AE462
Exercise 2-3

Date
4/02/2024

Instructor
Prof. G.M. Kamath

Group 11
Aryaman Badkul
Palmate Aditya Tukaram
Shrutikirti Singh
Prabuddha Singh
Shikha Sharma
Rahangdale Prachi Sanjay

Exercise 2

1. Calculation of loads as a function of span

Flight loads using Schrenk's Approximation

$$L_e(y) = \frac{4L}{\pi b}\sqrt{1-\left(\frac{2y}{b}\right)^2}$$

where,

L = total lift = weight (for n=1)

b=wing span

y=distance along span from aircraft centre-line

$$L_t(y) = \frac{2L}{b(1+\lambda)}\left[1-\frac{2y}{b}(1-\lambda)\right]$$

where,

λ=taper ratio of trapezoidal wing

$$L_s(y) = \frac{1}{2}[L_e + L_t]$$

Weight(W)= 612.842 kg=Lift(L)

b(wingspan length)=10880 mm =10.88 m

λ (Taper ratio)=0.6

Wing Area= S = 9.48878 m^2

After putting values we get,

$L_e(y)$=703.556 *$\sqrt{1 - \frac{y^2}{29.59}}$

$L_t(y)$=690.714 * [1- $\frac{y}{13.6}$]

The above equations of Le and Lt gives us Lift distribution per unit length. To better visualize this, we created sections throughout the wingspan. The length of each section along wingspan is 0.1 meter and the Ls used to calculate Lift of that section is average value of Ls at y_1 and y_1+0.1
Multiplying this avg Ls will give us lift in that section.

Since the half wingspan length is 5.44 m, we had total 54 sections. Values of $Lift_{section}$ and $y_{avg(section)}$ are tabulated on next slide

For Lift due to half wing,

integrating along spanwise direction, from root chord(y=0) to tip chord(y=b/2)

L_e= 613.1528 kg

L_t = 612.842 kg

L_s = 0.5(L_e+ L_t) = 612.9974 kg

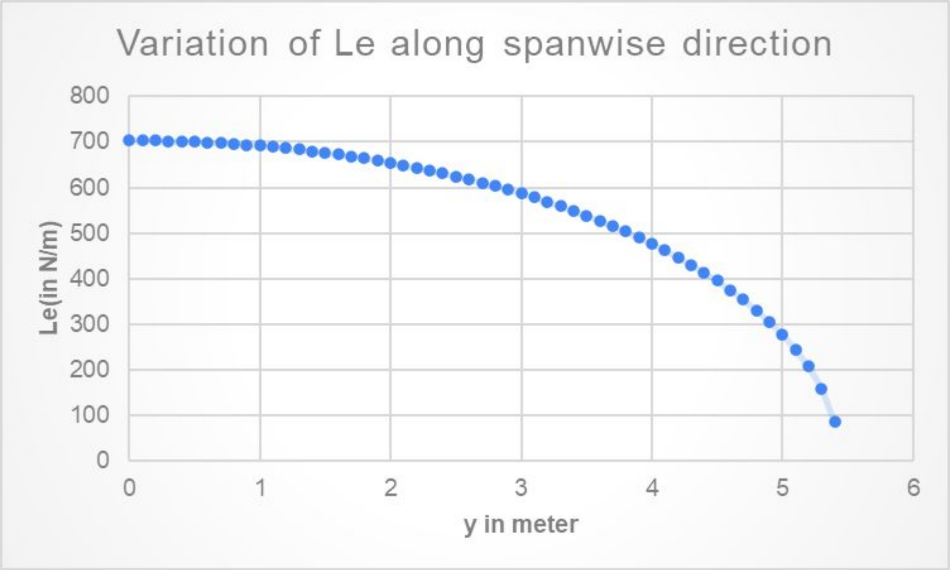
L_s =612.9974 kg ~ 612.842 kg = W

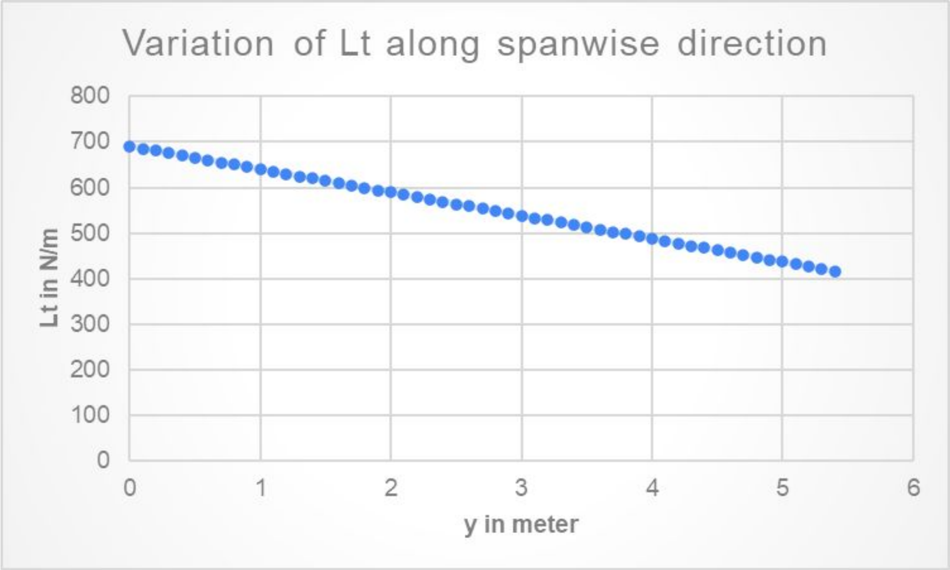
Hence, the total lift adds up to the weight of the aircraft.

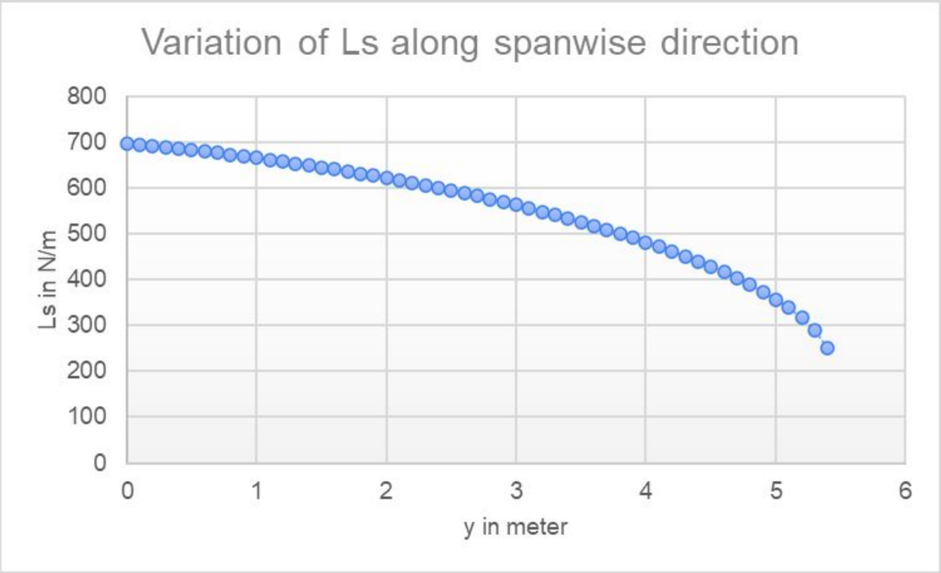
y(in m)	L(elliptic) N/m	L(trapezoidal) N/m	Ls	Section	y(avg for section)	Lift(Section) N
0	703.556	690.714	697.135	1	0.05	69.58355816
0.1	703.4371059	685.6352206	694.5361632	2	0.15	69.31772676
0.2	703.0803029	680.5564412	691.818372	3	0.25	69.03999086
0.3	702.4852284	675.4776618	688.9814451	4	0.35	68.75032622
0.4	701.6512764	670.3988824	686.0250794	5	0.45	68.44869638
0.5	700.5775936	665.3201029	682.9488483	6	0.55	68.1350524
0.6	699.2630759	660.2413235	679.7521997	7	0.65	67.80933263
0.7	697.7063619	655.1625441	676.434453	8	0.75	67.47146243
0.8	695.9058263	650.0837647	672.9947955	9	0.85	67.12135369
0.9	693.8595711	645.0049853	669.4322782	10	0.95	66.75890443
1	691.565415	639.9262059	665.7458104	11	1.05	66.38399823
1.1	689.0208819	634.8474265	661.9341542	12	1.15	65.99650355
continued						
5.1	244.7190071	431.69625	338.2076285	52	5.15	32.73922434
5.2	206.5362459	426.6174706	316.5768582	53	5.25	30.32746193
5.3	158.4060695	421.5386912	289.9723803	54	5.35	27.030432
5.4	84.81260766	416.4599118	250.6362597		5.45	

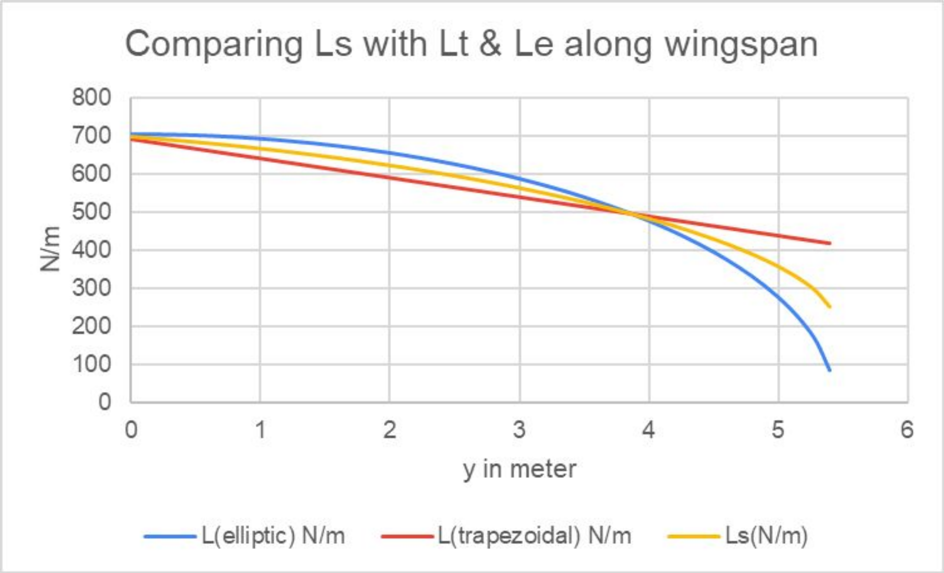
[Link for complete sheet]

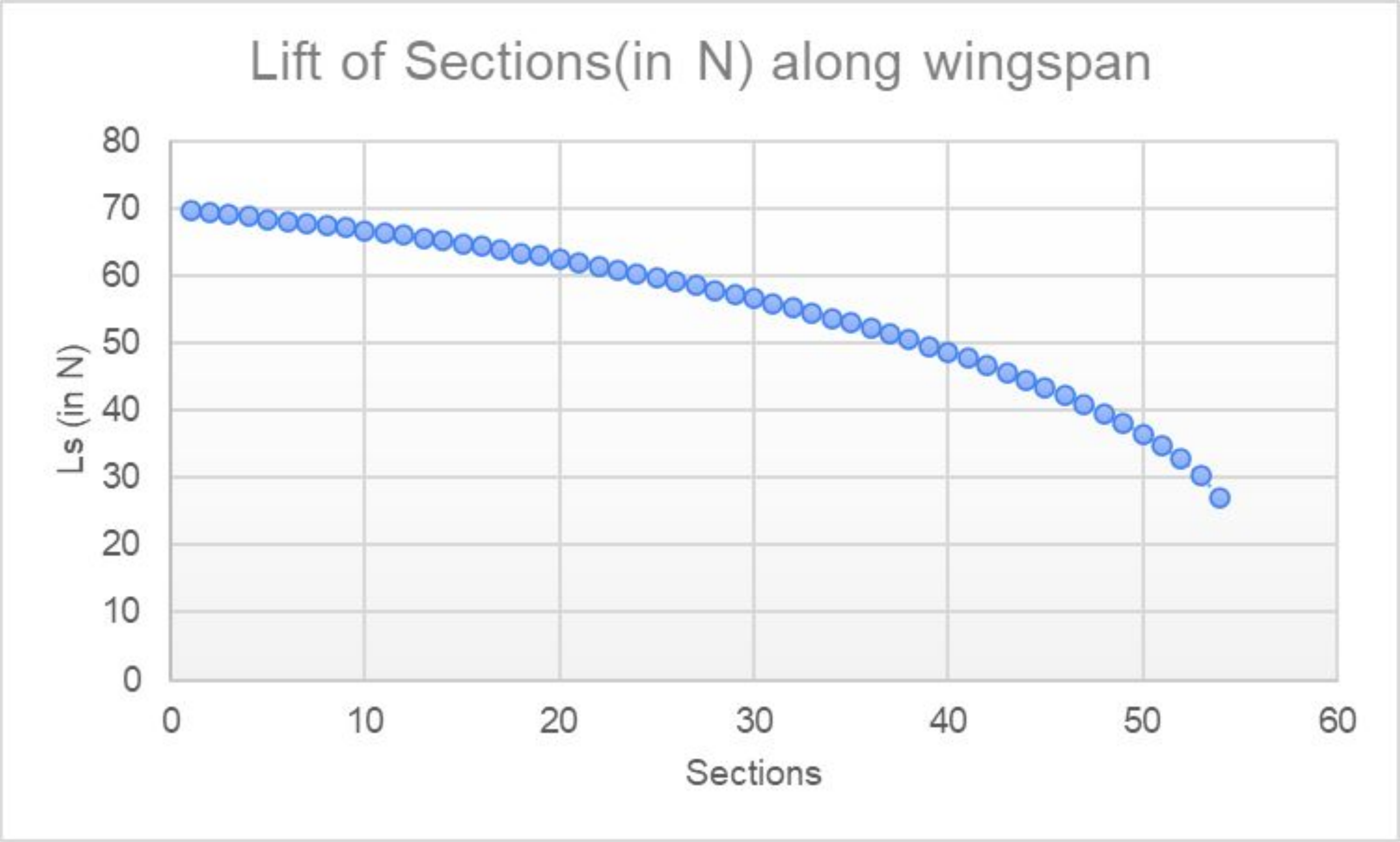
Plots


Variation of Le along spanwise direction
800
700
600
500
400
300
200
100
0
Le(in N/m)
0
1
2
3
4
5
6
y in meter


Variation of Lt along spanwise direction
800
700
600
500
400
300
200
100
0
Lt in N/m
0
1
2
3
4
5
6
y in meter


Variation of Ls along spanwise direction
800
700
600
500
400
300
200
100
0
Ls in N/m
0
1
2
3
4
5
6
y in meter


Comparing Ls with Lt & Le along wingspan
800
700
600
500
400
300
200
100
0
N/m
0
1
2
3
4
5
6
y in meter
L(elliptic) N/m
L(trapezoidal) N/m
Ls(N/m)


Lift of Sections(in N) along wingspan
80
70
60
50
40
30
20
10
0
Ls (in N)
0
10
20
30
40
50
60
Sections

2. Calculation of Gravity loads:

The Gravity load in an aircraft is the force of gravity experienced by the aircraft and its occupants during flight.

W (lbf)= Multiplier * Factor

Values of Multiplier(general aviation in this case) and Factor to be taken from chart below:

TABLE 10.4: Statistical weights of major components on various types of aircraft.

Component	Multiplier			Factor*
	Combat	Transport/ Bomber	General Aviation	
Main Wing	9.0	10.0	2.5	S_W
Horizontal Tail	4.0	5.5	2.0	S_W
Vertical Tail	5.3	5.5	2.0	S_W
Installed Engine	1.3	1.3	1.4	Uninstalled W_{engine}
Landing Gear	0.033 (Navy, 0.045)	0.043	0.057	W_{TO}
	(Navy, 0.045)			W_{TO}
Fuselage	4.8	5.0	1.4	$S_{fuse-wetted}$

***Note:** Areas have units of f^2 and weights have units of pounds.

Further, dead weight of wing structure, fuel/battery etc to be added

For the plane we designed in the course Aircraft Design-I,
Values of required parameters are as follows-

S_W = Wing Area = 9.48878 m^2

$W_{Electric\ Engine(motor)}$ = Weight of Uninstalled Motor= 23 kg

From historical data of E-811 (the first certified electric aircraft motor by EASA on May 18, 2020)

W_{To} = 612.842 kg

$S_{fuse\text{-}wetted}$ = Fuselage Wetted Area (the area which is in contact with the external airflow)= 10410000 mm^2= 10.41 m^2

Total Weight of these components= 368.18 kg
Dead Weight of Wing Structure:
(The wing weight represents about 22-27% of the empty weight)
For our Aircraft, W_{empty}= 356.75 kg
Hence, Dead W_{wing}=27% of 356.75kg= 96.32 kg
Weight of LIB Batteried(calculated in AE461)=142 kg
Total Gravity Loads= 368.2 +96.32 +142 = 606.52 kg
Total takeoff Weight= 612.842 kg

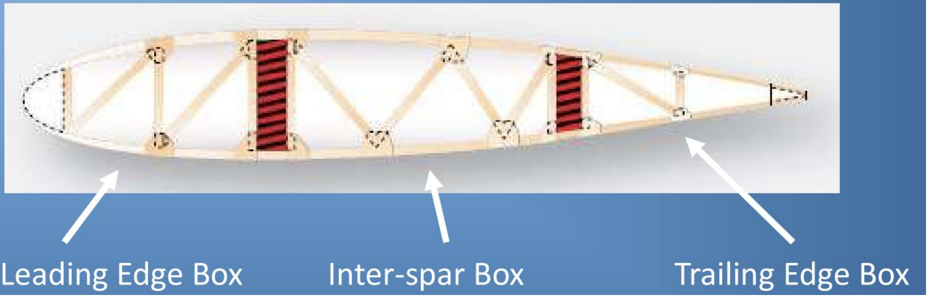
Hence, Total Gravity Loads= 606.52 kg ~ Total Weight

Component	Multiplier	Factor	Factor(corrected*)	Weight(lbf)	Weight(kg)
Main Wing	2.5	9.49 m^2	102.11 f^2	255.275	115.79
Horizontal Tail	2.0	9.49 m^2	102.11 f^2	204.22	92.63
Vertical Tail	2.0	9.49 m^2	102.11 f^2	204.22	92.63
Installed Engine	1.4	23 kg	23 kg	-	32.2
Landing Gear	0.057	612.842 kg	612.842 kg	-	34.93
Fuselage	1.4	10.41 m^2	112.01 f^2	156.814	71.13

*m^2 converted to f^2

Exercise 3: Preliminary design of wing box

- Preliminary Sizing considering only inter-spar box (the main load carrying portion of the wing)



In this exercise first we need to choose a spar cross section, material for spars and ribs and then calculate the variation of wing skin thickness & spar web thickness along wingspan using SF, BM and Torsion

[Excel sheet for Ex3]

Choosing a Material

Material taken: Aluminium Alloy 7075

G = 71.7 GPa

Max stress: 489.52 MPa

Torsional Shear Stress (fts) : 331 MPa

Reason- It can support stresses that are produced during high altitude flights

Choosing spar cross section

As discussed in class, we would be only considering inter-spar box since it carries the main portion of load.

Assuming front spar at 20% of the chord and rear spar at 60% of the chord, we calculated the interspar distance as 0.4*chord

To calculate height of box, we divided the cross section area of box by interspar length.

We assumed rib-spacing as 50 cm or 19.68 inches

y(in m)	chord length(m)	Interspar distance(w) (in m)	height of wing box (h) (in m)
0	1.1288365	0.4515346	0.1245406553
0.1	1.1205095	0.4482038	0.123621966
0.2	1.1121825	0.444873	0.1227032767
0.3	1.1038555	0.4415422	0.1217845874
0.4	1.0955285	0.4382114	0.1208658981
0.5	1.0872015	0.4348806	0.1199472087
0.6	1.0788745	0.4315498	0.1190285194
0.7	1.0705475	0.428219	0.1181098301
0.8	1.0622205	0.4248882	0.1171911408
0.9	1.0538935	0.4215574	0.1162724515
1	1.0455665	0.4182266	0.1153537621
1.1	1.0372395	0.4148958	0.1144350728
continued			
5.1	0.7124865	0.2849946	0.07860618932
5.2	0.7041595	0.2816638	0.0776875
5.3	0.6958325	0.278333	0.07676881068
5.4	0.6875055	0.2750022	0.07585012136

Calculation of Limit load distribution

We will calculate the limit load in each section,

For a section

$q(y)=L_s(\text{section})-W_t(\text{section})$

L_s(section)= lift of section calculated using Schrenk's approximation

W_t(section)=Weight of wing calculated considering trapezoidal wing

$$W_t(y)=\frac{2*W}{b(1+\lambda)} * \left[1-\frac{2y}{b}(1-\lambda)\right]$$

W=96 kg

b=10.88

λ=0.6

$W_t(y)=108.198 * [1- y/13.6]$

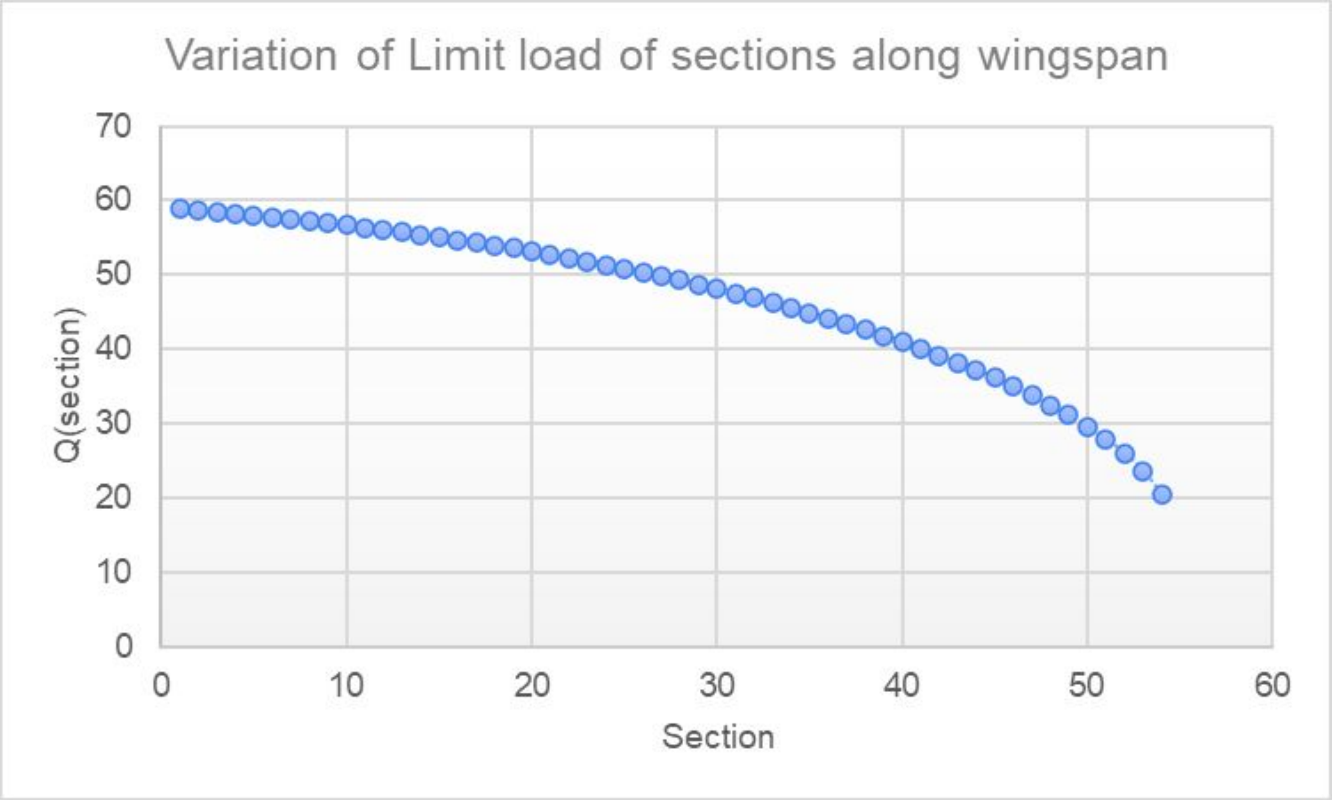
First we calculated $W_t(y)$, then we calculated W_t for each section by taking avg of W_t at two extreme y of that section

$W_t(\text{section})=0.5*(W_t(y_1)+W_t(y_1+0.1))$

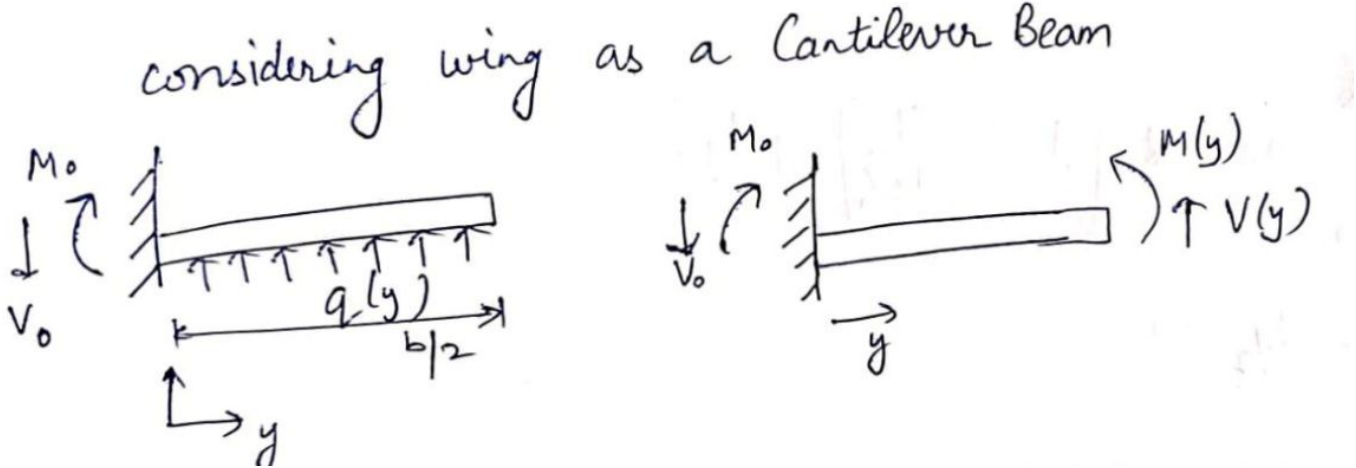
In Excel we have calculated limit load for all sections

