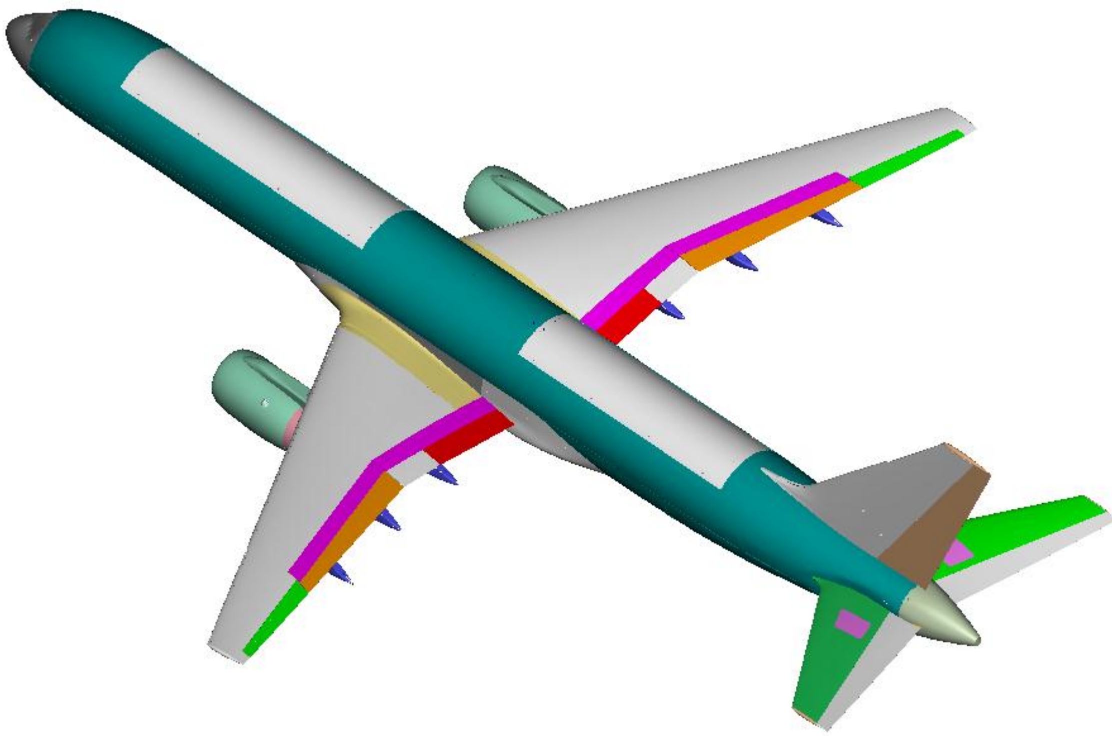
GTM-T2 Stress Analysis

Project Document # GTMP-4006

Model Number RBL-04-0426

5.5 % Dynamically Scaled Generic
Transport Model, Trainer #2



NASA Langley Research Center

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Model Number RBL-04-0426

5.5 % Dynamically Scaled Generic Transport Model

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NASA Langley Research Center

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INTRODUCTION

This report verifies the structural integrity of the 5.5% dynamically scaled Generic Transport Model Trainer number 2. This vehicle is a remotely piloted air vehicle that is used for Control Upset Prevention and Recovery (CUPR) element of the Aviation Safety Program (AvSP) and includes but is not limited to re-configurable controls, control upset recovery, collision avoidance, and flight controls research. The Generic Transport Model Trainer number 2 (a.k.a. GTM-T2) is fabricated from fiberglass composite sandwich assemblies made from ¼ inch honeycomb, balsa wood, and "West Systems" epoxy. The GTM-T2 vehicle is comprised of 6 major assemblies: the fuselage, the wing, the vertical tail, the horizontal tail, the nacelles & pylons, and the wing/fuselage fairing. GTM-T2 has a maximum weight of 55 pounds and has a maximum thrust capacity of 40 pounds from the two"AMT-180-SP" turbine jet engines. The engines are fueled with a 20:1 mixture of clear Kerosene "K1" and "AreoShell" turbine oil number 500 (MIL-PRF-23699 STD). The vehicle has a fuel capacity of about 1.8 gallons that is contained in two separate collapsible fuel bladders located at the vehicle's center of gravity. The GTM-T2 has nose gear steering, a retractable landing gear system, and a main landing gear braking system, all powered by an onboard compressed air bottle. The wing assembly bolts to the fuselage from below and houses the ten servos that control the 2 ailerons, 4 spoilers, and 4 flaps. The wing assembly has leading edge devices and thus the leading edge is in the "deployed" position. The empennage consists of a vertical tail assembly and a horizontal tail assembly. The vertical tail rudder is actuated by 2 servos located in the vertical tail internally. The horizontal tail has 4 elevators, 2 per side, actuated by 2 servos located internally. The horizontal tail has a manual-lockout and "trimmable". The fuselage contains the fuel system and all other associated electronics systems to operate the vehicle for a flight time of about 30 minutes depending on the fuel usage rate and onboard battery power drainage. The GTM-T2 has a landing/take-off speed of about 53mph with an 11.5-degree rotation angle and a test speed of 75mph. This analysis uses a speed of 100 mph for conservatism. The model has a thrust to weight ratio of approximately 0.75, so a vehicle speed feedback system is utilized to notify the pilots of the model's velocity.

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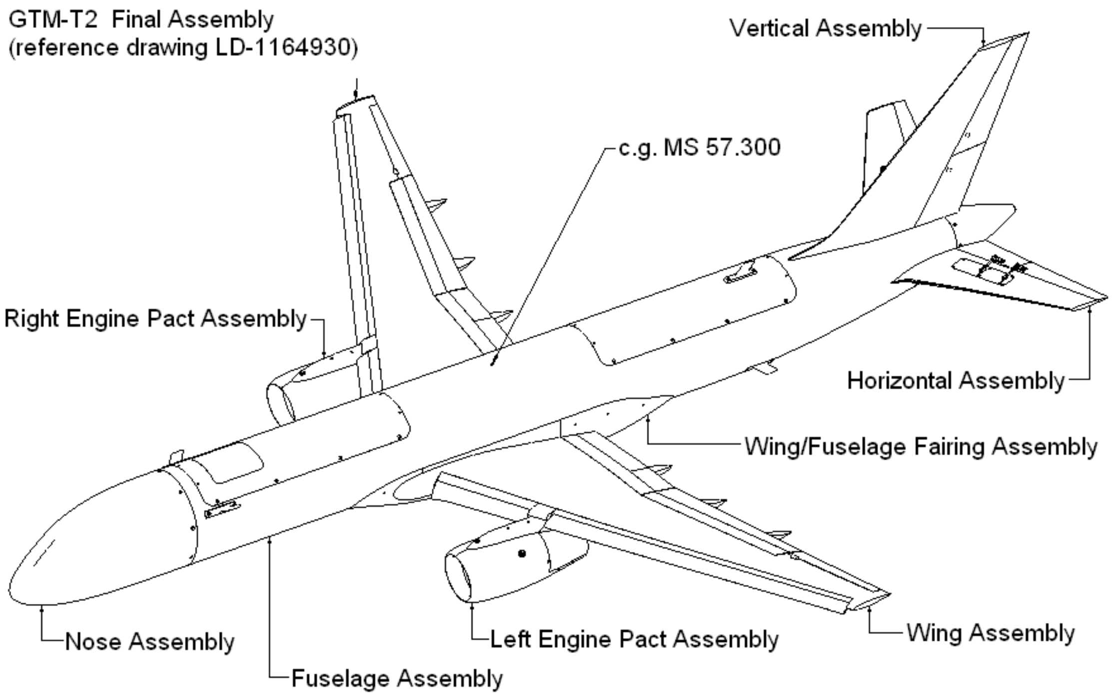
Weight Breakdown:

Part/Assembly	Weight (*lbs*)
Airframe	20.90
(Wing Assembly 9.6*lbs*)	
Primer & Paint	0.77
Nose & Main Landing Gear	3.50
Engines & Support Hardware	7.50
Servos (18)	2.00
All other electronics & wiring	6.80
Fuel Tanks	0.50
Fuel, Max. Capacity	9.75

TOTAL = 51.72 *lbs*

GTM-T2 Final Assembly:



Inertial Target Values:

Weight:	49.6 *lbs*	
Ixx (roll):	1.327 $sl\text{-}ft^2$	6146.33 $lb\text{-}in^2$
Izz (yaw):	5.454 $sl\text{-}ft^2$	19703.45 $lb\text{-}in^2$
Iyy (pitch)	4.254 $sl\text{-}ft^2$	25261.54 $lb\text{-}in^2$
Ixz	0.120 $sl\text{-}ft^2$	555.81 $lb\text{-}in^2$

Target C.G. = MS 57.300 *in*

Dynamic Pressure Calculation

A vehicle test velocity of 75 mph is dictated by the customer as a design requirement (reference document GTMP-1101). However, the dynamic pressure calculation for this analysis uses a speed of 100 mph for a conservative approach to this report.

$$q = \frac{1}{2} * \rho * V^2$$

Where: q = dynamic pressure, $\frac{lb}{ft^2}$

ρ (air density) = $0.0023769 \frac{slug}{ft^3}$

V (speed) = $146.7 \frac{ft}{\sec}$ (= 100 mph)

1 slug = $\frac{1lb. - \sec^2}{ft}$

So:

$$q = \frac{1}{2}\left(\frac{0.0023769 slug}{ft^3}\right)\left(\frac{146.7\, ft}{\sec}\right)^2 \left(\frac{\frac{1lb. - \sec^2}{ft}}{1slug}\right)$$

$$q = \frac{1}{2}\left(\frac{0.0023769 slug}{ft^3}\right)\left(\frac{146.7^2\ ft^2}{\sec^2}\right)\left(\frac{1}{1slug}\right)\left(\frac{1lb. - \sec^2}{ft}\right)$$

Therefore:

$q = 25.58 psf$ or $0.17761 psi$

Stall Speed Calculation

The following calculations were performed to illustrate the stall speed of the airplane.

The Equations:

$$V_{stall} = (5.46)\frac{\left(W/S\right)^{\frac{1}{2}}}{\left(dC_L\right)^{\frac{1}{2}}}$$ Ref. Don Brooks, Model Aviation magazine, September 2002, Page 66.

Where: V_{Stall} = stall speed, *mph*

W = weight, *oz*
S = wing area, ft^2
d = air density, *g/L*
C_L = lift coefficient, dimensionless

And:

$$V_{Stall} = \sqrt{\left(\frac{2W}{\rho_\infty S C_{L_{Max}}}\right)}$$

Ref. Anderson, Fundamentals of Aerodynamics, Page 43, 3rd Edition, McGraw Hill, New York, 2001,

Where: V_{Stall} = stall speed, m/s

W = weight, *N*
S = wing area, m^2
ρ_∞ = air density, kg/m^3
C_L = lift coefficient, dimensionless

Two different equations are used for stall speed calculation to verify the results. Both equations will be used to test a "fixed slats & flaps stowed" and "fixed slats & flaps deployed" a scenario.

Case 1-Stall Speed Where C_L = 1.77 (fixed slats & flaps stowed):

Reference NACA 427, April 6, 1932
Model Aviation Magazine Calculation:

$$V_{stall} = (5.46)\frac{\left(W/S\right)^{\frac{1}{2}}}{\left(dC_L\right)^{\frac{1}{2}}}$$

$$W = 55lb = 880oz$$

$$S = 6.03\,ft^2$$

$$d = 4.4255x10^{-5}\,\frac{lb}{in^3}\left(\frac{61.02374in^3}{1L}\right)\left(\frac{453.5924g}{1lb}\right) = 1.2297\frac{g}{L}$$

$$C_L = 1.77$$

$$V_{Stall} = (5.46)\frac{\left[\left(880oz/6.03\,ft^2\right)\right]^{\frac{1}{2}}}{\left[\left(1.2297\,g/L\right)(1.77)\right]^{\frac{1}{2}}} = 44.71mph$$

Case 2-Stall Speed Where C_L = 2.18 (fixed slats & flaps deployed):

Reference NACA 427, April 6, 1932
Model Aviation Magazine Calculations:

$$V_{stall} = (5.46)\frac{\left(W/S\right)^{\frac{1}{2}}}{\left(dC_L\right)^{\frac{1}{2}}}$$

$$W = 55lb = 880oz$$

$$S = 6.03\,ft^2$$

$$d = 4.4255x10^{-5}\,\frac{lb}{in^3}\left(\frac{61.02374in^3}{1L}\right)\left(\frac{453.5924g}{1lb}\right) = 1.2297\frac{g}{L}$$

$$C_L = 2.18$$

$$V_{Stall} = (5.46)\frac{\left[\left(880oz/6.03\,ft^2\right)\right]^{\frac{1}{2}}}{\left[\left(1.2297\,g/L\right)(2.18)\right]^{\frac{1}{2}}} = 40.29mph$$

Fundamentals of Aerodynamics Calculations where C_L =1.77:

$$V_{Stall} = \sqrt{\left(\frac{2W}{\rho_\infty S C_{L_{Max}}}\right)}$$

This equation is solved using the metric system units. The final answer will be converted into the English unit system.

$W = 55lb = 244.65N$
$\rho_\infty = 1.225 \, kg/m^3$
$S = 0.560205m^2$
$C_{Lmax} = 1.77$

$$V_{Stall} = \sqrt{\left(\frac{2(244.65N)}{\left(1.225 \, kg/m^3\right)\left(0.560205m^2\right)(1.77)}\right)} = 20.07 \, m/s$$

$$V_{Stall} = \left(20.07 \, m/s\right)\left(\frac{2.2269mph}{1m/s}\right) = 44.7mph$$

Fundamentals of Aerodynamics Calculation where C_L =2.18:

$$V_{Stall} = \sqrt{\left(\frac{2W}{\rho_\infty S C_{L_{Max}}}\right)}$$

This equation is solved using the metric system units. The final answer will be converted into the English unit system.

W=55lb=244.65N
$\rho_\infty = 1.225 \, kg/m^3$
$S = 0.560205m^2$
C_{Lmax}= 2.18

$$V_{Stall} = \sqrt{\left(\frac{2(244.65N)}{\left(1.225 \, kg/m^3\right)\left(0.560205m^2\right)(2.18)}\right)} = 18.09 \, m/s$$

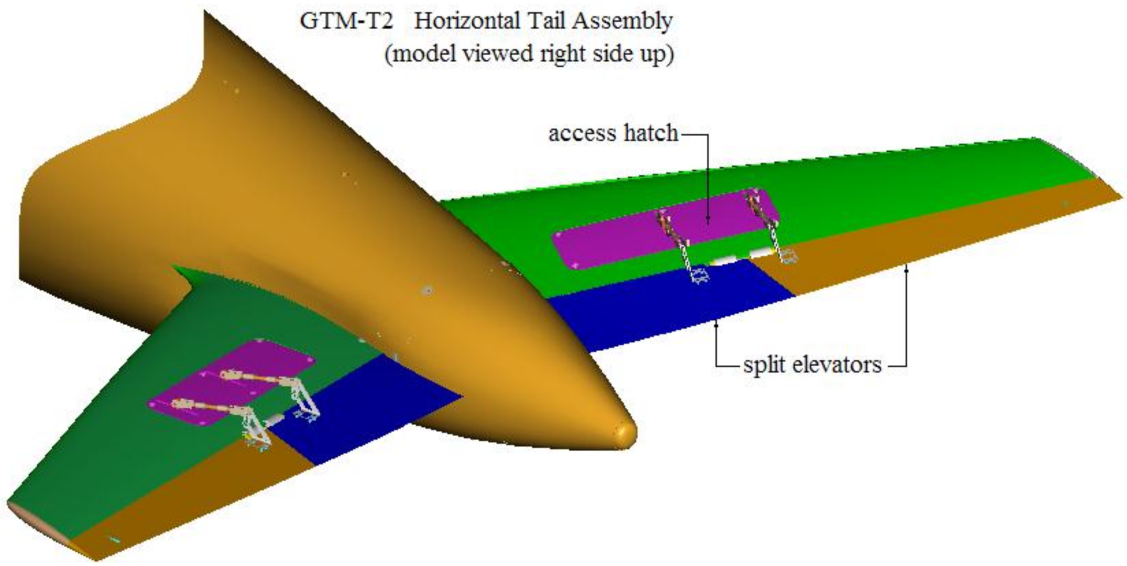
$$V_{Stall} = \left(18.09 \, m/s\right)\left(\frac{2.2269mph}{1m/s}\right) = 40.3mph$$

Stall Speed Conclusion

The stall speeds were verified for C_L = 1.77 (fixed slats & flaps stowed) and for C_L = 2.18 (fixed slats & flaps deployed) and the speeds were calculated using two different equations for each case. For the “fixed slats & flaps stowed” condition the average stall speed is 44.7 *mph.* For the “fixed slats & flaps deployed” condition the average stall speed calculated is 40.3 *mph*.

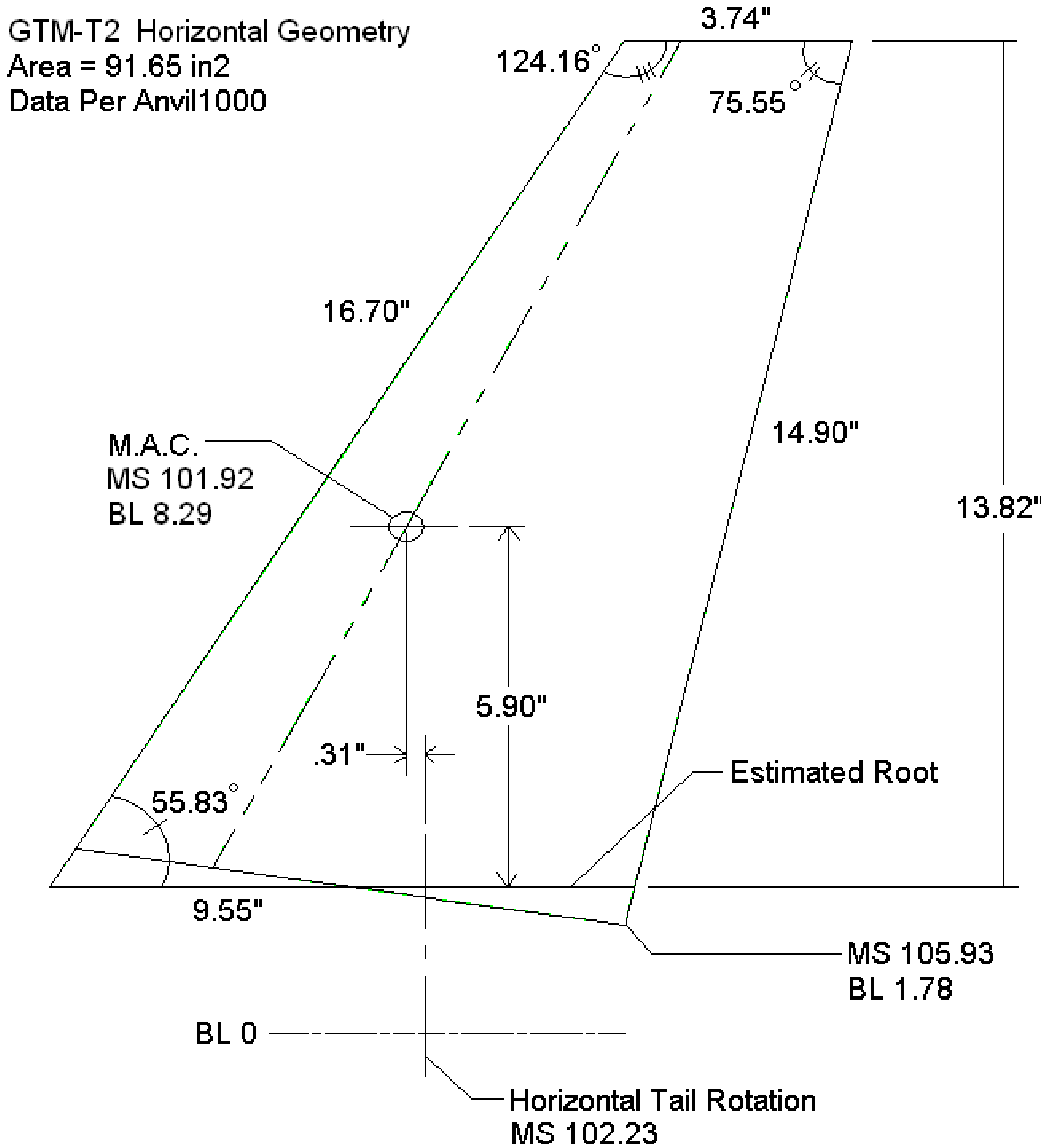
Horizontal Assembly Analysis

Calculations are done to prove the integrity of the horizontal tail assembly, the spin recovery lock-out system and their related mounting & assembly components. It should be noted that the O.E.M. airfoil shape for the horizontal surfaces is upside down when compared to the main wing surfaces.



Reference drawing LD-1164908

Horizontal Tail MAC Location:
It is assumed that the aero load acts at the quarter MAC (MS 101.92 & BL 8.29). The horizontal loading location is sketched below:



Reference drawing LD-1164918

The aerodynamic lift at 100*mph* on the horizontal assembly is,

$$F_{lift} = A \cdot C_L \cdot q$$
$$Assume, C_L = 1.44 \ (per\ the\ research\ dept.)$$
$$F_{lift} = (91.65in^2)(1.44)(25.58psf)\left(1ft^2 / 144in^2\right)$$
$$F_{lift} = 23.44lbs(per\ horizontal\ assembly)$$
$$F_{Lift\_Total} = (2)(23.44) = 46.88lbs$$

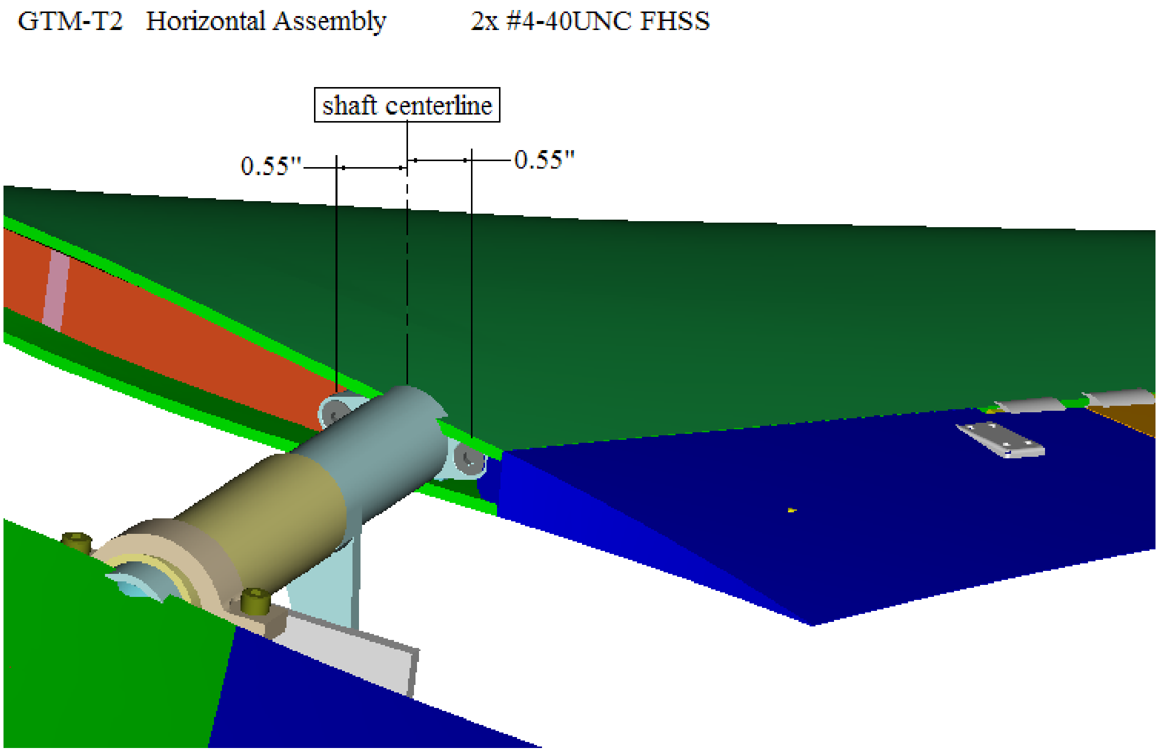
And from the above figure the offset distance to the spin recovery shaft is 0.31 *in*. And the resulting torque due to the aero load on the horizontal surfaces is:

$$Torque_{shaft\_pivot} = F_{Lift\_Total}(0.31in)$$
$$Torque_{shaft\_pivot} = (46.88lbs)(0.31in)$$
$$Torque_{shaft\_pivot} = 14.53in - lbs \ (for\ both\ horizontals)$$

And the elevator hinge moment loading is 2.21*in-lbs* for the outboard elevator and 3.24*in-lbs* for the inboard elevator (reference control surface hinge moments in this document). So the total shaft pivot torque is assumed to be:

$$Torque_{TOTAL\_shaft\_pivot} = 14.53 + 2(2.21) + 2(3.24)$$
$$Torque_{TOTAL\_shaft\_pivot} = 25.43in - lbs$$

A flange and bolt assembly connects the pivot shaft to the horizontal boss. The pivot arm is part of the shaft. Two #4-40 UNC FHSS transfer the torque generated from the horizontals. The sketch below gives the holes spacing:



Drawing Reference LD-1164908

The shear in the #4-40UNC FHSS is,

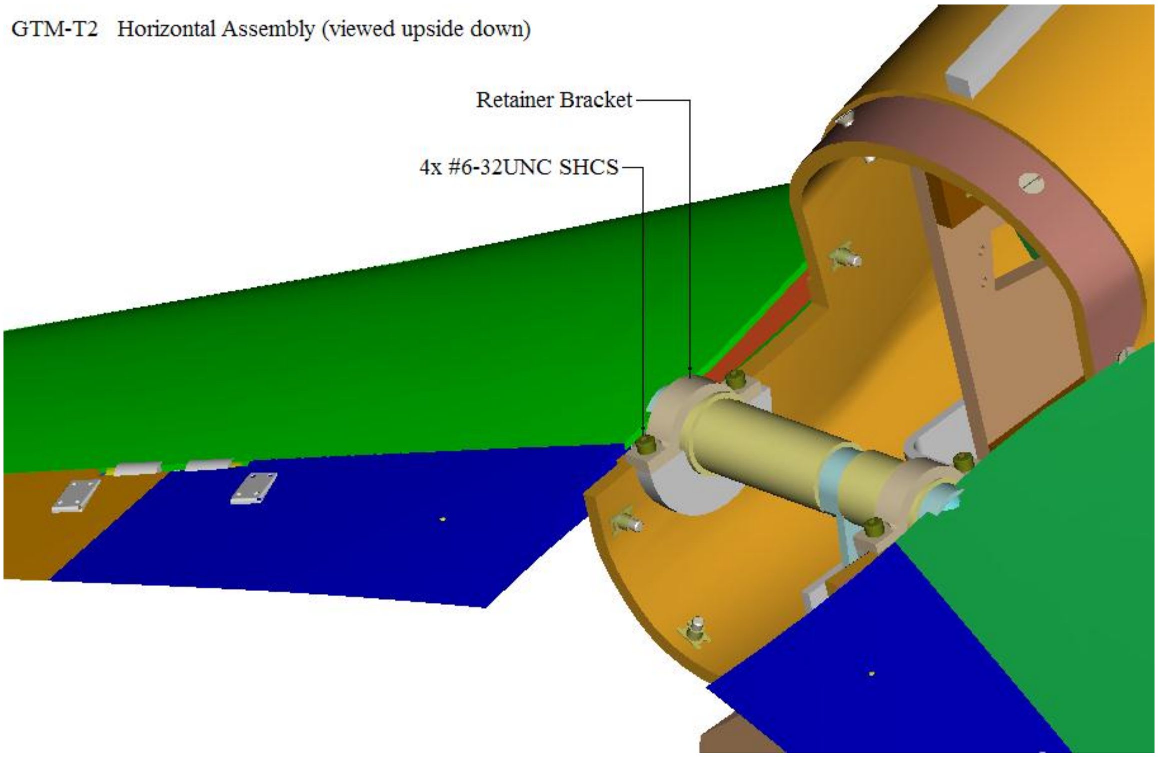
$$Torque_{TOTAL\_shaft\_pivot} = 25.43in - lbs = F_{shear}(0.55in) + F_{shear}(0.55in)$$
$$\therefore F_{shear}(1.1in) = 25.43in - lbs$$
$$F_{shear} = 23.12\ lbs\ \ per\ \ bolt$$

The shear strength, at the thread root, of a #4-40 UNC FHSS is 620*lbs* (reference Holo-Krome Technical Handbook, P/N 99004 10M HU, revised 12/90), therefore, the factor of safety is,

$$\boxed{F.S. = \frac{620lbs}{23.12lbs} = 26.8}$$

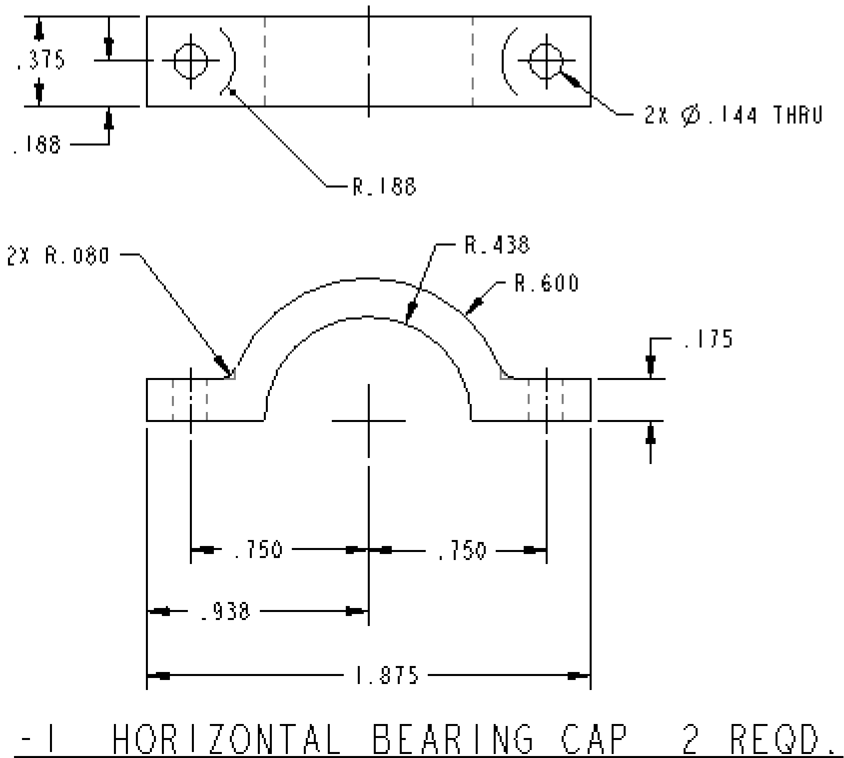
Pivot Shaft Retainer Brackets:

The retainer brackets, or bearing caps, position and hold the pivot shaft to the aft fuselage. A bracket is located on each side of the rear fuselage. Each bracket is retained by two #6-32 UNC SHCS as shown below:



Drawing Reference LD-1164908

It is assumed that the maximum load on each bearing cap is 23.44*lbs*, reference above analysis, and acts perpendicular to its mounting surface. A sketch of this bracket or bearing cap is given below:



Drawing Reference LD-1164915-1

<u>Bearing Caps, Bolt Tension:</u>

$$F_{\substack{bolt\\ \#6-32\\ SHCS}} = \frac{F_h}{(no.bolts)} = \frac{46.88lbs}{4bolts}$$

$$F_{\substack{bolt\\ \#6-32\\ SHCS}} = 11.72lbs$$

And, the factor of safety based on a tensile strength of 1640*lbs* for a #6-32 UNC SHCS, 1960 series, alloy steel construction, is:

$$\boxed{FS = \frac{1640lbs}{11.72lbs} = 140}$$

Bearing Caps, Bolt Head Shear Pull-Thru:

Assuming that the maximum bolt load is 11.72 *lbs* then,

$$F_{\substack{bolt \\ \#6-32 \\ SHCS}} = 11.72 lbs$$

$$and,$$

$$Area_{shear} = (flange\ \ thickness)(bolt\ \ head\ \ \ perimeter)$$

$$Area_{shear} = t[\pi D] = (0.175in)[\pi(0.217in)]$$

$$Area_{shear} = 0.119in^2$$

$$Then,$$

$$\tau = \frac{11.72lbs}{0.119in^2} = 98.5psi$$

The shear strength of 6061-T6 aluminum is 27*ksi*, as a result the factor of safety is:

$$\boxed{FS = \frac{27000psi}{98.5psi} = 274}$$

Bearing Caps, Cap Tension at the Base:

Assuming that the maximum load is 11.72*lbs* and using the smallest thickness, the area is,

$$Area_{M-M} = (no.\ of\ \ \ places)(width)(thickness) = n\,w\,t$$

$$Area_{M-M} = (2\ \ places)(0.385in)(0.16in)$$

$$Area_{M-M} = (2)(0.385)(0.16) = 0.1232in^2$$

Then the tensile stress is:

$$\sigma = \frac{F_{h\_totlal}}{Area_{M-M}} = \frac{(2\_bolts)(11.72lbs)}{(0.1232in^2)}$$

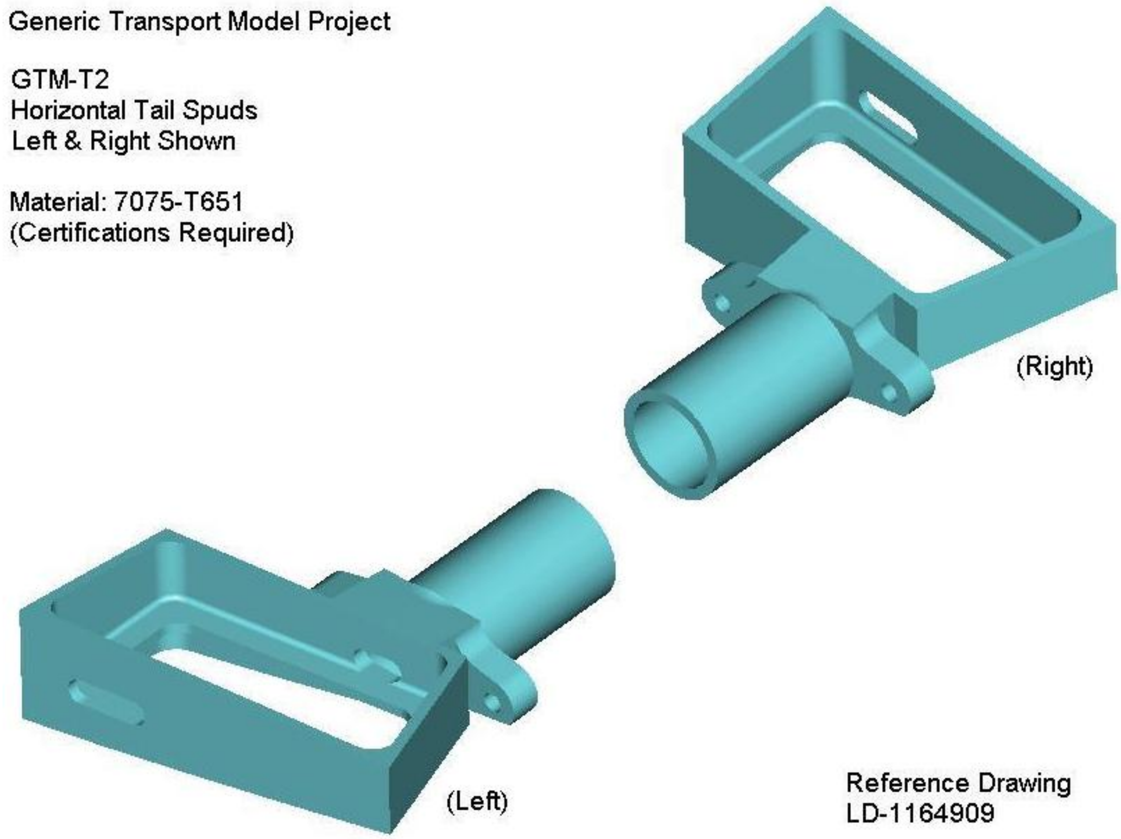
$$\sigma = 190.3psi$$

Then the factor of safety based on the yield strength of the 6061-T6 aluminum is,

$$\boxed{FS = \frac{42000psi}{190.3psi} = 221}$$

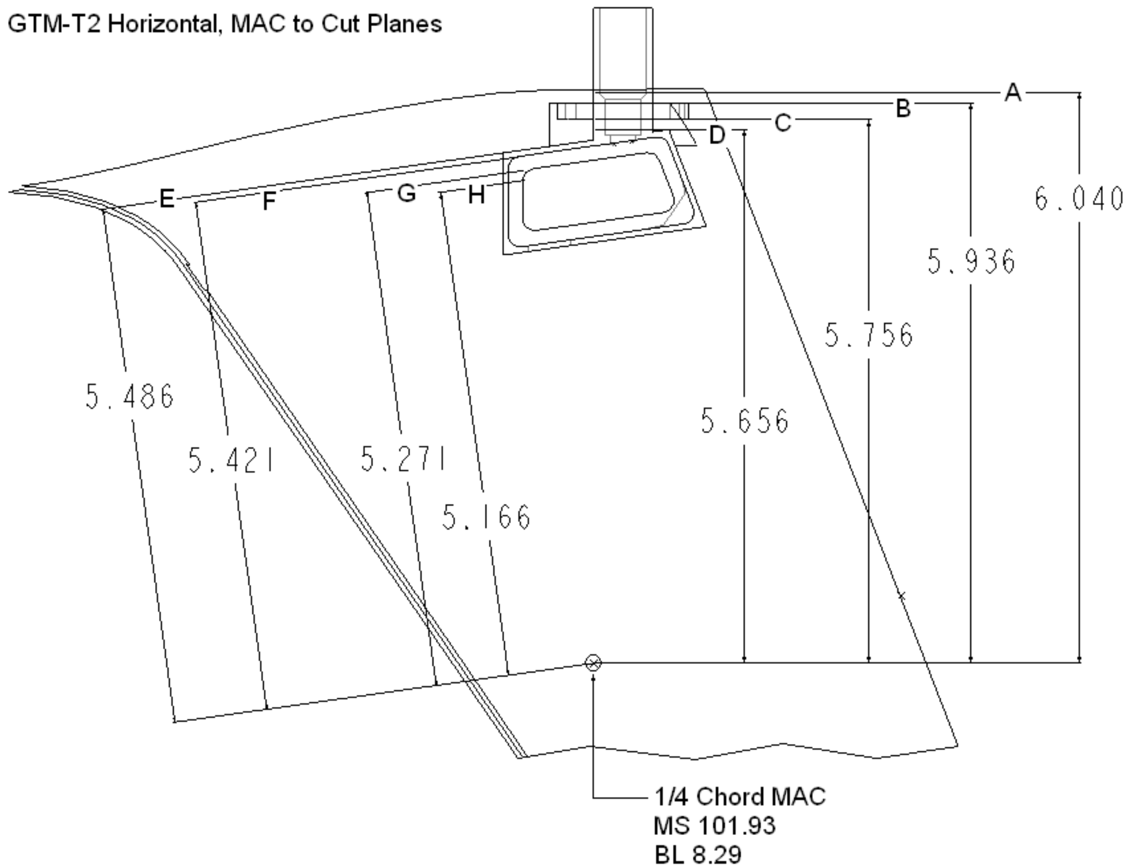
Horizontal Spud Stresses

The horizontal spud is located at the root of the horizontal assembly and is sandwiched between composite skins. These calculations were done to prove that the spud, made from aluminum 7075-T651, would withstand the load applied to the horizontal tail. The spuds are shown below:



Drawing reference LD-1164909-1

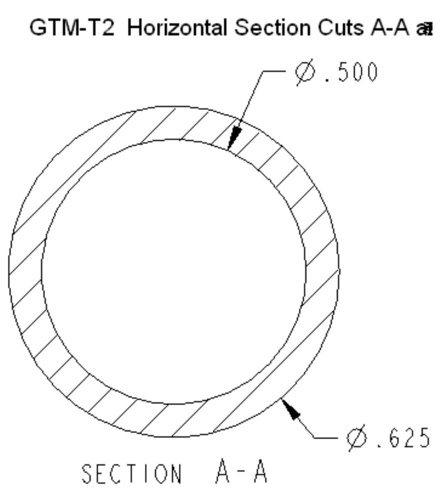
The analysis calculates the stresses at eight cross-sections. Each section A-A through H-H is analyzed individually. These calculations compute the force, moment, bending stress, and shear stress for each section displayed in the sketch below:



Drawing reference LD-1164909-1

The force is 23.44*lbs*. (see previous analysis).

<u>Section A-A:</u>



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT A-A

AREA =  1.1044646e-01 INCH^2

CENTER OF GRAVITY with respect to _A coordinate frame:
X   Y     1.3299000e+01  1.0222900e+02  INCH
INERTIA with respect to _A coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.1542548e+03 -1.5015677e+02
Iyx Iyy -1.5015677e+02  1.9538359e+01

POLAR MOMENT OF INERTIA:  1.1737932e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _A coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.4221708e-03  0.0000000e+00
Iyx Iyy  0.0000000e+00  4.4221817e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  4.4221708e-03  4.4221817e-03

POLAR MOMENT OF INERTIA:  8.8443526e-03 INCH^4
ROTATION MATRIX from _A orientation to PRINCIPAL AXES:
      1.0            0.00000
      0.00000        1.00000

ROTATION ANGLE from _A orientation to PRINCIPAL AXES (degrees):
about z axis          0.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  2.0009759e-01  2.0009783e-01  INCH


SECTION MODULI and corresponding points:
                    MODULUS                  1           2   COORD
about AXIS 1:  1.41509e-02 INCH^3       -3.4728e-12 -3.1250e-01   INCH
               1.41509e-02 INCH^3       -3.6575e-12  3.1250e-01   INCH
about AXIS 2:  1.42965e-02 INCH^3       -3.0932e-01 -4.4473e-02   INCH
               1.42965e-02 INCH^3        3.0932e-01  4.4473e-02   INCH
```

END PRO-ENGINEER SECTION DATA

Moment (section A-A):

$$M_{root} = Fd$$
$$M_{root} = (23.44lbf)(6.04in)$$
$$M_{root} = 141.58in \cdot lb$$

Bending Stress (section A-A):

Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.004422in^4}{0.0141509in^3} = 0.312in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(141.58in \cdot lb)(0.312in)}{0.004422in^4}$$
$$\sigma = 10004.8\,psi$$

Shear Stress (section A-A):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{23.44lbf}{0.11in^2}$$
$$\tau = 212.23\,psi$$

Principal Stress (section A-A):

$$\sigma_{A-A} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\sigma_{A-A} = \frac{10004.8\,psi}{2} \pm \sqrt{\left(\frac{10004.8\,psi}{2}\right)^2 + (212.23\,psi)^2}$$
$$\sigma_{A-A} = 5002.42 + 5006.92$$
$$\sigma_{A-A} = 10009.3\,psi$$

Maximum Shear Stress (section A-A):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{A-A}}{2}\right)^2 + \tau^2}$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{10009.3\,psi}{2}\right)^2 + (212.23\,psi)^2}$$

$$\tau_{\max} = 5009.2\,psi$$

Factor of Safety (section A-A):

The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

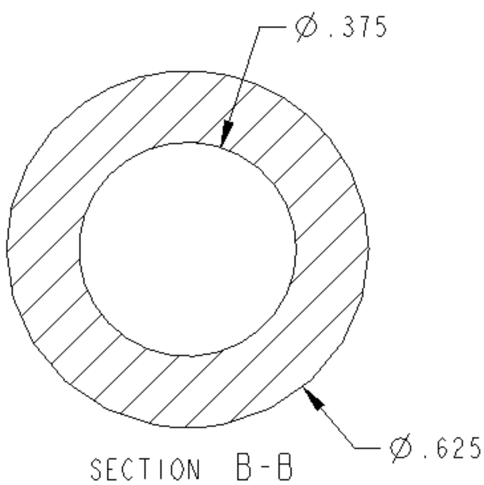
$$\boxed{FS = \frac{67000\,psi}{10009.3\,psi} = 6.7}$$

The ultimate shear strength for aluminum 7075-T651 is 42*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{42000\,psi}{5009.2\,psi} = 8.4}$$

<u>Section B-B:</u>

GTM-T2 Horizontal Section Cut B-B



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT B-B

AREA =  1.1044646e-01 INCH^2

CENTER OF GRAVITY with respect to _A coordinate frame:
X   Y     1.3299000e+01  1.0222900e+02  INCH

INERTIA with respect to _A coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.1542548e+03 -1.5015677e+02
Iyx Iyy -1.5015677e+02  1.9538359e+01

POLAR MOMENT OF INERTIA:  1.1737932e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _A coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.4221708e-03  0.0000000e+00
Iyx Iyy  0.0000000e+00  4.4221817e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  4.4221708e-03  4.4221817e-03

POLAR MOMENT OF INERTIA:  8.8443526e-03 INCH^4

ROTATION MATRIX from _A orientation to PRINCIPAL AXES:
       1.00000        0.00000
       0.00000        1.00000

ROTATION ANGLE from _A orientation to PRINCIPAL AXES (degrees):
about z axis          0.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  2.0009759e-01  2.0009783e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                  1          2   COORD
about AXIS 1:  1.41509e-02 INCH^3       -3.4728e-12 -3.1250e-01   INCH
               1.41509e-02 INCH^3       -3.6575e-12  3.1250e-01   INCH
about AXIS 2:  1.42965e-02 INCH^3       -3.0932e-01 -4.4473e-02   INCH
               1.42965e-02 INCH^3        3.0932e-01  4.4473e-02   INCH
```

END PRO-ENGINEER SECTION DATA

Moment (section B-B):

$$M_{root} = Fd$$
$$M_{root} = (23.44lbf)(5.936in)$$
$$M_{root} = 139.14in \cdot lb$$

Bending Stress (section B-B):

Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.004422in^4}{0.014151in^3} = 0.312in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(139.14in \cdot lb)(0.312in)}{0.004422in^4}$$
$$\sigma = 9832.6psi$$

Shear Stress (section B-B):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{23.44lbf}{0.11in^2}$$
$$\tau = 212.23psi$$

Principal Stress (section B-B):

$$\sigma_{B-B} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\sigma_{B-B} = \frac{9832.6psi}{2} \pm \sqrt{\left(\frac{9832.6psi}{2}\right)^2 + (212.23psi)^2}$$
$$\sigma_{B-B} = 4916.3 + 4920.9$$
$$\sigma_{B-B} = 9837.2psi$$

Maximum Shear Stress (section B-B):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{B-B}}{2}\right)^2 + \tau^2}$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{9837.2\,psi}{2}\right)^2 + (212.23\,psi)^2}$$

$$\tau_{\max} = 4923.2\,psi$$

Stress Concentration (section B-B):

Using "Stress Concentrations Factors", by Peterson, © 1974, figure 73:

D = 0.848
d = 0.625
r = 0.010
D/d = 0.848/0.625= 1.36 (assumed 1.50)
r/d = 0.010/0.625= .126
From figure 73: Kt = 1.66

Factor of Safety (section B-B):

The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

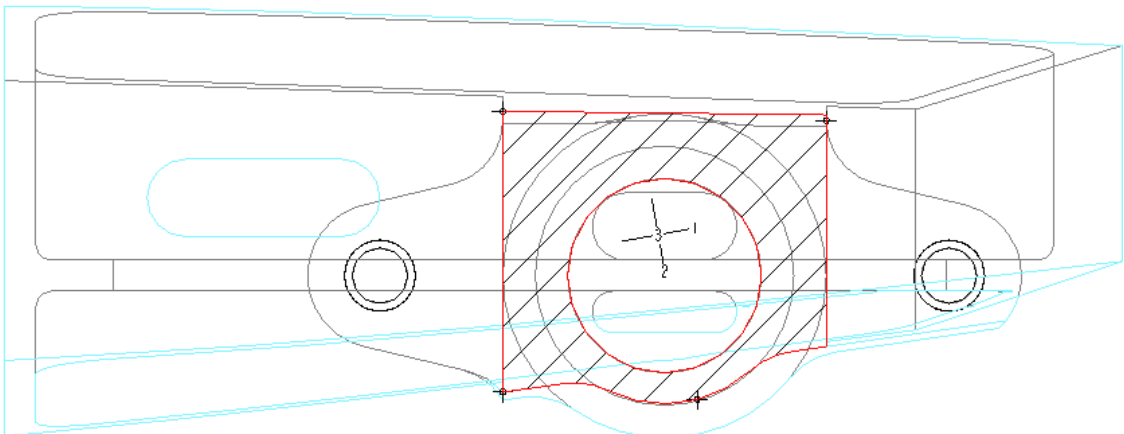
$$\boxed{FS = \frac{67000\,psi}{(1.66)\cdot 9837.2\,psi} = \frac{67000}{16329.7} = 4.1}$$

The ultimate shear strength for aluminum 7075-T651 is 42*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{42000\,psi}{4923.2\,psi} = 8.5}$$

Section C-C:

GTM-T2 Horizontal Cut C-C



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT C-C

AREA =  2.1500764e-01 INCH^2

CENTER OF GRAVITY with respect to _C coordinate frame:
X   Y     1.3378009e+01  1.0221687e+02  INCH

INERTIA with respect to _C coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.2464709e+03 -2.9401452e+02
Iyx Iyy -2.9401452e+02  3.8486270e+01

POLAR MOMENT OF INERTIA:  2.2849571e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _C coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.1795808e-03 -5.7150223e-04
Iyx Iyy -5.7150223e-04  6.1107866e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  6.0078110e-03  9.2825564e-03

POLAR MOMENT OF INERTIA:  1.5290367e-02 INCH^4

ROTATION MATRIX from _C orientation to PRINCIPAL AXES:
       0.17733       -0.98415
       0.98415        0.17733

ROTATION ANGLE from _C orientation to PRINCIPAL AXES (degrees):
```

```
about z axis         79.786

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.6715955e-01  2.0778149e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                   1           2   COORD
about AXIS 1:  2.08973e-02 INCH^3       -2.5340e-01 -2.8749e-01   INCH
               1.84241e-02 INCH^3        1.7905e-02  3.2608e-01   INCH
about AXIS 2:  2.65643e-02 INCH^3       -3.4944e-01  2.4549e-01   INCH
               2.58941e-02 INCH^3        3.5848e-01 -1.5884e-01   INCH
```

END PRO-ENGINEER SECTION DATA

Moment (section C-C):

$$M_{root} = Fd$$
$$M_{root} = (23.44lbf)(5.756in)$$
$$M_{root} = 134.9in \cdot lb$$

Bending Stress (section C-C):

Checking c and solving for σ.

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.006in^4}{0.01842in^3} = 0.326in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(134.9in \cdot lb)(0.326in)}{0.006in^4}$$
$$\sigma = 7323.1psi$$

Shear Stress (section C-C):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{23.44lbf}{0.215in^2}$$
$$\tau = 109.0psi$$

Principal Stress (section C-C):

$$\sigma_{C-C} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$

$$\sigma_{C-C} = \frac{7323.1 psi}{2} \pm \sqrt{\left(\frac{7323.1 psi}{2}\right)^2 + (109.0\, psi)^2}$$

$$\sigma_{C-C} = 3661.5 + 3663.2$$

$$\sigma_{C-C} = 7324.7\, psi$$

Maximum Shear Stress (section C-C):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{C-C}}{2}\right)^2 + \tau^2}$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{7324.7\, psi}{2}\right)^2 + (109.0\, psi)^2}$$

$$\tau_{\max} = 3664.0\, psi$$

Factor of Safety (section C-C):

The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

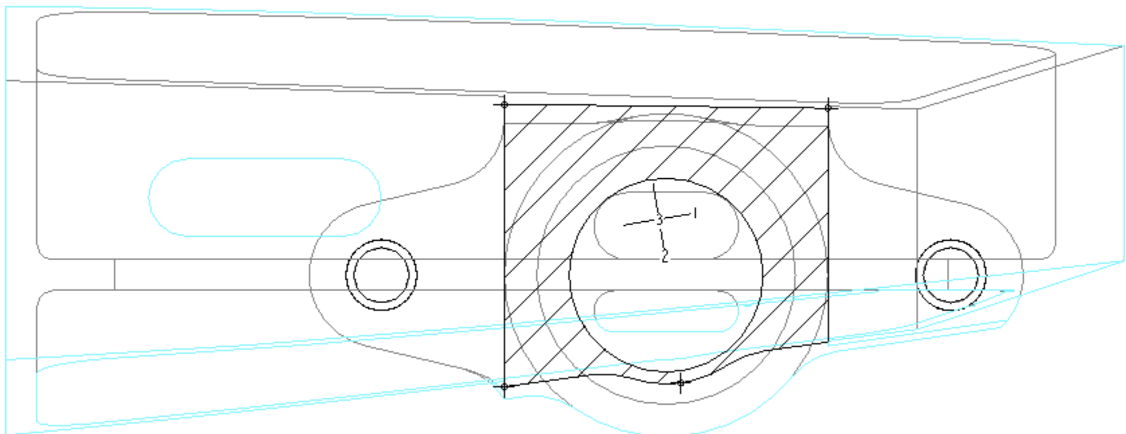
$$\boxed{FS = \frac{67000\, psi}{7324.7\, psi} = 9.2}$$

The ultimate shear strength for aluminum 7075-T651 is 42*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{42000\, psi}{3664.0\, psi} = 11.5}$$

Section D-D:

GTM-T2 Horizontal Cut D-D



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT D-D

AREA =  2.0860103e-01 INCH^2

CENTER OF GRAVITY with respect to _D coordinate frame:
X   Y     1.3407193e+01  1.0221519e+02  INCH

INERTIA with respect to _D coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.1794615e+03 -2.8587148e+02
Iyx Iyy -2.8587148e+02  3.7501858e+01

POLAR MOMENT OF INERTIA:  2.2169634e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _D coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.1784518e-03 -6.8776089e-04
Iyx Iyy -6.8776089e-04  5.2315819e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  5.1151698e-03  9.2948638e-03

POLAR MOMENT OF INERTIA:  1.4410034e-02 INCH^4
```

```
ROTATION MATRIX from _D orientation to PRINCIPAL AXES:
       0.16689       -0.98598
       0.98598        0.16689

ROTATION ANGLE from _D orientation to PRINCIPAL AXES (degrees):
about z axis          80.393

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.5659280e-01  2.1108788e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                      1          2   COORD
about AXIS 1:  1.90189e-02 INCH^3       -2.5742e-01 -2.6895e-01   INCH
               1.60116e-02 INCH^3       -1.0976e-02  3.1947e-01   INCH
about AXIS 2:  2.66665e-02 INCH^3       -3.4856e-01  2.6950e-01   INCH
               2.59958e-02 INCH^3        3.5755e-01 -1.5719e-01   INCH
```

END PRO-ENGINEER SECTION DATA

Moment (section D-D):

$$M_{root} = Fd$$
$$M_{root} = (23.44 lbf)(5.656 in)$$
$$M_{root} = 132.6 in \cdot lb$$

Bending Stress (section D-D):

Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.005 in^4}{0.0160 in^3} = 0.320 in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(132.6 in \cdot lb)(0.320 in)}{0.005 in^4}$$
$$\sigma = 8280.0\, psi$$

Shear Stress (section D-D):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{23.44 lbf}{0.209 in^2}$$
$$\tau = 112.4\, psi$$

Principal Stress (section D-D):

$$\sigma_{D-D} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\sigma_{D-D} = \frac{8280.0\, psi}{2} \pm \sqrt{\left(\frac{8280.0\, psi}{2}\right)^2 + (112.4\, psi)^2}$$
$$\sigma_{D-D} = 4140.0 + 4141.5$$
$$\sigma_{D-D} = 8281.6\, psi$$

Maximum Shear Stress (section D-D):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{D-D}}{2}\right)^2 + \tau^2}$$
$$\tau_{\max} = \pm\sqrt{\left(\frac{8281.6\, psi}{2}\right)^2 + (112.4\, psi)^2}$$
$$\tau_{\max} = 4142.3\, psi$$

Factor of Safety (section D-D):

The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

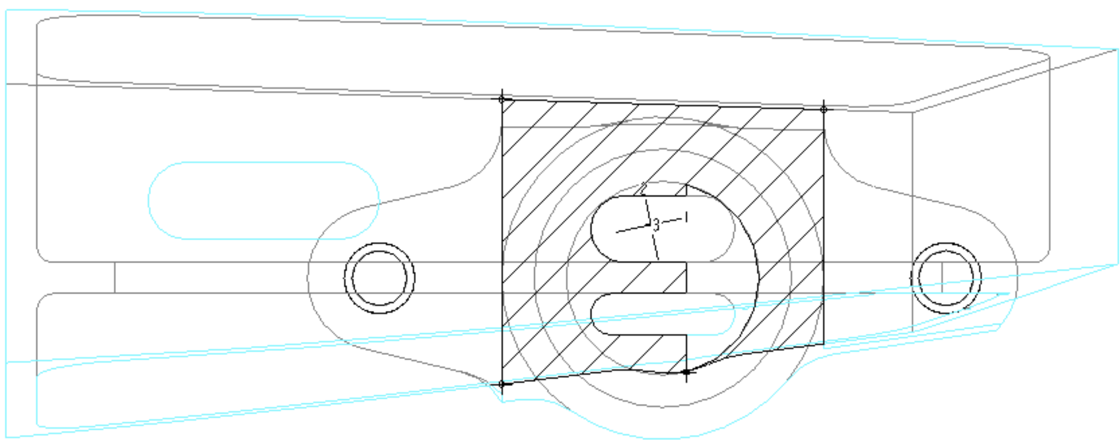
$$\boxed{FS = \frac{67000\, psi}{8281.6\, psi} = 8.1}$$

The ultimate shear strength for aluminum 7075-T651 is 42*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{42000\, psi}{4142.3\, psi} = 10.1}$$

Section E-E:

GTM-T2 Horizontal Cut E-E



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT E-E

AREA =  2.4448252e-01 INCH^2

CENTER OF GRAVITY with respect to _E coordinate frame:
X   Y    -1.3402125e+01 -5.8431156e-02  INCH

INERTIA with respect to _E coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.0593561e-02 -1.9062348e-01
Iyx Iyy -1.9062348e-01  4.3919111e+01

POLAR MOMENT OF INERTIA:  4.3929704e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _E coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.7588485e-03  8.3118773e-04
Iyx Iyy  8.3118773e-04  5.9051025e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  5.7334731e-03  9.9304780e-03

POLAR MOMENT OF INERTIA:  1.5663951e-02 INCH^4

ROTATION MATRIX from _E orientation to PRINCIPAL AXES:
      -0.20222        -0.97934
       0.97934        -0.20222
```

```
ROTATION ANGLE from _E orientation to PRINCIPAL AXES (degrees):
about z axis        101.667

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.5313871e-01  2.0153996e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  1.95604e-02 INCH^3        1.0636e-02 -2.9312e-01   INCH
               1.92975e-02 INCH^3       -2.3709e-01  2.9711e-01   INCH
about AXIS 2:  2.84874e-02 INCH^3       -3.4859e-01 -2.4288e-01   INCH
               2.63383e-02 INCH^3        3.7704e-01  1.5023e-01   INCH
```

END PRO-ENGINEER SECTION DATA

Moment (section E-E):

$$M_{root} = Fd$$
$$M_{root} = (23.44 lbf)(5.486 in)$$
$$M_{root} = 128.59 in \cdot lb$$

Bending Stress (section E-E):

Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.005 in^4}{0.0193 in^3} = 0.297 in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(128.59 in \cdot lb)(0.297 in)}{0.005 in^4}$$
$$\sigma = 6663.7 psi$$

Shear Stress (section E-E):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{23.44 lbf}{0.2444 in^2}$$
$$\tau = 95.9 psi$$

Principal Stress (section E-E):

$$\sigma_{E-E} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$

$$\sigma_{E-E} = \frac{6663.7\,psi}{2} \pm \sqrt{\left(\frac{6663.7\,psi}{2}\right)^2 + (95.9\,psi)^2}$$

$$\sigma_{E-E} = 3331.8 + 3333.2$$

$$\sigma_{E-E} = 6665.0\,psi$$

Maximum Shear Stress (section E-E):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{E-E}}{2}\right)^2 + \tau^2}$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{6665.0\,psi}{2}\right)^2 + (95.9\,psi)^2}$$

$$\tau_{\max} = 3333.9\,psi$$

Factor of Safety (section E-E):

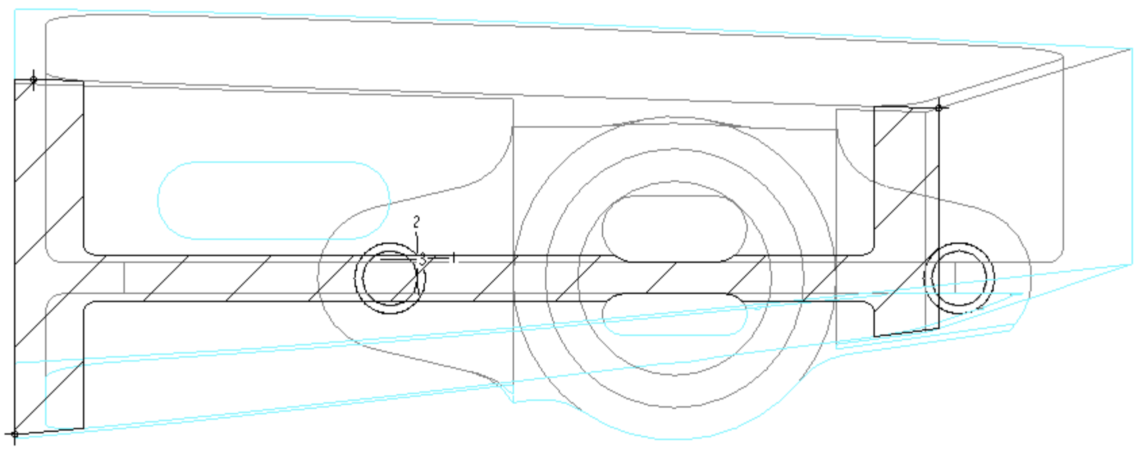
The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{67000\,psi}{6665.0\,psi} = 10.1}$$

The ultimate shear strength for aluminum 7075-T651 is 42*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{42000\,psi}{3333.9\,psi} = 12.6}$$

Section F-F:



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT F-F

AREA =  2.7509184e-01 INCH^2

CENTER OF GRAVITY with respect to _F coordinate frame:
X   Y    -1.3336303e+01 -5.4653420e-01  INCH

INERTIA with respect to _F coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.0697563e-01 -2.0019999e+00
Iyx Iyy -2.0019999e+00  4.8931969e+01

POLAR MOMENT OF INERTIA:  4.9138944e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _F coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.2480580e-01  3.0746408e-03
Iyx Iyy  3.0746408e-03  4.9628732e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  4.8840434e-03  1.2488463e-01

POLAR MOMENT OF INERTIA:  1.2976867e-01 INCH^4

ROTATION MATRIX from _F orientation to PRINCIPAL AXES:
      -0.02563        -0.99967
       0.99967        -0.02563
```

```
ROTATION ANGLE from _F orientation to PRINCIPAL AXES (degrees):
about z axis          91.469

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.3324499e-01  6.7377616e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                   1           2   COORD
about AXIS 1:  1.53311e-02 INCH^3       -7.9369e-01 -3.1857e-01   INCH
               1.33505e-02 INCH^3       -7.4010e-01  3.6583e-01   INCH
about AXIS 2:  1.57346e-01 INCH^3       -7.9369e-01 -3.1857e-01   INCH
               1.21894e-01 INCH^3        1.0245e+00  2.6490e-01   INCH
```

END PRO-ENGINEER SECTION DATA

Moment (section F-F):

$$M_{root} = Fd$$
$$M_{root} = (23.44lbf)(5.421in)$$
$$M_{root} = 127.1in \cdot lb$$

Bending Stress (section F-F):

Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.005in^4}{0.0133in^3} = 0.366in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(127.1in \cdot lb)(0.366in)}{0.005in^4}$$
$$\sigma = 9517.9\,psi$$

Shear Stress (section F-F):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{23.44lbf}{0.2751in^2}$$
$$\tau = 85.2\,psi$$

Principal Stress (section F-F):

$$\sigma_{F-F} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$

$$\sigma_{F-F} = \frac{9517.9\,psi}{2} \pm \sqrt{\left(\frac{9517.9\,psi}{2}\right)^2 + (85.2\,psi)^2}$$

$$\sigma_{F-F} = 4758.9 + 4759.7$$

$$\sigma_{F-F} = 9518.6\,psi$$

Maximum Shear Stress (section F-F):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{F-F}}{2}\right)^2 + \tau^2}$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{9518.6\,psi}{2}\right)^2 + (85.2\,psi)^2}$$

$$\tau_{\max} = 4760.1\,psi$$

Stress Concentration (section F-F):

Using “Stress Concentrations Factors”, by Peterson, © 1974, figure 73:

D = 0.551
d = 0.060
r = 0.030
D/d = 0.551/0.060= 9.18
r/d = 0.030/0.060= 0.50
The current ratio numbers are not present on Peterson (fig. 73) stress concentration graph; therefore it is assumed that the stress concentration factor is equal to 3.0.

Factor of Safety (section F-F):

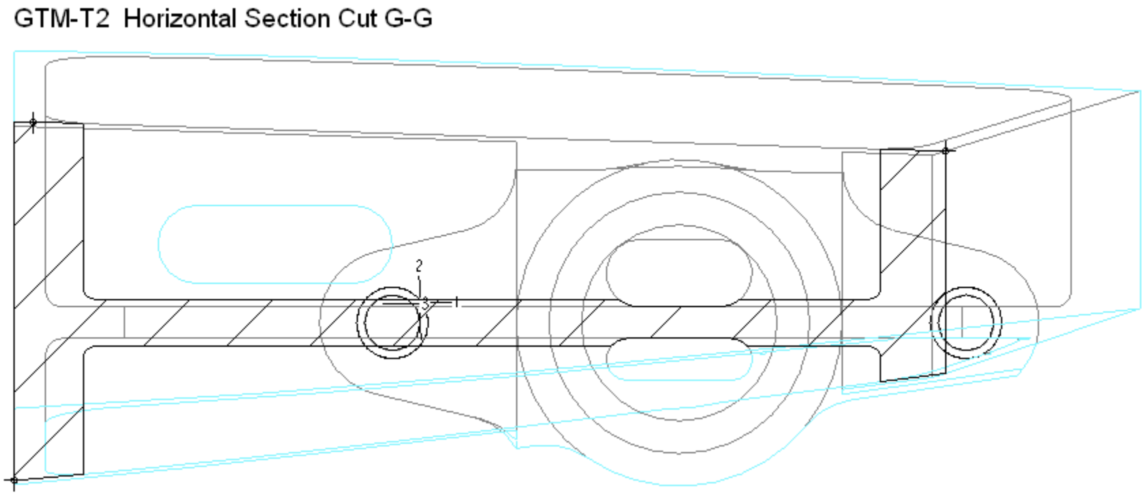
The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{67000\,psi}{(3.0)\cdot 9518.6\,psi} = \frac{67000\,psi}{28555.9} = 2.4}$$

The ultimate shear strength for aluminum 7075-T651 is 42*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{42000\,psi}{(3.0)\cdot 4760.1\,psi} = \frac{42000\,psi}{14280.2} = 3.0}$$

Section G-G:



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT G-G

AREA =  9.5439128e-02 INCH^2

CENTER OF GRAVITY with respect to _G coordinate frame:
X   Y    -1.3362873e+01 -5.6805991e-01  INCH

INERTIA with respect to _G coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.9727623e-02 -7.2278680e-01
Iyx Iyy -7.2278680e-01  1.7044490e+01

POLAR MOMENT OF INERTIA:  1.7144218e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _G coordinate frame:
(INCH^4)
INERTIA TENSOR:
```

```
Ixx Ixy  6.8930173e-02  1.6832645e-03
Iyx Iyy  1.6832645e-03  2.2717144e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.2292355e-03  6.8972652e-02

POLAR MOMENT OF INERTIA:  7.1201888e-02 INCH^4

ROTATION MATRIX from _G orientation to PRINCIPAL AXES:
      -0.02523       -0.99968
       0.99968       -0.02523

ROTATION ANGLE from _G orientation to PRINCIPAL AXES (degrees):
about z axis          91.446

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.5283216e-01  8.5011021e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  6.87181e-03 INCH^3       -7.6291e-01 -3.2440e-01   INCH
               6.19012e-03 INCH^3       -7.4564e-01  3.6013e-01   INCH
about AXIS 2:  8.69439e-02 INCH^3       -7.9330e-01 -3.2608e-01   INCH
               6.38130e-02 INCH^3        1.0809e+00  2.5457e-01   INCH
```

<u>END PRO-ENGINEER SECTION DATA</u>

Moment (section G-G):

$$M_{root} = Fd$$
$$M_{root} = (23.44lbf)(5.271in)$$
$$M_{root} = 123.55in \cdot lb$$

Bending Stress (section G-G):

Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.002in^4}{0.00619in^3} = 0.360in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(123.55in \cdot lb)(0.360in)}{0.002in^4}$$
$$\sigma = 19959.6\,psi$$

Shear Stress (section G-G):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{23.44lbf}{0.09544in^2}$$
$$\tau = 245.6\,psi$$

Principal Stress (section G-G):

$$\sigma_{G-G} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\sigma_{G-G} = \frac{19959.6\,psi}{2} \pm \sqrt{\left(\frac{19959.6\,psi}{2}\right)^2 + (245.6\,psi)^2}$$
$$\sigma_{G-G} = 9979.8 + 9982.8$$
$$\sigma_{G-G} = 19962.6\,psi$$

Maximum Shear Stress (section G-G):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{G-G}}{2}\right)^2 + \tau^2}$$
$$\tau_{\max} = \pm\sqrt{\left(\frac{19962.6\,psi}{2}\right)^2 + (245.6\,psi)^2}$$
$$\tau_{\max} = 9984.3\,psi$$

Factor of Safety (section G-G):

The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

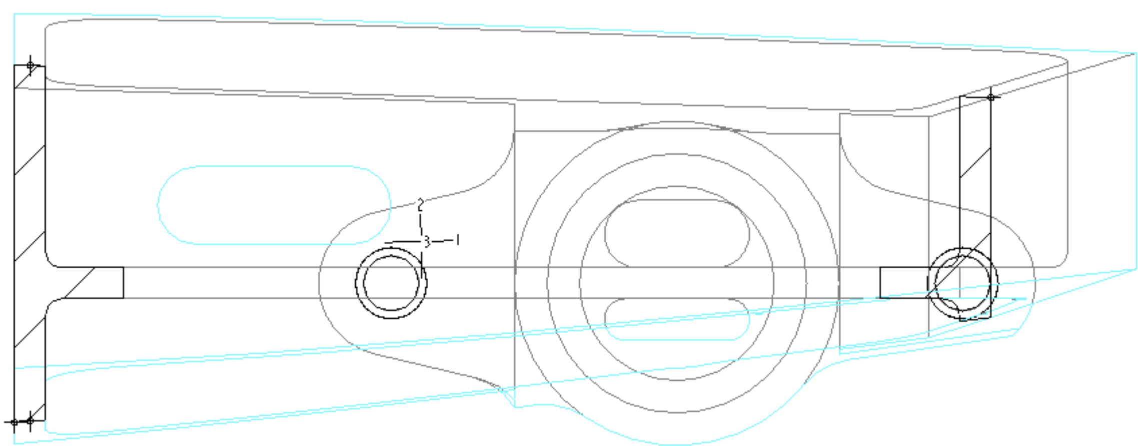
$$\boxed{FS = \frac{67000\,psi}{19962.6\,psi} = 3.4}$$

The ultimate shear strength for aluminum 7075-T651 is 42*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{42000\,psi}{9984.3\,psi} = 4.2}$$

Section H-H:

GTM-T2 Horizontal Cut H-H



Drawing reference LD-1164909-1

START PRO-ENGINEER SECTION DATA

```
GTM-T2 HORIZONTAL SPUD SECTION CUT H-H

AREA =  8.7064623e-02 INCH^2

CENTER OF GRAVITY with respect to _H coordinate frame:
X   Y    -1.3379502e+01 -5.7715051e-01  INCH

INERTIA with respect to _H coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.6890830e-02 -6.7062531e-01
Iyx Iyy -6.7062531e-01  1.5587803e+01

POLAR MOMENT OF INERTIA:  1.5684694e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _H coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  6.7889368e-02  1.6865443e-03
Iyx Iyy  1.6865443e-03  2.2702368e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.2269178e-03  6.7932687e-02

POLAR MOMENT OF INERTIA:  7.0159605e-02 INCH^4

ROTATION MATRIX from _H orientation to PRINCIPAL AXES:
```

```
      -0.02568       -0.99967
       0.99967       -0.02568

ROTATION ANGLE from _H orientation to PRINCIPAL AXES (degrees):
about z axis          91.471

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.5993048e-01  8.8332096e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                     1           2   COORD
about AXIS 1:  6.81399e-03 INCH^3      -7.6880e-01 -3.2682e-01   INCH
               6.21194e-03 INCH^3      -7.5120e-01  3.5849e-01   INCH
about AXIS 2:  8.50027e-02 INCH^3      -7.9918e-01 -3.2835e-01   INCH
               6.09513e-02 INCH^3       1.1145e+00  2.4866e-01   INCH
```

END PRO-ENGINEER SECTION DATA

Moment (section H-H):

$$M_{root} = Fd$$
$$M_{root} = (23.44lbf)(5.166in)$$
$$M_{root} = 121.1in \cdot lb$$

Bending Stress (section H-H):

Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.002in^4}{0.00621in^3} = 0.365in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(121.1in \cdot lb)(0.365in)}{0.002in^4}$$
$$\sigma = 19493.3psi$$

Shear Stress (section H-H):

$$\tau = \frac{F}{A}$$

$$\tau = \frac{23.44 lbf}{0.08706 in^2}$$

$$\tau = 269.2 psi$$

Principal Stress (section H-H):

$$\sigma_{H-H} = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$

$$\sigma_{H-H} = \frac{19493.3 psi}{2} \pm \sqrt{\left(\frac{19493.3 psi}{2}\right)^2 + (269.2 psi)^2}$$

$$\sigma_{H-H} = 9746.7 + 9750.4$$

$$\sigma_{H-H} = 19497.1 psi$$

Maximum Shear Stress (section H-H):

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_{H-H}}{2}\right)^2 + \tau^2}$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{19497.1 psi}{2}\right)^2 + (269.2 psi)^2}$$

$$\tau_{\max} = 9752.3 psi$$

Factor of Safety (section H-H):

The tensile yield strength for aluminum 7075-T651 is 67ksi. Thus the factor of safety is:

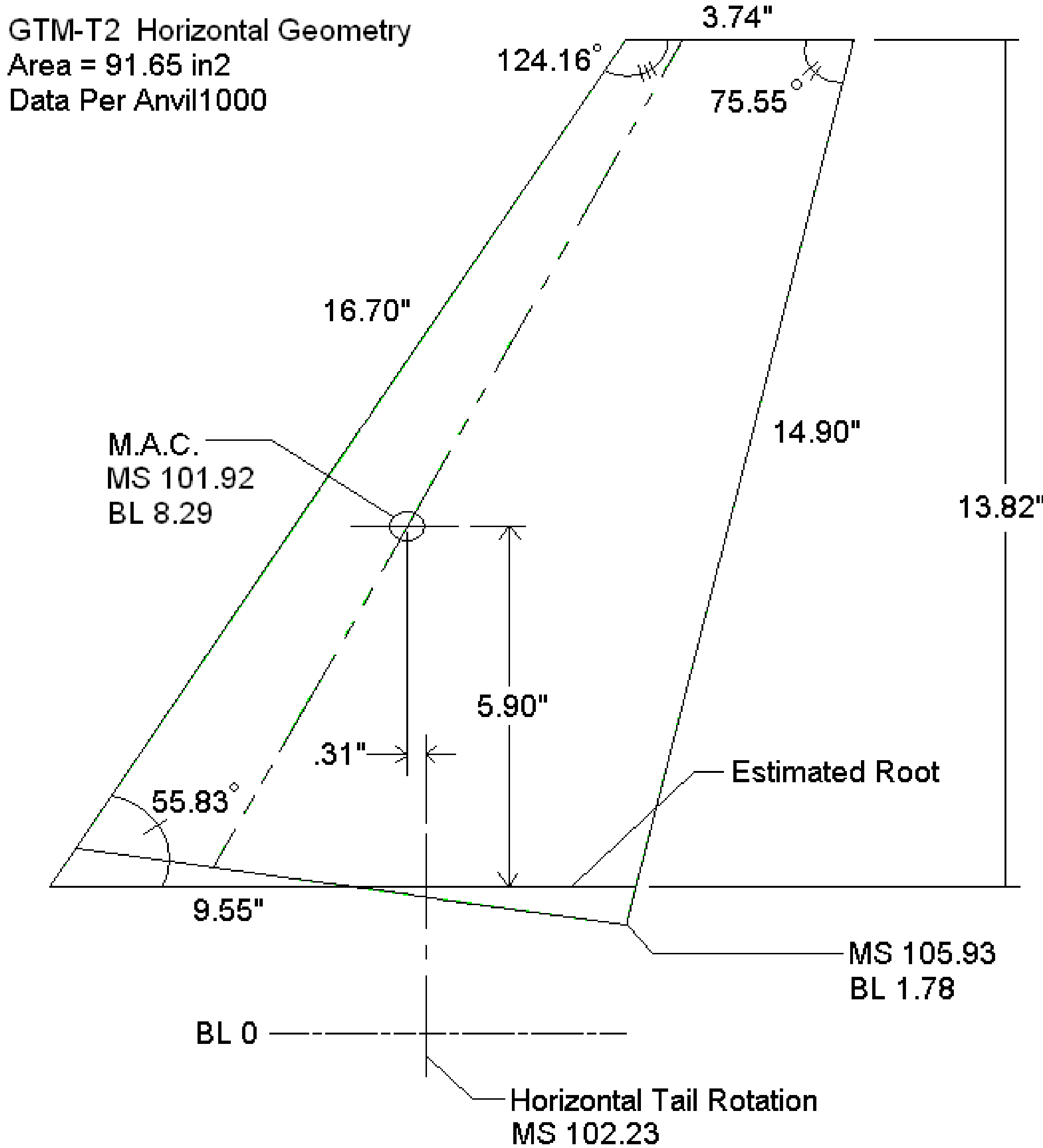
$$\boxed{FS = \frac{67000 psi}{19497.1 psi} = 3.4}$$

The ultimate shear strength for aluminum 7075-T651 is 42ksi. Thus the factor of safety is:

$$\boxed{FS = \frac{42000 psi}{9752.3 psi} = 4.3}$$

Horizontal Tail Skin Stresses

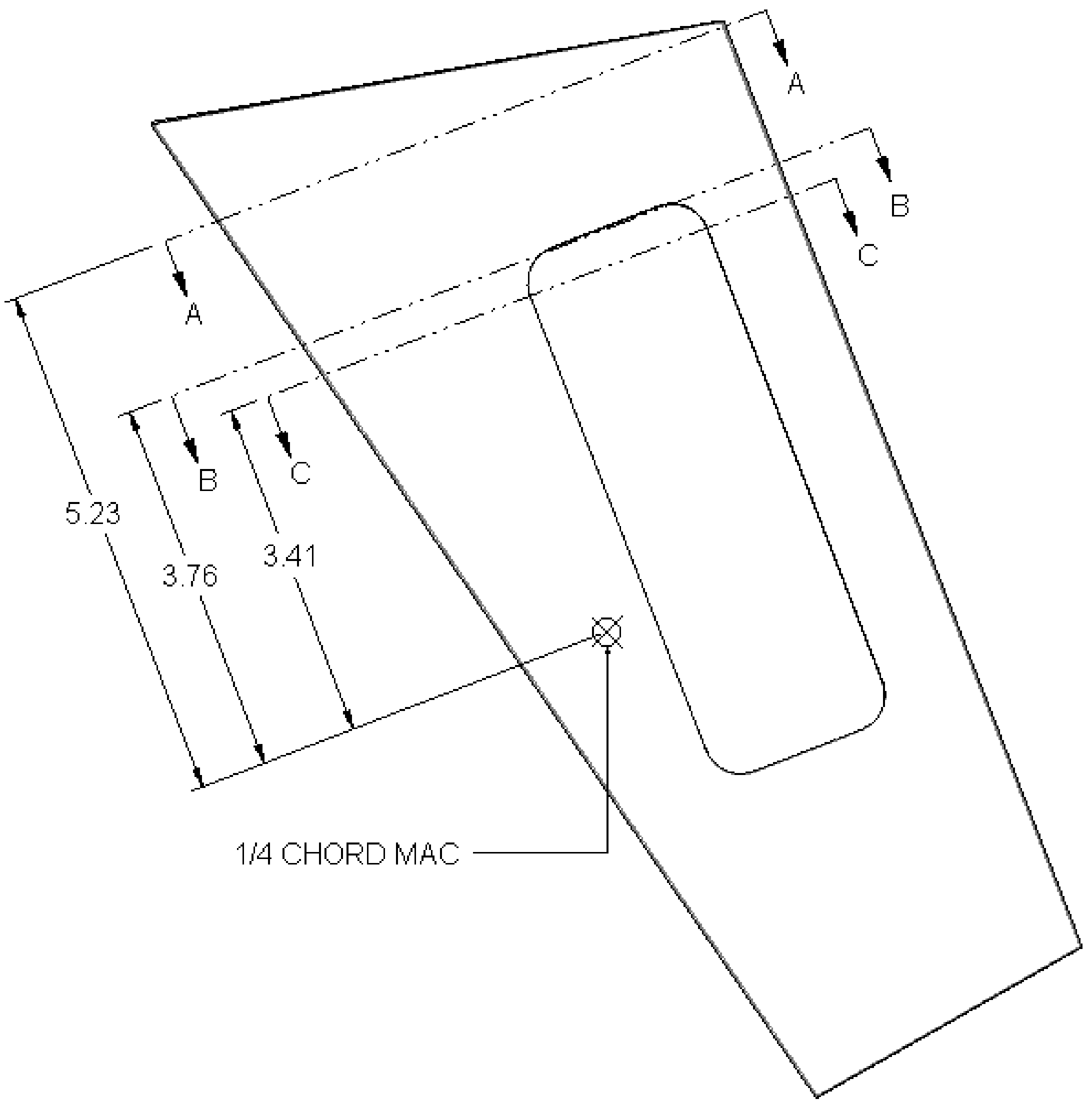
The horizontal skins are analyzed for combined shear and bending stress due to air loads (C_{Lmax}=1.44). Homogeneous material equations have been used in calculating a safety factor for the carbon fiber in bending. It is assumed that the aero load acts at the quarter MAC (MS 101.92 & BL 8.29). The horizontal loading location is sketched below:



Center of pressure location on the horizontal elevator, reference drawing LD-1164918

The servo hatch is not included in the analysis and it is assumed that it experiences little to no load. The skin section cuts are shown below:

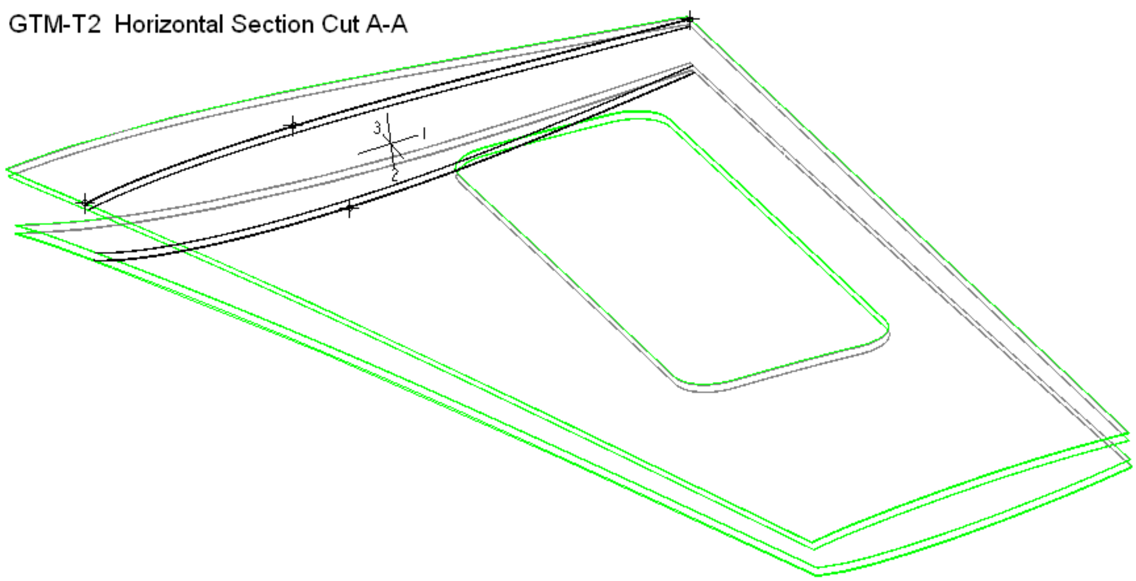
GTM -T2 Horizontal Section Cut Distances



Drawing reference no. LD-1164918

Wing Section Cut A-A Stress Analysis

The section cut A-A is shown below:



Drawing reference no. LD-1164918

The section cut A-A has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  1.6552986e-01 INCH^2

CENTER OF GRAVITY with respect to _AA coordinate frame:
X   Y     1.3377578e+01 -2.7155299e+00  INCH

INERTIA with respect to _AA coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.6295296e+00  6.0162271e+00
Iyx Iyy  6.0162271e+00  2.9646457e+01

POLAR MOMENT OF INERTIA:  3.1275986e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _AA coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.0889549e-01  2.9886003e-03
Iyx Iyy  2.9886003e-03  2.3299255e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.3276093e-02  4.0891865e-01

POLAR MOMENT OF INERTIA:  4.3219475e-01 INCH^4
```

```
ROTATION MATRIX from _AA orientation to PRINCIPAL AXES:
      -0.00775       -0.99997
       0.99997       -0.00775

ROTATION ANGLE from _AA orientation to PRINCIPAL AXES (degrees):
about z axis         90.444

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.7498757e-01  1.5717385e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                         1           2   COORD
about AXIS 1:  5.15252e-02 INCH^3       -8.2406e-01 -4.5174e-01   INCH
               4.86141e-02 INCH^3       -4.3248e-01  4.7879e-01   INCH
about AXIS 2:  1.50484e-01 INCH^3       -2.7174e+00 -3.3357e-01   INCH
               1.49424e-01 INCH^3        2.7366e+00 -2.8854e-01   INCH
```

