PEGA STOL WING KIT
Wing Stress Analysis

For:

DEDALIUS AVIATION inc.

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REVISION LEVEL	REVISION DATE	COMMENTS
REV 0	03/1999	Initial Analysis
REV 1	08/1999	Analysis including changes for production reasons.
REV 2	01/2001	Add a column in the recommendation table to indicate that the recommendation has been followed.

TABLE OF CONTENTS

List of Figures

List of Tables

List of Calculation

SECTION 1
INTRODUCTION

This Wing Stress Analysis report describes the procedure and all of the calculation used to calculate the stress on the PEGA-STOL wing kit. Following the calculation results, recommendations are given to make sure that the wing structure will resist within its flight envelope and its maneuver envelope.

This report is made up of the following sections:

- Section 2 lists the reference documents used in this report.
- Section 3 states the project description.
- Section 4 provides details about calculation.
- Section 5 lists the recommendations.

- Appendix A, contains photocopies of reference documents

SECTION 2
REFERENCE DOCUMENTS

The following documents were used in the structural analysis of the PEGA STOL WING KIT.

[1] Dedalius Aviation inc., PEGA-STOL Drawings, Rev.: 0, September 1998.

[2] FAR PART 23 and TP 10141F

[3] Analysis and Design of Flight Vehicle Structures, E.F. Bruhn, June 1973.

[4] Aircraft Design: A conceptual Approach, Daniel P. Raymer, AIAA Education Series, 1989.

[5] Aircraft Structures for Engineering students, Second Edition, T.H.G. Megson,1991.

[6] University of Sherbrooke, Aircraft Structures, Notes prepared by Alain Berry, 1995.

[7] Jorgensen Stell & Aluminum, Steel - Aluminum Stock List and Reference Book, 1991

[8] Fax from M.Gilles Boulanger about Rivets Shear Strength, December 1998

[9] Theory of wing section, Ira H.Abbott and Albert E.Von Doenhoff, June 1959

SECTION 3
PROJECT DESCRIPTION

GENERAL

The main objective of this project is to determine if the PEGA STOL WING KIT structure is correctly dimensioned to be able to resist to the external forces (Lift, Drag and Moment) in the worst cases.

To meet this objective a stress analysis on the wing is required. During this analysis, if a part of the structure does not meet stress requirements, a recommendation will be made to correct the defect.

CALCULATION SEQUENCE

The following figure shows the calculation sequence used in the calculation development to perform the stress analysis on the PEGA-STOL WING KIT.

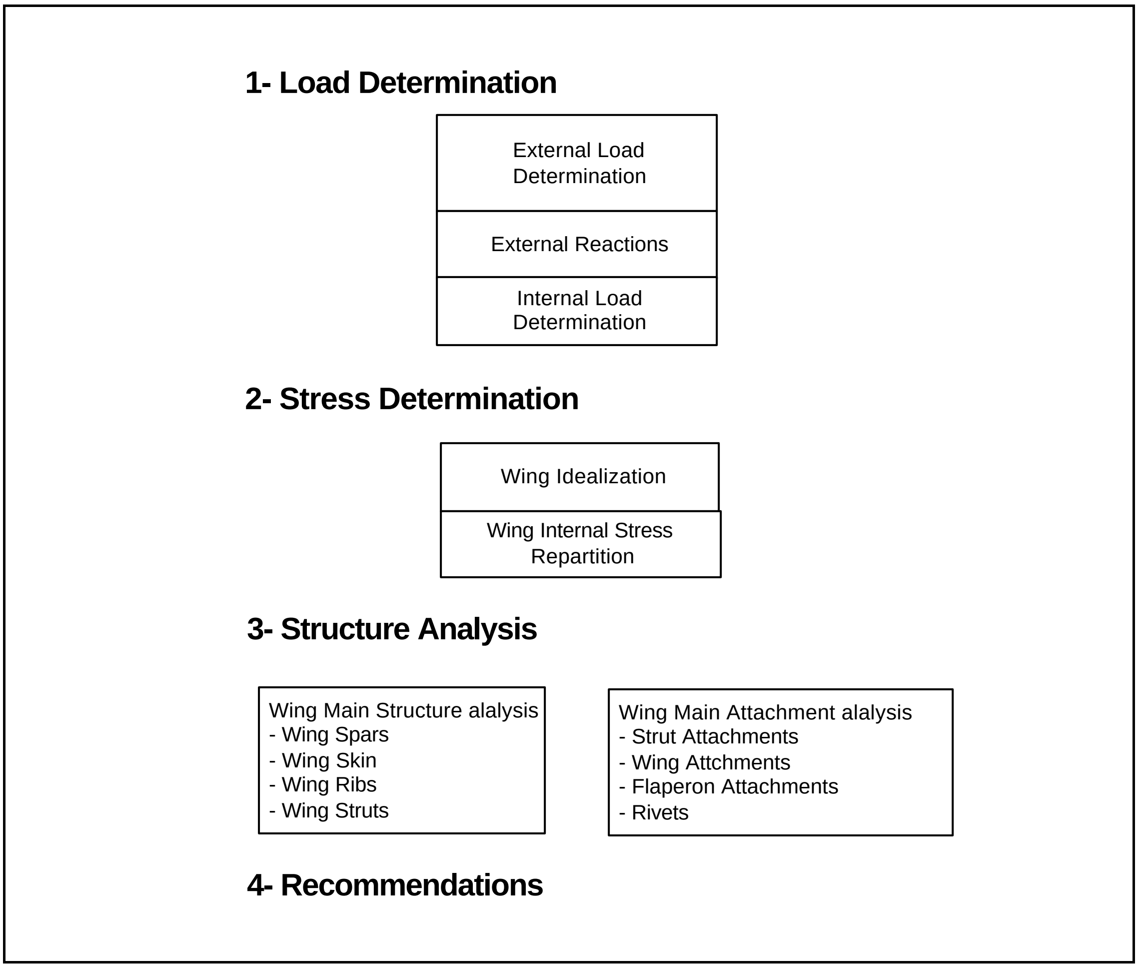

Figure 1: Calculation Sequence

SECTION 4
CALCULATION DEVELOPMENT

Aerodynamic Loads: Wing

References:

[1]
[2]
[4] p. 676 (See Appendix A)

Assumptions:

- NACA 2415 Airfoil
- Lift, drag and aerodynamic moment constant as a function of the wing span
- Maximum lift 1058 lbf
- High Stress Condition: maximum speed = 125 mph

Calculation:

Lift Coefficient:

$$C_L = \frac{L}{\frac{1}{2} r V^2 S_{ref}} = 0.244$$

where: Lift = L = 1058 lbf = 4706.22 N
Density (on ground) ρ = 1.225 kg/m^3
Wing Area = Sref = 28 ft X 3.88 ft = 108.65 ft^2 = 10.09 m^2
Airspeed = 125 mph = 55.88 m/sec

Drag Coefficient:

With the Airfoil data of the NACA 2415 (see Appendix A), the airfoil drag coefficient is determined with the lowest Reynolds Number available.