Section	Lift(Section) N	Wt (section) N	Q (section)(in N)
1	69.58355816	10.78002132	58.80353684
2	69.31772676	10.70046397	58.61726279
3	69.03999086	10.62090662	58.41908424
4	68.75032622	10.54134926	58.20897696
5	68.44869638	10.46179191	57.98690447
6	68.1350524	10.38223456	57.75281784
7	67.80933263	10.30267721	57.50665543
8	67.47146243	10.22311985	57.24834257
9	67.12135369	10.1435625	56.97779119
10	66.75890443	10.06400515	56.69489928
11	66.38399823	9.984447794	56.39955044
12	65.99650355	9.904890441	56.0916131
continued			
52	32.73922434	6.722596324	26.01662801
53	30.32746193	6.643038971	23.68442296
54	27.030432	6.563481618	20.46695039

Plots


Variation of Limit load of sections along wingspan
Q(section)
Section
0
10
20
30
40
50
60
70

Calculation of Shear force distribution



We are idealising the wing as a cantilever beam for all calculations in this report

Calculation of Shear force distribution

Shear force $V(y) = V_O - \int_0^y q(y)\,dy$ where, $V_O = \int_0^{b/2} q(y)\,dy$

We have calculated the expression for q(y) earlier

$$q(y)=351.778*\sqrt{1-\frac{y^2}{29.59}} + 236.439\left(1-\frac{y}{13.6}\right)$$

Integrating it from 0 to b/2, we get V_O

Hence V_O=2528 N

Now putting that value in V(y) expression we can get shear force distribution

Calculations are done in excel.(Integrated function is written as formula)

$$\int_0^y q(y)\,dy=\left(5.44*\sin^{-1}\left(\frac{y}{5.44}\right)+y*\sqrt{\frac{29.59-y^2}{29.59}}\right)*175.889 + \left(y-\frac{y^2}{27.2}\right)*236.439$$

OR(using sections)

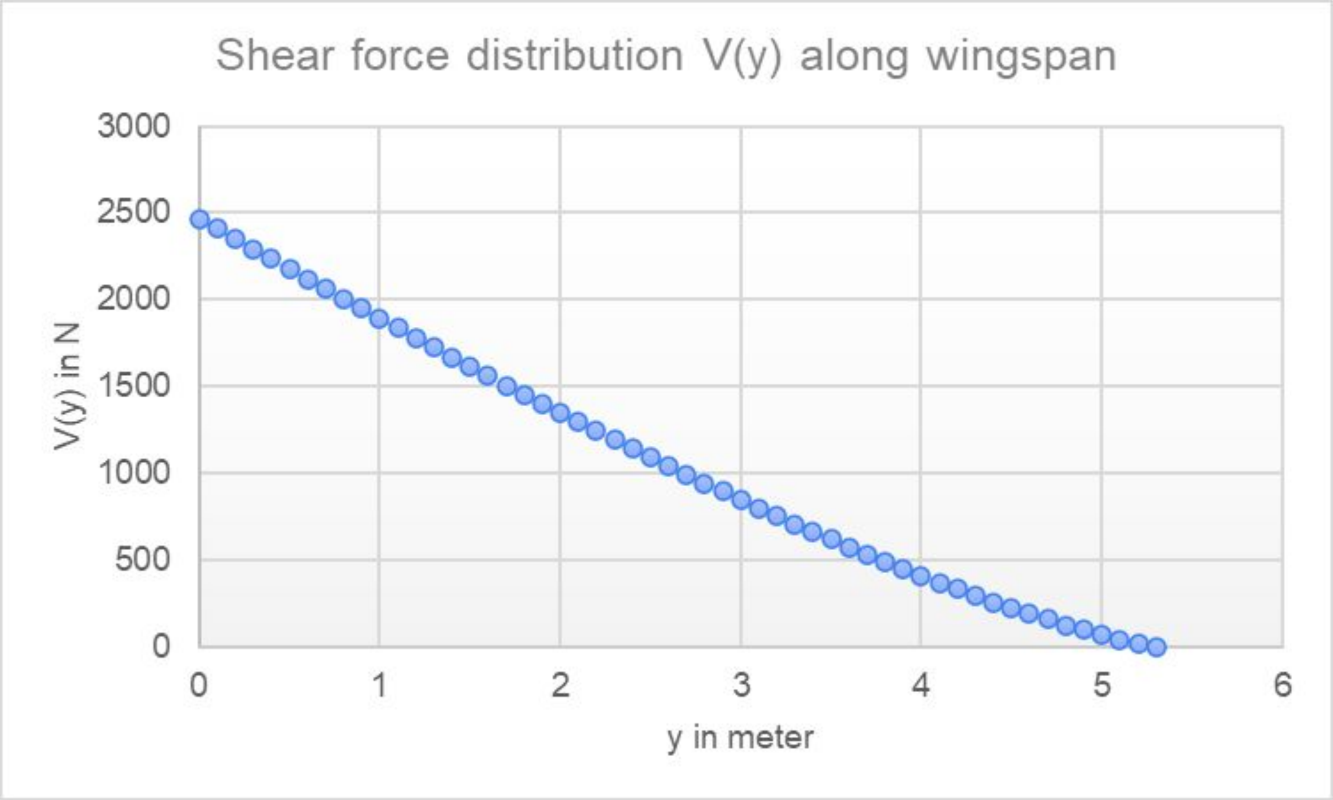
To calculate the V(y) at distance y from root chord, we can simply add the q(sections) of all the sections left to the y and then subtract that value from VO

We have done calculations using both methods and found that V(y) comes out to be the same.

Calculation of Shear force distribution

y(in m)	V(y) N
0	2468.964674
0.1	2410.347411
0.2	2351.928327
0.3	2293.71935
0.4	2235.732446
0.5	2177.979628
0.6	2120.472972
0.7	2063.22463
0.8	2006.246839
0.9	1949.551939
1	1893.152389
1.1	1837.060776
continued	
5.1	70.16800172
5.2	44.15137371
5.3	20.46695075
5.4	0.000000365469532

Plots


Shear force distribution V(y) along wingspan
3000
2500
2000
1500
1000
500
0
V(y) in N
0
1
2
3
4
5
6
y in meter

Calculation of Bending Moment

$$M(y) = M_O - \int_0^y q(y)*y\,dy, \text{ where } M_O = \int_0^{b/2} q(y)*y\,dy$$

$$M_O = \int_0^{b/2} 351.778*y*\sqrt{1 - \frac{y^2}{29.59}} + 236.439*y*\left(1 - \frac{y}{13.6}\right)$$

Integrating it from 0 to b/2, we get M_O

Hence M_O=6000 N

Now putting that value in M(y) expression we can get moment distribution

Calculations are done in excel.(Integrated function is written as formula)

$$\int_0^y q(y)*y\,dy = -3435*\left(1 - \frac{y^2}{29.59}\right)^{1.5} + 236.439*\left(\frac{y^2}{2} - \frac{y^3}{40.8}\right)$$

OR(using sections)

Similar approach to V(y), here to calculate moment of a section, we take cumulative sum of

$\sum_0^s q(s)*y(\text{section\_avg})$ of all the sections left to the y, and subtract it from M_O

Both Methods give the same result, can be seen in excel.

Calculation of Bending Moment

Since we have chosen the rib spacing as 50 cm, there will be 5 sections between two consecutive ribs.
In total there will be 10 ribs for half wingspan.

We need to know M(y) only at the cross sections of these ribs.

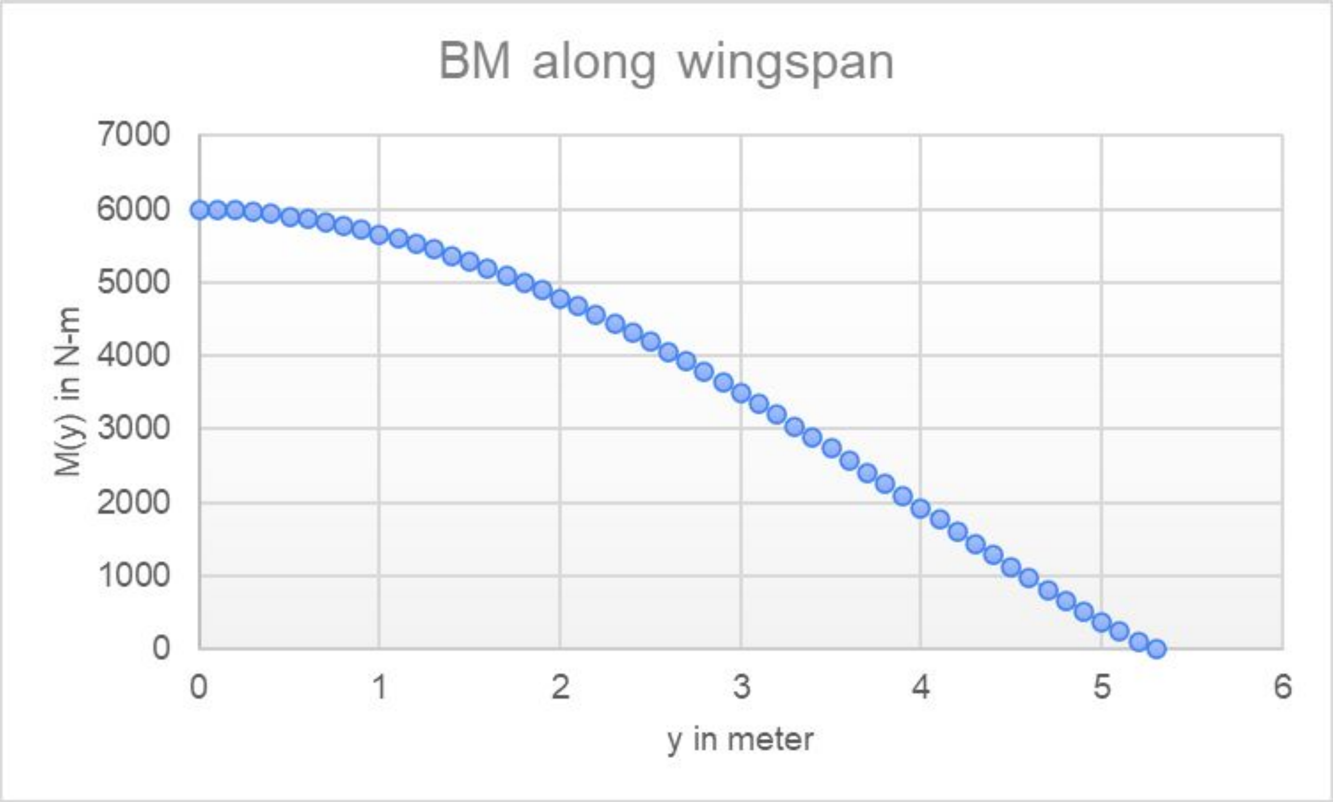
We have done calculation of M(y) at rib using section method which has been discussed on previous slides

Calculation of Bending Moment

y(in m)	M(y) in N-m
0	6001.481063
0.1	5992.688474
0.2	5978.083703
0.3	5957.710561
0.4	5931.616454
0.5	5899.852404
0.6	5862.473078
0.7	5819.536821
0.8	5771.105698
0.9	5717.245544
1	5658.026016
1.1	5593.520661
continued	
5.1	367.8270382
5.2	233.8414039
5.3	109.4981834
5.4	0

Ribs	Rib(location)in m	M(ribs) in N-m
1	0.5	5931.616454
2	1	5717.245544
3	1.5	5369.085009
4	2	4897.363147
5	2.5	4315.05845
6	3	3638.411546
7	3.5	2887.807307
8	4	2089.397641
9	4.5	1278.513371
10	5	508.999989

Plots



BM along wingspan
7000
6000
5000
4000
3000
2000
1000
0
M(y) in N-m
0
1
2
3
4
5
6
y in meter

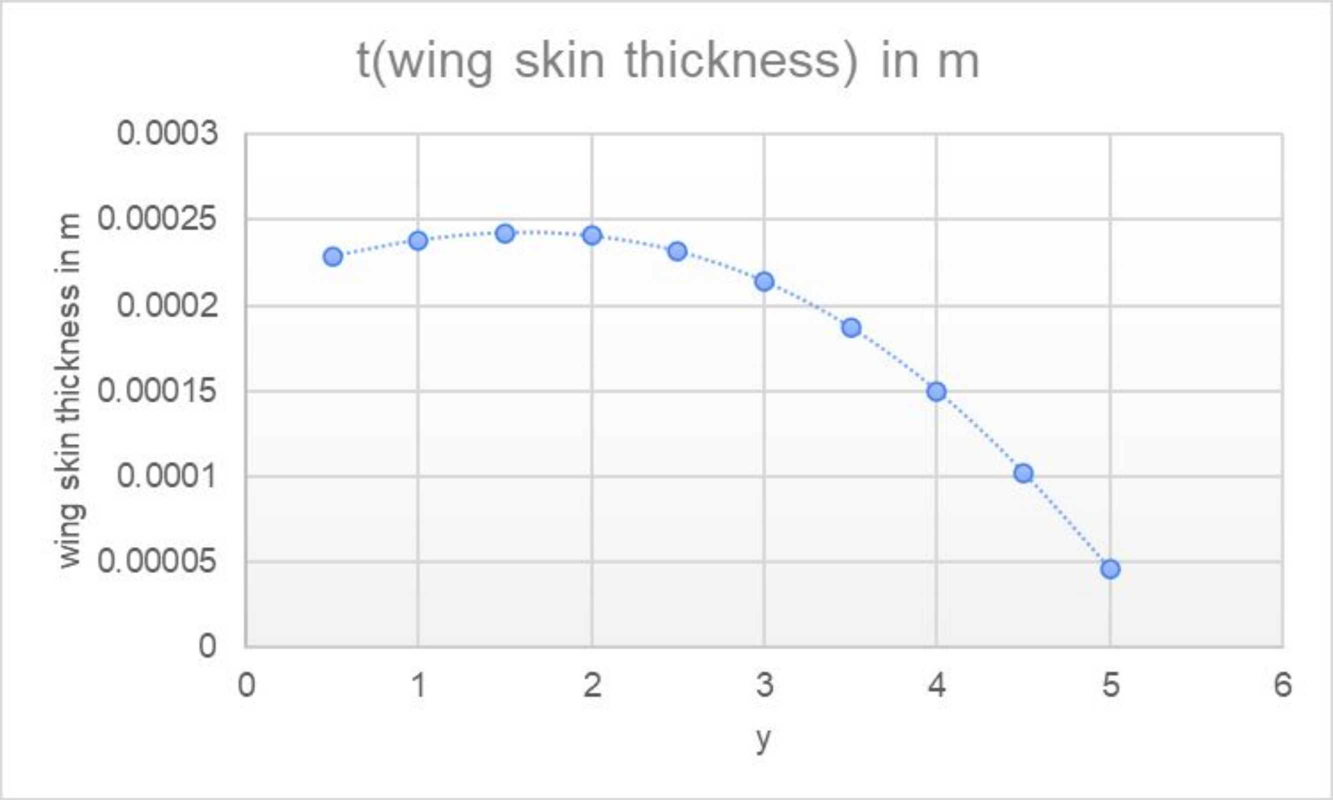
Calculation of Wing Skin thickness

$$t = \frac{M}{w * h * \sigma(tu)}$$

Where
t=wing skin thickness(in m)
M=Bending Moment at rig cross section
w=width of spar(in m)
h=height of spar(in m)
σ_{tu}=ultimate tensile strength of material =489.52 MPa

Rib(location)in m	t (wing skin thickness) in m
0.5	0.0002287784779
1	0.000238277403
1.5	0.0002425531287
2	0.0002406302212
2.5	0.0002314490873
3	0.0002138986055
3.5	0.0001868985855
4	0.0001495919642
4.5	0.000101806079
5	0.00004534834213

Plots


t(wing skin thickness) in m
wing skin thickness in m
0.0003
0.00025
0.0002
0.00015
0.0001
0.00005
0
0
1
2
3
4
5
6
y

Calculation of Torsion

Since the weight is acting on CG but the lift is acting at c/4, torque will be generated.
First we calculated the lift force acting at a cross section at distance y, and then multiplied it with the distance between cg and AC.
That Lead us to,

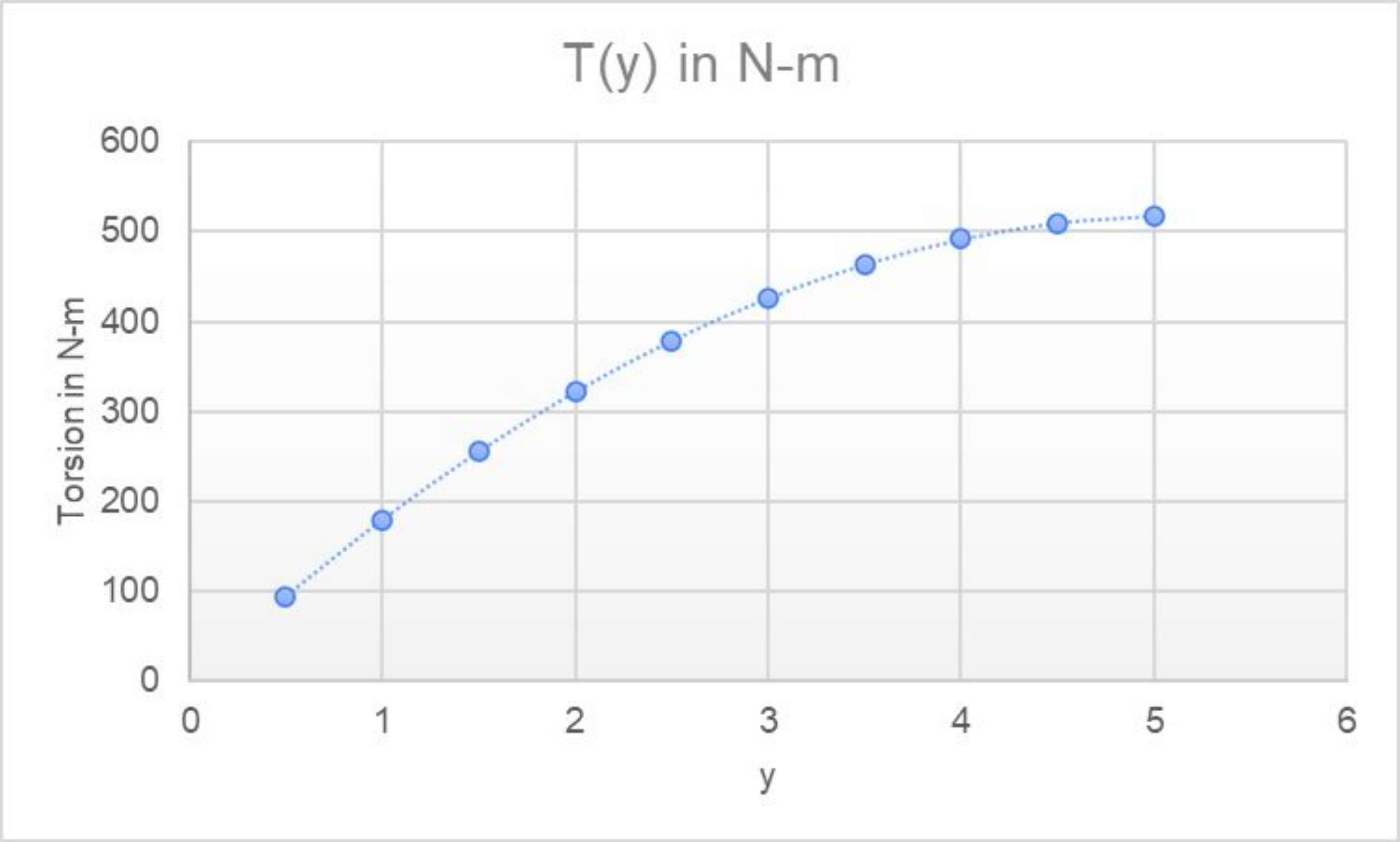
We are calculating Torsion in the cross section of a rig,
For a rig cross section, the weight is acting at CG of cross section but the lift is acting at AC(here c/4) Hence Torsion will be generated.

We will take the leading edge of airfoil as our reference to measure the CG and AC of cross section.
Torque about CG,
$T=L*(x_{CG}- x_{AC})$

Here L=Lift in Newton generated by all the sections left to the y $_{\text{(cross section of rig)}}$

Calculation for all rigs is has been done in excel

Plots



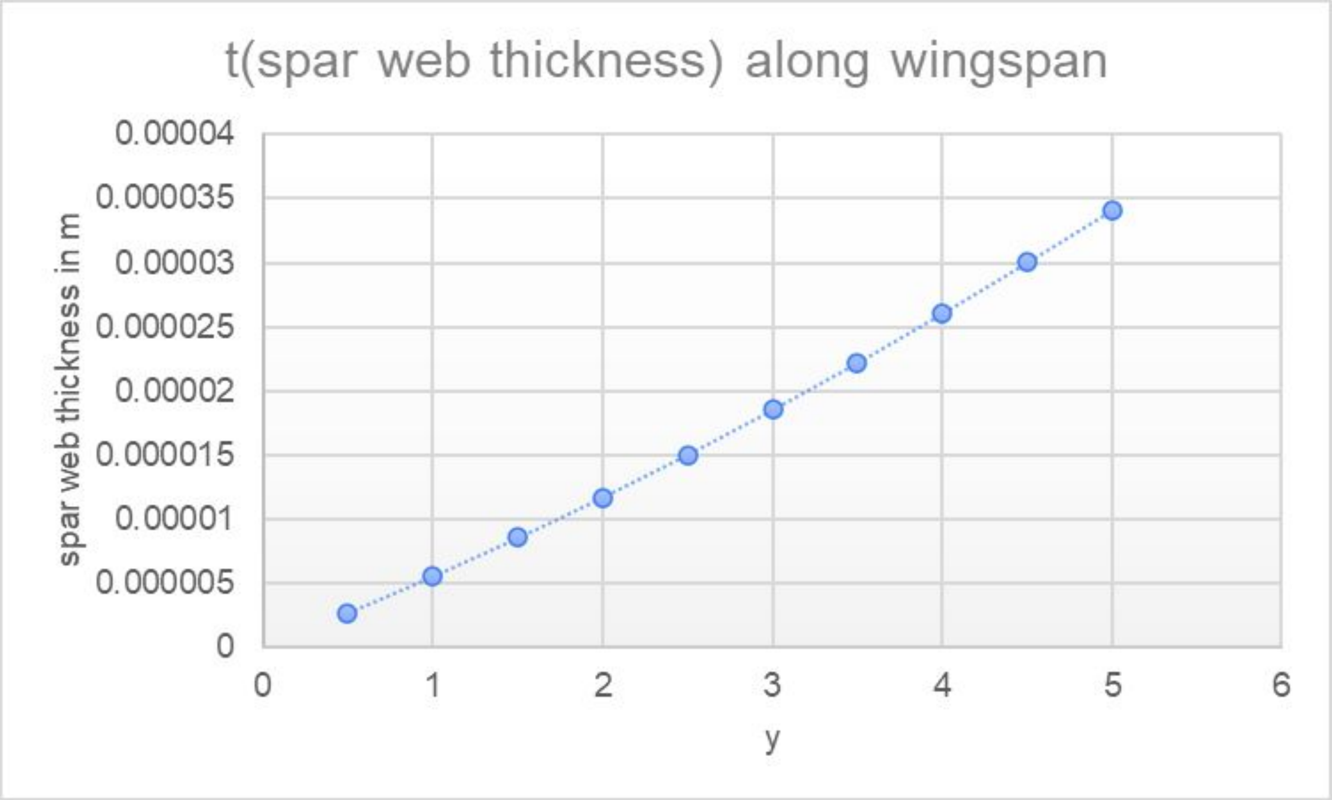
Calculation of Spar web thickness

$$f = \frac{T}{2a*b*t}$$

Where
t=Spar web thickness(in m)
T=Torsion at rig cross section
b=width of spar(in m)
a=height of spar(in m)
f=Min(Tensile strength, compressive strength)=331 MPa

Rib(location)in m	T(y) in N-m	t(spar web thickness) in m
0.5	94.52775834	0.000002695962921
1	179.8038226	0.000005541245411
1.5	255.6846667	0.000008541295277
2	322.0702632	0.00001170175337
2.5	378.8951008	0.00001502796691
3	426.1172236	0.00001852412366
3.5	463.7018823	0.0000219162542
4	491.5917106	0.000260258047
4.5	509.6391972	0.00003000846013
5	517.3978541	0.00003408636699

Plots



t(spar web thickness) along wingspan
0.00004
0.000035
0.00003
0.000025
0.00002
0.000015
0.00001
0.000005
0
spar web thickness in m
0
1
2
3
4
5
6
y

AE462

Exercise 4

__Date__
11/03/2024

__Instructor__
Prof. G.M. Kamath

__Group 11__
Aryaman Badkul
Palmate Aditya Tukaram
Shrutikirti Singh
Prabuddha Singh
Shikha Sharma
Rahangdale Prachi Sanjay

Content

Procedure Flowchart:

1. Pre-processing Part
- Modelling
- Part defining(material used)
- Assembly
- Application of Loads & Boundary Conditions
- See Results Obtained

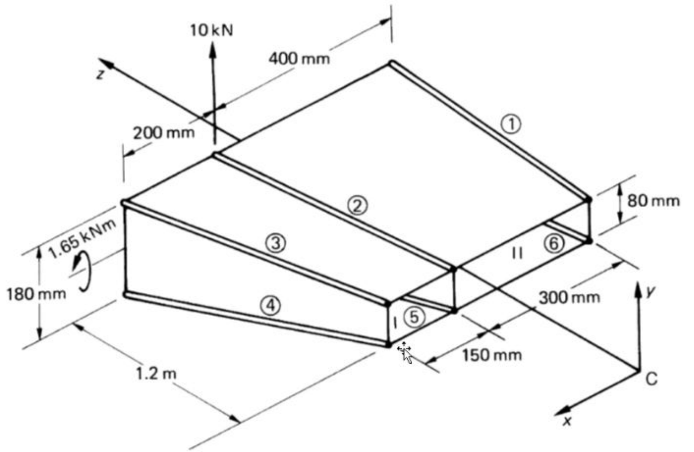
2. Post-processing Part
- Multiple Files will get generated
- Use .odp file to see the Results