END PRO-E DATA

The lift force on one wing assembly at 100*mph* (assume C_{Lmax}=1.44) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(91.65\,in^2\right)\cdot(1.44)\cdot(0.1776\,psi)$$
$$F_{Lift} = 23.4\ lbf$$

Therefore the shear stress at section cut A-A is:

$$\tau_{A-A} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{A-A} = \frac{23.4lbf}{0.1655in^2}$$
$$\tau_{A-A} = 141.6\,psi$$

The bending stress at section A-A due to the aero load is:

$$\sigma_{A-A} = \frac{M_{aeroload}}{z_{A-A}} = \frac{(23lbf)(5.23in)}{0.0486in^3}$$
$$\sigma_{A-A} = 2521.6\,psi$$

The maximum shear stress at section A-A due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{A-A}}{2}\right)^2 + \tau_{A-A}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{2521.6\,psi}{2}\right)^2 + (141.6\,psi)^2}$$

$$\tau_{Max} = 1268.7\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{17362\,psi}{1268.7\,psi} = 13.7}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\A-A}} = \frac{2521.6\,psi}{2} \pm \sqrt{\left(\frac{2521.6\,psi}{2}\right)^2 + (141.6\,psi)^2}$$

$$\sigma_{\substack{Max\\A-A}} = 1260.8 + 1268.7$$

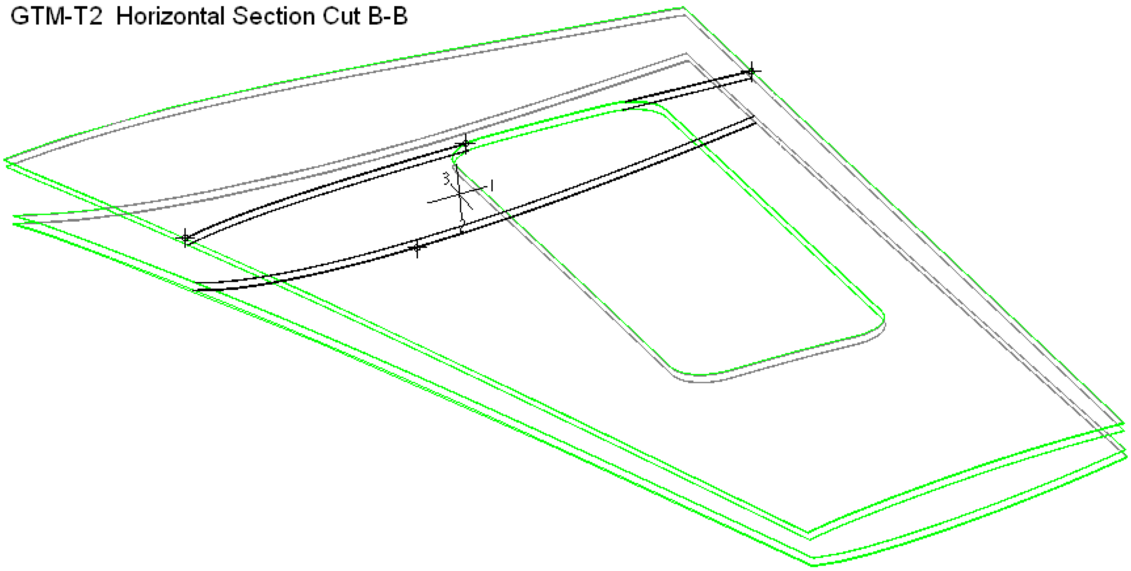
$$\sigma_{\substack{Max\\A-A}} = 2529.5\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{17362\,psi}{2529.5\,psi} = 6.86}$$

Wing Section Cut B-B Stress Analysis

The section cut B-B is shown below:



Drawing reference no. LD-1164918

The section cut B-B has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  1.3342595e-01 INCH^2

CENTER OF GRAVITY with respect to _BB coordinate frame:
X   Y     1.3523165e+01 -2.6542609e+00  INCH

INERTIA with respect to _BB coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.2590388e+00  4.8014074e+00
Iyx Iyy  4.8014074e+00  2.4416259e+01

POLAR MOMENT OF INERTIA:  2.5675298e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _BB coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.1903953e-01  1.2215341e-02
Iyx Iyy  1.2215341e-02  1.5856409e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.5365046e-02  3.1953089e-01
```

```
POLAR MOMENT OF INERTIA:  3.3489594e-01 INCH^4

ROTATION MATRIX from _BB orientation to PRINCIPAL AXES:
      -0.04019       -0.99919
       0.99919       -0.04019

ROTATION ANGLE from _BB orientation to PRINCIPAL AXES (degrees):
about z axis         92.303

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.3934916e-01  1.5475200e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                   1           2   COORD
about AXIS 1:  3.32502e-02 INCH^3        9.8701e-02 -4.6210e-01   INCH
               3.92953e-02 INCH^3       -4.1814e-01  3.9102e-01   INCH
about AXIS 2:  1.30776e-01 INCH^3       -2.4433e+00 -3.1241e-01   INCH
               1.19763e-01 INCH^3        2.6680e+00 -3.8202e-01   INCH
```

END PRO-E DATA

The lift force on one wing assembly at 100*mph* (assume C_{Lmax}=1.44) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(91.65\,in^2\right) \cdot (1.44) \cdot (0.1776\,psi)$$
$$F_{Lift} = 23.4\ lbf$$

Therefore the shear stress at section cut B-B is:

$$\tau_{B-B} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{B-B} = \frac{23.4 lbf}{0.1334 in^2}$$
$$\tau_{B-B} = 175.7\,psi$$

The bending stress at section B-B due to the aero load is:

$$\sigma_{B-B} = \frac{M_{aeroload}}{z_{B-B}} = \frac{(23.4 lbf)(3.76 in)}{0.0333 in^3}$$
$$\sigma_{B-B} = 2646.1 psi$$

The maximum shear stress at section B-B due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{B-B}}{2}\right)^2 + \tau_{B-B}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{2646.1psi}{2}\right)^2 + (175.7\,psi)^2}$$

$$\tau_{Max} = 1334.7\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{17362\,psi}{1334.7\,psi} = 13.0}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ B-B}} = \frac{2646.1psi}{2} \pm \sqrt{\left(\frac{2646.1psi}{2}\right)^2 + (175.7\,psi)^2}$$

$$\sigma_{\substack{Max \\ B-B}} = 1323.1 + 1334.7$$

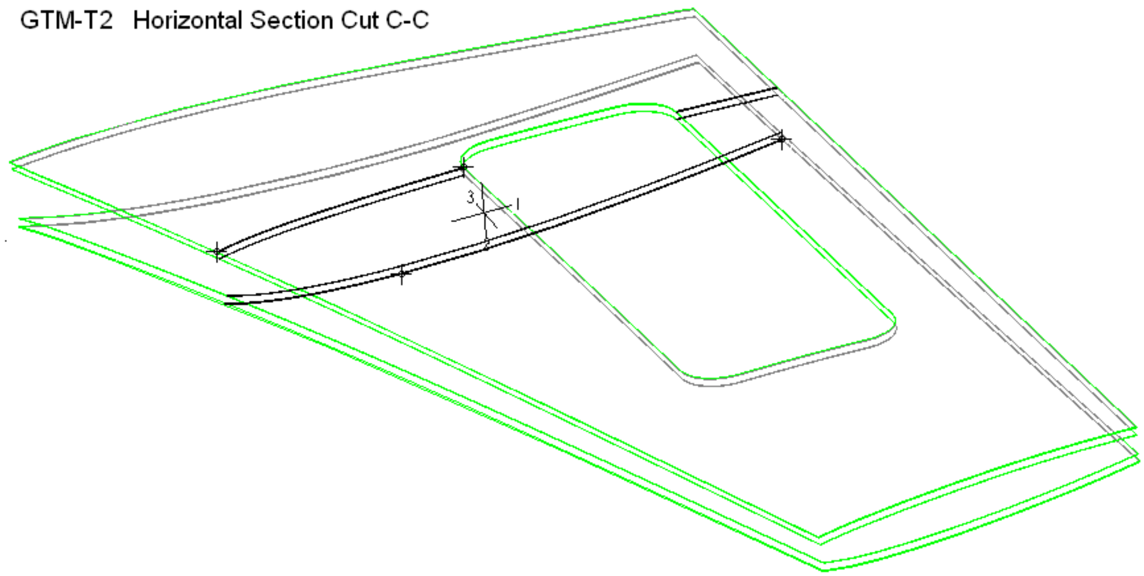
$$\sigma_{\substack{Max \\ B-B}} = 2657.7\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{17362\,psi}{2657.7\,psi} = 6.53}$$

Wing Section Cut C-C Stress Analysis

The section cut C-C is shown below:



Drawing reference no. LD-1164918

The section cut C-C has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  1.2361009e-01 INCH^2

CENTER OF GRAVITY with respect to _CC coordinate frame:
X   Y     1.3546980e+01 -2.6503493e+00  INCH

INERTIA with respect to _CC coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.1630943e+00  4.4523451e+00
Iyx Iyy  4.4523451e+00  2.2698836e+01

POLAR MOMENT OF INERTIA:  2.3861930e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _CC coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.9481354e-01  1.4220231e-02
Iyx Iyy  1.4220231e-02  1.3830926e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.3113090e-02  2.9553137e-01

POLAR MOMENT OF INERTIA:  3.0864446e-01 INCH^4
```

```
ROTATION MATRIX from _CC orientation to PRINCIPAL AXES:
      -0.05042       -0.99873
       0.99873       -0.05042

ROTATION ANGLE from _CC orientation to PRINCIPAL AXES (degrees):
about z axis         92.890

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.2570585e-01  1.5462327e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                        1           2   COORD
about AXIS 1:  2.75397e-02 INCH^3       -1.4539e-01 -4.7615e-01   INCH
               3.60843e-02 INCH^3       -7.6710e-01  3.6340e-01   INCH
about AXIS 2:  1.24792e-01 INCH^3       -2.3682e+00 -3.1282e-01   INCH
               1.10996e-01 INCH^3        2.6626e+00  6.4792e-02   INCH
```

<u>END PRO-E DATA</u>

The lift force on one wing assembly at 100*mph* (assume C_{Lmax}=1.44) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(91.65 in^2\right) \cdot (1.44) \cdot (0.1776 psi)$$
$$F_{Lift} = 23.4\ lbf$$

Therefore the shear stress at section cut C-C is:

$$\tau_{C-C} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{C-C} = \frac{23.4 lbf}{0.1236 in^2}$$
$$\tau_{C-C} = 189.6 psi$$

The bending stress at section C-C due to the aero load is:

$$\sigma_{C-C} = \frac{M_{aeroload}}{z_{C-C}} = \frac{(23.4 lbf)(3.41 in)}{0.0275 in^3}$$
$$\sigma_{C-C} = 2902.2 psi$$

The maximum shear stress at section C-C due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{C-C}}{2}\right)^2 + \tau_{C-C}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{2902.2\,psi}{2}\right)^2 + (189.6\,psi)^2}$$

$$\tau_{Max} = 1463.5\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{17362\,psi}{1463.5\,psi} = 11.9}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ C-C}} = \frac{2902.2\,psi}{2} \pm \sqrt{\left(\frac{2902.2\,psi}{2}\right)^2 + (189.6\,psi)^2}$$

$$\sigma_{\substack{Max \\ C-C}} = 1451.1 + 1463.5$$

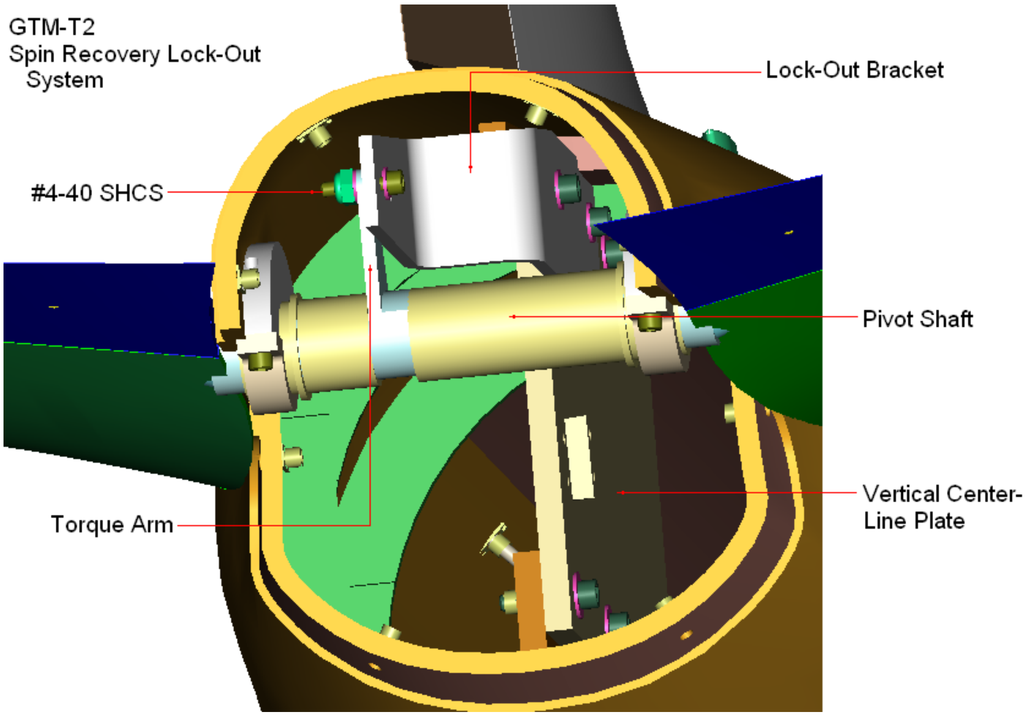
$$\sigma_{\substack{Max \\ C-C}} = 2914.6\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{17362\,psi}{2914.6\,psi} = 6.0}$$

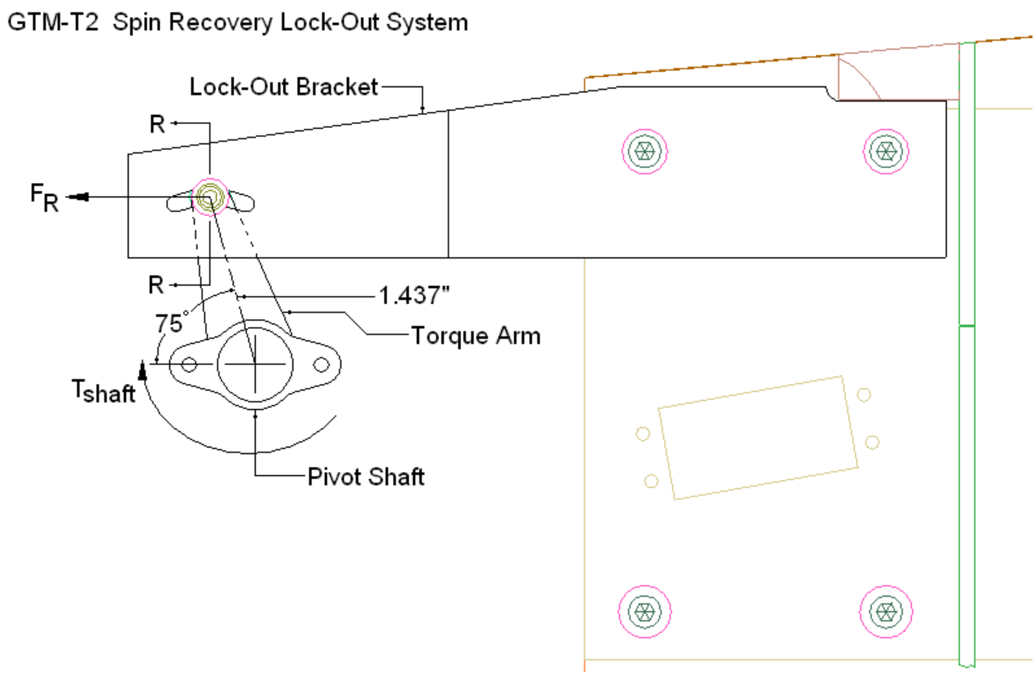
Spin Recovery Lock-Out System:

The spin recovery system is locked out using a "dog-legged" angled bracket. This bracket experiences the aero-loads generated from the horizontal tails. The bracket as shown below:



Drawing reference no. LD-1164951

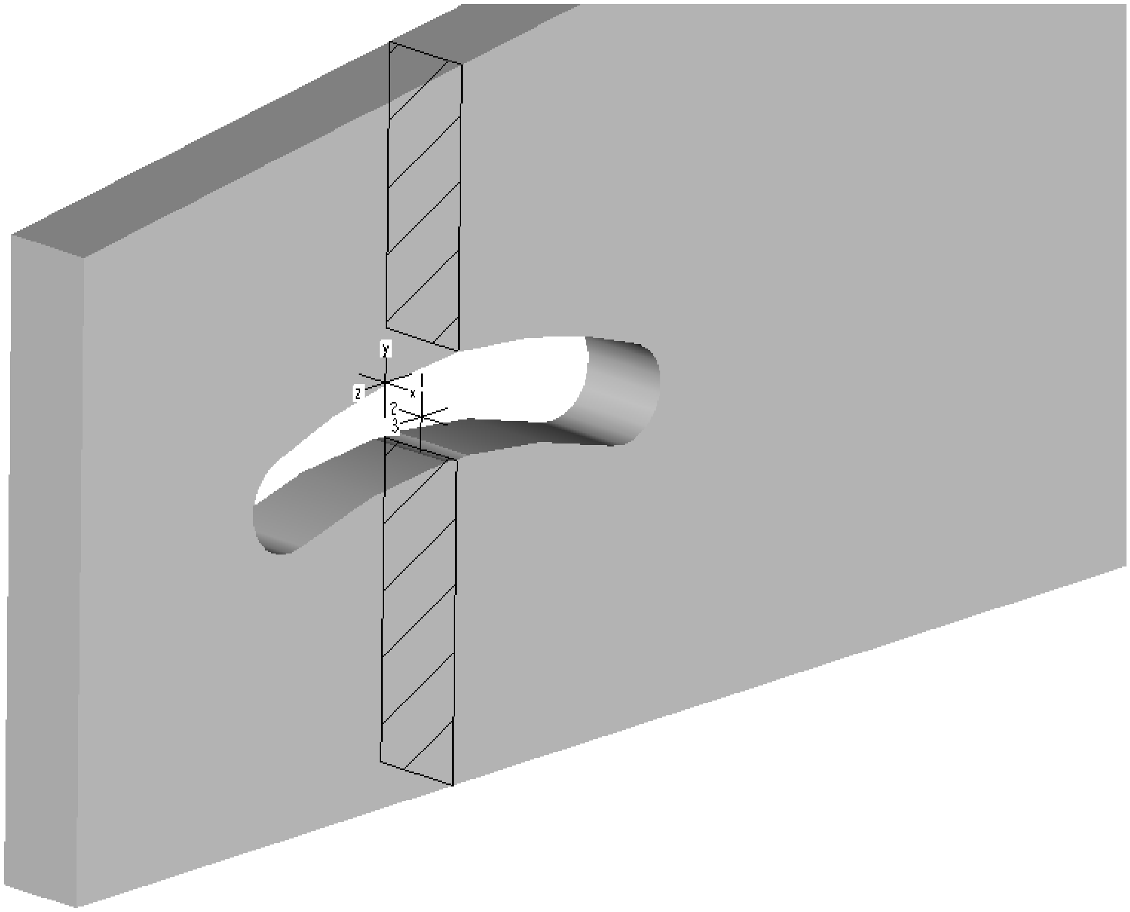
A side view is provided below with the appropriate dimensions for the analysis:



Drawing reference LD-1164951

And Section R-R is shown below:

GTM-T2   Section R-R, Spin Recovery Lock-Out Bracket



Drawing reference LD-1164920

And Section R-R has the following Pro-Engineer Properties:

<u>START PRO-E DATA</u>

```
AREA =  1.0067950e-01 INCH^2

CENTER OF GRAVITY with respect to _R coordinate frame:
X   Y     6.2500000e-02 -2.9802040e-02  INCH

INERTIA with respect to _R coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  8.9598462e-03  1.8752841e-04
Iyx Iyy  1.8752841e-04  5.2437240e-04

POLAR MOMENT OF INERTIA:  9.4842186e-03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _R coordinate frame:
(INCH^4)
```

```
INERTIA TENSOR:
Ixx Ixy  8.8704265e-03  0.0000000e+00
Iyx Iyy  0.0000000e+00  1.3109310e-04

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.3109310e-04  8.8704265e-03

POLAR MOMENT OF INERTIA:  9.0015196e-03 INCH^4

ROTATION MATRIX from _R orientation to PRINCIPAL AXES:
       0.00000       -1.00000
       1.00000        0.00000

ROTATION ANGLE from _R orientation to PRINCIPAL AXES (degrees):
about z axis          90.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.6084392e-02  2.9682585e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                  1          2   COORD
about AXIS 1:  2.09749e-03 INCH^3       -4.2198e-02 -6.2500e-02   INCH
               2.09749e-03 INCH^3       -4.2198e-02  6.2500e-02   INCH
about AXIS 2:  1.88653e-02 INCH^3       -4.7020e-01 -6.2500e-02   INCH
               1.85094e-02 INCH^3        4.7924e-01 -6.2500e-02   INCH
```

END PRO-E DATA

The maximum torque on the pivot shaft is assumed to be 19.52*in-lbs* (see Horizontal Assembly Analysis p. 21). So depending on a clockwise shaft rotation or a counter-clockwise shaft rotation the lock-out bracket is in tension or compression (respectively). From the side view sketch this force is,

$$T_{shaft} = F_R\, l$$

$$T_{shaft} = F_R\left(\left(1.437in\right)\sin 75°\right)$$

$$19.52in - lbs = F_R(1.388in)$$

$$therefore,$$

$$F_R = 14.1lbs$$

Bracket Tensile Stress at Section R-R:

Using a tensile load of 14.1*lbs* then the stress is,

$$\sigma_{R-R} = \frac{F_R}{Area} = \frac{14.1lbs}{0.10in^2}$$

$$\sigma_{R-R} = 139.7\ psi$$

Then the factor of safety based on the yield strength of the 6061-T6 aluminum is,

$$\boxed{FS = \frac{42000\,psi}{139.7\,psi} = 300.7}$$

<u>Pivot Arm Extension Lock-Down Bolt, Static Friction Analysis:</u>
The lock-out bracket attaches to the pivot arm extension via a #4-40 UNC SHCS and locking hex nut combination. The contact surfaces are metal to metal. The recommended torque for this nut & bolt combination is 16*in-lbs*, which produces a 630*lb* tensile load in the bolt (reference page 32, Holo-Krome Technical Handbook, P/N 99004 10M HU, revised 12/90, © 1990). And the equation for a static friction analysis is (reference equation 5-14, page 80, Fundamentals of Physics, Halliday & Resnick, revised printing, © 1974),

$$f_S = \mu_S N$$

where,

$$f_S = \text{force of the static friction } (lbs.)$$

$$\mu_S = \text{coefficient of static friction}, \text{ (dimensionless)}$$

$$N = \text{normal force } (lbs.)$$

And, the coefficient of friction is given as a range between 0.15 and 0.60 (reference table 8.1, page 306, Statics & Dynamics, Beer & Johnston, 3rd Edition, © 1977). A coefficient of friction of 0.15 is assumed for a conservative analysis. Therefore,

$$f_S = \mu_S N$$

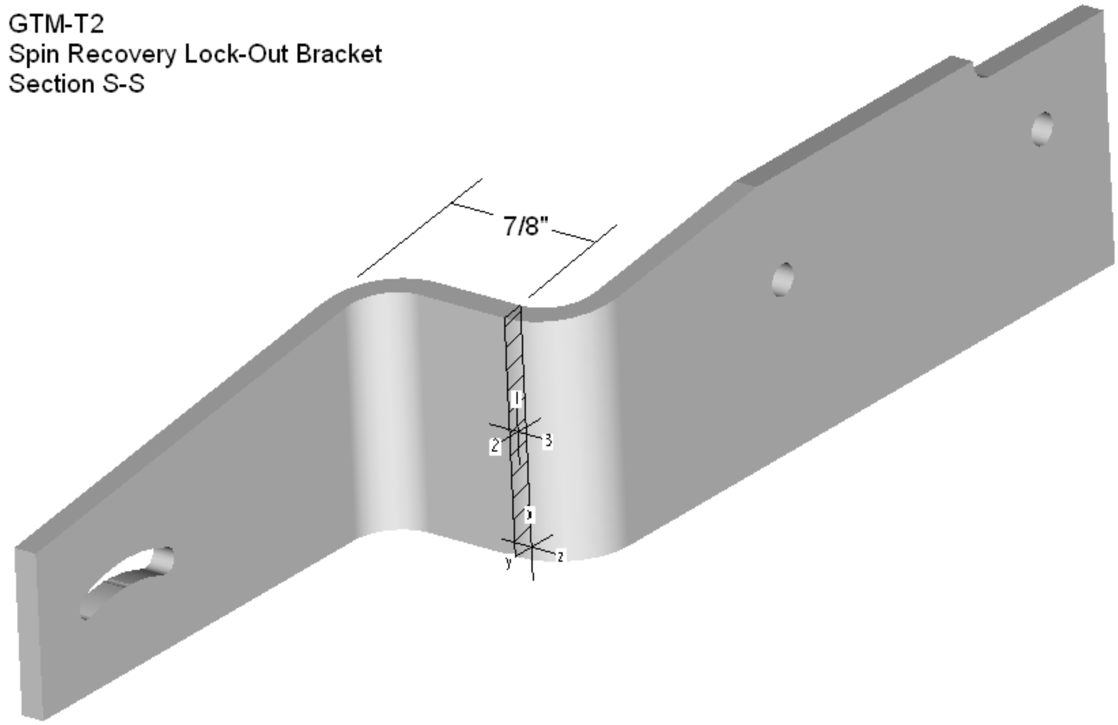
$$f_S = (0.15)(630lbs)$$

$$f_S = 94.5\ lbs$$

Since a force of 14.1*lbs* is applied to the connection then the factor of safety is,

$$\boxed{FS = \frac{94.5lbs}{14.1lbs} = 6.7}$$

Bracket Bending Stress at Section S-S:
The bracket is analyzed for bending at section S-S as shown below:



Drawing reference LD-1164920-6

The Pro-Engineer data for this section is as follows:

START PRO-E DATA

```
AREA =  1.5352248e-01 INCH^2

CENTER OF GRAVITY with respect to _S coordinate frame:
X   Y     6.1409981e-01  6.2355032e-02  INCH

INERTIA with respect to _S coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  7.9681425e-04 -5.8650486e-03
Iyx Iyy -5.8650486e-03  7.7196150e-02

POLAR MOMENT OF INERTIA:  7.7992964e-02 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _S coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.9989583e-04  1.3666844e-05
Iyx Iyy  1.3666844e-05  1.9299972e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.9988605e-04  1.9299982e-02
```

```
POLAR MOMENT OF INERTIA:  1.9499868e-02 INCH^4
ROTATION MATRIX from _S orientation to PRINCIPAL AXES:
       1.00000        0.00072
      -0.00072        1.00000

ROTATION ANGLE from _S orientation to PRINCIPAL AXES (degrees):
about z axis          0.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.6083218e-02  3.5456223e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                       1           2   COORD
about AXIS 1:  3.18318e-03 INCH^3       -6.1406e-01 -6.2794e-02   INCH
               3.16886e-03 INCH^3        6.0549e-01  6.3078e-02   INCH
about AXIS 2:  3.14258e-02 INCH^3       -6.1414e-01  6.2206e-02   INCH
               3.09955e-02 INCH^3        6.2267e-01 -6.1910e-02   INCH
```

END PRO-E DATA

From the above data the shear stress is,

$$\tau = \frac{F}{A}$$
$$\tau = \frac{14.1 lbs}{0.15 in^2}$$
$$\tau = 91.6\ psi$$

And the bending stress is,

$$\sigma = \frac{M}{z} = \frac{Fl}{z}$$
where,
$$F = 14.1 lbs$$
$$l = 0.875 in$$
$$z = 0.0031 in^3 \ (\text{Pro} - E\ data\ above)$$
$$\therefore \sigma = \frac{(14.1 lbs)(0.875 in)}{0.0031 in^3}$$
$$\sigma = 3883.2\, psi$$

And combining this for a total stress gives,

$$\sigma_{combined} = \frac{3883.2\,psi}{2} \pm \sqrt{\left(\frac{3883.2\,psi}{2}\right)^2 + (91.6\,psi)^2}$$
$$\sigma_{combined} = 1941.6 + 1943.7$$
$$\sigma_{combined} = 3885.3\,psi$$

Then the factor of safety based on the yield strength of the 6061-T6 aluminum is,

$$\boxed{FS = \frac{42000\,psi}{3885.3\,psi} = 10.8}$$

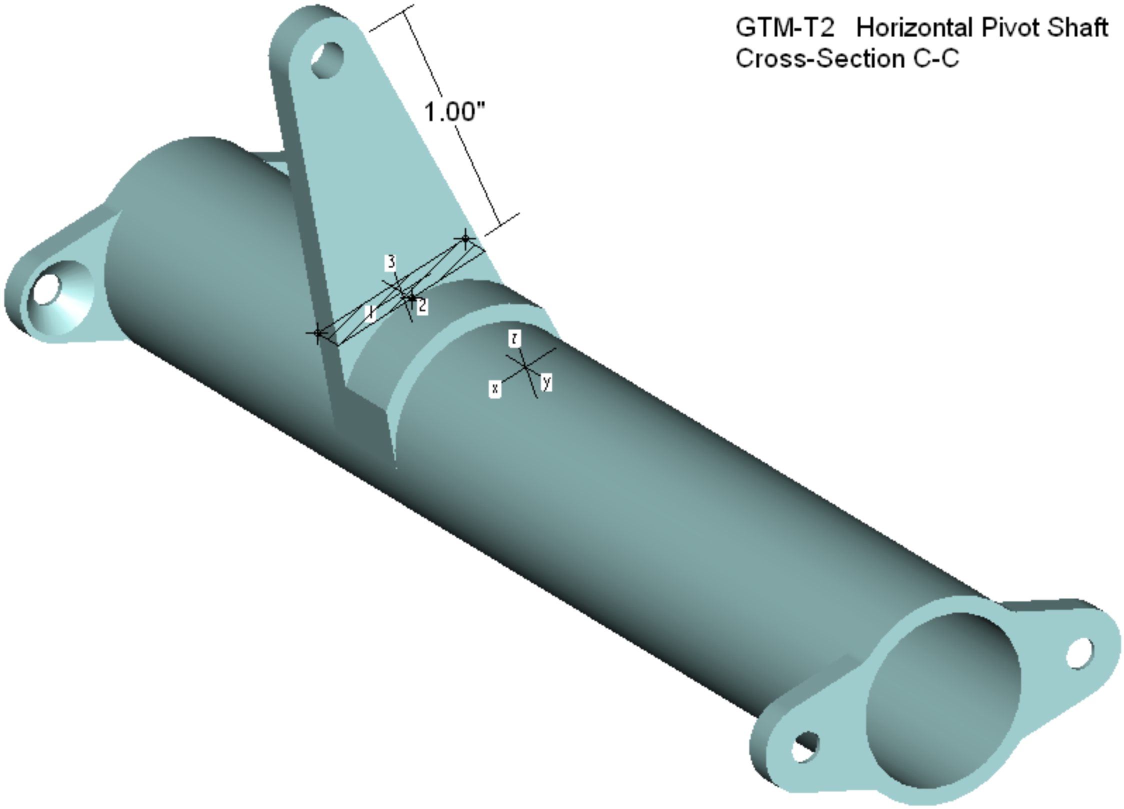
<u>Bracket Bolt Shear:</u>
The bracket attaches to the pivot shaft arm via a #4-40 UNC SHCS (see previous sketches). The bolt is in single shear when the pivot shaft rotates clockwise or counter-clockwise. The factor of safety based on the shear stress of the #4-40 UNC SHCS in the threaded area is:

$$\boxed{FS = \frac{665lbs\ (\sin gle\ \ shear)}{14.1lbs} = 47.3}$$

<u>Lock-Out Bracket Bolt Shear Analysis:</u>
The Lock-Out Bracket attaches to the vertical bulkhead with two #8-32 UNC SHCS with hex nuts. The bracket places a shear load on the bolts. It is assumed that only one bolt experiences the shear loading. So, the factor of safety based on a tensile strength of 1535*lbs* for a #8-32 UNC SHCS, 1960 series, alloy steel construction, is:

$$\boxed{FS = \frac{1535\,lbs}{14.1\,lbs} = 109.2}$$

<u>The Combined Stress at Section C-C on the Horizontal Pivot Shaft:</u>
Section C-C is analyzed for a combined stress due to shear and bending shown below,



Drawing reference LD-1164920

The Pro-Engineer data for section C-C is as follows:

<u>START PRO-E DATA</u>

```
AREA =  7.8014557e-02 INCH^2

CENTER OF GRAVITY with respect to _C coordinate frame:
X   Y     0.0000000e+00 -7.6850000e-01  INCH

INERTIA with respect to _C coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.6176374e-02  0.0000000e+00
Iyx Iyy  0.0000000e+00  2.5323613e-03

POLAR MOMENT OF INERTIA:  4.8708735e-02 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _C coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.0158145e-04  0.0000000e+00
Iyx Iyy  0.0000000e+00  2.5323613e-03
```

```
AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.0158145e-04  2.5323613e-03

POLAR MOMENT OF INERTIA:  2.6339427e-03 INCH^4

ROTATION MATRIX from _C orientation to PRINCIPAL AXES:
       1.00000        0.00000
       0.00000        1.00000

ROTATION ANGLE from _C orientation to PRINCIPAL AXES (degrees):
about z axis          0.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.6084392e-02  1.8016690e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  1.62530e-03 INCH^3       -3.1206e-01 -6.2500e-02   INCH
               1.62530e-03 INCH^3       -7.8586e-20  6.2500e-02   INCH
about AXIS 2:  8.11503e-03 INCH^3       -3.1206e-01 -6.2500e-02   INCH
               8.11503e-03 INCH^3        3.1206e-01 -6.2500e-02   INCH
```

<u>END PRO-E DATA</u>

The shear stress at section C-C is,

$$\tau = \frac{F}{A_{shear}} = \frac{14.1 lbs}{0.078 in^2}$$
$$\tau = 180.3\, psi$$

And, the bending stress at section C-C is,

$$\sigma = \frac{M}{z} = \frac{Fl}{z}$$
$$where,$$
$$F = 14.1 lbs$$
$$l = 1.00\, in$$
$$z = 0.0016\, in^3 \ (\mathrm{Pro} - E \ data \ above)$$
$$\therefore \sigma = \frac{(14.1 lbs)(1 in)}{0.0016 in^3}$$
$$\sigma = 8652.6\, psi$$

The combined stress at section C-C is,

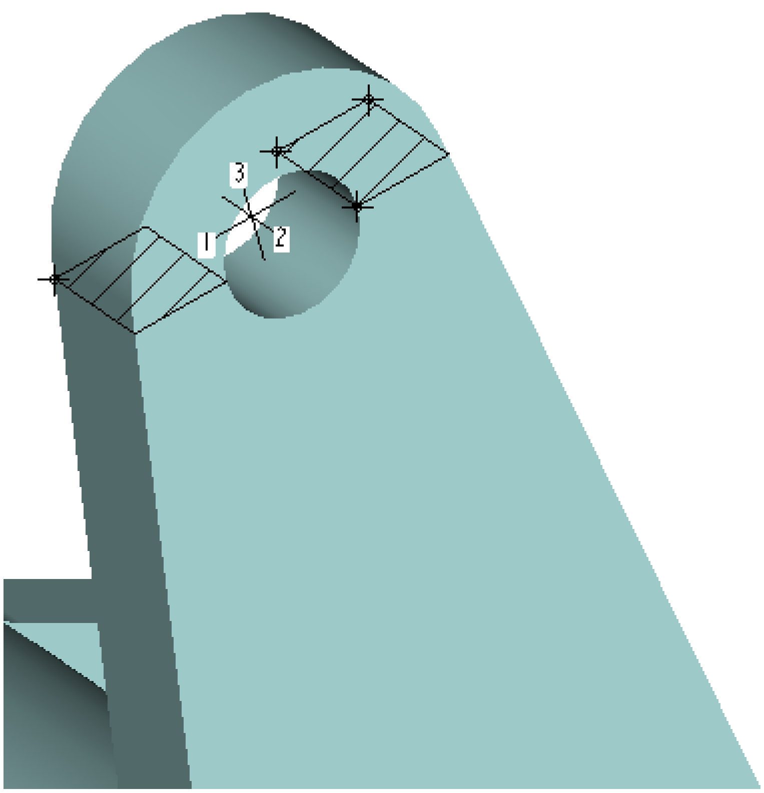
$$\sigma_{combined} = \frac{8652.6\,psi}{2} \pm \sqrt{\left(\frac{8652.6\,psi}{2}\right)^2 + (180.3\,psi)^2}$$
$$\sigma_{combined} = 4326.3 + 4330.0$$
$$\sigma_{combined} = 8656.3\,psi$$

Since the material is 7075-T651 aluminum with an ultimate strength of 62ksi, the factor of safety is,

$$\boxed{FS = \frac{62000\,psi}{8656.3\,psi} = 7.2}$$

Pivot shaft arm extension, shear tear-out:
For this analysis the shaft pivot arm is assumed to shear (tear-out) fail at section D-D shown below:

GTM-T2 Horizontal Pivot Shaft, Cross-Section D-D



Drawing reference LD-1164920

And the Pro-Engineer data is as follows:

START PRO-E DATA

```
AREA =  2.3022793e-02 INCH^2

CENTER OF GRAVITY with respect to _D coordinate frame:
X   Y     0.0000000e+00 -7.6850000e-01  INCH

INERTIA with respect to _D coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.3627061e-02  0.0000000e+00
Iyx Iyy  0.0000000e+00  3.0530400e-04

POLAR MOMENT OF INERTIA:  1.3932365e-02 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _D coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.9977595e-05  0.0000000e+00
Iyx Iyy  0.0000000e+00  3.0530400e-04

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.9977595e-05  3.0530400e-04

POLAR MOMENT OF INERTIA:  3.3528160e-04 INCH^4

ROTATION MATRIX from _D orientation to PRINCIPAL AXES:
       1.00000        0.00000
       0.00000        1.00000

ROTATION ANGLE from _D orientation to PRINCIPAL AXES (degrees):
about z axis          0.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.6084392e-02  1.1515618e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                   1           2   COORD
about AXIS 1:  4.79642e-04 INCH^3       -6.6000e-02 -6.2500e-02   INCH
               4.79642e-04 INCH^3       -6.6000e-02  6.2500e-02   INCH
about AXIS 2:  1.93119e-03 INCH^3       -1.5809e-01 -6.2500e-02   INCH
               1.93119e-03 INCH^3        1.5809e-01 -6.2500e-02   INCH
```

END PRO-E DATA

Again the force is 14.1*lbs* and the shear area is,

$$A_{shear} = (0.5in)(0.023in) = 0.0115in^2$$

Then the shear tear – out stress is,

$$\tau = \frac{14.1lbs}{0.0115in^2}$$

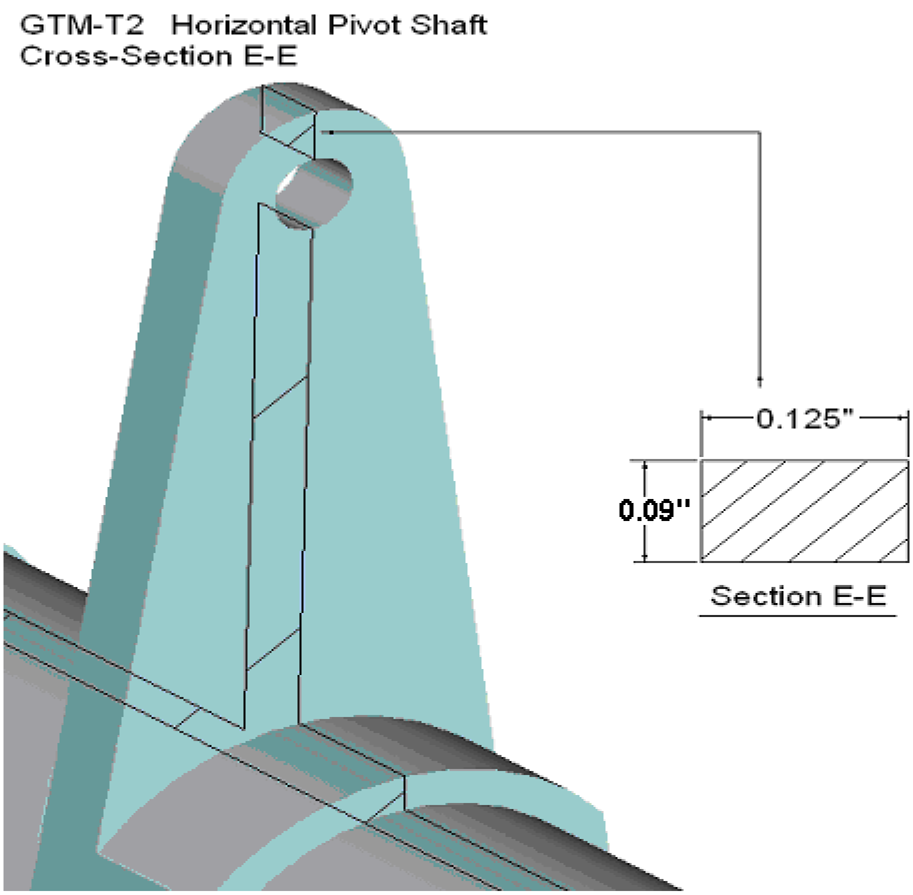
$$\tau = 1221.7\,psi$$

Since the material is 7075-T651 aluminum with an ultimate strength of 62*ksi*, the factor of safety is,

$$\boxed{FS = \frac{62000\,psi}{1221.7\,psi} = 50.8}$$

<u>Pivot Shaft Arm, Tensile Failure at the Top End:</u>

For this analysis the arm extension is assumed to tensile fail at section E-E shown below:



Drawing reference LD-1164920

Again, the force is 14.1*lbs* and the tensile area is,

$$A_{tensile} = (0.125in)(0.09in) = 0.0112in^2$$

Then the tensile stress is,

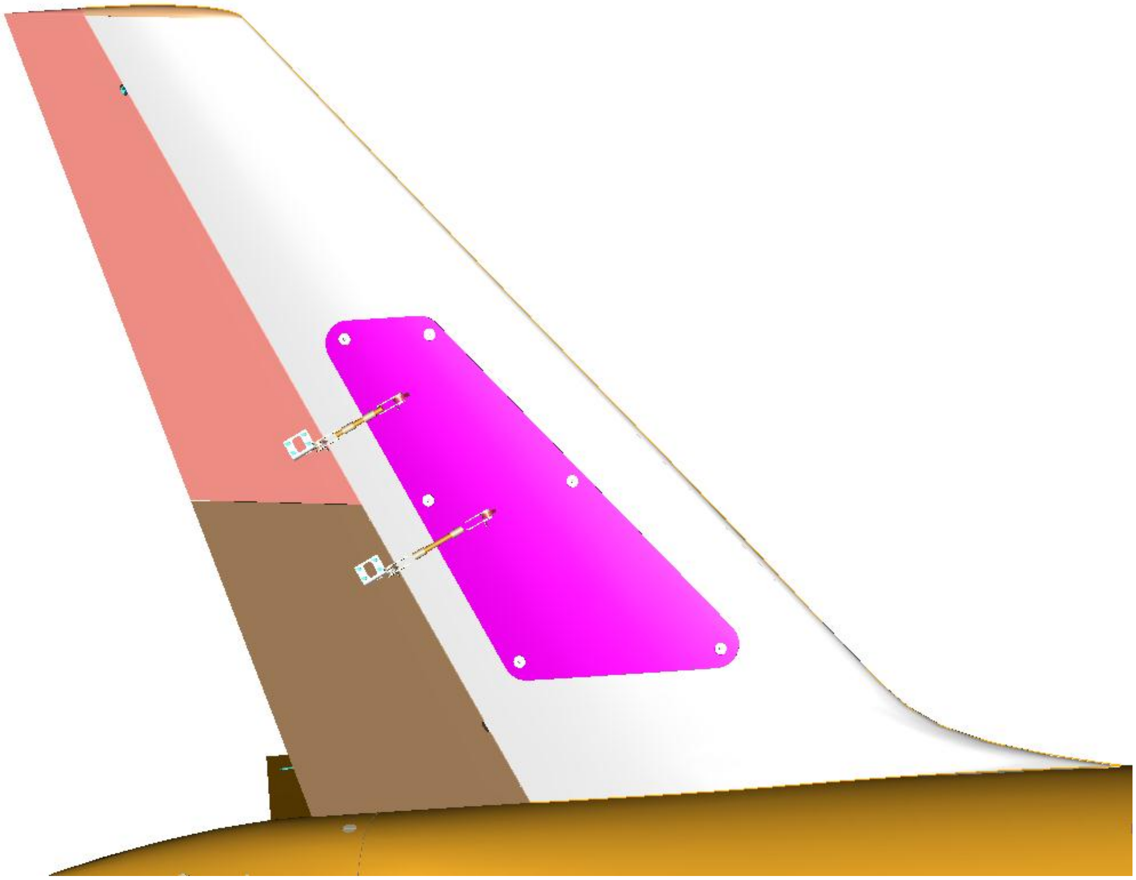
$$\sigma = \frac{14.1lbs}{0.0112in^2}$$

$$\sigma = 1250\,psi$$

Since the material is 7075-T651 aluminum with an ultimate strength of 62*ksi*, the factor of safety is,

$$\boxed{FS = \frac{62000\,psi}{1250\,psi} = 49.6}$$

Vertical Tail Analysis



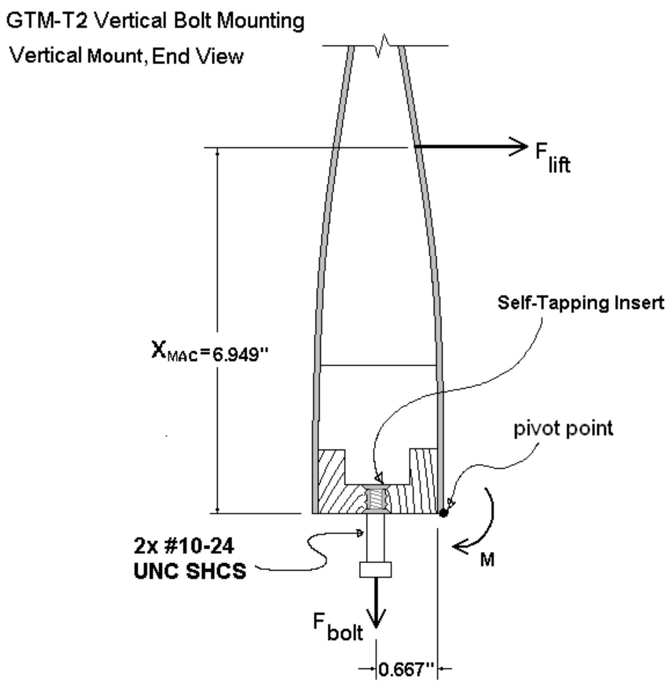
Drawing reference: LD-1164927 & LD-1164928

The vertical tail assembly is analyzed for bolt attachment loads and skin bending & shear stresses.

Aerodynamic Load

$$F_{Lift} = A \cdot C_L \cdot q$$
$$Assume,\ C_L = 1.61$$
$$F_{Lift} = (161.83in^2)(1.61)\left(0.17761\frac{lbf}{in^2}\right)$$
$$F_{Lift} = 46.28lbf$$

Wing Root



Drawing reference: LD-1164927

Moment about Pivot Point
$$M = F \cdot X_{mac}$$
$$M = (46.28lbf)(6.949in)$$
$$M = 321.6in \cdot lb$$

Force on Main Bolt

$$F_{\substack{bolt \\ Max}} = \frac{M \cdot r_{\max}}{\sum_{1}^{2} r_i^2} = \frac{321.6in \cdot lbs(0.667in)}{(0.667in)^2 + (0.667in)^2} = \frac{214.5lbs}{0.8898in^2}$$

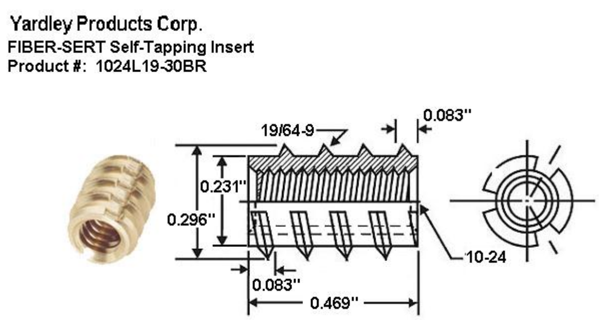
$$F_{\substack{bolt \\ Max}} = 241.1psi$$

The #10-24 UNC SHCS, alloy steel, has a 3150*psi* tensile strength, therefore, the factor of safety is:

$$\boxed{F.S. = \frac{3150\,psi}{241.1psi} = 13.1}$$

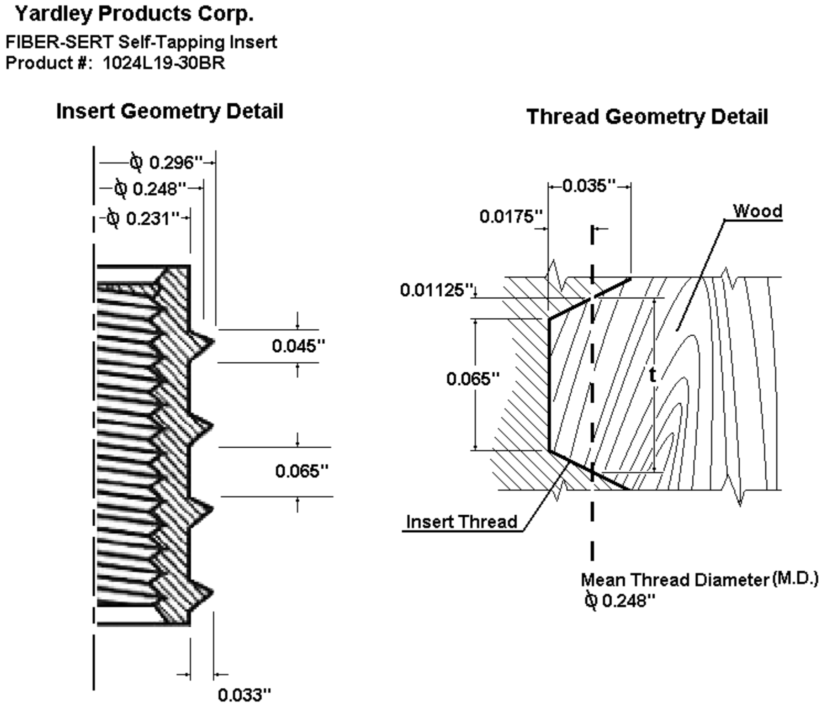
Self-Tapping Inserts Capacity

The vertical tail is attached to the fuselage by a two #10-24 machine screws, which threads into a self-tapping insert. The insert is mounted in the base of the tail, which is made of black cherry. It is assumed that the insert/wood interface is the weakest point of this assembly.



Drawing reference: LD-1164927

To estimate the inserts tension capacity, we assume that the wood at the mean diameter of the thread supports 100% of the shear load.



Drawing reference: LD-1164927

Area in shear

$t = 0.065in + 2(0.01125in) = 0.0875in$

$$A_1 = Circumference(t)$$
$$A_1 = \pi(M.D.)t$$
$$A_1 = \pi \cdot (0.248in) \cdot 0.0875in$$
$$A_1 = 0.06817in^2$$

$$A_s = \sum_{i=1}^{n} A_i \ \text{(where } n\text{, the number of thread raps = 4)}$$
$$A_s = 4(0.06817in^2)$$
$$A_s = 0.2727in^2$$

Maximum Insert Holding Force

The tensile strength of black cherry is 9050*psi* (reference project document number GTMP-6030).

$$\tau = \frac{F_{bolt_{Max}}}{A_{shear}} = \frac{241.1lbs}{0.2727in^2}$$

$$\tau = 884.1psi$$

$$\boxed{F.S. = \frac{9050\,psi}{884.1psi} = 10.2}$$

<u>Vertical Tail Shear Dowels</u>

Two ∅1/8*in* dowels are used to locate the vertical tail assembly and carry the shear loading. Only one dowel is assumed to carry the load.

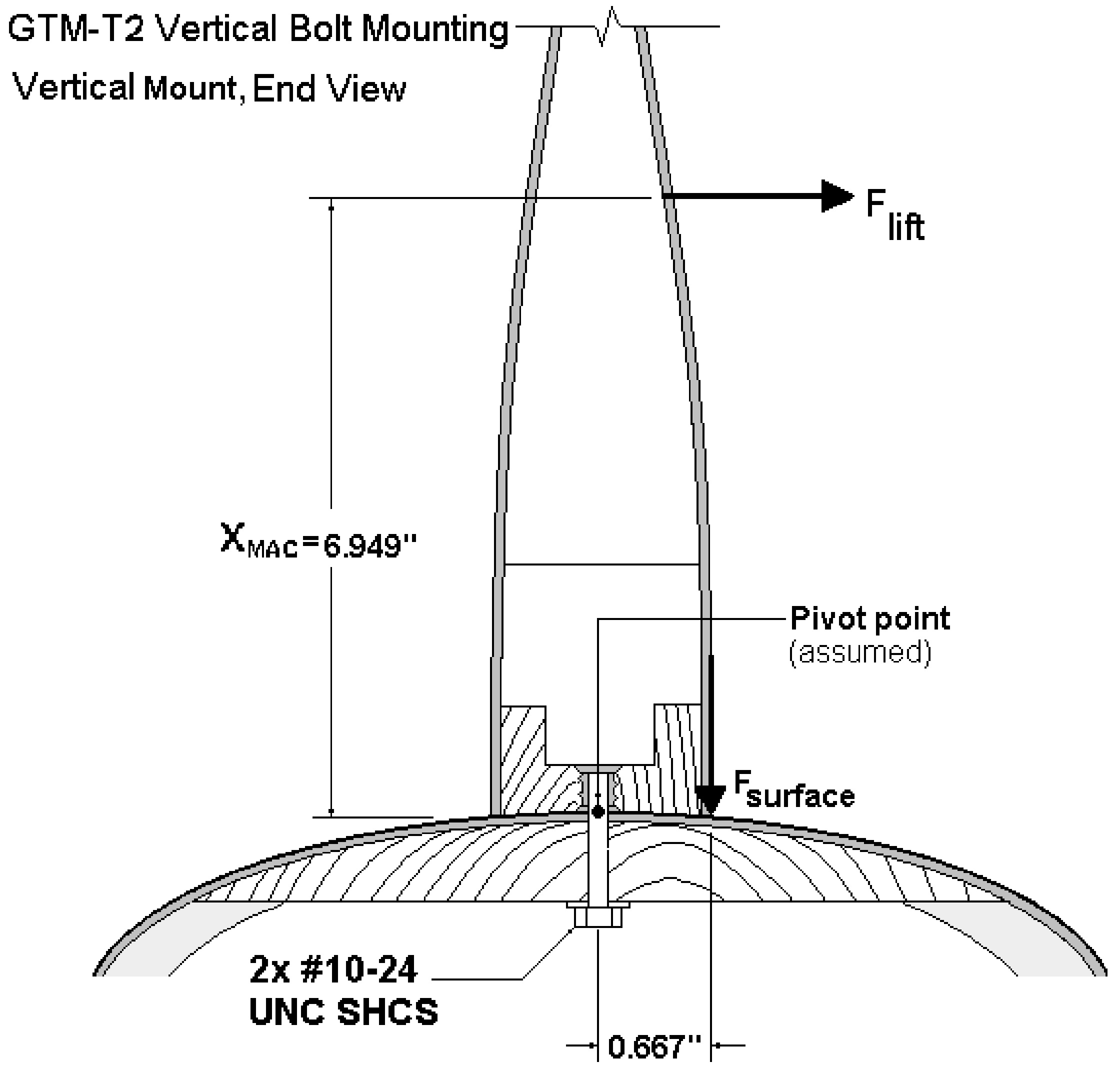
$F_{shear} = F_{aero} = 46.28lbf$

A ∅1/8in Holo-Krome dowel is rated for 1840*lbf* single shear, then the factor of safety is:

$$\boxed{F.S. = \frac{1840lbf}{46.28lbf} = 39.8}$$

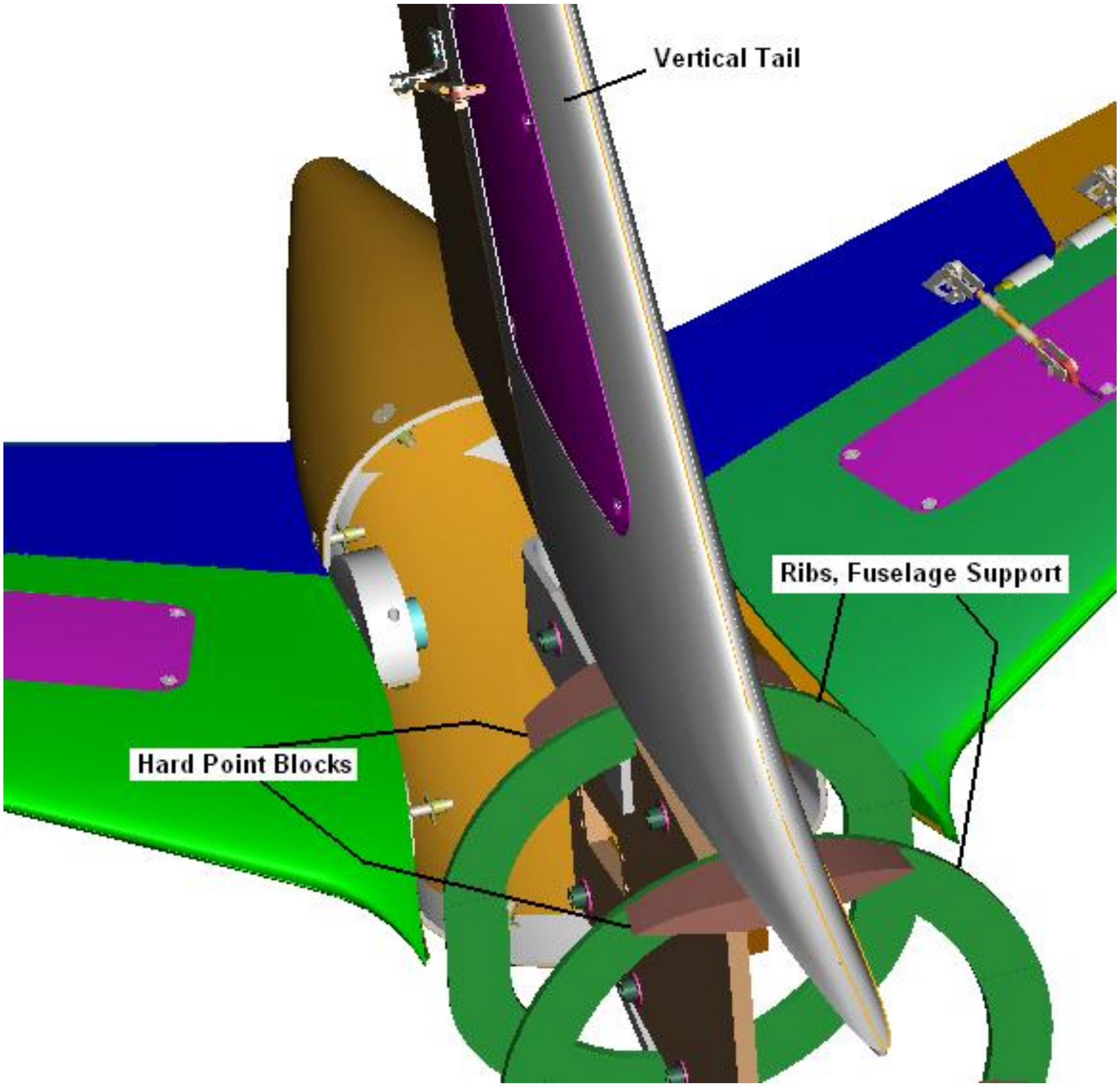
<u>Vertical Tail/Fuselage Edge (Bearing) Loading:</u>

The vertical stabilizer skin is constructed of carbon fiber and balsa and is 0.0725*in* thick. We make the conservative assumption that the skin of the vertical stabilizer carries the root load and transfers the entire moment load to the fuselage. It is also assumed that only 7.467 linear inches of the edge of the vertical tail root is in bearing loaded condition. Homogeneous material equations have been used in calculating a safety factor for the carbon fiber in bending.

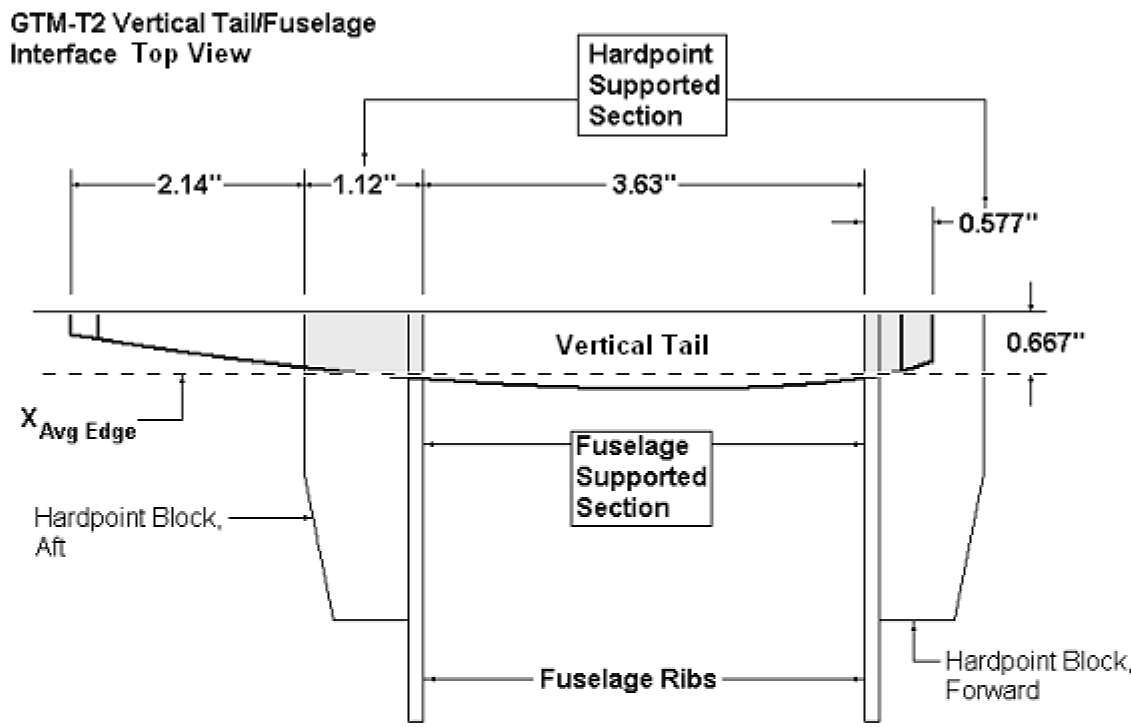


Drawing reference: LD-1164927

It is further assumed that the hard point blocks inside the fuselage bears 80% of the load exerted by the vertical skin edge, and that the unsupported fuselage skin carries the remaining 20%. From the Pro-E picture below the hard point and rib locations in relation to the vertical tail location are displayed.



Drawing reference: LD-1164927 & LD-1164928



Drawing reference: LD-1164927 & LD-1164928

For ease of calculation, we assume an average edge distance at which the entire load is concentrated.

Surface Edge Force

$$F_{\text{Vertical Edge}} = \frac{X_{MAC}(F_{lift})}{X_{Avg\ Edge}} = \frac{6.949in(46.28lbs)}{0.667in}$$

$$F_{\text{Vertical Edge}} = 482.1lbs$$

<u>Hard Point Block Bearing Stress:</u>

$$A_{Block} = 0.0725in(1.12in + 0.577in)$$

$$A_{Block} = 0.123in^2$$

$$F_{Block} = (0.80)F_{\text{Vertical Edge}} = 0.80(482.1lbs)$$

$$F_{Block} = 385.7lbs$$

$$\sigma_{Block} = \frac{F}{A} = \frac{385.7lbs}{0.123in^2}$$
$$\sigma_{Block} = 3134.9\,psi$$

The tensile strength of carbon fiber is 17362*psi* (reference project document no. GTMP-6030).

$$\boxed{FS = \frac{17362\,psi}{3134.9\,psi} = 5.5}$$

Fuselage Skin Bearing Stress:

$$A_{Fuselage} = (0.0725in)\cdot(3.63in)$$
$$A_{Fuselage} = 0.2632in^2$$

$$F_{Fuselage} = (0.20)F_{\text{Vertical Edge}} = 0.20(482.1lbs)$$
$$F_{Fuselage} = 96.4lbs$$

$$\sigma_{Block} = \frac{F}{A} = \frac{96.4lbs}{0.2632in^2}$$
$$\sigma_{Block} = 366.4\,psi$$

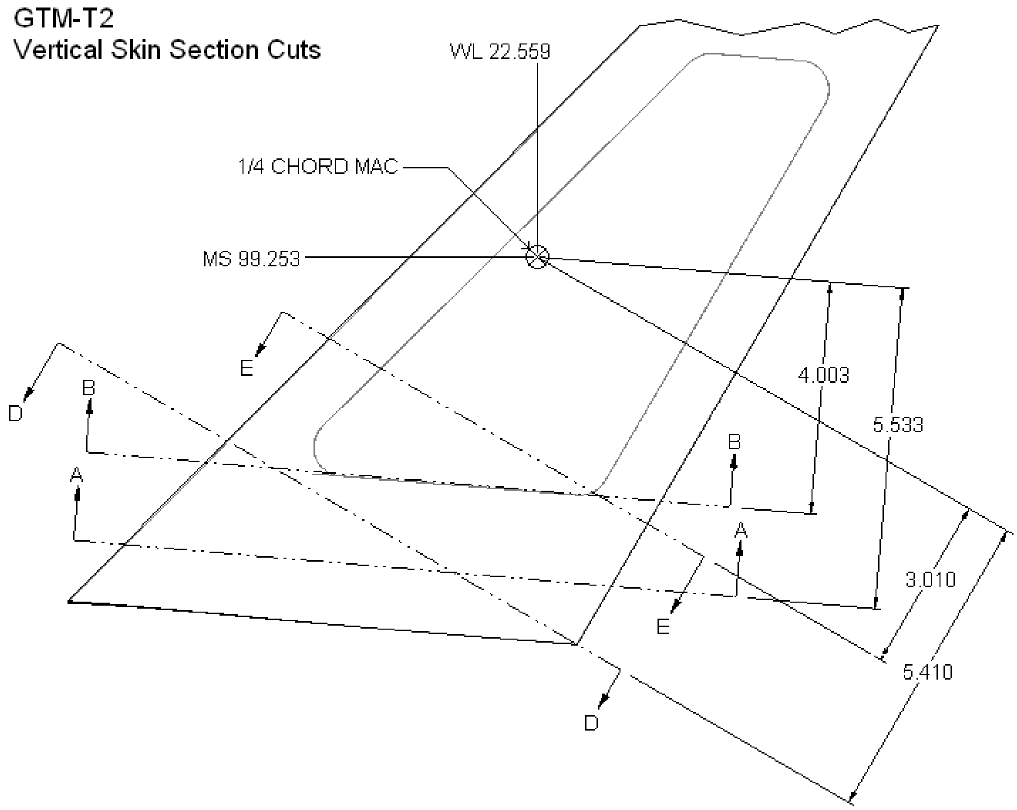
The tensile strength of carbon fiber is 17362*psi* (reference project document no. GTMP-6030).

$$\boxed{FS = \frac{17362\,psi}{366.4\,psi} = 47.4}$$

Vertical Tail Skin Analysis:

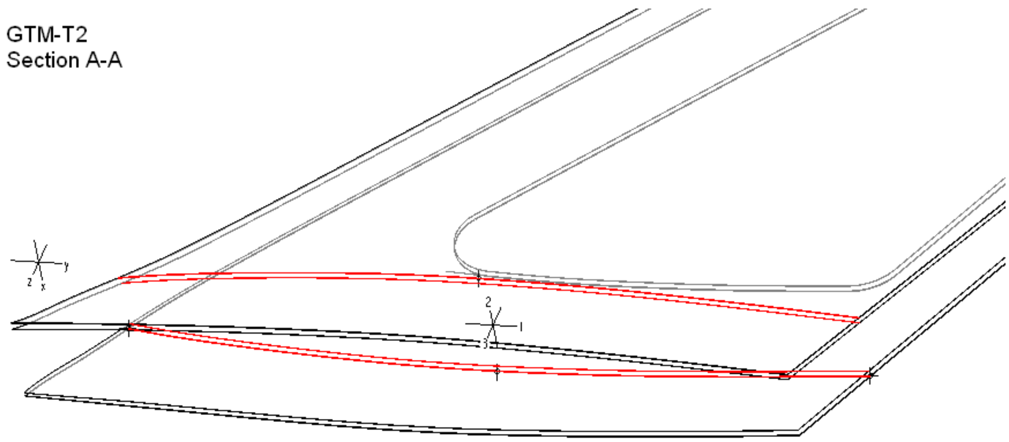
According to document GTMP-6030 a 0/90°, ±45°, ±45°, carbon fiber lay up has a tensile strength of 17362*psi*. The actual skin has a balsa core, which is assumed to add negligible strength to the overall assembly. Homogeneous material equations have been used in calculating a safety factor for the carbon fiber in bending.

It is assumed that the stress due to drag is negligible, and that the only significant loads are due to aerodynamic lift.



Drawing reference: LD-1164927

<u>Section AA</u>



Drawing reference: LD-1164927

START PRO-ENGINEER SECTION DATA

```
AREA =  2.5337968e-01 INCH^2

CENTER OF GRAVITY with respect to _AA coordinate frame:
X   Y     5.0197231e-01  5.0692090e+00  INCH

INERTIA with respect to _AA coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  8.0068914e+00 -6.4475085e-01
Iyx Iyy -6.4475085e-01  1.3403232e-01

POLAR MOMENT OF INERTIA:  8.1409237e+00 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _AA coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.4958243e+00  0.0000000e+00
Iyx Iyy  0.0000000e+00  7.0186668e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  7.0186668e-02  1.4958243e+00

POLAR MOMENT OF INERTIA:  1.5660109e+00 INCH^4

ROTATION MATRIX from _AA orientation to PRINCIPAL AXES:
       0.00000       -1.00000
       1.00000        0.00000

ROTATION ANGLE from _AA orientation to PRINCIPAL AXES (degrees):
about z axis         90.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  5.2630976e-01  2.4297097e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  1.09975e-01 INCH^3       -6.0129e-02 -6.3821e-01   INCH
               1.09974e-01 INCH^3       -6.0142e-02  6.3821e-01   INCH
about AXIS 2:  3.56885e-01 INCH^3       -4.1913e+00 -3.4121e-01   INCH
               3.55050e-01 INCH^3        4.2130e+00 -3.9344e-01   INCH
```

END PRO-ENGINEER SECTION DATA

Bending Stress

$$M_{A-A} = F_{areo} \cdot X_{A-A}$$
$$M_{A-A} = (46.28 lbs) \cdot (5.533 in) = 256.0 in \cdot lbs$$

$$\sigma = \frac{M}{z} = \frac{256.0 in \cdot lbs}{0.109974 in^3}$$
$$\sigma = 2328.2 psi$$

Shear Stress

$$\tau = \frac{F_{total}}{A_{shear}}$$
$$\tau = \frac{46.28 lbs}{0.2534 in^2}$$
$$\tau = 182.6 psi$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\tau_{\max} = \pm\sqrt{\left(\frac{2328.2 psi}{2}\right)^2 + (182.6 psi)^2}$$
$$\tau_{\max} = 1178.4 psi$$

Combined Stress

$$\sigma_{combined} = \frac{2328.2 psi}{2} \pm \sqrt{\left(\frac{2328.2 psi}{2}\right)^2 + (182.6 psi)^2}$$
$$\sigma_{combined} = 1164.11 + 1178.4$$
$$\sigma_{combined} = 2342.5 psi$$

Factor of Safety

The factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

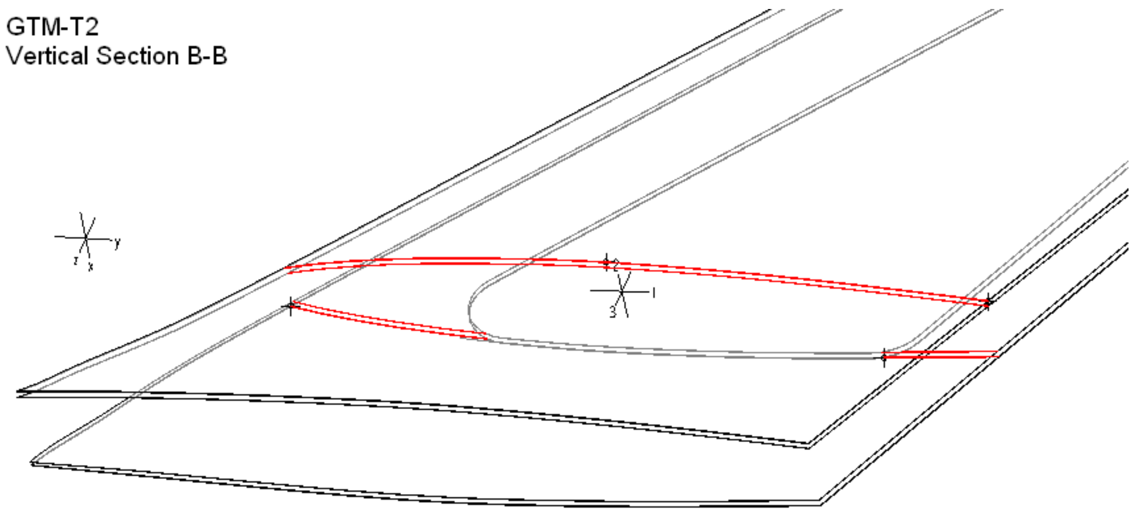
Shear stress:

$$\boxed{FS = \frac{17362 psi}{1178.4 psi} = 14.7}$$

Bending stress:

$$\boxed{FS = \frac{17362 psi}{2342.5 psi} = 7.4}$$

Section BB



Drawing reference: LD-1164927

START PRO-ENGINEER SECTION DATA

```
AREA =  1.6970587e-01 INCH^2

CENTER OF GRAVITY with respect to _BB coordinate frame:
X   Y     2.9010950e-01  5.9127713e+00  INCH

INERTIA with respect to _BB coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  7.0259120e+00 -2.7273411e-01
Iyx Iyy -2.7273411e-01  4.3319215e-02

POLAR MOMENT OF INERTIA:  7.0692313e+00 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _BB coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.0928482e+00  1.8371043e-02
Iyx Iyy  1.8371043e-02  2.9036171e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.8719015e-02  1.0931654e+00

POLAR MOMENT OF INERTIA:  1.1218844e+00 INCH^4

ROTATION MATRIX from _BB orientation to PRINCIPAL AXES:
      -0.01726       -0.99985
       0.99985       -0.01726

ROTATION ANGLE from _BB orientation to PRINCIPAL AXES (degrees):
about z axis          90.989
```

```
RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  4.1137353e-01  2.5380169e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                   1           2   COORD
about AXIS 1:  3.87922e-02 INCH^3       2.7850e+00 -7.4033e-01   INCH
               7.39479e-02 INCH^3      -1.0655e-01  3.8837e-01   INCH
about AXIS 2:  2.90497e-01 INCH^3      -3.7631e+00 -3.9538e-01   INCH
               2.66791e-01 INCH^3       4.0975e+00  8.6369e-02   INCH
```

END PRO-ENGINEER SECTION DATA

Bending Stress

$$M_{B-B} = F_{areo} \cdot X_{B-B}$$
$$M_{B-B} = (46.28lbs) \cdot (4.003in) = 185.3in \cdot lbs$$

$$\sigma = \frac{M}{z} = \frac{185.3in \cdot lbs}{0.03879in^3}$$
$$\sigma = 4775.2\,psi$$

Shear Stress

$$\tau = \frac{F_{total}}{A_{shear}}$$
$$\tau = \frac{46.28lbs}{0.1697in^2}$$
$$\tau = 272.7\,psi$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\tau_{\max} = \pm\sqrt{\left(\frac{4775.2\,psi}{2}\right)^2 + (272.7\,psi)^2}$$
$$\tau_{\max} = 2403.1psi$$

Combined Stress

$$\sigma_{combined} = \frac{4775.2\,psi}{2} \pm \sqrt{\left(\frac{4775.2\,psi}{2}\right)^2 + (272.7\,psi)^2}$$
$$\sigma_{combined} = 2387.6 + 2403.1$$
$$\sigma_{combined} = 4790.7\,psi$$

Factor of Safety
And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

Shear stress:

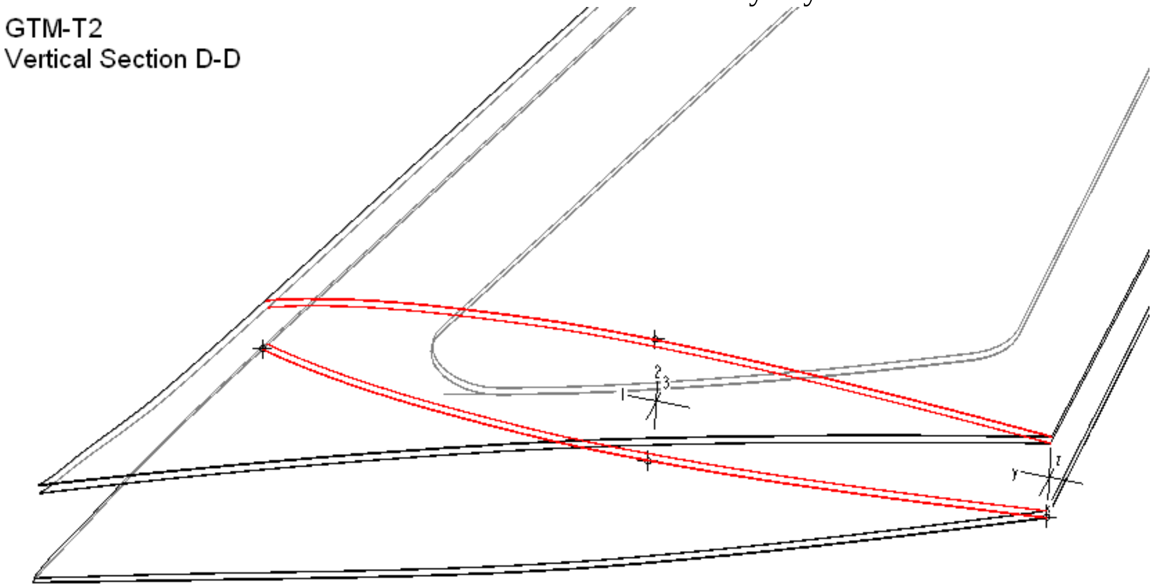
$$FS = \frac{17362\,psi}{2403.1\,psi} = 7.2$$

Bending stress:

$$FS = \frac{17362\,psi}{4790.7\,psi} = 3.6$$

Section DD

For conservatism it is assumed that the hatch does not carry any load.