The data gives a Cd = 0.007of for a Cl = 0.244. Unfortunately during maneuver (turn, loop, rapid pull up, gust loads etc.) the number of "g" increases. At a 85° turn we have a 6g load. At an airspeed of 125mph at an 85° banking turn is possible since the CL will be 1.5. This will be the worst condition for the structure and we will analyse it. With this information the airfoil Cd is 0.024 (Cl = 1.6)

We have:

$$C_D = \frac{D}{\frac{1}{2} r V^2 S_{ref}} = \frac{C_L^2}{pAR} + C_{D_{Airfoil}} = 0.137$$

Where: CL = 1.6
Aspect Ratio AR = b/c = 28 ft/ 3.88 ft = 7.22
Cdprofil = 0.024

We can determine the total wing drag:

$$D = C_D \frac{1}{2} rV^2 S_{ref} = 2643.8 N$$

Moment Coefficient:

With the same data the coefficient moment, Cm, is determined as -0.05. The aerodynamic moment acting on the wing is calculated as follow:

$$M = C_M \frac{1}{2} rV^2 S_{ref} \overline{c} = 1141.47 N * m$$

Where: CM = 0.05 (-) means a pitching moment which drops the nose
c = 1.183 m (means aerodynamic chord)

Main Results:

Total Lift acting on 1 wing = 4706.22 N / 2 = 2353.11 N = 529 lbf
Total Drag acting on 1 wing = 2643.8 N/2 = 1321.9 N = 297.17 lbf
Total Moment acting on 1 wing = 1141.47 N*m/2 = 570.735 N * m= 420.95 lbf * ft

Aerodynamic Loads: Flapperon (Flap + Aileron)

References:

[1]
[9] p. 483-485 (See Appendix A)

Assumptions:

- Flap = NACA 2418 and Aileron = NACA 2421
- Lift, drag constant as a function of the flapperon span
- High Stress Condition: maximum speed = 125 mph
- To calculate the aerodynamic load, aileron and flap are considered like a small wing

Calculation:

Lift Coefficient maximum for the aileron and flap:

$$C_L = \frac{L}{\frac{1}{2} r V^2 S_{ref}} = 1.2$$

where: Density (on ground) ρ = 1.225 kg/m^3
Flap Area = Sref = 12 in X 72 in = 864 in^2 = 0.557 m^2
Aileron Area = Sref = 10 in X 72 in = 720 in^2 = 0.464 m^2
Airspeed = 125 mph = 55.88 m/sec

With this information we can determine the lift supplied by the aileron and the flap:

Flap Lift = 1278 N = 290 lbf and Aileron Lift = 1064 N = 240 lbf

With the Airfoil data of the airfoil drag coefficient is determined with the lowest Reynolds Number available.

The data gives a Cd (flap) = 0.019 and Cd (aileron) = 0.026

We have:

$$C_D = \frac{D}{\frac{1}{2} r V^2 S_{ref}} = \frac{C_L^2}{pAR} + C_{D_{Airfoil}}$$

Where: CL (flap) = 1.2 and CL (aileron) = 1.2
Aspect Ratio AR = b/c; Flap = 72/12 = 6 and Aileron = 72/10 = 7.2
Cdprofil = 0.019 (flap) and 0.026 (aileron)

With this information we can find the total CD acting on the aileron and the flap:

CD (flap) = 0.095 and CD (aileron) = 0.089

We can determine the total aileron/flap drag:

$$D = C_D \frac{1}{2} r V^2 S_{ref}$$

Flap Drag = 23 lbf
Aileron Drag = 18 lbf

Main Results:

Total Force supplied by the aileron = 240 lbf and Total Force supplied by the flap = 290 lbf

Wing External Reactions

References:

[2]
[3] pages A19.8 - 10

Assumptions:

- Uniform distribution of the Lift, Drag and Moment
- Struts only work in tension or compression
- The wing spar - fuselage attaches only transmit forces (no moments)
- A loading factor of +6/-3 g will be considered for the LIFT (FAR PART 23 and TP10141F).

Calculation:

The following figure shows the wing dimensions and the general structural layout required to find the external reactions:

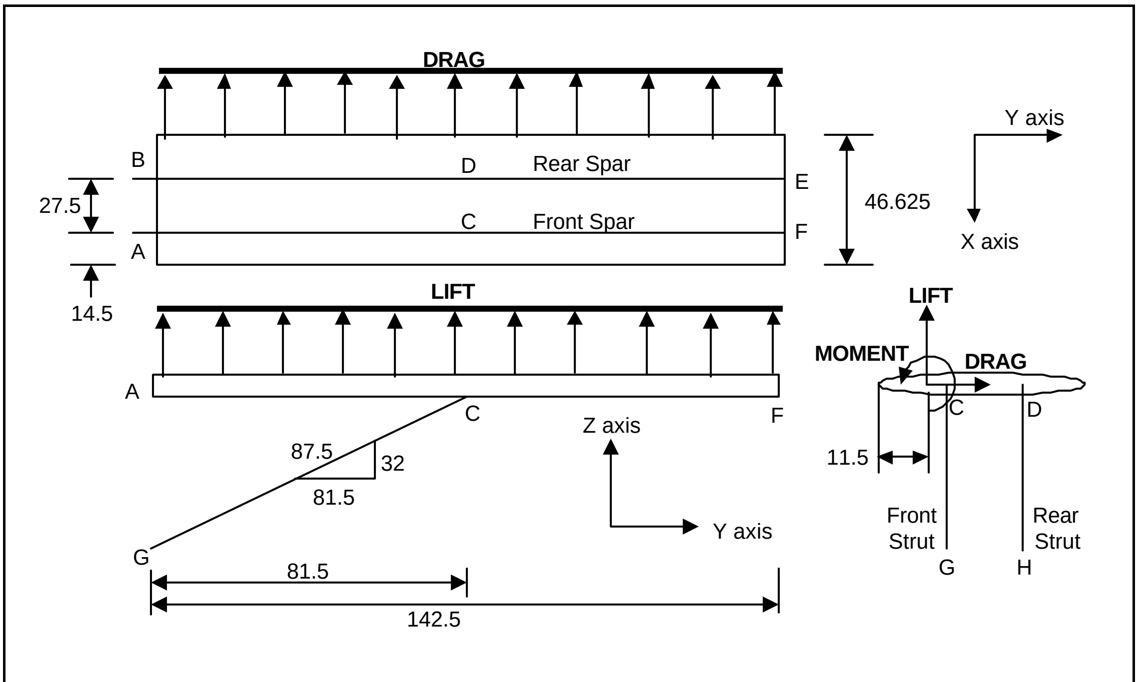

Figure 2: General Structural layout and dimensions

The external forces have been found previously and are (when the loading factor is included) :

LIFT (L):	529 lbf X 6g	= 3174 lbf
	or 3174 lbf / 142.5 in	= 22.27 lbf/in (uniform distribution)
DRAG (D):	297.17 lbf	= 297.17 lbf
	or 297.17 lbf / 142.5 in	= 2.09 lbf/in (uniform distribution)
MOMENT (M):	420.95 lbf*ft	= 420.95 lbf*ft
	or 420.95 lbf*ft / 142.5 in = 2.95 lbf*ft/in	

Lift Reactions:

Due to the LIFT the unknown external reactions (refer to figure 1) are Cz, Dz, Az, and Bz. To find the reactions first, take moments about Y axis through points A,F the LIFT axis.

