
### 3.3.2 Atmospheric Conditions

*Mars atmospheric profiles are listed in Appendix C. This appendix consists of four atmospheric profiles generated by different sources and for different locations on Mars. The data available with each profile is not necessarily the same.*

The first profile is a reference atmosphere supplied by JPL.[53] This data was generated for a latitude of -20°. It provides data on temperature, pressure, viscosity and density from just above the surface to nearly 10 km. From this data curve fits were generated for density and viscosity as a function of altitude. The atmospheric temperature, density and viscosity as a function of altitude, up to 10 km, is given in Equations 3-36, 3-37 and 3-38 respectively.

\[
\rho = 0.014694 - 0.001145h + 4.6638E - 5h^2 - 9.7737E - 7h^3 \tag{Equation 3-36}
\]

\[
\rho = 0.014694 - 0.001145h + 4.6638E - 5h^2 - 9.7737E - 7h^3 \tag{Equation 3-37}
\]

\[
\mu = 1.2024E - 5 - 1.30002E - 6h + 1.0525E - 6h^2 - 3.6507E - 7h^3 + 6.0536E - 8h^4 - 4.8317E - 9h^5 + 1.4911E - 10h^6 \tag{Equation 3-38}
\]
The second profile is a general atmospheric model that is not specific to any location. This was generated to provide a rough estimate of the atmospheric conditions at any location on the planet. It provides density, temperature, pressure and speed of sound data for elevations of -5km (below the mean surface level) to 120 km above the surface [125].

The third profile was generated using Mars-GRAM atmospheric simulation tool. This profile was generated for a specific location on Mars, Parana Valles (-25°, 11°). It contains information on density, temperature, pressure speed of sound and viscosity for altitudes of 2.38 km to 20 km. [22]

The fourth and last profile was also generated using the Mars-GRAM atmospheric simulation tool. This profile was generated for a specific location on Mars Utopia Planitia (57, 235). It contains information on density, temperature, pressure speed of sound and viscosity for altitudes of -1.74 km to 20 km [22].

Significant data was also collected on the Mars atmosphere during the recent Pathfinder mission. For the first 30 days, surface pressure at the landing site underwent substantial daily variations of 0.2 to 0.3 mbar, which were associated primarily with the large thermal tides in the thin Mars atmosphere. Daily pressure cycles were characterized by a significant pressure change throughout the day period. This is shown in Figure 3-20 [98] and the pressure change over a 30 day period is shown in Figure 3-21 [98].

Because of the thinness of the Martian atmosphere the solar attenuation was assumed to be a constant with a value of 0.85 over the complete altitude range. This attenuation was due mainly to dust particles within the atmosphere and therefore is subject of significant variations throughout the Martian year.

The near surface temperature on Mars is greatly influenced by the surface temperature cycle (surface heating during the day and radiative cooling at night due to the low density of the Martian atmosphere). At sunrise, the atmosphere is typically stable and cool dense air lies near the surface. As the surface warms, the air mass is heated and by early morning begins to rise. As the heating continues the atmosphere becomes unstable. This causes temperature fluctuations on the order of 15 to 20K, which are observed during the remainder of the morning and early afternoon. Later in the afternoon the surface cools, the instability decreases, and the temperature fluctuations reduce
Figure 3-20: Daily pressure variation (Pathfinder data)

Figure 3-21: Pressure variation over a one-month period (Pathfinder data)
3.3 Environmental Conditions for Flight on Mars

3.3.3 Dust Storms and Wind

The wind at or near the surface can range from 2 to 7 m/s, (based on Viking lander data). These winds have a strong diurnal and seasonal variation in both direction and magnitude. Wind speeds of up to and possibly greater than 50 m/s will occur above the surface boundary layer, this surface boundary layer is estimated to extend 10s of meters above the surface. Preliminary estimates of the Pathfinder wind data suggest that wind speeds were comparable with or lower than those measured by Viking Lander-1 at the same time of year. Speeds were generally less than 5 to 10 m/s, except during the passage of dust devils, and were often less than 1 m/s in the morning hours. This may be consistent with the lower slope at the Pathfinder site. [98]

For a one month period, pathfinder data shows that wind direction generally rotated in a clock-wise manner through a full 360°. Winds were consistently from the South in the late and early morning and then rotated steadily through West, North and East during the day period. The wind

By evening, the thermal convection subsides and the instability in the atmosphere is diminished. The atmosphere becomes stable again due to surface cooling during the night time period. Any major nighttime temperature fluctuations are caused by downslope winds that disturb the surface boundary layer.

Figure 3-22: Atmosphere temperature variation throughout a day (Pathfinder data)
direction at nighttime was very consistent but became more variable throughout the day. The wind direction is shown in Figure 3-23.

![Wind direction graph](image)

*Figure 3-23: Wind direction throughout the day (Pathfinder data)*

Dust storms tend to occur when Mars is near perihelion in its orbit, when the solar intensity is the greatest. It is believed that the greater intensity of solar radiation coupled with variations in the topology of Mars triggers the dust storms. The storms can last up several months and the opacity of the storms can be quite high. Due to the low atmospheric density these dust storms result in only minimal distribution and accumulation of debris. More information on dust storms, gathered for the Mars micromission aircraft program is given in Reference [21].

Dust devils are short term variations in measured surface pressure, wind velocity and air temperature over periods of tens of seconds to minutes. This is shown in Figure 3-24. Dust devils,
about 2 km width and a few kilometers high, have been observed in the tropics by the Viking orbiters.

*Figure 3-24: Measurements taken during a dust devil (Pathfinder data)*
In order to establish the aerodynamic feasibility of the SSA, we present a computational study of the unsteady flow about a 2D airfoil in pitch-plunge motion that mimics animal flapping flight aerodynamics. Reynolds number range of 100-100,000, pitch amplitude of ±30 deg., and plunge amplitude of 0.5c are used in the simulations. Frequency tuning based on bluff body vortex shedding, proposed recently by the authors, will be emphasized. In addition to the above frequency tuning approach, wing kinematics include an end-of-stroke pitching scheme that differs from those proposed previously by other investigators. Unlike previous flat plate airfoil studies, this new scheme enables using cambered airfoils providing greater amount of lift. Our flow simulation results show that the proposed frequency tuning procedure and the cambered wing orientation scheme during downstroke and upstroke may improve wing performance.

A parallel effort to understand vortex dynamics and force generation in flapping and pitching motion has also been in progress during the duration of the project. Because of the limitations of the experimental facility, only hover mode has been studied, with the expectation of extending the work to include forward flight in future studies. The present experimental results include only those flat plate airfoils; cambered airfoil studies are under way. The experimental work sheds light on the 3D effects which are not available from the present 2D computational work. CFD viscous flow simulations with 3D wing kinematics is outside the scope of the present project. The feasibility of and resource requirements for realistic 3D simulations are discussed elsewhere in this report.

4.1.1 Previous Work on Flapping Flight Aerodynamics

Low Reynolds number, unsteady aerodynamics of thin cambered wings representative of insect wings are of current interest because the inherent fluid dynamics phenomena are not well understood. High lift associated with insect flight, not predicted by conventional quasi-steady aerodynamics, has been a fascinating subject for many biologists. While these efforts are noteworthy, gaining a thorough understanding of insect wing flow control mechanisms, quantifying them and incorporating their desirable features into wing design have become critical to biomimetic flight. It is impossible to incorporate the numerous flight mechanisms that insects and birds employ in a laboratory experiment or a numerical simulation. Thus simplified models for likely individual mechanisms can shed light on the various associated fluid dynamic phenomena. Several aerodynamic models have been proposed to explain lift generation during the cyclic motion of the insect wing. Forward flight, requiring both lift and thrust, has been more easily analyzed than hovering flight. Studies on tethered live animals have been conducted to measure forces and visualize flow patterns. Experimental studies using models of insect species such as the fruit fly have also been undertaken.

Insects generate thrust and lift by controlling wing kinematics that include flapping and pitching during each stroke cycle. Other mechanisms such as camber control, wing flexure, fore-wing-hind-wing coordination, and variation of stroke plane inclination have also been observed. Animal flight aerodynamics have been investigated largely by two groups, biologists and aerody-
dynamicists. Nachtigall [78] has discussed the various aspects of insect flight based on his work on live insects using high speed photography. Azuma [8] has compiled the work of various investigators and provided rudimentary analytical tools for adapting insect flight features into engineering design. There exists a large body of work on insect aerodynamics in the Journal of Experimental Biology; however, though they provide good qualitative descriptions, because of the difficulties associated with doing experiments on small live animals, the reported quantitative results are often incomplete, and have large experimental error bands.

Some insects execute the "clap-and-fling" mechanism (Weis-Fogh mechanism) during the wing beat cycle. Insects that execute this mechanism start the stroke cycle with the wing surfaces in contact with each other. The beat cycle starts with the wing leading edges moving apart as if the wings were hinged at the trailing edges. After reaching the maximum angular displacement, the two wings separate at the trailing edges, change the sense of rotation and linear motion, and come together with the leading edges coming in contact first. A motion, in reverse to the fling motion is executed with the leading edges forming the hinge. The wings then rotate about the leading edge hinge in a clapping motion to complete the cycle. Weis-Fogh [141] and Lighthill [70] modeled the corresponding aerodynamics in terms of bound vortices using inviscid theory. Lighthill also applied a correction factor for viscous effects. Even though their model satisfied the Kelvin-Helmholtz theorem, according to which the circulation should remain zero, Weis-Fogh pointed out that the effect is not the common Magnus effect since the lift produced during the fling is in the opposite sense to that would be created by the Magnus effect. Other unsteady mechanisms are suggested for the creation of circulation of the right sense. The Weis-Fogh-Lighthill theory generated great interest at the time; however, closer examination revealed its limitations, not including the effect of viscosity being the most obvious. At the Reynolds numbers of interest, inviscid theories are not likely to be reliable. Later, Maxworthy [73] performed experiments to explain the Weis-Fogh mechanism in terms of the interaction of the leading edge vortices from the wings as they fling apart.

Not all insects execute the clap-and-fling motion described above. Dickinson et al. [30] recently proposed the "delayed-stall-rotational-lift-wake-capture" mechanism to explain lift generation during all phases of the wing beat cycle. During the up- and down-strokes, a leading edge vortex (LEV) forms and stays attached to the surface creating lift ("delayed stall"). As the wing approaches the end of each stroke, it rotates in preparation for the following stroke; this rotation coupled with the translation relative to the surrounding air creates lift ("rotational lift"). Finally, "wake capture" denotes the augmentation of the relative velocity and the resultant lift due to the wing wake flow as the wing reverses the direction of its linear motion at the ends of the strokes. They investigated the effects of wing rotation on unsteady aerodynamic performance using a flat plate airfoil geometrically scaled in wing planform, and dynamically scaled to the Reynolds number of fruit flies (Drosophila). Vest and Katz [136] investigated wing motion in pitch and heave using unsteady, three-dimensional potential flow model. Comparisons to limited experimental data showed agreement within the error bounds of the experiments. To provide basic steady state aerodynamic characteristics, experimental investigation of low-Reynolds number, low-aspect ratio flat-plate wings have been recently conducted by Torres and Mueller [133]. Four Planform shapes were considered. Lift, drag and pitching moment characteristics were compared. In another similar study, Sunada, et al. [124] compared low-Reynolds number data from 20 wings of different air foil shapes. Effects of camber, leading edge profile and surface
corrugation are reported. Laitone [61] has reported that the lift coefficient at low Reynolds numbers can be increased by using sharp leading edges. Vortical signatures of heaving and pitching airfoils were investigated by Freymuth [43] who observed a reverse Karman vortex street that resulted in thrust production. Experiments were done using a plunging and pitching NACA 0015 airfoil in a wind tunnel. Results are given for Re = 5,200 and 12,000 and reduced frequency, k = 2.7 and 2.9, respectively. The authors noted that the vortex street was not entirely laminar. Spedding et al. [121] investigated the vortex wakes generated by birds in level flight and deduced the generated forces from the wake structure. A closed-loop model and constant circulation model were proposed as two limiting cases, with most of the observed results lying between these two extremes. Although the Reynolds number regime of bird flight is different from that of insects, there are many aspects common to both.

Several computational fluid dynamics (CFD) studies of flapping wing aerodynamics, especially that of the fruit fly, have recently been published. Most of these studies dealt with rigid body motion of the wing in pitch (about the y-axis, Figure 4-1) and flap (about the X-axis), mimicking the two important motions of an insect wing. To avoid grid generation complexities, flat-plate wings were used in these studies. Ramamurti and Sandberg [90] conducted numerical simulation of insect flight and compared their results with experiments of Dickinson et al. [30]. They used the Euler equations, while others utilized the laminar Navier-Stokes equations [35, 36, 123]. Almost all of them used structured grid, with some using Cartesian grid [36] to simplify dynamic gridding necessary to accommodate rigid body motion. All used a flat plate airfoil with the planform shape resembling the fruit fly wing.

Emblemsvag and Candler [35] used a fruit fly model wing about 3.7 times thicker than the one used in the experiments, which had a thickness to chord ratio of approximately 2.5%. The computations revealed that wing thickness plays an important role in the magnitude of the lift force and the force peaks at stroke reversal. However, computer limitations prevented them from scaling the thickness of a fruit fly wing. Sun and Tang [123] created the grid around the wing using the O-H topology. They used a 12% thick flat plate with an elliptic airfoil section due to limitations of the numerical scheme to handle sharper edges and thinner airfoils. Their numerical algorithm is based on the method of artificial compressibility which introduces a pseudo time derivative of pressure into the continuity equation. Subiterations in pseudo time are used during each physical time step to achieve numerical accuracy. The grid consisted of 720,000 cells. Down- and up-strokes were treated as symmetrical producing the same magnitudes of forces, a deviation from animal wings which produce larger forces during the downstroke. Large values of lift and drag which changed with the location of the pitch axis were observed. A maximum

![Figure 4-1: The coordinate system used to describe wing motion. The pitch axis, y, sweeps in the XY plane of the inertial XYZ coordinate system, while the wing pitches about the y-axis of the wing-fixed xyz coordinate system](image)
steady state $C_L = 0.6$ was observed from the authors’ work and the work of other investigators. Twice as much lift was observed for the beating wing. Both Sun and Tang [123] and Emblemsvag and Candler [120] reported lift coefficients significantly lower than from experiments [30], while Ramamurty and Sandberg’s [90] computational results from mostly inviscid simulations agreed very well with the experiments, and concluded that viscous effects are minimal for the fruit fly wing. Clearly the simulation results from various studies show a large spread, with the more accurate models that include the viscous effects giving larger deviation from experiments. For the low Reynolds number regime of animal flight, including viscous effects will, however, be critical to capture all the important phenomena.

Slightly different explanations have been given by Dickinson et al. [30] and Sun and Tang [123] for the force peaks observed at the ends for the strokes; the former explaining it as Magnus effect and the latter as the effect of acceleration. Since the fruit fly wing has no line of symmetry parallel to the span-wise axis (y-axis, Figure 4-1), explaining the force peaks at the ends of the stroke for such an asymmetric wing is difficult. A planform that has symmetry along the span-wise axis, on the other hand, would be better suited to study the effect of wing rotation about the pitch axis.

Numerical simulations of an ultra low Reynolds number (Re), high angle of attack ($\alpha$), thin, cambered, two-dimensional airfoil representative of insect wings, have been conducted by Isaac, et al. [51] for two values of $\alpha$ (~30° and ~45°). The time-accurate, unsteady simulation results are analyzed and the phase relations of the leading edge and trailing edge vortex (LEV/TEV) dynamics to cyclical lift variation are established. The results show the fixed wing undergoing a high-lift phase during each lift cycle. At moderately high $\alpha$, the LEV dominates, and at higher $\alpha$ the LEV and TEV are equally dominant. The Strouhal number for all the cases considered is close to that of the Karman vortex shedding. The goals of this ongoing study are aspects such as leading edge and trailing edge vortex (LEV/TEV) dynamics, span-wise flow features, and dynamic camber variation during the wing beat, all common to most insects. The LEV and TEV play a dominant role even in cases in which forced wing oscillations are not imposed. Leading edge and trailing edge vortices were observed to form alternatively and shed to form the vortex street, the strengths of the vortices dependent on the angle-of-attack. The sharp leading and trailing edges accentuate the vortices originating at these locations. Lentink and Garritsma [69] compiled reduced frequencies of insects and showed that they lie in a narrow band around 0.22, supporting our results.

Anderson et al. [5] investigated thrust producing mechanisms of harmonically oscillating foils by flow visualization at $Re = 1100$, and by force measurements at $Re = 40,000$. Both leading edge and trailing edge vortices have been observed, and their interaction has been identified as a factor in thrust generation and the formation of a reverse Karman vortex street (KVS). Koochesfehani’s [59] work on NACA 0012 airfoil also examined the vortical pattern in the wake, and thrust generation, when oscillations by pitching the airfoil about the quarter chord point were imposed. A wake vortex pattern opposite to that of the KVS was observed with the wake behaving as a “jet” and producing thrust instead of drag for frequencies above a threshold. The corresponding mean velocity profiles showed a momentum excess in the wake indicating thrust. Most of the studies of leading and trailing edge vortices (LEV/TEV) have been done on oscillating airfoils, and the interpretations of the results tend to tie these vortices to the forced oscillations.
While the oscillations might accentuate the vortices, results from the work of our group suggest that unsteady vortex shedding from the leading and trailing edges similar to the Karman vortex shedding are present in non-oscillating foils. This periodic vortex shedding leads to oscillatory behavior of the lift and drag forces in ultra low Reynolds number flows. The dynamics of the LEV/TEV play an important role in the periodic nature of the forces.

Visbal and Shang [137] performed numerical computations of a pitching NACA 0015 airfoil. The basic vortical structure of their studies agreed with experimental flow visualization for chord-based Reynolds number, $Re_c = 45,000$. A leading edge vortex, a shear layer vortex and a trailing edge vortex were identified. The authors correctly observed that the counter clockwise vortex originating in the lower surface boundary layer was shed into the wake, while the clockwise vortex remained on the upper surface, as also observed in the experiments of Walker, et al. [139]. Maxworthy [73] has proposed that the leading edge vortex (LEV) on three-dimensional wings might be stabilized by spanwise flow and cause the LEV to spiral out towards the wing tip. Other works that address the various aspects of low Reynolds number fluid dynamics and the importance of vortex dynamics can be found in References [12, 142, 60, 9, 146, 42, 33, 55, 46]. Following the observation of Isaac et al. [51] that matching the reduced frequency to the Strouhal number might be used to advantage to increase the force levels on the wing, Taylor et al. [130] in a recent publication drew similar conclusions from their flow visualization experiments. However, quantitative data such as forces have not been presented to support the claim.

The present work is a continuation of our earlier work [51, 86, 117] on unsteady numerical simulations of an ultra low Reynolds number, high angle-of-attack, thin, cambered, two-dimensional airfoil representative of animal wings. The time-accurate, unsteady simulation results were analyzed and the phase relations of the leading edge and trailing edge vortex (LEV/TEV) dynamics to cyclical lift variation were established. The results show the fixed wing undergoing a high-lift phase during each lift cycle. At moderately high $\alpha$, the LEV dominates, and at higher $\alpha$ the LEV and TEV are equally dominant. The Strouhal number for all the cases considered was close to that of the Karman vortex shedding, a significant departure from some of the previous high Re results. It is interesting to note that the lift curve from our work is very similar to that of Claupeau et al. [12], reproduced in Reference [73]. These observations led us to propose that, by matching the reduced beat frequency of flapping wing UAV that incorporate wing kinematics to the Strouhal number of Karman vortex shedding, the wing can avoid operating in the low lift phase, explaining, at least in part, the high lift associated with insect flight. Later analysis of flight data of a large number of animals showed that their reduced beat frequencies indeed lie close to the Strouhal number of Karman vortex shedding. Greenwalt’s [46] suggestion that insects operate as resonant oscillators might be relevant to this concept.

In this report we present results from additional simulations at Reynolds number of 100,000 representative of the proposed Solid State Aircraft. These results complement the results presented in our previous publications [51, 86, 117]. Preliminary experimental results are also provided to shed more light on the fluid dynamic and flow control mechanisms of animal flight.

4.1.2 Two-dimensional Airfoil Analysis

To begin the two dimensional (2D) airfoil analysis three Computational Fluid Dynamics (CFD) codes were selected to be utilized to generate data that could then be used in the feasibility and
sizing analysis. These codes included XFOIL [145], a panel method code for quickly analyzing airfoils, written by Mark Drela at MIT, WIND [144], a CFD code that solves Reynolds-Averaged Navier-Stokes equations and FLUENT [41], a commercially available CFD flow solver.

XFOIL was initially validated against wind tunnel data for the NACA 2414 airfoil at a Reynolds number of 100,300. The results of this validation are shown in Figures 4-2 through 4-4. Figure 4-2 is a lift coefficient versus drag coefficient curve of XFOIL results and the measured wind tunnel results. This figure shows very good agreement between the calculated performance and the test data for this operational Reynolds number.

![Figure 4-2: Lift vs. drag coefficient validation data for XFOIL](image)

The next validation data generated was lift coefficient and pitching moment coefficient (C_m) versus angle of attack (alpha). The comparison between XFOIL and the measured wind tunnel data is shown in Figures 4-3 and 4-4 respectively for this data. Again the generated and experimental data are in close agreement. The Cm data for the wind tunnel data is somewhat higher than the XFOIL predicted data at angles of attack of 0' and greater.
With XFOIL validated in the Reynolds number regime that is of interest for the SSA, aerodynamic performance on a number of representative airfoils was generated. This data included lift coefficient versus drag coefficient (shown in Figure 4-5), lift coefficient and pitching moment coefficient versus angle of attack (shown in Figures 4-6 and 4-7 respectively). The airfoils utilized in this initial performance analysis included a number of Eppler airfoils (e376, e378, e377, e379, e471, e63 and e71) the Althaus AH6407 and the Gilbert Morris GM15SM. Cross-sections of each of these airfoils is shown in Figure 4-8 [132].
The airfoils evaluated were all thin airfoils. For the SSA application the wing thickness will need to be thin to both reduce the aircraft mass as well as conform to the properties of the materials utilized to construct the aircraft. The data generated on the airfoil performance showed little variation between the airfoils in the lift versus drag coefficient curves or the lift coefficient versus angle of attack. There was more variation in the pitch moment coefficient versus angle of attack than with the other quantities.
Figure 4-7: XFOIL generated pitching moment coefficient vs. angle of attack for various airfoils
Additional work was performed to determine two-dimensional airfoil performance using the WIND fluid dynamics code. This analysis was performed to evaluate the capabilities of the WIND CFD program as well as provide additional performance data on an airfoil not examined XFOIL results presented above. These additional calculations were performed for the Selig 1091 airfoil [132], which is a generic low speed airfoil representative of something that could be used for the SSA application. These were completed at Mach number of 0.19 and Reynolds number of 100,000, as with the XFOIL results. The analysis was performed to ascertain the lift and drag at high altitudes and low Mach numbers. The Selig 1091 cases were run 2D, steady state, turbulent, representing the SSA in gliding or soaring flight (see Figures Figure 4-9, Figure 4-10 and Figure 4-13). The computational grid (Figure 4-12) was created with GRID-GEN [87], and contains 299 points chord-wise and 100 points normal to the airfoil. Points were
clustered close to the airfoil surface to capture the boundary layer effects. Cl and Cd values were tabulated for various angles of attacks up to 24 degrees. XFOIL was then used to verify the values from WIND for the Re=100,000 case, shown in Figures 4-11 and 4-14. It was determined that XFOIL can simulate viscous effects in the boundary layer but becomes unreliable at high angle of attacks because of separation. Additional Cl and Cd values were found for the Re=200,000, 300,000, 400,000 and 500,000 cases using only XFOIL for Mach number 0.19. XFOIL showed that just very slight differences between lift and drag for the different Reynolds numbers although experimental data shows larger differences. This discrepancy could be a result of coarse gridding and unsteady viscous effects that are neglected in XFOIL.

Figure 4-9: Mach number contours around the Selig 1091 airfoil at Reynolds number 100,000; Mach number 0.19; and 6 degrees angle of attack
Figure 4-10: Static pressure contours around the Selig 1091 airfoil at Reynolds number 100,000; Mach number 0.19; and 6 degrees angle of attack

Figure 4-11: Lift coefficient vs. angle of attack for WIND and XFOIL computations
Figure 4-12: 299 x 100 grid used for the CFD calculations
Figure 4-13: Streamlines around the Selig 1091 airfoil at Reynolds number 100,000; Mach number 0.19; and 18 degrees of angle of attack

Figure 4-14: Drag coefficient vs. angle of attack for WIND and XFOIL computations
In addition to the airfoil shapes shown in Figure 4-8, other airfoil geometries were also considered. A flat plate geometry with a curved leading edge was examined as a potential means of producing a very thin airfoil. This shape would produce greater aerodynamic performance over a basic flap plate. The results of the analysis on the curved tip flat plate airfoil are shown in Figures 4-15 through 4-23. Figures 4-15 though 4-18 are for an angle of attack of 0°, Figures 4-19 through 4-22 are for an angle of attack of 15° and Figure 4-23 for an angle of attack of 10°. All cases were run at a chord Reynolds number of 100,000. These results were generated with the Fluent CFD analysis code [41].

**Figure 4-15: 0° angle of attack static pressure distribution**

**Figure 4-16: 0° angle of attack velocity vectors**
Figure 4-17: 0° angle of attack modified turbulent viscosity

Figure 4-18: 0° angle of attack lift coefficient
Chapter 4.0 SSA Design and Feasibility Analysis

4.1 Aerodynamic Design and Analysis

Figure 4-19: $15^\circ$ angle of attack static pressure variation

Figure 4-20: $15^\circ$ angle of attack modified turbulence intensity ($v_t$)
Figure 4-21: 15° angle of attack velocity vectors

Figure 4-22: Lift coefficient at 15° angle of attack (steady state value is around 1.15)
The figures that show lift coefficient as a function of time (Figures 4-11, 4-15 and 4-16) show an initial transient in the calculation. This initial variation is due to the iterative process used in the calculation. On all of these curves the value settles out to a number representative of the lift coefficient calculated under those operating conditions. These values are shown in Table 4-1.

**Table 4-1: Curved tip flat plate lift coefficient for various angles of attack**

<table>
<thead>
<tr>
<th>Angle of Attack (degrees)</th>
<th>Lift Coefficient</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.0°</td>
<td>0.755</td>
</tr>
<tr>
<td>10.0°</td>
<td>1.213</td>
</tr>
<tr>
<td>15.0°</td>
<td>1.151</td>
</tr>
</tbody>
</table>

Of the airfoils examined the Eppler 378 was selected as the best choice for further more detailed analysis. The performance data generated for it was subsequently utilized in the aircraft sizing and feasibility analysis. It should be noted that because of the unusual airfoil shape and the low quality of leading-edge definition, convergence problems with XFOIL were routinely encountered while analyzing this airfoil.

A 2D grid was created with GRIDGEN around the E378 airfoil cross-section, shown in Figure 4-24. Emphasis in generating the grid was placed on the leading edge which contained the highest concentration of points. The grid contained 29,950 points. Results were generated.
with angles of attack of 4° and 12° using both WIND and XFOIL. WIND generated Mach number graphs for angles of attack of 4° and 12° are shown in Figures 4-25 and 4-26 respectively. A summary of the lift coefficient, drag coefficient and pitch moment coefficient results from both codes is shown in Table 4-2.

Figure 4-24: Eppler 378 airfoil CFD calculation grid
Figure 4-25: Mach number contours for E378 airfoil at 4° angle of attack

Figure 4-26: Mach number contours for E378 airfoil at 12° angle of attack
An additional validation of the results produced for the Eppler 378 airfoil was performed. This validation used the same grid techniques and CFD analysis (used with the E378 airfoil). The validation was performed with the Selig 1223 airfoil, shown in Figure 4-27. The operational point utilized for the airfoil was a 6° angle of attack at a Reynolds number of 100600. This operational point was selected in order to compare directly with experimental data [132].

<table>
<thead>
<tr>
<th>Re=2.1e5</th>
<th>CL</th>
<th>CD</th>
<th>CM</th>
</tr>
</thead>
<tbody>
<tr>
<td>XFOIL 4 degrees</td>
<td>1.1061</td>
<td>0.01417</td>
<td>-0.1015</td>
</tr>
<tr>
<td>WIND 4 degrees</td>
<td>1.0122</td>
<td>0.0227</td>
<td>-0.1129</td>
</tr>
<tr>
<td>XFOIL 12 degrees</td>
<td>1.53</td>
<td>0.07316</td>
<td>-0.0843</td>
</tr>
<tr>
<td>WIND 12 degrees</td>
<td>1.725</td>
<td>0.0886</td>
<td>-0.111486</td>
</tr>
</tbody>
</table>

A grid similar to the one utilized with the E378 airfoil was generated. This grid also consisted of 29,950 points with the emphasis on the leading edge. This grid is shown in Figure 4-28.
The Mach number contours for this airfoil are shown in Figure 4-29 and the resulting performance data is given in Table 4-3. The data given in Table 4-3 shows good comparison between the predicted lift coefficient from both XFOIL and WIND and the wind tunnel data. XFOIL over estimated the lift coefficient by approximately 10% and WIND over estimated it by approximately 6.5%. For the Drag coefficient data, XFOIL produced significantly more accurate results. XFOIL under estimated the drag coefficient by 24% and WIND over estimated the drag coefficient by 146%. There was significant variation from the experimental data in the calculation of pitching moment. XFOIL over estimated the pitching moment by 671% and WIND over estimated it by 863%. Although there was some variation in the results generated from the experimental data, the critical lift coefficient value was reasonably close to the experimental value for both codes utilized. The drag coefficient generated by XFOIL was considerably closer than that generated by WIND. For the performance analysis the pitching moment is not a required value and therefore the inability for these codes to predict it will not affect the results.
4.1.2.1 Circular Arc CFD Airfoil Evaluation for SSA

A candidate circular arc airfoil was created using the geometry and grid generation package, GAMBIT, a preprocessor for FLUENT. A laminar flow simulation of the airfoil was first performed. The airfoil has 5% camber and 1% thickness. The leading and trailing edges are semi-
circles. A hybrid grid with 41,000 cells is used. The solution domain is an ellipse with its axis inclined 25 deg. with respect to airfoil chord line. This orientation was chosen to include as much of the wake as possible in the solution domain at large angles-of-attack. A hybrid mesh with a rectangular boundary-layer mesh close to the surface and an unstructured mesh everywhere else was used.

Figure 4-30: Hybrid mesh used in the present simulation. The axis of the elliptical outer boundary is inclined to capture as much of the wake as possible and make the outflow boundary condition more accurate. Note the large size of the domain compared to the airfoil.

Figure 4-31: Part of the grid close to the airfoil surface.
Figure 4-32: A magnified view of the hybrid grid at the airfoil leading edge

Figure 4-33: Velocity vectors; the LEV is still attached to the top surface
Chapter 4.0 SSA Design and Feasibility Analysis

4.1 Aerodynamic Design and Analysis

Figure 4-34: Static pressure distribution; the leading edge vortex can be clearly seen

Figure 4-35: Lift variation, each cycle represents a series of events consisting of leading edge and trailing edge formation, transport and shedding
4.1.3 Turbulent Flow Simulations at $Re_c = 100,000$

Based on the feedback from the July 2004 review meeting, new CFD simulations have been completed at $Re_c = 100,000$. Three cases corresponding to $\alpha = 5$ deg., 10 deg., and 15 deg. have been completed. These turbulent simulations using the Spalart-Almaras turbulence model show that at the lower angle-of-attack, the leading edge vortex is absent and the lift curve is flat. At the higher angle-of-attack, the lift curve shows periodic variations similar to those observed in the low-Re, laminar simulations. The main difference is in the frequency content and the period of the dominant frequencies that correspond to the leading and trailing edge vortex dynamics. Several secondary vortices are also observed at these higher Reynolds numbers, possibly contributing to the larger frequencies present in the lift curve. The grid is shown in Figure 4-37. Static pressure contours, velocity vectors and turbulent viscosity contours are shown in Figures 4-38 through 4-39 for $\alpha = 5^\circ$, Figures 4-41 through 4-43 for $\alpha = 10^\circ$ and Figures 4-44 though 4-46 for $\alpha = 15^\circ$. For the 5-degree $\alpha$ case, the flow is more or less steady and there is no large LEV present on the upper surface. For the largest $\alpha$ case, strong vortices can be seen on the upper surface. The turbulent viscosity contours show a large wake with alternating clockwise and counterclockwise vortices. The $c_l$ plot for $\alpha = 10^\circ$ shows a maximum value of 1.7. For the smallest $\alpha$ case $c_l = 0.9$ and for the largest $\alpha$ case, $c_l = 1.5$. 

*Figure 4-36: Drag variation; note the nearly 180 deg. phase difference between the lift and the drag peaks*
Chapter 4.0 SSA Design and Feasibility Analysis

4.1 Aerodynamic Design and Analysis

**Figure 4-37: Grid for turbulent simulations**

**Figure 4-38: Static pressure contours, \( \alpha = 5^\circ, \text{Re}_c = 100,000 \)**
**Figure 4-39: Velocity vectors, \( \alpha = 5^\circ, \text{Re}_c = 100,000 \)**

**Figure 4-40: Turbulent viscosity, \( \alpha = 5^\circ, \text{Re}_c = 100,000 \)**
Figure 4-41: Pressure contours, $\alpha = 10^\circ$, $Re_c = 100,000$. Note that the LEV is larger compared to the small angle-of-attack case. The LEV is just detaching from the surface and the TEV is forming at this instant.

Figure 4-42: Turbulent viscosity, $\alpha = 10^\circ$, $Re_c = 100,000$. The vortex cores show high turbulence levels.
Figure 4-43: Velocity vectors, $\alpha = 10^\circ$, $Re_c = 100,000$. The vortex structure is clearly visible. The vortex structure is characterized by several secondary vortices. These also manifest as high frequency components in the lift-curve.

Figure 4-44: Static pressure contours, $\alpha = 15^\circ$, $Re_c = 100,000$. 
Figure 4-45: Velocity vectors, $\alpha = 15^\circ$, $Re = 100,000$.

Figure 4-46: Turbulent viscosity, $\alpha = 15^\circ$, $Re_c = 100,000$. 
4.1.4 Performance of a Two-dimensional Pitching and Plunging Airfoil for a Solid State Aircraft

4.1.4.1 Analysis

Two important non-dimensional parameters, Reynolds number (Re_c) and reduced frequency (k), can be used to study performance of a flapping wing in hover. These can be expressed in terms of the chord length, c, the wing length, l_w (=b/2, semi span) and the flap angle, ψ, in radians, measured between the two extreme wing stroke positions. The mean translational velocity at the wing location midway between the root and the tip can be expressed in terms of the wing beat frequency, f, as follows:

\[ \bar{U} = l_w \psi f \]  
\[ \text{Equation 4-1} \]

A Reynolds number can be defined as:

\[ \text{Re}_c = \frac{\bar{U}c}{\nu} \]  
\[ \text{Equation 4-2} \]

where \( \nu \) is the fluid kinematic viscosity. Substituting for \( \bar{U} \) from Equation 4-1 gives the following for \( \text{Re}_c \):

\[ \text{Re}_c = \frac{l_w \psi fc}{\nu} \]  
\[ \text{Equation 4-3} \]

The reduced frequency is defined in terms of the wing beat frequency, \( f \), as:

\[ k = \frac{fc}{U} \]  
\[ \text{Equation 4-4} \]
Combining the above expression with Equation 4-1 yields the following:

\[ k = \frac{c}{l_a \Psi} \tag{Equation 4-5} \]

Note that the wing beat frequency, \( f_r \), does not appear explicitly in Equation 4-5 for the reduced frequency. The above analysis illustrates the importance of matching both the Reynolds number and the reduced frequency of the animal and a laboratory model. While Reynolds number matching was done in previous work, the importance of matching reduced frequency was not addressed. Note, from Equation 4-5 that frequency tuning is possible by simply adjusting the stroke amplitude, \( \Psi \).

Discussing the forces in terms of lift and drag as used in conventional aerodynamics can be misleading. The forces that are important in flapping mode are the vertical component that balances the weight and the horizontal component that provides thrust. These two components can be expressed using the lift and drag, for the downstroke and upstroke, respectively, as:

**Downstroke:**

\[ F_v = L_d \cos(\gamma_d) + D_d \sin(\gamma_d), \quad F_H = L_d \sin(\gamma_d) - D_d \cos(\gamma_d) \tag{Equation 4-6} \]

**Upstroke:**

\[ F_v = L_u \cos(\gamma_u) - D_u \sin(\gamma_u), \quad F_H = L_u \sin(\gamma_u) + D_u \cos(\gamma_u) \tag{Equation 4-7} \]

where subscripts \( d \) and \( u \) denote the downstroke and upstroke, respectively.

Current approaches to analyzing animal flight attempt to separate hovering flight from forward flight under the assumption that the two are governed by different mechanisms. In our work, we present a unified approach for hovering flight and forward flight; the two differ in the direction of the relative velocity vector and the orientation of the stroke plane and the body axis, both of which are control parameters that the animal uses to achieve the necessary vertical force to balance body weight and horizontal force to provide thrust for forward flight. While the animal is in forward flight, the velocity relative to the wing is the vector sum due to the instantaneous flapping motion and the translation of the body in the forward direction. Thus, in order to achieve favorable force magnitude and direction, the wing must undergo a wider range of motion to attain optimum angle-of-attack and stroke plane inclination.

Designing the vehicle to match the reduced frequency to the Strouhal number of the Karman vortex street (~0.22), justified by the mounting evidence from our flow simulation results, will require additional flow simulation and experimental work to address issues not considered here.

### 4.1.4.2 Numerical Simulation of Plunging/Pitching Airfoil

The simulations were performed using the computational fluid dynamics simulation package FLUENT. The capability to handle boundaries in rigid body motion and/or deforming boundaries has enabled simulation of the wing in pitch-plunge motion. Mesh updating at regular time intervals is referred to as "Dynamic Mesh" in FLUENT. To use dynamic mesh the conservation
equations need to account for mesh motion. The integral form of the conservation equation for a general scalar \( \phi \) on an arbitrary control volume \( V \) can be written as:

\[
\frac{\partial}{\partial t} \int_V \rho \phi dV + \int_{\partial V} \rho \phi (\bar{u} - \bar{u}_g) \cdot d\bar{A} = \int_V \nabla \phi \cdot d\bar{A} + \int_{\partial V} S_\phi dV
\]

Equation 4-8

The first term in Equation 4-8 can be discretized using the following first order formula.:

\[
\frac{\partial}{\partial t} \int_V \rho \phi dV = \frac{(\rho \phi V)^{n+1} - (\rho \phi V)^n}{\Delta t}
\]

Equation 4-9

where \( n \) and \( n+1 \) represent consecutive time levels. The volume is computed using the following first-order formula:

\[
V^{n+1} = V^n + \left( \frac{dV}{dt} \right)^n \Delta t
\]

Equation 4-10

The following form for \( dV/dt \) satisfies the grid conservation law.:

\[
\frac{dV}{dt} = \int_{\partial V} \bar{u}_g \cdot d\bar{A} = \sum_{j} n_j \bar{u}_{gj} \cdot \bar{A}_j
\]

Equation 4-11

where \( n_j \) is the number of control volume faces.

The mesh was updated using two methods, spring-based smoothing and local remeshing. In the spring-based smoothing method, the edge between any two mesh nodes is idealized to have properties of a linear spring. Thus the nodes can be thought of as being connected by a network of interconnected springs, the motion of each being governed by the force created when the nodes move as result of the motion at the boundaries. The new mesh is obtained as the equilibrium state caused by the spring forces. Hook's law gives the force on a mesh node:

\[
\bar{F}_i = \sum_{j} k_{ij} (\Delta \bar{x}_j - \Delta \bar{x}_i)
\]

Equation 4-12

where \( \Delta \bar{x}_i \) and \( \Delta \bar{x}_j \) denote displacement of node \( i \) and its neighbor \( j \), respectively. \( n_i \) is the number of nodes connected to node \( i \). The spring constant \( k_{ij} \) is given by the following expression.

\[
k_{ij} = \frac{1}{\sqrt{|\bar{x}_i - \bar{x}_j|}}
\]

Equation 4-13

At equilibrium, the net force on a node should be equal to zero, which provides the necessary relation for calculating the new node positions, and hence the new grid. The new node positions can be obtained by simultaneously solving the equilibrium equations for all the nodes. An iterative procedure can be implemented.

\[
\Delta \bar{x}_i^{n+1} = \frac{\sum_{j} k_{ij} \Delta \bar{x}_i^n}{\sum_{j} k_{ij}}
\]

Equation 4-14
where \( m \) denotes the iteration step. At convergence, the positions are updated as follows.

\[
\mathbf{x}_i^{n+1} = \mathbf{x}_i^n + \mathbf{\Delta x}_i^m,\text{converged}
\]

\[\text{Equation 4-15}\]

Spring-based smoothing is suitable for unstructured mesh. When the boundary displacement is large compared to the local cell size, cell quality can deteriorate. Cell quality in terms of cell size and skewness can be retained by agglomerating the cells in low cell-quality regions and remeshing to improve quality.

### 4.1.4.3 Results

Preliminary results from the simulations show that dynamic meshing can be successfully implemented under large amplitude oscillations in pitch and plunge. The results were obtained using a sine function for the oscillatory motion in both pitch and plunge. Other profiles for the oscillatory motion can be implemented by choosing appropriate user-defined functions. Top-hat profiles will be useful in mimicking end-of-stroke rotations for pitch control, where the wing does not pitch between end-of-stroke rotations (Figure 4-48). Figures 4-49 through 4-55 show results from simulations in which the plunge motion is up and down and the pitch motion is about the midchord point. It can be seen that the mesh gets modified as the wing undergoes pitch and plunge motion. Bottom surface vortices can be seen in the static pressure contour plot (Figure 4-51) and the velocity vector plot (Figure 4-52) that corresponds to the end of one cycle.

![Figure 4-48: Pitch schedule during half-cycle. A sinusoidal profile (top) has been used in the preliminary simulations. The top hat profile may have advantages over the sinusoidal profile.](image-url)
Figure 4-49: Starting mesh

Figure 4-50: Mesh at the end of one cycle
Chapter 4.0 SSA Design and Feasibility Analysis
4.1 Aerodynamic Design and Analysis

Figure 4-51: Static pressure contours at the end of one cycle

Figure 4-52: Velocity vectors at the end of one cycle
Figure 4-53: Mesh at phase angle $\pi/2$

Figure 4-54: Static pressure contours at phase angle $= \pi/2$
A different simulation of the airfoil in pitch/plunge in which the plunging motion was along the x-axis was conducted using Dynamic Mesh and animations of the velocity vectors (Animation Vec) and static pressure contours (Animation Pstat) have been created. These are included as separate avi files that accompany this report. These two animations clearly show vortex dynamics associated with the pitching and plunging motions of the wing. These preliminary simulations are meant to be technology demonstrators and therefore, optimization of the wing kinematics has not been done. Future studies will address these issues. These animations can be used as design tools to establish wing kinematics during the stroke cycle for optimum performance.

### 4.1.5 Experiments Using Flapping Wing Model

A wing and a pitching-flapping mechanism were built at UMR. The wing is of semi-elliptic planform chosen as a baseline. It is a flat plate with no camber and about 2.1 mm thick. The chord at the base is 50.8 mm and the length is 152.4 mm. It is painted black to minimize reflections of the laser light. A 50 µm platinum wire is mounted along the edges for hydrogen bubble flow visualization purposes. The whole setup is mounted on top of a water tank. Parameters such as flap and pitch amplitudes and phase relationship between the two needs to be established based on analysis and the results from the preliminary experiments. Modeling the wing after a specific animal was not done because a basic understanding of aerodynamic features was desired.

The initial experiments were done with the wing models constructed at UMR, so that any wing such as an IPMC wing can be tested when ready. Once we have the baseline rigid wing operational, the next step would be to use an aeroelastic wing to model the membrane-like structure of animal wings.
Figure 4-56: Schematic of the experimental setup

Figure 4-56 shows a basic schematic of the flow visualization and PIV setup. The laser light sheet will illuminate the hydrogen bubbles created by the platinum wire. In the setup shown, an end view of the flow in the plane of the light sheet will be imaged. Other views will be possible by changing the setup as desired.

4.1.5.1 Wing Motion

It is assumed that there is no deformation and no lagging and, thus, flapping is confined to the stroke plane. Any of the stroke plane orientations could have been investigated by altering the orientation of 'ground' in the calculations. Various fixed angles of attack, along with varying the time of pitching were investigated; however, the wing beat frequency and amplitude were held constant. Both flapping-only and pitching-and-flapping motions were investigated.
4.1.5.2 Flapping

It was assumed that a pair of wings would produce flow structures symmetrical with respect to the plane of symmetry, allowing investigation of only one wing. To model the motion of the flapping of one wing, a mechanism was designed as shown in Figure 4-57. A DC motor was connected behind the wheel (not shown). For the purpose of this experiment, the motor input voltage was kept at 5V, causing the motor to have a period of 4.47 ± 0.07 sec (13.4 ± 2.5 RPM). There was a pin on the wheel at a distance, r, from the center of the wheel (0.0327 ± 0.0002 m). As the wheel turned, a two-bar mechanism with a fixed pivot caused the wing to rotate about the pivot point, modeling a flapping motion.

The experiment was conducted in water for Reynolds number scaling. The tank was 0.3048 m×0.4318 m×0.762 m (12”×17”×30”). It was assumed that the tank was large enough to not interfere with the flow structures created during the flapping motion. The parameters l₁ (0.0637m ± 0.0002 m) and l₂ (0.330m ± 0.005m) were the lengths of the wing arm as labeled in Figure 4-57. \( \psi \), as shown in Figure 4-58, was the stroke amplitude and could be found from r and l₁ \( (\sin(\psi/2)=r/l₁) \). \( \psi \) was calculated to be 1.08 ± 0.01 radians (61.8º). The centroid of the wing was 0.0877 ± 0.0002 m from the tip of the wing. \( s \) was the distance the wing swept out in half a cycle and was calculated at the centroid by multiplying the distance between the centroid and the pivot by \( \psi \), where \( \psi \) is in radians. In one cycle, the tip of the wing traveled 2s, which was 0.523 ± 0.005 m. One flapping frequency, \( f \), that was tested was 0.232 ± 0.005 Hz (period = 4.3 ± 0.1 sec). The following equation can be used to calculate the mean velocity at the centroid:

\[
\overline{U} = 2lr \psi f
\]

where \( l \) is the distance from the pivot to the centroid. For the above frequency, this resulted in a mean velocity of the wing at the centroid, \( \overline{U} = 0.119 ± 0.003 \text{ m/s} \). The instantaneous velocity varied harmonically during the flapping cycle, zero at the flapping extremes and maximum at mid-stroke.