Drawing reference: LD-1164927

START PRO-ENGINEER SECTION DATA

```
AREA =  2.1027807e-01 INCH^2

CENTER OF GRAVITY with respect to _DD coordinate frame:
X   Y     2.5643853e-04  3.4858668e+00  INCH

INERTIA with respect to _DD coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.4100577e+00 -2.6178732e-04
Iyx Iyy -2.6178732e-04  5.4968569e-02

POLAR MOMENT OF INERTIA:  3.4650263e+00 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _DD coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  8.5491262e-01 -7.3817534e-05
Iyx Iyy -7.3817534e-05  5.4968555e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  5.4968548e-02  8.5491262e-01

POLAR MOMENT OF INERTIA:  9.0988117e-01 INCH^4

ROTATION MATRIX from _DD orientation to PRINCIPAL AXES:
       0.00009       -1.00000
       1.00000        0.00009

ROTATION ANGLE from _DD orientation to PRINCIPAL AXES (degrees):
about z axis         89.995

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  5.1128157e-01  2.0163405e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  8.82820e-02 INCH^3        4.0221e-02 -6.2265e-01   INCH
               8.82081e-02 INCH^3        4.0268e-02  6.2317e-01   INCH
about AXIS 2:  2.45293e-01 INCH^3       -3.4853e+00 -4.1043e-01   INCH
               2.45842e-01 INCH^3        3.4775e+00 -2.4388e-01   INCH
```

END PRO-ENGINEER SECTION DATA

Bending Stress

$$M_{D-D} = F_{areo} \cdot X_{D-D}$$
$$M_{D-D} = (46.28lbs) \cdot (5.410in) = 250.4in \cdot lbs$$

$$\sigma = \frac{M}{z} = \frac{250.4 in \cdot lbs}{0.08821 in^3}$$
$$\sigma = 2838.2 psi$$

Shear Stress

$$\tau = \frac{F_{total}}{A_{shear}}$$
$$\tau = \frac{46.28 lbs}{0.2103 in^2}$$
$$\tau = 220.1 psi$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\tau_{\max} = \pm\sqrt{\left(\frac{2838.2 psi}{2}\right)^2 + (220.1 psi)^2}$$
$$\tau_{\max} = 1436.1 psi$$

Combined Stress

$$\sigma_{combined} = \frac{2838.2 psi}{2} \pm \sqrt{\left(\frac{2838.2 psi}{2}\right)^2 + (220.1 psi)^2}$$
$$\sigma_{combined} = 1419.1 + 1436.1$$
$$\sigma_{combined} = 2855.2 psi$$

Factor of Safety

And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

Shear stress:

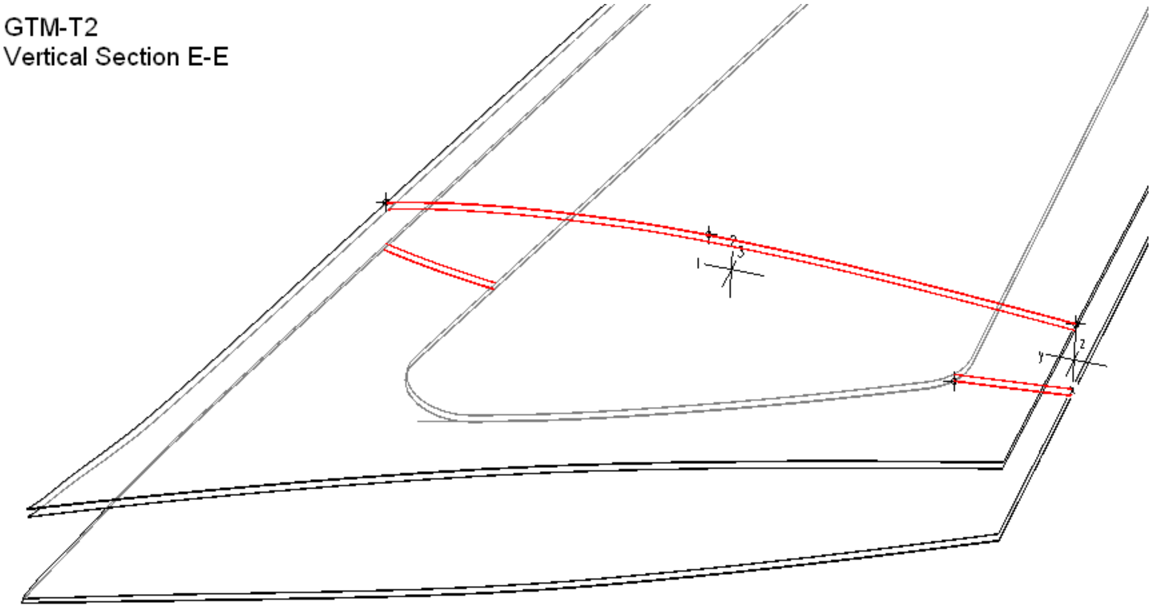
$$\boxed{FS = \frac{17362 psi}{1436.1 psi} = 12.1}$$

Bending stress:

$$\boxed{FS = \frac{17362 psi}{2855.2 psi} = 6.1}$$

Section E-E

GTM-T2
Vertical Section E-E



Drawing reference: LD-1164927

START PRO-ENGINEER SECTION DATA

```
AREA =  1.2746109e-01 INCH^2

CENTER OF GRAVITY with respect to _EE coordinate frame:
X   Y    -2.5131146e-01  3.1650127e+00  INCH

INERTIA with respect to _EE coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.8230068e+00  1.0231146e-01
Iyx Iyy  1.0231146e-01  2.5246119e-02

POLAR MOMENT OF INERTIA:  1.8482529e+00 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _EE coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  5.4619006e-01  9.2839977e-04
Iyx Iyy  9.2839977e-04  1.7196002e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.7194372e-02  5.4619169e-01

POLAR MOMENT OF INERTIA:  5.6338606e-01 INCH^4

ROTATION MATRIX from _EE orientation to PRINCIPAL AXES:
      -0.00176       -1.00000
       1.00000       -0.00176
```

```
ROTATION ANGLE from _EE orientation to PRINCIPAL AXES (degrees):
about z axis          90.101

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.6728598e-01  2.0700638e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  2.38799e-02 INCH^3       -2.0820e+00 -7.2004e-01   INCH
               5.39577e-02 INCH^3        2.2831e-01  3.1866e-01   INCH
about AXIS 2:  1.72614e-01 INCH^3       -3.1642e+00  1.2972e-01   INCH
               1.71988e-01 INCH^3        3.1758e+00 -3.5856e-03   INCH
```

END PRO-ENGINEER SECTION DATA

Bending Stress

$$M_{E-E} = F_{areo} \cdot X_{E-E}$$
$$M_{E-E} = (46.28lbs) \cdot (3.010in) = 139.3in \cdot lbs$$

$$\sigma = \frac{M}{z} = \frac{139.3in \cdot lbs}{0.02388in^3}$$
$$\sigma = 5832.9\,psi$$

Shear Stress

$$\tau = \frac{F_{total}}{A_{shear}}$$
$$\tau = \frac{46.28lbs}{0.1275in^2}$$
$$\tau = 363.1psi$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$
$$\tau_{\max} = \pm\sqrt{\left(\frac{5832.9\,psi}{2}\right)^2 + (363.1psi)^2}$$
$$\tau_{\max} = 2939.0\,psi$$

Combined Stress

$$\sigma_{combined} = \frac{5832.9\,psi}{2} \pm \sqrt{\left(\frac{5832.9\,psi}{2}\right)^2 + (363.1\,psi)^2}$$
$$\sigma_{combined} = 2916.5 + 2939.0$$
$$\sigma_{combined} = 5855.4\,psi$$

Factor of Safety
And, the factor of safety based on the carbon fiber yield strength, 17362*psi* (reference project document no. GTMP-6030) is,

Shear stress:

$$\boxed{FS = \frac{17362\,psi}{2939.0\,psi} = 5.9}$$

Bending stress:

$$\boxed{FS = \frac{17362\,psi}{5855.4\,psi} = 2.97}$$

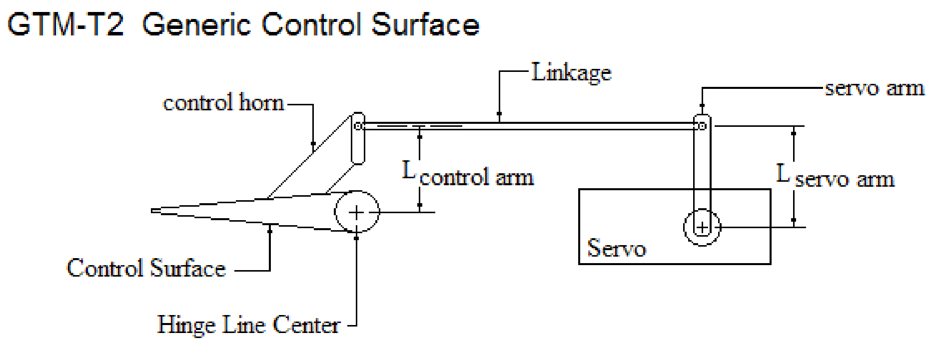
Control Surface Authority Analysis

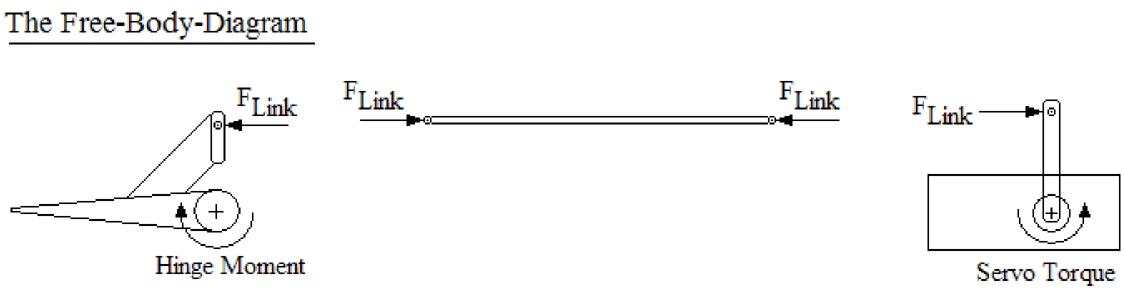
The control surface geometry and location for the GTM-T1 model are the same for the GTM-T2 vehicle. A control surface hinge moment equation is used to calculate the moment at the hinge line.

All the control surfaces are actuated via a direct linkage assembly where the actuating servo is located near the control surface's hinge line. The assembly consists of a #4-40 threaded rod, and "Sullivan" steel clevises and steel control horns.


A "Sullivan" steel clevis with retainer clip and a steel control horn.

A generic control surface "free-body-diagram" of this general set-up is given below:





Drawing reference: LD-1164931

In the sketch above the angle between the linkage and the arms is assumed to be 90 degrees. The control surface hinge moment calculations are done using the following equation (reference NACA TN 1400, August 1947):

$$H.M. = C_h \cdot q \cdot b \cdot \bar{c}^2$$

where,

$H.M. =$ *hinge moment* $(in - lbs)$

$C_h =$ *hinge moment coefficient*

$q =$ *dynamic pressure* (psi)

$b =$ *span* (in)

$\bar{c} =$ *MAC chord length* (in)

Since the above equation is used for an unswept surface the equation is modified as shown below for swept areas:

$$H.M. = C_h \cdot q \cdot S \cdot \bar{c}$$

where,

$H.M. = hinge\ moment\ \ (in - lbs)$

$C_h = hinge\ moment\ coefficient$

$q = dynamic\ \ pressure\ (psi)$

$S = control\ \ surface\ area\ \ (in^2)$

$\bar{c} = MAC\ chord\ length\ (in)$

And, this equation is similar, if not equal to, the "Ground Gust Loading" equation as presented on page 8.14 of the publication "Introduction to Aircraft Loads Analysis" by Paul Taylor, Mark Ray, Doug McKissack, and Larry Hanson, The Gulfstream Company, December 2003. And per the publication the hinge moment coefficient is usually 0.50 to 0.75 depending on the surface and deflection.

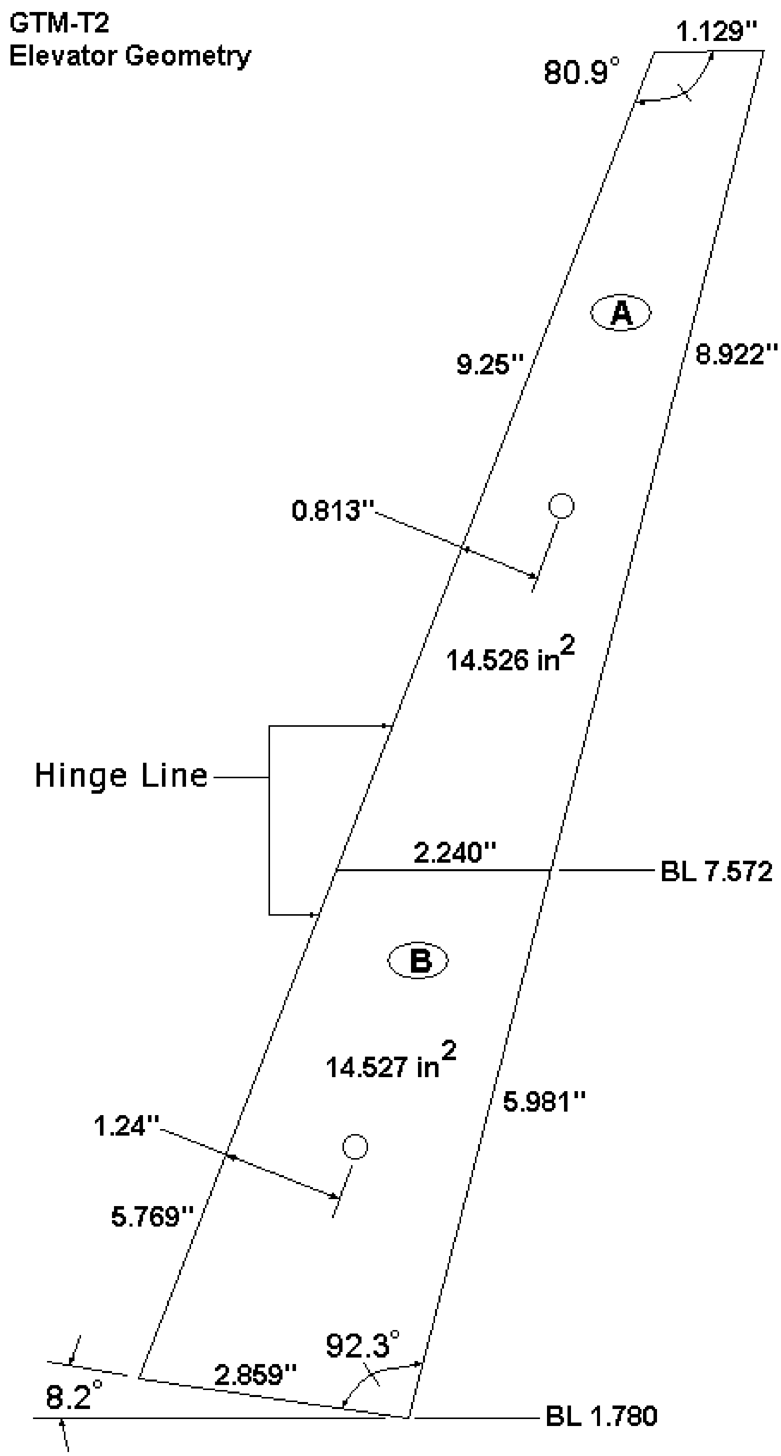
A spreadsheet is provided below detailing the servo types, their torques and the pertinent arm lengths.

	Servo	Servo Torque	Control Surface Horn Length	Servo Arm Length
Elevator-A	Volz "Wing-Maxx"	42 in-oz	0.65625	0.4375
Elevator-B	Volz "Wing-Maxx"	42 in-oz	0.65625	0.4375
Rudder-A	JR-3421	65 in-oz	0.65625	0.4375
Rudder-B	JR-8411	155 in-oz	0.65625	0.4375
Aileron	Volz "Wing-Maxx"	42 in-oz	0.65625	0.4375
Inboard Flap	JR-8411	155 in-oz	1.03125	0.78125
Outboard Flap	JR-8411	155 in-oz	1.03125	0.78125
Inboard Spoiler	JR-3421	65 in-oz	0.65625	0.4375
Outboard Spoiler	JR-3421	65 in-oz	0.65625	0.4375

The GTM-T2 control surface geometry spreadsheet

Split Elevator Servo Actuation Calculations:

The split elevator has two sections: area A, the outboard section, and area B, the inboard section. Each section is actuated by a "Volz Wing-Maxx" servo, which is located in the horizontal tail assembly just ahead of the hinge line. The manufacturer's rated torque is 42*in-oz*. The elevator geometry and centroid locations are shown below:



Drawing Reference: LD-1164939

Elevator Section A (outboard):

The elevator hinge moment is,

$$H.M._{elevator} = C_{h\_elevator} \cdot q \cdot S_{elevator} \cdot \bar{c}_{elevator}$$

$$where,$$

$$C_{h\_elevator} = 0.49 \text{ (At } 45°, \text{Ref GTMP} - 4039)$$

$$S_{elevator} = 14.526 in^2$$

And,

$$MAC = \frac{2}{3}\left(C_{R-A} + C_{T-A} - \frac{C_{R-A} C_{T-A}}{C_{R-A} + C_{T-A}} \right)$$

$$where,$$

$$C_{R-A} = wing\ root\ chord$$

$$C_{T-A} = wing\ tip\ chord$$

$$MAC = \frac{2}{3}\left(2.24in + 1.129in - \frac{(2.24in)(1.129in)}{(2.24in + 1.129in)} \right)$$

$$MAC = 1.75in$$

$$\bar{c}_{A-elevator} = 1.75in$$

$$\therefore H.M._{elevator} = 0.49 \cdot (0.17761 psi)(14.526 in^2)(1.75in)$$

$$H.M._{elevator} = 2.21in - lbs \Rightarrow 35.3in - oz$$

The elevator control horn offset is 0.656*in* and the servo arm length is 0.437*in* therefore, the servo torque is,

$$T = Fd$$

$$35.3in - oz = F(0.656in)$$

$$F_{elevator\_horn} = 53.8oz$$

$$And,$$

$$T_{elevator\_servo} = (53.8oz)(0.437in)$$

$$T_{elevator\_servo} = 23.5in - oz$$

The hinge moment calculation, using the coefficient 0.49, was determined to be 23.5*in-oz*. Since the actuation system is capable of an elevator hinge moment of 35.3*in-oz*. then this system is assumed to be valid.

Elevator Section B (inboard):

The elevator hinge moment is,

$$H.M._{elevator} = C_{h\_elevator} \cdot q \cdot S_{elevator} \cdot \bar{c}_{elevator}$$
$$where,$$
$$C_{h\_elevator} = 0.49 \text{ (At } 45°, \text{Ref GTMP} - 4039)$$
$$S_{elevator} = 14.527 in^2$$

And,

$$MAC = \frac{2}{3}\left(C_{R-B} + C_{T-B} - \frac{C_{R-B} C_{T-B}}{C_{R-B} + C_{T-B}} \right)$$
$$where,$$
$$C_{R-B} = wing \ root \ chord$$
$$C_{T-B} = wing \ tip \ chord$$
$$MAC = \frac{2}{3}\left(2.859in + 2.24in - \frac{(2.859in)(2.24in)}{(2.859in + 2.24in)} \right)$$
$$MAC = 2.56in$$
$$\bar{c}_{B-elevator} = 2.56in$$

$$\therefore H.M._{elevator} = 0.49 \cdot (0.17761 psi)(14.527 in^2)(2.56in)$$
$$H.M._{elevator} = 3.24in - lbs \Rightarrow 51.8in - oz$$

The elevator control horn offset is 0.656*in* and the servo arm length is 0.437*in* therefore, the servo torque is,

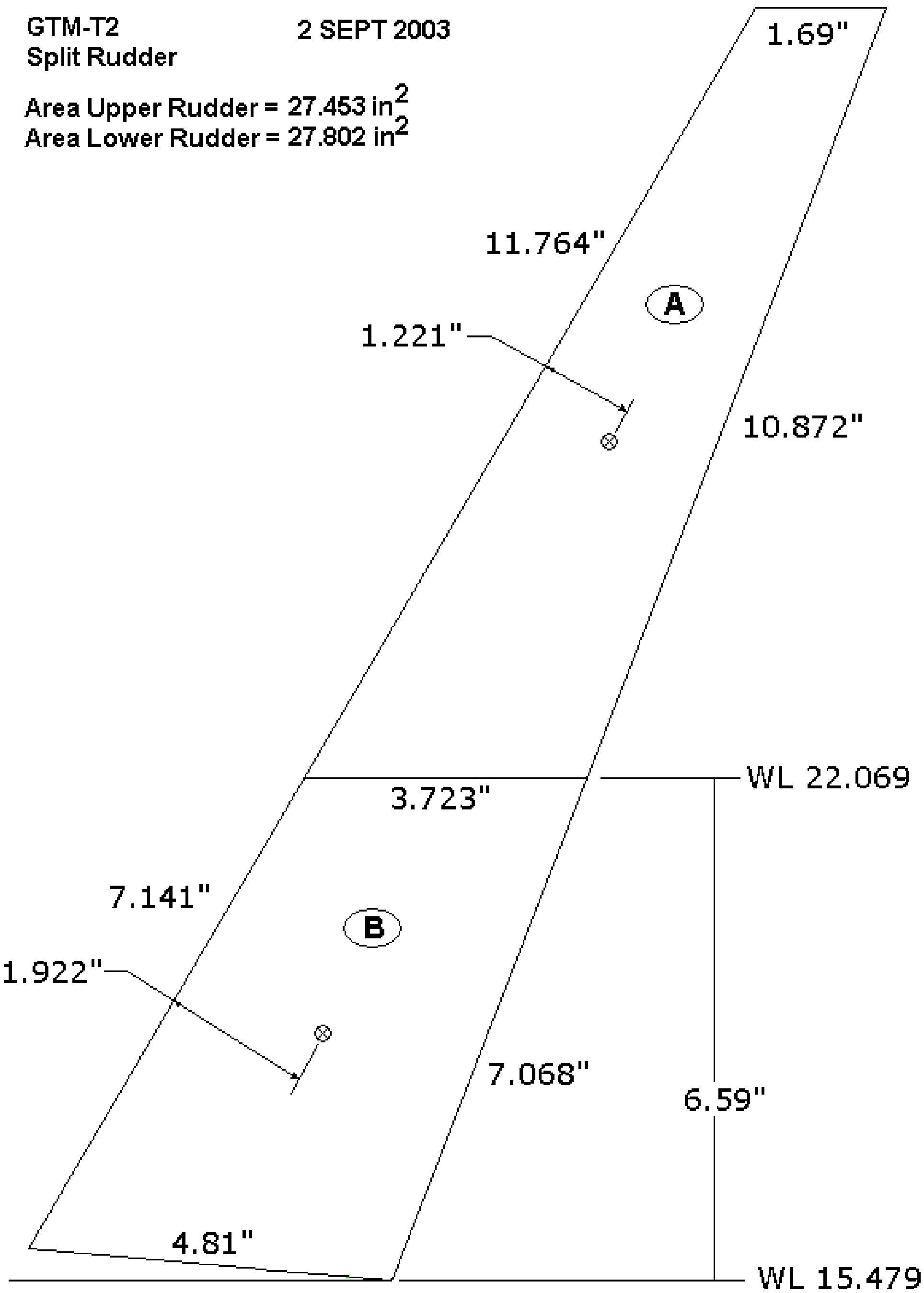
$$T = Fd$$
$$51.8in - oz = F(0.656in)$$
$$F_{elevator\_horn} = 79.0oz$$
$$And,$$
$$T_{elevator\_servo} = (79.0oz)(0.437in)$$
$$T_{elevator\_servo} = 34.5in - oz$$

The hinge moment calculation, using the coefficient 0.49, was determined to be 34.5*in-oz.* Since the actuation system is capable of an elevator hinge moment of 51.8*in-oz*. then this system is assumed to be valid.

Split Rudder Servo Actuation Calculation:

The split rudder has two sections: A and B. Section A is actuated by a "JR-3421" servo, and section B is actuated by a "JR-8411" servo. Both servos are located in the horizontal tail assembly just ahead of the hinge line. The manufacturer's rated torque for the"JR-

3421” is 65*in-oz*, and the rated torque for the“JR-8411” is 155*in-oz.* The rudder geometry and centroid locations are shown below:



Drawing references: LD-1164940

Rudder Section A:

The rudder hinge moment is,

$$H.M._{rudder} = C_{h\_rudder} \cdot q \cdot S_{rudder} \cdot \bar{c}_{rudder}$$

where,

$$C_{h\_rudder} = 0.35 \text{(Ref. GTMP} - 4039\text{)}$$

$$q = 0.17761 psi$$

$$S_{rudder} = 27.453 in^2$$

$$MAC = \frac{2}{3}\left(C_{R-A} + C_{T-A} - \frac{C_{R-A}C_{T-A}}{C_{R-A} + C_{T-A}} \right)$$

where,

$$C_{R-A} = wing\ root\ chord$$

$$C_{T-A} = wing\ tip\ chord$$

$$MAC = \frac{2}{3}\left(3.723in + 1.69in - \frac{(3.723in)(1.69in)}{(3.723in + 1.69in)} \right)$$

$$MAC = 2.83in$$

$$\bar{c}_{A-rudder} = 2.83in$$

$$\therefore H.M._{rudder} = 0.35 \cdot (0.17761psi)(27.453in^2)(2.83in)$$

$$H.M._{rudder} = 4.84in - lbs \Rightarrow 77.4in - oz$$

The rudder control horn offset is 0.656*in* and the servo arm length is 0.437*in* therefore, the servo torque is,

$$T = Fd$$

$$77.4in - oz = F(0.656in)$$

$$F_{rudder\_horn} = 117.9oz$$

And,

$$T_{rudder\_servo} = (117.9)(0.437)$$

$$T_{rudder\_servo} = 51.6in - oz$$

The hinge moment calculation, using the coefficient 0.35, was determined to be 51.6*in-oz.* Since the actuation system is capable of a rudder hinge moment of 77.4*in-oz*. then this system is assumed to be valid.

Rudder Section B:

The rudder hinge moment is,

$$H.M._{rudder} = C_{h\_rudder} \cdot q \cdot S_{rudder} \cdot \bar{c}_{rudder}$$

$$where,$$

$$C_{h\_rudder} = 0.35 \text{(Ref. GTMP}-4039)$$

$$q = 0.17761 psi$$

$$S_{rudder} = 27.802 in^2$$

$$MAC = \frac{2}{3}\left(C_{R-B} + C_{T-B} - \frac{C_{R-B} C_{T-B}}{C_{R-B} + C_{T-B}} \right)$$

$$where,$$

$$C_{R-B} = wing\ root\ chord$$

$$C_{T-B} = wing\ tip\ chord$$

$$MAC = \frac{2}{3}\left(4.81in + 3.723in - \frac{(4.81in)(3.723in)}{(4.81in + 3.723in)} \right)$$

$$MAC = 4.29in$$

$$\bar{c}_{B-rudder} = 4.29in$$

$$\therefore H.M._{rudder} = 0.35 \cdot (0.17761 psi)(27.802 in^2)(4.29in)$$

$$H.M._{rudder} = 7.41 in - lbs \Rightarrow 118.6 in - oz$$

The rudder control horn offset is 0.656*in* and the servo arm length is 0.437*in* therefore, the servo torque is,

$$T = Fd$$

$$118.6 in - oz = F(0.656 in)$$

$$F_{rudder\_horn} = 180.7 oz$$

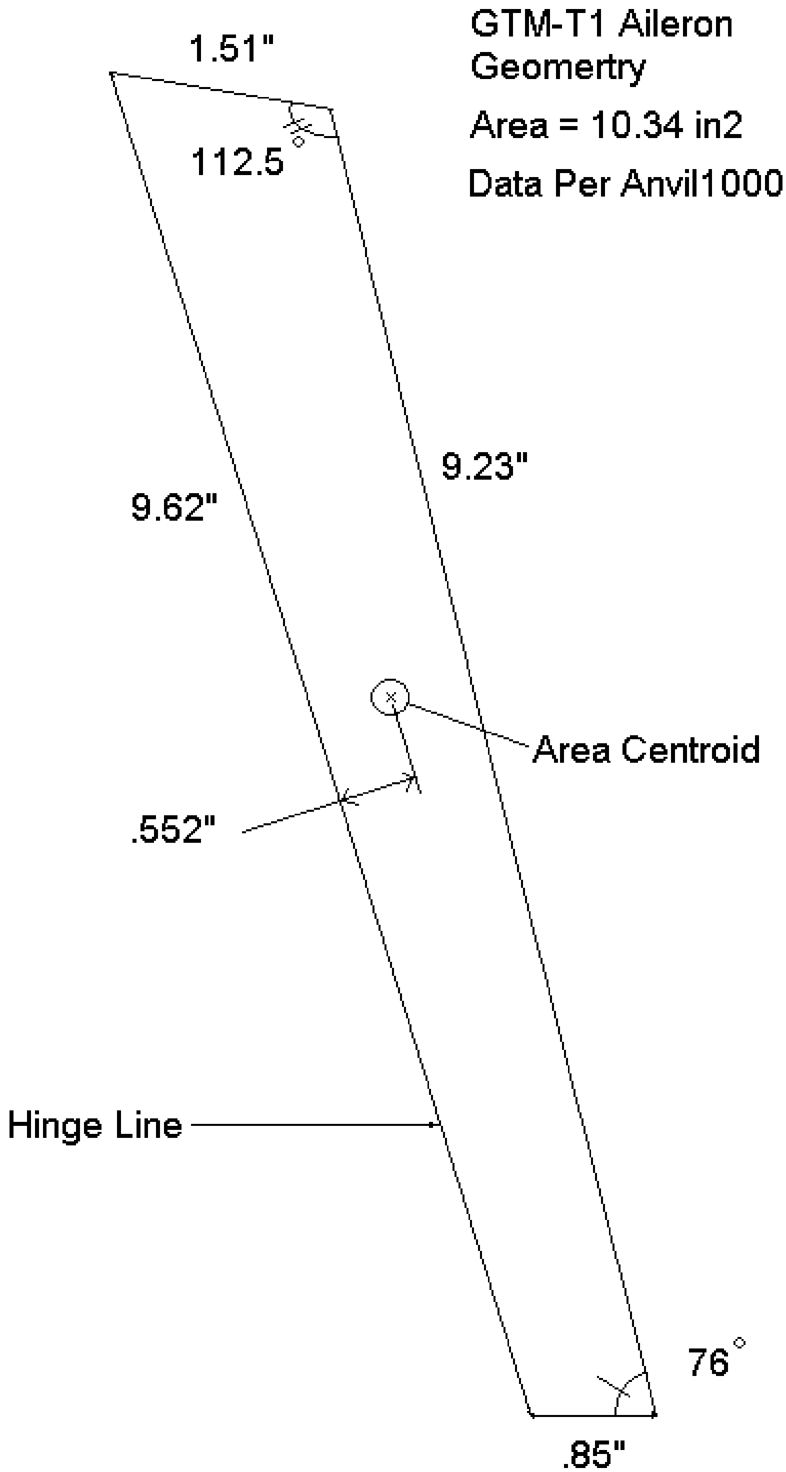
$$And,$$

$$T_{rudder\_servo} = (180.7)(0.437)$$

$$T_{rudder\_servo} = 79.1 in - oz$$

The hinge moment calculation, using the coefficient 0.35, was determined to be 79.1*in-oz*. Since the actuation system is capable of a rudder hinge moment of 118.6*in-oz*. then this system is assumed to be valid.

Aileron Servo Actuation Calculation:
Each aileron control surface is actuated by a "Volz Wing-Maxx" servo, located in the center body of the wing assembly. The manufacturer's rated torque is 42*in-oz*. The aileron geometry is shown below:



Drawing reference: LD-1164907

The aileron hinge moment is,

$$H.M._{aileron} = C_{h\_aileron} \cdot q \cdot S_{aileron} \cdot \bar{c}_{aileron}$$

where,

$$C_{h\_aileron} = 0.33(\text{Ref. GTMP}-4039)$$

$$q = 0.17761 psi$$

$$S_{aileron} = 10.34 in^2$$

$$MAC = \frac{2}{3}\left(C_R + C_T - \frac{C_R C_T}{C_R + C_T}\right)$$

where,

$$C_R = wing\ root\ chord$$

$$C_T = wing\ tip\ chord$$

$$MAC = \frac{2}{3}\left(1.51in + 0.85in - \frac{(1.51in)(0.85in)}{(1.51in + 0.85in)}\right)$$

$$MAC = 1.21in$$

$$\bar{c}_{aileron} = 1.21in$$

$$\therefore H.M._{aileron} = 0.33 \cdot (0.17761 psi)(10.34 in^2)(1.21in)$$

$$H.M._{aileron} = 0.733 in - lbs \Rightarrow 11.7 in - oz$$

The aileron control horn offset is 0.656*in* and the servo arm length is 0.437*in* therefore, the servo torque is,

$$T = Fd$$

$$11.7in - oz = F(0.656in)$$

$$F_{aileron\_horn} = 17.8 oz$$

And,

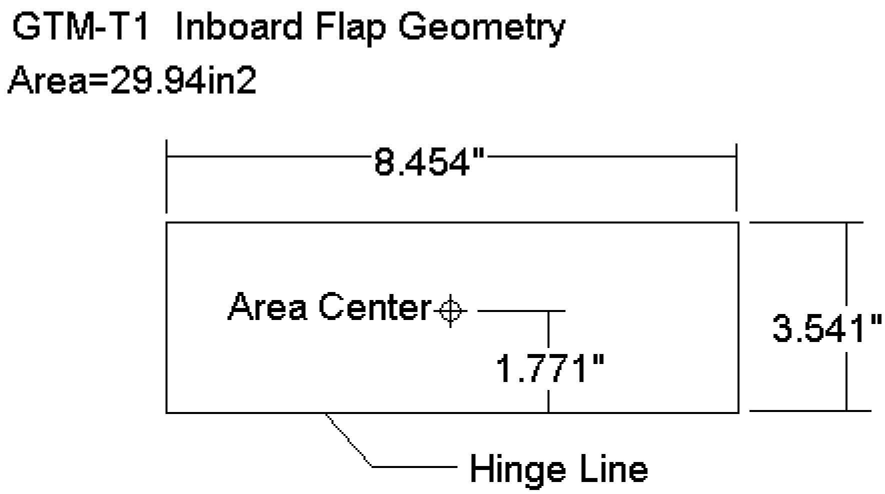
$$T_{aileron\_servo} = (17.8)(0.437)$$

$$T_{aileron\_servo} = 7.8 in - oz$$

The hinge moment calculation, using the coefficient 0.33, was determined to be 7.8*in-oz*. Since the actuation system is capable of an aileron hinge moment of 11.7*in-oz*. then this system is assumed to be valid.

Inboard Flap Servo Actuation Calculation:

The inboard flap control surface is actuated by a “JR-8411” servo located in the center body of the wing assembly. The manufacturer’s rated torque is 155*in-oz.* The inboard flap geometry is shown below:



Drawing reference: LD-1164907

The inboard flap hinge moment is,

$$H.M._{inboard\_flap} = C_{h\_in-flap}\, q\, S_{inboard\_flap}\, \bar{c}_{inboard\_flap}$$

$where,$

$C_{h\_in-flap} = 0.50$ $(assumed\ by\ the\ T.P.E.)$

$q = 0.17761 psi$

$S_{inboard\_flap} = 29.94 in^2$

$\bar{c}_{inboard\_flap} = 3.541 in$

$\therefore H.M._{inboard\_flap} = 0.50(0.17761 psi)(29.94 in^2)(3.541 in)$

$H.M._{inboard\_flap} = 9.4 in - lbs \Rightarrow 150.6 in - oz$

The inboard flap control horn offset is 1.031in and the servo arm length is 0.781in therefore the servo torque is,

$T = Fd$

$150.6 in - oz = F(1.031 in)$

$F_{in\_flap\_horn} = 73.1 oz$

$And,$

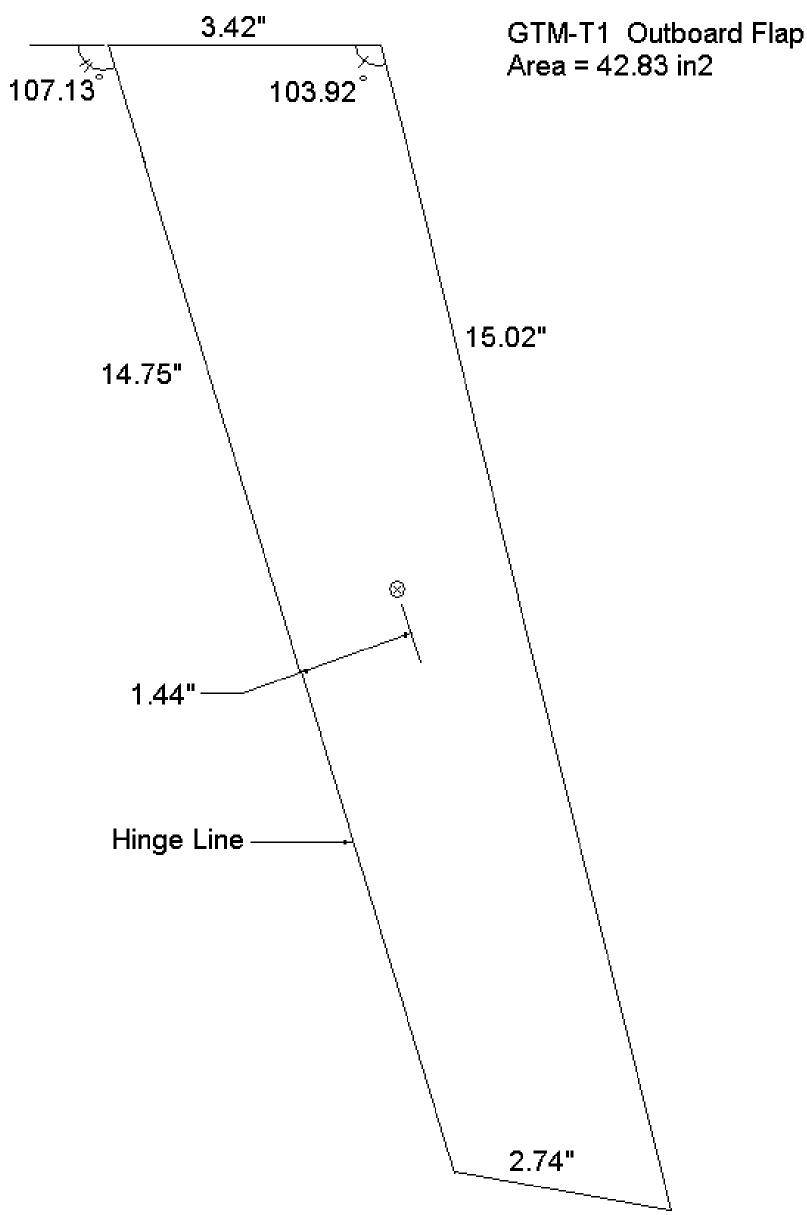
$T_{in\_flap\_servo} = (146.1)(0.781)$

$T_{in\_flap\_servo} = 114.1 in - oz$

The hinge moment calculation, using the coefficient 0.50, was determined to be114.1*in-oz*. Since the actuation system is capable of an inboard flap hinge moment of 150.6*in-oz*. then this system is assumed to be valid.

Outboard Flap Servo Actuation Calculation:

The outboard flap control surface is actuated by a "JR-8411" servo located in the center body of the wing assembly. The manufacturer's rated torque is 155*in-oz*. The outboard flap geometry is shown below:



Drawing reference: LD-1164907

The outboard flap hinge moment is,

$$H.M._{outboard\_flap} = C_{h\_outboard\_flap} \cdot q \cdot S_{outboard\_flap} \cdot \bar{c}_{outboard\_flap}$$

$$where,$$

$$C_{h\_outboard-flap} = 0.5 (\text{Ref. GTMP}-4039)$$

$$q = 0.17761 psi$$

$$S_{outboard\_flap} = 42.8 in^2$$

$$MAC = \frac{2}{3}\left(C_R + C_T - \frac{C_R C_T}{C_R + C_T}\right)$$

$$where,$$

$$C_R = wing\ root\ chord$$

$$C_T = wing\ tip\ chord$$

$$MAC = \frac{2}{3}\left(3.42in + 2.74in - \frac{(3.42in)(2.74in)}{(3.42in + 2.74in)}\right)$$

$$MAC = 3.09in$$

$$\bar{c}_{outboard\_flap} = 3.09in$$

$$\therefore H.M._{outboard\_flap} = 0.5 \cdot (0.17761 psi)(42.8 in^2)(3.09 in)$$

$$H.M._{outboard\_flap} = 11.7 in - lbs \Rightarrow 187.9 in - oz$$

The outboard flap control horn offset is 1.031*in* and the servo arm length is 0.781*in* therefore, the servo torque is,

$$T = Fd$$

$$187.9 in - oz = F(1.031 in)$$

$$F_{out\_flap\_horn} = 182.2 oz$$

$$And,$$

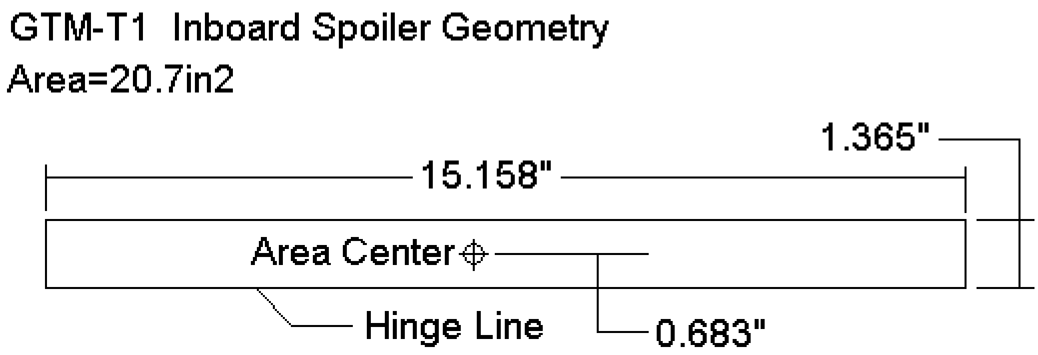
$$T_{out\_flap\_servo} = (182.2)(0.781)$$

$$T_{out\_flap\_servo} = 142.4 in - oz$$

The hinge moment calculation, using the coefficient 0.50, was determined to be 142.2*in-oz*. Since the actuation system is capable of an outboard flap hinge moment of 187.9*in-oz*. then this system is assumed to be valid.

Inboard Spoiler Servo Actuation Calculation:

And the inboard spoiler control surface is actuated by a "JR-3421" servo located in the center body of the wing assembly. The manufacturer's rated torque is 65*in-oz.* The inboard spoiler geometry is shown below:



Drawing reference: LD-1164907

The inboard spoiler hinge moment is,

$$H.M._{inboard-spoiler} = C_{h\_in-spoiler} \cdot q \cdot S_{inboard-spoiler} \cdot \bar{c}_{inboard-spoiler}$$

$$where,$$

$$C_{h\_in-spoiler} = 0.50 \ (assumed\ by\ the\ T.P.E.)$$

$$q = 0.17761 psi$$

$$S_{inboard-spoiler} = 20.7 in^2$$

$$\bar{c}_{inboard-spoiler} = 1.365 in$$

$$\therefore H.M._{inboard-spoiler} = 0.5(0.17761 psi)(20.7 in^2)(1.365 in)$$

$$H.M._{inboard-spoiler} = 2.51 in - lbs \Rightarrow 40.2 in - oz$$

The inboard spoiler control horn offset is 0.656*in* and the servo arm length is 0.437*in* therefore the servo torque is,

$$T = Fd$$

$$40.2 in - oz = F(0.656 in)$$

$$F_{in\_spoiler\_horn} = 61.2 oz$$

$$And,$$

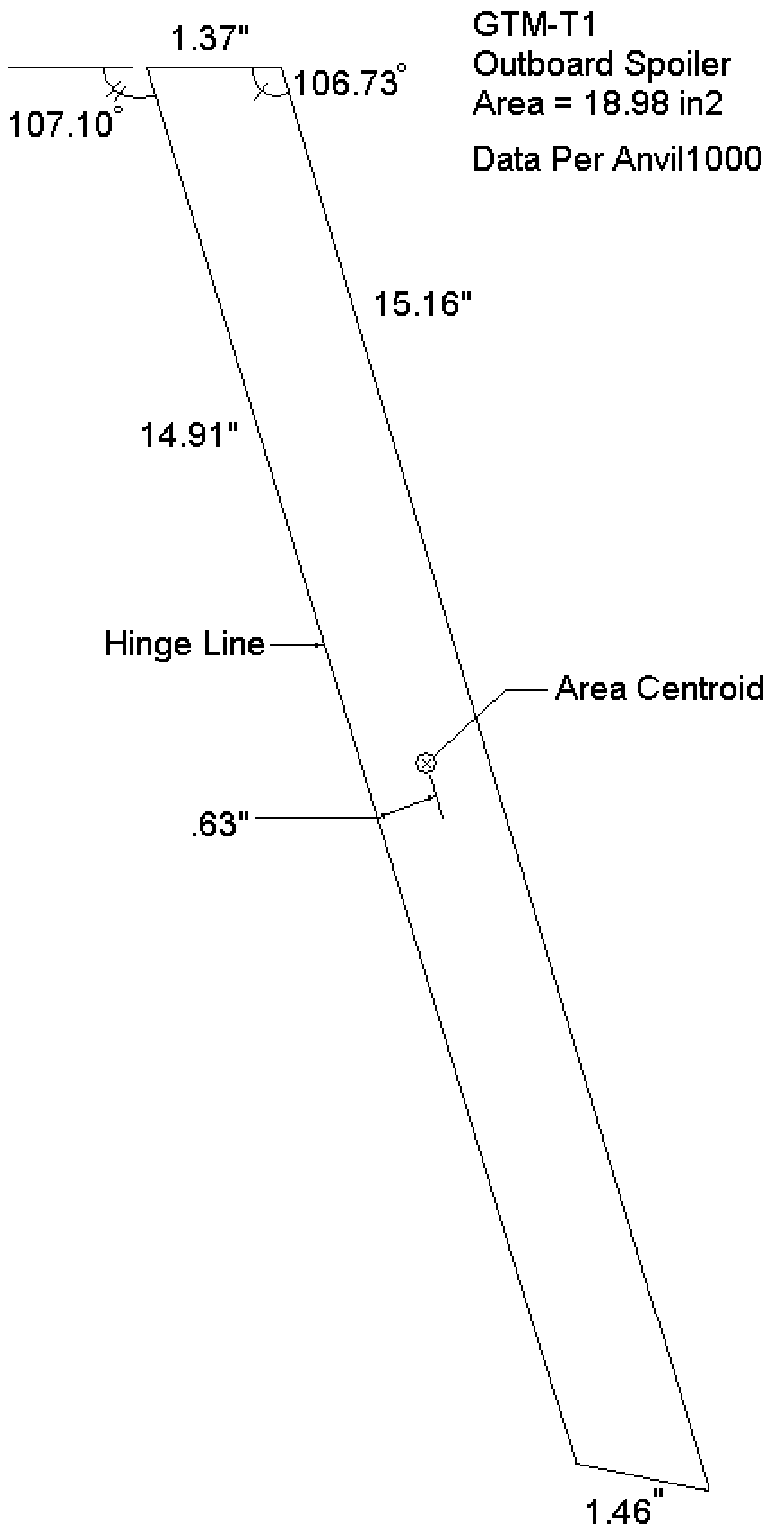
$$T_{in\_spoiler\_servo} = (61.2)(0.437)$$

$$T_{in\_spoiler\_servo} = 26.8 in - oz$$

The hinge moment calculation, using the coefficient 0.50, was determined to be 26.8*in-oz.* Since the actuation system is capable of an inboard spoiler hinge moment of 40.2*in-oz*. then this system is assumed to be valid.

Outboard Spoiler Servo Actuation Calculation:

The outboard spoiler control surface is actuated by a “JR-3421” servo located in the center body of the wing assembly. The manufacturer’s rated torque is 65*in-oz.* The outboard spoiler geometry is shown below:



Drawing reference: LD-1164907

The outboard spoiler hinge moment is,

$$H.M._{outboard\text{-}spoiler} = C_{h\_outboard\text{-}spoiler} \cdot q \cdot S_{outboard\text{-}spoiler} \cdot \bar{c}_{outboard\text{-}spoiler}$$

where,

$$C_{h\_outboard\text{-}spoiler} = 0.5 \text{(Ref. GTMP} - 4039\text{)}$$

$$q = 0.17761 psi$$

$$S_{outboard\text{-}spoiler} = 18.98 in^2$$

$$MAC = \frac{2}{3}\left(C_R + C_T - \frac{C_R C_T}{C_R + C_T} \right)$$

where,

$$C_R = wing\ root\ chord$$

$$C_T = wing\ tip\ chord$$

$$MAC = \frac{2}{3}\left(1.46in + 1.37in - \frac{(1.46in)(1.37in)}{(1.46in + 1.37in)} \right)$$

$$MAC = 1.42in$$

$$\bar{c}_{outboard\text{-}spoiler} = 1.42in$$

$$\therefore H.M._{outboard\text{-}spoiler} = 0.5 \cdot (0.17761psi)(18.98in^2)(1.42in)$$

$$H.M._{outboard\text{-}spoiler} = 2.39in - lbs \Rightarrow 38.3in - oz$$

The outboard spoiler control horn offset is 0.656*in* and the servo arm length is 0.437*in* therefore, the servo torque is,

$$T = Fd$$

$$38.3in - oz = F(0.656in)$$

$$F_{out\_spoiler\_horn} = 58.4oz$$

And,

$$T_{out\_spoiler\_servo} = (58.4)(0.437)$$

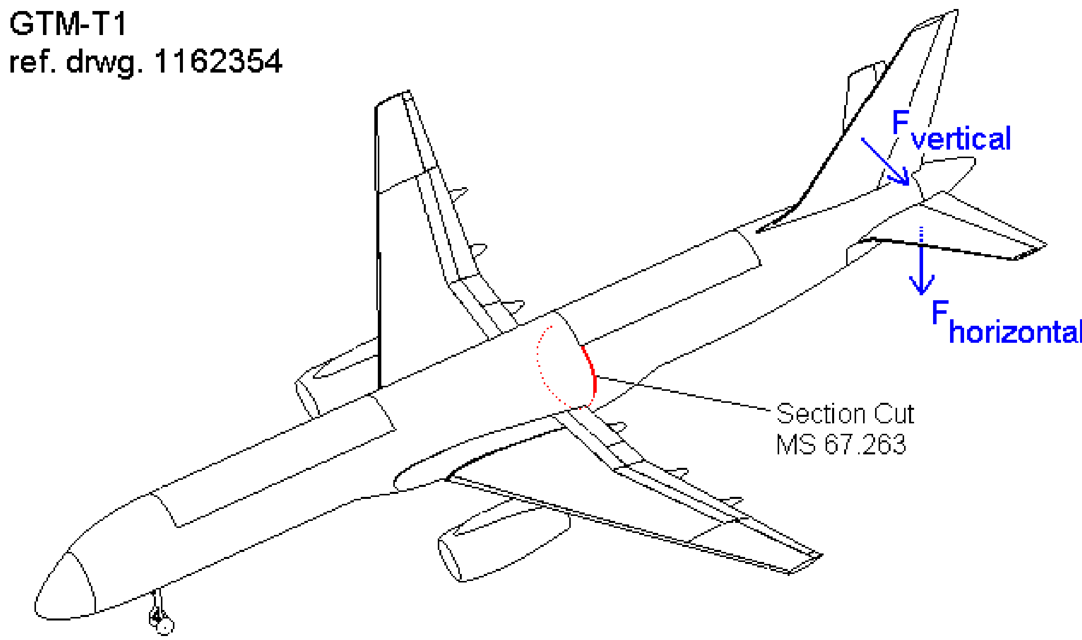
$$T_{out\_spoiler\_servo} = 25.5in - oz$$

The hinge moment calculation, using the coefficient 0.50, was determined to be 25.5*in-oz*. Since the actuation system is capable of an outboard spoiler hinge moment of 38.3*in-oz*. then this system is assumed to be valid.

Fuselage Analysis

Aft Fuselage Combined Stress Calculation:

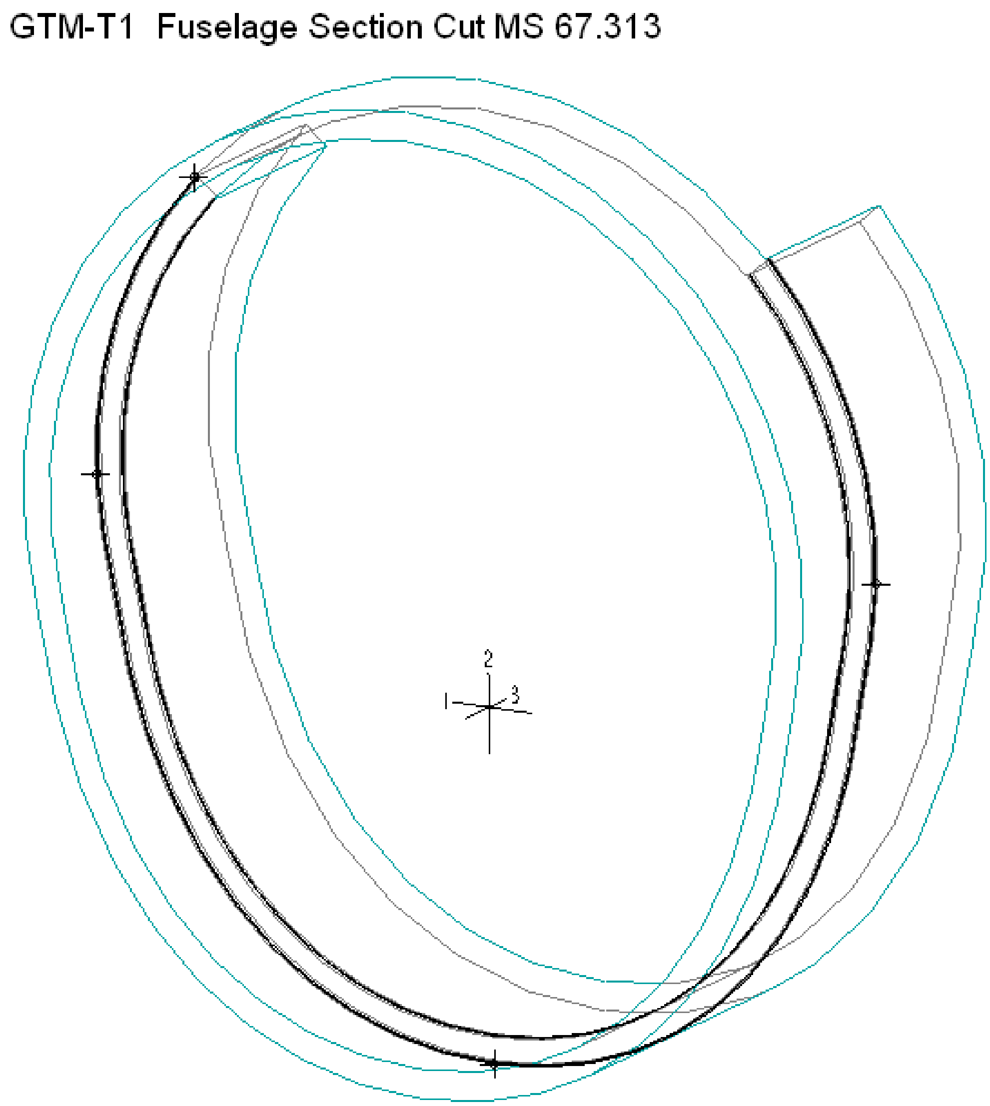
This analysis presents calculations of the fuselage stress on the 5.5% General Transport Model. The calculations were done to prove that the fuselage would withstand the bending and shear stress due to load on horizontals, and vertical tail.



Drawing Reference: LD-1164884

Sketch of the GTM-T2 and the section cut location of MS 67.263,
Where,

$$F_{Vertical} = 46.3 lbs. @ MS 99.25 \& WL 22.56$$
$$F_{Horizontal} = 23.4 lbs. @ MS 101.93 \& BL 8.29$$
$$F_{Horizontal\_Total} = (2)(23.4 lbs.)$$
$$F_{Horizontal\_Total} = 46.9 lbs. @ MS 101.93$$



Drawing reference: LD-1164884

Sketch of section cut at MS 67.313 with the aft fuselage hatch removed. It is assumed that the aft fuselage hatch contributes no load carrying ability to the section cut.

The MS 67.313 section properties are given below:

START PRO-ENGINEER DATA

```
 AREA =  6.5841185e-01 INCH^2

CENTER OF GRAVITY with respect to _MS_67-313 coordinate frame:
X   Y     0.0000000e+00  1.0960356e+01  INCH

INERTIA with respect to _MS_67-313 coordinate frame:  (INCH^4)
```

```
INERTIA TENSOR:
Ixx Ixy  8.2462178e+01  0.0000000e+00
Iyx Iyy  0.0000000e+00  5.5941491e+00

POLAR MOMENT OF INERTIA:  8.8056327e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _MS_67-313 coordinate
frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.3675619e+00  0.0000000e+00
Iyx Iyy  0.0000000e+00  5.5941491e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  3.3675619e+00  5.5941491e+00

POLAR MOMENT OF INERTIA:  8.9617110e+00 INCH^4

ROTATION MATRIX from _MS_67-313 orientation to PRINCIPAL AXES:
       1.00000        0.00000
       0.00000        1.00000

ROTATION ANGLE from _MS_67-313 orientation to PRINCIPAL AXES (degrees):
about z axis          0.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  2.2615645e+00  2.9148633e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                         1           2   COORD
about AXIS 1:  1.00739e+00 INCH^3        5.3071e-14 -3.3429e+00   INCH
               7.34229e-01 INCH^3        2.9918e+00  4.5865e+00   INCH
about AXIS 2:  1.37557e+00 INCH^3       -4.0668e+00  1.6693e+00   INCH
               1.37557e+00 INCH^3        4.0668e+00  1.6693e+00   INCH
```
END PRO-ENGINEER DATA

Vertical

Moment

$$M_{yt} = Fd$$
$$M_{yt} = (46.3lbf)(99.25in - 67.313in)$$
$$M_{yt} = 1478.7in \cdot lb$$

Bending Stresses

$$\sigma_y = \frac{M_y}{z_y}$$

$$\sigma_y = \frac{1478.7 in \cdot lb}{1.376 in^3}$$

$$\sigma_y = 1074.9 psi$$

and,

Horizontal

Moment

$$M_{zt} = Fd$$

$$M_{zt} = (46.9 lbf)(101.93 in - 67.313 in)$$

$$M_{zt} = 1623.5 in \cdot lb$$

Bending Stresses

$$\sigma_z = \frac{M_z}{z_z}$$

$$\sigma_z = \frac{1623.5 in \cdot lb}{0.734 in^3}$$

$$\sigma_z = 2211.2 psi$$

Combining vertical and horizontal stresses gives:

$$\sigma_{total} = \sqrt{(1074.9 psi)^2 + (2211.2 psi)^2}$$

$$\sigma_{total} = 2458.7 psi$$

Torsional Stress due to the Vertical Tail

The torsion "arm" for this calculation is:

L_v = WL 22.559 – WL 10.954 = 11.61*in*

And from Mechanical Engineering Design, Shigley, 3rd Edition, page 58:

$$\tau_{\max} = \frac{T r_{pt\,A}}{J}$$

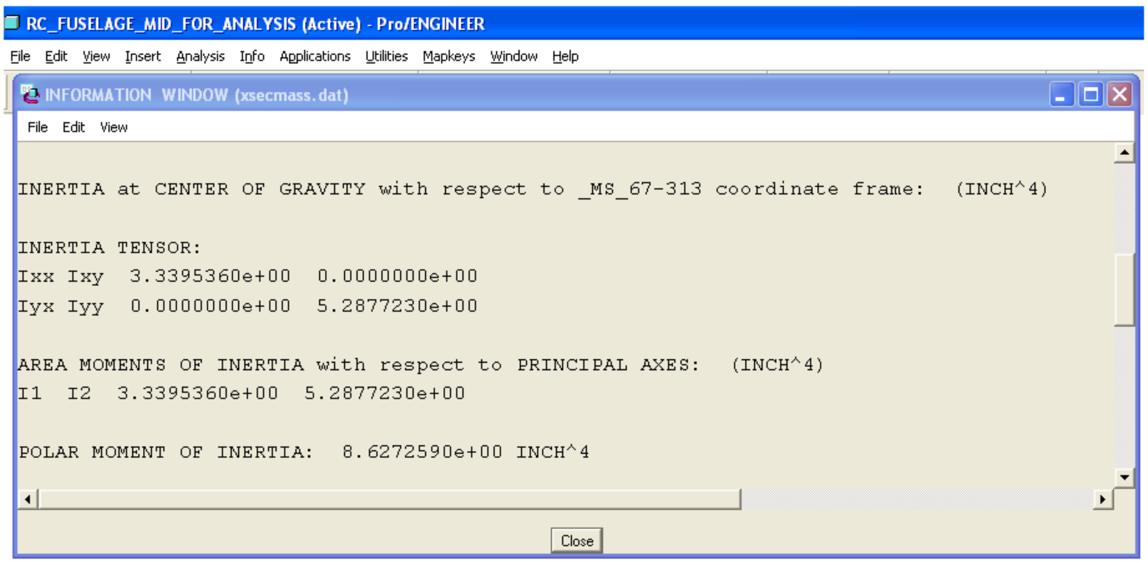
where,

$$r_{pt\,A} = \sqrt{(2.99in)^2 + (4.07in)^2} = 5.05in$$

$$T = (F_{vertical})(Lv) = (46.3lbs)(11.61in) = 537.3in - lbs$$

$$J = the\ \ polar\ \ moment\ \ of\ \ inertia = 8.627\ in^4\ \ (see\ below)$$

GTM-T1 Fuselage Cut at MS 67.313



RC_FUSELAGE_MID_FOR_ANALYSIS (Active) - Pro/ENGINEER

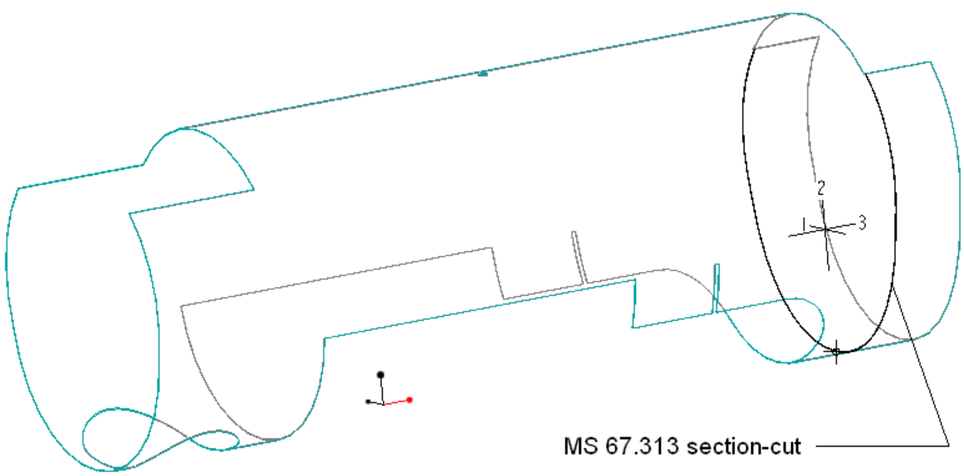
File Edit View Insert Analysis Info Applications Utilities Mapkeys Window Help

INFORMATION WINDOW (xsecmass.dat)

File Edit View

```
INERTIA at CENTER OF GRAVITY with respect to _MS_67-313 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.3395360e+00  0.0000000e+00
Iyx Iyy  0.0000000e+00  5.2877230e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  3.3395360e+00  5.2877230e+00

POLAR MOMENT OF INERTIA:  8.6272590e+00 INCH^4
```

Close



Drawing reference: LD-1164884

Therefore:

$$\tau_{\max} = \frac{(537.3in - lbs)(5.05in)}{(8.627in^4)}$$

$$\tau_{\max} = 314.5psi$$

Shear Stress

$$\tau = \frac{F_{total}}{A_{shear}}$$

$$\tau = \frac{\sqrt{(46.3lbs)^2 + (46.9lbs)^2}}{(0.658in^2)}$$

$$\tau = 100.16psi$$

Maximum Shear Stress

$$\tau_{Max} = \sqrt{\left(\frac{2458.7psi}{2}\right)^2 + (314.5psi + 100.05psi)^2}$$

$$\tau_{Max} = 1297.4psi$$

Factor of Safety

And the factor of safety based on an assumed ultimate strength of fiberglass of 11,000*psi* gives:

$$\boxed{FS = \frac{11000psi}{1297.4psi} = 8.5}$$

Combined Stress

$$\sigma_{combined} = \frac{2458.7psi}{2} \pm \sqrt{\left(\frac{2458.7psi}{2}\right)^2 + (314.5psi + 100.05psi)^2}$$

$$\sigma_{combined} = 1229.3 \pm 1297.4$$

$$\sigma_{combined} = 2526.8psi$$

Factor of Safety

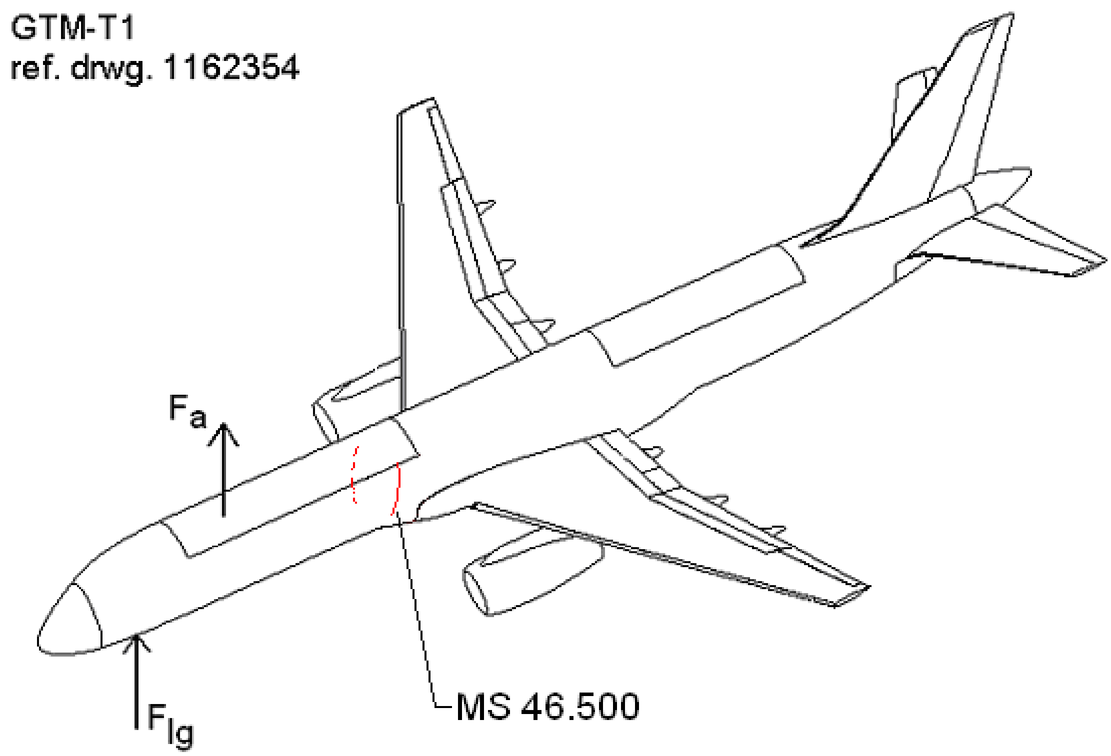
And the factor of safety based on an assumed ultimate strength of fiberglass of 11,000*psi* gives:

$$\boxed{FS = \frac{11000psi}{2526.8psi} = 4.4}$$

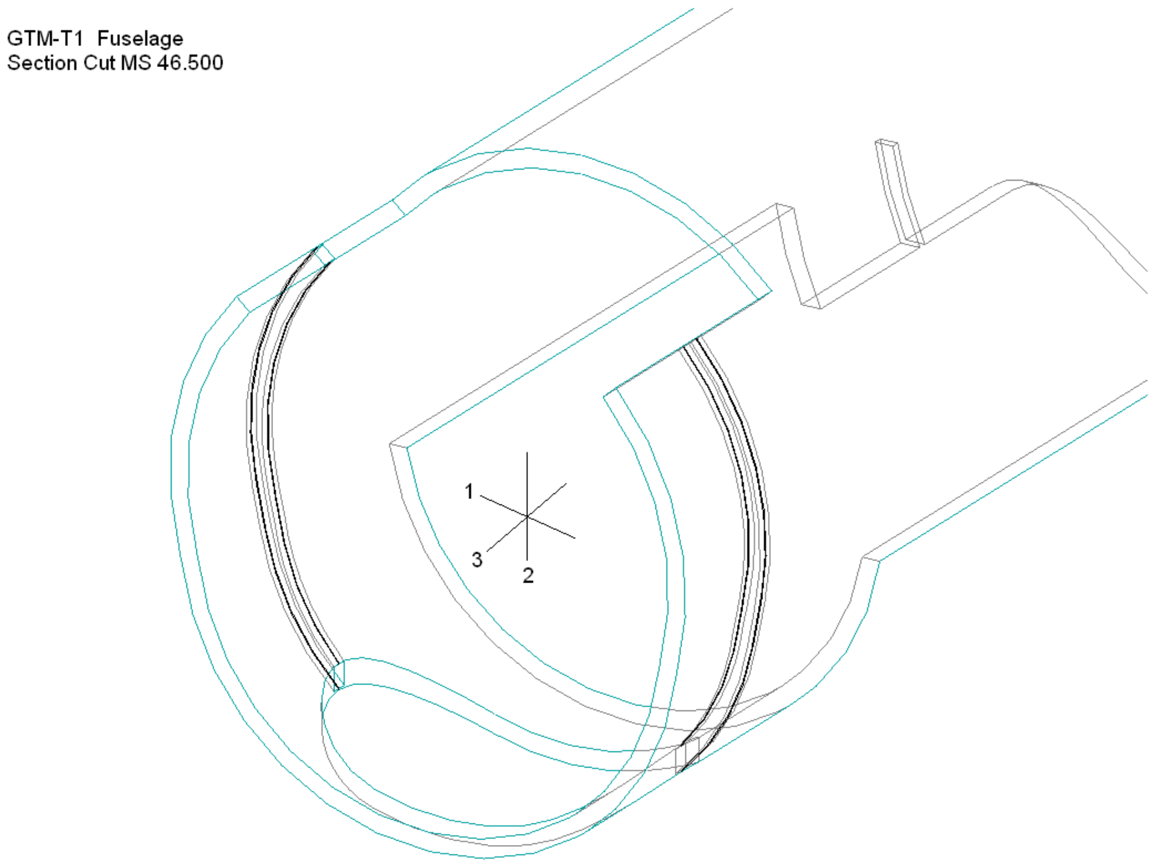
The calculations were done to prove that the fuselage would withstand the stresses due to loading on the horizontals, and the vertical tail.

<u>Forward Fuselage Combined Stress Calculation: (landing gear and aero load forces)</u>

The combined stress in the forward fuselage is calculated at MS 46.500 due to a large nose gear load upon landing and an aerodynamic load. It is assumed the hatch and the wing body-fairing do not carrying any significant load. The section cut at MS 46.500 intersects the lower fuselage access hole as shown in the sketches below:



Drawing reference: LD-1164884



Drawing reference LD-1164884

The MS 46.500 section properties are given below:

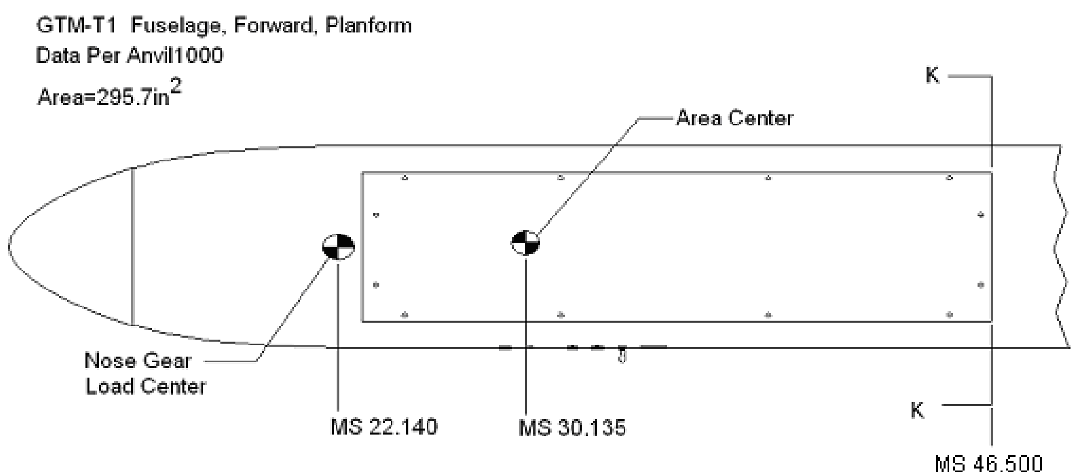
<u>START PRO-ENGINEER DATA</u>

```
GTM-T1 Fuselage MS 46.500 Section Properties  4May04

AREA =  3.3758962e-01 INCH^2

CENTER OF GRAVITY with respect to _MS_46-500 coordinate frame:
X   Y     1.2401569e+01 -1.2000000e+00  INCH

INERTIA with respect to _MS_46-500 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.7523229e+00  5.0239690e+00
Iyx Iyy  5.0239690e+00  5.3062361e+01

POLAR MOMENT OF INERTIA:  5.7814683e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _MS_46-500 coordinate
frame:  (INCH^4)
```

```
INERTIA TENSOR:
Ixx Ixy  4.2661938e+00  0.0000000e+00
Iyx Iyy  0.0000000e+00  1.1414475e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.1414475e+00  4.2661938e+00

POLAR MOMENT OF INERTIA:  5.4076413e+00 INCH^4

ROTATION MATRIX from _MS_46-500 orientation to PRINCIPAL AXES:
       0.00000       -1.00000
       1.00000        0.00000

ROTATION ANGLE from _MS_46-500 orientation to PRINCIPAL AXES (degrees):
about z axis          90.000

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.8387954e+00  3.5548866e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  3.62904e-01 INCH^3        2.9918e+00 -3.1453e+00   INCH
               3.56371e-01 INCH^3        2.7000e+00  3.2030e+00   INCH
about AXIS 2:  1.04913e+00 INCH^3       -4.0664e+00 -2.1903e-01   INCH
               1.04913e+00 INCH^3        4.0664e+00 -2.1903e-01   INCH
```

END PRO-ENGINEER DATA

A plan-view of the forward fuselage is shown below with its associated load points:



Drawing reference: LD-1164884

It is assumed that the aero load acts at the area center in the above sketch.

Assuming that C_L=1.3 then:

$$F_{aero} = AC_L q$$

$$F_{aero} = (295.7in^2)(1.3)(25.58psf)\left(\frac{1ft^2}{144in^2}\right)$$

$$F_{aero} = 68.29lbs$$

Also, assume that the force on the nose gear is due to a 2'g landing condition:

$$F_{\lg} = (2g's)(Model_{\max\_weight})$$

$$F_{\lg} = (2)(55lbs) = 110lbs$$

Then, the shear stress at section K-K is:

$$\tau = \frac{F_{\lg} + F_a}{Area_{shear}}$$

$$Area_{shear} = 0.337in^2$$

$$\tau = \frac{68.29lb + 110lb}{0.337in^2} = \frac{178.29lb}{0.337in^2}$$

$$\tau = 528.1psi$$

The bending stress due to aero load force is:

$$\sigma_1 = \frac{M_{Faero}}{z_1}$$

$$\sigma_1 = \frac{(68.29lb)(46.5in - 30.135in)}{0.356in^3} = \frac{1117.5in \cdot lb}{0.356in^3}$$

$$\sigma_1 = 3135.8psi$$

The bending stress due to the landing gear force is:

$$\sigma_2 = \frac{M_{F\lg}}{z_1}$$

$$\sigma_2 = \frac{(110lb)(46.5in - 22.14in)}{0.356in^3} = \frac{2679.6lb \cdot in}{0.356in^3}$$

$$\sigma_2 = 7519.1psi$$

Combining the bending stresses gives:

$$\sigma_{total} = 3135.8psi + 7519.1psi$$
$$\sigma_{total} = 10655psi$$

The total stress is:

$$\sigma_{combined} = \frac{10655psi}{2} \pm \sqrt{\left(\frac{10655psi}{2}\right)^2 + (528.1psi)^2}$$
$$\sigma_{combined} = 5327.5 \pm 5353.6$$
$$\sigma_{combined} = 10681psi$$

The factor of safety based on an assumed ultimate strength of fiberglass of 11,000*psi* gives:

$$\boxed{FS = \frac{11000psi}{10681psi} = 1.03}$$

Forward Fuselage Combined Stress Calculation: (From a side load only.)

Assume that the side load is from aerodynamic loading only and is of the same magnitude and location as the above analysis then:

$$\tau = \frac{F_a}{Area_{shear}}$$
$$Area_{shear} = 0.337in^2$$
$$\tau = \frac{68.29lb}{0.337in^2}$$
$$\tau = 202.3psi$$

And:

$$\sigma = \frac{M_{Faero}}{z_2}$$
$$\sigma = \frac{(68.29lb)(46.5in - 30.135in)}{1.049in^3} = \frac{1117.5lb \cdot in}{1.049in^3}$$
$$\sigma = 1065.2psi$$

The total stress is:

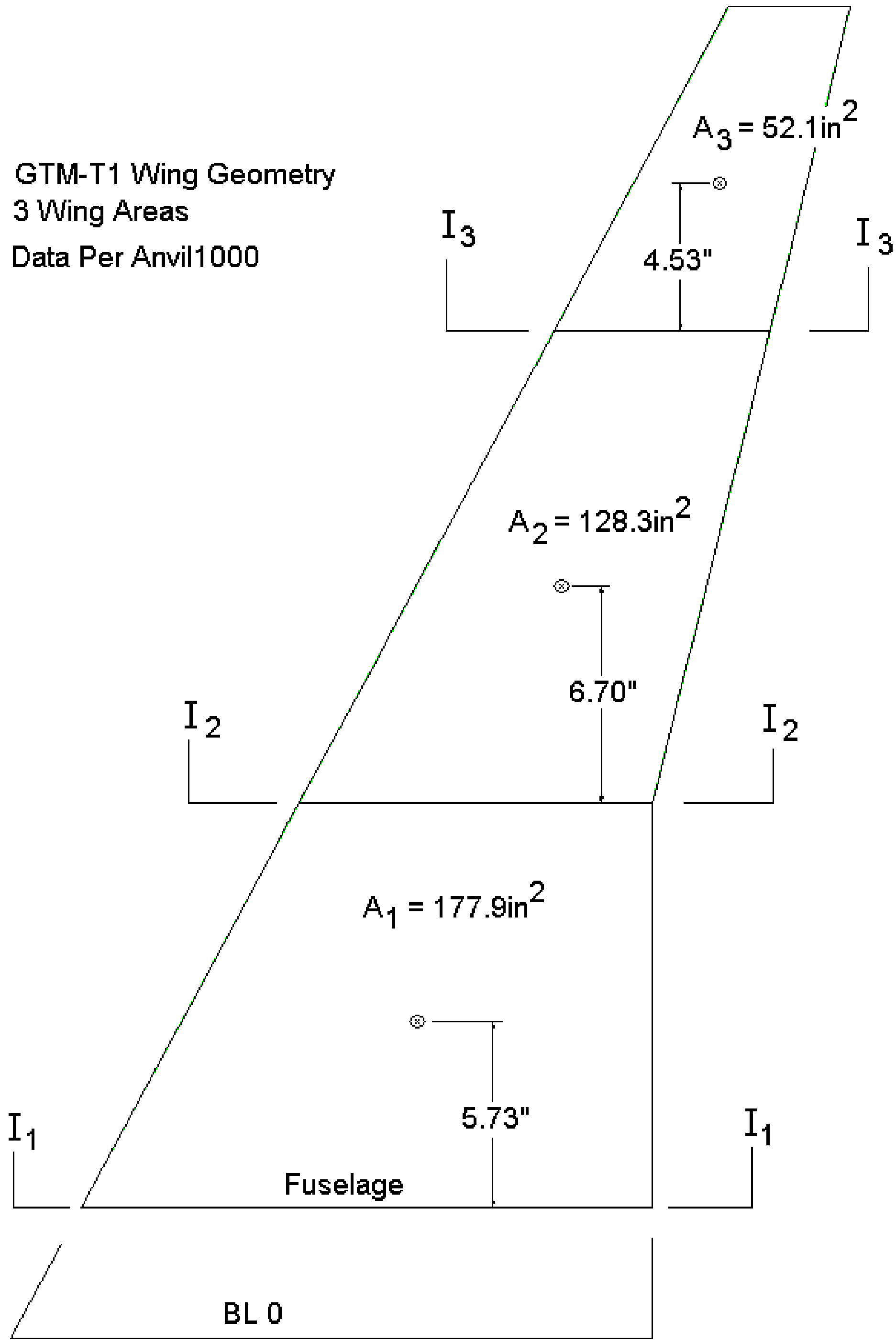
$$\sigma_{combined} = \frac{1065.2psi}{2} \pm \sqrt{\left(\frac{1065.2psi}{2}\right)^2 + (202.3psi)^2}$$

$$\sigma_{combined} = 532.6 \pm 569.7$$

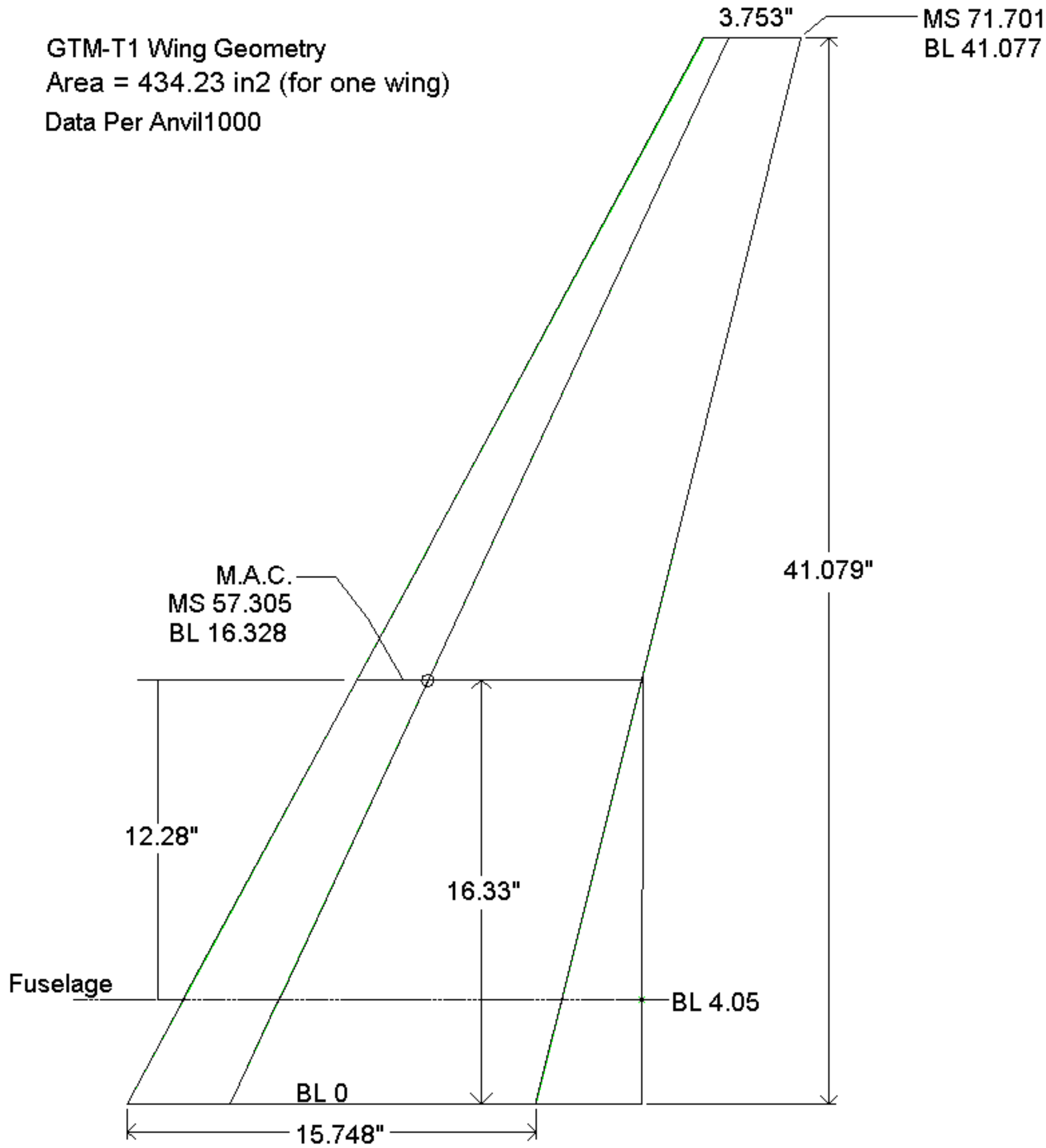
$$\sigma_{combined} = 1102.3psi$$

The factor of safety based on an assumed ultimate strength of fiberglass of 11,000psi gives:

$$\boxed{FS = \frac{11000psi}{1102.3psi} = 9.98}$$



Top view of the wing, drawing reference: LD –1164907.

