

Optimization for Load Alleviation of Truss-Braced Wing Aircraft With Variable Camber Continuous Trailing Edge Flap

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This paper focuses on load alleviation optimization for a high aspect ratio truss braced wing (TBW) aircraft. The TBW aircraft model is based on the Subsonic Ultra Green Aircraft Research (SUGAR) concept developed by Boeing, with the wing structures of the model modified to include a novel aerodynamic control surface known as the Variable Camber Continuous Trailing Edge Flap (VCCTEF). The purpose of the study is to investigate the effectiveness of a Performance Adaptive Aeroelastic Wing (PAAW) technology, specifically the VCCTEF, for alleviating load on the TBW wing during flight maneuver. The specific flight maneuver under consideration in this study is a 2.5g pull-up maneuver. Constrained gradient-based optimization is conducted to tailor the deflections of the VCCTEF such that bending moment along the wing is minimized at the 2.5g pull-up flight condition. Aerodynamic modeling for this study is conducted using a vortex-lattice method code called Vorlax. A non-linear finite element analysis (FEA) method is constructed for analyzing the structural deformation and resulting bending moment along the wing of the aircraft with the inclusion of effects from tension-stiffening due to axial loading in the truss. This study is the first phase of several, and involves optimization of a rigid wing aircraft for preliminary analysis. Future studies will incorporate flexible wing structures with aeroelastic interactions and deformations. The results of this first phase positively demonstrate the potential of utilizing the novel control surface on modern aircraft wing designs for shaping control in order to provide load alleviation during flight maneuver.

I. Introduction

With recent focus in the aviation industry on the need for reduced environmental impact and reduced fuel burn, demand for green technologies and designs is expected to increase. Large transport aircraft today use the conventional tube-and-wing design and only incremental improvements have been made in aerodynamic efficiency in the past few decades. In recent years, one such improvement is the use of lightweight materials such as composites, which have been shown to allow for a significant reduction in weight resulting in reduced drag and thus improved energy efficiency. The Boeing 787 is an example of a modern aircraft design that incorporates lightweight structures. Another opportunity for further efficiency improvements currently

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being investigated and pursued is in the form of high aspect ratio wing designs. However, as wing aspect ratio increases, the need for maintaining sufficient load carrying capacity becomes increasingly important. Traditional cantilever wing designs can only accommodate up to a certain aspect ratio beyond which the wing root bending moment becomes too large imposing structural and weight limitations on the wing design. Truss braced wing (TBW) aircraft concepts provide a structural solution for high aspect ratio wing designs. The long slender wing includes structural bracing via the use of a truss member that provides intermediate span supports in addition to the wing root attachment. These truss members generally support a portion of the spanwise load carried by the wing and are loaded in tension.

The Subsonic Ultra Green Aircraft Research (SUGAR) TBW aircraft concept is a Boeing developed N+3 aircraft configuration funded by the NASA ARMD Advanced Air Transport Technology (AATT) Project.^{1,2} The SUGAR TBW is designed to be aerodynamically efficient by employing an aspect ratio on the order of 19, which is significantly greater than the aspect ratio of conventional aircraft. The wings are braced at approximately mid-span by two trusses, and two smaller jury struts, one on each wing, provide additional reinforcement. Figure 1 shows an illustration of the SUGAR TBW aircraft concept.



Figure 1: Boeing SUGAR TBW aircraft concept.

Research into the TBW as a viable future generation aircraft is presently being conducted. Owing to its high aspect ratio wing constructed from flexible modern materials, significant bending deformations, twisting deformations, and aeroelastic interactions are expected for the aircraft. These deformations and aeroelastic interactions may result in adverse aerodynamic effects such as increased drag as the wing deforms to a non-optimal shape, as well as adverse structural effects as loading on the structure is increased during flight maneuvers. Previous conceptual studies, such as a 2010 study titled “Elastically Shaped Future Air Vehicle Concept,”³ were conducted to address such issues. The study produced results demonstrating potential aerodynamic and load alleviation benefits from using active control technology to tailor the wing shape during flight. A Performance Adaptive Aeroelastic Wing (PAAW) technology control surface known as the Variable Camber Continuous Trailing Edge Flap (VCCTEF) was proposed as a control effector^{3,4} to act as a wing shaping device. Several previous conceptual design studies have been conducted investigating the potential of the VCCTEF system for drag reduction at off-design cruise flight conditions for a flexible wing aircraft representative of a current generation commercial aircraft model.⁵⁻⁷ These studies produced results showing that a VCCTEF system does have potential for effectively reshaping the aircraft wing during flight for significant drag reduction benefits. Experimental wind tunnel studies were also conducted to show the benefit and potential of a VCCTEF system on a flexible wing.⁸⁻¹⁰ These previous studies were primarily focused on drag minimization and were performed for older generation aircraft designs. It is of interest to assess the capabilities of a PAAW system like the VCCTEF on more modern or future aircraft designs, and

to address not only drag minimization, but also the potential for using a wing shaping device to provide load alleviation during flight maneuvers such as pull-up or coordinated turn.

The TBW represents an N+3 testbed for evaluation of the load alleviation and drag reduction capabilities of the VCCTEF system. This current study is an initial assessment of the VCCTEF capabilities on the TBW. In particular, this study involves an investigation into whether the VCCTEF can be used to shape the wing such that the spanwise lift distribution is modified in such a way as to provide load alleviation on a rigid wing aircraft. Drag minimization for a flexible wing TBW is to be conducted in future studies. An aerodynamic model of the TBW is created using a vortex-lattice method, and an automatic geometry generation tool is used to deflect the flaps of the VCCTEF system. A non-linear finite element analysis is used to determine the wing deformations and internal bending moments. Constrained gradient-based optimization is performed to determine the optimal flap deflections resulting in minimized bending moment due to a 2.5g pull-up maneuver. This study represents a preliminary analysis of the performance benefit of the VCCTEF system and its utility in adaptive wing shaping for load alleviation during flight.

