

From: [REDACTED]
To: [REDACTED]
Subject: Fwd:
Date: Friday, October 29, 2021 8:31:59 PM
Attachments: [Cycle Dump.txt](#)
[N844X - 2021-10-27 21.17.58.png](#)

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Mike,

Here is Austin's write up. He is pretty busy now. I will edit it for our report.

Have a great weekend!

Jeff

----- Forwarded message -----

From: Austin Meyer [REDACTED]
Date: Wed, Oct 27, 2021 at 7:25 PM
Subject: Re:
To: Jeff Edwards [REDACTED]

OK let's just get this one out of the way first. X-Plane is based on a technique called "Blade Element Theory". It's a concept of breaking propeller blades into many elements, finding the air speed and angle of attack on each element, and then looking up the coefficients of lift and drag and pitch-moment and for the airfoil(s) on the propeller blade at that location. X-Plane does this for not just any propeller blades your airplane might have, but also for every wing and stabilizer! Once X-Plane has found all of these speed and coefficients, it then corrects for finite-wing effects, ground effect, Mach compression (though hopefully not both at the same time! ;-P) and bunch of other factors, and uses that to find the forces on each bit of the airplane. X-Plane then adds up all those little forces to get the total forces on the plane and uses that to compute the accelerations, and integrate those accelerations to velocities and velocities to positions. X-Plane has been doing this since 1995.

I entered an Evo into X-Plane and test-flew it against my actual airplane. The results were incredibly similar.

One interesting note is that flaps and ailerons both change the effective angle of attack and lift coefficient of the span-fraction of the wing to which they are attached, which causes the lift on any given section of wing to be affected by the flap or aileron, and that lift is applied at the approximate spanwise location that it applies in reality.

Given these things, I expected the effect of asymmetric aileron deflection (indeed, the most common kind there is!) and asymmetric flap deflection (hopefully less-common!) to be simulated fairly accurately.

Knowing this, I flew my Evo in X-Plane and induced a left flap failure, causing it to remain in the ZERO DEFLECTION position.

Flying an Evo at 90 knots in X-Plane and locking the LEFT flap at ZERO, and and RIGHT flap at the take-

off setting, it takes FULL RIGHT aileron and almost 20% RIGHT RUDDER to hold the wings level in roll.

Flying an Evo at 90 knots in X-Plane and locking the LEFT flap at ZERO, and and RIGHT flap at the FULL setting, there is NO CONTROL INPUT that will hold the wings level: The roll was extremely quick.

Check my math here for the moment in time described above if desired!

A lot of math (the propeller aloe is prolly the first 75% of the file), but it's all there!

austin

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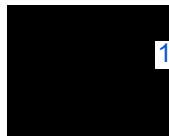
W. Jeffrey Edwards, Ph.D.
AvSafe, LLC



McCall, ID 83638



St. Louis, MO 63146-2032



cell

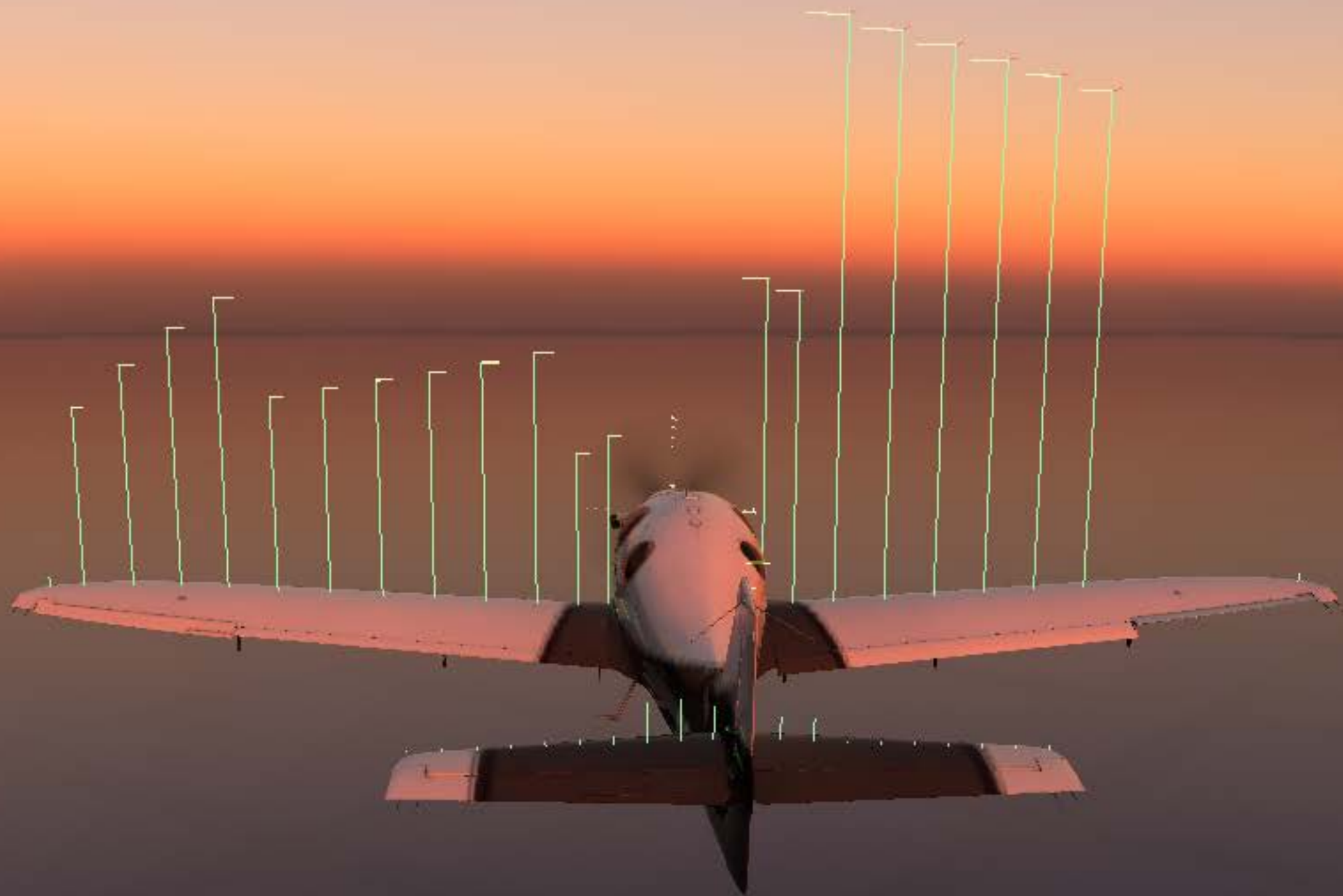
1 office

fax

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elev	ailrn	ruddr	nwhel
-0.1060	1.0000	0.1768	-4.3708
surf	surf	surf	steer

<Element Forces>



.....
X-Plane, by Austin Meyer

120000

Simulating Aircraft/Laminar Research/Lancair Evolution/N844X.acf
.....

x location positive aft x force positive right pitch/alpha pos nose up
y location positive right y force positive up roll pos right
z location positive up z force positive aft yaw/beta pos nose right

elevator, aileron, spoiler positive control surface up
rudder positive control surface right
drag-yaw positive control surface deployed
pitch cyclic prop pitch positive request nose up
roll cyclic prop pitch positive request nose right

Finite-wing and build-up for prop 1: Wing area for this wing= 0.1887

After any wing-joining, our area is 0.19 m.
After any wing-joining, our semi-length is 1.04 m.
After any wing-joining, our root chord is 0.20 m.
After any wing-joining, our tip chord is 0.20 m.
After any wing-joining, our leading-edge sweep is 4.1 deg.
After any wing-joining, our mean aero sweep is 4.1 deg.
After any wing-joining, our aspect ratio is 11.4494.
After any wing-joining, spanwise centroid is 0.4813 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.8902.
After any wing-joining, our delta-wing ratio is 0.0000.
Oswalds efficiency is therefore 0.9302

Finite-wing and build-up for Left Wing 1: Wing area for this wing= 0.7713

NOTE! I AM ADDING Left Wing 2 TO THE TIP OF Left Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=-1.0941, y=-0.2863, z=0.1517, loop=0/48)
NOTE! I AM ADDING Left Wing 3 TO THE TIP OF Left Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=-3.7672, y=-0.0761, z=0.2273, loop=0/48)
NOTE! I AM ADDING Left Wing 4 TO THE TIP OF Left Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-5.3387, y=0.0475, z=0.3943, loop=0/48)
After any wing-joining, our area is 5.55 m.
After any wing-joining, our semi-length is 5.10 m.
After any wing-joining, our root chord is 1.51 m.
After any wing-joining, our tip chord is 0.30 m.
After any wing-joining, our leading-edge sweep is 10.0 deg.
After any wing-joining, our mean aero sweep is 6.7 deg.
After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 1: Wing area for this wing= 0.7713

NOTE! I AM ADDING Right Wing 2 TO THE TIP OF Right Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=1.0941, y=-0.2863, z=0.1517, loop=0/48)
NOTE! I AM ADDING Right Wing 3 TO THE TIP OF Right Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=3.7672, y=-0.0761, z=0.2273, loop=0/48)
NOTE! I AM ADDING Right Wing 4 TO THE TIP OF Right Wing 1 (through any intermediate wings) TO

FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=5.3387, y=0.0475, z=0.3943, loop=0/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Wing 2: Wing area for this wing= 3.2281

NOTE! I AM ADDING Left Wing 1 TO THE ROOT OF Left Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=-1.0942, y=-0.2865, z=0.1524, loop=0/48)

NOTE! I AM ADDING Left Wing 3 TO THE TIP OF Left Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=-3.7672, y=-0.0761, z=0.2273, loop=0/48)

NOTE! I AM ADDING Left Wing 4 TO THE TIP OF Left Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-5.3387, y=0.0475, z=0.3943, loop=0/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 2: Wing area for this wing= 3.2281

NOTE! I AM ADDING Right Wing 1 TO THE ROOT OF Right Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=1.0942, y=-0.2865, z=0.1524, loop=0/48)

NOTE! I AM ADDING Right Wing 3 TO THE TIP OF Right Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=3.7672, y=-0.0761, z=0.2273, loop=0/48)

NOTE! I AM ADDING Right Wing 4 TO THE TIP OF Right Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=5.3387, y=0.0475, z=0.3943, loop=0/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Wing 3: Wing area for this wing= 1.4237

NOTE! I AM ADDING Left Wing 2 TO THE ROOT OF Left Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.6517, connecting at x=-3.7673, y=-0.0762, z=0.2286, loop=0/48)

NOTE! I AM ADDING Left Wing 4 TO THE TIP OF Left Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=-5.3387, y=0.0475, z=0.3943, loop=0/48)

NOTE! I AM ADDING Left Wing 1 TO THE ROOT OF Left Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-1.0942, y=-0.2865, z=0.1524, loop=1/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 3: Wing area for this wing= 1.4237

NOTE! I AM ADDING Right Wing 2 TO THE ROOT OF Right Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.6517, connecting at x=3.7673, y=-0.0762, z=0.2286, loop=0/48)

NOTE! I AM ADDING Right Wing 4 TO THE TIP OF Right Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=5.3387, y=0.0475, z=0.3943, loop=0/48)

NOTE! I AM ADDING Right Wing 1 TO THE ROOT OF Right Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=1.0942, y=-0.2865, z=0.1524, loop=1/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Wing 4: Wing area for this wing= 0.1280

NOTE! I AM ADDING Left Wing 3 TO THE ROOT OF Left Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=1.5517, connecting at x=-5.3401, y=0.0488, z=0.3932, loop=0/48)

NOTE! I AM ADDING Left Wing 2 TO THE ROOT OF Left Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=-3.7673, y=-0.0762, z=0.2286, loop=1/48)

NOTE! I AM ADDING Left Wing 1 TO THE ROOT OF Left Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-1.0942, y=-0.2865, z=0.1524, loop=2/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 4: Wing area for this wing= 0.1280

NOTE! I AM ADDING Right Wing 3 TO THE ROOT OF Right Wing 4 (through any intermediate wings)
TO FORM ONE CONTINUOUS WING! (S=1.5517, connecting at x=5.3401, y=0.0488, z=0.3932, loop=0/48)
NOTE! I AM ADDING Right Wing 2 TO THE ROOT OF Right Wing 4 (through any intermediate wings)
TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=3.7673, y=-0.0762, z=0.2286, loop=1/48)
NOTE! I AM ADDING Right Wing 1 TO THE ROOT OF Right Wing 4 (through any intermediate wings)
TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=1.0942, y=-0.2865, z=0.1524, loop=2/48)

After any wing-joining, our area is 5.55 m.
After any wing-joining, our semi-length is 5.10 m.
After any wing-joining, our root chord is 1.51 m.
After any wing-joining, our tip chord is 0.30 m.
After any wing-joining, our leading-edge sweep is 10.0 deg.
After any wing-joining, our mean aero sweep is 6.7 deg.
After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Stab: Wing area for this wing= 1.7065

After any wing-joining, our area is 1.71 m.
After any wing-joining, our semi-length is 2.04 m.
After any wing-joining, our root chord is 1.04 m.
After any wing-joining, our tip chord is 0.64 m.
After any wing-joining, our leading-edge sweep is 7.3 deg.
After any wing-joining, our mean aero sweep is 4.5 deg.
After any wing-joining, our aspect ratio is 4.8577.
After any wing-joining, spanwise centroid is 0.4610 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.7706.
After any wing-joining, our delta-wing ratio is 0.1311.
Oswalds efficiency is therefore 0.9819

Finite-wing and build-up for Right Stab: Wing area for this wing= 1.7065

After any wing-joining, our area is 1.71 m.
After any wing-joining, our semi-length is 2.04 m.
After any wing-joining, our root chord is 1.04 m.
After any wing-joining, our tip chord is 0.64 m.
After any wing-joining, our leading-edge sweep is 7.3 deg.
After any wing-joining, our mean aero sweep is 4.5 deg.
After any wing-joining, our aspect ratio is 4.8577.
After any wing-joining, spanwise centroid is 0.4610 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.7706.
After any wing-joining, our delta-wing ratio is 0.1311.
Oswalds efficiency is therefore 0.9819

Finite-wing and build-up for Vert Stab 1: Wing area for this wing= 1.9381

After any wing-joining, our area is 1.94 m.
After any wing-joining, our semi-length is 2.04 m.
After any wing-joining, our root chord is 1.68 m.
After any wing-joining, our tip chord is 0.58 m.
After any wing-joining, our leading-edge sweep is 38.7 deg.
After any wing-joining, our mean aero sweep is 32.7 deg.
After any wing-joining, our aspect ratio is 3.0476.
After any wing-joining, spanwise centroid is 0.4197 semi-spans. (this is area centroid fraction of

the way out to the tip)

After any wing-joining, our taper ratio is 0.5278.