Problem Statement

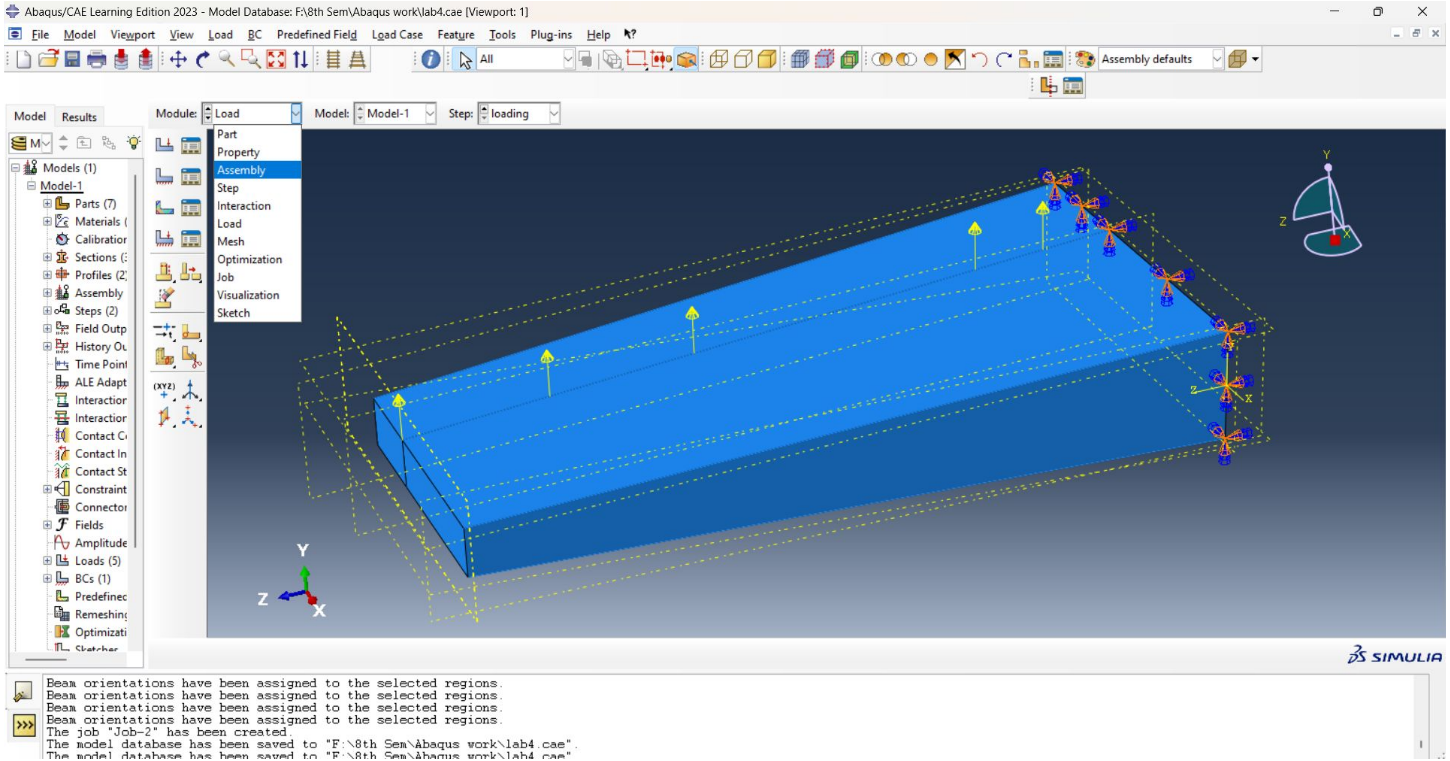
Example 23.4

A two-cell beam has singly symmetrical cross-sections 1.2 m apart and tapers symmetrically in the y direction about a longitudinal axis (Fig. 23.12). The beam supports loads which produce a shear force $S_y = 10$ kN and a bending moment $M_x = 1.65$ kN m at the larger cross-section; the shear load is applied in the plane of the internal spar web. If booms 1 and 6 lie on a plane which is parallel to the yz plane, calculate the forces in the booms and the shear flow distribution in the walls at the larger cross-section. The booms are assumed to resist all the direct stresses while the walls are effective only in shear. The shear modulus is constant throughout, the vertical webs are all 1.0 mm thick, while the remaining walls are all 0.8 mm thick:

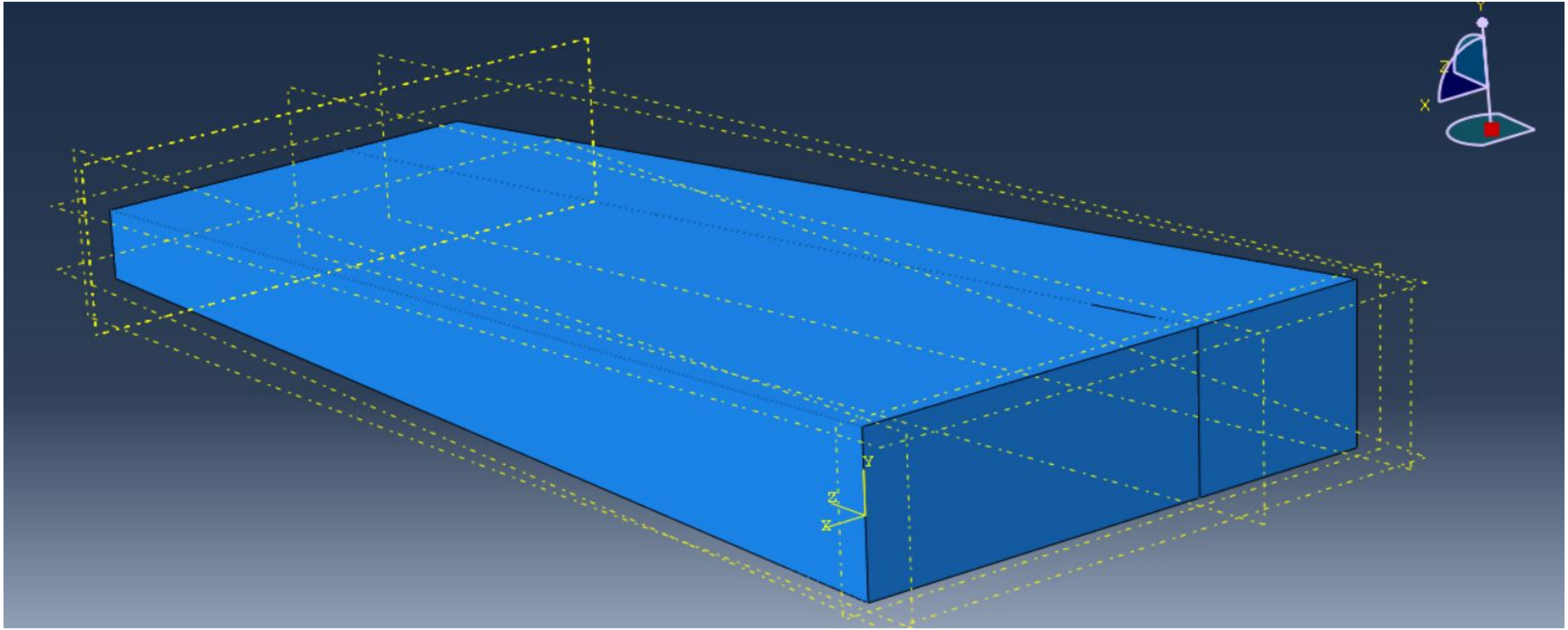
$$\text{Boom areas: } B_1 = B_3 = B_4 = B_6 = 600 \text{ mm}^2, \; B_2 = B_5 = 900 \text{ mm}^2$$



Link for CAE file- [Lab4_Group11]()
Abaqus Workspace

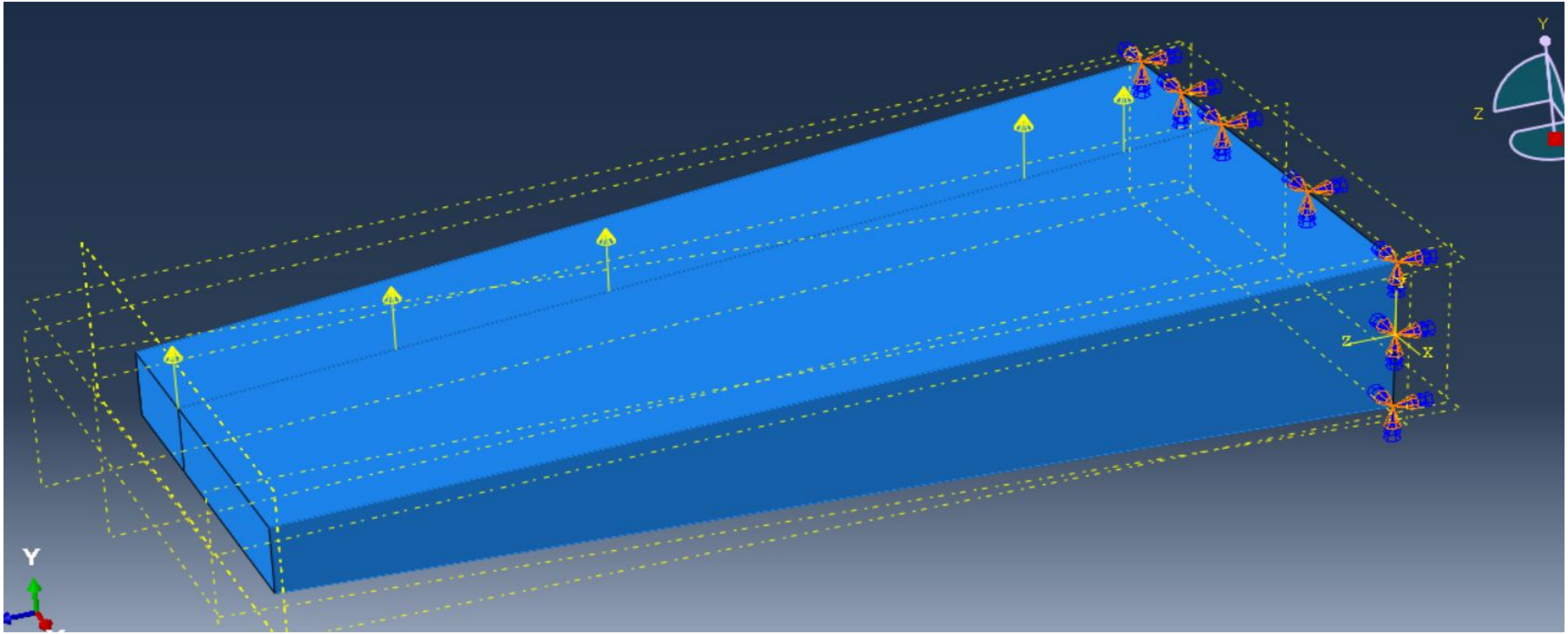


Model of a Two-Cell Beam

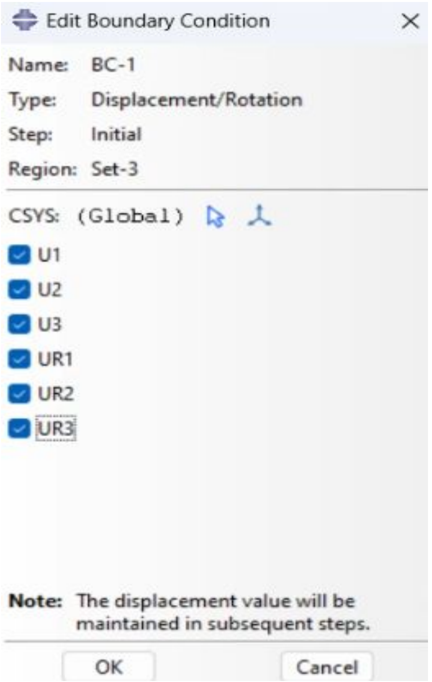


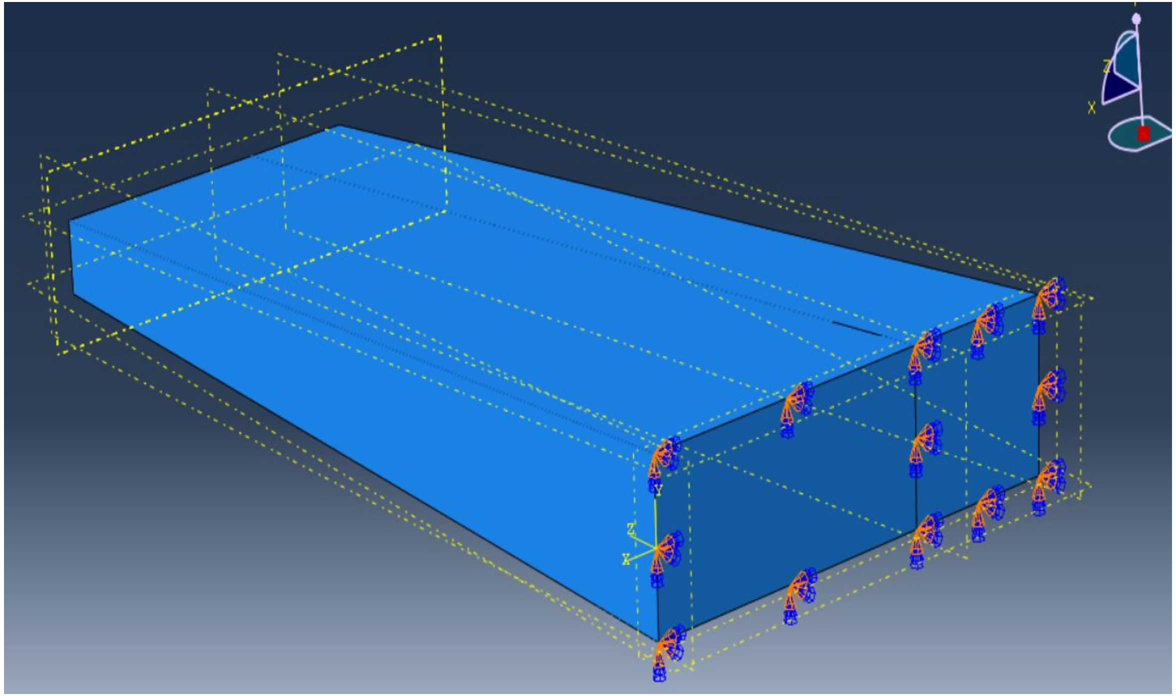
Loads Applied along with Boundary Conditions

Distance from root	Force(KN)
0.1	6
0.2	3.5
0.6	0.3
0.8	0.15
1	0.05

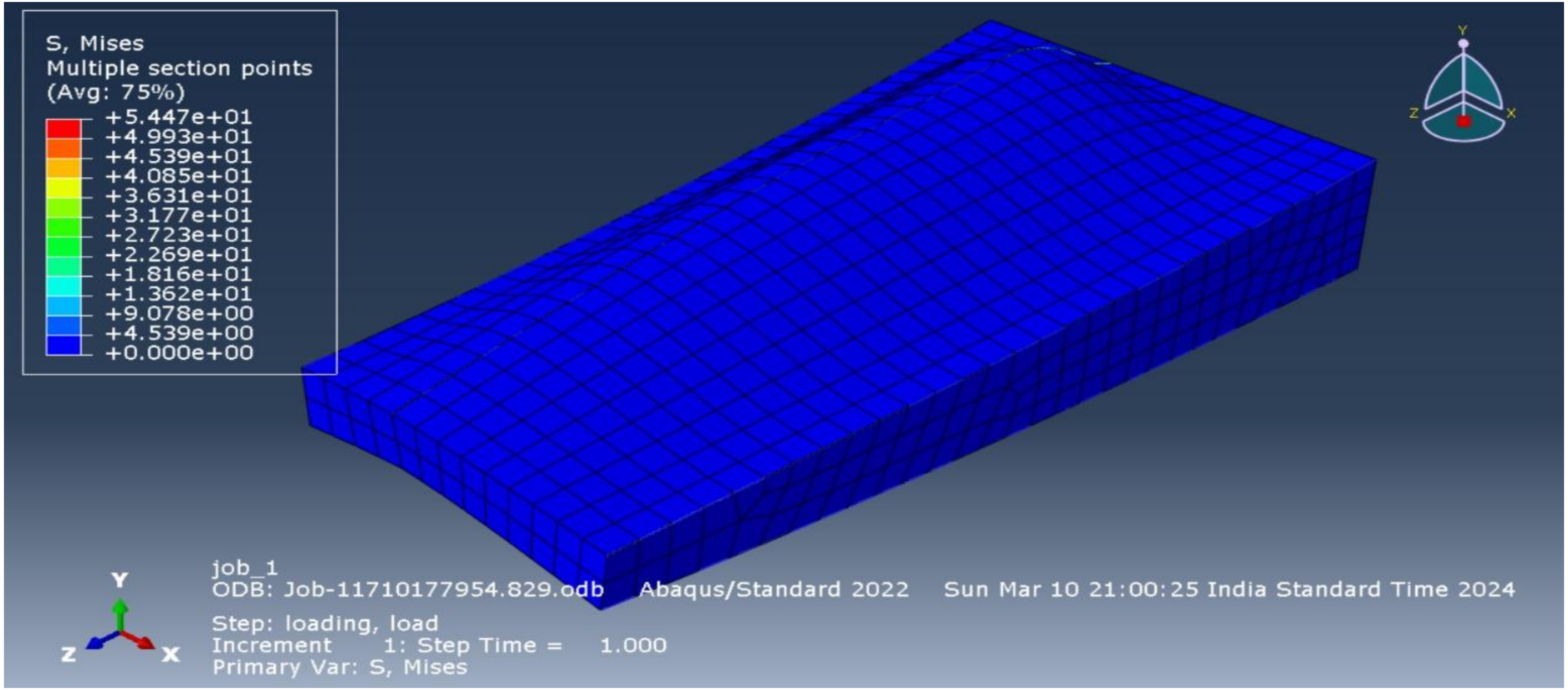


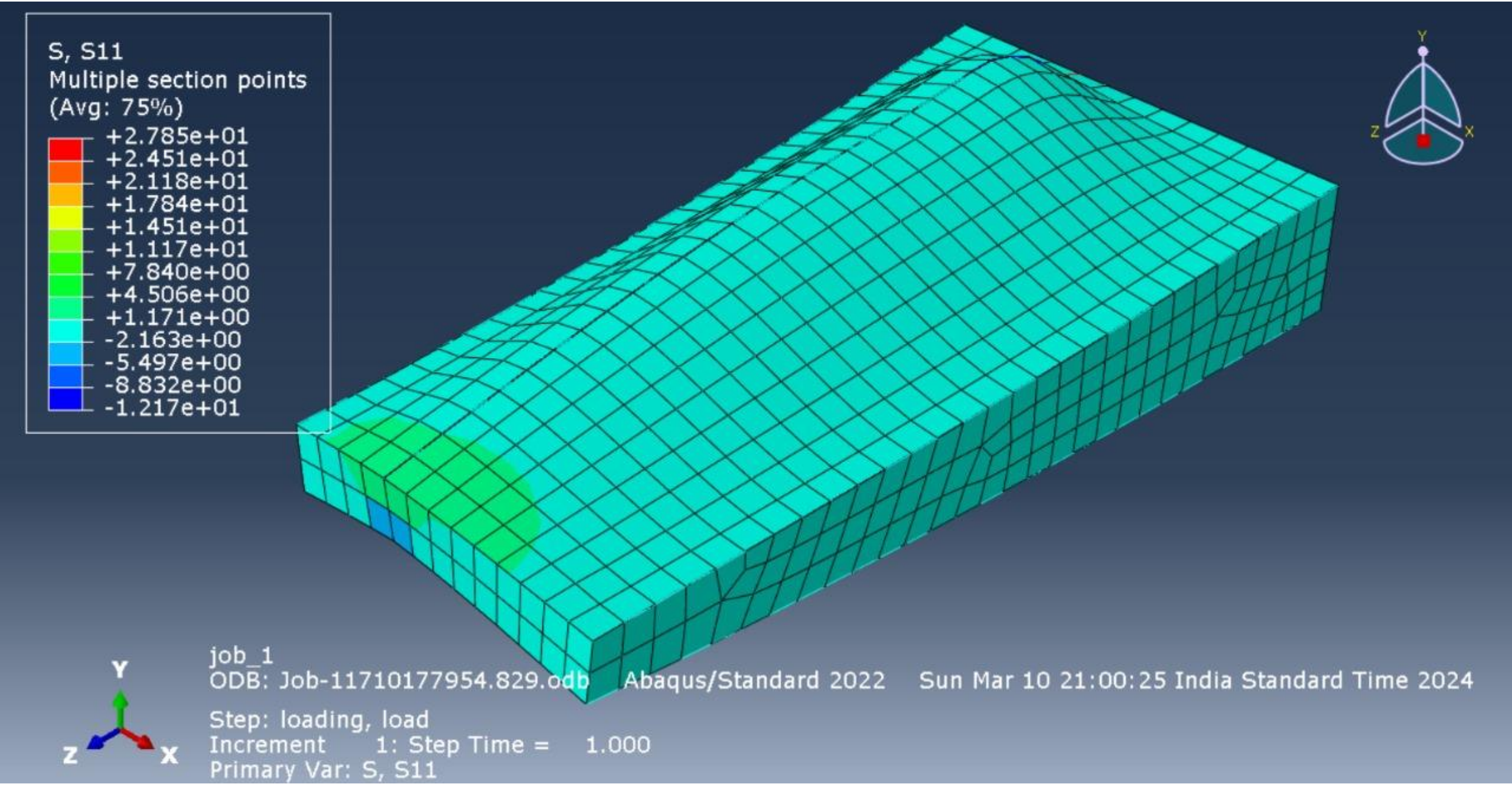
Boundary Conditions:

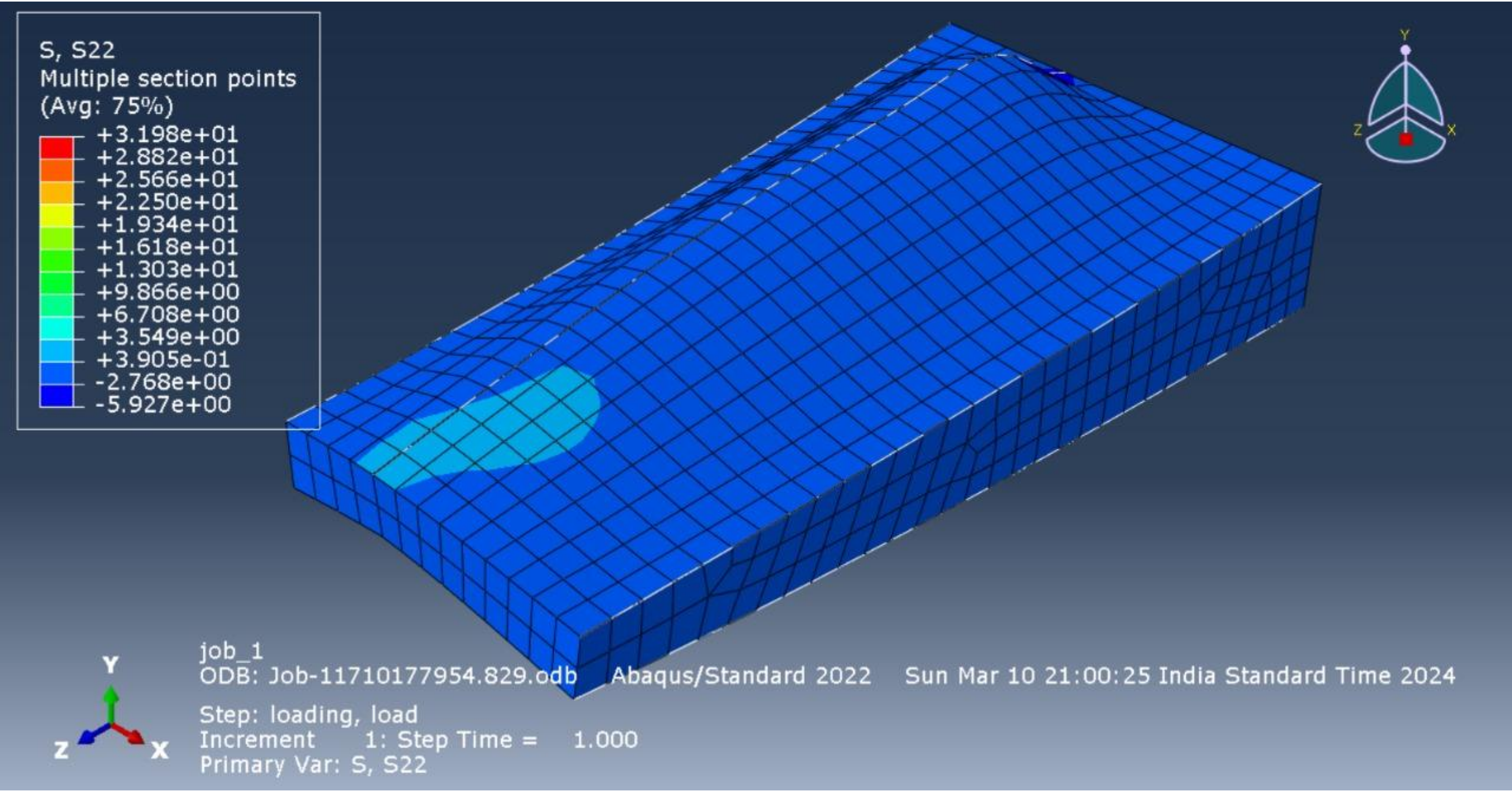


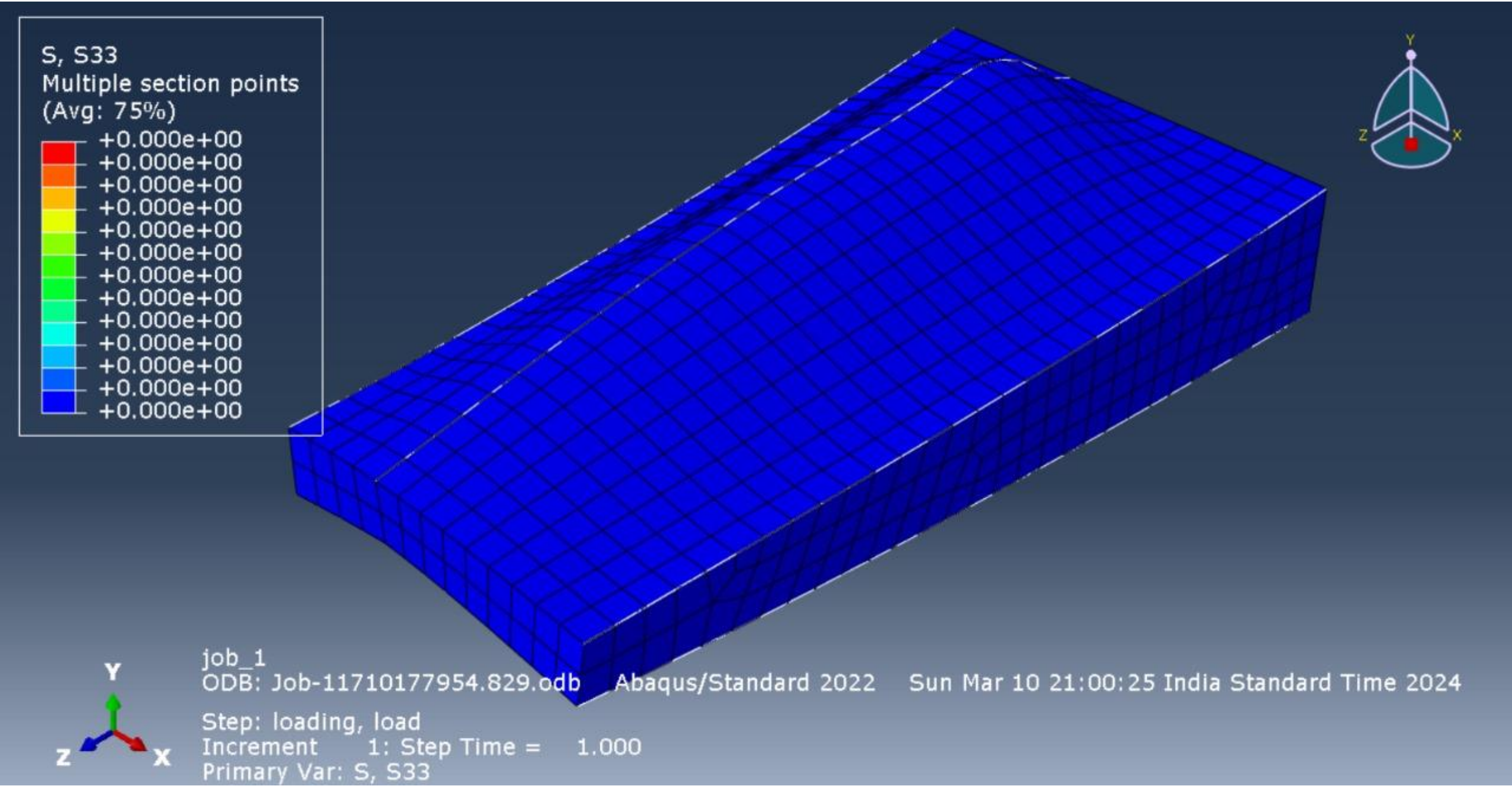


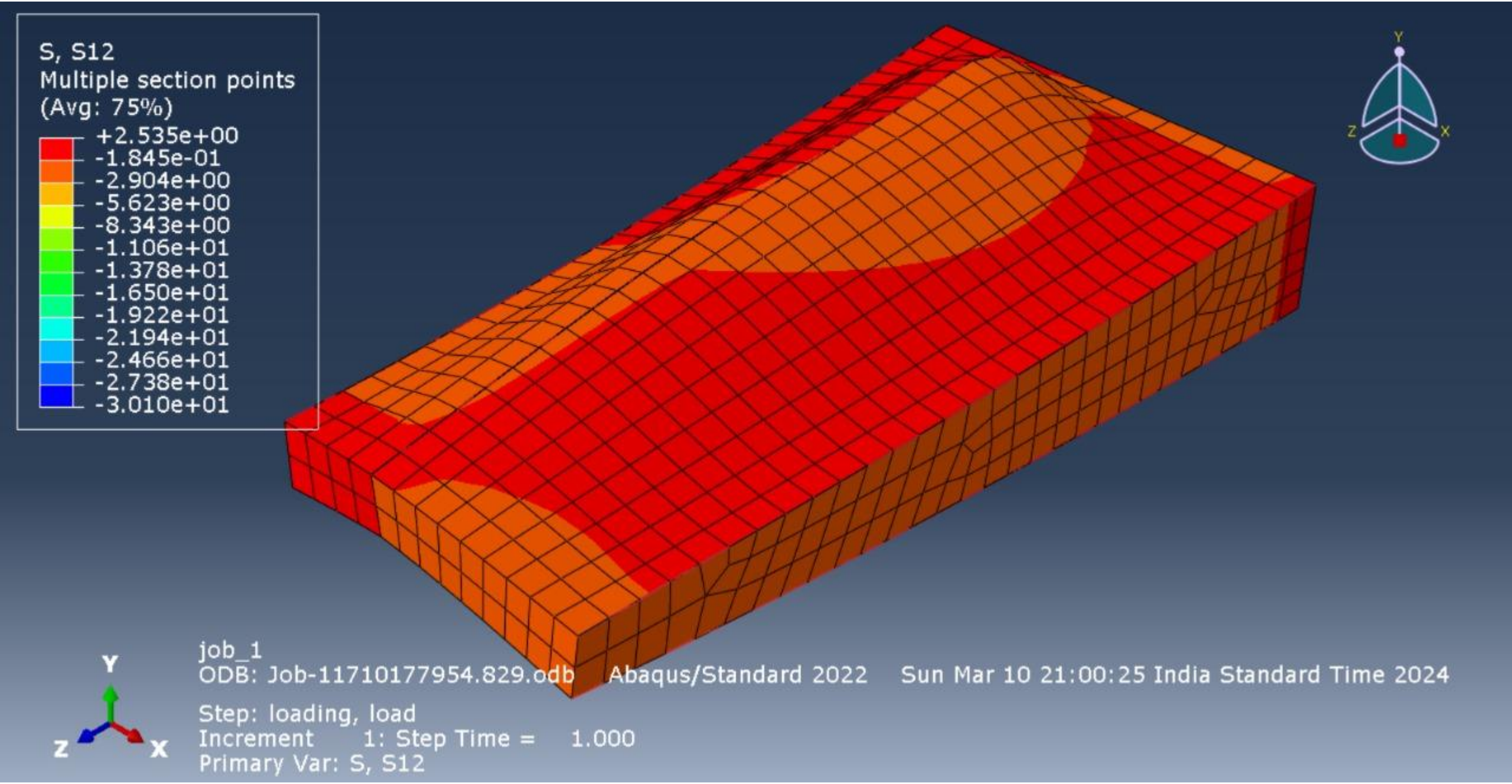
Stress Analysis:



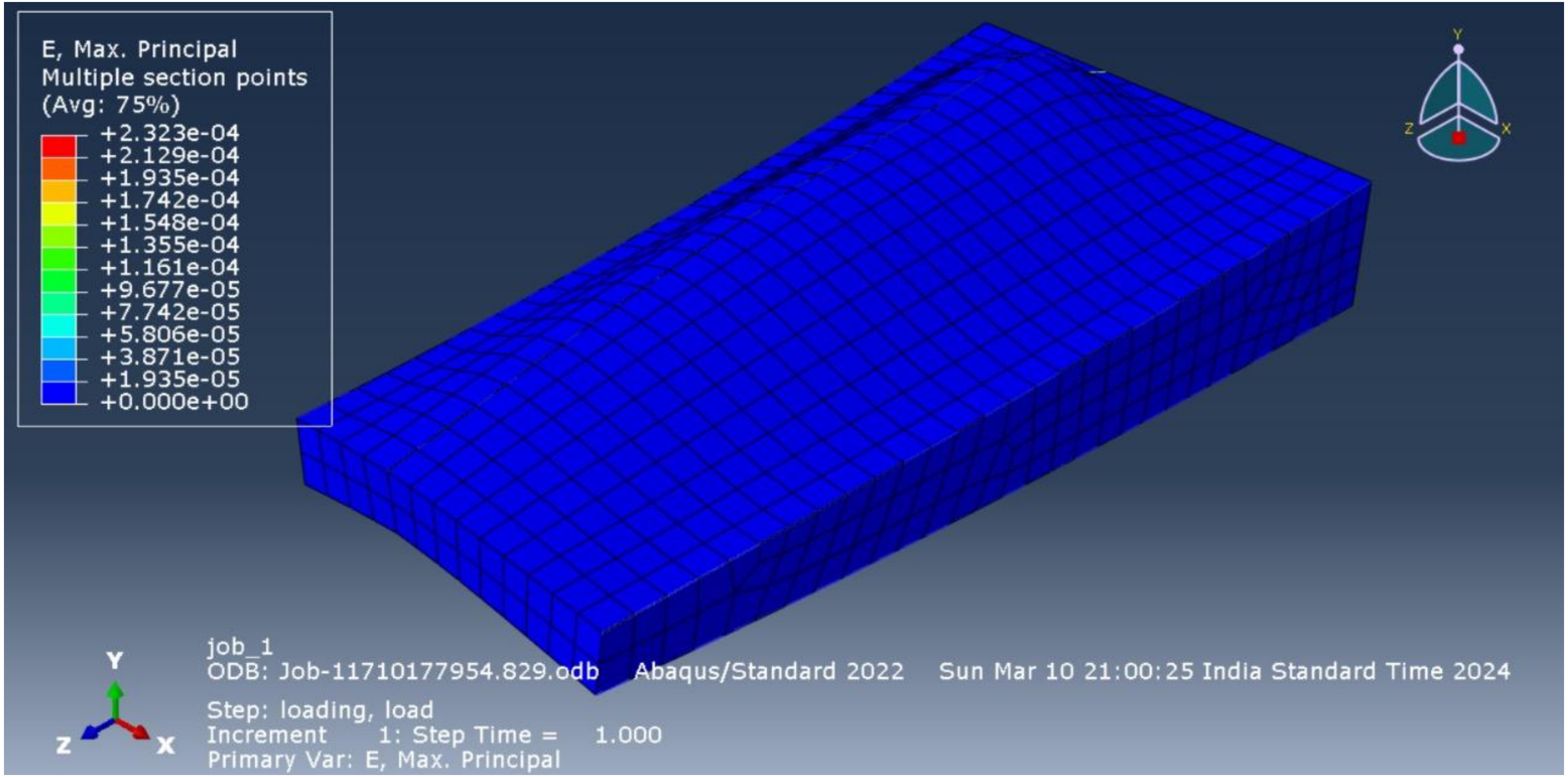

S, S11
Multiple section points
(Avg: 75%)
+2.785e+01
+2.451e+01
+2.118e+01
+1.784e+01
+1.451e+01
+1.117e+01
+7.840e+00
+4.506e+00
+1.171e+00
-2.163e+00
-5.497e+00
-8.832e+00
-1.217e+01
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: S, S11
Y
Z
X

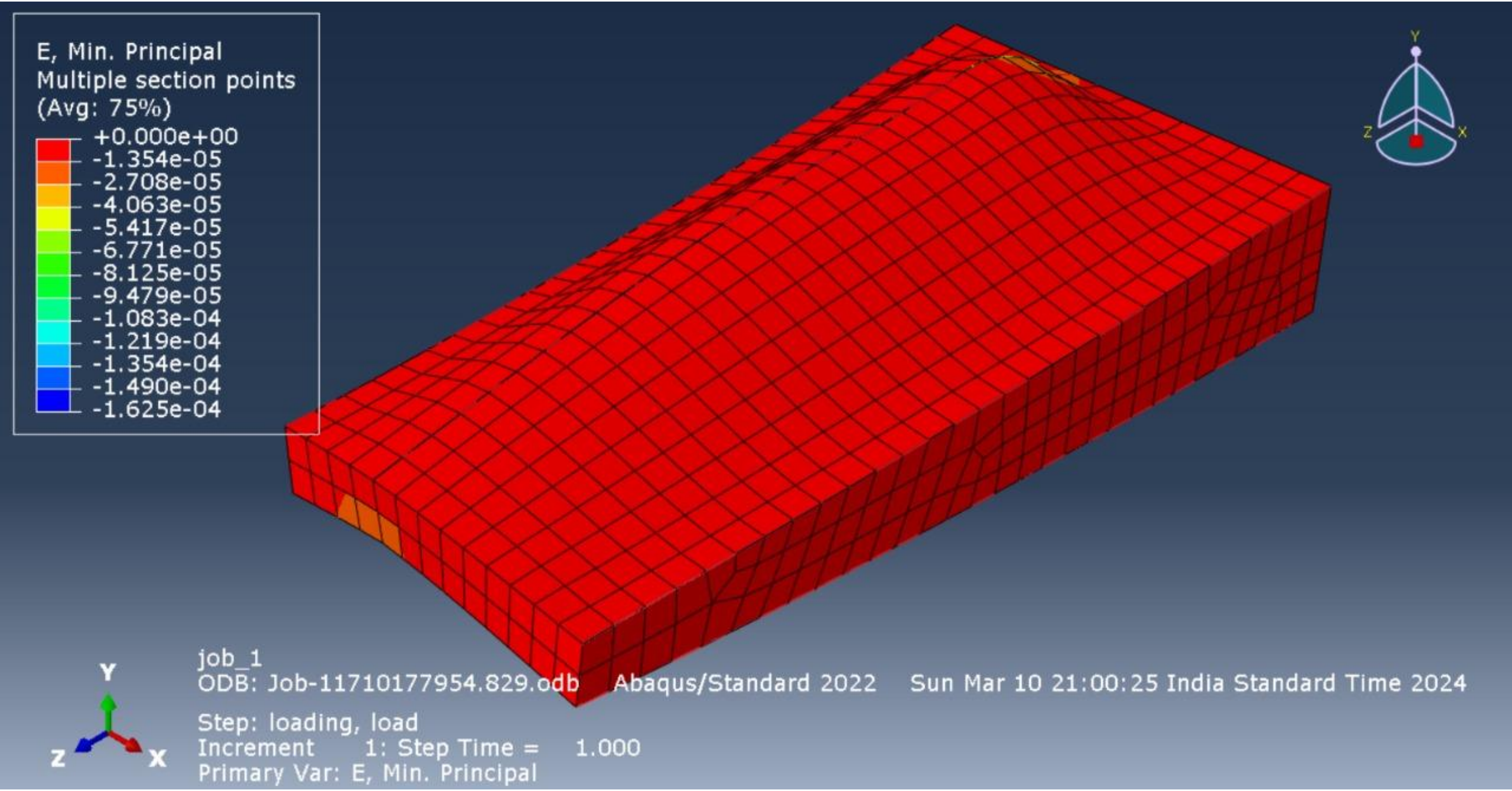

S, S22
Multiple section points
(Avg: 75%)
+3.198e+01
+2.882e+01
+2.566e+01
+2.250e+01
+1.934e+01
+1.618e+01
+1.303e+01
+9.866e+00
+6.708e+00
+3.549e+00
+3.905e-01
-2.768e+00
-5.927e+00
Y
Z
X
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: S, S22

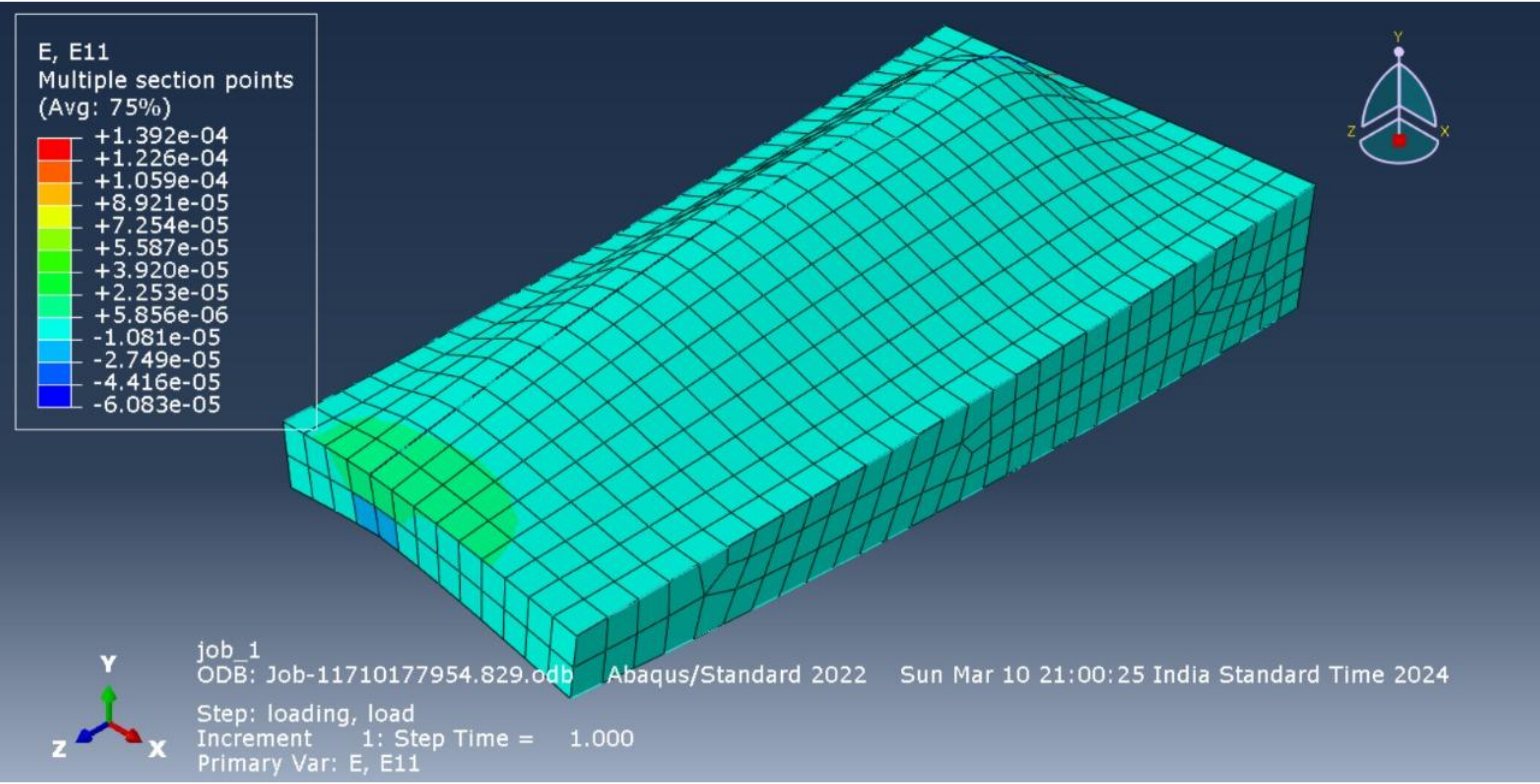

S, S33
Multiple section points
(Avg: 75%)
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
Y
Z
X
job_1
ODB: Job-11710177954.829.odb   Abaqus/Standard 2022   Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment   1: Step Time =   1.000
Primary Var: S, S33

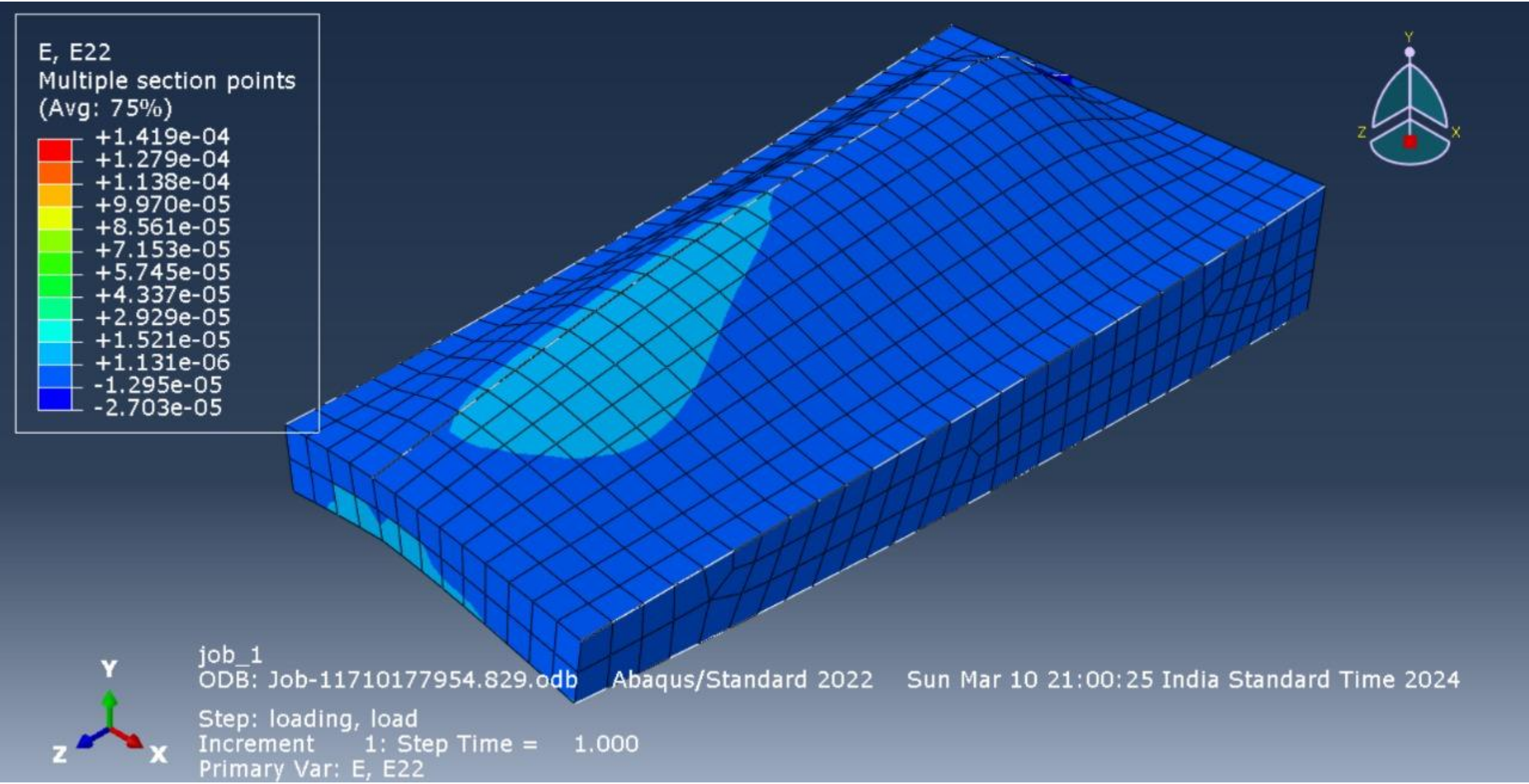

S, S12
Multiple section points
(Avg: 75%)
+2.535e+00
-1.845e-01
-2.904e+00
-5.623e+00
-8.343e+00
-1.106e+01
-1.378e+01
-1.650e+01
-1.922e+01
-2.194e+01
-2.466e+01
-2.738e+01
-3.010e+01
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: S, S12
Y
Z
X

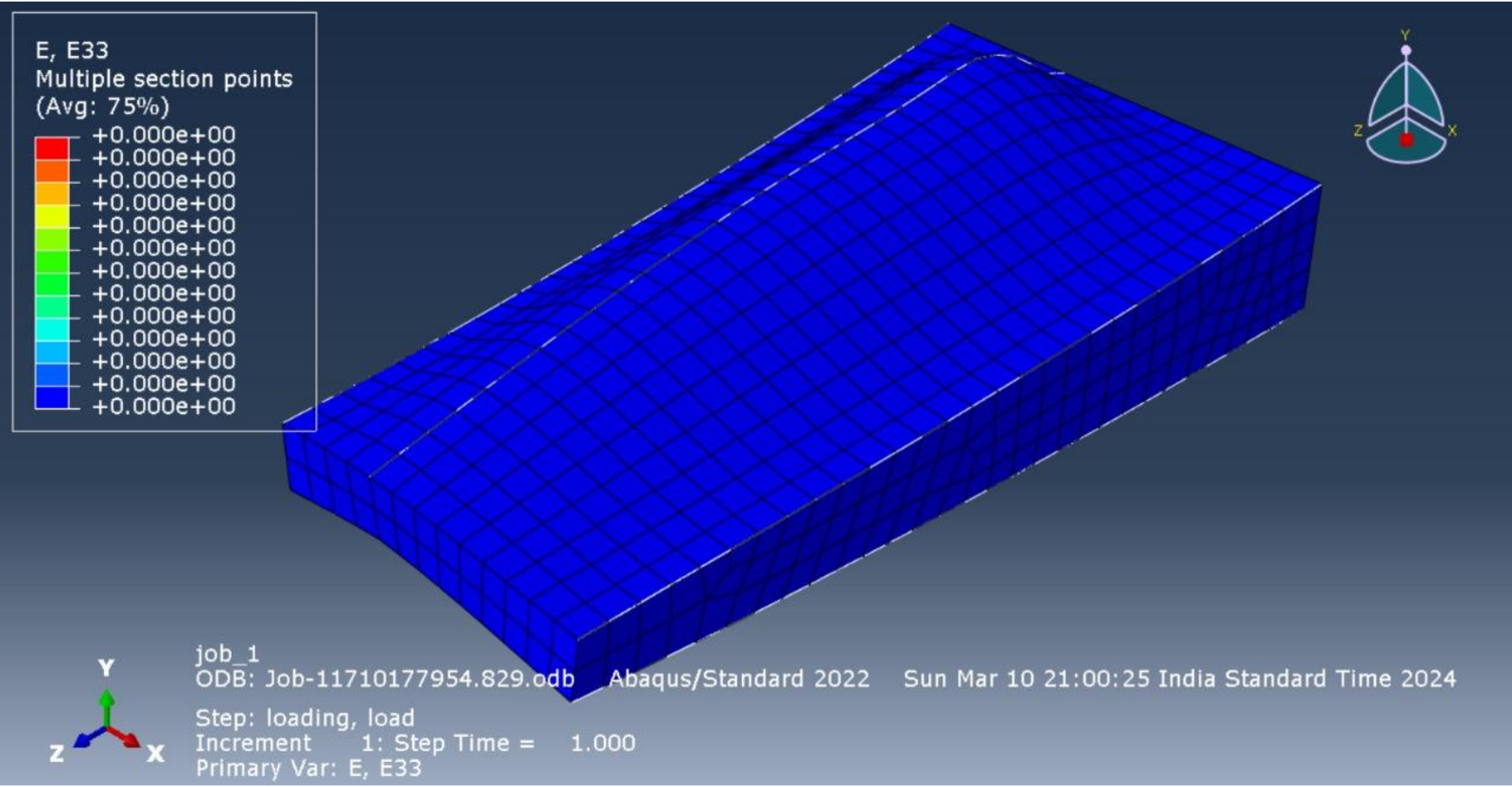
Strain Analysis:

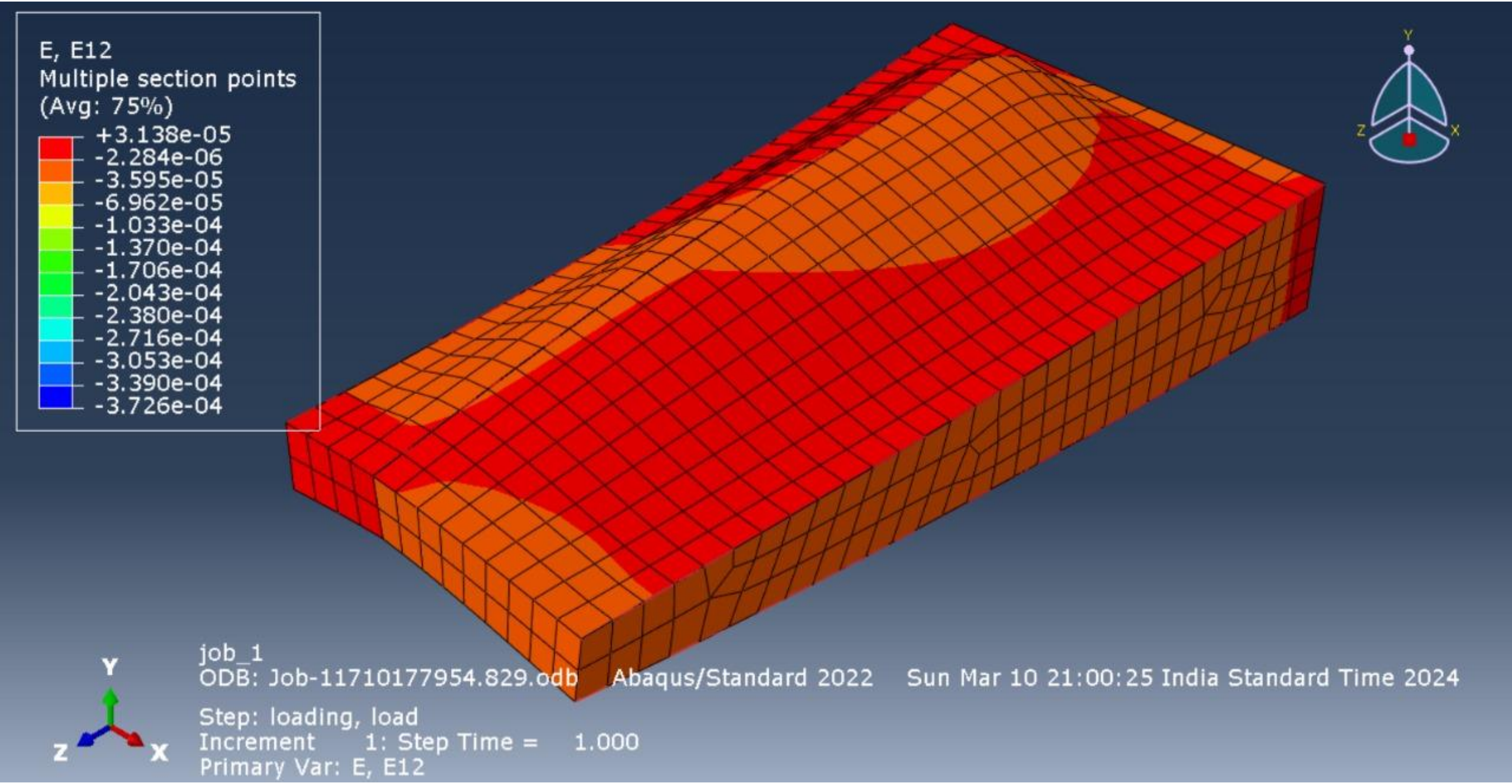



E, Min. Principal
Multiple section points
(Avg: 75%)
+0.000e+00
-1.354e-05
-2.708e-05
-4.063e-05
-5.417e-05
-6.771e-05
-8.125e-05
-9.479e-05
-1.083e-04
-1.219e-04
-1.354e-04
-1.490e-04
-1.625e-04
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: E, Min. Principal
Y
Z
X

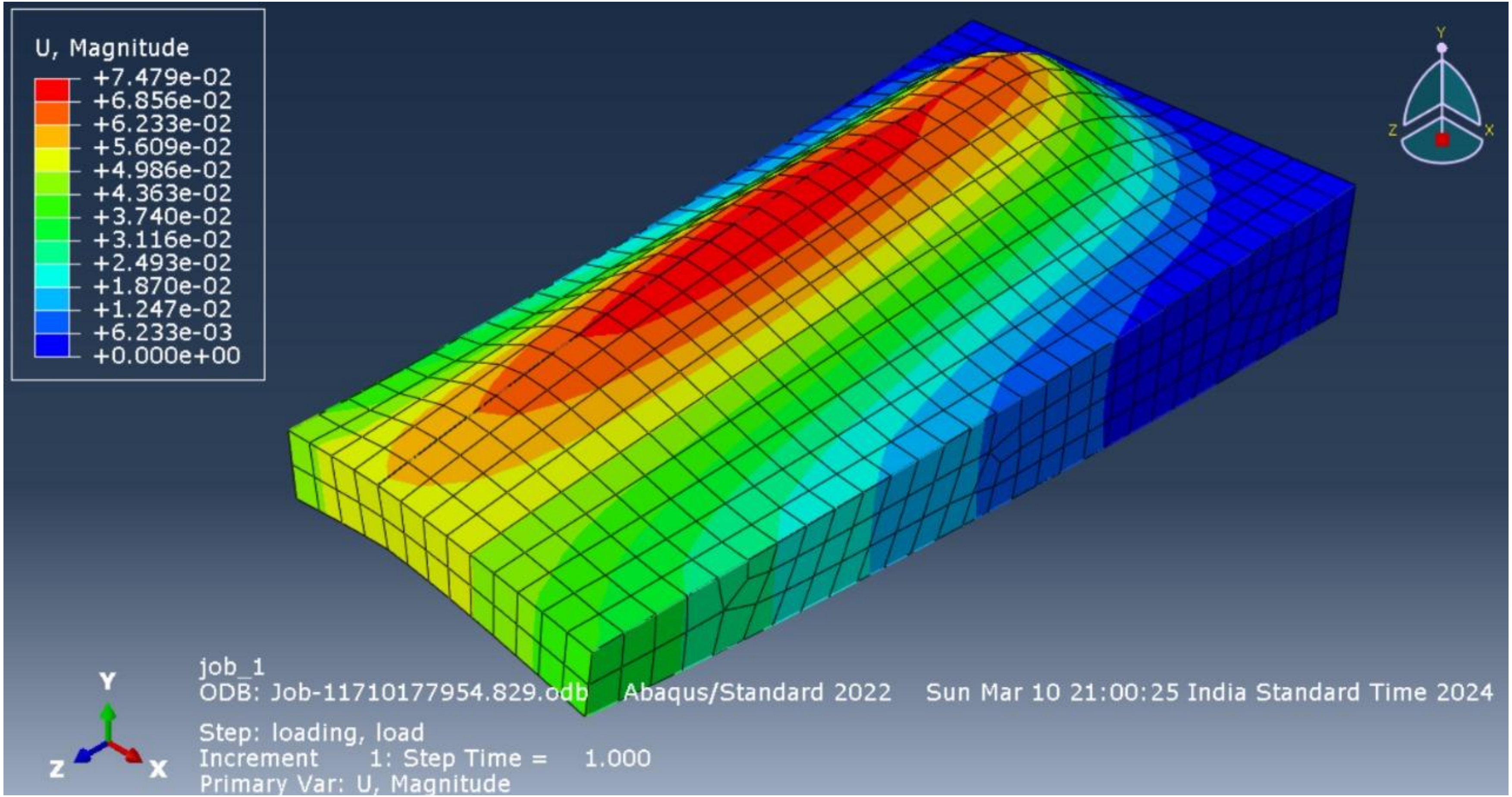

E, E11
Multiple section points
(Avg: 75%)
+1.392e-04
+1.226e-04
+1.059e-04
+8.921e-05
+7.254e-05
+5.587e-05
+3.920e-05
+2.253e-05
+5.856e-06
-1.081e-05
-2.749e-05
-4.416e-05
-6.083e-05
Y
Z
X
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: E, E11

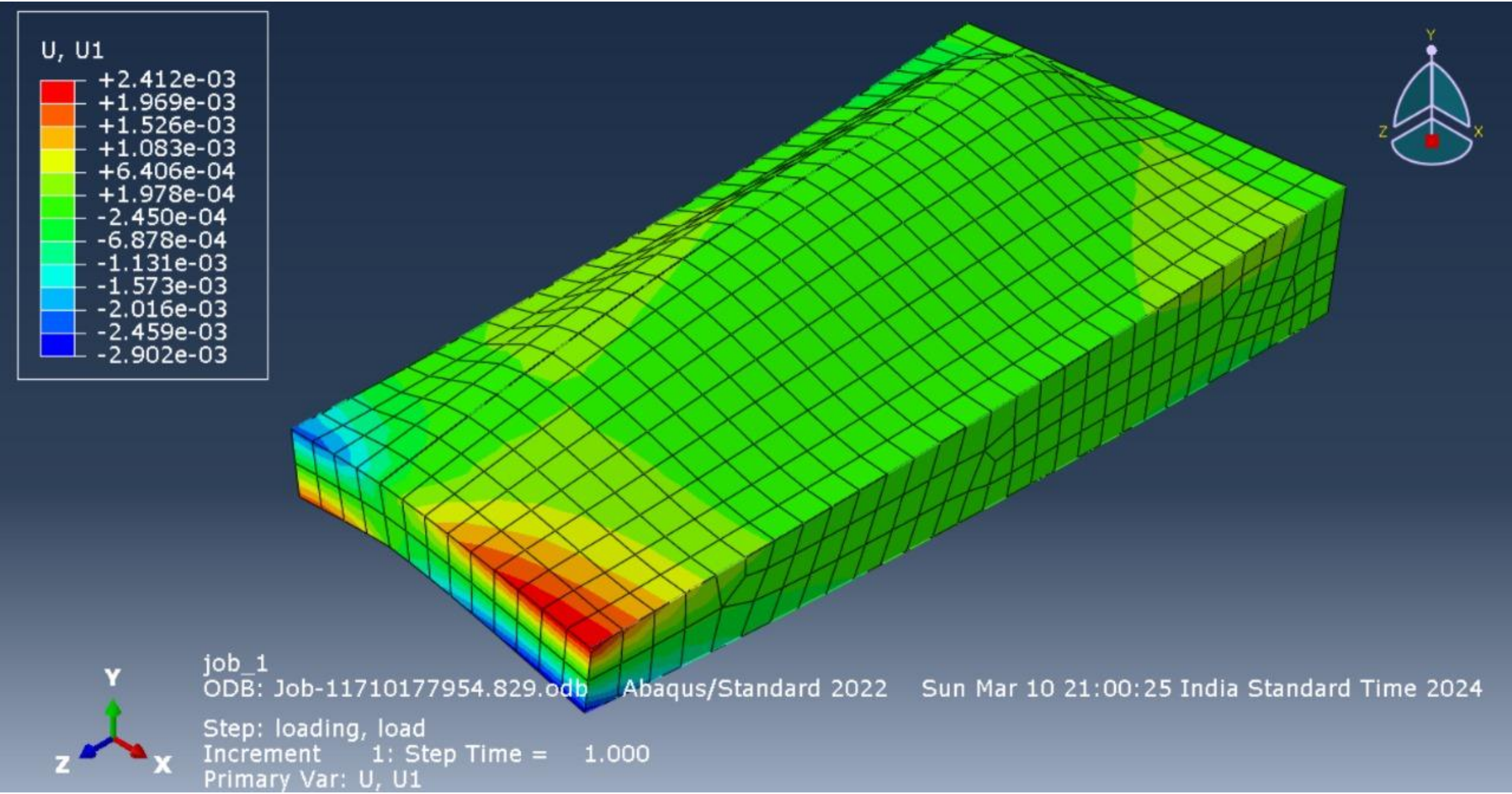

E, E22
Multiple section points
(Avg: 75%)
+1.419e-04
+1.279e-04
+1.138e-04
+9.970e-05
+8.561e-05
+7.153e-05
+5.745e-05
+4.337e-05
+2.929e-05
+1.521e-05
+1.131e-06
-1.295e-05
-2.703e-05
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: E, E22
Y
Z
X

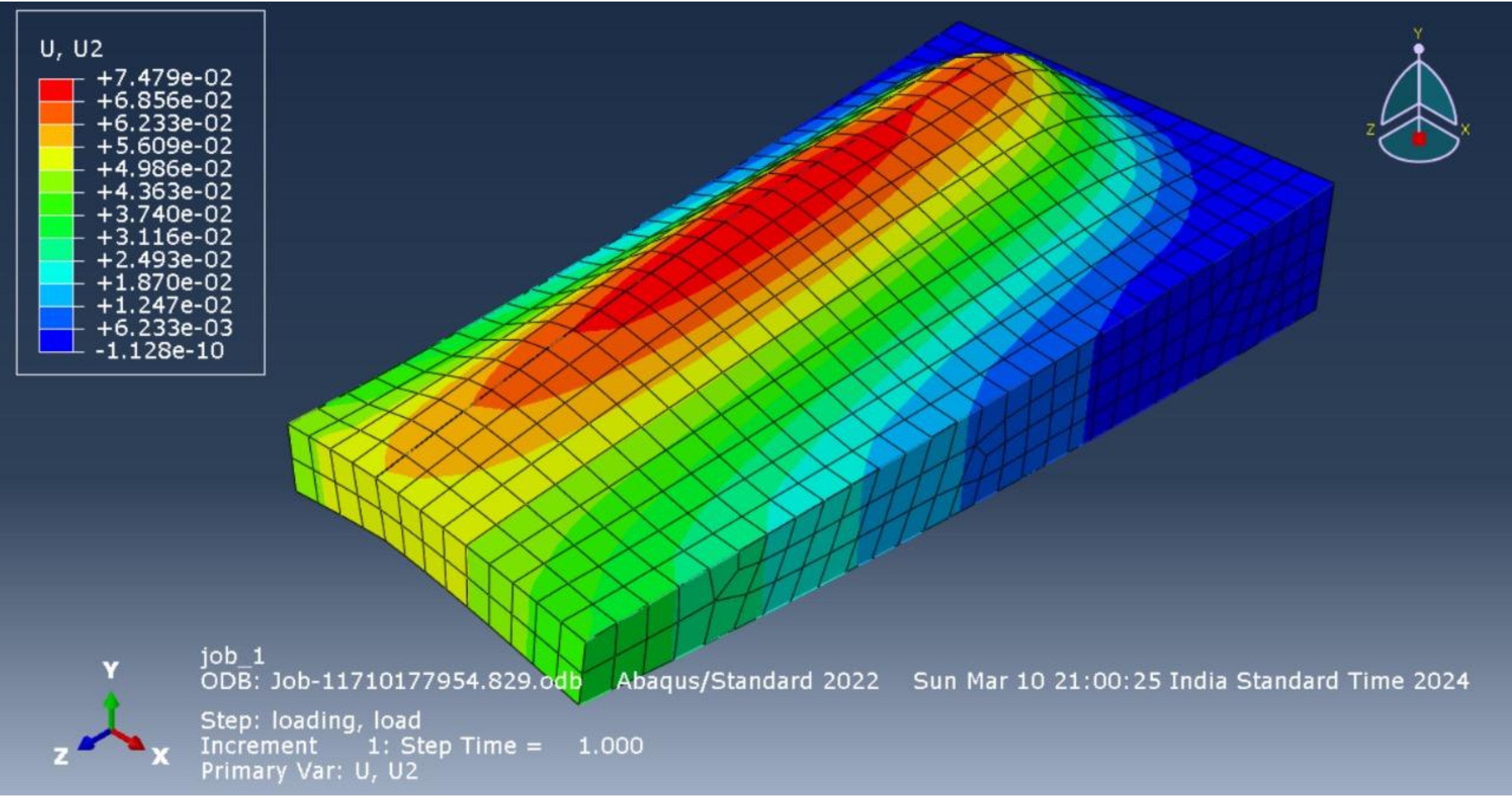

E, E33
Multiple section points
(Avg: 75%)
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
+0.000e+00
Y
Z
X
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: E, E33

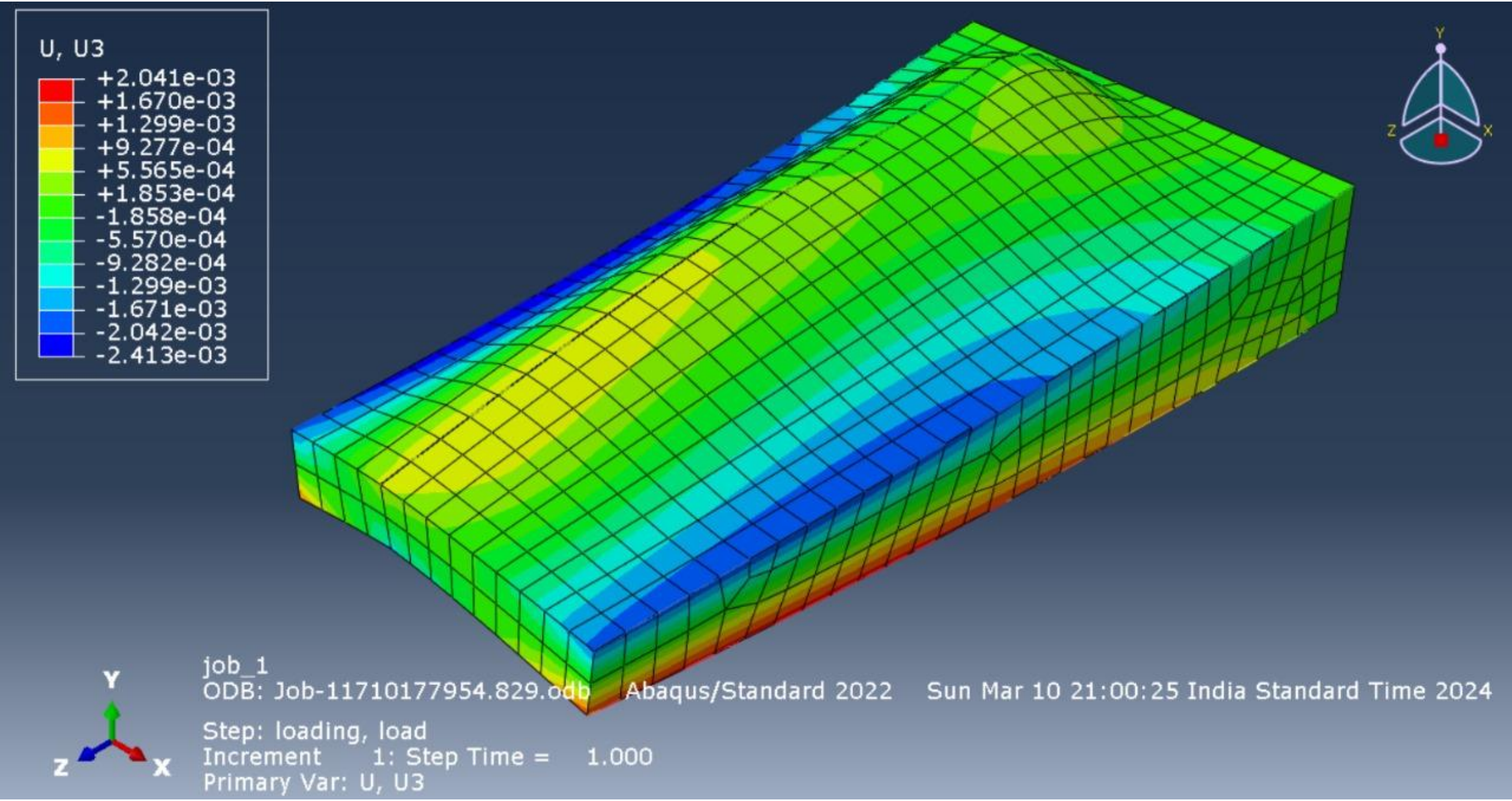

E, E12
Multiple section points
(Avg: 75%)
+3.138e-05
-2.284e-06
-3.595e-05
-6.962e-05
-1.033e-04
-1.370e-04
-1.706e-04
-2.043e-04
-2.380e-04
-2.716e-04
-3.053e-04
-3.390e-04
-3.726e-04
Y
Z
X
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: E, E12

Deflection Analysis:




U, U1
+2.412e-03
+1.969e-03
+1.526e-03
+1.083e-03
+6.406e-04
+1.978e-04
-2.450e-04
-6.878e-04
-1.131e-03
-1.573e-03
-2.016e-03
-2.459e-03
-2.902e-03
Y
Z
X
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: U, U1


U, U2
+7.479e-02
+6.856e-02
+6.233e-02
+5.609e-02
+4.986e-02
+4.363e-02
+3.740e-02
+3.116e-02
+2.493e-02
+1.870e-02
+1.247e-02
+6.233e-03
-1.128e-10
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: U, U2
Y
Z
X


U, U3
+2.041e-03
+1.670e-03
+1.299e-03
+9.277e-04
+5.565e-04
+1.853e-04
-1.858e-04
-5.570e-04
-9.282e-04
-1.299e-03
-1.671e-03
-2.042e-03
-2.413e-03
Y
Z
X
job_1
ODB: Job-11710177954.829.odb Abaqus/Standard 2022 Sun Mar 10 21:00:25 India Standard Time 2024
Step: loading, load
Increment 1: Step Time = 1.000
Primary Var: U, U3

Results Validation

At the larger cross-section

$$I_{xx} = 4 \times 600 \times 90^2 + 2 \times 900 \times 90^2 = 34.02 \times 10^6\,\text{mm}^4$$

The direct stress in a boom is given by Eq. (16.18) in which $I_{xy} = 0$ and $M_y = 0$, i.e.

$$\sigma_{z,r} = \frac{M_x y_r}{I_{xx}}$$

whence

$$P_{z,r} = \frac{M_x y_r}{I_{xx}} B_r$$

or

$$P_{z,r} = \frac{1.65 \times 10^6 y_r B_r}{34.02 \times 10^6} = 0.08 y_r B_r \qquad \text{(i)}$$

The value of $P_{z,r}$ is calculated from Eq. (i) in column ② of Table 23.2; $P_{x,r}$ and $P_{y,r}$ follow from Eqs (21.10) and (21.9), respectively in columns ⑤ and ⑥. The axial load P_r is given by $[②^2 + ⑤^2 + ⑥^2]^{1/2}$ in column ⑦ and has the same sign as $P_{z,r}$ (see Eq. (21.12)). The moments of $P_{x,r}$ and $P_{y,r}$, columns ⑩ and ⑪, are calculated for a moment centre at the mid-point of the internal web taking anticlockwise moments as positive.
From column ⑤

$$\sum_{r=1}^{6} P_{x,r} = 0$$

(as would be expected from symmetry).
From column ⑥

$$\sum_{r=1}^{6} P_{y,r} = 764.4\,\text{N}$$

From column ⑩

$$\sum_{r=1}^{6} P_{x,r}\eta_r = -117\,846\,\text{N mm}$$

From column ⑪

$$\sum_{r=1}^{6} P_{y,r}\xi_r = -43\,680\,\text{N mm}$$

From Eq. (21.15)

$$S_{x,w} = 0 \quad S_{y,w} = 10 \times 10^3 - 764.4 = 9235.6\,\text{N}$$

Also, since Cx is an axis of symmetry, $I_{xy} = 0$ and Eq. (20.6) for the 'open section' shear flow reduces to

$$q_b = -\frac{S_{y,w}}{I_{xx}} \sum_{r=1}^{n} B_r y_r$$

or

$$q_b = -\frac{9235.6}{34.02 \times 10^6} \sum_{r=1}^{n} B_r y_r = -2.715 \times 10^{-4} \sum_{r=1}^{n} B_r y_r \qquad \text{(ii)}$$

'Cutting' the top walls of each cell and using Eq. (ii), we obtain the q_b distribution shown in Fig. 23.13. Evaluating δ for each wall and substituting in Eq. (23.10) gives for cell I

$$\frac{d\theta}{dz} = \frac{1}{2 \times 36\,000G}(760q_{s,0,\text{I}} - 180q_{s,0,\text{II}} - 1314) \qquad \text{(iii)}$$

for cell II

$$\frac{d\theta}{dz} = \frac{1}{2 \times 72\,000G}(-180q_{s,0,\text{I}} + 1160q_{s,0,\text{II}} + 1314) \qquad \text{(iv)}$$

Taking moments about the mid-point of web 25 we have, using Eq. (23.13)

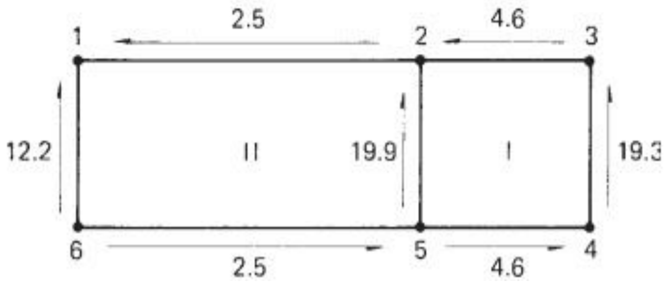
$$0 = -14.7 \times 180 \times 400 + 14.7 \times 180 \times 200 + 2 \times 36\,000q_{s,0,\text{I}} + 2 \times 72\,000q_{s,0,\text{II}} \\ -117\,846 - 43\,680$$

or

$$0 = -690\,726 + 72\,000q_{s,0,\text{I}} + 144\,000q_{s,0,\text{II}} \qquad \text{(v)}$$

Resultant Distribution of Shear Flow:

① Boom	② $P_{z,r}$ (N)	③ $\frac{\delta_{x_r}}{\delta_z}$	④ $\frac{\delta_{y_r}}{\delta_z}$	⑤ $P_{x,r}$ (N)	⑥ $P_{y,r}$ (N)	⑦ P_r (N)	⑧ ξ_r (mm)	⑨ η_r (mm)	⑩ $P_{x,r}\eta_r$ (N mm)	⑪ $P_{y,r}\xi_r$ (N mm)
1	2619.0	0	0.0417	0	109.2	2621.3	400	90	0	43 680
2	3928.6	0.0833	0.0417	327.3	163.8	3945.6	0	90	−29 457	0
3	2619.0	0.1250	0.0417	327.4	109.2	2641.6	200	90	−29 466	21 840
4	−2619.0	0.1250	−0.0417	−327.4	109.2	−2641.6	200	90	−29 466	21 840
5	−3928.6	0.0833	−0.0417	−327.3	163.8	−3945.6	0	90	−29 457	0
6	−2619.0	0	−0.0417	0	109.2	−2621.3	400	90	0	−43 680



Insights:

1. Material Behavior:

   Material properties such as modulus of elasticity, yield strength, and Poisson's ratio significantly influence stress and strain distributions.