II. Aircraft and VCCTEF Model

The TBW aircraft model used for this study is based on the Boeing SUGAR 765-095 aircraft, which was developed through a collaboration between the NASA Fixed Wing Project, Boeing Research and Technology, and a number of other organizations.¹¹ The aircraft has an aspect ratio of 19.56, with a wing span of 170 ft. It is designed to fly at a Mach number of 0.7, with an optimal cruise C_L of 0.766. The CAD geometry of the aircraft provided for this study had wings already deformed to a 1g loaded shape, instead of the unloaded jig shape. Therefore, any aerodynamic and structural finite element analysis of the aircraft would need to take this built-in 1g shape into account.

The TBW aircraft model was modified for this study to include the VCCTEF, as shown in Fig. 2. The number of spanwise flap sections was arbitrarily set to 10, and each flap was sized to be of approximately equal span. Between each pair of neighboring flaps is a 6" section of flexible supported material, or elastomer, joining the adjacent flaps and allowing for a continuous trailing edge with no drag producing gaps. Additionally, each flap section has two individually commanded chord-wise camber segments, as showing in Fig. 3, allowing for a spanwise distribution of variable camber.

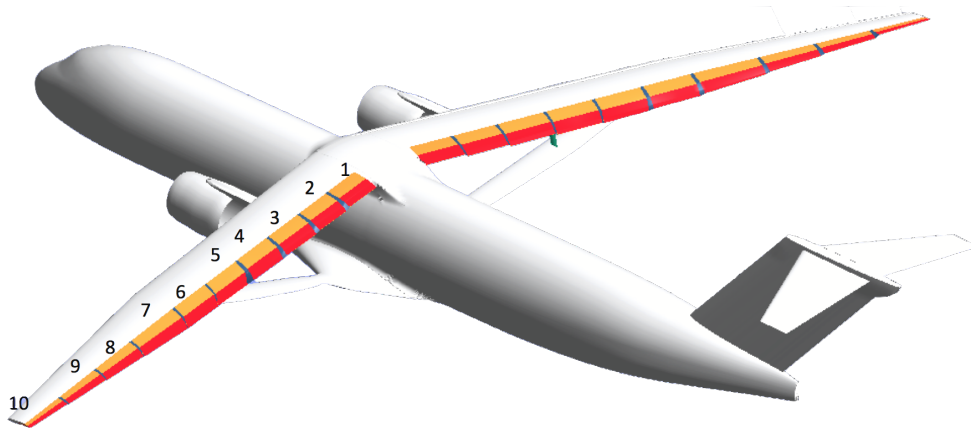


Figure 2: TBW aircraft with VCCTEF system.



Figure 3: Cross-section of variable camber flap section.

III. Modeling

III.A. Aerodynamic Model: Superposition Vortex-Lattice Approach

Aerodynamic analysis for the load alleviation optimization of the TBW with VCCTEF is performed using a vortex-lattice flow solver called Vorlax.¹² Although Vorlax is a low fidelity tool, it does provide a rapid method for estimation of aerodynamic force and moment coefficients given an input geometry. Therefore the vortex-lattice method is chosen for this study due to its computational efficiency. However, it is important to keep in mind the limitations of vortex-lattice aerodynamic modeling when assessing the results of this study. Vorlax is an inviscid, incompressible code not capable of capturing transonic effects such as shock formation. Although the flight condition for the TBW is transonic at $M = 0.7$, it was deemed that transonic effects would not play a critical role in the evaluation of load alleviation benefit. However, for future studies, particularly ones involving drag assessments, a transonic correction involving the integration of 2D transonic small-disturbance theory results will be included in the model. For conceptual design studies such as this one, particularly involving optimizations with many iterations, the trade-off in fidelity for a rapid solution is acceptable.

Vorlax models a lifting surface as a vortex sheet formed by the mean camber surface of the component. While this generally provides a reliable aerodynamic prediction for simple lifting surfaces such as a single cantilevered wing, the method has been found to be less reliable as more complex geometries are introduced, such as multiple lifting surfaces located in close proximity in the stream-wise direction like those on the TBW. However, due to the vortex-lattice method's basis in potential flow theory, the principal of superposition of aerodynamic solutions holds. A previous study showed the possibility of using various superposition combinations for the TBW aircraft.¹³ For this study, the TBW full configuration is decomposed into three components: 1) fuselage+wings+tail, 2) fuselage+truss+tail, 3) fuselage+tail. The aerodynamic solution for each configuration is obtained separately, and then the aerodynamic solution for the full configuration is obtained by adding the first two configurations and subtracting the third, as diagrammed in Fig. 4. This approach separates the wing and truss lifting surfaces, and therefore it is important to note that aerodynamic interference effects between bodies is not accounted for.

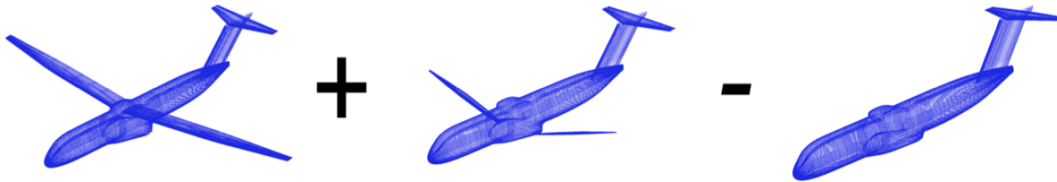


Figure 4: Aerodynamic superposition method for the TBW model.