After any wing-joining, our delta-wing ratio is 0.3648.

Oswalds efficiency is therefore 0.9995

reference area for total coefficients=14.5150 m*m. (total wing area of the craft, based on all flying surfaces with a dihedral of less than 45 degrees, meaning wings and horizontal stabs but not vertical stabs)

average mean aerodynamic chord= 1.02 m. (average mean aerodynamic chord of the craft, based on all flying surfaces with a dihedral of less than 45 degrees, meaning wings and horizontal stabs but not vertical stabs)

total wing span= 11.19 m. (wingspan of the craft, checking all flying surfaces with a dihedral of less than 45 degrees, meaning wings and horizontal stabs but not vertical stabs)

The prop 1 DISC AREA is broken down into rings for propwash distribution as follows:

element number 0, rad1= 0.00, rad2= 0.11, ring area= 0.0369 square meters

element number 1, rad1= 0.11, rad2= 0.21, ring area= 0.0996 square meters

element number 2, rad1= 0.21, rad2= 0.31, ring area= 0.1691 square meters

element number 3, rad1= 0.31, rad2= 0.42, ring area= 0.2375 square meters

element number 4, rad1= 0.42, rad2= 0.52, ring area= 0.3054 square meters

element number 5, rad1= 0.52, rad2= 0.62, ring area= 0.3733 square meters

element number 6, rad1= 0.62, rad2= 0.73, ring area= 0.4413 square meters

element number 7, rad1= 0.73, rad2= 0.83, ring area= 0.5100 square meters

element number 8, rad1= 0.83, rad2= 0.94, ring area= 0.5810 square meters

element number 9, rad1= 0.94, rad2= 1.04, ring area= 0.6568 square meters

TOTAL DISC AREA= 3.4109 square meters

en=0, el=0, element_mass= 1.131 veh_prop_ptr(en)->prop_mass=1.131, rad_arm_mtr= 0.07,
veh_prop_ptr(en)->prop_mi_rpm= 0.006 kg*m*m

en=0, el=1, element_mass= 1.006 veh_prop_ptr(en)->prop_mass=2.137, rad_arm_mtr= 0.18,
veh_prop_ptr(en)->prop_mi_rpm= 0.038 kg*m*m

en=0, el=2, element_mass= 1.203 veh_prop_ptr(en)->prop_mass=3.340, rad_arm_mtr= 0.28,
veh_prop_ptr(en)->prop_mi_rpm= 0.133 kg*m*m

en=0, el=3, element_mass= 1.336 veh_prop_ptr(en)->prop_mass=4.675, rad_arm_mtr= 0.39,
veh_prop_ptr(en)->prop_mi_rpm= 0.331 kg*m*m

en=0, el=4, element_mass= 1.390 veh_prop_ptr(en)->prop_mass=6.066, rad_arm_mtr= 0.49,
veh_prop_ptr(en)->prop_mi_rpm= 0.664 kg*m*m

en=0, el=5, element_mass= 1.390 veh_prop_ptr(en)->prop_mass=7.456, rad_arm_mtr= 0.59,
veh_prop_ptr(en)->prop_mi_rpm= 1.153 kg*m*m

en=0, el=6, element_mass= 1.349 veh_prop_ptr(en)->prop_mass=8.805, rad_arm_mtr= 0.70,
veh_prop_ptr(en)->prop_mi_rpm= 1.809 kg*m*m

en=0, el=7, element_mass= 1.178 veh_prop_ptr(en)->prop_mass=9.983, rad_arm_mtr= 0.80,
veh_prop_ptr(en)->prop_mi_rpm= 2.565 kg*m*m

en=0, el=8, element_mass= 0.871 veh_prop_ptr(en)->prop_mass=10.854, rad_arm_mtr= 0.90,
veh_prop_ptr(en)->prop_mi_rpm= 3.278 kg*m*m

en=0, el=9, element_mass= 0.476 veh_prop_ptr(en)->prop_mass=11.330, rad_arm_mtr= 1.01,
veh_prop_ptr(en)->prop_mi_rpm= 3.762 kg*m*m

BEFORE any tip weights, the prop 1 weighs 11.330 kg, and the MI is 3.762 kg*m*m

AFTER any tip weights, the prop 1 weighs 11.330 kg, and the MI is 3.762 kg*m*m

Critical altitude for recip and turboprop engines=14999 ft, rho at crit alt= 0.7711 kg/mmm, rho at SL= 1.2250 kg/mmm, engine power boosted to 1.5886 of original, spinning motor MI 7.2250 kg*m*m

Engine # 0, is mass 173.499 kg at x= 0.00, y= 0.03, z= -2.76

Prop # 0, is mass 11.330 kg at x= 0.00, y= 0.03, z= -2.76

Left Wing 1 Running wetted area 0.3085 square meters for all parts listed so far

of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 0.29 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 3.21 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Left Stab Running wetted area 5.1234 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 3.02 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 1.76 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 3.48 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Right Stab Running wetted area 5.8060 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 2.86 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.32 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 3.67 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Vert Stab 1 Running wetted area 6.5812 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 2.71 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.77 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 3.83 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Body 1 (fuselage):
The frontal area is 1.5802 square meters
The side area is 7.9058 square meters
The top area is 6.4788 square meters
The lateral centroid is 0.00 meters
The vertical centroid is 0.03 meters
The longitudinal centroid for lift at 0 slip is 2.00 meters
The longitudinal centroid for lift at 180 slip is 6.13 meters

Running wetted area 31.0750 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 1.34 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.25 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 2.56 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Body 2 (engine to body fairing):
The frontal area is 0.1317 square meters
The side area is 0.6998 square meters
The top area is 1.1839 square meters
The lateral centroid is 0.00 meters
The vertical centroid is 0.02 meters
The longitudinal centroid for lift at 0 slip is 0.61 meters
The longitudinal centroid for lift at 180 slip is 1.83 meters

Running wetted area 34.7864 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 1.27 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.17 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 2.45 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Body 22 (serpra e):
The frontal area is 0.2607 square meters
The side area is 0.0880 square meters
The top area is 0.4737 square meters
The lateral centroid is 0.00 meters
The vertical centroid is 0.00 meters
The longitudinal centroid for lift at 0 slip is 0.32 meters
The longitudinal centroid for lift at 180 slip is 0.53 meters

Running wetted area 36.1385 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 1.25 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.17 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 2.44 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Sum of all engines have mass of 173.50 kg (16% of total):
The radius of gyration of the engines in roll is 0.03 m, based on all engine masses and locations
The radius of gyration of the engines in pitch is 2.76 m, based on all engine masses and locations
The radius of gyration of the engines in yaw is 2.76 m, based on all engine masses and locations

Sum of all props have mass of 11.33 kg (1% of total):
The radius of gyration of the props in roll is 0.03 m, based on all prop masses and locations
The radius of gyration of the props in pitch is 2.76 m, based on all prop masses and locations
The radius of gyration of the props in yaw is 2.76 m, based on all prop masses and locations

Remainder of the craft has mass 903.79 (83% of total):
The radius of gyration of the shell in roll is 1.25 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces
The radius of gyration of the shell in pitch is 2.17 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces
The radius of gyration of the shell in yaw is 2.44 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Final combined results:
The radius of gyration in roll is 1.14 m
The radius of gyration in pitch is 2.28 m
The radius of gyration in yaw is 2.50 m

.....
X-Plane, by Austin Meyer
120000
Simulating Aircraft/Laminar Research/Lancair Evolution/N844X.acf
.....

x location positive aft x force positive right pitch/alpha pos nose up
y location positive right y force positive up roll pos right
z location positive up z force positive aft yaw/beta pos nose right

elevator, aileron, spoiler positive control surface up
rudder positive control surface right
drag-yaw positive control surface deployed
pitch cyclic prop pitch positive request nose up
roll cyclic prop pitch positive request nose right

Finite-wing and build-up for prop 1: Wing area for this wing= 0.1887

After any wing-joining, our area is 0.19 m.
After any wing-joining, our semi-length is 1.04 m.
After any wing-joining, our root chord is 0.20 m.
After any wing-joining, our tip chord is 0.20 m.
After any wing-joining, our leading-edge sweep is 4.1 deg.
After any wing-joining, our mean aero sweep is 4.1 deg.
After any wing-joining, our aspect ratio is 11.4494.
After any wing-joining, spanwise centroid is 0.4813 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.8902.
After any wing-joining, our delta-wing ratio is 0.0000.
Oswalds efficiency is therefore 0.9302

Finite-wing and build-up for Left Wing 1: Wing area for this wing= 0.7713

NOTE! I AM ADDING Left Wing 2 TO THE TIP OF Left Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=-1.0941, y=-0.2863, z=0.1517, loop=0/48)

NOTE! I AM ADDING Left Wing 3 TO THE TIP OF Left Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=-3.7672, y=-0.0761, z=0.2273, loop=0/48)

NOTE! I AM ADDING Left Wing 4 TO THE TIP OF Left Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-5.3387, y=0.0475, z=0.3943, loop=0/48)

After any wing-joining, our area is 5.55 m.
After any wing-joining, our semi-length is 5.10 m.
After any wing-joining, our root chord is 1.51 m.
After any wing-joining, our tip chord is 0.30 m.
After any wing-joining, our leading-edge sweep is 10.0 deg.
After any wing-joining, our mean aero sweep is 6.7 deg.
After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 1: Wing area for this wing= 0.7713

NOTE! I AM ADDING Right Wing 2 TO THE TIP OF Right Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=1.0941, y=-0.2863, z=0.1517, loop=0/48)

NOTE! I AM ADDING Right Wing 3 TO THE TIP OF Right Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=3.7672, y=-0.0761, z=0.2273, loop=0/48)

NOTE! I AM ADDING Right Wing 4 TO THE TIP OF Right Wing 1 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=5.3387, y=0.0475, z=0.3943, loop=0/48)

After any wing-joining, our area is 5.55 m.
After any wing-joining, our semi-length is 5.10 m.
After any wing-joining, our root chord is 1.51 m.
After any wing-joining, our tip chord is 0.30 m.
After any wing-joining, our leading-edge sweep is 10.0 deg.
After any wing-joining, our mean aero sweep is 6.7 deg.
After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Wing 2: Wing area for this wing= 3.2281

NOTE! I AM ADDING Left Wing 1 TO THE ROOT OF Left Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=-1.0942, y=-0.2865, z=0.1524, loop=0/48)

NOTE! I AM ADDING Left Wing 3 TO THE TIP OF Left Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=-3.7672, y=-0.0761, z=0.2273, loop=0/48)

NOTE! I AM ADDING Left Wing 4 TO THE TIP OF Left Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-5.3387, y=0.0475, z=0.3943, loop=0/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 2: Wing area for this wing= 3.2281

NOTE! I AM ADDING Right Wing 1 TO THE ROOT OF Right Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=3.9993, connecting at x=1.0942, y=-0.2865, z=0.1524, loop=0/48)

NOTE! I AM ADDING Right Wing 3 TO THE TIP OF Right Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.4230, connecting at x=3.7672, y=-0.0761, z=0.2273, loop=0/48)

NOTE! I AM ADDING Right Wing 4 TO THE TIP OF Right Wing 2 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=5.3387, y=0.0475, z=0.3943, loop=0/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.

After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)

After any wing-joining, our taper ratio is 0.6619.

After any wing-joining, our delta-wing ratio is 0.0287.

Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Wing 3: Wing area for this wing= 1.4237

NOTE! I AM ADDING Left Wing 2 TO THE ROOT OF Left Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.6517, connecting at x=-3.7673, y=-0.0762, z=0.2286, loop=0/48)

NOTE! I AM ADDING Left Wing 4 TO THE TIP OF Left Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=-5.3387, y=0.0475, z=0.3943, loop=0/48)

NOTE! I AM ADDING Left Wing 1 TO THE ROOT OF Left Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-1.0942, y=-0.2865, z=0.1524, loop=1/48)

After any wing-joining, our area is 5.55 m.

After any wing-joining, our semi-length is 5.10 m.

After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.

After any wing-joining, our leading-edge sweep is 10.0 deg.

After any wing-joining, our mean aero sweep is 6.7 deg.

