The Pennsylvania State University
The Graduate School
Department of Aerospace Engineering

ROTORCRAFT PERFORMANCE ENHANCEMENTS
DUE TO A LOWER-SURFACE MINATURE EFFECTOR

A Thesis in
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by
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ABSTRACT

Although the application of advanced structures and intelligent control systems on helicopters has seen a dramatic increase over the past two decades, the overall performance of helicopters, relative to fixed-wing aircraft, is somewhat stagnant. This is due to many factors, one of them being the lack of innovative aerodynamic devices that can operate in the unique environment of a rotor. Research over the past 20 years has shown that in order to have a performance increase with the rotor, it must adapt, or “morph,” to the changing environment around the azimuth. One method that is being researched is the use of Miniature Trailing-Edge Effectors (MiTEs) on the blades of the rotor. MiTEs are an extension of the passive high-lift device, the Gurney flap. Gurney flaps are small flat plates, between 0.5 to 5 percent chord, fitted perpendicular to the airfoil surface at or near the trailing edge of a wing or rotor blade. A MiTE is an active Gurney flap, which can be used to actively control the lift and moment distribution on a rotor blade. MiTEs also have the advantage of having very low actuator loads compared to those of traditional trailing-edge flaps. Experimental and validated computational fluid dynamics research has been done on MiTEs and an unsteady aerodynamic model was created for MiTEs placed at the trailing edge. In this work, this model has been modified to account for a MiTE placed at the trailing edge up to the 85 percent chord position. This aerodynamic model has also been incorporated into a rotor performance code to predict the effect of MiTEs on rotor performance and explore their ability to extend the flight envelope of the RAH-66 Comanche. The maximum velocity of Comanche was shown to have the potential increase of 20 percent with the increased use of transonic airfoils as facilitated through the use of MiTEs on the outboard section of the rotor blades. Investigations were also made on increasing the service ceiling of the Comanche, which showed a potential improvement of 8 percent with the use of MiTEs.
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Chapter 1

Introduction

Since the late 1970s, most of the effort in rotocraft blade design has been diverted from aerodynamics to structural dynamics. This is due to the increased use of composite material blades, for which dealing with the flexibility of these blades has become a priority for all rotorcraft manufacturers. With the introduction of bearingless rotors, the structural design of blades with favorable aeroelastic characteristics has become even more important. All the while, the aerodynamic design of blades has received somewhat less attention.

Overall the performance of rotorcraft is determined primarily by that of the rotor, and this rotor performance is driven by its aerodynamic characteristics. Aerodynamic design methodology dictates the planform design of the rotor blades are determined simultaneously with specially designed airfoil sections. In order for rotor performance to improve, a major change must take place in rotor design efforts to re-emphasize the importance of aerodynamics. Most basically, a rotorcraft airfoil must address the following considerations [1]:

1.) A high maximum lift coefficient, $c_{l_{max}}$. This allows a rotor with lower solidity and lighter weight. This will permit flight at high rotor thrusts and under high maneuver load factors.

2.) A high drag divergence Mach number, $M_{dd}$. This permits flight at high forward speeds without prohibitive power loss or increase in noise levels.

3.) A good lift-to-drag ratio over a wide range of Mach numbers. This gives the rotor a low profile power consumption and low autorotative rate of descent.

4.) A low pitching moment. This helps minimize blade torsion moments, minimize vibrations, and keep control loads to reasonable levels.
To an airfoil designer, these rules are conflicting and create a situation where they cannot all be simultaneously achieved with the use of a single element airfoil. What can be done and has been during the airfoil design process is compromising where one criterion can be maximized while not having a drastic effect on another. This traditional design process is further complicated by the complex environment of the rotor. A better understanding of this environment is required to design higher performance blade sections and rotors.

1.1 Description of the Rotor Environment

The rotor environment differs greatly from that experienced by the wings of fixed wing aircraft. Not only do the blades operate over a wide range of angles of attack and velocities in a short period of time, they also operate in the wakes of the preceding blades. Large gains were made once composite blades made it possible to vary the airfoil section along the span of the blade as this enabled rotors to better handle the rotor environment. The three areas of focus can be seen in Figure 1-1.[2]
The effects of forward flight at high advance ratios cause Regions I and II. Figure 1-2 shows the advancing side of the rotor has an area of high Mach number and therefore encounters the effects of compressibility. At the same time on the retreating side of the rotor, the blade is traveling in the same direction as the freestream. This essentially slows the flow relative to the blade and therefore high angles of attack are needed to produce the required amount lift. The inboard sections of the rotor on the retreating side can even experience reverse flow. [1]

Figure 1-1: Rotor Operating Environment. [2]
In hover, the requirements on the airfoils changes from those of forward flight. Due to the absence of forward flight velocity, the induced inflow through the rotor disk is increased and other surfaces such as wings or the fuselage do not produce lift. Therefore, the lift-to-drag ratio of the blade airfoils must be as large as possible to produce the required thrust at altitude for maneuverability and to sustain hover.

1.1.1 Region I: Compressibility on Advancing Blade

The blade tips on the advancing side operate at high Mach numbers and low angles of attack. The primary focus is to increase the drag divergence Mach number, $M_{dd}$. To achieve this, the airfoil should be a relatively thin and have a sharp leading edge. The drag divergence Mach number decreases as the pressure distribution peak is farther from the leading edge. This behavior can be incorporated as a design target to reduce supersonic velocities over the airfoil. If the surface Mach number is below a value of approximately $M=1.16$, the shock will be weak and the flow will remain almost isentropic reducing compressibility drag.

Figure 1-2: The rotor disk showing the areas of compressibility and reverse flow.[1]
Designing an airfoil to conform these requirements dictates that the lower surface, curvature changes rapidly toward the nose. As for the upper surface, it is more difficult to design since the rapidly changing curvature would oppose the design goals for Region II, which will be discussed later. Further design goals for Region I call for a mild pressure recovery after the shock and to minimize the trailing-edge pressure coefficient as much as possible without causing separation. The requirements also show a need of laminar flow up to at least 30% of the chord at Mach numbers below \( M_{\text{sl}} \) to achieve lower skin-friction drag.[3] 

1.1.2 Region II: Retreating Blade Stall

On the retreating side of the rotor, the blades are operating at lower velocities and very high angles of attack. In this region, while Mach numbers approach zero, \( c_{\text{max}} \) should be as high as possible for Mach numbers between 0.4 to 0.5 as compromise since designing for lower Mach numbers would hurt forward flight performance. Performance can be further enhanced if a high \( c_{\text{max}} \) is attainable at a wide range of Mach numbers. The traditional method to reach this design goal is to increase the \( M_{\text{sl}} \) so the boundary layer is less likely to separate at the shock or near the trailing-edge. Wortmann applied supercritical design concepts to rotorcraft airfoils to reduce shock strength and create more isentropic recompression by making the upper surface pressure distribution as flat as possible.[4] Also, the pitching moment must be kept at a minimum, therefore, the pressure distribution is constrained. This limits camber to the forward section of the airfoil and minimizes the amount overall although some small amounts of camber can be used in the aft section.
1.1.3 Region III: Hover

For the hover condition, moderate lift coefficients are needed over moderate Mach numbers. The lift-to-drag ratio should be maximized at a $\alpha$ of 0.6 for Mach numbers between 0 to 0.55. Since inflow is uniform around the azimuth, these requirements apply to all airfoil sections. Conservative target values of $\alpha$ have been traditionally set at 80 counts.[5] It seems that these can be easily achieved and drag could be further lowered without losing lift with attached flow to the trailing edge and laminar flow on the lower surface.

1.2 Airfoil Design

To properly handle all the conditions described in the rotor environment, a good airfoil selection is a major prerequisite to a successful rotorcraft. As stated before, this is difficult since the design of a helicopter is not a “point” design. Rotorcraft airfoils have gone through an evolution where theory and experiment have been used in parallel to best meet the specific operating requirements.

1.2.1 Initial and First Generation Airfoils

Juan de la Cierva first used symmetric Göttingen airfoil sections on his Autogiros.[6] He later switched to cambered airfoils with higher lift-to-drag ratios to increase performance but these airfoils also had higher pitch moments. The low torsional stiffness of early blades lead to aeroelastic twisting of the blades, major control problems, and ultimately structural divergence.[7] This created an airfoil selection criterion that avoided cambered airfoils and made primary use of the symmetrical NACA 0012 or NACA 0015 on the outboard section of rotor blades and the NACA 23012 on the inboard section. While these gave a good compromise in overall performance, it was known that
using cambered and thinner airfoils optimized along the length of the blade would meet the specific 
local quasi-steady aerodynamic requirements as seen in Figure 1-3.

![Diagram of airfoil regions]

Figure 1-3: The weighted quasi-steady aerodynamic requirements for the blades are such that 
different airfoils must be employed at different radial stations.[8]

1.2.2 Modern Airfoils

Many modern airfoils stem from the classic NACA series. The second generation rotorcraft 
airfoils showed significant changes from the baseline NACA airfoils. They were designed to meet the 
requirements of the inboard and blade tips regions, and matched to the advancing and retreating 
sides of the rotor disk.[9] The third and modern generation airfoils show modest improvements. It 
seems that airfoil developments have essentially hit a plateau in terms of 2-D sectional performance. 
This is because these airfoils are still designed for high lift, low drag, and low pitching moments at a 
single Mach number, Reynolds number, and under steady flow conditions. Figure 1-4 shows the 
geometric progression of airfoil design through the generations.
It is interesting to look at the historical trends of the blade loading coefficient, $C_T/\sigma$, shown in Figure 1-5. Also, the United States Army uses this metric as a way to measure the performance of helicopters and set their Technology Development Approach (TDA). The TDA goals are shown on Figure 1-5, which show 16% and 24% improvements in $C_T/\sigma$ by 2005 and 2010, respectively. The graph shows that the performance has not changed very much after 50 years of development. It should be noted, the lower values for combat helicopters are attributed to the larger stall margin needed for the maneuvering requirements.
1.3 Flow Control

To increase the performance of the existing single element airfoils, the use of passive and active flow control devices has been investigated by multiple agencies. Research has shown that perhaps a 5-10% increase in the rotor’s figure of merit and an expansion of the flight envelope of the helicopter.[8]

1.3.1 Boundary-Layer Control

One method of reducing profile drag that has been researched on various aircraft is boundary layer control. Here, the growth of unstable disturbances in the laminar boundary layer that induce a transition to turbulent flow are suppressed. Over the past few decades, many concepts of using suction and blowing (or combinations thereof) have been investigated to control the development of the boundary layer on aerodynamic surfaces. This, however, has many challenges in implementation on aircraft, not to mention helicopters due to the volume and structural restraints of
the rotor blades. Few of the concepts have been successful and even fewer have been used on production aircraft.[11] No boundary layer control devices have been used on helicopters due to the added weight, need of power and complexity of the required pumps and tubes to actuate the system. There have been numerous problems also of the pores becoming clogged due to adverse environmental conditions.

1.3.2 Zero-Mass Flow Synthetic Jets

The use of zero-mass synthetic jets (SJA) is concept of flow control that has been investigated in recent years.[12] These devices have shown increases in airfoil $C_{l_{\text{max}}}$ and decreases in profile drag. The synthetic jets are produced by a diaphragm inside the SJA that is vibrated to energize the flow without adding any mass to the system. A schematic can be seen in Figure 1-6. The interaction of the ejected flow with the external boundary layer energizes the flow and suppresses the onset of separation. While the exact effects of SJAs on the flow are still unknown, initial investigations have shown sizable increases in 2-D airfoil performance. SJAs also have an advantage over traditional boundary layer control methods, as SJAs do not require a supply of compressed air or the use of associated tubing, piping or valves.

![Figure 1-6: Schematic of a synthetic jet actuator (SJA).][8]
1.4 Miniature Trailing-Edge Effectors

Miniature Trailing-Edge Effectors, or MiTEs, are another flow control concept that have been considered for use on lifting surfaces.\[13\] Aerodynamically, MiTEs, depicted in Figure 1-7, are as effective as plain flaps, and high frequency deployments are achievable due to their small size. Experiments showed success with their use in flutter stabilization.\[14\] Seen in Figure 1-7, these MiTEs remained effective at frequencies exceeding 125 Hz. Also, their use has been investigated for rotor-blade control.\[15,16\] These studies positioned the MiTEs on the upper and lower surfaces of the airfoil upstream from the trailing-edge. This provides aerodynamic control at the outboard stations of the rotorblade, and reduces the high loading at the root of the blade.