4.1.5.3 Encoder

In order to monitor the wing’s location during its flapping motion, a hollow shaft incremental encoder was mounted on the shaft between the motor and the wheel. A cross-section of the arrangement can be seen in Figure 4-59. The shaft from the motor passed through the encoder and wheel, both of which...
used set screws to attach to the shaft. A 5V power supply was used to power the encoder. From
the encoder, two signals were used, Index and Channel A. Channel A outputted a pulse after a
specified angle of shaft rotation. The encoder used in this experiment pulsed 360 times per rota-
tion. Thus, a pulse was sent to Channel A after every degree of rotation. The Index pulsed only
once per 360° rotation. The signals from the Index and Channel A were sent to a data acquisi-
tion board (DAQ), installed on a computer. A LabView Virtual Instrument (VI) was used to count
pulses from Channel A and reset the count when the Index was triggered. The Index ensured that
missed degree counts did not propagate to the next cycle. The VI recorded the angular displace-
ment of the pin on the wheel and the Index count.

The location of the Index was experimentally deter-
mined by slowly turning the encoder shaft while
watching the index signal on an oscilloscope. The
signal was 5V when the index was triggered and 0V
otherwise. The orientation between the wheel and
the encoder was adjusted so that the Index was trig-
gered when the pin on the wheel was at the top of the
wheel's rotation. It should be noted that during the
first rotation the LabView VI could not ascertain the
encoder's orientation. It was only after the index had
been triggered at least once that the position could be
accurately determined. This was acceptable since it
took approximately two full rotations of the wheel,
or flaps of the wing, for the complete flow to
develop.

To provide pitching, along with flapping of the wing, a servo motor was installed between the
fixed pivot and the wing. The servo's range of rotation was about 120°. For the full range of rota-
tion, the angle of attack at midstroke was 30°. The servo's rate of rotation was not adjustable and
was approximately 32 ± 5 RPM. This resulted in the servo taking approximately 0.5 s to com-
plete the turn. To attach the wing to the servo motor, a plastic horn that was provided with the
motor was used.
The wheel that was attached to the DC motor (Figure 4-61) shows how the position of the wing could be related to the position of the pin. In Figure 4-61, \( \beta \) is the angle through which the pin rotates from the vertical as the wheel rotates counter clockwise. Since the encoder was installed with the Index corresponding to \( \beta = 0^\circ \), \( \beta \) is also the angular position reported by the LabView VI encoder counter. The wing was vertical when \( \beta \) equaled \( 0^\circ \) and \( 180^\circ \). The extremes of the wing’s flapping motion occurred when \( \beta \) equaled \( 90^\circ \) and \( 270^\circ \). The relationship between flapping position and \( \beta \) can be seen in Figure 4-61.

The servo was controlled using a modified controller used to control a robot. As previously mentioned, a LabView VI was used to count the angular position of the wheel. The VI also controlled a digital trigger signal from the DIO pins on the DAQ. This digital signal was sent to the controller. The digital signal triggered the servo as the wing approached its extremes. The approximate location \( \beta \) at which the servo was triggered is marked in Figure 4-62 by a dashed line. If pitching was symmetric with respect to the ends of the strokes, meaning that half of the pitching occurred on either side of the extreme flap position, the servo was triggered to turn counter-clockwise at \( \beta = 65^\circ \) and clockwise at \( \beta = 245^\circ \). These would be smaller for advanced rotation and greater for delayed rotation. For this experiment the maximum flapping angle from the mid-stoke was \( 30.9^\circ \). If the servo was triggered at \( \beta = 65^\circ \), the wing was \( 26.3^\circ \) from mid-stroke. For symmetric pitching, when the count was between 65 and 245 the digital trigger was high (5V), otherwise it was low (0V). The trigger signal with respect to angular position, \( \beta \), may be seen in Figure 4-62.
On the modified controller there was a Basic Stamp, which can be viewed as a programmable brain. The Basic Stamp was programmed to monitor the trigger signal and respond by positioning the servo. One of the input/output pins on the Basic Stamp was used to signal the servo to turn. Another pin was modified to read the trigger signal from the DAQ. This pin was also grounded on the DAQ board. The BASIC program that controlled the position of the servo motor is as follows:

```
Main
Hold1:
    IF IN2=0 THEN Hold1
    serout net,baud,["!1R101"]
    serin net,baud,[charn]
    pause 600
    serout net,baud,["!1R100"]
    serin net,baud,[charn]
Hold2:
    IF IN2=1 THEN Hold2
    serout net,baud,["!1R1FF"]
    serin net,baud,[charn]
    pause 600
    serout net,baud,["!1R100"]
    serin net,baud,[charn]
    Goto Main
```

The Main loop was repeated for each period of the wing’s motion. IN2, used in the third line of the BASIC program, was the input/output pin on the modified controller that received the trigger signal. Hexadecimal numbers, from position 01 to FF, were used to control the position of the wing. The first line of the Hold1 loop was repeated as long as the input signal was low (0). The position of the wing while Hold1 loop was being repeated was position FF. This position is shown in Figure 4-63, in which the wing is represented by the bold line as if viewed from the tip of the wing, the tip of the wing is marked by a dot, and the dashed line is the stroke plane. Thus, the angle of attack is $\alpha$. As soon as the trigger signal switched to high (1) the Hold1 loop would stop repeating. Thus, the program execution would move to the next line and the servo motor would rotate clockwise to position 01. This position is also shown in Figure 4-62. When the rotate signal was received by the servo, a confirmation was sent back to the modified controller. The program then waited 600 ms for the servo to be actuated, after which, the servo was instructed to hold in its last position with the 00 command. Then the Hold2 loop would be repeated until the trigger signal switched back to low, at which point the servo rotated back to position FF. That position was held after 600 ms had passed. Finally, the main loop was repeated for the next cycle.

The trigger signal and angle of attack with respect to encoder angular position, $\beta$, during the flapping motion can be seen in Figure 4-62. The trigger switched from low to high when $\beta$ reached $70^\circ$, which is when the wing is approaching the extreme of its flapping motion. At that
point the servo began to rotate to position 01. While the servo is rotating, the encoder is still turning and the wing flapping. The reverse occurred at the opposite extreme of the flap.

4.1.5.4 Flapping-Only Wing Motion
Flapping-only wing motion was investigated as a first step to test the signals from the strain gauges and observe their behavior. The wing was held at a fixed orientation by holding the servo used for pitching at a specified angle, so that the angle of attack was not zero. The servo was held at the fixed angle until programmed to do otherwise. 45°, 30°, 15°, and 8° cases were investigated.

4.1.5.5 Reynolds Number
This experiments were done at low Reynolds numbers to investigate hovering flight. Because the experiments modeled hovering flight, the wing had no mean linear displacement. As such the velocity used in the Reynolds number calculation was the mean velocity of the wing tip. The Reynolds number was calculated using:

$$Re = \frac{\bar{U}c}{\nu}$$

where \(c\) was the maximum chord length (0.0508 m (2\”)) and \(\nu\) was the kinematic viscosity (1.12×10^{-6} m^2/s (water)). For the sample frequency = 0.232, the Reynolds number was approximately 6000. Different ways of defining the Reynolds number can be found in the literature. Ellington [33] listed the Reynolds number for various animals. He divided them according to the types of flapping motion: horizontal, inclined, and vertical. From this list, a range of Reynolds numbers was found for each type of flight and they are listed in Table 4-4 [33]. Also, Ellington mentioned that birds and bats have Reynolds numbers that fall in the range of 10^4 to 10^5. Table 4-4 also lists the range for the calculated mean lift coefficient, \(C_L\), assuming quasi-steady conditions.

<table>
<thead>
<tr>
<th>Type of Flapping</th>
<th>Mean Lift Coefficient</th>
<th>Reynolds Number</th>
</tr>
</thead>
<tbody>
<tr>
<td>Horizontal</td>
<td>0.6 to 1.8</td>
<td>23 to 6,100</td>
</tr>
<tr>
<td>Inclined</td>
<td>3 to 6</td>
<td>1,900 to 15,000</td>
</tr>
<tr>
<td>Vertical</td>
<td>0</td>
<td>28,000</td>
</tr>
</tbody>
</table>

4.1.5.6 Force Measurements
To measure the force acting on the wing, a strain gage was attached near the base of the wing. Using the assumption that the force was applied at the centroid of the wing (or the pressure was uniform), the force on the wing could be calculated from the strain measurement. This assumption was necessary due to a limitation of the instrumentation. Planned future work will use load cells that can be used to measure forces directly regardless of its point of application.
Calibrated strain gages were used to measure the strain. The strain gage was arranged in a quarter bridge configuration with a resistance of $350.0 \pm 0.4\% \Omega$ and a nominal $2.1 \pm 10\%$ gage factor. The strain gage signal was read by a P-3500 strain gage indicator from Measurements Group. The bridge configuration and resistance determined how the strain gage was connected to the indicator. To use the indicator, it was zeroed, and the gage factor was entered. The readout on the indicator reported the strain in microstrains. To record the strain, the indicator converted the strain reading to volts and the signal was sent via a BNC cord to a data acquisition board (DAQ) installed in a computer. In a LabView VI the signal from the indicator was converted back to microstrains. The LabVIEW VI read and recorded the strain data at specified angular displacements of the wheel. The gage factor, resistance, excitation voltage of the indicator (2.00 VDC $\pm 0.1\%$), lead resistance (3 $\Omega$ $\pm 0.4\%$), and bridge configuration of the strain gage were entered in the VI in order to perform the conversion from volts to strain. Also entered was the input voltage limits. If the voltage limits were set too small the signal could be capped. As the voltage limits increased, the measurement resolution decreased. A voltage limit of $\pm 0.2V$ was acceptable for this application. High frequency noise in the strain data was filtered out using a Windowed low-pass FIR filter. The LabView VI first took a voltage measurement from the indicator when there is no force applied on the wing. This voltage, $V_0$, corresponded to zero strain. Precaution had to be taken to ensure that there was no strain on the wing when the $V_0$ measurement was recorded. The least strain occurred when the wing was stationary and vertical, thus minimizing bending stresses. If the wing was slightly off vertical, gravitational forces created bending stresses on the wing, thus changing the zero value. The VI then recorded the strain every 4 degrees of the wheel position as reported by the encoder. The strain indicator had provisions to adjust a signal amplification factor. This amplification factor had to be found experimentally. Throughout the experiment the strain indicator amplification factor was held constant. By calibrating the final LabView output to the actual strain, it was not necessary to know the strain indicator amplification factor.

The calibration was performed by first horizontally mounting the wing at its base. A perpendicular load was applied to the tip of the wing by suspending weights from a string attached to the tip. This resulted in a bending moment at the strain gage mounting location. The masses of the weights, $m$, that were tested ranged from 1.77 g (1/16 oz) to 24.8 g (7/8 oz), in increments of 1.77 g (1/16oz). It was assumed that the uncertainty in the weights was negligible. A light string was used to attach the weights, and it was assumed that the mass of the string was negligible. The wing was coated with black paint, and the thickness of the wing, $t$, used included the thickness of the paint. It was assumed that the effect of the paint on the thickness was negligible. The material used for the wing was Lucite-TUF. Very few references were found that provided a modulus of elasticity, $E$, for Lucite-TUF. Matweb.com listed four different acrylic sheets available from Lucite, with moduli of elasticity as follows: 3.03, 2.5, 2.96, 3.21 GPa. Thus, the average of these, $2.9 \pm .4$ GPa, was used in the strain calculation. The moment of inertia, $I$, for the wing cross section at the location of the strain gage was found from the wing thickness and the width at the gage mounting location. The moment arm, $b$, was the distance between where the
force was applied and the center of the strain gage. Assuming that the load was applied directly at the tip, the predicted strain on the gage was calculated using the values found in Table 4-5.

Table 4-5: Parameters used in strain calculations

<table>
<thead>
<tr>
<th></th>
<th>Wing Base Width</th>
<th>0.0502 ± 0.0004 m</th>
</tr>
</thead>
<tbody>
<tr>
<td>c</td>
<td>Wing Thickness</td>
<td>0.0025 ± 0.0002 m</td>
</tr>
<tr>
<td>t</td>
<td>Moment Arm</td>
<td>0.1773 ± 0.0002 m</td>
</tr>
<tr>
<td>b</td>
<td>Gravity</td>
<td>9.81 m/s²²</td>
</tr>
<tr>
<td>g</td>
<td>Moment of Inertia</td>
<td>(3.3 + .5) x 10⁻¹¹ m⁻⁴</td>
</tr>
<tr>
<td>l</td>
<td>Modulus of Elasticity</td>
<td>2.9 ± 0.4 GPa</td>
</tr>
<tr>
<td>m</td>
<td>Mass (Range)</td>
<td>1.77 - 58.35 g</td>
</tr>
</tbody>
</table>

The calculated strain was found using the following equation:

\[
\varepsilon = \frac{mgb}{IE}
\]

*Equation 4-18*

Using the values in Table 4-5 in Equation 4-18, the calculated strain was found to be \(m(640) \pm m(150) \mu\text{strains}\), where \(m\) was the mass of the suspended weight in ounces. From this equation the calculated strain was found for each mass, as shown in Table 4-6. This table also contains the measured strain as reported by the strain indicator and LabView VI.

Table 4-6: Strain gage calibration numbers

<table>
<thead>
<tr>
<th>weight (oz)</th>
<th>cale strain (ustrains)</th>
<th>strain uncert (ustrains)</th>
<th>Indicator (+ 3 ustrains)</th>
<th>LabVIEW (+ 10 ustrains)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>2000</td>
</tr>
<tr>
<td>1/16</td>
<td>40</td>
<td>9.375</td>
<td>43</td>
<td>-12000</td>
</tr>
<tr>
<td>1/8</td>
<td>80</td>
<td>18.75</td>
<td>86</td>
<td>-22000</td>
</tr>
<tr>
<td>3/16</td>
<td>120</td>
<td>28.125</td>
<td>132</td>
<td>-41000</td>
</tr>
<tr>
<td>1/4</td>
<td>160</td>
<td>37.5</td>
<td>175</td>
<td>-57000</td>
</tr>
<tr>
<td>5/16</td>
<td>200</td>
<td>46.875</td>
<td>222</td>
<td>-70000</td>
</tr>
<tr>
<td>3/8</td>
<td>240</td>
<td>56.25</td>
<td>269</td>
<td>-83000</td>
</tr>
<tr>
<td>7/16</td>
<td>280</td>
<td>65.625</td>
<td>315</td>
<td>-94000</td>
</tr>
<tr>
<td>1/2</td>
<td>320</td>
<td>75</td>
<td>363</td>
<td>-107000</td>
</tr>
<tr>
<td>9/16</td>
<td>360</td>
<td>84.375</td>
<td>411</td>
<td>-120000</td>
</tr>
</tbody>
</table>
Figure 4-64 shows the strain measured by the strain indicator versus the calculated strain. Ideally, there should be a 1 to 1 ratio between these two values. However, as can be seen in the figure, there is a 1.16 ± 0.01 ratio. The strain recorded by LabView versus the calculated strain is plotted in Figure 4-65. The uncertainty in strain reported by LabVIEW was 10%. By comparing the indicator and LabVIEW strains, the amplification factor can be found to be about 336.

![Figure 4-64: Calibration of strain indicator](image1)

![Figure 4-65: Measured vs. calculated strain](image2)

### Table 4-6: Strain gage calibration numbers

<table>
<thead>
<tr>
<th>weight (oz)</th>
<th>calc strain (ustrains)</th>
<th>strain uncert (ustrains)</th>
<th>Indicator (+ 3 ustrains)</th>
<th>LabVIEW (+ 10 ustrains)</th>
</tr>
</thead>
<tbody>
<tr>
<td>5/8</td>
<td>400</td>
<td>93.75</td>
<td>457</td>
<td>-138000</td>
</tr>
<tr>
<td>11/16</td>
<td>440</td>
<td>103.125</td>
<td>506</td>
<td>-149000</td>
</tr>
<tr>
<td>3/4</td>
<td>480</td>
<td>112.5</td>
<td>550</td>
<td>-159000</td>
</tr>
<tr>
<td>13/16</td>
<td>520</td>
<td>121.875</td>
<td>599</td>
<td>-169000</td>
</tr>
<tr>
<td>7/8</td>
<td>560</td>
<td>131.25</td>
<td>646</td>
<td>-184000</td>
</tr>
</tbody>
</table>
It is assumed that the main force on the wing is due to pressure. The centroid of half of an ellipse was found to be located at a distance \( \frac{4}{3\pi}b \), where \( b \) is the semi-major axis of the ellipse. Since \( b = 0.1524 \pm 0.0002 \) m \((6")\), the centroid was \( 0.06468 \pm 0.0002 \) m from the base of the wing. The force, \( F \), was calculated from the strain, \( \varepsilon \), to be:

\[
F = \frac{E\varepsilon l}{2l_c}
\]

where \( l_c \) was the distance between the centroid of the wing and the strain gage mounting location \((l_c = 0.090 \pm 0.002\) m). The normal force was then found as a function of strain to be \( F = (850)\varepsilon \pm (190)\varepsilon \).

### 4.1.5.7 Noise Filtering

There were many sources of noise in the experimental set-up. There was vibration caused by the DC motor, servo, the linkages and the wing. These vibrations may have resulted in noise in the data. Noise may have also been caused by the electronic circuits used to transfer the strain signal to the LabView VI. To remove noise from the data, a Finite-Impulse Response (FIR) filter was added to the LabVIEW VI. The FIR filter was a low pass Windowed FIR with 50 taps and pass band range of 2.25 to 0 Hz. This pass band range was appropriate because the frequency of the flapping ranged from 0.232 to 0.588 Hz.

### 4.1.5.8 Directions of Lift and Drag

The wing’s flapping velocity was tangent to the stroke plane. Thus, the lift and drag were perpendicular and parallel to the stoke plane, respectively. The normal force and angle of attack were used to find the measured lift and drag force. Figure 4-66 shows two orientations at mid-stroke of the wing with flapping only and no pitching. The arrows at the bottom of the figure point in the flapping direction. The normal force vectors are also shown for each orientation. If the strain gage is on the side of the wing marked with a box in the figure, the gage is in tension in A and compression in B. This resulted in the measured strain and thus the calculated normal force to oscillate during flapping between positive and negative. The component of the normal force perpendicular to the stroke plane, lift, also oscillated from positive to negative. At the same time, drag, acting in the opposite direction of motion, was always positive.
Figure 4-67 shows wing orientation at mid-stroke when pitching is included. Again the measured strain oscillated during flapping between positive and negative. However, the lift was positive in both cases as can be seen from the directions of the normal force in the two cases. The component of the force normal to the stroke plane points in the same direction in both cases. Drag was always positive because it acted opposite to the direction of motion.

The servo took approximately 0.6 s to complete the pitching motion. During that time approximately twelve data points were recorded. The total pitching angle was divided into twelve increments to approximate the angle of attack for the data points during pitching. These approximate angles of attack were then used to find lift and drag component from the normal force.

4.1.5.9 Estimating Tangential Force

To determine if the force tangential to the surface of the wing was negligible, the skin friction drag force experienced by a wing aligned with the free stream flow was estimated. The Blasius boundary layer solution for a flat plate gives the following expression for the skin friction drag coefficient:

\[ C_{D_f} = \frac{1.328}{\sqrt{Re}} \]

The drag due to skin friction, \( D_f \), experienced by one side of the wing was:

\[ D_f = \frac{1}{2} \rho U^2 S C_{D_f} \]

where \( \rho \) is the density of water (999 Kg/m\(^3\)) and \( S \) is the surface area of the wing (0.0122 m\(^2\)). Since the velocity of the wing was not constant along its length, the mean velocity at the centroid, 0.119 m/s, was used instead. To find the total drag due to both sides of the wing, \( D_f \) was multiplied by 2. For the sample frequency discussed above, 0.232 Hz, the total drag due to skin friction on a wing that was parallel to the free stream flow was 0.00295 N. This is an estimate of the total drag because the speeds at various locations on the wing are generally different from the mean speed of the wing at the centroid. The maximum normal force measured in this experiment was generally greater than 0.1 N. Since the estimated skin friction drag was less than 3% of the normal force, the tangential force was considered negligible and thus was not measured in the present set of experiments.
4.1.5.10 Flow Visualization

The tracer particles for flow visualization and PIV imaging were obtained using the hydrogen bubble technique. A small amount of salt was added to the water to enhance electrolysis. A thin (~50 µm diameter) platinum wire mounted along the edge of the wing, served as the cathode, and a graphite rod, submerged in the tank, served as the anode. An adjustable DC voltage was applied between the cathode and the anode. This induced an electrical current through the electrolyte, resulting in hydrogen and oxygen bubbles being formed through electrolysis. Because the particles were formed on the edge of the wing they were effective in following the flow within the vortices formed at the edge of the wing. Increasing the applied voltage increased the bubble number density. Many particles were desired to produce good flow visualization. A voltage of about 15V produced a sufficient number of particles.

In order to visualize flow in planes perpendicular to the span, the wing had to be viewed from the tip. To accomplish this, a mirror was positioned at a 45° angle below the tank with the camera axis horizontal to the ground (Figure 4-68). This arrangement allowed for chord-wise vortices to be clearly viewed. The video was taken using a Sony mini DV Digital Handycam, which used PIXELA Image-Mixer to transfer the digital video to an MPEG file. The resolution of the image was drastically reduced during video file transfer.

There were hydrogen bubbles distributed throughout the tank. To ensure that only particles in a small region were imaged, a thin plane was illuminated using a Boyce Scientific microscope fiber optic light. This was accomplished by shining the lights into the side of the tank with part of it blocked off, as shown in Figure 4-69. Then the video was taken using the configuration shown in Figure 4-68.
Figure 4-70: Preliminary photograph of the wake vortex system created by a flapping wing viewed from the wing tip. Hydrogen bubbles are used as tracer particles. The bright streak on the left is the wing tip. The image resolution has been degraded while importing from AVI format to jpg format.

Figure 4-71: Preliminary velocity vectors from PIV image interrogation of the water flow created by a flapping wing model conducted in the UMR Fluid Dynamics Lab.
The vortices created by the flow are clearly visible in Figure 4-71. The flow was seeded with ~25 µm diameter hydrogen bubbles created by hydrolysis using a platinum wire cathode. Phase-locked PIV data from such images together with force data from the load cells can be analyzed to assess the impact of leading edge and trailing edge vortices on the forces on the wing. A flow visualization picture (not shown) using an ~8 ns pulsed laser light sheet illuminating the flow from above showed the vortex structure at the trailing edge of a cambered plate airfoil at high angle-of-attack.

Figure 4-72: The photograph shows a wing fabricated in our lab by mounting a flexible membrane material on a frame made of thin stainless steel sheet. This wing was fabricated to study dynamic camber variation.
4.1.5.11 Experimental Results
Some preliminary results have been obtained from the experiments. These are presented as strain and force vs. flapping phase angle in degrees.

4.1.5.12 Forces on the Wing
As discussed previously, the strain measurements were recorded and converted to normal force, lift, and drag. Figures 46 and 47 graphically show the results of varying angle of attack, pitching timing, and flapping frequency. The stroke amplitude, $\psi$, was held constant at 61.8º. Thus, the wing’s angular position at the extremes when measured from the center of the stroke was 30.9º and -30.9º. These ends-of-stroke flapping positions are labeled in Figs. 46 and 47. Figs. 46 and 47 show forces over two complete cycles calculated from the filtered strain signals. Even though only two cycles are shown, cycle-to-cycle repeatability has been observed during quasi-steady operation (excluding starting and stopping transients) in all cases.

Figure 46 shows the calculated normal force, lift, and drag on the wing at fixed angles of attack ($\alpha$) of 8º, 15º, 30º, and 45º with flapping frequency of 0.232 ± 0.005 Hz (period = 4.3 ± 0.1 s). As the wing flapped back and forth, the normal force on the wing fluctuated between positive and negative as expected. The maximum force was observed in the middle and zero force approximately at the extremes of the flapping motion. The shape of the normal force signal is nearly symmetric with respect to its peak. However, the magnitude of the normal force peak was slightly greater for the positive flapping position than for the negative. This may be attributed to inaccurate zeroing of the LabView strain recorder. Or may be the flapping was slightly asymmetric, influenced by unintended out-of-plane wing motion. As $\alpha$ increased from 8º to 30º, the magnitudes of the normal force, lift, and drag peaks generally increased. In some cases, the lift at $\alpha = 15$º was greater than at $\alpha = 30$º. At $\alpha = 45$º the normal force and lift were less than at $\alpha = 30$º, indicating perhaps the onset of flow separation. However, the drag was the greatest for $\alpha = 45$º. This suggests an optimal angle of attack between 30º and 45º. This agrees with the observation of flying animals that during the mid-stroke the angle of attack with respect to the stroke plane is generally less than 40º [33].
Figure 4-74: Normal Force, Lift, and Drag for various Fixed Angle of Attack (flapping frequency, $f = 0.242 \text{ Hz}$)
The calculated forces for 30º fixed angle of attack were compared to those for symmetric pitching in Figure 47, at frequency, $f = 0.232 \pm 0.005$ Hz. In symmetric pitching the first half of pitching occurs before the wing changed flapping direction. Symmetric pitching began when the flapping position was 27º. Pitching phase is highlighted in Fig. 47 with shaded vertical bars. Figure 47 shows that pitching causes a spike in normal force and the force dropped to approximately zero after pitching was completed. This resulted in an increase in lift and drag during the pitching phase that was not seen in the flapping-only cases. A larger peak was present during mid-stroke as in the flapping-only cases. The peak normal force was greater for the flapping-only cases. However, because of pitching, the lift is always positive for symmetric pitching, whereas, the normal force and lift fluctuated between positive and negative for fixed angle of attack flapping. Obviously, flapping only motion without pitching would not be suitable for force generation in the desired direction to provide the necessary force to sustain hovering. Also, the drag was less with pitching than without. Also evident in Figure 46 is that around the extremes (30.9º and -30.9º) of the flapping-only cases, the normal force levels out around zero. This suggests a region of low force generation at the flapping extremes.
Figure 4-75: Fixed angle of attack and symmetric pitching (f = .232 Hz).
4.1.6 Technology Roadmap - Aerodynamic Analysis

Optimization of flapping wing aerodynamics: The sizing analysis for the SSA (see Section 4.3) revealed the critical nature of the details of the wing motion and airfoil shape during a flapping cycle. It was observed that the wing down-stroke provides the majority of lift in the cycle, while also reducing drag to a large extent. On the up-stroke significantly less lift is generated, and the drag is also at its highest levels. This points towards a potential direction for future study, namely optimization of the flapping wing aerodynamics (airfoil shape and characteristics, wing motion) to increase overall lift-drag ratios on the SSA. For example, altering the airfoil shape on the wing up-stroke to enhance its lift and reduce drag could provide significant benefits in the SSA performance. The fully active wing surface that can be achieved using the IPMC material allows the airfoil shape to be altered during a flapping cycle. For example, the airfoil camber could be altered on the up-stroke to produce an airfoil shape better suited for operation at negative angles of attack. Optimization of the wing motion, e.g. pitch angle distribution along the wingspan, would also be undertaken. The potential also exists to use the IPMC material to change the wing surface texture to manipulate the boundary layer in order to increase performance. DOD or NSF may provide potential funding sources for aerodynamic optimization studies in flapping wing flight.

4.1.6.1 Future Computer Simulations

1. Verify predictions on 2D airfoils using available airfoil performance codes.
2. Flapping wing CFD simulations are essential to a full understanding of the associated aerodynamics. However, careful planning, resource allocation, and sufficient lead time would be key components of such an effort, especially when undertaken in a university environment. Identifying CFD software capable of solving such a problem, and recruiting Ph. D.-level graduate students will be required. Recruiting Ph. D. students will require an uninterrupted project duration of at least three years. It is estimated that a six-million cell mesh would be the minimum acceptable mesh size for such a simulation. Published results based on the works various groups show that the results are qualitatively useful but quantitatively inadequate. Euler simulations are probably not of much value for this problem. The CFD simulation effort should go hand in hand with ongoing experiments on flapping wing aerodynamics.

4.1.6.2 Experiments on Flapping Wings

The solid state aircraft concept would benefit from systematic experiments on the aerodynamic performance of flapping wings. In particular, the lift and drag predictions on 2D airfoil sections used in the design analysis from the JAVAFOIL/XFOIL codes need to be experimentally verified. Subsonic flow facilities are available at various institutions of SSA team members (NASA Glenn, U. Missouri-Rolla WPI) to perform these studies. Dr. M. Shahinpoor has also developed a flapping wing demonstration model of approximately 1 meter wingspan that uses the IPMC materials. With further development this demonstration model could be used in wind tunnel tests to measure and confirm lift coefficient and drag coefficient values (both averaged and instantaneous) over a flapping cycle. Variation of parameters affecting SSA performance would include Reynolds number, wing motion, and aspect ratio effects.
4.1.6.3 Aerodynamic Stability Analysis

We have not specifically addressed aircraft stability issues in the Phase I and II design studies. Further study of both static and dynamic stability about the pitch, roll, and yaw axes are needed. This should concentrate first on static and dynamic pitch stability through characterization of center of pressure, quarter-chord moments trends for the flapping wing. Center of gravity envelope considerations would lead to verifying that the SSA would be stable in flight through a flapping cycle. Studies of this type would lead directly into consideration of control system requirements for the aircraft including the necessity for aerodynamic control (tail) surfaces.

4.1.6.4 Scale-model Demonstrator - Flight Tests

Future development of the solid state aircraft concept would benefit from development of a scale-model demonstrator vehicle capable of flight. The flapping wing demonstration model developed by Dr. M. Shahinpoor could be further developed for flight testing. This testing could proceed in several stages. A first stage would be a tethered flight test where the scale-model demonstrator (SSA SMD) is flown at the end of a tether in a circular flight path. A scale-model demonstrator of approximately 1 meter wingspan and 1 kg weight is anticipated. Tethered flight tests have been used extensively by Dr. David Olinger at WPI for unmanned micro-air vehicle (MAV) development. They have several advantages as a preliminary flight test stage, including that power can be supplied through the tether from a power source (battery). This would allow for initial testing of the SSA SMD even if the solar cell and battery technologies have not matured to a feasible level. The constrained flight in the tethered test also eases some of the control/stability requirements for the aircraft, although static and dynamic pitch stability can be studied and confirmed. This tethered flight test would also confirm the aerodynamics and flight capability features of the solid state aircraft. Free flight tests, in which control/stability issues would need to be further addressed, could then follow. DARPA may provide a potential funding source for the SSA SMD development and flight tests.

4.1.7 Quasi-steady, Blade Element Analysis of Flapping Wing Motion

4.1.7.1 Quasi-steady Analysis

The analysis used to determine the resultant lift, drag and thrust forces on the SSA’s flapping wing combines a quasi-steady analysis with a blade-element technique. A quasi-steady analysis ignores unsteady effects by effectively ‘freezing’ the wing motion at times separated by a specified time step $\Delta t$. The quasi-steady analysis is incorporated by dividing the dynamic wing motion into $i_{MAX} = 100$ equal time steps where $t_i = i \Delta t; \ i = 1, 2, \ldots, i_{MAX}$. The local, instantaneous freestream velocity $V = \sqrt{V_{flight}^2 + V_{flap}^2}$ oriented at an angle $\beta = \tan^{-1}(V_{flap}/V_{flight})$ (with respect to the horizontal flight path) results due to a combination of the flight velocity and wing flapping velocity. The various resultant velocities and forces during an wing upstroke are shown in Figure 4-76. In the quasi-steady analysis, the flapping velocity is given by:

$$V_{flap} = V_{MAX} \sin\left(\frac{2\pi t_i}{i_{MAX}}\right)$$

*Equation 4-22*
where $V_{\text{MAX}}$ is the maximum wing velocity at each wing section. In order to determine the $V_{\text{MAX}}$ values, the wing is assumed to take on a parabolic shape (see Figure 4-77 and Appendix D) at all time instances during its motion.

The angle $\beta$ varies along the wing span due to variation in $V_{\text{flap}}$ since the flapping amplitude increases as one moves towards the wing tip. The effective angle of attack felt by the wing section is given by:

$$\alpha_{\text{EFF}} = \alpha - \beta$$  \hspace{1cm} \text{Equation 4-23}$$

where $\alpha$ is the geometric angle of attack (pitch angle). The 2D airfoil analysis codes JAVAFOIL/ XFOIL are steady flow solvers. In order to analyze the flapping wing motion, JAVAFOIL/ XFOIL are run in a ‘quasi-steady mode’ by inputting the parameters $\Re_{\text{eff}} N_{\infty}$, and $\alpha_{\text{EFF}}$ described above so that the effect of the flapping motion is accounted for. The solvers return the local lift and drag coefficients $C_l$ and $C_d$ which are then post-processed using:

$$L' = C_i \frac{1}{2} \rho V^2_c \cos \theta$$

$$D' = C_d \frac{1}{2} \rho V^2_c$$

Here $c$ is the local chordlength and $\theta$ is the wing deflection angle (shown in Figure 4-77) given by:

$$\theta = \tan^{-1}\left(z_k \tan \theta_{\text{MAX}} \right) / b$$

$$\hspace{1cm} \text{Equation 4-25}$$

where $z_k$ is the local spanwise location, $\theta_{\text{MAX}}$ is the maximum wing deflection angle, and $b$ is the wingspan.
To determine the local, vertical lift $L$ (perpendicular to flight velocity vector), and local, horizontal thrust $T$ (parallel to flight velocity vector) we use:

$$L = L' \cos \beta + D' \sin \beta$$

$$T = L' \sin \beta - D' \cos \beta$$

**Equation 4-26**

### 4.1.7.2 Blade Element Analysis

The blade element analysis is incorporated by dividing the aircraft wingspan into $k_{MAX} = 100$ equal length sections where $z_k = k \Delta z$; $k = 1, 2, ..., k_{MAX}$. Lift and drag coefficient data is obtained from the JAVAFOIL/XFOIL solvers at each section using the procedure described in the previous discussion. The quasi-steady analysis is incorporated by dividing the dynamic wing motion into $T = 100$ equal time steps as described earlier.

The above analysis calculates the parasitic drag force using the JAVAFOIL solver. Induced drag effects are also calculated using Prandtl’s finite wing theory through:

$$D_i = C_{d,i} \frac{1}{2} \rho V^2 c = \frac{C_i^2}{\pi e(AR)} \frac{1}{2} \rho V^2 c$$

**Equation 4-27**

The average lift over the entire flapping motion cycle and wing span is then given by:

$$\bar{L} = \frac{1}{i_{MAX} \Delta t} \sum_{i=1}^{i_{MAX}} \sum_{k=1}^{k_{MAX}} (L' \cos \beta + D' \sin \beta) S_k \Delta t$$

where $S_k$ is the local area of wing section $k$ along the wingspan.

The average thrust over the entire flapping motion cycle and wing span is given by:

$$\bar{T} = \frac{1}{i_{MAX} \Delta t} \sum_{i=1}^{i_{MAX}} \sum_{k=1}^{k_{MAX}} (L' \sin \beta - D' \cos \beta - D_i) S_k \Delta t$$

**Equation 4-29**

In order to achieve steady, level flight it is required that
\[ \mathcal{L} = W \]

\[ \overline{T} \geq 0 \]

where \( W \) is the aircraft weight. The quasi-steady, blade element analysis for the SSA wing motion is incorporated into the power production and sizing analysis spreadsheet described in Section 4.3.

### 4.1.7.3 Validation of JAVAFOIL Code

The JAVAFOIL code used in the aerodynamic analysis of the SSA wing was validated by comparing simulation results for lift and drag coefficients to previous experiments on low Reynolds number airfoils [84]. JAVAFOIL [50] utilizes a higher order panel method with linearly varying vorticity for lift coefficient calculation. In order to calculate parasitic drag, a boundary layer integral method is used starting at stagnation points with appropriate criteria for transition and separation [37]. JAVAFOIL [50] is most accurate for Reynolds numbers (based on airfoil chordlength) of \( 5 \times 10^5 < Re < 2 \times 10^7 \). As a result it was necessary to validate the performance of the JAVAFOIL code, particularly at Reynolds numbers on the order of 100,000 in the Mars environment. Reynolds numbers in the Earth (\( Re \sim 450,000 \)) and Venus (\( Re \sim 10^6 – 10^7 \)) environments are in a higher Re range where JAVAFOIL’s accuracy has been previously confirmed.

Pelletier and Mueller [84] conducted wind tunnel experiments on two-dimensional flat plate airfoils with various levels of camber for \( 80,000 < Re < 140,000 \). JAVAFOIL results for lift, drag, and lift-drag ratio for a 4% cambered flat plate at \( Re = 140,000 \) is compared to Pelletier and Mueller’s data in Figures 4-78, 4-79, and 4-80. The camber (4%) and airfoil thickness (~2%) are reasonable values for the SSA wing.

Good agreement is observed between the JAVAFOIL and experiment results for lift coefficient values at low angles of attack. However, JAVAFOIL overpredicts the lift coefficient above the stall angle for both positive and negative angles of attack. Larger difference in drag coefficient data are observed low angles of attack. This is reflected in the lift-drag coefficient data comparison of Figure 4-80. The same comparisons are repeated in Figures 4-81, 4-82, and 4-83 for a 0% camber flat plate at a lower \( Re = 80,000 \). There is better agreement in drag coefficient and lift-drag ratio data for lower \( Re \). The results of Figures 4-81, 4-82, and 4-83 suggest that JAVAFOIL can provide accurate aerodynamic performance data as long as the effective angle of attack \( \alpha_{eff} = \alpha - \beta \) is maintained within a range of \(-10^0 < \alpha_{eff} < 10^0 \) during the wing flapping motion.
Chapter 4.0 SSA Design and Feasibility Analysis
4.1 Aerodynamic Design and Analysis

Figure 4-78: Lift coefficient validation data for a cambered plate, 1.93% thickness, 4% camber, leading edge radius of .6, Re= 140,000

Figure 4-79: Drag coefficient validation data for a cambered plate, 1.93% thickness, 4% camber, leading edge radius of .6, Re= 140,000
Figure 4-80: Lift-drag ratio versus angle of attack for a cambered plate: 1.93% thickness, 4% camber, leading edge radius of .6, Re=140,000

Figure 4-81: Lift coefficient validation data for a flat plate: 1.96% thickness, 0% camber, leading edge radius of .6, Re=80,000
4.1.7.4 Airfoil Shape Optimization

An study was conducted using JAVAFOIL find an optimum airfoil shape at a Re = 100,000 in the Mars environment at $\alpha_{EFF} = 4^\circ$. Five different airfoil parameters; maximum thickness, maximum thickness location, maximum camber, maximum camber location, and leading edge radius, were varied in the study in order to optimize the lift-drag ratio of the airfoil. The optimum airfoil shape is shown in Figure 4-84 with maximum thickness = 2% chord, maximum thickness location = 50% chord, maximum camber = 5% chord, maximum camber location =

---

*Figure 4-82: Drag coefficient validation data for a flat plate: 1.96% thickness, 0% camber, leading edge radius of .6, Re=80,000*

*Figure 4-83: Lift-to-drag ratio versus effective angle of attack for a flat plate: 1.96% thickness, 0% camber, leading edge radius of .6, Re=80,000*
20% chord, and leading edge radius = 0.15% chord. Lift and drag coefficient data is presented in Figure 4-85 for the optimum airfoil shape over a wider range of $\alpha_{eff}$.

![Figure 4-84: Final optimized airfoil geometry: 2% thickness and 50% location, 5% camber at 20% location, 0.15% leading edge radius, and Re=100,000](image)

![Figure 4-85: Lift and drag coefficients for optimum airfoil: 2% thickness and 50% location, 5% camber at 20% location, 0.15% leading edge radius, and Re=100,000](image)

### 4.2 Solar Array Power Production

#### 4.2.1 Incident Solar Radiation

The incident solar radiation on the SSA is the key factor in determining how much usable power can be produced by the solar array. The calculation of this incident radiation is critical to determining the feasibility and capabilities of the SSA. The calculation of incident solar radiation is performed in similar manner regardless of the planet of operation. It will depend on the position of the wing, latitude of operation, time of day and time of year. Using these factors and the available solar flux (which will be dependent on the planet of operation and flight altitude) the incident solar radiation on a given segment of the wing can be calculated. The following is a general
method for calculating the solar flux on the wing. The method can be applied to each potential planet of operation.

The solar radiation environment is the most significant environmental factor that drives the design and capabilities of the SSA. However, unlike other design factors such as the winds, the incident solar radiation is very predictable and can be modeled with significant accuracy. The solar radiation environment is constantly changing. As the solar elevation angle changes throughout the day the available output power from the solar array will vary. This change in elevation angle is also dependent on the time of year as well as the latitude. The solar intensity also changes throughout the year. This is dependent on the orbital eccentricity of the planet that causes slight variations in the distance of the planet from the Sun.

To determine the amount of power produced by the solar array the incident flux on the array must be known. This flux is dependent on the shape and position of the wing. Other variables that determine the array output include time of day, time of year, latitude and orientation of the aircraft. All of these factors must be considered when determining the incident solar radiation on a given photovoltaic cell. The environmental factors that influence the output of the solar array are shown in Figure 4-86.

The solar flux at the orbital location (\( S_{Im} \)) for Venus Earth and Mars is given in Chapter 3: Operational Environments, in Tables 3-1, 3-3, and 3-9, respectively in the environmental section. The actual flux will vary throughout the year as the planet orbits the Sun. This variation is caused by the planet’s orbit not being completely circular. The variation in orbital radius (\( r_{orb} \)) from the mean orbital radius (\( r_{orbm} \)) is represented by the eccentricity (\( \varepsilon \)) of the orbit (also given in table 1,3, and 6). The actual solar flux (or intensity, SI) in W/m² for a specific day of the year is determined by Equations 4-31 and 4-32.

\[
SI = S_{Im} \left( \frac{r_{orbm}^2}{r_{orb}^2} \right)
\]

\textit{Equation 4-31}
The orbital radius at a specific time of the year is given by the following equation.

$$r_{orb} = \frac{(1-\varepsilon^2)}{(1+\varepsilon \cos (\alpha))}$$

Equation 4-32

The day angle ($\alpha$) is defined as $0^\circ$ at perihelion.

The power incident on the array is given by the normal component of the incident flux given in Equation 4-31. Due to the shape and motion of the SSA wing and the variation in the sun elevation angle and position throughout the day this incident angle will be different along the array (from the root of the wing to the tip) and will vary throughout the day. Determining this incident angle is a critical factor in modeling the output power for the array.

It is assumed that the SSA wing is a flap plate that is tapered and swept from the wing to the tip. The wing motion consists of flapping and twisting. This geometry is shown in Figure 4-87. The solar array is assumed to be on the upper surface of the wing. There is no solar array on lower surface, although for operation on Venus this may be a desirable addition due to the high albedo of the planet.
Because of the variation in angle of the array between the root and tip of the wing, the solar flux will need to be calculated incrementally and then summed over the surface of the wing. The geometry for calculating the flux on an incremental section of the solar array is given in Figure 4-88. It was assumed that there was no curvature or variation in array angle along the chord from the tip to the trailing edge of the wing. Therefore, a strip of array from the leading edge of the wing to the trailing edge would all be at the same angle to the sun at any given moment regardless of the wing twist or bending.

The incident power \( P_n \) on the array in W/m² for a specific time during the day is given by Equation 4-33 where \( \tau \) is the attenuation of the solar flux due to the atmosphere, \( \theta_l \) is the local sun elevation angle as seen from a specific segment of array which has an inclination angle of \( \beta \) and an orientation angle of \( \alpha \). The orientation angle is represented by the position of the aircraft. To orient the aircraft the angle sign for \( \beta \) is positive (in the range of \( 0 \) to \( \pi/2 \)) for portions of the array facing south when the wing is bent and negative for portions facing north (in the range of \( 0 \) to \( -\pi/2 \)). If the aircraft is flying north to south the east facing portion of the array is in the positive region. Also it should be noted that if during bending a portion of the array is shadowed, the solar elevation angle is less than the array inclination angle \( (\theta_l < \beta) \), it will not produce any power.

\[
P_n = \sum_{i} SI(1 - \tau) \sin(\theta_i)
\]

*Equation 4-33*
The local solar elevation angle ($\theta_i$) is the elevation to the sun as seen from a specific segment of the solar array. This angle will depend on the inclination of the solar array segment ($\beta$), this is the angle of the specific solar array cells to the horizontal. For $\beta = 0^\circ$ the array segment is horizontal and for $\beta = 90^\circ$ the array segment is vertical. The local solar elevation angle is also based on the solar elevation angle ($\theta$) relative to the surface (or horizontal), the latitude ($\phi$) of the airship location, the declination angle ($\delta$) of the planet (which is based on the time of year) and the geometry and orientation of the airship. Using Figure 4-88 the local solar elevation angle can be derived based on the position of the sun and the inclination angle ($\beta$) of a given array segment on the aircraft. From this figure the following relationships can be derived.

\[
\sin(\theta_i) = \frac{x}{a}
\]

\[
x = b(\tan(\theta) \cos(\beta) - \sin(\omega_i) \sin(\beta))
\]

\[
a = \frac{b}{\cos(\theta)}
\]

Substituting Equations 4-35 and 4-36 into Equation 4-34 yields a relation for the local solar elevation angle given in Equation 4-37.

\[
\theta_i = \sin^{-1}[\sin(\theta) \cos(\beta_i) - \sin(\omega_i) \sin(\beta_i) \cos(\theta)]
\]
Chapter 4.0 SSA Design and Feasibility Analysis

4.2 Solar Array Power Production

The solar hour angle \( \omega \) is a function of the time of day and is given by Equation 4-41, where the time of day is represented as a fraction of one day and is given as a number between 0 and 1 and represents. The solar hour angle is defined as being zero at noon \( (t = 0.5) \), positive before noon, and negative after noon with each hour representing 15° of rotation.

\[
\omega = -2\pi t + \pi
\]

The local hour angle \( \omega_l \) is based on the position of the sun as well as the orientation angle \( \alpha \) of the airship. The local hour angle is given by Equation 4-42 where the aircraft orientation angle is defined as being 0 when the airship is positioned West to East and \( \pi/2 \) when the airship is positioned North to South.

\[
\omega_l = \alpha + \frac{\pi}{2} - 2\pi t
\]

4.2.2 Solar Array Output

The wing shape will affect the area available for the solar array and therefore the power that can be produced by the array. The wing shape is designed to provide an efficient aerodynamic wing for flapping flight. The wing shape used in this analysis is based on the pteradon. This was the largest flying creature and had a thin membrane wing. This membrane wing is very similar to the type of flapping wing that is proposed for the SSA. For the power analysis the wing shape is shown in Figure 4-89.

The key aspects of the geometry for the solar array output are the chord distribution and the shape when bending. The chord distribution is given by Equation 4-43. This distribution is based on the wing planform shown in Figure 4-89. The chord length given by Equation 4-43 is based on the distance from the root of the wing to the tip \( (r) \) and is normalized to 1. That is the root is station 0 and the tip is at a distance of 1 from the root. To get the actual chord length at a given station the equation is multiplied by the wing section length \( (R) \), the distance from the root to the tip.
The chord distribution based on Equation 4-43 is shown in Figure 4-90. This distribution is normalized to 1 and represents the wing shape shown in Figure 4-89. Based on the mean chord length this distribution produces an aspect ratio of approximately 10.