$\Sigma My/AF=0$: $3(Cz + Az) + 30.5(Dz + Bz) = 0.0$ [1]

Secondly take moments about X axis through points A,B.

$\Sigma Mx/AB=0$: $81.5(Cz + Dz) + 71.25L = 0.0$ [2]

Thirdly take the summation of Z forces.

$\Sigma Fz=0$: $Az + Bz + Cz + Dz + L = 0.0$ [3]

We have 3 equations 4 unknowns ! We have to find another equation. If the LIFT force was located between the two spars Cz = Dz and Az = Bz, if the LIFT force was located directly on the point Cz = L and Dz = 0. With this arrangement, the distribution of the LIFT is as follow:

$27.5Cz = 30.5L$ => $Cz = 30.5/27.5L = 1.11L$ => 111% of the lift acts on the front spar (up)

$27.5Dz = 3.0L$ => $Dz = 3.0/27.5L = 0.11L$ => 11% of the lift acts on the rear spar (down)

Recommendation 1:
Use only one strut (front) but shift the front spar 3 inches toward the leading edge. With this configuration 100% of the lift is on the front spar instead of 111% and no rear strut would be necessary. This would save weight.

So we can rewrite the above equations using 111% of the lift (up) on the front spar and 11% of lift (down) for the rear spar.

Front spar:

$\Sigma Mx/AB=0$: $81.5Cz + 71.25(1.11L) = 0.0$ [4]
$Cz = -3080.05$ lbf (down)

$\Sigma Fz=0$: $Az + Cz + 1.11L = 0.0$ [5]
$Az = -443.09$ lbf (down)

Rear spar:

$\Sigma Mx/AB=0$: $81.5Dz - 71.25(.11L) = 0.0$ [6]
$Dz = 305.3$ lbf (up)

$\Sigma Fz=0$: $Bz + Dz - .11L = 0.0$ [7]
$Bz = 43.84$ lbf (up)

So the reactions in the z axis due to the LIFT are:

$Az = -443.09$ lbf $Bz = 43.84$ lbf $Cz = -3080.05$ lbf $Dz = 305.3$ lbf

These values follow all of the above equations. With these values the reactions in the front and rear struts can be found:

Front Strut:

(32/87.5)CG = Cz => CG = 8422 lbf in **tension** in the front strut

Rear Strut:

(32/87.5)DH = Dz => DH = 835 lbf in **compression** in the rear strut

The strut induces reactions at the root of the wing, so reactions Ay and By are as follow :

Ay = (81.5/87.5)CG => Ay = 7844.49 lbf (left)
By = (81.5/87.5)DH => By = 780.53 lbf (right)

Drag Reactions:

Due to the DRAG the unknown external reactions (refer to figure 1) are Ay, Ax, and By these reactions will be added at the Ay and By previously found (due to the Lift). To find the reactions take moments about Z axis through point A.

$\Sigma Mz/A=0$: 27.5By - (142.5/2)D = 0.0 [1]
By = 769.94 lbf (right)

And sum the force in Y axis.

$\Sigma Fy=0$: Ay + By = 0.0 [2]
Ay = -769.94 lbf (left)

And finally sum the force in X axis.

$\Sigma Fx=0$: Ax - D = 0.0 [3]
Ax = 297.17 lbs (front)

Moment Reactions:

Due to the MOMENT the unknown external reactions (refer to figure 1) are Az, Bz, Cz and Dz these reactions will be added at the Az, Bz, Cz and Dz previously found (due to the Lift).

The struts will react at the moment from C to F plus a half of the moment from C to A.

Cz = -Dz	=> (27.5/12)Cz = 61M/142.5 + 0.5(81.5)M/142.5	=> Cz = 131.16 lbf
Az = -Bz	=> (27.5/12)Az = 0.5(81.5)M/142.5	=> Az = 52.53 lbf

With these values additional reactions in the front and rear struts can be found:

Front Strut:

(32/87.5)CG = Cz => CG = 358.64 lbf in **compression** in the front strut

Rear Strut:

(32/87.5)DH = Dz => DH = 358.64 lbf in **tension** in the rear strut

Like we know, the strut induces reactions at the root of the wing, so we will get reactions Ay and By.

Ay = (81.5/87.5)CG => Ay = 385.04 lbf (right)
By = (81.5/87.5)DH => By = 385.04 lbf (left)

Main Results:

Front Strut Total Tension = 8063.36 lbf
Rear Strut Total Compression = 477 lbf

Wing Attachment Point A:
Ax = 297 lbf (Drag action)
Ay = 8229.39 lbf (Lift action (7844.49) + Drag Action (769.94) - Moment action (385.04))
Az = -443.09 lbf (Lift Action)

Wing Attachment Point B:
By = 1165.43 lbf (Lift action (780.53) + Drag Action (769.94) - Moment action (385.04))
Bz = 43.84 (Lift Action)

Wing Internal Reactions

References:

[6]

Assumptions:

- Uniform distribution of the Lift, Drag and Moment
- Struts only work in tension or compression
- The wing spar - fuselage attaches only transmit forces (no moments)

Calculation:

The following figure shows the distribution of the shears and moments throughout the wing. The following curves are determined with the following equations:

$$V(x) = \int F(x)dx \quad M(x) = \int V(x)dx$$

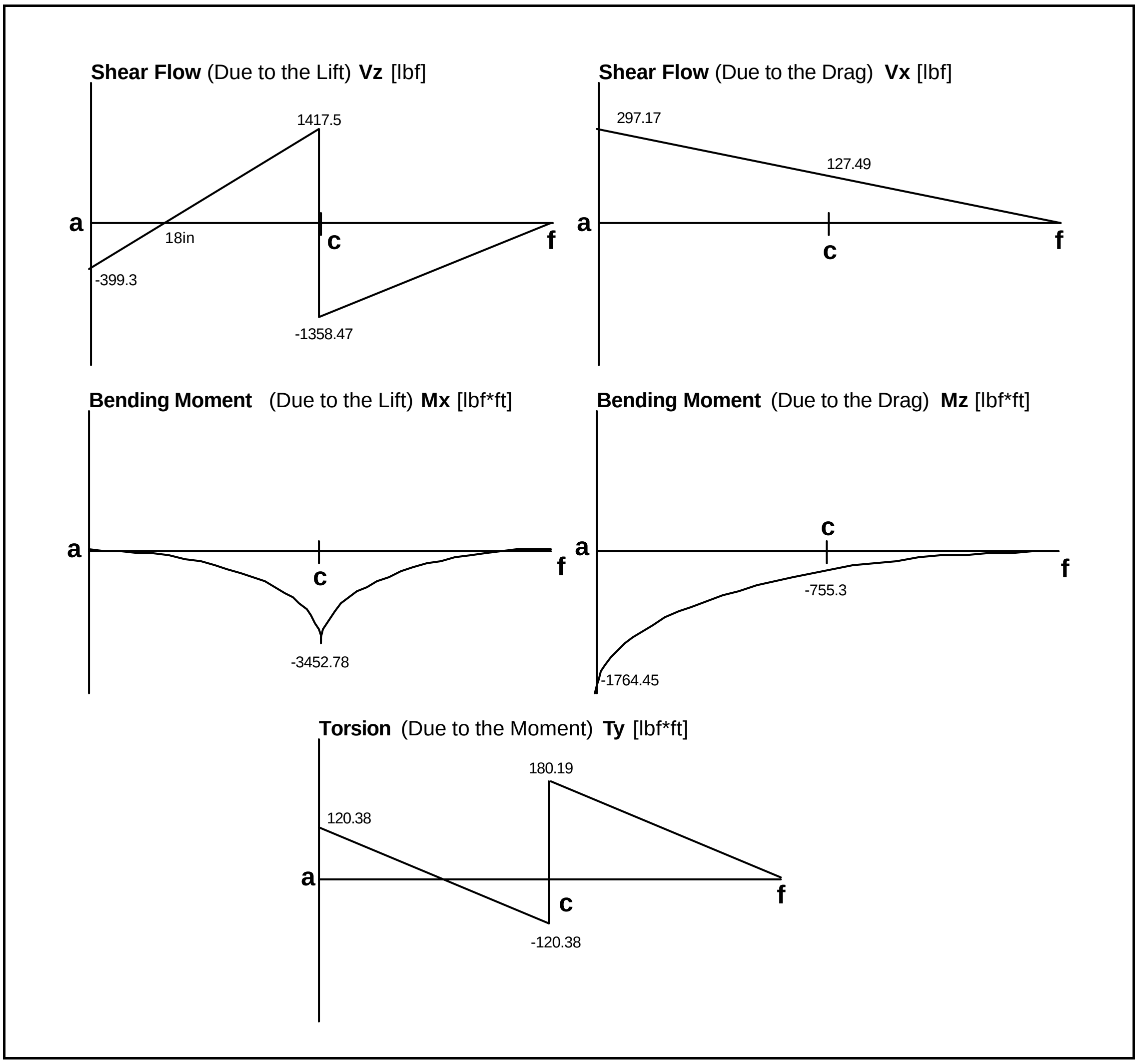

Figure 3: Shear Force, Bending Moment & Torsion Diagrams

Main Results:

With these diagrams we can determine the internal reactions acting on the wing as a function of the wing span. Point C corresponds to the strut attachment. The wing structure at this point must resist to the higher stress condition. This point is critical and a detailed stress analysis will be done.

The internal forces at the point C (strut attachment) are:

- Shear forces: Vz = 1417.5 lbf Vx = 127.49 lbf
- Bending Moments: Mx = 3452.78 lbf*ft Mz = 755.3 lbf
- Torsion Moments: Ty = 180.19 lbf*ft

Wing Idealization

References:

[5] Section 8.8
[6] Section 5.5

Assumptions:

- The spar flanges carry direct stresses
- The skin carries shear stresses
- Most of the stresses are carried by the part of the wing between the front and the rear spar
- The airfoil is symmetrical

Calculation:

The following figure shows the wing idealization. The spar flanges are merged into mass points S1 and S2. These points will carry the direct stresses and the skins the shear stresses.

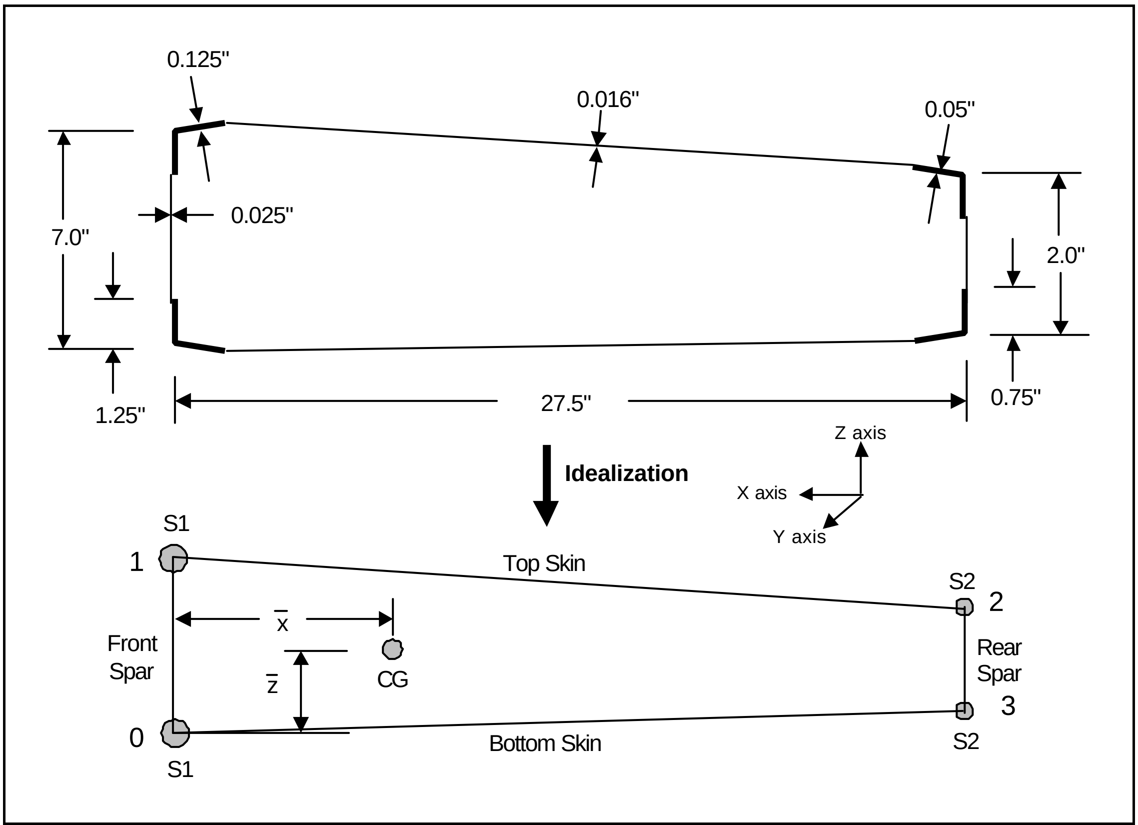

Figure 4: Wing Idealization

Section Properties:

The areas of the points S1 and S2 are:

S1 = 0.125” X 1.25” X 2 = 0.3125 in^2 and S2 = 0.05” X 0.75” X 2 = 0.075 in^2

The position of the center of gravity is:

$$\bar{x} = \frac{\sum A_i x_i}{\sum A_i} = \frac{2 \times 0.3125 \times 0 + 2 \times 0.075 \times 27.5}{2 \times 0.3125 + 2 \times 0.075} = 5.32`` \quad \bar{z} = \frac{\sum A_i x_i}{\sum A_i} = 3.5`` \text{ (By Symmetry)}$$

The moments of inertia are:

$$I_{xx} = \sum A_i z_i^2 = 2 \times 0.3125 \times 3.5^2 + 2 \times 0.075 \times 1.0^2 = 7.80625 in^4$$
$$I_{zz} = \sum A_i x_i^2 = 2 \times 0.3125 \times 5.32^2 + 2 \times 0.075 \times 22.18^2 = 91.4819 in^4$$

To be able to determine the stress acting on the spars and the skins, we must find the internal forces acting about the center of gravity CG point. To calculate this we take the internal reactions previously calculated about the ¼ of chord, and transpose them about the center of gravity, the CG point.