![Figure 1-7: Concept of MiTE\[13\]](image)

1.4.1 Background

MiTEs are essentially an actively actuated Gurney flap. The Gurney flap was first used in 1971 by race car driver Dan Gurney as an experimental fix during the testing of a poorly performing racecar. The device is beneficial for providing racecars with an efficient, alterable means to increase the downward force on the wings.\[17\] This downward force is used to increase the traction of the vehicle, thereby increasing cornering speeds.
To gain a better understanding of this new aerodynamic flap, Gurney brought the flap to the attention of McDonnell Douglas aerodynamicist, Robert Liebeck. He investigated the aerodynamics of the Gurney flap and introduced their application to the aircraft industry. In this investigation, Liebeck hypothesized the flow structure around a Gurney flap as shown in Figure 1-8, where two counter-rotating vortices form behind the flap and effectively increase the airfoil’s camber near the trailing edge.[18]

Experiments were done to investigate the effects of Gurney flaps with respect to the thickness of the boundary layer.[19] It was observed that Gurney flaps were effective when the heights were at the same scale as the boundary layer. When the boundary layer was significantly thicker than the flap, there was essentially no effect on the lift of the airfoil. The study postulated that the attached counter-rotating vortices act as a means for the airfoil to attain an "off-the-surface pressure recovery," allowing for a large discontinuity in the surface pressure at the trailing edge. This effectively shifts the Kutta condition downstream and below the physical trailing edge.

In other experiments, laser Doppler anemometry (LDA) was used to obtain detailed flow structures near Gurney flaps.[20-22] These results displayed the von Kármán vortex street that forms downstream of the Gurney flap and when time averaged, resulted in Liebeck's hypothesized flow

Figure 1-8: Liebeck’s hypothesized flow structure around MiTE.[18]
structure. Also, the frequency of these oscillations was observed to depend on the height of the Gurney flap and boundary layer thickness.

There has been some interest in using MiTEs for vibration control on rotorcraft. The major concern is their ability to achieve increments in the lift and pitching moment at high frequencies. As Gurney flaps are effective on both the upper and lower surfaces, MiTEs can achieve twice the amplitude of the lift and pitching moments if used on both surfaces. After the flow separates however, they become increasingly submerged into the boundary layer and have thus has no affect on the aerodynamic loads. This behavior is also true for MiTEs. Also, Gurney flaps have been shown to be effective if placed upstream of the trailing edge but the performance gains are not as great.[15]

Table 1-1 shows the effects of a 1 percent chord lower-surface Gurney flap on a variety of airfoils. The $\Delta c_{\ell,\text{max}}$ over the baseline airfoil remains fairly consistent amongst all the airfoils displayed. Figure 1-9 plots $\Delta c_{\ell,\text{max}}$ versus $c_{\ell,\text{max}}$ of the baseline airfoil. It is seen that the effects of the Gurney flap decreases as the baseline $c_{\ell,\text{max}}$ increases. The effects of a static Gurney flap can be roughly approximated by this trend.

Table 1-1: Summary of 1 percent chord height Gurney flap effect to $c_{\ell,\text{max}}$. [19,23-25,35]

<table>
<thead>
<tr>
<th>Airfoil</th>
<th>M</th>
<th>Re</th>
<th>$c_{\ell,\text{max}}$</th>
<th>$\Delta c_{\ell,\text{max}}$</th>
<th>$\Delta c_{\ell,\text{max}}$ (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>VC-OPT</td>
<td>0.755</td>
<td>5.0E+06</td>
<td>1.025</td>
<td>0.175</td>
<td>17.1%</td>
</tr>
<tr>
<td>S903</td>
<td>0.1</td>
<td>1.0E+06</td>
<td>1.15</td>
<td>0.200</td>
<td>17.4%</td>
</tr>
<tr>
<td>S7055</td>
<td>0.1</td>
<td>3.0E+05</td>
<td>1.19</td>
<td>0.140</td>
<td>11.8%</td>
</tr>
<tr>
<td>HQ17</td>
<td>0.1</td>
<td>5.0E+05</td>
<td>1.423</td>
<td>0.191</td>
<td>13.4%</td>
</tr>
<tr>
<td>HQ17</td>
<td>0.1</td>
<td>1.0E+05</td>
<td>1.438</td>
<td>0.189</td>
<td>13.1%</td>
</tr>
<tr>
<td>M06-13-128</td>
<td>0.1</td>
<td>2.0E+05</td>
<td>1.47</td>
<td>0.129</td>
<td>8.8%</td>
</tr>
<tr>
<td>Göttingen 797</td>
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<td>2.5E+05</td>
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</tr>
<tr>
<td>LA203A</td>
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<td>2.5E+05</td>
<td>1.52</td>
<td>0.179</td>
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</tr>
<tr>
<td>S1223</td>
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<td>2.0E+06</td>
<td>2.11</td>
<td>0.091</td>
<td>4.3%</td>
</tr>
</tbody>
</table>
In rotorcraft applications, Gurney flaps have been used on the horizontal tail to increase its authority for high-powered climbs. They have also been investigated for the improvement of rotor performance for both helicopters and wind turbines. One study used a model helicopter with Gurney flaps showed increases in the rotor performance at high thrusts and in forward flight. However, it was also observed that at lower thrusts and hover, the Gurney flaps decrease the performance. It is because of this decrease in performance that passive Gurney flaps are not practical for overall performance gains.[27]

1.4.2 Rotor Applications

MiTEs have many potential applications on rotorcraft. The ability to actuate Gurney flaps makes them a very capable aerodynamic device on rotorcraft.
1.4.2.1 Individual Blade Control

For helicopters, vibration reduction is a major area of research in effort to improve the ride qualities over that of current vibration control systems. Methods of higher harmonic control (HHC) and Individual Blade Control (IBC) have been investigated for these applications. The idea behind these concepts is to create aerodynamics loads at frequencies corresponding to those that are felt in the fuselage and therefore cancel the vibrations. The requirement for these loads is on the order of 4/rev or about 20 Hz. In particular for IBC, active flaps have been considered in numerous studies. MiTEs could be used in place of these active flaps as they appear to be ideal in providing the required changes in lift and moments required. An additional advantage of using MiTEs are that they provide this potential with significantly lower actuator loads and are insensitive to compressibility effects.[23,28]

1.4.2.2 Rotor Performance Enhancement

A major study was conducted recently by Michael Kinzel to investigate the use of MiTEs on rotorcraft. The goals of the project were to first model the aerodynamic characteristics of MiTEs, and then investigate the potential rotor performance improvements.[26] This includes increasing the maximum flight speed, achievable rotor thrusts, thrusting performance, maneuver performance and payload capabilities. These gains would indirectly increase the cruise performance. MiTEs show to be most effective for transonic airfoils, as they provide a more efficient configuration for high-speed flows, while still providing high lift when needed. These increases in the maximum lift indirectly improve the cruise performance of the helicopter, by allowing the blade to be designed for transonic flow, without losses in the maximum flight speed and high thrust performance. Deployment schedules of MiTEs were also studied, showing that MiTEs are equally effective using on/off actuation or gradual deployment. Finally, the performance gains that MiTEs provide at high speeds
and for high thrusts depend more on the capability to increase the lift and prevent stall. Figure 1-10 shows the increase in the maximum forward speed of a MiTE-equipped helicopter.

MiTEs provide lift and pitching moment increments that are comparable to those of plain flaps. One advantage of a MiTE in this application is that the actuation loads required are much less. The decreased actuation loads are due to the hinge moment being essentially zero compared to those of a trailing-edge flap. A MiTE’s low inertia also allows for high frequency response.

All of the performance analysis done in this initial research was done on MiTEs placed at the trailing-edge of the airfoil. The study that continues in the following chapters extends this research to MiTEs placed upstream of the trailing-edge. Previous experiments showed a decreased in the
effectiveness of an upstream MiTE; however, this placement is advantageous in that it allows the retracted MiTE to be buried within the airfoil.

1.4.2.3 Other Potential Applications of MiTEs

There are a wide range of other applications of MiTEs to rotorcraft. These devices can be used for any function where IBC systems can provide benefits. IBC systems are often suggested for their usage to alter the rotor wake for noise control, for which purpose a MiTE could deploy to affect the tip vortices and their interactions with other rotor blades. The vortex miss distance from the blade can be increased and, therefore, the noise generated by blade-vortex interactions reduced. In another application, the vortices can influence and reduce the angle of attack on the retreating blades to delay stall.
Chapter 2

Experimental and CFD Investigations

In order for an aerodynamic model to be formulated, a number of experimental studies were utilized for understanding and validation. Likewise, due to the complex nature of a MiTE on a rotorcraft, a single wind tunnel experiment was not feasible. Therefore, multiple experiments were investigated and used in validating a Computational Fluid Dynamics (CFD) simulation that was then used as a “virtual” wind tunnel.

2.1 Static Wind Tunnel Testing

The first experiment was that of an airfoil tested in a wind tunnel with and without Gurney flaps attached at and near the trailing edge.[35] Surface pressure measurements were used to determine the lift and pitching moment of the airfoil, while a wake-traverse probe was used to obtain the drag. The airfoil was initially tested in its baseline state then with Gurney flaps of 0.005c, 0.01c and 0.02c in height located at 0.9c, 0.95c and 1.0c. All of these configurations were tested with natural and fixed transition, a chord Reynolds number of 1.0x10^6, and a Mach number of less than 0.2. While previous tests of Gurney flaps used a force balance, this particular experiment’s surface pressure integration showed how the flaps affected the pressure distributions to achieve higher lift coefficients.

2.1.1 Wind Tunnel Description and Airfoil Model

The wind tunnel used for the static airfoil testing was the Penn State Low-Speed, Low-Turbulence Wind Tunnel. This facility is a closed throat, single-return atmospheric wind tunnel that...
has a maximum test section velocity of approximately 220 ft/s. The test section is rectangular measuring 3.25 ft high and 4.75 ft wide with filleted corners, as seen in Figure 2-1.[29]

Figure 2-1: The 3.25' x 4.75' cross section of the S903 airfoil placed in the Penn State Low-Turbulence, Low-Speed Wind Tunnel.

The surface pressure measurements on the model are reduced to pressure coefficients and then used to determine the sectional force and moment (about the quarter-chord) coefficients. The sectional profile drag coefficients are obtained through use of the wake traversing probe and total pressures using momentum theory.[30,31]

The airfoil model used was the 12-percent thick S903 airfoil.[32] This airfoil was designed to explore the effects of airfoil thickness and surface roughness on the maximum lift coefficient of wind turbines. While this airfoil was designed for large amounts of laminar flow, it is similar to many rotorcraft airfoils in terms of thickness ratio and camber distribution. Figure 2-2 compares the S903 airfoil with the Boeing-Vertol VR-12, a well-known rotorcraft airfoil.[33]
2.1.2 Experimental Results

Figures 2-3 and 2-4 show the pressure distributions for the baseline airfoil, the airfoil with a 0.02c high lower-surface Gurney flap at 1.0c and at 0.9c. At a constant lift coefficient of 0.7, the three configurations are compared in Figure 2-3, while in Figure 2-4, they are compared at their maximum lift coefficients of 1.15, 1.35 and 1.5 respectively.

Figure 2-2: The S903 airfoil compared to the VR-12 airfoil

Figure 2-3: Pressure distributions at $\alpha = 0.7$ with and without Gurney flaps
From Figure 2W3, it is shown that the Gurney flap greatly increases the pressure on the lower surface of the airfoil upstream of the flap. On the upper surface of the airfoil, by alleviating the pressures on the trailing edge with the Gurney flaps, the leading-edge suction peak is reduced and the start of pressure recovery is moved aft by approximately 20 percent chord. Along with the delay, the pressure at which recovery begins is much lower than that of the original airfoil. This relaxing of the pressure recovery allows the airfoil to have more favorable gradients over the forward part of the airfoil.

As seen in Figure 2W4, large gains are made in terms of maximum lift coefficient between that of the baseline airfoil and those of the flapped model. This is due to the large increases in lower-surface pressure. At higher angles of attack, the Gurney flap becomes more effective due to the thinning of the lower-surface boundary layer. The lowering of pressure levels in the pressure recovery on the upper surface also aided in greater maximum lift coefficients attained and moved the point of separation downstream by 15 percent chord. The cause of differing lift coefficients between

Figure 2W4: Pressure distributions at \( \ell_{\text{max}} \) with and without Gurney flaps
the two Gurney flap locations is mostly due to the earlier pressure gain on the lower surface for the upstream Gurney flap. This results in a loss of lift production from the location of the flap to the trailing edge.