The next aspect of the wing shape that affects power output is the bending shape. For this initial analysis on wing output power a parabolic bending shape was used for the wing during the flapping motion. This is based on the equation of a parabola given by Equation 4-44. The constant “a” is determined by selecting the maximum wing tip angle ($\beta_R$) desired for a given flap cycle. The expression for “a” is given in Equation 4-45. With “a” known the angle at any point along the wing can be calculated ($\beta_i$). This angle is given by Equation 4-46.

$$c_i = (0.36279 - 1.2479r_i + 2.7697r_i^2 + 3.6957r_i^3 - 21.759r_i^4 + 27.501r_i^5 - 11.321r_i^6)R$$  \[Equation 4-43\]

$$y_i = \frac{x_i^2}{4a}$$  \[Equation 4-44\]

$$a = \frac{R}{4 \tan(\beta_R)(1 - \frac{1}{\sqrt{2}})}$$  \[Equation 4-45\]
Equations 4-45 and 4-46 can be combined to provide an expression for the segment tangent angle that is only dependent on the desired tip end angle and the wing section length. This is given in Equation 4-47.

\[
\beta_i = \tan^{-1}\left(\frac{x_i}{4\alpha(1 - \frac{1}{\sqrt{2}})}\right)
\]

\[
\beta_i = \tan^{-1}\left(\frac{x_i \tan(\beta_R)}{R}\right)
\]

To determine the available power at the various proposed operational locations (Venus, Earth, and Mars) some assumptions have to be made on the capabilities of the solar array and its power control system. These assumptions are listed below.

- Solar Cell Efficiency (\(\eta_{sc}\)) 10%
- Solar Cell Fill Factor (\(S_{ff}\)) 80%
- Power Conditioning Efficiency (\(\eta_{pcon}\)) 95%

The 10% solar cell efficiency is the baseline efficiency selected for the analysis. It is representative of an Amorphous Silicon solar array. The efficiency of a solar array is not a constant and will vary with its operating temperature. This relationship between solar cell efficiency and temperature has been well documented and this effect is consistent for all types of solar cells. For example, a typical commercial silicon solar cell efficiency will increase from 13% to 17% when cooled from 25° C to -25° C. This translates into a 30% increase in performance. The opposite
is also true if the solar cell is operated at an elevated temperature. An example of this is shown in Figure 4-91 for various types of solar cells. This figure shows a strong trend of increasing efficiency with temperature for six different types of solar cells.

![Figure 4-91: Effect of temperature on efficiency of various types of solar cells](image)

The assumed efficiency of 10% is applicable to the Earth and Mars environments and is probably conservative due to the cold temperatures at the upper altitudes on Earth and on Mars. Also it would be applicable at the upper levels of the Venus atmosphere as well where the temperatures are cool. However, at lower levels within the Venus atmosphere at and below the cloud layer, the temperature of the atmosphere begins to rise significantly. This is shown in the environmental section of the report. To account for this rise in operating temperature on the efficiency of the solar cells ($\eta_{sc}$) a linear approximation of efficiency as a function of cell temperature ($T$) was used. This approximation is given by Equation 4-48.

\[
\eta_{sc} = 0.33781 - 0.0006902 T
\]

Equation 4-48

Another issue that is specific to the Venus environment is the adsorption spectrum of the solar cells, illustrated in Figure 4-92. As you descent through Venus’s atmosphere, beneath the cloud layer, the available light becomes increasingly limited to the red side of the spectrum. The majority of the blue side gets absorbed. This can become a problem for the operation of a solar cell with this red spectrum light.
Figure 4-92 shows that the red portion of the spectrum is not absorbed by the solar cell material and is transmitted through to the substrate. This transmission begins to happen between the green and yellow wavelengths, well before the red wavelength. Because of this characteristic, present day commercial thin film solar cells would not operate well, if at all, within the red light which dominates the lower regions of Venus’s atmosphere. To operate significantly below the cloud layer would require the development of a new type of solar cell that can utilize the red portion of the light spectrum.

Based on the equations given above for calculating the output power of the array and the environmental conditions at the various planets, baseline power available curves were generated for Venus, Earth and Mars. The curves shown in Figures 4-93 through 4-103 give available power in watts per square meter of solar array. These curves are based on the wing in a horizontal position. The curves represent the available power at different times of the year and different latitudes. Four times of the year were plotted, vernal equinox, summer solstice, autumnal equinox and winter solstice. These dates represent the maximum and minimum points for the year as well as the times of equal day night cycle lengths. Figure 4-93 is the power available for Venus at different latitudes. Since the day length is longer than one year on Venus there are no plots...
showing the available power as a function of time of year. A single day length covers the entire year. Therefore, the yearly change in output power is the same as the change throughout a day period. Figures 4-94 through 4-98 show the power available on Earth and Figures 4-99 through 4-103 show the power available on Mars.

Because the wing flapping cycle can include extended periods of gliding the output power for the wing in the horizontal position is necessary to estimate the available power during this gliding phase. The gliding phase power is critical since it is used to charge the battery for the next wing flap cycle. Also since the wing flapping profile is cyclical with equal motion on the up and down strokes the horizontal position represents the mean or average power available over a given flap cycle and mission period.

**Figure 4-93: Venus: available power throughout a day at various latitudes**

**Figure 4-94: Earth: available power throughout the day at 0° latitude**
Chapter 4.0 SSA Design and Feasibility Analysis
4.2 Solar Array Power Production

Figure 4-95: Earth: available power throughout the day at 20° latitude

Figure 4-96: Earth: available power throughout the day at 40° latitude
Figure 4-97: Earth: available power throughout the day at 60° latitude

Figure 4-98: Earth: available power throughout the day at 80° latitude
Chapter 4.0 SSA Design and Feasibility Analysis
4.2 Solar Array Power Production

Figure 4-99: Mars: available power throughout the day at 0° Latitude

Figure 4-100: Mars: available power throughout the day at 20° latitude
Figure 4-101: Mars: available power throughout the day at 40° latitude

Figure 4-102: Mars: available power throughout the day at 60° latitude
Chapter 4.0 SSA Design and Feasibility Analysis

4.2 Solar Array Power Production

Figure 4-103: Mars: available power throughout the day at 80° latitude

With the wing in the horizontal position the output power curves, given in Figures 4-93 through 4-103, are valid for the aircraft facing any direction at the given latitude and time of year. However, if the wing is not in the horizontal position then there can be variation in the output power depending on the position of the wings and the orientation of the aircraft. This variation in output power with wing angle and direction is shown in Figures 4-104 through 4-108. These figures are for Earth flight on June 21st and December 21st at latitudes of 0°, 30°, and 60°. The aircraft direction was varied from East-West orientation to North-South orientation. The wing bending was described by the wing tip angle, as given by Equations 4-45 through 4-48, of which angles of 0°, 30°, and 60° were analyzed.

Figure 4-103 shows the available power at a latitude of 0° which at this latitude (the equator) are the same irrespective of the time of year. From the figure it can be seen that the aircraft attitude has little effect on the output power. There is a greater effect due to the wing angle. The variation due to wing angle peaks at noon, when the sun angle is high, with an approximate 20% decrease in output power between the wing at a horizontal position and the wing tip at an angle of 60°. As the sun elevation angle decreases the effect of wing angle on output power is reduced.

Figures 4-105 and 4-106 represent the available power on June 21st at latitudes of 30° and 60° respectively. June 21st represents the longest day of the year in the Northern hemisphere and therefore the peak solar availability for the year.
Figure 4-104: Available power at 0° latitude for various attitudes and wing angles

Figure 4-105: Available power: June 21st, 30° latitude for various attitudes and wing angles
The available power shows about the same distribution in Figures 4-105 and 4-106 as it does in Figure 4-104. The largest effect is due to a change in wing angle and it occurs at midday. As latitude increases there is a greater variation in the shape of the power available curves with changes in aircraft attitude at high wing angles.

Figures 4-107 and 4-108 show the change in available power for December 21st, which represents the day with the lowest solar availability. The dates of June 21st and December 21st represent the extremes in solar availability and elevation angle throughout the year. The power available at any other time throughout the year will fall between these two extremes. Figure 4-106 shows the greatest variation in available power with changes in aircraft attitude and maximum wing angle. From this figure it can be seen that there are significant variations in available power throughout the whole daytime period. From these figures it can be seen that the lower the sun angle the greater the variation in available power with wing motion and attitude.

The effect of variations in attitude and wing angle on available power, given in Figures 4-104 through 4-108, were generated for flight on Earth. However, the trends seen are also applicable to flight on Venus and Mars. The absolute values of available power will be different but the variations with latitude will be similar. Because Mars has a similar declination angle as Earth the seasonal variations will be similar. However, there will be little seasonal change on Venus due to its very small declination angle.
4.2.2.1 Solar Array Power Distribution

In addition to the total power that is produced by the solar array the actual distribution of this power along the wing is important. The power production distribution determines how the array is wired as well as how the overall power is managed and distributed along the wing. As an example of the power production distribution along the wing, output power curves were gener-
ated along the wing length under various flight conditions on Earth. These curves represent output power in watts per square meter from the root of the wing to the tip. These output power curves were generated for various wing angles, aircraft attitudes, operational latitudes, time of year and for different times throughout the day. These are shown in Figures 4-109 through 4-112. As with the available power, these figures are for flight on Earth. The trends and profile characteristics shown for flight on Earth will also be applicable for flight on Venus and Mars.

Changing latitude does not have a significant effect on the output profiles of the wing. This can be seen from Figures 4-109 and 4-110. The change in latitude from 0° to 45° produce little change in the output profile curves, only the magnitude of the output power profiles was effected.

**Figure 4-109: Output power variation along the wing on June 21st, 0° latitude, 0° attitude**
Figure 4-110: Output power variation along the wing on June 21st, 45° latitude, 0° attitude

Figure 4-111: Output power variation along the wing on June 21st, 45° latitude, 90° attitude
The change in aircraft attitude however, had a much greater effect on the wing output profile throughout the flapping motion. The effect of attitude on the output profile can be seen by comparing Figures 4-110 and 4-111 which are for the same time of year and latitude but differ in the direction in which the aircraft is pointed. These curves represent the extremes in the orientation of the aircraft from east-west (in Figure 4-110) to north-south (in Figure 4-111). When the aircraft is orientated north-south the power profiles over the wing sections are much more symmetrical. For east-west flight the variation in output power between the both wings can be significant.

The last effect that is shown is that for the time of year. Figure 4-112 shows flight on December 21st, which is the day of lowest solar elevation angle and shortest day time period in the Northern hemisphere. Figures 4-111 and 4-112 show the dramatic effect of shorter day length and lower solar elevation angle on the output power profile along the wing.

4.3 Sizing Analysis

4.3.1 Analysis Approach

The feasibility and sizing analysis for the solid-state aircraft is based on an energy balance conducted over one flapping cycle. The energy collected during the cycle must be sufficient to power the aircraft and meet the power needs for moving the wing over the desired range and motion. A diagram of this energy balance approach is show in Figure 4-113.
The power available and power required are determined through the aircraft properties such as its aerodynamics and mass and the environmental conditions it is flying in. Because the aircraft is solar powered the environmental conditions directly affect the available power instead of just the power required as with conventionally powered aircraft.

### 4.3.2 Aerofoil Properties

Like all flapping wing flyers in nature the solid-state aircraft will operate within a low Reynolds number flight regime. This is due mainly to its required low wing loading and the potential for high altitude operation, where the air density is low. The Reynolds number for the vehicle is based on the chord length \( c \), velocity of air over the wing \( V \) and air density and viscosity \( \rho, \mu \). The Reynolds number (given in Equation 4-49) is a means of gauging the aerodynamic performance and capabilities of a vehicle. Vehicles or devices operating under the same Reynolds number operate aerodynamically similar. The Reynolds number is a nondimensional number representing a ratio of the dynamic forces to viscous forces of the fluid flow over the given object.

\[
R_e = \frac{\rho V c}{\mu}
\]

The altitude at which the solid-state aircraft is flying at will establish the density and viscosity as well as the flight speed. Since the aircraft mass is assumed to be evenly distributed over its wing surface and it is assumed that the SSA will scale proportionally, the wing loading is constant regardless of the vehicle size. Therefore changing the aircraft's size or wingspan affects the chord length (for a given aspect ratio) but it does not affect the cruse velocity. Cruse velocity is however affected by the atmospheric conditions and therefore will vary with altitude. This can be seen in Figure 4-114 for various wing loading values.
Based on the velocities shown in Figure 4-114 the cruise flight Reynolds numbers for the aircraft are shown in Figure 4-115. This figure represents Reynolds number per wing chord length. To get the actual flight Reynolds number the values in the graph would need to be multiplied by the chord length in meters. Reynolds numbers below approximately 2.00E5 are considered to be low and produce low Reynolds number aerodynamic effects.

Airfoils within this low Reynolds number flight regime have characteristics that are somewhat different than those used in higher Reynolds number operation. The main issue, which at low Reynolds numbers is highly variable, is boundary layer stability. That is where the boundary layer separates from the surface or transitions to turbulent flow. These boundary layer effects will in turn affect the wings lift generating capability and control.

Initially a curved flat plate airfoil was used for the solid-state aircraft analysis. Because of the desired low wing loading the airfoil thickness will be kept to a minimum. With a wing chord length much larger than the wing thickness the curved flat plate provided a good initial approximation of the lifting characteristics of the wing. Data on a curved flat plate is shown in Figures 4-115 and 4-116 [74].
Regressions were performed on the lift coefficient ($C_l$) versus angle of attack ($\alpha$) data plotted in Figure 4-116. The curve fits generated by the regressions are given in Equations 4-50 through 4-53.

\[ C_l = -0.010263 + 0.18651\alpha - 0.0062394\alpha^2 \]

Equation 4-50
Chapter 4.0 SSA Design and Feasibility Analysis
4.3 Sizing Analysis

Figure 4-117: Lift coefficient vs. drag coefficient for a curved plat plate at various low Reynolds numbers

As with the data in Figure 4-116, a regression was performed for the data used to generate Figure 4-117. The resulting curve fits of lift coefficient versus drag coefficient for various Reynolds numbers are given in Equations 4-54 through 4-56.

\[ C_l = -0.092355 + 0.22017\alpha_0 - 0.0098994\alpha^2 \]  
\textit{Equation 4-51}

\[ C_l = -0.037329 + 0.19727\alpha_0 - 0.008449\alpha^2 \]  
\textit{Equation 4-52}

\[ C_l = -0.029874 + 0.20066\alpha_0 - 0.0086644\alpha^2 \]  
\textit{Equation 4-53}

\[ C_d = 0.045079 - 0.041756C_l - 0.0027412C_l^2 + 0.049651C_l^3 \]  
\[ -0.10393C_l^4 + 0.065285C_l^5 \]  
\textit{Equation 4-54}

\[ C_d = 0.044899 - 0.02599C_l - 0.13674C_l^2 + 0.41452C_l^3 \]  
\[ -0.51246C_l^4 + 0.24663C_l^5 \]  
\textit{Equation 4-55}
As the wing design evolved and more detail was required by the analysis other airfoil shapes were considered beyond the curved flat plate. The Eppler E 377 airfoil has a number of characteristics that are required by the SSA design. The airfoil is thin with a high cambered surface. It has good low Reynolds number performance and is capable of producing lift at negative angles of attack, down to an angle of attack of $-5^\circ$. The cross section of this airfoil is shown in Figure 4-118. This airfoil was used in the subsequent performance analysis of the aircraft.

\[ \text{Re} = 42,000 \quad C_d = 0.044885 - 0.032763C_i - 0.058057C_i^2 + 0.23791C_i^3 - 0.3715C_i^4 + 0.22436C_i^5 \quad \text{Equation 4-56} \]

The operating characteristics of the airfoil are shown in Figure 4-119. This figure shows the lift and drag coefficients as a function of angle of attack. These results were generated for a Rey-
nolds number of 450,000. From this figure it can be seen that positive lift can be generated from an angle of attack of $-5^\circ$ up to an angle of attack of $10^\circ$. This ability of producing positive lift over a wide range of angles of attack is necessary in order to provide positive thrust over the complete flap cycle. In order to easily utilize this airfoil lift and drag performance data in the analysis a regression was performed to curve fit the data and represent it with a polynomial function. These equations representing the lift and drag coefficients as a function of angle of attack ($\alpha$) are given by Equations 4-57 and 4-58 respectively.

\[
C_l = 0.61754 + 0.13536\alpha + 0.0010655\alpha^2 - 5.4562E - 4\alpha^3 - 5.8176E - 6\alpha^4 + 1.1606E - 6\alpha^5 \\
\]

\[
C_d = 0.022043 - 0.0067314\alpha + 6.6412E - 4\alpha^2 + 2.4297E - 5\alpha^3 - 1.2504E - 6\alpha^4 - 4.7105E - 8\alpha^5 \\
\]

### 4.3.3 Power and Energy Required

The power required (P) by the solid state aircraft to maintain flight can be broken into two main categories, power to overcome drag ($P_d$) and power to move the wings ($P_m$) plus the power for the various operational systems and payload. A diagram of the SSA power consumption mechanisms is shown in Figure 4-120.

The movement of the wings consists of the acceleration and deceleration associated with flapping the wing at the desired flapping frequency and through the desired angle of motion. The drag on the vehicle consists of parasite drag, that is the drag associated with the vehicle moving through the air and the induced drag, the drag generated due to the production of lift. The total power that the SSA needs in order to fly is given by Equation 4-59.
The energy requirements for the SSA are dependent on the way the vehicle will operate. The operating scheme consists of a wing flap followed by a period of gliding flight. During the wing flap altitude and velocity will be gained and subsequently lost during the gliding portion of the flight. This flap / glide cycle continues to maintain the SSA near its desired mission altitude. There are two main aspects that define this cycle, the rate at which a flap is performed and the glide time between flaps. These characteristics will depend on the lift to drag of the SSA and the incident power available. Examples of some of the potential flap/glide combinations are shown in Figure 4-121. The cycles shown in the figure are some of the many combinations that can be utilized. Determining the optimum combination of these characteristics under different flight conditions is a main goal of this analysis.

The energy consumed during a flap glide cycle \( (E) \) is the continuous power consumed during the flap \( (P_r) \) multiplied by the time duration of the flap \( (t_f) \).

\[
E = P_r t_f
\]

\textit{Equation 4-60}

However, since there is a period of time between flaps when no power is consumed, the continuous power level required from the solar arrays \( (P) \) is the energy consumed during a flap divided by the total flap time and glide time \( (t_g) \) for one complete cycle.

\[
P = \frac{E}{t_f + t_g}
\]

\textit{Equation 4-61}
4.3.3.1 Power Required Due to Motion

The movement of the wings is the mechanism for generating thrust for the aircraft. The force and associated power needed to move the wing is therefore a key aspect to the vehicle operation. The power required for motion will be much greater than the power to overcome the drag of the vehicle.

The energy required to move the mass of the wing can be easily calculated based on the geometry of the wing, mass distribution along the wing and the flapping rate. These parameters, which include the wing length, flapping frequency and the angle through which the wing moves during the flap cycle, are illustrated in Figure 4-122.

They can be varied to try and optimize the wing design and operation. The optimization consists of maximizing lift while minimizing power required. The loading along the wing due to the wing motion is based on the mass distribution along the wing. The wing geometry is discussed in the power production segment and shown in Figure 4-122. The mass distribution and corresponding loading are shown in Figure 4-123.
As discussed in the power production section of this report, it is assumed that the wing will bend in a parabolic profile given by Equation 4-44. Therefore, each segment of the wing will be at a different angle and velocity throughout the flap motion. The inboard sections of the wing will traverse through a much smaller angle at a much lower velocity than the outboard sections of the wing. For the analysis the wing motion was determined by selecting the maximum angle the wing tip will traverse throughout the flap cycle. By selecting the maximum tip angle the focus of the parabola can be determined for a given wing length, as given by equation Equation 4-45. Based on the maximum tip angle and parabola focus distance the wing shape can be defined.

The maximum angle traversed by any segment of the wing inboard of the tip can be determined based on the distance the segment is from the wing root \(x_i\). This angle is given by equation Equation 4-47. A plot of the maximum wing angle along the wing is shown in Figure 4-124. This plot is for a 5 m wingspan and a maximum tip wing angle of 60°. Utilizing this angle enables the speed of each wing segment to be calculated which is then used to determine the velocity vector of the given wing segment at any point along the flap cycle.

**Figure 4-123: Force and mass distribution along a wing section (for a 5 m wingspan)**
The wing loading due to motion is based on the acceleration of the wing mass. The maximum deflection angle and flapping frequency that describe the wing motion are variables that can be altered to optimize the wing motion. These values can be varied to determine the optimum motion characteristics for the wing for given flight conditions and vehicle size. The wing acceleration was assumed to be a constant throughout the flapping motion. The acceleration goes to 0 at the top and bottom of the flap stroke, when the direction of the wing is reversed. It also changes from negative to positive as the wing passes through the horizontal location. This acceleration profile throughout the flap cycle is demonstrated by Figure 4-125. The acceleration profile for the wing is given by Figure 4-124. Since the acceleration is constant this profile represents the absolute value of the acceleration along the wing while it is moving. The actual sign or convention of the acceleration will change based on whether the wing is speeding up or slowing down at a particular point in the cycle as shown in Figure 4-125.
The curves shown in Figure 4-123 are used to show the force and mass distributions for a
generic case, not specific to a particular flapping frequency or angle. The curve profiles should
be the same for all operating conditions and wing sizes only the absolute values should be
affected by a change in these parameters. The overall magnitude of the force distribution curve
will be greatly influenced by the size and operating conditions. An example of this is shown in
Figure 4-126. This figure shows the force distribution along the wing for wing section sizes of 3
and 5 meters (1/2 the wing span), maximum flap angles of 45° and 60° and flap durations of 4
and 6 seconds.

The force ($F_i$) needed to move a given wing segment is given by Equation 4-62, where $m_i$ is the
mass of an incremental piece of the wing corresponding to a mean radial distance of $r_i$, $\beta_i$ is the
angle through which the wing segment will move during the acceleration and $f$ is the flapping
frequency in cycles per second. Some of these variables and the basic force balance for moving
the wing are illustrated in Figures 4-122 and 4-126. The wing section is considered to be one
half the wing length (from the center body to the tip). The total force ($F$) needed to move the
wings is given by Equation 4-63.

$$F_i = m_i \frac{2\beta_i r_i}{(1/f)^2}$$

\textbf{Equation 4-62}

$$F = 2 \sum_{i=0}^{R} F_i$$

\textbf{Equation 4-63}
The absolute values associated with the force and mass curves given in Figure 4-120 are dependent on the size of the wing, flapping rate and maximum angle through which the wing will move. The combination of these variables has to be optimized to maximize the amount of lift generated by the wing while operating at a power level that can be supplied by the solar array.

The work performed to move a segment of the wing ($W_i$) is given by Equation 4-64. The work or energy consumed by moving the whole wing through a complete flap cycle is given by Equation 4-65.

\[ W_i = 4F_i \beta_i r_i \]  

\[ W = 2 \sum_{i=0}^{i=R} W_i \]
The total work performed (or energy consumed), under a given operational condition and wing geometry, is the area under the force – distance traveled curve. Examples are shown in Figure 4-128 for various flap durations, wing lengths and maximum tip flap angle. This figure shows some important trends that will have an effect on the sizing and limitations of the SSA. As the vehicle size increases there is an exponential increase in the total energy required. This can be seen from the figure where the wing section (half of the total wing span) is increased from 3 to 5 meters in length with the same maximum end angle and flap frequency. Also increasing the flap duration significantly reduced the energy required, as would be expected. This is because the acceleration rate is greatly reduced as the flap duration increases.
Figure 4-128: Force along the wing as a function of distance traveled by each segment of the wing section

\[ P_m = \frac{W}{t_f} \quad \text{Equation 4-66} \]

4.3.3.2 Power Required Due to Drag

The drag due to lift generation and flight through the atmosphere is the second power consuming mechanism being considered in this analysis. For any vehicle that flies through the air, there are two main components of drag. These are parasite drag and induced drag. The parasite drag is the drag produced due to the movement of the vehicle through the air. For the thin shape of the SSA this drag will be mainly due to the surface friction of the air moving over the vehicle surface. The effect of surface friction can be represented by a surface friction drag coefficient \( c_f \). The induced drag is the drag produced through the generation of lift. This type of drag results from the generation of vortices by the wing. Induced drag is the main component of drag generated by a surface that generates lift. This drag is also represented by a drag coefficient \( c_d \). A diagram illustrating the drag acting on the vehicle is shown in Figure 4-129.
The induced drag coefficient is related to the amount of lift generated by the wing as seen by the lift and drag curves of the airfoil given in Figure 4-119. The induced drag coefficient of the wing will depend on the angle of attack the wing or each individual segment of the wing is operating at. Since the wing will be twisting and flapping the induced drag coefficient will vary along the wing and throughout the flap motion. Because of this variation in the drag coefficient and there for drag along the wing, the total wing drag is determined by summing up the drag on small segments of the wing. This drag \( D_i \) is then averaged over the complete flap cycle \( t \) to determine the drag force that must be overcome in order for the SSA to remain in flight.

Based on these two types of drag, the total SSA vehicle drag during cruise can be represented by Equation 4-67, where \( S_i \) is the wing segment area, \( \rho \) is the atmospheric density and \( V_i \) is the velocity at the wing segment. This velocity will vary along the wing because the flap velocity is different from the root to the tip as well as throughout the flap cycle.

\[
\bar{D} = \frac{1}{t} \rho \sum_{i=0}^{R} \sum_{j=0}^{R} V_{ij}^2 (c_{ij} 2S_i + c_{ij} S_i)
\]

\textit{Equation 4-67}

The parasite drag coefficient was assumed to have a value of 0.008. This value was selected based on the estimated surface roughness of the wing. As discussed previously the value of the induced drag coefficient will depend on the angle of attack of the wing. Since one of the key aspects and capabilities of the SSA is the ability to twist and bend the wing during flight the angle of attack along the wing is continually variable and can be controlled during flight. A goal of the analysis was determining what the wing twist profile would need to be throughout the flap cycle. The angle of attack, which is used to determine the twist profile, is controllable and treated as an input in the analysis. The determination of the angle of attack along the wing and
throughout the flap cycle is based on generating sufficient lift to maintain altitude and overcome the vehicle drag as well as minimizing any negative thrust that is generated during the flapping motion. The details, on how the angle of attack profile is determined along the wing as well as throughout the flap cycle, are given in the lift generation section of this report.

The atmospheric density will depend on the flight altitude and planetary location in which the SSA is operating. During cruise the velocity is the flight velocity \( V \) however, during a flap the velocity \( V_f \) is the flight velocity plus a component due to the motion of the wing \( V_m \). Since the wing is moving in an arc, the velocity at each radial station is different and will increase along the wing. The velocity along the wing \( V_{mij} \), at a station \( i \), and time \( j \) due to the wing motion during a flap is dependent on the location along the wing and the moment of time during the flap. This velocity is given by Equation 4-68. This is based on the maximum wing velocity at that location \( V_{max} \). This maximum velocity occurs when the wing passes through the horizontal location during the flap cycle and is given by Equation 4-69. It is also dependent on the angle the wing is at with respect to the horizontal \( \beta_{ij} \). This angle also varies with radial location and time. The total velocity at a given location on the wing during the flap \( V_{fi} \) is given by Equation 4-70.

\[
V_{mij} = \sqrt{V_{max}^2 + \frac{8}{t_f} V_{maxj} \tan^{-1} \left( \frac{r_i \tan(\beta_{ij})}{R} \right)}
\]

\[
V_{maxi} = \frac{8r_i}{t_f} \tan^{-1} \left( \frac{r_i \tan(\beta)}{R} \right)
\]

\[
V_{fi} = \sqrt{V_{mij}^2 + V^2}
\]

The velocity due to the motion of the wing is plotted in Figures 4-130 and 4-131 for a number of radial stations along the wing length from the root \( (r=0) \) to the tip \( (r=R) \) and for maximum wing angles of 60° and 45°. From these figures it can be seen that the change in the maximum flap angle produces a change in the magnitude of the velocity profile along the wing, as would be expected. The shape and distribution of the velocity profile remains the same at a given wing station regardless of the maximum angle the wing is traversing.
From Equations 4-68 and 4-69 it can be seen that other factors that will influence the velocity along the wing are the frequency or flap rate and the wing length. Figure 4-132 shows the velocity due to motion along the wing for flapping frequency of 2. This is one half the frequency used in calculating the curves in Figures 4-130 and 4-131.
By comparing Figures 4-130 and 4-132 it can be seen that decreasing the flapping frequency produces a significant increasing in wing velocity. This trend is also obvious but does illustrate the fact that increasing the flapping rate can significantly increase the velocity (and subsequently lift) generated. However, this extra velocity comes at a price. There is a significant increase in the force and therefore power required to move the wing at the accelerated rates (as shown and discussed in Figures 4-127 and 4-128).

By increasing the wing length the velocity at a given wing station will increase. This is shown in Figure 4-133. This figure is for a 10 m wingspan vehicle. As the wingspan is increased the wing must move at a faster rate to traverse the same maximum wing angle within the same flapping rate. The difference in wing station speed can be seen by comparing Figures 4-130 and 4-133.

To determine the continuous power consumed during a flap cycle due to the aircraft drag ($P_d$), the drag given in Equation 4-67 is multiplied by the velocity given by Equation 4-70. The resulting power is a summation over the length of the wing from the root to the tip and over the total flap cycle.

\[
P_d = \frac{1}{l} \rho \sum_{j=0}^{i=R} \sum_{i=0}^{j=0} V_{fi}^3 (c_f 2 S_i + c_d q_i S_i)
\]

*Equation 4-71*
The total power consumed or required during the flapping motion is the summation of Equations 4-66 and 4-71. The total energy consumed during the flap cycle is given by Equation 4-72. This total energy required \( E_r \) must be equal to or less than the energy produced throughout the flap/glide cycle described under the power production section.

\[
E_r = (P_m + P_d)t_f
\]

**Equation 4-72**

### 4.3.4 Lift Generation

The SSA lift force is generated by flapping the wings. This lift is generated in a similar fashion to that of an aircraft in which the airflow over the wing causes a pressure difference above and below the wing. This pressure difference produces an upward force thereby producing lift. However, since the wings of the SSA are moving there are unsteady flow phenomena that will occur with the SSA that do not occur with conventional fixed wing flight. For this analysis the airflow is assumed to be steady and the lift generation is based on the wing angle of attach and air velocity during the flapping motion.

The amount of lift that is generated will depend on the angle of attack of the wing, the speed at which the SSA is flying and the speed of the wing motion. The total average lift force \( \bar{L} \) generated by the wings over a flap cycle is given by Equation 4-73. This equation represents the summation of the lift generated by incremental wing segments along the wing and averaged over the flap cycle.

![Figure 4-133: Wing segment velocity due to motion for 60° maximum flap angle. 4s flap period & 10 m wingspan vehicle](image-url)
Chapter 4.0 SSA Design and Feasibility Analysis
4.3 Sizing Analysis

Based on the wing angle there will be both a horizontal and vertical component to the lift. The horizontal component will need to be large enough to overcome the vehicle drag and the vertical component will need to be greater than the weight of the vehicle in order to gain altitude for the subsequent glide period. A diagram of the lift force generated is shown in Figure 4-134. The angle or orientation of the lift force and its magnitude will vary along the wing and throughout the flap cycle.

\[
\bar{L} = \frac{\rho}{t} \sum_{j=0}^{j_{\text{max}}} \sum_{i=R}^{i_{\text{max}}} V_{ij}^2 c_{ij} S_i
\]

*Equation 4-73*

As the wing moves through its flap cycle the motion of the wing will change the angle of attack and velocity along the wing from root to tip. This change in angle of attack and velocity will produce a varying lift at each segment along the wing. This variation in lift can be somewhat controlled by having the wing twist at a specified amount during the flapping motion in order to maximize the lift and thrust that is generated. The angle of attack is initially dependent on the resultant velocity vector generated by the flapping motion of the wing. This resultant velocity vector will vary along the wing from root to tip. The angle of this velocity vector to the horizontal is shown in Figure 4-135. This figure is based on the velocity profile given in Figure 4-130 with an aircraft cruise speed of 10 m/s. The velocity vector angle shown in Figure 4-135 would be the angle of attack along the wing and throughout the flap cycle if the wing is held in a horizontal position. That is there was no twisting during the flapping motion.
As would be expected, the angle of attack due to the wing flapping motion increases from the root to the tip with the increase in vertical velocity. This significant variation and increase in angle of attack would stall the airfoil over the majority of the flap motion and along most of the wing-span. In Figure 4-119 it was shown that the Eppler 377 airfoil (that for one reason was chosen because of its ability to operate over a wide angle of attack range) can produce positive lift at angles of attack between $-5^\circ$ and $10^\circ$. So to maintain the wing within this angle of attack range it must twist continuously throughout the flapping motion. This need to twist and bend the wing to optimize its performance is one of the main reasons that flapping flight has been very difficult to reproduce efficiently. This capability is easily accomplished by birds and mammal using muscle systems that give them many degrees of freedom in the wing motions they can produce. However, it is much more difficult to reproduce with a mechanical system. The characteristics of the IPMC material give us the ability to reproduce this fine and complex motion that nature has perfected and truly optimize a wing for flapping flight.

To determine what this twist motion should be the resultant forces (horizontal and vertical) generated by the wing must be considered. The vertical component of the lift force is needed to counter the weight of the aircraft as well as gain altitude if needed. The horizontal component is needed to offset the drag produced by the aircraft during flight. The first approach would be to twist the wing to maintain an angle of attack that produces the optimal performance from the airfoil. For the Eppler 377 this would be an angle of attack of $7^\circ$. This maximizes lift and minimizes induced drag. However, this approach would not produce a vehicle that could maintain flight. The reason for this is illustrated in Figure 4-136. During the downward stroke the resultant force produced by the lift generation would be directed forward and there would be a component that produces lift and one that produces thrust. On the upward stroke however, the resultant lift force would be directed backwards. This would produce a lift component and a negative
thrust or drag component. The thrust and drag produced over the complete flap cycle would cancel out, providing no net thrust.

![Diagram](image1)

**Figure 4-136: Lift force generation due to flapping**

To overcome this problem the twisting of the wing must be done to optimize forward thrust and lift not airfoil performance. To accomplish this the analysis was set up to provide a specific twisting profile for both the up and down stroke. On the upstroke and down-stroke an upper and lower limit is set for the angle of attack. If the angle of attack due to the velocity vectors is greater or less than the limits then that segment of the wing is twisted to meet either the upper or lower limit. By setting these limits the angle of attack range can be maintained within the stall limits of the airfoil and the amount of negative thrust can be controlled on the upstroke. The determination of the optimum angle of attack range on both the upstroke and down stroke is an iterative process that is adjusted during the analysis. An example of the angle of attack profile and the subsequent wing twist profile at different stations along the wing length are shown in Figures 4-137 and 4-138. These figures were generated for an angle of attack range of 0 to 8 degrees for the upstroke and 0 to –0.1 degrees for the down-stroke.

![Diagram](image2)

**Figure 4-137: Wing angle of attack throughout the flap cycle at various stations**
Figure 4-137 shows the wing angle of attack with the specified limits imposed. By comparing this figure to Figure 4-135, which is the wing angle of attack under similar conditions but without the limits imposed, the effect of these limits can be seen. The angle of attack for the majority of the wing remains at the ideal 8° (for the Eppler 377 airfoil) on the down-stroke. On the upstroke the wing angle of attack is held at –0.1° for most of the wing. This produces some lift and thrust but mainly it eliminates the negative thrust from the upstroke motion.

To achieve this angle of attack distribution during the flapping motion the wing must twist at different rates and to different degrees along its length throughout the flap cycle. The required twisting needed to achieve this angle of attack profile is shown in Figure 4-138. This figure shows that the required wing twisting is significant and increases from the root to the tip, as would be expected. In addition to optimizing the wing’s performance, this information on the wing twisting sets the requirements for the wing and the IPMC material. By knowing and understanding the amount of twisting that is needed the material capabilities and response rate can be evaluated to determine if it is capable of achieving this motion. Also it sets the requirements for the electrode control grid that is incorporated onto the IPMC material and is utilized to move and control the wing. The fineness of the grid will depend on the motion that is required. So knowing the twisting requirements will enable various grid densities to be tested and determine what level of fidelity is necessary to match these twisting requirements.

As mentioned previously the angle of attack range can be adjusted during the analysis. The main adjustment occurs on the upstroke. Since on the down-stroke there is no negative thrust generation, the angle limits are set for the maximum performance of the airfoil. On the upstroke however, there is a balance between how much, if any, negative thrust can be tolerated and the amount of aerodynamic performance that is desired from the airfoil. It is easy to see why having an airfoil with good negative angle of attack characteristic is critical for flapping flight. The bet-
The induced drag coefficient (discussed in the previous section) and lift are dependent on the angle of attack. The relationship for lift and drag coefficient as a function of angle of attack for the Eppler 377 airfoil is given in Equations 4-57 and 4-58 respectively. These relationships are plotted in Figures 4-139 and 4-140 for the angle of attack values given in Figure 4-137.

Figure 4-139 shows the induced drag coefficient as a function of the flap cycle for various stations along the wing length. This figure shows that the induced drag coefficient, and therefore drag is the greatest during the upstroke. This is because, as discussed previously, the wing angle is not at an optimum for the airfoil’s aerodynamic performance. On the down-stroke the drag coefficient levels out for most of the wing due to the angle of attack being fixed at 8°. The inboard section of the wing has a lower drag coefficient than the rest of the wing because it is operating at a lower angle of attack due to its lower flap velocity.

![Figure 4-139: Induced drag coefficient along the wing & through the flap cycle](image)

The lift coefficient curves, shown in Figure 4-140 are similar in characteristics to the induced drag curves. The lift coefficient flattens off and is a maximum during the down-stroke.

On the upstroke the lift coefficient is about half what it is on the down-stroke. Again this is because of the off-performance operation of the airfoil due to the limitation placed on the angle of attack. The lift coefficient on the inboard portion of the wing is much less than further out toward the tip due to its lower flap velocity and hence lower angle of attack.

The lift generated by each incremental segment of the wing, from root to tip will vary throughout the flap cycle. From Equation 4-73 it can be seen that this variation will depend on a number of factors that include the wing chord distribution, velocity and lift coefficient which are also varying as functions of the wing length and or flap cycle. The lift generated at various stations
along the wing is plotted in Figure 4-141. The absolute value of the lift generated is not significant for this figure because it is based on the incremental strip of the wing over which the lift was calculated. This incremental wing area is arbitrary. However the distribution or lift profile is of interest. The lift profile shown in Figure 4-141 follows a similar pattern to the lift coefficient curves in Figure 4-140. The lift at each wing station plotted peaks at the center of the downstroke when the wing is moving with its greatest velocity and drops off significantly during the upstroke. Unlike the lift coefficient curves the value of lift generated on the down-stroke and upstroke varies at the different wing stations examined. The greatest lift is generated at the mid span station (0.5 wing station). This is due to the combination of wing velocity and chord distribution. The wing section closest to the wing tip produced the least amount of lift even though it is moving the fastest. This is due to the tapering off of the wing chord toward the tip. The wing area at this section is significantly less than that for the other stations. The 0.95 station was used instead of the 1.0 station because there is effectively no wing area and therefore lift at this last station.

![Figure 4-140: Lift coefficient along the wing & through the flap cycle](image)

The total drag (parasite plus induced) is plotted in Figure 4-142. This figure is done in a similar fashion as that for the lift shown in Figure 4-141. As would be expected due to the off optimal operation of the airfoil the drag on the upstroke is greater than that on the down-stroke. On the down-stroke the drag increases and peaks as the wing moves through the horizontal position where the velocity is a maximum. The only curve that does not follow this pattern is for the inboard most station plotted (0.25 wing station). At this wing station the drag continues to decrease and reaches a minimum at the horizontal position. The difference between this station’s drag and the drag of the stations further outboard can be explained by looking at the angle of attack curves in Figure 4-137.
In Figure 4-137 the angle of attack of the inboard most station does not achieve the maximum angle of attack limit set at 8°. Therefore as the wing moves through its downward motion the angle of attack at this inboard station continues to increase due to the increase in wing velocity. As the angle of attack is increasing over this range the induced drag coefficient is decreasing. The decreasing drag coefficient is what produces the drop in drag throughout the downward motion at this inboard station. For the other stations there is no effect of decreasing drag coefficient because their angle of attack once it reaches 8° is held there by the twisting of the wing.
The total lift and drag, represented by Equations 4-67 and 4-72, are determined by summing up the incremental lift and drag generated along the wing throughout the flap cycle. The summation of the lift and drag as a function of the flap cycle are shown in Figure 4-143.

**Figure 4-143: Total lift and drag generated throughout the flap cycle**

Figure 4-143 represents the lift and drag generated along one wing section, from root to tip. As with the other figures in this section the absolute values are not that important since this was generated for an example case. However, the trends that are shown represent the lift and drag pattern that is generation throughout the flap cycle. For the complete wing section the downstroke provides the majority of lift in the cycle. During this portion of the cycle the drag is also minimized. On the upstroke significantly less lift is generated and the drag is at its highest. These curves point out a direction for potential airfoil and aeronautical research and development. If an airfoil or a method of aerodynamically altering the characteristics of the airfoil on the upstroke to enhance its lift generating capability and or reduce its drag profile, significant benefits can be obtained in the SSA’s performance. Having a fully active surface for the wing that is capable of being altered provides some interesting possibilities as to how the wing or aerodynamics may be altered to increase performance on the upstroke. One potential method for achieving this increased performance might be to alter the camber of the airfoil during the flapping motion. Producing an airfoil shape that is better suited for negative angle of attack operation. Another option may be to use the IPMC material to change its surface texture, affecting the boundary layer or flow over the wing to potentially increase performance.

The total lift generated throughout the flap cycle only tells part of the story. As discussed previously, this total lift is a vector and has a horizontal (thrust) and vertical component. The horizontal component has to be greater, on average, than the average drag throughout the flap cycle. And the vertical component, on average throughout the flap cycle, has to be greater than the weight of the vehicle. A plot of the vertical and horizontal components of the lift force is shown in Figure 4-144.
The vertical lift shown in Figure 4-144 closely follows the pattern given in Figure 4-143 for the total lift generated by the wing. However, the horizontal component of the lift or thrust rises and declines at a much higher rate providing a spike in the thrust generation during each flap cycle. The thrust does become slightly negative during the upstroke but this is more than offset by the positive thrust generated on the down-stroke.

The distribution of the vertical lift and thrust forces along the wing can also provide some insight into the SSA’s flight performance. The vertical lift and thrust for various stations along the wing length (from root to top) are plotted in Figures 4-145 and 4-146 respectively. The curves for vertical lift at stations along the wing follow a similar shape as the total lift curve and the vertical lift curve shown in Figures 4-143 and 4-144. Toward the inboard portion of the wing there is some positive vertical lift generated on the upstroke. The vertical lift peaks around mid-span and decreases toward the wing tip. This is mainly because of the reduction in wing area toward the tip due to the tapering of the wing chord.
Figure 4-145: Vertical lift at various stations along with wing & throughout the flap cycle

The thrust generated at various points along the wing follows a similar pattern to the total thrust generation curve shown in Figure 4-144. Unlike the vertical lift, the majority of the thrust is generated further outboard on the wing. For the curves shown 0.75 wing station shows the maximum amount of thrust generated on the down stroke. Moving further toward the tip (0.95 station) the thrust does decrease somewhat but it is still greater than that generated at the midboard station. Figures 4-145 and 4-146 demonstrate an interesting point. The mid and inboard stations play a greater role in providing vertical lift to overcome the mass of the aircraft whereas the outboard section of the wing provides the majority of the thrust generated by the aircraft.

Figure 4-146: Thrust at various stations along the wing & throughout the flap cycle
The SSA will operate by flapping its wings thereby gaining altitude and then gliding back to its original altitude where it flap its wings again to continue the cycle. This lift cycle is illustrated in Figure 4-147. The gliding portion of the cycle is used to collect energy that is stored for the subsequent flap. As long as sufficient solar energy can be collected during the glide portion to power the flapping motion the cycle can continue indefinitely.

![Figure 4-147: Lift/glide cycle for the SSA operation](image)

The distance the SSA will move upward \(Y\) due to a flap of the wings is given by Equation 4-74, which is based on the gravitational constant \(g\) of the planet on which the SSA is flying, the mass \(m\) and total weight \(W\) of the SSA and the vertical component of the lift generated \(L_v\):

\[ Y = \frac{1}{2} t_f^2 \left( \frac{L_y - W}{m} \right) \left( \frac{L_y - W}{mg} + 1 \right) \]

**Equation 4-74**

The horizontal distance \(X\) the vehicle will move while it descends back to its starting altitude is based on its glide slope. The glide slope is dependent on the lift to drag value for the SSA. This horizontal distance is given in Equation 4-75.

\[ X = Y \frac{C_l}{C_d + 2C_f} \]

**Equation 4-75**

Using the horizontal distance traveled, \(X\), and the cruse velocity \(V\), given by Equation 4-76, the glide time \(t_g\) can be calculated. The glide time is given in Equation 4-77. As discussed under the power production section the total energy available is based on the output of the solar array over the flap and glide period of the cycle.

\[ V = \sqrt{\frac{2W}{\rho C_s S}} \]

\[ t_g = \frac{X}{V} \]

**Equation 4-76**

**Equation 4-77**
4.4 Results

The analysis was used to generate data on the flight capabilities of the SSA under various conditions and at each of the planetary locations being considered, Venus, Earth and Mars. For a given flight location and operating conditions the power available and required over a complete flap/glide cycle is calculated. The methods for calculating these quantities are described in detail in the analysis section. The power required and available for flight is dependent on the flapping frequency, the flap angle and the glide duration. The results are based on an iterative process that evaluates the energy consumed versus the energy collected through a complete flap/glide cycle to determine the flapping frequency and glide duration that balances these quantities. Which produces an aircraft that can collect enough energy over the flap glide cycle to overcome the drag on the plane and move the wings at the required rate. This process is illustrated in Figure 4-148.

![Figure 4-148: Iterative diagram for glide/flap analysis](image)

4.4.1 Operation on Earth

The first series of results were generated to determine what the maximum achievable altitude is for a given size SSA at a specific location (planet, latitude) at a certain time of year. Since these vehicles are solar powered their performance will be significantly affected by the solar elevation angle, which is dependent on latitude, time of year and time of day. The effects of these parameters on the aircraft’s performance has been demonstrated through a number of previous studies for conventional solar-powered aircraft [14, 94, 62, 15, 63]. For the maximum altitude calculations it was assumed that the aircraft was flying at noon, which provides the highest sun angle and therefore greatest performance. A number of assumptions were used to generate these initial results. These assumptions, summarized in Table 4-7, are the baseline operating conditions...
Chapter 4.0 SSA Design and Feasibility Analysis

4.4 Results

and were used to produce the initial set of results shown in Figures 4-149 through 4-178 for Earth operation.