III.B. Geometry Generation and Flap Deflection

A geometry generation tool was developed in order to automatically generate a mesh of the TBW geometry for input into Vorlax. An example of the generated aircraft mesh used for this study is shown in Fig. 5. As can be seen from the figure, engine nacelles and pylons are removed from the geometry. While the weight of the engines is accounted for in the lg shape of the wing, the aerodynamic effects and interferences of the engines and nacelles are neglected. For the purpose of this study, only the unimpeded aerodynamic effects of the VCCTEF are considered without the added complexity of engine interference. The small jury struts were also removed from the model for simplicity.

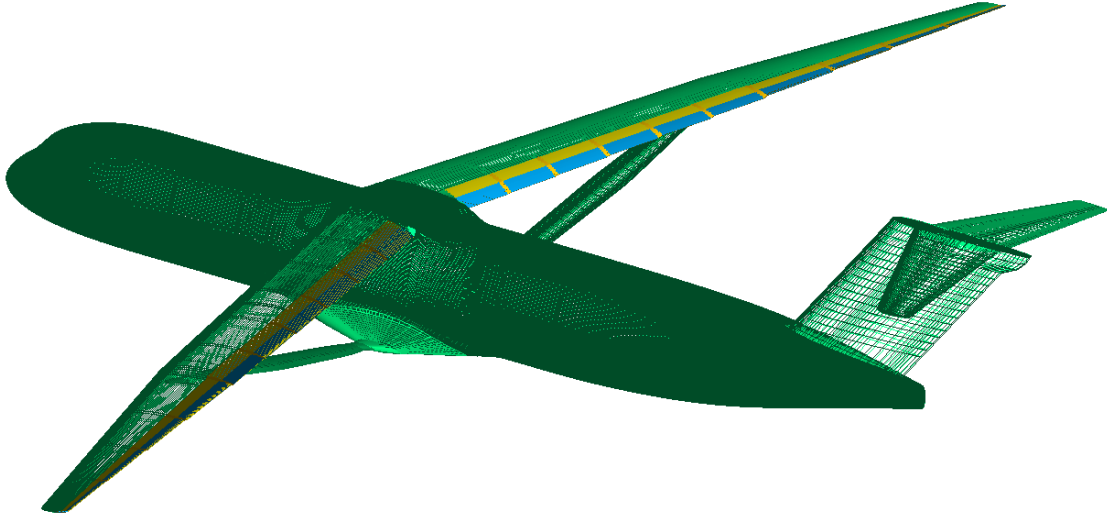


Figure 5: Mesh of TBW geometry created using geometry generation tool.

Modification to the shape of the wing due to deflections of the cambered segments in the VCCTEF is accounted for in the geometry generation tool. Flap deflections are modeled by defining the hinge line for the flap and then rotating the aft portion of the wing section about that hinge line by the provided deflection angle. The result is a new wing section with the modified camber.

III.C. Non-Linear Finite Element Analysis

The finite element analysis (FEA) for this study uses stick beam models for both the wing and the truss, defined along the elastic axes for each component. Each component beam is divided into n elements with 6 degrees of freedom at each node, and the FEA is used to numerically approximate the solution of the governing structural equations through discretization into matrix equations.^{14–16} A couple of modifications are made to the general FEA method in order to account for some unique features of the TBW model. Firstly, the elastic axis definitions for the wing and truss, as supplied by Boeing, do not connect. That is, the end node of the truss that should join to the wing does not actually lie on the wing elastic axis. Therefore, a master-slave relationship is implemented such that compatibility in deformation between the end node of the truss and the corresponding joint node on the wing is imposed. Secondly, the FEA for this study includes the effects of geometric non-linearity due to tension stiffening in the truss member. The presence of axial loading in the structure causes an increase in bending and torsional stiffness. The effect of the tensile force in the structure is included as additive terms to both the bending and torsion components of the structural stiffness matrix,

$$K_i^s = \int_0^{l_i} \begin{bmatrix} N_u'^T E A N_u' & 0 & 0 & -N_u'^T E A e_y N_v'' \\ 0 & N_\theta'^T (GJ + Tk^2) N_\theta' & 0 & 0 \\ 0 & 0 & N_w''^T E I_{yy} N_w'' + N_w'^T T N_w' & -N_w''^T E I_{yz} N_v'' \\ -N_v''^T E A e_y N_u' & 0 & -N_v''^T E I_{yz} N_w'' & N_v''^T E I_{zz} N_v'' + N_v'^T T N_v' \end{bmatrix} \quad (1)$$

where the subscript i indicates the i^{th} beam element; N_u , N_v , and N_θ are the flap-wise bending, chord-wise bending, and torsional FEA shape functions with the primes indicating the order of derivative; T is the tensile force; and k^2 is the radius of gyration of the element such that $I_{xx} = Ak^2$. Since the total tensile force T in the truss is not known prior to the solution, the problem is non-linear and is solved using an iterative method, as outlined in Fig. 6. The tension in the truss is first initialized to zero and the FEA static solution is found. From the resulting static deformation, the tensile force in the truss elements can be calculated. The structural stiffness matrix is updated with the calculated tensile force and the FEA solution is recomputed. This process is repeated until convergence is achieved. In this study, convergence is defined as being reached when the vertical deflection of the wing tip is no longer significantly changing between

iterations. A detailed discussion and analysis of the tension stiffening non-linear FEA as implemented for the TBW is provided in a previous study.¹⁷

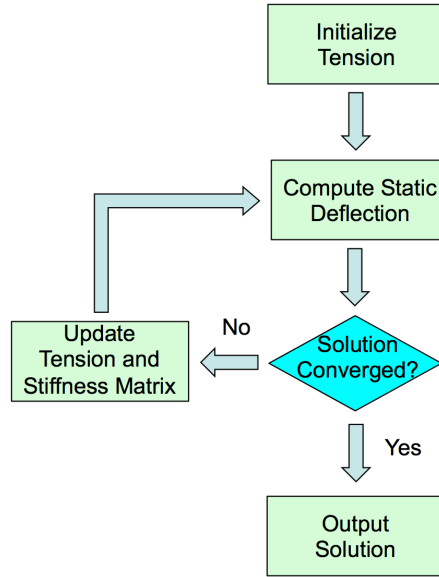


Figure 6: Flowchart outlining stiffness matrix updating process for structure with tension stiffening.