Figure 2-5 shows the influence of Gurney flap height on the aerodynamic characteristics of the airfoil. A 0.02c Gurney flap placed at the trailing edge achieves a 29 percent increase in maximum lift coefficient compared to the baseline airfoil. The shorter flaps also show gains in maximum lift coefficient and were proportional to their size. As the boundary layer thins on the lower surface with increasing angles of attack, the Gurney flap on that surface becomes more effective and causes the lift-curve slope to increase. Conversely, as the angle of attack decreases, the effect of the Gurney flap decreases. This results in the lift curves merging with that of the baseline airfoil as the negative maximum lift value is approached. The moment curves behaved in a similar manner with increasingly negative angles of attack. At angles of attack for which the Gurney flaps are effective, the increased aft loading due to the flap results in increasing nose-down pitching moment coefficients compared with the baseline airfoil. The change in the pitching moment is also proportional to the Gurney flap height. Although the magnitude of pitching moment coefficients might be of concern, the actually moments remain small for MiTEs deployed on the retreating blade side due to low dynamic pressure. Within the low-drag region just as with the lift and pitching moments, the minimum drag rises approximately proportional to the Gurney-flap height.
Due to the physical constraints of MiTE deployments normal to the surface, the flap must be placed far enough upstream to be completely buried within the airfoil when not deployed. Figure 2-6 shows the aerodynamic gains achieved with Gurney flaps are only moderately reduced by upstream MiTEs. The maximum achievable lift coefficient is reduced by approximately 10 percent as a 0.02c flap is moved to 0.90c from the trailing-edge location. Similar changes are seen with the pitching moment coefficient. The changes in drag, however, seem to only change with the Gurney flap height and not with its location.
Although the S903 is similar to the VR-12 rotorcraft airfoil, the S903 was designed to make use of natural laminar flow to achieve low drag. To explore how the airfoil aerodynamics would be altered on an airfoil such as the VR-12, measurements were also made with transition fixed at 0.02c on the upper surface and 0.05c on the lower surface of the airfoil. The primary interest is the aerodynamic behavior near the maximum lift coefficient, the turbulator grit was sized to the critical roughness height for angles of attack approaching stall.[34] Thus, at lower angles of attack, the grit is too small to force transition. Figure 2-7 shows this by the converging drag curves of the free and forced transition configurations at lower angles of attack. Also, it is seen that the while the drag increases with fixed transition, the lift and moment curves are essentially unaffected. With fixed transition, the drag increases due to the Gurney flap is not nearly as large as it is with free transition, especially in the case of the baseline airfoil that has significant amounts of laminar flow. Again, the fixed transition results are more representative of an operational rotorcraft environment. The effect of Gurney flaps in terms of height and location are summarized in Figure 2-8.
Figure 2-7: Aerodynamic characteristics with natural transition and transition fixed at 0.02c on the upper surface and 0.05c on the lower.

Figure 2-8: $\Delta c_{lmax}$ with varying Gurney flap heights and chordwise locations.
Concerns about whether or not MiTEs can be used as control devices with up and down deployment are addressed in Figure 2-9. The effects of a 0.01c high Gurney flap on the upper surface with those of one on the lower surface are compared. The Gurney flap on the upper surface becomes more effective as the angle of attack decreases and the boundary layer on the upper surface thins. At higher angles of attack, the aerodynamic characteristics merge to that of the baseline airfoil. The opposite is true for a lower-surface Gurney flap where it becomes more effective with increasing angle of attack. At midrange angles of attack, Gurney flaps still have a significant effect on lift and moment.

![Graph](image)

Figure 2-9: Comparison of the aerodynamic characteristics with a 0.01c high Gurney flap located at the trailing edge on the upper and lower surfaces.

Finally, a 17 percent chord plain flap deflected at 13 degrees achieves the same increment in maximum lift as a Gurney flap that is located at 0.9c and is 0.02c in height as seen in Figure 2-10. Comparing a MiTE to the plain flap shows a large decrease in the required actuator loads due to the decrease in hinge moments and inertial loads.
2.2 NASA Ames Dynamic Wind Tunnel Testing

Experiments investigating an oscillating airfoil with a Gurney flap were conducted at the NASA Ames Research Center in the Compressible Dynamic Stall Facility (CDSF) using the VR-12 rotorcraft airfoil.[36-38] The point of these tests was to increase the understanding of Gurney flaps in compressible dynamic stall. The test matrix consisted of an airfoil with a Gurney flap located at the trailing edge oscillated at reduced frequencies, \( k \), of 0.0, 0.05 and 0.1 where the reduced frequency is defined as

\[
k = \frac{\omega k}{2V}
\]
The airfoil rotates about the quarter chord such that $\alpha(t) = 10^\circ + 10^\circ \sin(\omega t)$. The Reynolds number ranged from $0.7 \times 10^6$ to $1.6 \times 10^6$ with Mach numbers of 0.2, 0.3 and 0.4. The Gurney flap heights investigated were 0.0085c, 0.0135c and 0.024c.

The lift, pressure drag and pitching moment calculations were done through the use of surface pressure integrations. The surface pressure measurements were done using 20 Kulite unsteady-pressure transducers that measured the absolute pressure. Typically the sampling rate was 4 kHz/channel, and totally 40,000 samples/channel were recorded and stored according to bins based on the angle of attack, with increments ranging from 0.02 to 0.08 degrees. The data was stored in 800 bins and then averaged. A minimal standard deviation was observed giving confidence that uncertainty is low. Using the geometry and instantaneous angle of attack, the lift, pressure drag and pitching moment coefficients was calculated. The viscous drag is neglected, as this quantity is relatively small at angles of attack corresponding to dynamic stall. The pressures on the faces of the Gurney flap were not measured.

### 2.2.1 Wind Tunnel Description and Airfoil Model

The CDSF, shown in Figure 2-11, is an in-draft wind tunnel with a test section measuring 35 cm high, 25 cm wide and 100 cm long and designed to operate at Mach numbers up to 0.5.[37] The mounting system has the capability to continuously vary the airfoil pitch angle at amplitudes varying from 2 to 10 degrees, and using mean airfoil pitch angles of 5, 10 or 15 degrees. The mean pitch angle can also be set to other angles by manually adjusting the angle of attack assembly.[37,38]
The chord length of the model is 15.2 cm and its span is 25 cm.[38] Baseline cases that have zero droop are of interest for this study. These results are used in the validation the CFD modeling of MiTEs. Fabrication requires that the trailing edge be constructed as shown in Figure 2-12, and the leading edge surface of the model is relatively rough. Comparisons of static polars of the VR-12 airfoil as designed and that of the fabricated model for this experiment are shown in Figure 2-13. These polars are generated using MSES,[39] at conditions similar to those of the wind tunnel, with the transitions fixed near the leading edge.
Figure 2.12 shows the predicted pressure distributions of the designed and fabricated airfoils agree well at an angle of attack of 10 degrees. This gives confidence that the fabricated model will predict the correct characteristics of the VRW12 airfoil in dynamic stall. Minor

Figure 2.13: Comparison of the static polars of the designed VR-12 airfoil to that fabricated for experiments, as calculated by MSES at M=0.4, Re =1.4e6.

Figure 2.14 shows the predicted pressure distributions of the designed and fabricated airfoils agree well at an angle of attack of 10 degrees. This gives confidence that the fabricated model will predict the correct characteristics of the VR-12 airfoil in dynamic stall. Minor
discrepancies will occur because of the fabricated model having an increased stall angle of attack, and a lower $\epsilon_{1,\text{max}}$ than the designed VR-12 airfoil.

![Figure 2-14: Comparison of the pressure distribution the designed VR-12 airfoil to that fabricated for experiments](image)

**2.2.2 Experimental Results**

Quasi-steady results at $M=0.3$ are shown in Figure 2-15 for the airfoil in the positive-pitch stroke. These results are useful since they display the static effects of the Gurney flap post stall. This is of importance for rotor modeling purposes where these post-stall Gurney flap effects are used for dynamic stall modeling. Recall that the experiments do not measure the Gurney flap pressures or the viscous drag; thus, as observed in Figure 2-15, for lower angles of attack the Gurney flap shows a decrease in the pressure drag. If these additional quantities are measured, the airfoil using a Gurney flap would in fact increase the drag as observed in the static-airfoil wind-tunnel experiments. Finally notice that the Gurney flap continues to be effective post stall.
The results from the oscillating airfoil experiments are presented in Figures 2-15 to 2-18 for a 0.0135c height Gurney flap at Mach numbers of 0.3 and 0.4 and a reduced frequency of 0.05. In Figure 2-16, the lift coefficient is plotted against the angle of attack. It can be observed that in dynamic stall, this Gurney flap increases maximum lift by 22 percent. Gurney flaps also have the effect of decreasing the stall angle of attack with an increase in flap height.[38] This decreased stall angle of attack is expected, as a similar behavior occur statically.
The drag is compared in Figure 2-17, where nearly constant increases are observed at angles less than stall. Post stall, the Gurney flap results in a much larger increase in drag. This is due to the increased vortex effects and because the airfoil is in a deeper stall when compared to the baseline airfoil.
A better comparison is made in the plot of $\alpha$ versus $\alpha'$ in Figure 2-18 as it shows the increase in aerodynamic efficiency at high lifts. These results are consistent with the static results, and if the profile drag were measured, would show that the aerodynamic efficiency is increased at high lifts using Gurney flaps.

![Figure 2-18: Comparison of the drag polars for the VR-12 baseline airfoil and with a 0.0135c height Gurney flap in dynamic stall at a reduced frequency of 0.05](image)

The pitching-moment coefficient is compared in Figure 2-19. It can be seen that there is nearly an offset in an increased nose-down pitching moment. With delaying stall to higher lifts results in the nose-down pitching moments being decreased at high lifts.
2.3 Wind Tunnel Experimental Conclusions

From these two experiments, the understanding of Gurney flaps behavior has been improved. These results suggest that Gurney flaps can provide similar lift increases upstream of the trailing edge, when compared to the traditional placement at the trailing edge of an airfoil. There is a performance penalty associated with this upstream placement, but such a concept is essential for the implementation of actuating MiTEs. Lift and moment changes are achievable in either the positive or negative directions, depending on the surface which the Gurney flap is placed.

In considering the effects of a Gurney flaps to oscillating airfoils, the results resemble those found statically. Furthermore, the increase in $\alpha_{\text{max}}$ and the change in $\alpha_{\text{stall}}$ for an oscillating airfoil are similar to static airfoils. Finally, at higher lifts, the Gurney flap is observed to increase the aerodynamic efficiency, while simultaneously lower pitching moment by delaying the effects of dynamic stall to increased lifts.

![Comparison of the pitching moment for the VR-12 baseline airfoil and with a 0.0135c height Gurney flap in dynamic stall at a reduced frequency of 0.05](image)
2.4 CFD Investigations

Ideally, more experimental testing would be done to investigate the application of MiTEs. These experiments, however, are not practical for initial investigations due to their cost and complexity where the test would involve an oscillating airfoil, at various flow conditions and having flaps that deploy at high frequencies. This being the situation, computational methods are the best tool for this initial stage of research. CFD calculations are most valuable in investigating complex problems such as these where one can gain a better understanding of the physical flow characteristics present.

The NASA CFD code, OVERFLOW2, was validated and used for the analysis of Gurney flaps and MiTEs.[26] OVERFLOW2 is a Reynolds averaged Navier-Stokes (RANS) solver that utilizes structured Chimera overset grids that allow for relative motion between bodies.[40,41] These simulations use the Spalart-Allmaras one-equation turbulence model, which is specified for all boundary layers which are modeled as fully turbulent.[42] An inherent consequence of this method is that the laminar regions are also modeled at turbulent. This simplification is because OVERFLOW2 does not incorporate a transition model. While this is not completely modeling the true characteristics of the flow, helicopter airfoils are typically designed using a fully turbulent model and thus the simplification is an acceptable approximation.[8]

2.4.1 Gurney Flapped Airfoils

Due to the environment of the helicopter rotor, the performance of MiTEs at a range of Mach numbers needs to be investigated. Transonic experiments and CFD research have shown that the effects of Gurney flaps are similar in transonic flows, as observed in low-speed experiments. Unfortunately, data does not exist for Gurney flaps at varied Mach numbers on a consistent airfoil. This data is needed to successfully model the effects of Gurney flaps and MiTEs in order to apply...
them to rotorcraft. To do this, CFD was used to analyze the Gurney flaps at several Mach Numbers. The airfoils used were the Boeing-Vertol high-lift VR-12 and transonic VR-15 airfoils which are two heavily used rotorcraft airfoils.