*Table 4-7: Analysis baseline assumptions*

<table>
<thead>
<tr>
<th>Operating Conditions</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Time of Day</td>
<td>12:00 pm (noon)</td>
</tr>
<tr>
<td>Maximum Wing Flap Angle</td>
<td>60°</td>
</tr>
<tr>
<td>Payload Specific Mass</td>
<td>0.25 kg/m²</td>
</tr>
<tr>
<td>Payload Continuous Power Requirement</td>
<td>100 W</td>
</tr>
<tr>
<td>Aircraft Flight Direction</td>
<td>North-South</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Configuration &amp; Aerodynamics</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Aspect Ratio</td>
<td>8</td>
</tr>
<tr>
<td>Parasite Drag Coefficient</td>
<td>0.0008</td>
</tr>
<tr>
<td>Upstroke Max/Min Angle of Attack</td>
<td>0° / -5°</td>
</tr>
<tr>
<td>Down-stroke Max/Min Angle of Attack</td>
<td>8° / 0°</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Power System</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Solar Cell Specific Mass [6,7]</td>
<td>0.12 kg/m²</td>
</tr>
<tr>
<td>Solar Cell Efficiency [6,7]</td>
<td>10 %</td>
</tr>
<tr>
<td>Solar Cell Fill Factor</td>
<td>80 %</td>
</tr>
<tr>
<td>Battery Depth of Discharge</td>
<td>50 %</td>
</tr>
<tr>
<td>Battery Specific Power [8,9]</td>
<td>250 W-Hr/kg</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Ionic Polymer Metal Composite</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>IPMC Specific Mass</td>
<td>2 kg/m²</td>
</tr>
<tr>
<td>IPMC Efficiency</td>
<td>20%</td>
</tr>
</tbody>
</table>

The initial results generated under the Phase I portion of the program and the beginning of the Phase II portion focused on determining the flapping frequency that would produce the minimum required power, thereby maximizing the aircraft’s range and altitude capability. This was a logical approach for the first round of modeling. The wing aerodynamics and lift generating capability were considered constant over the flap motion. Average values for lift and drag were utilized. As the detail and refinement in the modeling capability increased throughout the Phase II portion of the program the actual lift and drag generation throughout the flapping cycle was calculated and utilized directly in the power required and power produced calculation. This additional layer of complexity in the analysis insured that sufficient lift and thrust was being generated in order to overcome the aircraft drag and weight while trying to minimize required power.
Because of the number of variables involved in the calculation of the optimal flap and glide durations, a set of baseline operational conditions was selected. These baseline conditions, given in Table 4-8 were utilized to demonstrate the trends in the sizing analysis and are used to show how varying each parameter affects the aircraft's operational conditions and flight capabilities.

Table 4-8: Baseline operational conditions for Earth flight

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Altitude</td>
<td>10 km</td>
</tr>
<tr>
<td>Date</td>
<td>3/21</td>
</tr>
<tr>
<td>Wingspan</td>
<td>10 m</td>
</tr>
<tr>
<td>Latitude</td>
<td>30°N</td>
</tr>
</tbody>
</table>

Figure 4-149: Required and available energy as a function of flap duration for the baseline operational conditions

The wing angle of attack, which in the updated analysis changed between the up and down motion, could now produce negative thrust (drag) on the upstroke if the optimal wing angle of attack was utilized to minimize the drag coefficient. Because of this a tradeoff was required between power consumption and the ability to produce sufficient thrust over the complete flap cycle. This completely changed the optimization of the glide and flap duration. The original
results produced an optimum flap/glide ratio by minimizing the required power. This turned out to occur when the glide duration went to zero. However, utilizing the updated lift and drag generation model, the optimum operational flap-glide ratio no longer occurred at a glide duration of zero. At a cycle with zero glide duration, the flapping rate was insufficient to generate the required forward thrust to overcome the drag. Therefore, the operational flap-glide cycle required a quicker flapping motion and a subsequent gliding portion was required in order to produce sufficient forward thrust. This trend is illustrated in Figures 4-149 and 4-150.

Figure 4-150: Required and available thrust as a function of flap duration for the baseline operational conditions

Figure 4-149 represents the energy available and energy required over a range of flap durations. As shown in this figure, the required energy over the complete flap-glide cycle is strongly dependent on the flapping rate. At a flap rate of approximately 3 seconds, the available energy equals the required energy. Beyond this point, the required energy is less than the available energy. The required energy slowly decreases reaching a minimum when the glide duration is zero. The available power also slowly decreases with increasing flapping duration. This is due to the decreasing glide duration, reducing the overall flap-glide period thereby reducing the total energy that can be collected over the cycle. The glide duration, which is based on the lift generated during the flapping motion, also decreases significantly with increasing flap duration (as should be expected). The slower the flapping motion, the less the total lift generated and therefore the less the time that the aircraft can glide between flaps. The glide duration does show a small drop and then a subsequent increase until the flap duration reaches approximately four sec-
onds. This is due to the aerodynamics of the wing and the angle of attack limitations placed on the wing from the constraints of the airfoil selection.

The required and available thrusts are also a factor when determining the flap-glide cycle duration. As discussed previously minimizing the required power may not result in an aircraft that can maintain flight. This is illustrated in Figure 4-150. This figure shows the required and available thrust as a function of flap duration. The available thrust decreases significantly as the flap duration increases. At a flap duration of approximately 6.8, the available thrust equals the required thrust. At flap durations greater than this there will not be sufficient thrust to overcome the aircraft drag and maintain flight. The crossover between available and required thrust occurs before the glide duration reaches zero. Figure 4-151 shows a magnified view of the crossover between the required and available thrust from Figure 4-150.

![Figure 4-151: Magnified view of the required thrust and power crossover](image)

From Figure 4-151, the corresponding glide duration for the point where the required and available thrust are equal is approximately 1.25 seconds. Therefore, for the flight conditions specified (baseline operational conditions) the optimal flap-glide cycle would be a flap duration of 6.8 seconds and a glide duration of 1.25 seconds. This also sets the flight conditions for the vehicle and establishes a operational design point for the specified aircraft size flying under the specified conditions. Although the absolute value of these determined flight parameters will change as the flight conditions and aircraft size are varied the overall trends shown in Figures 4-149 through 4-151 will remain consistent for all the conditions evaluated.
Depending on the flight conditions a situation can arise in which there is no solution. That is the power required is more than the power available and or the required thrust is more than the available thrust. An example of this is shown in Figure 4-152.

The required and available power, shown in Figure 4-152, are for the same conditions used to generate Figure 4-149 except the flight altitude was increased to 14.5 km. From this figure it can be seen that the available energy curve never crosses the required energy curve, even as the glide duration reaches zero. Therefore under these flight conditions the aircraft would not be capable of operating at the specified altitude.

Figure 4-152 demonstrates how variations in these baseline parameters can significantly affect the flight capabilities of the SSA. By changing the aircraft size, flight location, time of year or altitude the flight envelope of the SSA will also change. To determine what affect each of these parameters has on the operation on the SSA each was varied from the baseline operational configuration. The effect of changing the latitude and time of year are shown in Figures 4-153 through 4-156. As should be expected changing the latitude and or time of year has no affect on the thrust generation produced at a given flap-glide cycle or on the relationship between the flap rate and the corresponding glide cycle. In the following figures the difference between the thrust available and the thrust required (designated delta thrust) is plotted on each graph. This curve is the same on each figure. Similarly the glide duration curve is also plotted and is the same for each figure.
Figures 4-153 though 4-156 show the difference between the available energy and the required energy (delta energy) for various latitudes from 0° up to 75° N and for four times of the year, March 21st, June 21st, September 21st and December 21st. These dates were chosen because they represent key points during the year, vernal equinox, summer solstice autumnal equinox and winter solstice respectively. The baseline conditions for wingspan and altitude were used to generate all of these figures. Negative values of delta energy and or delta thrust occur when the required energy or thrust is greater than the available energy or thrust. Under these conditions the vehicle cannot maintain flight. Combinations of flap and glide durations that produce positive delta energy and delta thrust values are operating points that, for the specified flight conditions, the SSA is capable of maintaining flight.

*Figure 4-153: Delta energy for various latitudes on 3/21 (10 m wingspan, 10 km altitude)*
Figure 4-154: Delta energy for various latitudes on 6/21 (10 m wingspan, 10 km altitude)

Figure 4-155: Delta energy for various latitudes on 9/21 (10 m wingspan, 10 km altitude)
As with Figure 4-152, the curve representing the difference between available thrust and required thrust, in the above graphs, becomes negative at a flap duration of 6.8 seconds. For this altitude and aircraft size flap durations above 6.8 seconds will not produce sufficient thrust for the aircraft to fly. In Figure 4-153 there is sufficient energy at latitudes up to 60° N to operate for the March 21st date. The lower latitudes have a larger flap duration range they can operate under. As the latitudes increase the flap duration range of operation decreases. The 0° latitude curve provides the largest operational range with a flap duration of approximately 2.8 seconds up to the thrust limited 6.8 seconds. The highest latitude examined, 75°, has no operational points. Therefore the vehicle would not be capable at operating at these high latitudes during this date.

The second graph in the series, Figure 4-154, shows the difference in available and required energy for June 21st, the day of highest solar elevation angle in the northern hemisphere. At this date the latitude of 30° produces the greatest operational range, with flap durations from approximately 2.75 seconds to the thrust limited 6.8 seconds. At this time of year flight is possible at the 75° latitude location.

As would be expected the figure for September 21st is almost identical to the March 21st figure. Figure 4-156, which shows the difference in available and required energy for December 21st, has the most limited latitude range. The 45° N latitude location has a very small operational range between approximately 5.25 seconds to 6.1 seconds. At latitudes beyond this the aircraft cannot operate during this time of the year. This set of figures demonstrates the effect that the operational latitude and time of year have on the aircraft.

The remaining two parameters, aircraft size and flight altitude, will have a significant affect on the available and required thrust. Also aircraft size will affect the required energy. The next
series of graphs will demonstrate how the aircraft size and flight altitude affect the operational range. Figures 4-157 through 4-167 show the difference in energy available and energy required (delta energy) and the difference in thrust available and thrust required (delta thrust) for various size aircraft. The aircraft sizes examined ranged from a wingspan of 10 m to 50 m. These graphs plot delta energy or delta thrust as a function of flap duration. Also plotted is the glide duration as a function of flap duration. This shows the limits on the flap/glide cycle. Once the glide duration goes to zero the flap duration can no longer be increased. The graphs vary the altitude from 10 km to 25 km to demonstrate how this change in operational altitude affects the energy and thrust availability and requirements for a given size aircraft. The remaining operational conditions (date and latitude) utilize the baseline values of March 31st at a latitude of 30° N.

![Graph showing delta energy and glide duration for various altitudes and aircraft sizes.](image)

**Figure 4-157: Delta energy at various altitudes for a 10 m wingspan aircraft**
Figures 4-157 and 4-158 show the delta energy and thrust, respectively, for a 10 m wingspan aircraft at various altitudes up to 25 km. From these figures it can be seen that there is sufficient thrust generation capability throughout this altitude range. However, as the altitude increases the flap duration decreases in order to generate sufficient lift to maintain flight. Because of this the energy required to move the wings is greater than that which can be collected by the solar array over the complete flap / glide cycle. Therefore based on the delta energy curves shown in Figure 4-157 there is insufficient energy available to maintain flight above the at the 15 km altitude level. For this case the energy availability is the limiting factor.

The next two figures, 4-159 and 4-160, are for a slightly larger aircraft with a 15 m wingspan. Figure 4-159 shows the delta energy and Figure 4-160 shows the delta thrust. As with the results for the 10 m wingspan, the limiting factor is the energy availability. Although for the 15 m wingspan aircraft the 15 km altitude level did provide a solution in which there was sufficient energy and thrust generated over the flap / glide cycle to maintain flight at this altitude. Above this altitude there was insufficient energy to maintain flight. From Figure 4-159 it can be seen that flap durations of between approximately 5 and 7 seconds enabled the aircraft to collect enough over the complete cycle and produce sufficient thrust.

The next six figures show similar trends in the delta energy and delta thrust as the aircraft size is increased. These figures show results for aircraft with wingspans of 20 to 50 m.
Figures 4-159 and 4-160 are for a 20 m wingspan aircraft. For this size aircraft the operating range at the 15 km altitude level has increased. The applicable flap duration at this altitude is now from approximately 5 seconds to 9.5 seconds. However there is still insufficient energy collected over the complete flap/glide cycle for the aircraft to attain the 20 km altitude level.
The next aircraft size shown is for a 30 m wingspan, given in Figures 4-163 and 4-164. At this size there is still insufficient energy collected throughout the cycle to achieve the 20 km altitude level. Comparing these figures with those given above for the smaller sized aircraft it can be seen that as altitude is increased the aircraft size necessary to operate at the increased altitude increases at a nonlinear rate. The last set of charts, which represent the delta energy and thrust for a 50 m wingspan aircraft show that there is an operational area at the 20 km altitude level. This operational range goes from flapping rate of approximately 9.5 seconds to 15 seconds. Within this range there is sufficient thrust and energy collected for the aircraft to operate. Of the sizes examined the 15 m wingspan was the smallest capable of operating at the 15 km altitude level. To increase the altitude to 20 km required an aircraft with at wingspan of 50 m more than three times that needed to fly at 15 km. This exponential increase in aircraft size with increasing altitude is due to the decrease in atmospheric density with altitude. The lower the density the more difficult it is to generate lift and the faster the wings need to move for a given size aircraft. This shows how sensitive the capabilities of a given size aircraft are to atmospheric density and therefore flight altitude.

Figure 4-161: Delta energy at various altitudes for a 20 m wingspan aircraft
Chapter 4.0 SSA Design and Feasibility Analysis

4.4 Results

249

**Figure 4-162:** Delta thrust at various altitudes for a 20 m wingspan aircraft

**Figure 4-163:** Delta energy at various altitudes for a 30 m wingspan aircraft
Figure 4-164: Delta thrust at various altitudes for a 30 m wingspan aircraft

Figure 4-165: Delta energy at various altitudes for a 50 m wingspan aircraft
The results shown in Figures 4-153 through 4-166 demonstrate how the baseline operational conditions affect the flight capabilities of the aircraft. A consistent means for measuring the aircraft performance capability under these conditions is by comparing the maximum achievable altitude for a given size aircraft operating under specific operational conditions. From the figures in this section it can be seen that there is a specific flap / glide cycle that will produce the optimal performance for the aircraft and allow it to achieve the maximum altitude.

By using maximum achievable altitude as the sole guideline for determining aircraft performance, a comparison can be made of the effect of changing operational conditions as well as performance characteristics of the various components that make up the SSA. The following figures show the aircraft size necessary to achieve a given altitude under different operational conditions. Each figure represents a specific latitude and has the maximum achievable altitude plotted as a function of aircraft wingspan. As mentioned previously the aircraft sizes proportionately with wingspan. That is the baseline aspect ration or 8 is maintained for each wingspan shown. Each maximum altitude figure has four curves plotted representing four times during the year, Winter solstice (December 21st), summer solstice (June 21st), vernal equinox (March 21st) and autumnal equinox (September 21st). The latitudes examined are separated by 15° and include 0°, 15°, 30°, 45°, 60° and 75°. This latitude and time range provides a complete operational envelope for flight on Earth of a given size aircraft.
Figure 4-167: Maximum achievable altitude for 0° latitude throughout the year

Figure 4-168: Maximum achievable altitude for 15° latitude throughout the year
Figure 4-169: Maximum achievable altitude for 30° latitude throughout the year

Figure 4-170: Maximum achievable altitude for 45° latitude throughout the year
Figures 4-167 through 4-172 show how significantly the capabilities of the aircraft change as the size is increased. Also for any given size aircraft at a specific latitude there is a significant effect on its performance based on the time of year it is operating. As should be expected the effect of time of year has on the aircraft performance is more pronounced at higher latitudes than closer to the Equator.
the equator. All the curves shown follow the same pattern. There is a rapid increase at smaller aircraft sizes that tapers off as the aircraft size increases. This is due mainly to the exponential decrease in atmospheric density with altitude.

The maximum achievable altitude, shown in Figure 4-167, is for 0° latitude. At this latitude there is little variation in maximum altitude throughout the year. This is because at the equator the day length or variation in elevation angle throughout the day is constant throughout the year. There is a slight change in achievable altitude for the different times of the year shown. This is due to the change in solar attenuation at different times of the year.

As the latitude increases the achievable altitude curve for December 21st begins to decrease significantly and provides no flight solutions for the 75° latitude graph. The curves for the remaining dates also shift downward with increasing altitude but not as significantly as the winter solstice curve. The autumnal and vernal equinox curves remain fairly close to one another at all latitude shown. This would be expected since they both represent days with similar solar elevation angles.

The next series of figures plots the flap and glide durations for the maximum altitude curves shown in the previous figures. As would be expected the flap and glide durations increase as the aircraft size increases. At lower latitudes the flap duration is greater than the glide duration over the complete range of aircraft sizes. This changes at higher latitudes where the glide durations are greater than the flap durations. This is because at the higher latitudes the aircraft's maximum altitude is lower, for a given aircraft size. This means that much more lift is generated per flap cycle enabling larger glide durations. Also it should be pointed out that the curves shown, especially those for the glide durations show variations as they trend toward larger values. These variations are produced by the iteration scheme used to calculate the flight conditions for maximum altitude. This iteration error is based on the convergence accuracy of the iteration. It is variations in this convergence that produce the oscillations in the flap and glide curves.
Figure 4-173: Flap & glide durations for maximum altitude operation at 0° latitude

Figure 4-174: Flap & glide durations for maximum altitude operation at 15° latitude
Chapter 4.0 SSA Design and Feasibility Analysis

4.4 Results

Figure 4-175: Flap & glide durations for maximum altitude operation at 30° latitude

Figure 4-176: Flap & glide durations for maximum altitude operation at 45° latitude
4.4.2 Operation on Venus

Figure 4-177: Flap & glide durations for maximum altitude operation at 60° latitude

Figure 4-178: Flap & glide durations for maximum altitude operation at 75° latitude
The conditions on Venus provide a suitable environment for the operation of the solid-state aircraft. As described in the environmental section of this report, Venus has a thick atmosphere and abundant solar energy above the cloud layer. The environmental conditions of Venus were used in the analysis to determine the flight range that would be possible there. The basic assumptions used in the results generated for Earth operation were also used for Venus operation. The trends shown in the Earth results for required and available thrust and required and available energy are consistent for any planet of operation and therefore do not need to be repeated for flight within the Venus atmosphere. Also the trends shown under the Earth results for delta energy and thrust for various aircraft sizes will also apply to flight on Venus.

There are however some significant differences between Earth's and Venus's environment that will effect the operation and capabilities of the aircraft. The first main difference is that the maximum declination angle on Venus is only 3° compared to 23° on Earth. This small declination angle basically eliminates the effect of seasons on the flight range of the aircraft. Because of this results were generated for only one time of year on Venus. For the results presented the worst case was utilized, winter solstice in the northern hemisphere.

The next main difference is the structure of the Venus atmosphere. As shown in the environmental section, Venus's atmosphere is thick (90 bar at the surface) and has a cloud layer that shrouds the planet. This cloud layer extends from approximately 45 km to 65 km in altitude. Although the solar intensity at Venus is about twice what it is on Earth, this cloud layer obscures a significant amount of solar energy. This reduced solar intensity has a significant effect on the flight altitude range of the SSA. Also the temperature within the Venus atmosphere increases significantly from the upper levels of the atmosphere to the surface. Flight within the lower levels of the atmosphere where the temperature is very high can present many materials problems for the aircraft. One serious concern is the operation of the solar array within this high temperature environment. As shown in the power production section, the efficiency of the solar array will decrease as its operating temperature increases. The analysis for flight on Venus takes into account this reduction in solar cell efficiency as well as the attenuation caused by the atmosphere. The combination of these two effects produces a lower limit on the flight envelope for the SSA. Previous studies of solar powered vehicles on Venus have demonstrated this upper and lower limit on altitude for the operation of the aircraft [64, 67, 67, 17, 19].

Even though the efficiency of the solar cells and the atmospheric attenuation were considered in the analysis, the lower limit values produced may still be optimistic. There are other aspects of the atmosphere that can have a significant effect of the aircraft's ability to operate in and below the cloud layer. The first of these is that the clouds contain sulfuric acid. This highly corrosive acid can cause erosion problems with the materials on the aircraft. Some of the materials may be resistant to the acid and it may be possible to coat others to enable them to operate within that environment. The materials chosen for the SSA will need to be selected with the chemical composition of the atmosphere in mind. For example, the IPMC requires a conductive layer or coating on the outside of the polymer for it to operate. Presently the conductive material used is usually Platinum which is itself resistant to corrosion by sulfuric acid. A list of metals that have a high resistance to sulfuric acid include [17]:

- Zirconium
Also in addition to the pure metals listed above there are a number of alloys that have been
developed for corrosion resistance [19]. Some examples of these include:

- Stainless Steels: Stainless steels are iron-base alloys containing at least 12%
  chromium. Maximum corrosion protection occurs, generally, with the highest
  chromium content, which may range up to about 30%. Nickel can also be added
  to the alloy for enhanced corrosion resistance include Nickel is particularly use-
  ful in promoting increased resistance to mineral acids.

- Nickel Alloys: Nickel-base alloys generally are extremely effective corrosion-
  resistant materials in service environments that range from subzero to elevated
  temperatures. Nickel-base alloys are known for their ability to resist severe oper-
  ating conditions involving liquid or gaseous environments, high stresses, and
  combinations of these factors. The principal elements used in nickel-base alloys
  are copper (as in the Monels) and chromium plus aluminum (as in super-alloys
  such as the Hastelloy or Inconel/Incoloy materials). Molybdenum in nickel sub-
  stantially improves resistance to non-oxidizing acids. Commercial alloys for
  ambient temperature applications have employed up to 28% molybdenum for
  service in severe nonoxidizing solutions of hydrochloric, phosphoric, and hydro-
  fluoric acids as well as in sulfuric acid.

Corrosion resistant materials will need to be utilized either directly as components of the SSA
(such as the wiring or plating on the IPMC) or as a thin coating to protect other non-resistant
materials.

The second issue is the light spectrum of the sunlight beneath the cloud layer. As you move
deeper into the atmosphere the spectrum of the available sunlight becomes dominantly red.
However, it is the blue portion of the spectrum that is mostly utilized by present day photovolta-
ics. This can pose a significant problem for generating power deep within the atmosphere. In
order to utilize the wavelengths of light that are available as you move deeper into the atmo-
sphere, new types of solar cell would need to be developed. These cells would both have to oper-
ate at high temperatures and be capable of utilizing the red end of the visible light spectrum.
Although this could be a significant factor in limiting the performance of the solar array, the
reduction in efficiency due to the shifting or the light spectrum at lower levels of the atmosphere
was not considered in the analysis. The main reason for this is that there was no data available
on how this spectrum would change with altitude. So at altitudes much below the cloud layer the
results shown may be overly optimistic based on the solar array's ability to produce power.

The results for flight on Venus of the SSA are given in Figures 4-179 through 4-181. The results
were generated for the same range of altitudes used in the Earth flight analysis. Figure 4-179
shows the maximum and minimum altitude range for various size (wingspans) aircraft at differ-
ent latitudes. There is almost no effect of latitude on the minimum flight altitude. The minimum altitude is mainly determined by the lack of available solar power from the array. As discussed previously this reduction in output power of the solar array is due to the attenuation of the atmosphere and the operating temperature of the solar array.

The upper altitude limit is much more affected by latitude. As with the Earth results as the latitude is increased the maximum achievable altitude decreases. Aircraft sizes of 10 m wingspans or greater are capable of operating above the cloud layer at most latitudes examined. Only the 75° latitude level required a larger size aircraft to operate at that altitude. The ability to fly above the clouds provides a number of science and operational benefits and should be achievable with reasonably sized aircraft.

The flap and glide durations corresponding to the maximum and minimum flight altitude curves given in Figure 4-179 are shown in Figures 4-180 and 4-181 respectively.

In Figure 4-180 the flap durations are generally longer in time than the glide periods for all latitudes except 75°. The reason the 75° latitude level has much greater glide durations is that the aircraft are operating at a lower altitude for a given size. This is shown in Figure 4-179. At 75° latitude the maximum altitude is much less than at the other latitudes. Therefore the operation of the aircraft at this lower latitude or denser portion of the atmosphere, generates significantly more lift during the flapping motion. This enables the greater glide durations seen in Figure 4-180.
In contrast to the flap and glide durations for the maximum altitude, the flap and glide durations shown in Figure 4-181 for the minimum altitude operation are very consistent for all the latitudes examined. This is due to the consistency in the minimum altitude for all of the flight latitudes. At the minimum flight altitude the glide durations are significantly longer than the flap durations. Again this should be expected due to the dense atmosphere at that altitude. Significant amounts of lift can be generated by each flap of the wing thereby enabling the aircraft to glide for extended periods of time. The ability to generate lift at the lower altitude range is significant. The flapping motion can be fairly slow and still generate a large amount of lift and thrust. This operation within the very dense lower levels of the Venus atmosphere poses a unique capability. Because the motion of the wing does not need to be rapid to generate lift the power requirements are significantly reduced. If a solar array can be devised that could operate deep within Venus's atmosphere it would not need to be highly efficient to provide the power needed by the aircraft. In fact an array operating at only a few percent efficiency would still be capable of generating sufficient power for the aircraft to fly. The development of a solar array that could operate at high temperatures and within the red dominated spectrum could potentially enable SSA to operate within the lower levels of the atmosphere, potentially even to just above the surface.
Another positive aspect of operating deep within the atmosphere is that due to scattering the available solar energy comes uniformly from all directions. To take advantage of this the aircraft could have solar cells on both the upper and lower surfaces of the wing. Potentially doubling the output power of the array and somewhat making up for the reduced efficiency. The added weight of the extra array would not be of much consequence since the generation of lift at such high atmospheric density is significantly enhanced.

If materials, coatings and the photovoltaic cells can be produced that could operate within the lower levels of the Venus atmosphere. The ideal place to fly a flapping winged vehicle may not be high up above the clouds but down low near the surface.

### 4.4.3 Operation on Mars

At the beginning of the study Mars was identified as a potential location for operation of the SSA. However, as the work on the vehicle design proceeded it became evident that the ability to fly on Mars would be marginal at best. The initial stage of the analysis focused on flight on Earth. However, there are some direct parallels between solar powered flight on Earth and solar powered flight on Mars. Previous work on solar powered flight on Mars [16] have shown that a solar powered aircraft capable of flight near the surface of Mars would be very similar in size as a similar type aircraft that is capable of flight on Earth at approximately 30 km altitude. The similarity between flight near the surface of Mars and at 30 km on Earth occurs because the atmospheric density is about equal at those two locations. There are other factors besides atmospheric density the play a critical role in establishing the capabilities of a solar power air-
Solid State Aircraft

craft. First among these would be solar intensity. The solar intensity on Mars is about half what it is on Earth. However with regard to the capabilities of a solar powered aircraft on Mars, the reduced solar intensity is offset by the reduced gravitational force. The lower operational power due to lower solar intensity and the lower power requirement due to the lower gravitational force tend to cancel out.

Considering this parallel between high altitude solar power flight on Earth and solar powered flight on Mars, the Earth based results were initially used to gage the applicability of the SSA for flight on Mars. As the Earth based results were produced it became evident that the SSA was not capable of flying at altitudes up to or greater than 30 km. From the Earth results it can be seen that the maximum achievable altitude for the range of aircraft sizes examined was just over 21 km. This is significantly less than the 30 km altitude needed to be comparable to solar powered flight on Mars.

Even though the Earth results indicated that flight on Mars would not be possible with the baseline aircraft, the analysis was set up to model the Mars environment and examine the potential for flying the SSA on Mars. The same range of aircraft sizes that were used for evaluating flight on Earth and Venus were used on Mars. The analysis was begun at an altitude of 0 km (just above the surface) for each of the aircraft sizes. These results are given in Table 4-9.

As expected, there was no flight solution over the range of aircraft sizes evaluated for the baseline aircraft configuration and assumptions. The aircraft performance and flight capability can be enhanced by changing or increasing some of the various characteristics used to size the vehicle. The main factors that can be adjusted to increase the performance of the SSA include:

- Solar Cell Efficiency
- IPMC Operational Efficiency
- IPMC Specific Mass (kg/m2)

There are considerably more material and aerodynamic factors that could have also been adjusted but these were deemed the most influential to the vehicles performance. The range through which each variable could be adjusted and the various combinations of these variables produced too large a design space to evaluate. However, since the point of this exercise is to determine if performance enhancements could produce a vehicle capable of flight on Mars there was no need to evaluate the complete design space. The main objective of this analysis is to

Table 4-9: Flight results for the baseline SSA operating on Mars

<table>
<thead>
<tr>
<th>Wingspan</th>
<th>Maximum Altitude Results</th>
</tr>
</thead>
<tbody>
<tr>
<td>4 m</td>
<td>No Solution</td>
</tr>
<tr>
<td>6 m</td>
<td>No Solution</td>
</tr>
<tr>
<td>8 m</td>
<td>No Solution</td>
</tr>
<tr>
<td>10 m</td>
<td>No Solution</td>
</tr>
<tr>
<td>15 m</td>
<td>No Solution</td>
</tr>
<tr>
<td>20 m</td>
<td>No Solution</td>
</tr>
<tr>
<td>30 m</td>
<td>No Solution</td>
</tr>
<tr>
<td>50 m</td>
<td>No Solution</td>
</tr>
</tbody>
</table>
determine what and or how great an improvement would be necessary to get each size vehicle to operate near the surface on Mars. The results of this exercise are summarized in Table 4-10 for a flight altitude of 0.5 km above the surface of Mars.

Table 4-10: Performance increase required for flight on Mars

<table>
<thead>
<tr>
<th>Wingspan</th>
<th>Parameter</th>
<th>Performance Values Used</th>
<th>Percent Change in Performance over the Baseline Configuration</th>
</tr>
</thead>
<tbody>
<tr>
<td>4 m</td>
<td>Array Efficiency IPMC</td>
<td>37%</td>
<td>+270%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>50%</td>
<td>+150%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>0.5 kg/m²</td>
<td>- 75%</td>
</tr>
<tr>
<td>6 m</td>
<td>Array Efficiency IPMC</td>
<td>36%</td>
<td>+260%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>43%</td>
<td>+115%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>0.70 kg/m²</td>
<td>- 65%</td>
</tr>
<tr>
<td>8 m</td>
<td>Array Efficiency IPMC</td>
<td>36%</td>
<td>+260%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>41%</td>
<td>+105%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>0.85 kg/m²</td>
<td>- 57.5%</td>
</tr>
<tr>
<td>10 m</td>
<td>Array Efficiency IPMC</td>
<td>31%</td>
<td>+210%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>41%</td>
<td>+105%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>0.9 kg/m²</td>
<td>- 55%</td>
</tr>
<tr>
<td>15 m</td>
<td>Array Efficiency IPMC</td>
<td>26%</td>
<td>+160%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>40%</td>
<td>+100%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>1.0 kg/m²</td>
<td>- 50%</td>
</tr>
<tr>
<td>20 m</td>
<td>Array Efficiency IPMC</td>
<td>22%</td>
<td>+120%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>34%</td>
<td>+70%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>1.0 kg/m²</td>
<td>- 50%</td>
</tr>
<tr>
<td>30 m</td>
<td>Array Efficiency IPMC</td>
<td>20%</td>
<td>+100%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>33%</td>
<td>+65%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>1.15 kg/m²</td>
<td>-42.5%</td>
</tr>
<tr>
<td>50 m</td>
<td>Array Efficiency IPMC</td>
<td>18%</td>
<td>+80%</td>
</tr>
<tr>
<td></td>
<td>Efficiency</td>
<td>26%</td>
<td>+30%</td>
</tr>
<tr>
<td></td>
<td>IPMC Specific Mass</td>
<td>1.25 kg/m²</td>
<td>-37.5%</td>
</tr>
</tbody>
</table>

These results do not represent a comprehensive look at what combination of improvements in performance would have achieved the goal of flight near the surface of Mars. Other combinations may have also been successful in producing a configuration that could be capable of flight. However it does demonstrate that with increasing technology development, reducing mass and increasing efficiency, will expand the capabilities of the SSA and potentially enable it to fly within the Martian environment.

The data in Table 4-10 show that as the aircraft size is increased the required performance enhancements are reduced. However, operating very large aircraft (especially on another planet),
Solid State Aircraft

bring their own set of logistical problems. Deployment is probably the most critical factor in the operation of such large vehicle.

Some additional analysis was performed to further evaluate how the increase in performance of the solar array and IPMC material can potentially enable flight on Mars and to lay out a road map of performance requirements needed to enable flight on Mars and enhance the flight range on both Earth and Venus.

A set of baseline values were selected for the critical performance parameters, these are shown in Table 4-11. These values are slightly better than those used in the baseline Earth and Venus analysis and represent possible near-term improvements in the technologies. As seen in Table 4-10 the combination of these performance values alone is not sufficient to enable any of the vehicles sizes evaluated to operate on Mars.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Array Efficiency</td>
<td>15 %</td>
</tr>
<tr>
<td>IPMC Efficiency</td>
<td>25 %</td>
</tr>
<tr>
<td>IPMC Specific Mass</td>
<td>1.50 kg/m²</td>
</tr>
</tbody>
</table>

Table 4-11: Baseline performance values for development evaluation

To determine which parameter has the greatest influence on the SSA's performance each value will be increased, while the other two are held constant, and the size of the aircraft capable of flight just above the surface of Mars will be calculated. The results of this analysis are show in Figures 4-182 and 4-183.

Figure 4-182 shows the effect on wingspan of a change in efficiency of the solar cells and the IPMC material. Also plotted in this figure is the required wingspan for different IPMC specific mass values. Both the solar cell and IPMC efficiency was varied from the baseline value shown in Table 4-11 to 100%. Although this full range is not necessarily realistic it does illustrate what efficiency values would be required for significant reductions in aircraft size needed to achieve flight on Mars. From these curves it can be seen that the increase in solar cell efficiency has a greater effect in reducing the aircraft size than the increase in IPMC material efficiency. Reductions in specific mass of the IPMC material also have a significant effect on the required aircraft size. This figure demonstrates that improvements in any of the three factors can have a significant effect on reducing the required aircraft size for flight on Mars.
To better illustrate how each of these factors affects the aircraft size, the percent change in each of the parameters (solar cell efficiency, IPMC efficiency and IPMC specific mass) was plotted against the corresponding percent change in wingspan. The percent change in the parameters was from the baseline values given in Table 4-11. These results are shown in Figure 4-183. This figure illustrates that reducing the specific mass of the IPMC material has the greatest effect on reducing aircraft size. This seems logical since the IPMC material constitutes a significant portion of the mass of the vehicle. Also because the wings on the vehicle need to move, any reduction in their mass will translate directly into a power savings because of the reduced power needed to accelerate and decelerate the wing.

The curves for solar cell efficiency and IPMC efficiency are fairly similar. Initially, up to an increase in performance of approximately 30%, changes in either parameter produce similar results. Beyond this point, increases in solar cell efficiency have a greater effect on reducing the required wingspan.
These figures demonstrate what types of improvements in critical performance values would be necessary to operate a reasonable sized SSA on Mars. But beyond the Mars application these results also indicate how a greater altitude range could be achieved on both Earth and Venus. The previous discussion indicates that if a given configuration SSA could fly near the surface of Mars it would also be capable of flying at approximately 30 km on Earth. These figures indicate what types of performance increases would be needed to operate at that altitude on Earth. This is significantly above the maximum altitude, of 21 km, that was determined using the baseline parameters for flight on Earth. So by considering the component performance increases shown in Figures 4-182 and 4-183 significant improvements in altitude capability (for flight on Earth or Venus) or a reduction in the size of aircraft needed to fly at a specific altitude can be achieved by advancements in solar cell and IPMC technologies. As a path for further development of these materials the following list is in order of their benefit to the performance of the SSA.

1. Decrease the weight of the IPMC material
2. Increase the efficiency of the solar array
3. Increase the efficiency of the IPMC material
Appendix A: List of References


84. Pelletier, A. and Mueller, T.J. "Low Reynolds number aerodynamics of low-aspect-ratio,
85. Personal Correspondence with Viktor.V.Kerzhanovich, JPL, January 2000.
86. Planetary Exploration for Biomimetics: An Entomopter for Flight on Mars Exploration,
88. Pterosaur Flight, web site: http://www.nurseminerva.co.uk/adapt/pterosaur.htm, March
2005.
90. Ramamurti, R. and Sandberg, W. C., "A Three-Dimensional Computational Study of the
92. Rayner, J. "Modelling Flapping Flight," School of Biology, University of Leeds, web site,
Gel Polymer Battery Power Supply System for an Unmanned Aerial Vehicle," NASA TM-
Aerial Vehicles," Presented at the Intersociety Energy Conversion Engineering Conference,
96. Sadeghipour, K., Salomon, R. and Neogi, S., (1992), Development of A Novel Electro-
chemically Active Membrane and Smart Material based Vibration Sensor/Damper, Smart
Materials and Structures Journal, Institute of Physics Publication, Philadelphia, Pa., vol. 1,
pp. 172-179
98. Schofield,J.T., Barnes,J.R., Crisp,D., Haberle, R.M., Larsen,S.,Magalhaes, J.A., Murphy,
J.R., Seiff, A., Wilson,G.,"The Mars Pathfinder Atmospheric Structuer Investigation/Meteo-
rology (ASI/MET) Experiment.
Application of Electrically Controlled Polymeric Gels, Smart Materials and Structures,
Institute of Physics Publication, Philadelphia, Pa., vol. 1, pp.95-100
ment Simulation of the 2D Collapse of A Polyelectrolyte Gel Disk, Proc.(1993) SPIE North
American Conference on Smart Structures and Materials, February 93, Albuquerque, NM,
vol. 1916, pp. 14-22
Controlled Polymeric Muscles As Active Materials in Adaptive Structures, Int. J. Smart
Materials & Structures, Institute of Physics Publication, Philadelphia, Pa., vol.1, no.1,
pp.44-54


Appendix B: Presentations, Media Exposure, and Future Development Interest

Presentations and Media Exposure

• ABCNEWS.com

• [PDF] Solid State Aircraft Concept Overview
  Jenkins Ohio Aerospace Institute Phil.jenkins@grc.nasa.gov Curtis Smith Ohio Aerospace Institute curtissmith@oai.org Dr. Kakkattukuzhy Isaac University of ... doi.ieeecomputersociety.org/10.1109/EH.2004.49 - Similar pages

• e4engineering.com - Engineering news, engineering information and ... Leading the research, Kakkattukuzhy Isaac, professor of mechanical and aerospace engineering at the university, said the research will be refined over the ... www.e4engineering.com/story.aspx?uid=44d5fe8d-a1b1-4376-abf0-439267403d8f&type=news - 31k - Cached - Similar pages


• [PDF] Newsletter
  Page 1. WW-EAP Newsletter, Vol. 6, No. 1, June 2004 1 FROM THE EDITOR Yoseph Bar-Cohen, yosi@jpl.nasa.gov It is my pleasure to report ... ndeaa.jpl.nasa.gov/nasa-nde/newsltr/WW-EAP_Newsletter6-1.pdf - Similar pages

• Interview with Discover Channel Canada / Science Channel US. Segment will be shown on "Discoveries this Week" in September 2005.

• NIAC 2004 Annual Meeting presentation on the status of the Solid State Aircraft Project is hosted on the AICS Research Inc. Website: http://www.aics-research.com/lectures/niac04/
Solid State Aircraft


- Interview with the Futures Channel, to be released for educational programming Summer 2005. http://www.thefutureschannel.com/


- A design firm in London, AUKETT TYHERLEIGH, requested images from the SSA video for their Design Prima exhibition. The SSA will be put in the Products for Prophecy exhibition. This Products of Prophecy feature is the collective thought of more than 35 design visionaries, (including Jasper Morrison, Ross Lovegrove, John Pawson, and Ron Arad) forecasting what objects or images of the late 20th century might be most likely to be prophetic indicators of cultural ambition and aspiration in the 21st century. In effect, a bit of fortune telling. The SSA will also be featured in a special on Discovery Channel Canada.

Future Development Interest

- Interest in the SSA program has been expressed by the Technology Assessment Office at Wright-Patterson Airforce Base. OAI forwarded the animated video and will plan on a team visit soon after the conclusion of the program. The Pentagon also has expressed an interest in the vehicle and additional information is pending. The Solid State team is going to pursue additional development resources that include DARPA, the Army Research Lab (ARL), the Pentagon, and other funding sources as the opportunity presents itself.
## Appendix C: Planetary Atmosphere Data

### Mean Standard Atmosphere for Venus (JPL Model)

<table>
<thead>
<tr>
<th>H (Km)</th>
<th>T (K)</th>
<th>P (bar)</th>
<th>( \rho ) (kg/m³)</th>
<th>( U ) (m/s)</th>
<th>( \mu ) (Pa·s)</th>
<th>( ν ) (s/m²)</th>
<th>( C_p ) (J/kg·K)</th>
<th>( C_p/C_v )</th>
<th>A (m/s)</th>
<th>K (W/m²·K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>735.3</td>
<td>92.1</td>
<td>64.79</td>
<td>0.6</td>
<td>3.35E-05</td>
<td>5.17E-07</td>
<td>1181</td>
<td>1.193</td>
<td>410</td>
<td>0.0588</td>
</tr>
<tr>
<td>1</td>
<td>727.7</td>
<td>86.45</td>
<td>61.56</td>
<td>0.7</td>
<td>3.12E-05</td>
<td>5.07E-07</td>
<td>1177</td>
<td>1.194</td>
<td>408</td>
<td>0.0575</td>
</tr>
<tr>
<td>2</td>
<td>720.2</td>
<td>81.09</td>
<td>58.45</td>
<td>0.8</td>
<td>2.89E-05</td>
<td>4.95E-07</td>
<td>1172</td>
<td>1.195</td>
<td>406</td>
<td>0.0561</td>
</tr>
<tr>
<td>3</td>
<td>712.4</td>
<td>76.01</td>
<td>55.47</td>
<td>0.9</td>
<td>2.67E-05</td>
<td>4.81E-07</td>
<td>1168</td>
<td>1.196</td>
<td>404</td>
<td>0.0548</td>
</tr>
<tr>
<td>4</td>
<td>704.6</td>
<td>71.2</td>
<td>52.62</td>
<td>1.0</td>
<td>2.44E-05</td>
<td>4.63E-07</td>
<td>1163</td>
<td>1.197</td>
<td>402</td>
<td>0.0534</td>
</tr>
<tr>
<td>5</td>
<td>696.8</td>
<td>66.65</td>
<td>49.87</td>
<td>1.2</td>
<td>2.21E-05</td>
<td>4.43E-07</td>
<td>1159</td>
<td>1.198</td>
<td>400</td>
<td>0.0521</td>
</tr>
<tr>
<td>6</td>
<td>688.8</td>
<td>62.35</td>
<td>47.24</td>
<td>1.3</td>
<td>2.18E-05</td>
<td>4.62E-07</td>
<td>1155</td>
<td>1.199</td>
<td>398</td>
<td>0.0511</td>
</tr>
<tr>
<td>7</td>
<td>681.1</td>
<td>58.28</td>
<td>44.71</td>
<td>1.9</td>
<td>2.16E-05</td>
<td>4.83E-07</td>
<td>1151</td>
<td>1.200</td>
<td>396</td>
<td>0.0501</td>
</tr>
<tr>
<td>8</td>
<td>673.6</td>
<td>54.44</td>
<td>42.26</td>
<td>2.4</td>
<td>2.13E-05</td>
<td>5.04E-07</td>
<td>1146</td>
<td>1.200</td>
<td>393</td>
<td>0.0490</td>
</tr>
<tr>
<td>9</td>
<td>665.8</td>
<td>50.81</td>
<td>39.95</td>
<td>3.4</td>
<td>2.11E-05</td>
<td>5.27E-07</td>
<td>1142</td>
<td>1.201</td>
<td>391</td>
<td>0.0480</td>
</tr>
<tr>
<td>10</td>
<td>658.2</td>
<td>47.39</td>
<td>37.72</td>
<td>4.5</td>
<td>2.08E-05</td>
<td>5.51E-07</td>
<td>1138</td>
<td>1.202</td>
<td>389</td>
<td>0.0470</td>
</tr>
<tr>
<td>11</td>
<td>650.6</td>
<td>44.16</td>
<td>35.58</td>
<td>6.3</td>
<td>2.25E-05</td>
<td>6.33E-07</td>
<td>1134</td>
<td>1.203</td>
<td>387</td>
<td>0.0462</td>
</tr>
<tr>
<td>12</td>
<td>643.2</td>
<td>41.12</td>
<td>33.54</td>
<td>8.2</td>
<td>2.42E-05</td>
<td>7.23E-07</td>
<td>1129</td>
<td>1.204</td>
<td>385</td>
<td>0.0455</td>
</tr>
<tr>
<td>13</td>
<td>635.5</td>
<td>38.26</td>
<td>31.6</td>
<td>10.8</td>
<td>2.60E-05</td>
<td>8.22E-07</td>
<td>1125</td>
<td>1.205</td>
<td>383</td>
<td>0.0447</td>
</tr>
<tr>
<td>14</td>
<td>628.1</td>
<td>35.57</td>
<td>29.74</td>
<td>13.4</td>
<td>2.77E-05</td>
<td>9.31E-07</td>
<td>1120</td>
<td>1.206</td>
<td>381</td>
<td>0.0440</td>
</tr>
<tr>
<td>15</td>
<td>620.8</td>
<td>33.04</td>
<td>27.95</td>
<td>16.1</td>
<td>2.94E-05</td>
<td>1.05E-06</td>
<td>1116</td>
<td>1.207</td>
<td>379</td>
<td>0.0432</td>
</tr>
<tr>
<td>16</td>
<td>613.3</td>
<td>30.66</td>
<td>26.27</td>
<td>19.6</td>
<td>2.91E-05</td>
<td>1.11E-06</td>
<td>1111</td>
<td>1.208</td>
<td>377</td>
<td>0.0425</td>
</tr>
<tr>
<td>17</td>
<td>605.2</td>
<td>28.43</td>
<td>24.68</td>
<td>22.0</td>
<td>2.88E-05</td>
<td>1.17E-06</td>
<td>1106</td>
<td>1.209</td>
<td>374</td>
<td>0.0417</td>
</tr>
<tr>
<td>18</td>
<td>597.1</td>
<td>26.33</td>
<td>23.18</td>
<td>24.5</td>
<td>2.84E-05</td>
<td>1.23E-06</td>
<td>1101</td>
<td>1.211</td>
<td>372</td>
<td>0.0410</td>
</tr>
<tr>
<td>19</td>
<td>589.3</td>
<td>24.36</td>
<td>21.74</td>
<td>26.0</td>
<td>2.81E-05</td>
<td>1.29E-06</td>
<td>1096</td>
<td>1.212</td>
<td>369</td>
<td>0.0402</td>
</tr>
<tr>
<td>20</td>
<td>580.7</td>
<td>22.52</td>
<td>20.39</td>
<td>27.6</td>
<td>2.78E-05</td>
<td>1.36E-06</td>
<td>1091</td>
<td>1.213</td>
<td>367</td>
<td>0.0395</td>
</tr>
<tr>
<td>21</td>
<td>572.4</td>
<td>20.79</td>
<td>19.11</td>
<td>28.8</td>
<td>2.76E-05</td>
<td>1.44E-06</td>
<td>1085</td>
<td>1.214</td>
<td>365</td>
<td>0.0388</td>
</tr>
<tr>
<td>22</td>
<td>564.3</td>
<td>19.17</td>
<td>17.88</td>
<td>29.9</td>
<td>2.74E-05</td>
<td>1.53E-06</td>
<td>1079</td>
<td>1.216</td>
<td>362</td>
<td>0.0382</td>
</tr>
<tr>
<td>23</td>
<td>556.0</td>
<td>17.66</td>
<td>16.17</td>
<td>30.6</td>
<td>2.71E-05</td>
<td>1.62E-06</td>
<td>1074</td>
<td>1.217</td>
<td>360</td>
<td>0.0375</td>
</tr>
<tr>
<td>24</td>
<td>547.5</td>
<td>16.25</td>
<td>15.62</td>
<td>31.3</td>
<td>2.69E-05</td>
<td>1.72E-06</td>
<td>1068</td>
<td>1.219</td>
<td>357</td>
<td>0.0369</td>
</tr>
<tr>
<td>25</td>
<td>539.2</td>
<td>14.93</td>
<td>14.57</td>
<td>32.3</td>
<td>2.67E-05</td>
<td>1.83E-06</td>
<td>1062</td>
<td>1.220</td>
<td>355</td>
<td>0.0362</td>
</tr>
<tr>
<td>26</td>
<td>530.7</td>
<td>13.7</td>
<td>13.59</td>
<td>33.3</td>
<td>2.64E-05</td>
<td>1.94E-06</td>
<td>1056</td>
<td>1.222</td>
<td>352</td>
<td>0.0357</td>
</tr>
</tbody>
</table>
## Solid State Aircraft