After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 3: Wing area for this wing= 1.4237

NOTE! I AM ADDING Right Wing 2 TO THE ROOT OF Right Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.6517, connecting at x=3.7673, y=-0.0762, z=0.2286, loop=0/48)

NOTE! I AM ADDING Right Wing 4 TO THE TIP OF Right Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=5.3387, y=0.0475, z=0.3943, loop=0/48)

NOTE! I AM ADDING Right Wing 1 TO THE ROOT OF Right Wing 3 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=1.0942, y=-0.2865, z=0.1524, loop=1/48)

After any wing-joining, our area is 5.55 m.
After any wing-joining, our semi-length is 5.10 m.
After any wing-joining, our root chord is 1.51 m.
After any wing-joining, our tip chord is 0.30 m.
After any wing-joining, our leading-edge sweep is 10.0 deg.
After any wing-joining, our mean aero sweep is 6.7 deg.
After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Wing 4: Wing area for this wing= 0.1280

NOTE! I AM ADDING Left Wing 3 TO THE ROOT OF Left Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=1.5517, connecting at x=-5.3401, y=0.0488, z=0.3932, loop=0/48)

NOTE! I AM ADDING Left Wing 2 TO THE ROOT OF Left Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=-3.7673, y=-0.0762, z=0.2286, loop=1/48)

NOTE! I AM ADDING Left Wing 1 TO THE ROOT OF Left Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=-1.0942, y=-0.2865, z=0.1524, loop=2/48)

After any wing-joining, our area is 5.55 m.
After any wing-joining, our semi-length is 5.10 m.
After any wing-joining, our root chord is 1.51 m.
After any wing-joining, our tip chord is 0.30 m.
After any wing-joining, our leading-edge sweep is 10.0 deg.
After any wing-joining, our mean aero sweep is 6.7 deg.
After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Right Wing 4: Wing area for this wing= 0.1280

NOTE! I AM ADDING Right Wing 3 TO THE ROOT OF Right Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=1.5517, connecting at x=5.3401, y=0.0488, z=0.3932, loop=0/48)

NOTE! I AM ADDING Right Wing 2 TO THE ROOT OF Right Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=4.7797, connecting at x=3.7673, y=-0.0762, z=0.2286, loop=1/48)

NOTE! I AM ADDING Right Wing 1 TO THE ROOT OF Right Wing 4 (through any intermediate wings) TO FORM ONE CONTINUOUS WING! (S=5.5510, connecting at x=1.0942, y=-0.2865, z=0.1524, loop=2/48)

After any wing-joining, our area is 5.55 m.
After any wing-joining, our semi-length is 5.10 m.
After any wing-joining, our root chord is 1.51 m.

After any wing-joining, our tip chord is 0.30 m.
After any wing-joining, our leading-edge sweep is 10.0 deg.
After any wing-joining, our mean aero sweep is 6.7 deg.
After any wing-joining, our aspect ratio is 9.2364.
After any wing-joining, spanwise centroid is 0.4425 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.6619.
After any wing-joining, our delta-wing ratio is 0.0287.
Oswalds efficiency is therefore 0.9704

Finite-wing and build-up for Left Stab: Wing area for this wing= 1.7065

After any wing-joining, our area is 1.71 m.
After any wing-joining, our semi-length is 2.04 m.
After any wing-joining, our root chord is 1.04 m.
After any wing-joining, our tip chord is 0.64 m.
After any wing-joining, our leading-edge sweep is 7.3 deg.
After any wing-joining, our mean aero sweep is 4.5 deg.
After any wing-joining, our aspect ratio is 4.8577.
After any wing-joining, spanwise centroid is 0.4610 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.7706.
After any wing-joining, our delta-wing ratio is 0.1311.
Oswalds efficiency is therefore 0.9819

Finite-wing and build-up for Right Stab: Wing area for this wing= 1.7065

After any wing-joining, our area is 1.71 m.
After any wing-joining, our semi-length is 2.04 m.
After any wing-joining, our root chord is 1.04 m.
After any wing-joining, our tip chord is 0.64 m.
After any wing-joining, our leading-edge sweep is 7.3 deg.
After any wing-joining, our mean aero sweep is 4.5 deg.
After any wing-joining, our aspect ratio is 4.8577.
After any wing-joining, spanwise centroid is 0.4610 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.7706.
After any wing-joining, our delta-wing ratio is 0.1311.
Oswalds efficiency is therefore 0.9819

Finite-wing and build-up for Vert Stab 1: Wing area for this wing= 1.9381

After any wing-joining, our area is 1.94 m.
After any wing-joining, our semi-length is 2.04 m.
After any wing-joining, our root chord is 1.68 m.
After any wing-joining, our tip chord is 0.58 m.
After any wing-joining, our leading-edge sweep is 38.7 deg.
After any wing-joining, our mean aero sweep is 32.7 deg.
After any wing-joining, our aspect ratio is 3.0476.
After any wing-joining, spanwise centroid is 0.4197 semi-spans. (this is area centroid fraction of the way out to the tip)
After any wing-joining, our taper ratio is 0.5278.
After any wing-joining, our delta-wing ratio is 0.3648.
Oswalds efficiency is therefore 0.9995

reference area for total coefficients=14.5150 m*m. (total wing area of the craft, based on all flying surfaces with a dihedral of less than 45 degrees, meaning wings and horizontal stabs but not vertical stabs)
average mean aerodynamic chord= 1.02 m. (average mean aerodynamic chord of the craft, based on all flying surfaces with a dihedral of less than 45 degrees, meaning wings and horizontal stabs but not vertical

stabs)

total wing span= 11.19 m. (wingspan of the craft, checking all flying surfaces with a dihedral of less than 45 degrees, meaning wings and horizontal stabs but not vertical stabs)

The prop 1 DISC AREA is broken down into rings for propwash distribution as follows:

element number 0, rad1= 0.00, rad2= 0.11, ring area= 0.0369 square meters
element number 1, rad1= 0.11, rad2= 0.21, ring area= 0.0996 square meters
element number 2, rad1= 0.21, rad2= 0.31, ring area= 0.1691 square meters
element number 3, rad1= 0.31, rad2= 0.42, ring area= 0.2375 square meters
element number 4, rad1= 0.42, rad2= 0.52, ring area= 0.3054 square meters
element number 5, rad1= 0.52, rad2= 0.62, ring area= 0.3733 square meters
element number 6, rad1= 0.62, rad2= 0.73, ring area= 0.4413 square meters
element number 7, rad1= 0.73, rad2= 0.83, ring area= 0.5100 square meters
element number 8, rad1= 0.83, rad2= 0.94, ring area= 0.5810 square meters
element number 9, rad1= 0.94, rad2= 1.04, ring area= 0.6568 square meters

TOTAL DISC AREA= 3.4109 square meters

en=0, el=0, element_mass= 1.131 veh_prop_ptr(en)->prop_mass=1.131, rad_arm_mtr= 0.07,
veh_prop_ptr(en)->prop_mi_rpm= 0.006 kg*m*m
en=0, el=1, element_mass= 1.006 veh_prop_ptr(en)->prop_mass=2.137, rad_arm_mtr= 0.18,
veh_prop_ptr(en)->prop_mi_rpm= 0.038 kg*m*m
en=0, el=2, element_mass= 1.203 veh_prop_ptr(en)->prop_mass=3.340, rad_arm_mtr= 0.28,
veh_prop_ptr(en)->prop_mi_rpm= 0.133 kg*m*m
en=0, el=3, element_mass= 1.336 veh_prop_ptr(en)->prop_mass=4.675, rad_arm_mtr= 0.39,
veh_prop_ptr(en)->prop_mi_rpm= 0.331 kg*m*m
en=0, el=4, element_mass= 1.390 veh_prop_ptr(en)->prop_mass=6.066, rad_arm_mtr= 0.49,
veh_prop_ptr(en)->prop_mi_rpm= 0.664 kg*m*m
en=0, el=5, element_mass= 1.390 veh_prop_ptr(en)->prop_mass=7.456, rad_arm_mtr= 0.59,
veh_prop_ptr(en)->prop_mi_rpm= 1.153 kg*m*m
en=0, el=6, element_mass= 1.349 veh_prop_ptr(en)->prop_mass=8.805, rad_arm_mtr= 0.70,
veh_prop_ptr(en)->prop_mi_rpm= 1.809 kg*m*m
en=0, el=7, element_mass= 1.178 veh_prop_ptr(en)->prop_mass=9.983, rad_arm_mtr= 0.80,
veh_prop_ptr(en)->prop_mi_rpm= 2.565 kg*m*m
en=0, el=8, element_mass= 0.871 veh_prop_ptr(en)->prop_mass=10.854, rad_arm_mtr= 0.90,
veh_prop_ptr(en)->prop_mi_rpm= 3.278 kg*m*m
en=0, el=9, element_mass= 0.476 veh_prop_ptr(en)->prop_mass=11.330, rad_arm_mtr= 1.01,
veh_prop_ptr(en)->prop_mi_rpm= 3.762 kg*m*m
BEFORE any tip weights, the prop 1 weighs 11.330 kg, and the MI is 3.762 kg*m*m
AFTER any tip weights, the prop 1 weighs 11.330 kg, and the MI is 3.762 kg*m*m

Critical altitude for recip and turboprop engines=14999 ft, rho at crit alt= 0.7711 kg/mmm, rho at SL= 1.2250 kg/mmm, engine power boosted to 1.5886 of original, spinning motor MI 7.2250 kg*m*m

Engine # 0, is mass 173.499 kg at x= 0.00, y= 0.03, z= -2.76

Prop # 0, is mass 11.330 kg at x= 0.00, y= 0.03, z= -2.76

Left Wing 1 Running wetted area 0.3085 square meters for all parts listed so far

Running radius of gyration of the shell in roll is 0.95 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Running radius of gyration of the shell in pitch is 0.32 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Running radius of gyration of the shell in yaw is 0.90 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Right Wing 1 Running wetted area 0.6170 square meters for all parts listed so far

Running radius of gyration of the shell in roll is 0.95 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 3.48 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Right Stab Running wetted area 5.8060 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 2.86 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.32 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 3.67 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Vert Stab 1 Running wetted area 6.5812 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 2.71 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.77 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 3.83 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Body 1 (fuselage):

The frontal	area	is 1.5802 square meters
The side	area	is 7.9058 square meters
The top	area	is 6.4788 square meters
The lateral	centroid	is 0.00 meters
The vertical	centroid	is 0.03 meters
The longitudinal centroid for lift at	0 slip	is 2.00 meters
The longitudinal centroid for lift at	180 slip	is 6.13 meters

Running wetted area 31.0750 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 1.34 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.25 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 2.56 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Body 2 (engine to body fairing):

The frontal	area	is 0.1317 square meters
The side	area	is 0.6998 square meters
The top	area	is 1.1839 square meters
The lateral	centroid	is 0.00 meters
The vertical	centroid	is 0.02 meters
The longitudinal centroid for lift at	0 slip	is 0.61 meters
The longitudinal centroid for lift at	180 slip	is 1.83 meters

Running wetted area 34.7864 square meters for all parts listed so far
Running radius of gyration of the shell in roll is 1.27 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in pitch is 2.17 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces
Running radius of gyration of the shell in yaw is 2.45 m, based on mass-distribution across the shell
of the craft with a 20% weighting per square meter on all of the flying surfaces

Body 22 (serpra e):

The frontal	area	is 0.2607 square meters
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The side area is 0.0880 square meters
The top area is 0.4737 square meters
The lateral centroid is 0.00 meters
The vertical centroid is 0.00 meters
The longitudinal centroid for lift at 0 slip is 0.32 meters
The longitudinal centroid for lift at 180 slip is 0.53 meters

Running wetted area 36.1385 square meters for all parts listed so far

Running radius of gyration of the shell in roll is 1.25 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Running radius of gyration of the shell in pitch is 2.17 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Running radius of gyration of the shell in yaw is 2.44 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Sum of all engines have mass of 173.50 kg (16% of total):

The radius of gyration of the engines in roll is 0.03 m, based on all engine masses and locations

The radius of gyration of the engines in pitch is 2.76 m, based on all engine masses and locations

The radius of gyration of the engines in yaw is 2.76 m, based on all engine masses and locations

Sum of all props have mass of 11.33 kg (1% of total):

The radius of gyration of the props in roll is 0.03 m, based on all prop masses and locations

The radius of gyration of the props in pitch is 2.76 m, based on all prop masses and locations

The radius of gyration of the props in yaw is 2.76 m, based on all prop masses and locations

Remainder of the craft has mass 903.79 (83% of total):

The radius of gyration of the shell in roll is 1.25 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

The radius of gyration of the shell in pitch is 2.17 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

The radius of gyration of the shell in yaw is 2.44 m, based on mass-distribution across the shell of the craft with a 20% weighting per square meter on all of the flying surfaces

Final combined results:

The radius of gyration in roll is 1.14 m

The radius of gyration in pitch is 2.28 m

The radius of gyration in yaw is 2.50 m

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==AT TIME=
54.19=====
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flight control ptch:-0.0949 ratio
flight control hdng: 0.2397 ratio
flight control roll: 1.0000 ratio

h : 4243 m
T : -13 deg C
sigma : 0.6519 sigma ratio

rho : 0.7985 kg/cubic meter, density ratio: 0.6519
vx : -55.5
vy : -0.2
vz : 18.7
mach-1 : 323.6 m/s
airspeed: 58.6 m/s
mach : 0.1812

psi : 254.4 deg
the : 4.2 deg
phi : -2.3 deg

alpha : 4.3 deg (pos nose up)
beta : 3.2 deg (pos nose right)