2.4.1.1 VR-12: 0.01c Gurney Flap located at 1.0c

The VR-12 airfoil is representative of inboard regions on the rotor. The Gurney flap effects at various Mach numbers for the VR-12 are seen in Figure 2-20.

Figure 2-20: Gurney flap effects with Mach number variations for the VR-12 airfoil (a) lift increment (b) drag increment (c) moment increment
The incremental changes in lift, drag and pitching moment are plotted against a scaled angle of attack, $\alpha'$, which is defined in Eq. 2.2. This scaling easily compares the effects of Gurney flaps between positive and negative stall of the baseline airfoil at various Mach numbers. Thus, $\alpha'$ is scaled from 0.0 to 1.0, where 0.0 is negative stall for the baseline airfoil, and 1.0 is where it reaches positive stall.

$$\alpha' = \frac{\alpha - \alpha_{\text{Stall}}}{\alpha_{\text{Stall}} - \alpha_{\text{Stall}}}$$  \hspace{1cm} 2.2

When using this scaling, the effects of the Gurney flap appear to be independent of the Mach number. As $\alpha' \to 0$, the effect of the Gurney flap approaches zero as it becomes ineffective. Also, when using this angle of attack scaling, the increases in the absolute lift, drag, and pitching moment are nearly constant and are insensitive to changes in the Mach number. These results show that Gurney flaps are just dependent on $\alpha'$, this effectively removes the dependence on Mach number and therefore we have Eq. 2.3.

$$\Delta \epsilon_{l,\text{max}}(M, \alpha) = \Delta \epsilon_{l,GF}(M, \alpha')$$  \hspace{1cm} 2.3

This relation provides a useful method with a reasonable degree accuracy to interpolate the effects of a Gurney flap at different Mach numbers. This method is also valid for the drag and pitching moment effects of the Gurney flap.

### 2.4.1.2 VR-15: 0.01c Gurney Flap located at 1.0c

The VR-15 airfoil is a transonic airfoil representative of the outboard radial stations of a helicopter rotor blade. The lift, drag and pitching moment effects can be seen in Figure 2-21 again utilizing the $\alpha'$ scaling factor for the angle of attack. The effects are similar at both Mach numbers,
however, not at good at the VR-12 results. This is a result of using a small number of discrete angles of attack and would cause $\alpha_{+\text{Stall}}$ and $\alpha_{-\text{Stall}}$ not to be fully resolved.

Figure 2-21: Gurney flap effects with Mach number variations for the VR-15 airfoil (a) lift increment (b) drag increment (c) moment increment

2.4.2 Dynamic MiTE CFD Results

MiTEs are intended to perform two functions in their application to rotorcraft. The first is that of fuselage vibration control which requires deployment to occur at or near 4/rev (20Hz). The
second application is to increase rotor performance which requires 1/rev deployments (5Hz). At this frequency, the MiTEs increase the sectional $c_{\text{max}}$ to delay the onset of dynamic stall. CFD calculations were conducted at both frequencies to study the aerodynamics at a constant angle of attack.

### 2.4.2.1 VR-12 Airfoil: 0.01c height MiTE at 1.0c

The first case was that of a VR-12 airfoil with a MiTE height of 0.01c at the trailing edge where deployments were done at numerous frequencies and flow conditions. The results can be seen in Figure 2-22 through Figure 2-24. The normalized lift is compared to Theodorsen’s theory in Figure 2-23(a).[43] The CFD shows excellent agreement in both amplitude and phase-lag trends. One difference is that the normalized lift magnitude for the MiTE approaches 0.4 as $k \to \infty$, while Theodorsen’s function approaches 0.5. There is also a difference at high Mach numbers between the CFD results and Theodorsen’s theory which is due to the incompressible assumptions made in it. The CFD results also show that when the Mach number is increasing, as $k \to 0$, the lift magnitude and phase lags are increased which follows compressible-unsteady theory. The phase difference at low Mach numbers is due to the Theodorsen function only accounting for circulatory loads where the apparent mass effects would be a factor at this particular flow condition. These apparent mass loads have a 90 degree phase lead that becomes more evident for high frequency flapping. The CFD data also correlates very well with the lift data of a NACA 0012 at a Mach number of 0.1 seen in Figure 2-23.[14]
Figure 2-22: Force and moment for the sinusoidal deployment of a 0.01c MITE located 1.0c, compared to static wind-tunnel data. (a) $\omega=5$Hz, ($k=0.14$) (b) $\omega=20$Hz, ($k=0.54$)
Figure 2.23: Effect of the lift with changes in deployment frequency for a MiTE located at 1.0c, at various free-stream conditions, and compared to incompressible theories.

Figure 2.24: Effect of the drag and pitching moment with changes in deployment frequency of a MiTE located at 1.0c for various free-stream conditions.
2.4.2.2 VR-12 Airfoil: 0.02c height MiTE at 0.9c

The case of a VR-12 model using a MiTE height of 0.02c positioned at 0.9c is investigated at numerous frequencies and flow conditions. The results can be seen in Figure 2-26 through Figure 2-28.

Figure 2-25 shows the instantaneous streamlines for the airfoil with a MiTE deploying at 5Hz.

At this lower deployment frequency, notice that in the retracted position, the flow near the MiTE is stable. This is not the case when the deployment frequency is increased where visualizations show that the flow is unattached downstream of flap. This is a result of subsequent deployments that occur before the flow has time to reattach. As flap deploys, a strong vortex forms downstream of the MiTE and convects to the trailing edge of the airfoil. The time required for this vortex convection is a significant portion of the deployment cycle, especially for the 20Hz case.

This vortex has tremendous effects to the aerodynamics as shown in Figure 2-26, where the two deployment frequencies are plotted and the lift can be compared to Theodorsen's function. Observing the lift plots, notice that there are major deviations between the CFD results and theory.
This does not discredit the CFD results, as the unsteady flow characteristics for this case clearly violate assumptions made in Theodersen's function. These assumptions are that the conditions are an inviscid, incompressible, attached flow with only small disturbances. The flow around these MiTEs show a large, unattached vortex that develops downstream of the MiTE, which suggests that the solutions cannot be validated against analyses developed for conventional flaps and wings.

With further investigations of Figure 2-26, upon comparison of the lift amplitudes of CFD and theory, the predicted amplitudes are comparable to the theory at both frequencies. There are small positive offsets in the lift, suggesting that the MiTEs are shedding additional vortices into the wake that induce upward velocities on the airfoil, and effectively increase the angle of attack. It is also noticed that the MiTE does not achieve the maximum lift until the vortex reaches the trailing edge of the airfoil. This is a result of the strong vortex that develops as the MiTE deploys, creating a low-pressure region on the lower surface of the airfoil and counter produces lift. This is similar, but in the opposing direction, to the vortex lift observed in dynamic stall.[1] Furthermore, flow visualizations show that when the vortex reaches the trailing edge, the low-pressure core entrains air from the suction surface and creates additional circulation over the entire airfoil. This entrained air also balances the pressure and reduces the vortex strength, enabling the maximum lift to finally be achieved. These combined effects attribute for these sharp increases in lift. The drag is noticed to slightly increase for the low frequency case, where the flow has the time to reattach. For the high-frequency case, where the time required for the vortex to reach the trailing edge approaches that of the deployment cycle, the flow does not have time to reattach and the drag is significantly increased.
Figure 2.26: Force and moment for the sinusoidal deployment of a 0.02c MiTE located 0.9c, compared to static wind-tunnel data. (a) $\omega=5$Hz ($k=0.14$) (b) $\omega=20$Hz ($k=0.54$)
Shown in Figure 2-27 is the normalized lift and phase angles compared to Theodorsen’s function. Here the phase lags between the full deployment position and the maximum lift, and between the fully retracted position and the minimum lift, are plotted separately as they vary significantly. The lift amplitudes predicted by the CFD are larger than those of predicted by theory and the phase lags also differ from those predicted by theory. Also, with increasing deployment frequency, the phase lag increases which differs from what is predicted from theory. As a final note in terms of the lift of the system, the phase lag at the minimum lift position is much larger than that at the maximum lift. This is a result of a vortex system that develops behind the MiTE on the lower surface of the airfoil.

![Graph showing lift and phase angles](image)

Figure 2-27: Effect of the lift with changes in deployment frequency of a MiTE located at 0.9c for various free-stream conditions compared to incompressible theories

Figures 2-28 and 2-29 summarize the drag and moment calculations in the reduced frequency domain, with variations of the free-stream conditions. Notice that one case is simulated at an angle of attack of 10 degrees, which unintentionally, falls in between the stall angles of attack of the baseline airfoil and one with a Gurney flap. This is representative of a type of dynamic stall occurring to a static airfoil, and would account for the large increases in the drag. The pitching
moments for these cases can also be observed, where there appears to be little dependence on the deployment frequency.

Figure 2-28: Drag effect of MiTE located at 0.9c with changes in the deployment frequency for various free-stream conditions.

Figure 2-29: Pitching Moment effect of MiTE located at 0.9c with changes in the deployment frequency for various free-stream conditions.
2.5 CFD Conclusions

When MiTEs are positioned at the trailing edge, the results agree well with unsteady aerodynamic theories that are valid for plain flaps. They also appear to remain effective through high deployment frequencies. When MiTEs are positioned upstream of the trailing edge, strong vortices are created on the pressure surface of the airfoil. This vortex delays the development of lift and creates large phase lags between the flap position, and the resulting aerodynamic forces and moments. This positioning of MiTEs show possible decreases in the sectional performance at high deployment frequencies; however, this appears to mostly be a factor at low-speed conditions. These results show that the aerodynamic responses are as expected, and that MiTEs can sufficiently delay stall to higher lift coefficients.
Chapter 3

Aerodynamic Modeling

To apply the results of the previous chapter to the design and application of MiTEs to rotorcraft, an analytic aerodynamic model must be used to produce computationally fast results to be used in a rotor performance program. The aerodynamic response of a MiTE depends on the deployment frequency, free-stream velocity, Mach number, and angle of attack at which the MiTEs are deployed.[26] Since the primary focus of this research is of MiTEs delaying the onset of stall and extending the flight envelope, dynamic stall modeling must be included.

The well-validated unsteady aerodynamics and dynamic stall model, Leishman-Beddoes, is used to account for the unsteady aerodynamics and dynamic stall of the oscillating airfoil to which MiTEs are attached. Modeling is done through a wide range of angles of attack and freestream conditions to first quantify the changes due to Gurney flaps when compared to the OVERFLOW CFD results. These results are then used to modify the Hariharan-Leishman unsteady flapped airfoil model to predict the unsteady aerodynamics during a MiTE deployment. This modified Hariharan-Leishman model along with the Leishman-Beddoes unsteady aerodynamics and dynamic stall models are used to predict the performance of a well-equipped rotor. Further modifications were made to both the flap and dynamic stall models to account for the behavior of an upstream MiTE.

3.1 Gurney Flap Effects

The effects of a Gurney flap at various Mach numbers and angles or attack are initially investigated to begin a base for the aerodynamic model of MiTEs. For simple plain flaps, thin-airfoil theory is used. Unfortunately, due to the sizeable amount separated flow associated with Gurney
flaps and MiTEs, thin-airfoil theory cannot be used accurately. So, for Gurney flaps, experimental data or CFD needs to be used to ascertain the effects and then used to produce a simplified method of calculating the effects.

With the use of the scaled angle of attack, \( \alpha' \), seen in the previous chapter, Gurney flaps seemed to have little dependence on Mach number. Therefore, all that is needed are the aerodynamic characteristics of the Gurney flap at a few flow conditions and the characteristics of the baseline airfoil to calculate the effect on lift, drag, and pitching moment at a range of Mach numbers. To bound the effects at a Mach number of zero, the Gurney flap effects are assumed to be constant. At higher Mach numbers, it is assumed that the drag increment remains constant and that the lift increase from the Gurney flap approaches zero.

Outside of the region of positive and negative stall, the effects of the Gurney flap are also defined. At or below negative stall, the Gurney flap has no effect on lift, drag or pitching moment due to it being submerged in the boundary layer. For positive stall however, there is a continuing influence. The lift gradually increases to a value of 0.7 of that seen in the range of \( 0.0 < \alpha' < 1.0 \). This gain in lift uses an exponential function and then diminishes as the angle of attack approaches 90°. The drag of the Gurney flap differs slightly as there is an initial decrease after the baseline airfoil stalls. At angles slightly higher than \( \alpha' = 1.0 \), the drag increment from the Gurney flap is 50 percent of that at positive stall. Exponential functions are used to smooth the effect as drag diminishes to zero as the angle of attack approaches 90°. Sample results of this can be seen in Figure 3-1.
3.2 Dynamic Stall

A major concern of rotorcraft operation and factor that limits the flight envelope is dynamic stall. This is caused by the large changes of angle of attack around the azimuth in a high speed condition. Since MiTEs are used in alleviating blade stall, the modeling of their aerodynamic characteristics in dynamic stall is of great interest.