<table>
<thead>
<tr>
<th>H Km</th>
<th>T K</th>
<th>P (bar)</th>
<th>ρ kg/m³</th>
<th>U m/s</th>
<th>µ Pa•s</th>
<th>ν s/m²</th>
<th>Cp J/kg•K</th>
<th>Cp/Cv</th>
<th>A m/s</th>
<th>K W/m•K</th>
</tr>
</thead>
<tbody>
<tr>
<td>27</td>
<td>522.3</td>
<td>12.56</td>
<td>12.65</td>
<td>34.0</td>
<td>2.61E-05</td>
<td>2.06E-06</td>
<td>1049</td>
<td>1.223</td>
<td>350</td>
<td>0.0352</td>
</tr>
<tr>
<td>28</td>
<td>513.8</td>
<td>11.49</td>
<td>11.77</td>
<td>34.6</td>
<td>2.58E-05</td>
<td>2.19E-06</td>
<td>1043</td>
<td>1.225</td>
<td>347</td>
<td>0.0346</td>
</tr>
<tr>
<td>29</td>
<td>505.6</td>
<td>10.5</td>
<td>10.93</td>
<td>35.0</td>
<td>2.55E-05</td>
<td>2.33E-06</td>
<td>1036</td>
<td>1.226</td>
<td>345</td>
<td>0.0341</td>
</tr>
<tr>
<td>30</td>
<td>496.9</td>
<td>9.85</td>
<td>10.15</td>
<td>35.5</td>
<td>2.52E-05</td>
<td>2.48E-06</td>
<td>1030</td>
<td>1.228</td>
<td>342</td>
<td>0.0336</td>
</tr>
<tr>
<td>31</td>
<td>488.3</td>
<td>8.729</td>
<td>9.406</td>
<td>36.0</td>
<td>2.49E-05</td>
<td>2.65E-06</td>
<td>1023</td>
<td>1.230</td>
<td>339</td>
<td>0.0331</td>
</tr>
<tr>
<td>32</td>
<td>479.9</td>
<td>7.94</td>
<td>8.704</td>
<td>36.4</td>
<td>2.46E-05</td>
<td>2.83E-06</td>
<td>1016</td>
<td>1.232</td>
<td>337</td>
<td>0.0325</td>
</tr>
<tr>
<td>33</td>
<td>472.7</td>
<td>7.211</td>
<td>8.041</td>
<td>36.7</td>
<td>2.44E-05</td>
<td>3.03E-06</td>
<td>1010</td>
<td>1.234</td>
<td>334</td>
<td>0.0320</td>
</tr>
<tr>
<td>34</td>
<td>463.4</td>
<td>6.537</td>
<td>7.42</td>
<td>36.9</td>
<td>2.41E-05</td>
<td>3.25E-06</td>
<td>1003</td>
<td>1.236</td>
<td>332</td>
<td>0.0314</td>
</tr>
<tr>
<td>35</td>
<td>455.5</td>
<td>5.917</td>
<td>6.831</td>
<td>37.3</td>
<td>2.38E-05</td>
<td>3.48E-06</td>
<td>996</td>
<td>1.238</td>
<td>329</td>
<td>0.0309</td>
</tr>
<tr>
<td>36</td>
<td>448.0</td>
<td>5.346</td>
<td>6.274</td>
<td>37.6</td>
<td>2.35E-05</td>
<td>3.75E-06</td>
<td>990</td>
<td>1.240</td>
<td>326</td>
<td>0.0304</td>
</tr>
<tr>
<td>37</td>
<td>439.9</td>
<td>4.822</td>
<td>5.762</td>
<td>38.2</td>
<td>2.33E-05</td>
<td>4.04E-06</td>
<td>983</td>
<td>1.242</td>
<td>324</td>
<td>0.0300</td>
</tr>
<tr>
<td>38</td>
<td>432.5</td>
<td>4.342</td>
<td>5.276</td>
<td>38.7</td>
<td>2.30E-05</td>
<td>4.36E-06</td>
<td>977</td>
<td>1.244</td>
<td>321</td>
<td>0.0295</td>
</tr>
<tr>
<td>39</td>
<td>425.1</td>
<td>3.903</td>
<td>4.823</td>
<td>39.7</td>
<td>2.28E-05</td>
<td>4.72E-06</td>
<td>970</td>
<td>1.246</td>
<td>319</td>
<td>0.0291</td>
</tr>
<tr>
<td>40</td>
<td>417.6</td>
<td>3.501</td>
<td>4.404</td>
<td>40.7</td>
<td>2.25E-05</td>
<td>5.11E-06</td>
<td>964</td>
<td>1.248</td>
<td>316</td>
<td>0.0286</td>
</tr>
<tr>
<td>41</td>
<td>410.0</td>
<td>3.135</td>
<td>4.015</td>
<td>42.6</td>
<td>2.23E-05</td>
<td>5.55E-06</td>
<td>958</td>
<td>1.250</td>
<td>314</td>
<td>0.0282</td>
</tr>
<tr>
<td>42</td>
<td>403.5</td>
<td>2.802</td>
<td>3.646</td>
<td>44.5</td>
<td>2.21E-05</td>
<td>6.05E-06</td>
<td>953</td>
<td>1.252</td>
<td>311</td>
<td>0.0277</td>
</tr>
<tr>
<td>43</td>
<td>397.1</td>
<td>2.499</td>
<td>3.303</td>
<td>47.4</td>
<td>2.18E-05</td>
<td>6.61E-06</td>
<td>947</td>
<td>1.253</td>
<td>309</td>
<td>0.0273</td>
</tr>
<tr>
<td>44</td>
<td>391.2</td>
<td>2.226</td>
<td>2.985</td>
<td>50.3</td>
<td>2.16E-05</td>
<td>7.24E-06</td>
<td>942</td>
<td>1.255</td>
<td>306</td>
<td>0.0268</td>
</tr>
<tr>
<td>45</td>
<td>385.4</td>
<td>1.979</td>
<td>2.693</td>
<td>54.2</td>
<td>2.14E-05</td>
<td>7.95E-06</td>
<td>936</td>
<td>1.257</td>
<td>304</td>
<td>0.0264</td>
</tr>
<tr>
<td>46</td>
<td>379.7</td>
<td>1.756</td>
<td>2.426</td>
<td>57.4</td>
<td>2.11E-05</td>
<td>8.71E-06</td>
<td>930</td>
<td>1.259</td>
<td>302</td>
<td>0.0260</td>
</tr>
<tr>
<td>47</td>
<td>373.1</td>
<td>1.556</td>
<td>2.186</td>
<td>59.4</td>
<td>2.09E-05</td>
<td>9.55E-06</td>
<td>923</td>
<td>1.261</td>
<td>299</td>
<td>0.0256</td>
</tr>
<tr>
<td>48</td>
<td>366.4</td>
<td>1.375</td>
<td>1.967</td>
<td>61.0</td>
<td>2.06E-05</td>
<td>1.05E-05</td>
<td>917</td>
<td>1.264</td>
<td>297</td>
<td>0.0251</td>
</tr>
<tr>
<td>49</td>
<td>358.6</td>
<td>1.213</td>
<td>1.769</td>
<td>61.2</td>
<td>2.04E-05</td>
<td>1.15E-05</td>
<td>910</td>
<td>1.266</td>
<td>294</td>
<td>0.0247</td>
</tr>
<tr>
<td>50</td>
<td>350.5</td>
<td>1.066</td>
<td>1.594</td>
<td>60.9</td>
<td>2.01E-05</td>
<td>1.26E-05</td>
<td>904</td>
<td>1.268</td>
<td>292</td>
<td>0.0243</td>
</tr>
<tr>
<td>51</td>
<td>342.0</td>
<td>0.9347</td>
<td>1.432</td>
<td>60.2</td>
<td>1.97E-05</td>
<td>1.38E-05</td>
<td>895</td>
<td>1.272</td>
<td>288</td>
<td>0.0239</td>
</tr>
<tr>
<td>52</td>
<td>333.3</td>
<td>0.8167</td>
<td>1.284</td>
<td>59.4</td>
<td>1.94E-05</td>
<td>1.51E-05</td>
<td>886</td>
<td>1.276</td>
<td>284</td>
<td>0.0235</td>
</tr>
<tr>
<td>53</td>
<td>323.0</td>
<td>0.7109</td>
<td>1.153</td>
<td>59.3</td>
<td>1.90E-05</td>
<td>1.65E-05</td>
<td>877</td>
<td>1.279</td>
<td>281</td>
<td>0.0231</td>
</tr>
<tr>
<td>54</td>
<td>312.8</td>
<td>0.616</td>
<td>1.032</td>
<td>59.2</td>
<td>1.87E-05</td>
<td>1.81E-05</td>
<td>868</td>
<td>1.283</td>
<td>277</td>
<td>0.0227</td>
</tr>
<tr>
<td>55</td>
<td>302.3</td>
<td>0.5314</td>
<td>0.9207</td>
<td>59.9</td>
<td>1.83E-05</td>
<td>1.99E-05</td>
<td>859</td>
<td>1.287</td>
<td>273</td>
<td>0.0223</td>
</tr>
<tr>
<td>56</td>
<td>291.8</td>
<td>0.4559</td>
<td>0.8183</td>
<td>60.5</td>
<td>1.80E-05</td>
<td>2.20E-05</td>
<td>851</td>
<td>1.290</td>
<td>270</td>
<td>0.0219</td>
</tr>
</tbody>
</table>
### Appendix C: Planetary Atmosphere Data

<table>
<thead>
<tr>
<th>H Km</th>
<th>T K</th>
<th>P (bar)</th>
<th>$\rho$ kg/m$^3$</th>
<th>U m/s</th>
<th>$\mu$ Pa·s</th>
<th>$\nu$ m$^2$/s</th>
<th>Cp J/kg·K</th>
<th>Cp/Cv</th>
<th>A m/s</th>
<th>K W/m·K</th>
</tr>
</thead>
<tbody>
<tr>
<td>57</td>
<td>282.5</td>
<td>0.3891</td>
<td>0.7212</td>
<td>62.7</td>
<td>1.77E-05</td>
<td>2.45E-05</td>
<td>844</td>
<td>1.294</td>
<td>266</td>
<td>0.0215</td>
</tr>
<tr>
<td>58</td>
<td>275.2</td>
<td>0.3306</td>
<td>0.6289</td>
<td>65.0</td>
<td>1.73E-05</td>
<td>2.76E-05</td>
<td>836</td>
<td>1.297</td>
<td>263</td>
<td>0.0212</td>
</tr>
<tr>
<td>59</td>
<td>268.7</td>
<td>0.2796</td>
<td>0.5448</td>
<td>71.1</td>
<td>1.70E-05</td>
<td>3.12E-05</td>
<td>829</td>
<td>1.301</td>
<td>259</td>
<td>0.0208</td>
</tr>
<tr>
<td>60</td>
<td>262.8</td>
<td>0.2357</td>
<td>0.4694</td>
<td>77.2</td>
<td>1.67E-05</td>
<td>3.56E-05</td>
<td>821</td>
<td>1.304</td>
<td>256</td>
<td>0.0204</td>
</tr>
<tr>
<td>61</td>
<td>258.7</td>
<td>0.2008</td>
<td>0.40525</td>
<td>85.4</td>
<td>1.66E-05</td>
<td>4.09E-05</td>
<td>818</td>
<td>1.306</td>
<td>355</td>
<td>0.0201</td>
</tr>
<tr>
<td>62</td>
<td>254.5</td>
<td>0.1659</td>
<td>0.3411</td>
<td>92.0</td>
<td>1.64E-05</td>
<td>4.82E-05</td>
<td>815</td>
<td>1.307</td>
<td>253</td>
<td>0.0197</td>
</tr>
<tr>
<td>63</td>
<td>250.0</td>
<td>0.14075</td>
<td>0.2927</td>
<td>94.0</td>
<td>1.63E-05</td>
<td>5.57E-05</td>
<td>811</td>
<td>1.309</td>
<td>252</td>
<td>0.0194</td>
</tr>
<tr>
<td>64</td>
<td>245.4</td>
<td>0.1156</td>
<td>0.2443</td>
<td>94.5</td>
<td>1.62E-05</td>
<td>6.62E-05</td>
<td>808</td>
<td>1.310</td>
<td>250</td>
<td>0.0190</td>
</tr>
<tr>
<td>65</td>
<td>243.2</td>
<td>0.09765</td>
<td>0.2086</td>
<td>95.0</td>
<td>1.61E-05</td>
<td>7.69E-05</td>
<td>805</td>
<td>1.312</td>
<td>249</td>
<td>0.0187</td>
</tr>
<tr>
<td>66</td>
<td>241.0</td>
<td>0.0797</td>
<td>0.1729</td>
<td>94.4</td>
<td>1.59E-05</td>
<td>9.21E-05</td>
<td>802</td>
<td>1.314</td>
<td>247</td>
<td>0.0184</td>
</tr>
<tr>
<td>67</td>
<td>238.2</td>
<td>0.06709</td>
<td>0.14695</td>
<td>93.8</td>
<td>1.58E-05</td>
<td>1.07E-04</td>
<td>799</td>
<td>1.315</td>
<td>246</td>
<td>0.0180</td>
</tr>
<tr>
<td>68</td>
<td>235.4</td>
<td>0.05447</td>
<td>0.121</td>
<td>93.2</td>
<td>1.57E-05</td>
<td>1.29E-04</td>
<td>795</td>
<td>1.317</td>
<td>244</td>
<td>0.0177</td>
</tr>
<tr>
<td>69</td>
<td>232.6</td>
<td>0.04569</td>
<td>0.102465</td>
<td>92.6</td>
<td>1.55E-05</td>
<td>1.52E-04</td>
<td>792</td>
<td>1.318</td>
<td>243</td>
<td>0.0173</td>
</tr>
<tr>
<td>70</td>
<td>229.8</td>
<td>0.0369</td>
<td>0.08393</td>
<td>92.0</td>
<td>1.54E-05</td>
<td>1.83E-04</td>
<td>789</td>
<td>1.320</td>
<td>241</td>
<td>0.0170</td>
</tr>
<tr>
<td>71</td>
<td>227.0</td>
<td>0.03083</td>
<td>0.07084</td>
<td>89.4</td>
<td>1.53E-05</td>
<td>2.15E-04</td>
<td>786</td>
<td>1.322</td>
<td>239</td>
<td>0.0167</td>
</tr>
<tr>
<td>72</td>
<td>224.1</td>
<td>0.02476</td>
<td>0.05775</td>
<td>86.8</td>
<td>1.52E-05</td>
<td>2.61E-04</td>
<td>783</td>
<td>1.324</td>
<td>238</td>
<td>0.0164</td>
</tr>
<tr>
<td>73</td>
<td>221.4</td>
<td>0.02061</td>
<td>0.04854</td>
<td>84.2</td>
<td>1.50E-05</td>
<td>3.08E-04</td>
<td>779</td>
<td>1.325</td>
<td>236</td>
<td>0.0161</td>
</tr>
<tr>
<td>74</td>
<td>218.6</td>
<td>0.01645</td>
<td>0.03933</td>
<td>81.6</td>
<td>1.48E-05</td>
<td>3.76E-04</td>
<td>776</td>
<td>1.327</td>
<td>235</td>
<td>0.0158</td>
</tr>
<tr>
<td>75</td>
<td>215.4</td>
<td>0.01363</td>
<td>0.03298</td>
<td>79.0</td>
<td>1.47E-05</td>
<td>4.44E-04</td>
<td>773</td>
<td>1.329</td>
<td>233</td>
<td>0.0155</td>
</tr>
<tr>
<td>76</td>
<td>212.1</td>
<td>0.01081</td>
<td>0.02663</td>
<td>74.6</td>
<td>1.45E-05</td>
<td>5.44E-04</td>
<td>770</td>
<td>1.331</td>
<td>231</td>
<td>0.0151</td>
</tr>
<tr>
<td>77</td>
<td>208.7</td>
<td>0.00891</td>
<td>0.022235</td>
<td>70.2</td>
<td>1.44E-05</td>
<td>6.45E-04</td>
<td>767</td>
<td>1.333</td>
<td>230</td>
<td>0.0148</td>
</tr>
<tr>
<td>78</td>
<td>205.3</td>
<td>0.00701</td>
<td>0.01784</td>
<td>65.8</td>
<td>1.43E-05</td>
<td>7.96E-04</td>
<td>763</td>
<td>1.334</td>
<td>228</td>
<td>0.0145</td>
</tr>
<tr>
<td>79</td>
<td>201.2</td>
<td>0.00589</td>
<td>0.01485</td>
<td>61.4</td>
<td>1.41E-05</td>
<td>9.46E-04</td>
<td>760</td>
<td>1.336</td>
<td>227</td>
<td>0.0142</td>
</tr>
<tr>
<td>80</td>
<td>197.1</td>
<td>0.00476</td>
<td>0.01186</td>
<td>57.0</td>
<td>1.39E-05</td>
<td>1.17E-03</td>
<td>757</td>
<td>1.338</td>
<td>225</td>
<td>0.0139</td>
</tr>
<tr>
<td>81</td>
<td>193.5</td>
<td>0.00378</td>
<td>0.009793</td>
<td>52.4</td>
<td>1.38E-05</td>
<td>1.41E-03</td>
<td>755</td>
<td>1.340</td>
<td>223</td>
<td>0.0138</td>
</tr>
<tr>
<td>82</td>
<td>189.9</td>
<td>0.00281</td>
<td>0.007725</td>
<td>47.8</td>
<td>1.36E-05</td>
<td>1.77E-03</td>
<td>752</td>
<td>1.341</td>
<td>222</td>
<td>0.0136</td>
</tr>
<tr>
<td>83</td>
<td>186.9</td>
<td>0.00227</td>
<td>0.006326</td>
<td>43.2</td>
<td>1.35E-05</td>
<td>2.14E-03</td>
<td>750</td>
<td>1.343</td>
<td>220</td>
<td>0.0135</td>
</tr>
<tr>
<td>84</td>
<td>183.8</td>
<td>0.00173</td>
<td>0.004926</td>
<td>38.6</td>
<td>1.34E-05</td>
<td>2.72E-03</td>
<td>747</td>
<td>1.344</td>
<td>219</td>
<td>0.0133</td>
</tr>
<tr>
<td>85</td>
<td>181.0</td>
<td>0.00139</td>
<td>0.004007</td>
<td>34.0</td>
<td>1.33E-05</td>
<td>3.31E-03</td>
<td>745</td>
<td>1.346</td>
<td>217</td>
<td>0.0132</td>
</tr>
<tr>
<td>86</td>
<td>178.2</td>
<td>0.00105</td>
<td>0.003088</td>
<td>30.4</td>
<td>1.31E-05</td>
<td>4.25E-03</td>
<td>743</td>
<td>1.347</td>
<td>215</td>
<td>0.0131</td>
</tr>
<tr>
<td>H Km</td>
<td>T K</td>
<td>P (bar)</td>
<td>ρ kg/m^3</td>
<td>U m/s</td>
<td>μ Pa*s</td>
<td>ν m^2</td>
<td>Cp J/kg*K</td>
<td>Cp/Cv</td>
<td>A m/s</td>
<td>K W/m*K</td>
</tr>
<tr>
<td>------</td>
<td>------</td>
<td>--------</td>
<td>----------</td>
<td>--------</td>
<td>--------</td>
<td>-------</td>
<td>-----------</td>
<td>-------</td>
<td>-------</td>
<td>---------</td>
</tr>
<tr>
<td>87</td>
<td>175.9</td>
<td>0.00084</td>
<td>0.002493</td>
<td>26.8</td>
<td>1.30E-05</td>
<td>5.21E-03</td>
<td>740</td>
<td>1.349</td>
<td>214</td>
<td>0.0129</td>
</tr>
<tr>
<td>88</td>
<td>173.6</td>
<td>0.00063</td>
<td>0.001898</td>
<td>23.2</td>
<td>1.29E-05</td>
<td>6.78E-03</td>
<td>738</td>
<td>1.350</td>
<td>212</td>
<td>0.0128</td>
</tr>
<tr>
<td>89</td>
<td>171.5</td>
<td>0.0005</td>
<td>0.001525</td>
<td>19.6</td>
<td>1.27E-05</td>
<td>8.35E-03</td>
<td>735</td>
<td>1.352</td>
<td>211</td>
<td>0.0126</td>
</tr>
<tr>
<td>90</td>
<td>169.4</td>
<td>0.00037</td>
<td>0.001151</td>
<td>16.0</td>
<td>1.26E-05</td>
<td>1.09E-02</td>
<td>733</td>
<td>1.352</td>
<td>209</td>
<td>0.0125</td>
</tr>
<tr>
<td>91</td>
<td>168.3</td>
<td>0.0003</td>
<td>0.000917</td>
<td>15.0</td>
<td>1.26E-05</td>
<td>1.38E-02</td>
<td>734</td>
<td>1.353</td>
<td>209</td>
<td>0.0125</td>
</tr>
<tr>
<td>92</td>
<td>167.2</td>
<td>0.00022</td>
<td>0.000684</td>
<td>14.0</td>
<td>1.27E-05</td>
<td>1.85E-02</td>
<td>734</td>
<td>1.353</td>
<td>210</td>
<td>0.0126</td>
</tr>
<tr>
<td>93</td>
<td>167.2</td>
<td>0.00017</td>
<td>0.000542</td>
<td>13.0</td>
<td>1.27E-05</td>
<td>2.34E-02</td>
<td>735</td>
<td>1.353</td>
<td>210</td>
<td>0.0126</td>
</tr>
<tr>
<td>94</td>
<td>167.2</td>
<td>0.00013</td>
<td>0.0004</td>
<td>12.0</td>
<td>1.27E-05</td>
<td>3.18E-02</td>
<td>735</td>
<td>1.353</td>
<td>211</td>
<td>0.0126</td>
</tr>
<tr>
<td>95</td>
<td>168.2</td>
<td>0.0001</td>
<td>0.000315</td>
<td>11.0</td>
<td>1.28E-05</td>
<td>4.04E-02</td>
<td>736</td>
<td>1.353</td>
<td>211</td>
<td>0.0127</td>
</tr>
<tr>
<td>96</td>
<td>169.2</td>
<td>7.5E-05</td>
<td>0.000231</td>
<td>10.8</td>
<td>1.28E-05</td>
<td>5.52E-02</td>
<td>736</td>
<td>1.352</td>
<td>211</td>
<td>0.0127</td>
</tr>
<tr>
<td>97</td>
<td>170.6</td>
<td>6E-05</td>
<td>0.000183</td>
<td>10.6</td>
<td>1.28E-05</td>
<td>7.00E-02</td>
<td>737</td>
<td>1.352</td>
<td>212</td>
<td>0.0127</td>
</tr>
<tr>
<td>98</td>
<td>172.0</td>
<td>4.5E-05</td>
<td>0.000135</td>
<td>10.4</td>
<td>1.28E-05</td>
<td>9.53E-02</td>
<td>737</td>
<td>1.352</td>
<td>212</td>
<td>0.0127</td>
</tr>
<tr>
<td>99</td>
<td>173.7</td>
<td>3.6E-05</td>
<td>0.000107</td>
<td>10.2</td>
<td>1.29E-05</td>
<td>1.21E-01</td>
<td>738</td>
<td>1.352</td>
<td>213</td>
<td>0.0128</td>
</tr>
<tr>
<td>100</td>
<td>175.4</td>
<td>2.7E-05</td>
<td>7.89E-05</td>
<td>10.0</td>
<td>1.29E-05</td>
<td>1.63E-01</td>
<td>738</td>
<td>1.352</td>
<td>213</td>
<td>0.0128</td>
</tr>
</tbody>
</table>
### Earth Standard Atmosphere

<table>
<thead>
<tr>
<th>Altitude (m)</th>
<th>Temp. °K</th>
<th>Pressure mBar</th>
<th>Density (kg/m³)</th>
<th>Speed of Sound (m/s)</th>
<th>Viscosity (kg/m s)</th>
<th>Conductivity (kcal m s °K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>288.15</td>
<td>1013.3</td>
<td>1.225</td>
<td>340.3</td>
<td>1.789E-5</td>
<td>0.6053E-5</td>
</tr>
<tr>
<td>200</td>
<td>286.85</td>
<td>989.5</td>
<td>1.202</td>
<td>339.5</td>
<td>1.78E-5</td>
<td>0.6029E-5</td>
</tr>
<tr>
<td>400</td>
<td>285.55</td>
<td>966.1</td>
<td>1.179</td>
<td>338.8</td>
<td>1.777E-5</td>
<td>0.6004E-5</td>
</tr>
<tr>
<td>600</td>
<td>284.25</td>
<td>943.2</td>
<td>1.156</td>
<td>338.0</td>
<td>1.771E-5</td>
<td>0.5980E-5</td>
</tr>
<tr>
<td>800</td>
<td>282.95</td>
<td>920.8</td>
<td>1.134</td>
<td>337.2</td>
<td>1.764E-5</td>
<td>0.5955E-5</td>
</tr>
<tr>
<td>1000</td>
<td>281.65</td>
<td>898.8</td>
<td>1.112</td>
<td>336.4</td>
<td>1.758E-5</td>
<td>0.5931E-5</td>
</tr>
<tr>
<td>1200</td>
<td>280.35</td>
<td>877.2</td>
<td>1.090</td>
<td>335.7</td>
<td>1.752E-5</td>
<td>0.5906E-5</td>
</tr>
<tr>
<td>1400</td>
<td>279.05</td>
<td>856.0</td>
<td>1.069</td>
<td>334.9</td>
<td>1.745E-5</td>
<td>0.5881E-5</td>
</tr>
<tr>
<td>1600</td>
<td>277.75</td>
<td>835.3</td>
<td>1.048</td>
<td>334.1</td>
<td>1.739E-5</td>
<td>0.5857E-5</td>
</tr>
<tr>
<td>1800</td>
<td>276.45</td>
<td>814.9</td>
<td>1.027</td>
<td>333.3</td>
<td>1.732E-5</td>
<td>0.5832E-5</td>
</tr>
<tr>
<td>2000</td>
<td>275.15</td>
<td>795.0</td>
<td>1.007</td>
<td>332.5</td>
<td>1.726E-5</td>
<td>0.5807E-5</td>
</tr>
<tr>
<td>2200</td>
<td>273.86</td>
<td>775.5</td>
<td>0.987</td>
<td>331.7</td>
<td>1.720E-5</td>
<td>0.5784E-5</td>
</tr>
<tr>
<td>2400</td>
<td>272.56</td>
<td>756.3</td>
<td>0.967</td>
<td>331.0</td>
<td>1.713E-5</td>
<td>0.5759E-5</td>
</tr>
<tr>
<td>2600</td>
<td>271.26</td>
<td>737.6</td>
<td>0.947</td>
<td>330.2</td>
<td>1.707E-5</td>
<td>0.5733E-5</td>
</tr>
<tr>
<td>2800</td>
<td>269.96</td>
<td>719.2</td>
<td>0.928</td>
<td>329.4</td>
<td>1.700E-5</td>
<td>0.5708E-5</td>
</tr>
<tr>
<td>3000</td>
<td>268.66</td>
<td>701.2</td>
<td>0.909</td>
<td>328.6</td>
<td>1.694E-5</td>
<td>0.5683E-5</td>
</tr>
<tr>
<td>3200</td>
<td>267.36</td>
<td>683.6</td>
<td>0.891</td>
<td>327.8</td>
<td>1.687E-5</td>
<td>0.5658E-5</td>
</tr>
<tr>
<td>3400</td>
<td>266.06</td>
<td>666.3</td>
<td>0.872</td>
<td>327.0</td>
<td>1.681E-5</td>
<td>0.5634E-5</td>
</tr>
<tr>
<td>3600</td>
<td>264.76</td>
<td>649.4</td>
<td>0.854</td>
<td>326.2</td>
<td>1.674E-5</td>
<td>0.5609E-5</td>
</tr>
<tr>
<td>3800</td>
<td>263.47</td>
<td>632.8</td>
<td>0.837</td>
<td>325.4</td>
<td>1.668E-5</td>
<td>0.5584E-5</td>
</tr>
<tr>
<td>4000</td>
<td>262.17</td>
<td>616.6</td>
<td>0.819</td>
<td>324.6</td>
<td>1.661E-5</td>
<td>0.5559E-5</td>
</tr>
<tr>
<td>4200</td>
<td>260.87</td>
<td>600.7</td>
<td>0.802</td>
<td>323.8</td>
<td>1.655E-5</td>
<td>0.5534E-5</td>
</tr>
<tr>
<td>4400</td>
<td>259.57</td>
<td>585.2</td>
<td>0.785</td>
<td>323.0</td>
<td>1.648E-5</td>
<td>0.5508E-5</td>
</tr>
<tr>
<td>4600</td>
<td>258.27</td>
<td>570.0</td>
<td>0.769</td>
<td>322.2</td>
<td>1.642E-5</td>
<td>0.5483E-5</td>
</tr>
<tr>
<td>4800</td>
<td>256.97</td>
<td>555.1</td>
<td>0.752</td>
<td>321.4</td>
<td>1.635E-5</td>
<td>0.5458E-5</td>
</tr>
<tr>
<td>5000</td>
<td>255.68</td>
<td>540.5</td>
<td>0.736</td>
<td>320.5</td>
<td>1.628E-5</td>
<td>0.5433E-5</td>
</tr>
<tr>
<td>5200</td>
<td>254.38</td>
<td>526.2</td>
<td>0.721</td>
<td>319.7</td>
<td>1.622E-5</td>
<td>0.5408E-5</td>
</tr>
<tr>
<td>Altitude (m)</td>
<td>Temp. °K</td>
<td>Pressure mBar</td>
<td>Density (kg/m³)</td>
<td>Speed of Sound (m/s)</td>
<td>Viscosity (kg/m s)</td>
<td>Conductivity (kcal m s °K)</td>
</tr>
<tr>
<td>-------------</td>
<td>----------</td>
<td>---------------</td>
<td>-----------------</td>
<td>----------------------</td>
<td>-------------------</td>
<td>-------------------------</td>
</tr>
<tr>
<td>5400</td>
<td>253.08</td>
<td>512.3</td>
<td>0.705</td>
<td>318.9</td>
<td>1.615E-5</td>
<td>0.5383E-5</td>
</tr>
<tr>
<td>5600</td>
<td>251.78</td>
<td>498.6</td>
<td>0.690</td>
<td>318.1</td>
<td>1.608E-5</td>
<td>0.5357E-5</td>
</tr>
<tr>
<td>5800</td>
<td>250.48</td>
<td>485.2</td>
<td>0.675</td>
<td>317.3</td>
<td>1.602E-5</td>
<td>0.5332E-5</td>
</tr>
<tr>
<td>6000</td>
<td>249.191</td>
<td>472.2</td>
<td>0.660</td>
<td>316.5</td>
<td>1.595E-5</td>
<td>0.5307E-5</td>
</tr>
<tr>
<td>6200</td>
<td>247.89</td>
<td>459.4</td>
<td>0.646</td>
<td>315.6</td>
<td>1.588E-5</td>
<td>0.5282E-5</td>
</tr>
<tr>
<td>6400</td>
<td>246.59</td>
<td>446.9</td>
<td>0.631</td>
<td>314.8</td>
<td>1.582E-5</td>
<td>0.5256E-5</td>
</tr>
<tr>
<td>6600</td>
<td>245.59</td>
<td>434.7</td>
<td>0.617</td>
<td>314.0</td>
<td>1.575E-5</td>
<td>0.5231E-5</td>
</tr>
<tr>
<td>6800</td>
<td>244.00</td>
<td>422.7</td>
<td>0.604</td>
<td>313.1</td>
<td>1.568E-5</td>
<td>0.5205E-5</td>
</tr>
<tr>
<td>7000</td>
<td>242.70</td>
<td>411.1</td>
<td>0.590</td>
<td>312.3</td>
<td>1.561E-5</td>
<td>0.5180E-5</td>
</tr>
<tr>
<td>7500</td>
<td>239.46</td>
<td>383.0</td>
<td>0.572</td>
<td>310.2</td>
<td>1.544E-5</td>
<td>0.5116E-5</td>
</tr>
<tr>
<td>8000</td>
<td>236.22</td>
<td>356.5</td>
<td>0.526</td>
<td>308.1</td>
<td>1.527E-5</td>
<td>0.5052E-5</td>
</tr>
<tr>
<td>8500</td>
<td>232.97</td>
<td>331.6</td>
<td>0.496</td>
<td>306.0</td>
<td>1.510E-5</td>
<td>0.4988E-5</td>
</tr>
<tr>
<td>9000</td>
<td>229.73</td>
<td>308.0</td>
<td>0.467</td>
<td>303.8</td>
<td>1.493E-5</td>
<td>0.4924E-5</td>
</tr>
<tr>
<td>9500</td>
<td>226.49</td>
<td>285.8</td>
<td>0.440</td>
<td>301.7</td>
<td>1.475E-5</td>
<td>0.4859E-5</td>
</tr>
<tr>
<td>10000</td>
<td>223.25</td>
<td>265.0</td>
<td>0.414</td>
<td>299.5</td>
<td>1.458E-5</td>
<td>0.4794E-5</td>
</tr>
<tr>
<td>11000</td>
<td>216.77</td>
<td>227.0</td>
<td>0.365</td>
<td>295.2</td>
<td>1.442E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>12000</td>
<td>216.65</td>
<td>194.0</td>
<td>0.312</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>13000</td>
<td>216.65</td>
<td>165.8</td>
<td>0.267</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>14000</td>
<td>216.65</td>
<td>141.7</td>
<td>0.228</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>15000</td>
<td>216.65</td>
<td>121.1</td>
<td>0.195</td>
<td>295.1</td>
<td>1.442E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>16000</td>
<td>216.65</td>
<td>103.5</td>
<td>0.166</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>17000</td>
<td>216.65</td>
<td>88.5</td>
<td>0.142</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>18000</td>
<td>216.65</td>
<td>75.7</td>
<td>0.122</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>19000</td>
<td>216.65</td>
<td>64.7</td>
<td>0.104</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>20000</td>
<td>216.65</td>
<td>55.3</td>
<td>0.0889</td>
<td>295.1</td>
<td>1.422E-5</td>
<td>0.4664E-5</td>
</tr>
<tr>
<td>22000</td>
<td>218.57</td>
<td>40.5</td>
<td>0.0645</td>
<td>296.4</td>
<td>1.433E-5</td>
<td>0.4702E-5</td>
</tr>
<tr>
<td>24000</td>
<td>220.56</td>
<td>29.7</td>
<td>0.0469</td>
<td>297.8</td>
<td>1.444E-5</td>
<td>0.4742E-5</td>
</tr>
<tr>
<td>26000</td>
<td>222.54</td>
<td>21.9</td>
<td>0.0343</td>
<td>299.1</td>
<td>1.454E-5</td>
<td>0.4782E-5</td>
</tr>
</tbody>
</table>
## Appendix C: Planetary Atmosphere Data

<table>
<thead>
<tr>
<th>Altitude (m)</th>
<th>Temp. °K</th>
<th>Pressure mBar</th>
<th>Density (kg/m³)</th>
<th>Speed of Sound (m/s)</th>
<th>Viscosity (kg/m s)</th>
<th>Conductivity (kcal m s °K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>28000</td>
<td>224.53</td>
<td>16.2</td>
<td>0.0251</td>
<td>300.4</td>
<td>1.465E-5</td>
<td>0.4820E-5</td>
</tr>
<tr>
<td>30000</td>
<td>226.51</td>
<td>12.0</td>
<td>0.0184</td>
<td>301.7</td>
<td>1475E-5</td>
<td>0.4859E-5</td>
</tr>
<tr>
<td>35000</td>
<td>236.51</td>
<td>5.75</td>
<td>0.0085</td>
<td>308.3</td>
<td>1.529E-5</td>
<td>0.5058E-5</td>
</tr>
<tr>
<td>40000</td>
<td>250.35</td>
<td>2.87</td>
<td>0.0040</td>
<td>317.2</td>
<td>1.601E-5</td>
<td>0.5330E-5</td>
</tr>
<tr>
<td>50000</td>
<td>270.65</td>
<td>.798</td>
<td>0.0010</td>
<td>329.8</td>
<td>1.704E-5</td>
<td>0.5721E-2</td>
</tr>
<tr>
<td>100000</td>
<td>210.02</td>
<td>3e-4</td>
<td>5e-7</td>
<td>---</td>
<td>---</td>
<td>---</td>
</tr>
</tbody>
</table>
Solid State Aircraft

**JPL Reference Mars Atmosphere for -20° Latitude**

Mars Atmosphere Model

\[ \text{coxZ} = 0.7 \]

\[ \text{Lat} = -20 \quad Z, \text{deg} \quad 41.9298101 \]

<table>
<thead>
<tr>
<th>H, km</th>
<th>T, K</th>
<th>P, Pa</th>
<th>( p, \text{g/m}^3 )</th>
<th>( \mu, \text{Pa}\text{.s} )</th>
<th>( v, \text{m}^2/\text{s} )</th>
<th>( 1/v )</th>
</tr>
</thead>
<tbody>
<tr>
<td>9.8750</td>
<td>205</td>
<td>273.6</td>
<td>6.968</td>
<td>1.04E-05</td>
<td>0.00150</td>
<td>667</td>
</tr>
<tr>
<td>9.6250</td>
<td>206</td>
<td>280.2</td>
<td>7.100</td>
<td>1.05E-05</td>
<td>0.00148</td>
<td>677</td>
</tr>
<tr>
<td>9.3750</td>
<td>207</td>
<td>286.8</td>
<td>7.234</td>
<td>1.05E-05</td>
<td>0.00146</td>
<td>687</td>
</tr>
<tr>
<td>9.1250</td>
<td>208</td>
<td>293.6</td>
<td>7.369</td>
<td>1.06E-05</td>
<td>0.00144</td>
<td>696</td>
</tr>
<tr>
<td>8.8750</td>
<td>209</td>
<td>300.6</td>
<td>7.507</td>
<td>1.06E-05</td>
<td>0.00142</td>
<td>706</td>
</tr>
<tr>
<td>8.6250</td>
<td>209</td>
<td>307.6</td>
<td>7.683</td>
<td>1.06E-05</td>
<td>0.00138</td>
<td>723</td>
</tr>
<tr>
<td>8.3750</td>
<td>210</td>
<td>314.8</td>
<td>7.826</td>
<td>1.07E-05</td>
<td>0.00136</td>
<td>733</td>
</tr>
<tr>
<td>8.1250</td>
<td>211</td>
<td>322.2</td>
<td>7.970</td>
<td>1.07E-05</td>
<td>0.00135</td>
<td>743</td>
</tr>
<tr>
<td>7.8750</td>
<td>212</td>
<td>329.7</td>
<td>8.117</td>
<td>1.08E-05</td>
<td>0.00133</td>
<td>753</td>
</tr>
<tr>
<td>7.6250</td>
<td>213</td>
<td>337.3</td>
<td>8.266</td>
<td>1.08E-05</td>
<td>0.00131</td>
<td>764</td>
</tr>
<tr>
<td>7.3750</td>
<td>214</td>
<td>345.0</td>
<td>8.416</td>
<td>1.09E-05</td>
<td>0.00129</td>
<td>774</td>
</tr>
<tr>
<td>7.1250</td>
<td>215</td>
<td>352.9</td>
<td>8.569</td>
<td>1.09E-05</td>
<td>0.00127</td>
<td>785</td>
</tr>
<tr>
<td>6.8750</td>
<td>216</td>
<td>361.0</td>
<td>8.724</td>
<td>1.10E-05</td>
<td>0.00125</td>
<td>795</td>
</tr>
<tr>
<td>6.6250</td>
<td>217</td>
<td>369.2</td>
<td>8.880</td>
<td>1.10E-05</td>
<td>0.00124</td>
<td>806</td>
</tr>
<tr>
<td>6.3750</td>
<td>218</td>
<td>377.5</td>
<td>9.039</td>
<td>1.11E-05</td>
<td>0.00122</td>
<td>817</td>
</tr>
<tr>
<td>6.1250</td>
<td>218</td>
<td>386.0</td>
<td>9.243</td>
<td>1.11E-05</td>
<td>0.00120</td>
<td>835</td>
</tr>
<tr>
<td>5.8750</td>
<td>219</td>
<td>394.7</td>
<td>9.407</td>
<td>1.11E-05</td>
<td>0.00118</td>
<td>847</td>
</tr>
<tr>
<td>5.6250</td>
<td>220</td>
<td>403.5</td>
<td>9.574</td>
<td>1.12E-05</td>
<td>0.00117</td>
<td>858</td>
</tr>
<tr>
<td>5.3750</td>
<td>221</td>
<td>412.5</td>
<td>9.743</td>
<td>1.12E-05</td>
<td>0.00115</td>
<td>869</td>
</tr>
<tr>
<td>5.1250</td>
<td>222</td>
<td>421.6</td>
<td>9.914</td>
<td>1.13E-05</td>
<td>0.00114</td>
<td>881</td>
</tr>
<tr>
<td>4.8750</td>
<td>223</td>
<td>430.9</td>
<td>10.087</td>
<td>1.13E-05</td>
<td>0.00112</td>
<td>892</td>
</tr>
<tr>
<td>4.6250</td>
<td>224</td>
<td>440.4</td>
<td>10.262</td>
<td>1.14E-05</td>
<td>0.00111</td>
<td>904</td>
</tr>
<tr>
<td>4.3750</td>
<td>224</td>
<td>450.0</td>
<td>10.487</td>
<td>1.14E-05</td>
<td>0.00108</td>
<td>924</td>
</tr>
<tr>
<td>4.1250</td>
<td>225</td>
<td>459.9</td>
<td>10.669</td>
<td>1.14E-05</td>
<td>0.00107</td>
<td>936</td>
</tr>
<tr>
<td>3.8750</td>
<td>226</td>
<td>469.9</td>
<td>10.853</td>
<td>1.14E-05</td>
<td>0.00105</td>
<td>948</td>
</tr>
<tr>
<td>3.6250</td>
<td>227</td>
<td>480.0</td>
<td>11.039</td>
<td>1.15E-05</td>
<td>0.00104</td>
<td>960</td>
</tr>
</tbody>
</table>
## Appendix C: Planetary Atmosphere Data

<table>
<thead>
<tr>
<th>H, km</th>
<th>T, K</th>
<th>P, Pa</th>
<th>p, g/m³</th>
<th>µ, Pa·s</th>
<th>v, m²/s</th>
<th>1/v</th>
</tr>
</thead>
<tbody>
<tr>
<td>3.3750</td>
<td>227</td>
<td>490.4</td>
<td>11.278</td>
<td>1.15E-05</td>
<td>0.00102</td>
<td>981</td>
</tr>
<tr>
<td>3.1250</td>
<td>228</td>
<td>501.0</td>
<td>11.470</td>
<td>1.15E-05</td>
<td>0.00101</td>
<td>994</td>
</tr>
<tr>
<td>2.8750</td>
<td>228</td>
<td>511.8</td>
<td>11.717</td>
<td>1.15E-05</td>
<td>0.00099</td>
<td>1015</td>
</tr>
<tr>
<td>2.6250</td>
<td>229</td>
<td>522.8</td>
<td>11.917</td>
<td>1.16E-05</td>
<td>0.00097</td>
<td>1028</td>
</tr>
<tr>
<td>2.3750</td>
<td>229</td>
<td>534.0</td>
<td>12.172</td>
<td>1.16E-05</td>
<td>0.00095</td>
<td>1050</td>
</tr>
<tr>
<td>2.1250</td>
<td>229</td>
<td>545.4</td>
<td>12.433</td>
<td>1.16E-05</td>
<td>0.00093</td>
<td>1073</td>
</tr>
<tr>
<td>1.8750</td>
<td>229</td>
<td>557.1</td>
<td>12.699</td>
<td>1.16E-05</td>
<td>0.00091</td>
<td>1095</td>
</tr>
<tr>
<td>1.6250</td>
<td>229</td>
<td>569.0</td>
<td>12.971</td>
<td>1.16E-05</td>
<td>0.00089</td>
<td>1119</td>
</tr>
<tr>
<td>1.3750</td>
<td>228</td>
<td>581.2</td>
<td>13.308</td>
<td>1.15E-05</td>
<td>0.00087</td>
<td>1153</td>
</tr>
<tr>
<td>1.1250</td>
<td>227</td>
<td>593.8</td>
<td>13.655</td>
<td>1.15E-05</td>
<td>0.00084</td>
<td>1188</td>
</tr>
<tr>
<td>0.8750</td>
<td>226</td>
<td>606.6</td>
<td>14.012</td>
<td>1.14E-05</td>
<td>0.00082</td>
<td>1224</td>
</tr>
<tr>
<td>0.6375</td>
<td>228</td>
<td>619.1</td>
<td>14.174</td>
<td>1.15E-05</td>
<td>0.00081</td>
<td>1228</td>
</tr>
<tr>
<td>0.4500</td>
<td>230</td>
<td>629.0</td>
<td>14.276</td>
<td>1.16E-05</td>
<td>0.00082</td>
<td>1226</td>
</tr>
<tr>
<td>0.3250</td>
<td>231</td>
<td>635.7</td>
<td>14.365</td>
<td>1.17E-05</td>
<td>0.00081</td>
<td>1229</td>
</tr>
<tr>
<td>0.2375</td>
<td>232</td>
<td>640.4</td>
<td>14.408</td>
<td>1.17E-05</td>
<td>0.00081</td>
<td>1228</td>
</tr>
<tr>
<td>0.1750</td>
<td>233</td>
<td>643.7</td>
<td>14.422</td>
<td>1.18E-05</td>
<td>0.00082</td>
<td>1224</td>
</tr>
<tr>
<td>0.1300</td>
<td>234</td>
<td>646.1</td>
<td>14.414</td>
<td>1.18E-05</td>
<td>0.00082</td>
<td>1218</td>
</tr>
<tr>
<td>0.0950</td>
<td>234</td>
<td>648.0</td>
<td>14.456</td>
<td>1.18E-05</td>
<td>0.00082</td>
<td>1222</td>
</tr>
<tr>
<td>0.0675</td>
<td>235</td>
<td>649.5</td>
<td>14.427</td>
<td>1.19E-05</td>
<td>0.00082</td>
<td>1214</td>
</tr>
<tr>
<td>0.0450</td>
<td>236</td>
<td>650.7</td>
<td>14.393</td>
<td>1.19E-05</td>
<td>0.00083</td>
<td>1207</td>
</tr>
<tr>
<td>0.0275</td>
<td>237</td>
<td>651.6</td>
<td>14.353</td>
<td>1.20E-05</td>
<td>0.00083</td>
<td>1198</td>
</tr>
<tr>
<td>0.0150</td>
<td>238</td>
<td>652.3</td>
<td>14.307</td>
<td>1.20E-05</td>
<td>0.00084</td>
<td>1190</td>
</tr>
<tr>
<td>0.0066</td>
<td>239</td>
<td>652.7</td>
<td>14.257</td>
<td>1.21E-05</td>
<td>0.00085</td>
<td>1181</td>
</tr>
<tr>
<td>0.0016</td>
<td>244</td>
<td>653.0</td>
<td>13.970</td>
<td>1.23E-05</td>
<td>0.00088</td>
<td>1135</td>
</tr>
</tbody>
</table>
**Solid State Aircraft**