IV. Optimization

With the aerodynamic and non-linear FEA models built, it is possible to consider shape optimization of the wing for maneuver load alleviation. In this case, shape optimization is achieved through deflection of the VCCTEF system, resulting in a change of bending, twist, and camber along the wing span. For this study, the maneuver load being considered is a 2.5g pull-up maneuver. While a pull-up maneuver involves an elevator deflection to initiate the maneuver, the change in lift due to the elevator deflection is typically small compared to the total C_L and is ignored. For this study, the aircraft is analyzed at the final total C_L resulting from the 2.5g pull-up maneuver. For the TBW, C_L at 1g flight is 0.766, therefore the C_L for following analysis is $C_L = (2.5)(0.766) = 1.915$. As the lift on the wing is increased due to pull-up, the flap-wise bending moment on the wing also increases and may become the critical load on the wing. For this load alleviation optimization study, the goal is to identify a VCCTEF deflection resulting in a reduction of the wing flap-wise bending moment at the increased 2.5g loading.

IV.A. Load Alleviation: Minimization of Bending Moment

In general, for a cantilever wing the maximum flap-wise bending moment occurs at the wing root. However, the TBW has the addition of a truss member and can no longer be assumed to behave in the same manner as a single cantilever. Therefore, the first step in this analysis is to determine where on the clean wing (VCCTEF stowed) the flap-wise bending moment is critical at 2.5g. The bending moment along the wing is calculated in the local beam element axis system using the FEA solution at 2.5g loading and the FEA element shape function for bending. The beam element axis system, which lies along the wing elastic axis, is defined as shown in Fig. 7. The x-axis points towards the wing tip, the z-axis points upwards, and the y-axis is orthogonal to the x-z plane. Following the derivation by J. C. Houbolt,¹⁸ the bending moment M_y is defined as positive when it causes compression on the upper surface of the wing and tension on the lower surface.

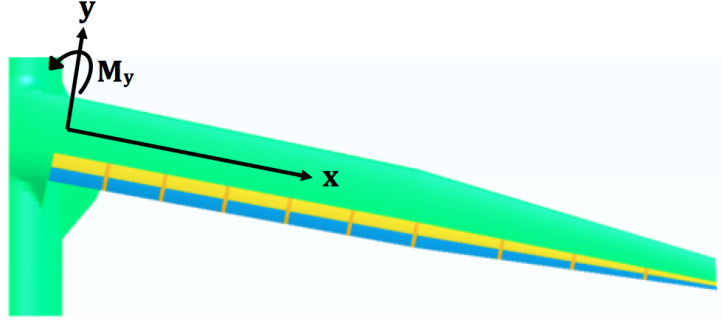


Figure 7: Local wing axis system and positive flap-wise bending moment.

The flap-wise bending deformation for each wing beam element is approximated as,

$$W_i(\eta) = \begin{bmatrix} \phi_1(\eta) & \phi_2(\eta) & \phi_3(\eta) & \phi_4(\eta) \end{bmatrix} \begin{bmatrix} w_{1_i} \\ w'_{1_i} \\ w_{2_i} \\ w'_{2_i} \end{bmatrix} = N_w(\eta)w_i \quad (2)$$

where the w_i vector contains the vertical deflection and slope at the i^{th} element nodes, and the vector $N_w(\eta)$ contains the Hermite polynomial shape functions given by,

$$N_w^T = \begin{bmatrix} \phi_1(\eta) \\ \phi_2(\eta) \\ \phi_3(\eta) \\ \phi_4(\eta) \end{bmatrix} = \begin{bmatrix} 1 - 3\eta^2 + 2\eta^3 \\ l [\eta - 2\eta^2 + \eta^3] \\ 3\eta^2 - 2\eta^3 \\ l [-\eta^2 + \eta^2] \end{bmatrix} \quad (3)$$

where $\eta \in [0, 1]$ is the local coordinate and l is the element length.

The flap-wise bending moment for a beam element is defined as,

$$M_y = EI_{yy} \frac{d^2 W}{d^2 x} = EI_{yy} \left(\frac{1}{l^2} \right) \frac{d^2 W}{d^2 \eta} = EI_{yy} \left(\frac{1}{l^2} \right) \frac{d^2 N_w(\eta)}{d^2 \eta} w_i \quad (4)$$

Therefore, the local flap-wise bending moment along the i^{th} beam element is calculated as,

$$M_{y_i} = E_i I_{yy_i} \left(\frac{1}{l_i^2} \right) \begin{bmatrix} -6 + 12\eta & l(-4 + 6\eta) & 6 - 12\eta & l(-2 + 6\eta) \end{bmatrix} \begin{bmatrix} w_{1_i} \\ w'_{1_i} \\ w_{2_i} \\ w'_{2_i} \end{bmatrix} \quad (5)$$

The flap-wise bending moment along the span of the wing for 2.5g is shown in Fig. 8. As mentioned in Section III, the wing model provided for the analysis already has 1g loaded shape built in, and it is unknown what the bending moment is for this 1g load. Therefore, the values for the bending moment at 2.5g are not absolute values, but rather incremental values obtained by using the incremental loading between 1g and 2.5g in the FEA. It is clear from the figure that the maximum M_y occurs not at the wing root, but rather at the juncture of the wing and the truss.^a This is because the truss constrains the wing at the juncture point, acting as an effective root. Thus, for the load alleviation optimization, it is the flap-wise bending moment at the wing/truss juncture that is the objective to be minimized, herein called JBM for juncture bending moment.