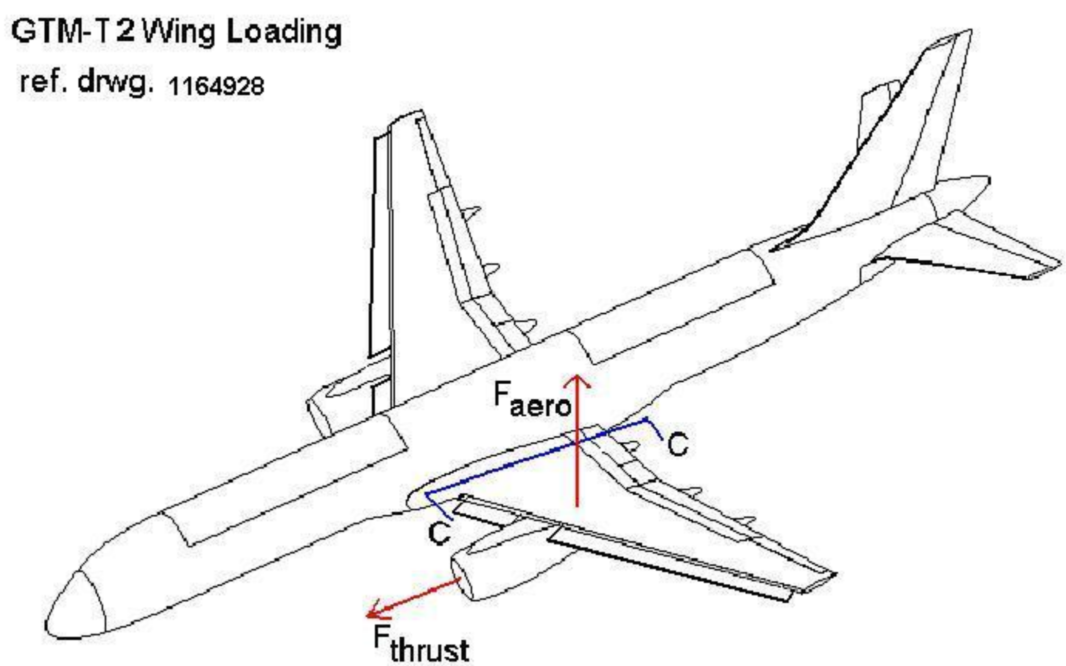


Location of center of pressure – Top view, drawing reference: LD-1164907

Wing Bending Stresses:

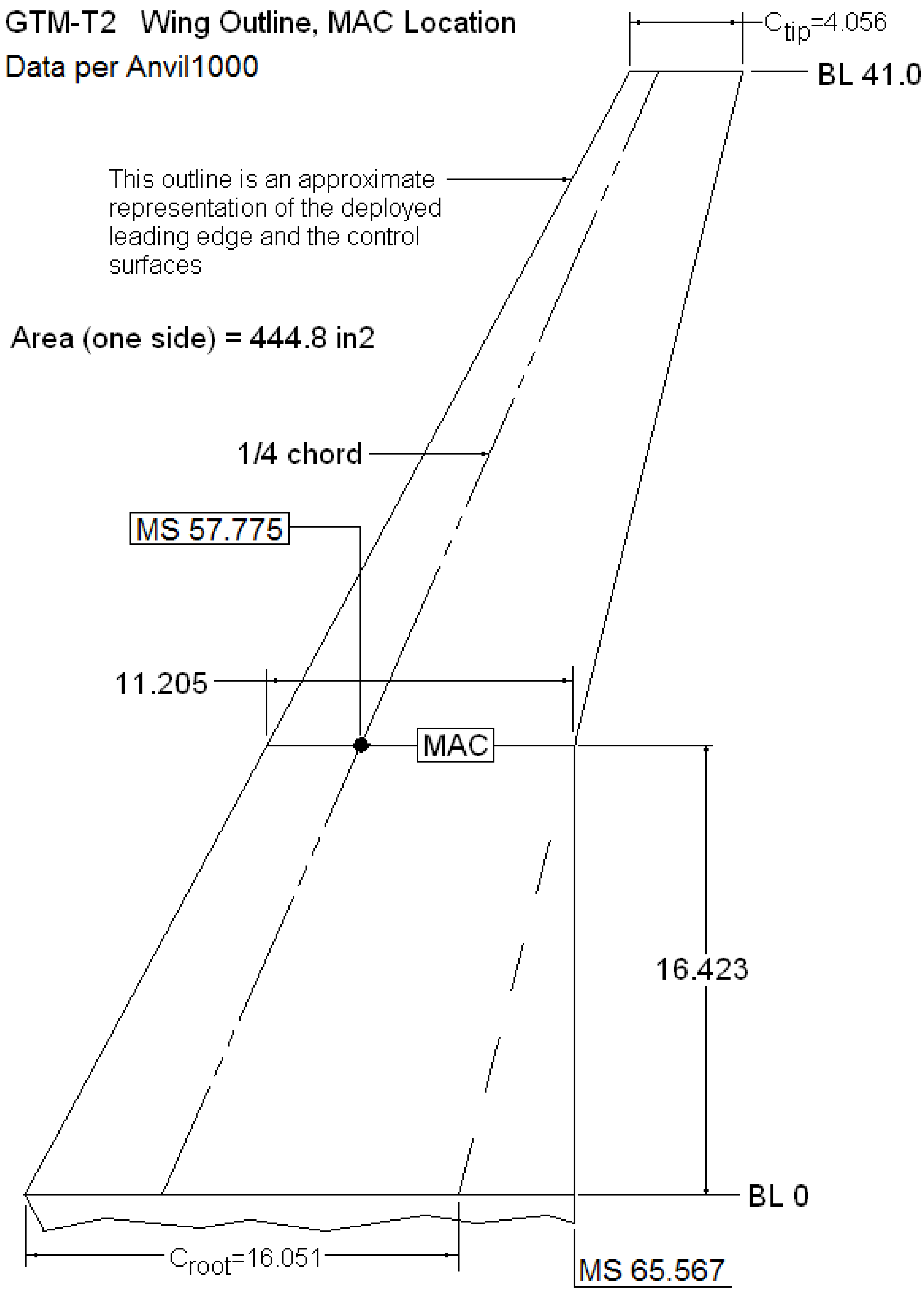
Wing Bending stress at BL 4.125

The maximum stress is calculated from the shear stress and the bending stresses generated by the aerodynamic load and the engine loads. The wing root stresses are calculated using the coefficient of lift where: C_L = 1.77 (fixed slats & flaps stowed) and C_L = 2.18 (fixed slats & flaps deployed). The reference for these coefficients was obtained from the SAWA Weight Engineers Handbook, Society of Allied Weight Engineers, © 1976, page 10.6. These values are also reference to the document "NACA-TR-427". Homogeneous material equations have been used in calculating a safety factor for the carbon fiber in bending. The weight of the wing and the engine are neglected in this analysis. The wing is analyzed at section C-C as sketched below:



Drawing reference: LD-1164928, LD-1162340, and LD-1162348

It is assumed that the F_{aero} load acts at the ¼ chord line on the *MAC* as shown below:



Drawing reference: LD-1164907-1

MAC Calculations:

$$MAC = \frac{2}{3}\left(C_R + C_T - \frac{C_R C_T}{C_R + C_T}\right)$$

$$where,$$

$$C_R = wing\ root\ chord$$

$$C_T = wing\ tip\ chord$$

$$MAC = \frac{2}{3}\left(16.051in + 4.056in - \frac{(16.051in)(4.056in)}{(16.051in + 4.056in)}\right)$$

$$MAC = 11.246\ in$$

And the location of the *MAC* is,

$$X_{MAC} = \frac{b}{6}\left(\frac{1+2\lambda}{1+\lambda}\right)$$

$$where,$$

$$b = 82in\ wing\ span$$

$$\lambda = {C_T}\!\left/{C_R}\right. = 0.25269in$$

$$X_{MAC} = \frac{82in}{6}\left(\frac{1+2(0.25269in)}{1+(0.25269in)}\right)$$

$$X_{MAC} = 16.423\ in$$

The Lift Force:

Case 1- C_L = 1.77 (fixed slats & flaps stowed):

The calculation for the entire wing with extended leading edge slats and the fuselage is indicated below.

$$F_{Lift} = A \cdot C_L \cdot q$$

$$F_{Lift} = \left(889.6in^2\right)\cdot(1.77)\cdot(0.17761psi)$$

$$F_{Lift} = 279.7lbf$$

F_{lift} acting on each wing at 100*mph* is 139.8*lbf*

Therefore:

F_{aero} = 139.8 *lbs* and F_{thrust} = 20.5 *lbs**

**Note: The engine test stand results yielded values higher than 20lbs*

Case 2- C_L = 2.18 (fixed slats & flaps deployed):

The calculation for the entire wing with extended leading edge slats and the fuselage is indicated below.

$$F_{Lift} = A \cdot C_L \cdot q$$

$$F_{Lift} = \left(889.6in^2\right) \cdot \left(2.18\right) \cdot \left(0.17761psi\right)$$

$$F_{Lift} = 344.4lbf$$

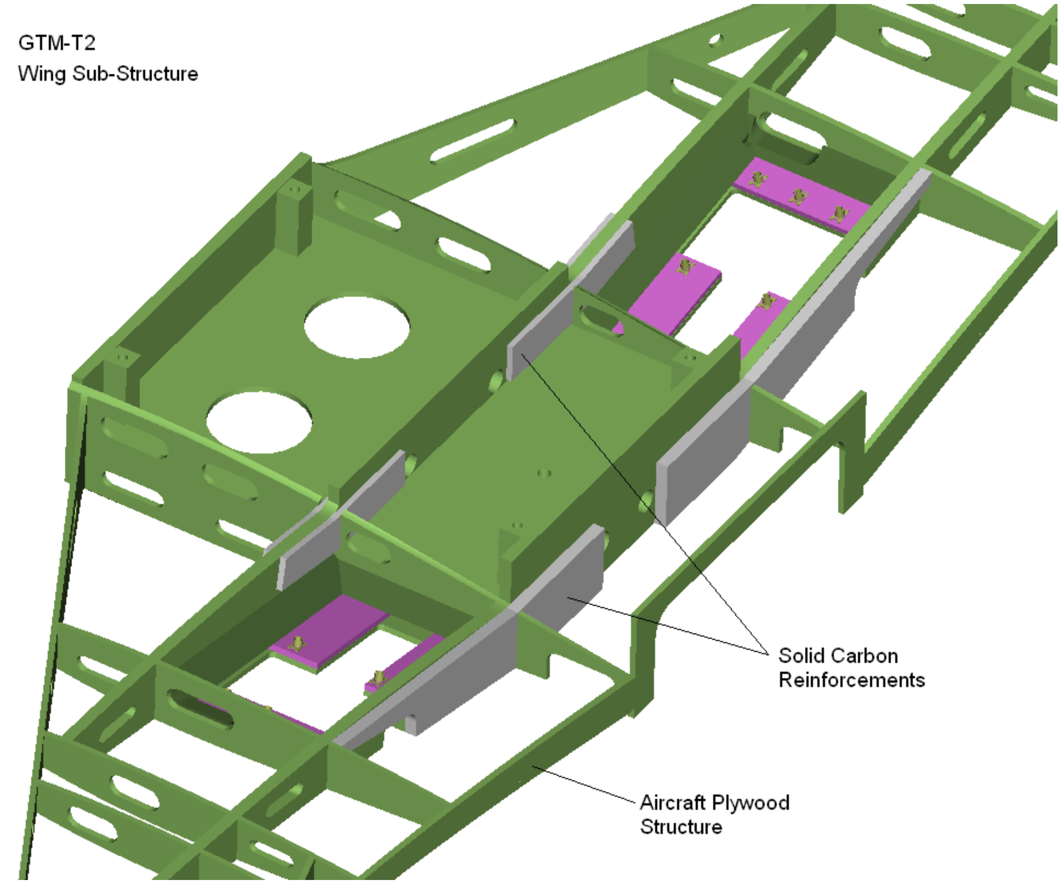
F_{lift} acting on each wing at 100mph is 172.2*lbf*

Therefore:

F_{aero} = 172.2 *lbs* and F_{thrust} = 20.5 *lbs**

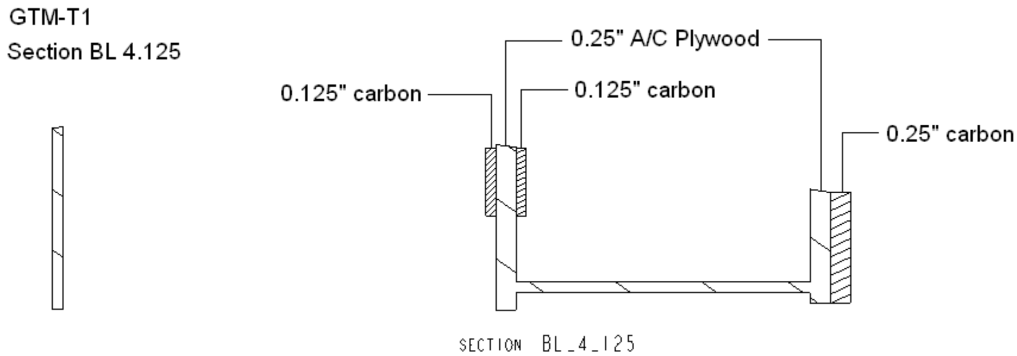
**Note: The engine test stand results yielded values higher than 20lbs*

An isometric view of the wing structure with its reinforced spars is shown below:



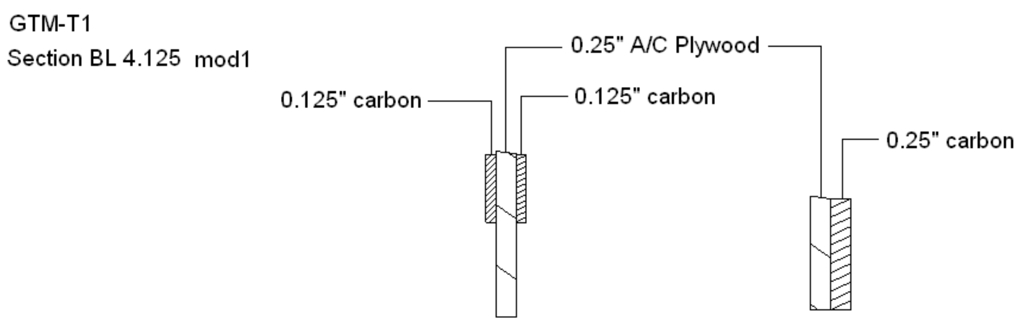
Drawing reference: LD-1164895-1

It is assumed that section C-C is located at BL 4.125 inches and does not include the wing skins. The section is shown below:



Drawing Reference: LD-1164914, LD-1164895-1

The section is modified by deleting the 1/8 *in* thick spar features at the ends of the section. Also, the main gear floor plate is deleted since it is assumed that these features do not contribute significantly to the load carry capability. The modified section cut is shown below with its carbon reinforcement:



Drawing Reference: LD-1164914, LD-1164895-1

Since the section cut consist of different types of materials (i.e. carbon & wood) the carbon features need to be converted into the appropriate wooded dimensions. Assuming that the wood has an elastic modulus of 1.7E6*psi* and assuming that the carbon (0/90 degree orientation) has an elastic modulus of 6.0E6*psi* then, the equivalent stiffness ration is:

$$\frac{E_{carbon0/90}}{E_{A/C\ plywood}} = \frac{6.0E6}{1.7E6} = 3.529$$

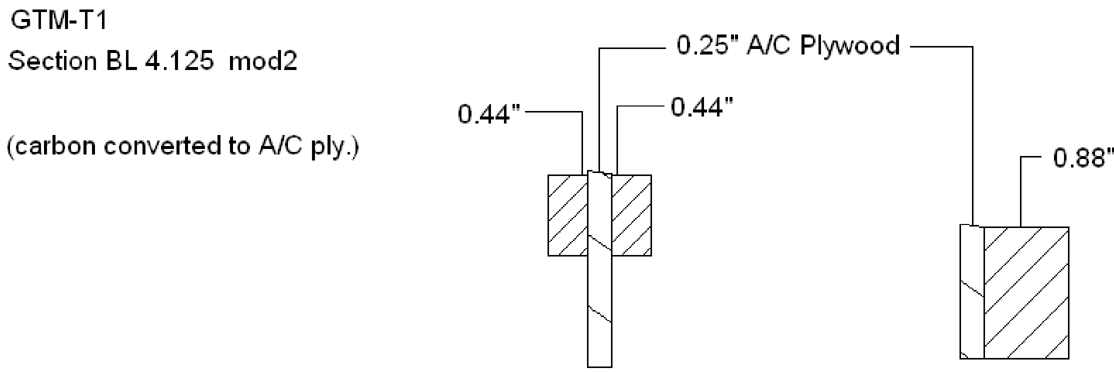
Resizing the carbon parts to represent A/C plywood gives the following sketch:

NOTE:

$(0.125\ carbon)(3.529) = 0.44in$ A/C plywood

And

$(0.25\ carbon)(3.529) = 0.88in$ A/C plywood



Drawing Reference: LD-1164914, LD-1164895-1

Therefore from the Pro-Engineer Solid Modeling Program this section has the following properties as shown below:

START PRO-ENGINEER SECTION DATA

```
BL 4.125 Section Properties                        7MAY04

AREA =  2.7202820e+00 INCH^2

CENTER OF GRAVITY with respect to _BL_4_125 coordinate frame:
X   Y     7.8287484e+00  1.0059057e+00  INCH

INERTIA with respect to _BL_4_125 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.5032209e+00 -1.9704350e+01
Iyx Iyy -1.9704350e+01  1.7927892e+02

POLAR MOMENT OF INERTIA:  1.8278214e+02 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _BL_4_125 coordinate
frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  7.5071384e-01  1.7178233e+00
Iyx Iyy  1.7178233e+00  1.2554734e+01

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  5.0580275e-01  1.2799645e+01
```

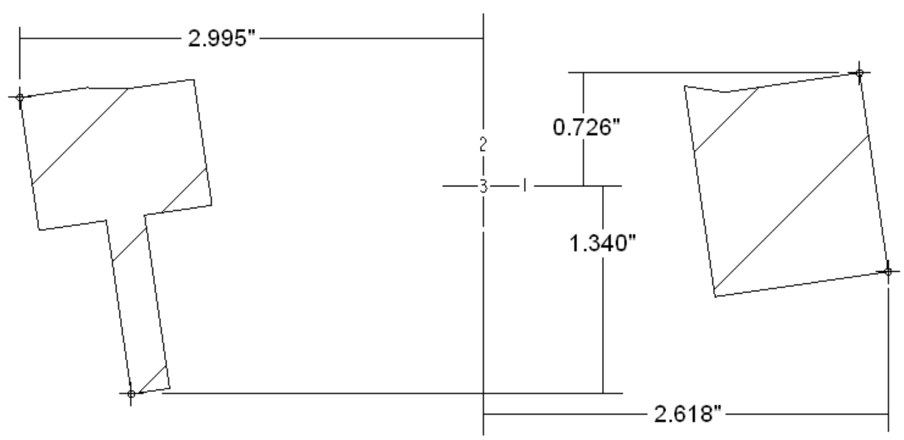
```
POLAR MOMENT OF INERTIA:  1.3305448e+01 INCH^4

ROTATION MATRIX from _BL_4_125 orientation to PRINCIPAL AXES:
       0.98999        0.14114
      -0.14114        0.98999

ROTATION ANGLE from _BL_4_125 orientation to PRINCIPAL AXES (degrees):
about z axis          -8.114

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  4.3120484e-01  2.1691621e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                        1           2   COORD
about AXIS 1:  3.77252e-01 INCH^3       -2.2773e+00 -1.3408e+00   INCH
               6.96633e-01 INCH^3        2.4351e+00  7.2607e-01   INCH
about AXIS 2:  4.27340e+00 INCH^3       -2.9952e+00  5.7712e-01   INCH
               4.88791e+00 INCH^3        2.6186e+00 -5.6092e-01   INCH
```

<u>END PRO-ENGINEER SECTION DATA</u>

Where, the “MODULUS”, $z = I/c$ when using the stress equation: $(Mc)/I$

And, the sketch relating to this data is given below:

GTM-T2 BL 4.125 section cut, mod2a
section rotated 8.114deg CCW



Drawing Reference: LD-1164914, LD-1164895-1

Checking the numbers (ref. PRO-ENIGINEER SECTION DATA):

From Line 16:

$$I_1 = 0.5058 in^4 \; - used \;\; for \;\; stresses \; due \; to \; aero-loading, positive \& negative$$

$$I_2 = 12.7996 in^4 \; - used \;\; for \;\; stress \; due \; to \; the \; engine$$

And from line 28:

$$c_2 = 2.995 in - used \; for \; the \; tension \; in \; the \; structure \; due \; to \; engine \; thrust \; rotating \; about \; axis \; 2$$

Checking this gives:

$$z_2 = \frac{I_2}{c_2} = \frac{12.799 in^4}{2.995 in} = 4.273 in^3 \; - used \;\; for \;\; stress \; due \; to \; engine \;\; thrust$$

This value equals the “MODULUS” value on line 28 from the Pro-E data.

For positive wing bending, from line 26:

$$c_1 = 1.34 in$$

Checking this gives:

$$z_1 = \frac{I_1}{c_1} = \frac{0.5058 in^4}{1.3408 in} = 0.377 in^3 \; - used \;\; for \;\; stress \; due \;\; positve \; aero-loading$$

This value equals the “MODULUS” value on line 26 from the Pro-E data.

For negative wing bending, from line 27:

$$c_1 = 0.726 in$$

Checking this gives:

$$z_1 = \frac{I_1}{c_1} = \frac{0.5058 in^4}{0.726 in} = 0.696 in^3 \; - used \;\; for \;\; stress \; due \; negative \; aero-loading$$

This value equals the “MODULUS” value on line 27 from the Pro-E data.

Positive and Negative Wing Bending:

Case 1- C_L = 1.77 (fixed slats & flaps stowed)

From line 2 of the “PRO-ENGINEER SECTION DATA” the shear stress at section C-C is:

$$\tau_{C-C} = \frac{F_{shear}}{A_{shear}}$$

$$\tau_{C-C} = \frac{F_{aero} + F_{thrust}}{A_{shear}}$$

$$\tau_{C-C} = \frac{\sqrt{(139.8 lb)^2 + (20.5 lb)^2}}{2.72 in^2} = \frac{141.3 lb}{2.72 in^2}$$

$$\tau_{C-C} = 52 \, psi$$

The bending stress at section C-C due to the aero load for positive wing bending:

$$\sigma_{\substack{x\\C-C}} = \frac{M_{aeroload}}{z_{x-x}} = \frac{(139.8lb)(BL16.33in - BL4.125in)}{0.377in^3} = \frac{1706.7lb \cdot in}{0.377in^3}$$
$$\sigma_{\substack{x\\C-C}} = 4524.0\,psi$$

The bending stress at section C-C due to the aero load for negative wing bending:

$$\sigma_{\substack{x\\C-C}} = \frac{M_{aeroload}}{z_{x-x}} = \frac{(139.8lb)(BL16.33in - BL4.125in)}{0.696in^3} = \frac{1706.7lb \cdot in}{0.696in^3}$$
$$\sigma_{\substack{x\\C-C}} = 2449.9\,psi$$

The bending stress at section C-C due to engine thrust:

$$\sigma_{\substack{y\\C-C}} = \frac{M_{thrust}}{z_{y-y}} = \frac{(20.5lb)(BL14.2in)}{4.273in^3} = \frac{291.1lb \cdot in}{4.273in^3}$$
$$\sigma_{\substack{y\\C-C}} = 68\,psi$$

Therefore for positive wing bending,

$$\sigma_{\substack{total\\(+)wing}} = \sqrt{(4524.0)^2 + (68)^2} = 4524.4\,psi$$

And for negative wing bending,

$$\sigma_{\substack{total\\(-)wing}} = \sqrt{(2449.9)^2 + (68)^2} = 2450.8\,psi$$

So, for positive wing bending:

$$\sigma_{\substack{(+)Max\\C-C}} = \frac{4524.4\,psi}{2} \pm \sqrt{\left(\frac{4524.4\,psi}{2}\right)^2 + (52\,psi)^2}$$
$$\sigma_{\substack{(+)Max\\C-C}} = 2262.2\,psi + 2262.8\,psi$$
$$\sigma_{\substack{(+)Max\\C-C}} = 4525.0\,psi$$

So, for negative wing bending:

$$\sigma_{\substack{(-)Max\\C-C}} = \frac{2450.8\,psi}{2} \pm \sqrt{\left(\frac{2450.8\,psi}{2}\right)^2 + (52\,psi)^2}$$

$$\sigma_{\substack{(-)Max\\C-C}} = 1225.4\,psi + 1226.5\,psi$$

$$\sigma_{\substack{(-)Max\\C-C}} = 2451.9\,psi$$

<u>Case 2- C_L = 2.18 (fixed slats & flaps deployed)</u>

From line 2 of the "PRO-ENGINEER SECTION DATA" the shear stress at section C-C is:

$$\tau_{C-C} = \frac{F_{shear}}{A_{shear}}$$

$$\tau_{C-C} = \frac{F_{aero} + F_{thrust}}{A_{shear}}$$

$$\tau_{C-C} = \frac{\sqrt{(172.2lb)^2 + (20.5lb)^2}}{2.72in^2} = \frac{173.4lb}{2.72in^2}$$

$$\tau_{C-C} = 64\,psi$$

The bending stress at section C-C due to the aero load for positive wing bending:

$$\sigma_{\substack{x\\C-C}} = \frac{M_{aeroload}}{z_{x-x}} = \frac{(172.2lb)(BL16.33in - BL4.125in)}{0.377in^3} = \frac{2101.9lb \cdot in}{0.377in^3}$$

$$\sigma_{\substack{x\\C-C}} = 5572.0\,psi$$

The bending stress at section C-C due to the aero load for negative wing bending:

$$\sigma_{\substack{x\\C-C}} = \frac{M_{aeroload}}{z_{x-x}} = \frac{(172.2lb)(BL16.33in - BL4.125in)}{0.696in^3} = \frac{2101.9lb \cdot in}{0.696in^3}$$

$$\sigma_{\substack{x\\C-C}} = 3017.3\,psi$$

The bending stress at section C-C due to engine thrust:

$$\sigma_{\substack{y\\C-C}} = \frac{M_{thrust}}{z_{y-y}} = \frac{(20.5lb)(BL14.2in)}{4.273in^3} = \frac{291.1lb \cdot in}{4.273in^3}$$

$$\sigma_{\substack{y\\C-C}} = 68\,psi$$

Therefore for positive wing bending:

$$\sigma_{\substack{total\\(+)wing}} = \sqrt{(5572.0\,psi)^2 + (68\,psi)^2} = 5572.4\,psi$$

And for negative wing bending:

$$\sigma_{\substack{total\\(-)wing}} = \sqrt{(3017.3\,psi)^2 + (68\,psi)^2} = 3018.1\,psi$$

So, for positive wing bending:

$$\sigma_{\substack{(+)Max\\C-C}} = \frac{5572.4\,psi}{2} \pm \sqrt{\left(\frac{5572.4\,psi}{2}\right)^2 + (64\,psi)^2}$$

$$\sigma_{\substack{(+)Max\\C-C}} = 2786.2\,psi + 2786.9\,psi$$

$$\sigma_{\substack{(+)Max\\C-C}} = 5573.2\,psi$$

So, for negative wing bending:

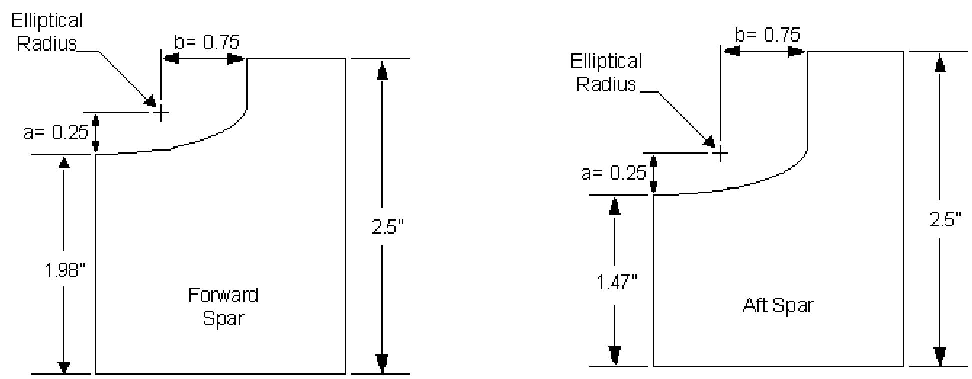
$$\sigma_{\substack{(-)Max\\C-C}} = \frac{3018.1\,psi}{2} \pm \sqrt{\left(\frac{3018.1\,psi}{2}\right)^2 + (64\,psi)^2}$$

$$\sigma_{\substack{(-)Max\\C-C}} = 1509.1\,psi + 1510.4\,psi$$

$$\sigma_{\substack{(-)Max\\C-C}} = 3019.4\,psi$$

A stress concentration factor is needed for the elliptical geometry is needed for the final calculations. From the wing spar geometry the following values are given:

GTM-T2 BL4.125 Stress Concentration
Geometry, Forward & Aft Spars



Drawing Reference: LD-1164894
Using “Stress Concentrations Factors”, by Peterson, © 1974, figure 77:

<u>Forward Spar:</u>
D = 2.495*in*
d = 1.98*in*
a = 0.25*in*
b = 0.75*in*
b/d = 0.75/1.98 = 0.38*in*
b/a = 0.75/0.25 = 3.0*in*
From figure 77: Kt = 1.15

<u>Aft Spar:</u>
D = 2.495*in*
d = 1.47*in*
a =0 .25*in*
b = 0.75*in*
b/d = 0.75/1.47 = 0.51*in*
b/a = 0.75/0.25 = 3.0*in*
From figure 77: Kt = 1.085

Therefore, the average stress concentration value is:

$$Kt_{avg} = (1.15 + 1.09)/2 = 1.12$$

Maximum Wing Bending Stress:

Case 1- C_L = 1.77 (fixed slats & flaps stowed):

For the maximum positive wing bending stress:

$$\sigma_{\underset{(+)wing}{\max}} = Kt_{avg}(\sigma_{\underset{C-C}{(+)Max}}) = (1.12)(4525.0) = 5068\,psi$$

For the maximum negative wing bending stress:

$$\sigma_{\underset{(-)wing}{\max}} = Kt_{avg}(\sigma_{\underset{C-C}{(-)Max}}) = (1.12)(2451.9) = 2746\,psi$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for positive wing bending with C_L = 1.77, at 100*mph*, at 139.8 *lbs*/wing is:

$$\boxed{FS = \frac{7578\,psi}{5068\,psi} = 1.5}$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for negative wing bending with C_L = 1.77, at 100*mph*, at 139.8 *lbs*/wing is:

$$\boxed{FS = \frac{7578\,psi}{2746\,psi} = 2.76}$$

Case 2- C_L = 2.18 (fixed slats & flaps deployed):

For the maximum positive wing bending stress:

$$\sigma_{\underset{(+)wing}{\max}} = Kt_{avg}(\sigma_{\underset{C-C}{(+)Max}}) = (1.12)(5573.2) = 6242\,psi$$

For the maximum negative wing bending stress:

$$\sigma_{\underset{(-)wing}{\max}} = Kt_{avg}(\sigma_{\underset{C-C}{(-)Max}}) = (1.12)(3019.4) = 3382\,psi$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for positive wing bending with C_L = 2.18, at 100*mph*, at 172.2 *lbs*/wing is:

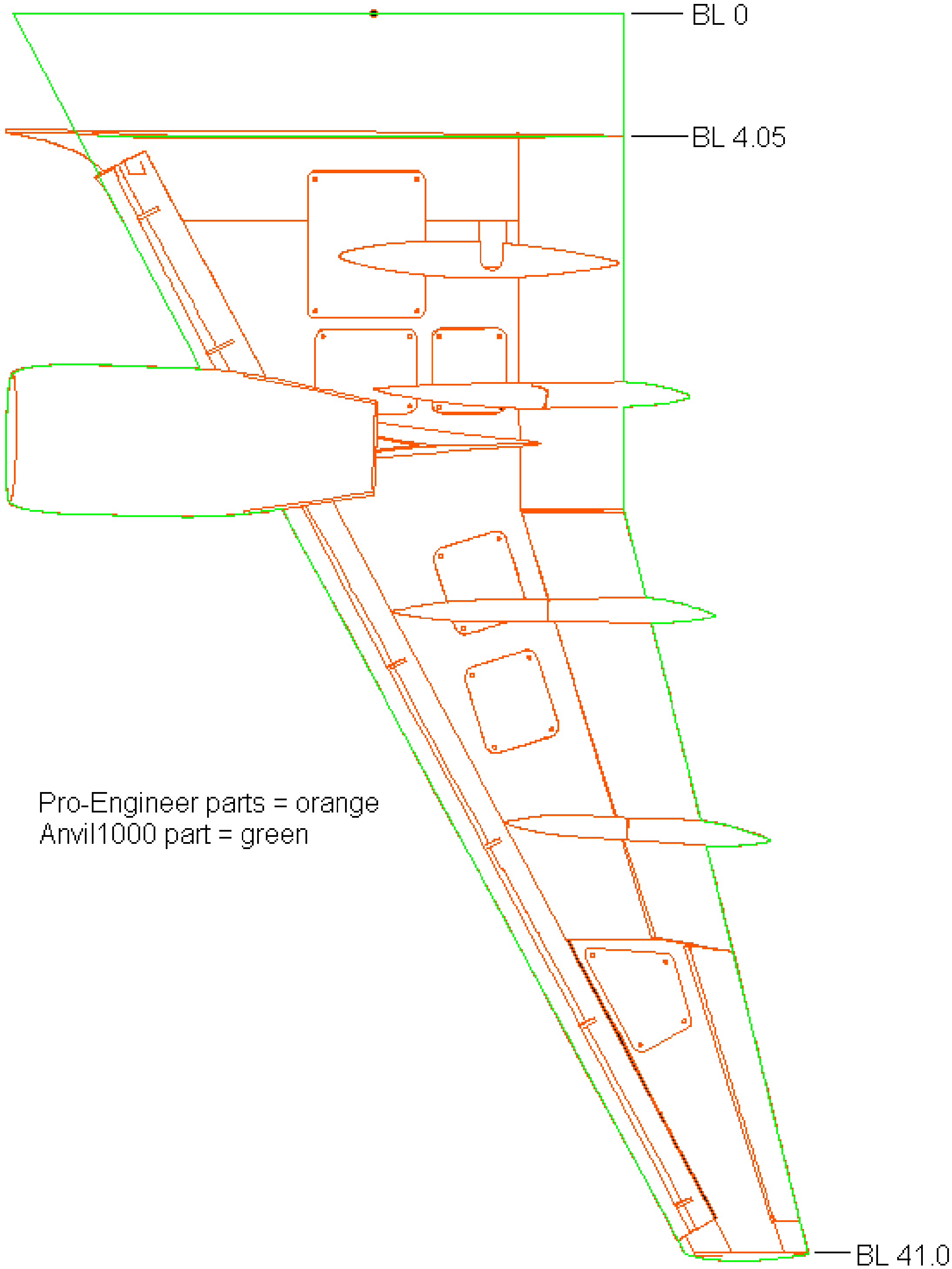
$$\boxed{FS = \frac{7578\,psi}{6242\,psi} = 1.21}$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for negative wing bending with C_L = 2.18, at 100mph, at 172.2 *lbs*/wing is:

$$\boxed{FS = \frac{7578\,psi}{3382\,psi} = 2.24}$$

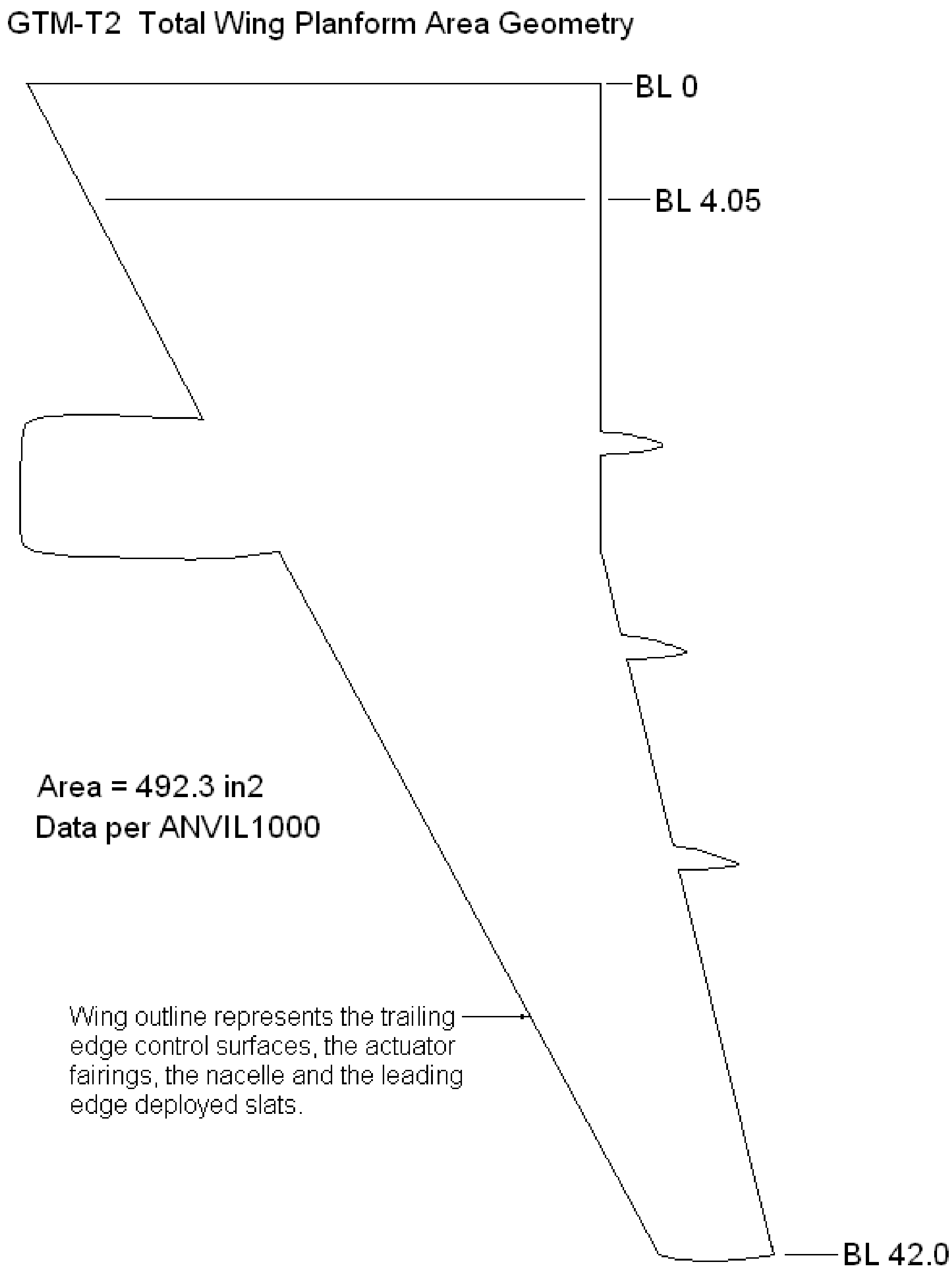
Additional analysis has shown that the surface area could be increased due to additional surface area created by the increased complexity of T2 wing design as illustrated below.

GTM-T2 Wing Outline Comparision



Drawing Number LD-1164906

The total surface area for the T2 wing is shown below.



Drawing Number LD-1164906

Lift Force (Using 984.6in^2 Wing Area):

Wing outline represents the trailing edge control surfaces, the actuator fairings, the nacelle and the leading edge deployed slats.

Case 1- C_L = 1.77 (fixed slats & flaps stowed):

The calculation for the entire wing with extended leading edge slats and the fuselage is indicated below.

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = (984.6in^2) \cdot (1.77) \cdot (0.17761psi)$$
$$F_{Lift} = 309.5lbf$$

F_{lift} acting on each wing at 100mph is 154.8*lbf*

Therefore:

F_{aero} = 154.8*lbs* and F_{thrust} = 20.5 *lbs**

Case 2- C_L = 2.18 (fixed slats & flaps deployed):

The calculation for the entire wing with extended leading edge slats and the fuselage is indicated below.

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = (984.6in^2) \cdot (2.18) \cdot (0.17761psi)$$
$$F_{Lift} = 381.2lbf$$

F_{lift} acting on each wing at 100mph is 190.6*lbf*

Therefore:

F_{aero} = 190.6*lbs* and F_{thrust} = 20.5 *lbs**

**Note: The engine test stand results yielded values higher than 20lbs*

Positive and Negative Wing Bending (Using 984.6in^2 Wing Area)

MAC calculations and Pro-Engineer data are the same as the previous analysis.

Case 1- C_L = 1.77 (fixed slats & flaps stowed)

From line 2 of the "PRO-ENGINEER SECTION DATA" the shear stress at section C-C is:

$$\tau_{C-C} = \frac{F_{shear}}{A_{shear}}$$

$$\tau_{C-C} = \frac{\sqrt{(154.8lb)^2 + (20.5lb)^2}}{2.72in^2} = \frac{156.15lb}{2.72in^2}$$

$$\tau_{C-C} = 57.4\,psi$$

The bending stress at section C-C due to the aero load for positive wing bending:

$$\sigma_{\substack{x \\ C-C}} = \frac{M_{aeroload}}{z_{x-x}} = \frac{(154.8lb)(BL16.33in - BL4.125in)}{0.377in^3} = \frac{1888.9lb \cdot in}{0.377in^3}$$

$$\sigma_{\substack{x \\ C-C}} = 5007\,psi$$

The bending stress at section C-C due to the aero load for negative wing bending:

$$\sigma_{\substack{x \\ C-C}} = \frac{M_{aeroload}}{Z_{x-x}} = \frac{(154.8lb)(BL16.33in - BL4.125in)}{0.696in^3} = \frac{1888.9lb \cdot in}{0.696in^3}$$

$$\sigma_{\substack{x \\ C-C}} = 2711.5\,psi$$

The bending stress at section C-C due to engine thrust:

$$\sigma_{\substack{y \\ C-C}} = \frac{M_{thrust}}{z_{y-y}} = \frac{(20.5lb)(BL14.2in)}{4.273in^3} = \frac{291.1lb \cdot in}{4.273in^3}$$

$$\sigma_{\substack{y \\ C-C}} = 68\,psi$$

Therefore for positive wing bending,

$$\sigma_{\substack{total \\ (+)wing}} = \sqrt{(5007\,psi)^2 + (68\,psi)^2} = 5007.5\,psi$$

And for negative wing bending,

$$\sigma_{\substack{total \\ (-)wing}} = \sqrt{(2711.5psi)^2 + (68psi)^2} = 2712.3psi$$

So, for positive wing bending:

$$\sigma_{\substack{(+)Max \\ C-C}} = \frac{5007.5psi}{2} \pm \sqrt{\left(\frac{5007.5psi}{2}\right)^2 + (57.4psi)^2}$$

$$\sigma_{\substack{(+)Max \\ C-C}} = 2503.7 + 2504.4$$

$$\sigma_{\substack{(+)Max \\ C-C}} = 5008.1psi$$

So, for negative wing bending:

$$\sigma_{\substack{(-)Max \\ C-C}} = \frac{2712.3psi}{2} \pm \sqrt{\left(\frac{2712.3psi}{2}\right)^2 + (57.4psi)^2}$$

$$\sigma_{\substack{(-)Max \\ C-C}} = 1356.2 + 1357.4$$

$$\sigma_{\substack{(-)Max \\ C-C}} = 2713.5psi$$

<u>Case 2- C_L = 2.18 (fixed slats & flaps deployed)</u>

From line 2 of the "PRO-ENGINEER SECTION DATA" the shear stress at section C-C is:

$$\tau_{C-C} = \frac{F_{shear}}{A_{shear}}$$

$$\tau_{C-C} = \frac{\sqrt{(190.6lb)^2 + (20.5lb)^2}}{2.72in^2} = \frac{191.7lb}{2.72in^2}$$

$$\tau_{C-C} = 70.5psi$$

The bending stress at section C-C due to the aero load for positive wing bending:

$$\sigma_{\substack{x \\ C-C}} = \frac{M_{aeroload}}{z_{x-x}} = \frac{(190.6lb)(BL16.33in - BL4.125in)}{0.377in^3} = \frac{2326.3lb \cdot in}{0.377in^3}$$

$$\sigma_{\substack{x \\ C-C}} = 6166.8psi$$

The bending stress at section C-C due to the aero load for negative wing bending:

$$\sigma_{\substack{x \\ C-C}} = \frac{M_{aeroload}}{z_{x-x}} = \frac{(190.6lb)(BL16.33in - BL4.125in)}{0.696in^3} = \frac{2326.3lb \cdot in}{0.696in^3}$$

$$\sigma_{\substack{x \\ C-C}} = 3339.6\,psi$$

The bending stress at section C-C due to engine thrust:

$$\sigma_{\substack{y \\ C-C}} = \frac{M_{thrust}}{z_{y-y}} = \frac{(20.5lb)(BL14.2in)}{4.273in^3} = \frac{291.1lb \cdot in}{4.273in^3}$$

$$\sigma_{\substack{y \\ C-C}} = 68\,psi$$

Therefore for positive wing bending:

$$\sigma_{\substack{total \\ (+)wing}} = \sqrt{(6166.8\,psi)^2 + (68\,psi)^2} = 6167.2\,psi$$

And for negative wing bending:

$$\sigma_{\substack{total \\ (-)wing}} = \sqrt{(3339.6\,psi)^2 + (68\,psi)^2} = 3340.2\,psi$$

So, for positive wing bending:

$$\sigma_{\substack{(+)Max \\ C-C}} = \frac{6167.2\,psi}{2} \pm \sqrt{\left(\frac{6167.2\,psi}{2}\right)^2 + (70.5\,psi)^2}$$

$$\sigma_{\substack{(+)Max \\ C-C}} = 3083.6 + 3084.4$$

$$\sigma_{\substack{(+)Max \\ C-C}} = 6168.0\,psi$$

So, for negative wing bending:

$$\sigma_{\substack{(-)Max \\ C-C}} = \frac{3340.2\,psi}{2} \pm \sqrt{\left(\frac{3340.2\,psi}{2}\right)^2 + (70.5\,psi)^2}$$

$$\sigma_{\substack{(-)Max \\ C-C}} = 1670.1 + 1671.6$$

$$\sigma_{\substack{(-)Max \\ C-C}} = 3341.7\,psi$$

Maximum Wing Bending Stress (Using 984.6in^2 Wing Area)

The average stress concentration value is the same as the prior analysis, where:

$$Kt_{avg} = (1.15 + 1.09)/2 = 1.12$$

Case 1- C_L = 1.77 (fixed slats & flaps stowed):

For the maximum positive wing bending stress:

$$\sigma_{\substack{\max \\ (+)wing}} = Kt_{avg}(\sigma_{\substack{(+)Max \\ C-C}}) = (1.12)(5008.1) = 5609\,psi$$

For the maximum negative wing bending stress:

$$\sigma_{\substack{\max \\ (-)wing}} = Kt_{avg}(\sigma_{\substack{(-)Max \\ C-C}}) = (1.12)(2713.5) = 3039\,psi$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for positive wing bending with C_L = 1.77, at 100*mph*, at 154.8*lbs*/wing is:

$$\boxed{FS = \frac{7578\,psi}{5609\,psi} = 1.35}$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for negative wing bending with C_L = 1.77, at 100*mph*, at 154.8*lbs*/wing is:

$$\boxed{FS = \frac{7578\,psi}{3039\,psi} = 2.49}$$

Case 2- C_L = 2.18 (fixed slats & flaps deployed):

For the maximum positive wing bending stress:

$$\sigma_{\substack{\max \\ (+)wing}} = Kt_{avg}(\sigma_{\substack{(+)Max \\ C-C}}) = (1.12)(6168.0) = 6908\,psi$$

For the maximum negative wing bending stress:

$$\sigma_{\substack{\max \\ (-)wing}} = Kt_{avg}(\sigma_{\substack{(-)Max \\ C-C}}) = (1.12)(3341.7) = 3743\,psi$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for positive wing bending with C_L = 2.18, at 100mph, at 190.6*lbs*/wing is:

$$\boxed{FS = \frac{7578\,psi}{6908\,psi} = 1.10}$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* (reference project document number GTMP-6031), for negative wing bending with C_L = 2.18, at 100mph, at 190.6*lbs*/wing is:

$$\boxed{FS = \frac{7578\,psi}{3743\,psi} = 2.02}$$

Positive Wing Loading, Maximum Velocity and Bank Angle with F.S. = 1.01

Calculations for maximum stresses are given, assuming yield strength of aircraft plywood to be 7578*psi*. The shear stress & stress due to engine thrust at Section C-C is considered negligible. The positive wing loading conditions, with a stress concentration factor (Kt_{avg} = 1.12), is considered below:

$$F.S. = 1.01 = \frac{7578}{\sigma_{allow}}$$

$$\therefore \sigma_{allow} = 7503\,psi = \sigma_{total} K_t$$

$$7503 = \sigma_{total}(1.12)$$

$$therefore,$$

$$\sigma_{total} = 6699\,psi$$

$$and,$$

$$6699\,psi = \frac{M}{z}$$

Where M is the bending moment at BL 4.125 and $z = I / c$, then:

$$6699\,psi = \frac{M}{z} = \frac{F_{Lift}\left(BL16.33in - BL4.125in\right)}{0.377in^3} = F_{Lift}(32.35in^2)$$

$$And\ \ solving\ for\ F_{Lift}\ (\,for\ one\ wing)\ gives:$$

$$F_{Lift} = 207.1\,{}^{lbs}\!/\!{}_{wing}$$

For both wings, the lift force = 414.2 *lbs* this equates to a 1 *"g" load* of:

$$"g"\ loading = \frac{414.2lbs}{55lbs} = 7.53$$

<u>Case 1-C_L = 1.77 (fixed slats & flaps stowed):</u>

Velocity is a variable to the equation for lift, which is:
Equation reference: "Introductory to Aircraft Performance" James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2002

$$L = {}^{1}\!/\!{}_{2}(\rho)(V^2)(S)(C_L)$$

Where:
ρ = .00237 *slugs/ft*^{3}
C_L= 1.77(Assumed Maximum)
S = 6.027 ft^2
L in *lbs*
V in *ft/s*

Solving for Velocity V, from the lift equation yields:

$$L = 414.2lbs = 0.5(0.00237\,{}^{slugs}\!/\!{}_{ft^3})(V^2)(6.027\,ft^2)(1.77)$$

$$And\ \ solving\ \ for\ V\ \ gives,$$

$$V^2 = 32765.5$$

$$V = \sqrt{32765.5}$$

$$V = 181.0\,ft/\sec = 123mph$$

For the T2 flying at a speed of 123*mph*, the load factor n, is equivalent to:

$$n = \frac{L}{W_{\max}} = \frac{414.2lbs}{55\,lbs} = 7.53$$

The bank angle calculations assume constant altitude, constant turning radius, and constant turning rate.

Equation reference: "Introductory to Aircraft Performance" James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2001

$$Bank\ Angle\ \theta \Rightarrow \cos\theta = \frac{1}{n}$$

$$\theta = \cos^{-1}\left(\frac{1}{7.53}\right) = 82.4^{\circ}$$

The corresponding turning radius for a load factor of 7.53 is calculated using the following equations:

$$R = \frac{2W}{\rho\, g\, S\, C_L}\left[\frac{n}{\sqrt{n^2-1}}\right] = \frac{2(55lbs)}{(0.00237\frac{slugs}{ft^3})(32.2\ ft/s^2)(6.027\ ft^2)(1.77)}\left[\frac{7.53}{\sqrt{7.53^2-1}}\right]$$

$$R = (135.118)(1.00894)$$

$$R = 136.3\,ft$$

<u>Case 2- C_L = 2.18 (fixed slats & flaps deployed):</u>

Velocity is a variable to the equation for lift, which is:
Equation reference: "Introductory to Aircraft Performance" James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2002

$$L = \frac{1}{2}(\rho)(V^2)(S)(C_L)$$

Where:
ρ = .00237 *slugs/ft*^{3}
C_L= 2.18 (Assumed Maximum)
S = 6.027 *ft*^{2}
L in *lbs*
V in *ft/s*

Solving for Velocity V, from the lift equation yields:

$$L = 414.2 = 0.5(0.00237\ slugs/ft^3)(V^2)(6.027\ ft^2)(2.18)$$

$$Ans \backslash d\ \ solving\ \ for\ V\ \ gives,$$

$$V^2 = 26603.2$$

$$V = \sqrt{26603.2}$$

$$V = 163.0\ ft/\sec = 111 mph$$

For the T2 flying at a speed of 111 *mph*, the load factor n, is equivalent to:

$$n = \frac{L}{W_{\max}} = \frac{414.2\,lbs}{55\,lbs} = 7.53$$

The bank angle calculations assume constant altitude, constant turning radius, and constant turning rate.
Equation reference: “Introductory to Aircraft Performance” James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2001

$$Bank\ Angle\ \theta \Rightarrow \cos\theta = \frac{1}{n}$$

$$\theta = \cos^{-1}\left(\frac{1}{7.53}\right) = 82.4^{\circ}$$

The corresponding turning radius for a load factor of 7.53 is calculated using the following equations:

$$R = \frac{2W}{\rho\, g\, S\, C_L}\left[\frac{n}{\sqrt{n^2-1}}\right] = \frac{2(55lbs)}{(0.00237\,\frac{slugs}{ft^3})(32.2\ ft/s^2)(6.027\ ft^2)(2.18)}\left[\frac{7.53}{\sqrt{7.53^2-1}}\right]$$

$$R = (109.706)(1.00894)$$

$$R = 111\,ft$$

Negative Wing Loading, Maximum Velocity and Bank Angle with F.S. = 1.01

Calculations for maximum stresses are given, assuming yield strength of aircraft plywood to be 7578*psi*. The shear stress & stress due to engine thrust at Section C-C is considered negligible. The positive wing loading conditions, with a stress concentration factor (Kt_{avg} = 1.12) is considered below:

$$F.S. = 1.01 = \frac{7578}{\sigma_{allow}}$$

$$\therefore \sigma_{allow} = 7503\,psi = \sigma_{total} K_t$$

$$7503 = \sigma_{total}(1.12)$$

$$therefore,$$

$$\sigma_{total} = 6699\,psi$$

$$and,$$

$$6699\,psi = \frac{M}{z}$$

Where *M* is the bending moment at BL 4.125 and z = c / I, then,

$$6699\,psi = \frac{M}{z} = \frac{F_{Lift}\left(BL16.33in - BL4.125in\right)}{0.696in^3} = F_{Lift}\,(17.52in)$$

$$And\ \ solving\ \ for\ F_{Lift}\ (\ for\ one\ wing)\ \ gives:$$

$$F_{Lift} = 382\,{}^{lbs}\!/\!{}_{wing}$$

For both wings, the lift force = 765 *lbs*. This equates to a 1*"g" load* of:

$$"g"\ loading = \frac{765lbs}{55lbs} = 13.9$$

<u>Case 1- C_L = 1.77 (fixed slats & flaps stowed):</u>

Velocity is a variable to the equation for lift, which is:
Equation reference: "Introductory to Aircraft Performance" James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2002

$$L = {}^1\!/\!{}_2\,(\rho)(V^2)(S)(C_L)$$

Where:
ρ = .00237 *slugs/ft*^{3}
C_L= 1.77(Assumed Maximum)
S = 6.027 *ft*^{2}
L in *lbs*
V in *ft/s*

Solving for Velocity V, from the lift equation yields:

$$L = 765lb = 0.5(0.00237\,{}^{slug}\!/\!{}_{ft^3})(V^2)(6.027)(1.77)$$

$$And\ \ solving\ \ for\ V\ \ gives,$$

$$V^2 = 60494.9$$

$$V = \sqrt{60494.9}$$

$$V = 246\,ft/\sec = 168mph$$

For the T2 flying at a speed of 168 *mph*, the load factor n, is equivalent to:

$$n = \frac{L}{W_{\max}} = \frac{765lbs}{55\,lbs} = 13.9$$

The bank angle calculations assume constant altitude, constant turning radius, and constant turning rate.

Equation reference: "Introductory to Aircraft Performance" James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2001

$$Bank\ Angle\ \theta \Rightarrow \cos\theta = \frac{1}{n}$$

$$\theta = \cos^{-1}\left(\frac{1}{13.9}\right) = 85.9^{\circ}$$

The corresponding turning radius for a load factor of 13.9 is calculated using the following equations:

$$R = \frac{2W}{\rho\, g\, S\, C_L}\left[\frac{n}{\sqrt{n^2-1}}\right] = \frac{2(55 lbs)}{(.00237\frac{slugs}{ft^3})(32.2\ {ft}/{s^2})(6.027\ ft^2)(1.77)}\left[\frac{13.9}{\sqrt{13.9^2-1}}\right]$$

$$R = (135.118)(1.0026)$$

$$R = 135.5\, ft$$

<u>Case 2-C_L = 2.18 (fixed slats & flaps deployed):</u>

Velocity is a variable to the equation for lift, which is:
Equation reference: "Introductory to Aircraft Performance" James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2002

$$L = {1}/{2}(\rho)(V^2)(S)(C_L)$$

Where:
ρ = .00237 *slugs/ft*^{3}
C_L= 2.18 (Assumed Maximum)
S = 6.027 *ft*^{2}
L in *lbs*
V in *ft/s*

Solving for Velocity *V*, from the lift equation yields:

$$L = 765 lb = 0.5(0.00237\ {slug}/{ft^3})(V^2)(6.027\ ft^2)(2.18)$$

$$And\ \ solving\ \ for\ V\ \ gives,$$

$$V^2 = 49117.4$$

$$V = \sqrt{49117.4}$$

$$V = 222\ ft/\sec = 151 mph$$

For the T2 flying at a speed of 151 *mph*, the load factor *n*, is equivalent to:

$$n = \frac{L}{W_{\max}} = \frac{765 lbs}{55\, lbs} = 13.9$$

The bank angle calculations assume constant altitude, constant turning radius, and constant turning rate.

Equation reference: "Introductory to Aircraft Performance" James F. Marchman III, professor of Aerospace and Ocean Engineering Virginia Tech, 2001

$$Bank\ Angle\ \theta \Rightarrow \cos\theta = \frac{1}{n}$$

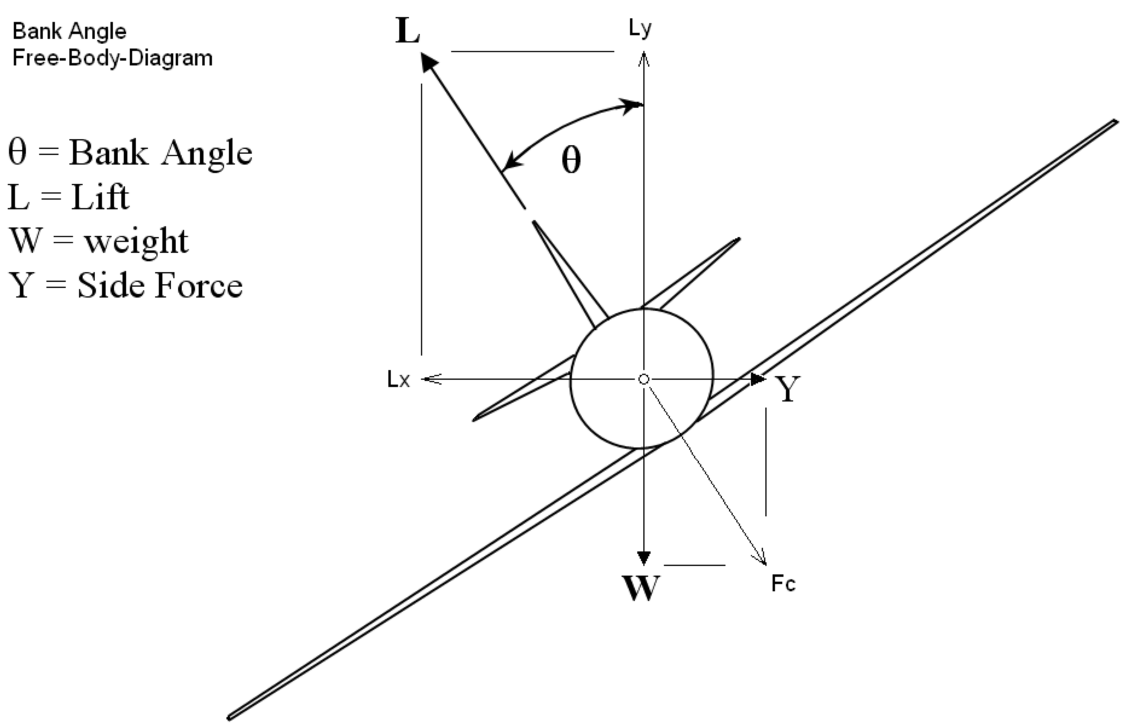
$$\theta = \cos^{-1}\left(\frac{1}{13.9}\right) = 85.9^{\circ}$$

The corresponding turning radius for a load factor of 13.9 is calculated using the following equation:

$$R = \frac{2W}{\rho\, g\, S\, C_L}\left[\frac{n}{\sqrt{n^2 - 1}}\right] = \frac{2(55 lbs)}{(.00237\frac{slugs}{ft^3})(32.2\ {ft}/{s^2})(6.027\ ft^2)(2.18)}\left[\frac{13.9}{\sqrt{13.9^2 - 1}}\right]$$

$$R = (109.706)(1.0026)$$

$$R = 110\, ft$$



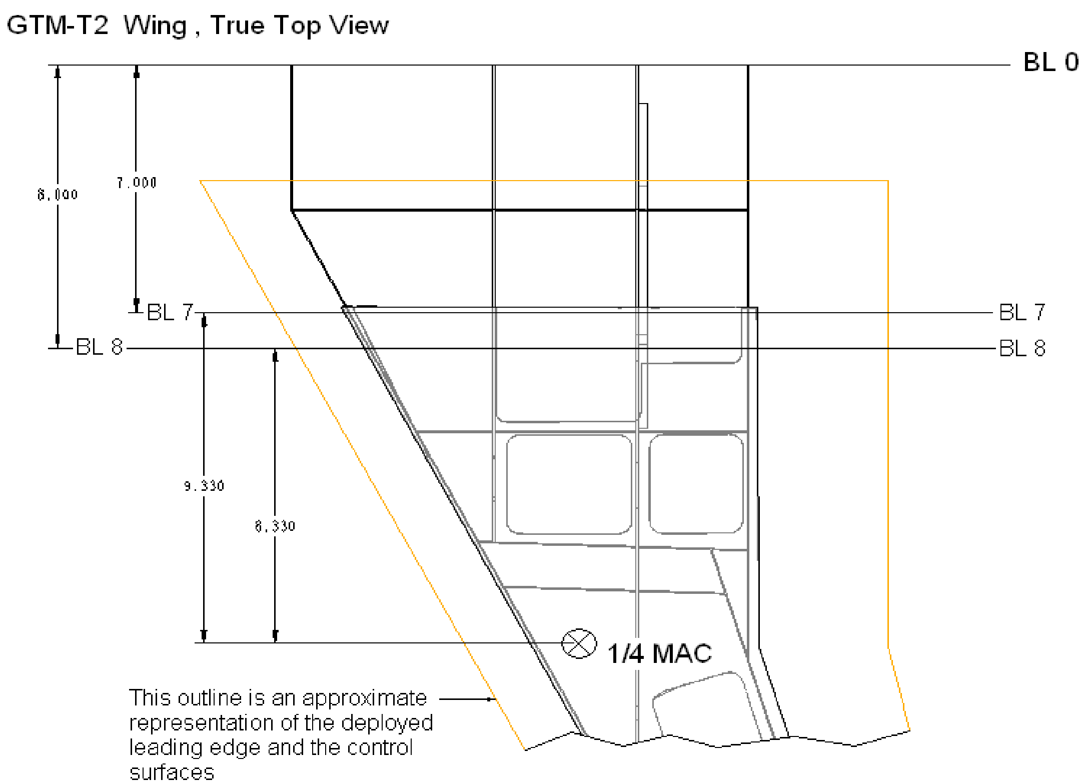
Wing Section Cuts

Homogeneous material equations have been used in calculating a safety factor for the carbon fiber in the bending stress calculation.

Wing Section Cut BL_7 Stress Analysis

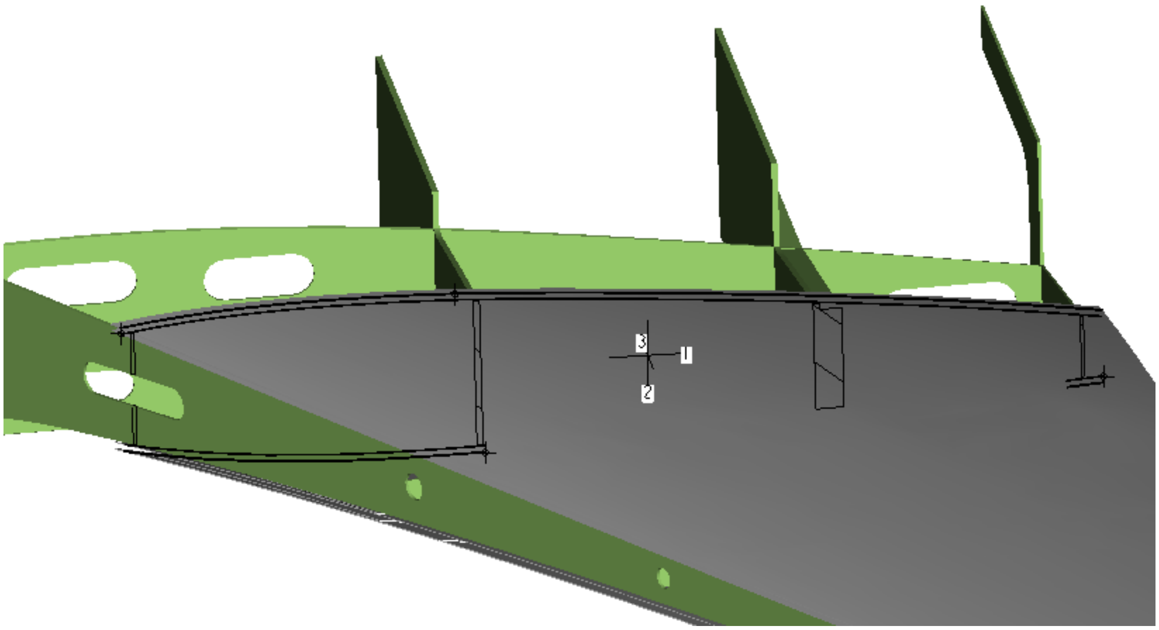
Case 1: C_{Lmax}=1.77

It is assumed that the aero load acts at the quarter MAC (BL 16.423) as shown below:



Drawing reference no. LD-116489

The section cut BL_7 is shown below:



Drawing reference no. LD-1164893

The section cut BL_7 has the following Pro-Engineer properties:

<u>START PRO-E DATA</u>

```
AREA =  9.6760354e-01 INCH^2

CENTER OF GRAVITY with respect to _BL_7 coordinate frame:
X   Y     1.0408306e+01  5.6954123e+01  INCH

INERTIA with respect to _BL_7 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.1475332e+03 -5.7338283e+02
Iyx Iyy -5.7338283e+02  1.0516017e+02

POLAR MOMENT OF INERTIA:  3.2526934e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _BL_7 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  8.8477986e+00  2.0863528e-01
Iyx Iyy  2.0863528e-01  3.3693022e-01

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  3.3181881e-01  8.8529100e+00

POLAR MOMENT OF INERTIA:  9.1847288e+00 INCH^4

ROTATION MATRIX from _BL_7 orientation to PRINCIPAL AXES:
      -0.02449       -0.99970
```

```
      0.99970       -0.02449

ROTATION ANGLE from _BL_7 orientation to PRINCIPAL AXES (degrees):
about z axis         91.403

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  5.8560095e-01  3.0247835e+00  INCH


SECTION MODULI and corresponding points:

                   MODULUS                   1          2   COORD
about AXIS 1:  3.80645e-01 INCH^3       -2.1719e+00 -8.7173e-01   INCH
               3.15811e-01 INCH^3       -1.8339e+00  1.0507e+00   INCH
about AXIS 2:  1.49160e+00 INCH^3       -5.9352e+00 -6.3250e-01   INCH
               1.72236e+00 INCH^3        5.1400e+00  5.7629e-01   INCH
```

END PRO-E DATA

The lift force on one wing at 100*mph* (assume C_{Lmax}=1.77 for deployed slats and stowed flaps, reference NACA TR 427) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3 in^2\right) \cdot (1.77) \cdot (0.1776 psi)$$
$$F_{Lift} = 155\ lbf$$

Therefore the shear stress at section cut BL_7 is:

$$\tau_{BL\_7} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{BL\_7} = \frac{155 lbs}{0.9676 in^2}$$
$$\tau_{BL\_7} = 160 psi$$

The bending stress at section BL_7 due to the aero load is:

$$\sigma_{BL\_7} = \frac{M_{aeroload}}{z_{BL\_7}} = \frac{(155 lbf)(9.33 in)}{0.3158 in^3}$$
$$\sigma_{BL\_7} = 4572.2 psi$$

The maximum shear stress at section BL_7 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{BL\_7}}{2}\right)^2 + \tau_{BL\_7}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{4572.2\,psi}{2}\right)^2 + (160\,psi)^2}$$

$$\tau_{Max} = 2292\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{2292\,psi} = 13.1}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ BL\_7}} = \frac{4572.2\,psi}{2} \pm \sqrt{\left(\frac{4572.2\,psi}{2}\right)^2 + (160\,psi)^2}$$

$$\sigma_{\substack{Max \\ BL\_7}} = 2286.1 + 2291.7$$

$$\sigma_{\substack{Max \\ BL\_7}} = 4578\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{4578\,psi} = 6.57}$$

Wing Section Cut BL_7 Stress Analysis

Case 2: C_{Lmax}=2.18

It is assumed that the aero load acts at the quarter MAC (BL 16.423).
The section cut BL_7 is shown above. The section cut BL_7 has Pro-Engineer properties also shown above. The lift force on one wing at 100*mph* (assume C_{Lmax}=2.18 for deployed slats and deployed flaps, reference NACA TR 427) is,

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$

$$F_{Lift} = \left(492.3\,in^2\right) \cdot (2.18) \cdot (0.1776\,psi)$$

$$F_{Lift} = 191\ lbf$$

Therefore the shear stress at section cut BL_7 is:

$$\tau_{BL\_7} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{BL\_7} = \frac{191 lbs}{0.9676 in^2}$$
$$\tau_{BL\_7} = 197\,psi$$

The bending stress at section BL_7 due to the aero load is:

$$\sigma_{BL\_7} = \frac{M_{aeroload}}{z_{BL\_7}} = \frac{(191 lbf)(9.33 in)}{0.3158 in^3}$$
$$\sigma_{BL\_7} = 5631.3\,psi$$

The maximum shear stress at section BL_7 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{BL\_7}}{2}\right)^2 + \tau_{BL\_7}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{5631.3\,psi}{2}\right)^2 + (197\,psi)^2}$$
$$\tau_{Max} = 2823\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{2823\,psi} = 10.65}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ BL\_7}} = \frac{5631.3\,psi}{2} \pm \sqrt{\left(\frac{5631.3\,psi}{2}\right)^2 + (197\,psi)^2}$$
$$\sigma_{\substack{Max \\ BL\_7}} = 2815.7 + 2822.5$$
$$\sigma_{\substack{Max \\ BL\_7}} = 5638\,psi$$

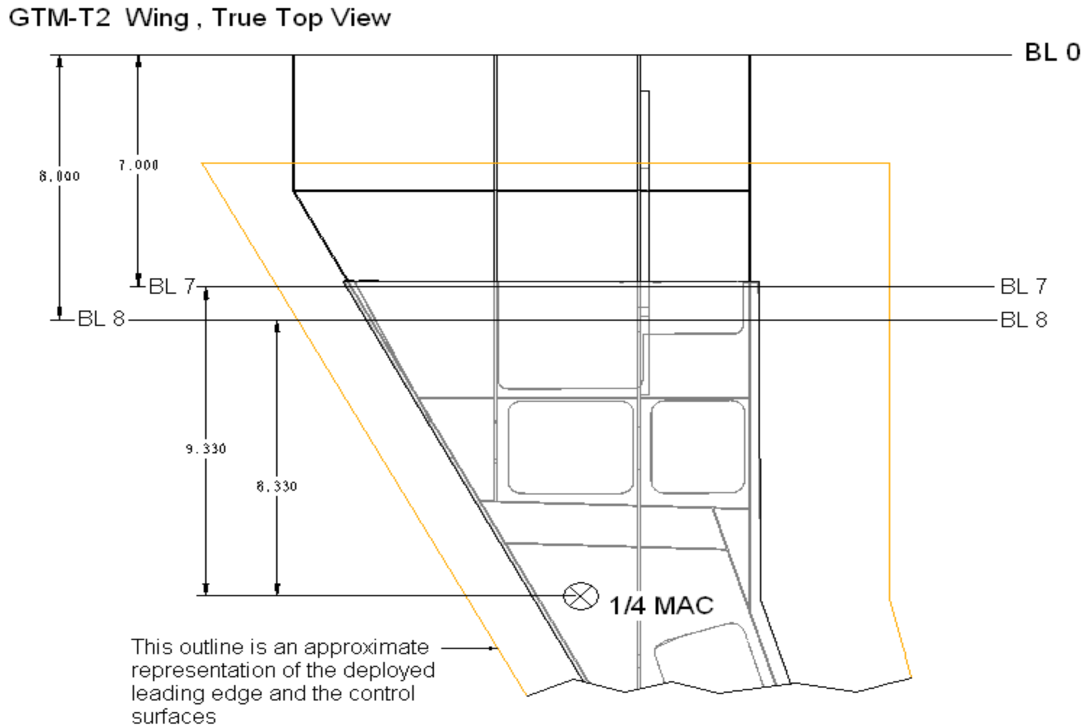
And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{5638\,psi} = 5.33}$$

Wing Section Cut BL_8 Stress Analysis

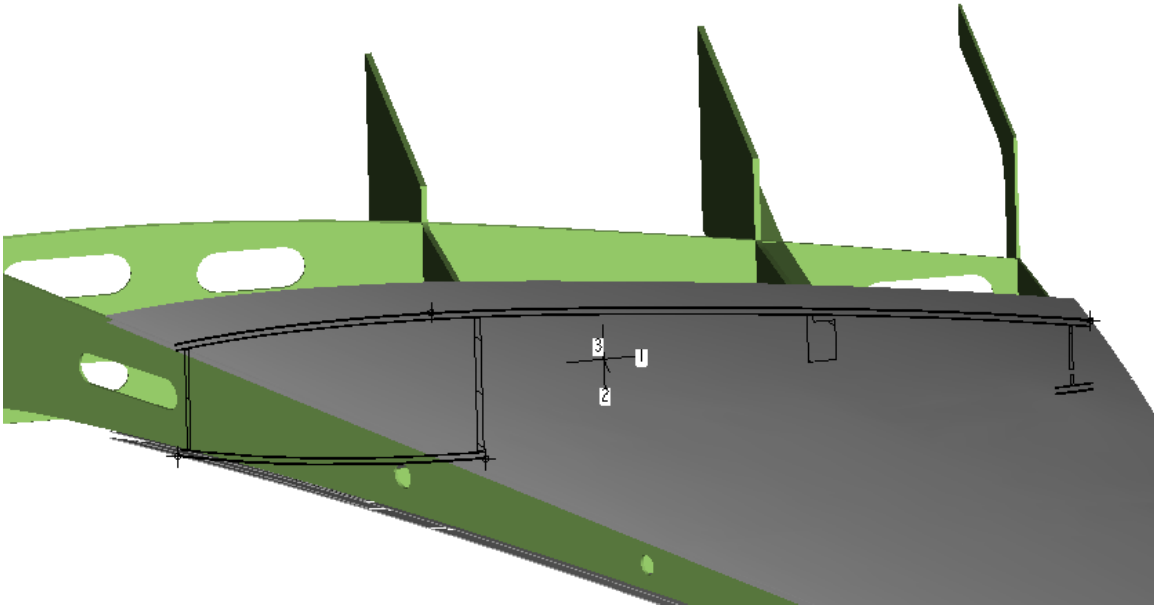
Case 1: C_{Lmax}=1.77

It is assumed that the aero load acts at the quarter MAC (BL 16.423) as shown below



Drawing reference no. LD-1164893

The section cut BL_8 is shown below:



Drawing reference no. LD-1164893

The section cut BL_8 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  7.1503938e-01 INCH^2

CENTER OF GRAVITY with respect to _BL_8 coordinate frame:
X   Y     1.0696010e+01  5.6481969e+01  INCH

INERTIA with respect to _BL_8 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.2879725e+03 -4.3207808e+02
Iyx Iyy -4.3207808e+02  8.2015191e+01

POLAR MOMENT OF INERTIA:  2.3699877e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _BL_8 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  6.8446824e+00 -1.0012630e-01
Iyx Iyy -1.0012630e-01  2.1137919e-01

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.0986818e-01  6.8461934e+00

POLAR MOMENT OF INERTIA:  7.0560616e+00 INCH^4
```

```
ROTATION MATRIX from _BL_8 orientation to PRINCIPAL AXES:
       0.01509       -0.99989
       0.99989        0.01509

ROTATION ANGLE from _BL_8 orientation to PRINCIPAL AXES (degrees):
about z axis          89.135

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  5.4176172e-01  3.0942799e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  2.70935e-01 INCH^3       -1.9584e+00 -7.7461e-01   INCH
               1.94317e-01 INCH^3       -1.4034e+00  1.0800e+00   INCH
about AXIS 2:  1.38719e+00 INCH^3       -4.9353e+00  6.8617e-01   INCH
               1.22213e+00 INCH^3        5.6019e+00  1.0370e-01   INCH
```

<u>END PRO-E DATA</u>

The lift force on one wing at 100*mph* (assume C_{Lmax}=1.77 for deployed slats and stowed flaps, reference NACA TR 427) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3 in^2\right) \cdot (1.77) \cdot (0.1776\, psi)$$
$$F_{Lift} = 155\ lbf$$

Therefore the shear stress at section cut BL_8 is:

$$\tau_{BL\_8} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{BL\_8} = \frac{155 lbs}{0.715 in^2}$$
$$\tau_{BL\_8} = 217\, psi$$

The bending stress at section BL_8 due to the aero load is:

$$\sigma_{BL\_8} = \frac{M_{aeroload}}{z_{BL\_8}} = \frac{(155 lbf)(8.33 in)}{0.1943 in^3}$$
$$\sigma_{BL\_8} = 6635\, psi$$

The maximum shear stress at section BL_8 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{BL\_8}}{2}\right)^2 + \tau_{BL\_8}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{6635\,psi}{2}\right)^2 + (217\,psi)^2}$$

$$\tau_{Max} = 3325\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{3325\,psi} = 9.04}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\BL\_8}} = \frac{6635\,psi}{2} \pm \sqrt{\left(\frac{6635\,psi}{2}\right)^2 + (217\,psi)^2}$$

$$\sigma_{\substack{Max\\BL\_8}} = 3317.2 + 3324.3$$

$$\sigma_{\substack{Max\\BL\_8}} = 6642\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{6642\,psi} = 4.53}$$

Wing Section Cut BL_8 Stress Analysis

Case 2: C_{Lmax}=2.18

It is assumed that the aero load acts at the quarter MAC (BL 16.423).
The section cut BL_8 is shown above. The section cut BL_8 has Pro-Engineer properties also shown above. The lift force on one wing at 100*mph* (assume C_{Lmax}=2.18 for deployed slats and deployed flaps, reference NACA TR 427) is,

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$

$$F_{Lift} = \left(492.3\,in^2\right)\cdot(2.18)\cdot(0.1776\,psi)$$

$$F_{Lift} = 191\ lbf$$

Therefore the shear stress at section cut BL_8 is:

$$\tau_{BL\_8} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{BL\_8} = \frac{191 lbs}{0.715 in^2}$$
$$\tau_{BL\_8} = 267\ psi$$

The bending stress at section BL_8 due to the aero load is:

$$\sigma_{BL\_8} = \frac{M_{aeroload}}{z_{BL\_8}} = \frac{(191 lbf)(8.33 in)}{0.1943 in^3}$$
$$\sigma_{BL\_8} = 8172\ psi$$

The maximum shear stress at section BL_8 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{BL\_8}}{2}\right)^2 + \tau_{BL\_8}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{8172\, psi}{2}\right)^2 + (267\, psi)^2}$$
$$\tau_{Max} = 4095\ psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\, psi}{4095\, psi} = 7.35}$$

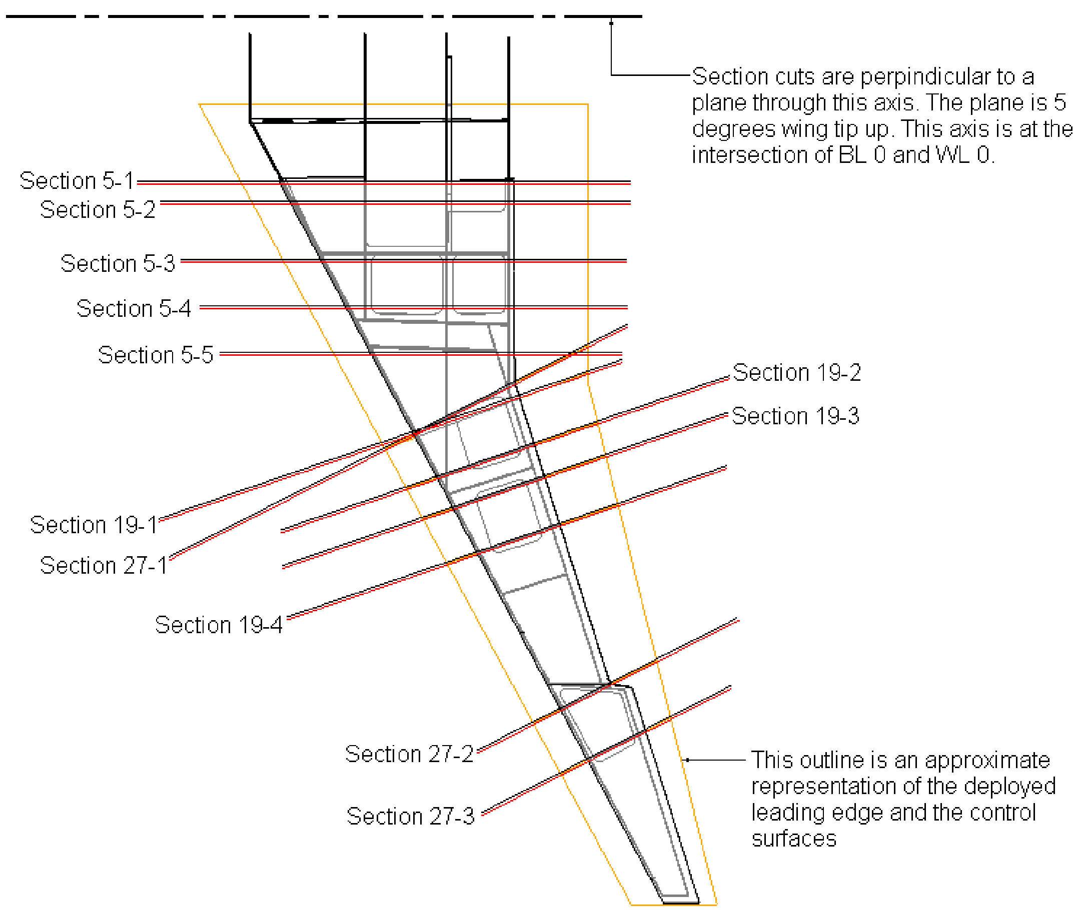
Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ BL\_8}} = \frac{8172\, psi}{2} \pm \sqrt{\left(\frac{8172\, psi}{2}\right)^2 + (267\, psi)^2}$$
$$\sigma_{\substack{Max \\ BL\_8}} = 4085.6 + 4094.3$$
$$\sigma_{\substack{Max \\ BL\_8}} = 8180\ psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{8180\,psi} = 3.68}$$

GTM-T2 Wing Section Cut Locations, 5 degrees WL 0 Offset

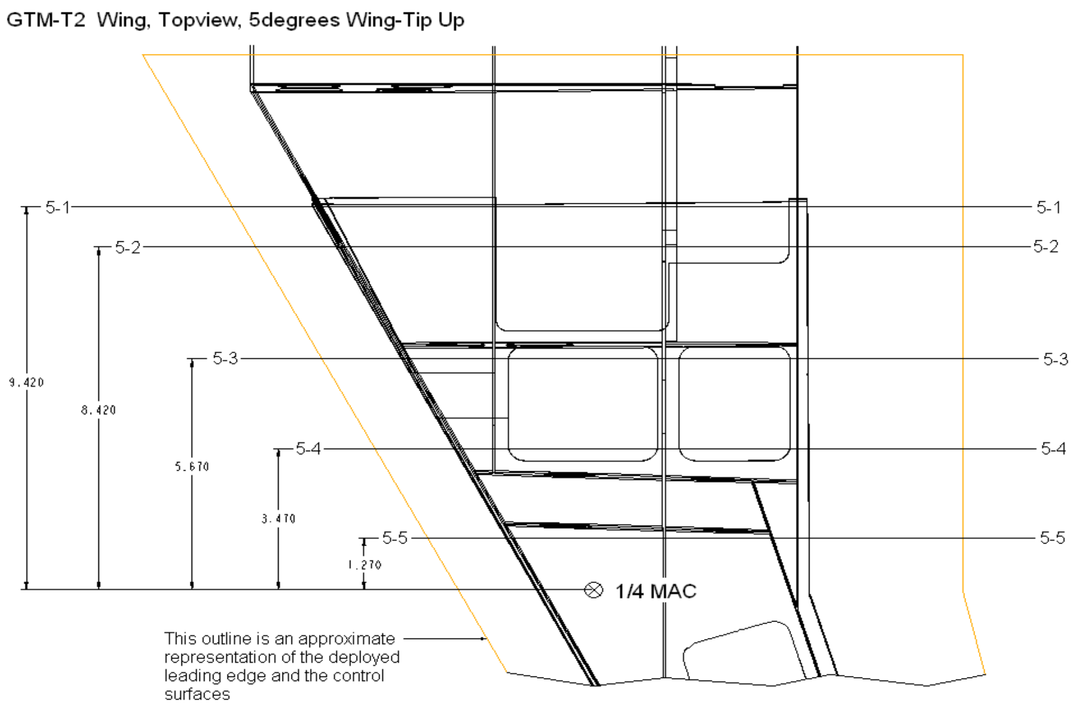


Drawing reference no. LD-1164893

Wing Section Cut 5_1 Stress Analysis

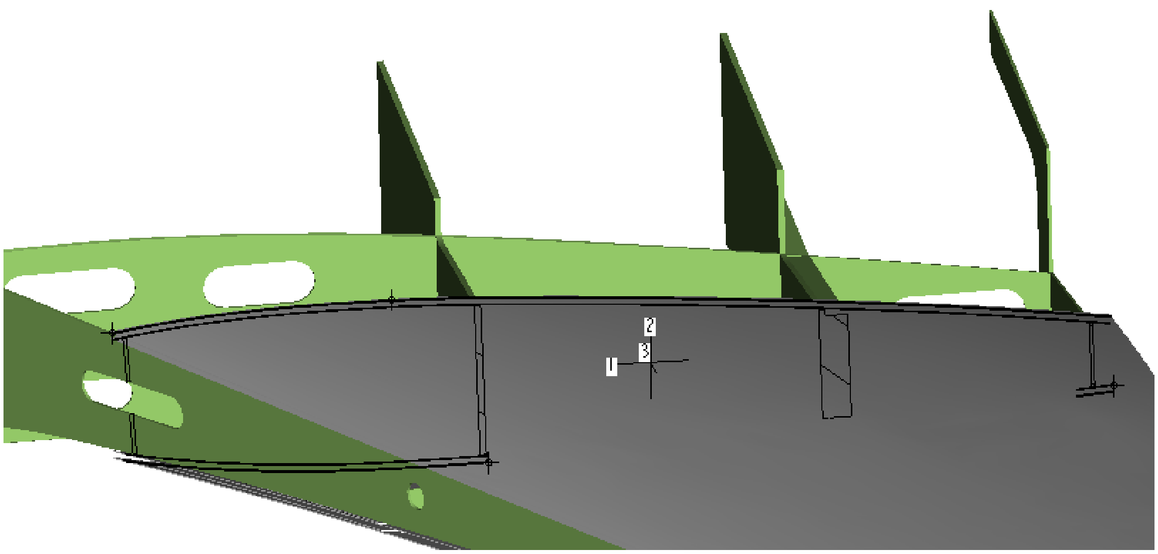
Case 1: C_{Lmax}=1.77

It is assumed that the aero load acts at the quarter MAC (BL 16.423) as shown below:



Drawing reference no. LD-1164893

The section cut 5_1 is shown below:



Drawing reference no. LD-1164893

The section cut 5_1 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  9.5724860e-01 INCH^2

CENTER OF GRAVITY with respect to _5_1 coordinate frame:
X   Y    -5.6939827e+01  9.7646023e+00  INCH

INERTIA with respect to _5_1 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.1600116e+01  5.3199842e+02
Iyx Iyy  5.3199842e+02  3.1123549e+03

POLAR MOMENT OF INERTIA:  3.2039550e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _5_1 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.2889447e-01 -2.2678968e-01
Iyx Iyy -2.2678968e-01  8.8171714e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  3.2283943e-01  8.8232265e+00

POLAR MOMENT OF INERTIA:  9.1460659e+00 INCH^4

ROTATION MATRIX from _5_1 orientation to PRINCIPAL AXES:
       0.99964       -0.02669
       0.02669        0.99964

ROTATION ANGLE from _5_1 orientation to PRINCIPAL AXES (degrees):
about z axis          1.529

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  5.8073890e-01  3.0359970e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1          2   COORD
about AXIS 1:  3.07293e-01 INCH^3       1.8174e+00 -1.0506e+00   INCH
               3.63880e-01 INCH^3       2.8813e+00  8.8721e-01   INCH
about AXIS 2:  1.71136e+00 INCH^3      -5.1557e+00 -5.6440e-01   INCH
               1.47245e+00 INCH^3       5.9922e+00  6.8928e-01   INCH
```