Figure 3-2 shows the aerodynamic and flow characteristics of dynamics stall in stages. On the left are the force and moment trends during dynamic stall while the corresponding flow stages are shown on the right. A notable characteristic of dynamic stall is the large increase in $\alpha_{\text{max}}$. This is caused by two mechanisms, the first being lags in the unsteady response of pressure and the boundary layer which in turn cause a lag in separation as the angle attack increases. The second
mechanism is seen in stage 2-3 where the leading-edge vortex convects over the upper surface of the airfoil and creates the additional lift. It should be noticed however there is a large increase in both drag and pitching moment during this stage. After the vortex convects off the surface of the airfoil, a hard stall occurs along with the flow being completely separated on the upper surface. To model this, the Leishman-Beddoes dynamic stall model was used.

Figure 3-2: Force, moment, and flow characteristics of dynamic stall. [1]
3.2.1 Leishman-Beddoes Dynamic Stall Model

The Leishman-Beddoes method is a semi-empirical model and was used for the dynamic stall aerodynamic modeling of the baseline airfoils, and those fitted with Gurney flaps and MiTEs.[44] For MiTEs, the Hariharan-Leishman unsteady flapped airfoil model will be coupled with dynamic stall to describe the unsteady aerodynamics. Both of these models are based on indicial functions that show the response to step changes in angle of attack and/or MiTE deployment. First, the focus will be on an oscillating airfoil.

3.2.1.1 Oscillating Airfoil with Attached Flow

The response to an indicial step of the angle of attack is approximated as

\[
\phi^C_\alpha(M, s) = 1.0 - A_1 e^{(-b_1 \beta^2 s)} - A_2 e^{(-b_2 \beta^2 s)}
\]

3.1

The empirical constants are set to \( A_1 = 0.3, A_2 = 0.7, b_1 = 0.14, \) and \( b_2 = 0.53 \) as recommend in Ref. [44]. The indicial response function seen in Eq. 3.1 is solved using the convolution integral and then discretized to obtain the following deficiency functions that relate to the pitch rate and time history.

\[
X_n = X_{n-1} \exp(-b_1 \beta^2 \Delta s) + \Delta \alpha \cdot A_1 \exp(-b_1 \beta^2 \Delta s / 2)
\]

3.2

\[
Y_n = Y_{n-1} \exp(-b_2 \beta^2 \Delta \tau) + \Delta \alpha \cdot A_2 \exp(-b_2 \beta^2 \Delta s / 2)
\]

3.3

These results account for the deficiency of the angle of attack and therefore, an effective attack, \( \alpha_e \), can be defined.

\[
\alpha_e = \alpha_n - X_n - Y_n
\]

3.4
Using lookup tables produced from experiments and computer codes, the circulatory lift, 
\( \epsilon^C_{f,\alpha}(M, \alpha_s) \), can be determined. The non-circulatory or apparent mass effects are calculated as

\[
\epsilon^l_{N,\alpha} = \frac{4K_a T_j}{M} \left( \frac{\Delta \alpha_s}{\Delta t} - D_n \right)
\]

where the deficiency function, \( D_n \), is calculated as

\[
D_n = D_{n-1} \exp \left( \frac{\Delta t}{K_a T_j} \right) + \left( \frac{\Delta \alpha_s - \Delta \alpha_{s-1}}{\Delta t} \right) \exp \left( - \frac{\Delta t}{K_a T_j} \right)
\]

and the time constant, \( T_j \), is defined

\[
T_j = \frac{c}{a}
\]

This time constant represents the non-dimensional time required for pressure disturbances to propagate along the chord length of the airfoil. Finally, the factor \( K_a \) depends on the Mach number and given by

\[
K_a = \frac{0.75}{(1 - M) + \pi \beta^2 M^2 (A_1 b_1 + A_2 b_2)}
\]

### 3.2.1.2 Oscillating Airfoil In Dynamic Stall

To account for cases where the flow is separated and dynamic stall is occurring, the Leishman-Beddoes’ method is used to model the unsteady, nonlinear aerodynamics. The method utilizes the Kirchhoff/Helmholtz model of the static stall of a flat plate to calculate the lift. The lift is determined as a function of the angle of attack and the separation point, \( f \)

\[
\epsilon^C = \epsilon_{\alpha a} \left( \frac{1 + \sqrt{f}}{2} \right)^2 (\alpha - \alpha_s)
\]

The separation point can be approximated by
To calculate the values of $\alpha_1$, $S_1$, and $S_3$, known static data is required to determine their values. To a first order approximation, the lag in the pressure response is calculated by

$$f = \begin{cases} 
1.0 - 0.3 \exp \left[ \frac{(\alpha - \alpha_1)}{S_1} \right] & \text{for } \alpha \leq \alpha_1 \\
0.04 - 0.66 \exp \left[ \frac{(\alpha_1 - \alpha)}{S_2} \right] & \text{for } \alpha > \alpha_1
\end{cases}$$ 

3.10

The value $T_p$ is a time constant that relates to the reaction time of the unsteady pressure response on the airfoil surface. This is a function of Mach number and independent of the airfoil characteristics. $T_p$ is determined experimentally or with the use of unsteady calculations. A new effective angle of attack is defined that incorporates this lag of the pressure response

$$\alpha_f = \frac{\epsilon_f}{\epsilon_{\alpha,n}} + \alpha_\zeta$$

3.13

Using this effective angle of attack, the first order separation point, $f_s$, is calculated using Eq. 3.10. The unsteady boundary layer response is modeled and used to determine the unsteady separation point by

$$f'' = f' - D_{f,n}$$

3.14

Where the deficiency function $D_{f,n}$ is calculated using

$$D_{f,n} = D_{f,n-1} \exp \left( \frac{\Delta f}{T_f} \right) + \left( f_\alpha - f_{n-1}' \right) \exp \left( \frac{\Delta f}{2T_f} \right)$$

3.15
The time constant $T_f$ is a time constant related to unsteady boundary layer responses. This is a function of Mach number and is airfoil dependent. It is therefore recommended that $T_f$ be determined experimentally or with an unsteady boundary layer code. The final unsteady circulatory lift is calculated with using the Kirchhoff/Helmholtz model as

$$\epsilon^C_{\alpha, f, \alpha_n} = \epsilon_{\alpha, \alpha_n} \left( \frac{1 + \sqrt{f''}}{2} \right) (\alpha_e - \alpha_c)$$

3.16

This result from the Kirchhoff/Helmholtz model can be mapped back to the airfoil data tables [45]

$$\epsilon^C_{\alpha, f} = \frac{f'' + \sqrt{f''} + 1}{f + \sqrt{f} + 1} \epsilon_{\alpha, \alpha_n} (\alpha_e, M)$$

3.17

The dynamic stall model also accounts for the leading-edge vortex effects. This vortex is assumed to be fueled by the value $\epsilon_v$, which is the difference between the linearized lift and the lift achieved in dynamic stall. Using the Kirchhoff/Helmholtz

$$\epsilon_v = \epsilon_{\alpha, \alpha_n}(M)(\alpha_f - \alpha_c) - \epsilon_f(M, \alpha_f) = \epsilon^C_{\alpha, \alpha_n}(1 - K_{N, \alpha})$$

3.18

where, the factor $K_N$ is determined by

$$K_{N, \alpha} = \frac{(1 + \sqrt{f''})^2}{4}$$

3.19

This difference drives the strength of the vortex and therefore, the forces normal to the airfoil

$$\epsilon^V_{\alpha, \alpha_n} = \epsilon^V_{\alpha, \alpha_n} \exp \left( \frac{\Delta \alpha}{T_v} \right) + \left( \epsilon_{\alpha, \alpha_n} - \epsilon_{\alpha, \alpha_n-1} \right) \exp \left( \frac{\Delta \alpha}{2T_v} \right)$$

3.20

The vortex decay rate is defined by the time constant $T_v$, while $T_{\alpha_n}$ is a non-dimensional time representative of the time required for the vortex to convect past the trailing-edge of the airfoil. These constants are relatively airfoil independent and the values for the constants can be found in Ref. 46. Defining the term $\tau_v$ to be the non-dimensional time after which the vortex is formed
\[ \tau_v = T_{vl} + \frac{2(1 - f^*)}{St} \] 3.21

Where \( St \) is the Strouhal number for the formation of the leading-edge vortices, and is typically set at 0.19. Therefore, Eq. 3.20 is used while \( \tau_v \) is less than \( T_{vl} \) and when \( \tau_v \) becomes greater than \( T_{vl} \) the vortex is no longer gaining strength and \( T_{vl} \) is empirically modified by a factor of 0.5.

These vortex effects are resolved into lift and drag components and the final lift is determined by superimposing the models by

\[ \epsilon_{l,n} = \epsilon_{l,n}^C + \epsilon_{l,n}^I + \epsilon_{l,n}^\epsilon \] 3.22

Where \( \epsilon_{l,n}^C \) is defined by Eq. 3.17, the apparent mass, \( \epsilon_{l,n}^I \), by Eq. 3.5 and the vortex lift is determined by Eq. 3.20.

The drag is calculated using

\[ \epsilon_{d,n} = \epsilon_{d,0} + \epsilon_{d}^P + \epsilon_{d,n}^I + \epsilon_{d,n}^\epsilon + \epsilon_\epsilon \] 3.23

Where \( \epsilon_{d,n}^\epsilon \) is the drag component of the leading vortex term and the remaining variables are determined in the following equations.

\[ \epsilon_{d,0} + \epsilon_{d}^P = \epsilon_{d,static} (\alpha, M) \] 3.24

\[ \epsilon_{d,n}^I = \epsilon_{N,n}^I \sin(\alpha) \] 3.25

\[ \epsilon_\epsilon = \eta \epsilon_{\epsilon} (M) \alpha_c^2 \sqrt{f^*} \cos(\alpha) \] 3.26

The models were applied to a baseline VR-12 airfoil, and a VR-12 with a 0.01c height Gurney flap placed at the trailing-edge. These results are shown below in Figure 3-3 along with the results from the oscillating airfoil experiment in Section 2.2 and the CFD results from Section 2.4. The static-airfoil data used were generated in MSES for the baseline airfoil, at comparable conditions to the oscillating airfoil wind-tunnel experiment. The data is then used in the Leishman-Beddoes model to predict the Gurney flap effects. Comparing the predictions of dynamic stall with the data
from experiment and the CFD shows good agreement. It should also be noted that the magnitude lift and drag are predicted well by the model. These results show that the dynamic stall model can be extended to Gurney flaps.

![Figure 3-3: Results of Leishman-Beddoes dynamic stall model for the VR-12 airfoil with and without a Gurney flap compared with wind tunnel and CFD results. [26]](image)

### 3.3 Unsteady Flapping Model

To model the unsteady aerodynamics of MiTEs, the Harihara-Leishman (H-L) unsteady flapped airfoil model is used.[47, 48] This model incorporates compressibility, variable deployment schedules, and local freestream conditions. However, since the model was developed for plain flaps and quasi-steady airfoil theory, it must be modified empirically using CFD results.[26] Furthermore, for upstream MiTEs, the vortex formulation and advection on the lower surface of the airfoil seen in Section 2.4.2.2 must be accounted for in the model. This is done by use of a vortex model similar to that used in the dynamic stall model in the previous section.
3.3.1 Indicial Response of MiTEs

Even though MiTEs differ from plain flaps aerodynamically, the H-L model can still be applied. Figure 3-4 shows the indicial responses calculated from CFD as the MiTE is deflected at various Mach numbers. The average of these responses resembles those of which the H-L unsteady flapped airfoil model is based upon. Therefore, this model is indeed applicable to MiTEs.

![Indicial Responses at various Mach numbers calculated by CFD](image)

In order to apply the H-L model to MiTEs, the indicial responses must be approximated in some manner. The technique of R.T. Jones to approximate the Wagner function was chosen to accomplish this.[1] This technique uses the response at \( s \to 0 \) and \( s \to \infty \) and then the mean of the intermediate response is estimated by using a two-term exponential approximation. The response as
$s \rightarrow \infty$ is determined by the incremental effect of the Gurney flap found in Section 2.4.1.1 as a function of Mach number and angle of attack.