^aThe discontinuity in the bending moment is due to the wing elastic axis not being straight. At the juncture node the local beam axis changes direction, resulting in a slightly different local bending moment at the node for each adjoining element.

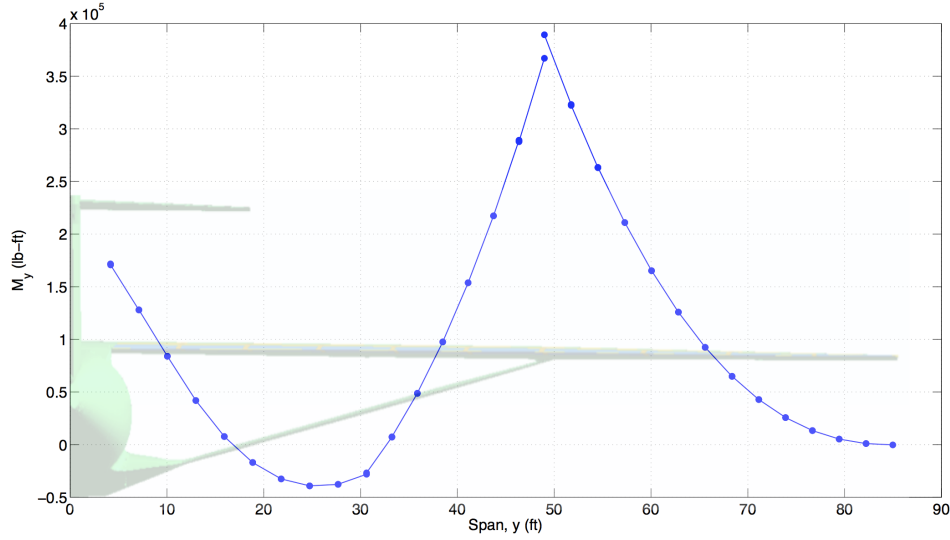


Figure 8: Flap-wise bending moment along clean wing.

IV.B. Constraints

In order to achieve a feasible result, the minimization of flap-wise bending moment at the wing/truss juncture is subject to several constraints. Firstly, the total C_L for the aircraft is fixed at the 2.5g value of $C_L = 1.915$. Also, due to the elastomer material between adjacent flaps, it is assumed that there is some relative limit on flap deflection. Therefore, a constraint of ± 2 degrees between adjacent VCCTEF sections is imposed. If no further constraints were imposed, the optimizer would drive the solution to one where the JBM is reduced with no restrictions on the shape of the lift distribution on the wing. This could result in a solution where the VCCTEF is deflected in such a way that the lift distribution is pushed to a triangular shape with the C_l at the root increasing beyond any limit since Vorlax is not capable of predicting stall. If the C_l at the root were to exceed the stall limit, then the solution would be unacceptable, even if JBM was successfully minimized. Therefore, a constraint on local section C_l value is imposed. Aerodynamic stall data for the TBW is not available, but it is assumed that a wing section will stall when the aircraft is at an angle of attack of approximately $\alpha = 12$ degrees, which is a typical value for current commercial aircraft. Following from this stall assumption, $\alpha = 10$ degrees is assumed to be a conservative limit for where non-linear aerodynamics begins. Thus the TBW spanwise C_l distribution at $\alpha = 10$ degrees is calculated using Vorlax, and the C_l value at the root location is determined to be the critical value, $C_{l,critical}$. Throughout the optimization, C_l values at spanwise locations along the wing are monitored and constrained to remain below $C_{l,critical}$. Finally, as the optimization proceeds, it is desirable that lift load does not shift from the wing to other components that are not as well designed to carry lift, such as the truss. This constraint is imposed by holding the aircraft angle of attack constant, ensuring that the total lift values being carried by any one component will not change. The constraint angle of attack, α_0 , is the TBW angle of attack for clean wing at $C_L = 1.915$ as calculated using the aerodynamic model.

IV.C. Optimization Method

With the objective function and all constraints determined, the full optimization problem is posed as follows,

$$\begin{aligned}
 &\text{Minimize: } JBM(\delta) \\
 &\text{Subject to: } C_{L,total} = C_{L,2.5g} = 1.915 \\
 &\quad \pm 2 \text{ degrees between adjacent flaps} \\
 &\quad \max(C_l(y)) \leq C_{l,critical} \\
 &\quad \alpha = \alpha_0
 \end{aligned}$$

where it is indicated that JBM is a function of the VCCTEF deflections, δ , which are the design variables for the problem. The solution to the optimization problem is acceptable if all constraints are either not violated, or active within a small buffer of ± 0.01 , which is used to ease convergence.

Optimization is carried out using a gradient-based constrained optimization method called the Method of Feasible Directions (MFD).^{19–21} MFD takes into account the gradient of the objective function as well as the gradient of each constraint to determine a feasible (no violated constraints) and useable (reduction in objective function) search direction. Gradients for the problem are approximated using forward finite difference. A flowchart outlining the optimization procedure is given in Fig. 9.

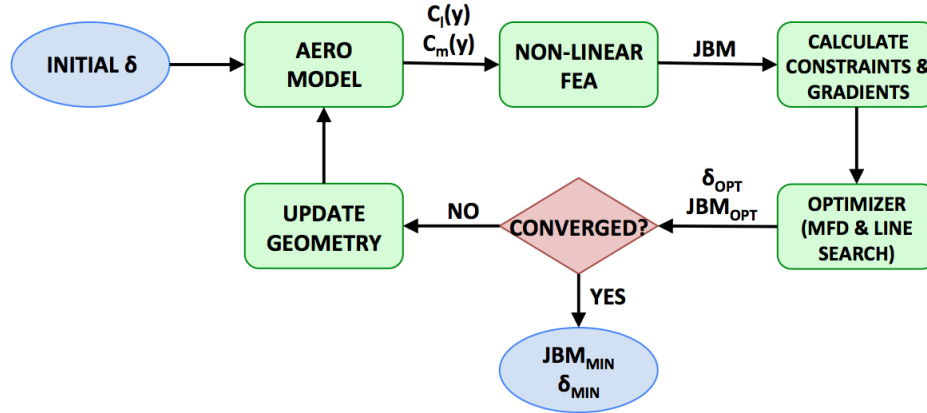


Figure 9: TBW load alleviation optimization procedure.