P : -0.2 deg/second
Q : -0.8 deg/second
R : -0.9 deg/second

tire height above ground : 4243 m
resting height of aircraft above ground: 1 m

prop 1 connecting to prop 1, el=0, Cl=-0.2959, V= 41.0, clQ_total=496.8863, clQ_x_arm=24.8488
prop 1 connecting to prop 1, el=1, Cl= 0.5336, V= 51.0, clQ_total=1882.3137, clQ_x_arm=232.6629
prop 1 connecting to prop 1, el=2, Cl= 0.1853, V= 77.9, clQ_total=3007.8311, clQ_x_arm=514.0424
prop 1 connecting to prop 1, el=3, Cl= 0.2622, V= 94.7, clQ_total=5359.7178, clQ_x_arm=1337.2028
prop 1 connecting to prop 1, el=4, Cl= 0.2558, V= 112.8, clQ_total=8612.9834, clQ_x_arm=2801.1724
prop 1 connecting to prop 1, el=5, Cl= 0.2167, V= 131.6, clQ_total=12364.3076, clQ_x_arm=4864.4004
prop 1 connecting to prop 1, el=6, Cl= 0.1676, V= 150.9, clQ_total=16179.8281, clQ_x_arm=7344.4893
prop 1 connecting to prop 1, el=7, Cl= 0.1184, V= 169.8, clQ_total=19594.0371, clQ_x_arm=9905.1455
prop 1 connecting to prop 1, el=8, Cl= 0.0694, V= 188.1, clQ_total=22049.9805, clQ_x_arm=11992.6982
prop 1 connecting to prop 1, el=9, Cl= 0.0166, V= 205.1, clQ_total=22748.2070, clQ_x_arm=12656.0137
clQ_x_arm/clQ_total= 0.5564, root_to_centroid= 0.5564, centroid_to_tip= 0.4436, AR_mult= 0.8873

prop 1 connecting to prop 1, el=0, Cl= 0.1783, V= 44.5, clQ_total=353.2754, clQ_x_arm=17.6683
prop 1 connecting to prop 1, el=1, Cl= 0.7850, V= 55.6, clQ_total=2778.8865, clQ_x_arm=381.5100
prop 1 connecting to prop 1, el=2, Cl= 0.4102, V= 84.9, clQ_total=5733.3315, clQ_x_arm=1120.1213
prop 1 connecting to prop 1, el=3, Cl= 0.4108, V= 102.1, clQ_total=10014.0312, clQ_x_arm=2618.3667
prop 1 connecting to prop 1, el=4, Cl= 0.3600, V= 120.3, clQ_total=15224.9863, clQ_x_arm=4963.2969
prop 1 connecting to prop 1, el=5, Cl= 0.2941, V= 139.2, clQ_total=20923.4844, clQ_x_arm=8097.4707
prop 1 connecting to prop 1, el=6, Cl= 0.2283, V= 158.5, clQ_total=26661.5000, clQ_x_arm=11827.1807
prop 1 connecting to prop 1, el=7, Cl= 0.1689, V= 177.4, clQ_total=31979.9648, clQ_x_arm=15816.0293
prop 1 connecting to prop 1, el=8, Cl= 0.1136, V= 195.7, clQ_total=36329.5391, clQ_x_arm=19513.1680
prop 1 connecting to prop 1, el=9, Cl= 0.0575, V= 212.5, clQ_total=38927.4727, clQ_x_arm=21981.2051
clQ_x_arm/clQ_total= 0.5647, root_to_centroid= 0.5647, centroid_to_tip= 0.4353, AR_mult= 0.8707

prop 1 connecting to prop 1, el=0, Cl= 0.1153, V= 43.9, clQ_total=221.9152, clQ_x_arm=11.1003
prop 1 connecting to prop 1, el=1, Cl= 0.7535, V= 54.8, clQ_total=2485.7390, clQ_x_arm=350.6738
prop 1 connecting to prop 1, el=2, Cl= 0.3825, V= 83.7, clQ_total=5167.6943, clQ_x_arm=1021.1628
prop 1 connecting to prop 1, el=3, Cl= 0.3952, V= 100.9, clQ_total=9188.6758, clQ_x_arm=2428.5063
prop 1 connecting to prop 1, el=4, Cl= 0.3514, V= 119.1, clQ_total=14171.1602, clQ_x_arm=4670.6240
prop 1 connecting to prop 1, el=5, Cl= 0.2896, V= 137.9, clQ_total=19682.4492, clQ_x_arm=7701.8320
prop 1 connecting to prop 1, el=6, Cl= 0.2257, V= 157.3, clQ_total=25267.5312, clQ_x_arm=11332.1348
prop 1 connecting to prop 1, el=7, Cl= 0.1667, V= 176.2, clQ_total=30446.0918, clQ_x_arm=15216.0557

prop 1 connecting to prop 1, el=8, Cl= 0.1105, V= 194.5, clQ_total=34628.3633, clQ_x_arm=18770.9863
prop 1 connecting to prop 1, el=9, Cl= 0.0533, V= 211.4, clQ_total=37011.0352, clQ_x_arm=21034.5234
clQ_x_arm/clQ_total= 0.5683, root_to_centroid= 0.5683, centroid_to_tip= 0.4317, AR_mult= 0.8633

prop 1 connecting to prop 1, el=0, Cl=-0.3745, V= 40.6, clQ_total=615.8473, clQ_x_arm=30.7969
prop 1 connecting to prop 1, el=1, Cl= 0.4891, V= 50.3, clQ_total=1855.1344, clQ_x_arm=216.6899
prop 1 connecting to prop 1, el=2, Cl= 0.1468, V= 77.0, clQ_total=2724.5977, clQ_x_arm=434.0558
prop 1 connecting to prop 1, el=3, Cl= 0.2379, V= 93.6, clQ_total=4809.7744, clQ_x_arm=1163.8677
prop 1 connecting to prop 1, el=4, Cl= 0.2405, V= 111.6, clQ_total=7806.4873, clQ_x_arm=2512.3884
prop 1 connecting to prop 1, el=5, Cl= 0.2071, V= 130.4, clQ_total=11328.2285, clQ_x_arm=4449.3462
prop 1 connecting to prop 1, el=6, Cl= 0.1615, V= 149.7, clQ_total=14947.3457, clQ_x_arm=6801.7720
prop 1 connecting to prop 1, el=7, Cl= 0.1138, V= 168.7, clQ_total=18183.5137, clQ_x_arm=9228.8975
prop 1 connecting to prop 1, el=8, Cl= 0.0648, V= 187.0, clQ_total=20450.9062, clQ_x_arm=11156.1816
prop 1 connecting to prop 1, el=9, Cl= 0.0111, V= 204.0, clQ_total=20914.2637, clQ_x_arm=11596.3711
clQ_x_arm/clQ_total= 0.5545, root_to_centroid= 0.5545, centroid_to_tip= 0.4455, AR_mult= 0.8911

Left Wing 1 connecting to Left Wing 1, el=0, Cl= 0.6640, V= 60.8, clQ_total=2452.7180,
clQ_x_arm=66.2251
Left Wing 1 connecting to Left Wing 1, el=1, Cl= 0.6117, V= 57.8, clQ_total=4497.4668,
clQ_x_arm=231.8771
Left Wing 1 connecting to Left Wing 2, el=0, Cl= 0.5709, V= 58.6, clQ_total=6456.8418,
clQ_x_arm=530.0029
Left Wing 1 connecting to Left Wing 2, el=1, Cl= 0.5618, V= 58.6, clQ_total=8384.4980,
clQ_x_arm=993.3415
Left Wing 1 connecting to Left Wing 2, el=2, Cl= 0.5527, V= 58.6, clQ_total=10280.4941,
clQ_x_arm=1616.2900
Left Wing 1 connecting to Left Wing 2, el=3, Cl= 0.5436, V= 58.6, clQ_total=12144.8145,
clQ_x_arm=2393.2305
Left Wing 1 connecting to Left Wing 2, el=4, Cl= 0.5345, V= 58.6, clQ_total=13977.4658,
clQ_x_arm=3318.5544
Left Wing 1 connecting to Left Wing 2, el=5, Cl= 0.5253, V= 58.6, clQ_total=15778.4570,
clQ_x_arm=4386.6572
Left Wing 1 connecting to Left Wing 3, el=0, Cl= 0.8867, V= 58.5, clQ_total=18817.8594,
clQ_x_arm=6441.3813
Left Wing 1 connecting to Left Wing 3, el=1, Cl= 0.8635, V= 58.5, clQ_total=21777.0215,
clQ_x_arm=8671.8311
Left Wing 1 connecting to Left Wing 3, el=2, Cl= 0.8460, V= 58.4, clQ_total=24658.5781,
clQ_x_arm=11067.6924
Left Wing 1 connecting to Left Wing 3, el=3, Cl= 0.8398, V= 58.0, clQ_total=27486.8125,
clQ_x_arm=13638.9482
Left Wing 1 connecting to Left Wing 4, el=0, Cl= 0.4871, V= 52.6, clQ_total=28836.0391,
clQ_x_arm=14935.4482
Left Wing 1 connecting to Left Wing 4, el=1, Cl= 0.4756, V= 41.2, clQ_total=29642.7324,
clQ_x_arm=15731.0361
clQ_x_arm/clQ_total= 0.5307, root_to_centroid= 0.5307, centroid_to_tip= 0.4693, AR_mult= 0.9386

Right Wing 1 connecting to Right Wing 1, el=0, Cl= 1.1059, V= 59.4, clQ_total=3898.2764,
clQ_x_arm=105.2534
Right Wing 1 connecting to Right Wing 1, el=1, Cl= 1.1917, V= 56.3, clQ_total=7670.2554,
clQ_x_arm=410.8341
Right Wing 1 connecting to Right Wing 2, el=0, Cl= 1.2475, V= 58.4, clQ_total=11926.2305,
clQ_x_arm=1058.3959
Right Wing 1 connecting to Right Wing 2, el=1, Cl= 1.2428, V= 58.4, clQ_total=16167.3477,
clQ_x_arm=2077.8066
Right Wing 1 connecting to Right Wing 2, el=2, Cl= 1.2383, V= 58.4, clQ_total=20394.2578,
clQ_x_arm=3466.6006
Right Wing 1 connecting to Right Wing 2, el=3, Cl= 1.2340, V= 58.4, clQ_total=24607.7773,

clQ_x_arm=5222.5513
Right Wing 1 connecting to Right Wing 2, el=4, Cl= 1.2300, V= 58.4, clQ_total=28808.7637,
clQ_x_arm=7343.6709
Right Wing 1 connecting to Right Wing 2, el=5, Cl= 1.2263, V= 58.4, clQ_total=32998.1602,
clQ_x_arm=9828.2480
Right Wing 1 connecting to Right Wing 3, el=0, Cl= 0.0139, V= 58.4, clQ_total=33045.6367,
clQ_x_arm=9860.3428
Right Wing 1 connecting to Right Wing 3, el=1, Cl= 0.0238, V= 58.2, clQ_total=33126.1094,
clQ_x_arm=9920.9990
Right Wing 1 connecting to Right Wing 3, el=2, Cl= 0.0258, V= 57.6, clQ_total=33211.6367,
clQ_x_arm=9992.1094
Right Wing 1 connecting to Right Wing 3, el=3, Cl= 0.0127, V= 57.0, clQ_total=33252.7344,
clQ_x_arm=10029.4727
Right Wing 1 connecting to Right Wing 4, el=0, Cl= 0.4765, V= 49.6, clQ_total=34425.6289,
clQ_x_arm=11156.5332
Right Wing 1 connecting to Right Wing 4, el=1, Cl= 0.4647, V= 36.4, clQ_total=35042.4531,
clQ_x_arm=11764.8672
clQ_x_arm/clQ_total= 0.3357, root_to_centroid= 0.3357, centroid_to_tip= 0.6643, AR_mult= 1.3285

Left Wing 2 connecting to Left Wing 2, el=0, Cl= 0.5709, V= 58.6, clQ_total=1959.3848,
clQ_x_arm=298.1309
Left Wing 2 connecting to Left Wing 2, el=1, Cl= 0.5618, V= 58.6, clQ_total=3887.0408,
clQ_x_arm=761.4694
Left Wing 2 connecting to Left Wing 2, el=2, Cl= 0.5527, V= 58.6, clQ_total=5783.0371,
clQ_x_arm=1384.4180
Left Wing 2 connecting to Left Wing 2, el=3, Cl= 0.5436, V= 58.6, clQ_total=7647.3569,
clQ_x_arm=2161.3584
Left Wing 2 connecting to Left Wing 2, el=4, Cl= 0.5345, V= 58.6, clQ_total=9480.0078,
clQ_x_arm=3086.6824
Left Wing 2 connecting to Left Wing 2, el=5, Cl= 0.5253, V= 58.6, clQ_total=11280.9990,
clQ_x_arm=4154.7852
Left Wing 2 connecting to Left Wing 1, el=0, Cl= 0.6640, V= 60.8, clQ_total=13733.7070,
clQ_x_arm=4221.0054
Left Wing 2 connecting to Left Wing 1, el=1, Cl= 0.6117, V= 57.8, clQ_total=15778.4561,
clQ_x_arm=4386.6572
Left Wing 2 connecting to Left Wing 3, el=0, Cl= 0.8867, V= 58.5, clQ_total=18817.8574,
clQ_x_arm=6441.3813
Left Wing 2 connecting to Left Wing 3, el=1, Cl= 0.8635, V= 58.5, clQ_total=21777.0195,
clQ_x_arm=8671.8311
Left Wing 2 connecting to Left Wing 3, el=2, Cl= 0.8460, V= 58.4, clQ_total=24658.5762,
clQ_x_arm=11067.6924
Left Wing 2 connecting to Left Wing 3, el=3, Cl= 0.8398, V= 58.0, clQ_total=27486.8086,
clQ_x_arm=13638.9482
Left Wing 2 connecting to Left Wing 4, el=0, Cl= 0.4871, V= 52.6, clQ_total=28836.0352,
clQ_x_arm=14935.4482
Left Wing 2 connecting to Left Wing 4, el=1, Cl= 0.4756, V= 41.2, clQ_total=29642.7285,
clQ_x_arm=15731.0361
clQ_x_arm/clQ_total= 0.5307, root_to_centroid= 0.5307, centroid_to_tip= 0.4693, AR_mult= 0.9386

Right Wing 2 connecting to Right Wing 2, el=0, Cl= 1.2475, V= 58.4, clQ_total=4255.9849,
clQ_x_arm=647.5668
Right Wing 2 connecting to Right Wing 2, el=1, Cl= 1.2428, V= 58.4, clQ_total=8497.1016,
clQ_x_arm=1666.9775
Right Wing 2 connecting to Right Wing 2, el=2, Cl= 1.2383, V= 58.4, clQ_total=12724.0127,
clQ_x_arm=3055.7715
Right Wing 2 connecting to Right Wing 2, el=3, Cl= 1.2340, V= 58.4, clQ_total=16937.5312,
clQ_x_arm=4811.7222