**General Mars Atmosphere Model (NASA Langley)**

<table>
<thead>
<tr>
<th>Altitude(ell), km</th>
<th>Altitude (surf), km</th>
<th>Density (kg/m³)</th>
<th>Pressure (N/m²)</th>
<th>Temperature (K°)</th>
<th>Speed of Sound (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.00E+00</td>
<td>-5.50E+00</td>
<td>1.44E-02</td>
<td>7.91E+02</td>
<td>2.87E+02</td>
<td>2.68E+02</td>
</tr>
<tr>
<td>1.00E+00</td>
<td>-4.50E+00</td>
<td>1.38E-02</td>
<td>7.39E+02</td>
<td>2.81E+02</td>
<td>2.65E+02</td>
</tr>
<tr>
<td>2.00E+00</td>
<td>-3.50E+00</td>
<td>1.31E-02</td>
<td>6.88E+02</td>
<td>2.74E+02</td>
<td>2.62E+02</td>
</tr>
<tr>
<td>3.00E+00</td>
<td>-2.50E+00</td>
<td>1.25E-02</td>
<td>6.40E+02</td>
<td>2.68E+02</td>
<td>2.59E+02</td>
</tr>
<tr>
<td>4.00E+00</td>
<td>-1.50E+00</td>
<td>1.19E-02</td>
<td>5.95E+02</td>
<td>2.62E+02</td>
<td>2.56E+02</td>
</tr>
<tr>
<td>5.00E+00</td>
<td>-5.00E-01</td>
<td>1.13E-02</td>
<td>5.52E+02</td>
<td>2.56E+02</td>
<td>2.53E+02</td>
</tr>
<tr>
<td>6.00E+00</td>
<td>5.00E-01</td>
<td>1.07E-02</td>
<td>5.11E+02</td>
<td>2.49E+02</td>
<td>2.50E+02</td>
</tr>
<tr>
<td>7.00E+00</td>
<td>1.50E+00</td>
<td>1.01E-02</td>
<td>4.72E+02</td>
<td>2.43E+02</td>
<td>2.47E+02</td>
</tr>
<tr>
<td>8.00E+00</td>
<td>2.50E+00</td>
<td>9.60E-03</td>
<td>4.35E+02</td>
<td>2.37E+02</td>
<td>2.43E+02</td>
</tr>
<tr>
<td>9.00E+00</td>
<td>3.50E+00</td>
<td>9.07E-03</td>
<td>4.00E+02</td>
<td>2.31E+02</td>
<td>2.40E+02</td>
</tr>
<tr>
<td>1.00E+01</td>
<td>4.50E+00</td>
<td>8.56E-03</td>
<td>3.68E+02</td>
<td>2.25E+02</td>
<td>2.37E+02</td>
</tr>
<tr>
<td>1.10E+01</td>
<td>5.50E+00</td>
<td>7.98E-03</td>
<td>3.37E+02</td>
<td>2.21E+02</td>
<td>2.35E+02</td>
</tr>
<tr>
<td>1.20E+01</td>
<td>6.50E+00</td>
<td>7.37E-03</td>
<td>3.08E+02</td>
<td>2.19E+02</td>
<td>2.34E+02</td>
</tr>
<tr>
<td>1.30E+01</td>
<td>7.50E+00</td>
<td>6.80E-03</td>
<td>2.82E+02</td>
<td>2.17E+02</td>
<td>2.33E+02</td>
</tr>
<tr>
<td>1.40E+01</td>
<td>8.50E+00</td>
<td>6.27E-03</td>
<td>2.58E+02</td>
<td>2.15E+02</td>
<td>2.32E+02</td>
</tr>
<tr>
<td>1.50E+01</td>
<td>9.50E+00</td>
<td>5.78E-03</td>
<td>2.35E+02</td>
<td>2.13E+02</td>
<td>2.31E+02</td>
</tr>
<tr>
<td>1.60E+01</td>
<td>1.05E+01</td>
<td>5.32E-03</td>
<td>2.15E+02</td>
<td>2.11E+02</td>
<td>2.30E+02</td>
</tr>
<tr>
<td>1.70E+01</td>
<td>1.15E+01</td>
<td>4.90E-03</td>
<td>1.96E+02</td>
<td>2.09E+02</td>
<td>2.29E+02</td>
</tr>
<tr>
<td>1.80E+01</td>
<td>1.25E+01</td>
<td>4.50E-03</td>
<td>1.78E+02</td>
<td>2.07E+02</td>
<td>2.28E+02</td>
</tr>
<tr>
<td>1.90E+01</td>
<td>1.35E+01</td>
<td>4.14E-03</td>
<td>1.63E+02</td>
<td>2.05E+02</td>
<td>2.27E+02</td>
</tr>
<tr>
<td>2.00E+01</td>
<td>1.45E+01</td>
<td>3.80E-03</td>
<td>1.48E+02</td>
<td>2.04E+02</td>
<td>2.26E+02</td>
</tr>
<tr>
<td>2.10E+01</td>
<td>1.55E+01</td>
<td>3.48E-03</td>
<td>1.34E+02</td>
<td>2.02E+02</td>
<td>2.25E+02</td>
</tr>
<tr>
<td>2.20E+01</td>
<td>1.65E+01</td>
<td>3.18E-03</td>
<td>1.22E+02</td>
<td>2.01E+02</td>
<td>2.24E+02</td>
</tr>
<tr>
<td>2.30E+01</td>
<td>1.75E+01</td>
<td>2.91E-03</td>
<td>1.11E+02</td>
<td>1.99E+02</td>
<td>2.23E+02</td>
</tr>
<tr>
<td>2.40E+01</td>
<td>1.85E+01</td>
<td>2.66E-03</td>
<td>1.01E+02</td>
<td>1.98E+02</td>
<td>2.22E+02</td>
</tr>
<tr>
<td>2.50E+01</td>
<td>1.95E+01</td>
<td>2.43E-03</td>
<td>9.12E+01</td>
<td>1.96E+02</td>
<td>2.22E+02</td>
</tr>
<tr>
<td>2.60E+01</td>
<td>2.05E+01</td>
<td>2.22E-03</td>
<td>8.26E+01</td>
<td>1.95E+02</td>
<td>2.21E+02</td>
</tr>
<tr>
<td>2.70E+01</td>
<td>2.15E+01</td>
<td>2.02E-03</td>
<td>7.48E+01</td>
<td>1.94E+02</td>
<td>2.20E+02</td>
</tr>
<tr>
<td>2.80E+01</td>
<td>2.25E+01</td>
<td>1.84E-03</td>
<td>6.77E+01</td>
<td>1.92E+02</td>
<td>2.19E+02</td>
</tr>
</tbody>
</table>
## Appendix C: Planetary Atmosphere Data

<table>
<thead>
<tr>
<th>Altitude (ell), km</th>
<th>Altitude (surf), km</th>
<th>Density (kg/m³)</th>
<th>Pressure (N/m²)</th>
<th>Temperature (K°)</th>
<th>Speed of Sound (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.90E+01</td>
<td>2.35E+01</td>
<td>1.68E-03</td>
<td>6.13E+01</td>
<td>1.91E+02</td>
<td>2.18E+02</td>
</tr>
<tr>
<td>3.00E+01</td>
<td>2.45E+01</td>
<td>1.53E-03</td>
<td>5.54E+01</td>
<td>1.90E+02</td>
<td>2.18E+02</td>
</tr>
<tr>
<td>3.10E+01</td>
<td>2.55E+01</td>
<td>1.39E-03</td>
<td>5.00E+01</td>
<td>1.88E+02</td>
<td>2.17E+02</td>
</tr>
<tr>
<td>3.20E+01</td>
<td>2.65E+01</td>
<td>1.26E-03</td>
<td>4.51E+01</td>
<td>1.87E+02</td>
<td>2.16E+02</td>
</tr>
<tr>
<td>3.30E+01</td>
<td>2.75E+01</td>
<td>1.15E-03</td>
<td>4.07E+01</td>
<td>1.86E+02</td>
<td>2.15E+02</td>
</tr>
<tr>
<td>3.40E+01</td>
<td>2.85E+01</td>
<td>1.04E-03</td>
<td>3.67E+01</td>
<td>1.84E+02</td>
<td>2.15E+02</td>
</tr>
<tr>
<td>3.50E+01</td>
<td>2.95E+01</td>
<td>9.46E-04</td>
<td>3.31E+01</td>
<td>1.83E+02</td>
<td>2.14E+02</td>
</tr>
<tr>
<td>3.60E+01</td>
<td>3.05E+01</td>
<td>8.58E-04</td>
<td>2.98E+01</td>
<td>1.81E+02</td>
<td>2.13E+02</td>
</tr>
<tr>
<td>3.70E+01</td>
<td>3.15E+01</td>
<td>7.77E-04</td>
<td>2.68E+01</td>
<td>1.80E+02</td>
<td>2.12E+02</td>
</tr>
<tr>
<td>3.80E+01</td>
<td>3.25E+01</td>
<td>7.03E-04</td>
<td>2.41E+01</td>
<td>1.79E+02</td>
<td>2.12E+02</td>
</tr>
<tr>
<td>3.90E+01</td>
<td>3.35E+01</td>
<td>6.36E-04</td>
<td>2.16E+01</td>
<td>1.78E+02</td>
<td>2.11E+02</td>
</tr>
<tr>
<td>4.00E+01</td>
<td>3.45E+01</td>
<td>5.75E-04</td>
<td>1.94E+01</td>
<td>1.77E+02</td>
<td>2.10E+02</td>
</tr>
<tr>
<td>4.10E+01</td>
<td>3.55E+01</td>
<td>5.19E-04</td>
<td>1.74E+01</td>
<td>1.75E+02</td>
<td>2.09E+02</td>
</tr>
<tr>
<td>4.20E+01</td>
<td>3.65E+01</td>
<td>4.69E-04</td>
<td>1.56E+01</td>
<td>1.74E+02</td>
<td>2.09E+02</td>
</tr>
<tr>
<td>4.30E+01</td>
<td>3.75E+01</td>
<td>4.23E-04</td>
<td>1.40E+01</td>
<td>1.73E+02</td>
<td>2.08E+02</td>
</tr>
<tr>
<td>4.40E+01</td>
<td>3.85E+01</td>
<td>3.81E-04</td>
<td>1.25E+01</td>
<td>1.72E+02</td>
<td>2.07E+02</td>
</tr>
<tr>
<td>4.50E+01</td>
<td>3.95E+01</td>
<td>3.43E-04</td>
<td>1.12E+01</td>
<td>1.71E+02</td>
<td>2.06E+02</td>
</tr>
<tr>
<td>4.60E+01</td>
<td>4.05E+01</td>
<td>3.09E-04</td>
<td>1.00E+01</td>
<td>1.69E+02</td>
<td>2.06E+02</td>
</tr>
<tr>
<td>4.70E+01</td>
<td>4.15E+01</td>
<td>2.78E-04</td>
<td>8.95E+00</td>
<td>1.68E+02</td>
<td>2.05E+02</td>
</tr>
<tr>
<td>4.80E+01</td>
<td>4.25E+01</td>
<td>2.50E-04</td>
<td>7.99E+00</td>
<td>1.67E+02</td>
<td>2.04E+02</td>
</tr>
<tr>
<td>4.90E+01</td>
<td>4.35E+01</td>
<td>2.25E-04</td>
<td>7.12E+00</td>
<td>1.66E+02</td>
<td>2.04E+02</td>
</tr>
<tr>
<td>5.00E+01</td>
<td>4.45E+01</td>
<td>2.02E-04</td>
<td>6.35E+00</td>
<td>1.65E+02</td>
<td>2.03E+02</td>
</tr>
<tr>
<td>5.10E+01</td>
<td>4.55E+01</td>
<td>1.81E-04</td>
<td>5.65E+00</td>
<td>1.63E+02</td>
<td>2.02E+02</td>
</tr>
<tr>
<td>5.20E+01</td>
<td>4.65E+01</td>
<td>1.62E-04</td>
<td>5.03E+00</td>
<td>1.62E+02</td>
<td>2.01E+02</td>
</tr>
<tr>
<td>5.30E+01</td>
<td>4.75E+01</td>
<td>1.45E-04</td>
<td>4.47E+00</td>
<td>1.61E+02</td>
<td>2.01E+02</td>
</tr>
<tr>
<td>5.40E+01</td>
<td>4.85E+01</td>
<td>1.30E-04</td>
<td>3.98E+00</td>
<td>1.60E+02</td>
<td>2.00E+02</td>
</tr>
<tr>
<td>5.50E+01</td>
<td>4.95E+01</td>
<td>1.16E-04</td>
<td>3.53E+00</td>
<td>1.59E+02</td>
<td>1.99E+02</td>
</tr>
<tr>
<td>5.60E+01</td>
<td>5.05E+01</td>
<td>1.04E-04</td>
<td>3.13E+00</td>
<td>1.57E+02</td>
<td>1.98E+02</td>
</tr>
<tr>
<td>5.70E+01</td>
<td>5.15E+01</td>
<td>9.26E-05</td>
<td>2.77E+00</td>
<td>1.57E+02</td>
<td>1.98E+02</td>
</tr>
<tr>
<td>5.80E+01</td>
<td>5.25E+01</td>
<td>8.24E-05</td>
<td>2.46E+00</td>
<td>1.56E+02</td>
<td>1.97E+02</td>
</tr>
<tr>
<td>Altitude (ell), km</td>
<td>Altitude (surf), km</td>
<td>Density (kg/m(^3))</td>
<td>Pressure (N/m(^2))</td>
<td>Temperature (K°)</td>
<td>Speed of Sound (m/s)</td>
</tr>
<tr>
<td>-------------------</td>
<td>---------------------</td>
<td>-----------------------</td>
<td>-----------------------</td>
<td>-----------------</td>
<td>----------------------</td>
</tr>
<tr>
<td>5.90E+01</td>
<td>5.35E+01</td>
<td>7.32E-05</td>
<td>2.18E+00</td>
<td>1.56E+02</td>
<td>1.97E+02</td>
</tr>
<tr>
<td>6.00E+01</td>
<td>5.45E+01</td>
<td>6.51E-05</td>
<td>1.93E+00</td>
<td>1.55E+02</td>
<td>1.97E+02</td>
</tr>
<tr>
<td>6.10E+01</td>
<td>5.55E+01</td>
<td>5.78E-05</td>
<td>1.71E+00</td>
<td>1.54E+02</td>
<td>1.96E+02</td>
</tr>
<tr>
<td>6.20E+01</td>
<td>5.65E+01</td>
<td>5.13E-05</td>
<td>1.51E+00</td>
<td>1.54E+02</td>
<td>1.96E+02</td>
</tr>
<tr>
<td>6.30E+01</td>
<td>5.75E+01</td>
<td>4.56E-05</td>
<td>1.34E+00</td>
<td>1.53E+02</td>
<td>1.96E+02</td>
</tr>
<tr>
<td>6.40E+01</td>
<td>5.85E+01</td>
<td>4.04E-05</td>
<td>1.18E+00</td>
<td>1.53E+02</td>
<td>1.95E+02</td>
</tr>
<tr>
<td>6.50E+01</td>
<td>5.95E+01</td>
<td>3.59E-05</td>
<td>1.04E+00</td>
<td>1.52E+02</td>
<td>1.95E+02</td>
</tr>
<tr>
<td>6.60E+01</td>
<td>6.05E+01</td>
<td>3.18E-05</td>
<td>9.21E-01</td>
<td>1.52E+02</td>
<td>1.95E+02</td>
</tr>
<tr>
<td>6.70E+01</td>
<td>6.15E+01</td>
<td>2.82E-05</td>
<td>8.14E-01</td>
<td>1.51E+02</td>
<td>1.94E+02</td>
</tr>
<tr>
<td>6.80E+01</td>
<td>6.25E+01</td>
<td>2.50E-05</td>
<td>7.18E-01</td>
<td>1.50E+02</td>
<td>1.94E+02</td>
</tr>
<tr>
<td>6.90E+01</td>
<td>6.35E+01</td>
<td>2.21E-05</td>
<td>6.34E-01</td>
<td>1.50E+02</td>
<td>1.94E+02</td>
</tr>
<tr>
<td>7.00E+01</td>
<td>6.45E+01</td>
<td>1.96E-05</td>
<td>5.59E-01</td>
<td>1.49E+02</td>
<td>1.93E+02</td>
</tr>
<tr>
<td>7.10E+01</td>
<td>6.55E+01</td>
<td>1.73E-05</td>
<td>4.93E-01</td>
<td>1.49E+02</td>
<td>1.93E+02</td>
</tr>
<tr>
<td>7.20E+01</td>
<td>6.65E+01</td>
<td>1.53E-05</td>
<td>4.34E-01</td>
<td>1.48E+02</td>
<td>1.92E+02</td>
</tr>
<tr>
<td>7.30E+01</td>
<td>6.75E+01</td>
<td>1.36E-05</td>
<td>3.82E-01</td>
<td>1.48E+02</td>
<td>1.92E+02</td>
</tr>
<tr>
<td>7.40E+01</td>
<td>6.85E+01</td>
<td>1.20E-05</td>
<td>3.37E-01</td>
<td>1.47E+02</td>
<td>1.92E+02</td>
</tr>
<tr>
<td>7.50E+01</td>
<td>6.95E+01</td>
<td>1.06E-05</td>
<td>2.96E-01</td>
<td>1.46E+02</td>
<td>1.91E+02</td>
</tr>
<tr>
<td>7.60E+01</td>
<td>7.05E+01</td>
<td>9.35E-06</td>
<td>2.61E-01</td>
<td>1.46E+02</td>
<td>1.91E+02</td>
</tr>
<tr>
<td>7.70E+01</td>
<td>7.15E+01</td>
<td>8.25E-06</td>
<td>2.29E-01</td>
<td>1.45E+02</td>
<td>1.91E+02</td>
</tr>
<tr>
<td>7.80E+01</td>
<td>7.25E+01</td>
<td>7.28E-06</td>
<td>2.02E-01</td>
<td>1.45E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>7.90E+01</td>
<td>7.35E+01</td>
<td>6.42E-06</td>
<td>1.77E-01</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>8.00E+01</td>
<td>7.45E+01</td>
<td>5.66E-06</td>
<td>1.56E-01</td>
<td>1.44E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.10E+01</td>
<td>7.55E+01</td>
<td>4.99E-06</td>
<td>1.37E-01</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.20E+01</td>
<td>7.65E+01</td>
<td>4.40E-06</td>
<td>1.20E-01</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.30E+01</td>
<td>7.75E+01</td>
<td>4.00E-06</td>
<td>1.09E-01</td>
<td>1.42E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.40E+01</td>
<td>7.85E+01</td>
<td>3.51E-06</td>
<td>9.55E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.50E+01</td>
<td>7.95E+01</td>
<td>3.08E-06</td>
<td>8.39E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.60E+01</td>
<td>8.05E+01</td>
<td>2.70E-06</td>
<td>7.36E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.70E+01</td>
<td>8.15E+01</td>
<td>2.37E-06</td>
<td>6.47E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>8.80E+01</td>
<td>8.25E+01</td>
<td>2.08E-06</td>
<td>5.68E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
</tbody>
</table>
## Appendix C: Planetary Atmosphere Data

<table>
<thead>
<tr>
<th>Altitude (ell), km</th>
<th>Altitude (surf), km</th>
<th>Density (kg/m³)</th>
<th>Pressure (N/m²)</th>
<th>Temperature (K°)</th>
<th>Speed of Sound (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>8.90E+01</td>
<td>8.35E+01</td>
<td>1.83E-06</td>
<td>4.99E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.00E+01</td>
<td>8.45E+01</td>
<td>1.60E-06</td>
<td>4.38E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.10E+01</td>
<td>8.55E+01</td>
<td>1.41E-06</td>
<td>3.85E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.20E+01</td>
<td>8.65E+01</td>
<td>1.24E-06</td>
<td>3.38E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.30E+01</td>
<td>8.75E+01</td>
<td>1.09E-06</td>
<td>2.97E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.40E+01</td>
<td>8.85E+01</td>
<td>9.55E-07</td>
<td>2.61E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.50E+01</td>
<td>8.95E+01</td>
<td>8.39E-07</td>
<td>2.30E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.60E+01</td>
<td>9.05E+01</td>
<td>7.37E-07</td>
<td>2.02E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.70E+01</td>
<td>9.15E+01</td>
<td>6.48E-07</td>
<td>1.78E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.80E+01</td>
<td>9.25E+01</td>
<td>5.69E-07</td>
<td>1.56E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>9.90E+01</td>
<td>9.35E+01</td>
<td>5.01E-07</td>
<td>1.37E-02</td>
<td>1.43E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>1.00E+02</td>
<td>9.45E+01</td>
<td>4.40E-07</td>
<td>1.21E-02</td>
<td>1.44E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>1.01E+02</td>
<td>9.55E+01</td>
<td>3.87E-07</td>
<td>1.06E-02</td>
<td>1.44E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>1.02E+02</td>
<td>9.65E+01</td>
<td>3.40E-07</td>
<td>9.35E-03</td>
<td>1.44E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>1.03E+02</td>
<td>9.75E+01</td>
<td>2.99E-07</td>
<td>8.22E-03</td>
<td>1.44E+02</td>
<td>1.89E+02</td>
</tr>
<tr>
<td>1.04E+02</td>
<td>9.85E+01</td>
<td>2.63E-07</td>
<td>7.24E-03</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.05E+02</td>
<td>9.95E+01</td>
<td>2.32E-07</td>
<td>6.37E-03</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.06E+02</td>
<td>1.01E+02</td>
<td>2.04E-07</td>
<td>5.61E-03</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.07E+02</td>
<td>1.02E+02</td>
<td>1.79E-07</td>
<td>4.94E-03</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.08E+02</td>
<td>1.03E+02</td>
<td>1.58E-07</td>
<td>4.35E-03</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.09E+02</td>
<td>1.04E+02</td>
<td>1.39E-07</td>
<td>3.83E-03</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.10E+02</td>
<td>1.05E+02</td>
<td>1.22E-07</td>
<td>3.37E-03</td>
<td>1.44E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.11E+02</td>
<td>1.06E+02</td>
<td>1.07E-07</td>
<td>2.97E-03</td>
<td>1.45E+02</td>
<td>1.90E+02</td>
</tr>
<tr>
<td>1.12E+02</td>
<td>1.07E+02</td>
<td>9.42E-08</td>
<td>2.62E-03</td>
<td>1.45E+02</td>
<td>1.91E+02</td>
</tr>
<tr>
<td>1.13E+02</td>
<td>1.08E+02</td>
<td>8.27E-08</td>
<td>2.31E-03</td>
<td>1.46E+02</td>
<td>1.91E+02</td>
</tr>
<tr>
<td>1.14E+02</td>
<td>1.09E+02</td>
<td>7.27E-08</td>
<td>2.04E-03</td>
<td>1.47E+02</td>
<td>1.92E+02</td>
</tr>
<tr>
<td>1.15E+02</td>
<td>1.10E+02</td>
<td>6.39E-08</td>
<td>1.80E-03</td>
<td>1.47E+02</td>
<td>1.92E+02</td>
</tr>
<tr>
<td>1.16E+02</td>
<td>1.11E+02</td>
<td>5.62E-08</td>
<td>1.59E-03</td>
<td>1.48E+02</td>
<td>1.92E+02</td>
</tr>
<tr>
<td>1.17E+02</td>
<td>1.12E+02</td>
<td>4.95E-08</td>
<td>1.41E-03</td>
<td>1.49E+02</td>
<td>1.93E+02</td>
</tr>
<tr>
<td>1.18E+02</td>
<td>1.13E+02</td>
<td>4.36E-08</td>
<td>1.25E-03</td>
<td>1.50E+02</td>
<td>1.93E+02</td>
</tr>
</tbody>
</table>
### Solid State Aircraft

<table>
<thead>
<tr>
<th>Altitude(ell), km</th>
<th>Altitude (surf), km</th>
<th>Density (kg/m³)</th>
<th>Pressure (N/m²)</th>
<th>Temperature (K°)</th>
<th>Speed of Sound (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.19E+02</td>
<td>1.14E+02</td>
<td>3.84E-08</td>
<td>1.10E-03</td>
<td>1.50E+02</td>
<td>1.94E+02</td>
</tr>
<tr>
<td>1.20E+02</td>
<td>1.15E+02</td>
<td>3.39E-08</td>
<td>9.78E-04</td>
<td>1.51E+02</td>
<td>1.94E+02</td>
</tr>
<tr>
<td>1.21E+02</td>
<td>1.16E+02</td>
<td>2.99E-08</td>
<td>8.68E-04</td>
<td>1.52E+02</td>
<td>1.95E+02</td>
</tr>
<tr>
<td>1.22E+02</td>
<td>1.17E+02</td>
<td>2.64E-08</td>
<td>7.70E-04</td>
<td>1.52E+02</td>
<td>1.95E+02</td>
</tr>
<tr>
<td>1.23E+02</td>
<td>1.18E+02</td>
<td>2.34E-08</td>
<td>6.83E-04</td>
<td>1.53E+02</td>
<td>1.96E+02</td>
</tr>
<tr>
<td>1.24E+02</td>
<td>1.19E+02</td>
<td>2.07E-08</td>
<td>6.07E-04</td>
<td>1.54E+02</td>
<td>1.96E+02</td>
</tr>
<tr>
<td>1.25E+02</td>
<td>1.20E+02</td>
<td>1.83E-08</td>
<td>5.40E-04</td>
<td>1.54E+02</td>
<td>1.96E+02</td>
</tr>
</tbody>
</table>
### Mars-GRAM Generated Atmosphere Profile for -25° Latitude, 11° Longitude

<table>
<thead>
<tr>
<th>Height (km)</th>
<th>Density (kg/m³)</th>
<th>Temperature (K)</th>
<th>Pressure (Pa)</th>
<th>Speed of Sound (m/s)</th>
<th>Viscosity (kg/m s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.38</td>
<td>1.25E-02</td>
<td>252.4</td>
<td>594.8</td>
<td>251.11</td>
<td>1.29E-05</td>
</tr>
<tr>
<td>2.5</td>
<td>1.24E-02</td>
<td>251.8</td>
<td>589.2</td>
<td>250.82</td>
<td>1.28E-05</td>
</tr>
<tr>
<td>2.75</td>
<td>1.22E-02</td>
<td>250.7</td>
<td>577.9</td>
<td>250.27</td>
<td>1.28E-05</td>
</tr>
<tr>
<td>3</td>
<td>1.20E-02</td>
<td>249.7</td>
<td>566.9</td>
<td>249.77</td>
<td>1.27E-05</td>
</tr>
<tr>
<td>3.25</td>
<td>1.18E-02</td>
<td>248.6</td>
<td>556</td>
<td>249.22</td>
<td>1.27E-05</td>
</tr>
<tr>
<td>3.5</td>
<td>1.17E-02</td>
<td>247.5</td>
<td>545.4</td>
<td>248.66</td>
<td>1.26E-05</td>
</tr>
<tr>
<td>3.75</td>
<td>1.15E-02</td>
<td>246.5</td>
<td>534.9</td>
<td>248.16</td>
<td>1.26E-05</td>
</tr>
<tr>
<td>4</td>
<td>1.13E-02</td>
<td>245.4</td>
<td>524.7</td>
<td>247.61</td>
<td>1.25E-05</td>
</tr>
<tr>
<td>4.25</td>
<td>1.12E-02</td>
<td>244.3</td>
<td>514.6</td>
<td>247.05</td>
<td>1.24E-05</td>
</tr>
<tr>
<td>4.5</td>
<td>1.10E-02</td>
<td>243.2</td>
<td>504.8</td>
<td>246.49</td>
<td>1.24E-05</td>
</tr>
<tr>
<td>4.75</td>
<td>1.08E-02</td>
<td>242.2</td>
<td>495.1</td>
<td>245.99</td>
<td>1.23E-05</td>
</tr>
<tr>
<td>5</td>
<td>1.07E-02</td>
<td>241.1</td>
<td>485.6</td>
<td>245.43</td>
<td>1.23E-05</td>
</tr>
<tr>
<td>5.25</td>
<td>1.05E-02</td>
<td>240.5</td>
<td>475.4</td>
<td>245.12</td>
<td>1.22E-05</td>
</tr>
<tr>
<td>5.5</td>
<td>1.03E-02</td>
<td>239.9</td>
<td>465.4</td>
<td>244.82</td>
<td>1.22E-05</td>
</tr>
<tr>
<td>5.75</td>
<td>1.01E-02</td>
<td>239.3</td>
<td>455.6</td>
<td>244.51</td>
<td>1.22E-05</td>
</tr>
<tr>
<td>6</td>
<td>9.90E-03</td>
<td>238.7</td>
<td>446</td>
<td>244.2</td>
<td>1.22E-05</td>
</tr>
<tr>
<td>6.25</td>
<td>9.71E-03</td>
<td>238.1</td>
<td>436.6</td>
<td>243.9</td>
<td>1.21E-05</td>
</tr>
<tr>
<td>6.5</td>
<td>9.53E-03</td>
<td>237.6</td>
<td>427.4</td>
<td>243.64</td>
<td>1.21E-05</td>
</tr>
<tr>
<td>6.75</td>
<td>9.35E-03</td>
<td>237</td>
<td>418.4</td>
<td>243.33</td>
<td>1.21E-05</td>
</tr>
<tr>
<td>7</td>
<td>9.18E-03</td>
<td>236.4</td>
<td>409.6</td>
<td>243.02</td>
<td>1.20E-05</td>
</tr>
<tr>
<td>7.25</td>
<td>9.01E-03</td>
<td>235.8</td>
<td>401</td>
<td>242.72</td>
<td>1.20E-05</td>
</tr>
<tr>
<td>7.5</td>
<td>8.84E-03</td>
<td>235.2</td>
<td>392.6</td>
<td>242.41</td>
<td>1.20E-05</td>
</tr>
<tr>
<td>7.75</td>
<td>8.68E-03</td>
<td>234.6</td>
<td>384.3</td>
<td>242.1</td>
<td>1.19E-05</td>
</tr>
<tr>
<td>8</td>
<td>8.51E-03</td>
<td>234</td>
<td>376.2</td>
<td>241.79</td>
<td>1.19E-05</td>
</tr>
<tr>
<td>8.25</td>
<td>8.36E-03</td>
<td>233.4</td>
<td>368.3</td>
<td>241.48</td>
<td>1.19E-05</td>
</tr>
<tr>
<td>8.5</td>
<td>8.20E-03</td>
<td>232.8</td>
<td>360.5</td>
<td>241.17</td>
<td>1.19E-05</td>
</tr>
<tr>
<td>8.75</td>
<td>8.05E-03</td>
<td>232.2</td>
<td>353</td>
<td>240.86</td>
<td>1.18E-05</td>
</tr>
<tr>
<td>9</td>
<td>7.90E-03</td>
<td>231.6</td>
<td>345.5</td>
<td>240.54</td>
<td>1.18E-05</td>
</tr>
</tbody>
</table>
## Solid State Aircraft

<table>
<thead>
<tr>
<th>Height (km)</th>
<th>Density (kg/m³)</th>
<th>Temperature (K)</th>
<th>Pressure (Pa)</th>
<th>Speed of Sound (m/s)</th>
<th>Viscosity (kg/m s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>9.25</td>
<td>7.75E-03</td>
<td>231</td>
<td>338.3</td>
<td>240.23</td>
<td>1.18E-05</td>
</tr>
<tr>
<td>9.5</td>
<td>7.61E-03</td>
<td>230.4</td>
<td>331.1</td>
<td>239.92</td>
<td>1.17E-05</td>
</tr>
<tr>
<td>9.75</td>
<td>7.47E-03</td>
<td>229.9</td>
<td>324.2</td>
<td>239.66</td>
<td>1.17E-05</td>
</tr>
<tr>
<td>10</td>
<td>7.33E-03</td>
<td>229.3</td>
<td>317.3</td>
<td>239.35</td>
<td>1.17E-05</td>
</tr>
<tr>
<td>10.25</td>
<td>7.18E-03</td>
<td>228.8</td>
<td>310.5</td>
<td>239.09</td>
<td>1.16E-05</td>
</tr>
<tr>
<td>10.5</td>
<td>7.04E-03</td>
<td>228.4</td>
<td>303.7</td>
<td>238.88</td>
<td>1.16E-05</td>
</tr>
<tr>
<td>10.75</td>
<td>6.90E-03</td>
<td>228</td>
<td>297.1</td>
<td>238.67</td>
<td>1.16E-05</td>
</tr>
<tr>
<td>11</td>
<td>6.76E-03</td>
<td>227.6</td>
<td>290.7</td>
<td>238.46</td>
<td>1.16E-05</td>
</tr>
<tr>
<td>11.25</td>
<td>6.63E-03</td>
<td>227.1</td>
<td>284.4</td>
<td>238.2</td>
<td>1.16E-05</td>
</tr>
<tr>
<td>11.5</td>
<td>6.50E-03</td>
<td>226.7</td>
<td>278.2</td>
<td>237.99</td>
<td>1.15E-05</td>
</tr>
<tr>
<td>11.75</td>
<td>6.37E-03</td>
<td>226.3</td>
<td>272.2</td>
<td>237.78</td>
<td>1.15E-05</td>
</tr>
<tr>
<td>12</td>
<td>6.24E-03</td>
<td>225.9</td>
<td>266.3</td>
<td>237.57</td>
<td>1.15E-05</td>
</tr>
<tr>
<td>12.25</td>
<td>6.12E-03</td>
<td>225.4</td>
<td>260.5</td>
<td>237.3</td>
<td>1.15E-05</td>
</tr>
<tr>
<td>12.5</td>
<td>6.00E-03</td>
<td>225</td>
<td>254.9</td>
<td>237.09</td>
<td>1.15E-05</td>
</tr>
<tr>
<td>12.75</td>
<td>5.88E-03</td>
<td>224.6</td>
<td>249.3</td>
<td>236.88</td>
<td>1.14E-05</td>
</tr>
<tr>
<td>13</td>
<td>5.76E-03</td>
<td>224.2</td>
<td>243.9</td>
<td>236.67</td>
<td>1.14E-05</td>
</tr>
<tr>
<td>13.25</td>
<td>5.65E-03</td>
<td>223.8</td>
<td>238.6</td>
<td>236.46</td>
<td>1.14E-05</td>
</tr>
<tr>
<td>13.5</td>
<td>5.53E-03</td>
<td>223.3</td>
<td>233.5</td>
<td>236.19</td>
<td>1.14E-05</td>
</tr>
<tr>
<td>13.75</td>
<td>5.42E-03</td>
<td>222.9</td>
<td>228.4</td>
<td>235.98</td>
<td>1.13E-05</td>
</tr>
<tr>
<td>14</td>
<td>5.32E-03</td>
<td>222.5</td>
<td>223.5</td>
<td>235.77</td>
<td>1.13E-05</td>
</tr>
<tr>
<td>14.25</td>
<td>5.21E-03</td>
<td>222.1</td>
<td>218.6</td>
<td>235.56</td>
<td>1.13E-05</td>
</tr>
<tr>
<td>14.5</td>
<td>5.11E-03</td>
<td>221.6</td>
<td>213.9</td>
<td>235.29</td>
<td>1.13E-05</td>
</tr>
<tr>
<td>14.75</td>
<td>5.01E-03</td>
<td>221.2</td>
<td>209.2</td>
<td>235.08</td>
<td>1.13E-05</td>
</tr>
<tr>
<td>15</td>
<td>4.91E-03</td>
<td>220.8</td>
<td>204.7</td>
<td>234.87</td>
<td>1.12E-05</td>
</tr>
<tr>
<td>15.25</td>
<td>4.80E-03</td>
<td>220.4</td>
<td>200.1</td>
<td>234.66</td>
<td>1.12E-05</td>
</tr>
<tr>
<td>15.5</td>
<td>4.70E-03</td>
<td>219.9</td>
<td>195.6</td>
<td>234.39</td>
<td>1.12E-05</td>
</tr>
<tr>
<td>15.75</td>
<td>4.61E-03</td>
<td>219.5</td>
<td>191.1</td>
<td>234.18</td>
<td>1.12E-05</td>
</tr>
<tr>
<td>16</td>
<td>4.51E-03</td>
<td>219.1</td>
<td>186.8</td>
<td>233.96</td>
<td>1.11E-05</td>
</tr>
<tr>
<td>16.25</td>
<td>4.42E-03</td>
<td>218.7</td>
<td>182.6</td>
<td>233.75</td>
<td>1.11E-05</td>
</tr>
<tr>
<td>16.5</td>
<td>4.33E-03</td>
<td>218.3</td>
<td>178.5</td>
<td>233.54</td>
<td>1.11E-05</td>
</tr>
</tbody>
</table>
### Appendix C: Planetary Atmosphere Data

<table>
<thead>
<tr>
<th>Height (km)</th>
<th>Density (kg/m³)</th>
<th>Temperature (K)</th>
<th>Pressure (Pa)</th>
<th>Speed of Sound (m/s)</th>
<th>Viscosity (kg/m s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>16.75</td>
<td>4.24E-03</td>
<td>217.8</td>
<td>174.5</td>
<td>233.27</td>
<td>1.11E-05</td>
</tr>
<tr>
<td>17</td>
<td>4.15E-03</td>
<td>217.4</td>
<td>170.5</td>
<td>233.05</td>
<td>1.11E-05</td>
</tr>
<tr>
<td>17.25</td>
<td>4.06E-03</td>
<td>217</td>
<td>166.7</td>
<td>232.84</td>
<td>1.10E-05</td>
</tr>
<tr>
<td>17.5</td>
<td>3.98E-03</td>
<td>216.6</td>
<td>162.9</td>
<td>232.62</td>
<td>1.10E-05</td>
</tr>
<tr>
<td>17.75</td>
<td>3.90E-03</td>
<td>216.1</td>
<td>159.2</td>
<td>232.36</td>
<td>1.10E-05</td>
</tr>
<tr>
<td>18</td>
<td>3.82E-03</td>
<td>215.7</td>
<td>155.6</td>
<td>232.14</td>
<td>1.10E-05</td>
</tr>
<tr>
<td>18.25</td>
<td>3.74E-03</td>
<td>215.3</td>
<td>152.1</td>
<td>231.93</td>
<td>1.09E-05</td>
</tr>
<tr>
<td>18.5</td>
<td>3.66E-03</td>
<td>214.9</td>
<td>148.7</td>
<td>231.71</td>
<td>1.09E-05</td>
</tr>
<tr>
<td>18.75</td>
<td>3.59E-03</td>
<td>214.5</td>
<td>145.3</td>
<td>231.49</td>
<td>1.09E-05</td>
</tr>
<tr>
<td>19</td>
<td>3.51E-03</td>
<td>214</td>
<td>142.1</td>
<td>231.22</td>
<td>1.09E-05</td>
</tr>
<tr>
<td>19.25</td>
<td>3.44E-03</td>
<td>213.6</td>
<td>138.8</td>
<td>231.01</td>
<td>1.09E-05</td>
</tr>
<tr>
<td>19.5</td>
<td>3.37E-03</td>
<td>213.2</td>
<td>135.7</td>
<td>230.79</td>
<td>1.08E-05</td>
</tr>
<tr>
<td>19.75</td>
<td>3.30E-03</td>
<td>212.8</td>
<td>132.6</td>
<td>230.57</td>
<td>1.08E-05</td>
</tr>
<tr>
<td>20</td>
<td>3.23E-03</td>
<td>212.4</td>
<td>129.7</td>
<td>230.36</td>
<td>1.08E-05</td>
</tr>
</tbody>
</table>
### Mars-GRAM Generated Atmosphere Profile for 57° Latitude, 2.35° Longitude

<table>
<thead>
<tr>
<th>Height (km)</th>
<th>Density (kg/m³)</th>
<th>Temperature (K)</th>
<th>Pressure (Pa)</th>
<th>Speed of Sound (m/s)</th>
<th>Viscosity (kg/m s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>-1.74</td>
<td>2.82E-02</td>
<td>168.3</td>
<td>896.7</td>
<td>205.05</td>
<td>8.44E-06</td>
</tr>
<tr>
<td>-1.5</td>
<td>2.72E-02</td>
<td>168.8</td>
<td>865.8</td>
<td>205.36</td>
<td>8.47E-06</td>
</tr>
<tr>
<td>-1.25</td>
<td>2.63E-02</td>
<td>169.3</td>
<td>842.1</td>
<td>205.66</td>
<td>8.49E-06</td>
</tr>
<tr>
<td>-1</td>
<td>2.55E-02</td>
<td>169.8</td>
<td>819</td>
<td>205.97</td>
<td>8.52E-06</td>
</tr>
<tr>
<td>-0.75</td>
<td>2.48E-02</td>
<td>170.3</td>
<td>796.5</td>
<td>206.27</td>
<td>8.55E-06</td>
</tr>
<tr>
<td>-0.5</td>
<td>2.40E-02</td>
<td>170.8</td>
<td>774.7</td>
<td>206.57</td>
<td>8.57E-06</td>
</tr>
<tr>
<td>-0.25</td>
<td>2.33E-02</td>
<td>171.3</td>
<td>753.4</td>
<td>206.87</td>
<td>8.60E-06</td>
</tr>
<tr>
<td>0</td>
<td>2.26E-02</td>
<td>171.7</td>
<td>732.8</td>
<td>207.11</td>
<td>8.62E-06</td>
</tr>
<tr>
<td>0.25</td>
<td>2.19E-02</td>
<td>172.2</td>
<td>712.7</td>
<td>207.42</td>
<td>8.65E-06</td>
</tr>
<tr>
<td>0.5</td>
<td>2.12E-02</td>
<td>172.7</td>
<td>693.1</td>
<td>207.72</td>
<td>8.68E-06</td>
</tr>
<tr>
<td>0.75</td>
<td>2.06E-02</td>
<td>173.2</td>
<td>674.1</td>
<td>208.02</td>
<td>8.71E-06</td>
</tr>
<tr>
<td>1</td>
<td>2.00E-02</td>
<td>173.7</td>
<td>655.6</td>
<td>208.32</td>
<td>8.73E-06</td>
</tr>
<tr>
<td>1.25</td>
<td>1.94E-02</td>
<td>174.2</td>
<td>637.6</td>
<td>208.62</td>
<td>8.76E-06</td>
</tr>
<tr>
<td>1.5</td>
<td>1.88E-02</td>
<td>174.7</td>
<td>620.1</td>
<td>208.92</td>
<td>8.79E-06</td>
</tr>
<tr>
<td>1.75</td>
<td>1.82E-02</td>
<td>175.2</td>
<td>603.1</td>
<td>209.21</td>
<td>8.81E-06</td>
</tr>
<tr>
<td>2</td>
<td>1.77E-02</td>
<td>175.7</td>
<td>586.6</td>
<td>209.51</td>
<td>8.84E-06</td>
</tr>
<tr>
<td>2.25</td>
<td>1.71E-02</td>
<td>176.1</td>
<td>570.5</td>
<td>209.75</td>
<td>8.86E-06</td>
</tr>
<tr>
<td>2.5</td>
<td>1.66E-02</td>
<td>176.6</td>
<td>554.8</td>
<td>210.05</td>
<td>8.89E-06</td>
</tr>
<tr>
<td>2.75</td>
<td>1.61E-02</td>
<td>177.1</td>
<td>539.6</td>
<td>210.35</td>
<td>8.92E-06</td>
</tr>
<tr>
<td>3</td>
<td>1.56E-02</td>
<td>177.6</td>
<td>524.8</td>
<td>210.64</td>
<td>8.94E-06</td>
</tr>
<tr>
<td>3.25</td>
<td>1.52E-02</td>
<td>178.1</td>
<td>510.4</td>
<td>210.94</td>
<td>8.97E-06</td>
</tr>
<tr>
<td>3.5</td>
<td>1.47E-02</td>
<td>178.6</td>
<td>496.4</td>
<td>211.24</td>
<td>9.00E-06</td>
</tr>
<tr>
<td>3.75</td>
<td>1.43E-02</td>
<td>179.1</td>
<td>482.8</td>
<td>211.53</td>
<td>9.02E-06</td>
</tr>
<tr>
<td>4</td>
<td>1.38E-02</td>
<td>179.6</td>
<td>469.5</td>
<td>211.83</td>
<td>9.05E-06</td>
</tr>
<tr>
<td>4.25</td>
<td>1.34E-02</td>
<td>180.1</td>
<td>456.7</td>
<td>212.12</td>
<td>9.08E-06</td>
</tr>
<tr>
<td>4.5</td>
<td>1.30E-02</td>
<td>180.6</td>
<td>444.1</td>
<td>212.41</td>
<td>9.11E-06</td>
</tr>
<tr>
<td>4.75</td>
<td>1.26E-02</td>
<td>181</td>
<td>431.9</td>
<td>212.65</td>
<td>9.13E-06</td>
</tr>
<tr>
<td>5</td>
<td>1.22E-02</td>
<td>181.5</td>
<td>420.1</td>
<td>212.94</td>
<td>9.15E-06</td>
</tr>
</tbody>
</table>
## Appendix C: Planetary Atmosphere Data