   Higher modulus of elasticity leads to stiffer behavior and lower deflections under load.

   Variations in material properties, such as defects or imperfections, can cause localized stress concentrations.

   Understanding material behavior allows for optimization of material selection to improve overall structural performance.

Insights:

2. Structural Integrity:

   Critical points of stress concentrations often occur at areas subjected to high loads.

   Stress concentrations can lead to material failure or fatigue over time, impacting structural integrity.

   Simulation results provide insights into potential failure modes such as yielding, buckling, or fracture, aligning with theoretical expectations.

Insights:

3. Performance Enhancement:

    Design modifications such as adding ribs or reinforcements can enhance structural performance by distributing loads more effectively.

    Enhancing geometric features to promote uniform stress distribution can improve overall structural integrity and longevity.

    Performance enhancements based on FEA insights lead to more resilient and reliable structures in service.

Thank You

AE462

Lab Exercise 5

Aircraft Wing Analysis

Date
14/03/2024

Instructor
Prof. G.M. Kamath

Group 11
Aryaman Badkul
Palmate Aditya Tukaram
Shrutikirti Singh
Prabuddha Singh
Shikha Sharma
Rahangdale Prachi Sanjay

Exercise 4:

Content

Procedure Flowchart:

1. Pre-processing Part
   - Modelling
   - Part defining(material used)
   - Assembly
   - Application of Loads & Boundary Conditions
   - See Results Obtained

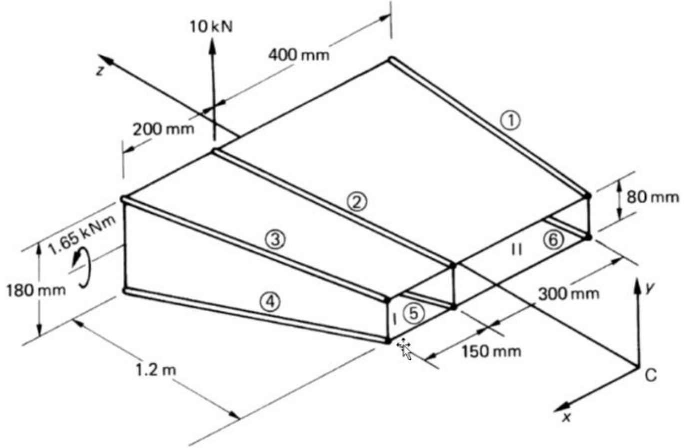
2. Post-processing Part
   - Multiple Files will get generated
   - Use .odp file to see the Results

Problem Statement

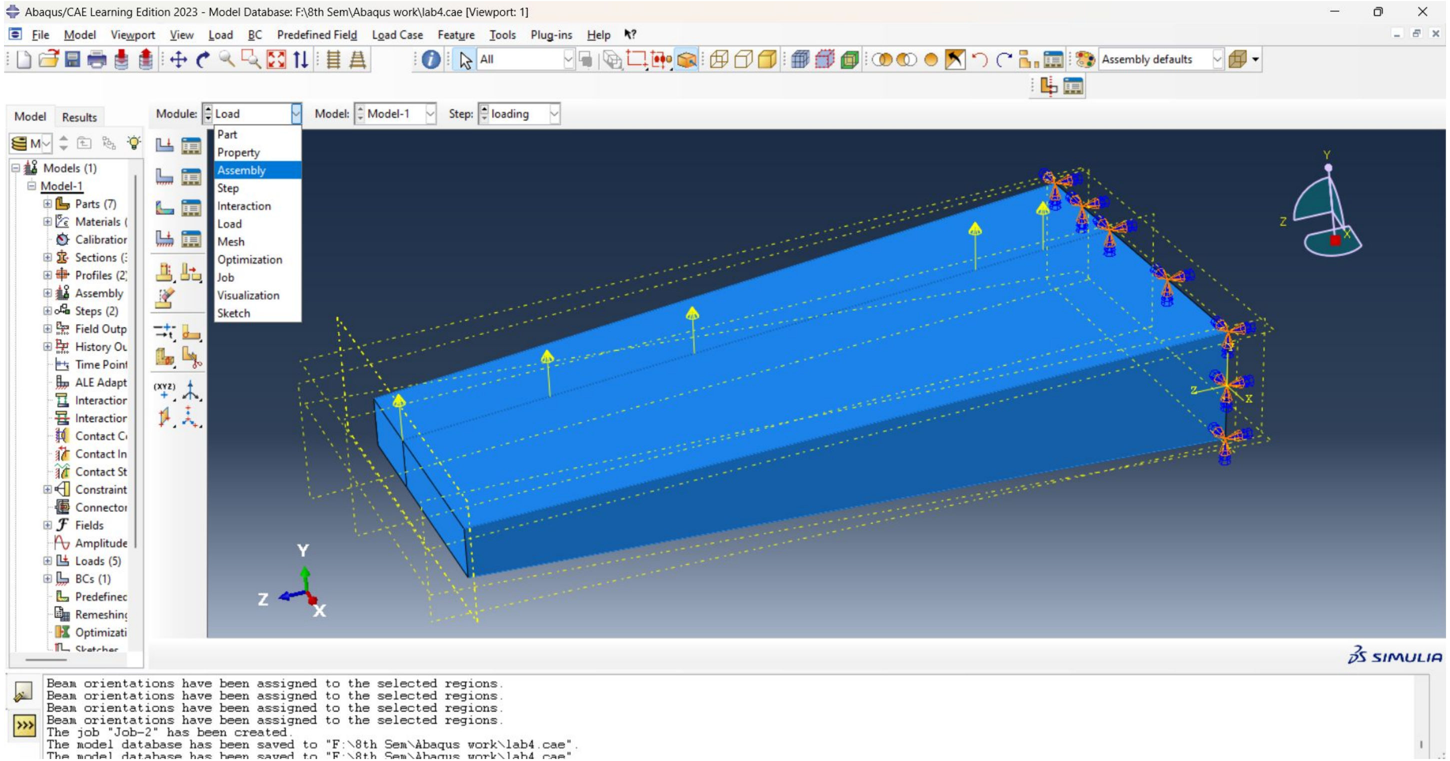
Example 23.4

A two-cell beam has singly symmetrical cross-sections 1.2 m apart and tapers symmetrically in the y direction about a longitudinal axis (Fig. 23.12). The beam supports loads which produce a shear force $S_y = 10$ kN and a bending moment $M_x = 1.65$ kN m at the larger cross-section; the shear load is applied in the plane of the internal spar web. If booms 1 and 6 lie on a plane which is parallel to the yz plane, calculate the forces in the booms and the shear flow distribution in the walls at the larger cross-section. The booms are assumed to resist all the direct stresses while the walls are effective only in shear. The shear modulus is constant throughout, the vertical webs are all 1.0 mm thick, while the remaining walls are all 0.8 mm thick:

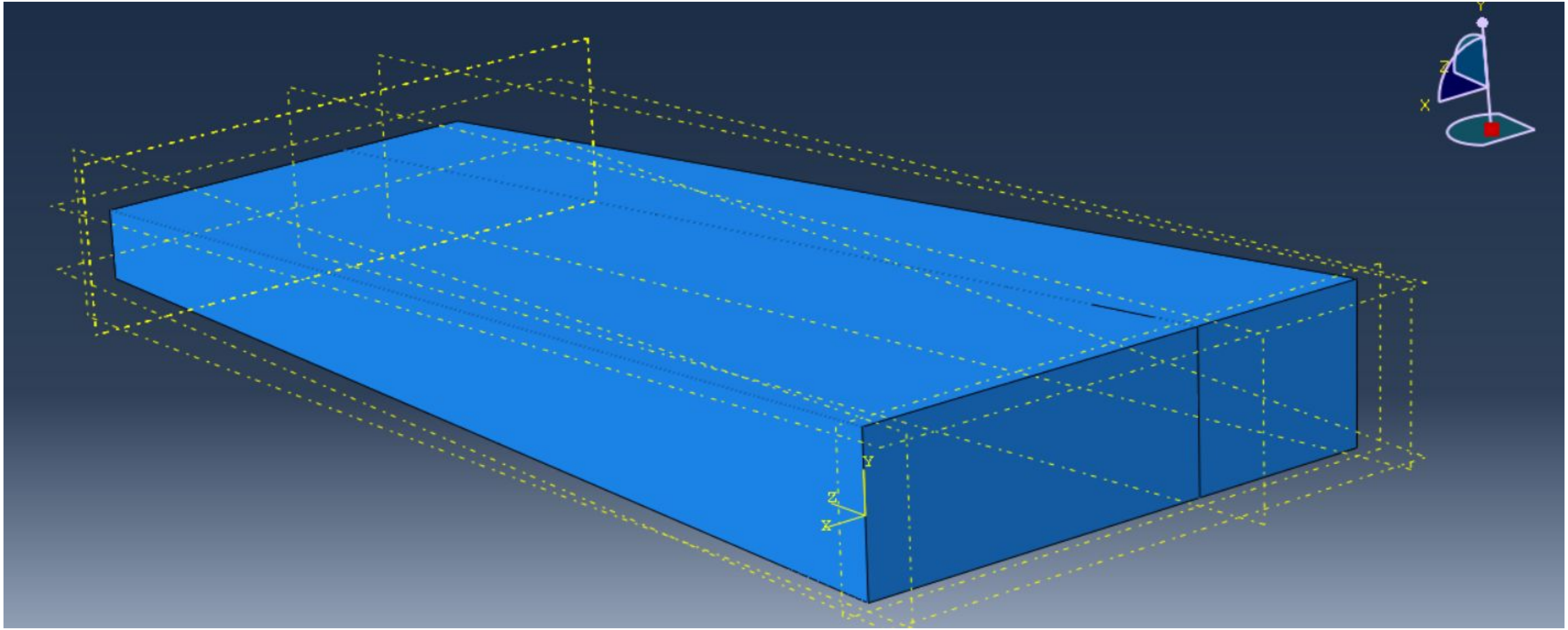
$$\text{Boom areas: } B_1 = B_3 = B_4 = B_6 = 600 \text{ mm}^2, \; B_2 = B_5 = 900 \text{ mm}^2$$



Link for CAE file- [Lab4_Group11]
Abaqus Workspace

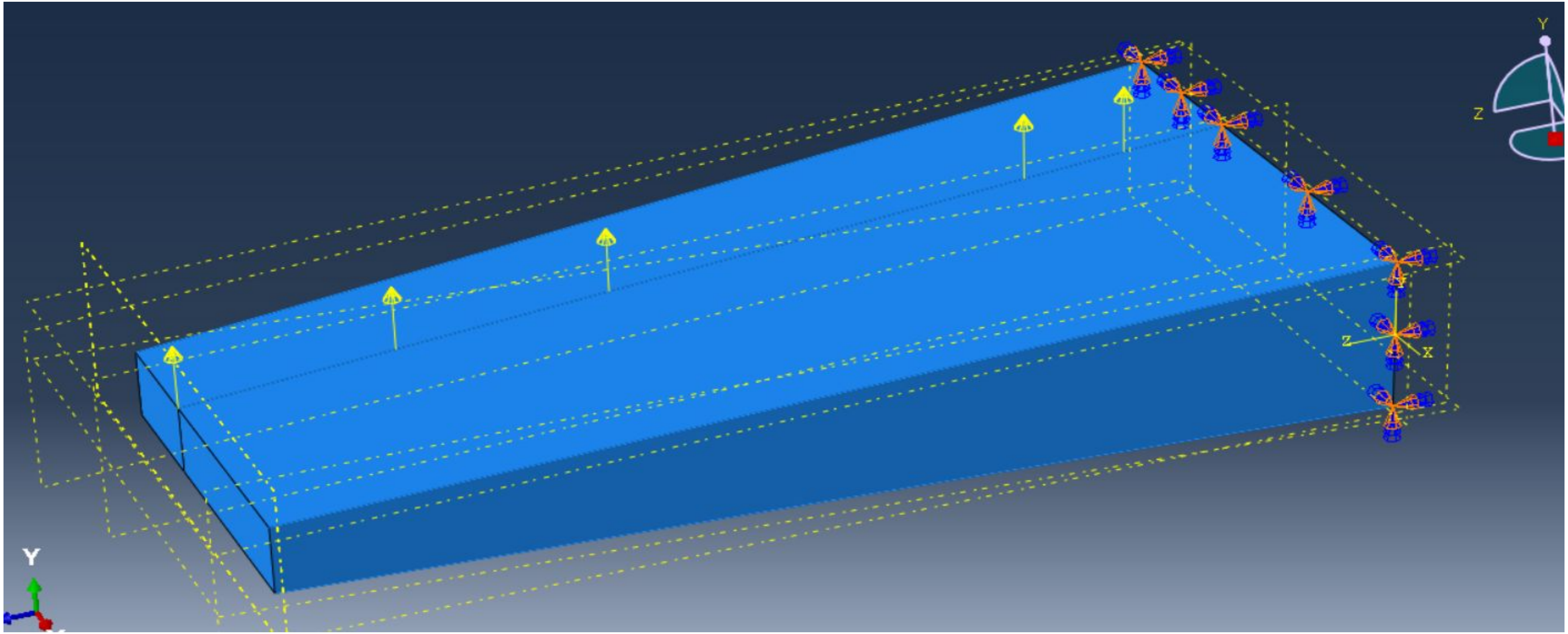


Model of a Two-Cell Beam

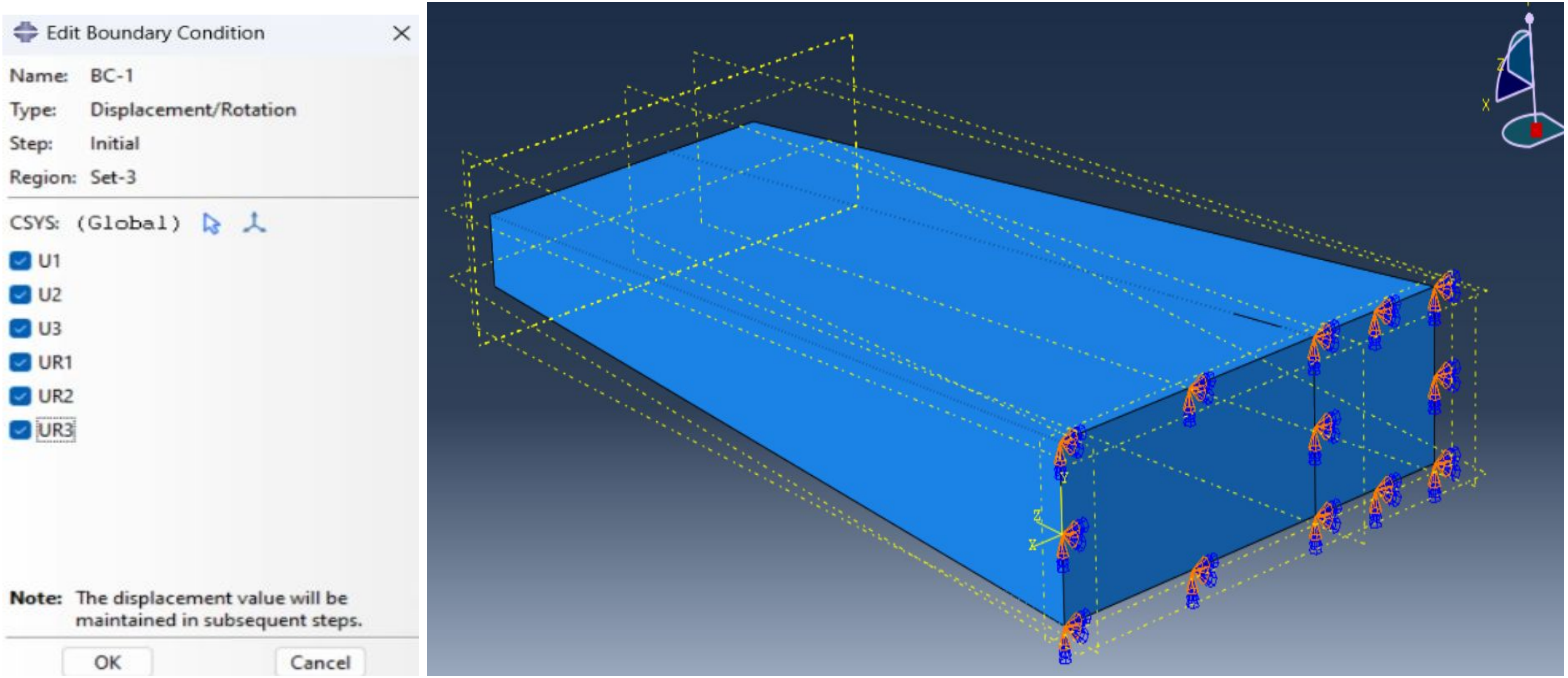


Loads Applied along with Boundary Conditions

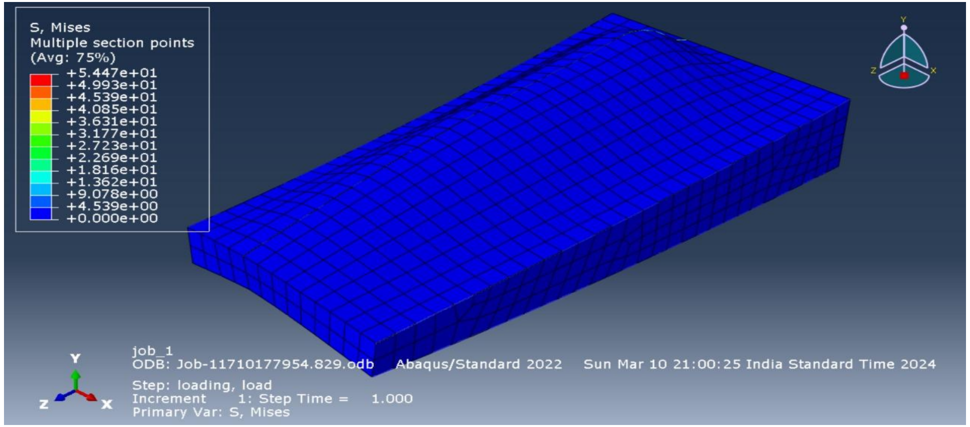
Distance from root	Force(KN)
0.1	6
0.2	3.5
0.6	0.3
0.8	0.15
1	0.05

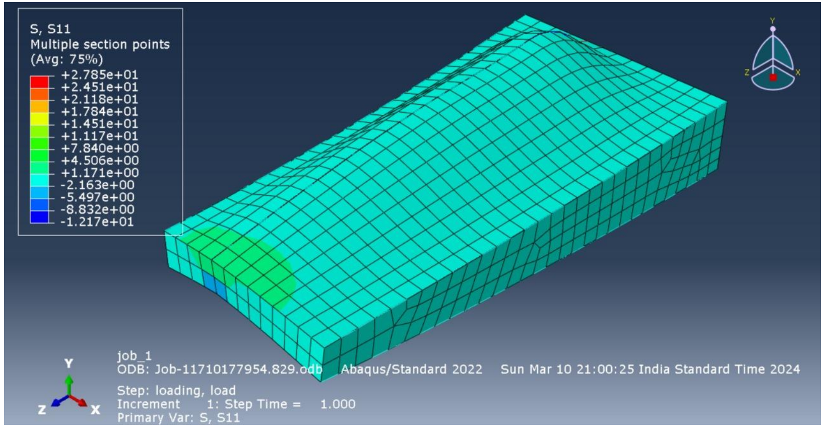


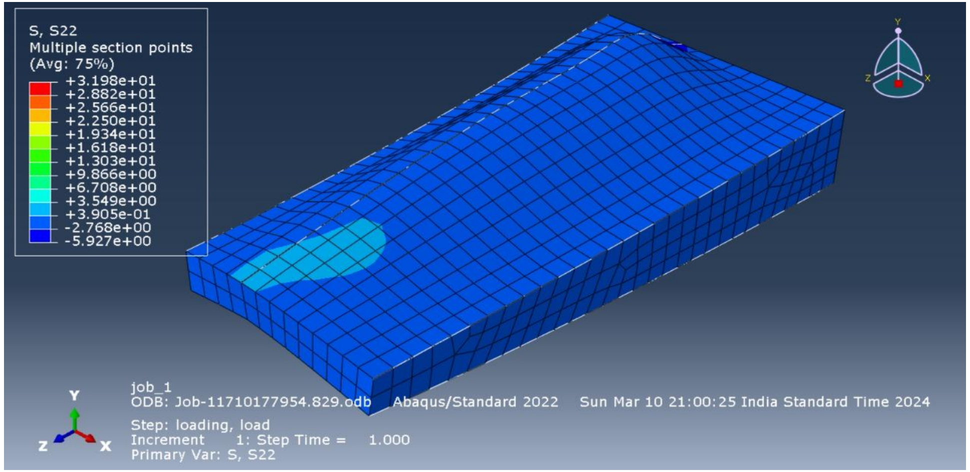
Boundary Conditions:

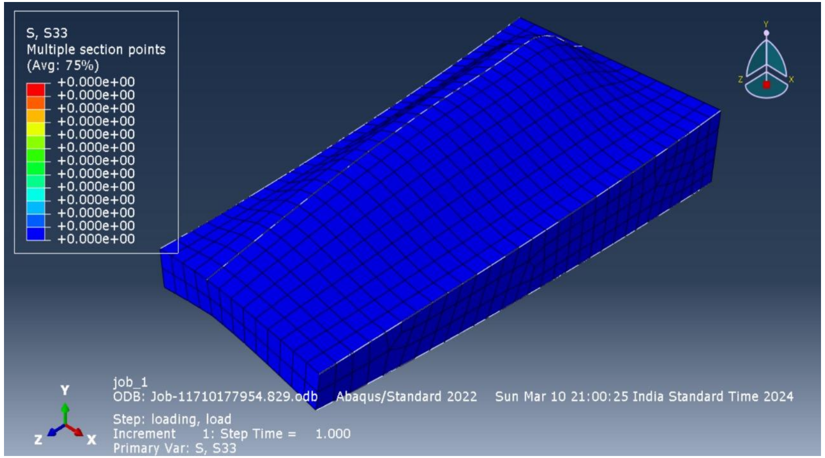


Stress Analysis:

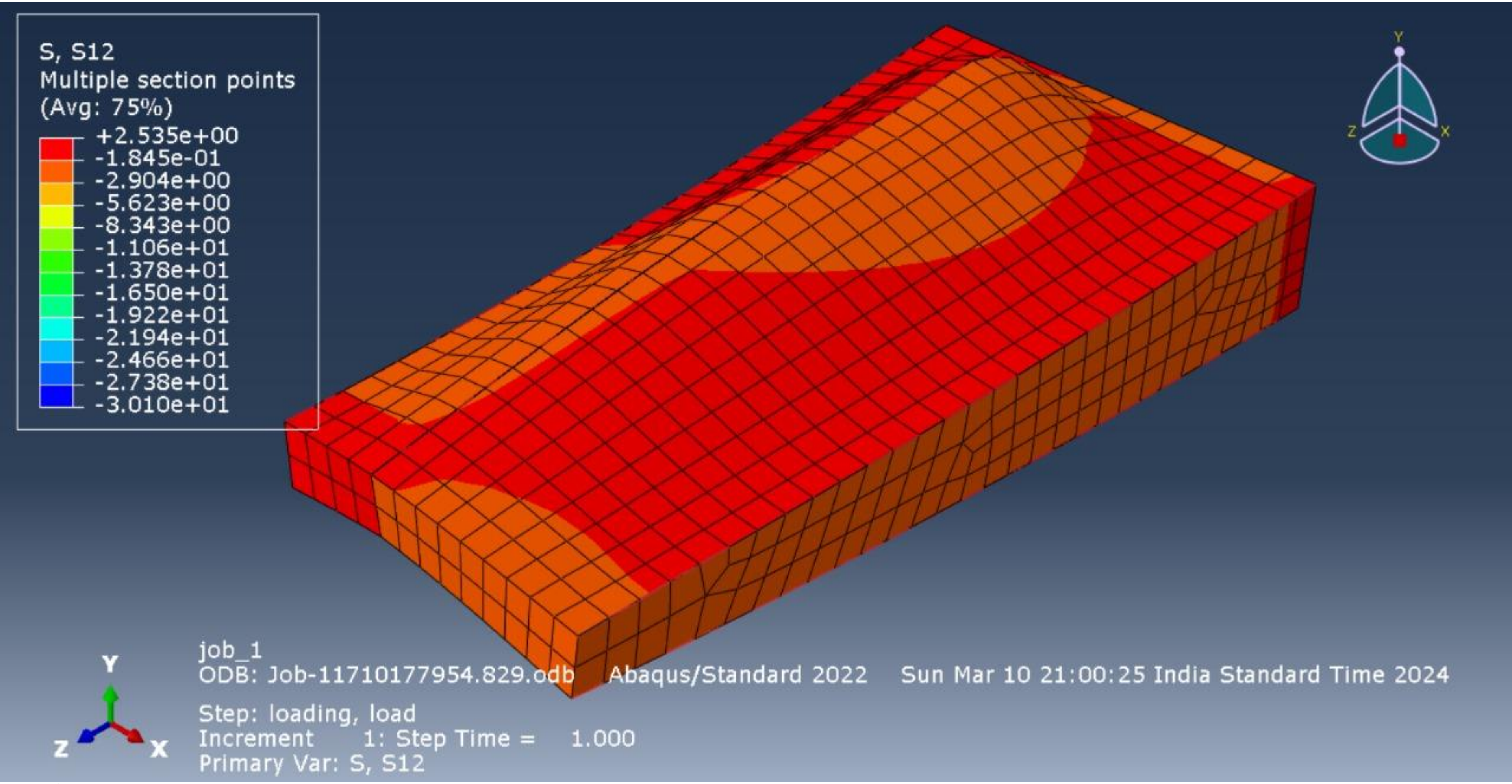






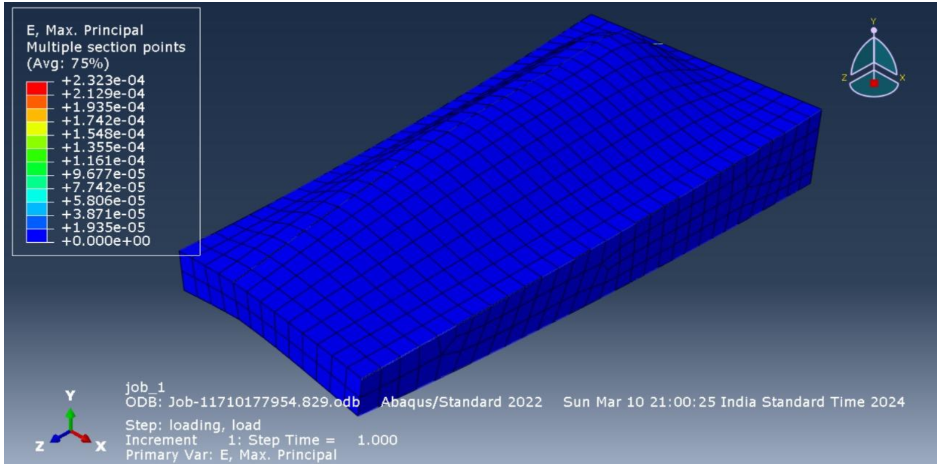


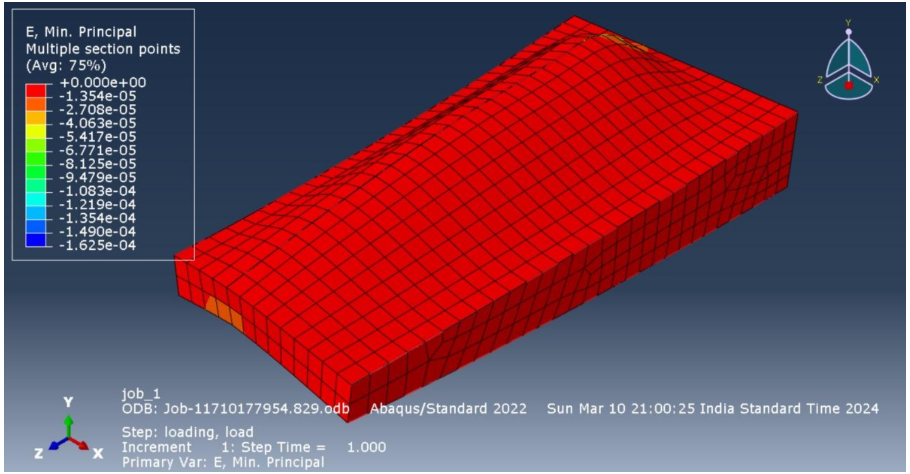
S11, S22 and S33 are principal stresses in three directions.