Right Wing 2 connecting to Right Wing 2, el=4, Cl= 1.2300, V= 58.4, clQ_total=21138.5176,
clQ_x_arm=6932.8418
Right Wing 2 connecting to Right Wing 2, el=5, Cl= 1.2263, V= 58.4, clQ_total=25327.9121,
clQ_x_arm=9417.4189
Right Wing 2 connecting to Right Wing 1, el=0, Cl= 1.1059, V= 59.4, clQ_total=29226.1777,
clQ_x_arm=9522.6670
Right Wing 2 connecting to Right Wing 1, el=1, Cl= 1.1917, V= 56.3, clQ_total=32998.1562,
clQ_x_arm=9828.2480
Right Wing 2 connecting to Right Wing 3, el=0, Cl= 0.0139, V= 58.4, clQ_total=33045.6328,
clQ_x_arm=9860.3428
Right Wing 2 connecting to Right Wing 3, el=1, Cl= 0.0238, V= 58.2, clQ_total=33126.1055,
clQ_x_arm=9920.9990
Right Wing 2 connecting to Right Wing 3, el=2, Cl= 0.0258, V= 57.6, clQ_total=33211.6328,
clQ_x_arm=9992.1094
Right Wing 2 connecting to Right Wing 3, el=3, Cl= 0.0127, V= 57.0, clQ_total=33252.7305,
clQ_x_arm=10029.4727
Right Wing 2 connecting to Right Wing 4, el=0, Cl= 0.4765, V= 49.6, clQ_total=34425.6250,
clQ_x_arm=11156.5332
Right Wing 2 connecting to Right Wing 4, el=1, Cl= 0.4647, V= 36.4, clQ_total=35042.4492,
clQ_x_arm=11764.8672
clQ_x_arm/clQ_total= 0.3357, root_to_centroid= 0.3357, centroid_to_tip= 0.6643, AR_mult= 1.3285

Left Wing 3 connecting to Left Wing 3, el=0, Cl= 0.8867, V= 58.5, clQ_total=3039.4121,
clQ_x_arm=2054.7290
Left Wing 3 connecting to Left Wing 3, el=1, Cl= 0.8635, V= 58.5, clQ_total=5998.5752,
clQ_x_arm=4285.1787
Left Wing 3 connecting to Left Wing 3, el=2, Cl= 0.8460, V= 58.4, clQ_total=8880.1328,
clQ_x_arm=6681.0400
Left Wing 3 connecting to Left Wing 3, el=3, Cl= 0.8398, V= 58.0, clQ_total=11708.3662,
clQ_x_arm=9252.2959
Left Wing 3 connecting to Left Wing 2, el=0, Cl= 0.5709, V= 58.6, clQ_total=13667.7412,
clQ_x_arm=9550.4219
Left Wing 3 connecting to Left Wing 2, el=1, Cl= 0.5618, V= 58.6, clQ_total=15595.3975,
clQ_x_arm=10013.7607
Left Wing 3 connecting to Left Wing 2, el=2, Cl= 0.5527, V= 58.6, clQ_total=17491.3945,
clQ_x_arm=10636.7090
Left Wing 3 connecting to Left Wing 2, el=3, Cl= 0.5436, V= 58.6, clQ_total=19355.7148,
clQ_x_arm=11413.6494
Left Wing 3 connecting to Left Wing 2, el=4, Cl= 0.5345, V= 58.6, clQ_total=21188.3672,
clQ_x_arm=12338.9736
Left Wing 3 connecting to Left Wing 2, el=5, Cl= 0.5253, V= 58.6, clQ_total=22989.3594,
clQ_x_arm=13407.0762
Left Wing 3 connecting to Left Wing 4, el=0, Cl= 0.4871, V= 52.6, clQ_total=24338.5859,
clQ_x_arm=14703.5762
Left Wing 3 connecting to Left Wing 4, el=1, Cl= 0.4756, V= 41.2, clQ_total=25145.2793,
clQ_x_arm=15499.1641
Left Wing 3 connecting to Left Wing 1, el=0, Cl= 0.6640, V= 60.8, clQ_total=27597.9883,
clQ_x_arm=15565.3838
Left Wing 3 connecting to Left Wing 1, el=1, Cl= 0.6117, V= 57.8, clQ_total=29642.7383,
clQ_x_arm=15731.0361
clQ_x_arm/clQ_total= 0.5307, root_to_centroid= 0.5307, centroid_to_tip= 0.4693, AR_mult= 0.9386

Right Wing 3 connecting to Right Wing 3, el=0, Cl= 0.0139, V= 58.4, clQ_total=47.4847,
clQ_x_arm=32.0993
Right Wing 3 connecting to Right Wing 3, el=1, Cl= 0.0238, V= 58.2, clQ_total=127.9575,
clQ_x_arm=92.7551
Right Wing 3 connecting to Right Wing 3, el=2, Cl= 0.0258, V= 57.6, clQ_total=213.4831,

clQ_x_arm=163.8651
Right Wing 3 connecting to Right Wing 3, el=3, Cl= 0.0127, V= 57.0, clQ_total=254.5810,
clQ_x_arm=201.2288
Right Wing 3 connecting to Right Wing 2, el=0, Cl= 1.2475, V= 58.4, clQ_total=4510.5562,
clQ_x_arm=848.7906
Right Wing 3 connecting to Right Wing 2, el=1, Cl= 1.2428, V= 58.4, clQ_total=8751.6738,
clQ_x_arm=1868.2013
Right Wing 3 connecting to Right Wing 2, el=2, Cl= 1.2383, V= 58.4, clQ_total=12978.5850,
clQ_x_arm=3256.9951
Right Wing 3 connecting to Right Wing 2, el=3, Cl= 1.2340, V= 58.4, clQ_total=17192.1055,
clQ_x_arm=5012.9458
Right Wing 3 connecting to Right Wing 2, el=4, Cl= 1.2300, V= 58.4, clQ_total=21393.0918,
clQ_x_arm=7134.0654
Right Wing 3 connecting to Right Wing 2, el=5, Cl= 1.2263, V= 58.4, clQ_total=25582.4863,
clQ_x_arm=9618.6426
Right Wing 3 connecting to Right Wing 4, el=0, Cl= 0.4765, V= 49.6, clQ_total=26755.3828,
clQ_x_arm=10745.7031
Right Wing 3 connecting to Right Wing 4, el=1, Cl= 0.4647, V= 36.4, clQ_total=27372.2090,
clQ_x_arm=11354.0371
Right Wing 3 connecting to Right Wing 1, el=0, Cl= 1.1059, V= 59.4, clQ_total=31270.4746,
clQ_x_arm=11459.2852
Right Wing 3 connecting to Right Wing 1, el=1, Cl= 1.1917, V= 56.3, clQ_total=35042.4531,
clQ_x_arm=11764.8662
clQ_x_arm/clQ_total= 0.3357, root_to_centroid= 0.3357, centroid_to_tip= 0.6643, AR_mult= 1.3285

Left Wing 4 connecting to Left Wing 4, el=0, Cl= 0.4871, V= 52.6, clQ_total=1349.2369,
clQ_x_arm=1296.5054
Left Wing 4 connecting to Left Wing 4, el=1, Cl= 0.4756, V= 41.2, clQ_total=2155.9294,
clQ_x_arm=2092.0930
Left Wing 4 connecting to Left Wing 3, el=0, Cl= 0.8867, V= 58.5, clQ_total=5195.3315,
clQ_x_arm=4146.8174
Left Wing 4 connecting to Left Wing 3, el=1, Cl= 0.8635, V= 58.5, clQ_total=8154.4941,
clQ_x_arm=6377.2676
Left Wing 4 connecting to Left Wing 3, el=2, Cl= 0.8460, V= 58.4, clQ_total=11036.0508,
clQ_x_arm=8773.1289
Left Wing 4 connecting to Left Wing 3, el=3, Cl= 0.8398, V= 58.0, clQ_total=13864.2842,
clQ_x_arm=11344.3848
Left Wing 4 connecting to Left Wing 2, el=0, Cl= 0.5709, V= 58.6, clQ_total=15823.6592,
clQ_x_arm=11642.5107
Left Wing 4 connecting to Left Wing 2, el=1, Cl= 0.5618, V= 58.6, clQ_total=17751.3145,
clQ_x_arm=12105.8496
Left Wing 4 connecting to Left Wing 2, el=2, Cl= 0.5527, V= 58.6, clQ_total=19647.3105,
clQ_x_arm=12728.7979
Left Wing 4 connecting to Left Wing 2, el=3, Cl= 0.5436, V= 58.6, clQ_total=21511.6309,
clQ_x_arm=13505.7383
Left Wing 4 connecting to Left Wing 2, el=4, Cl= 0.5345, V= 58.6, clQ_total=23344.2812,
clQ_x_arm=14431.0625
Left Wing 4 connecting to Left Wing 2, el=5, Cl= 0.5253, V= 58.6, clQ_total=25145.2734,
clQ_x_arm=15499.1650
Left Wing 4 connecting to Left Wing 1, el=0, Cl= 0.6640, V= 60.8, clQ_total=27597.9805,
clQ_x_arm=15565.3848
Left Wing 4 connecting to Left Wing 1, el=1, Cl= 0.6117, V= 57.8, clQ_total=29642.7305,
clQ_x_arm=15731.0371
clQ_x_arm/clQ_total= 0.5307, root_to_centroid= 0.5307, centroid_to_tip= 0.4693, AR_mult= 0.9386

Right Wing 4 connecting to Right Wing 4, el=0, Cl= 0.4765, V= 49.6, clQ_total=1172.9060,
clQ_x_arm=1127.0653

Right Wing 4 connecting to Right Wing 4, el=1, Cl= 0.4647, V= 36.4, clQ_total=1789.7316,
 clQ_x_arm=1735.3997
 Right Wing 4 connecting to Right Wing 3, el=0, Cl= 0.0139, V= 58.4, clQ_total=1837.2063,
 clQ_x_arm=1767.4939
 Right Wing 4 connecting to Right Wing 3, el=1, Cl= 0.0238, V= 58.2, clQ_total=1917.6791,
 clQ_x_arm=1828.1498
 Right Wing 4 connecting to Right Wing 3, el=2, Cl= 0.0258, V= 57.6, clQ_total=2003.2047,
 clQ_x_arm=1899.2598
 Right Wing 4 connecting to Right Wing 3, el=3, Cl= 0.0127, V= 57.0, clQ_total=2044.3026,
 clQ_x_arm=1936.6234
 Right Wing 4 connecting to Right Wing 2, el=0, Cl= 1.2475, V= 58.4, clQ_total=6300.2778,
 clQ_x_arm=2584.1851
 Right Wing 4 connecting to Right Wing 2, el=1, Cl= 1.2428, V= 58.4, clQ_total=10541.3945,
 clQ_x_arm=3603.5957
 Right Wing 4 connecting to Right Wing 2, el=2, Cl= 1.2383, V= 58.4, clQ_total=14768.3057,
 clQ_x_arm=4992.3896
 Right Wing 4 connecting to Right Wing 2, el=3, Cl= 1.2340, V= 58.4, clQ_total=18981.8242,
 clQ_x_arm=6748.3403
 Right Wing 4 connecting to Right Wing 2, el=4, Cl= 1.2300, V= 58.4, clQ_total=23182.8105,
 clQ_x_arm=8869.4600
 Right Wing 4 connecting to Right Wing 2, el=5, Cl= 1.2263, V= 58.4, clQ_total=27372.2051,
 clQ_x_arm=11354.0371
 Right Wing 4 connecting to Right Wing 1, el=0, Cl= 1.1059, V= 59.4, clQ_total=31270.4707,
 clQ_x_arm=11459.2852
 Right Wing 4 connecting to Right Wing 1, el=1, Cl= 1.1917, V= 56.3, clQ_total=35042.4492,
 clQ_x_arm=11764.8662
 clQ_x_arm/clQ_total= 0.3357, root_to_centroid= 0.3357, centroid_to_tip= 0.6643, AR_mult= 1.3285

Left Stab connecting to Left Stab, el=0, Cl= 0.2049, V= 56.3, clQ_total=649.1673, clQ_x_arm=32.4629
 Left Stab connecting to Left Stab, el=1, Cl= 0.2608, V= 56.7, clQ_total=1487.2842, clQ_x_arm=158.1804
 Left Stab connecting to Left Stab, el=2, Cl= 0.2678, V= 61.4, clQ_total=2496.3325, clQ_x_arm=410.4424
 Left Stab connecting to Left Stab, el=3, Cl= 0.0999, V= 60.4, clQ_total=2860.2031, clQ_x_arm=537.7971
 Left Stab connecting to Left Stab, el=4, Cl= 0.0730, V= 58.8, clQ_total=3112.7053, clQ_x_arm=651.4230
 Left Stab connecting to Left Stab, el=5, Cl= 0.0264, V= 58.6, clQ_total=3203.3892, clQ_x_arm=701.2990
 Left Stab connecting to Left Stab, el=6, Cl= 0.0270, V= 58.5, clQ_total=3295.6106, clQ_x_arm=761.2428
 Left Stab connecting to Left Stab, el=7, Cl= 0.0270, V= 58.5, clQ_total=3387.8616, clQ_x_arm=830.4309
 Left Stab connecting to Left Stab, el=8, Cl= 0.0274, V= 58.5, clQ_total=3481.6118, clQ_x_arm=910.1185
 Left Stab connecting to Left Stab, el=9, Cl= 0.0274, V= 58.5, clQ_total=3575.2698, clQ_x_arm=999.0933
 clQ_x_arm/clQ_total= 0.2794, root_to_centroid= 0.2794, centroid_to_tip= 0.7206, AR_mult= 1.4411