The task of determining $s \rightarrow 0$ is not trivial since it depends on the immediate response of the circulatory and non-circulatory loads. When $s=0$, apparent mass effects dominate the response of a MiTE. In the original H-L flapped airfoil model, the apparent mass loads are created by the flap moving a virtual mass of air and locally induced cambers on the flap. For MiTEs, the surrounding flow is affected in a different manner by the accumulation of vortices and flow blockage. Therefore, the H-L model is modified by using CFD to determine the effective plain flap size to approximate the apparent mass effects.

3.3.2 Modified Hariharan-Leishman Unsteady Flapped Airfoil Model

3.3.2.1 Modified Hariharan-Leishman Model Normal Force

Using the H-L flapped airfoil model, the forces and moments of MiTEs can be predicted. The normal force of MiTEs may be defined as

$$
\epsilon_N(s) = \epsilon_N^C(s) + \epsilon_{N,\delta}^I(s) + \epsilon_{N,\delta}^I(s) + \epsilon_N^I(s)
$$

Where $\epsilon_N^C(s)$ is the circulatory load, $\epsilon_{N,\delta}^I(s)$ and $\epsilon_{N,\delta}^I(s)$ are the apparent mass effects associated with the deployment of the MiTE, and $\epsilon_N^I(s)$ is the load due to the vortex formulation and advection on the lower surface on the airfoil. The first two components are given by

$$
\epsilon_N^C(s) = \Delta \epsilon_{N,GF}^C \delta_{\text{eff}}^I(s)
$$

$$
\epsilon_{N,\delta}^I(s) = \frac{2(1 - \epsilon_{\text{eff}}^I)}{M} T_{N\delta} (K_{N\delta}^n - K_{N\delta}^n)
$$

Where
\[ \delta_{\text{eff}}(t) = \delta^n - X^n_1 - Y^n_1 \]  

3.30

The geometric height of the MiTE at time \( n \) is defined as \( \delta^n \) and the variables \( X^n_1, Y^n_1, K^n_{N\delta}, \) and \( K'_{N\delta} \) are deficiency functions that account for the time history of the system.

\[ X^n_1 = X^{n-1}_1 \exp(-b_1\beta^2\Delta\tau) + A_1\Delta\delta \]  

3.31

\[ Y^n_1 = Y^{n-1}_1 \exp(-b_2\beta^2\Delta\tau) + A_2\Delta\delta \]  

3.32

\[ K^n_{N\delta} = \frac{\Delta\delta}{\Delta\tau} \]  

3.33

\[ K'_{N\delta} = K'^{n-1}_{N\delta} \exp \left( -\frac{\Delta\tau}{T'_{N\delta}} \right) \left( K^n_{N\delta} - K^{n-1}_{N\delta} \right) \]  

3.34

Where the time constant used for the flap position, \( T'_{N\delta} \), is

\[ T'_{N\delta} = \frac{2M(1 - \varepsilon_{\text{eff}})}{(1 - M) + 2F_{10}\beta M^2(A_1b_2 + A_2b_1)} \]  

3.35

The effective plain-flap hinge location is defined as \( \varepsilon_{\text{eff}} \) and is defined as the length in semi-chords, from the mid-chord to the location of the effective flap hinge. This is adjusted to account for an equivalent apparent mass produced by the MiTE. It also should be noted that the constant of \( A_1, A_2, b_1, \) and \( b_2 \) are not the same constants as in the dynamic stall model. The constants “\( F^n_{\alpha} \)” are geometric constants and are determined by \( \varepsilon_{\text{eff}} \).

The model also accounts for the flap rate of the MiTEs. This is apparent mass effect as the surrounding flow is affected depending on the rate of deployment. Here, \( \varepsilon_{\text{eff}} \) is used to capture the effects of the flap rate

\[ \varepsilon'_{N\delta}(t) = \frac{(1 - \varepsilon_{\text{eff}})^2}{2M} T'_{N\delta} \left( K^n_{N\delta} - K'^{n}_{N\delta} \right) \]  

3.36

Where the deficiency functions, \( K^n_{N\delta} \& K'_{N\delta} \), and time constant, \( T'_{N\delta} \), are defined as
Modified Hariharan-Leishman Model Pitching Moment

The pitching moment is also modeled using the H-L flapped airfoil method. The components of the unsteady pitching moment are given by

\[ \epsilon_m(t) = \epsilon_m^C(t) + \epsilon_{m,\delta}(t) + \epsilon_{m,\delta}^I(t) + \epsilon_m^J(t) \]

where the first two terms are

\[ \epsilon_m^C(t) = \Delta \epsilon_{m,G} \delta_{eff,m}(t) \]
\[ \epsilon_{m,\delta}(t) = \frac{-1}{2M} (1 - e_{eff}) T_{m\delta} \left( K_{m\delta}^u - K_{m\delta}^{u-1} \right) \]

And \( \delta_{eff,m}(t) \), deficiency functions, and time constants are defined as

\[ \delta_{eff,m}(t) = \delta^u - X_3^u \]
\[ X_3^u = X_3^{u-1} + \exp(-b_3 \beta^2 \Delta t) + A_3 \Delta \delta \]
\[ K_{m\delta}^u = \frac{\Delta \delta}{\Delta t} \]
\[ K_{m\delta}^{u-1} = K_{m\delta}^{u-1} \exp \left( \frac{-\Delta t}{T_{m\delta}} \right) - \left( K_{m\delta}^u - K_{m\delta}^{u-1} \right) \]
\[ T_{m\delta}^I = \frac{M(1 - e_{eff}) (2 + e_{eff})}{3(1 - M) + 2(F_4 + F_{10}) \beta M^2 (A_3 b_3)} \]
As in Eq. 3.36, the flap rate can be incorporated into the unsteady pitching moment calculations. Yet again, this is an apparent mass effect depending on the surrounding flow and the rate of deployment of the MiTE.

\[
\epsilon_{N\delta}^{\prime}(\tau) = \frac{2\left(1 - e_{\text{eff}}\right)^3 - 3\left(1 - e_{\text{eff}}\right)^2 - 2(12e_{\text{eff}} - 4)}{24M} T_{m\delta}^\prime \left( K_{m\delta}^u - K_{m\delta}^{u'} \right)
\]

Where the deficiency functions, \( K_{m\delta}^u \) & \( K_{m\delta}^{u'} \), and time constant, \( T_{m\delta}^\prime \), are defined as

\[
K_{m\delta}^u = \frac{\Delta \left( \dot{\delta} / V \right)}{\Delta t}
\]

\[
K_{m\delta}^{u'} = K_{m\delta}^{u-1} \exp \left( -\frac{\Delta \left( \dot{\delta} / V \right)}{T_{m\delta}^\prime} \right) - \left( K_{m\delta}^u - K_{m\delta}^{u-1} \right)
\]

\[
T_{m\delta}^\prime = \frac{2M \left[ 1 + e_{\text{eff}} \right]^3 - 1.5 \left[ 1 - e_{\text{eff}} \right]^2 - \left( 12e_{\text{eff}} - 4 \right)}{9(1 - M) \left[ 1 - e_{\text{eff}} \right] + 6F_1 - F_3 - \left( e_{\text{eff}} + 0.5 \right) F_4 + 0.5F_{11} B M^2 / A_3 B_3}
\]

The final term is the additional pitching moment from an upstream MiTE and the lower surface vortex that is produced. This will be discussed in a later section just as with the addition normal force.

### 3.3.2.3 Modified Hariharan-Leishman Model Drag Force

The unsteady drag is given as

\[
\epsilon_d = \epsilon_{d, p} + \epsilon_s + \Delta \epsilon_{d, GF} \left( \frac{\delta^u}{\delta} \right) + \epsilon_{d, u}^{\prime}
\]

Where the pressure drag, \( \epsilon_{d, p} \), and the leading-edge suction force, \( \epsilon_s \), are defined as

\[
\epsilon_{d, p} = \epsilon_N \alpha + \Delta \epsilon_{d, \text{eff}}(s) \delta_{tf, \text{eff}}
\]

\[
\epsilon_s = \frac{2\pi}{\beta} A_0^2
\]
The third term is the additional pressure drag on the surface of the MiTE, which is scaled as a portion of the Gurney flap drag. The final term in the equation is the additional drag from the vortex produced from an upstream MiTE and will be defined in the next section.

The term $\epsilon_{f, \text{eff}}$ in Eq. 3.53 the effective plain flap force and $\delta_{pf, \text{eff}}$ is the effective plain flap deflection angle. The $A$ term in the leading-edge suction equation, Eq. 3.54, is the leading term from quasi-steady thin-airfoil theory. There needs to be an addition to this term due to the MiTE and added to the result of Eq. 3.54

$$\Delta A_{e,MiTE} = \cos^{-1}(\epsilon_{\text{eff}}) \sin \left( \frac{\delta_{pf, \text{eff}}}{\pi} \right)$$ 3.55

And the effective plain flap angle is defined as

$$\delta_{pf, \text{eff}} = \frac{\beta \delta_{\text{eff}}}{2F_{10}}$$ 3.56

### 3.3.3 Additional Effects from Upstream MiTEs

To account for the effects of MiTEs located upstream of the trailing edge, a vortex model similar to that used in the dynamic stall model is used (Eq. 3.20). This model accounts for the vortex scrubbing witnessed on the lower surface of the airfoil in Section 2.4.2.2. The strength of this vortex may be determined as

$$\epsilon_{N, \sigma}^{(N, \sigma)} = \epsilon_{N, \sigma}^{(N, \sigma)} \exp \left( \frac{\Delta s}{T_{v, \sigma}} \right) + (\epsilon_{N, \sigma}^{(N, \sigma)} - \epsilon_{N, \sigma-1}^{(N, \sigma-1)}) \exp \left( \frac{\Delta s}{2T_{v, \sigma}} \right)$$ 3.57

The time constant, $T_{v, \sigma}$ accounts for the vortex decay and MiTE position, $\sigma$

$$T_{v, \sigma} = T_{vd} + \frac{2(1 - \epsilon \sqrt{f_{\sigma}})}{Sf}$$ 3.58
The unknown term, $T_{oh}$, that are used to determine this time constant are varied to match CFD results found in Section 2.4.2.2. There are also apparent mass effects associated with the vortex that must be accounted for

$$
\epsilon_{N,a}^{l \prime} = \frac{4K_a T_{r,l}}{M} \left( \frac{\Delta \delta}{\Delta t} - D_\eta \right)
$$

Combining the circulatory and apparent mass effects gives the total force effect of the vortex on the lower surface of the airfoil

$$
\epsilon_N^{l \prime} = \epsilon_{N,a}^{l \prime} + \epsilon_{N,a}^{l \prime} \Delta T
$$

The pitching moment will also be affected by this vortex. Based on methods used in Ref. 44, the center of pressure was formulated as

$$
CP_{el} = 0.20 \left[ 1 - \cos \left( \frac{\pi T_{el}}{T_{el,l}} \right) \right]
$$

Therefore, the pitching moment is

$$
\epsilon_m^{l \prime} = -CP_{el} \epsilon_N^{l \prime}
$$

There is an additional modification to the Hariharan-Leishman Unsteady Flapped Airfoil Model that must be made in order to adequately model the effects of an upstream MiTE. The effective plain flap location, $e_{eff}$, must be corrected depending of the location of the MiTE. Using the CFD results, this term is modified empirically as

$$
e_{eff} = 0.9 \sqrt{\epsilon}
$$

The formulation of this term should be investigated further in the future as it has a large impact on the Modified H-L Model.
3.3.4 Parameters of Modified Harihan-Leishman Model

The parameters $A_1$, $A_2$, $A_3$, $b_1$, $b_2$, $b_3$ and $\epsilon_{\text{eff}}$ that were discussed in previous sections are calibrated to match a sinusoidal deployment result. This enables the model to achieve a reasonable agreement in various operating conditions. The parameters are further adjusted to fit the intermediate response to an indicial step deployment. Since the flows are fundamentally different during deployment and retraction, two sets of parameters are used. The values used for a MiTE placed at the trailing edge are seen in Table 3-1.