Forces about ¼ chord:

Vz = 1417.5 lbf Vx = 127.49 lbf
Mx = 3452.78 lbf*ft Mz = 755.3 lbf
Ty = 180.19 lbf*ft

Transposition about the CG point:

Vz, Vx, Mx and Mz will stay the same. But the torsion Ty will change since the Vz is not applied on the CG but on the ¼ chord.

Ty = 1417.5 lbf X (5.32 + 3.0)”/12 ft - 180.19 lbf*ft = 802.61 lbf*ft

Main Results:

- Section Properties:
  - S1 = 0.3125 in^2 and S2 = 0.075 in^2
  - CG location: x = 5.32 in and z = 3.5 in
  - Inertia: I_{xz} = 0.0, I_{xx} = 7.8062 in^4 and I_{zz} = 91.4819 in^4
- Forces apply about CG:
  - Vz = 1417.5 lbf Vx = 127.49 lbf
  - Mx = 3452.78 lbf*ft Mz = 755.3 lbf
  - Ty = 802.61 lbf*ft

Wing Section Stress Distribution (Skins + Spar Flanges)

References:

[5] Section 8.8
[6] Section 5.5

Assumptions:

- Spar flanges carry all of the direct stresses (supplied by bending moment Mx and My)
- Skins carry all of the shear stresses (supplied by Ty. Vz)
- Vx is neglected for shear stress loading

Calculation:

Direct Stresses σy (in y direction):

The direct stress acting on the spar flanges, due to the bending moment Mx and Mz, is found using the following equation:

$$s_y = -\frac{M_x(I_{zz}z - I_{xz}x)}{I_{xx}I_{zz} - I_{xz}^2} - \frac{M_z(I_{xx}x - I_{xz}z)}{I_{xx}I_{zz} - I_{xz}^2}$$

And by applying this equation to the PEGA-STOL wing we obtain:

$$s_y = -\frac{M_x \times Z}{I_{xx}} - \frac{M_z \times X}{I_{zz}} = -\frac{3452.78 \times Z}{7.8062} - \frac{-755.3 \times X}{91.4819}$$

For S1 at X = 3.5in and Z = 3.5in; compression stress

$$s_y = -\frac{3452.78 \times 12 \times 3.5}{7.8062} - \frac{-755.3 \times 12 \times 3.5}{91.4819} = -18230.36\,psi$$

For S1 at X = 3.5in and Z = -3.5in; tension stress

$$s_y = -\frac{3452.78 \times 12 \times -3.5}{7.8062} - \frac{-755.3 \times 12 \times -3.5}{91.4819} = 18923.89\,psi$$

For S2 at X = -24in and Z = 1.0in; compression stress

$$s_y = -\frac{3452.78 \times 12 \times 1.0}{7.8062} - \frac{-755.3 \times 12 \times -24}{91.4819} = -7685.56\,psi$$

For S2 at X = -24in and Z = -1.0in; tension stress

$$s_y = -\frac{3452.78 \times 12 \times -1.0}{7.8062} - \frac{-755.3 \times 12 \times -24}{91.4819} = 2929.94\,psi$$

<u>Shear Stresses τy (in skin plan direction):</u>

The shear stress acting on the skins, due to the shear forces Vx, Vz and the torsion moment Ty is found with the following equations:

Contribution of the shear forces Vx and Vy:

$$q_s = -\left(\frac{V_x I_{xx} - V_z I_{xz}}{I_{xx} I_{zz} - I_{xz}^2}\right)\left(\int_0^s t_D x ds + \sum_{r=1}^{n} S_r x_r\right) - \left(\frac{V_z I_{zz} - V_x I_{xz}}{I_{xx} I_{zz} - I_{xz}^2}\right)\left(\int_0^s t_D z ds + \sum_{r=1}^{n} S_r z_r\right) + q_0$$

by applying this equation to the PEGA-STOL wing we obtain:

Vx contribution section:

$$q_s = \frac{-V_z}{I_{xx}} \sum_{r=1}^{r=n} S_r z_r + q_0$$

$$q_{01} = 0$$

$$q_{12} = q_{01} - \frac{0.3125 \times 1417.5 \times 3.5}{7.8062} = -198.61 psi \times in$$

$$q_{30} = q_{12} = -198.61 psi \times in \text{ (by symmetry)}$$

$$q_{23} = q_{12} - \frac{0.075 \times 1417.5 \times 1.0}{7.8062} = -212.23 psi \times in$$

Taking the moments about the center of gravity where the Vz is applied:

$$\sum M_{y/CG} = 0$$

$$q_{12} \times 27.5 \times 3.5 \times 2 + q_{23} \times 2 \times 27.5 + (27.5 \times (7+2)/2) \times q_0 = 0$$

$$q_0 = 403.27 psi \times in$$

With this we can find the total shear stress due to Vx:

q01 = 403.27 psi*in (front spar)
q12 = 204.66 psi*in (upper skin)
q23 = 191.04 psi*in (rear spar)
q30 = 204.66 psi*in (lower skin)

Contribution of the torsion moment Ty:

$$q_s = \frac{T_y}{2A_{wing\_section}} = \frac{802.61 \times 12}{2 \times 123.75} = 38.91 psi * in$$

Main Results:

The internal stress acting on the high stressed wing section is:

Front spar upper cap:	18230 psi (compression)
Front spar lower cap:	18924 psi (tension)
Front spar web:	(403.27 psi*in + 38.91psi*in) / 0.025in = 17687 psi (shear)
Rear spar upper cap:	7686 psi (compression)
Rear spar lower cap:	2930 psi (tension)
Rear spar web:	(191.04 psi*in + 38.91psi*in) / 0.025in = 9198 psi (shear)
Upper skin:	(204.66 psi*in + 38.91psi*in) / 0.016in = 15223 psi (shear)
Lower skin:	(204.66 psi*in + 38.91psi*in) / 0.016in = 15223 psi (shear)

Material Mechanical Properties and Factor of Safety

References:

[2]
[7] and [8]

Calculation:

The minimal factor of safety used is 1.5 and it is specified by the FAR PART 23 and TP10141F. So to validate a piece of the wing structure, the following equations will be applied:

$$s_y \geq s_{calc} \times FS \text{ and } t_y \geq t_{calc} \times FS$$

Where:
- σ_y = Direct Stress Material Yield Strength Limit [psi]
- τ_y = Shear Stress Material Yield Strength Limit [psi] = σ_y / 2 (Tresca)
- σ_{calc} = Direct Stress Calculated [psi]
- τ_{calc} = Shear Stress Calculated [psi]
- FS = Minimal Factor of Safety (1.5)

The value of Yield Strength is function of the material used for the construction, for this wing, 6061-T6 Aluminum Alloy is used. The following figure shows the mechanical properties of the materials used :