Right Stab connecting to Right Stab, el=0, Cl= 0.0885, V= 55.4, clQ_total=271.8307, clQ_x_arm=13.5960
 Right Stab connecting to Right Stab, el=1, Cl= 0.0344, V= 35.8, clQ_total=315.8506, clQ_x_arm=20.1990
 Right Stab connecting to Right Stab, el=2, Cl= 0.0298, V= 35.8, clQ_total=353.8976, clQ_x_arm=29.7108
 Right Stab connecting to Right Stab, el=3, Cl=-0.1634, V= 55.0, clQ_total=848.0238, clQ_x_arm=202.6549
 Right Stab connecting to Right Stab, el=4, Cl=-0.1333, V= 58.2, clQ_total=1299.7620,
 clQ_x_arm=405.9369
 Right Stab connecting to Right Stab, el=5, Cl=-0.0954, V= 58.1, clQ_total=1621.1200,
 clQ_x_arm=582.6836
 Right Stab connecting to Right Stab, el=6, Cl=-0.0943, V= 58.0, clQ_total=1938.2026,
 clQ_x_arm=788.7870
 Right Stab connecting to Right Stab, el=7, Cl=-0.0938, V= 58.0, clQ_total=2253.5085,
 clQ_x_arm=1025.2659
 Right Stab connecting to Right Stab, el=8, Cl=-0.0953, V= 58.0, clQ_total=2573.7144,
 clQ_x_arm=1297.4402
 Right Stab connecting to Right Stab, el=9, Cl=-0.0947, V= 58.0, clQ_total=2892.1807,
 clQ_x_arm=1599.9822
 clQ_x_arm/clQ_total= 0.5532, root_to_centroid= 0.5532, centroid_to_tip= 0.4468, AR_mult= 0.8936

Vert Stab 1 connecting to Vert Stab 1, el=0, Cl=-0.1997, V= 33.6, clQ_total=226.0763, clQ_x_arm=11.3083
 Vert Stab 1 connecting to Vert Stab 1, el=1, Cl=-0.1758, V= 44.7, clQ_total=578.0395, clQ_x_arm=64.1028
 Vert Stab 1 connecting to Vert Stab 1, el=2, Cl=-0.1496, V= 29.1, clQ_total=704.8666, clQ_x_arm=95.8096
 Vert Stab 1 connecting to Vert Stab 1, el=3, Cl=-0.1279, V= 29.1, clQ_total=812.8089,
 clQ_x_arm=133.5894
 Vert Stab 1 connecting to Vert Stab 1, el=4, Cl=-0.1058, V= 44.7, clQ_total=1024.0295,
 clQ_x_arm=228.6386
 Vert Stab 1 connecting to Vert Stab 1, el=5, Cl=-0.0728, V= 43.6, clQ_total=1162.2666,
 clQ_x_arm=304.6690
 Vert Stab 1 connecting to Vert Stab 1, el=6, Cl=-0.0161, V= 43.3, clQ_total=1192.3567,
 clQ_x_arm=324.2275
 Vert Stab 1 connecting to Vert Stab 1, el=7, Cl=-0.0127, V= 46.9, clQ_total=1220.2073,
 clQ_x_arm=345.1154
 Vert Stab 1 connecting to Vert Stab 1, el=8, Cl=-0.0076, V= 46.9, clQ_total=1237.0159,
 clQ_x_arm=359.4028
 Vert Stab 1 connecting to Vert Stab 1, el=9, Cl=-0.2112, V= 46.9, clQ_total=1701.5531,
 clQ_x_arm=800.7131
 clQ_x_arm/clQ_total= 0.4706, root_to_centroid= 0.4706, centroid_to_tip= 0.5294, AR_mult= 1.0588

Left Wing 1 height above ground: 4258.46 m
 hgt_in_cords is 2917.1699
 hgt_in_spans is 417.6726
 setting ground effect cl multiplier to 1.0000
 setting ground effect cdi multiplier to 1.0000
 setting ground effect wash multiplier to 1.0000

Right Wing 1 height above ground: 4258.52 m
 hgt_in_cords is 2917.2151
 hgt_in_spans is 417.6790
 setting ground effect cl multiplier to 1.0000
 setting ground effect cdi multiplier to 1.0000
 setting ground effect wash multiplier to 1.0000

Left Wing 2 height above ground: 4258.50 m
 hgt_in_cords is 3479.4600
 hgt_in_spans is 417.6766
 setting ground effect cl multiplier to 1.0000
 setting ground effect cdi multiplier to 1.0000
 setting ground effect wash multiplier to 1.0000

Right Wing 2 height above ground: 4258.69 m
 hgt_in_cords is 3479.6204
 hgt_in_spans is 417.6959
 setting ground effect cl multiplier to 1.0000
 setting ground effect cdi multiplier to 1.0000
 setting ground effect wash multiplier to 1.0000

Left Wing 3 height above ground: 4258.54 m
 hgt_in_cords is 4459.6367
 hgt_in_spans is 417.6808
 setting ground effect cl multiplier to 1.0000
 setting ground effect cdi multiplier to 1.0000
 setting ground effect wash multiplier to 1.0000

Right Wing 3 height above ground: 4258.91 m

hgt_in_cords is 4460.0205
hgt_in_spans is 417.7168
setting ground effect cl multiplier to 1.0000
setting ground effect cdi multiplier to 1.0000
setting ground effect wash multiplier to 1.0000

Left Wing 4 height above ground: 4258.54 m

hgt_in_cords is 7290.0366
hgt_in_spans is 417.6809
setting ground effect cl multiplier to 1.0000
setting ground effect cdi multiplier to 1.0000
setting ground effect wash multiplier to 1.0000

Right Wing 4 height above ground: 4258.98 m

hgt_in_cords is 7290.7905
hgt_in_spans is 417.7242
setting ground effect cl multiplier to 1.0000
setting ground effect cdi multiplier to 1.0000
setting ground effect wash multiplier to 1.0000

Left Stab height above ground: 4258.76 m

hgt_in_cords is 4893.0820
hgt_in_spans is 1042.7100
setting ground effect cl multiplier to 1.0000
setting ground effect cdi multiplier to 1.0000
setting ground effect wash multiplier to 1.0000

Right Stab height above ground: 4258.84 m

hgt_in_cords is 4893.1763
hgt_in_spans is 1042.7301
setting ground effect cl multiplier to 1.0000
setting ground effect cdi multiplier to 1.0000
setting ground effect wash multiplier to 1.0000

---RUNNING PROPULSIVE FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Before all propulsion:

f_axil= 0.0 N, pos aft, L= 0.0 N-m, pos right
f_side= 0.0 N, pos right, M= 0.0 N-m, pos up
f_nrml= 0.0 N, pos up, N= 0.0 N-m, pos right

---RUNNING PROPULSIVE FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Before propeller:

f_axil= 0.0 N, pos aft, L= 0.0 N-m, pos right
f_side= 0.0 N, pos right, M= 0.0 N-m, pos up
f_nrml= 0.0 N, pos up, N= 0.0 N-m, pos right

.....
.....

prop 1

.....
.....
The prop is turning clockwise at 209.3 rad per sec
psi of the prop hub : -0.0 m
the of the prop hub : -0.0 m
phi of the prop hub : -0.0 m

X of the prop hub : 0.00 m
Y of the prop hub : 0.03 m
Z of the prop hub : 0.00 m

vx_air_on_acf : 3.3 m/s
vy_air_on_acf : 4.4 m/s
vz_air_on_acf : 58.4 m/s

vx_air_on_hub after propwash and rotation rates : 3.3 m/s
vy_air_on_hub after propwash and rotation rates : 4.5 m/s
vz_air_on_hub after propwash and rotation rates : 58.4 m/s

Geometric Vz (air into the disc, pure geo) : 58.4 m/s
Geometric Vx (air attacking to the right) : 3.3 m/s
Geometric Vy (air attacking up from below) : 4.5 m/s

Total 'mixed' propwash INCREASE in speed behind prop : 4.6 m/s, pos in from ahead

.....
prop 1, ELEMENT INDEX 1 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

check-point is buried inside of Body 1 (fuselage)
BODY BURYING multiplies V by 0.7071
So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s
So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: -2.6 m/s BEHIND the disc
propwash speed increment at ring: -1.3 m/s AT the disc after mult by 0.5019, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 40.0 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 9.0 m/s (from omega of 209.31 rad/sec, rad arm of
0.05, swept arm of 0.02, and sweep of 0.0)
total element speed: 41.0 m/s
corkscrew path from disk: 77.3 deg (arctan(40.0/ 9.0))
element pitch: 68.9 deg (prop= 11.6 + incidence= 57.3 + cyclic)
blade element geometric angle of attack: -8.4 deg

For Reynolds number: rho= 0.7985, chord= 7.52, dynamic_viscosity =16.5794 millionths (divide by
this to get Re in Meg), Re_meg= 0.3624 million at this element

element center at 5% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -8.4, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -8.0

For lookup, alpha airflow= -8.4, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -8.0

For lo RE using Reynolds number= 0.1000 million

For hi RE using Reynolds number= 0.5000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: -0.3104 and -0.3104, interpolated Cl=-0.3104

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0284 and 0.0195, interpolated Cd= 0.0226

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.1266 yields Prandtl-Glauert Cl multiplier of 1.0081 to take Cl to -0.3129

Mach of 0.1266 yields Prandtl-Glauert Cm multiplier of 1.0081 to take Cm to -0.0706

Mach of 0.1266 with a Cl-based drag divergence Mach number of 0.7187 takes the Cd to 0.0226 (increase of 0.0000)

final Cl -0.3129

final Cd 0.0226

final Cm -0.0706

finite-wing lift multiplier of 0.9440 takes cl to -0.2954

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0029

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0029, which takes total Cd to 0.0255

cl multiplied by 1.0000 to get to -0.2954 to lower pressure from Mach 0.9 to 1.3.

lift=-0.2954 * 0.0190 * sqrt(41.0) * 0.7985 / 2 = -3.8 N

drag= 0.0255 * 0.0190 * sqrt(41.0) * 0.7985 / 2 = 0.3 N

momt=-0.0706 * 0.0190 * sqrt(41.0) * 0.7985 / 2 * 0.18= -0.2 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: -3.8 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 0.3 N

lift-only force in disc coords lateral right: 3.7 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: 0.8 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.1 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.3 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 3.6 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: 1.1 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.05 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= -0.2 N*m, running drag torque on the shaft= -0.2 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= 1.1 N,
running thrust normal to the disc in disc coordinate system= -1.1 N

totl force in aircraft coords lateral right: 3.6 (positive pulling the aircraft right)

totl force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)

totl force in aircraft coords axial aft: 1.1 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.08, M PRP on COG acf coord= 0.1)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: 1.1 (Z acf= -2.76, N PRP on COG acf coord= 0.0)

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prop 1, ELEMENT INDEX 1 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

check-point is buried inside of Body 1 (fuselage)
BODY BURYING multiplies V by 0.7071
So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s
So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 1.6 m/s BEHIND the disc
propwash speed increment at ring: 0.8 m/s AT the disc after mult by 0.5014, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 42.1 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 14.5 m/s (from omega of 209.31 rad/sec, rad arm of 0.05, swept arm of 0.02, and sweep of 0.0)
total element speed: 44.5 m/s
corkscrew path from disk: 71.0 deg (arctan(42.1/ 14.5))
element pitch: 68.9 deg (prop= 11.6 + incidence= 57.3 + cyclic)
blade element geometric angle of attack: -2.1 deg

For Reynolds number: rho= 0.7985, chord= 8.17, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 0.3937 million at this element

element center at 5% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -2.1, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -2.4
For lookup, alpha airflow= -2.1, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -2.4
For lo RE using Reynolds number= 0.1000 million
For hi RE using Reynolds number= 0.5000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1874 and 0.1874, interpolated Cl= 0.1874
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0184 and 0.0081, interpolated Cd= 0.0108
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.1375 yields Prandtl-Glauert Cl multiplier of 1.0096 to take Cl to 0.1892
Mach of 0.1375 yields Prandtl-Glauert Cm multiplier of 1.0096 to take Cm to -0.0707
Mach of 0.1375 with a Cl-based drag divergence Mach number of 0.7311 takes the Cd to 0.0108 (increase of 0.0000)

final Cl 0.1892
final Cd 0.0108
final Cm -0.0707

finite-wing lift multiplier of 0.9431 takes cl to 0.1784
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0011
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0011, which takes total Cd to 0.0119

cl multiplied by 1.0000 to get to 0.1784 to lower pressure from Mach 0.9 to 1.3.

$$\begin{aligned}\text{lift} &= 0.1784 * 0.0190 * \text{sqr}(44.5) * 0.7985 / 2 &= 2.7 \text{ N} \\ \text{drag} &= 0.0119 * 0.0190 * \text{sqr}(44.5) * 0.7985 / 2 &= 0.2 \text{ N} \\ \text{momt} &= -0.0707 * 0.0190 * \text{sqr}(44.5) * 0.7985 / 2 * 0.18 &= -0.2 \text{ N-m}\end{aligned}$$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 2.7 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 0.2 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 2.5 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -0.9 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 0.1 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.2 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 2.6 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -0.7 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.05 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 0.1 N*m, running drag torque on the shaft= -0.1 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -0.7 N,
running thrust normal to the disc in disc coordinate system= -0.4 N

totl force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 2.6 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -0.7 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.05, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 0.1)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -0.7 (Z acf= -2.76, N PRP on COG acf coord= -0.0)

.....
prop 1, ELEMENT INDEX 1 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

check-point is buried inside of Body 1 (fuselage)

BODY BURYING multiplies V by 0.7071

So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s

So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 0.8 m/s BEHIND the disc

propwash speed increment at ring: 0.4 m/s AT the disc after mult by 0.4981, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 41.7 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 13.7 m/s (from omega of 209.31 rad/sec, rad arm of