Table 3-1: Modified H-L model parameters for a MiTE at the trailing edge

<table>
<thead>
<tr>
<th>M</th>
<th>$\epsilon_{\text{eff}}$</th>
<th>$A_1$</th>
<th>$A_2$</th>
<th>$A_3$</th>
<th>$b_1$</th>
<th>$b_2$</th>
<th>$b_3$</th>
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<tbody>
<tr>
<td>MiTE Deploys</td>
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<td></td>
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<tr>
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<td>0.20</td>
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<td>0.20</td>
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<td>0.40</td>
<td>0.20</td>
<td>0.02</td>
<td>1.75</td>
</tr>
<tr>
<td>0.3</td>
<td>0.9</td>
<td>0.38</td>
<td>0.30</td>
<td>0.50</td>
<td>0.50</td>
<td>0.02</td>
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<tr>
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<td>1.00</td>
<td>0.50</td>
<td>0.07</td>
<td>1.50</td>
</tr>
</tbody>
</table>

These parameters also need to be adjusted slightly for an upstream MiTE as the location of the MiTE changes and $\epsilon_{\text{eff}}$ varies.

3.3.5 Results of the Modified Harihan-Leishman Model

Using the completed model, the first case of a MiTE placed at the trailing edge of the airfoil is investigated.
3.3.5.1 Trailing Edge MiTE

The first case investigates the effects of trailing-edge MiTE at a low Mach number of 0.1 and a reduced frequency of \( k = 0.14 \) (1/rev) seen in Figure 3.5.

![Graphs showing comparisons between CFD and modified H-L unsteady flapping model for \( M=0.1, \alpha=0 \) deg, \( k=0.14 \) (a) lift (b) drag (c) moment [26]](image)

These data show that the model agrees well with the CFD for the lift, drag, and pitching moment at these conditions. Figure 3.6 shows results from when the reduced frequency is increased to 0.56 (4/rev).
This case again shows good agreement between the CFD and the H-L model. Next is to compare the model with CFD when there are changes in angle of attack. To do this, the angle of attack is set at an angle of attack of 10 degrees. The results are shown in Figure 3-7.

Figure 3-6: Comparisons between CFD and modified H-L unsteady flapping model for $M=0.1$, $\alpha=0$ deg, $\kappa=0.56$ (a) lift (b) drag (c) moment [26]
In this case, there is good agreement between the CFD and the H-L model in terms of lift and pitching moment. There are however slight discrepancies between the drags. This was to be expected as it is hard to both model and experimentally measure drag especially with the cases at hand. A high Mach number case is then investigated to see the compressibility effects. Here, the freestream Mach number is increased to 0.6 and the results are seen in Figure 3-8.
Again, the lift and pitching moment correlate well between the CFD and H-L model. The discrepancy in drag is seen again.

Finally, the modified H-L model was used to calculate the lift amplitude while the Mach number was varied. The results of the model, seen in Figure 3-9, show good agreement with those predicted with CFD seen in Figure 2-23. This means that the model is capable of being used for vibration control purposes as it captures the effects to lift from both changes in Mach number and reduced frequency.

Figure 3-8: Comparisons between CFD and modified H-L unsteady flapping model for $M=0.6$, $\alpha=0$ deg, $k=0.5$ (a) lift (b) drag (c) moment [26]
3.3.5.2 Upstream MiTE

The model is now tested with a MiTE located upstream of the trailing edge of the airfoil. The first case is that of a MiTE placed at 0.90c and a height of 0.02c at a Mach number of 0.45 and a reduced frequency of $k = 0.14$ (1/rev).
In this case, there is good agreement between the CFD and the H-L model in terms of lift and pitching moment. There are however slight discrepancies between the drags as seen for the trailing-edge MiTE. The reduced frequency of the MiTE deployment was increased to 0.56 (4/rev) and the results are seen in Figure 3-11
Again, there is good correlation between the CFD and the model for the lift and pitching moment with some discrepancy in terms of drag. The location of the MiTE is now varied from 0.85c to 1.00c at a freestream Mach number of 0.6 and a reduced frequency of 1.0.

Figure 3-11: Comparisons between CFD and modified H-L unsteady flapping model of a 2% MiTE placed at 0.9c for $M=0.45$, $\alpha=0$ deg, $k=0.56$ (a) lift (b) drag (c) moment
This case shows an agreement between the CFD and the upstream model for lift and pitching moment. These results of lift and pitching moment for the upstream MiTEs at varying location, Mach number, and reduced frequency give good confidence in their validity.

### 3.4 MiTE Aerodynamic Model

The effects of the unsteady aerodynamics, dynamic stall, and deployment of MiTEs discussed in the previous sections can be put together to produce a complete aerodynamic model of MiTEs. To better understand the entire model, a flowchart is used as a visual aid and can be seen in Figure 3-13.
Figure 3.13: Flowchart of MiTE Aerodynamic Model

1. Static Airfoil Force and Moment Data
   - Oscillating Airfoil
     - Leishman-Beddoes Compressible Unsteady Aero Model
       \( f = F, M, q, \alpha \)
     - Indicial Response Formulation, \( \epsilon_{n}^{\alpha} \)
     - Unsteady Apparent Mass Loads
     - Unsteady Circulatory Loads
     - MITE Deployment or Gurney Flap

2. Determine Maximum \( \Delta c_{L}, \Delta c_{D}, \Delta c_{m} \) Due to Gurney Flap
   - Specified MITE Deployment Schedule
   - Determine Reduced Time of Deployment
     - Trailing-Edge Location of MITE or Gurney Flap
     - Upstream Location of MITE or Gurney Flap
     - Effects of Vortex Formulations and Advection
     - Apparent Mass Effects
     - "Equivalent" Flap Size & Deflection
       - Modified Hartharan-Leishman Compressible Unsteady Flapped Airfoil Model
       - Unsteady MITE Contribution

3. Check for Static Airfoil Stall
   - Uninstalled
   - Stall
     - Leishman-Beddoes Dynamic Stall Aerodynamic Model
     - Kirchoff/Helmholtz
     - Pressure Lags - Unsteady Separation Point
     - Unsteady Circulatory Loads
     - Unsteady Apparent Mass Loads
     - Leading-Edge Vortex Contribution
     - Unsteady Airfoil Aerodynamics w/ Dynamic Stall and MITE Contributions
Following this flowchart, the aerodynamic characteristics of a MiTE may be completely modeled including the deployment and dynamic stall. Figure 3-14 shows the lift response of a MiTE including the effects of dynamic stall.

![Figure 3-14: Results of the MiTE Model](image)

The comparison of this lift response with that of the CFD results shows good correlation and gives confidence in the MiTE model. It should be noted however that the drag results of the CFD and that of the model do not match. This is due to the Modified Hariharan-Leishman model failing to fully capture the initial drag rise due to the indicial step deployment of the MiTE. Nevertheless, the overall results are satisfactory in providing a fairly accurate and efficient method in calculating the aerodynamics of MiTEs.
3.5 Summary

The Lesihman-Beddoes dynamic stall model was applied to an airfoil with a Gurney flap to capture the effects during dynamic stall. This model was compared with experimental results showing the accuracy of the dynamic stall model for airfoils with and without Gurney flaps. Since the dynamic stall models are based on static airfoil data, the baseline data sets are modified to obtain the airfoil characteristics when using Gurney flaps throughout a full range of angles of attack and Mach numbers by use of CFD results. Then, a modified Hariharan-Leishman unsteady flapped airfoil model is used to calculate the unsteady aerodynamics of MiTEs. This model also includes a vortex model to capture the effects of vortex formulation and advection of an upstream MiTE.

These models produce lift and pitching moment results that are valid for large angles of attack, Mach numbers and varying location of the MiTE. The drag results, however, showed discrepancies and require refinement.
Chapter 4
MiTE Rotor Performance

The application of Gurney flaps to helicopter rotors shows potential to increase the rotor performance at high thrust levels and in forward flight. This is, however, accompanied by a reduction in performance in low thrust and low-speed flight conditions. [48] These performance investigations were then extended by the use of trailing-edge MiTEs. [25] These MiTEs were able to selectively deploy along the blade span and around the azimuth of the rotor showing increases in rotor performance throughout the flight envelope. This particular research also suggests an application to vibration control. The current objective is to investigate the possible rotor performance enhancements of upstream MiTEs.

4.1 ROTOR Program

The rotor performance enhancements due to MiTEs are investigated using a modified version of ROTOR, a rotor performance analysis code based on blade-element theory.[3] This program predicts the required power for hover and level forward flight with a trimmed rotor state. The rotor is trimmed in the longitudinal and vertical directions as well as trimming the pitching and rolling moments to zero. This program does not model a tail rotor, so the rotorcraft is not trimmed in the lateral direction or yawing moment.

The inflow model used in the computer program is a non-uniform cosine downwash model. Blade flapping with a hinge offset is also modeled although lead-lag dynamics are neglected. The aerodynamic loads are calculated and then combined with the inertial forces. These combined forces
are then integrated along the span of the blade and around the azimuth to determine the rotor thrust, torque, and power.

4.1.1 Generation of Airfoil Data Tables

In order for Rotor to compute the aerodynamic loads, a table of airfoil force coefficients spanning a range of angles of attack and Mach numbers is needed. For MiTE performance calculations, two tables are needed for each airfoil. One table is needed for the baseline characteristics and another for the airfoil equipped with a Gurney flap.

The baseline airfoil characteristics were determined through the use of MSES.[38] This provided a consistent set of results over a range of operation of the rotor. It was used to predict \( \alpha \), \( \alpha_b \) \( and \( c_w \) for the baseline airfoil from positive to negative stall and at Mach numbers from 0.0 to 1.0. The sectional characteristics are assumed to be independent of Reynolds number. This is supported by the fact that MSES predicted transition forward of 20 percent chord over most of the operational range. The wind tunnel tests in Chapter 2 also support these results by the small difference seen in lift coefficients between natural and fixed transition. Therefore, transition was set to 2 percent chord on the upper surface and 7 percent on the lower in MSES. With these characteristics of the flow, the data generated in MSES at the Reynolds number 4.0x10^6, which is the average value at blade mid-span around the azimuth. To generate the tables for the Gurney flapped airfoils, the \( \Delta c_l \), \( \Delta c_d \) and \( \Delta c_w \) values from CFD were added to the baseline tables at various Mach numbers.

The Rotor program requires airfoil data at angles of attack ranging from \( -180^\circ < \alpha < 180^\circ \). To obtain this needed data beyond positive and negative stall, the airfoil data from MSES was blended into that of the NACA 0012 airfoil at high angles of attack. [49] In situations where the airfoil is approaching stall and MSES was unable to converge, the Kirchhoff/Helmholtz model is used to predict stall based on a combination of converged
predictions and the NACA 0012 data. This sort of situation usually occurs at Mach numbers greater than the critical Mach number. In the analysis of the rotor, this data usually don’t have to be implemented since it occurs at relatively high Mach numbers on the advancing side of the rotor where stall is typically avoided.

4.1.2 Aerodynamic Models

4.1.2.1 Oscillating Airfoil Model

To model the aerodynamics during the rotor analyses, the models are separated into two categories. The first category is the modeling of the baseline airfoil without a MiTE. There are three different types of models that can be used: quasi-steady, unsteady, or dynamic stall. The quais-steady model is based on the airfoil data and the angle of attack is based on the pitch rate, downwash, blade flapping, and other motions. The unsteady aerodynamic model modifies the quasi-steady angle of attack based on indicial methods as discussed in Section 3.2.1.1.

The Leishman-Beddoes dynamic stall model is used to determine additional unsteady effects such as lags in the separation and pressure responses, and considers the effects of the leading–edge vortices as discussed in Section 3.2.1.2. In the rotor performance code, the dynamic stall parameters are based upon the experimental data of the NACA 0012 airfoil since data is available. This approximation is due to the fact that the separation and pressure time constants, $T_f$ and $T_p$, are somewhat airfoil-independent parameters. For future projects where the rotor is being designed, more appropriate constants should be selected that are more representative of the actual airfoil.
4.1.2.2 Oscillating MiTE Model

The second category of aerodynamic models is one that pertains to the modeling of MiTEs. The program is set up to use a separate set of data that represents the same airfoil with the specific Gurney flap being used attached to the airfoil. This provides the needed $\Delta \epsilon_{l,GF}$ and $\Delta \epsilon_{d,GF}$ as a function of angle of attack and Mach number.

The user has the ability to select a steady linear model or unsteady flapped airfoil model. The linear model assumes there are no lags and uses linear interpolations between the baseline and the Gurney flap data sets. This is based on the instantaneous height of the MiTE.

The unsteady flapped airfoil model incorporates the model found in Section 3.3 into the rotor performance code. The model adds the unsteady effects of the motion of the MiTE. This determines the lead and lag of the system and its effects overall. The effects of the upstream MiTEs, where there is the formulation and advection of vortices on the lower surface of the airfoil, are also accounted for.