END PRO-E DATA

The lift force on one wing at 100*mph* (assume C_{Lmax}=1.77 for deployed slats and stowed flaps, reference NACA TR 427) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3in^2\right) \cdot \left(1.77\right) \cdot \left(0.1776\,psi\right)$$
$$F_{Lift} = 155\ lbf$$

Therefore the shear stress at section cut 5_1 is:

$$\tau_{5\_1} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_1} = \frac{155lbs}{0.957in^2}$$
$$\tau_{5\_1} = 162\,psi$$

The bending stress at section 5_1 due to the aero load is:

$$\sigma_{5\_1} = \frac{M_{aeroload}}{z_{5\_1}} = \frac{(155lbf)(9.42in)}{0.307in^3}$$
$$\sigma_{5\_1} = 4745\,psi$$

The maximum shear stress at section 5_1 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_1}}{2}\right)^2 + \tau_{5\_1}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{4745\,psi}{2}\right)^2 + \left(162\,psi\right)^2}$$
$$\tau_{Max} = 2378\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{2378\,psi} = 12.65}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ 5\_1}} = \frac{4745\,psi}{2} \pm \sqrt{\left(\frac{4745\,psi}{2}\right)^2 + \left(162\,psi\right)^2}$$
$$\sigma_{\substack{Max \\ 5\_1}} = 2372.1 + 2377.6$$
$$\sigma_{\substack{Max \\ 5\_1}} = 4750\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{4750\,psi} = 6.33}$$

Wing Section Cut 5_1 Stress Analysis

Case 2: C_{Lmax}=2.18

It is assumed that the aero load acts at the quarter MAC (BL 16.423).
The section cut 5_1 is shown above. The section cut 5_1 has Pro-Engineer properties also shown above. The lift force on one wing at 100*mph* (assume C_{Lmax}=2.18 for deployed slats and deployed flaps, reference NACA TR 427) is,

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3\,in^2\right)\cdot\left(2.18\right)\cdot\left(0.1776\,psi\right)$$
$$F_{Lift} = 191\ lbf$$

Therefore the shear stress at section cut 5_1 is:

$$\tau_{5\_1} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_1} = \frac{191 lbs}{0.957 in^2}$$
$$\tau_{5\_1} = 200\,psi$$

The bending stress at section 5_1 due to the aero load is:

$$\sigma_{5\_1} = \frac{M_{aeroload}}{z_{5\_1}} = \frac{(191 lbf)(9.42 in)}{0.307 in^3}$$
$$\sigma_{5\_1} = 5844\,psi$$

The maximum shear stress at section 5_1 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_1}}{2}\right)^2 + \tau_{5\_1}^{\ 2}}$$
$$\tau_{Max} = \sqrt{\left(\frac{5844\,psi}{2}\right)^2 + \left(200\,psi\right)^2}$$
$$\tau_{Max} = 2929\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{2929\,psi} = 10.27}$$

Finally, the combined maximum stress is,

$$\sigma_{\underset{5\_1}{Max}} = \frac{5844\,psi}{2} \pm \sqrt{\left(\frac{5844\,psi}{2}\right)^2 + (200\,psi)^2}$$

$$\sigma_{\underset{5\_1}{Max}} = 2921.6 + 2928.4$$

$$\sigma_{\underset{5\_1}{Max}} = 5850\,psi$$

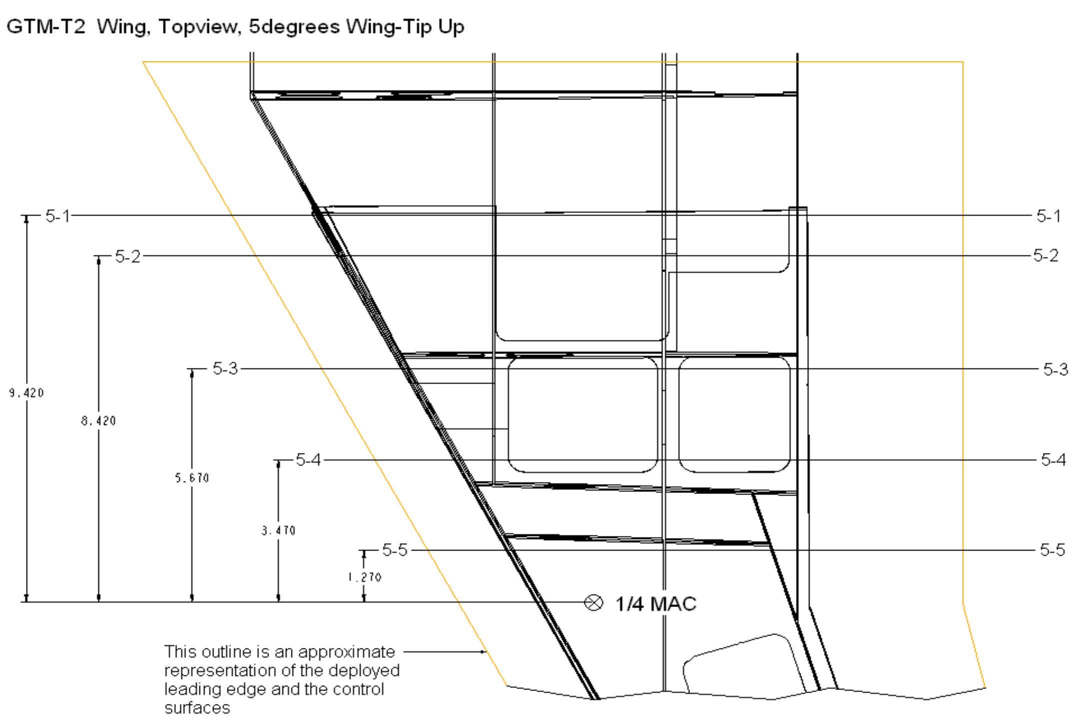
And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{5850\,psi} = 5.14}$$

Wing Section Cut 5_2 Stress Analysis

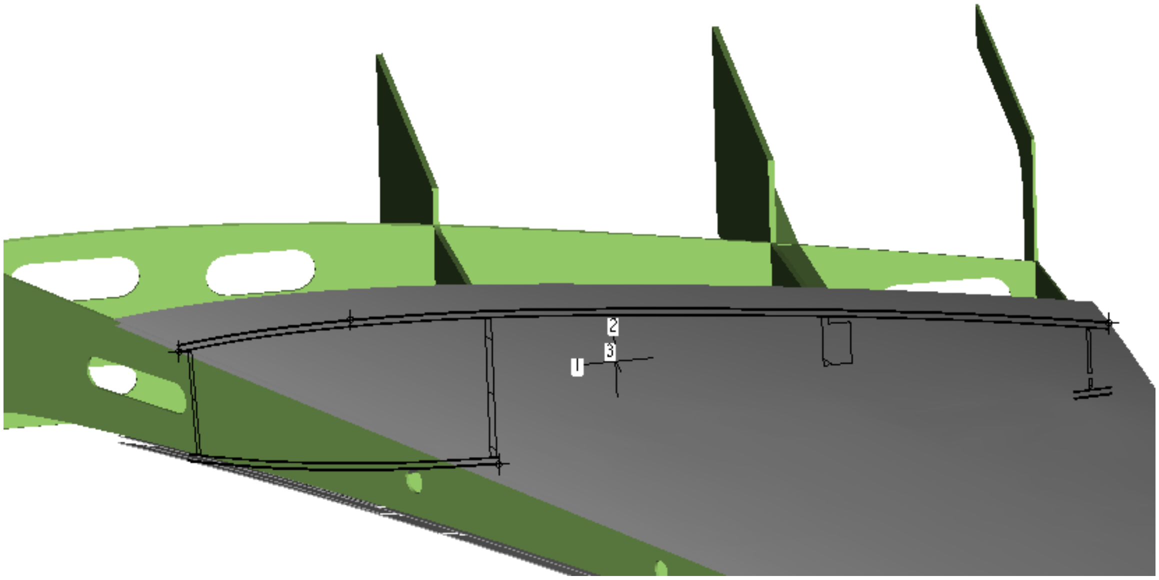
Case 1: C_{Lmax}=1.77

It is assumed that the aero load acts at the quarter MAC (BL 16.423) as shown below:



Drawing reference no. LD-1164893

The section cut 5_2 is shown below:



Drawing reference no. LD-1164893

The section cut 5_2 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  7.1138540e-01 INCH^2

CENTER OF GRAVITY with respect to _5_2 coordinate frame:
X   Y    -5.6467840e+01  9.9609028e+00  INCH

INERTIA with respect to _5_2 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  7.0791002e+01  4.0021702e+02
Iyx Iyy  4.0021702e+02  2.2751698e+03

POLAR MOMENT OF INERTIA:  2.3459608e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _5_2 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.0763768e-01  8.3602712e-02
Iyx Iyy  8.3602712e-02  6.8342304e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.0658309e-01  6.8352850e+00

POLAR MOMENT OF INERTIA:  7.0418681e+00 INCH^4
```

```
ROTATION MATRIX from _5_2 orientation to PRINCIPAL AXES:
       0.99992         0.01261
      -0.01261         0.99992

ROTATION ANGLE from _5_2 orientation to PRINCIPAL AXES (degrees):
about z axis          -0.723

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  5.3888354e-01  3.0997441e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                  1           2   COORD
about AXIS 1:  1.91659e-01 INCH^3        1.3865e+00 -1.0779e+00   INCH
               2.62639e-01 INCH^3        2.9819e+00  7.8657e-01   INCH
about AXIS 2:  1.21713e+00 INCH^3       -5.6159e+00 -9.7121e-02   INCH
               1.38258e+00 INCH^3        4.9439e+00  6.0101e-01   INCH
```

<u>END PRO-E DATA</u>

The lift force on one wing at 100*mph* (assume C_{Lmax}=1.77 for deployed slats and stowed flaps, reference NACA TR 427) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3 in^2\right) \cdot (1.77) \cdot (0.1776 psi)$$
$$F_{Lift} = 155\ lbf$$

Therefore the shear stress at section cut 5_2 is:

$$\tau_{5\_2} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_2} = \frac{155 lbs}{0.7114 in^2}$$
$$\tau_{5\_2} = 218 psi$$

The bending stress at section 5_2 due to the aero load is:

$$\sigma_{5\_2} = \frac{M_{aeroload}}{z_{5\_2}} = \frac{(155 lbf)(8.42 in)}{0.192 in^3}$$
$$\sigma_{5\_2} = 6800 psi$$

The maximum shear stress at section 5_2 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_2}}{2}\right)^2 + \tau_{5\_2}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{6800\,psi}{2}\right)^2 + (218\,psi)^2}$$

$$\tau_{Max} = 3407\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{3407\,psi} = 8.83}$$

Finally, the combined maximum stress is,

$$\sigma_{\underset{5\_2}{Max}} = \frac{6800\,psi}{2} \pm \sqrt{\left(\frac{6800\,psi}{2}\right)^2 + (218\,psi)^2}$$

$$\sigma_{\underset{5\_2}{Max}} = 3399.6 + 3406.5$$

$$\sigma_{\underset{5\_2}{Max}} = 6806\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{6806\,psi} = 4.42}$$

Wing Section Cut 5_2 Stress Analysis

Case 2: C_{Lmax}=2.18

It is assumed that the aero load acts at the quarter MAC (BL 16.423).
The section cut 5_2 is shown above. The section cut 5_2 has Pro-Engineer properties also shown above. The lift force on one wing at 100*mph* (assume C_{Lmax}=2.18 for deployed slats and deployed flaps, reference NACA TR 427) is,

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$

$$F_{Lift} = (492.3\,in^2) \cdot (2.18) \cdot (0.1776\,psi)$$

$$F_{Lift} = 191\ lbf$$

Therefore the shear stress at section cut 5_2 is:

$$\tau_{5\_2} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_2} = \frac{191 lbs}{0.7114 in^2}$$
$$\tau_{5\_2} = 268 psi$$

The bending stress at section 5_2 due to the aero load is:

$$\sigma_{5\_2} = \frac{M_{aeroload}}{z_{5\_2}} = \frac{(191 lbf)(8.42 in)}{0.192 in^3}$$
$$\sigma_{5\_2} = 8374 psi$$

The maximum shear stress at section 5_2 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_2}}{2}\right)^2 + \tau_{5\_2}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{8374 psi}{2}\right)^2 + (268 psi)^2}$$
$$\tau_{Max} = 4196 psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080 psi}{4196 psi} = 7.17}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ 5\_2}} = \frac{8374 psi}{2} \pm \sqrt{\left(\frac{8374 psi}{2}\right)^2 + (268 psi)^2}$$
$$\sigma_{\substack{Max \\ 5\_2}} = 4187.0 + 4195.6$$
$$\sigma_{\substack{Max \\ 5\_2}} = 8384 psi$$

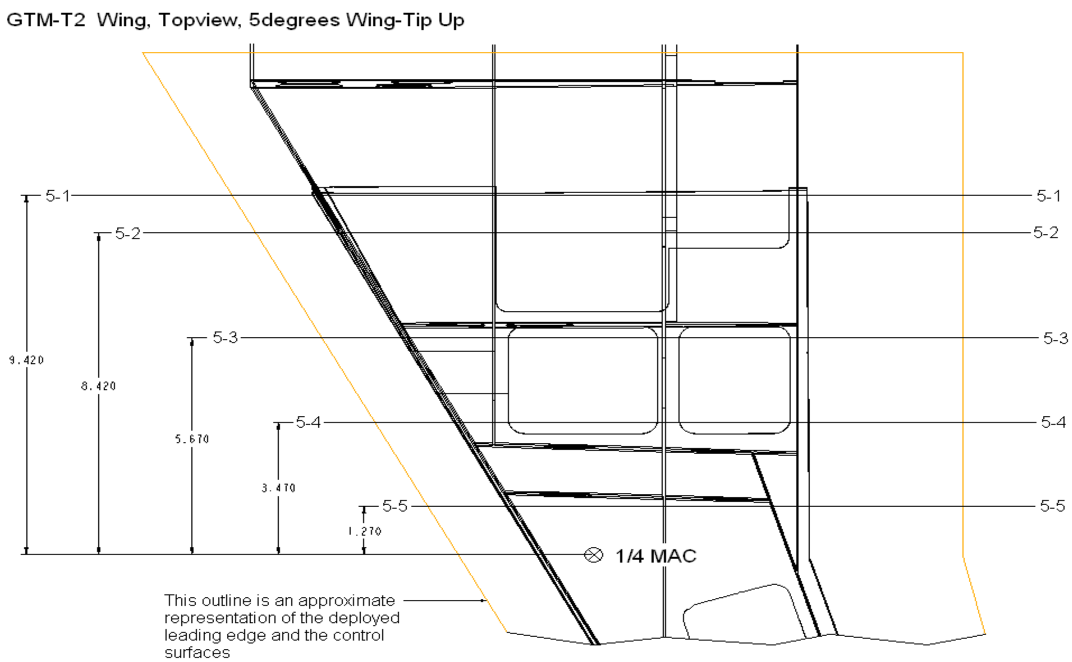
And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080 psi}{8384 psi} = 3.59}$$

Wing Section Cut 5_3 Stress Analysis

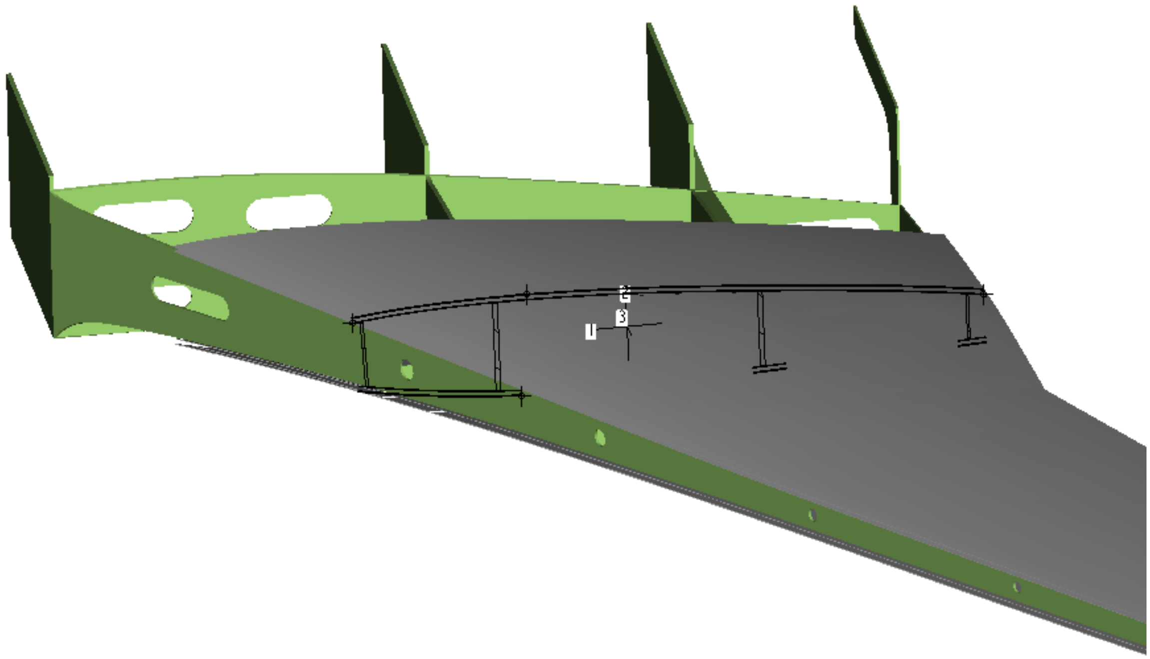
Case 1: C_{Lmax}=1.77

It is assumed that the aero load acts at the quarter MAC (BL 16.423) as shown below:



Drawing reference no. LD-1164893

The section cut 5_3 is shown below:



Drawing reference no. LD-1164893

The section cut 5_3 has the following Pro-Engineer properties:

<u>START PRO-E DATA</u>

```
AREA =  5.3828240e-01 INCH^2

CENTER OF GRAVITY with respect to _5_3 coordinate frame:
X   Y    -5.6931473e+01  1.0018368e+01  INCH

INERTIA with respect to _5_3 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  5.4150780e+01  3.0705905e+02
Iyx Iyy  3.0705905e+02  1.7488421e+03

POLAR MOMENT OF INERTIA:  1.8029928e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _5_3 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.2461360e-01  4.4055712e-02
Iyx Iyy  4.4055712e-02  4.1651373e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.2413330e-01  4.1656176e+00

POLAR MOMENT OF INERTIA:  4.2897509e+00 INCH^4
```

```
ROTATION MATRIX from _5_3 orientation to PRINCIPAL AXES:
       0.99994        0.01090
      -0.01090        0.99994

ROTATION ANGLE from _5_3 orientation to PRINCIPAL AXES (degrees):
about z axis          -0.625

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  4.8021869e-01  2.7818558e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                       1           2   COORD
about AXIS 1:  1.40031e-01 INCH^3        1.5789e+00 -8.8647e-01   INCH
               2.00659e-01 INCH^3        1.4234e+00  6.1863e-01   INCH
about AXIS 2:  8.07818e-01 INCH^3       -5.1566e+00 -7.4551e-03   INCH
               1.05595e+00 INCH^3        3.9449e+00  4.2543e-01   INCH
```

<u>END PRO-E DATA</u>

The lift force on one wing at 100*mph* (assume C_{Lmax}=1.77 for deployed slats and stowed flaps, reference NACA TR 427) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3 in^2\right) \cdot \left(1.77\right) \cdot \left(0.1776 psi\right)$$
$$F_{Lift} = 155\ lbf$$

Therefore the shear stress at section cut 5_3 is:

$$\tau_{5\_3} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_3} = \frac{155 lbs}{0.5383 in^2}$$
$$\tau_{5\_3} = 288 psi$$

The bending stress at section 5_3 due to the aero load is:

$$\sigma_{5\_3} = \frac{M_{aeroload}}{z_{5\_3}} = \frac{(155 lbf)(5.67 in)}{0.140 in^3}$$
$$\sigma_{5\_3} = 6267 psi$$

The maximum shear stress at section 5_3 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_3}}{2}\right)^2 + \tau_{5\_3}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{6267\,psi}{2}\right)^2 + (288\,psi)^2}$$

$$\tau_{Max} = 3147\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{3147\,psi} = 9.56}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\5\_3}} = \frac{6267\,psi}{2} \pm \sqrt{\left(\frac{6267\,psi}{2}\right)^2 + (288\,psi)^2}$$

$$\sigma_{\substack{Max\\5\_3}} = 3133.3 + 3146.5$$

$$\sigma_{\substack{Max\\5\_3}} = 6280\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{6280\,psi} = 4.79}$$

Wing Section Cut 5_3 Stress Analysis

Case 2: C_{Lmax}=2.18

It is assumed that the aero load acts at the quarter MAC (BL 16.423).
The section cut 5_3 is shown above. The section cut 5_3 has Pro-Engineer properties also shown above. The lift force on one wing at 100*mph* (assume C_{Lmax}=2.18 for deployed slats and deployed flaps, reference NACA TR 427) is,

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$

$$F_{Lift} = (492.3\,in^2) \cdot (2.18) \cdot (0.1776\,psi)$$

$$F_{Lift} = 191\ lbf$$

Therefore the shear stress at section cut 5_3 is:

$$\tau_{5\_3} = \frac{F_{shear}}{A_{shear}}$$

$$\tau_{5\_3} = \frac{191lbs}{0.5383in^2}$$

$$\tau_{5\_3} = 355psi$$

The bending stress at section 5_3 due to the aero load is:

$$\sigma_{5\_3} = \frac{M_{aeroload}}{z_{5\_3}} = \frac{(191lbf)(5.67in)}{0.140in^3}$$

$$\sigma_{5\_3} = 7718psi$$

The maximum shear stress at section 5_3 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_3}}{2}\right)^2 + \tau_{5\_3}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{7718psi}{2}\right)^2 + (355psi)^2}$$

$$\tau_{Max} = 3876psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080psi}{3876psi} = 7.76}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max \\ 5\_3}} = \frac{7718psi}{2} \pm \sqrt{\left(\frac{7718psi}{2}\right)^2 + (355psi)^2}$$

$$\sigma_{\substack{Max \\ 5\_3}} = 3859.1 + 3875.3$$

$$\sigma_{\substack{Max \\ 5\_3}} = 7735psi$$

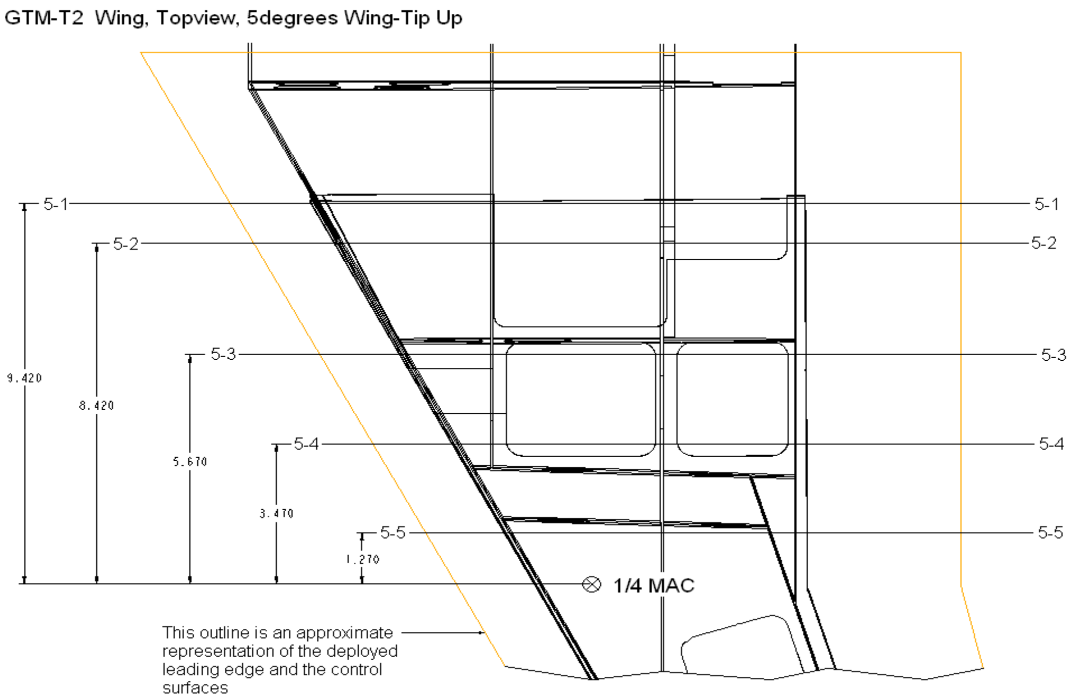
And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080psi}{7735psi} = 3.89}$$

Wing Section Cut 5_4 Stress Analysis

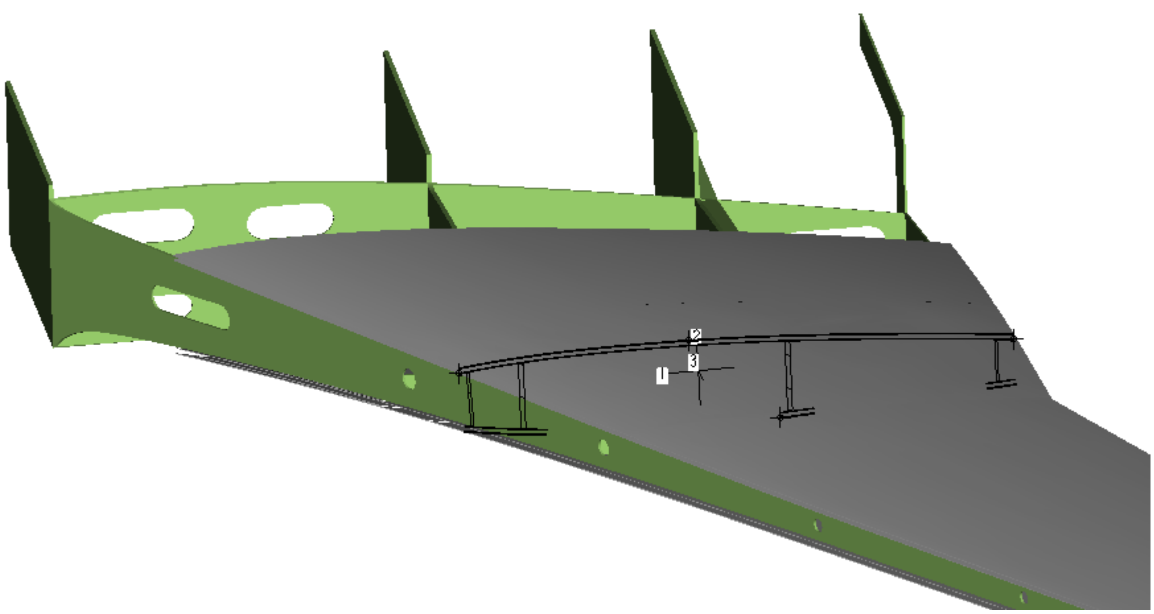
Case 1: C_{Lmax}=1.77

It is assumed that the aero load acts at the quarter MAC (BL 16.423) as shown below:



Drawing reference no. LD-1164893

The section cut 5_4 is shown below:



Drawing reference no. LD-1164893

The section cut 5_4 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  4.4262273e-01 INCH^2

CENTER OF GRAVITY with respect to _5_4 coordinate frame:
X   Y    -5.7575499e+01  1.0084483e+01  INCH

INERTIA with respect to _5_4 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.5082190e+01  2.5703306e+02
Iyx Iyy  2.5703306e+02  1.4701157e+03

POLAR MOMENT OF INERTIA:  1.5151979e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _5_4 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  6.8876417e-02  3.7827672e-02
Iyx Iyy  3.7827672e-02  2.8487596e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  6.8361767e-02  2.8492742e+00

POLAR MOMENT OF INERTIA:  2.9176360e+00 INCH^4

ROTATION MATRIX from _5_4 orientation to PRINCIPAL AXES:
       0.99991        0.01360
      -0.01360        0.99991

ROTATION ANGLE from _5_4 orientation to PRINCIPAL AXES (degrees):
about z axis         -0.779

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.9299750e-01  2.5371741e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                   1          2   COORD
about AXIS 1:  8.51305e-02 INCH^3      -1.1334e+00 -8.0302e-01   INCH
               1.46878e-01 INCH^3       1.1671e-01  4.6543e-01   INCH
about AXIS 2:  6.29322e-01 INCH^3      -4.5275e+00  2.8475e-02   INCH
               8.33802e-01 INCH^3       3.4172e+00  2.9024e-01   INCH
```

END PRO-E DATA

The lift force on one wing at 100*mph* (assume C_{Lmax}=1.77 for deployed slats and stowed flaps, reference NACA TR 427) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3in^2\right)\cdot\left(1.77\right)\cdot\left(0.1776\,psi\right)$$
$$F_{Lift} = 155\ lbf$$

Therefore the shear stress at section cut 5_4 is:

$$\tau_{5\_4} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_4} = \frac{155lbs}{0.443in^2}$$
$$\tau_{5\_4} = 350\,psi$$

The bending stress at section 5_4 due to the aero load is:

$$\sigma_{5\_4} = \frac{M_{aeroload}}{z_{5\_4}} = \frac{(155lbf)(3.47in)}{0.0851in^3}$$
$$\sigma_{5\_4} = 6309\,psi$$

The maximum shear stress at section 5_4 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_4}}{2}\right)^2 + \tau_{5\_4}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{6309\,psi}{2}\right)^2 + \left(350\,psi\right)^2}$$
$$\tau_{Max} = 3174\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{3174\,psi} = 9.48}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\5\_4}} = \frac{6309\,psi}{2} \pm \sqrt{\left(\frac{6309\,psi}{2}\right)^2 + \left(350\,psi\right)^2}$$
$$\sigma_{\substack{Max\\5\_4}} = 3154.2 + 3173.5$$
$$\sigma_{\substack{Max\\5\_4}} = 6328\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{6328\,psi} = 4.75}$$

Wing Section Cut 5_4 Stress Analysis

Case 2: C_{Lmax}=2.18

It is assumed that the aero load acts at the quarter MAC (BL 16.423).
The section cut 5_4 is shown above. The section cut 5_4 has Pro-Engineer properties also shown above. The lift force on one wing at 100*mph* (assume C_{Lmax}=2.18 for deployed slats and deployed flaps, reference NACA TR 427) is,

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3\,in^2\right) \cdot (2.18) \cdot (0.1776\,psi)$$
$$F_{Lift} = 191\ lbf$$

Therefore the shear stress at section cut 5_4 is:

$$\tau_{5\_4} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_4} = \frac{191lbs}{0.443in^2}$$
$$\tau_{5\_4} = 431psi$$

The bending stress at section 5_4 due to the aero load is:

$$\sigma_{5\_4} = \frac{M_{aeroload}}{z_{5\_4}} = \frac{(191lbf)(3.47in)}{0.0851in^3}$$
$$\sigma_{5\_4} = 7770\,psi$$

The maximum shear stress at section 5_4 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_4}}{2}\right)^2 + \tau_{5\_4}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{7770\,psi}{2}\right)^2 + (431\,psi)^2}$$
$$\tau_{Max} = 3909\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{3909\,psi} = 7.70}$$

Finally, the combined maximum stress is,

$$\sigma_{\underset{5\_4}{Max}} = \frac{7770\,psi}{2} \pm \sqrt{\left(\frac{7770\,psi}{2}\right)^2 + (431\,psi)^2}$$

$$\sigma_{\underset{5\_4}{Max}} = 3884.8 + 3908.6$$

$$\sigma_{\underset{5\_4}{Max}} = 7794\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

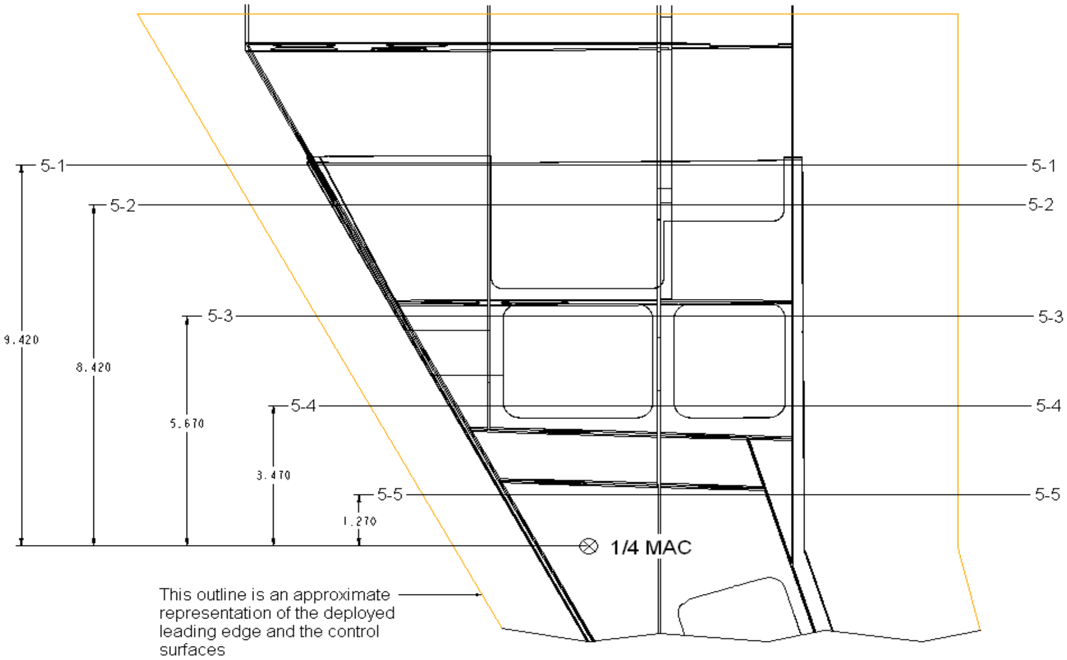
$$\boxed{FS = \frac{30080\,psi}{7794\,psi} = 3.86}$$

Wing Section Cut 5_5 Stress Analysis

Case 1: C_{Lmax}=1.77

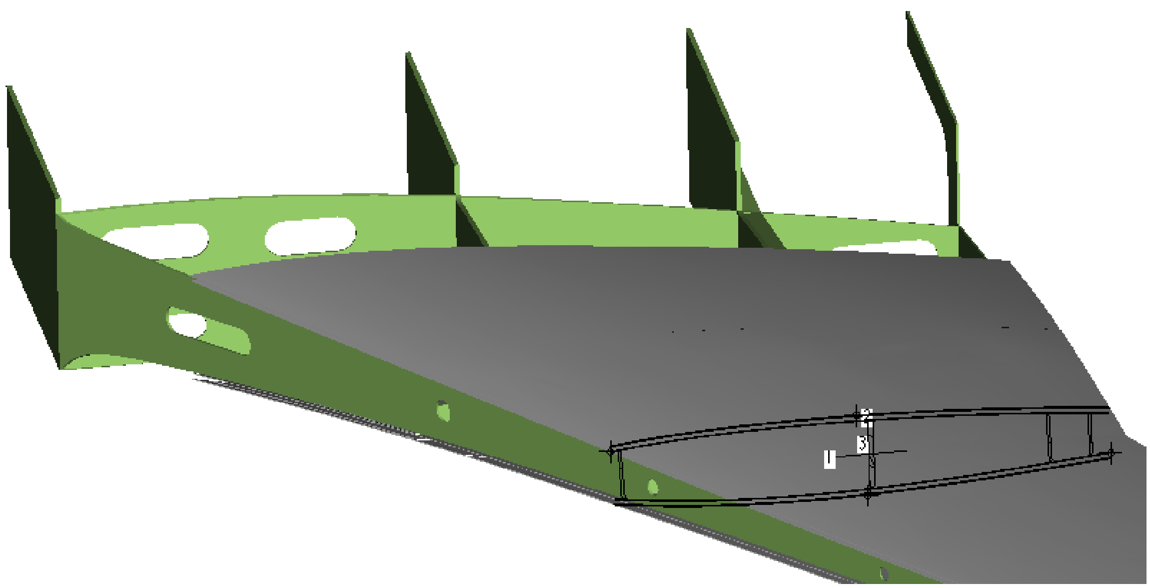
It is assumed that the aero load acts at the quarter MAC (BL 16.423) as shown below:

GTM-T2 Wing, Topview, 5degrees Wing-Tip Up



Drawing reference no. LD-1164893

The section cut 5_5 is shown below:



Drawing reference no. LD-1164893

The section cut 5_5 has the following Pro-Engineer properties:

<u>START PRO-E DATA</u>

```
AREA =  4.7561930e-01 INCH^2

CENTER OF GRAVITY with respect to _5_5 coordinate frame:
```

```
X   Y    -5.8827685e+01  9.9229867e+00  INCH

INERTIA with respect to _5_5 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.6916422e+01  2.7764777e+02
Iyx Iyy  2.7764777e+02  1.6479108e+03

POLAR MOMENT OF INERTIA:  1.6948272e+03 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _5_5 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  8.4252010e-02  6.7444106e-03
Iyx Iyy  6.7444106e-03  1.9367339e+00

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  8.4227456e-02  1.9367584e+00

POLAR MOMENT OF INERTIA:  2.0209859e+00 INCH^4

ROTATION MATRIX from _5_5 orientation to PRINCIPAL AXES:
       0.99999        0.00364
      -0.00364        0.99999

ROTATION ANGLE from _5_5 orientation to PRINCIPAL AXES (degrees):
about z axis         -0.209

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  4.2082071e-01  2.0179388e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                     1           2   COORD
about AXIS 1:  1.47090e-01 INCH^3        3.5101e-02 -5.7263e-01   INCH
               1.50596e-01 INCH^3        1.4094e-01  5.5929e-01   INCH
about AXIS 2:  5.89303e-01 INCH^3       -3.2865e+00 -2.7746e-01   INCH
               5.53429e-01 INCH^3        3.4996e+00  3.4747e-01   INCH
```
END PRO-E DATA

The lift force on one wing at 100*mph* (assume C_{Lmax}=1.77 for deployed slats and stowed flaps, reference NACA TR 427) is:

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3\,in^2\right) \cdot \left(1.77\right) \cdot \left(0.1776\,psi\right)$$
$$F_{Lift} = 155\ lbf$$

Therefore the shear stress at section cut 5_5 is:

$$\tau_{5\_5} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_5} = \frac{155lbs}{0.4756in^2}$$
$$\tau_{5\_5} = 326\,psi$$

The bending stress at section 5_5 due to the aero load is:

$$\sigma_{5\_5} = \frac{M_{aeroload}}{z_{5\_5}} = \frac{(155lbf)(1.27in)}{0.147in^3}$$
$$\sigma_{5\_5} = 1336\,psi$$

The maximum shear stress at section 5_5 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_5}}{2}\right)^2 + {\tau_{5\_5}}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{1336\,psi}{2}\right)^2 + (326\,psi)^2}$$
$$\tau_{Max} = 743\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{743\,psi} = 40.48}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\5\_5}} = \frac{1336\,psi}{2} \pm \sqrt{\left(\frac{1336\,psi}{2}\right)^2 + (326\,psi)^2}$$
$$\sigma_{\substack{Max\\5\_5}} = 668.2 + 743.2$$
$$\sigma_{\substack{Max\\5\_5}} = 1412\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{1412\,psi} = 21.31}$$

<u>Wing Section Cut 5_5 Stress Analysis</u>

Case 2: C_{Lmax}=2.18

It is assumed that the aero load acts at the quarter MAC (BL 16.423).
The section cut 5_5 is shown above. The section cut 5_5 has Pro-Engineer properties also shown above. The lift force on one wing at 100*mph* (assume C_{Lmax}=2.18 for deployed slats and deployed flaps, reference NACA TR 427) is,

$$F_{Lift} = A_{wing} \cdot C_{L\max} \cdot q_{100mph}$$
$$F_{Lift} = \left(492.3in^2\right) \cdot \left(2.18\right) \cdot \left(0.1776psi\right)$$
$$F_{Lift} = 191\ lbf$$

Therefore the shear stress at section cut 5_5 is:

$$\tau_{5\_5} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{5\_5} = \frac{191lbs}{0.4756in^2}$$
$$\tau_{5\_5} = 401psi$$

The bending stress at section 5_5 due to the aero load is:

$$\sigma_{5\_5} = \frac{M_{aeroload}}{z_{5\_5}} = \frac{(191lbf)(1.27in)}{0.147in^3}$$
$$\sigma_{5\_5} = 1646psi$$

The maximum shear stress at section 5_5 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{5\_5}}{2}\right)^2 + \tau_{5\_5}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{1646psi}{2}\right)^2 + \left(401psi\right)^2}$$
$$\tau_{Max} = 916psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080psi}{918psi} = 32.86}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\5\_5}} = \frac{1646\,psi}{2} \pm \sqrt{\left(\frac{1646\,psi}{2}\right)^2 + (401\,psi)^2}$$

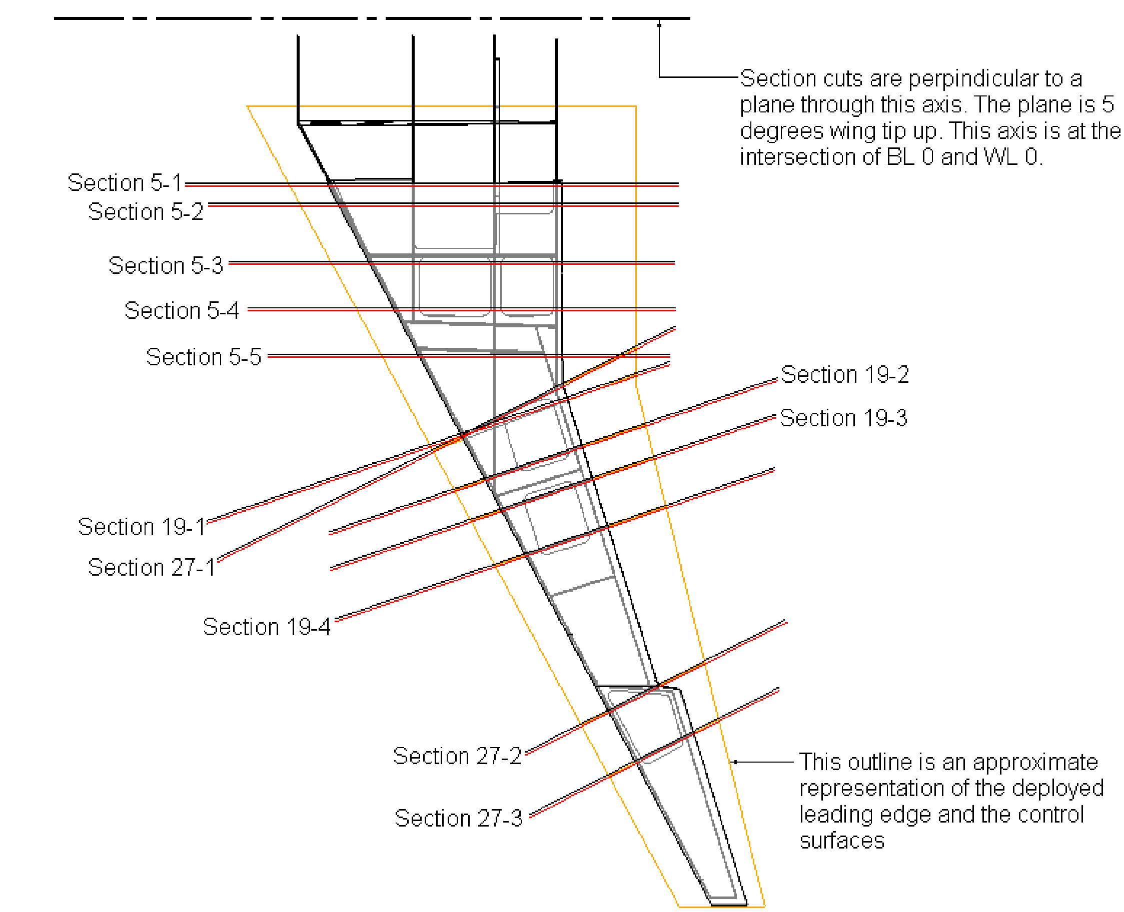
$$\sigma_{\substack{Max\\5\_5}} = 822.9 + 915.3$$

$$\sigma_{\substack{Max\\5\_5}} = 1739\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{1739\,psi} = 17.31}$$

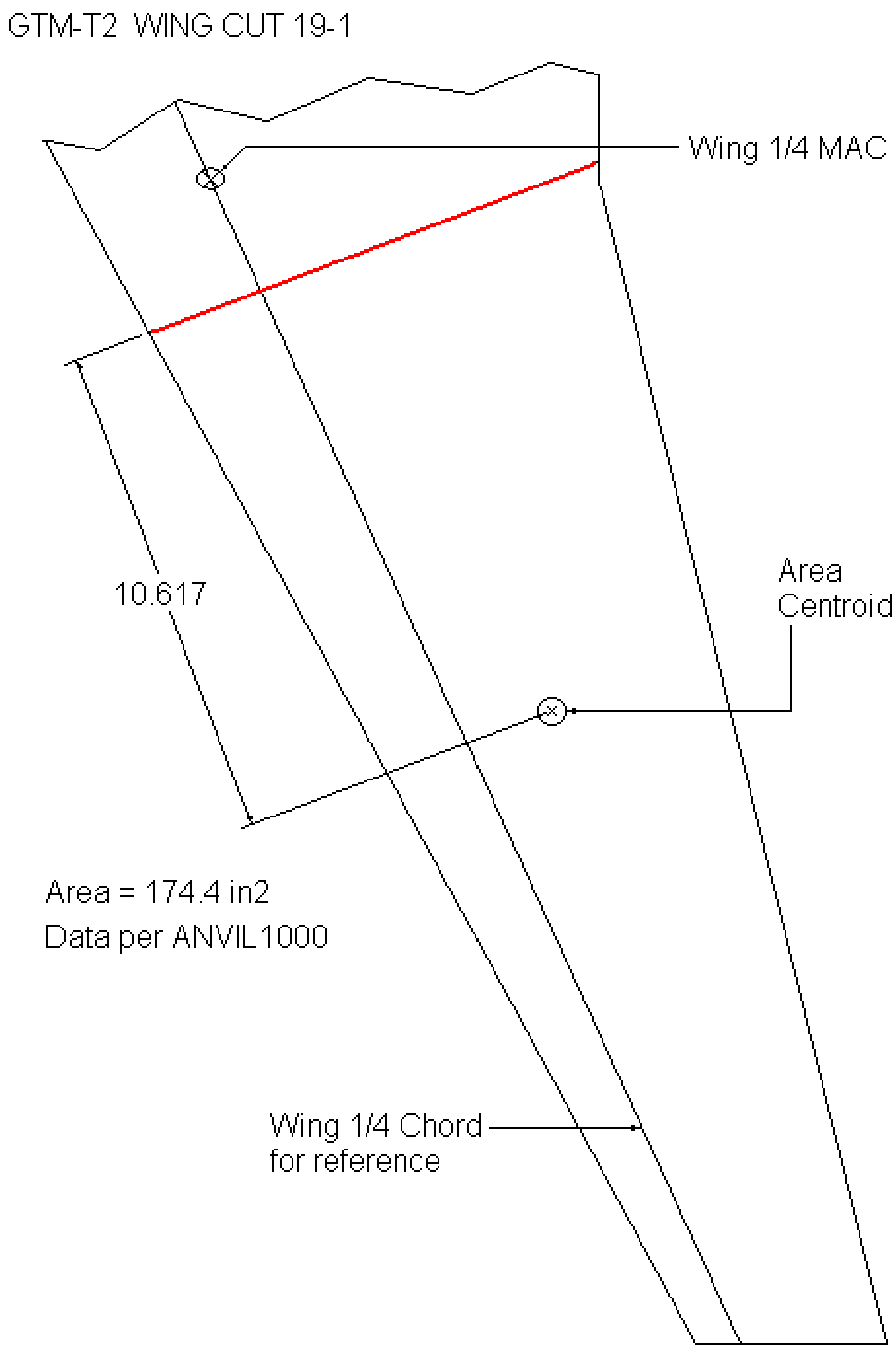
GTM-T2 Wing Section Cut Locations, 5 degrees WL 0 Offset



Drawing reference no. LD-1164893

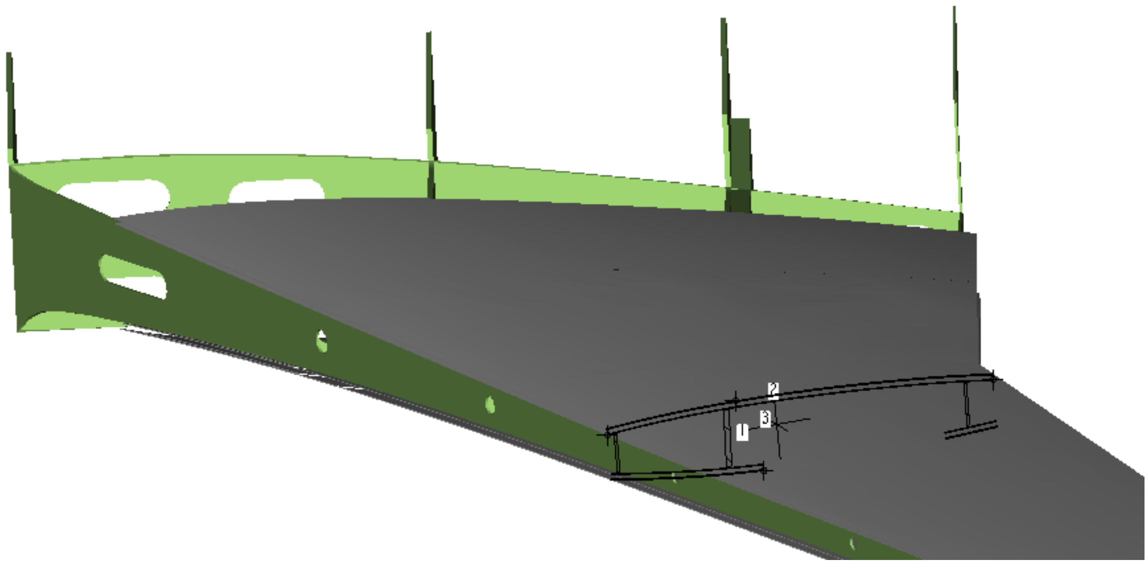
Wing Section Cut 19_1 Stress Analysis:

Since this area is outside of the quarter MAC, it is assumed that the aero load acts at the area centroid of the panel as shown below:



Drawing reference no. LD-1164907

The section cut 19_1 is shown below:



Drawing reference no. LD-1164893

The section cut 19_1 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  2.9740161e-01 INCH^2

CENTER OF GRAVITY with respect to _19_1 coordinate frame:
X   Y    -1.0005537e+01 -3.5284094e+00  INCH

INERTIA with respect to _19_1 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.5048549e+00 -1.0537315e+01
Iyx Iyy -1.0537315e+01  2.9808272e+01

POLAR MOMENT OF INERTIA:  3.4313127e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _19_1 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  8.0230202e-01 -3.7957952e-02
Iyx Iyy -3.7957952e-02  3.5168984e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  3.3295390e-02  8.0417562e-01

POLAR MOMENT OF INERTIA:  8.3747101e-01 INCH^4

ROTATION MATRIX from _19_1 orientation to PRINCIPAL AXES:
       0.04930       -0.99878
       0.99878        0.04930
```

```
ROTATION ANGLE from _19_1 orientation to PRINCIPAL AXES (degrees):
about z axis          87.174

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.3459573e-01  1.6443861e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2    COORD
about AXIS 1:  5.76861e-02 INCH^3       2.3380e-01 -5.7718e-01   INCH
               8.22687e-02 INCH^3       5.2274e-01  4.0472e-01   INCH
about AXIS 2:  2.69552e-01 INCH^3      -2.9834e+00  8.7493e-02   INCH
               3.49136e-01 INCH^3       2.3033e+00  2.5050e-01   INCH
```

END PRO-E DATA

The lift force on the plan form area at 100*mph* (assume C_{Lmax}=1.77, reference NACA TR 427) is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = \left(174.4\,in^2\right) \cdot \left(1.77\right) \cdot \left(0.1776\,psi\right)$$
$$F_{Lift} = 55\,lbf$$

Therefore the shear stress at section cut 19_1 is:

$$\tau_{19\_1} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{19\_1} = \frac{55\,lbs}{0.2974\,in^2}$$
$$\tau_{19\_1} = 185\,psi$$

The bending stress at section 19_1 due to the aero load is:

$$\sigma_{19\_1} = \frac{M_{aeroload}}{z_{19\_1}} = \frac{(55\,lbf)(10.617\,in)}{0.0577\,in^3}$$
$$\sigma_{19\_1} = 10123\,psi$$

The maximum shear stress at section 19_1 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{19\_1}}{2}\right)^2 + \tau_{19\_1}{}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{10123\,psi}{2}\right)^2 + \left(185\,psi\right)^2}$$
$$\tau_{Max} = 5065\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{5065\,psi} = 5.94}$$

Finally, the combined maximum stress is,

$$\sigma_{\underset{19\_1}{Max}} = \frac{10123\,psi}{2} \pm \sqrt{\left(\frac{10123\,psi}{2}\right)^2 + (185\,psi)^2}$$

$$\sigma_{\underset{19\_1}{Max}} = 5061.3 + 5064.7$$

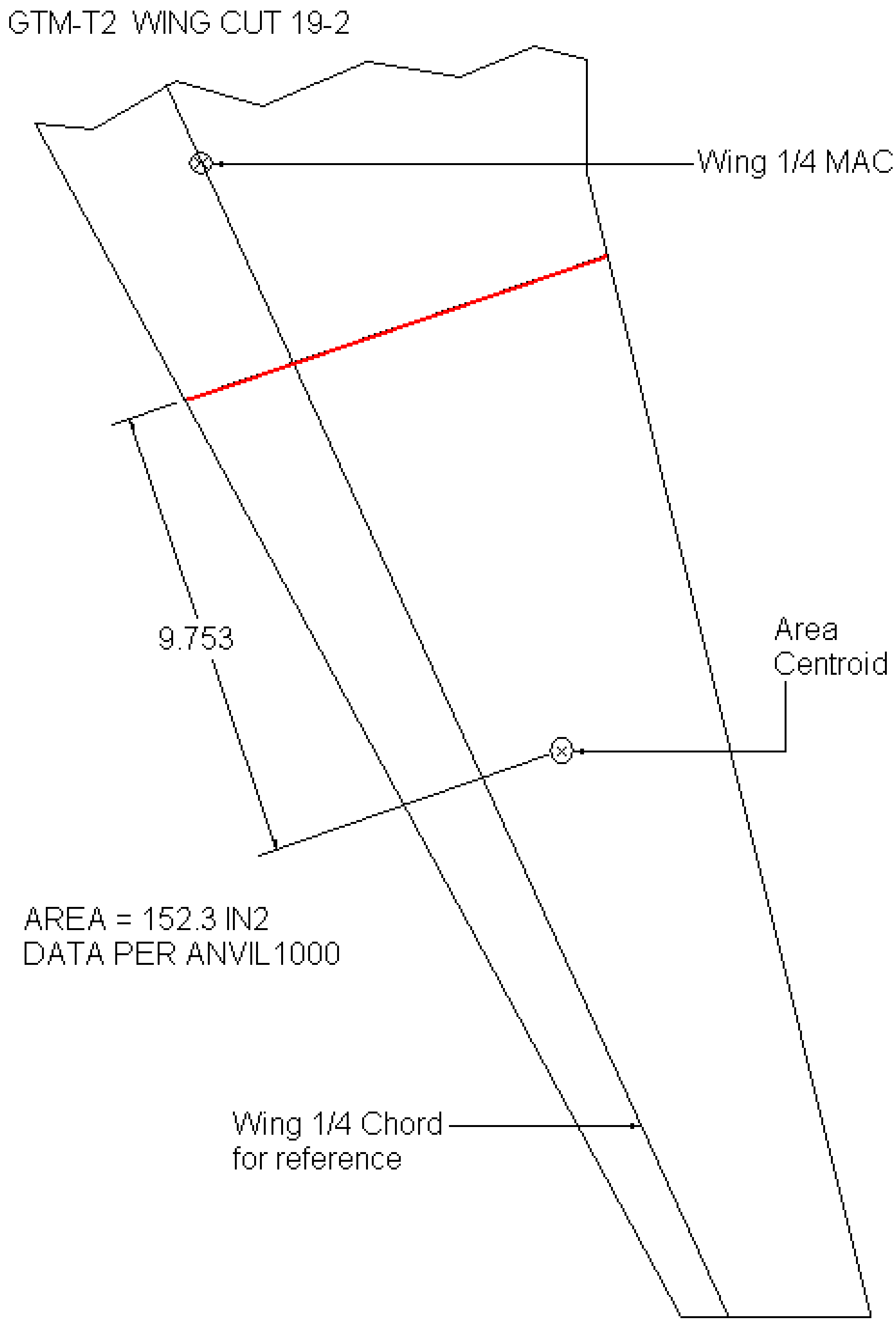
$$\sigma_{\underset{19\_1}{Max}} = 10126\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{10126\,psi} = 2.97}$$

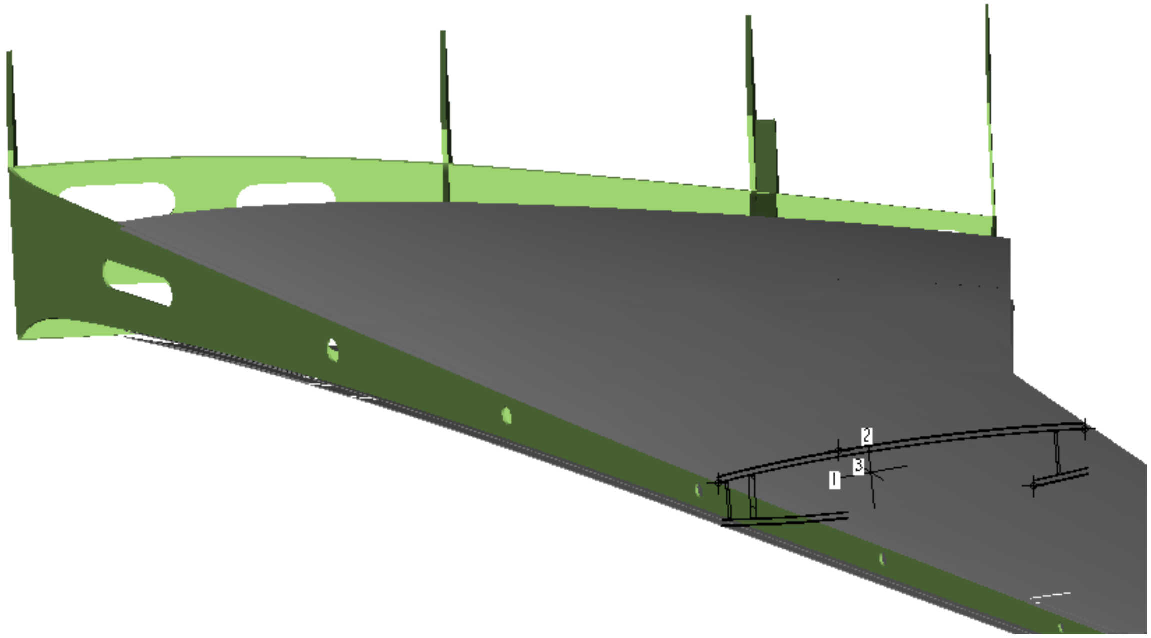
Wing Section Cut 19_2 Stress Analysis:

Since this area is outside of the quarter MAC, it is assumed that the aero load acts at the area centroid of the panel as shown below:



Drawing reference no. LD-1164907

The section cut 19_2 is shown below:



Drawing reference no. LD-1164893

The section cut 19_2 has the following Pro-Engineer properties:

<u>START PRO-E DATA</u>

```
AREA =  2.5687989e-01 INCH^2

CENTER OF GRAVITY with respect to _19_2 coordinate frame:
X   Y    -9.9999850e+00 -3.6030721e+00  INCH

INERTIA with respect to _19_2 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  4.0482982e+00 -9.2900409e+00
Iyx Iyy -9.2900409e+00  2.5713033e+01

POLAR MOMENT OF INERTIA:  2.9761331e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _19_2 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  7.1345043e-01 -3.4487119e-02
Iyx Iyy -3.4487119e-02  2.5121196e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  2.3397616e-02  7.1517401e-01

POLAR MOMENT OF INERTIA:  7.3857162e-01 INCH^4

ROTATION MATRIX from _19_2 orientation to PRINCIPAL AXES:
```

```
       0.04992       -0.99875
       0.99875        0.04992

ROTATION ANGLE from _19_2 orientation to PRINCIPAL AXES (degrees):
about z axis          87.139

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.0180105e-01  1.6685561e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                        1           2   COORD
about AXIS 1:  4.35563e-02 INCH^3        -2.1089e+00 -5.3718e-01   INCH
               6.64673e-02 INCH^3         3.9075e-01  3.5202e-01   INCH
about AXIS 2:  2.51110e-01 INCH^3        -2.8481e+00  6.7161e-02   INCH
               3.57143e-01 INCH^3         2.0025e+00  2.1163e-01   INCH
```

END PRO-E DATA

The lift force on the plan form area at 100*mph* (assume C_{Lmax}=1.77, reference NACA TR 427) is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = \left(152.3 in^2\right) \cdot (1.77) \cdot (0.1776 psi)$$
$$F_{Lift} = 48 lbf$$

Therefore the shear stress at section cut 19_2 is:

$$\tau_{19\_2} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{19\_2} = \frac{48 lbs}{0.2569 in^2}$$
$$\tau_{19\_2} = 186 psi$$

The bending stress at section 19_2 due to the aero load is:

$$\sigma_{19\_2} = \frac{M_{aeroload}}{z_{19\_2}} = \frac{(48 lbf)(9.753 in)}{0.0436 in^3}$$
$$\sigma_{19\_2} = 10721 psi$$

The maximum shear stress at section 19_2 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{19\_2}}{2}\right)^2 + \tau_{19\_2}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{10721 psi}{2}\right)^2 + (186\, psi)^2}$$

$$\tau_{Max} = 5364\, psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\, psi}{5364\, psi} = 5.61}$$

Finally, the combined maximum stress is,

$$\sigma_{\underset{19\_2}{Max}} = \frac{10721 psi}{2} \pm \sqrt{\left(\frac{10721 psi}{2}\right)^2 + (186\, psi)^2}$$

$$\sigma_{\underset{19\_2}{Max}} = 5360.4 + 5363.7$$

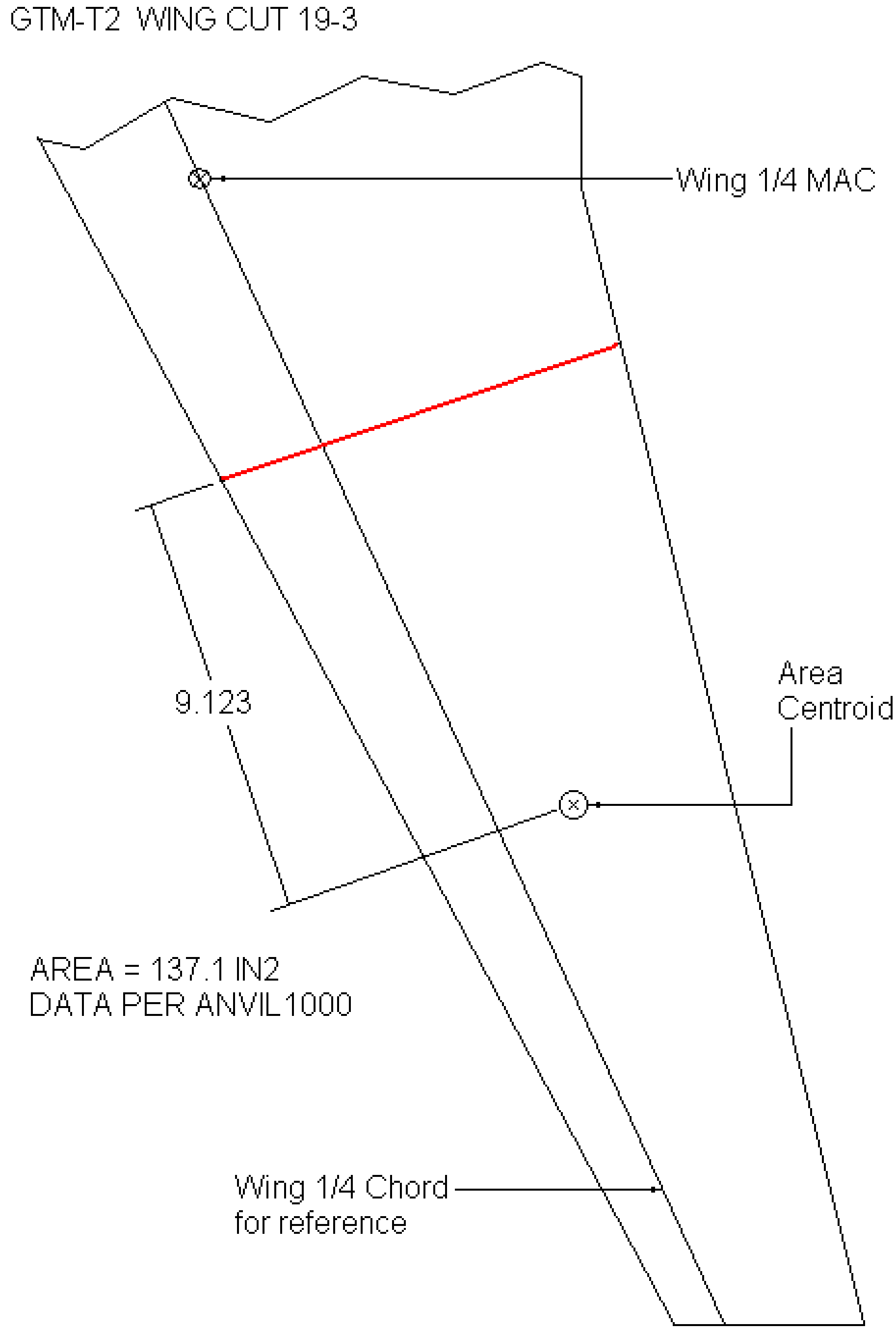
$$\sigma_{\underset{19\_2}{Max}} = 10724\, psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\, psi}{10724\, psi} = 2.80}$$

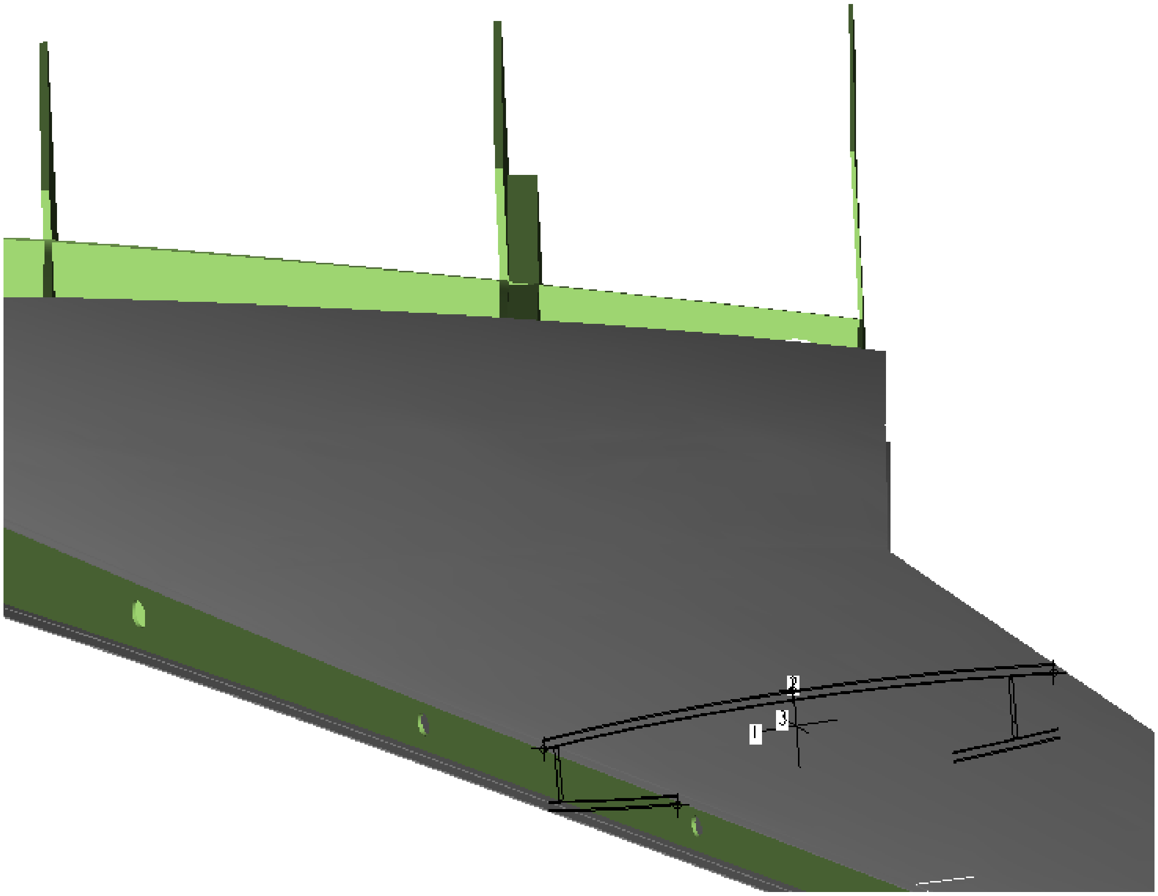
Wing Section Cut 19_3 Stress Analysis:

Since this area is outside of the quarter MAC, it is assumed that the aero load acts at the area centroid of the panel as shown below:



Drawing reference no. LD-1164907

The section cut 19_3 is shown below:



Drawing reference no. LD-1164893

The section cut 19_3 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  2.0063497e-01 INCH^2

CENTER OF GRAVITY with respect to _19_3 coordinate frame:
X   Y    -1.0022853e+01 -4.0980607e+00  INCH

INERTIA with respect to _19_3 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.8729789e+00 -8.2545095e+00
Iyx Iyy -8.2545095e+00  2.0173949e+01

POLAR MOMENT OF INERTIA:  2.4046928e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _19_3 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  5.0349490e-01 -1.3576335e-02
```

```
Iyx Iyy -1.3576335e-02  1.8643396e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.8263543e-02  5.0387475e-01

POLAR MOMENT OF INERTIA:  5.2213830e-01 INCH^4

ROTATION MATRIX from _19_3 orientation to PRINCIPAL AXES:
       0.02797       -0.99961
       0.99961        0.02797

ROTATION ANGLE from _19_3 orientation to PRINCIPAL AXES (degrees):
about z axis         88.397

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.0170964e-01  1.5847399e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  3.48223e-02 INCH^3        1.1062e+00 -5.2448e-01   INCH
               5.92737e-02 INCH^3        1.0350e-02  3.0812e-01   INCH
about AXIS 2:  2.18007e-01 INCH^3       -2.3113e+00  1.1888e-01   INCH
               2.24349e-01 INCH^3        2.2459e+00  1.3378e-01   INCH
```

END PRO-E DATA

The lift force on the plan form area at 100*mph* (assume C_{Lmax}=1.77, reference NACA TR 427) is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = \left(137.1 in^2\right) \cdot (1.77) \cdot (0.1776 psi)$$
$$F_{Lift} = 43 lbf$$

Therefore the shear stress at section cut 19_3 is:

$$\tau_{19\_3} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{19\_3} = \frac{43 lbs}{0.201 in^2}$$
$$\tau_{19\_3} = 215 psi$$

The bending stress at section 19_3 due to the aero load is:

$$\sigma_{19\_3} = \frac{M_{aeroload}}{z_{19\_3}} = \frac{(43lbf)(9.123in)}{0.0348in^3}$$

$$\sigma_{19\_3} = 11292\,psi$$

The maximum shear stress at section 19_3 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{19\_3}}{2}\right)^2 + \tau_{19\_3}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{11292\,psi}{2}\right)^2 + (215\,psi)^2}$$

$$\tau_{Max} = 5650\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{5650\,psi} = 5.32}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\19\_3}} = \frac{11292\,psi}{2} \pm \sqrt{\left(\frac{11292\,psi}{2}\right)^2 + (215\,psi)^2}$$

$$\sigma_{\substack{Max\\19\_3}} = 5645.8 + 5649.9$$

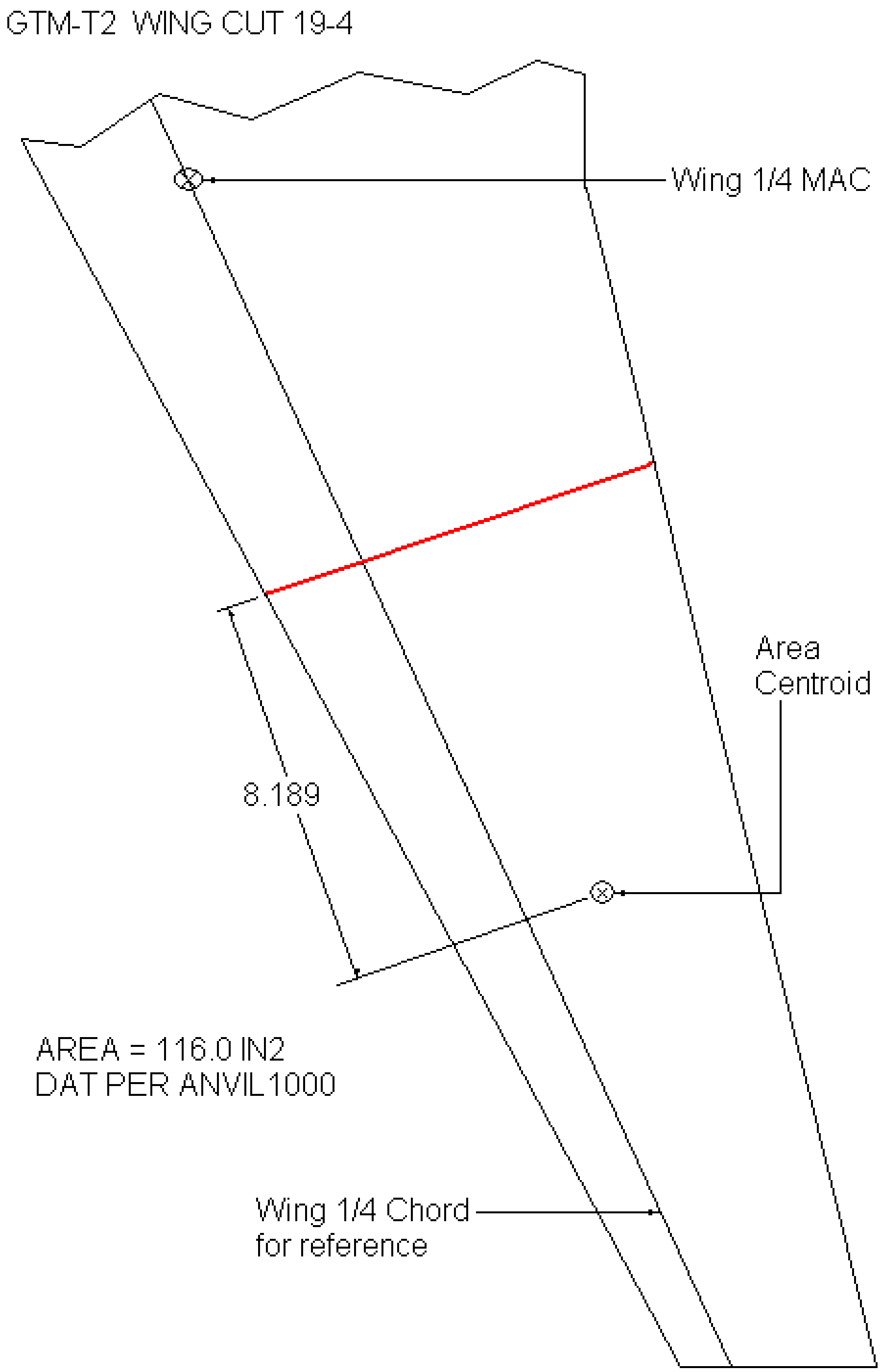
$$\sigma_{\substack{Max\\19\_3}} = 11296\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{11296\,psi} = 2.66}$$

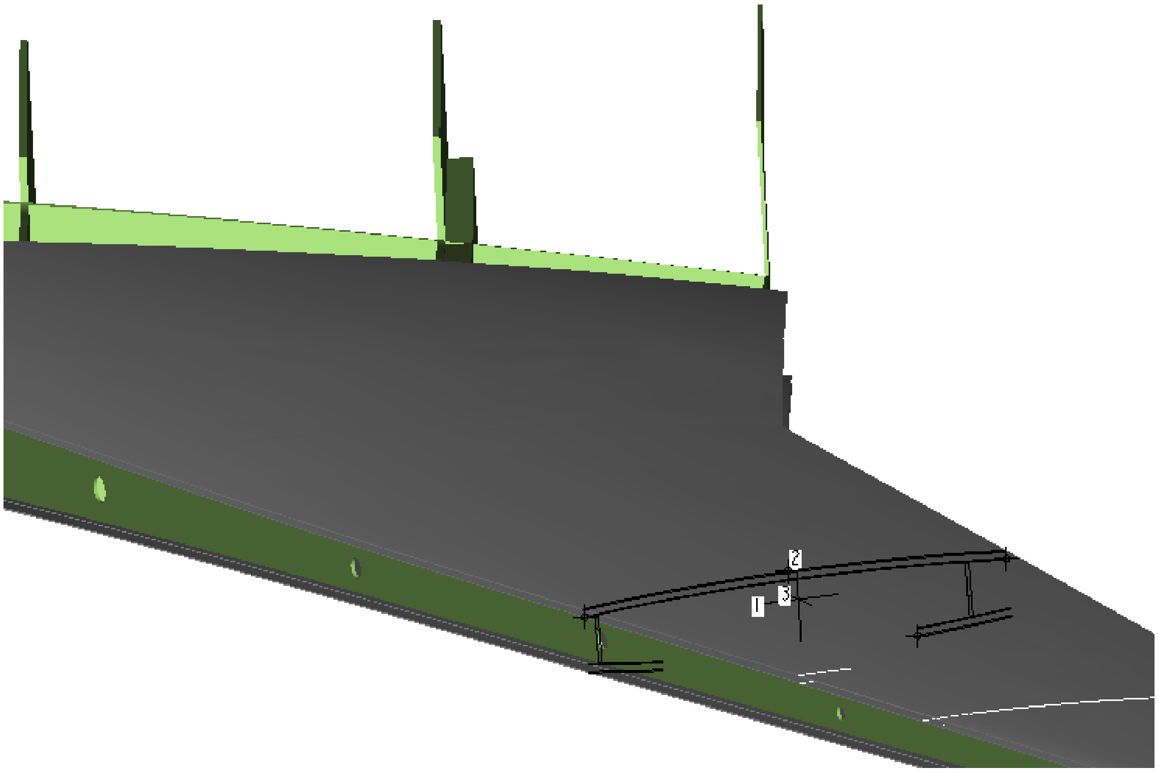
Wing Section Cut 19_4 Stress Analysis:

Since this area is outside of the quarter MAC, it is assumed that the aero load acts at the area centroid of the panel as shown below:



Drawing reference no. LD-1164907

The section cut 19_4 is shown below:



Drawing reference no. LD-1164893

The section cut 19_4 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  1.7518846e-01 INCH^2

CENTER OF GRAVITY with respect to _19_4 coordinate frame:
X   Y    -1.0022018e+01 -4.3068467e+00  INCH

INERTIA with respect to _19_4 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.6114796e+00 -7.5673767e+00
Iyx Iyy -7.5673767e+00  1.7609522e+01

POLAR MOMENT OF INERTIA:  2.1221002e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _19_4 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  3.6192145e-01 -5.6656341e-03
Iyx Iyy -5.6656341e-03  1.3445218e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.3353129e-02  3.6201354e-01
```