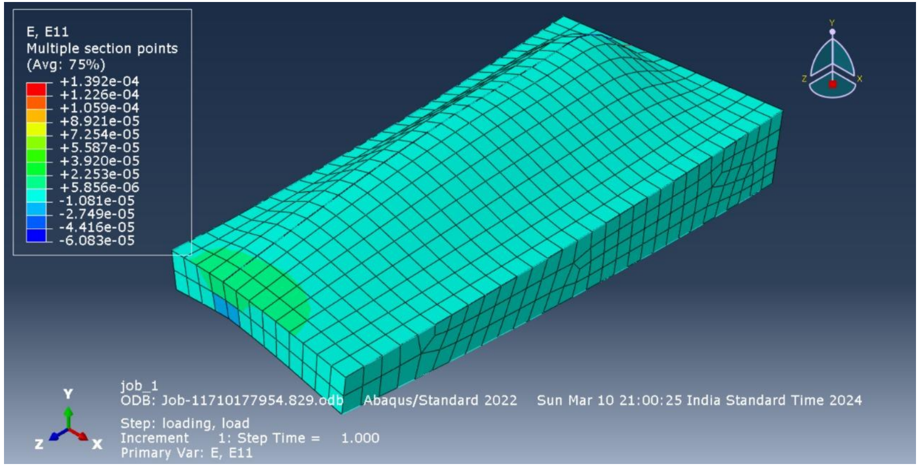


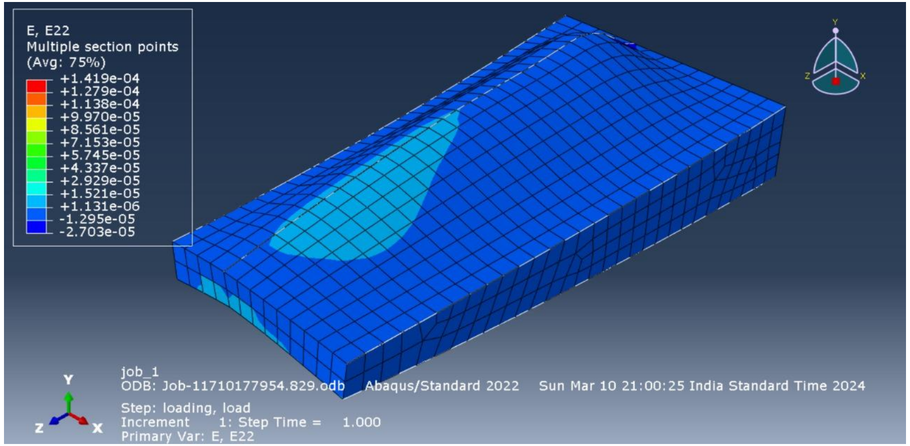
S12 is the shear stress acting along y axis

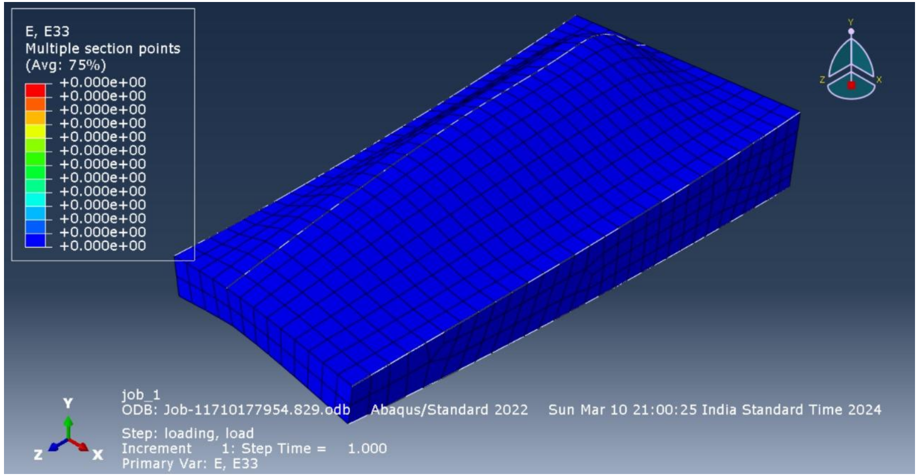
Strain Analysis:

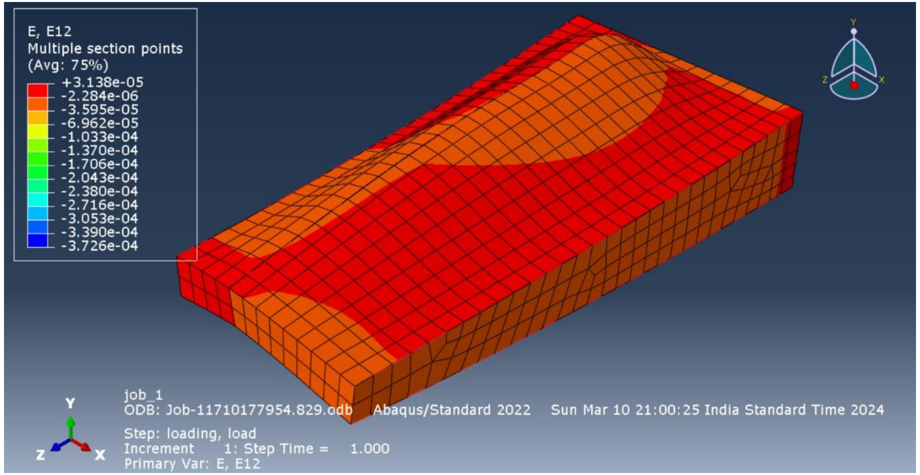






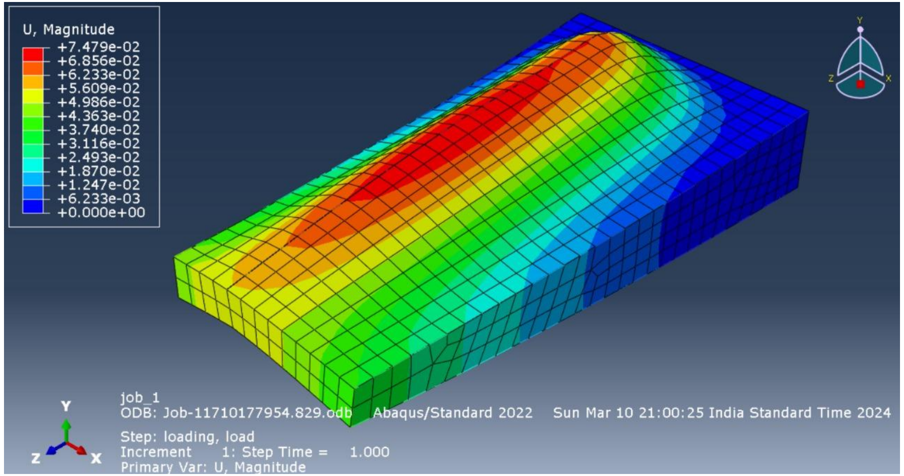


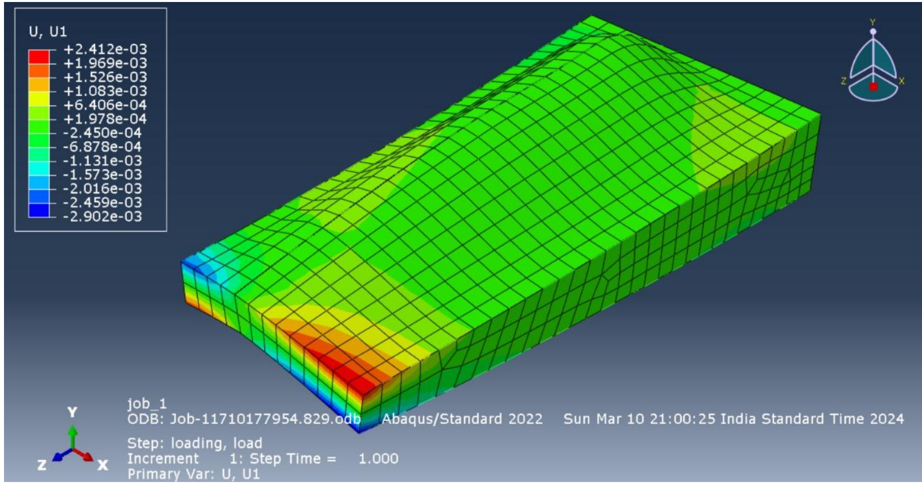


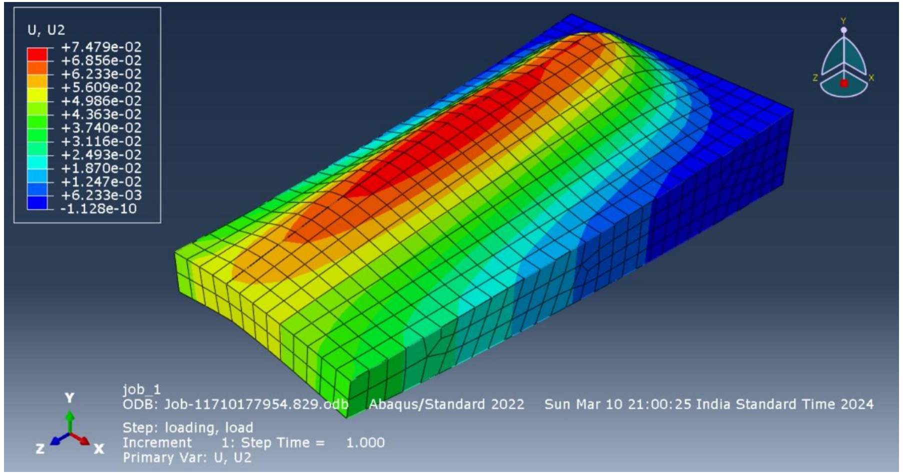


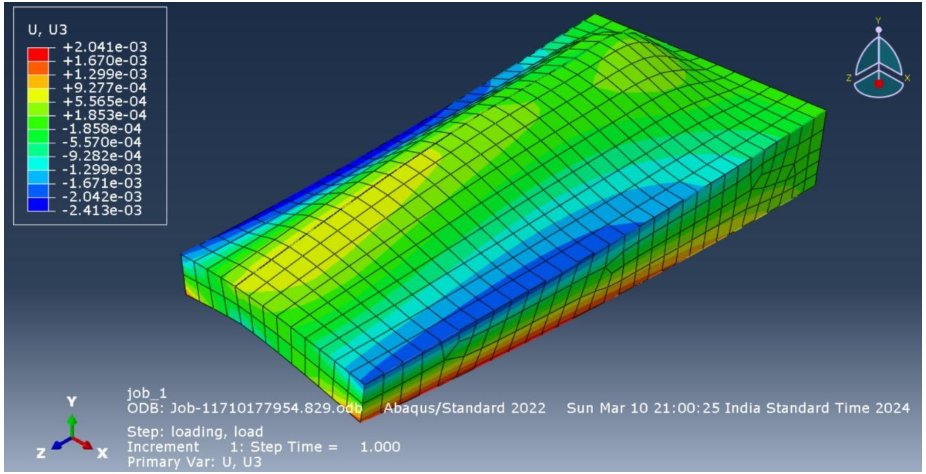
E11, E22, and E33 refer to the three principal components of strain in a three-dimensional stress state.
E12 represents the strain in the direction parallel to the axis 2 due to loading in the direction of axis 1.

Deflection Analysis:









Insights:

1. **Material Behavior:**
   Material properties such as modulus of elasticity, yield strength, and Poisson's ratio significantly influence stress and strain distributions.
   Higher modulus of elasticity leads to stiffer behavior and lower deflections under load.
   Variations in material properties, such as defects or imperfections, can cause localized stress concentrations.
   Understanding material behavior allows for optimization of material selection to improve overall structural performance.
2. **Structural Integrity:**
   Critical points of stress concentrations often occur at areas subjected to high loads.
   Stress concentrations can lead to material failure or fatigue over time, impacting structural integrity.
   Simulation results provide insights into potential failure modes such as yielding, buckling, or fracture, aligning with theoretical expectations.
3. **Performance Enhancement:**
   Design modifications such as adding ribs or reinforcements can enhance structural performance by distributing loads more effectively.
   Enhancing geometric features to promote uniform stress distribution can improve overall structural integrity and longevity.
   Performance enhancements based on FEA insights lead to more resilient and reliable structures in service.

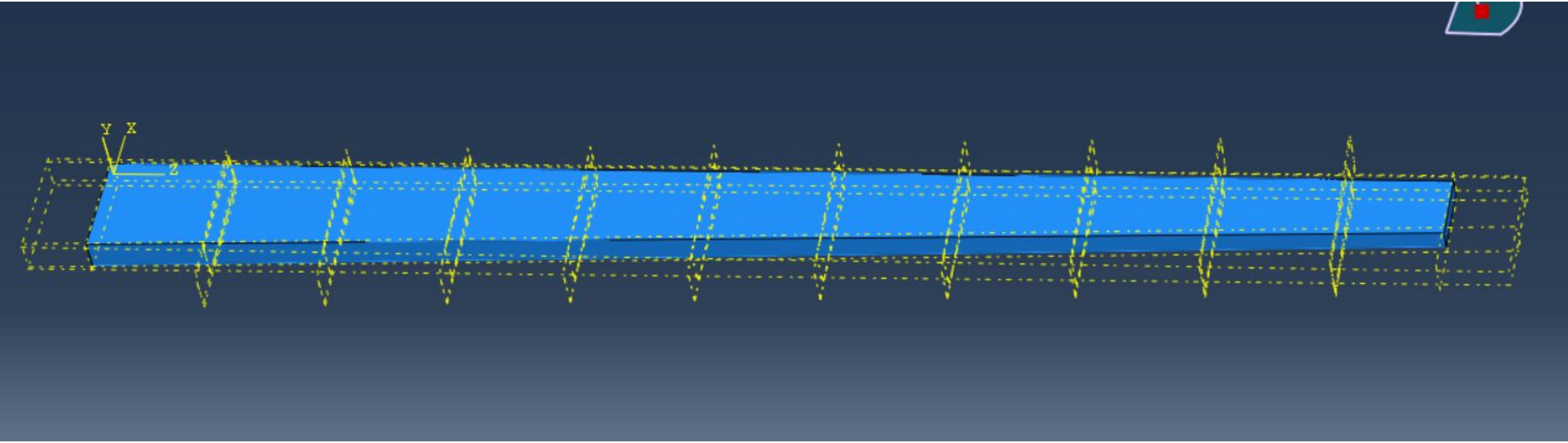
Exercise 5
Aircraft Wing Analysis

Problem Statement

Prepare the finite element model in Abaqus of your aircraft wing and carry out the analysis with the loads determined in an earlier exercise.

Prepare a report in the form of a powerpoint presentation and include all details of the model and results. This includes finite elements used for various parts, boundary conditions, loads applied, and the results obtained (deflections, stresses and strains). Most importantly, some kind of insights and validation of the results should be also be included.

Wing Model



A Tapered wing with 4 booms

Max web skin thickness and max spar web thickness has been used for shell skin for entire wing(ignoring the variation of these thickness along wingspan)

Loads on wing

We have used total 11 ribs in this wing, which divides the wing in 11 sections.
We've calculated Load(Q) generated for each section ($Lift_{section}$-$Weight_{section}$)
And torsion(T) in each rib about its CG.
Now since we are idealizing the wing cross section with inter spar box. We need to modify the loads accordingly.
We are applying loads on front and rear spars(say La(front spar) Lb(rear spar))
The summation of these loads(La+Lb) in for a section should equal to $Q_{section}$
And the torsion due to these two loads about the CG of wing box should equal to torsion(T) that we calculated earlier.
Since we have idealized the wing cross section with inter spar box(by equating cross sectional areas to calculate height of box) the wing box itself represents the entire wing and hence the CG should lie at the center of wing box

Hence for each section we get,
La+Lb=Q
(La-Lb)(interspar length/2)=T

Where
Q=Lift-Weight
T(about wing CG)=L(chord/4)

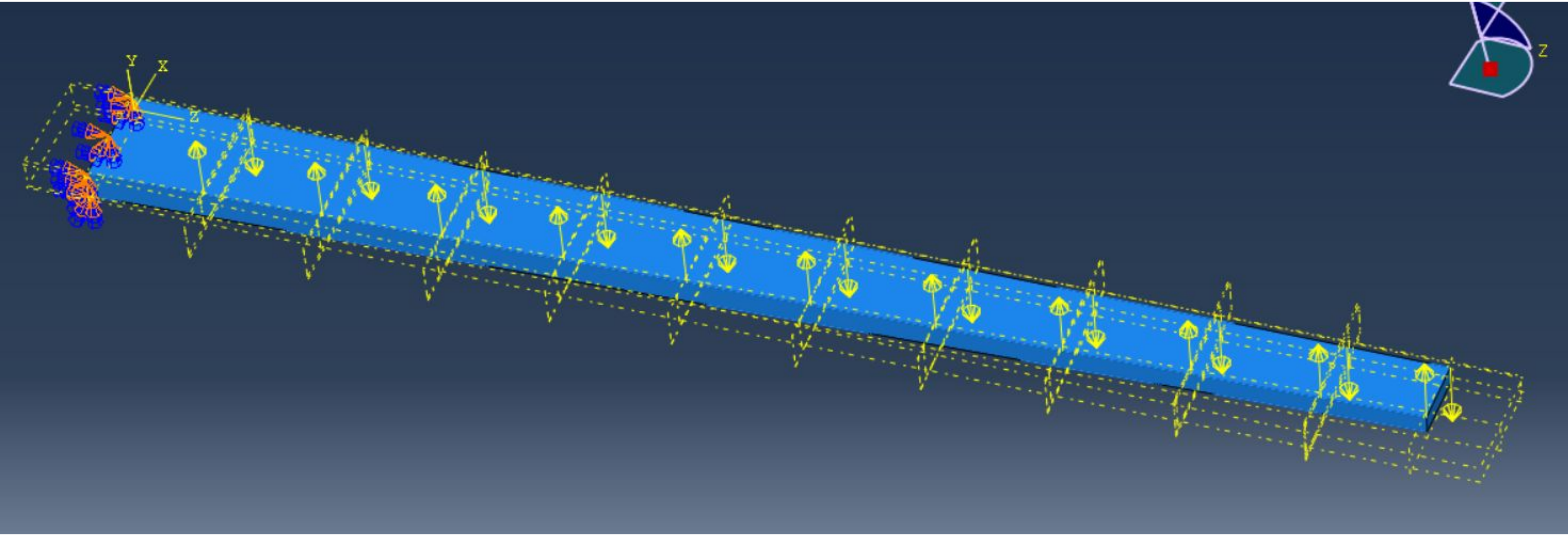
Loads on wing

Link- Excel for Loads on wing box

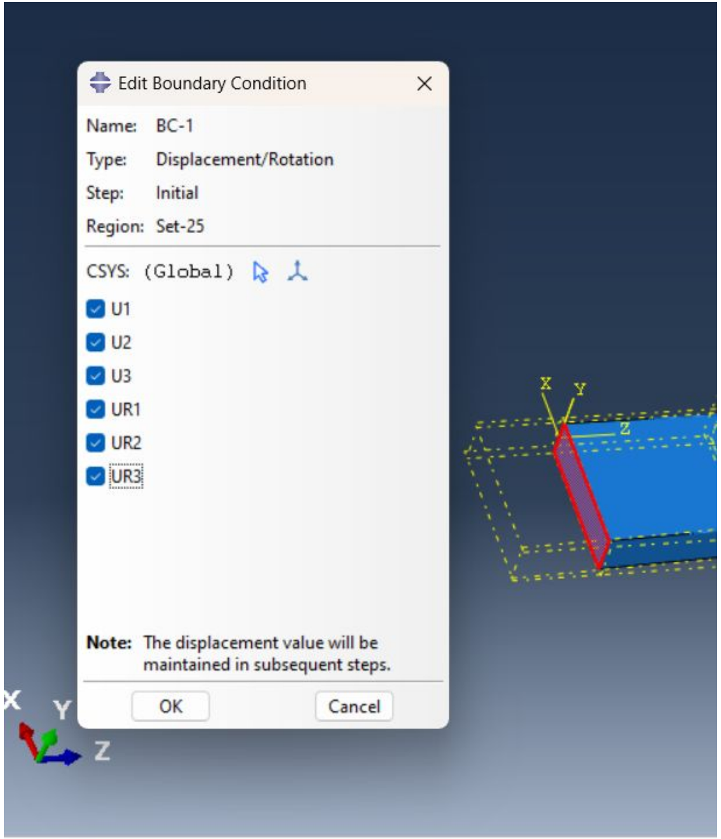
Using two eqn we can find La(force on front spar) and Lb(force on rear spar) for each section(these loads would be applied at ribs)

Rib(location)in m	La(in N)	Lb(in N)
0.5	361.7305691	-69.69480384
1	569.6130056	-283.4324993
1.5	770.8658731	-492.0756911
2	964.4351128	-694.6544475
2.5	1149.149368	-890.1390665
3	1323.655777	-1077.400982
3.5	1486.312187	-1255.154875
4	1634.982595	-1421.862314
4.5	1766.5754	-1575.546705
5	1875.636132	-1713.349466
5.4	1936.413019	-1808.640468

Loads Applied along with Boundary Conditions

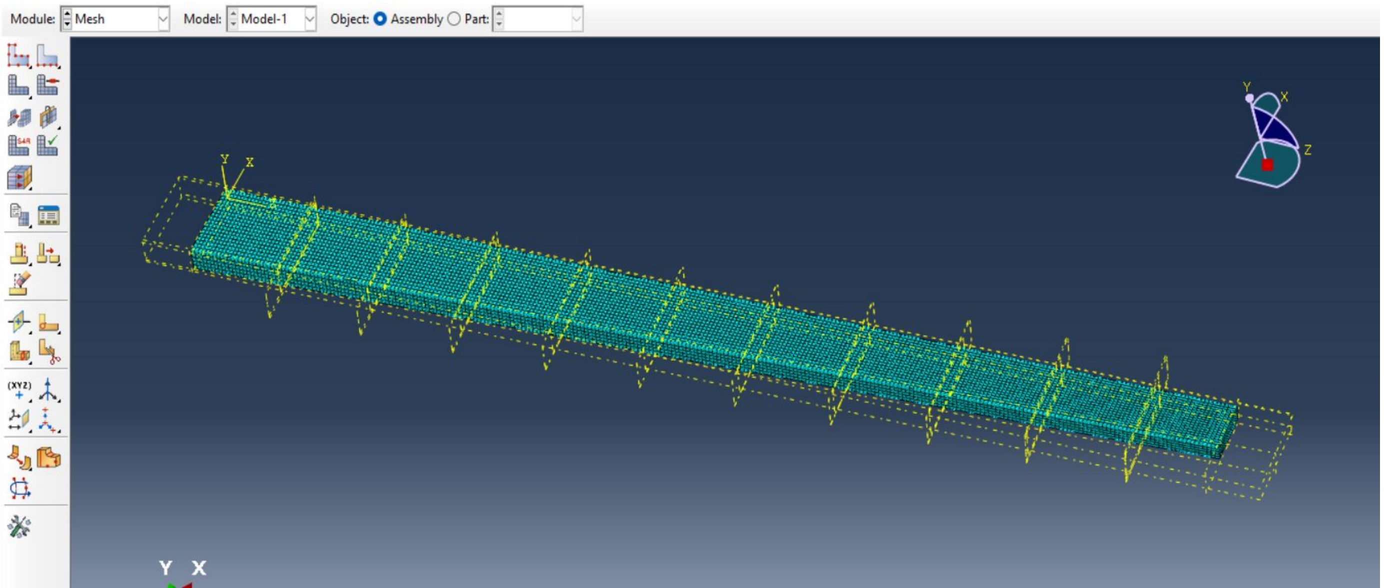


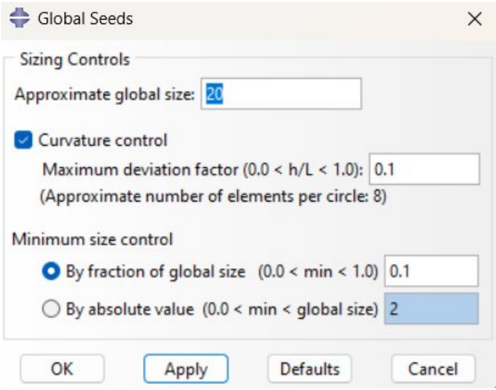
Boundary Conditions:



Meshing

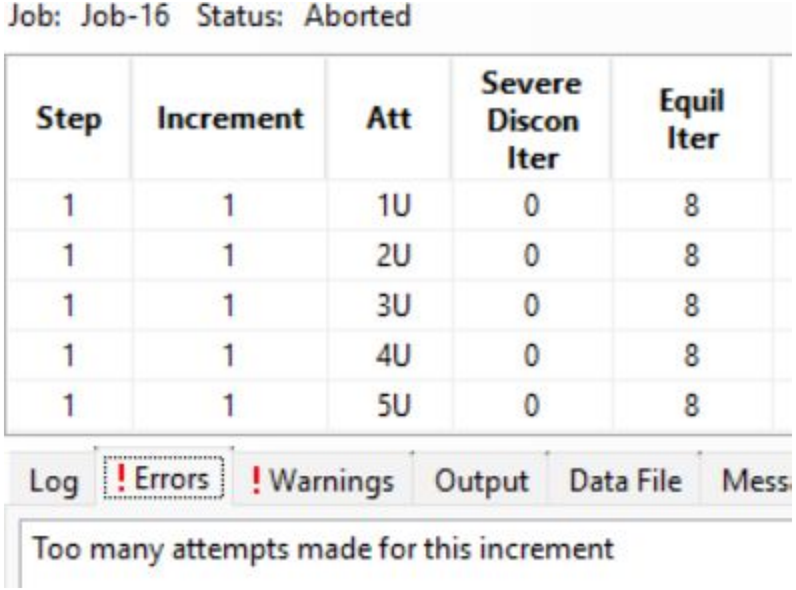
For our earlier model, we were facing meshing error(some elements were of unusable quality)
We recreated the model using similar approach and seed size 20(mm)



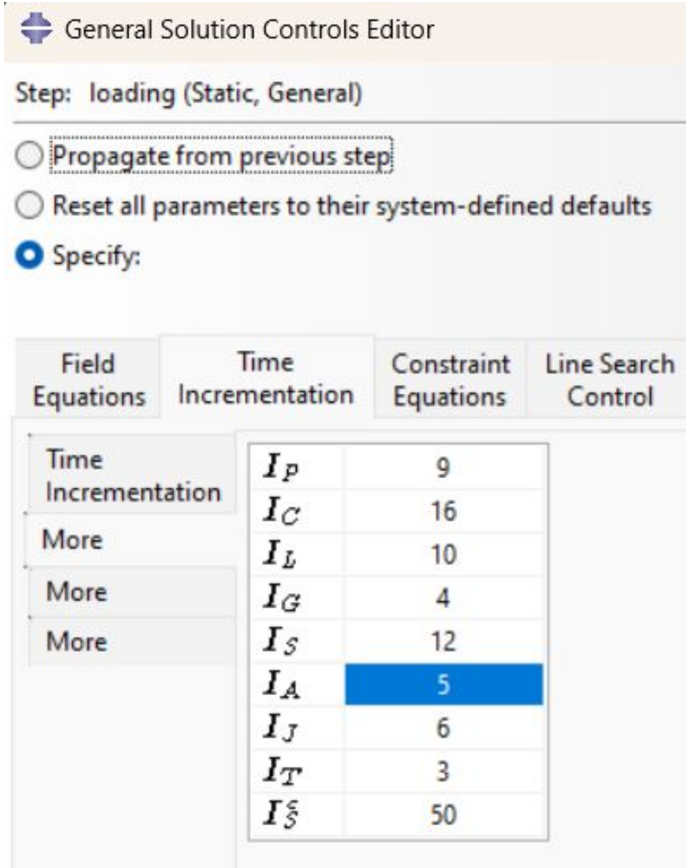


Visualization(errors)

We were facing ‘too many attempts for one increment’ error hence we tried changing(increasing) the time increment value



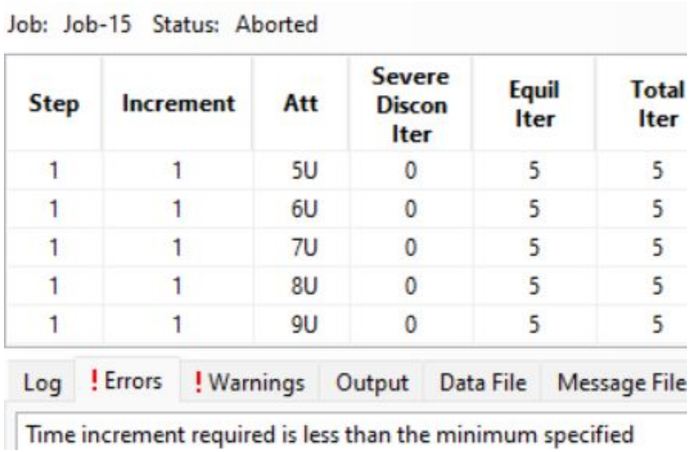
Step	Increment	Att	Severe Discon Iter	Equil Iter
1	1	1U	0	8
1	1	2U	0	8
1	1	3U	0	8
1	1	4U	0	8
1	1	5U	0	8



I_P	9
I_C	16
I_L	10
I_G	4
I_S	12
I_A	5
I_J	6
I_T	3
I_S^{ξ}	50

Visualization(errors)

but then faced error ‘time increment required is less than the minimum specified’
We tried reducing the min increment size and initial increment size and increasing the max no of increments

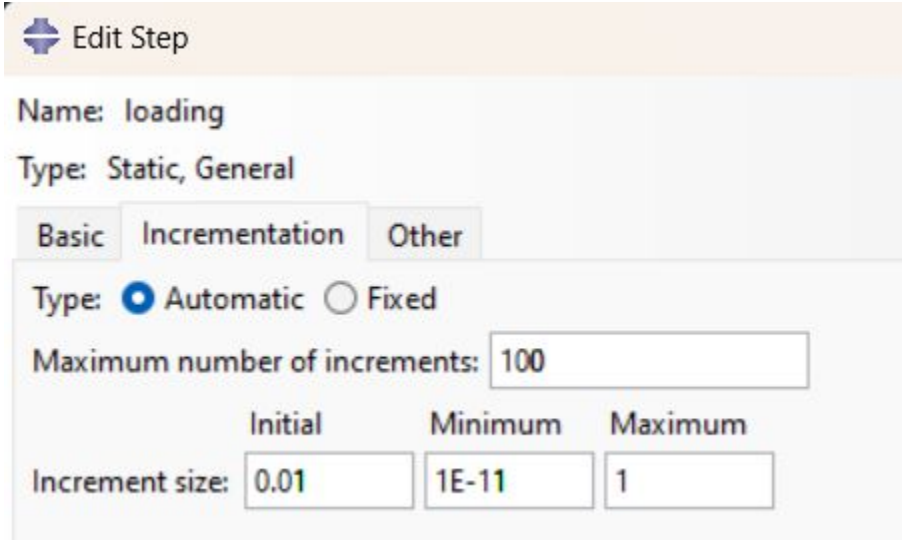


Job: Job-15 Status: Aborted

Step	Increment	Att	Severe Discon Iter	Equil Iter	Total Iter
1	1	5U	0	5	5
1	1	6U	0	5	5
1	1	7U	0	5	5
1	1	8U	0	5	5
1	1	9U	0	5	5

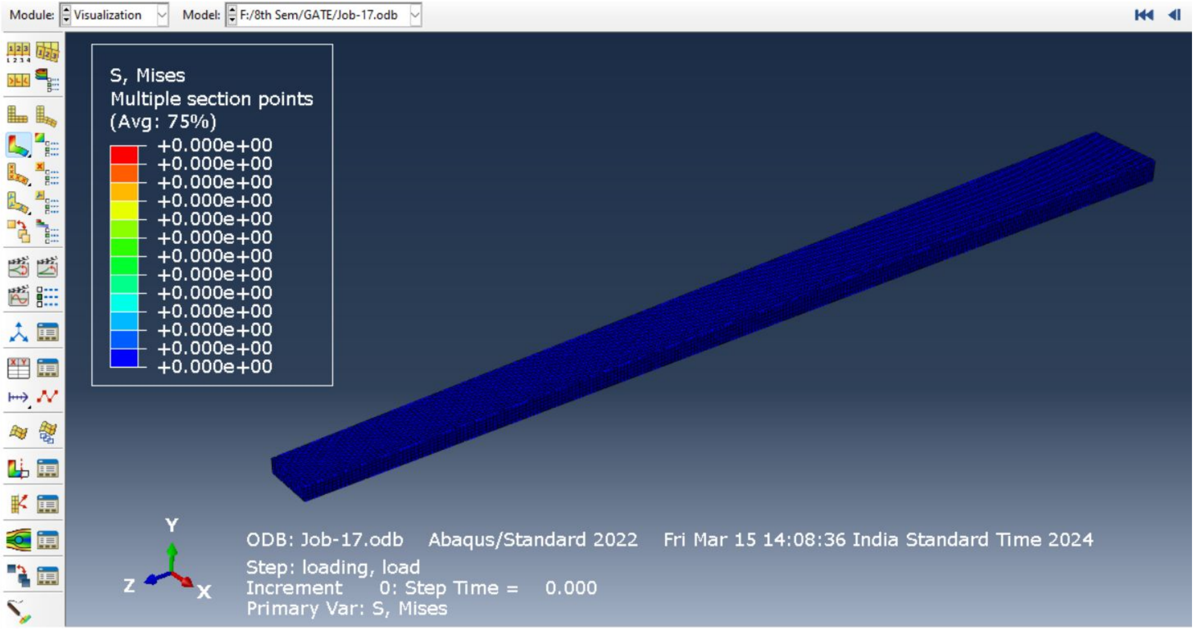
Log | ! Errors | ! Warnings | Output | Data File | Message File

Time increment required is less than the minimum specified



Visualization(errors)

Another reason could have been meshing, hence we made finer meshing for all corners but still same error occurred.



We have tried all the solutions available on internet for potential reasons for such error but unable to solve this issue
Link for cae file of this model Lab 5 Group 11

We solved the meshing error(due to which we couldn’t submit on Thu) by making new model but, since we could successfully mesh the model, there are hardly any problems with model(and hence I didn’t modeled again from scratch)

Thank You

AE462
Exercise 6

Date
18/04/2024

Instructor
Prof. G.M. Kamath

Group 11
Aryaman Badkul
Palmate Aditya Tukaram
Shrutikirti Singh
Prabuddha Singh
Shikha Sharma
Rahangdale Prachi Sanjay

EXERCISE 6 Calculation of Buckling Margins

Calculation of Critical Stress for Wing Skin:

From Chapter A18, the equation for the elastic instability of flat sheet in compressio[n] is,

$$\sigma_{cr} = \frac{\pi^2 k_c E}{12(1-\nu_e^2)} \left(\frac{t}{b}\right)^2 \quad \text{------- (C5.1)}$$

Where k_c = buckling coefficient which depends on edge boundary conditions and sheet aspect ratio (a/b)

E = modulus of elasticity

ν_e = elastic Poisson's ratio

b = short dimension of plate or loaded edge

t = sheet thickness

For our case,

E=71.1GPa

ν_e = 0.3

t = 0.0000372 m

b = spar distance (as already calculated in previous report)

For Calculation of K_c

- Firstly, we calculated the value of a/b, where a = 0.5 (rib spacing); b = interspar distance.
- Then we got the value of k_c using the attached plot.
  Here, we took the clamped case as both ends are fixed.

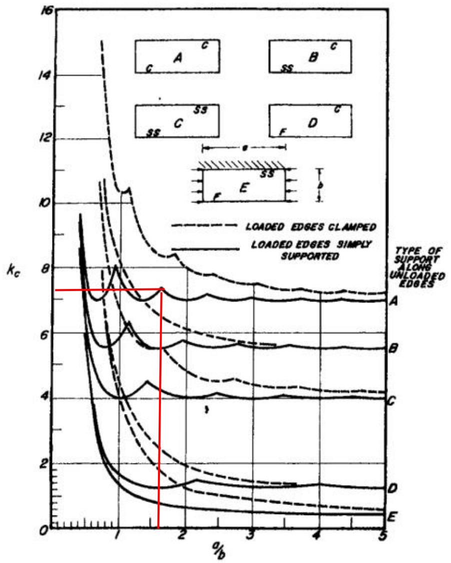
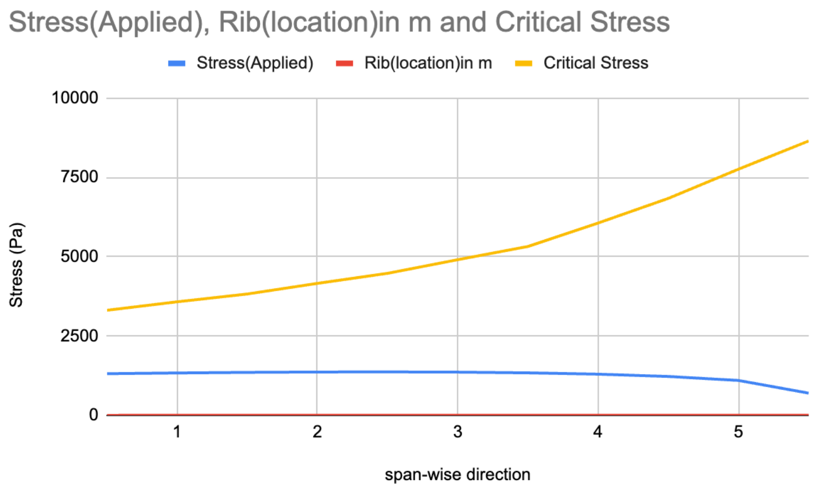
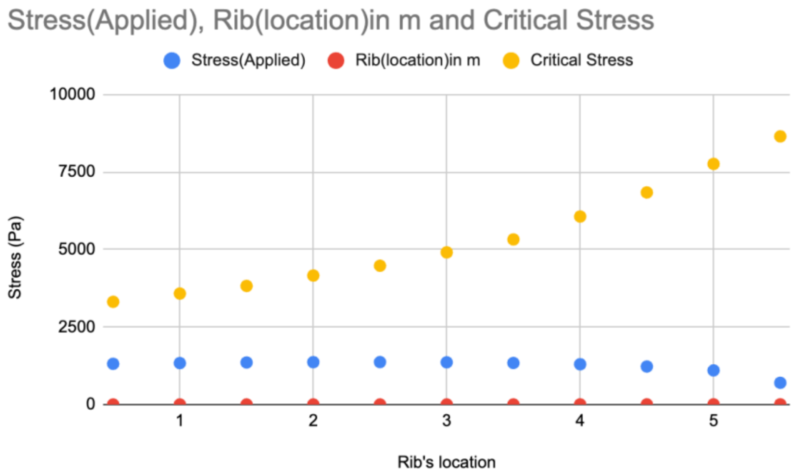


Fig. C5. 2 (Ref. 1) Compressive-buckling coefficients for flat rectangular plates.

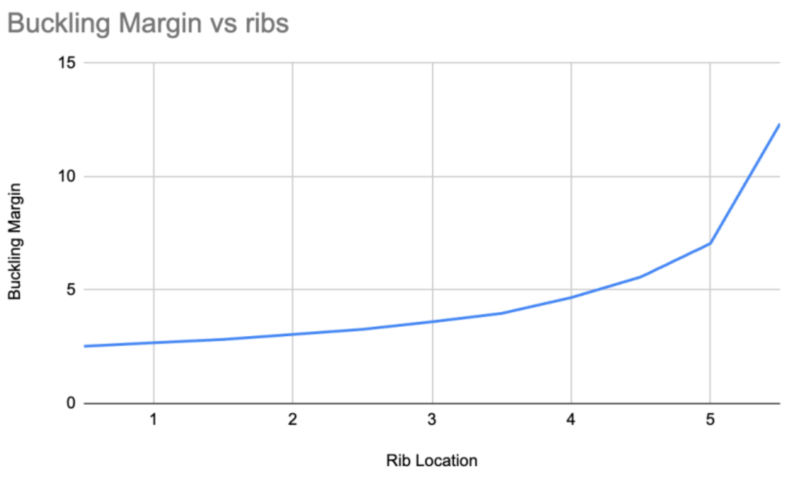
Plot of Applied Stress and Critical Stress

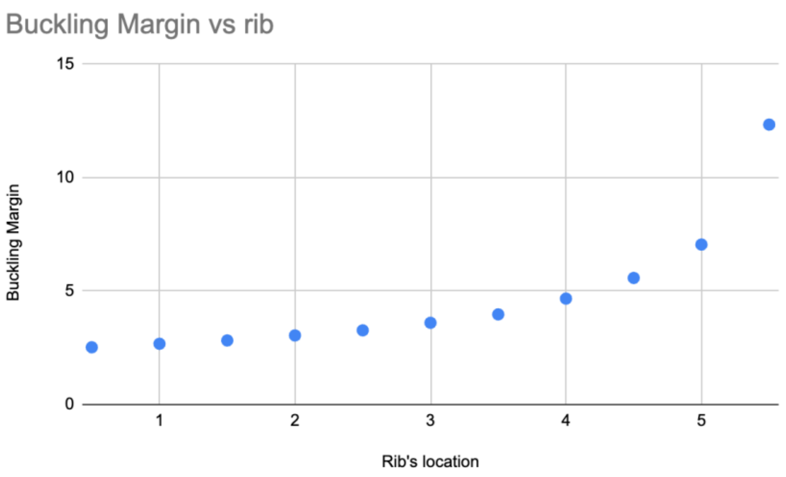




Calculation of Buckling Margin:

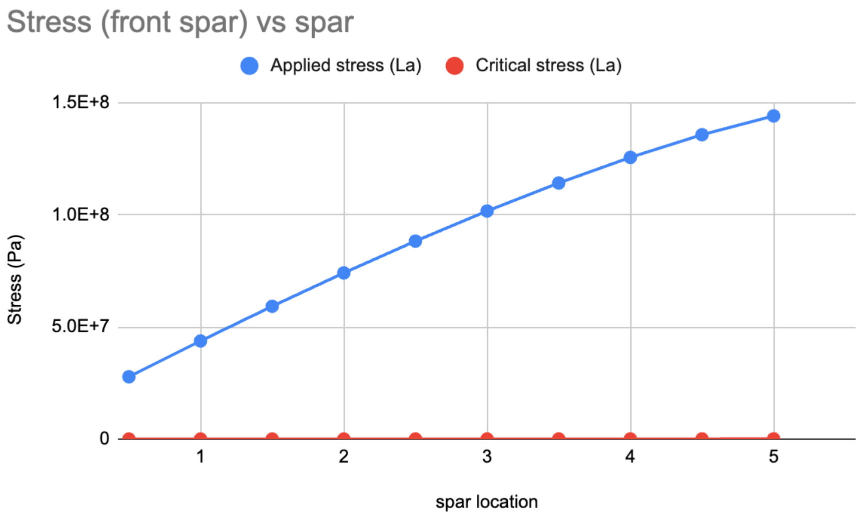
Buckling Margin = critical stress/ applied stress

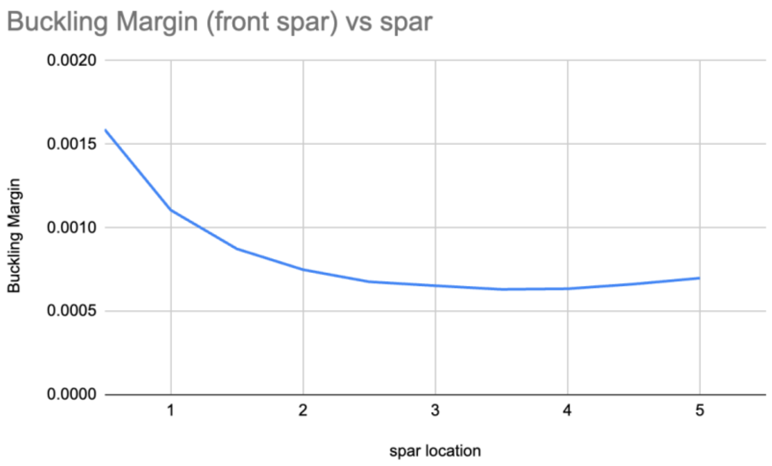




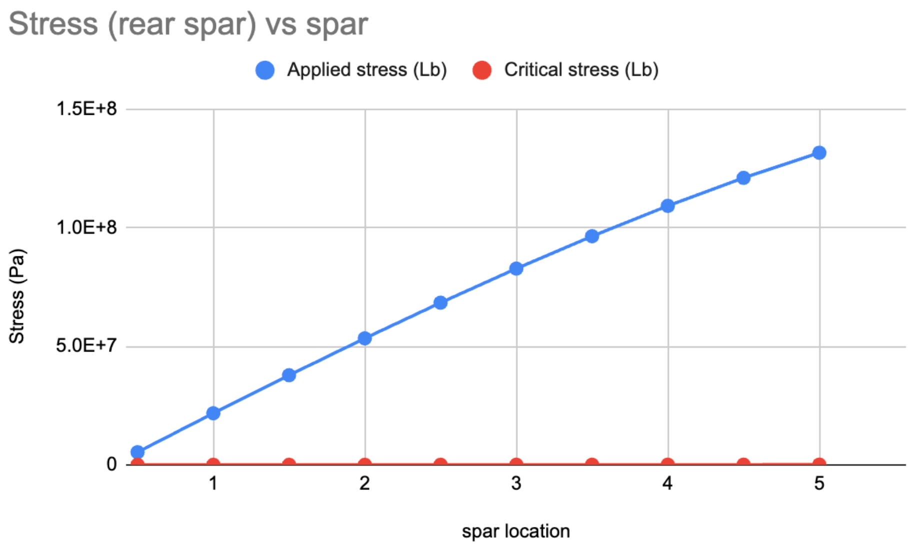
Similarly for front spar and rear spar

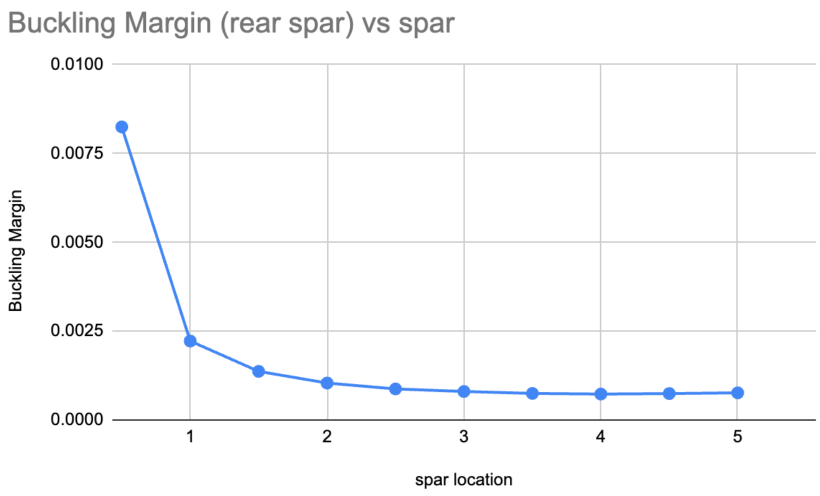
Plots for front spar:





Plots for rear spar:





Assuming that the wing-fuselage joint has two lugs each on the top and bottom (see sketch and figure in attachments), calculate the forces in each of the lugs and based on these forces, check for the lug failure margins for **axial failure, shear failure and bearing failure.** The methodology as discussed in class is provided in the relevant sections of Bruhn's text attached here. Use BOTH Method 1 and Method 2 as given in the attachment.

The material to be used is Al2124 and its properties are provided in the attachments. The initial trial dimensions for the lug are provided in the attachment. (Note: this is just a typical set of dimensions and so this could be too strong or too weak depending on the type of aircraft you are designing)

Deliverables: Prepare a presentation which is clear and self-explanatory which includes the details of the load used for the design, the methodology used and the margins obtained based on it for the final lug dimensions (NOT the initial dimensions given!).

Axial failure

Failure in Tension

$$P_{u(tension)} = F_{tu}\,(2R - D)t \quad \text{- - - - - - - - - - (2)}$$

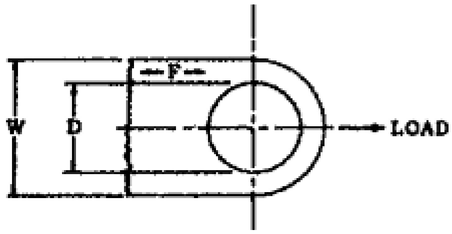
where F_{tu} = ultimate tensile strength of plate material.

R=75mm
D=25mm
t=25mm
F_{tu}=489.52 MPa
$P_u = F_{tu}*(0.15-0.05)*0.025 = 1.2238 * 10^6$ N
Tension across the net section

$$P_u = K_t\, F_{tu}\, A_t \quad \text{- - - - - - - - - - - - - - - - (6)}$$

where K_t is the stress concentration factor as found from Fig. D1.12 and Table D1.3. F_{tu} = ultimate tensile strength of the material and A_t = net tension area.

$P_u = K_t F_{tu} A_t$
$A_t = (0.15-0.05)*0.025$
F_{tu}=489.52 MPa



For W/D=150mm/50mm=3
K_t=0.9
P_u=1.1014*10^6 N

Shear Failure

Failure by bolt shear

$$P_{u(\text{bolt shear})} = F_{su} \cdot A \cdot 2 \quad \text{- - - - - - - - - - - (1)}$$

where F_{su} = ultimate shearing stress for bolt material.
A = cross-sectional area of bolt.

P_u=F_{su}*A*2
A=Bolt area=pi*0.025^2=0.0019625
F_{su}= 331 MPa
P_u=1.299* 10^6 N

Bearing Failure

Failure by bearing of bolt bushing

$$P_u = F_{br}\, D\, t \quad \text{- - - - - - - - - - - - - - - -}$$

where F_{br} = allowable bearing stress
D = diameter of bushing
t = plate thickness

P_u=F_{br}*D*t
P_u=489.52*10^6*0.05*0.025=6.119*10^5 N

Shear out Bearing Strength

$$P_{bru} = K_{bru} F_{tu} A_{br}$$ - - - - - - - - - - - - -

The values of K_{bru}, the shear-bearing efficiency factor, is given by curves in Fig D1.13.

$P_{bru}=K_{bru}*F_{tu}*A_{br}$

$P_{bru}=0.6*489.52*10^6*(0.05*0.025)=3.67*10^5$ N