V. Results

V.A. VCCTEF Design Variable Parametrization

The design variables for the optimization problem are the deflection angles of the VCCTEF. As outlined in Section II, the VCCTEF system is made up of 10 individual spanwise flap sections, each with two chord-wise segments controlling the variable camber. For this study, a circular arc camber shape is assumed. Therefore, it is the deflection angle of the trailing edge chord-wise flap segment, δ_2 , that is varied to achieve load alleviation, with the other flap segment following in a circular arc such that $\delta_1 = \frac{1}{2}\delta_2$. The deflection angles of the chord-wise segments are illustrated in Fig. 10.



Figure 10: Deflection angles of VCCTEF chord-wise flaps.

One way to approach the optimization problem is to have all 10 values of δ_2 be independent design variables. However, the forward finite difference method used to approximate the gradient of the objective function requires that the aerodynamic model be run once for every design variable, which can result in significant run time and increased cost as the number of design variables is increased. Therefore, for this analysis, the deflections are parametrized using a shape function in order to reduce the number of design variables for the problem. The shape function used is a Chebyshev cubic polynomial,

$$\delta_2 = c_1 + c_2\tau + c_3(2\tau^2 - 1) + c_4(4\tau^3 - 3\tau) \quad (6)$$

where,

$$\tau = \frac{\text{Flap No.} - 1}{\text{No. of Flaps} - 1} \quad (7)$$

Therefore, there are only four design variables for the optimization problem, namely the coefficients of the parametrization function, c_1 , c_2 , c_3 , and c_4 .

V.B. Load Alleviation Results

The flap-wise bending moment along the wing beam axis before and after optimization is shown in Fig. 11, and the percent and absolute value reductions from clean wing to optimized wing are given in Table 1. In the figure the 36.7% reduction in JBM is clearly seen. In fact, the optimization results in a reduction of flap-wise bending moment along the majority of the wing span. However, there is some increase in bending moment at the wing root. This is not surprising, as it is expected that the optimization would drive the lift distribution to increase at the root in order to alleviate the loading at the joint. However, as shown in Table 1, the absolute increase in root bending moment (RBM) is approximately 1.6 times less than the absolute decrease in JBM. Also, the final RBM value is not significantly larger than the final moment value at the critical joint location. Therefore, the increase in RBM is considered acceptable in relation to the overall reduction in JBM.

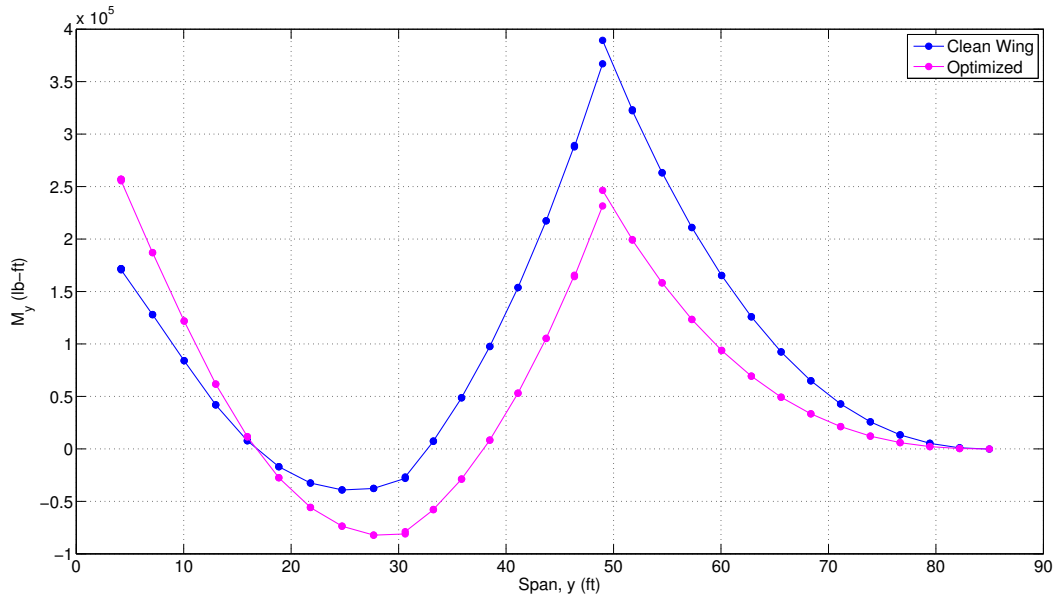


Figure 11: Flap-wise bending moment along wing span, clean wing and optimized.

Table 1: Reduction in bending moments from clean wing to optimized wing, percent difference and absolute difference.

	Percent Reduction, %	Absolute Value Reduction (kb-ft)
JBM	36.7	1.43×10^5
RBM	-49.7	-8.53×10^4

The optimized flap deflection resulting from the JBM minimization is shown in Fig. 12. The figure shows the total deflection angle of the trailing edge camber segment, with the understanding that the other camber segment follows in a circular arc. Since it is expected that the optimizer would drive the flaps to a shape that moves the wing loading inboard towards the root, the resulting deflection shape of flaps down inboard toward flaps up outboard follows what is expected. Figure 13 shows the deflected flaps on the aircraft model, with the deflections magnified 2x for visibility.