```
POLAR MOMENT OF INERTIA:  3.7536667e-01 INCH^4

ROTATION MATRIX from _19_4 orientation to PRINCIPAL AXES:
       0.01625       -0.99987
       0.99987        0.01625

ROTATION ANGLE from _19_4 orientation to PRINCIPAL AXES (degrees):
about z axis          89.069

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  2.7608244e-01  1.4375060e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2    COORD
about AXIS 1:  2.77874e-02 INCH^3       -1.1292e+00 -4.8055e-01   INCH
               4.88590e-02 INCH^3        8.3735e-02  2.7330e-01   INCH
about AXIS 2:  1.77337e-01 INCH^3       -2.0414e+00  1.3014e-01   INCH
               1.74081e-01 INCH^3        2.0796e+00  8.6765e-02   INCH
```

<u>END PRO-E DATA</u>

The lift force on the plan form area at 100*mph* (assume C_{Lmax}=1.77, reference NACA TR 427) is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = \left(116.0\,in^2\right)\cdot\left(1.77\right)\cdot\left(0.1776\,psi\right)$$
$$F_{Lift} = 37lbf$$

Therefore the shear stress at section cut 19_4 is:

$$\tau_{19\_4} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{19\_4} = \frac{37lbs}{0.1752in^2}$$
$$\tau_{19\_4} = 208\,psi$$

The bending stress at section 19_4 due to the aero load is:

$$\sigma_{19\_4} = \frac{M_{aeroload}}{z_{19\_4}} = \frac{(37lbf)(8.189in)}{0.0278in^3}$$
$$\sigma_{19\_4} = 10747\,psi$$

The maximum shear stress at section 19_4 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{19\_4}}{2}\right)^2 + \tau_{19\_4}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{10747\,psi}{2}\right)^2 + (208\,psi)^2}$$

$$\tau_{Max} = 5378\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{5378\,psi} = 5.59}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\19\_4}} = \frac{10747\,psi}{2} \pm \sqrt{\left(\frac{10747\,psi}{2}\right)^2 + (208\,psi)^2}$$

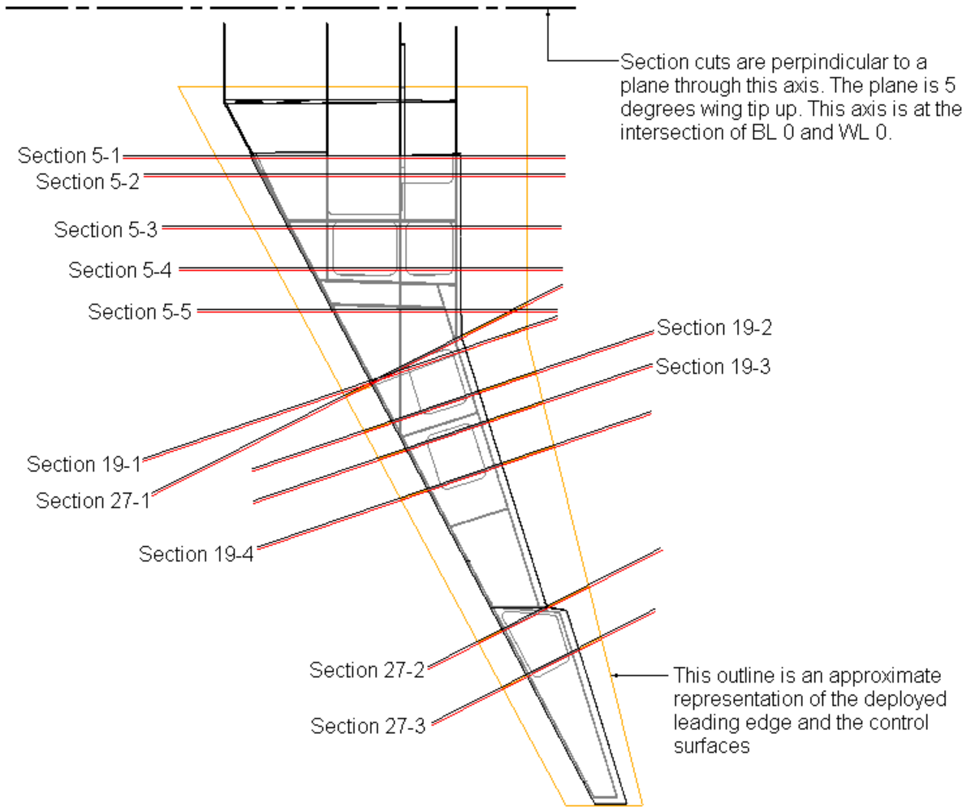
$$\sigma_{\substack{Max\\19\_4}} = 5373.4 + 5377.5$$

$$\sigma_{\substack{Max\\19\_4}} = 10751\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{10751\,psi} = 2.80}$$

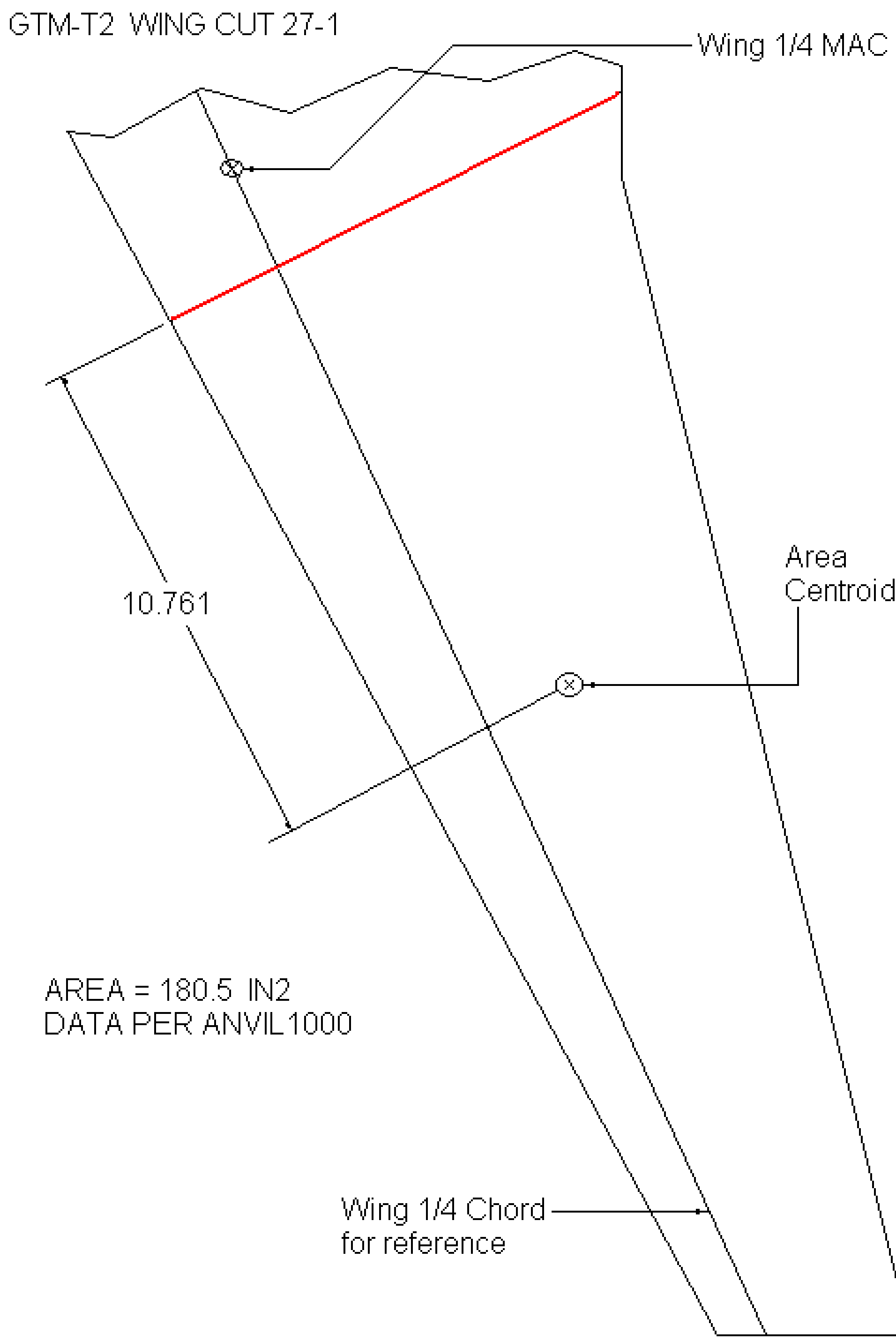
GTM-T2 Wing Section Cut Locations, 5 degrees WL 0 Offset



Drawing reference no. LD-1164893

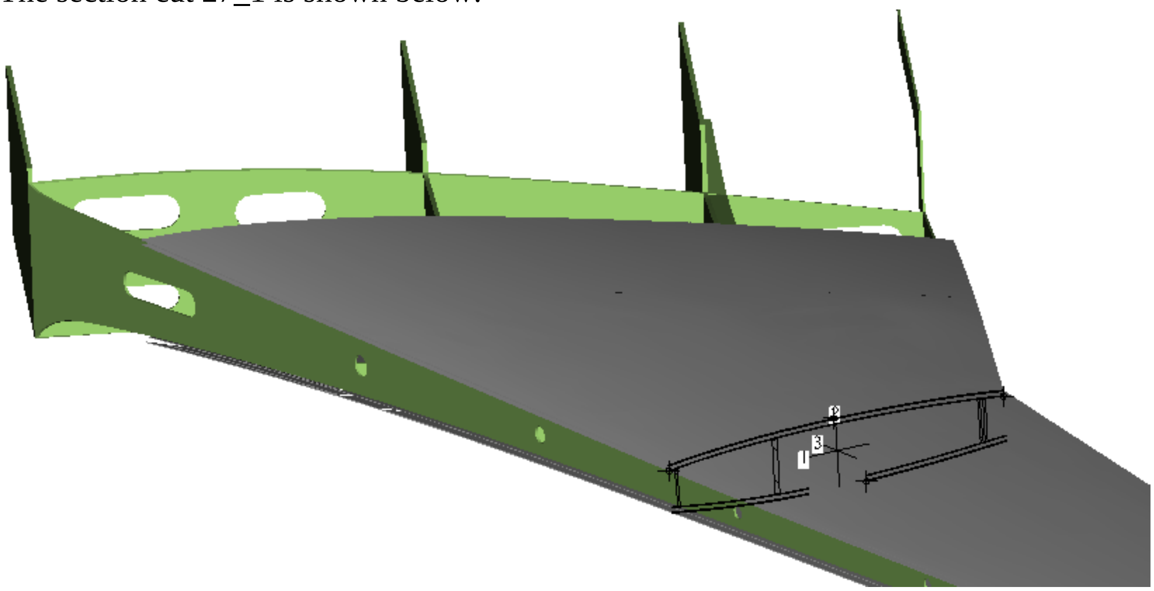
Wing Section Cut 27_1 Stress Analysis:

Since this area is outside of the quarter MAC, it is assumed that the aero load acts at the area centroid of the panel as shown below:



Drawing reference no. LD-1164907

The section cut 27_1 is shown below:



Drawing reference no. LD-1164893

The section cut 27_1 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  3.6750900e-01 INCH^2

CENTER OF GRAVITY with respect to _27_1 coordinate frame:
X   Y    -9.9486974e+00 -1.0626281e+00  INCH

INERTIA with respect to _27_1 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.4707412e+00 -3.8990786e+00
Iyx Iyy -3.8990786e+00  3.6420914e+01

POLAR MOMENT OF INERTIA:  3.7891655e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _27_1 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.0557580e+00 -1.3859504e-02
Iyx Iyy -1.3859504e-02  4.6130255e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  4.5940037e-02  1.0559482e+00

POLAR MOMENT OF INERTIA:  1.1018882e+00 INCH^4

ROTATION MATRIX from _27_1 orientation to PRINCIPAL AXES:
       0.01372       -0.99991
       0.99991        0.01372

ROTATION ANGLE from _27_1 orientation to PRINCIPAL AXES (degrees):
```

```
about z axis         89.214

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  3.5355883e-01  1.6950687e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  8.77920e-02 INCH^3       -4.3361e-01 -5.2328e-01   INCH
               9.88715e-02 INCH^3        2.4018e-02  4.6464e-01   INCH
about AXIS 2:  3.94876e-01 INCH^3       -2.6741e+00  2.5084e-01   INCH
               3.94861e-01 INCH^3        2.6742e+00  2.3852e-01   INCH
```

<u>END PRO-E DATA</u>

The lift force on the plan form area at 100*mph* (assume C_{Lmax}=1.77, reference NACA TR 427) is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = \left(180.5 in^2\right) \cdot \left(1.77\right) \cdot \left(0.1776 psi\right)$$
$$F_{Lift} = 57 lbf$$

Therefore the shear stress at section cut 27_1 is:

$$\tau_{27\_1} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{27\_1} = \frac{57 lbs}{0.3675 in^2}$$
$$\tau_{27\_1} = 154 psi$$

The bending stress at section 27_1 due to the aero load is:

$$\sigma_{27\_1} = \frac{M_{aeroload}}{z_{27\_1}} = \frac{(57 lbf)(10.761 in)}{0.0878 in^3}$$
$$\sigma_{27\_1} = 6955 psi$$

The maximum shear stress at section 27_1 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{27\_1}}{2}\right)^2 + {\tau_{27\_1}}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{6955 psi}{2}\right)^2 + \left(154 psi\right)^2}$$
$$\tau_{Max} = 3481 psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{3481\,psi} = 8.64}$$

Finally, the combined maximum stress is,

$$\sigma_{\underset{27\_1}{Max}} = \frac{6955\,psi}{2} \pm \sqrt{\left(\frac{6955\,psi}{2}\right)^2 + (154\,psi)^2}$$

$$\sigma_{\underset{27\_1}{Max}} = 3477.7 + 3481.1$$

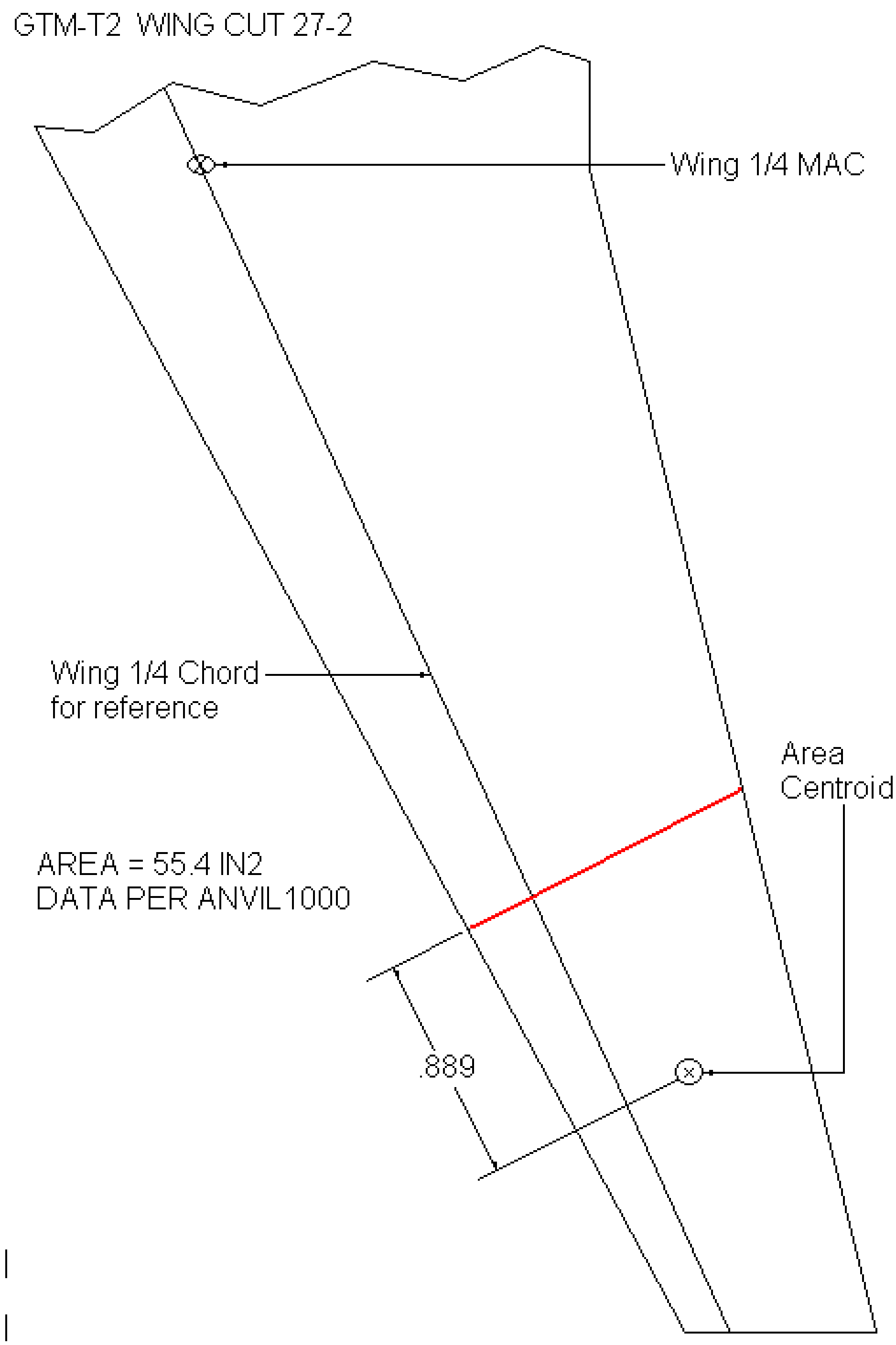
$$\sigma_{\underset{27\_1}{Max}} = 6959\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{6958\,psi} = 4.32}$$

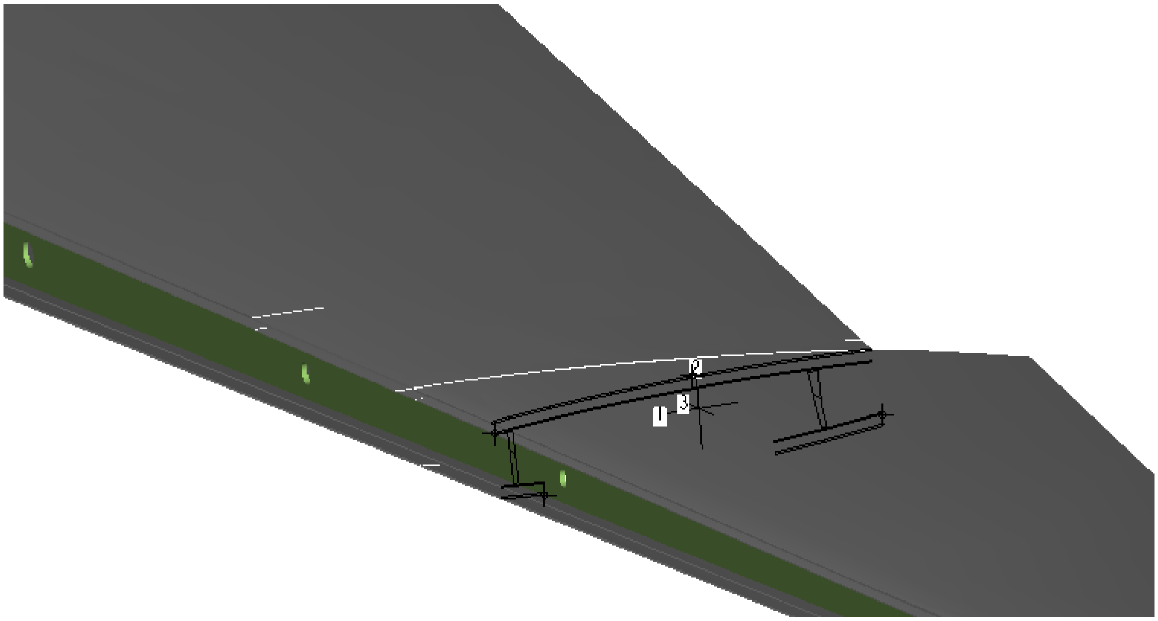
Wing Section Cut 27_2 Stress Analysis:

Since this area is outside of the quarter MAC, it is assumed that the aero load acts at the area centroid of the panel as shown below:



Drawing reference no. LD-1164907

The section cut 27_2 is shown below:



Drawing reference no. LD-1164893

The section cut 27_2 has the following Pro-Engineer properties:

START PRO-E DATA

```
AREA =  1.2285282e-01 INCH^2

CENTER OF GRAVITY with respect to _27_2 coordinate frame:
X   Y    -9.9559840e+00 -6.6918094e-02  INCH

INERTIA with respect to _27_2 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.9900042e-02 -8.1852223e-02
Iyx Iyy -8.1852223e-02  1.2182689e+01

POLAR MOMENT OF INERTIA:  1.2282590e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _27_2 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  9.9349903e-02 -3.3153220e-06
Iyx Iyy -3.3153220e-06  5.3191733e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  5.3191732e-03  9.9349903e-02

POLAR MOMENT OF INERTIA:  1.0466908e-01 INCH^4

ROTATION MATRIX from _27_2 orientation to PRINCIPAL AXES:
       0.00004       -1.00000
```

```
      1.00000        0.00004

ROTATION ANGLE from _27_2 orientation to PRINCIPAL AXES (degrees):
about z axis         89.998

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  2.0807960e-01  8.9927218e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1          2   COORD
about AXIS 1:  1.29790e-02 INCH^3        1.1099e+00 -4.0983e-01   INCH
               2.43212e-02 INCH^3        2.7392e-02  2.1871e-01   INCH
about AXIS 2:  7.90796e-02 INCH^3       -1.2563e+00 -2.2430e-01   INCH
               7.01573e-02 INCH^3        1.4161e+00  4.2660e-02   INCH
```

END PRO-E DATA

The lift force on the plan form area at 100*mph* (assume C_{Lmax}=1.77, reference NACA TR 427) is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = \left(55.4\,in^2\right) \cdot \left(1.77\right) \cdot \left(0.1776\,psi\right)$$
$$F_{Lift} = 17\,lbf$$

Therefore the shear stress at section cut 27_2 is:

$$\tau_{27\_2} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{27\_2} = \frac{17\,lbs}{0.1229\,in^2}$$
$$\tau_{27\_2} = 142\,psi$$

The bending stress at section 27_2 due to the aero load is:

$$\sigma_{27\_2} = \frac{M_{aeroload}}{z_{27\_2}} = \frac{(17\,lbs)(0.889\,in)}{0.013\,in^3}$$
$$\sigma_{27\_2} = 1193\,psi$$

The maximum shear stress at section 27_2 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{27\_2}}{2}\right)^2 + \tau_{27\_2}{}^2}$$

$$\tau_{Max} = \sqrt{\left(\frac{1193\,psi}{2}\right)^2 + (142\,psi)^2}$$

$$\tau_{Max} = 613\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{613\,psi} = 49.1}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\27\_2}} = \frac{1193\,psi}{2} \pm \sqrt{\left(\frac{1193\,psi}{2}\right)^2 + (142\,psi)^2}$$

$$\sigma_{\substack{Max\\27\_2}} = 596.5 + 613.1$$

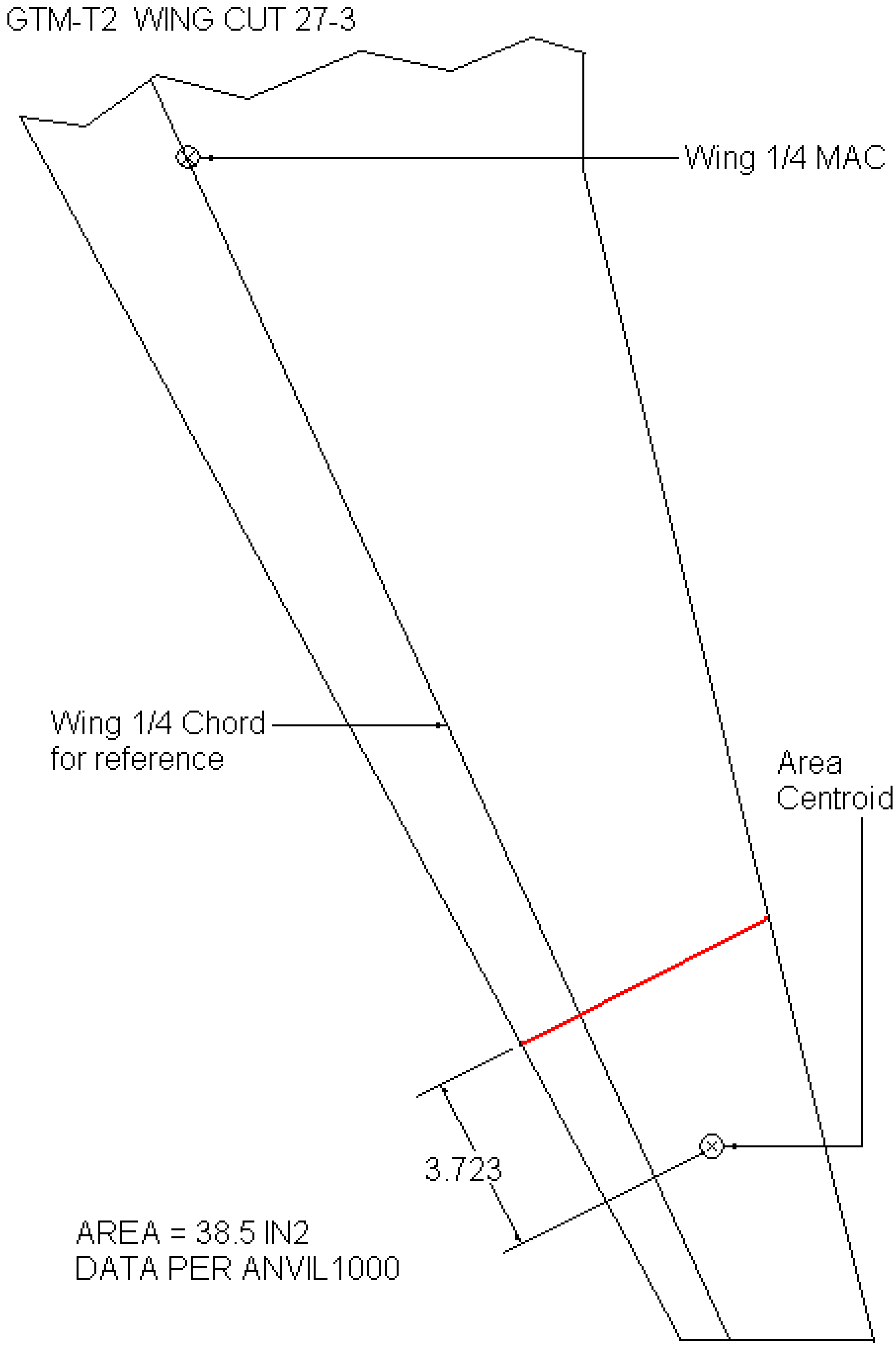
$$\sigma_{\substack{Max\\27\_2}} = 1210\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{1210\,psi} = 24.9}$$

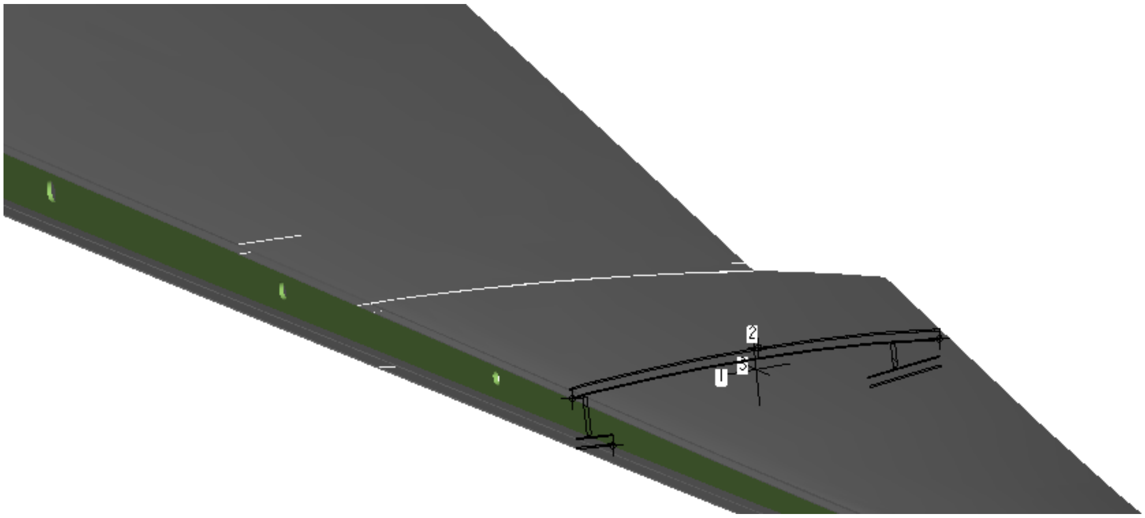
Wing Section Cut 27_3 Stress Analysis:

Since this area is outside of the quarter MAC, it is assumed that the aero load acts at the area centroid of the panel as shown below:



Drawing reference no. LD-1164907

The section cut 27_3 is shown below:



Drawing reference no. LD-1164893

The section cut 27_3 has the following Pro-Engineer properties:

<u>START PRO-E DATA</u>

```
AREA =  1.1745803e-01 INCH^2

CENTER OF GRAVITY with respect to _27_3 coordinate frame:
X   Y    -9.9749177e+00 -2.3895746e-01  INCH

INERTIA with respect to _27_3 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.3668724e-01 -2.8254737e-01
Iyx Iyy -2.8254737e-01  1.1690104e+01

POLAR MOMENT OF INERTIA:  1.1826791e+01 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _27_3 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.2998031e-01 -2.5766415e-03
Iyx Iyy -2.5766415e-03  3.1492242e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  3.0968999e-03  1.3003264e-01

POLAR MOMENT OF INERTIA:  1.3312954e-01 INCH^4

ROTATION MATRIX from _27_3 orientation to PRINCIPAL AXES:
       0.02030       -0.99979
       0.99979        0.02030

ROTATION ANGLE from _27_3 orientation to PRINCIPAL AXES (degrees):
```

```
about z axis         88.837

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.6237615e-01  1.0521674e+00  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  7.80694e-03 INCH^3        1.2430e+00 -3.9669e-01   INCH
               1.87036e-02 INCH^3       -3.8416e-02  1.6558e-01   INCH
about AXIS 2:  8.47604e-02 INCH^3       -1.5341e+00 -1.3536e-02   INCH
               8.44539e-02 INCH^3        1.5397e+00  2.8643e-02   INCH
```

END PRO-E DATA

The lift force on the plan form area at 100*mph* (assume C_{Lmax}=1.77, reference NACA TR 427) is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = \left(38.5 in^2\right) \cdot \left(1.77\right) \cdot \left(0.1776 psi\right)$$
$$F_{Lift} = 12 lbf$$

Therefore the shear stress at section cut 27_3 is:

$$\tau_{27\_3} = \frac{F_{shear}}{A_{shear}}$$
$$\tau_{27\_3} = \frac{12 lbs}{0.1175 in^2}$$
$$\tau_{27\_3} = 103 psi$$

The bending stress at section 27_3 due to the aero load is:

$$\sigma_{27\_3} = \frac{M_{aeroload}}{z_{27\_3}} = \frac{(12 lbs)(3.723 in)}{0.0078 in^3}$$
$$\sigma_{27\_3} = 5772 psi$$

The maximum shear stress at section 27_3 due to the aero load is:

$$\tau_{Max} = \sqrt{\left(\frac{\sigma_{27\_3}}{2}\right)^2 + {\tau_{27\_3}}^2}$$
$$\tau_{Max} = \sqrt{\left(\frac{5772 psi}{2}\right)^2 + \left(103 psi\right)^2}$$
$$\tau_{Max} = 2888 psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

$$\boxed{FS = \frac{30080\,psi}{2888\,psi} = 10.4}$$

Finally, the combined maximum stress is,

$$\sigma_{\substack{Max\\27\_3}} = \frac{5772\,psi}{2} \pm \sqrt{\left(\frac{5772\,psi}{2}\right)^2 + (103\,psi)^2}$$

$$\sigma_{\substack{Max\\27\_3}} = 2885.9 + 2887.8$$

$$\sigma_{\substack{Max\\27\_3}} = 5774\,psi$$

And, the factor of safety based on the carbon fiber yield strength, 30080*psi* (reference project document no. GTMP-6030) is,

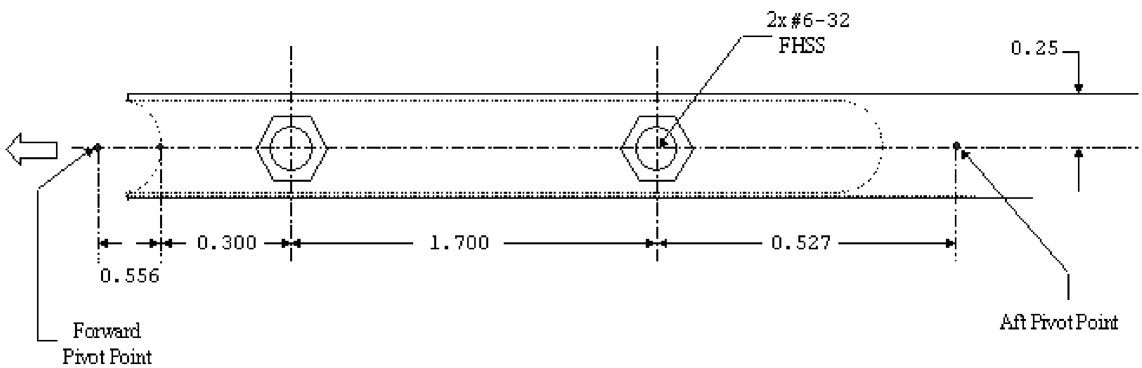
$$\boxed{FS = \frac{30080\,psi}{5774\,psi} = 5.21}$$

Wing Tip Air Data Boom

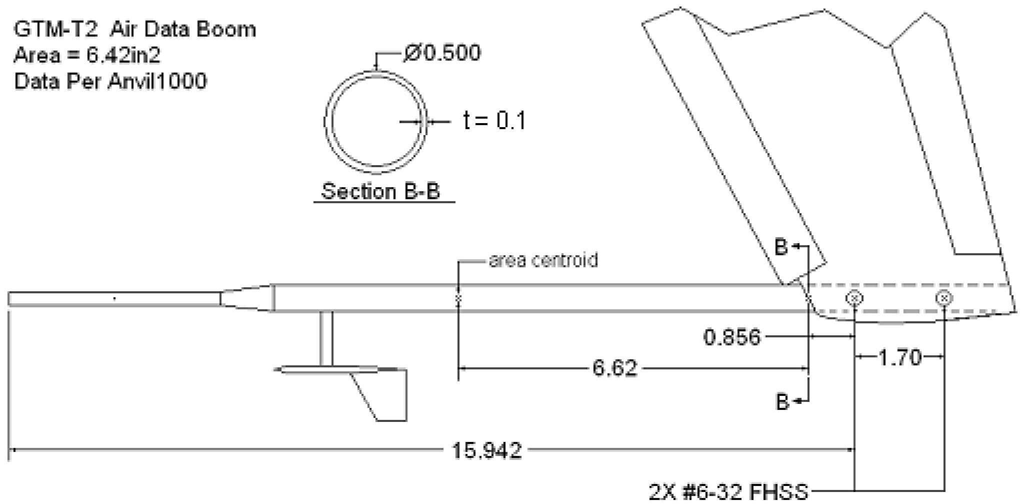
The wing tip is bolted to the underside of each wing tip using two #6-32UNC flat head socket screws (FHSS) and lock nuts. The air data booms are parallel to BL-0 and WL-0.

Bolt Tension

The maximum bolt tension is calculated at the assumed forward pivot point and the aft pivot point. The aerodynamic load is assumed to act at the area centroid. The area centroid is located at 6.62*in* away from the forward pivot point. The assumed forward and aft pivot points locations are sketched below:



Reference Drawing LD-1164949



Reference Drawing LD-1164949

Assuming that the wing is at a 100*mph* load, the air data boom area is 6.62 in^2 and the C_L=1.3.

$$F_{Lift} = A \cdot C_L \cdot q$$

Where:

F_{Lift} = force of lift, *lbs*
A = Plan form area, in^2
C_L = 1.3
q = 25.58*psf* or 0.17761*psi*

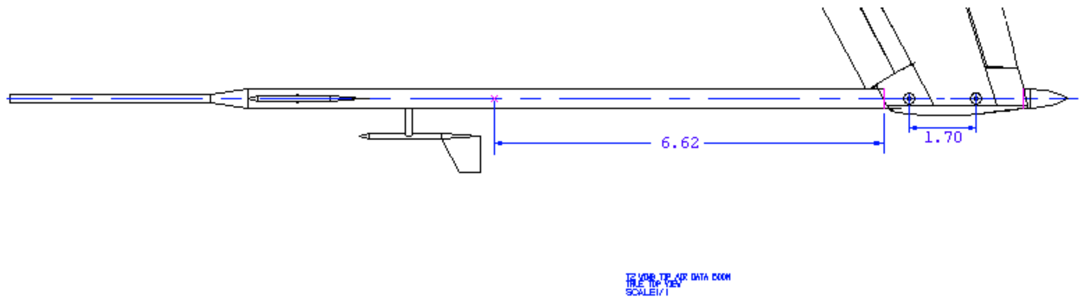
Then:

$$F_{Lift} = A \cdot C_L \cdot q$$

$$F_{Lift} = \left(6.62 in^2\right) \cdot (1.3) \cdot (0.17761\, psi)$$

$$F_{Lift} = 1.53 lbf$$

Forward Pivot Point (Located 6.62*in* **from area centroid):**



Reference Drawing LD-1164949

<u>Bolt Tensile Strength Forward Pivot Point</u>

$$FF_{\substack{Max\_bolt \\ Tension}} = \frac{M \cdot l \cdot r_{\max}}{\sum_{1}^{2} r_i^2}$$

$$FF_{\substack{Max\_bolt \\ Tension}} = \frac{(1.53lbs)(6.62in)(0.856in + 1.70in)}{(0.856in)^2 + (0.856in + 1.70in)^2}$$

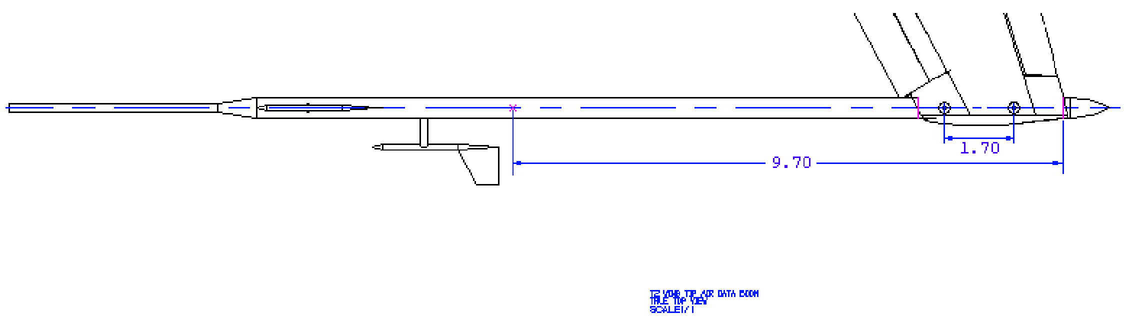
$$FF_{\substack{Max\_bolt \\ Tenion}} = \frac{25.89lbs}{7.27}$$

$$FF_{\substack{Max\_bolt \\ Tension}} = 3.56lbs.$$

The factor of safety based on a 1320*lbs* tensile strength of a #6-32UNC FHSS is:

$$\boxed{FS_{\#6-32FHSS} = \frac{1320lbs}{3.56lbs} = 370.5}$$

Aft Pivot Point (Located 9.70*in* from area centroid):



Reference Drawing LD-1164949

Bolt Tensile Strength Aft Pivot Point

$$FA_{\substack{Max\_bolt \\ Tension}} = \frac{M \cdot l \cdot r_{\max}}{\sum_{1}^{2} r_i^2}$$

$$FA_{\substack{Max\_bolt \\ Tension}} = \frac{(1.53lbs)(9.7in)(0.527in + 1.70in)}{(0.527in)^2 + (0.527in + 1.70in)^2}$$

$$FA_{\substack{Max\_bolt \\ Tenion}} = \frac{33.05lbs}{5.24}$$

$$FA_{\substack{Max\_bolt \\ Tension}} = 6.31lbs.$$

The factor of safety based on a 1320*lbs* tensile strength of a #6-32UNC FHSS is:

$$\boxed{FS_{\#6-32FHSS} = \frac{1320lbs}{6.31lbs} = 209.2}$$

Bolt-Head Shear Pull Through

The #6-32UNC FHSS have a diameter = 0.265*in* as measured by author. From the maximum bolt tension calculation above, the shear pull-through stress is:

Forward Pivot Point:

$$FF_{shear} = FF_{\substack{\max\_bolt \\ tension}} = 3.56lbs.$$

$$Area_{shear} = (median\_screw\_head\_diameter)(plate\_thickness)$$

$$Area_{shear} = \pi(d)(t) = \pi(0.202in)(0.0625in) = 0.040in^2$$

$$\therefore \tau = \frac{3.56lbs}{0.040in^2} = 89.80\,psi$$

The 1/16*in*, 5-layer plywood has at tensile strength of 7578*psi* (reference project document number GTMP-6031), therefore the factor of safety based on the plywood's tensile strength is,

$$\boxed{FS_{\substack{Plywood \\ Shear}} = \frac{7578\,psi}{89.80\,psi} = 84.4}$$

Aft Pivot Point:

$$FA_{shear} = FA_{\substack{\max\_bolt \\ tension}} = 6.31lbs.$$

$$Area_{shear} = (median\_screw\_head\_diameter)(plate\_thickness)$$

$$Area_{shear} = \pi(d)(t) = \pi(0.202)(0.0625) = 0.040in^2$$

$$\therefore \tau = \frac{6.31lbs}{0.040in^2} = 159.1psi$$

The 1/16*in*, 5-layer plywood has at tensile strength of 7578*psi* (reference project document number GTMP-6031), therefore the factor of safety based on the plywood's tensile strength is,

$$\boxed{FS_{\substack{Plywood \\ Shear}} = \frac{7578psi}{159.1psi} = 47.6}$$

<u>Model# 100400 Bending Stress</u>

$$\sigma_x = \frac{M_{aero}c}{I}$$

$$Where:$$

$$M = F_{Lift} \cdot l$$

$$c = \frac{d}{2}$$

$$I = \frac{\pi}{64}(d_o^{\ 4} - d_i^{\ 4})$$

$$F_{Lift} = A \cdot C_L \cdot q$$

$$F_{Lift} = \left(6.62in^2\right) \cdot (1.3) \cdot (0.17761psi)$$

$$F_{Lift} = 1.53lbf$$

$$c = \frac{d}{2} = \frac{0.5in}{2} = 0.25in$$

$$I = \frac{\pi}{64}(d_o^{\ 4} - d_i^{\ 4})$$

$$I = \frac{\pi}{64}((0.5in)^4 - (0.4in)^4)$$

$$I = 0.002in^4$$

$$\sigma_x = \frac{M_{aero}c}{I} = \frac{(1.53lbs)(6.62in)(0.25in)}{0.002in^4} = \frac{2.53lbs}{0.002in^2}$$
$$\sigma_x = 1398.0\,psi$$

Shear Stress:

$$\tau = \frac{F_{Lift}}{A_{Tube}}$$
$$Where:$$
$$A_{Tube} = \pi \cdot r_o{}^2 - \pi \cdot r_i{}^2$$
$$A_{Tube} = \pi \cdot (0.25in)^2) - (\pi \cdot (0.2in)^2$$
$$A_{Tube} = 0.0707in^2$$
$$\tau = \frac{1.53lbf}{0.0707in^2}$$
$$\tau = 21.62\,psi$$

Maximum Shear Stress:

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma_x}{2}\right)^2 + \tau^2}$$
$$\tau_{\max} = \pm\sqrt{\left(\frac{1398.0\,psi}{2}\right)^2 + (21.62\,psi)^2}$$
$$\tau_{\max} = 699.3\,psi$$

Factor of Safety:

The yield strength for annealed 304 stainless steel is 31200*psi*. Thus the factor of safety is:

$$\boxed{FS = \frac{31200\,psi}{699.3\,psi} = 44.6}$$

Principal Stress:

$$\sigma_{1,2} = \frac{\sigma_x}{2} \pm \sqrt{\left(\frac{\sigma_x}{2}\right)^2 + \tau^2}$$

$$\sigma_{1,2} = \frac{1398.0\,psi}{2} \pm \sqrt{\left(\frac{1398.0\,psi}{2}\right)^2 + (21.62\,psi)^2}$$

$$\sigma_{1,2} = 698.98 + 699.3$$

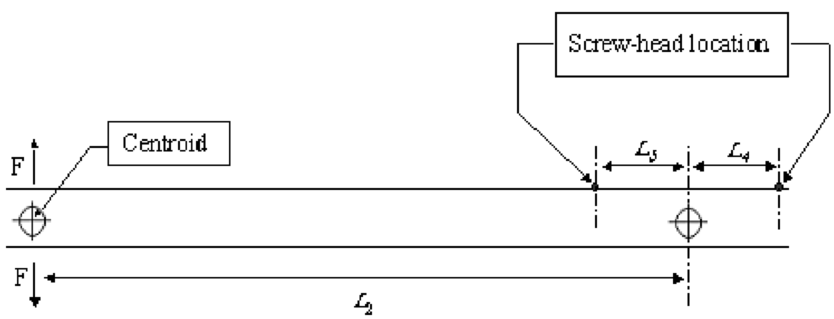
$$\sigma_{1,2} = 1398.3\,psi$$

The yield strength for annealed 304 stainless steel is 31200 *psi*. Thus the factor of safety is:

$$\boxed{FS = \frac{31200\,psi}{1389.3\,psi} = 22.5}$$

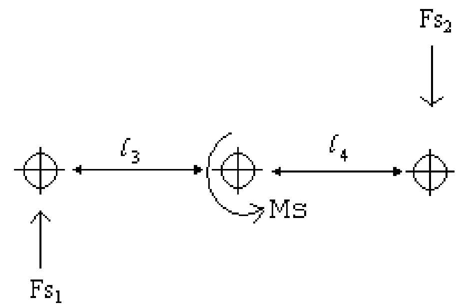
<u>Bolt Shear Load</u>: (Side load only)

It is assumed that an aggressive yaw maneuver will generate an aerodynamic side load on the air data boom. This will create a couple at the bolt attachment thus putting the fasteners in shear. A sketch of this condition is shown below.



Reference Drawing LD-1164949

This is an exploded view of the above picture displaying the screw force locations and the moment direction, as well as the distance (ℓ_3 and ℓ_4) from the center point to each screw head.



Reference Drawing LD-1164949

$$M_2 = F \cdot (l_2) = Fs_1 \cdot (l_3) + Fs_2 \cdot (l_4)$$

Assume,

$$Fs_1 = Fs_2 \text{ and } l_3 = l_4$$

Therefore,

$$F \cdot (l_2) = 2 \cdot (Fs_{1,2} \cdot l_{3,4})$$

$$(1.53lbs) \cdot (6.62in + 0.856in + 0.85in) = 2 \cdot (Fs \cdot 0.85in)$$

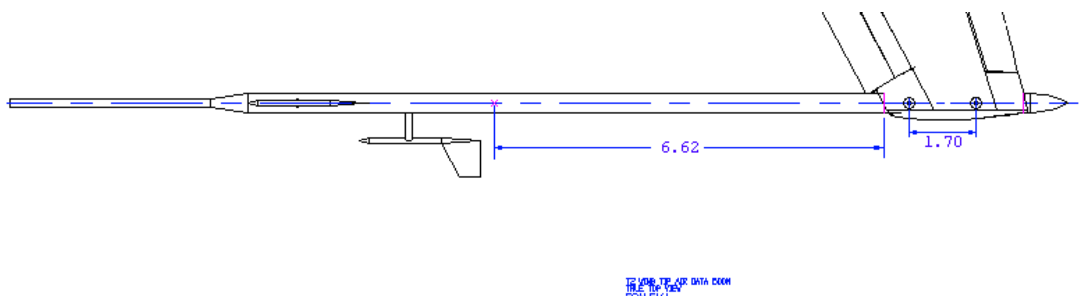
$$\therefore Fs = 7.49lbs$$

Therefore the factor of safety based on a shear strength of 940 *lbs* (threaded area) is, (what kind of screw are we using?)

$$\boxed{FS = \frac{940\,psi}{7.49\,psi} = 125.4}$$

<u>Stress Calculations for the Probe Tube at the Leading Edge of the Wing</u>

The maximum bending stress of the probe tube is assumed to be in front of the mounting boss on the end of the probe tube assembly as shown below:



Reference Drawing LD-1164949

The area centroid is located 6.62 *in* from this location. The entire air probe data boom assembly, less the boss geometry, has a calculated plan form area of 6.42 in^2. The two free air vanes are assumed to have negligible impact on the total aerodynamic load since they “free-wheel” relative to the main air data housing (tube). And, for cylindrical geometry, a coefficient of lift of 1.3 is assumed. A probe tube is analyzed for a 150 *mph* air stream velocity.

At 150 *mph* the dynamic pressure is,

$$q = \frac{1}{2} * \rho * V^2$$

Where, q = dynamic pressure (lbs./ft^2)

ρ (air density) = 0.0023769 slug/ft^3

V (speed) = 150 mph = 220 ft/sec

And,

$$1 slug = \frac{1 lb. - \sec^2}{ft}$$

So,

$$q = \frac{1}{2}\left(\frac{.0023769 slug}{ft^3}\right)\left(\frac{220\, ft}{\sec}\right)^2\left(\frac{\frac{1 lb. - \sec^2}{ft}}{1 slug}\right)$$

$$q = \frac{1}{2}\left(\frac{.0023769 slug}{ft^3}\right)\left(\frac{220^2\ ft^2}{\sec^2}\right)\left(\frac{1}{1 slug}\right)\left(\frac{1 lb. - \sec^2}{ft}\right)$$

Therefore,

q = 57.5 *psf* = 0.399 *psi*

Therefore the aerodynamic lift is,

$$F_{Lift} = A \cdot C_L \cdot q$$
$$F_{Lift} = (6.42 in^2) \cdot (1.3) \cdot (0.399\, psi)$$
$$F_{Lift} = 3.33\ lbf$$

The tube has an outside diameter of 0.50 *in* and the inside diameter is 0.40 *in* then,

The shear stress is,

$$\tau = \frac{F_{Lift}}{A_{Tube}}$$

$Where:$

$$A_{Tube} = \pi(r_o{}^2 - r_i{}^2)$$

$$A_{Tube} = \pi[(0.25in)^2 - (0.2in)^2]$$

$$A_{Tube} = \pi[0.0625 - 0.04] = 0.0707in^2$$

$$\tau = \frac{3.33lbf}{0.070in^2}$$

$$\tau = 47.0\,psi$$

The bending stress is,

$$\sigma = \frac{M_{aero}c}{I}$$

$Where:$

$$M_{aero} = F_{Lift}(l)$$

$where:$

F_{Lift} =3.33 lbs

l=6.62 in

$$c = \frac{d}{2} = \frac{0.50in}{2} = 0.25\ in$$

$$I = \frac{\pi}{64}(d_o{}^4 - d_i{}^4) = \frac{\pi}{64}((0.50in)^4 - (0.40in)^4) = \frac{\pi}{64}(0.0625in^4 - 0.0256in^4)$$

$$I = 0.0018\ in^4$$

Therefore,

$$\sigma = \frac{M_{aero}c}{I} = \frac{(3.33lbs)(6.62in)(0.25in)}{0.0018in^4} = \frac{5.51lbs}{0.0018in^2}$$

$$\sigma = 3042.6\,psi$$

Maximum Shear Stress:

$$\tau_{\max} = \pm\sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$

$$\tau_{\max} = \pm\sqrt{\left(\frac{3042.6\,psi}{2}\right)^2 + \left(47\,psi\right)^2}$$

$$\tau_{\max} = 1522\,psi$$

The ultimate shear strength for annealed 304 stainless steel is 73200*psi*. Thus the factor of safety is:

$$\boxed{FS = \frac{73200\,psi}{1522\,psi} = 48.1}$$

Principal Stress:

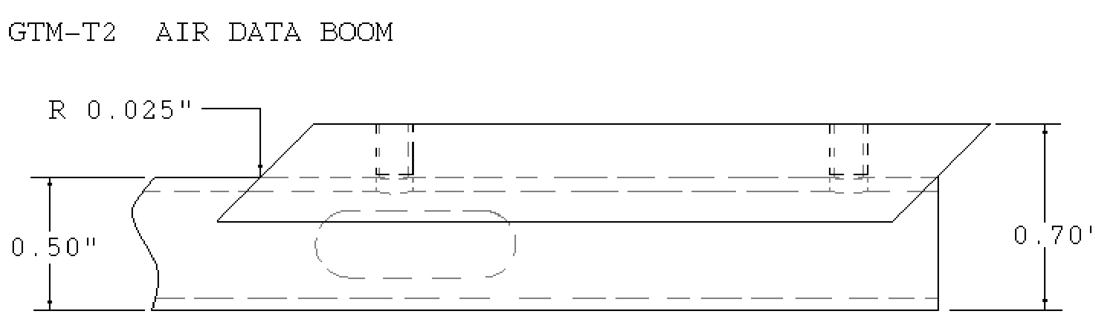
$$\sigma = \frac{\sigma}{2} \pm \sqrt{\left(\frac{\sigma}{2}\right)^2 + \tau^2}$$

$$\sigma = \frac{3042.6\,psi}{2} \pm \sqrt{\left(\frac{3042.6\,psi}{2}\right)^2 + \left(47\,psi\right)^2}$$

$$\sigma = 1521.3 + 1522$$

$$\sigma = 3043.3\,psi$$

And, for a conservative analysis, a stress concentration factor is needed for where the tube body meets the mounting boss as shown below:



Drawing reference: 101100_01

In the above sketch, a 0.025*in* radius is assumed due to the manufacturing technique used in the fabrication process. Therefore, referencing "Stress Concentration Factors" by Peterson, © 1974, figure 73, the following can be used:

D = 0.70
d = 0.50
r = 0.025

So, r/d = 0.05 and D/d = 1.40 (assume 1.5)

Therefore, from figure 73 (see Appendix) the stress concentration factor is read as:

$K_t = 2.2$

Then, the maximum bending stress is,

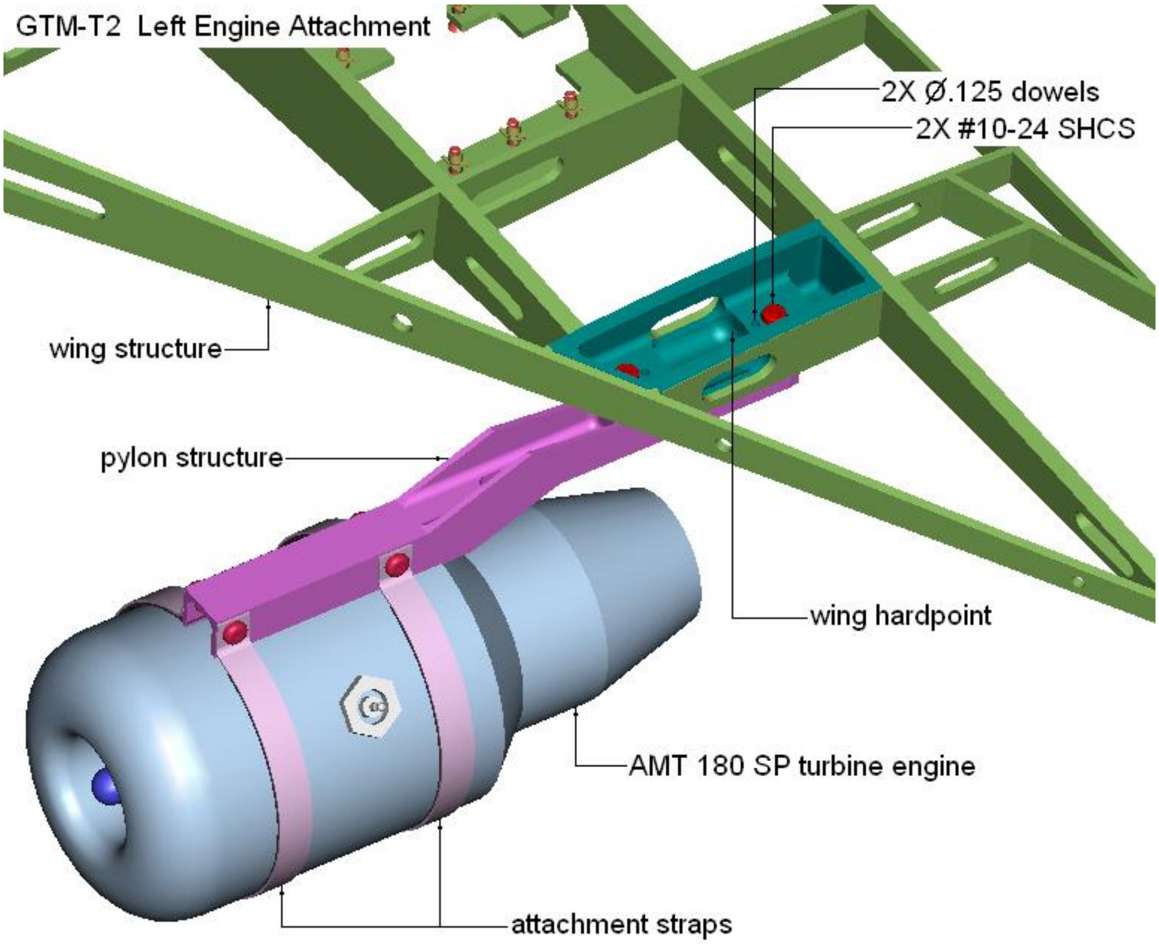
$$\sigma_{MAX} = K_t\,\sigma = (2.2)(3043.3 psi)$$
$$\sigma_{MAX} = 6695.3\ psi$$

The tensile yield strength for annealed 304 stainless steel is 31200*psi* and thus the factor of safety is:

$$\boxed{FS = \frac{31200\,psi}{6695.3\ psi} = 4.66}$$

Engine Pact Attachment Analysis:

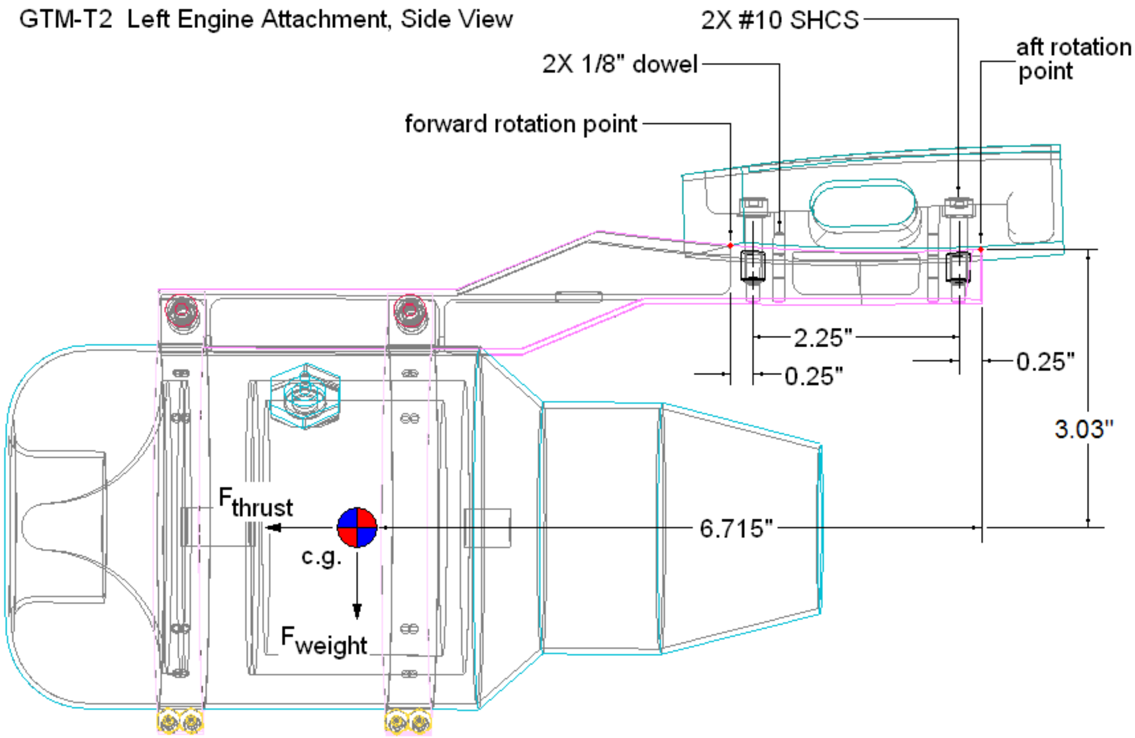
The engine pacts refer to the left and right engine, nacelle and pylon assemblies. They are attached to the wing via an aluminum pylon sub-structure. Loads are delivered from the "AMT180-SP" turbine engine only. The weight of the turbine engine is 3.2*lbs*. The available thrust of the turbine engine is assumed to be 20.5 *lbs*, which was obtained from the engine test program performed on the GTM program. The isometric view below shows this arrangement without the wing, nacelle or pylons skins:



Drawing reference: LD-1164922

Wing Hard-point to Pylon Attachment, Bolt Stress & Dowel Shear Load:

The wing hard-point to pylon attachment bolts is analyzed below. It is assumed that the bolts carry no shear load since the pylon locates to the wing hard-point with two eighth-inch dowels. There are two #10-24 UNC SHCS that attach the pylon to the hard-point. The maximum engine thrust is assumed to be 20.5*lbs*. A side view of this arrangement is given below:



Drawing reference: LD-1164922

<u>Bolt Tension, Forward Rotation Point, Engine Thrust Loading:</u>

The moment at the forward pivot point due to engine thrust is,

$$M_{forward-pt} = F_{thrust}(3.03in)$$
$$M_{forward-pt} = (20.5lbs)(3.03in)$$
$$M_{forward-pt} = 62.1\ in - lbs$$

Computing the maximum bolt tension at the forward located pivot point,

$$F_{\substack{Max\\Tension}} = \frac{Mr_{\max}}{\sum_{1}^{2} r_i^2}$$
$$F_{\substack{Max\\Tension}} = \frac{(62.1in - lbs)(0.25in + 2.25in)}{(0.25in)^2 + (0.25in + 2.25in)^2}$$
$$F_{\substack{Max\\Tension}} = \frac{155.3}{6.3125}$$
$$F_{\substack{Max\\Tension}} = 24.6\ lbs.$$

Then the factor of safety based on the #10-24UNC SHCS ("Holo-Krome" manufactured) is:

$$\boxed{FS = \frac{3150\ lbs}{24.6\ lbs} = 128}$$

<u>Dowel Pin Shear due to Engine Thrust:</u>
Assume the entire shear load is experienced by one dowel pin due to the maximum thrust of the engine. The 0.125*in* dowel pin has a single shear strength of 1840*lbs*., so the factor of safety is,

$$\boxed{FS = \frac{1840lbs}{20.5lbs} = 89}$$

<u>Bolt Tension, Forward Rotation Point, Ground Impact Loading:</u>
This analysis will identify localized tension in the engine attachment bolts during a less than ideal landing. Half the models weight during a 10 "g" landing will be analyzed.

The moment at the forward pivot point due a ground impact load is,

$$M_{forward-pt} = F_{ground}(10"g")(3.03in)$$
$$Assume: F_{ground} = (1/2)(Model\ \ Weight) = 27.5\ lbs$$
$$M_{forward-pt} = (27.5lbs)(10)(3.03in)$$
$$M_{forward-pt} = 833.3\ in - lbs$$

Computing the maximum bolt tension at the forward located pivot point,

$$F_{\substack{Max\\Tension}} = \frac{Mr_{\max}}{\sum_{1}^{2} r_i^2}$$
$$F_{\substack{Max\\Tension}} = \frac{(833.3in - lbs)(0.25in + 2.25in)}{(0.25in)^2 + (0.25in + 2.25in)^2}$$
$$F_{\substack{Max\\Tension}} = \frac{2083.3lbs}{6.3125}$$
$$F_{\substack{Max\\Tension}} = 330\ lbs.$$

Then the factor of safety based on the #10-24UNC SHCS ("Holo-Krome" manufactured) is:

$$\boxed{FS = \frac{3150\ lbs}{330\ lbs} = 9.5}$$

Bolt Tension, Aft Rotation Point:

This analysis will identify localized tension in the engine attachment bolts during a less than ideal landing. The engine weight during a 10 ”g” landing will be analyzed.

The moment at the aft pivot point due to an assumed 10“g” maneuvering load is,

$$M_{aft-pt} = F_{weight}(10g)(6.715in)$$
$$where:$$
$$F_{weight} = weight\ \ of\ \ engine$$
$$M_{aft-pt} = (3.2lbs)(10)(6.715in)$$
$$M_{aft-pt} = 215\ in-lbs$$

Computing the maximum bolt tension with pivoting at the aft rotation point gives,

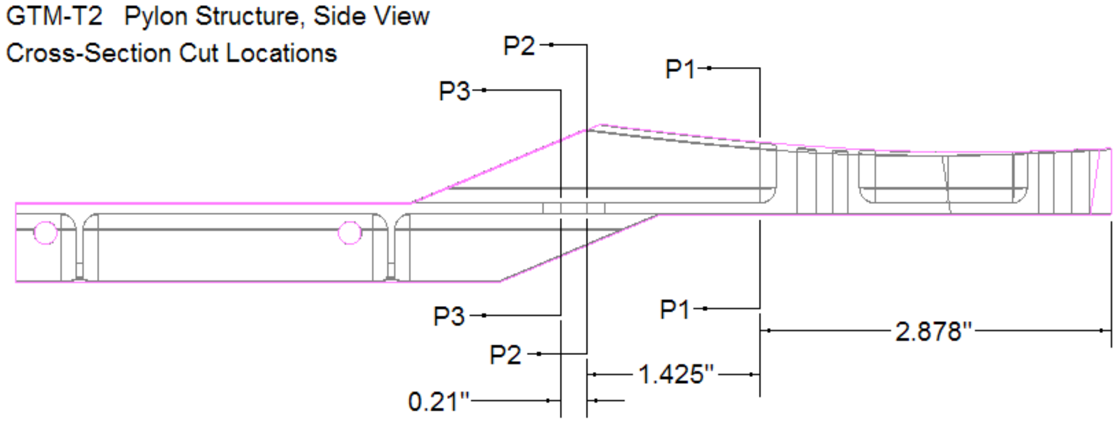
$$F_{\substack{Max\\Tension}} = \frac{Mr_{\max}}{\sum_{1}^{2} r_i^2}$$
$$F_{\substack{Max\\Tension}} = \frac{(215in-lbs)(0.25in+2.25in)}{(0.25in)^2+(0.25in+2.25in)^2}$$
$$F_{\substack{Max\\Tension}} = \frac{537.5lbs}{6.3125}$$
$$F_{\substack{Max\\Tension}} = 85\ lbs.$$

Then the factor of safety based on the #10-24UNC SHCS (“Holo-Krome” manufactured) is:

$$\boxed{FS = \frac{3150\ lbs}{85\ lbs} = 37}$$

Pylon Bending Stresses:

The pylon structure is analyzed for a combined loading of bending and shear. The pylon is analyzed in three locations as sketched below:

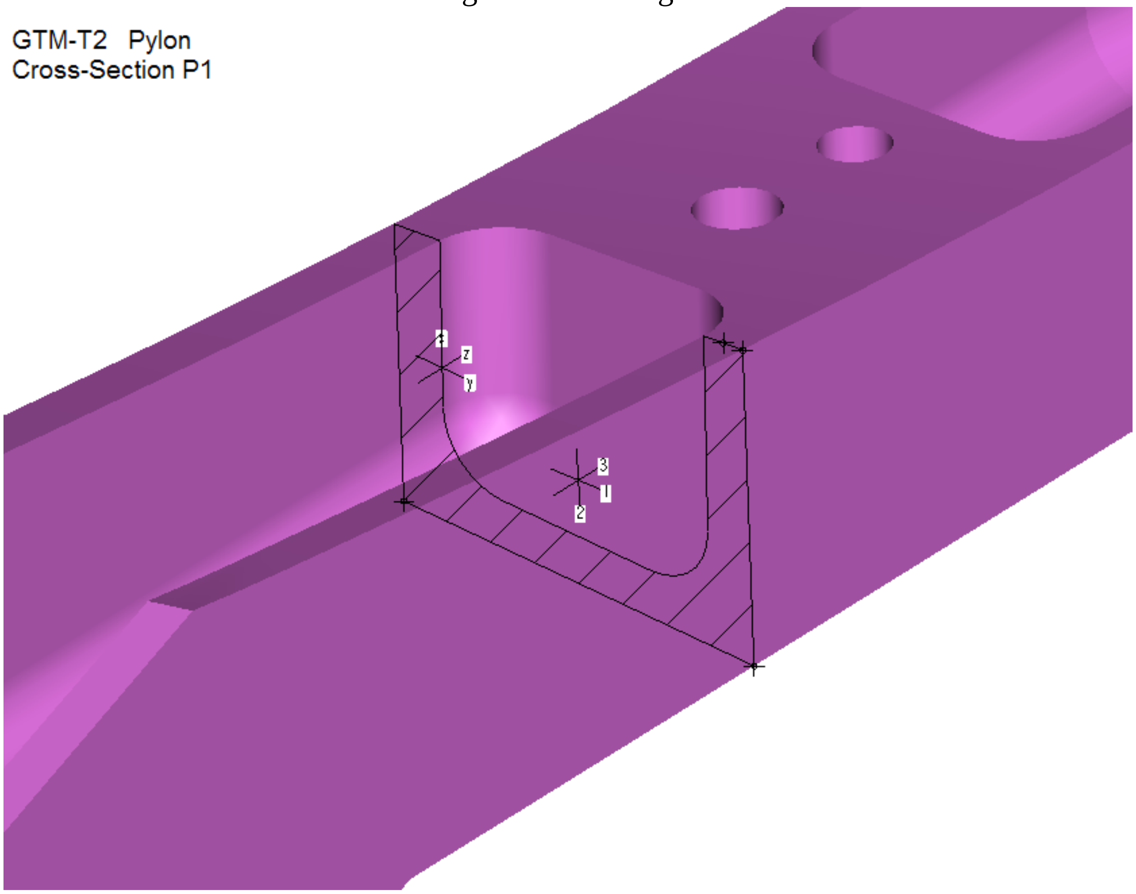


Drawing reference: LD-1164911

And, it is assumed that the tensile stresses at the cross-section cuts are due to the engine thrust and the shear stresses at due to a 10"*g*" engine weight loading. Also, it is assumed that the bending stresses at the cross-sections are due to both the engine thrust and 10"*g*" engine weight loading. The moment arm for engine thrust is assumed to be conservative since it is located above the neutral bending axis location for all three cross-sections.

Stress Calculations at Cross-Section P1:

The cross-section P1 and its Pro-Engineer data are given below:



Drawing reference: LD-1164911

START DATA

```
AREA =  1.5972256e-01 INCH^2

CENTER OF GRAVITY with respect to _P1 coordinate frame:
X   Y    -9.2616002e-02  2.8946572e-01  INCH

INERTIA with respect to _P1 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.6306578e-02  3.8559236e-03
Iyx Iyy  3.8559236e-03  6.0151183e-03

POLAR MOMENT OF INERTIA:  3.2321696e-02 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _P1 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  1.2923360e-02 -4.2610356e-04
```

```
Iyx Iyy -4.2610356e-04  4.6450623e-03

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  4.6231876e-03  1.2945235e-02

POLAR MOMENT OF INERTIA:  1.7568423e-02 INCH^4

ROTATION MATRIX from _P1 orientation to PRINCIPAL AXES:
       0.05127       -0.99868
       0.99868        0.05127

ROTATION ANGLE from _P1 orientation to PRINCIPAL AXES (degrees):
about z axis         87.061

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  1.7013264e-01  2.8468976e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                     1           2   COORD
about AXIS 1:  1.22599e-02 INCH^3        3.3937e-01 -3.7710e-01   INCH
               2.13444e-02 INCH^3        3.6102e-01  2.1660e-01   INCH
about AXIS 2:  3.33546e-02 INCH^3       -3.8811e-01  1.7814e-01   INCH
               3.39569e-02 INCH^3        3.8123e-01 -3.7910e-01   INCH
```

<u>END DATA</u>

Shear Stress:

$$\tau_{P1} = \frac{F_{weight} @10''g''}{A_{\substack{shear \\ \sec P1}}} = \frac{3.2lbs(10)}{0.15in^2}$$

$$\tau_{P1} = 200.3psi$$

Tensile Stress:

$$\sigma_{P1} = \frac{F_{thrust}}{A_{\substack{tensile \\ \sec P1}}} = \frac{20.5lbs}{0.15in^2}$$

$$\sigma_{P1} = 128.3psi$$

Bending Stress due to F_{thrust}:

$M_{thrust} = (F_{thrust})(3.03in) = (20.5lbs)(3.03in) = 62.1$ *in-lbs*

$$\sigma_{\substack{thrust \\ P1}} = \frac{M_{thrust}}{z_{P1}} = \frac{62.1in - lbs}{0.0122in^3}$$

$$\sigma_{\substack{thrust \\ P1}} = 5066.5psi$$

Bending Stress due to F_{weight}:

M_{weight} = (F_{weight})(10"g')(6.715*in*-2.878*in*) = (3.2*lbs*)(10)(3.837*in*) = 123 *in-lbs*

$$\sigma_{\underset{P1}{weight}} = \frac{M_{weight}}{z_{P1}} = \frac{123 in - lbs}{0.0122 in^3}$$
$$\sigma_{\underset{P1}{weight}} = 10015\, psi$$

Then,

$$\sigma_{\underset{P1}{Total}} = 128.3 + 5066.5 + 10015$$
$$\sigma_{\underset{P1}{Total}} = 15210\, psi$$

Therefore the combined stress is,

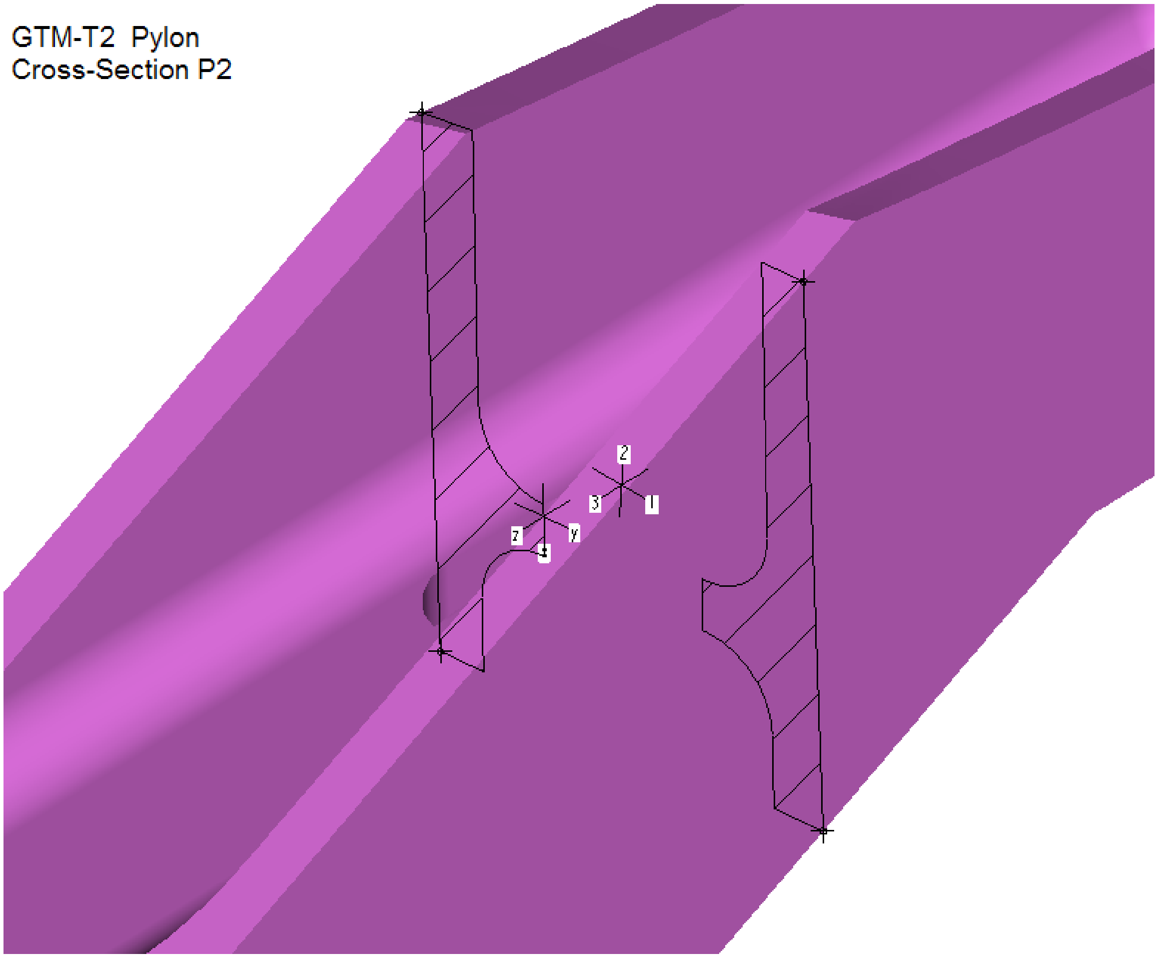
$$\sigma_{\underset{P1}{Max}} = \frac{15210\, psi}{2} \pm \sqrt{\left(\frac{15210\, psi}{2}\right)^2 + \left(200.3\, psi\right)^2}$$
$$\sigma_{\underset{P1}{Max}} = 7605.0 + 7607.6$$
$$\sigma_{\underset{P1}{Max}} = 15213\, psi$$

And, the material is aluminum 7075-T651 with a yield strength of 62000*psi*, so the factor of safety for pylon cross-section P1 is,

$$\boxed{FS_{P1-P1} = \frac{62000\, psi}{15213\, psi} = 4.07}$$

Stress Calculations at Cross-Section P2:

The cross-section P2 and its Pro-Engineer data are given below:



Drawing reference: LD-1164911

START DATA

```
AREA =  2.0807710e-01 INCH^2

CENTER OF GRAVITY with respect to _P2 coordinate frame:
X   Y    -1.1995735e-01  1.5606283e-01  INCH

INERTIA with respect to _P2 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.5854255e-02  3.1894272e-03
Iyx Iyy  3.1894272e-03  1.6651118e-02

POLAR MOMENT OF INERTIA:  4.2505372e-02 INCH^4
```

```
INERTIA at CENTER OF GRAVITY with respect to _P2 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.0786411e-02 -7.0595976e-04
Iyx Iyy -7.0595976e-04  1.3656937e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.3587705e-02  2.0855643e-02

POLAR MOMENT OF INERTIA:  3.4443348e-02 INCH^4

ROTATION MATRIX from _P2 orientation to PRINCIPAL AXES:
       0.09760       -0.99523
       0.99523        0.09760

ROTATION ANGLE from _P2 orientation to PRINCIPAL AXES (degrees):
about z axis          84.399

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  2.5554119e-01  3.1659179e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2    COORD
about AXIS 1:  2.85747e-02 INCH^3       -3.2078e-01 -4.7552e-01   INCH
               2.51287e-02 INCH^3        3.1669e-01  5.4072e-01   INCH
about AXIS 2:  4.87637e-02 INCH^3       -4.2769e-01  4.4890e-01   INCH
               4.89856e-02 INCH^3        4.2575e-01 -4.0231e-01   INCH
```

<u>END DATA</u>

Shear Stress:

$$\tau_{P2} = \frac{F_{weight} @ 10" g"}{A_{\substack{shear \\ \sec P2}}} = \frac{3.2 lbs(10)}{0.208 in^2}$$

$$\tau_{P2} = 154\, psi$$

Tensile Stress:

$$\sigma_{P2} = \frac{F_{thrust}}{A_{\substack{tensile \\ \sec P2}}} = \frac{20.5 lbs}{0.208 in^2}$$

$$\sigma_{P2} = 99\, psi$$

And, the stress concentration factor (fig. 86, Stress Concentration Factors, Peterson, © 1974) for this tensile case is,

Hole diameter, a = 0.31*in*
Material width, w = 0.75*in*

a/w = 0.31/0.75 = 0.413 therefore assume a/w = 0.415

Then, K_{tg} = 3.82

So,

$$\sigma_{\substack{tensile\\ \max\\ P2}} = K_t \sigma_{P2} = (3.82)(99\,psi)$$

$$\sigma_{\substack{tensile\\ \max\\ P2}} = 378\,psi$$

Bending Stress due to F_{thrust}:

M_{thrust} = (F_{thrust})(3.03*in*) = (20.5*lbs*)(3.03*in*) = 62.1 *in-lbs*

$$\sigma_{\substack{thrust\\ P2}} = \frac{M_{thrust}}{z_{P1}} = \frac{62.1in-lbs}{0.025in^3}$$

$$\sigma_{\substack{thrust\\ P2}} = 2472\,psi$$

Bending Stress due to F_{weight}:

M_{weight} = (F_{weight})(10"g')(6.715*in*-2.878*in*-1.425*in*) = (3.2*lbs*)(10)(2.412*in*) = 77.2 *in-lbs*

$$\sigma_{\substack{weight\\ P2}} = \frac{M_{weight}}{z_{P2}} = \frac{77.2in-lbs}{0.025in^3}$$

$$\sigma_{\substack{weight\\ P2}} = 3072\,psi$$

Then,

$$\sigma_{\substack{Total\\ P2}} = 378\,psi + 2472\,pso + 3072\,psi$$

$$\sigma_{\substack{Total\\ P2}} = 5922\,psi$$

Therefore the combined stress is,

$$\sigma_{\substack{Max\\P2}} = \frac{5922\,psi}{2} \pm \sqrt{\left(\frac{5922\,psi}{2}\right)^2 + (154\,psi)^2}$$

$$\sigma_{\substack{Max\\P2}} = 2960.8 + 2964.8$$

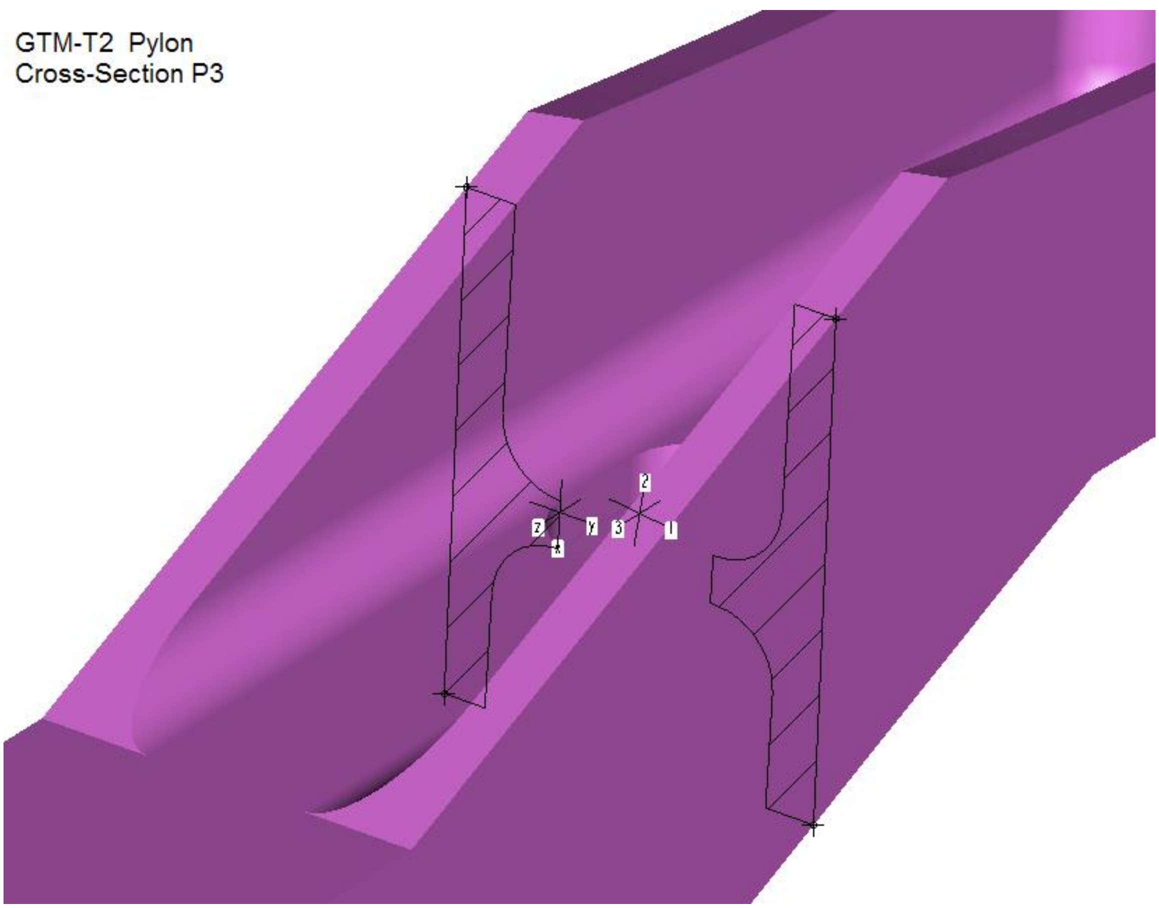
$$\sigma_{\substack{Max\\P2}} = 5925.6\,psi$$

And, the material is aluminum 7075-T651 with a yield strength of 62000psi, so the factor of safety for pylon cross-section P2 is,

$$\boxed{FS_{P2-P2} = \frac{62000\,psi}{5954\,psi} = 10.5}$$

Stress Calculations at Cross-Section P3:

The cross-section P3 and its Pro-Engineer data are given below:



Drawing reference: LD-1164911

START DATA

```
AREA =  2.0920433e-01 INCH^2

CENTER OF GRAVITY with respect to _P3 coordinate frame:
X   Y    -5.0460369e-02  1.5884366e-01  INCH

INERTIA with respect to _P3 coordinate frame:  (INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.6201840e-02  7.9375520e-04
Iyx Iyy  7.9375520e-04  1.3805440e-02

POLAR MOMENT OF INERTIA:  4.0007280e-02 INCH^4

INERTIA at CENTER OF GRAVITY with respect to _P3 coordinate frame:
(INCH^4)

INERTIA TENSOR:
Ixx Ixy  2.0923341e-02 -8.8308229e-04
Iyx Iyy -8.8308229e-04  1.3272754e-02

AREA MOMENTS OF INERTIA with respect to PRINCIPAL AXES:  (INCH^4)
I1  I2  1.3172145e-02  2.1023950e-02

POLAR MOMENT OF INERTIA:  3.4196095e-02 INCH^4

ROTATION MATRIX from _P3 orientation to PRINCIPAL AXES:
       0.11320       -0.99357
       0.99357        0.11320

ROTATION ANGLE from _P3 orientation to PRINCIPAL AXES (degrees):
about z axis         83.500

RADII OF GYRATION with respect to PRINCIPAL AXES:
R1  R2  2.5092442e-01  3.1700917e-01  INCH


SECTION MODULI and corresponding points:

                    MODULUS                    1           2   COORD
about AXIS 1:  2.63816e-02 INCH^3       -3.1240e-01 -4.9929e-01   INCH
               2.50031e-02 INCH^3        3.0906e-01  5.2682e-01   INCH
about AXIS 2:  4.81939e-02 INCH^3       -4.3624e-01  4.4191e-01   INCH
               4.85663e-02 INCH^3        4.3289e-01 -4.1438e-01   INCH
```

END DATA

Shear Stress:

$$\tau_{P3} = \frac{F_{weight} @ 10"g"}{A_{\substack{shear \\ \sec P3}}} = \frac{3.2lbs(10)}{0.209in^2}$$

$$\tau_{P3} = 153psi$$

Tensile Stress:

$$\sigma_{P3} = \frac{F_{thrust}}{A_{\substack{tensile \\ \sec\ P3}}} = \frac{20.5lbs}{0.209in^2}$$

$$\sigma_{P3} = 98\,psi$$

And, the stress concentration factor (fig. 86, Stress Concentration Factors, Peterson, © 1974) for this tensile case is,

Hole diameter, a = 0.31*in*
Material width, w = 0.751*in*

a/w = 0.31/0.75 = 0.413 therefore assume a/w = .415

Then, K_{tg} = 3.82

So,

$$\sigma_{\substack{tensile \\ \max \\ P3}} = K_t \sigma_{P3} = (3.82)(98\,psi)$$

$$\sigma_{\substack{tensile \\ \max \\ P3}} = 374\,psi$$

Bending Stress due to F_{thrust}:

M_{thrust} = (F_{thrust})(3.03*in*) = (20.5*lbs*)(3.03*in*) = 62.1 *in-lbs*

$$\sigma_{\substack{thrust \\ P3}} = \frac{M_{thrust}}{z_{P3}} = \frac{62.1in - lbs}{0.025in^3}$$

$$\sigma_{\substack{thrust \\ P3}} = 2484\,psi$$

Bending Stress due to F_{weight}:

M_{weight} = (F_{weight})(10”g’)(6.715*in*-2.878*in*-1.425*in*-0.21*in*)
= (3.2*lbs*)(10)(2.202*in*)
= 70.5 *in-lbs*

$$\sigma_{\substack{weight \\ P3}} = \frac{M_{weight}}{z_{P3}} = \frac{70.5in - lbs}{0.025in^3}$$
$$\sigma_{\substack{weight \\ P3}} = 2818\,psi$$

Then,

$$\sigma_{\substack{Total \\ P3}} = 374\,psi + 2484\,psi + 2818\,psi$$
$$\sigma_{\substack{Total \\ P3}} = 5677\,psi$$

Therefore the combined stress is,

$$\sigma_{\substack{Max \\ P3}} = \frac{5677\,psi}{2} \pm \sqrt{\left(\frac{5677\,psi}{2}\right)^2 + (153\,psi)^2}$$
$$\sigma_{\substack{Max \\ P3}} = 2838.4 + 2842.5$$
$$\sigma_{\substack{Max \\ P3}} = 5681\,psi$$

And, the material is aluminum 7075-T651 with a yield strength of 62000*psi*, so the factor of safety for pylon cross-section P3 is,

$$\boxed{FS_{P3-P3} = \frac{62000\,psi}{5681\,psi} = 10.9}$$

Wing Assembly Attachment:

The wing assembly including the engine pacts, has a weight of 18.60 *lbs*., bolts to sub-structure locate in the center of the fuselage. The wing assembly is attached using four #10-32UNF SHCS, which locate into "TEE" nuts embedded in the fuel bay floor plate. The wing assembly "locks-into" the fuselage assembly therefore the bolts are assumed to be primarily in tension.