<table>
<thead>
<tr>
<th>Height (km)</th>
<th>Density (kg/m³)</th>
<th>Temperature (K)</th>
<th>Pressure (Pa)</th>
<th>Speed of Sound (m/s)</th>
<th>Viscosity (kg/m·s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>5.25</td>
<td>1.19E-02</td>
<td>181.8</td>
<td>409</td>
<td>213.12</td>
<td>9.17E-06</td>
</tr>
<tr>
<td>5.5</td>
<td>1.16E-02</td>
<td>182.1</td>
<td>398.2</td>
<td>213.3</td>
<td>9.19E-06</td>
</tr>
<tr>
<td>5.75</td>
<td>1.12E-02</td>
<td>182.4</td>
<td>387.8</td>
<td>213.47</td>
<td>9.20E-06</td>
</tr>
<tr>
<td>6</td>
<td>1.09E-02</td>
<td>182.7</td>
<td>377.5</td>
<td>213.65</td>
<td>9.22E-06</td>
</tr>
<tr>
<td>6.25</td>
<td>1.06E-02</td>
<td>183</td>
<td>367.6</td>
<td>213.82</td>
<td>9.24E-06</td>
</tr>
<tr>
<td>6.5</td>
<td>1.03E-02</td>
<td>183.3</td>
<td>357.9</td>
<td>214</td>
<td>9.25E-06</td>
</tr>
<tr>
<td>6.75</td>
<td>1.00E-02</td>
<td>183.5</td>
<td>348.5</td>
<td>214.11</td>
<td>9.26E-06</td>
</tr>
<tr>
<td>7</td>
<td>9.77E-03</td>
<td>183.8</td>
<td>339.3</td>
<td>214.29</td>
<td>9.28E-06</td>
</tr>
<tr>
<td>7.25</td>
<td>9.49E-03</td>
<td>184.1</td>
<td>330.3</td>
<td>214.46</td>
<td>9.29E-06</td>
</tr>
<tr>
<td>7.5</td>
<td>9.23E-03</td>
<td>184.4</td>
<td>321.6</td>
<td>214.64</td>
<td>9.31E-06</td>
</tr>
<tr>
<td>7.75</td>
<td>8.97E-03</td>
<td>184.7</td>
<td>313.2</td>
<td>214.81</td>
<td>9.33E-06</td>
</tr>
<tr>
<td>8</td>
<td>8.72E-03</td>
<td>185</td>
<td>304.9</td>
<td>214.99</td>
<td>9.34E-06</td>
</tr>
<tr>
<td>8.25</td>
<td>8.48E-03</td>
<td>185.3</td>
<td>296.9</td>
<td>215.16</td>
<td>9.36E-06</td>
</tr>
<tr>
<td>8.5</td>
<td>8.24E-03</td>
<td>185.6</td>
<td>289.1</td>
<td>215.34</td>
<td>9.38E-06</td>
</tr>
<tr>
<td>8.75</td>
<td>8.01E-03</td>
<td>185.8</td>
<td>281.4</td>
<td>215.45</td>
<td>9.39E-06</td>
</tr>
<tr>
<td>9</td>
<td>7.79E-03</td>
<td>186.1</td>
<td>274</td>
<td>215.62</td>
<td>9.40E-06</td>
</tr>
<tr>
<td>9.25</td>
<td>7.57E-03</td>
<td>186.4</td>
<td>266.8</td>
<td>215.8</td>
<td>9.42E-06</td>
</tr>
<tr>
<td>9.5</td>
<td>7.36E-03</td>
<td>186.7</td>
<td>259.8</td>
<td>215.97</td>
<td>9.43E-06</td>
</tr>
<tr>
<td>9.75</td>
<td>7.16E-03</td>
<td>187</td>
<td>252.9</td>
<td>216.15</td>
<td>9.45E-06</td>
</tr>
<tr>
<td>10</td>
<td>6.96E-03</td>
<td>187.3</td>
<td>246.3</td>
<td>216.32</td>
<td>9.47E-06</td>
</tr>
<tr>
<td>10.25</td>
<td>6.77E-03</td>
<td>187.5</td>
<td>239.9</td>
<td>216.43</td>
<td>9.48E-06</td>
</tr>
<tr>
<td>10.5</td>
<td>6.59E-03</td>
<td>187.6</td>
<td>233.8</td>
<td>216.49</td>
<td>9.48E-06</td>
</tr>
<tr>
<td>10.75</td>
<td>6.41E-03</td>
<td>187.8</td>
<td>227.8</td>
<td>216.61</td>
<td>9.49E-06</td>
</tr>
<tr>
<td>11</td>
<td>6.24E-03</td>
<td>188</td>
<td>221.9</td>
<td>216.72</td>
<td>9.50E-06</td>
</tr>
<tr>
<td>11.25</td>
<td>6.08E-03</td>
<td>188.2</td>
<td>216.2</td>
<td>216.84</td>
<td>9.51E-06</td>
</tr>
<tr>
<td>11.5</td>
<td>5.92E-03</td>
<td>188.4</td>
<td>210.7</td>
<td>216.95</td>
<td>9.53E-06</td>
</tr>
<tr>
<td>11.75</td>
<td>5.76E-03</td>
<td>188.5</td>
<td>205.3</td>
<td>217.01</td>
<td>9.53E-06</td>
</tr>
<tr>
<td>12</td>
<td>5.60E-03</td>
<td>188.7</td>
<td>200</td>
<td>217.13</td>
<td>9.54E-06</td>
</tr>
<tr>
<td>12.25</td>
<td>5.46E-03</td>
<td>188.9</td>
<td>194.9</td>
<td>217.24</td>
<td>9.55E-06</td>
</tr>
<tr>
<td>12.5</td>
<td>5.31E-03</td>
<td>189.1</td>
<td>189.9</td>
<td>217.36</td>
<td>9.56E-06</td>
</tr>
<tr>
<td>Height (km)</td>
<td>Density (kg/m³)</td>
<td>Temperature (K)</td>
<td>Pressure (Pa)</td>
<td>Speed of Sound (m/s)</td>
<td>Viscosity (kg/m s)</td>
</tr>
<tr>
<td>------------</td>
<td>----------------</td>
<td>----------------</td>
<td>--------------</td>
<td>---------------------</td>
<td>-------------------</td>
</tr>
<tr>
<td>12.75</td>
<td>5.17E-03</td>
<td>189.3</td>
<td>185</td>
<td>217.47</td>
<td>9.57E-06</td>
</tr>
<tr>
<td>13</td>
<td>5.03E-03</td>
<td>189.4</td>
<td>180.2</td>
<td>217.53</td>
<td>9.58E-06</td>
</tr>
<tr>
<td>13.25</td>
<td>4.90E-03</td>
<td>189.6</td>
<td>175.6</td>
<td>217.64</td>
<td>9.59E-06</td>
</tr>
<tr>
<td>13.5</td>
<td>4.77E-03</td>
<td>189.8</td>
<td>171.1</td>
<td>217.76</td>
<td>9.60E-06</td>
</tr>
<tr>
<td>13.75</td>
<td>4.64E-03</td>
<td>190</td>
<td>166.7</td>
<td>217.87</td>
<td>9.61E-06</td>
</tr>
<tr>
<td>14</td>
<td>4.52E-03</td>
<td>190.2</td>
<td>162.4</td>
<td>217.99</td>
<td>9.62E-06</td>
</tr>
<tr>
<td>14.25</td>
<td>4.40E-03</td>
<td>190.3</td>
<td>158.2</td>
<td>218.04</td>
<td>9.63E-06</td>
</tr>
<tr>
<td>14.5</td>
<td>4.28E-03</td>
<td>190.5</td>
<td>154.2</td>
<td>218.16</td>
<td>9.64E-06</td>
</tr>
<tr>
<td>14.75</td>
<td>4.16E-03</td>
<td>190.7</td>
<td>150.2</td>
<td>218.27</td>
<td>9.65E-06</td>
</tr>
<tr>
<td>15</td>
<td>4.05E-03</td>
<td>190.9</td>
<td>146.4</td>
<td>218.39</td>
<td>9.66E-06</td>
</tr>
<tr>
<td>15.25</td>
<td>3.95E-03</td>
<td>190.9</td>
<td>142.7</td>
<td>218.39</td>
<td>9.66E-06</td>
</tr>
<tr>
<td>15.5</td>
<td>3.85E-03</td>
<td>191</td>
<td>139</td>
<td>218.45</td>
<td>9.66E-06</td>
</tr>
<tr>
<td>15.75</td>
<td>3.75E-03</td>
<td>191</td>
<td>135.5</td>
<td>218.45</td>
<td>9.66E-06</td>
</tr>
<tr>
<td>16</td>
<td>3.66E-03</td>
<td>191</td>
<td>132.1</td>
<td>218.45</td>
<td>9.66E-06</td>
</tr>
<tr>
<td>16.25</td>
<td>3.56E-03</td>
<td>191.1</td>
<td>128.7</td>
<td>218.5</td>
<td>9.67E-06</td>
</tr>
<tr>
<td>16.5</td>
<td>3.47E-03</td>
<td>191.1</td>
<td>125.4</td>
<td>218.5</td>
<td>9.67E-06</td>
</tr>
<tr>
<td>16.75</td>
<td>3.38E-03</td>
<td>191.1</td>
<td>122.3</td>
<td>218.5</td>
<td>9.67E-06</td>
</tr>
<tr>
<td>17</td>
<td>3.30E-03</td>
<td>191.2</td>
<td>119.2</td>
<td>218.56</td>
<td>9.68E-06</td>
</tr>
<tr>
<td>17.25</td>
<td>3.21E-03</td>
<td>191.2</td>
<td>116.1</td>
<td>218.56</td>
<td>9.68E-06</td>
</tr>
<tr>
<td>17.5</td>
<td>3.13E-03</td>
<td>191.2</td>
<td>113.2</td>
<td>218.56</td>
<td>9.68E-06</td>
</tr>
<tr>
<td>17.75</td>
<td>3.05E-03</td>
<td>191.2</td>
<td>110.3</td>
<td>218.56</td>
<td>9.68E-06</td>
</tr>
<tr>
<td>18</td>
<td>2.97E-03</td>
<td>191.3</td>
<td>107.5</td>
<td>218.62</td>
<td>9.68E-06</td>
</tr>
<tr>
<td>18.25</td>
<td>2.90E-03</td>
<td>191.3</td>
<td>104.8</td>
<td>218.62</td>
<td>9.68E-06</td>
</tr>
<tr>
<td>18.5</td>
<td>2.82E-03</td>
<td>191.3</td>
<td>102.1</td>
<td>218.62</td>
<td>9.68E-06</td>
</tr>
<tr>
<td>18.75</td>
<td>2.75E-03</td>
<td>191.4</td>
<td>99.5</td>
<td>218.67</td>
<td>9.69E-06</td>
</tr>
<tr>
<td>19</td>
<td>2.68E-03</td>
<td>191.4</td>
<td>97</td>
<td>218.67</td>
<td>9.69E-06</td>
</tr>
<tr>
<td>19.25</td>
<td>2.61E-03</td>
<td>191.4</td>
<td>94.5</td>
<td>218.67</td>
<td>9.69E-06</td>
</tr>
<tr>
<td>19.5</td>
<td>2.54E-03</td>
<td>191.5</td>
<td>92.1</td>
<td>218.73</td>
<td>9.69E-06</td>
</tr>
<tr>
<td>19.75</td>
<td>2.48E-03</td>
<td>191.5</td>
<td>89.8</td>
<td>218.73</td>
<td>9.69E-06</td>
</tr>
<tr>
<td>20</td>
<td>2.42E-03</td>
<td>191.5</td>
<td>87.5</td>
<td>218.73</td>
<td>9.69E-06</td>
</tr>
</tbody>
</table>
Aerodynamic Analysis of the Solid-State Aircraft

A Major Qualifying Project Report

Submitted to the Faculty of

Worcester Polytechnic Institute

In Partial Fulfillment of the Requirements for the

Degree of Bachelor of Science

By

Andrew Day

Lindsey Robbins

April 25, 2005

Professor D. Olinger
Advisor

Anthony Colozza
Northland Scientific
Ohio Aerospace Institute
Abstract

The purpose of our project was to improve the aerodynamic analysis of the Solid State Aircraft. The Solid State Aircraft uses ionic polymeric metal composite materials for its wings. An ionic polymeric metal composite is a material that deforms in the presence of an electric field and returns to its original shape once the electric field is removed. By constructing a large wing out of this material it may be possible to create a flapping wing aircraft. Thin photovoltaic material and thin film lithium ion batteries layered on top of the ionic polymeric metal composite allow the wing to use solar energy as a power source.

The Solid State Aircraft has many potential applications. It could perform high resolution imagery, atmospheric sampling, magnetic field imagery, or provide a communications relay. It could perform its functions in either Earth’s atmosphere or the atmospheres of other planets.

Our first task in improving the aerodynamic analysis was to determine an optimal airfoil shape for the projected flight regimes of Earth, Mars, and Venus. Because the Reynolds numbers involved were especially low in some environments, we had to validate that our airfoil analysis software remained reliable at these lower Reynolds numbers. Using this software we created an optimal airfoil shape for the flapping motion. We created a spreadsheet that calculated the lift, thrust, drag, and the amount of power consumed by the flapping wing using a quasi-steady, blade element approach. We also incorporated Tony Colozza’s previous work which calculated the power available to the aircraft from the solar arrays. By using this spreadsheet to alter the flapping motion and compare the power available to the aircraft versus the power consumed, we were able to
determine the various altitudes, latitudes, and the times of year the aircraft would be capable of flying for the various planetary environments.
Acknowledgements

We would like to take this opportunity to thank those without whom this project would not have been possible. We would like to thank our project advisor, David Olinger, for his many hours of guidance. We would like to thank our SSA project team for providing us with the opportunity to be a part of such a fascinating research project, and in particular thank Tony Colozza for his help and guidance in our efforts. We would also like to thank the IGSD of WPI and Toni Rusnak and Dr. David Kankam of the NASA Glenn Research Center for providing the connection between WPI and NASA that made this partnership possible. Lastly, we acknowledge fellowship support from a NASA Workforce Training grant through the Massachusetts Space Grant Consortium, to help finance part of our stay in Cleveland.
# Table of Contents

ABSTRACT ............................................................................................................................I

ACKNOWLEDGEMENTS .................................................................................................III

TABLE OF CONTENTS ........................................................................................................IV

TABLE OF FIGURES .......................................................................................................... V

TABLE OF TABLES ........................................................................................................... VIII

CHAPTER 1: INTRODUCTION ............................................................................................ 1

CHAPTER 2: LITERATURE REVIEW .................................................................................. 4
  2.1 IPMC, PHOTOVOLTAICS, AND LITHIUM ION BATTERIES ........................................ 4
  2.2 SSA FLIGHT APPLICATIONS .................................................................................... 7
  2.3 HISTORY OF FLAPPING FLIGHT ............................................................................ 7
  2.4 AERODYNAMIC FUNDAMENTALS .......................................................................... 9
  2.5 FLIGHT ON MARS, EARTH OR VENUS .................................................................. 18
  2.6 JAVAFOIL CODE .................................................................................................. 20

CHAPTER 3: METHODS AND RESULTS ........................................................................ 24
  3.1.1 METHODS FOR JAVAFOIL VALIDATION ............................................................ 24
  3.1.2 RESULTS OF JAVAFOIL VALIDATION ............................................................... 25
  3.2.1 METHODS FOR GEOMETRIC OPTIMIZATION .................................................. 29
  3.2.2 RESULTS OF GEOMETRIC OPTIMIZATION ...................................................... 30
  3.3.1 WING FLAP OPTIMIZATION ............................................................................ 54
  3.3.2 DETAILS OF SPREADSHEET CALCULATIONS .................................................... 56
  3.3.3 WING MOTION DETAILS ................................................................................... 65
  3.3.4 DRAG ANALYSIS ............................................................................................... 67
  3.4.1 IPMC EFFICIENCY ............................................................................................ 73

CHAPTER 4: CONCLUSIONS AND RECOMMENDATIONS ......................................... 78
  4.1 EARTH CONCLUSIONS ......................................................................................... 78
  4.2 MARS CONCLUSIONS ............................................................................................ 88
  4.3 VENUS CONCLUSIONS .......................................................................................... 90
  4.4 EFFICIENCY AND AERODYNAMICS CONCLUSIONS ............................................. 93

REFERENCES ..................................................................................................................... 95
Table of Figures

FIGURE 1. SOLID STATE AIRCRAFT CONCEPT ................................................................. 1
FIGURE 2. DEFORMATION OF IPMC FROM REFERENCE 1 ............................................ 4
FIGURE 3. THIN FILM PHOTOVOLTAICS FROM REFERENCE 1 .................................. 5
FIGURE 4. THIN FILM LITHIUM ION BATTERY FROM REFERENCE 1 ....................... 6
FIGURE 5. MATERIAL LAYERS IN THE SSA WING FROM REFERENCE 1 ................... 6
FIGURE 6. LEONARDO DA VINCI’S ORNITHOPTER DESIGN FROM REFERENCE 3 .. 7
FIGURE 7. PROJECT ORNITHOPTER AT TAKE-OFF FROM REFERENCE 3 ............... 8
FIGURE 8. AERODYNAMIC FORCES ON AN AIRCRAFT ........................................... 10
FIGURE 9. TYPICAL AIRFOIL AIRFLOW .................................................................... 10
FIGURE 10. ANGLE OF ATTACK DIFFERENTIATION ................................................. 11
FIGURE 11. AIRFOIL STALL VISUALIZATION ............................................................ 12
FIGURE 12. SHEAR FORCES ON WING .................................................................... 12
FIGURE 13. PRESSURE FORCES ON WING ............................................................. 13
FIGURE 14. WING TIP VORTICES – INDUCED DRAG .............................................. 14
FIGURE 15. PLANFORM AREA FROM REFERENCE 1 .................................................. 14
FIGURE 16. AIRFOIL DURING DOWNSTROKE ......................................................... 16
FIGURE 17. AIRFOIL DURING UPSTROKE ................................................................ 17
FIGURE 18. MARS, EARTH AND VENUS FROM REFERENCE 1 ................................ 18
FIGURE 19. L/D VERSUS REYNOLDS NUMBER FOR VARIOUS FLYING OBJECTS 20
FIGURE 20. PANEL METHOD DIAGRAM FROM REFERENCE 8 ................................. 21
FIGURE 21. BOUNDARY LAYER DIAGRAM .............................................................. 22
FIGURE 22. DRAG COEFFICIENT VALIDATION DATA FOR A CAMBERED PLATE: 1.93%
    THICKNESS, 4% CAMBER, LEADING EDGE RADIUS OF .6, Re=140,000 ............... 25
FIGURE 23. LIFT COEFFICIENT VALIDATION DATA FOR A CAMBERED PLATE, 1.93%
    THICKNESS, 4% CAMBER, LEADING EDGE RADIUS OF .6, Re=140,000 .............. 26
FIGURE 24. LIFT-TO-DRAG RATIO VERSUS ANGLE OF ATTACK FOR A CAMBERED PLATE:
    1.93% THICKNESS, 4% CAMBER, LEADING EDGE RADIUS OF .6, Re=140,000 ....... 26
FIGURE 25. DRAG COEFFICIENT VALIDATION DATA FOR A FLAT PLATE: 1.96% THICKNESS,
    0% CAMBER, LEADING EDGE RADIUS OF .6, Re=80,000 .................................... 27
FIGURE 26. LIFT COEFFICIENT VALIDATION DATA FOR A FLAT PLATE: 1.96% THICKNESS,
    0% CAMBER, LEADING EDGE RADIUS OF .6, Re=80,000 .................................... 27
FIGURE 27. LIFT-TO-DRAG RATIO VERSUS ANGLE OF ATTACK FOR A FLAT PLATE: 1.96%
    THICKNESS, 0% CAMBER, LEADING EDGE RADIUS OF .6, Re=80,000 ............... 28
FIGURE 28. EFFECT OF THICKNESS ON LIFT COEFFICIENT: CAMBER OF 5%, LEADING EDGE
    RADIUS of .15%, REYNOLDS NUMBER=100,000 ................................................. 31
FIGURE 29. EFFECT OF THICKNESS ON DRAG COEFFICIENT: CAMBER OF 5%, LEADING
    EDGE RADIUS of .15%, REYNOLDS NUMBER=100,000 ......................................... 31
FIGURE 30. EFFECT OF THICKNESS ON L/D: CAMBER OF 5%, LEADING EDGE RADIUS
    OF .15%, REYNOLDS NUMBER=100,000 .......................................................... 32
FIGURE 31. EFFECT OF CAMBER ON LIFT COEFFICIENT: THICKNESS OF 2%, LEADING EDGE
    RADIUS OF .15%, REYNOLDS NUMBER=100,000 ................................................ 33
FIGURE 32. EFFECT OF CAMBER ON DRAG COEFFICIENT: THICKNESS OF 2%, LEADING
    EDGE RADIUS OF .15%, REYNOLDS NUMBER=100,000 ....................................... 34
FIGURE 33. EFFECT OF CAMBER ON L/D: THICKNESS OF 2%, LEADING EDGE RADIUS OF
    .15%, REYNOLDS NUMBER=100,000 .......................................................... 34
FIGURE 55. EFFECT OF CAMBER LOCATION ON L/D: 2% THICKNESS, 10% CAMBER, .15% LEADING EDGE RADIUS, Re=450,000. .......................................................... 51
FIGURE 56. OPTIMIZED AIRFOIL GEOMETRY FOR EARTH APPLICATION: 2% THICKNESS AT 30% LOCATION, 10% CAMBER AT 40% LOCATION, .1% LEADING EDGE RADIUS, Re=450,000.................................................. 52
FIGURE 57. LIFT AND DRAG COEFFICIENTS FOR EARTH APPLICATION: 2% THICKNESS AT 30% LOCATION, 10% CAMBER AT 40% LOCATION, .1% LEADING EDGE RADIUS, Re=450,000.................................................. 53
FIGURE 58. PERFORMANCE RATIO FOR EARTH APPLICATION: 2% THICKNESS AT 30% LOCATION, 10% CAMBER AT 40% LOCATION, .1% LEADING EDGE RADIUS, Re=450,000.................................................. 53
FIGURE 59. FLAPPING MOTION .................................................................................. 54
FIGURE 60A/B. EFFECT OF FLAPPING WING ON LIFT AND DRAG ................................... 57
FIGURE 61. VELOCITY GEOMETRY ........................................................................... 58
FIGURE 62. FLAPPING VELOCITIES ......................................................................... 60
FIGURE 63. NORMALIZED FLAPPING VELOCITY AND ACCELERATION FOR ONE FLAPPING MOTION .................................................................................................................. 61
FIGURE 64. MOTION COMPARISON FOR PARABOLIC VERSUS STRAIGHT LINE WING MOTION ...................................................................................................................... 62
FIGURE 65. SWEEP ANGLE Δ VS SPAN FOR VARIOUS MAXIMUM DEFLECTION ANGLES (ΘMAX) .................................................................................................................. 63
FIGURE 66. SWEEP ANGLE VS DEFLECTION ANGLE .................................................. 64
FIGURE 67. INWARD DEFLECTION ANGLE THETA .................................................................................................................. 65
FIGURE 68A/B. VARIATION OF CAMBER AT DIFFERENT FLAP PHASES ..................... 66
FIGURE 69. CHORDLENGTH DISTRIBUTION ALONG SPAN FROM ROOT TO TIP........... 68
FIGURE 70. OSCILLOSCOPE READINGS ..................................................................... 74
FIGURE 71. SAMPLE SETUP WITH GRAPH PAPER .................................................... 75
FIGURE 72. GRAPH PAPER MEASUREMENT ON TELEVISION SCREEN .................... 76
FIGURE 73. POWER AVAILABLE BY MONTH OF THE YEAR ........................................ 79
FIGURE 74. POWER AVAILABLE BY LATITUDE ...................................................... 80
FIGURE 75. POWER VERSUS ALTITUDE FOR PLANET EARTH AT 23 DEGREES LATITUDE ON SUMMER SOLSTICE ................................................................. 81
FIGURE 76. LIFT-TO-DRAG RATIO VERSUS ALTITUDE FOR PLANET EARTH AT 23 DEGREES LATITUDE ON SUMMER SOLSTICE ................................................................. 81
FIGURE 77. POWER VERSUS ALTITUDE FOR PLANET EARTH AT 60 DEGREES LATITUDE ON SUMMER SOLSTICE ................................................................. 82
FIGURE 78. LIFT-TO-DRAG RATIO VERSUS ALTITUDE FOR PLANET EARTH AT 60 DEGREES LATITUDE ON SUMMER SOLSTICE ................................................................. 83
FIGURE 79. POWER VERSUS ALTITUDE FOR PLANET EARTH AT 85 DEGREES LATITUDE ON SUMMER SOLSTICE ................................................................. 83
FIGURE 80. LIFT-TO-DRAG RATIO VERSUS ALTITUDE FOR PLANET EARTH AT 85 DEGREES LATITUDE ON SUMMER SOLSTICE ................................................................. 84
FIGURE 81. POWER VERSUS ALTITUDE FOR PLANET EARTH AT 0 DEGREES LATITUDE ON WINTER SOLSTICE ................................................................. 85
FIGURE 82. LIFT-TO-DRAG RATIO VERSUS ALTITUDE FOR PLANET EARTH AT 0 DEGREES LATITUDE ON WINTER SOLSTICE ................................................................. 86
FIGURE 83. POWER VERSUS ALTITUDE FOR PLANET EARTH AT 23 DEGREES LATITUDE ON WINTER SOLSTICE ................................................................. 86
Table of Tables

TABLE 1. DENSITY (kg/m^3) VS ALTITUDE (KM) FOR VENUS, EARTH AND MARS........ 19
TABLE 2. DESIGN PARAMETERS DEFINED ........................................................................ 23
TABLE 3. EFFECT OF AIRFOIL THICKNESS RATIO AND REYNOLDS NUMBER ON ZERO LIFT DRAG^{10} (PAGE 46-47) ........................................................................ 28
Chapter 1: Introduction

Recent advancements in polymers, solar energy technologies, and thin-film batteries have made possible an aircraft powered by solar energy and propelled by flapping wings. This concept, the “Solid State” Aircraft (see Figure 1), has many potential applications including planetary exploration. The name “Solid State” refers to its unique characteristic of being a flexible moving structure, while at the same time containing no mechanical parts. With its lightweight structure and lack of mechanical parts, the aircraft would be ideal for deployment in the atmospheres of planetary bodies. The aircraft could be equipped to serve a number of different purposes, from gathering scientific data to serving as a communications platform.

Figure 1. Solid State Aircraft Concept

The use of an ionic polymeric metal composite (IPMC) is a key feature of the Solid State Aircraft concept. When an electric field is applied to this material, it has the
ability to deform. Once the electric field is removed, the material returns to its original shape. This deformation process resembles a flexible artificial muscle.

Thin film photovoltaic arrays will capture and convert solar energy while lithium-ion batteries will provide adequate storage to produce the electric field, enabling the wing to exhibit a flapping motion. A complex grid of electrodes controlled by a central processor will distribute this current to create a controllable electric field that dictates the motion of the wing.

The Solid State Aircraft design team is currently undertaking the conceptual design of the SSA.

The SSA team consists of the following people:

<table>
<thead>
<tr>
<th>Name</th>
<th>Institution</th>
<th>Role</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mr. Anthony Colozza</td>
<td>NASA Glenn Research Center/OAI</td>
<td>Conceptual Design/Power/Propulsion</td>
</tr>
<tr>
<td>Mr. Phillip Jenkines</td>
<td>Ohio Aerospace Institute</td>
<td>Power, Solar Cells and Batteries</td>
</tr>
<tr>
<td>Dr. Mohsen Shahinpoor</td>
<td>University of New Mexico</td>
<td>IPMC Development</td>
</tr>
<tr>
<td>Teryn Dalbello</td>
<td>University of Toledo/ICOMP</td>
<td>Aerodynamics</td>
</tr>
<tr>
<td>Dr. K.M. Isaac</td>
<td>University of Missouri/Rolla</td>
<td>Aerodynamics/CFD</td>
</tr>
<tr>
<td>Curtis Smith</td>
<td>Ohio Aerospace Institute</td>
<td>Project Management</td>
</tr>
<tr>
<td>Terri Deacey</td>
<td>Ohio Aerospace Institute</td>
<td>Project Management</td>
</tr>
<tr>
<td>Dr. David Olinger</td>
<td>Worcester Polytechnic Institute</td>
<td>Aerodynamics</td>
</tr>
</tbody>
</table>

Tony Colozza was our mentor at NASA Glenn Research Center, but our actual work took place at the Ohio Aerospace Institute. The SSA project is funded by the NASA Institute for Advanced Concept (NIAC) in the Phase II project. Our project focused on improving the aerodynamic analysis of the Solid State Aircraft. The first step of the project was to validate JavaFOIL, the aerodynamic analysis software used in this project. JavaFOIL is a 2D code that uses the vortex panel method and boundary layer theory and is well established for higher Reynolds numbers. However, since our range of projected flight environments included low Reynolds number regimes, we needed to study the accuracy of JavaFOIL at these conditions. The validated JavaFoil code was used to
optimize 2D airfoil shapes for the SSA by studying the effects of the camber, camber location, thickness, thickness location, and leading edge radius. The combination of these five characteristics that produced the highest lift to drag ratio over a wide range of angles of attack was identified. Once the airfoil shape was optimized we created an Excel spreadsheet that calculated the lift, thrust, and drag of a flapping wing using a quasi-steady blade element based analysis. This spreadsheet, which builds on Tony Colozza’s previous analysis\textsuperscript{2}, also calculates how much energy the flapping wing requires to maintain its flapping motion as well as the available energy in various planetary environments. By changing the characteristics of the flapping motion we were able to find a motion that produced an increased amount lift and thrust, while expending less energy. By comparing power requirements to the available power for the prescribed flapping motion we determined the various altitudes, latitudes, and times of the year the aircraft would be capable of flying in various planetary environments.

Since the Solid State Aircraft is still in the conceptual design stage, the results from this project are intended to aid the Solid State Aircraft design team as they further develop the concept.
Chapter 2: Literature Review

2.1 IPMC, Photovoltaics, and Lithium Ion Batteries

The key component of the flapping wing on the SSA is the ionic polymeric metal composite (IPMC)\(^1\). The IPMC consists of a polymer ion exchange membrane (PIEM) that allows movement of charged ions. The projected PIEM is a perfluorinate alkene that uses water to allow movement of ions. This membrane is then layered with a conductive coating, attaching the cathode to one side and the anode to the other. When a difference in potential occurs, an electric field is created and the IMPC deforms. In Figure 2, the electric potential forces the cations to one side of the material. The higher concentration of molecules causes this side of the IPMC to expand. Conversely the other side, having a lower concentration of molecules, contracts. This causes the material to bend. The amount of deformation is proportional to the strength of the electric field; the stronger the field, the greater the deformation.

Figure 2. Deformation of IPMC from Reference 1
Thin film photovoltaic solar cells will cover the surface of the SSA wing. Currently, similar solar cells are used on satellites and existing solar cell technology can be utilized with little modification\(^1\). At only a few thousands of an inch thick, the current solar arrays allow the flexibility the flapping wing will need to succeed\(^1\) (See Figure 3).

![Figure 3. Thin Film Photovoltaics from Reference 1](image)

Current thin film photovoltaic technology is capable of providing approximately one kilowatt of power per kilogram, operating at approximately 8%-10% efficiency. The results from our analysis will confirm that this efficiency is high enough for the needs of the SSA. Energy from the solar cells will be stored in thin film lithium-ion batteries. These batteries, seen in Figure 4, are well suited for applications in all the projected environments for the SSA. Similar to the photovoltaic array, they are lightweight and flexible. The batteries can also be charged and discharged frequently and quickly. They are rather insensitive to temperature change and have very little self-discharge\(^1\). Further development of the solar cells and batteries is still required for the SSA application, however this is outside the scope of our project.
When the stored energy is released it will be sent through a grid of electrodes. By controlling the amount of voltage that passes through the wing, the electric field and shape of the wing can also be controlled. A central processor will be used to distribute the desired voltages at the appropriate time to dictate the shape of the wing at each stage of its flapping motion. A schematic of the material layers for the SSA wing is shown in Figure 5.
2.2 SSA Flight Applications

There are many potential applications for the SSA in the planetary environments of Mars, Venus, and Earth:

- High-resolution surface imagery providing information necessary for mapping and geological characterization.
- Atmospheric sampling and analysis to determine the atmospheric composition.
- Magnetic field imagery that can provide information on the tectonic history of the planet as well as its geology and geophysics.
- Inter-aircraft networking to relay important information between solid state aircraft and communications satellites.

2.3 History of Flapping Flight

Engineers have been interested in mimicking flapping flight for centuries. In 1490, Leonardo da Vinci built an ornithopter model with thin membranes and a fingerlike structure similar to a bat. An ornithopter is an aircraft that generates thrust and lift from flapping wings (see Figure 6).

Figure 6. Leonardo da Vinci's Ornithopter Design from Reference 3
Da Vinci’s model never accomplished successful flight, but his innovative designs would inspire future studies. Also, he determined that the feathers of flapping creatures were not necessary to produce the lift and thrust.

In 1874 Alphonse Penaud demonstrated the first working flapping wing aircraft. His small scale ornithopter was powered by the twisting of a rubber band and would later be manufactured as a toy. In 1929 Alexander Lippisch created a human powered ornithopter that could perform a powered glide after being towed to the appropriate altitude. Thirty years later Emil Hartman reproduced a similar flying vehicle.

Percival Spencer created a number of engine powered ornithopters in the 1960’s, including a remotely piloted ornithopter that never achieved steady flight. It would be 1991 before a remotely piloted flapping aircraft achieved successful flight.

![Project Ornithopter at Take-off from Reference 3](image)

In 1999, the Project Ornithopter team designed an engine-powered piloted aircraft based on the earlier smaller remotely piloted version that was able to accelerate by
flapping its wings to lift-off speed (see Figure 7). The wings were joined in the center of
the aircraft body and moved up and down via pylons connected to the drivetrain\(^3\).

The Solid-State Aircraft Team intends to take this concept a step further, by using
solar energy and mimicking the intricate motion of a true flapping wing. This is now
possible due to the previously discussed advances in photovoltaics and composite
materials.

2.4 Aerodynamic Fundamentals

Before we can discuss the nature of flapping wing flight it is necessary to
understand how basic aerodynamic concepts apply to a rigid fixed wing. We will also
define some key terms that we will use throughout the report.

Any aircraft, both flapping and conventional, has four forces acting on it: weight, lift, drag, and thrust (see Figure 8). Weight is the force caused by gravity. Lift is the force
that the wing must produce to counteract the weight. Drag is the force air exerts on a
moving object to hinder its motion. Thrust is the force that must be produced to overcome the drag force.

The front edge of an airfoil is known as the leading edge. The leading edge has a
radius of curvature known as the leading edge radius. The back edge is the trailing edge. The straight-line distance connecting the two is referred to as the chordline. The curved line located midway between the upper and lower surface is the camber line and is measured as the largest vertical distance from the camber line to the chordline. The coordinates X,Y,Z represent distances along the chordline, perpendicular to the chordline, and along the wing span.
The Bernoulli equation is essential in understanding how a wing creates lift. Bernoulli’s equation states that the sum of the static pressure and the dynamic pressure equals the stagnation pressure, which remains constant.

\[ P_0 = P + \frac{1}{2} \rho v^2 = \text{constant} \] (1)

As the velocity of a fluid increases, the static pressure decreases. On a typical airfoil, the upper surface has a slight curve to it. As the air passes over the top of the wing, this curved surface forces the air to accelerate. Since the air on top is traveling faster than the air on the bottom, the upper surface experiences a lower pressure. The pressure difference across the top and bottom surfaces of the wing creates an upward force (see Figure 9).
The angle of attack is the angle between the airfoil’s chordline and the oncoming freestream flow. Zero, positive, and negative angles of attack are shown in Figure 10.

Figure 10. Angle of Attack Differentiation

Increasing the angle of attack increases lift. However, there is a limit to how much lift an airfoil can produce. In Figure 11A, the flow remains attached to the surface of the airfoil even as the angle of attack increases (Figure 11B). If the angle of attack is increased above the stall angle, the flow will separate and lift will drop dramatically (Figure 11C). This is known as airfoil stall.
Figure 11. Airfoil Stall Visualization

Although the wing creates lift, there is also a drag force on the wing. The two sources of drag are parasitic drag, which includes both viscous and pressure drag, and induced drag. Viscosity is a fluid property which characterizes a fluid's ability to resist shear deformation. As the temperature of a gas such as air decreases its viscosity also decreases. As the fluid flows over the airfoil, a parallel shear force is exerted upon the wing surface (see Figure 12) as a result of friction between the airfoil and the fluid.

Figure 12. Shear Forces on Wing

As mentioned earlier, wings generate lift by creating a pressure difference between the upper and lower surfaces. These pressure forces are normal to the airfoil surface as shown in Figure 13. Most of the surface is fairly horizontal. Since the pressure force is normal to the surface most of the pressure forces act vertically. However, there are some pressure forces that act in the horizontal direction. These horizontal pressure forces cause pressure drag shown in Figure 13. In general, blunt bodies are dominated by pressure drag, whereas streamline bodies such as airfoils are dominated by viscous drag.
Induced drag is caused by circulation downstream of the wing, creating a trailing vortex on the wing tips. The vortices induce a small velocity component in the downward direction called downwash, as seen in Figure 14. The drag created by the presence of downwash is called the induced drag.
Figure 14. Wing Tip Vortices – Induced Drag

Induced drag is calculated using the lift coefficient, $C_l$, which will be described later, and the aspect ratio, $AR$, as shown in Equation 2. The aspect ratio is defined as the wingspan, $b$, squared, divided by the planform area, $S$, from Equation 3. The planform shape of the solid state aircraft is below in Figure 15, similar to a pteranodon wing.

\[ AR = \frac{b^2}{S} \quad (3) \]

The lift and drag coefficients are non-dimensional numbers that relate the amount of lift and drag created to the fluid density, the velocity, and the planform area. The two coefficients are defined as the following:
\[ C_l = \frac{L}{\frac{1}{2} \rho V_{\infty}^2 S} \]  \hspace{1cm} (4)

\[ C_d = \frac{D}{\frac{1}{2} \rho V_{\infty}^2 S} \]  \hspace{1cm} (5)

where \( L \) is the lift force, \( D \) is the drag force, the Greek letter \( \rho \) is density, \( V_{\infty} \) is the freestream velocity, and \( S \) is the planform area.

The ratio of lift to drag describes the aerodynamic efficiency. High lift-to-drag ratios indicate an efficient wing whereas low lift-to-drag ratios indicate a less efficient wing.

Similar to the lift and drag coefficients is the moment coefficient as defined below, where \( M \) is the aerodynamic moment exerted about some point \( x \) on the airfoil and \( c \) is the chord length.

\[ C_{M_x} = \frac{M_x}{\frac{1}{2} \rho V^2 S c} \]  \hspace{1cm} (6)

No matter how efficient the wing is, there will always be both parasitic and induced drag that hinder an aircraft’s ability to maintain the speed needed to create the appropriate lift. To overcome this drag, planes use propellers or jet engines. Birds use the flapping motion of their wings. As the wing flaps, the effective freestream velocity over the wing increases as seen in Figure 16 which shows the airfoil motion during an upstroke. The effective angle of attack for our moving airfoil is the angle between the effective velocity and the chordline. This angle is used in the airfoil analysis software when determining lift and drag. \( \beta \) is the angle between the effective velocity and the
flight velocity. The geometric angle of attack is the sum of the effective angle of attack and β and is the angle between the flight velocity and the chordline.

Figure 16. Airfoil During Downstroke
During the downstroke (Figure 16) the wing slopes forward with respect to the aircraft’s horizontal flight velocity. This translates some of the vertical lift into horizontal thrust to overcome drag. Towards the wingtip the angle $\beta$ is larger than towards the body. Because this angle is higher towards the wingtip, it is the wingtip that is responsible for creating the bulk of the thrust in order for the bird to maintain the speed necessary to remain in flight.

Maneuverability refers to an aircraft’s ability to intentionally change direction. Stability is the aircraft’s ability control or counteract flight disturbances, caused by a gust of wind, for example. Maneuverability and stability are both desirable feature of any aircraft, an aircraft with high maneuverability is less stable, whereas an aircraft with high stability is less maneuverable.
Evolution in birds has shown that maneuverability is more favorable than stability. Over millions of years scientists have seen a decrease in the size of birds’ tail feathers, which help the bird maintain its course when yawing occurs. With less stabilization, birds must actively counteract any disturbances by altering their flapping motion, which requires more energy. Over time however, the cost of energy expenditure versus maneuverability in either catching food or escaping prey have led to less developed stabilization methods and more developed maneuverability.

For airplanes, the question of maneuverability versus stability is dependent on the function of the plane itself. Smaller planes, such as fighter jets, need maneuverability to dogfight. Larger planes, however, rely on their stability for long range flight (by using less energy for active stabilization). For the solid-state aircraft, stability would take priority. Because our focus is on lift, thrust, drag, and power, analyses on stability and maneuverability will not be included in this project.

2.5 Flight on Mars, Earth or Venus

In order for a wing to create lift there has to be an atmosphere present. This limits aircraft flight to planets that contain an appropriately dense atmosphere. The first projected applications of the solid-state aircraft are in the atmospheres of Mars, Venus, and the Earth (See Figure 18).

Figure 18. Mars, Earth and Venus from Reference 1
Mars has a significantly lower atmospheric density than Earth does. Venus on the other hand has a higher atmospheric density than Earth, full of thick corrosive gases. Those gases block much of the sunlight and extend approximately 66km above the planet’s surface. The SSA will most likely fly above the cloud layer in order to acquire the necessary solar energy to power flight. Table 1 compares the atmospheric densities of the three planetary environments at various altitudes.

<table>
<thead>
<tr>
<th>Altitude (km)</th>
<th>Venus (kg/m^3)</th>
<th>Earth (kg/m^3)</th>
<th>Mars (kg/m^3)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>61.6</td>
<td>1.11</td>
<td>0.0136</td>
</tr>
<tr>
<td>5</td>
<td>49.8</td>
<td>0.736</td>
<td>0.010</td>
</tr>
<tr>
<td>10</td>
<td>37.7</td>
<td>0.414</td>
<td>0.0</td>
</tr>
<tr>
<td>50</td>
<td>1.62</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>66</td>
<td>0.181</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>

The Reynolds number is a non-dimensional number that compares inertial and viscous effects in a fluid. Equation 7 defines the Reynolds number using the fluid density $\rho$, fluid velocity $V$, characteristic length $L$ (generally the average chord length for a wing), and the viscosity $\mu$, and is as follows:

$$Re = \frac{\rho VL}{\mu}$$  \hspace{1cm} (7)

Projected Reynolds numbers range from 100,000 (just above the surface of Mars) to 10,000,000 (in the gaseous clouds of Venus).

The Reynolds number is useful in creating dynamically similar environments. This principle is used when creating scaled versions to be tested in wind tunnels. The forces on a large model can be predicted by placing a smaller model in a wind tunnel and
matching the two Reynolds numbers.

![Figure 19. L/D Versus Reynolds Number for Various Flying Objects](image)

A Reynolds number of 100,000 is lower than conventional aircraft Reynolds numbers in Earth applications, which are on the order of $10^6 – 10^7$. Low Reynolds numbers lead to many disadvantageous aerodynamic effects such as lower lift-to-drag (L/D) ratios, as seen in Figure 19. As discussed earlier, when the angle of attack increases, the lift increases until the flow begins to separate and lift is lowered. In lower Reynolds number applications the flow separation occurs at lower angles of attack.

2.6 JavaFOIL Code

To determine the most efficient airfoil design for the SSA flapping wing, an aerodynamic analysis program called JavaFOIL was used. Possible configurations of airfoil thickness, thickness location, camber, camber location, and leading edge radius were analyzed. This program helped us predict the lift and drag on potential two-dimensional airfoil shapes. JavaFOIL uses potential flow theory and boundary layer theory\(^6\) to analyze the flow over an airfoil.
The panel method, part of the potential flow theory, is an analytical tool that helps to study the flow field around an airfoil in two dimensions. Computations using the panel method are done to determine the local inviscid velocity along the surface of the airfoil for any given angle of attack.

The boundary layer parameters of the airfoil shape can be determined with the integral method, which solves multiple differential equations using the coordinates starting at the stagnation point and tracing along the upper and lower surfaces of the airfoil. The boundary layer integral method is derived from the conservation of momentum; as the velocity distribution changes, the momentum also changes, creating the lift and drag.

Potential flow theory assumes an irrotational flow, where the vorticity is zero at every point in the flow. This flow can be described by a velocity potential $\varphi$ and the panel method is an analytical tool that helps to study the flow field around an airfoil in two dimensions. The panel method divides the airfoil into straight panels, each having their own velocity (see Figure 20). These velocities can be solved using simple geometry to relate the panels’ positions leading to a system of linear equations.

Figure 20. Panel Method Diagram from Reference 8
The velocity distribution was found using the panel method and potential flow theory, while drag can be found using the integral method and boundary layer theory. A boundary layer is the thin layer of fluid close to the surface of the airfoil. Due to the no-slip condition, when the flow is in direct contact with the surface its velocity is zero. In the boundary layer the velocity varies and eventually reaches the local velocity outside the boundary layer (see Figure 21). When the velocity distribution changes with angle of attack, the drag does as well. The boundary layer method is most accurate at Reynolds numbers between 500,000 to 20,000,000. The solid state aircraft conditions fall outside this range for the Mars application and JavaFOIL therefore must be validated to ensure accurate results.

![Figure 21. Boundary Layer Diagram](image)

Since JavaFOIL uses inviscid flow theory and fairly rudimentary boundary layer theory to analyze airfoil performance, it has limitations. The program code has difficulty handling flow separation. As discussed in the basic aerodynamics section, stall occurs when a wing’s angle of attack exceeds a certain limit leading to flow separation that diminishes lift and drastically increases drag. Intuition in interpreting JavaFOIL results will be important to ensure accurate predictions.
2.7 Design Summary – Solid State Aircraft

The design summary for the Solid State Aircraft is shown in Table 2.

Table 2. Design Parameters Defined

<table>
<thead>
<tr>
<th></th>
<th>Mars</th>
<th>Earth</th>
<th>Venus</th>
</tr>
</thead>
<tbody>
<tr>
<td>Weight (N)</td>
<td>145.47</td>
<td>382.59</td>
<td>348.27</td>
</tr>
<tr>
<td>Span (m)</td>
<td>10</td>
<td>10</td>
<td>10</td>
</tr>
<tr>
<td>Flight Velocity</td>
<td>Varied with Altitude</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Battery Efficiency</td>
<td>20%</td>
<td>20%</td>
<td>20%</td>
</tr>
<tr>
<td>IPMC Efficiency</td>
<td>10%</td>
<td>10%</td>
<td>10%</td>
</tr>
<tr>
<td>Area</td>
<td>9.9299101</td>
<td>9.9299101</td>
<td>9.9299101</td>
</tr>
<tr>
<td>Aspect Ratio</td>
<td>10.070585</td>
<td>10.070585</td>
<td>10.070585</td>
</tr>
<tr>
<td>Wing Loading (N/m²)</td>
<td>30.6072</td>
<td>30.6072</td>
<td>30.6072</td>
</tr>
</tbody>
</table>
Chapter 3: Methods and Results

Our objectives in this project were to validate the JavaFOIL aerodynamic analysis software for low Reynolds numbers, optimize a 2D airfoil shape, create a spreadsheet to calculate lift, thrust, and drag for a flapping wing, optimize this flapping motion, and incorporate these advances into an existing power analysis developed by Tony Colozza to determine the flight regimes of the Solid State Aircraft in different planetary applications.

3.1.1 Methods for JavaFOIL Validation

The low Reynolds number of approximately 100,000 that exists on Mars presented an uncertainty with the JavaFOIL analysis software’s accuracy. JavaFOIL was designed initially to quickly analyze aerodynamic coefficients. Since the JavaFOIL website\(^6\) states that this module is most reliable in the Reynolds number regime between 500,000 and 20,000,000 we needed to ensure JavaFOIL’s accuracy at Reynolds numbers as low as 100,000. Pelletier and Mueller\(^7\) built several thin, flat plate models, some with 4% camber in order to model the flight of micro-aerial vehicles at low Reynolds numbers with thin wings. Using experimental data from Pelletier and Mueller\(^7\) consisting of lift and drag coefficients at varying camber levels, we created the dynamically similar airfoil in JavaFOIL in order compare the experimental lift and drag coefficients to the coefficients computed by JavaFOIL to determine JavaFOIL’s accuracy. Because Pelletier and Mueller used end plates to eliminate wingtip effects, results could be compared to the 2D analysis of JavaFOIL.
3.1.2 Results of JavaFOIL Validation

Our findings show JavaFOIL’s difficulty to handle stall. In comparing the curves we found consistent results up to a certain angle of attack. Beyond this the results were not accurate.