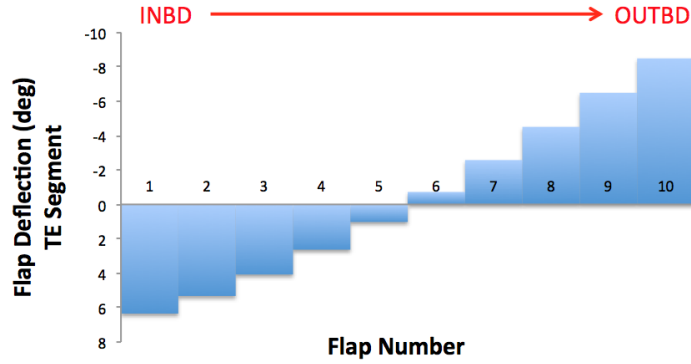


Figure 12: Optimized deflections of trailing edge flap segments.

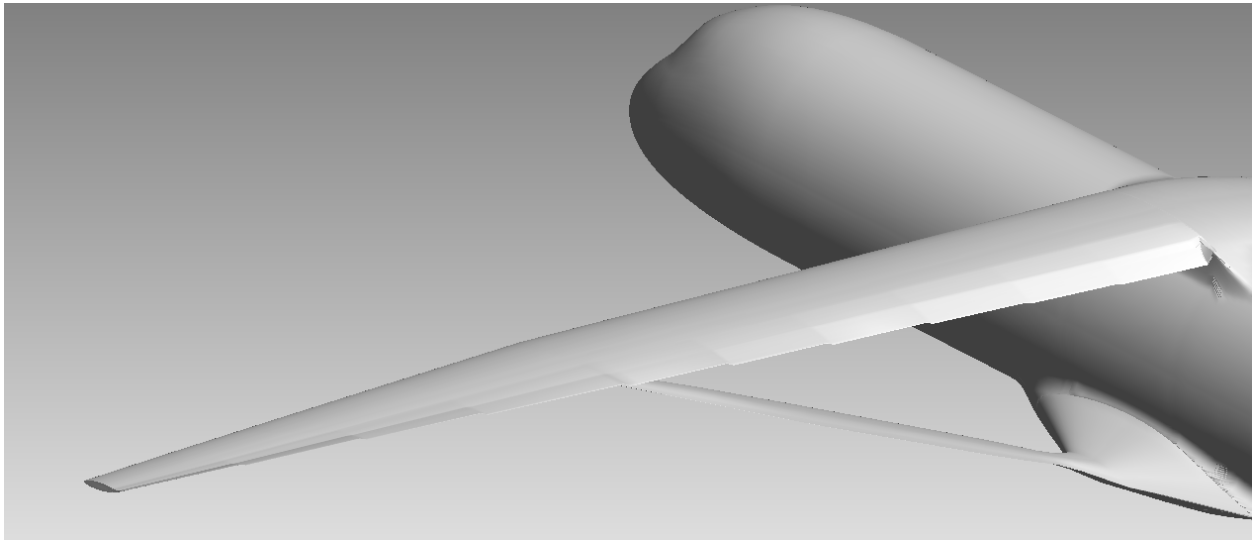


Figure 13: Optimized VCCTEF deflection, magnified two times for visibility.

Figure 14 shows the 2.5g spanwise lift distribution on the wing before and after optimization. The optimized lift distribution is of the expected shape, with lift increasing inboard of the wing/truss juncture and decreasing outboard of the juncture. Also included on the plot is the lift distribution for the wing at the $\alpha = 10$ degree limit, which is included for visualization of the effect of the $Cl_{critical}$ constraint. The optimized lift distribution does increase slightly above the constraint lift distribution, but this is an artifact of the ± 0.01 buffer that was imposed on the constraint, as mentioned in Section IV.B. The impact of the fixed α constraint is visualized in Fig. 15 and Table 2. As can be seen from the results, the truss does not take on any extra lift as a result of the optimization. Also, as desired, the total lift on the wing also remains unchanged with the optimal deflection of the flaps only resulting in a modification to the shape of the lift distribution.