Wing Assembly Attachment, Bolt Loads:

Assuming that the vehicle is in high-speed dive pull out and the wing assembly experiences a 6"g" pull-out loading.

$$W_{wing+engines} = 18.6\ lbs$$

$$W_{wing+engines} @ 6"g" = (6)(18.6) = 111.6\ lbs$$

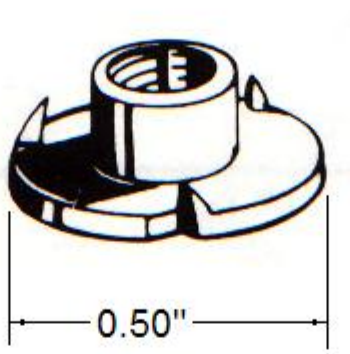
$$\therefore F_{\substack{each \\ bolt}} = \frac{111.6\ lbs}{4\ bolts} = 27.9\ lbs\ \ per\ bolt$$

And, the #10-32UNF SHCS have a tensile capability of 3600 *lbs*. so the factor of safety is:

$$\boxed{FS_{\substack{1/4-20 \\ SHCS}} = \frac{3600\ lbs}{27.9\ lbs} = 129}$$

<u>Blind-Nut, shear pull through stress:</u>

Four blind-nuts are used in the fuel bay floor for the wing assembly attachment. The blind-nuts are #10-32UNF and are from "Dubro", catalog number 584. This style of blind-nut has an approximate head diameter of 0.50*in*. A photo of this nut is given below:



$$\tau = \frac{F_{bolt}}{A_{shear}}$$

$$A_{shear} = C\,t = \pi\,D\,t$$

$$where, D = 0.50\,in$$

$$and,\ t = 0.125\,in\ (fuel\ \ bay\ \ floor - plate\ thickness)$$

$$So,$$

$$\tau = \frac{27.9\ lbs}{\pi(0.50in)(0.125in)} = \frac{27.9}{0.196}$$

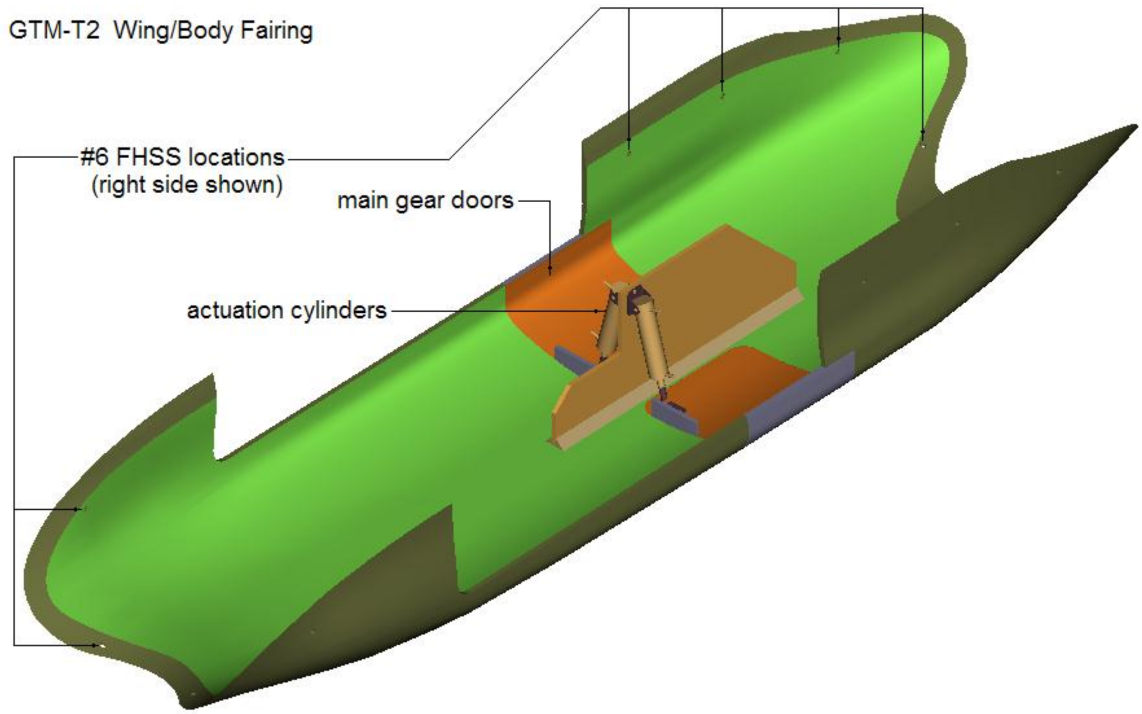
$$\tau = 142.1\ \ psi$$

The factor of safety based on aircraft plywood yield strength equivalent to 7578*psi* is,

$$\boxed{FS = \frac{7578\ \ psi}{142.1\ \ psi} = 53.3}$$

Wing Fairing Attachment

The fairing is a 1/4*in* thick fiberglass composite assembly (weight = 0.735 *lbs* including the 2 air cylinders and a mounting plate) that interfaces at the intersection of the lower fuselage and wing attachment location. The wing fairing covers the lower fuselage and wing wiring and pneumatic plumbing and it is assumed to provide no load carrying ability to the vehicles structural system, but it is design to withstand aerodnamic loading. It is attached by twelve #6-32UNC flat head nylon socket screws as shown below.



Drawing reference: LD-1164925

The wing fairing is approximately 9.84*in* wide by 39.38*in* long.

$$Area = (9.84in)(39.38in) = 387.6in^2$$
$$Therefore,$$
$$F_{lift} = A \cdot C_L \cdot q$$
$$Assume, C_L = 1.3$$
$$then,$$
$$F_{lift} = (387.6in^2)(1.3)(0.17761psi)$$
$$F_{lift} = 89.5lbs$$

And assuming that this load acts in the center of the fairing part then due to the lay-out of the #6 screws the following is assumed:

Four #6 screws are in tensile loading (2 forward & 2 aft) and,
Eight #6 screws are in shear loading (2 forward & 6 aft).

Therefore,

$$F_{\substack{screw\\tensile}} = \frac{89.5lbs}{4screws} = 22.4lbs$$

$$and,$$

$$F_{\substack{screw\\shear}} = \frac{89.5lbs}{8screws} = 11.2lbs$$

Assume Nylon 6 properties, 10*ksi* tensile strength. Assume that the tensile and shear areas of the #6-32 UNC screw is 0.00909in^2 (per Holo-Krome Technical Handbook, P/N 99004 10M HU, © 1990, revised 12/90), then the nylon screws rated tensile strength is,

$$TensileStrength_{\substack{Nylon\\\#6-32\\FHSS}} = (10000psi)\cdot(0.00909in^2) = 90.9lbs$$

Therefore the factor of safety on the tensile load for a #6-32UNC flat head nylon screw is:

$$\boxed{FS_{\substack{\#6-32\\FHSS}} = \frac{90.9lbs}{22.4lbs} = 4.1}$$

And the factor of safety for the shear stress, based on the tensile capability of a #6-32UNC flat head nylon screw is:

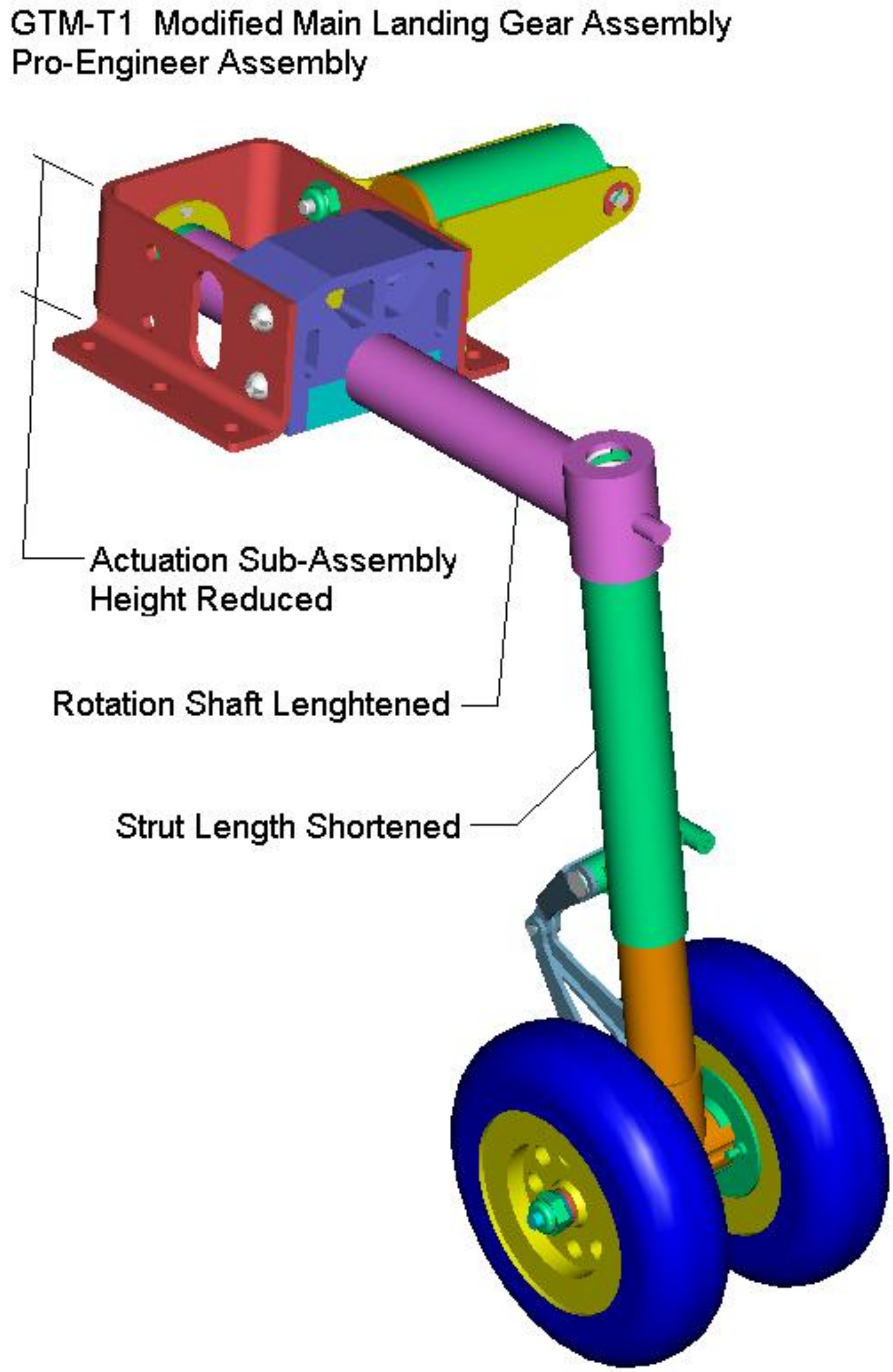
$$\boxed{FS_{\substack{\#6-32\\FHSS}} = \frac{90.9lbs}{11.2lbs} = 8.1}$$

The Landing Gear System

The landing gear system consists of six primary assemblies: one nose gear assembly, two main gear assemblies, one air bottle reservoir with tubing, one deployment valve & servo assembly, and one braking & servo assembly. The landing gear assemblies are storable and deployable within the vehicle via small pneumatic actuator cylinders. The main landing gear assemblies house the drum braking system, which is an expandable o-ring design when operated. All components are off-the-shelf and used as purchased with the exception of the two main gear assemblies. These assemblies required modifications so they could be incorporated into the scaled wing geometry of the model. These modifications include reducing the actuation sub-assembly height, increasing the rotation shaft length, shortening the strut length, and installing a stiffer strut spring.

Main Landing Gear Analysis

The main gear assembly is an off-the-shelf "Robart #630 w/90degree strut" unit with modifications as shown below (T2 main gear assembly design is the same as the T1 design):

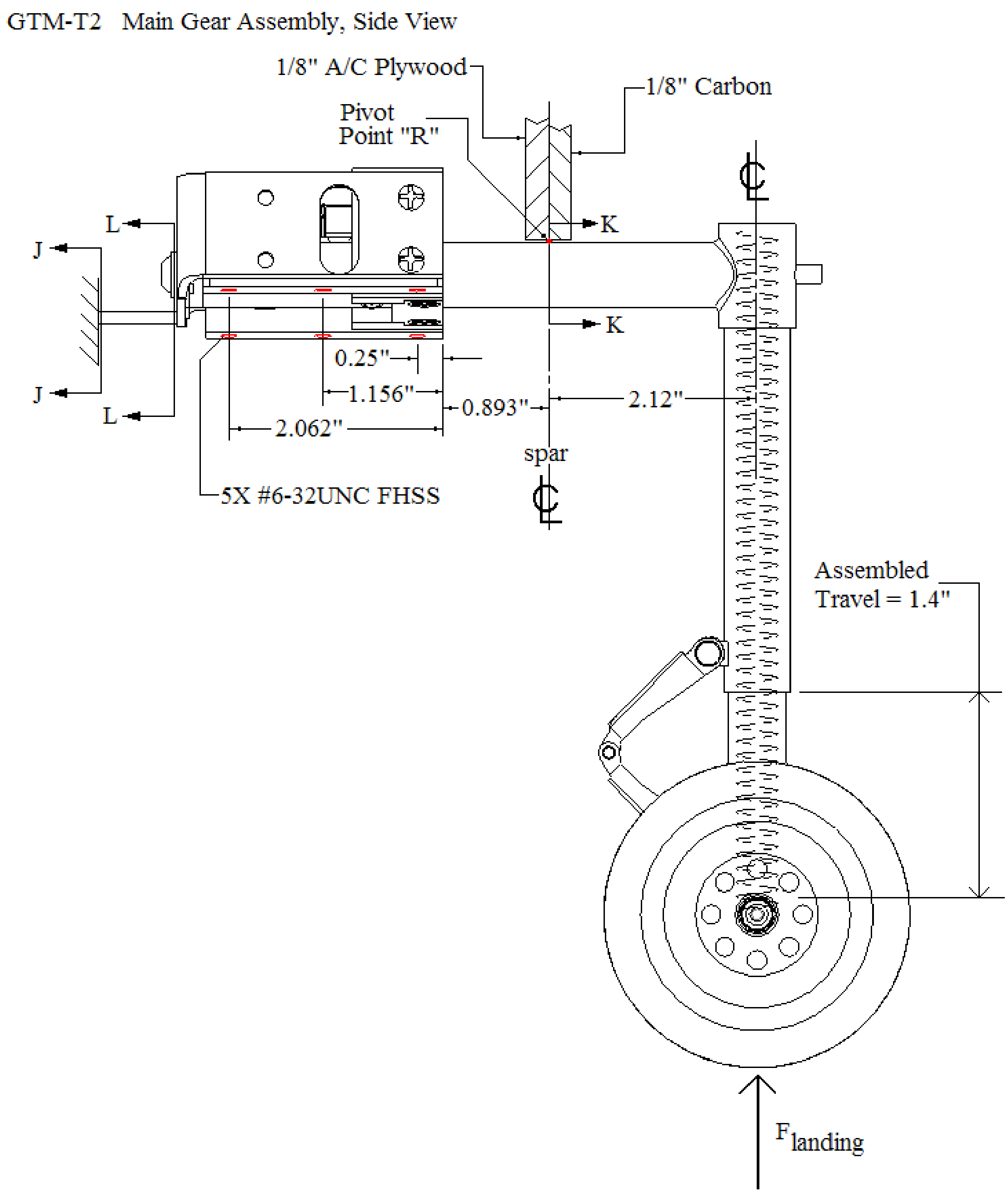


Drawing reference: LD-1164933

The main gear load analysis assumes a 2.5 *g* vertical load on the assembly. It is further assumed that this entire load goes into one main gear assembly.
Maximum aircraft weight at 2.5 *g*:

$$W = 2.5 \times 55 lbs = 137.5 lbs$$

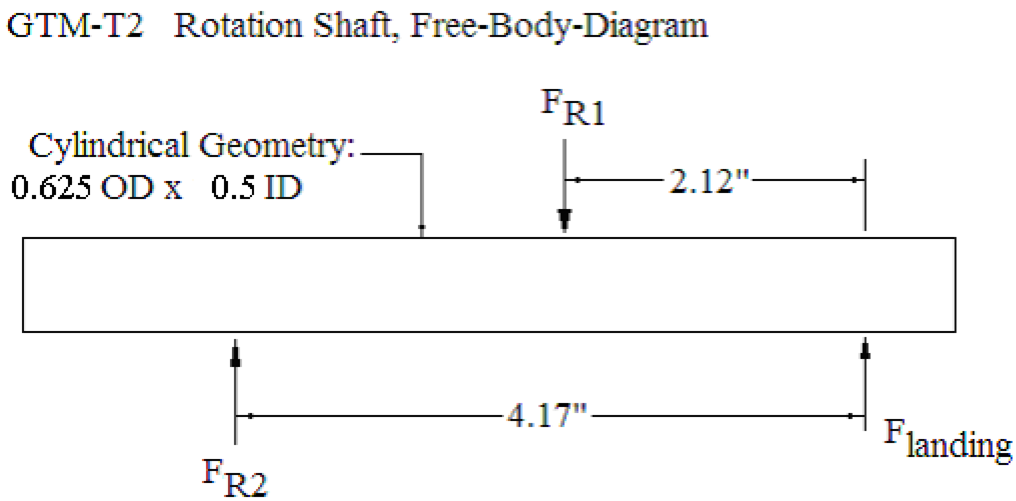
Shown below is a diagram of the mounted main gear assembly. The stresses are calculated for the cross sections shown.



Drawing reference: LD-1164933

Rotation Shaft Bending Stresses at Section K-K, case 1:

As the aircraft lands, the vertical load is transferred from the wheels to the rotation shaft. An aircraft plywood and carbon fiber wing spar acts as a bump stop and a pivot point, which continues the load transfer to the main gear assembly then to the mounting plate. Shown below is a free body diagram of the shaft with equivalent reaction forces and their respective positions.



Drawing reference: LD-1164933

F_{R2} is assumed to react at the center of the actuation housing.
Summing the moments about F_{R1} gives (right-hand-rule applies):

$$+\uparrow \sum M_{F_{R1}} = F_{landing}(2.12in) - F_{R2}(4.17in - 2.12in) = 0$$

$$Assume,\ F_{landing} = (2.5)\cdot W_{MAX} = 137.5\ lbs$$

$$Then,$$

$$(137.5lbs)(2.12in) = F_{R2}(2.05in)$$

$$F_{R2} = 142\ lbs$$

And, summing the forces,

$$+\uparrow \sum F_y = F_{landing} + F_{R2} - F_{R1} = 0$$

$$137.5lbs + 142lbs = F_{R1}$$

$$\therefore F_{R1} = 279.8\ lbs$$

Assuming that this shaft is simply-supported with an intermediate-load, then per Shigley, Mechanical Engineering Design, 3rd Edition, page 642, the moment at F_{R1} is,

$$M_{F_{R1}} = \frac{Fbx}{l} = \frac{(F_{R1})(2.12in)(2.05in)}{4.17in} = \frac{(279.8lbs)(2.12in)(2.05in)}{4.17in}$$

$$M_{F_{R1}} = 291.5\ in-lbs$$

Stress at Section K-K:

Shear Stress:

$$\tau = \frac{F_{R1}}{A_{shear}}$$

$$where\ A_{shear}\ is:$$

$$A_{shear} = \pi \cdot \left(\frac{D_o}{2}\right)^2 - \pi \cdot \left(\frac{d_i}{2}\right)^2$$

$$A_{shear} = \pi\left(\frac{0.625in}{2}\right)^2 - \pi\left(\frac{0.5in}{2}\right)^2 = 0.3068in^2 - 0.1964in^2$$

$$A_{shear} = 0.1104\,in^2$$

$$then,$$

$$\tau = \frac{279.8lbs}{0.1104in^2}$$

$$\tau = 2533.0\ \ psi$$

Bending Stress:

$$\sigma_{k-k} = \frac{M_{F_{R1}}c}{I}$$

Where,

$$I = \frac{\pi}{64}\left(D_o^{\,4} - d_i^{\,4}\right)$$

$$I = \frac{\pi}{64}\left[(0.625)^4 - (0.5)^4\right] = 0.00442\,in^4$$

And,

$$c = \frac{0.625}{2}in = 0.3125in$$

Therefore,

$$\sigma_{k-k} = \frac{M_{F_{R1}} c}{I} = \frac{(291.5in - lbs)(0.3125in)}{0.00442in^4}$$
$$\sigma_{k-k} = 20599.3 \ psi$$

Then the combined stress is,

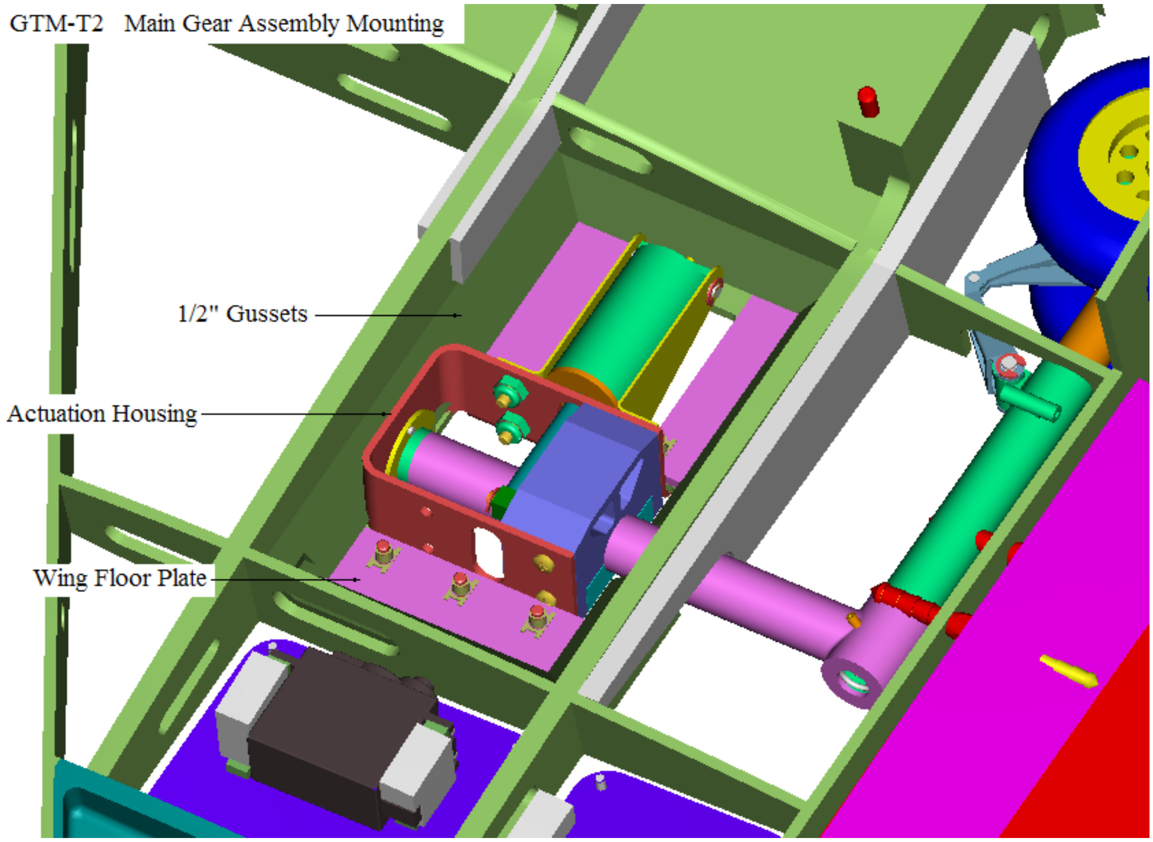
$$\sigma_{combined} = \frac{20599.3psi}{2} + \sqrt{\left(\frac{20599.3psi}{2}\right)^2 + (2533.0psi)^2}$$
$$\sigma_{combined} = 10299.6 + 10606.6$$
$$\sigma_{combined} = 20906.2 \ psi$$

The ultimate strength of the 15-5 H1025 steel is 155ksi, as a result the factor of safety is calculated below as:

$$\boxed{FS_{k-k} = \frac{155\,000psi}{20906.2psi} = 7.4}$$

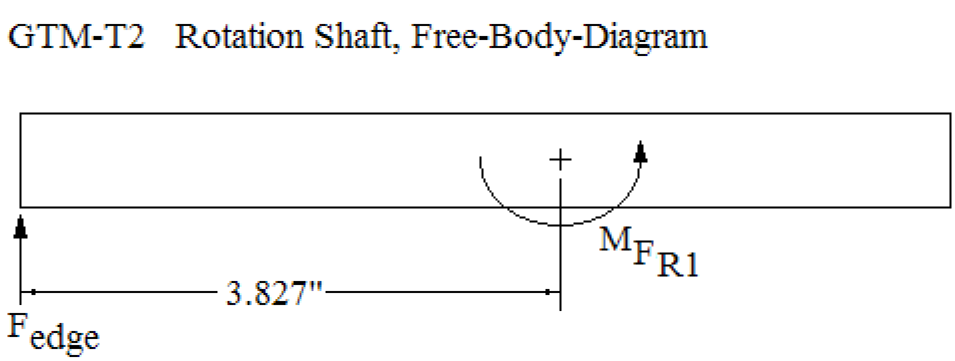
Stress at Section J-J:

The actuation housing is mechanically attached to a wing floor plate as shown below:



Drawing reference: LD-1164906 and LD-1164933

The shear force (F_{edge}) in the gusset shown above is derived from the free-body-diagram below:



Drawing reference: LD-1164933

The shear force is assumed to be F_{edge}, which is, computes as,

$$M_{F_{R1}} = F_{edge}(3.827in)$$
$$291.5in - lbs = F_{edge}(3.827in)$$
$$\therefore F_{edge} = 76.2\ lbs$$

The shear area for cross-section J-J is,

$$A_{shear} = (gusset\ length)(gusset\ height)$$
$$A_{shear} = (6.2in)(0.50in)$$
$$A_{shear} = 3.1\ in^2$$

And, the shear stress is,

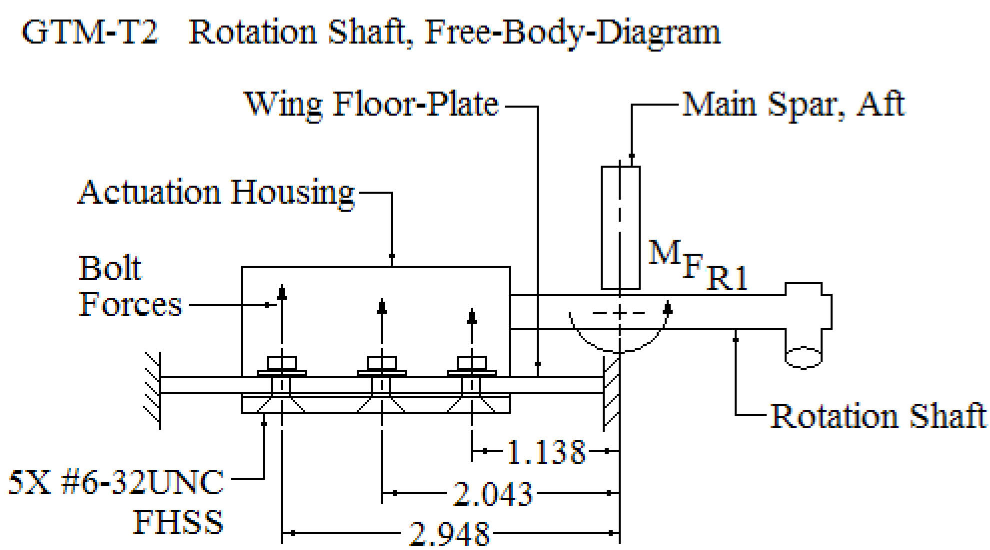
$$\tau_{J-J} = \frac{F_{edge}}{A} = \frac{76.2lbs}{3.1in^2}$$
$$\tau_{J-J} = 24.6\ psi$$

The assumed shear stress for HYSOL 9359.3 is 1000*psi*. Therefore, the factor of safety is,

$$\boxed{FS_{J-J} = \frac{1000\,psi}{24.6\,psi} = 40.7}$$

Bolt Attachment Load:

The main gear actuation housing maximum bolt load is calculated per the free-body-diagram:



Drawing reference: LD-1164906 and LD-1164933

$$F_{\underset{\max}{bolt}} = \frac{M_{F_{R1}}\, r_{\max}}{\sum_{1}^{5} r_i^2}$$

$$\sum_{1}^{5} r_i^2 = (1.138in)^2 + (1.138in)^2 + (2.043in)^2 + (2.948in)^2 + (2.948in)^2 = 24.16\ \ in^2$$

$$r_{\max} = 2.948\ in$$

$$\therefore F_{\underset{\max}{bolt}} = \frac{(291.5in - lbs)(2.948in)}{24.16in^2}$$

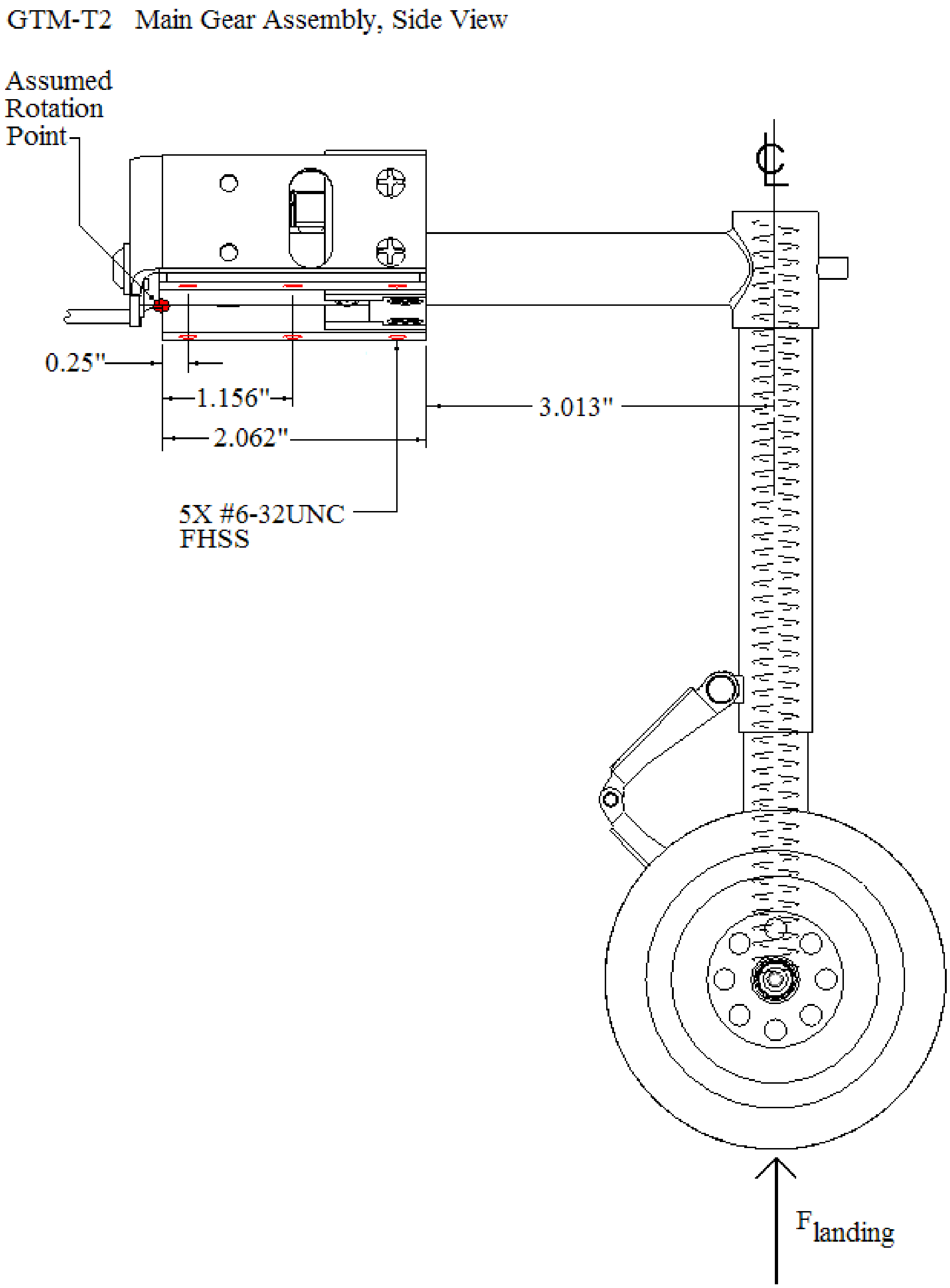
$$F_{\underset{\max}{bolt}} = 35.6\,lbs$$

The factor of safety for the #6-32UNC FHSS based on a tensile strength of 1320*lbs* as per the “Holo-Krome” technical handbook, P/N 99004, revised 12/90, © 1990, is,

$$\boxed{FS_{\underset{FHSS}{\#6-32UNC}} = \frac{1320lbs}{35.6lbs} = 37.1}$$

Bolt Attachment Load:

Each main gear assembly is bolted to the internal wing structure using five #6-32UNC flat head socket screws and lock nut plates. The slight angle at which the mechanism is bolted to the wing is assumed to have a negligible influence on the bolt loads. The maximum bolt tension is calculated. In this analysis it is assumed that the aft main spar does not contact the rotation shaft. The assumed pivot point location is shown below:



Drawing reference: LD-1164933

Assuming that the force of landing is 3*g*'s and the entire model weight load is on one main gear assembly then,

$$F_{landing} = (3"g")(Weight_{\max}) = (3)(55lbs) = 165lbs$$

$$then,$$

$$F_{\substack{Max\_bolt \\ Tension}} = \frac{Mr_{\max}}{\sum_{1}^{5} r_i^2}$$

$$F_{\substack{Max\_bolt \\ Tension}} = \frac{(165lbs)(3.013in + 2.062in)(2.062in)}{2(0.25in)^2 + (1.156in)^2 + 2(2.062in)^2}$$

$$F_{\substack{Max\_bolt \\ Tenion}} = \frac{1726.7}{9.965}$$

$$F_{\substack{Max\_bolt \\ Tension}} = 173.3\ lbs.$$

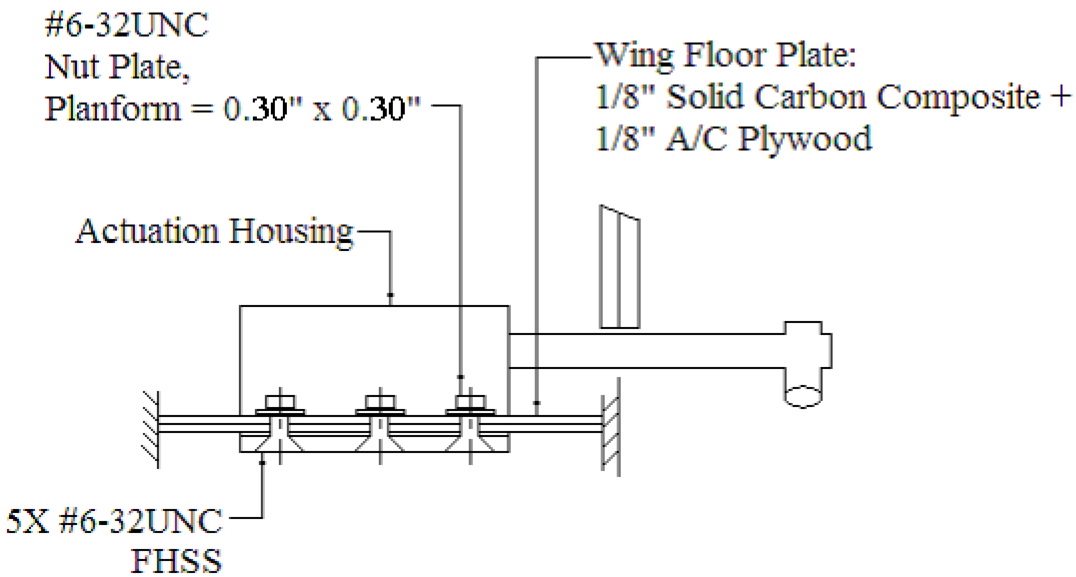
The factor of safety for the #6-32UNC FHSS based on a tensile strength of 1320*lbs* as per the "Holo-Krome" technical handbook, P/N 99004, revised 12/90, © 1990, is,

$$\boxed{FS_{\substack{\#6-32UNC \\ FHSS}} = \frac{1320lbs}{173.3lbs} = 7.62}$$

Nut-Plate Shear Pull-Thru Stress:

Shown below is a diagram of the flange, from the main gear mounting plate, attached to the nut-plates.

GTM-T2 Main Gear Assembly Attachment



Drawing reference: LD-1164906 and LD-1164933

The conservative nut plate perimeter is 0.3*in* x 0.3*in* as measured by E Muller on 11/09/04.

$$A_{shear} = (nut - plate\ \ perimeter)(wing\ \ floor - plate\ thickness)$$
$$A_{shear} = (2(0.3) + 2(0.3))(2(0.125))$$
$$A_{shear} = 0.30\ in^2$$

Therefore, the shear pull-thru stress is,

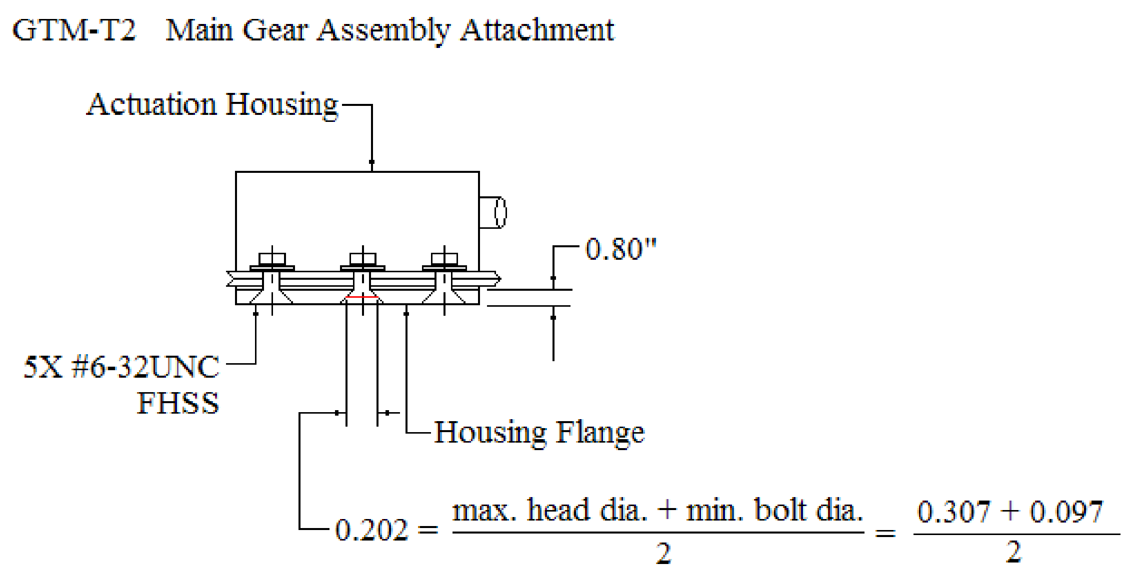
$$\tau = \frac{F_{bolt}}{A_{shear}} = \frac{173.3 lbs}{0.30 in^2}$$
$$\tau = 577.7\,psi$$

The wing floor plate (solid carbon composites & A/C plywood) is assumed to have a minimum ultimate strength of 7578psi (the strength of the A/C plywood) so, the factor of safety based on the plywood's ultimate strength is,

$$\boxed{FS = \frac{7578\,psi}{577.7\,psi} = 13.1}$$

#6 Flat-Head Shear Pull-Through Stress:

The #6 flat head screw locates on the flanged geometry of the actuation housing. This flange is 0.80 *in* thick aluminum. The median diameter for the shear area is shown below:



Drawing reference: LD-1164906 and LD-1164933

The shear stress is,

$$\tau = \frac{F_{bolt}}{A_{shear}} = \frac{F_{bolt}}{\pi D t} = \frac{173.3 lbs}{\pi(0.202 in)(0.8 in)}$$

$$\tau = \frac{173.3 lbs}{0.51 in^2}$$

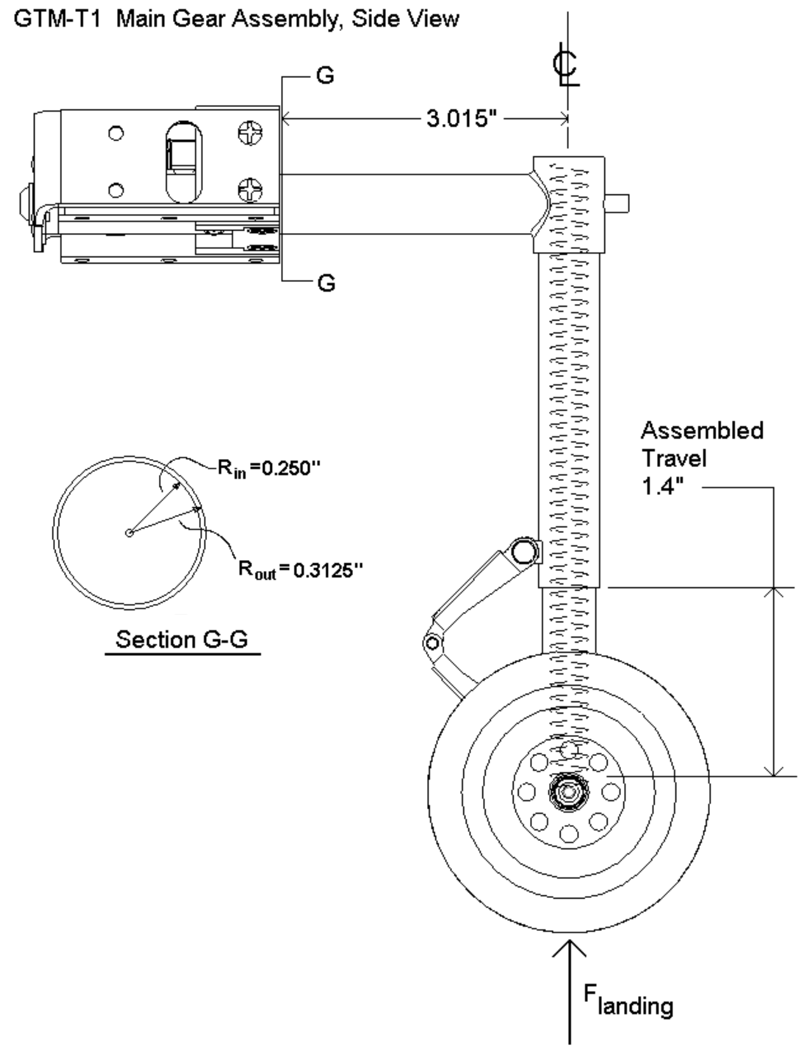
$$\tau = 339.8\ psi$$

The ultimate strength of 6061-T6 aluminum is 27*ksi*, as a result the factor of safety is:

$$\boxed{FS = \frac{27000\,psi}{339.8\,psi} = 79.5}$$

Rotation Shaft Bending Stress, Case 2:

In this analysis it is assumed that the aft main spar does not contact the rotation shaft



Drawing reference: LD-1164933

Assume: $F_{landing} = (3g's)(Weight_{\max}) = (3)(55lbs) = 165\ lbs$

Calculating the shear stress:

$$\tau_{G-G} = \frac{F_{\lg}}{A_{Shear}}$$

Where,

$$A_{Shear} = \pi(r_o^{\ 2} - r_i^{\ 2})$$

$$A_{shear} = \pi\left(0.3125in\right)^2 - \pi\left(0.25in\right)^2 = 0.3067in^2 - 0.1964in^2$$

$$A_{shear} = 0.1104\,in^2$$

$$\tau_{G-G} = \frac{165lbs}{0.1104in^2}$$
$$\tau_{G-G} = 1494.0\,psi$$

And the bending stress is:

$$\sigma_{G-G} = \frac{M_{landing} \cdot c}{I} = \frac{(165lbs)(3.015in)(0.3125in)}{\frac{\pi}{4}\left[(0.3125in)^4 - (0.25in)^4\right]}$$
$$\sigma_{G-G} = \frac{155.5lbs}{0.0044in^2}$$
$$\sigma_{G-G} = 35154.8\,psi$$

And the combined stress is:

$$\sigma_{\substack{Max\\G-G}} = \frac{35154.8\,psi}{2} \pm \sqrt{\left(\frac{35154.8\,psi}{2}\right)^2 + (1494.0\,psi)^2}$$
$$\sigma_{\substack{Max\\G-G}} = 17577.4 + 17640.8$$
$$\sigma_{\substack{Max\\G-G}} = 35218.2\,psi$$

And the factor of safety based on the yield strength of 15-5 ph1025 steel is145*ksi* therefore:

$$\boxed{FS_{G-G} = \frac{145000\,psi}{35218.2\,psi} = 4.12}$$

Main Gear Spring Compression:

The O.E.M. springs for the main landing gear assemblies have an uncompressed length of 10.125*in* and an initial compression of 1.7*in* for assembly into the strut housing. With a maximum designed travel of 1.4*in* in the final main gear assembly, the total load required to “bottom-out” the strut is 20*lbs* (determined from load testing). Thus the spring rate (k) is,

$$F = kx$$
$$20lbs = k(1.7in + 1.4in) = (3.1in)k$$
$$\therefore k = 6.45 \,{}^{lbs}\!/_{in}$$

And, assuming that the model (weight max. = 55*lbs*) lands on one main gear assembly then, this corresponds to a landing load of 0.36 g's (20/55=0.36), which is substantially less than the assumed 2*g* landing that the vehicle could experience. So, a "stiffer" spring is used in the main gear assemblies to mitigate the risk of "hard landings". The spring has the following specifications:

Part No.	O.D.	Wire Dia.	Coils/inch	Length	Constant
9637K29	0.48"	0.080"	6.2	10.50	920

(Reference McMaster-Carr)
Material: Hard-Drawn Steel

The spring was cut to a (uncompressed) length of 10.50*in* long.
Per McMaster-Carr: spring rate is the "Constant" divided by the number of coils:

$$k = \frac{920lbs}{(10.5in)(6.2)}$$
$$k = 14.13lbs/in$$

The compressed spring length is,

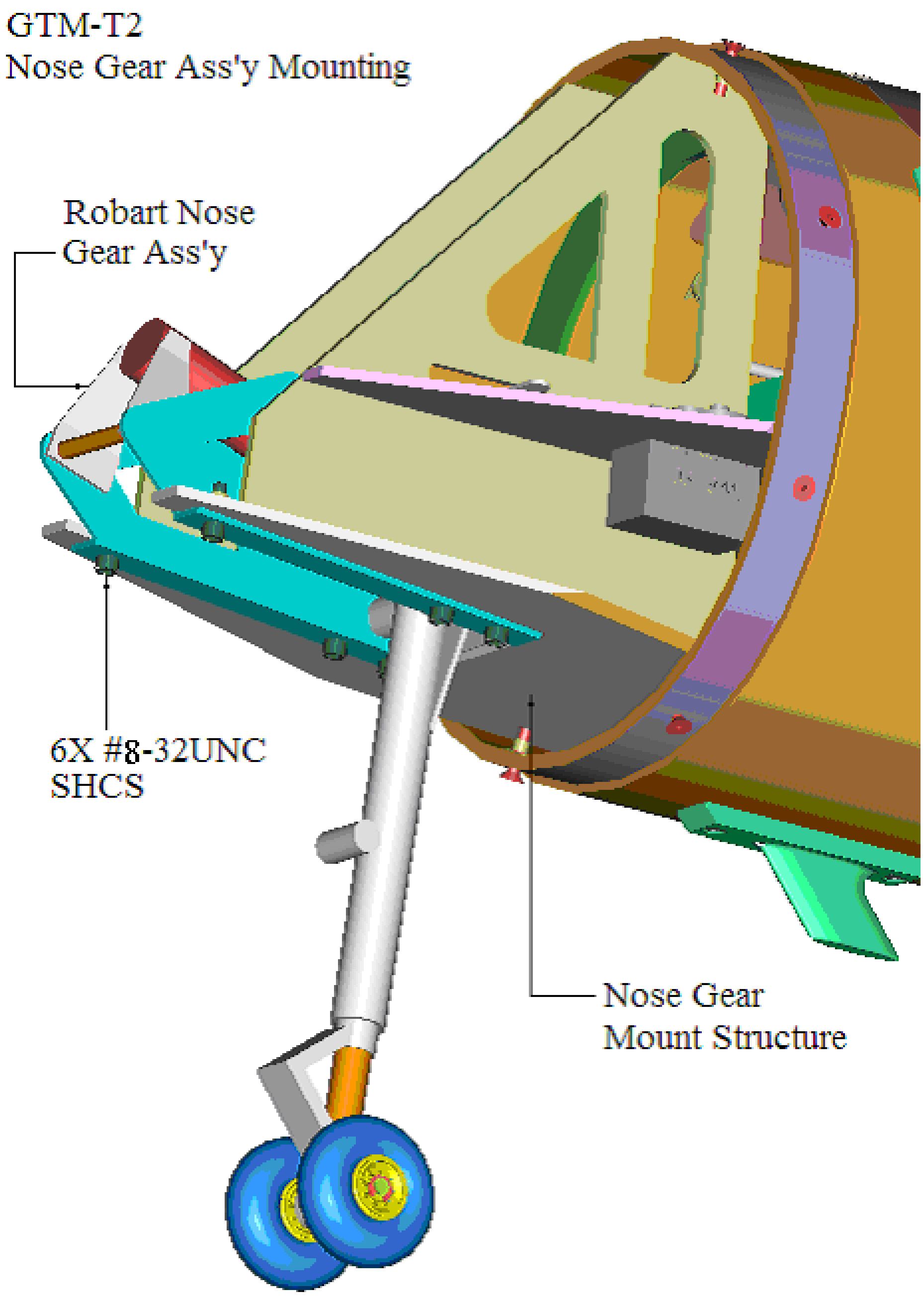
$$l_{deflected} = (6.2)(0.080)(10.50)$$
$$l_{deflected} = 5.208in$$
$$Then,$$
$$l_{compressed} = 10.50 - 5.208 = 5.292in$$
$$And,$$
$$F = kl_{deflected} = (14.13 \, {}^{lbs}\!/_{in})(5.208in)$$
$$F = 73.6lbs$$

Computing the "*g*" loading available gives:

$$g_{\max\_loading} = \frac{73.6lbs}{55lbs}$$
$$g_{\max\_loading} = 1.34$$

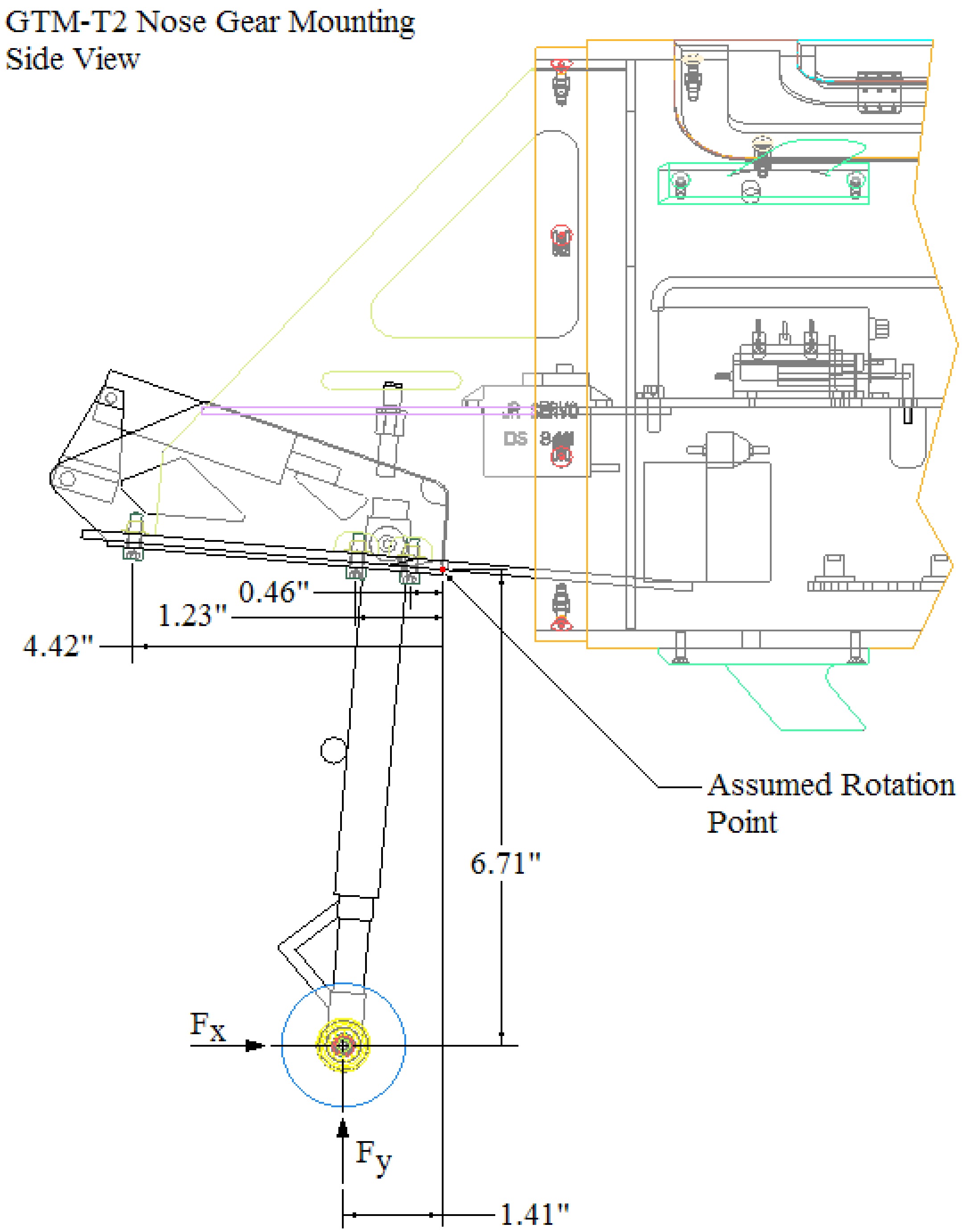
Nose Landing Gear

The nose gear assembly (Robart #158S) mounts to the fuselage structure as shown below:



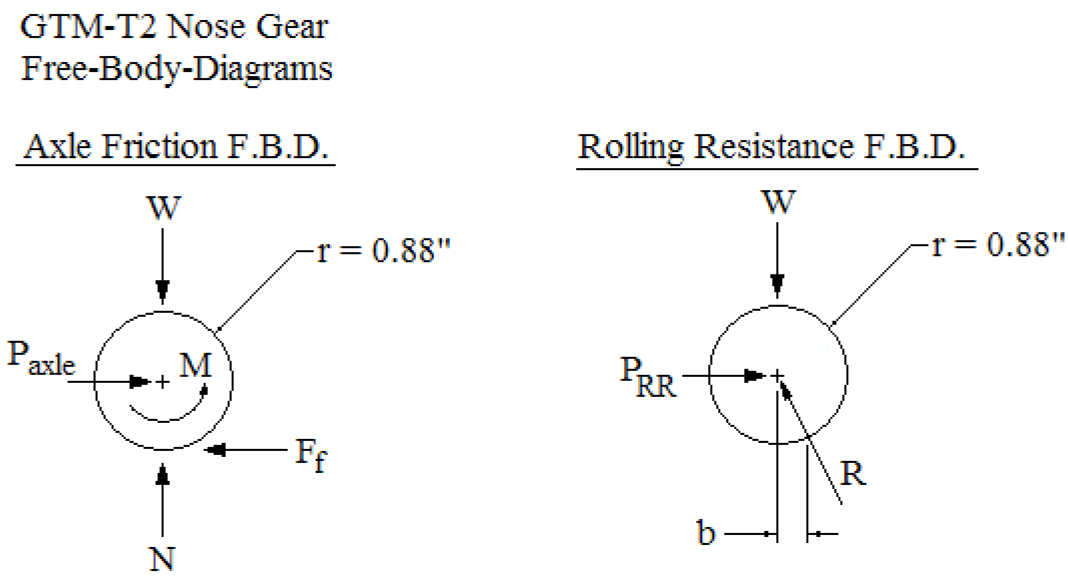
Drawing reference: LD-1164932

The nose gear assembly mounts to the bottom of the sub-structure. The assumed rotation point is shown below for bolt tension and structure bending loads upon landing:



Drawing reference: LD-1164932

Assume that the nose wheel assembly rotates about the location indicated in the above sketch. Also, for this analysis (i.e. bolt tension) assume that the vertical load does not impart tension into the six attachment bolts. And, assume that the horizontal force (F_X) is derived from: a 45 *mph* stall/landing speed, nose wheel axle friction, and nose wheel tire rolling friction. See the sketches below (reference Statics & Dynamics, Beer & Johnston, 3rd Edition, page 334):



Drawing reference: LD-1164932

From the above sketches:

$W = weight$, maximum

$N = normal \ \ force$

$F_f \ = force \ of \ \ friction = \mu N$

$M \ = wheel \ \ moment$

$r = wheel \ \ radius$

$R = folling \ \ force$

$b = coefficient \ of \ \ rolling \ \ friction \ (inches)$

$P_{RR} \ = horizontal \ force \ due \ to \ rolling$ resistance

$P_{axle} = horizontal \ force \ on \ the \ axle$

Assume that the coefficient of friction (reference Mechanical Design, Shigley, 3rd Edition, Table 14-1, molded phenolic plastic on steel, page 547) is:

$\mu = 0.25$

Then,

$$F_f \ = \mu N$$

$$F_f \ = (0.25)(4.5lbs)$$

$$F_f \ = 1.125lb$$

And assuming the force of friction equals the P_{axle} then,

$$F_f = P_{axle} = 1.125\ lbs$$

Also, assume that the coefficient of rolling friction is (reference Statics & Dynamics, Beer & Johnston, 3rd Edition, "Values of the coefficient of rolling friction varies from about 0.01 in. or 0.25mm for a steel wheel on a steel rail to 5.0 in. or 125mm for the same wheel on soft ground.", page 334):

$$b = 0.125\ inches$$

And the "rolling force" is given as (above reference, equation (8.10), page 334):

$$P_{RR}\, r = W\, b$$

Then,

$$P_{RR}\, r = W\, b$$
$$P_{RR} = \frac{(4.5lbs)(0.125in)}{0.88in}$$
$$P_{RR} = 0.639lbs$$

Finally, the last horizontal force to be accounted for is the force due to the models speed of 45*mph*. The impulse/momentum equation is used:

$$F_s \Delta t = m\Delta v$$
$$where,$$
$$F_s = \text{ force due to the model's speed}$$
$$m = \text{mass of the model}$$
$$\Delta v = \text{the change in the model's speed}$$
$$\Delta t = \text{the time for "}F\text{" to equal zero}$$

In the above equation the change in velocity is assumed to be 45 *mph* and the change in time is assumed to be 2.0 *sec*. So,

$$F_s\,\Delta t = m\,\Delta v$$

$$F_s = \frac{(55\,lb)\left(\dfrac{1\ lbf - \sec^2}{32.3\,lb - ft}\right)\left(\dfrac{45\,miles}{hour}\right)\left(\dfrac{1\ hour}{3600\ \sec}\right)\left(\dfrac{5280\ \ ft}{1\ mile}\right)}{(2\ \sec)}$$

$$F_s = \frac{(1.708\dfrac{lbf - \sec^2}{ft})(66\dfrac{ft}{\sec})}{2\ \sec}$$

$$F_s = 56.19\,lbs$$

<u>Summation of the horizontal forces:</u>

$$F_{horizontal} = P_{axle} + P_{RR} + F_s$$

$$F_{horizontal} = 1.125\,lbs + 0.639\,lbs + 56.19\,lbs$$

$$F_{horizontal} = 58.0\ lbs$$

And, the maximum bolt tension is,

$$F_{\substack{Max\_bolt\\Tension}} = \frac{Mr_{\max}}{\sum_1^6 r_i^2} = \frac{(F_{horizontal})(l)(r_{\max})}{\sum_1^6 r_i^2}$$

$$F_{\substack{Max\_bolt\\Tension}} = \frac{(58.0\,lbs)(6.71\,in)(4.42\,in)}{2(0.46\,in)^2 + 2(1.23\,in)^2 + 2(4.42\,in)^2}$$

$$F_{\substack{Max\_bolt\\Tenion}} = \frac{1718.9}{42.52}$$

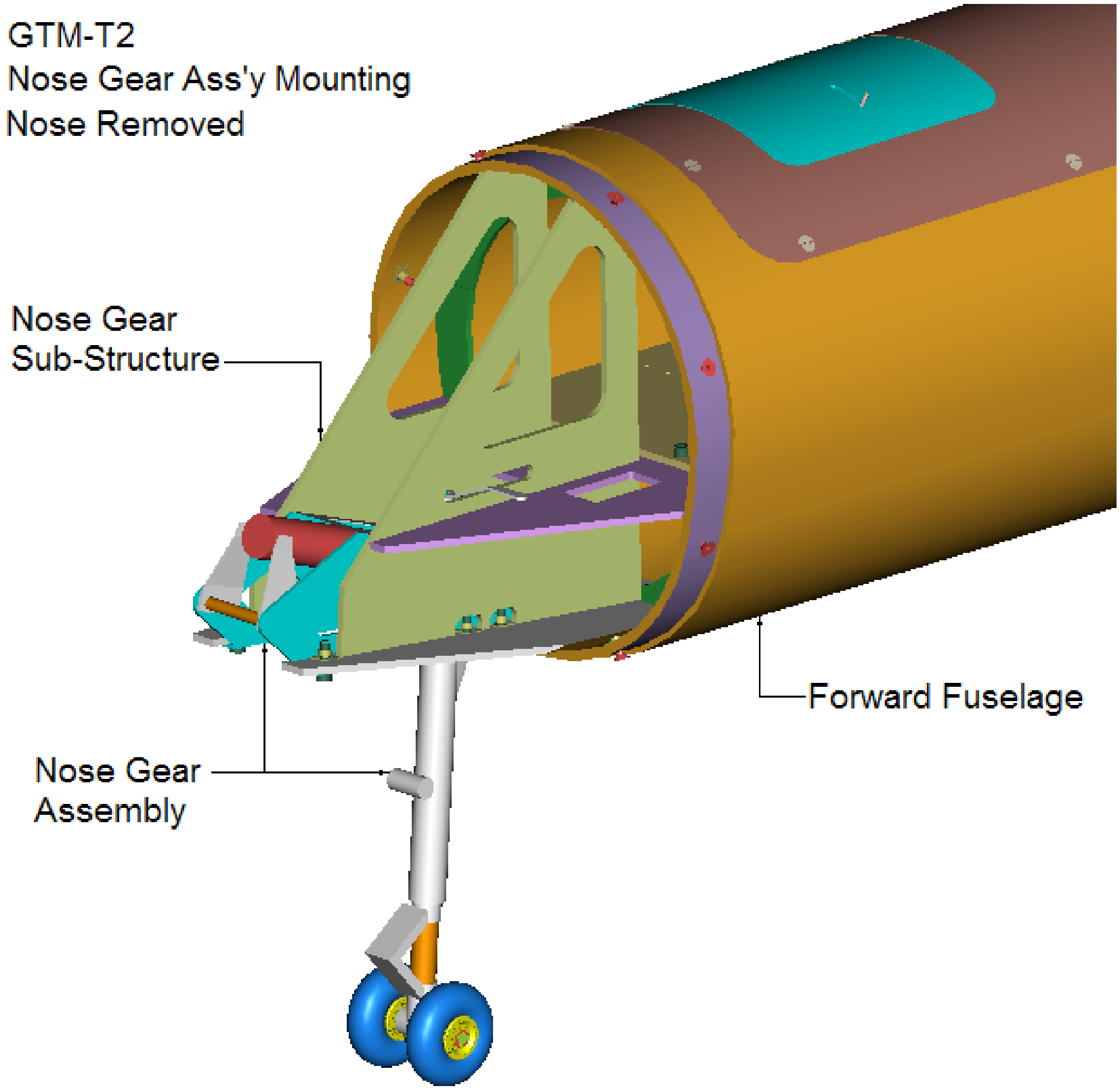
$$F_{\substack{Max\_bolt\\Tension}} = 40.4\,lbs.$$

And, so the factor of safety based on a 2520*lbs* tensile strength of a #8-32UNC socket head cap screw is,

$$\boxed{FS_{\#8-32SHCS} = \frac{2520\,lbs}{40.4\,lbs} = 62.3}$$

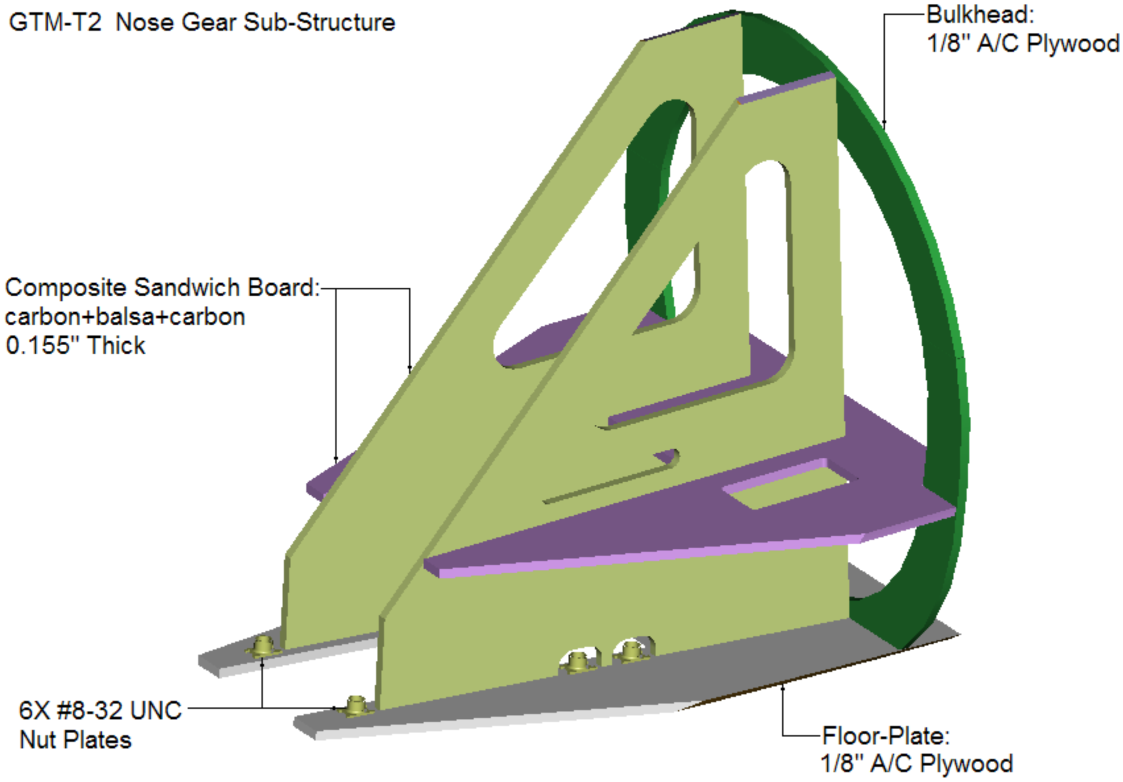
Nose Gear Assembly Mounting Analysis:

The nose gear assembly is an unmodified "Robart, Inc." assembly, part number 158S. It attaches to a composite sub-structure on the forward part of the fuselage as shown below.



Drawing reference: LD-1164901 and LD-1164932

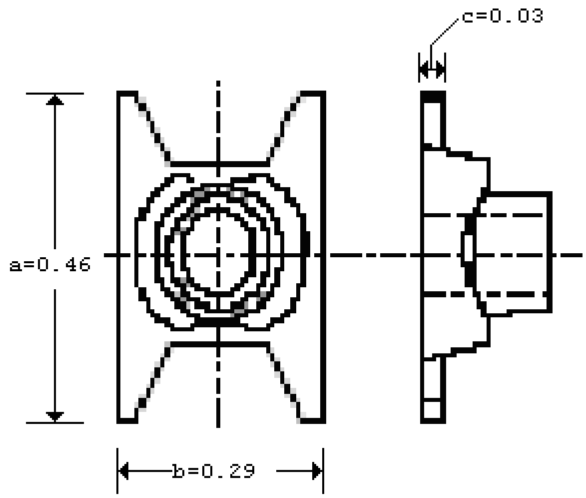
The nose gear assembly attaches to 1/8*in* aircraft plywood with six #8-32 UNC socket head cap screws. This sub-structure assembly on the forward part of the fuselage is an assembly of aircraft plywood and composite sandwich board. The composite sandwich board is made from 2 layers of 0.005*in* carbon cloth, 0.125*in* balsa, and 2 layers of 0.005*in* carbon cloth. The carbon cloth has a 0/90 and +45/-45 degree orientation on each side of the sandwich board. The sketch below gives the general layout and orientation:



Drawing reference: LD-1164901

Nut Plate Shear Pull-Through:
The landing gear is attached to the nose gear assembly with six #8-32 SHCS, which thread into nut plates. Originally the steel plate was "floating" as shown in the picture below, but it was modified by removing floating portion.

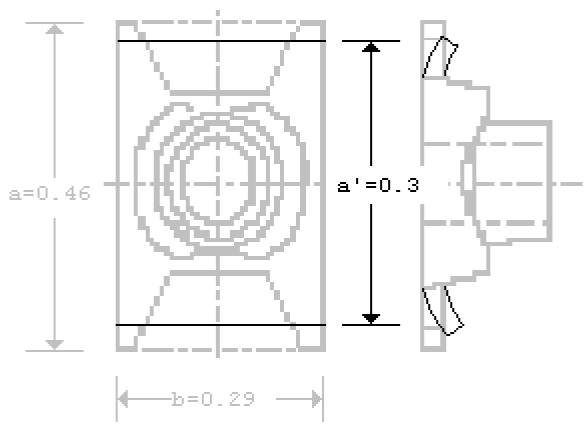
Floating Nut Plate Assembly
Drawing reference: LD-1164901



Modified Nut Plate (housing removed)
Drawing reference: LD-1164901

The sketch above shows the modified plate, with measurements, that was installed on the nose gear plate.

As the plate is drawn through the plywood, the tabs are assumed to bend as shown in the sketch below. It is assumed that this bending would decrease the length of the plate by 0.16*in.*



Drawing reference: LD-1164901

Therefore,

The perimeter is:

$$P = (2a' + 2b)$$
$$P = (2)(0.30in) + (2)(0.29in)$$
$$P = 1.18in$$

The shear area is:

$$A_s = P(t)$$
$$Where: t\ (plywood\ \ thickness) = 0.125in$$
$$A_s = (1.18in)(0.125in)$$
$$A_s = 0.148in^2$$

So the shear pull through stress is:

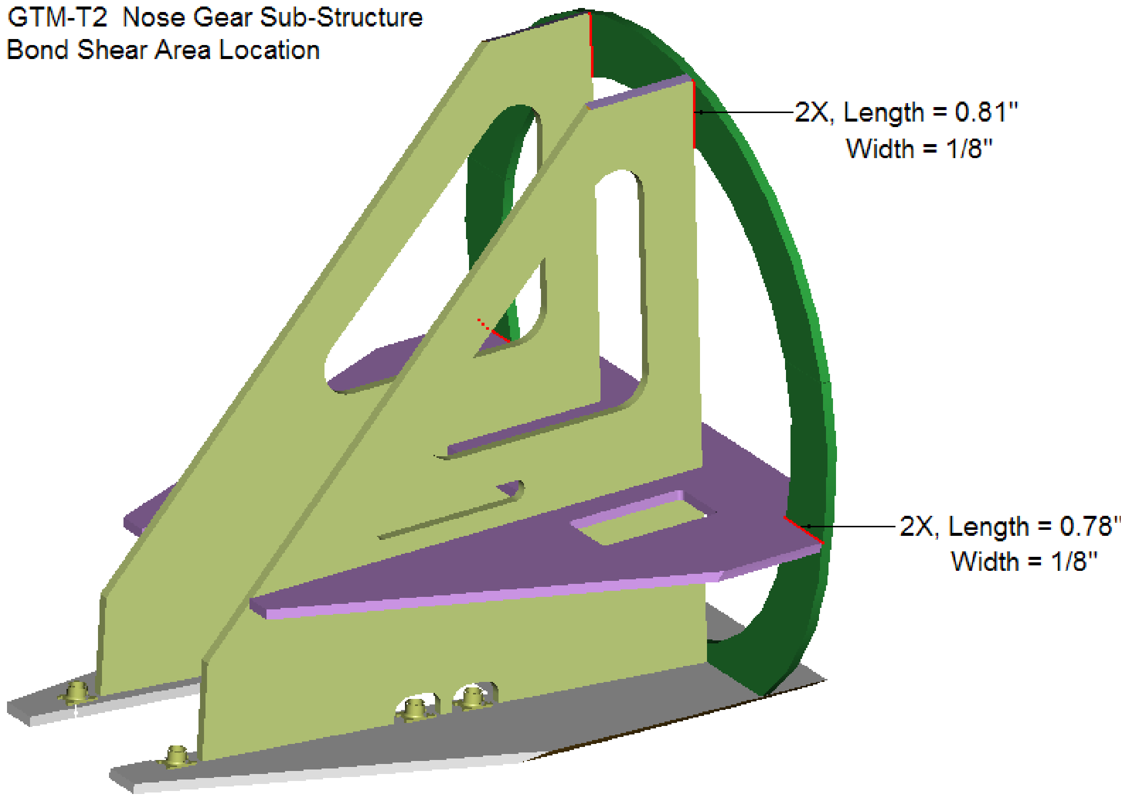
$$\therefore \tau = \frac{F_{Bolt\_Load}}{A_s}$$
$$\tau = \frac{56.19lbs}{0.148in^2}$$
$$\tau = 380.9\,psi$$

The 0.125*in* A/C plywood has a tensile strength of 7578*psi* (reference project document number GTMP-6031), so the factor of safety is:

$$\boxed{FS = \frac{7578\,psi}{380.9\,psi} = 19.9}$$

Mounting Plate Shear Stress:

The mounting plate is 0.125*in* plywood and bonded to the inside rib of the forward fuselage bay. The bond areas are indicated in the sketch below:



Drawing reference: LD-1164901

$$Area_{shear} = (2\,places)(0.81in)(0.125in) + (2\,places)(0.78in)(0.125in)$$
$$Area_{shear} = 0.398in^2$$

Assuming that the entire model weight and 3 "*g*" force then,

$$F_{shear} = (3g)(Weight_{\max}) = (2)(55lbs) = 165lbs$$

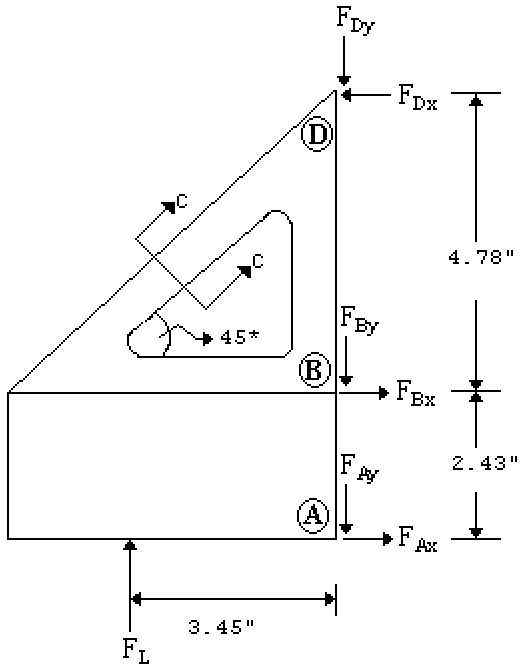
$$\therefore \tau = \frac{F_{shear}}{Area_{shear}} = \frac{165lbs}{0.398in^2}$$
$$\tau = 415.1psi$$

And the factor of safety based on the "Hysol 9359.3" structural adhesive's lap shear strength of 2167*psi* (Reference 14.2% BWB LSV Material Characterization Study, June 5, 2002) is:

$$\boxed{FS_{\substack{adhesive\\shear}} = \frac{2167\,psi}{415.1psi} = 5.22}$$

Mounting Plate Assembly-Column Buckling:

The mounting plates are analyzed for bending and compression stresses under assumed maximum loads. The Free Body Diagram is given below.