Our JavaFOIL test results for the flat plate and the cambered plate showed that the lift and drag coefficients were indeed similar to Pelletier and Mueller’s experimental data at low angles of attack.

Figures 22, 23 and 24 show the correlation between the drag coefficient, the lift coefficient and the lift-to-drag ratio versus the angle of attack, for a circular arced airfoil. As the angle of attack leaves the range of -10 to 10 degrees, JavaFOIL diverges from the experimental data. Figure 24 shows large peak in the L/D coefficient due to the unusually low drag coefficients values which can be seen in Figure 22 for the low angles of attack of zero to three degrees.

![Drag Coefficient Validation Data for a Cambered Plate: 1.93% Thickness, 4% Camber, Leading Edge Radius of .6, Re=140,000](image-url)
Figures 25, 26 and 27 show the drag coefficient, the lift coefficient and the lift-to-drag ratio versus angle of attack for a flat plate. The data shows consistent correlation at the lower angles of attack, similar to the cambered plate.
Figure 25. Drag Coefficient Validation Data for a Flat Plate: 1.96% Thickness, 0% Camber, Leading Edge Radius of .6, Re=80,000

Figure 26. Lift Coefficient Validation Data for a Flat Plate: 1.96% Thickness, 0% Camber, Leading Edge Radius of .6, Re=80,000
Pelletier and Mueller ran the cambered plate at a Reynolds number of 140,000 and the flat plate at 80,000. The cambered plate produced a higher lift coefficient because it was cambered and a lower drag coefficient because of the elevated Reynolds number. Aerodynamicists Abbot and Von Doenhoff compiled experimental data from the NACA four-digit family (NACA 0002, 0004, 0006, and 0008 airfoils) and approximated a linear function to describe the relationship between airfoil thickness and Reynolds number on drag. Lower Reynolds numbers create greater drag coefficients, as shown in Table 3.

Table 3. Effect of Airfoil Thickness ratio and Reynolds Number on Zero Lift Drag¹⁰ (page 46-47)

<table>
<thead>
<tr>
<th>Re</th>
<th>Zero Lift Cd</th>
<th>Cd Increase per % Thickness</th>
</tr>
</thead>
<tbody>
<tr>
<td>2000</td>
<td>0.0656</td>
<td>0.0013</td>
</tr>
<tr>
<td>6000</td>
<td>0.0362</td>
<td>0.0008</td>
</tr>
<tr>
<td>600000</td>
<td>0.004</td>
<td>0.0002</td>
</tr>
</tbody>
</table>

The results of these tests made it possible to move on to the next step in the optimization process. JavaFOIL was validated for angles of attack from approximately
-10 to 10 degrees, according to the comparison with Pelletier and Mueller’s experimental data. Past ten degrees, the lift coefficients stray farther apart from each other, but in our analysis and development of the flight model, the effective angle of attack is never intended to reach ten degrees.

JavaFOIL was validated up to higher angles of attack with airfoils of greater thickness, but for the case of the solid-state airfoil with a thickness of no greater than 2% of the chordline, JavaFOIL data can be used from -10 to 10 degrees angle of attack. Therefore, when designing the flapping motion of the SSA it was important to keep our angles of attack within this range.

3.2.1 Methods for Geometric Optimization

After validating JavaFOIL against existing experimental data, we used it to determine the optimum airfoil shape for different flight conditions. At a Reynolds number of approximately 100,000 (Mars application) and 450,000 (Earth application), we varied five different wing shape parameters: thickness, thickness location, camber, camber location, and leading edge radius. We knew from the requirements of the wing material and also from basic aerodynamics that some of these parameters had little room for alteration. In order to keep the wing flexible and reduce the weight as much as possible we needed to keep the airfoil thickness to a minimum. Higher camber levels also yield lower stall angles and low Reynolds numbers make separation occur at lower angles of attack. This forced us to keep the camber level fairly low in order to maintain lift over a wide range of angle of attack.

Despite our confidence in JavaFOIL’s results we sent our optimized airfoil to K.M. Isaac to be tested using WIND, and Fluent, two more sophisticated computational
fluid dynamics packages. This provided us with data to compare against those produced by JavaFOIL.

WIND and Fluent are two computational fluid dynamics software programs that use Navier-Stokes equation solvers. Although these complex programs are more accurate, they take a significant amount of time to run. JavaFOIL, although less accurate, performs the analysis more efficiently. This allows the user to execute hundreds of tests in the time it would take to perform one test using WIND or Fluent. WIND and Fluent were used merely to check JavaFOIL’s results at critical angles of attack during the validation and optimization processes.

3.2.2 Results of Geometric Optimization

The geometric optimization was done by a controlled iterative process to keep all the parameters constant except the one being tested. This means that when the thickness percentage was being tested, the Reynolds number, camber percentage, camber location, thickness location, and leading edge radius all remained constant at a realistic value. The first round of tests was run at a Reynolds number of 100,000 to simulate the Mars environment.
Figure 28. Effect of Thickness on Lift Coefficient: Camber of 5%, Leading Edge Radius of .15%, Reynolds number=100,000

Figure 28 shows the lift coefficient clearly increases with thickness, but due to our material limitations, 2% of the chord length is as thick as the wing can reasonably be.

Figure 29. Effect of Thickness on Drag Coefficient: Camber of 5%, Leading Edge Radius of .15%, Reynolds number=100,000
In Figure 29, each thickness follows the same trend for drag coefficients, especially in the low angle of attack range. As the angle of attack increases, the drag coefficient also increases steadily, regardless of the thickness.

![Figure 30. Effect of Thickness on L/D: Camber of 5%, Leading Edge Radius of .15%, Reynolds number=100,000](image)

The lift-to-drag ratio follows the same trend as the lift coefficient in that they increase as the thickness increases, as shown in Figure 30. Again, we are limited to only 2% of the chord length.
A study of the effect of airfoil camber on Cl, Cd, and L/D ratios is presented in Figures 31, 32 and 33. In Figure 31, the peak lift coefficient occurs at 3 or 4 degrees angle of attack for cambers 1% - 5%, but at 7 degrees for the maximum camber of 6%. It is possible that JavaFOIL failed in predicting stall at this angle, so the higher angles of attack were disregarded. Although the angle of attack makes this graph inconclusive to in determining an optimum angle, it reveals that the lift coefficient increases as the camber increases.
Figure 32. Effect of Camber on Drag Coefficient: Thickness of 2%, Leading Edge Radius of .15%, Reynolds number=100,000

In Figure 32, the drag coefficient drastically increases past 6 degrees angle of attack.

Figure 33. Effect of Camber on L/D: Thickness of 2%, Leading Edge Radius of .15%, Reynolds number=100,000

The maximum L/D’s occur from 2-4 degrees angle of attack for all camber sizes, as shown in Figure 33.
The location of the maximum camber along the airfoil chord has a significant effect on airfoil performance. In Figure 34, 5% or 10% camber locations show linear variations of $C_l$ versus angle of attack while other camber locations show nonlinear trends. Figures 35 and 36 show the effect of angle of attack and camber location on $C_D$ and $L/D$. 

Figure 34. Effect of Camber Location on Lift Coefficient: Thickness of 2%, Camber of 5%, Leading Edge Radius of .15%, Reynolds number=100,000

Figure 35. Effect of Camber Location on Drag Coefficient: Thickness of 2%, Camber of 5%, Leading Edge Radius of .15% Reynolds number=100,000
In Figure 35, drag coefficients for the 20% and 30% camber locations remain low at lower angles of attack and Cd for all the camber locations increase significantly past 5 degrees.

In Figure 36, the highest L/D values come from the camber locations at 20% or 30% at around 4 degrees angle of attack.

In Figure 36, the highest L/D values come from the camber locations at 20% or 30% at around 4 degrees angle of attack.
In Figure 37, the leading edge radii of .1, .2, .3, .4% of the chordlength follow the same trend from 0-10 degrees angle of attack. Using a thin airfoil limits the size of the leading edge radius. Although any of these sizes of the leading edge radii could be implemented in the final design, structural stability of the airfoil would need to be studied.

![Graph showing the effect of leading edge radius on drag coefficient.]

**Figure 38. Effect of Leading Edge Radius on Drag Coefficient: Thickness of 2%, Camber of 5%, Camber Location of 20%, Reynolds number=100,000**

The drag coefficients for each of the various LER values follow the same trend, as shown in Figure 38.
In Figure 39, the leading edge radius creates the best performance around 2-4 degrees angle of attack with LER values of .1, .2, .3 or .4%.

In determining the best airfoil geometry, all the JavaFOIL tests were run using a Reynolds number of 100,000. This value is for Mars, the first desired atmosphere of deployment, which has a low density. The remainder of the parameters was chosen from previous experience with JavaFOIL and our intuition. The shape of our first optimized airfoil is shown below in Figure 40, along with the lift and drag coefficients in Figure 41. This geometry is shown below with a 2% thickness at 50% location, 5% camber at 5% location, and .15% leading edge radius at a Reynolds number of 100,000.
This geometry was sent to Dr. K.M. Isaac, who ran the same airfoil geometry in FLUENT. Isaac reported back with computational fluid dynamics graphs of pressure, flow visualization and velocity vectors, as well as $C_x$ and $C_y$ (normal and axial) components. To calculate the lift and drag coefficients, we used the following equations:

$$C_l = C_y \cos(\alpha) - C_z \sin(\alpha) \quad (8)$$

$$C_d = C_y \sin(\alpha) + C_z \cos(\alpha) \quad (9)$$

These lift and drag coefficients from FLUENT are compared to JavaFOIL’s results in Figure 42. The lift coefficients at zero degrees angle of attack and fifteen degrees angle of attack were significantly lower in JavaFOIL. Also, at ten degrees angle of attack, JavaFOIL’s lift coefficient was slightly lower than Fluent’s prediction, which is favorable since the maximum lift coefficient is around ten degrees in both JavaFOIL and Fluent.
The drag coefficient in Fluent was significantly higher than JavaFOIL’s. Overall, when it comes to the performance ratio of L/D, JavaFOIL predicted higher L/D’s than Fluent, as seen below in Figure 43. This is because dividing the lower lift coefficient by a significantly lower drag coefficient in JavaFOIL produces a larger L/D value.
Above, in Figure 44, is a graph of the static pressure distribution that K.M. Isaac produced from his Fluent test. This shows the airfoil at ten degrees angle of attack at a Reynolds number of 100,000. The blue and green on the upper surface indicate low pressure, while the oranges and reds on the lower surface indicate much higher pressures.

Since the spreadsheet was calculating a higher drag than expected with this geometry, we re-designed the shape to incorporate a lower drag coefficient. We focused on finding the smallest drag coefficient with the greatest lift coefficient in the configurations. The new optimized geometry was created by further studying the optimized parameters and pairing them together in configurations for JAVAFOIL. The airfoil shape and performance characteristics are given below in Figure 45.
With the first optimized geometry, we used a high angle of attack of about nine degrees because that is where the highest lift coefficient occurred. Unfortunately for our design, nine degrees experienced an extremely high drag coefficient that would significantly hinder thrust production in flight. In this new geometry, however, we use a lower angle of attack of around four degrees because that is where the lowest drag coefficient occurs as well as a reasonable lift coefficient. Also the camber location was moved back to 20% from its original placement of 5%, to avoid extra drag. The coefficients improved and can be seen in Figure 46.
After testing the feasibility of this airfoil shape in the environment of Mars using Tony Colozza’s propulsion analysis spreadsheet, we preliminarily determined that the operation of the aircraft was marginal, even with the lower drag and higher L/D ratio. The atmosphere of Mars proved to be too thin for this flapping aircraft, requiring significantly more power than is available. As a result, we performed an additional optimization study using JavaFOIL focusing on the Earth environment. The approximate Reynolds number for Earth applications was calculated to be 450,000, which not only improved the performance characteristics but also enhanced the accuracy of JavaFOIL. The following graphs show the effects of important geometric parameters on the lift coefficient, drag coefficient and L/D versus angle of attack at Reynolds number 450,000.
Figure 47 shows the maximum lift coefficient increases with thickness percentage, however due to material constraints we are limited to 2%.

Figure 47. Effect of Thickness on Lift Coefficient: 5% Camber, .15% Leading Edge Radius, Re=450,000.
Figure 48. Effect of Thickness on Drag Coefficient: 5% Camber, .15% Leading Edge Radius, Re=450,000.

The drag coefficients for all three selected thicknesses are relatively the same, as seen in Figure 48. At very high angles of attack, the higher thicknesses yield higher drag coefficients, but our focus is on the lower angles of attack where the highest lift coefficients and L/D ratios occur.
In Figure 49, L/D over a large range of angles of attack increases with thickness, similar to the lift coefficient. Again, we are limited to 2% thickness, which shows a maximum L/D of almost 80 at angles of attack from 0 degrees to 2 degrees.
In Figure 50, the lift coefficient increases as the camber percentage increases. Increasing the curvature with the lower Reynolds number had similar results, but with lower Reynolds numbers we were restricted to a lower camber to avoid stall. At this Reynolds number of 450,000 we are able to increase the camber to 10% and increase performance significantly.
The drag coefficients again follow a trend that is independent of maximum camber. As can be seen in Figure 51, for each camber amount, the drag coefficient reaches its minimum at low angles of attack, and then rises fairly linearly through our working range.
Figure 52. Effect of Camber on L/D: 2% Thickness, .15% Leading Edge Radius, Re=450,000.

In Figure 52, the performance ratio of L/D appears to increase with camber percentage increase. Since we are now able to use higher cambers, we chose the highest camber value of 10% for the optimized airfoil
In Figure 53, the effect of maximum camber location on the lift coefficient is presented. The lowest drag coefficients occurred between 0 and 5 degrees, as shown in Figure 54. The maximum lift coefficient at 30 and 40% location occurs in the same angle of attack range, as seen in Figure 53. The highest overall lift coefficient occurs at a camber location of 80% with a negative angle of attack, which is not our primary range of angle of attack. Therefore, the larger camber location percentages were not considered.
Figure 54. Effect of Camber Location on Drag Coefficient: 2% Thickness, 10% Camber, .15% Leading Edge Radius, $Re=450,000$.

In Figure 54, the drag coefficient curves are similar except for the extremes of 10% and 80% locations.

Figure 55. Effect of Camber Location on L/D: 2% Thickness, 10% Camber, .15% Leading Edge Radius, $Re=450,000$. 
In Figure 55, the effect of maximum camber location on the L/D ratio is presented. In the previous two figures, the greater camber location percentages have extreme lift and drag coefficients which cause the L/D values to be higher than would be expected in the real world. The camber percentages between 20% and 40% are the most realistic and correspond to the optimized values from the previous two figures.

The optimized airfoil for a Reynolds number of 450,000 was designed to have a thickness 2% of the chord length with a camber of 10% of the chord length. The maximum thickness location was at 30% and the maximum camber location was at 40%. The leading edge radius was .1% of the chord length to stabilize the thickness. The optimized airfoil geometry, as well as L/D and the lift and drag coefficients, are shown below in Figures 56, 57 and 58.

![Optimized Airfoil Geometry for Earth Application](image)

**Figure 56.** Optimized Airfoil Geometry for Earth Application: 2% Thickness at 30% Location, 10% Camber at 40% Location, .1% Leading Edge Radius, Re=450,000
From all of our optimization trials, it was clear that when the Reynolds number is 450,000 we see a much better airfoil performance in between 0 and 5 degrees, compared to the lower Reynolds number of 100,000 which can be seen in Figure 46. We chose three degrees to be our optimum effective angle of attack for the flight of the SSA. If the
aircraft designed is capable of flying on Earth, then the applications on planet Venus will likely succeed due to the higher atmospheric density and increased Reynolds numbers. Because of this, an optimized airfoil for Venus will not be made at this time.

**3.3.1 Wing Flap Optimization**

In order to model the wing flapping motion, a quasi-steady blade element approach which effectively freezes the wing motion at each instance of time was used. Figure 59 illustrates the motion of the flapping wing. At each of the one hundred phases of the wing flap motion, the wing is also split into one hundred sections of equal span from root to tip. By summing the lift, thrust and drag for all of the one hundred sections along the span we found the lift, thrust, and drag on each individual fixed wing during the flapping motion. By multiplying these values by the incremental time the wing spent at each stage along the flapping motion, summing all of the individual fixed wings, and dividing by the time elapsed for one flapping motion we found the average lift, thrust, and drag experienced during one flapping cycle. Further discussion of the calculations of lift, thrust, and drag can be found later in Section 3.3.2.

![Flapping Motion](image)

**Figure 59. Flapping Motion**
The developed spreadsheet also calculated how much power the wing would consume in undergoing the prescribed flapping motion as well as how much solar power was available. By comparing the required power to the available power we can determine the achievable flight regimes for the Solid State Aircraft in different planetary atmospheres.

Power consumption was calculated based on the motion of each section of the wing along its flapping motion. A force is required to accelerate the mass of each individual wing section. The force needed to accelerate the mass added to the force required to overcome aerodynamic drag resulted in the total power consumed.

The available power data came from the existing power analyses developed by Tony Colozza. These analyses provided the environmental characteristics for Earth, Mars and Venus. Solar intensities can vary significantly in different environments on different days. The spreadsheet allows the user to choose the planet, day of the year, and time of day. Using Tony’s previous studies and integrating them with our flapping motion and optimized airfoils, we determined how much excess power would be available over the course of one flapping motion.

The spreadsheet calculated the lift, thrust, drag, power usage, and available power based on a number of independent variables. The flap duration, the flapping angle, the airfoil characteristics during the upstroke and downstroke, and the flap curvature were some of the parameters of the flapping motion that were altered to determine the most efficient flap motion. Our final goal was to determine the potential flight regimes of the SSA.
3.3.2 Details of Spreadsheet Calculations

The basic principle in the quasi-steady blade element approach that calculated the lift, thrust, and drag on a flapping wing was to divide the wing into one hundred distinct smaller sections along the span and also divide the flapping motion into one hundred different fixed wings during one flapping motion. Since both wings moved in an identical fashion we calculated the lift, thrust, and drag on one wing and multiplied the result by two.

The forces on each section of the wing were summed to find the total force on each wing. Those sums were multiplied by the approximate time the wing spent at each position divided by the time for one flapping motion to find the average lift, thrust, and drag during one cycle of the flapping motion.

The first force we calculated was the lift. Equation 10 gives the lift of an airfoil where $L'$ and $D'$ are the lift and drag generated on the stationary wing. Because the wing is flapping the angle $\beta$ relates these two forces to the true vertical lift force $L$, shown in Figure 60.

$$L = L' \cos \beta + D' \sin \beta$$  \hspace{1cm} (10)
Figure 60a/b. Effect of Flapping Wing on Lift and Drag
The wings were averaged in order to find the average lift that occurred during one flapping motion, resulting in Equation 11.

$$AverageLift = \frac{1}{T_f} \sum_{0}^{F} \sum_{0}^{S} (L\cos{\beta} + D\sin{\beta})T_w$$

In this equation S denotes the number of sections along the span of the wing and F denotes the number of individual wings the flapping motion was broken down into. Both of these values are one hundred in our case. $T_w$ denotes the time that the individual wing stays at each point along its flapping motion. When divided by the total time for one flapping motion, $T_f$, the equation yields the average lift over one flapping motion.

$L'$ for one section of an individual wing can be derived from Equation 4 and is given by equation 12, where l is the effective chordlength.

$$L' = C_i \frac{1}{2} \rho V_\infty^2 l S \cos{\theta} \cos{\beta}$$

The effective velocity (Figure 61), $V_\infty$, is the hypotenuse of the triangle created from the flight velocity and the flapping velocity and is calculated using Equation 13.

\[ V_\infty = \sqrt{V_{flight}^2 + V_{flap}^2} \]

The flight velocity is the horizontal velocity of the aircraft. To calculate this number we used the lift coefficient equation and made a few assumptions. By fixing the wing in its horizontal position we determined what velocity it would take to keep the
wing at steady level flight. The lift created must equal the aircraft’s weight. Rearranging the lift coefficient in Equation 14, we solved for the flight velocity (Equation 15).

\[ Cl = \frac{L}{\frac{1}{2} \rho v^2 S} \]  

(14)

\[ V_{flight} = \sqrt{\frac{W}{\frac{1}{2} \rho ClS}} \]  

(15)

The flapping velocity is caused by the wings movement up and down in order to create the desired thrust. During this flapping motion, the wing moves faster towards the tip than it does towards the root of the wing. However, an individual section along the span is not going to have a constant velocity. When the wing section reaches the top or bottom of its motion it must change directions. This requires a force, and consequently an acceleration. The flapping velocity at the top and bottom of the flapping motion is zero. As it accelerates it passes through the horizontal position where it begins to decelerate in order to change directions. When the wing is horizontal its velocity is at its maximum value, as seen in Figure 62.
Figure 62. Flapping Velocities
A sine function was incorporated in order to represent this velocity variation at each section along the span of the wing. A graph of the normalized velocity and acceleration over the course of one flapping motion is shown below in Figure 63.

**Figure 63. Normalized Flapping Velocity and Acceleration for One Flapping Motion**

The equation to describe the sinusoidal velocity function is as follows:

\[
FlappingVelocity = MaxVelocity \sin\left(\frac{2\pi}{100}\right)
\]

\[
FlappingAcceleration = -MaxAcceleration \cos\left(\frac{2\pi}{100}\right)
\]  

(16)

To find the maximum velocity we initially approximated the distance each wing section traveled as an arc of a circle, using the sweep angle input and the time per flap cycle. Both are input from the user. By knowing the time and the distance the wing traveled, the average velocity over one flapping motion was derived. Integrating the sine
function, it was determined that the average value of the sine function is equal to the maximum value multiplied by $2/\pi$. So to find the maximum value, we multiplied the average by $\pi/2$.

After further research and refinement we came to the conclusion that this approach to approximate the velocity was not as accurate as we first anticipated. In Figure 64, the blue shaded area is the actual area swept by the moving parabolic wing and the red area is the additional area swept by the wing by approximating the wing as a straight line moving through an arc. This substantially overestimated the distance traveled by each segment of the wing and therefore overestimated the flapping velocity.

![Figure 64. Motion Comparison for Parabolic versus Straight Line Wing Motion](image)

To account for this we used a different sweep angle, $\delta$, along the span. By doing so every section of the wing was still passing through an arc, but a different sweep angle
for each section greatly improved the accuracy of the distances traveled. This angle increased along the span and was dependant upon the maximum deflection angle $\theta$ that was input by the user. The deflection angle $\theta$ is shown in Figure 68 and will be discussed shortly. This input dictated how large or small the flapping motion was by changing the parabolic coefficient.

![Sweep Angle $\delta$ vs Span for Various Maximum Deflection Angles ($\theta_{max}$)](image)

Figure 65. Sweep Angle $\delta$ vs Span for Various Maximum Deflection Angles ($\theta_{max}$)

As seen in Figure 65, when the maximum deflection angle is 75 degrees or less, the sweep angle along the span appears to increase approximately linearly. Because there is little chance the deflection angle will be any larger than 75 degrees, a maximum sweep angle is the only thing necessary to calculate the sweep angle at any point along the span. Calculations were made to find this maximum sweep angle for a number of parabolic coefficients in order to run a linear regression to determine a representative polynomial function for the maximum sweep angle (Figure 66) being dependent only upon the maximum deflection angle theta. Once the maximum sweep angle is known, the linear variation of the sweep angle along the span is easily calculated.
With a more accurate sweeping angle we could much more accurately calculate the distance each wing section traveled and consequently the flapping velocity of the flapping motion.

Angle $\theta$ depends on the overall shape of the wing as it flaps. A parabolic profile was assumed. The equation of a parabola is

$$y = \frac{x^2}{4a}$$  \hspace{1cm} (17)

where $a$ is the parabolic coefficient and the $y$-position of the focus. Looking at the wing head on (in the flow direction), portions of the lift are also being deflected inwards (and outwards when the wings are pointing downwards). From Figure 67, the vertical portion of the lift (L) is multiplied by the cosine of the angle between L and L’. $\theta_{\text{max}}$ is the maximum deflection angle entered into the spreadsheet.
The angle $\theta$ is calculated for every section along the span of the wing using Equation 18, where $x_i$ is the portion of the span the lift is being calculated on, $R$ is the entire span of one wing, and $\theta_{\text{max}}$ is the maximum deflection angle.

$$\theta = \tan^{-1}\left[\frac{x_i \tan \theta_{\text{max}}}{R}\right]$$  \hspace{1cm} (18)

3.3.3 Wing Motion Details

We modified the upstroke to differ from the downstroke by varying the camber and the effective angle of attack. If the upstroke was the same as the downstroke the rearward horizontal force on the upstroke would negate the forward horizontal force on the downstroke, resulting in zero thrust. In consideration of this we incorporated a variable camber. On the downstroke the camber remains constant. On the upstroke the camber starts at the maximum camber, reaches its minimum camber at the horizontal position (which is negative in our case) and returns to the maximum camber at the top position. This minimum camber depends on its distance from the root. The root of the wing produces mostly lift and very little thrust. Similarly, because the wing produces a larger amount of thrust towards the tip, its minimum camber reaches a negative camber. This allows the tip of the wing to produce positive thrust on the upstroke. This forward
thrust is at the expense of some of the lift, requiring a faster glide velocity to maintain the appropriate lift. However, it was clear that the thrust benefit far outweighed the cost in lift reduction. Figure 68 shows how the camber changes along the wing’s flapping motion near the root (1/100 the wingspan) out to the wingtip (100/100 the wingspan).

Figure 68a/b. Variation of Camber at Different Flap Phases

Drop Factor = 1

Drop Factor = 2
Additionally, the spreadsheet allows for the user to determine how quickly the camber falls towards its minimum value. The camber varies according to a sine function. The sine function is multiplied by a scalar value called the drop factor. Figure 68a shows a drop factor of one and Figure 68b shows a drop factor of two.

During the downstroke the effective angle of attack remains at the same value. This value was predetermined through our airfoil optimization to give us the best lift to drag ratio. Our intent in doing this was to provide maximum lift and thrust during the downstroke. However, on the upstroke this also had to change in order to minimize the negative thrust. To take this into consideration, during the upstroke we reduced the effective angle of attack in the same fashion that we reduced the camber level. When the camber of the airfoil was at its maximum the effective angle of attack was also at its maximum. When the camber was at its minimum level, the effective angle was also at its minimum, both of which were negative.

To improve accuracy of the lift coefficient over the range of cambers and angles of attack we performed a linear regression that yielded a polynomial function. By plugging in the camber as the independent variable the equation yielded the appropriate lift coefficient.

3.3.4 Drag Analysis

The drag on the airfoil, which will be calculated in another section, also had an impact on the lift. During the downstroke, the drag contributed to the overall lift, whereas during the upstroke the drag detracted from the overall lift as shown in Figure 60b.

The area in this case is the chordlength l multiplied by the incremental span s. The chordlength varies across the span, having a maximum at the base and slowly becoming
zero at the tip as seen in Figure 69. The value at each span distance is taken off a table in the spreadsheet. The incremental span remains constant at 1% of the input wing length.

![Chordlength Distribution Along Span from Root to Tip](image)

**Figure 69. Chordlength Distribution Along Span from Root to Tip**

Adding each piece along the span results in the total lift for one point of the wing’s flapping motion.

\[
\sum_{0}^{s} Cl \frac{1}{2} \rho V^2 l_s \cos \theta \cos \beta + D \sin \beta
\]  

(19)

To calculate the average lift for one flapping motion one would then multiply each wing by the incremental time, sum these lift numbers, and divide it by the time for one flapping motion as shown in Eq(20).

This incremental time was approximated using a cosine function. When we broke the flapping motion into individual wings we broke it up according to the flap angle. Because the wing spends more time at the top and bottom of the flap cycle (due to the
slower velocities) it would have been less accurate to have averaged the individual wings.

Although not entirely representative of the actual time the wing spends at each increment, the cosine function gives a good approximation. Equation 20 shows the average lift after combining Eq(11) and Eq(12).

$$\text{Average Lift} = \frac{1}{T_F} \sum_{0}^{E} \sum_{0}^{S} (\text{Cl} \left[ \frac{1}{2} \rho V \infty^2 ls \cos \theta \cos \beta + D' \sin \beta \right]) T_w$$  \hspace{1cm} (20)

From Figures 17 and 18 one can see that the thrust the wing produces is the horizontal component of $L'$. The thrust equation is therefore:

$$T = L' \sin \beta$$  \hspace{1cm} (21)

To derive the equation for the average thrust produced by the flapping wing during one flapping period we used a method similar to how we derived the lift equation.

$$\text{Average Thrust} = \frac{1}{T_F} \sum_{0}^{E} \sum_{0}^{S} \text{Cl} \left[ \frac{1}{2} \rho V \infty^2 ls \sin \beta T_w \right]$$  \hspace{1cm} (22)

Notable differences between the final lift and thrust equations include the use of a cosine function for angle beta in order to find the horizontal component of $L'$. Also, angle $\theta$ does not appear in the equation. This is because the inward and outward deflections of the lift have no effect on the thrust created by the wing.

$$D = D' \cos \beta$$  \hspace{1cm} (23)

Equation 23 is the drag force as seen in Figures 16 and 17. The drag force is comprised of the induced drag force and the parasitic drag force, taken from Eq(2) and Eq(5), resulting in the drag force on one section of the wing being:

$$D = (Cd + C_d) \frac{1}{2} \rho V \infty^2 ls \cos \beta$$  \hspace{1cm} (24)
After substituting the equation for the induced drag and summing each section of the wing and averaging each wing similarly to both the lift and thrust equations the average drag on the wing over one flapping motion is shown is Eq(25).

\[
\text{Average Drag} = \frac{1}{T_F} \sum_{i=0}^{E} \sum_{j=0}^{S} T_{ij} \left( C_d \frac{C_l^2}{\pi A R} \right) \frac{1}{2} \rho V \infty^2 s \cos \beta
\]  

(25)

The power the aircraft required for one flapping motion was based on the power required to move the mass of the wing and the power required to overcome the drag experienced by the flapping motion.

Power is the product of a force and a velocity.

\[
P = FV
\]

(26)

To calculate the power consumed by the wing mass movement we again broke the wing length into one hundred pieces. By tracking the mass of each section we were able to find out how much power each required during its flapping motion.

\[
F_s = MA
\]

(27)

Substituting for the force in the power equation yields:

\[
P_s = M_s A_s V_s
\]

(28)

The mass of each section was calculated by multiplying the area density, \( \rho_{\text{area}} \), of the aircraft layers (IPMC, PV array, and Li Batteries) by the chord length, \( l \), and the incremental span length, \( s \), of one percent of the wing section length.

\[
M_s = \rho_{\text{area}} s l
\]

(29)

The velocity of each section varied throughout the flapping motion. An average velocity was calculated similarly to how the flapping velocity was calculated. We approximated the path traveled by each section to be a circular arc, with an increasing
sweep angle from the root to the tip. By calculating the distance traveled and dividing it by the flap cycle we were able to find the average velocity.

\[ V_{avg} = \frac{4\pi R_s \delta}{360T_F} \]  \hspace{1cm} (30)

In this equation \( R_s \) is the radial distance from the root to the span section, angle \( \delta \) is the sweep angle, and \( T_F \) is the time period for one flapping motion. The distance is multiplied by two to incorporate the distance traveled on both the downstroke and the upstroke.

To find the magnitude of the average acceleration we took the maximum velocity the wing section experienced and divided it by the change in time. The maximum velocity, since we used a trigonometric function, was the average velocity multiplied by \( \frac{\pi}{2} \). The time it took to achieve this velocity was one quarter of the flapping period, which led the acceleration of the mass to be as follows:

\[ A_s = \frac{8\pi^2 R_s \delta}{360T_F^2} \]  \hspace{1cm} (31)

Combining these yields the power required for one section.

\[ P_s = \frac{\rho_{area}sl2\pi}{T_F} \left( \frac{4R_s \delta}{360T_F} \right)^2 \]  \hspace{1cm} (32)

In order to find the power consumed by both wing lengths we summed each individual section and multiplied it by two.

\[ P_{Max} = 2\sum_{0}^{8} \frac{\rho_{area}sl2\pi}{T_F} \left( \frac{4R_s \delta}{360T_F} \right)^2 \]  \hspace{1cm} (33)

The power consumed by the drag force was:

\[ P_{Drag} = DV_{Avg} \]  \hspace{1cm} (34)
The drag force was taken from previous calculations and multiplied by the average velocity. To find the total power consumed due to the drag force we had to sum the power consumed on each section of the wing, multiply it by its incremental time $T_W$ and divide by the flap cycle $T_F$.

$$P_{\text{drag}} = \sum_{0}^{F} \sum_{0}^{S} D \frac{4\pi R_s \delta}{360T_F} T_W \frac{1}{T_F}$$

(35)

Combining the two sources of power consumption yields the total power consumption of the aircraft.

$$P_{\text{total}} = \frac{2}{\eta} \sum_{0}^{S} \sum_{0}^{S} \rho_{\text{area}} s l 2\pi \left( \frac{4R_s \delta}{360T_F} \right)^2 + \sum_{0}^{F} \sum_{0}^{S} D \frac{4\pi R_s \delta}{360T_F} T_W \frac{1}{T_F}$$

(36)

The IPMC transforms the electrical energy stored in the batteries into the mechanical energy of the moving wing. However, this process is not efficient, requiring an increase in the power produced. The projected efficiency of the IPMC material is approximately 20%. This means of the power supplied from the batteries, only 20% of the power is converted into the wing movement, with the other 80% wasted. To account for the inefficiency we divided the total power required by the efficiency, $\eta$, essentially increasing the required power by a factor of five.

It was critical to determine whether or not the aircraft would be able to generate enough power to undergo its flapping motion. The solar cell’s generation of electricity depended on the solar cell’s efficiency, the planet it was flying on, the altitude, the latitude of its position, the day and time of year, and also the flapping motion of the wing.

Tony Colozza had previously performed analysis of the power production capabilities of the SSA. Combining his analyses with our updated flapping profile we
were able to compare the power production and consumption to determine the aircraft’s potential in different planetary atmospheres.

### 3.4.1 IPMC Efficiency
As an additional task, we wanted to determine the efficiency of the IPMC material in the laboratory. In the spreadsheet, the power available is calculated using an efficiency of 20% for the IPMC. Samples from the University of New Mexico fabricated by Dr. Mohsen Shahinpoor were tested to measure the required power and efficiency.

In order to calculate the efficiency of the IPMC, we set up a laboratory experiment that connected the IPMC to a voltage generator that supplied 6V to the circuit. Two probes from an oscilloscope were connected to the circuit to measure the voltage, current and corresponding power.
The power measurement was read directly from the oscilloscope (Figure 70), as was the period for an entire flapping motion.

![Figure 70. Oscilloscope Readings](image)

We also determined the ideal amount of power the wing would need to move through the proper distance had the IPMC been 100% efficient. Since we knew the mass and dimensions of the wing, as well as the fact that gravity was the only force acting on it, we were able to calculate the theoretical power required to move the wing. By comparing the theoretical power and the measured power, we determined the efficiency of the IPMC.

Determining the theoretical power required to move the mass of the wing also required measurement of the distance traveled by the wing. We did this by mounting
millimeter graphing paper on a backdrop just behind the flapping wing as shown in Figure 71.
A camera helped us to magnify the process on a television screen so we could mark the distance on the televised graph paper (Figure 72).

![Figure 72. Graph Paper Measurement on Television Screen](image)

We multiplied the mass by the area by the force of gravity to determine the force. Multiplying this force by the distance the wing traveled resulted in the work done by the system. Dividing the work by the time it took for one flap from top to bottom equaled the power it should take to move that material. See a sample calculation below.
This test was run on another sample piece to ensure accuracy. The next sample resulted in a 0.0011% efficiency. Based on our tests, the spreadsheet efficiency of 20% seemed like an unreasonable number. We have only worked with the material for a short time. Philip Jenkins admitted that he had a difficult time trying to get the material to flap and destroyed multiple samples, whereas Moshen Shahinpoor, at the University of New Mexico, has made significant progress with the material. Regardless, the discrepancy between our simple calculations and the projected efficiency was strikingly large. It will be necessary to run additional tests to determine an accurate efficiency.
Chapter 4: Conclusions and Recommendations

4.1 Flight Envelope on Earth

The low altitudes of Earth produce a Reynolds number of approximately 450,000 which creates much higher performance characteristics. The power available in this altitude range of 0km – 15km is significantly higher than the power available on Mars, as it is farther away from the sun. From running tests with the spreadsheet, the Solid State Aircraft was able to fly in several different flight regimes.

On the spreadsheet, we chose the latitude to remain at a constant for each run. In Tony Colozza’s previous work\(^1\), he represented the four seasons graphically in regards to latitude and time of day to determine the greatest power available. Past zero degrees latitude, the summer months during the middle of the day all create the maximum power so in the spreadsheet, we used the month of June and determined the greatest power available from the sun is at 23 degrees latitude. The second and third runs were with 60 degrees latitude and 85 degrees latitude, respectively, both with significantly less power available.
In Figures 73 and 74, graphical representations of the power available by month of the year and latitude are shown. The power available by month was determined by setting the spreadsheet to 10km, 23 degrees latitude and the 21\textsuperscript{st} day of the month to remain constant, while the month was tested from 1-12 and the power available was recorded.

![Figure 73. Power Available by Month of the Year](image)

The power available by latitude was determined by setting the spreadsheet to 10km, the month of June and the 21\textsuperscript{st} day of the month to remain constant, while the latitude was tested from 0 – 90 degrees and the power available was recorded.
For determining the flight regimes that the Solid State Aircraft can successfully fly in, we entered in the latitude, month and day of the year to remain constant for the entire length of the test. We began at altitude of 0 km and read the calculations on the spreadsheet for power consumed, lift, drag, thrust and L/D. If the power consumed was greater than the power available, the aircraft would not be capable of flying at that altitude. However, if the power consumed was less than the power available, the lift was greater than the weight, and the thrust was greater than the drag, then the aircraft was indeed capable of flying in that regime.

For each altitude, there are characteristics that the spreadsheet can vary in order to increase the performance at a cost of more power. The glide velocity could be increased in order to gain more lift, the drop factor (the rate at which the camber flattens out on the upstroke) could be increased to gain more thrust, and the respective time for the upstroke and downstroke could be increased or decreased as needed to satisfy the requirements for successful flight.
The first run, completed at 23 degrees latitude (optimal), resulted in successful flight from 0km – 12km, most consistent up to 11km (see Figure 75 below).

![Figure 75. Power Versus Altitude for Planet Earth at 23 degrees latitude on Summer Solstice](image)

Past 12km, the aircraft was unable to create enough power to achieve the sufficient amounts of lift and thrust. Also important to note is the L/D ratio, as it is almost constant in the range of successful flight (see Figure 76).

![Figure 76. Lift-to-Drag Ratio Versus Altitude for Planet Earth at 23 degrees latitude on Summer Solstice](image)
At a latitude of 60 degrees, much less power was available than the optimal condition of 23 degrees latitude in the previous case (see Figure 77).

Figure 77. Power Versus Altitude for Planet Earth at 60 degrees latitude on Summer Solstice

The aircraft can now only maintain enough lift and thrust up to 11 kilometers in altitude because the power available is not as high. The lift-to-drag ratio is also slightly less in this flight regime for the same reason (Figure 78).
Next we tested the aircraft at a latitude of 85 degrees, which only has 600W available for power according to Figure 74. The SSA was unable to fly higher than 7 kilometers high at this latitude as a result of power requirements, as shown in Figure 79.
The lift-to-drag ratio is also slightly lower again than the previous cases of lower latitudes. It is also apparent that when the aircraft is capable of flying, in all three cases, L/D is almost constant up until the point where it can no longer fly (Figure 80).

![Figure 80. Lift-to-Drag Ratio Versus Altitude for Planet Earth at 85 degrees latitude on Summer Solstice](image)

The three flight regimes just presented were all run on the Summer Solstice. To provide a contrast, the next three flight regimes were run on the Winter Solstice to see how the opposite seasons affect the performance of the vehicle. In order to directly compare the seasons, we tested the aircraft at the same latitudes, however at 85 degrees (which is above the artic circle) there was zero power available so we included 0 degrees latitude as winter’s optimal latitude.

Figure 81 shows the power available compared to power required for altitudes 0 – 15 kilometers at 0 degrees latitude. This flight regime is very similar to the run on the Summer Solstice at 23 degrees latitude as they are both taken at the optimal latitude where the most power is available. The available power is almost identical and they are both capable of flying up to 11 kilometers.
The main difference between the two seasons in this case is that the L/D ratio is significantly lower in the wintertime (see Figure 82). The range past eleven kilometers altitude shows an elevated L/D ratio, which can be disregarded, because the ratio was shown to increase with an unavailable amount of power. Since the available power at those altitudes is significantly lower than the power required to create those lift and drag values (as shown in Figure 81), the change in trend would not in fact occur.
The run that was completed at 23 degrees latitude, however, shows more of a drastic difference and resembles the summertime run at 60 degrees latitude (see Figure 83). The SSA is capable of flying in this flight regime up to 10 kilometers high.

The L/D is also similar to the 60 degree latitude run from summertime (Figure 84).
The last run for planet Earth was at 60 degrees latitude on the Winter Solstice. This flight regime had less than 100 W of available power and therefore the SSA is incapable of flying in this environment (Figure 85). In order to satisfy the lift and thrust requirements to sustain flight in this environment, the aircraft would need to complete the entire flap cycle in one second, which requires significantly more power than 100W.
In conclusion, the Solid State Aircraft is capable of flight on Earth as long as it flies in a region with sufficient power. Flying at 60 degrees latitude in winter would not be possible, however flying at 5-7 kilometers at 15-35 degrees latitude makes flight possible. In the future, if more improvements in solar cell technology and thin-film lithium ion batteries become available, the efficiency of the entire aircraft will increase, thus increasing the performance in these flight regimes.

4.2 Flight Envelope on Mars

After performing many feasibility checks on the airfoil for the Mars atmosphere, we concluded that at the present time the application of the Solid State Aircraft will not succeed on Mars. The thin atmosphere required an extremely short flap cycle to produce the appropriate lift and thrust. This exponentially drove the power requirement past the power available by an order of magnitude.

![Figure 86. Power Versus Altitude for Planet Mars at 0 latitude on Summer Solstice](image)

Figure 86 shows the optimum power available in blue, which is nonzero, but very small in comparison to the power consumed shown in pink. The power required to create
enough lift and thrust to overcome the weight and drag of the aircraft exceeds the power available significantly. In order to achieve sufficient lift and thrust, the flap cycle time was drastically reduced to only .35 seconds on the downstroke and .22 seconds on the upstroke. The required velocity was almost 100 m/s and the drop factor was extremely large.

Just as with planet Earth, we compared the Summer Solstice data to the Winter Solstice data to determine how the seasons affected the flight plan. As opposed to planet Earth, though, the power available did not change with the season. Figure 87 shows the Power versus Altitude for Mars on the Winter Solstice, with the optimum power available in blue, again nonzero, and the power consumed in pink.

![Figure 87. Power Versus Altitude on Planet Mars at 0 latitude on Winter Solstice](image)

The clear conclusion for the Mars flight regime is that the Solid State Aircraft is incapable of flight at this time. In the future, it may be possible for this application to work on Mars, but many improvements must take place. The efficiencies of the IPMC and solar cells must increase. Also, a more refined aerodynamic analysis of the flapping motion could improve conditions.
4.3 Flight Envelope on Venus

Our initial intuition was to fly above the dense clouds of Venus which are 66 kilometers high. The problem with this is that the density above the clouds of Venus is prohibitively low. Thrust is the main problem in this area, because at 70 kilometers, the solar intensity is very high and the power available is great. To generate enough thrust to overcome the drag, however, uses more power than is available.

The planet of Venus was tested on the Summer Solstice and also the Winter Solstice to compare the seasons.

![Figure 88. Power Versus Altitude for Planet Venus on Summer Solstice](image)

Figure 88 shows that the SSA would be capable of flying in this flight regime from 4 – 65 kilometers. This range is misleading for the thick cloud layer on Venus that reaches up to an altitude of 66 kilometers, blocks much of the light that would be necessary to power the aircraft. Figure 89 shows that from 4 – 50 kilometers (inside the cloud layer where the atmospheric density ranges from 10-70 kg/m^3) the L/D ratio is approximately 16, much higher than the other planets. This is because the wings can flap
more slowly and the dense atmosphere creates lift and thrust with less power required.
The aircraft is still capable of flying in the upper layer of the dense clouds but it requires
more power, and as soon as the aircraft leaves the cloud layer at 66 kilometers, flight is
impossible. Although there is much power available, the density is too low and requires
too much power to sustain lift and thrust.

Figure 89. Lift-to-Drag Ratio Versus Altitude for Planet Venus on Summer Solstice
Analysis of the winter solstice case yielded similar results. Figure 90 shows that at 66 kilometers, the power consumed surpasses the power available.

![Power Versus Altitude for Planet Venus on Winter Solstice](image)

According to Figure 91, the L/D ratio again is approximately 16 from 10 – 40 kilometers and drops toward the upper layer of the dense clouds. The amount of power consumed on the Winter Solstice is slightly less than on the Summer Solstice.
Venus is the favored planet of the three studied to fly on because the lift-to-drag ratio is the highest and the power available compared to power consumed is advantageous. As with the other planets, increasing the efficiency of the components of the Solid State Aircraft will improve the performance of the aircraft in any given flight regime. The intense winds of Venus are an additional concern. As a result, we recommend additional study of the atmospheric conditions of Venus.

### 4.4 Aerodynamic Conclusions

With increases in the efficiencies of the solar cells and the IPMC, the Solid State Aircraft will be able to fly at even higher altitudes and a greater range of latitudes. A solar cell efficiency of 20% is projected in the next 10-20 years. This will effectively double the available power. In the near future however, there are a number of ways to improve the efficiency of the aircraft.

Modifications can be made to the structure of the aircraft to reduce the power consumption. The Lithium ion batteries could potentially be put in the fuselage of the
aircraft. Moving this mass from the wing to the fuselage means that the IPMC will no longer have to move the mass of the batteries, which provide approximately 20% of the wing’s current mass. In addition to the movement of the mass, the stiffness of the batteries also hinders the motion. It has been speculated that even though the thin film lithium-ion batteries are flexible, they are still not flexible enough to flap as part of an integrated system with the photovoltaic array and the IPMC. This would increase the power required to move the wing. Moving the battery to the fuselage would not only eliminate this problem, but reduce the amount of power required to accelerate the mass.

Initially, we assumed a parabolic wing motion. Ideally, a comparison of other profiles could provide insight into the most efficient flap profile.

We are confident that our aerodynamic and power analyses will help the Solid State Aircraft Team advance into further stages of design and development.
References


References [8], [9] and [10] were used to check our aerodynamics basic fundamentals.