0.05, swept arm of 0.02, and sweep of 0.0)
 total element speed: 43.9 m/s
 corkscrew path from disk: 71.8 deg (arctan(41.7/ 13.7))
 element pitch: 68.9 deg (prop= 11.6 + incidence= 57.3 + cyclic)
 blade element geometric angle of attack: -3.0 deg

For Reynolds number: rho= 0.7985, chord= 8.05, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.3880 million at this element

element center at 5% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -3.0, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -3.1

For lookup, alpha airflow= -3.0, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -3.1

For lo RE using Reynolds number= 0.1000 million

For hi RE using Reynolds number= 0.5000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1209 and 0.1209, interpolated Cl= 0.1209

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0188 and 0.0087, interpolated Cd= 0.0116

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.1356 yields Prandtl-Glauert Cl multiplier of 1.0093 to take Cl to 0.1220

Mach of 0.1356 yields Prandtl-Glauert Cm multiplier of 1.0093 to take Cm to -0.0707

Mach of 0.1356 with a Cl-based drag divergence Mach number of 0.7378 takes the Cd to 0.0116 (increase of 0.0000)

final Cl 0.1220

final Cd 0.0116

final Cm -0.0707

finite-wing lift multiplier of 0.9426 takes cl to 0.1150

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0005

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0005, which takes total Cd to 0.0120

cl multiplied by 1.0000 to get to 0.1150 to lower pressure from Mach 0.9 to 1.3.

lift= 0.1150 * 0.0190 * sqr(43.9) * 0.7985 / 2 = 1.7 N

drag= 0.0120 * 0.0190 * sqr(43.9) * 0.7985 / 2 = 0.2 N

momt=-0.0707 * 0.0190 * sqr(43.9) * 0.7985 / 2 * 0.18= -0.2 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element

airflow path: 1.7 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element

airflow path: 0.2 N

lift-only force in disc coords lateral right: 1.6 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -0.5 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.1 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.2 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 1.7 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -0.4 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.05 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 0.1 N*m, running drag torque on the shaft= 0.0 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -0.4 N, running thrust normal to the disc in disc coordinate system= -0.1 N

total force in aircraft coords lateral right: 1.7 (positive pulling the aircraft right)

total force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)

total force in aircraft coords axial aft: -0.4 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= -0.02, M PRP on COG acf coord= 0.1)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -0.4 (Z acf= -2.76, N PRP on COG acf coord= -0.0)

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prop 1, ELEMENT INDEX 1 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

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check-point is buried inside of Body 1 (fuselage)

BODY BURYING multiplies V by 0.7071

So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s

So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: -3.1 m/s BEHIND the disc

propwash speed increment at ring: -1.6 m/s AT the disc after mult by 0.4986, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 39.7 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 8.2 m/s (from omega of 209.31 rad/sec, rad arm of 0.05, swept arm of 0.02, and sweep of 0.0)

total element speed: 40.5 m/s

corkscrew path from disk: 78.4 deg (arctan(39.7/ 8.2))

element pitch: 68.9 deg (prop= 11.6 + incidence= 57.3 + cyclic)

blade element geometric angle of attack: -9.5 deg

For Reynolds number: rho= 0.7985, chord= 7.45, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.3586 million at this element

element center at 5% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -9.5, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= -8.9

For lookup, alpha airflow= -9.5, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= -8.9

For lo RE using Reynolds number= 0.1000 million

For hi RE using Reynolds number= 0.5000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: -0.3931 and -0.3936, interpolated Cl=-0.3934
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0322 and 0.0232, interpolated Cd= 0.0264
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.1253 yields Prandtl-Glauert Cl multiplier of 1.0079 to take Cl to -0.3966
Mach of 0.1253 yields Prandtl-Glauert Cm multiplier of 1.0079 to take Cm to -0.0706
Mach of 0.1253 with a Cl-based drag divergence Mach number of 0.7103 takes the Cd to 0.0264
(increase of 0.0000)

final Cl -0.3966
final Cd 0.0264
final Cm -0.0706

finite-wing lift multiplier of 0.9442 takes cl to -0.3744
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0047
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0047, which takes total Cd to 0.0311
cl multiplied by 1.0000 to get to -0.3744 to lower pressure from Mach 0.9 to 1.3.

lift=-0.3744 * 0.0190 * sqrt(40.5) * 0.7985 / 2 = -4.7 N
drag= 0.0311 * 0.0190 * sqrt(40.5) * 0.7985 / 2 = 0.4 N
momt=-0.0706 * 0.0190 * sqrt(40.5) * 0.7985 / 2 * 0.18= -0.2 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: -4.7 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 0.4 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 4.6 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: 0.9 (positive pulling airplane aft or helicopter down)
drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.1 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.4 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 4.5 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: 1.3 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.05 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= -0.2 N*m, running drag torque on the shaft=-0.2 N*m
thrust normal to the disc in disc coordinate system, from the total disc axial force above= 1.3 N,
running thrust normal to the disc in disc coordinate system= -1.4 N

totl force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 4.5 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: 1.3 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=-0.05, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=0.03, M PRP on COG acf coord= 0.1)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: 1.3 (Z acf=-2.76, N PRP on COG acf coord= -0.1)

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prop 1, ELEMENT INDEX 2 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

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check-point is buried inside of Body 1 (fuselage)

BODY BURYING multiplies V by 0.7071

So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s

So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 4.5 m/s BEHIND the disc

propwash speed increment at ring: 2.2 m/s AT the disc after mult by 0.5057, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 43.5 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 30.6 m/s (from omega of 209.31 rad/sec, rad arm of 0.16, swept arm of 0.02, and sweep of -16.7)

total element speed: 51.0 m/s

corkscrew path from disk: 54.8 deg (arctan(43.5/ 30.6))

element pitch: 57.4 deg (prop= 11.6 + incidence= 45.8 + cyclic)

blade element geometric angle of attack: 2.6 deg

For Reynolds number: rho= 0.7985, chord= 8.81, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.4244 million at this element

element center at 15% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= 2.6, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= 1.8

For lookup, alpha airflow= 2.6, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= 1.8

For lo RE using Reynolds number= 0.1000 million

For hi RE using Reynolds number= 0.5000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.5582 and 0.5582, interpolated Cl= 0.5582

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0180 and 0.0073, interpolated Cd= 0.0093

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.1575 yields Prandtl-Glauert Cl multiplier of 1.0126 to take Cl to 0.5653

Mach of 0.1575 yields Prandtl-Glauert Cm multiplier of 1.0126 to take Cm to -0.0709

Mach of 0.1575 with a Cl-based drag divergence Mach number of 0.6935 takes the Cd to 0.0093 (increase of 0.0000)

final Cl 0.5653

final Cd 0.0093

final Cm -0.0709

finite-wing lift multiplier of 0.9440 takes cl to 0.5336

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0096

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0096, which takes total Cd to 0.0189

cl multiplied by 1.0000 to get to 0.5336 to lower pressure from Mach 0.9 to 1.3.

lift= 0.5336 * 0.0180 * sqr(51.0) * 0.7985 / 2 = 9.9 N

drag= 0.0189 * 0.0180 * sqr(51.0) * 0.7985 / 2 = 0.4 N

momt=-0.0709 * 0.0180 * sqr(51.0) * 0.7985 / 2 * 0.17= -0.2 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
 airflow path: 9.9 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
 airflow path: 0.4 N

lift-only force in disc coords lateral right: -8.1 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -5.7 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.2 (positive pulling airplane right or helicopter right)
 drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 drag-only force in disc coords axial aft: 0.3 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -8.3 (positive pulling airplane right or helicopter right)
 total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 total force in disc coords axial aft: -5.4 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.16 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 1.3 N*m, running drag torque on the shaft= 1.1 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -5.4 N,
 running thrust normal to the disc in disc coordinate system= 4.0 N

total force in aircraft coords lateral right: -8.3 (positive pulling the aircraft right)
 total force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)
 total force in aircraft coords axial aft: -5.4 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)
 considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.19, M PRP on COG acf coord= -0.9)
 considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -5.4 (Z acf= -2.76, N PRP on COG acf coord= -0.1)

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 prop 1, ELEMENT INDEX 2 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

check-point is buried inside of Body 1 (fuselage)
 BODY BURYING multiplies V by 0.7071
 So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s
 So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs, but not this one)
 propwash speed increment behind ring: 8.2 m/s BEHIND the disc
 propwash speed increment at ring: 4.1 m/s AT the disc after mult by 0.5042, and turb mult of 0.0312 is added on this as well
 total incoming airspeed: 45.4 m/s (geometric inflow + speed increment above + flapping)
 speed due to rotation and translation of disc: 36.1 m/s (from omega of 209.31 rad/sec, rad arm of 0.16, swept arm of 0.02, and sweep of -16.7)
 total element speed: 55.6 m/s
 corkscrew path from disk: 51.5 deg (arctan(45.4/ 36.1))
 element pitch: 57.4 deg (prop= 11.6 + incidence= 45.8 + cyclic)
 blade element geometric angle of attack: 5.9 deg

For Reynolds number: $\rho = 0.7985$, chord = 9.61, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), $Re_{meg} = 0.4629$ million at this element

element center at 15% span: (wgt root foil = 0.0000, wgt mid foil = 1.0000, wgt tip foil = 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow = 5.9, alpha zero-lift = -4.5, finite wing alpha multiplier = 0.8861, slat alpha divisor = 1.0000, alpha lookup = 4.7

For lookup, alpha airflow = 5.9, alpha zero-lift = -4.5, finite wing alpha multiplier = 0.8861, slat alpha divisor = 1.0000, alpha lookup = 4.7

For lo RE using Reynolds number = 0.1000 million

For hi RE using Reynolds number = 0.5000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.8201 and 0.8201, interpolated Cl = 0.8201

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0192 and 0.0092, interpolated Cd = 0.0102

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm = -0.0700

Mach of 0.1718 yields Prandtl-Glauert Cl multiplier of 1.0151 to take Cl to 0.8325

Mach of 0.1718 yields Prandtl-Glauert Cm multiplier of 1.0151 to take Cm to -0.0711

Mach of 0.1718 with a Cl-based drag divergence Mach number of 0.6668 takes the Cd to 0.0102 (increase of 0.0000)

final Cl 0.8325

final Cd 0.0102

final Cm -0.0711

finite-wing lift multiplier of 0.9431 takes cl to 0.7851

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0212

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0212, which takes total Cd to 0.0313

cl multiplied by 1.0000 to get to 0.7851 to lower pressure from Mach 0.9 to 1.3.

lift = $0.7851 * 0.0180 * \sqrt{55.6} * 0.7985 / 2 = 17.4 \text{ N}$

drag = $0.0313 * 0.0180 * \sqrt{55.6} * 0.7985 / 2 = 0.7 \text{ N}$

momt = $-0.0711 * 0.0180 * \sqrt{55.6} * 0.7985 / 2 * 0.17 = -0.3 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element airflow path: 17.4 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element airflow path: 0.7 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: 13.6 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -10.8 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: 0.4 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.5 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: 14.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -10.3 (positive pulling airplane aft or helicopter down)

radial arm of this element = 0.16 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock) = 2.2 N*m, running drag torque on the shaft = 3.3 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -10.3 N,
running thrust normal to the disc in disc coordinate system= 14.3 N

totl force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 14.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -10.3 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
0.16, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
0.03, M PRP on COG acf coord= -1.2)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -10.3 (Z acf=
-2.76, N PRP on COG acf coord= -1.7)

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prop 1, ELEMENT INDEX 2 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
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check-point is buried inside of Body 1 (fuselage)
BODY BURYING multiplies V by 0.7071
So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s
So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 7.6 m/s BEHIND the disc
propwash speed increment at ring: 3.7 m/s AT the disc after mult by 0.4943, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 45.0 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 35.3 m/s (from omega of 209.31 rad/sec, rad arm of
0.16, swept arm of 0.02, and sweep of -16.7)
total element speed: 54.8 m/s
corkscrew path from disk: 51.9 deg (arctan(45.0/ 35.3))
element pitch: 57.4 deg (prop= 11.6 + incidence= 45.8 + cyclic)
blade element geometric angle of attack: 5.5 deg

For Reynolds number: rho= 0.7985, chord= 9.48, dynamic_viscosity =16.5794 millionths (divide by
this to get Re min Meg), Re_meg= 0.4564 million at this element

element center at 15% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= 5.5, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8853, slat alpha
divisor= 1.0000, alpha lookup= 4.4
For lookup, alpha airflow= 5.5, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8853, slat alpha
divisor= 1.0000, alpha lookup= 4.4
For lo RE using Reynolds number= 0.1000 million
For hi RE using Reynolds number= 0.5000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.7883 and 0.7883, interpolated Cl= 0.7883
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0189 and 0.0088, interpolated Cd= 0.0099
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.1693 yields Prandtl-Glauert Cl multiplier of 1.0147 to take Cl to 0.7999
Mach of 0.1693 yields Prandtl-Glauert Cm multiplier of 1.0147 to take Cm to -0.0710
Mach of 0.1693 with a Cl-based drag divergence Mach number of 0.6700 takes the Cd to 0.0099

(increase of 0.0000)

final Cl 0.7999
final Cd 0.0099
final Cm -0.0710

finite-wing lift multiplier of 0.9426 takes cl to 0.7540
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0197
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0197, which takes total Cd to 0.0296
cl multiplied by 1.0000 to get to 0.7540 to lower pressure from Mach 0.9 to 1.3.