Table 1: Material Mechanical Properties

Aluminum	**Yield Strength [psi]**	**Young Modulus [X 10^6 psi]**
2024-T0 clad	11 000	
2024-T3 clad	45 000	10.7
6061-T0 clad	8 000	
6061-T6 clad	37 000	10.1
Rivets	**Shear Strength [lbs]**	
Avdell (Avex) 5056 Al 1/8 dia.	210	
Avdell (Avex) 5056 Al 5/32 dia.	305	
SS - 4130 1/8 dia.	615	
SS - 4130 5/32 dia.	976	
Bolts	**Shear Strength [lbs]**	
Steel Bolt AN 3/16 (0.190)	2126	
Steel Bolt AN ¼	3680	
Steel Bolt AN 5/16	5750	
Steel Bolt AN 3/8	8280	
Steel Bolt AN 7/16	11250	

Structure Analysis: Front Spar Cap (A-202)

References:

[1] DWG No A-20

Assumptions:

- Spar cap carries all of the direct stresses (supply by bending moment Mx and My)

Calculation:

The stress acting on the upper and lower spar caps are:

Front spar upper cap:	18230 psi (compression)
Front spar lower cap:	18924 psi (tension)

The area of the cross-section is:
0.125'' X (1.25'' + 1.25''- 0.125'') = A_t = 0.296875in^2

The minimum area of the cross-section is the previous area minus the rivet holes:
0.125'' X (1.25''-5/32'' + 1.25'' -5/32'' - 0.125'') = A_e = 0.2578125 in^2

The direct stress acting on the cap is:

$$s_e = s_t \times \frac{A_t}{A_e}$$

For the upper cap:

$$s_e = 18230\text{psi} \times \frac{0.296875}{0.2578125} = 20992\,psi$$

For the lower cap:

$$s_e = 18924\text{psi} \times \frac{0.296875}{0.2578125} = 21791\,psi$$

The factor of safety, using Al-6061-T6 are:

For the upper cap:

$$FS = \frac{s_y}{s_{calc}} = \frac{37000\,psi}{20992\,psi} = 1.76 \succ 1.5$$

For the lower cap:

$$FS = \frac{s_y}{s_{calc}} = \frac{37000\,psi}{21791\,psi} = 1.69 \succ 1.5$$

The factor of safety, using Al-2024-T3 are:

For the upper cap:

$$FS = \frac{s_y}{s_{calc}} = \frac{45000psi}{20992psi} = 2.14 \succ 1.5$$

For the lower cap:

$$FS = \frac{s_y}{s_{calc}} = \frac{45000psi}{21791psi} = 2.06 \succ 1.5$$

Recommendation 2:
Even if using Al-6061-T6 the FS > 1.5, it will be better for the spar caps to use Al-2024-T3 and increase the FS.

For the rivets on the spar caps:

The maximum total force acting on the spar cap is:

$$F = s_{calc} \times A_e = 21791psi \times 0.2578125in^2 = 5617lbs$$

Rivet type: A-5 pitch 1.5'' for a total of 95 rivets

95 rivets X 305 lbs/rivet = 28975 lbs   FS = 5.2

Recommendation 3:
Rivet type: A-5 pitch 2'' for a total of 71 rivets -> FS = 3.8

Main Results:

Spar cap FS = 2 (including recommendation)
Rivet FS = 3.8 (including recommendation)

Structure Analysis: Upper & Lower Skins (A-801… A-806)

References:

[1] DWG No A-80
[3] C5.7 Buckling of flat rectangular plates under shear loads

Assumptions:

- Skins carry all of the shear stresses

Calculation:

The shear stresses acting on the skins have previously been determined and are:

Upper skin = 15223 psi (shear)
Lower skin = 15223 psi (shear)

Buckling Verification:

The critical elastic shear buckling stress for flat plates with various boundary conditions is given by the following equation:

$$t_{cr} = \frac{p^2 k_s E}{12(1-n_e^2)_e}\left(\frac{t}{b}\right)^2$$

Where: k_s = buckling coefficient which depends on edge boundary conditions and sheet aspect ratio (a/b)
E = modulus of elasticity
V_e = elastic Poisson's ratio
b = short dimension of plate or loaded edge
t = sheet metal thickness

If we apply the previous equation to our wing we get:

k_s: from fig. C5.11 (see appendix A) with a/b = 27.5''/12''=2.3 -> k_s = 6.2
E: 10.1 X 106 psi (6061-T6)
Ve: 0.3 (Al)
t: 0.016''

With these we obtain:

$$t_{cr} = \frac{p^2 \times 6.2 \times 10.1 \times 10^6\, psi}{12(1-0.3_e^2)_e}\left(\frac{0.016''}{12''}\right)^2 = 100.62\, psi$$

It is practically impossible to avoid buckling at this wing section location. Even if we add stringers or change the thickness of the plate. **Buckling will appear at this section of the wing.**

Factor of safety:

$$FS = \frac{t_y}{t_{calc}} = \frac{18500 psi}{15223 psi} = 1.2 \prec 1.5$$

The factor of safety is not acceptable. Instead of using 0.016'' 6061-T6, we will try 0.020" 6061-T6. With this the stress in the panel becomes:

15223psi X (0.016/0.020) = 12178.4psi and

$$FS = \frac{t_y}{t_{calc}} = \frac{18500 psi}{12178.4 psi} = 1.52 \succ 1.5$$

Recommendation 4:
For upper and lower skins, from STN 60 3/8 to STN 105 3/8 put a skin of 6061-T6 0.020''. So we will have five upper skins and three lower skins.

Skin Rivets:

Rivet: A-4 pitch 40mm (1.6")

The rivets around a section of skin (between the two spars and two ribs) have to support the following total force:

Total force:	12178.4 psi X ((2 X 27.5'' + 2 X 12'') X 0.020'') = 19242 lbs
Total force of rivets:	(2 X 27.5'' + 2 X 12'')/1.6 = 50 X 210 lbs/rivet = 10368.75 lbs

FS = 10368/19242 = 0.53 (not enough)

If we use A-5 rivet type pitch 25 near of the strut attachment we will obtain:

Total force of rivets: (2 X 27.5'' + 2 X 12'')/1.0 = 79 X 305 lbs/rivet = 24095 lbs

FS = 24095/19242 = 1.3 (acceptable)

Recommendation 5:
Between STN 70 7/8 and STN 93 3/8 use A-5 pitch 25 rivets on the ribs and on the spar junctions.

Main Results:

Buckling in the skin near of the strut.
Skin FS: 1.52 (including recommendation).
Rivet FS: 1.3 (acceptable and including recommendation).

Structure Analysis: Wing Mid Rib (W-12.3)

References:

[1] DWG No A-60

Assumptions:

- Wing ribs carry only shear stress

Calculation:

The maximal locally (just after the front spar and at the strut STN) shear stress acting on the mid-rib is 442.18 psi*in/0.016in = 27636 psi

$$FS = \frac{t_y}{t_{calc}} = \frac{18500 psi}{27636 psi} = 0.67$$

This FS is not enough so we will keep the same mat'l but 0.040in thick:
442.18 psi*in/0.040in = 11054.5 psi

$$FS = \frac{t_y}{t_{calc}} = \frac{18500 psi}{11054.5 psi} = 1.67$$

Now we will change the mat'l (2024-T3) and use 0.032 thick. So the shear stress will become 442.18 psi*in/0.032in = 13818 psi

$$FS = \frac{t_y}{t_{calc}} = \frac{22500 psi}{13818 psi} = 1.63$$

Recommendation 6:
For ribs located at STN 70 7/8, 81 3/8 and 93 3/8, use 2024-T3 0.032'' thick.