Drawing reference: LD-1164901

The landing force is found from the impulse equals momentum equation.

$$F \cdot \Delta t = m \cdot \Delta v$$

Where it is assumed that:

$$\Delta t = 2\sec$$

and,

m = model weight at 1"*g*"

or

$$m = (\frac{55lbs}{1})(\frac{1s^2}{32.2\,ft}) = 1.708\frac{lbs \cdot s^2}{ft}$$

and it is assumed that,

$$\Delta v = landing\ \ speed$$
$$\Delta v = 45mph = 66\ ft/\sec$$

Therefore:

$$F \cdot \Delta t = m \cdot \Delta v$$

$$F = \frac{\left(1.708\frac{lbs \cdot s^2}{ft}\right) \cdot \left(66\frac{ft}{s}\right)}{2s}$$

$$F = 56.37 lbs$$

From the FBD:

Sum of the horizontal force is,

$$+\rightarrow \sum F_x = 0$$

$$F_{Ax} + F_{Bx} - F_{Dx} = 0$$

Sum of the vertical force is,

$$+\uparrow \sum F_y = 0$$

$$F_L - F_{Ay} - F_{By} - F_{Dy} = 0$$

$$F_{Ay} + F_{By} + F_{Dy} = 56.4$$

Sum of the moment about point A is,

$$+ccw \sum M_a = 0$$

$$-F_L(3.45) - F_{Bx}(2.43) + F_{Dx}(4.78 + 2.43) = 0$$

$$-F_{Bx}(2.43) + F_{Dx}(7.21) = 56.4 \cdot (3.45)$$

$$-F_{Bx}(2.43) + F_{Dx}(7.21) = 194.58$$

By entering the coefficients from the three resultant equations into a matrix gives:

	F_A	F_B	F_D	*Sum*
$\sum F_x$	1	1	−1	0
$\sum F_y$	1	1	1	56.4
$\sum M_a$	0	−2.43	7.21	194.58

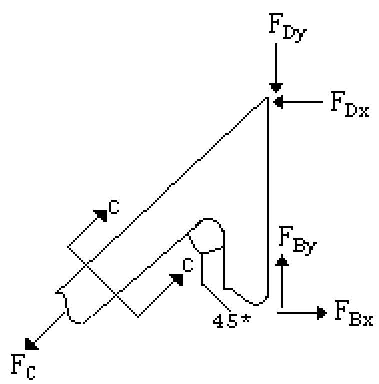
Using "reduced row echelon form" (a.k.a. RREF) simultaneously solves all three equations giving:

F_A	F_B	F_D	Sum
1	0	0	24.6
0	1	0	3.6
0	0	1	28.2

So that:

$$F_A = 24.6 lbs$$
$$F_B = 3.6 lbs$$
$$F_D = 28.2 lbs$$

The compression at 45^o leg of the structure is found by finding the forces at F_C by taking the corner inclusive of F_D and F_B:



Drawing reference: LD-1164901

Sum of the horizontal force is,

$$+\rightarrow \sum F_x = 0$$
$$-F_{Dx} + F_{Bx} - F_{Cx}(\cos 45) = 0$$
$$-(28.2lbs) + (3.6lbs) - F_C(\cos 45) = 0$$
$$F_C(\cos 45) = -24.6$$
$$F_C = -34.8lbs$$

And the critical buckling of this element is:

$$P_{critical} = \frac{n\pi^2 EI}{l^2}$$

Where:

n = 1 (end condition constant, ref. Mechanical Engineering Design, Shigley, 3rd Edition, © 1977, page 116, table 3-1 for a fixed/fixed conservative value)

E (elastic modulus of carbon) = $6x10^6 psi$

$I_{x\text{-}x}$ (c-c connection from above) = $1/12(b \text{ x } h^3)$

b (skin thickness) = (4*plys*)(0.015*in/ply*) = 0.06*in*

h (section c-c width) = 0.644*in*

l (unsupported length) = 3.146*in*

$$P_{critical} = \frac{n\pi^2 EI}{l^2}$$

$$I_{x-x} = \frac{1}{12}(b \cdot h^3) = \frac{(0.06 \cdot 0.644^3)}{12} = 0.0013in^4$$

$$Therefore:$$

$$P_{critical} = \frac{(1)\pi^2(6x10^6)(1.3x10^{-3})}{(3.146)^2}$$

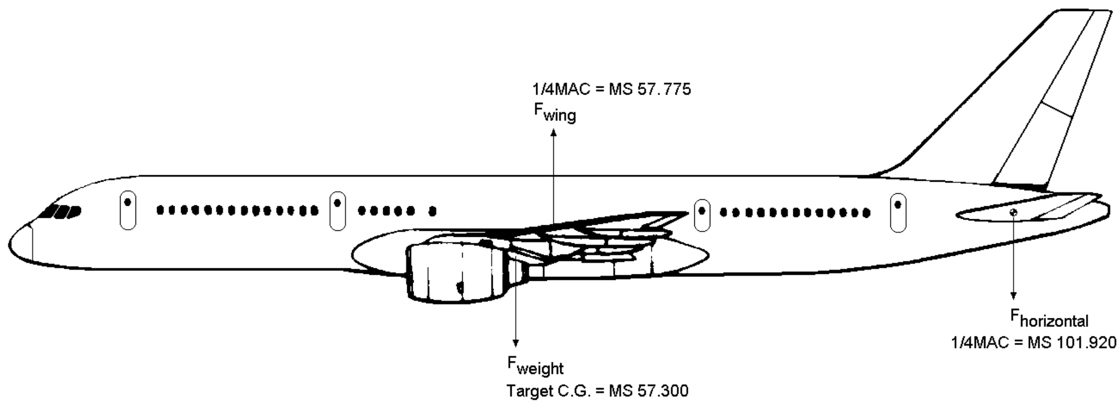
$$P_{critical} = 7778.2lbs$$

The factor of safety based on $P_{critical}$ is:

$$\boxed{FS = \frac{P_{critical}}{F_C} = \frac{7778.2lbs}{34.8lbs} = 223.5}$$

GTM-T2 Horizontal Tail Stress due to a High Speed Dive-Pull-Out:

The horizontal tail spud is analyzed for bending stresses due to a load generated from a high speed dive pull-out. The load generated on the tail spud is assumed to be generated from the moment due to the separation distance of the model's center of gravity and the aerodynamic lift location, as shown below:



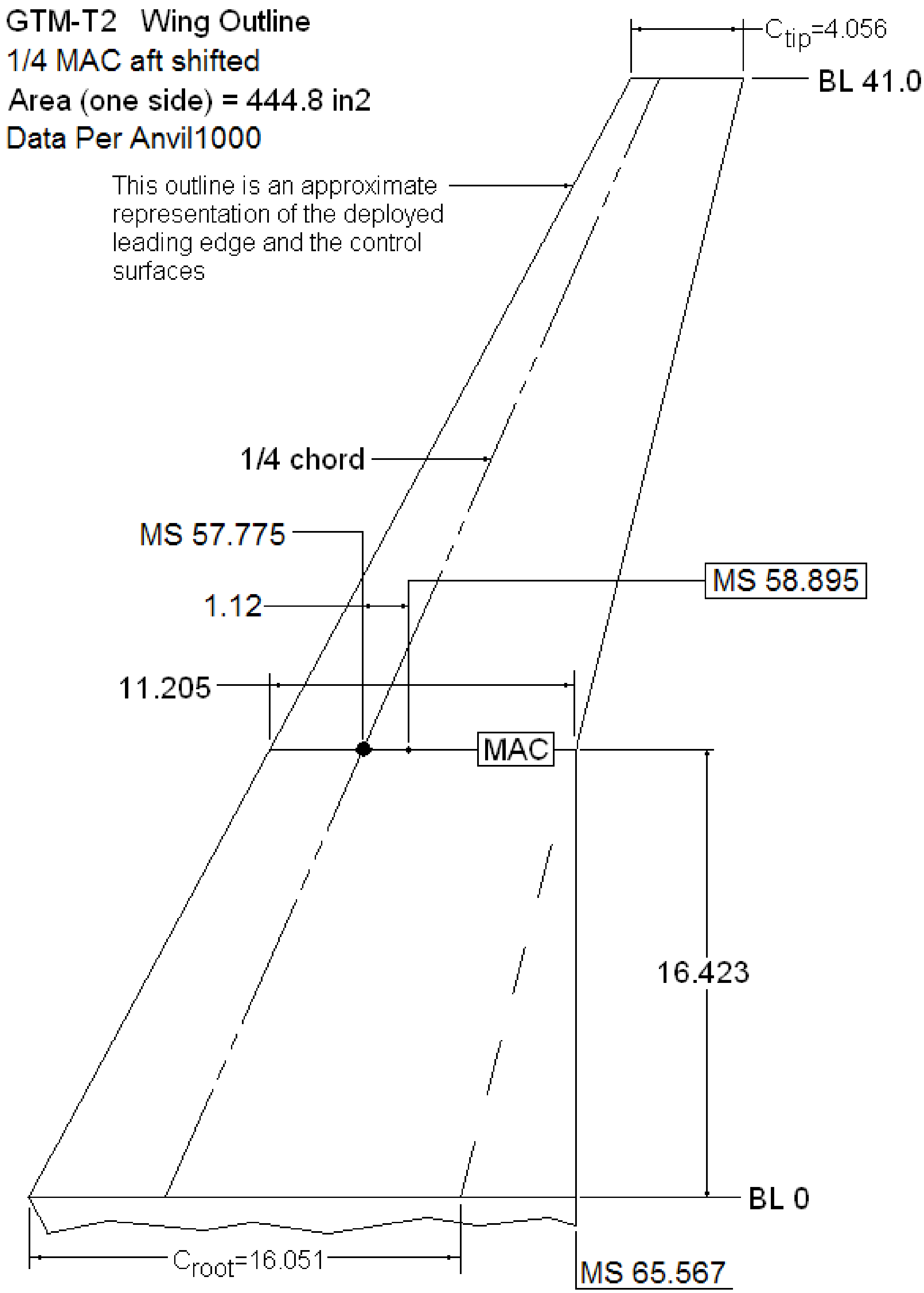
Drawing Ref. LD-1164928

Since the location of the model's weight and the wing's aerodynamic center are practically on the same model station, the following assumptions are used to provide a conservative analysis for the load calculation of the horizontal tail. It should be noted that the horizontal tail airfoil is upside down when compared to the model's main wing.

<u>Assumptions:</u>

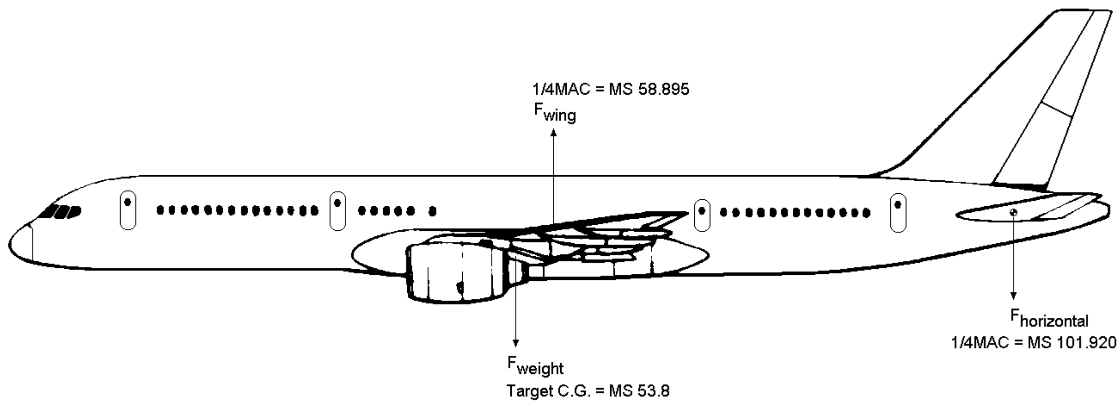
1. The highest horizontal tail loads are at the bottom of the dive-pull-out as shown in the sketch above.
2. The lift force on the horizontal tail remains at its 1/4MAC location.
3. The model weight is 55 *lbs*. therefore the fuel bladders are completely full.
4. The model's center of gravity is shifted forward 3.5 *in* due to fuel slosh from the dive maneuver.
5. This is a conservative analysis since assumptions 3 and 4 are in direct conflict.
6. The main wing's aerodynamic center has shifted aft by 10% of the 1/4MAC chord.
7. The wing assembly's weight is neglected.
8. The model experiences a 4 "*g*" loading due to this dive-pull-out maneuver.
9. The model reaches a dive speed of approximately 100*mph*.
10. The wings flaps and spoilers are in their stowed position.
11. The landing gears are in their stowed position.

The 10% aft shift in the wings aerodynamic center is shown below on a sketch:



Drawing Ref. LD-1164907

Summing the moments about the wings "new" aerodynamic center gives the sketch below:



Drawing Ref. LD-1164928

(Right hand rule applies)

$$+\uparrow \sum M_{wing\ aero\ center} = +F_{weight}(4"g")(58.895-53.8) - F_{horizontal}(101.92-58.895) = 0$$

$$therefore,$$

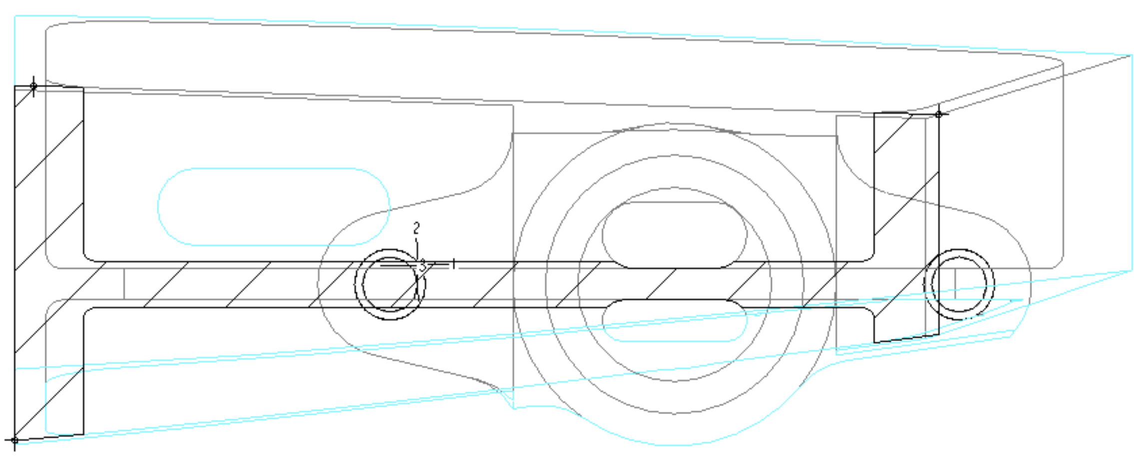
$$(55lbs)(4)(5.1) - F_{horizontal}(43.03) = 0$$

$$and,$$

$$F_{horizontal} = 26.1\ lbs.$$

And this can be used to calculate the horizontal spud bending stress. The bending stress for the horizontal spud is calculated from the cross-section with the lowest factor of safety shown below (reference F-F of the horizontal spud analysis):

GTM-T2 Horizontal Section Cut F-F



Drawing reference LD-1164909-1

The following data was calculated using Pro-Engineer Software

Moment (section F-F):

$$M = F_{Horizontal} d$$
$$M = (26.1lbf)(5.421in)$$
$$M = 141.2in \cdot lb$$

Bending Stress (section F-F):
Checking c and solving for σ:

$$z = \frac{I}{c} \Rightarrow c = \frac{I}{z}$$
$$c = \frac{0.005in^4}{0.0133in^3} = 0.366in$$
$$\sigma = \frac{Mc}{I}$$
$$\sigma = \frac{(141.2in \cdot lb)(0.366in)}{0.005in^4}$$
$$\sigma = 10578.6\,psi$$

Shear Stress (section F-F):

$$\tau = \frac{F}{A}$$
$$\tau = \frac{26.1lbf}{0.2751in^2}$$
$$\tau = 94.7\,psi$$

Principal Stress (section F-F):

$$\sigma_{1,2} = \frac{\sigma_x}{2} \pm \sqrt{\left(\frac{\sigma_x}{2}\right)^2 + \tau^2}$$
$$\sigma_{1,2} = \frac{10578.6\,psi}{2} \pm \sqrt{\left(\frac{10578.6\,psi}{2}\right)^2 + (94.7\,psi)^2}$$
$$\sigma_{1,2} = 5289.3 + 5290.1$$
$$\sigma_{1,2} = 10579.4\,psi$$

And, due to a “sloping-transitional-geometry” the stress concentration factor is assumed to be equal to 1.5

Then,

$$\sigma_{\max} = (10579.4)(1.5)$$

$$\sigma_{\max} = 15869.2\,psi$$

Factor of Safety (section F-F):

The tensile yield strength for aluminum 7075-T651 is 67*ksi*. Thus the factor of safety is:

$$\boxed{FS = \frac{67000\,psi}{15869.2\,psi} = 4.22}$$

And, at this moment in time, at the bottom of the dive-pull-out maneuver, the wings experience an upward force. This force is found by summation of the forces from the sketch above.

$$+\uparrow \sum Forces = +F_{wing} - F_{weight} - F_{horizontal} = 0$$

$$F_{wing} - (55lbs)(4) - 26.1lbs = 0$$

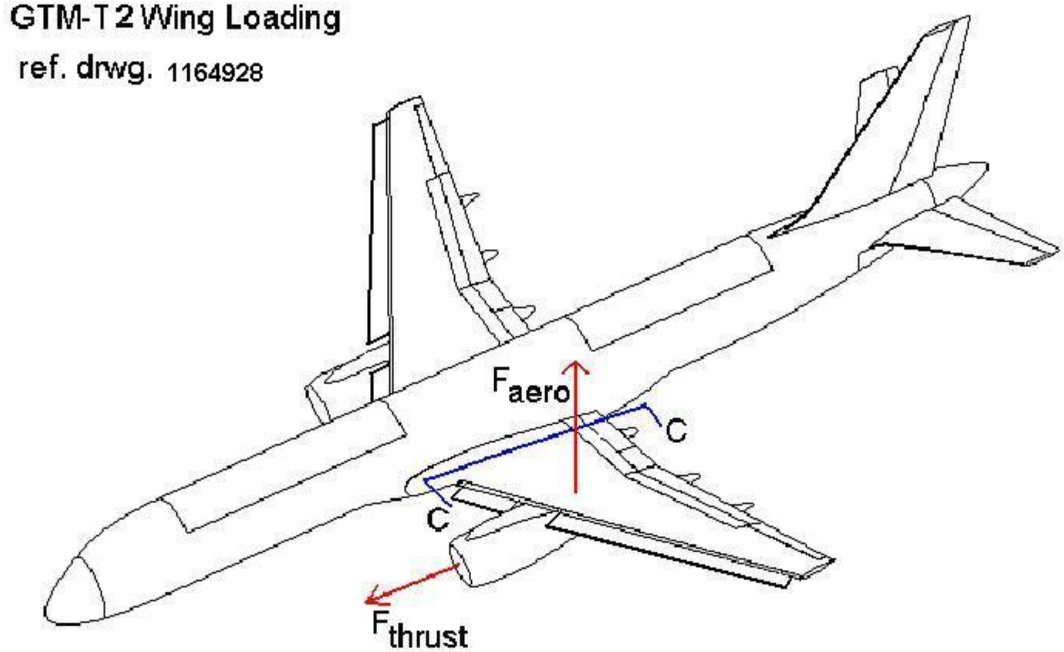
$$then,$$

$$F_{wing} = 246.1\ lbs.\ (total)$$

$$or,$$

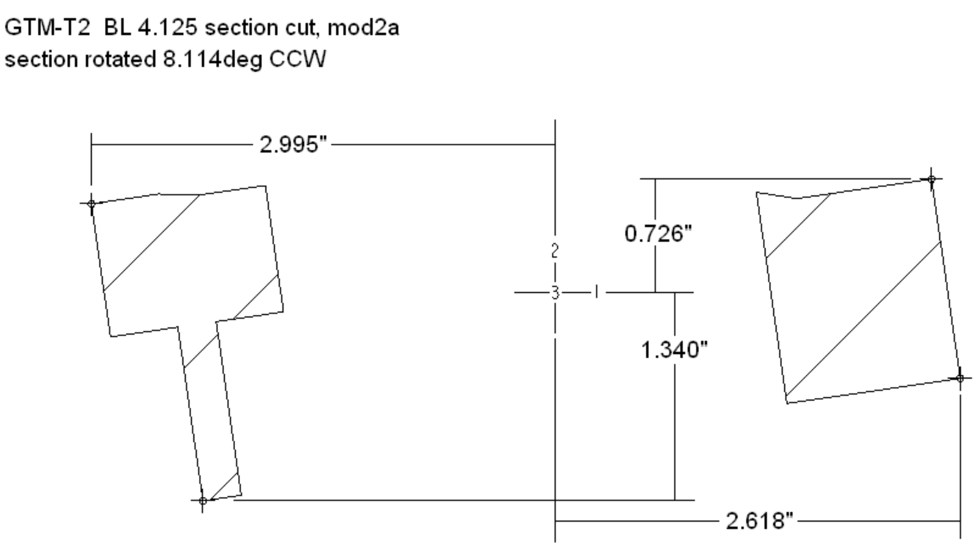
$$F_{wing,each} = 123.1\ lbs.\ (per\ \ wing)$$

And, this force can be used to calculate the wing root bending stress. The maximum stress is calculated from the shear stress and the bending stresses generated by this aerodynamic load. The load due to engine thrust is neglected. Homogeneous material equations have been used in calculating a safety factor for the aircraft grade plywood in bending. The weight of the wing and the engine are neglected in this analysis. The wing is analyzed at section C-C (BL 4.125) below:



Drawing reference: LD-1164928, LD-1162340, and LD-1162348

In this evaluation, we have taken into account the fact that the wing loading is asymmetrically distribution between the two main spars. It has also been assumed that the leading and the trailing wing spars do not carry a significant portion of the wing loading. Thus these two spars have not been taken into account in this analysis. The aerodynamic load is still assumed to be located at BL 16.423. The BL 4.125 cross-section cut is shown below:



Drawing Reference: LD-1164914, LD-1164895-1

The moment for this section is,

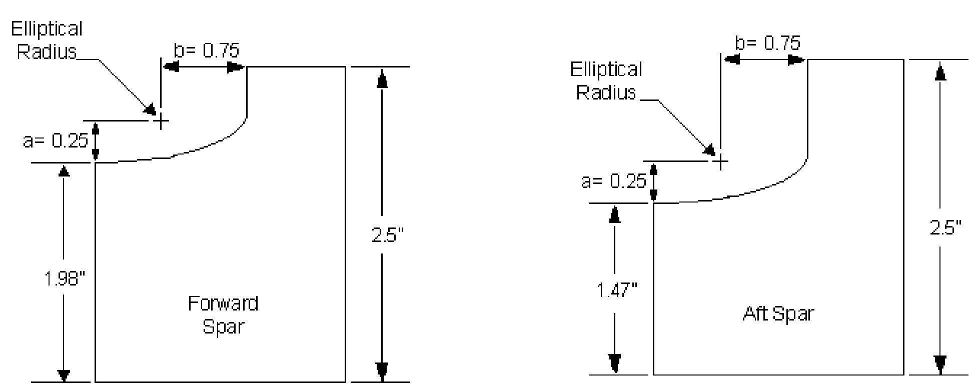
$$M = (F_{horiz})(x_{MAC})$$
$$M = (123.1)(BL\ 16.423 - BL\ 4.125)$$
$$M = 1513.3 lbf$$

And, the bending stress is (see Pro-Engineer data output in the wing analysis section of this report for the "z" modulus value),

$$\sigma = \frac{Mc}{I} = \frac{M}{z}$$
$$\sigma = \frac{1513.3 lbs}{0.377 in^2} = 4014\, psi$$

Also, a stress concentration value is needed. The geometry is given below:

GTM-T2 BL4.125 Stress Concentration Geometry, Forward & Aft Spars



Drawing Reference: LD-1164894

Where,

<u>Forward Spar Geometry:</u>
D = 2.495 *in*
d = 1.98 *in*
a = 0.25 *in*
b = 0.75 *in*
b/d = 0.75/1.98 = 0.38
b/a = 0.75/0.25 = 3.0

Kt = 1.15 (Ref. Fig 77, Peterson, Stress Concentration Factors, © 1974)

<u>Aft Spar Geometry:</u>
D = 2.495 *in*
d = 1.47 *in*
a =0 .25 *in*
b = 0.75 *in*
b/d = 0.75/1.47 = 0.51
b/a = 0.75/0.25 = 3.0
Kt = 1.085 (Ref. Fig 77, Peterson, Stress Concentration Factors, © 1974)

For a conservative analysis the largest stress concentration factor is used, therefore the maximum stress is,

$$\sigma_{Max} = \sigma K_t = (4014)(1.15)$$
$$\sigma_{Max} = 4616\,psi$$

Factor of Safety:

The A/C plywood has a tensile strength of 7578*psi* (reference project document number GTMP-6033), so the factor of safety is:

$$\boxed{F.S. = \frac{7578\,psi}{4616\,psi} = 1.64}$$

Shear Stress at BL 4.125:
It is assumed that all shear loads go through the two main spars (see Pro-Engineer data output in the wing analysis section of this report for the shear area value).

$$\tau = \frac{F_L}{A_{shear}}$$
$$\tau = \frac{123.1lbs}{2.72in^2}$$
$$\tau = 45.3\,psi$$

The A/C plywood has a tensile strength of 7578*psi* (reference project document number GTMP-6031), so the factor of safety is:

$$\boxed{F.S. = \frac{7578\,psi}{45.3\,psi} = 167.5}$$

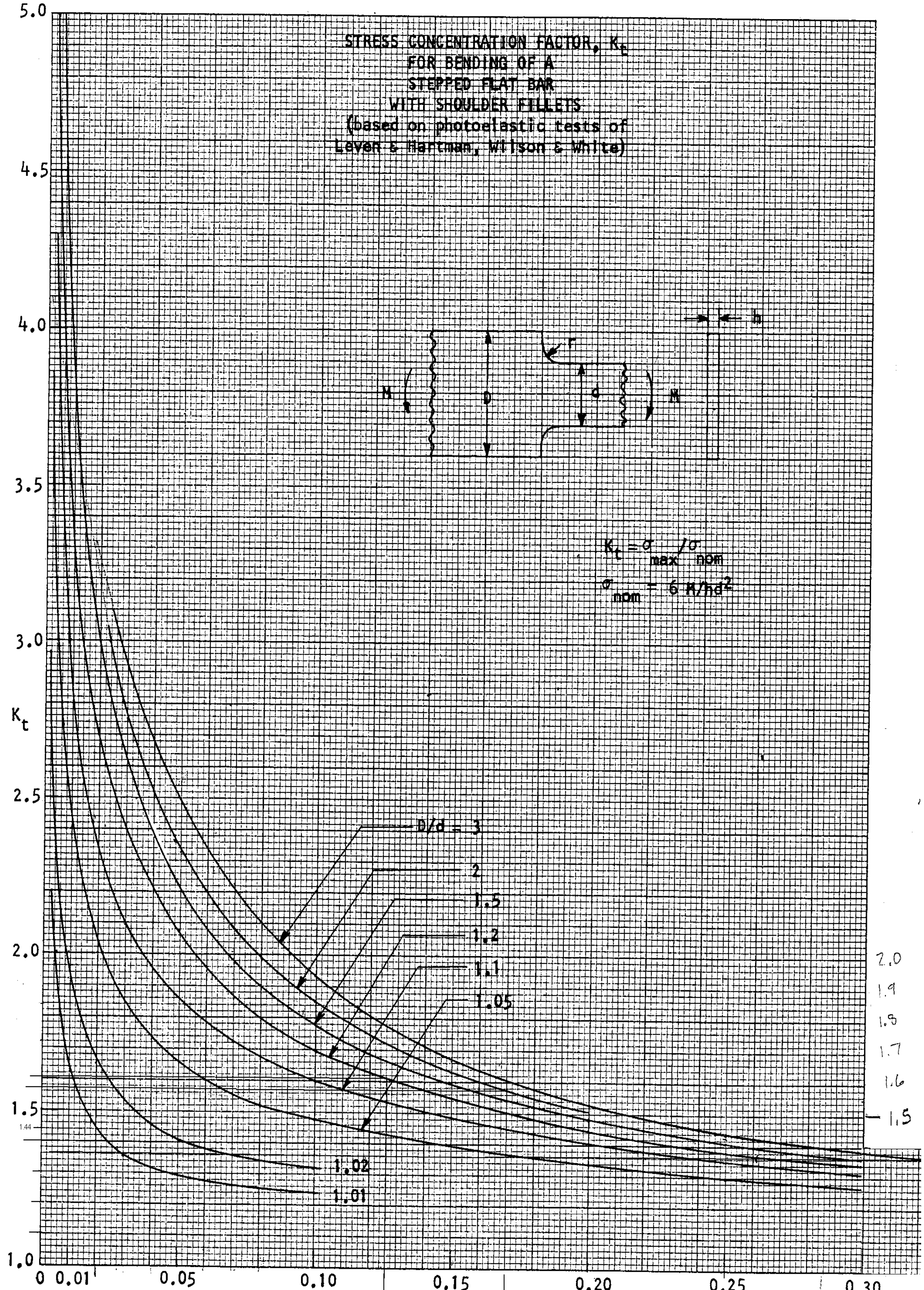
APPENDIX A

Material Data

Engineering Data

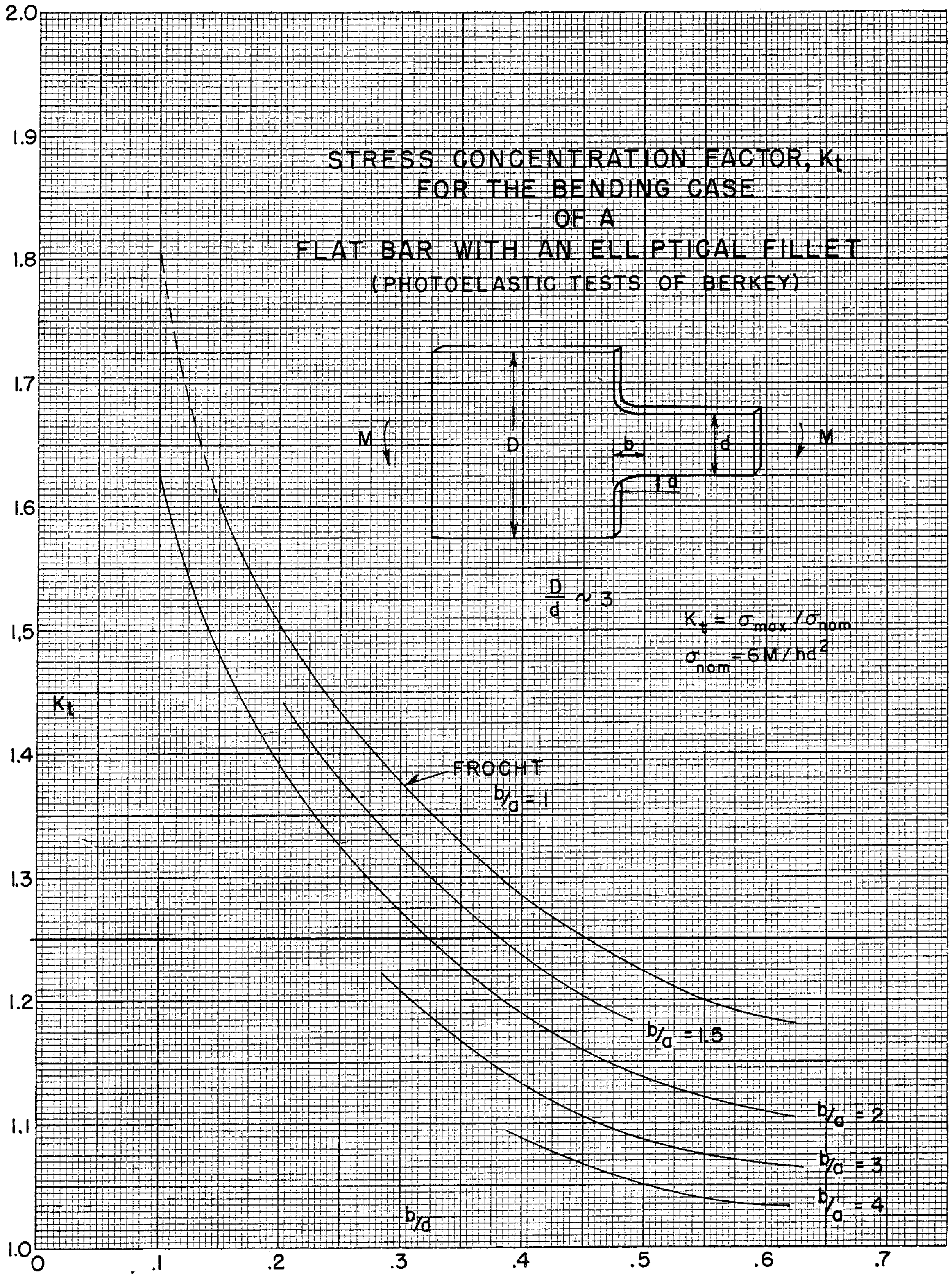
Researcher Supplied Data

FIG. 73



REF. STRESS CONCENTRATION FACTORS, PETERSON, ©1974

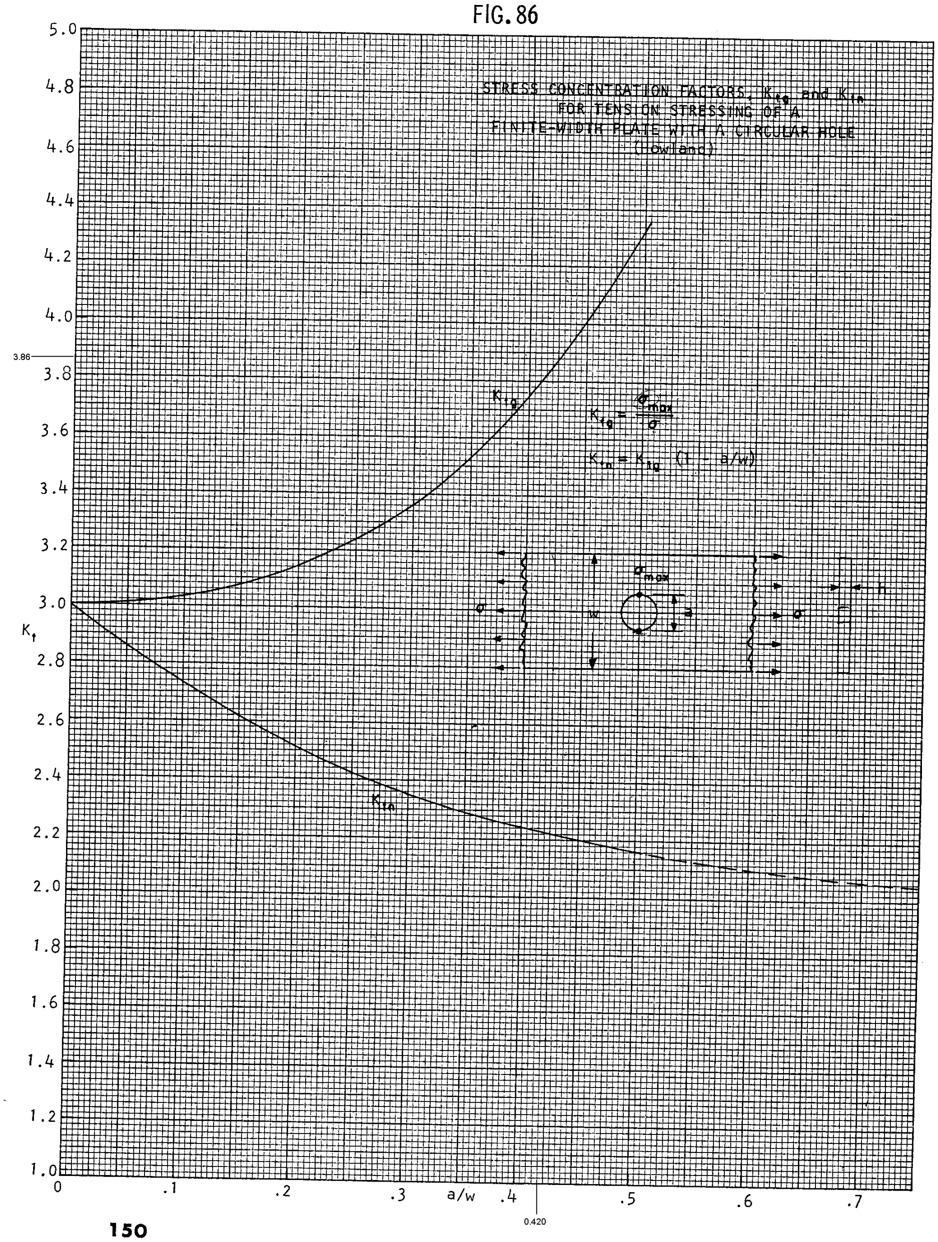
FIG. 77



102

REF. STRESS CONCENTRATION FACTORS, PETERSON, ©1974

FIG. 86



150

REF. STRESS CONCENTRATION FACTORS, PETERSON © 1974

Table 2.6.6.0(b). Design Mechanical and Physical Properties of 15-5PH Stainless Steel Bar and Forging

Specification	AMS 5659						
Form	Bar and forging						
Condition	H900	H925	H1025	H1075	H1100	H1150	H1150M[a]
Thickness or diam., in.	≤12	≤12	≤12	≤12	≤12	≤12	≤12
Basis	S	S	S	S	S	S	S
Mechanical Properties:							
F_{tu}, ksi:							
L	190	170	155	145	140	135	115
T	190	170	155	145	140	135	115
F_{ty}, ksi:							
L	170	155	145	125	115	105	75
T	170	155	145	125	115	105	75
F_{cy}, ksi:							
L	...	...	143	...	...	99	...
T	...	...	143	...	...	99	...
F_{su}, ksi	...	...	97	...	...	85	...
F_{bru}^{b}, ksi:							
(e/D = 1.5)	...	...	263	...	...	230	...
(e/D = 2.0)	...	...	332	...	...	293	...
F_{bry}^{b}, ksi:							
(e/D = 1.5)	...	...	211	...	...	166	...
(e/D = 2.0)	...	...	250	...	...	201	...
e, percent:							
L	10	10	12	13	14	16	18
T	6	7	8	9	10	11	14
RA, percent:							
L	35	38	45	45	45	50	55
T	20	25	32	33	34	35	35
E, 10^3 ksi	28.5						
E_c, 10^3 ksi	29.2						
G, 10^3 ksi	11.2						
μ	0.27						
Physical Properties:							
ω, lb/in.^{3}	0.283						
C, Btu/(lb)(°F)	...						
K and α	See Figure 2.6.6.0						

a H1150M condition is not covered by AMS 5659; properties reflect producers' guaranteed minimum tensile properties.
b Bearing values are "dry pin" values per Section 1.4.7.1.

Table 3.2.3.0(b₁). Design Mechanical and Physical Properties of 2024 Aluminum Alloy Sheet and Plate

Specification	AMS 4037 and AMS-QQ-A-250/4																	AMS-QQ-A-250/4		
Form	Sheet					Plate												Sheet		Plate
Temper	T3					T351												T361		
Thickness, in.	0.008-0.009	0.010-0.128		0.129-0.249		0.250-0.499		0.500-1.000		1.001-1.500		1.501-2.000		2.001-3.000		3.001-4.000		0.020-0.062	0.063-0.249	0.250-0.500
Basis	S	A	B	A	B	A	B	A	B	A	B	A	B	A	B	A	B	S	S	S
Mechanical Properties:																				
F_{tu}, ksi:																				
L	64	64	65	65	66	64	66	63	65	62	64	62	64	60	62	57	59	68	69	67
LT	63	63	64	64	65	64	66	63	65	62	64	62	64	60	62	57	59	67	68	66
ST	...	...	...	...	...	...	...	...	...	...	...	...	...	52[a]	54[a]	49[a]	51[a]	...	...	...
F_{ty}, ksi:																				
L	47	47	48	47	48	48	50	48	50	47	50	47	49	46	48	43	46	56	56	54
LT	42	42	43	42	43	42	44	42	44	42	44	42	44	42	44	41	43	50	51	49
ST	...	...	...	...	...	...	...	...	...	...	...	...	...	38[a]	40[a]	38[a]	39[a]	...	...	...
F_{cy}, ksi:																				
L	39	39	40	39	40	39	41	39	41	39	40	38	40	37	39	35	37	47	48	46
LT	45	45	46	45	46	45	47	45	47	44	46	44	46	43	45	41	43	53	54	52
ST	...	...	...	...	...	...	...	...	...	...	...	...	...	46	48	44	47	...	...	...
F_{su}, ksi	39	39	40	40	41	38	39	37	38	37	38	37	38	35	37	34	35	42	42	41
F_{bru}[b], ksi:																				
(e/D = 1.5)	104	104	106	106	107	97	100	95	98	94	97	94	97	91	94	86	89	111	112	109
(e/D = 2.0)	129	129	131	131	133	119	122	117	120	115	119	115	119	111	115	106	109	137	139	135
F_{bry}[b], ksi:																				
(e/D = 1.5)	73	73	75	73	75	72	76	72	76	72	76	72	76	72	76	70	74	82	84	81
(e/D = 2.0)	88	88	90	88	90	86	90	86	90	86	90	86	90	86	90	84	88	97	99	96
e, percent (S-basis):																				
LT	10	c	...	c	...	12	...	8	...	7	...	6	...	4	...	4	...	8	9	9[d]
E, 10^3 ksi	10.5					10.7												10.5		10.7
E_c, 10^3 ksi	10.7					10.9												10.7		10.9
G, 10^3 ksi	4.0					4.0												4.0		4.0
μ	0.33					0.33												0.33		0.33
Physical Properties:																				
ω, lb/in.^{3}	0.100																			
C, K, and α	See Figure 3.2.3.0																			

a Caution: This specific alloy, temper, and product form exhibits poor stress-corrosion cracking resistance in this grain direction. It corresponds to an SCC resistance rating of D, as indicated in Table 3.1.2.3.1(a).
b Bearing values are "dry pin" values per Section 1.4.7.1. See Table 3.1.2.1.1.
c See Table 3.2.3.0(c).
d 10% for 0.500 inch.

Table 3.6.2.0(b$_1$). Design Mechanical and Physical Properties of 6061 Aluminum Alloy Sheet

Specification	AMS 4026 and AMS-QQ-A-250/11		AMS-QQ-A-250/11	AMS 4025, AMS 4027 and AMS-QQ-A-250/11	
Form	Sheet				
Temper	T4		T42[a]	T6 and T62[b]	
Thickness, in.	0.010-0.249		0.010-0.249	0.010-0.249	
Basis	A	B	S	A	B
Mechanical Properties:					
F_{tu}, ksi:					
L	...	...	...	42	43
LT	30	32	30	42	43
F_{ty}, ksi:					
L	...	...	...	36	38
LT	16	18	14	35	37
F_{cy}, ksi:					
L	...	...	...	35	37
LT	16	18	...	36	38
F_{su}, ksi	20	21	...	27	28
F_{bru}, ksi:					
(e/D = 1.5)	48	51	...	67	69
(e/D = 2.0)	63	67	...	88	90
F_{bry}, ksi:					
(e/D = 1.5)	22	25	...	50	53
(e/D = 2.0)	26	29	...	58	61
e, percent (S-basis):					
LT	c	...	c	c	...
E, 10^3 ksi	9.9				
E_c, 10^3 ksi	10.1				
G, 10^3 ksi	3.8				
μ	0.33				
Physical Properties:					
ω, lb/in.^{3}	0.098				
C, K, and α	See Figure 3.6.2.0				

a Design allowables were based upon data obtained from testing samples of material, supplied in the O or F temper, which were heat treated to demonstrate response to heat treatment by suppliers. Properties obtained by the user may be lower than those listed if the material has been formed or otherwise cold or hot worked, particularly in the annealed temper, prior to solution heat treatment.

b Design allowables were based upon data obtained from testing T6 sheet and from testing samples of sheet, supplied in the O or F temper, which were heat treated to T62 temper to demonstrate response to heat treatment by suppliers. Properties obtained may be lower than those listed if the material has been formed or otherwise cold worked, particularly in the annealed temper, prior to solution heat treatment.

c See Table 3.6.2.0(b$_3$).

Table 3.6.2.0(b₂). Design Mechanical and Physical Properties of 6061 Aluminum Alloy Plate

Specification	AMS 4026 and AMS-QQ-A-250/11				AMS-QQ-A-250/11		AMS 4025, AMS 4027 and AMS-QQ-A-250/11					
Form	Plate											
Temper	T451				T42[a]		T651 and T62[b]					
Thickness, in.	0.250-2.000		2.001-3.000		0.250-1.000	1.001-3.000	0.250-2.000		2.001-3.000		3.001-4.000	4.001-6.000[d]
Basis	A	B	A	B	S	S	A	B	A	B	S	S
Mechanical Properties:												
F_{tu}, ksi:												
L	...	...	...	...	...	...	42	43	...	...	...	...
LT	30	32	30	32	30	30	42	43	42	43	42	40
F_{ty}, ksi:												
L	...	...	...	...	...	...	36	38	...	...	...	...
LT	16	18	16	18	14	14	35	37	35	37	35	35
F_{cy}, ksi:												
L	...	...	...	...	...	...	35	37	...	...	...	...
LT	16	18	...	...	...	...	36	38	...	...	...	...
F_{su}, ksi	20	21	...	...	...	...	27	28	...	...	...	...
F_{bru}, ksi:												
(e/D = 1.5)	48	51	...	...	...	...	67	69	...	...	...	...
(e/D = 2.0)	63	67	...	...	...	...	88	90	...	...	...	...
F_{bry}, ksi:												
(e/D = 1.5)	22	25	...	...	...	...	50	53	...	...	...	...
(e/D = 2.0)	26	29	...	...	...	...	58	61	...	...	...	...
e, percent (S-basis):												
LT	[c]	...	16	...	18	16	[c]	...	6	...	6	6
E, 10^3 ksi	9.9											
E_c, 10^3 ksi	10.1											
G, 10^3 ksi	3.8											
μ	0.33											
Physical Properties:												
ω, lb/in.^{3}	0.098											
C, K, and α	See Figure 3.6.2.0											

Table 3.7.4.0(d). Design Mechanical and Physical Properties of 7075 Aluminum Alloy Bar, Rod, and Shapes; Rolled, Drawn, or Cold-Finished

Specification	AMS 4122, AMS 4123, AMS 4186, AMS 4187, and AMS-QQ-A-225/9								AMS 4124 and AMS-QQ-A-225/9	
Form	Bar, rod, and shapes: rolled, drawn, or cold-finished									
Temper	T6, T651, and T62[a]								T73[b,c] or T7351[c]	
Thickness[d], in.	≤1.000		1.001-2.000		2.001-3.000		3.001-4.000		0.375-2.000	2.001-3.000
Basis	A	B	A	B	A	B	A	B	S	S
Mechanical Properties:										
F_{tu}, ksi:										
L	77	79	77	79	77	79	77	79	68	68
LT	77[e]	79[e]	75[e]	77[e]	72[e]	74[e]	69[e]	71[e]	...	65[e,f]
F_{ty}, ksi:										
L	66	68	66	68	66	68	66	68	56	56
LT	66[e]	68[e]	66[e]	68[e]	63[e]	65[e]	60[e]	62[e]	...	52[e,f]
F_{cy}, ksi:										
L	64	66	64	66	64	66	64	66	54	54
LT	...	...	...	...	...	...	...	...	...	55[f]
F_{su}, ksi	46	47	46	47	46	47	46	47	42	40
F_{bru}, ksi:										
(e/D = 1.5)	100	103	100	103	100	103	100	103	101	101
(e/D = 2.0)	123	126	123	126	123	126	123	126	131	131
F_{bry}, ksi:										
(e/D = 1.5)	86	88	86	88	86	88	86	88	81	81
(e/D = 2.0)	92	95	92	95	92	95	92	95	100	100
e, percent (S-basis):										
L	7	...	7	...	7	...	7	...	10	10
E, 10^3 ksi	10.3									
E_c, 10^3 ksi	10.5									
G, 10^3 ksi	3.9									
μ	0.33									
Physical Properties:										
ω, lb/in.^{3}	0.101									
C, K, and α	See Figure 3.7.4.0									

a Design allowables were based upon data obtained from testing of T6 and T651 material and from samples of material, supplied in the O or F temper, which were heat treated to T62 temper to demonstrate response to heat treatment by suppliers.
b Design allowables were based upon data obtained from testing T73 and T7351 temper material and from testing samples of material, supplied in the O or F temper, which were heat treated to T73 temper to demonstrate response to heat treatment by suppliers.
c Bearing values are "dry pin" values per Section 1.4.7.1.
d For rounds (rod) maximum diameter is 4 inches; for square bar, maximum size is 3½ inches; for rectangular bar, maximum thickness is 3 inches with corresponding width of 6 inches; for rectangular bar less than 3 inches in thickness, maximum width is 10 inches.
e Caution: This specific alloy, temper, and product form exhibits poor stress-corrosion cracking resistance in this grain direction. It corresponds to an SCC resistance rating of D, as indicated in Table 3.1.2.3.1(a).
f ST grain direction.

Table 3.7.4.0(g). Design Mechanical and Physical Properties of 7075 Aluminum Alloy Extrusion

Specification	QQ-A-200/11														
Form	Extrusion (rod, bar, and shapes)														
Temper	T6, T6510, T6511, and T62[a]														
Cross-Sectional Area, in.^{2}	≤20												>20, ≤32	≤32	
Thickness, in.[b]	≤0.249		0.250-0.499		0.500-0.749		0.750-1.499		1.500-2.999		3.000-4.499			4.500-5.000	
Basis	A	B	A	B	A	B	A	B	A	B	A	B	S	A	B
Mechanical Properties:															
F_{tu}, ksi:															
L	78	82	81	85	81	85	81	85	81	85	81	84	78	78	81
LT	75	79	78	82	77	81	75	79	71	75	67	69	64	63	65
ST	...	...	...	...	...	...	...	...	67[c]	71[c]	67[c]	69[c]	64[c]	63[c]	65[c]
F_{ty}, ksi:															
L	70	74	73	77	72	76	72	76	72	76	71	74	70	68	71
LT	66	70	69	72	67	71	65	69	61	65	56	59	55	52	55
ST	...	...	...	...	...	...	...	...	56[c]	59[c]	55[c]	58[c]	55[c]	52[c]	55[c]
F_{cy}, ksi:															
L	70	74	73	77	72	76	72	76	72	76	71	74	70	68	71
LT	72	76	74	78	73	77	71	75	67	71	62	64	61	57	60
ST	...	...	...	...	...	...	...	...	62	66	62	64	61	57	60
F_{su}, ksi	41	44	43	45	43	45	43	45	42	44	40	42	39	38	40
F_{bru}[d], ksi:															
(e/D = 1.5)	111	117	115	121	115	120	113	119	110	115	106	110	102	101	105
(e/D = 2.0)	140	148	146	153	145	152	144	151	141	148	137	142	132	131	136
F_{bry}[d], ksi:															
(e/D = 1.5)	92	97	96	101	94	99	93	98	89	94	84	88	83	79	83
(e/D = 2.0)	108	114	113	119	111	117	110	116	106	112	101	105	100	95	100
e, percent (S-basis):															
L	7	...	7	...	7	...	7	...	7	...	7	...	6	6	...
E, 10^3 ksi	10.4														
E_c, 10^3 ksi	10.7														
G, 10^3 ksi	4.0														
μ	0.33														
Physical Properties:															
ρ, lb/in.^{3}	0.101														
C, K, and α	See Figure 3.7.4.0														

a Design allowables were based upon data obtained from testing T6, T6510, and T6511 temper extrusions and from testing samples of extrusion supplied in the O or F temper, which were heat treated to T62 temper to demonstrate response to heat treatment by suppliers. Properties obtained by the user may be lower than those listed if the material has been formed or otherwise cold worked, particularly in the annealed temper, prior to solution heat treatment.

b The mechanical properties are to be based upon the thickness at the time of quench.

b Caution: This specific alloy, temper, and product form exhibits poor stress-corrosion cracking resistance in this grain direction. It corresponds to an SCC resistance rating of D, as indicated in Table 3.1.2.3.1(a).

d Bearing values are "dry pin" values per Section 1.4.7.1.

More About Cast Nylon 6

Sheets, Rods, and Hollow Rods

Common applications for this material include: general purpose wear applications and use in food contact environments. It's also good for bearings, wheels, gears, and machined parts.
Meet the following standards: FDA and USDA compliant and meets 3-A sanitary standard requirements.

Tensile Strength:	12,000 psi per ASTM D638
Impact Strength:	0.4 ft.-lbs./in. per ASTM D256 Type A
Coefficient of Friction:	0.20 per PTM 5007
Dielectric Strength:	500 V/mil per ASTM D149
Hardness:	Rockwell R: 115 per ASTM D785
Coefficient of Thermal Expansion:	3.5×10^{-5} in./in./°F per ASTM E831
Weather Resistance:	Not weather resistant.
Processing:	*Machinability:* Can be easily machined. *Molding:* Not recommended. *Welding:* Not recommended. *Thermoforming:* Not recommended.
Scratch Resistance:	Good.
Chemical Resistance:	Use with hydrocarbons, ketones, and ethers. Do not use with strong acids, strong alkalies, or chlorinated solvents.

Discs

Common applications for this material include: general purpose wear applications and use in food contact environments.
Meet the following standards: FDA and USDA compliant and meets 3-A sanitary standard requirements.

Tensile Strength:	10,000-13,500 psi per ASTM D638
Impact Strength:	0.7-0.9 ft.-lbs./in. per ASTM D256
Coefficient of Friction:	0.22
Dielectric Strength:	500-600 V/mil per ASTM D149
Hardness:	Rockwell R: 115-125 per ASTM D785
Coefficient of Thermal Expansion:	5×10^{-5} in./in./°F per ASTM D696
Weather Resistance:	Can be used outdoors. It does, however, absorb water and has limited UV resistance.
Processing:	*Machinability:* Can be easily machined. *Molding:* Not recommended. *Welding:* Not recommended. *Thermoforming:* Not recommended.
Scratch Resistance:	Average to good.
Chemical Resistance:	Use with acetone, acetylene, ammonia, isobutyl alcohol, and sodium chloride. Do not use with strong acids or strong alkalies.

Rectangular Bars

Common applications for this material include: general purpose wear applications and use in food contact environments.
Meet the following standards: FDA and USDA compliant and meets 3-A sanitary standard requirements.

Tensile Strength:	10,000-13,500 psi per ASTM D638
Impact Strength:	0.7-0.9 ft.-lbs./in. per ASTM D256
Coefficient of Friction:	0.22
Dielectric Strength:	500-600 V/mil per ASTM D149
Hardness:	Rockwell R: 115-125 per ASTM D785
Coefficient of Thermal Expansion:	5×10^{-5} in./in./°F per ASTM D696
Weather Resistance:	Can be used outdoors. It does, however, absorb water.
Processing:	*Machinability:* Can be easily machined with standard tooling. *Molding:* Not recommended. *Welding:* Not recommended. *Thermoforming:* Not recommended.
Scratch Resistance:	Average to good.
Chemical Resistance:	Use with acetone, acetylene, ammonia, isobutyl alcohol, sodium chloride, chlorinated solvents, and alcohols. Do not use with strong acids or strong alkalies.

This information is to advise you on current technical knowledge for comparative purposes only. It is given without obligation or liability. No warranty of fitness for a particular purpose or application is made.

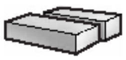
TABLE 2 Tensile Requirements

Temper Designation	Standard Name	Diameter or Distance Between Parallel Surfaces, in. [mm]		Tensile Strength min, ksi [MPa]	Yield Strength at 0.5 % Extension under Load min, ksi [MPa]	Elongation,[A] min, %
		Rod and Wire				
O60	soft	1 [25] and under		48 [330]	20 [140]	15
		over 1 [25] to 2 [50]		44 [305]	18 [125]	20
		over 2 [50]		40 [275]	15 [105]	25
H02	half-hard	½ [12] and under		57 [395]	25 [170]	7[B]
		over ½ [12] to 1 [25]		55 [380][C]	25 [170]	10
		over 1 [25] to 2 [50]		50 [345]	20 [140]	15
		over 2 [50] to 4 [100], and		45 [310]	15 [105]	20
		over 4 [100]		40 [275]	15 [105]	20
H04	hard	1⁄16 [1.6] to 3⁄16 [4], incl.		80 [550]	45 [310]	
		over 3⁄16 [4] to ½ [12], incl.		70 [480]	35 [240]	4
		over ½ [12] to ¾ [18], incl.		65 [450]	30 [205]	6
		Bar				
	Standard Name	Thickness, in. [mm]	Width, in. [mm]			
O60	soft anneal	1 [25] and under	6 [150] and under	44 [305]	18 [125]	20
		over 1 [25]	6 [150] and under	40 [275]	15 [105]	25
H02	half-hard	½ [12] and under	1 [25] and under	50 [345]	25 [170]	10
		½ [12] and under	over 1 [25] to 6 [150]	45 [310]	17 [115]	15
		over ½ [12] to 2 [50]	2 [50] and under	45 [310]	17 [115]	15
		over ½ [12] to 2 [50]	over 2 [50] to 6 [150]	40 [275]	15 [105]	20
		over 2 [50]	over 2 [50] to 4 [100]	40 [275]	15 [105]	20

[A] In any case, a minimum gage length of 1 in. [25 mm] shall be used. SI elongation values are based on a gage length of 5.65 times the square root of the area for dimensions greater than 2.5 mm.
[B] For product furnished in coils the elongation shall be 4 % min.
[C] If product is specified for thread rolling applications, the minimum tensile strength shall be 52 ksi [350 MPa].

WHITE DELRIN SPECIFICATIONS

Rectangular Bars and Square Bars

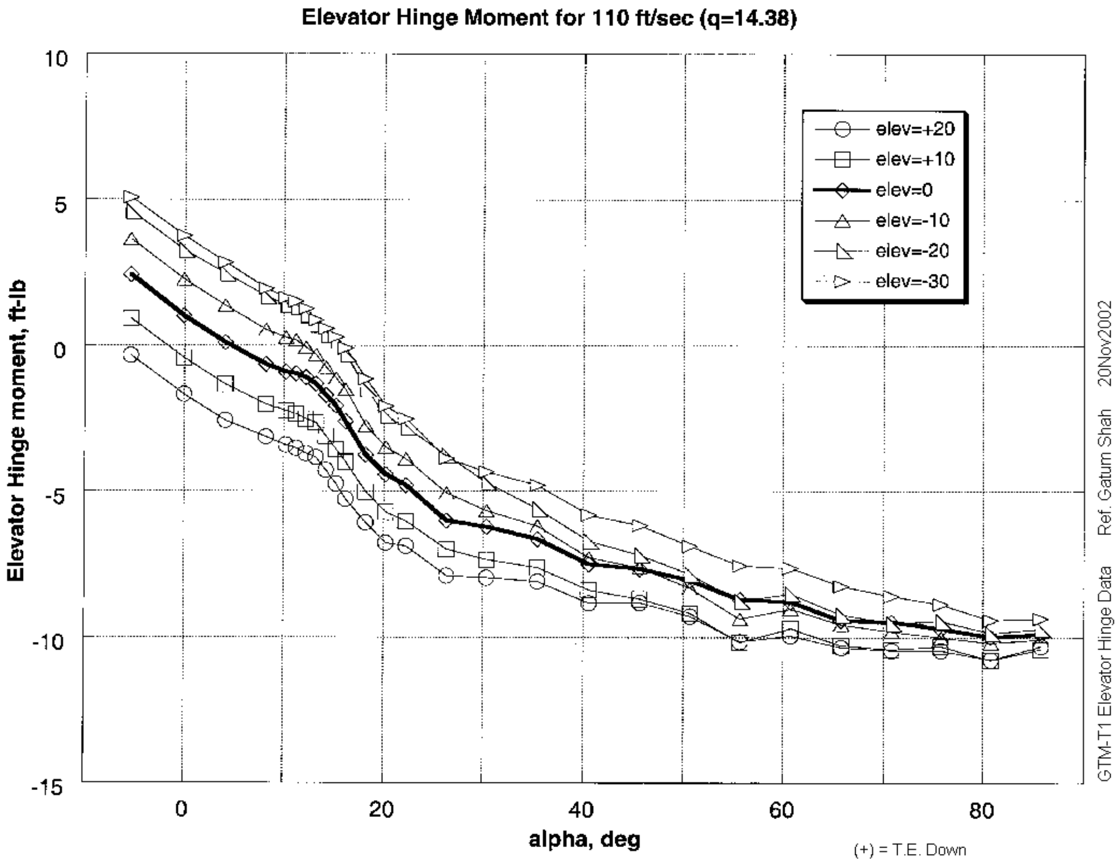


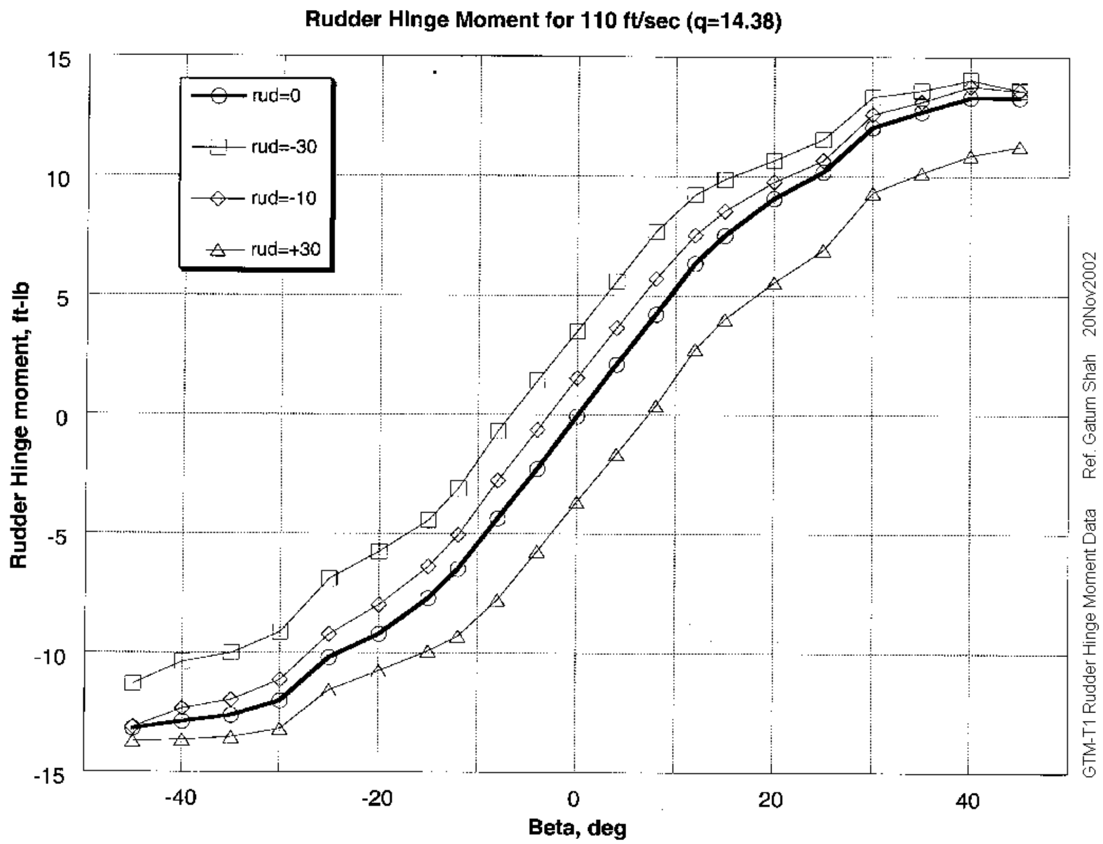
Common applications for this material include bearings, bushings, fuel system parts, gears, pulleys, springs, sprockets, rollers, valve seats, and seals.

Meet the following standards: UL94HB for flammability; FDA 21 CFR 177.2480.

Tensile Strength:	10,000 psi
Impact Strength:	2.3 ft.-lbs./in.
Coefficient of Friction:	0.35
Dielectric Strength:	500 V/mil
Durometer:	Rockwell M: 94
Coefficient of Thermal Expansion:	6.8×10^{-5} in./in./°F
Weather Resistance:	Use indoors.
Processing:	*Machinability:* Easy to machine. *Molding:* Can be molded using injection or extruded molding. *Welding:* Can be welded using ultrasonic or spin welding. *Thermoforming:* Not recommended.
Scratch Resistance:	Not scratch resistant.
Chemical Resistance:	High chemical resistance. Use with alcohols, aliphatics, aromatics, aldehydes, ketones, ethers, esters, oils, greases, gasoline, diesel and methanol based fuels, and agricultural chemicals. Do not use with strong acids, strong bases, ethyl acetate, bleach, chloroform, strong oxidizing agents, phenol, mineral acids, and ethylene glycol.

Reference: www.mcmaster.com


Elevator Hinge Moment for 110 ft/sec (q=14.38)
10
5
0
-5
-10
-15
Elevator Hinge moment, ft-lb
0
20
40
60
80
alpha, deg
elev=+20
elev=+10
elev=0
elev=-10
elev=-20
elev=-30
GTM-T1 Elevator Hinge Data Ref. Gatum Shah 20Nov2002
(+) = T.E. Down



Rudder Hinge Moment for 110 ft/sec (q=14.38)
rud=0
rud=-30
rud=-10
rud=+30
Rudder Hinge moment, ft-lb
Beta, deg
15
10
5
0
-5
-10
-15
-40
-20
0
20
40
GTM-T1 Rudder Hinge Moment Data Ref. Gatum Shah 20Nov2002

Hysol 9359.3 Lap Shear Specs Per ASTM standard D1002

Specimen: Hysol 9359.3 R.T. Cure

Manufacturers Lap Shear Strength Specs: 4500 psi

Specimen	Failure Load (lbs.)	Tensile Lap Shear Strength (psi)
b-1	1173	2346
b-2	1101	2202
b-3	1048	2096
b-4	1037	2074
b-5	1058	2116
Average		**2167**

Room Temperatured Cured for 6 Days
Room Temperatured Tested

Aluminum tabs were prepared according to a modified Boeing Specication developed for the LSV Project

Data Per 14.2% BWB LSV Materials Characterization Study, June 5,2002

Jumbo BB
Digi
Jumbo Speed BB
Digi

Source:
Multiplex USA
http://www.multiplexrc.com/servo_mcv2

	Profi BB Digi	Profi BB *Speed* Digi	Power BB Digi	Power BB *Speed* Digi	Jumbo BB Digi	Jumbo BB *Speed* Digi
Transit speed 40° 4 Cells	0.15 sec	0.11 sec	0.29 sec	0.23 sec	0.24 sec	0.19 sec
Torque 4 Cells	62 oz-in	48 oz-in	99 oz-in	78 oz-in	204 oz-in	166 oz-in
Holding Power Max 5° deflection 4 Cells	**113 oz-in**	**113 oz-in**	**227 oz-in**	**177 oz-in**	**368 oz-in**	**283 oz-in**
Transit speed 40° 5 Cells	**0.12 sec**	**0.09 sec**	**0.23 sec**	**0.19 sec**	**0.19 sec**	**0.14 sec**
Torque 5 Cells	**78 oz-in**	**59 oz-in**	**123 oz-in**	**96 oz-in**	**255 oz-in**	**205 oz-in**
Holding Power Max 5° deflection 5 Cells	**141 oz-in**	**141 oz-in**	**283 oz-in**	**221 oz-in**	**460 oz-in**	**354 oz-in**
Weight (g) Weight (oz) LxWxH (in)	48 g 1.69 oz 1.42x1.54x0.75	48 g 1.69 oz 1.42x1.54x0.75	82 g 2.89 oz 1.85x1.85x0.91	82 g 2.89 oz 1.85x1.85x0.91	158 g 5.57 oz 2.36x2.36x1.18	158 g 5.57 oz 2.36x2.36x1.18
Price Part No.	$82.95 Buy Now! 6 5374	$82.95 Buy Now! 6 5338	$109.37 Buy Now! 6 5376	$111.51 Buy Now! 6 5383	$128.35 Buy Now! 6 5377	$133.91 Buy Now! 6 5384

Servo Holding Power Test for JR8411 Digital Servo

Project No. GTMP-6003

Overview

Prepared by: Greg Howland, Lockheed

This document describes the work done to evaluate the holding torque of a JR8411 digital servo for use on the 5.5% Civil Transport (dynamically scaled) Model.

The purpose of this study is to determine if the JR8411 digital servo has sufficient holding torqre for use on the horizontal incidence on the Generic Transport Model (GTM T2). During the stress analysis of the model, it was determined that with the current linkages in the model, that the horizontal stabilizer incidence servo could see loads higher than its operational rated specifications. Under worst-case maximum loads, the servo could realize a load of 200 oz-inches. The servo is rated to operate at 155 oz inches. This is more than sufficient to trim the horizontal stabilizer incidence during a normal flight; however it was necessary to verify that the servo could hold position with a 200 oz inch load applied. Conditions where a load of 200 oz inches is realized by the servo do not require the servo to overcome the load, but to maintain position without being back driven.

Setup and Testing

A JR8411 Servo was mounted in a rigid fixture. A moment arm was mounted to the servo output shaft. Loads were applied to the moment arm at a distance of 8.1 inches from the axis of rotation of the servo shaft. The servo was connected to a JR Reciever and powered by a JR Extra 3000 5 cell battery. The signal was sent with a JR Transmitter. A load of 700 grams (24.7oz) was applied to the moment arm and the servo maintained position. 24.7 oz at a distance of 8.1 inches equates to 200 oz inches. Another load of 1000 grams (35.27 oz) was applied to the 8.1 inch moment arm and the servo again maintained position. 35.27 oz at a distance of 8.1 inches equates to 286 oz inches.

Conclusion

The JR8411 Servo was required to have a holding torque of 200 oz inches to be sufficient for use on the GTM T2. The servo was loaded to 286 oz inches and maintained position. It was therefore determined that the JR8411 servo is adequate for use on actuation of the GTM T2 horizontal stabilizer incidence.

Note: Although these requirements are for the final design (FD1) they where used in some capacity to design and analyze the GTM-T2.

MODEL REQUIREMENTS FOR 5.5%-SCALE CIVIL TRANSPORT RPV

I. General Information:

- This model requirements summary is intended to be an iterative and interactive document. Many of the requirements, while being relatively rigid, may have some flexibility on a case-by-case basis. In situations where meeting the absolute numbers presents a significant design difficulty, or is beyond the capability of available hardware, alternate requirements may be acceptable, but such situations should be communicated as soon as practical to the research team so appropriate trade-offs can be studied and discussed.
- Model geometry/weight/inertia requirements are targeted to represent a medium-range twin engine commercial transport, and are based on the Boeing 757-200 configuration
- Dynamic scaling match to full-scale flight condition: 200,000 lbs at 13,000 ft
- *Dimensions*

Wing Reference Area:	5.9018 ft^2	Wing MAC:	0.9153 ft
Wing Span:	6.8488 ft	Model Length:	8.5388 ft

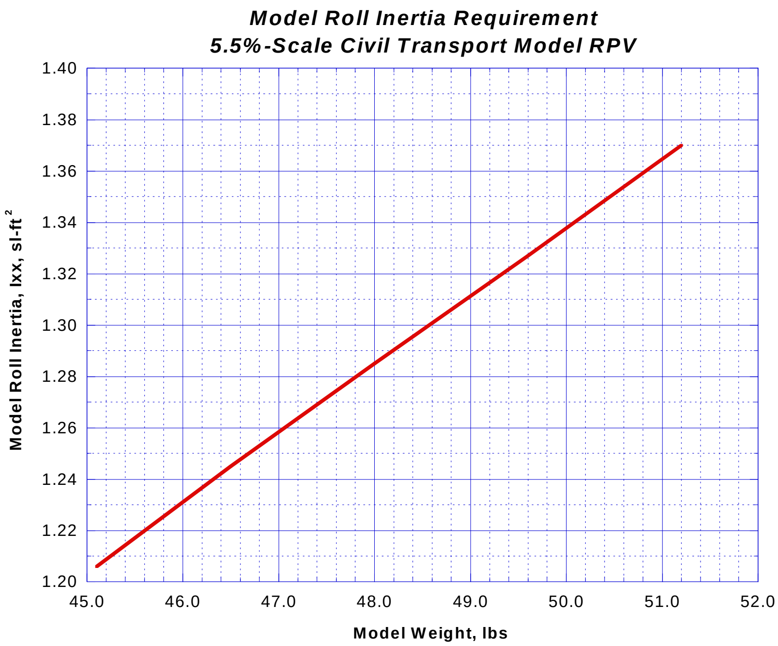
- *Target Maximum Mass Properties*

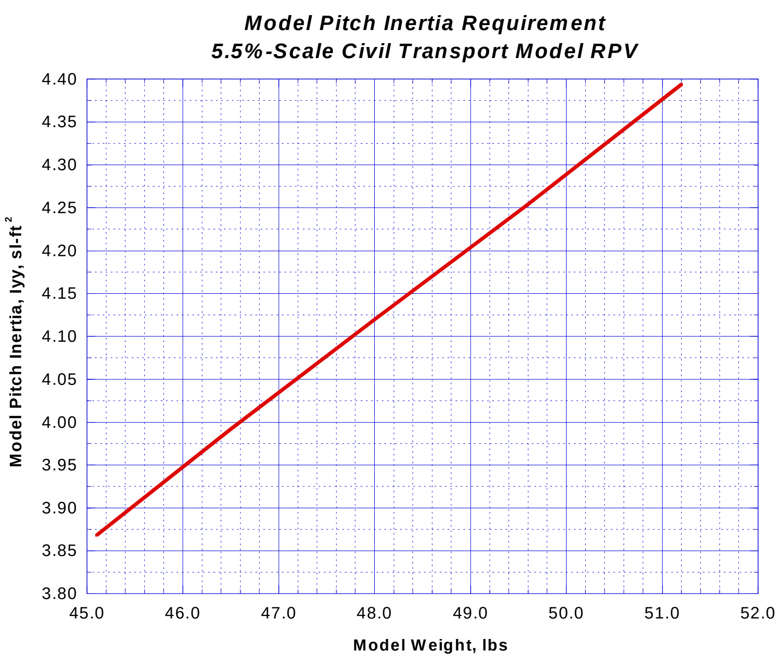
Weight:	49.6 lbs		
Ixx (roll):	1.327 sl-ft^2	Izz (yaw):	5.454 sl-ft^2
Iyy: (pitch)	4.254 sl-ft^2	Ixz:	0.120 sl-ft^2

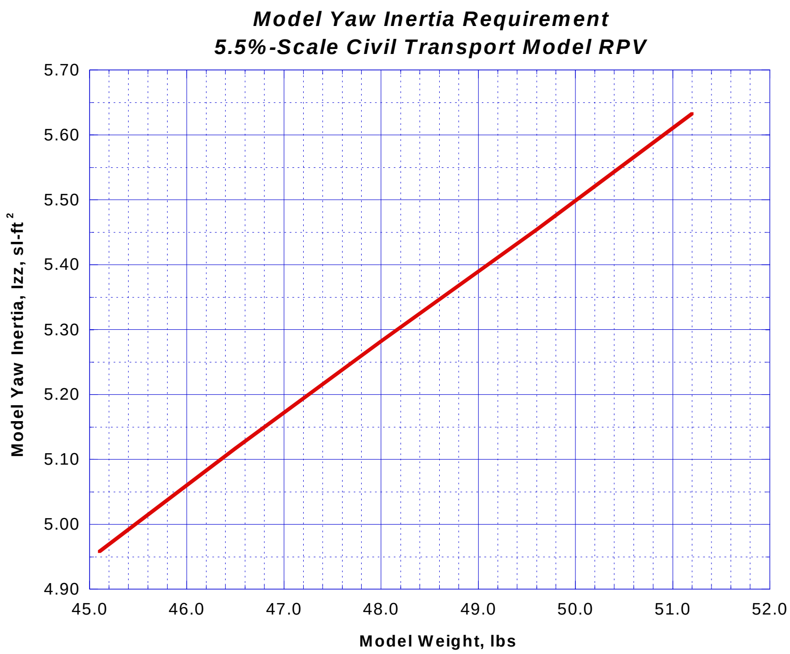
It is desired that the model be as light as possible, with the above being the maximum acceptable weight. The target inertia values listed above are based on that maximum weight. Since the target inertia values are a function of model weight, the actual target inertias will be determined by the final model weight, and are shown in the attached plots. Inertias are desired to be within 5% of the eventual target values.
A final weight greater than the maximum is undesirable for several reasons. It means higher takeoff/landing and flight speeds, resulting in more control difficulty and greater safety risk; and results in a dynamic scaling match to a heavier full-scale airplane at a higher altitude. This pushes us away from the typical weight and altitude conditions where loss-of-control accidents happen, diminishing the realism of the research, and hence its value.

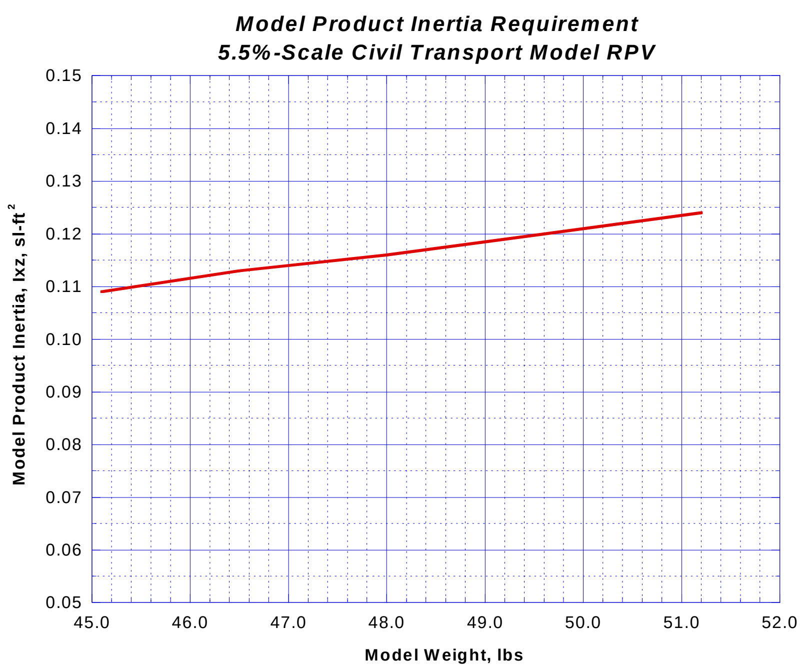
- Rigidity – the basic airframe structure must be relatively rigid in order to minimize aeroelastic effects. Vertical deflection at the wingtip during flight condition loads should be within +/- 0.25 inches. Control surfaces should not deflect more than +/- 0.05 inches under flight condition loads.
- Symmetry – under both no-load and flight condition loads, the configuration should be symmetrical (side-to-side) to within +/- 0.125 inches.

- Multiplier for FCS frame rates: 4.26 (Model FCS rate = 4.26 * Full-Scale FCS rate)
- Model flight altitude: Sea level to 1,500 feet
  - Model flight test speed: 60 to 110 ft/sec
  - Model takeoff/landing speeds: as low as possible
  - Flight duration: 10 minutes minimum
  - Center of Gravity:
    - Nominal c.g. should be 20-25% MAC
    - Provide information on the feasibility of a ballasting provision to be able to vary c.g. from 10% to 40% MAC (approximately a 3-inch range)


Model Roll Inertia Requirement
5.5%-Scale Civil Transport Model RPV
Model Roll Inertia, Ixx, sl-ft²
1.40
1.38
1.36
1.34
1.32
1.30
1.28
1.26
1.24
1.22
1.20
45.0
46.0
47.0
48.0
49.0
50.0
51.0
52.0
Model Weight, lbs


Model Pitch Inertia Requirement
5.5%-Scale Civil Transport Model RPV
Model Pitch Inertia, Iyy, sl-ft²
4.40
4.35
4.30
4.25
4.20
4.15
4.10
4.05
4.00
3.95
3.90
3.85
3.80
45.0
46.0
47.0
48.0
49.0
50.0
51.0
52.0
Model Weight, lbs


Model Yaw Inertia Requirement
5.5%-Scale Civil Transport Model RPV
Model Yaw Inertia, Izz, sl-ft²
5.70
5.60
5.50
5.40
5.30
5.20
5.10
5.00
4.90
45.0
46.0
47.0
48.0
49.0
50.0
51.0
52.0
Model Weight, lbs


Model Product Inertia Requirement
5.5%-Scale Civil Transport Model RPV
Model Product Inertia, Ixz, sl-ft²
0.15
0.14
0.13
0.12
0.11
0.10
0.09
0.08
0.07
0.06
0.05
45.0
46.0
47.0
48.0
49.0
50.0
51.0
52.0
Model Weight, lbs

<u>II. Data System, up- and down-link, with minimum frequencies as noted</u>

	Item	**Frequency (min)**
1.	Angles of attack and sideslip	120 Hz
2.	Airspeed	120 Hz
3.	Model attitudes (□□□□□□□)	240 Hz
4.	Model angular rates (p, q, r)	240 Hz
5.	Linear accelerations (Ax, Ay, Az) (along aircraft centerline, as close to c.g. as practical)	240 Hz
6.	Linear accelerations (Ax, Ay, Az) at wingtip	240 Hz
7.	Linear accelerations (Ax, Ay, Az) at tail	240 Hz
8.	Throttle command, left and right engines	30 Hz
9.	Thrust (or directly-thrust-related) output, left and right engines	60 Hz
10.	Position commands to control surfaces	240 Hz
11.	Position feedback from all control surfaces	240 Hz
12.	GPS position output	30 Hz
13.	Landing gear position (up/down)	15 Hz
14.	Telemetered video from on-board camera	TBD (separate system)

NOTE: Final frequencies should be multiples of lower ones (e.g., 15,30, 60, 120, 240)

III. Propulsion

- Two independently-operated jet turbine engines, one per wing
- Minimal thrust (< 1 lb) at idle power (per engine)
- Controllable/sustainable at any thrust level up to 20 lbs (per engine)

IV. Recovery Systems

The requirements for a recovery system are primarily for a range safety and survivability standpoint, not so much a research requirement. **Therefore, they will most likely be determined based on the flight test location, and its host organization's requirements, as well as Langley ASRB requirements**. At this point, it is expected that the recovery system(s) must fulfill two requirements:

1. Spin/Loss-of-Control recovery – recover the aircraft from a spin or loss-of-control situation and return to a stable flight condition (such as a release-able spin chute or other aerodynamic stabilization system)
2. Range Safety – following an unrecoverable situation (e.g., lost communication, engine failure/malfunction, structural failure), ensure that the vehicle a) does not depart the test area; and b) can be recovered with minimal damage if possible. (such as a recovery parachute, or autonomous landing capability)

Any single or multiple system which meets these requirements may be acceptable.

V. Video:

On-board video camera with a forward-looking view, e.g., flight deck location, vertical tail, etc.

VI. Control Surfaces:

- All actuators should be accessible for repair or replacement
- Range and rate requirements are **somewhat** flexible; i.e., if meeting the given requirements poses extreme complexity or difficulty in design, hardware capability/availability, etc., there may be some negotiation room on a case-by-case basis
- Control surfaces should be directly driven from actuators when possible (i.e., minimize linkages between actuator and surface, thereby minimizing slop and freeplay)
- High-lift system (Flap) is intended to be a simple system to allow low-speed take-off/landing and some stabilized low-speed/high-alpha flight. It is not required to be representative of the actual flap system of the 757.

Location	Surface	Range-desired	Range-minimum	Rate
Left Wing	Outboard Aileron	± 30º	± 26º	250 º/sec
Left Wing	Inboard Aileron	± 30º	± 26º	250 º/sec
Left Wing	Outboard Spoiler	0º to 60º	0º to 60º	315 º/sec
Left Wing	Inboard Spoiler	0º to 60º	0º to 60º	315 º/sec
Left Wing	Flap	TBD		TBD
Left Wing	Allow for addition of a bolt-on leading-edge slat (future use)			
Right Wing	Allow for addition of a bolt-on leading-edge slat (future use)			
Right Wing	Flap	TBD		TBD
Right Wing	Inboard Spoiler	0º to 60º	0º to 60º	315 º/sec
Right Wing	Outboard Spoiler	0º to 60º	0º to 60º	315 º/sec
Right Wing	Inboard Aileron	± 30º	± 26º	250 º/sec
Right Wing	Outboard Aileron	± 30º	± 26º	250 º/sec
Left Horizontal	Outboard Elevator	-45º to +30º (+TED)	-36º to +30º (+TED)	200 º/sec
Left Horizontal	Inboard Elevator	-45º to +30º (+TED)	-36º to +30º (+TED)	200 º/sec
Left Horizontal	Stabilizer	-20º to +10º (+TED)	-20º to +10º (+TED)	100 º/sec
Right Horizontal	Stabilizer	-20º to +10º (+TED)	-20º to +10º (+TED)	100 º/sec

Right Horizontal	Inboard Elevator 200 º/sec	-45º to +30º (+TED)	-36º to +30º (+TED)
Right Horizontal	Outboard Elevator 200 º/sec	-45º to +30º (+TED)	-36º to +30º (+TED)
Vertical Tail	Upper Rudder 250 º/sec	± 45º	± 36º
Vertical Tail	Lower Rudder 250 º/sec	± 45º	± 36º
Fuselage/Wing	Landing Gear System		

Prioritization for control surfaces:

If all of the above controls cannot be accommodated because of weight/space/etc. restrictions, reduce the requirements based on the following prioritization (i.e., 1st item is 1st to be eliminated):

1. Split stabilizer – have left and right stabilizer move as a single surface instead of independently
2. Split ailerons – replace inboard and outboard ailerons with a single aileron on each wing
3. Split elevator – replace replace inboard and outboard elevators with a single elevator on each stabilizer
4. Split rudder – replace upper and lower rudders with a single rudder
5. Independent spoiler control – link spoiler movement with aileron command/position (would eliminate channel requirements, but probably not the spoiler actuators themselves)