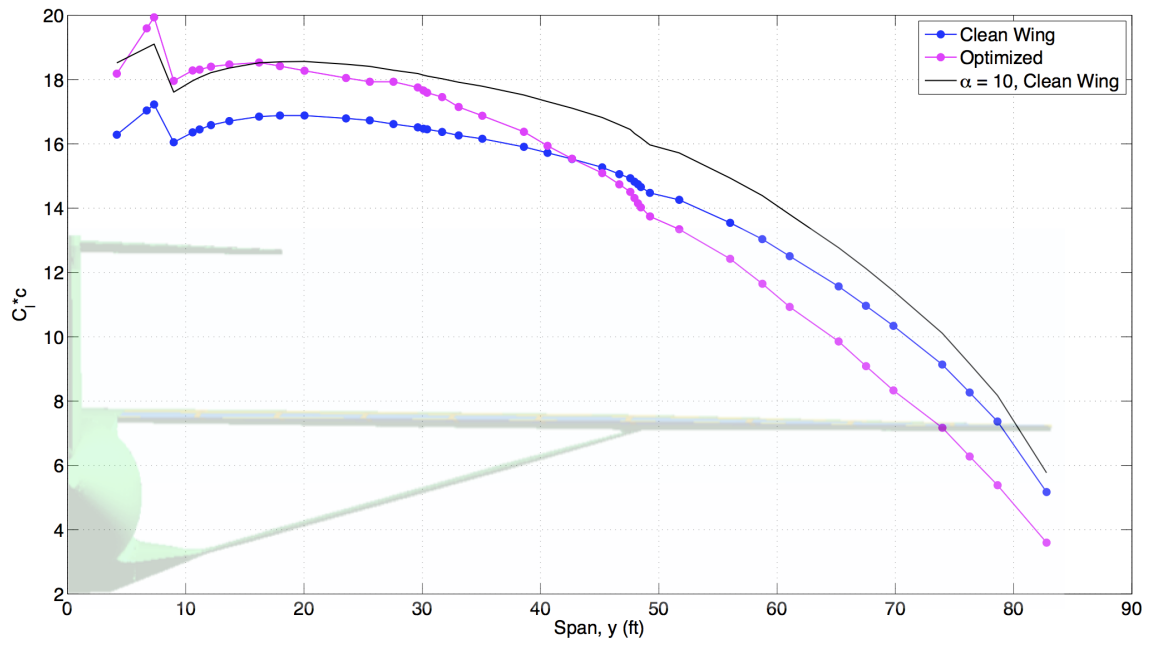


Figure 14: Spanwise lift distribution along wing, clean wing and optimized.

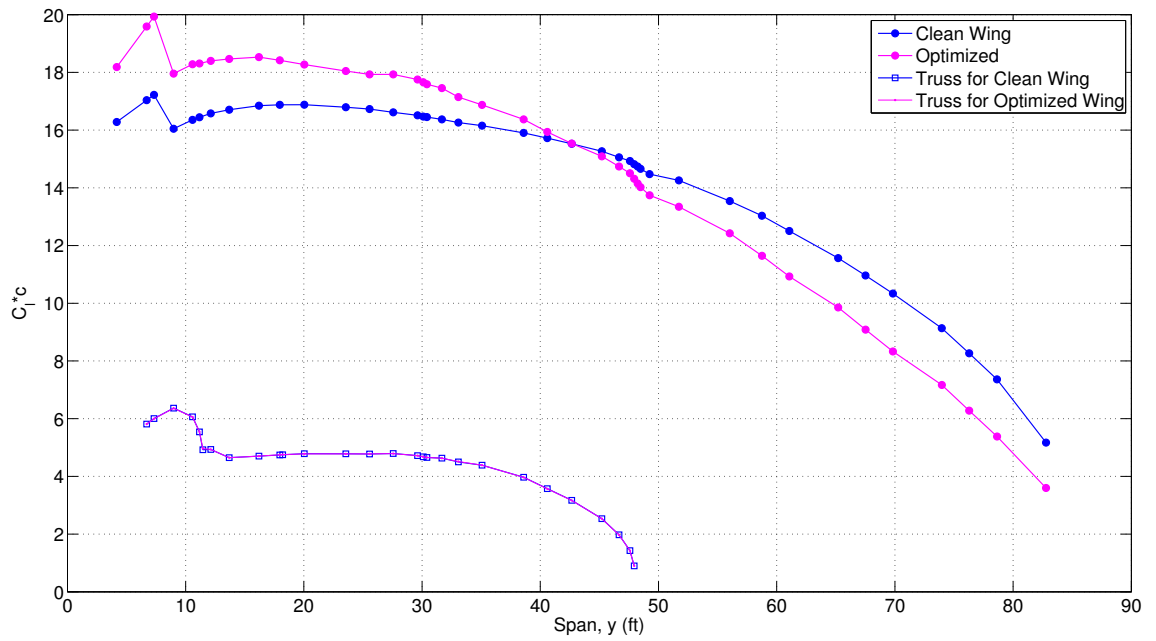


Figure 15: Spanwise lift distribution along wing, clean wing and optimized, for both wing and truss.

Table 2: Angle of attack and lift, clean wing and optimized.

	Angle of Attack (deg)	C_L on Wing	C_L on Truss
Clean Wing	8.67	1.49	0.25
Optimized	8.67	1.48	0.25

VI. Conclusion

This paper presents a study into the use of a PAAW technology, specifically a VCCTEF, for wing shaping in order to provide load alleviation during a 2.5g pull-up flight maneuver. The aircraft model used in this study is a Truss Braced Wing aircraft with rigid wings. The specific objective for measuring load alleviation is to minimize bending moment at the critical location of the wing/truss juncture point on the wing through deflection of the VCCTE flaps. The analysis involved using a vortex-lattice method to determine the aerodynamic loading on the wing and truss structures of the aircraft, and a non-linear FEA to calculate the bending moment along the wing. The non-linear component of the FEA arises from the fact that the truss member is axially loaded, resulting in tension-induced stiffening of the structure. Several constraints were imposed on the problem in order to ensure that the results were realistic. These constraints included a limit on the local lift load on the wing to ensure that any one wing section does not stall, and a constraint to fix the angle of attack to ensure that total load on the wing does not change or transfer to the weaker truss member. It was shown that within the bounds of the constraints, and the limitations of the lower fidelity aerodynamic modeling tool, the VCCTEF system could be used effectively to reduce the maximum bending moment on the wing due to the load from a 2.5g pull-up maneuver. An optimal flap deflection configuration found by the optimizer resulted in a 36.7% reduction in flap-wise bending moment at the wing/truss joint location. While this optimal flap deflection does also result in a 49.7% increase in the flap-wise bending moment at the wing root, the absolute increase at the root is 1.7 times less than the absolute decrease at the wing/truss joint. Furthermore, the root bending moment after optimization does not exceed the original maximum bending moment on the wing, and in fact is only marginally larger than the optimum value of the wing/truss joint moment, and so is not considered critical.

The ability to provide load alleviation during flight maneuver is desirable because it allows for lighter structures to be used in the construction of the aircraft, which then results in higher efficiency and lower cost. This study illustrates the potential of a PAAW system such as the VCCTEF for effectively providing flight maneuver load alleviation through active wing shaping. Future studies will involve analyzing the effectiveness of the VCCTEF on a flexible wing TBW aircraft, including both load alleviation from other flight maneuvers, as well as drag minimization at off-design flight conditions.

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