4.1.3 MiTE Deployment Scheme

The primary use of MiTEs in this research is stall alleviation as a means to increase rotor performance. The objective is initially to have the MiTE retracted at lower angles of attack when additional lift is not needed. When the lift requirements increase, MiTEs deploy to achieve higher lift without a stall drag penalty. The MiTE remains deployed as need to provide additional lift and then retracts when the lift-to-drag ratio drops below that of the baseline airfoil. The deployment schedule the MiTE is shown in Figure 4-1.
Implemented into the rotor performance code, the behavior of the MiTE around the azimuth can be seen in Figure 4-2. Here, the position of the MiTE is plotted against the $\alpha$ and Mach number. For this particular case, the unsteady model was used to determine $\alpha$.  

Figure 4-1: Deployment schedule of MiTEs
4.2 Rotor Performance Analysis

The rotor performance comparisons are based upon the Boeing/Sikorsky RAH-66 Comanche helicopter. The rotor blades utilize the VR-12 airfoil on the inboard sections and transition to a SSC-A09 airfoil on the outboard sections. As the break point for the airfoil sections was not published, it was assumed to be located at $r/R=0.75$. Table 4-1 shows the other parameters of the RAH-66 Comanche used in the performance analysis.
The MiTE used in these studies is located at 0.90c and has a height of 0.02c. They are also assumed to span the entire span of the blade. The dynamic stall model and the unsteady flapped airfoil model is used in performance predictions. This is done to capture as much of the effects as possible of the lower surface vortex associated with the upstream MiTE.

### 4.2.1 Importance of Unsteady Modeling

In previous research of trailing-edge MiTEs, the linear interpolation model was used to determine the effects of MiTEs, since there is only a small difference between it and the unsteady model. As shown in Figure 4-3, this is not the case with an upstream MiTE, for which the required power of rotor over a range of advance ratios is influenced by the use of the unsteady MiTE model.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
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<tbody>
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<tr>
<td>Rotor Diameter</td>
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</tr>
<tr>
<td>Hub Radius</td>
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</tr>
<tr>
<td>Hinge Offset</td>
<td>1.5 ft</td>
</tr>
<tr>
<td>Blade Chord</td>
<td>1.1 ft</td>
</tr>
<tr>
<td>Linear Twist</td>
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</tr>
<tr>
<td>Tip Speed</td>
<td>700 ft/s</td>
</tr>
<tr>
<td>Blade Weight</td>
<td>2.9 lb/ft</td>
</tr>
<tr>
<td>Equiv. Flat Plate Drag</td>
<td>18 ft²</td>
</tr>
<tr>
<td>Base Weight</td>
<td>10600 lb</td>
</tr>
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</table>
This dictates that the unsteady aerodynamic model should be modeled to capture the additional power required by the rotor.

Another aspect of the rotor that is affected by the aerodynamic model chosen is that of the MiTE deployment around the azimuth. Figure 4-4 shows a MiTE deployment comparison of a rotor in high speed flight ($\mu = 0.4$) using the quasi-steady model versus the dynamic stall with the unsteady MiTE model. Here, MiTEs are fitted to the outboard airfoil only and deployed as indicated by the MiTE is deployed in the shaded region. Again, sizable differences are seen between the two cases as to where the MiTE deploys and retracts.
4.2.2 Pitching Moment Concern

A major concern in the rotorcraft community concerning rotor blades is that of the pitching moment. Due to the concern over pith-link loads in dynamic stall, the pitching moment needs to be kept to a minimum. The pitching moment coefficient does increase as MiTEs are deployed, but whether or not the moment actually increases depends on where on the rotor, in span and azimuth, the MiTEs are located. If the load is beyond the limits of the blade, MiTEs will not be able to be used. Figure 4-5 shows the maximum pitching moment on each side of the rotor over a range of advance ratios. These are normalized by the maximum pitching moment on the advancing side of the rotor, which is taken to be the limiting case. It is seen that even with the MiTEs fully deployed in hover, the pitching moment is still lower than the limiting case.
4.2.3 Forward Speed Performance

In previous research, it has been shown that MiTEs are capable of increasing the maximum speed of the rotorcraft.\[25\] This is consequence of a MiTE’s ability to delay stall on the retreating blade. This reduced drag and increased the lift-to-drag ratio allows greater speeds. It was also demonstrated that the greatest effect is due to the increased its maximum lift on the transonic airfoil. Hartwich also showed that changing the airfoil distribution across the blade had the possibility of increasing the maximum speed of the rotorcraft.\[3\] It was therefore decided to explore changing the airfoil distribution on the rotor of the Comanche and use MiTEs along the span of the blade to produce a greater maximum speed. The results can be seen in Figure 4-6
Starting at the initial radial transition of $r/R=0.75$, the use of the SSC-A09 transonic airfoil was increased by moving the breakpoint inboard. This reduced the compressibility drag on the advancing blade by the increasing use of the transonic airfoil and alleviating stall on the retreating blade with use of the MiTEs. The maximum speed gradually increased from 175 kts to 210 kts before the speed decreasing again, a consequence of increased retreating blade stall. This increase shows a possible increase of 20 percent in the maximum forward speed of the Comanche.

4.2.4 Maximum Altitude Performance

Another scenario involves increase use of the inboard airfoil, the VR-12, and MiTEs that span the length of the rotor blade to increase the service ceiling of the Comanche helicopter. The same method was used as in the maximum speed case, where the radial transition point is moved on
the rotor blade. This time however, instead of moving the transition point inboard, its moved outboard. The results are seen in Figure 4-7.

![Service Ceiling Increase of Comanche](image)

**Figure 4-7: Service Ceiling Increase of Comanche**

Here, the rotor benefits from the additional $\phi$ of both the rotor and the MiTE. As was expected, the radial distribution moved outboard to the tip. The service ceiling increased from 14,980 ft to 16180 ft, an increase of 8 percent.

### 4.3 Advanced Designs

Although large gains in performance have been seen in these studies, there is potential for even greater gains. The following are few areas that can room for improvement in order to optimize the performance of MiTEs.
4.3.1 Rotors Designed with MiTEs

One area that has potential to produce large gains in performance through the use of MiTEs is that of geometric configuration. All of the situations analyzed in this research used existing rotors. Clearly, additional gains could be achieved if a rotor was designed utilizing MiTEs from the onset. Sections 4.2.3 and 4.2.4 did preliminary investigations into this and showed the potential increases possible through the use of MiTEs and by changing the existing airfoil distribution along the span of the blade.

Another potential area of improvement is in the design of airfoils used on the rotor blades. Up until now, rotorcraft airfoils have been selected based upon their static characteristics. It was seen in Chapter 2 that the unsteady characteristics of the airfoils did not necessarily correlate well with the static characteristics. It would then be very beneficial to investigate the design of airfoils for an unsteady environment and the use of MiTEs. This possibly has the potential of performance increase above those seen here in this research.

4.3.2 Deployment Schemes

The MiTE deployment scheme used was that of deploying when more $c_l$ is needed to avoid stall and retracting once the lift-to-drag ratio was less than that of the baseline airfoil. Initially, an instantaneous deployment scheme was used. This deployment schedule was smoothed to produce a model that accounts for realistic deployment time.

Currently, there are other research efforts underway that are tailoring the lift over the rotor to produce an optimal distribution and reduce the induced power requirements. These require a deployment scheme that is capable of a partial deployment of a MiTE, something that is not possible with the current deployment scheme, but is easily attainable.
4.4 Summary

The use of upstream MiTE was investigated for the performance enhancement and extension of the flight envelope for rotorcraft. Using the rotor performance code ROTOR, the importance of unsteady aerodynamics modeling was shown by the difference in power requirements between the models. Furthermore, the difference in MiTE deployment was shown with the unsteady case having greater use of MiTEs over the azimuth of the rotor disk. The issue of increased pitching moment on the blades due to MiTE deployment was shown not to be an issue. This was shown by that the fact the pitching moment increase due to MiTEs is less than that of the baseline airfoil on the advancing blade during high speed flight.

As for performance enhancements, the RAH-66 Comanche helicopter was used as the model helicopter. It was shown that using MiTEs and altering the airfoil distribution on the rotor yielded large increases in maximum speed. A large increase of 8 percent in service ceiling altitude was achieved by again altering the airfoil distribution and the use of MiTEs. Further improvements are possible by a number of methods such as the design of airfoils on the rotor and different deployment schemes of the MiTEs.
Chapter 5

Conclusion

5.1 Summary of Results

MiTEs placed at the trailing edge and upstream of the trailing edge have been shown to be an effective rotor performance enhancement device. A series of experimental, numerical, and analytical investigations of MiTEs support this and show the aerodynamic potential.

5.1.1 Wind Tunnel Experiments

Both of the experiments presented in Chapter 2 investigated the aerodynamic characteristics of the passive form of MiTEs, the Gurney flap. The first wind tunnel experiment was a static wind tunnel test at the Penn State Low-Speed, Low-Turbulence Wind Tunnel where the Gurney flap varied in height and chord-wise locations. They were found to provide similar lift increases upstream of the trailing edge of the airfoil. It was found, however, that performance penalties were incurred when the Gurney flap was moved forward.

The second wind tunnel experiment took place at NASA Ames’ Compressible Dynamic Stall Facility and analyzed an oscillating airfoil. This experiment had a Gurney flap on the model and showed similar aerodynamic effects through dynamic stall as were shown in the static experiment. At higher coefficients of lift, the Gurney flap increased the aerodynamic efficiency and demonstrated the ability to decrease the pitching moments at which the baseline airfoil stalls.
5.1.2 CFD Investigations

MiTEs were investigated using the CFD program OVERFLOW2. These studies involved MiTEs placed at the trailing edge and upstream of the trailing edge. The results showed that the chordwise location of the MiTE greatly affected the unsteady aerodynamics. When MiTEs are placed at the trailing-edge, the results agree with unsteady aerodynamic theories that are valid for plain flaps. As the MiTE was moved upstream, strong vortical flows were created on the lower surface of the airfoil. This vortex caused delays in the development of lift and large phase lags between the resulting aerodynamic forces and MiTE deployment position. This also showed possible decreases in the airfoil’s sectional performance characteristics at high deployment frequencies and low speeds. Nevertheless, CFD showed that MiTEs delay stall to significantly higher lift coefficients. These qualities make these devices highly desirable on rotorcraft for performance enhancement and vibration control.

5.1.3 Aerodynamic Modeling

The Leishman-Beddoes unsteady aerodynamics and dynamic stall models were used to model the effects of dynamic stall of an airfoil with Gurney flaps. The model was compared with experimental results from the NASA Ames test and showed good correlation between the two sets of data. For the baseline airfoil characteristics, MSES was used and then the addition effects of the Gurney flaps were determined from the CFD data all throughout a range of angles of attack and Mach numbers.

For MiTEs placed at the trailing-edge, a modified version of the Hariharan-Leishman unsteady flapped airfoil model was used to capture the unsteady aerodynamics of a MiTE deployment. This model was further modified for MiTEs located upstream of the trailing edge by modeling the lower surface vortex and associated phase lags. The results of this model are also in
agreement with those of the CFD data for both lift and pitching moment, although discrepancies in drag show the need for more research. The unsteady aerodynamic model, dynamic stall model, and unsteady flapped airfoil model were then coupled and showed good agreement with CFD to model MiTEs in dynamic stall.

5.1.4 MiTE Rotor Performance

The aerodynamic models were implemented into a blade-element rotor performance code to investigate the effects of MiTEs on rotorcraft. The model helicopter used was the Boeing/Sikorsky RAH-66 Comanche and the rotor blades utilized MiTEs placed upstream of the trailing edge. First, the importance of unsteady aerodynamic modeling was investigated. This showed differences in both power required and MiTE deployment around the azimuth of the rotor confirming the need for adequate unsteady modeling. The pitching moment of MiTEs was investigated and showed that MiTEs did not exceed the pitching moment seen on baseline rotor blades during forward flight. It was also shown that the maximum forward speed and service ceiling can be increased through the use of MiTEs and varying the airfoil distribution along the span of the rotor blade.

5.2 Conclusions

With the results presented of MiTEs placed upstream of the trailing edge, there are numerous applications to rotorcraft. Segmenting the devices along the span of the blade would provide azimuthal control of the lift distribution for minimizing induced drag, vibration control and increasing rotor performance. All of these benefits would be realized with a device that possess actuation loads many orders of magnitude less than conventional individual blade control systems.
List of References