Main Results:

Ribs FS = 1.67 (including recommendation)

Structure Analysis: Wing Attachments (A-206, A-203, A-404)

References:

[1] DWG No A-40 and A-20
Table 1 of this document

Calculation:

The load acting on the attachments are:

Wing Attachment Point A:
Ax = 297 lbf (Drag action)
Ay = 8229.39 lbf (Lift action (7844.49) + Drag Action (769.94) - Moment action (385.04))
Az = -443.09 lbf (Lift Action)

Wing Attachment Point B:
By = 1165.43 lbf (Lift action (780.53) + Drag Action (769.94) - Moment action (385.04))
Bz = 43.84 (Lift Action)

The shear force is determined by finding the magnitude of the above force vectors:

$$F_{TA} = \sqrt{A_y^2 + A_z^2} = \sqrt{8229.4^2 + 443.1^2} = 8241.\ lbf$$

$$F_{TB} = \sqrt{B_y^2 + B_z^2} = \sqrt{1165.4^2 + 43.8^2} = 1166\ lbf$$

The Factor of safety for the AN bolts can be found AN 5/16 with a an ultimate shear strength (Us) of 5750 lbf are presently used:

At point A:

$$FS = \frac{U_S}{F_{TA}} = \frac{5750}{8241} = 0.69 \prec 1.5$$

Since the factor of safety is less than 1.5 we must take a larger diameter for AN bolt. If we use AN 1/2 we will have a FS of:

$$FS = \frac{U_S}{F_{TA}} = \frac{14700}{8241} = 1.78 \succ 1.5$$

Recommendation 7:
Use AN ½ for the front spar wing attachment bolt, modify holes of pieces A206 and A-203 according to that.

At point B:

$$FS = \frac{U_S}{F_{TA}} = \frac{5750}{1166} = 4.93 \succ 1.5$$

Since the factor of safety is greater than 1.5 but we can optimize the attachment bolt by using AN ¼ . With this bolt we will have the following factor of safety:

$$FS = \frac{U_S}{F_{TA}} = \frac{3680}{1166} = 13.2 \succ 1.5$$

Recommendation 8:
Use AN 1/4 for the rear spar attachment bolt, modify holes of piece A-404 according to that.

Now we will check if the pieces A-206 and A-203 will resist to the load of the front spar wing attachment. The stress transmitted in these parts is determined by taking the Fta dividing it by the effective area:

Effective area = (1.5 - 0.5) X (0.063 + 0.063) = 0.126 in^2

$$s = \frac{F_{TA}}{A_e} = \frac{8241}{0.126} = 65404 \text{ psi}$$

The factor of safety is:

$$FS = \frac{s_y}{s_{calc}} = \frac{37000 psi}{65404 psi} = 0.56 \prec 1.5$$

Since the FS is less than 1.5 we must modify the piece A-206. If we put the height = 2.5 in and the material 2024-T3. We will get:

Effective area = (2.5 - 0.5) X (0.063 + 0.063) = 0.252 in^2

$$s = \frac{F_{TA}}{A_e} = \frac{8241}{0.252} = 32702.4 \text{ psi}$$

The factor of safety is:

$$FS = \frac{s_y}{s_{calc}} = \frac{45000 psi}{32702 psi} = 1.4 (acceptable)$$

Recommendation 9:
For pieces A-206, Fitting doubler, change material for 2024-T3 and modify height quote (1.5) by 2.5. Use material 2024-T3 for A-203: Front spar root fitting.

Recommendation 10:
For pieces A-206, A-203 (Fitting doubler + Front spar root fitting) drill when assembled, and drill to fit the AN 1/2 dia.

Now we will verify if the number of rivets used to hold this fitting are sufficient. The rivets used are SS - 4130 5/32 dia. 976 lbs. The rivets must maintain 8241 lbs. So the FS is:

$$FS = \frac{U_s \times nb \text{ of rivet}}{F_{AT}} = \frac{976 \times 6}{8241} = 0.71 \prec 1.5$$

Since the FS is less than 1.5 we have to increase the number of SS rivet. If we use three rows of 4 rivets we will get:

$$FS = \frac{U_s \times nb \text{ of rivet}}{F_{AT}} = \frac{976 \times 12}{8241} = 1.42 \approx 1.5$$

Recommendation 11:
A-206: Instead of using 2 rows of 3 rivets, use 3 rows of 4 rivets SS-5.

Structure Analysis: Wing Struts + Attachments (A-2030, A-2010, A-2020)

References:

[1] DWG No A-200
Table 1 of this document

Calculation:

The load acting on the struts are:

Front Strut Total Tension = 8063.36 lbf
Rear Strut Total Compression = 477 lbf

We will study only the front strut since the rear strut is almost unloaded.

The effective area of the strut is approximately:

A_e = 2 X 1/16 + 7 X 1/8 = 1 in2

$$s = \frac{F_S}{A_e} = \frac{8063}{1} = 8063 \text{ psi}$$

The factor of safety is:

$$FS = \frac{s_y}{s_{calc}} = \frac{37000\,psi}{8063\,psi} = 4.5 \succ 1.5$$

From DWG A-200 four AN4 are used to attach fitting to strut member. Since bolts are the same size, it will be assume that each of the 4 bolts transfers ¼ of the total fitting load (2015.75 lbf). The following figure shows the load in the fitting and the strut member.

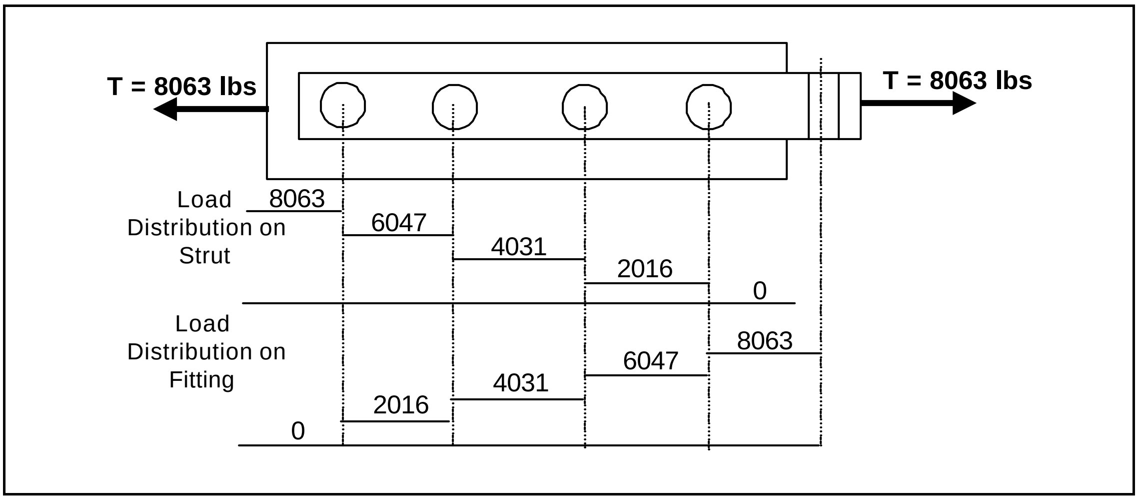

Figure 5: Strut / Fitting Load Distribution

Check the Shear Strength in the bolt AN-5 used to attach the strut:

The shear strength of AN-5 is 5750 lbf. This bolt is in double shear so the FS is:

$$FS = \frac{U_s}{T/2} = \frac{5750}{8063/2} = 1.42 \approx 1.5$$

Check the shear strength in the 4 bolts AN-4 used to attach the strut:

The shear strength of AN-4 is 3680 lbf. This bolt is in double shear and reacts to 2015.75 lbf so the FS is:

$$FS = \frac{U_s}{T/2} = \frac{3680}{2015.75/2} = 3.75 \succ 1.5$$

If you use only 3 bolts, the bolt will react to 2687.7 lbf and:

$$FS = \frac{U_s}{T/2} = \frac{3680}{2687.7/2} = 2.7 \succ 1.5$$

Recommendation 12:
A-151: Use only 3 bolts AN4 and reduce the length of the fitting according to that and use only two bolts for the rear strut.

Check shear bearing tear out of the fitting:

The effective area (A-2020) Ae = (1-3/16) X (1-5/16) = 0.56 in^2

$$s = \frac{T}{A_e} = \frac{8063}{0.56} = 14398 \text{ psi}$$

The factor of safety is:

$$FS = \frac{s_y}{s_{calc}} = \frac{37000\,psi}{14398\,psi} = 2.6 \succ 1.5$$

Structure Analysis: Front Spar Strut Attachment / Fitting (A-101 / A-102)

References:

[1] DWG No A-10
Table 1 of this document

Calculation:

The tension load acting on the spar fitting is: Front Strut Total Tension = 8063.36 lbf

Verification of the Tension Stress:

The effective area is: A_e = (1.5 - 5/16) X 3/16 = 0.223 in^2

$$s = \frac{F_S}{A_e} = \frac{8063}{0.223} = 36212 \text{ psi}$$

The factor of safety is:

$$FS = \frac{s_y}{s_{calc}} = \frac{37000\,psi}{36212\,psi} = 1.02 \prec 1.5$$

The factor of safety is not greater than 1.5, the design must be improved. If we use 2024-T3 instead of 6061-T6.

The factor of safety is:

$$FS = \frac{s_y}{s_{calc}} = \frac{45000\,psi}{36212\,psi} = 1.3(acceptable)$$

Recommendation 13:
A-102: Change material to 2024-T3.

Verification of the Shear Stress transmit in the three AN3-5A bolts:

The ultimate single shear strength for AN3 is 2126 lbf. The total force that can resist these three bolts are (3 X 2126 = 6078 lbf). So the factor of safety becomes :

$$FS = \frac{F_{bolt}}{F_{strut}} = \frac{6078lbf}{8063lbf} = 0.75 \prec 1.5$$

The factor of safety is not greater than 1.5, the design must be improved. Instead of using AN3 we will use AN4-5A (3680 lbs). The total force that the bolts will be able to resist will become (3 X 3680 = 11040 lbf) and the factor of safety is now:

$$FS = \frac{F_{bolt}}{F_{strut}} = \frac{11040lbf}{8063lbf} = 1.4 \approx 1.5$$

Recommendation 14:
A-102: Use three AN4-5A instead of three AN3-5A. A-101: Use two AN4-5A instead of two AN3-5A. Modify the three 3/16 dia holes to 1/4 dia to fit with A-102.

Structure Analysis: Flaperon Arms / Attach (A-501 - C-202)

References:

[1] DWG No C-20, C-30 and A-50
Table 1 of this document

Calculation:

Total force supplied by the aileron = 240 lbf and Total Force supplied by the flap = 290 lbf. For the flap (worst case) two AN3-6 bolts (20126 lbf) in double shear are used to maintain it. So we will have a very large factor of safety of:

$$FS = \frac{F_{bolt}}{F_{flap}} = \frac{2 \times 2126 lbf}{290 lbf \ / 2} = 30 \succ 1.5$$

No recommendation.

SECTION 5

RECOMMENDATIONS

This section summarizes all of the recommendations raised during the stress analysis. Also a cross-reference with the report page number is given. Finally the last column indicates the implementation date of the modification on the PEGA-STOL wing kit.

Table 2: Recommendations

Recommendations:	Page:	Implementation Date:
Recommendation 1: Use only one strut (the front strut) but shift the front spar 3 inches toward the leading edge, with this configuration 100% of the lift on the front spar instead of 111% and no rear strut (save weight).	10	Not Implemented.
Recommendation 2: Even if using Al-6061-T6 the FS > 1.5, it will be better for the spar caps to use Al-2024-T3 and increase the FS.	22	Summer 1999
Recommendation 3: Rivet type: A-5 pitch 2'' for a total of 71 rivets -> FS = 3.8	22	Summer 1999
Recommendation 4: For upper and lower skins, put a skin of 6061-T6 0.020''.	24	Summer 1999
Recommendation 5: Between STN 70 7/8 and STN 93 3/8 use A-5 pitch 25 rivets on the ribs and on the spar junctions.	24	Summer 1999
Recommendation 6: For ribs located at STN 70 7/8, 81 3/8 and 93 3/8, use 2024-T3 0.032'' thick.	25	Summer 1999
Recommendation 7: Use AN ½ for the front spar wing attachment bolt, modify holes of pieces A-206 and A-203 according to that.	26	Summer 1999
Recommendation 8: Use AN 1/4 for the rear spar attachment bolt, modify holes of piece A-404 accordingly.	27	Summer 1999
Recommendation 9: For pieces A-206, Fitting doubler, change material for 2024-T3 and modify height quote (1.5) by 2.5. Use material 2024-T3 for A-203: Front spar root fitting.	27	Summer 1999
Recommendation 10: For pieces A-206, A-203 (Fitting doubler + Front spar root fitting) drill when assembled, and drill to fit the AN 1/2 dia.	27	Summer 1999
Recommendation 11: A-206: Instead of using 2 rows of 3 rivets, use 3 rows of 4 rivets SS-5. **Recommendation 12:** A-151: Use only 3 bolts AN4 and reduce the length of the fitting accordingly and use only two bolts for the rear strut.	28 30	Summer 1999 Summer 1999
Recommendation 13: A-102: Change material for 2024-T3.	31	Summer 1999
Recommendation 14: A-102: Use three AN4-5A instead of three AN3-5A. A-101: Use two AN4-5A instead of two AN3-5A. Modify the three 3/16 dia holes to 1/4 dia to fit with A-102.	31	Summer 1999

APPENDIX A

REFERENCES DATA

This Appendix regroups applicable document photocopies of useful data.