lift= $0.7540 * 0.0180 * \text{sqr}(54.8) * 0.7985 / 2 = 16.2 \text{ N}$
drag= $0.0296 * 0.0180 * \text{sqr}(54.8) * 0.7985 / 2 = 0.6 \text{ N}$
momt= $-0.0710 * 0.0180 * \text{sqr}(54.8) * 0.7985 / 2 * 0.17 = -0.3 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 16.2 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 0.6 N

lift-only force in disc coords lateral right: 12.8 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -10.0 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.4 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.5 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 13.2 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -9.5 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.16 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 2.1 N*m, running drag torque on the shaft= 5.4 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -9.5 N,
running thrust normal to the disc in disc coordinate system= 23.8 N

totl force in aircraft coords lateral right: 13.2 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -9.5 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= -0.13, M PRP on COG acf coord= -0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -9.5 (Z acf= -2.76, N PRP on COG acf coord= -1.7)

.....
prop 1, ELEMENT INDEX 2 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

check-point is buried inside of Body 1 (fuselage)

BODY BURYING multiplies V by 0.7071

So speed is changed to vx= 2.3 vy= 3.2 vz= 41.3 m/s

So final speed is vx= 2.3 vy= 3.2 vz= 41.3 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 41.3 m/s (includes propwash from any OTHER discs, but not this one)
 propwash speed increment behind ring: 4.0 m/s BEHIND the disc
 propwash speed increment at ring: 2.0 m/s AT the disc after mult by 0.4958, and turb mult of 0.0312 is added on this as well
 total incoming airspeed: 43.3 m/s (geometric inflow + speed increment above + flapping)
 speed due to rotation and translation of disc: 29.8 m/s (from omega of 209.31 rad/sec, rad arm of 0.16, swept arm of 0.02, and sweep of -16.7)
 total element speed: 50.3 m/s
 corkscrew path from disk: 55.4 deg (arctan(43.3/ 29.8))
 element pitch: 57.4 deg (prop= 11.6 + incidence= 45.8 + cyclic)
 blade element geometric angle of attack: 2.0 deg

For Reynolds number: rho= 0.7985, chord= 8.70, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.4192 million at this element

element center at 15% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
 For lookup, alpha airflow= 2.0, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= 1.3
 For lookup, alpha airflow= 2.0, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= 1.3
 For lo RE using Reynolds number= 0.1000 million
 For hi RE using Reynolds number= 0.5000 million
 mid foil: Cl lookups for lo and hi Reynolds numbers: 0.5116 and 0.5116, interpolated Cl= 0.5116
 mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0180 and 0.0072, interpolated Cd= 0.0094
 mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
 Mach of 0.1556 yields Prandtl-Glauert Cl multiplier of 1.0123 to take Cl to 0.5179
 Mach of 0.1556 yields Prandtl-Glauert Cm multiplier of 1.0123 to take Cm to -0.0709
 Mach of 0.1556 with a Cl-based drag divergence Mach number of 0.6982 takes the Cd to 0.0094 (increase of 0.0000)

final Cl 0.5179
 final Cd 0.0094
 final Cm -0.0709

finite-wing lift multiplier of 0.9442 takes cl to 0.4890
 e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0080
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0080, which takes total Cd to 0.0174
 cl multiplied by 1.0000 to get to 0.4890 to lower pressure from Mach 0.9 to 1.3.

lift= 0.4890 * 0.0180 * sqr(50.3) * 0.7985 / 2 = 8.9 N
 drag= 0.0174 * 0.0180 * sqr(50.3) * 0.7985 / 2 = 0.3 N
 momt=-0.0709 * 0.0180 * sqr(50.3) * 0.7985 / 2 * 0.17= -0.2 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element airflow path: 8.9 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element airflow path: 0.3 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: -7.3 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -5.0 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.2 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.3 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -7.5 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -4.8 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.16 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 1.2 N*m, running drag torque on the shaft= 6.6 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -4.8 N,
running thrust normal to the disc in disc coordinate system= 28.6 N

total force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
total force in aircraft coords lateral up: -7.5 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -4.8 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.16, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= -0.2)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -4.8 (Z acf= -2.76, N PRP on COG acf coord= -1.0)

.....
prop 1, ELEMENT INDEX 3 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 2.6 m/s BEHIND the disc
propwash speed increment at ring: 1.3 m/s AT the disc after mult by 0.5095, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 59.7 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 51.1 m/s (from omega of 209.31 rad/sec, rad arm of 0.26, swept arm of 0.01, and sweep of -7.4)
total element speed: 77.9 m/s
corkscrew path from disk: 49.4 deg (arctan(59.7/ 51.1))
element pitch: 47.3 deg (prop= 11.6 + incidence= 35.7 + cyclic)
blade element geometric angle of attack: -2.1 deg

For Reynolds number: rho= 0.7985, chord= 14.73, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.7096 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -2.1, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -2.4

For lookup, alpha airflow= -2.1, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -2.4

For lo RE using Reynolds number= 0.5000 million

For hi RE using Reynolds number= 1.0000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1871 and 0.1959, interpolated Cl= 0.1908

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0081 and 0.0078, interpolated Cd= 0.0079

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.2409 yields Prandtl-Glauert Cl multiplier of 1.0303 to take Cl to 0.1966

Mach of 0.2409 yields Prandtl-Glauert Cm multiplier of 1.0303 to take Cm to -0.0721

Mach of 0.2409 with a Cl-based drag divergence Mach number of 0.7303 takes the Cd to 0.0079 (increase of 0.0000)

final Cl 0.1966

final Cd 0.0079

final Cm -0.0721

finite-wing lift multiplier of 0.9440 takes cl to 0.1856

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0012

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0012, which takes total Cd to 0.0091

cl multiplied by 1.0000 to get to 0.1856 to lower pressure from Mach 0.9 to 1.3.

lift= 0.1856 * 0.0196 * sqrt(77.9) * 0.7985 / 2 = 8.8 N

drag= 0.0091 * 0.0196 * sqrt(77.9) * 0.7985 / 2 = 0.4 N

momt=-0.0721 * 0.0196 * sqrt(77.9) * 0.7985 / 2 * 0.19= -0.6 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 8.8 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 0.4 N

lift-only force in disc coords lateral right: -6.7 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -5.8 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.3 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.3 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -7.0 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -5.4 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.26 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 1.8 N*m, running drag torque on the shaft= 8.4 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -5.4 N,
running thrust normal to the disc in disc coordinate system= 34.0 N

totl force in aircraft coords lateral right: -7.0 (positive pulling the aircraft right)

totl force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)

totl force in aircraft coords axial aft: -5.4 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.29, M PRP on COG acf coord= -1.7)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -5.4 (Z acf= -2.76, N PRP on COG acf coord= -1.0)

.....
prop 1, ELEMENT INDEX 3 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 7.3 m/s BEHIND the disc
propwash speed increment at ring: 3.7 m/s AT the disc after mult by 0.5071, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 62.1 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 58.9 m/s (from omega of 209.31 rad/sec, rad arm of 0.26, swept arm of 0.01, and sweep of -7.4)
total element speed: 84.9 m/s
corkscrew path from disk: 46.5 deg (arctan(62.1/ 58.9))
element pitch: 47.3 deg (prop= 11.6 + incidence= 35.7 + cyclic)
blade element geometric angle of attack: 0.8 deg

For Reynolds number: rho= 0.7985, chord= 16.04, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 0.7727 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= 0.8, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= 0.2
For lookup, alpha airflow= 0.8, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= 0.2
For lo RE using Reynolds number= 0.5000 million
For hi RE using Reynolds number= 1.0000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.4150 and 0.4238, interpolated Cl= 0.4198
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0072 and 0.0070, interpolated Cd= 0.0071
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.2623 yields Prandtl-Glauert Cl multiplier of 1.0363 to take Cl to 0.4350
Mach of 0.2623 yields Prandtl-Glauert Cm multiplier of 1.0363 to take Cm to -0.0725
Mach of 0.2623 with a Cl-based drag divergence Mach number of 0.7065 takes the Cd to 0.0071 (increase of 0.0000)

final Cl 0.4350
final Cd 0.0071
final Cm -0.0725

finite-wing lift multiplier of 0.9431 takes cl to 0.4103
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0058
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0058, which takes total Cd to 0.0129
cl multiplied by 1.0000 to get to 0.4103 to lower pressure from Mach 0.9 to 1.3.

lift= 0.4103 * 0.0196 * sqr(84.9) * 0.7985 / 2 = 23.2 N

$$\text{drag} = 0.0129 * 0.0196 * \text{sqr}(84.9) * 0.7985 / 2 = 0.7 \text{ N}$$

$$\text{momt} = -0.0725 * 0.0196 * \text{sqr}(84.9) * 0.7985 / 2 * 0.19 = -0.8 \text{ N-m}$$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
 airflow path: 23.2 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
 airflow path: 0.7 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: 16.8 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -15.9 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
 drag-only force in disc coords lateral up: 0.5 (positive pulling airplane up or helicopter aft)
 drag-only force in disc coords axial aft: 0.5 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
 total force in disc coords lateral up: 17.3 (positive pulling airplane up or helicopter aft)
 total force in disc coords axial aft: -15.4 (positive pulling airplane aft or helicopter down)

radial arm of this element = 0.26 m
 drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock) = 4.5 N*m, running drag torque on the shaft = 12.9 N*m
 thrust normal to the disc in disc coordinate system, from the total disc axial force above = -15.4 N, running thrust normal to the disc in disc coordinate system = 49.5 N

total force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
 total force in aircraft coords lateral up: 17.3 (positive pulling the aircraft up)
 total force in aircraft coords axial aft: -15.4 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf = 0.26, L PRP on COG acf coord = 0.0)
 considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf = 0.03, M PRP on COG acf coord = -2.2)
 considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -15.4 (Z acf = -2.76, N PRP on COG acf coord = -5.0)

.....
 prop 1, ELEMENT INDEX 3 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

So final speed is vx = 3.3 vy = 4.5 vz = 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
 propwash speed increment behind ring: 6.6 m/s BEHIND the disc
 propwash speed increment at ring: 3.2 m/s AT the disc after mult by 0.4905, and turb mult of 0.0312 is added on this as well
 total incoming airspeed: 61.6 m/s (geometric inflow + speed increment above + flapping)
 speed due to rotation and translation of disc: 57.7 m/s (from omega of 209.31 rad/sec, rad arm of 0.26, swept arm of 0.01, and sweep of -7.4)
 total element speed: 83.7 m/s
 corkscrew path from disk: 46.9 deg (arctan(61.6/ 57.7))
 element pitch: 47.3 deg (prop = 11.6 + incidence = 35.7 + cyclic)
 blade element geometric angle of attack: 0.4 deg

For Reynolds number: $\rho = 0.7985$, chord = 15.83, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), $Re_{meg} = 0.7624$ million at this element

element center at 25% span: (wgt root foil = 0.0000, wgt mid foil = 1.0000, wgt tip foil = 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow = 0.4, alpha zero-lift = -4.5, finite wing alpha multiplier = 0.8853, slat alpha divisor = 1.0000, alpha lookup = -0.1

For lookup, alpha airflow = 0.4, alpha zero-lift = -4.6, finite wing alpha multiplier = 0.8853, slat alpha divisor = 1.0000, alpha lookup = -0.2

For lo RE using Reynolds number = 0.5000 million

For hi RE using Reynolds number = 1.0000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.3872 and 0.3961, interpolated Cl = 0.3919

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0072 and 0.0070, interpolated Cd = 0.0071

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm = -0.0700

Mach of 0.2588 yields Prandtl-Glauert Cl multiplier of 1.0353 to take Cl to 0.4057

Mach of 0.2588 yields Prandtl-Glauert Cm multiplier of 1.0353 to take Cm to -0.0725

Mach of 0.2588 with a Cl-based drag divergence Mach number of 0.7094 takes the Cd to 0.0071 (increase of 0.0000)

final Cl 0.4057

final Cd 0.0071

final Cm -0.0725

finite-wing lift multiplier of 0.9426 takes cl to 0.3824

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0051

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0051, which takes total Cd to 0.0122

cl multiplied by 1.0000 to get to 0.3824 to lower pressure from Mach 0.9 to 1.3.

lift = $0.3824 * 0.0196 * \sqrt{83.7} * 0.7985 / 2 = 21.0 \text{ N}$

drag = $0.0122 * 0.0196 * \sqrt{83.7} * 0.7985 / 2 = 0.7 \text{ N}$

momt = $-0.0725 * 0.0196 * \sqrt{83.7} * 0.7985 / 2 * 0.19 = -0.8 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element airflow path: 21.0 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element airflow path: 0.7 N

lift-only force in disc coords lateral right: 15.3 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -14.4 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.5 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.5 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 15.8 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -13.9 (positive pulling airplane aft or helicopter down)

radial arm of this element = 0.26 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock) = 4.1 N*m, running drag torque on the shaft = 17.0 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -13.9 N,
running thrust normal to the disc in disc coordinate system= 63.3 N

totl force in aircraft coords lateral right: 15.8 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -13.9 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
-0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
-0.23, M PRP on COG acf coord= 1.0)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -13.9 (Z acf=
-2.76, N PRP on COG acf coord= -5.0)

.....
prop 1, ELEMENT INDEX 3 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 2.0 m/s BEHIND the disc
propwash speed increment at ring: 1.0 m/s AT the disc after mult by 0.4929, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 59.4 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 50.0 m/s (from omega of 209.31 rad/sec, rad arm of
0.26, swept arm of 0.01, and sweep of -7.4)
total element speed: 77.0 m/s
corkscrew path from disk: 49.9 deg (arctan(59.4/ 50.0))
element pitch: 47.3 deg (prop= 11.6 + incidence= 35.7 + cyclic)
blade element geometric angle of attack: -2.6 deg

For Reynolds number: rho= 0.7985, chord= 14.55, dynamic_viscosity =16.5794 millionths (divide by
this to get Re in Meg), Re_meg= 0.7007 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -2.6, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -2.8
For lookup, alpha airflow= -2.6, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -2.9
For lo RE using Reynolds number= 0.5000 million
For hi RE using Reynolds number= 1.0000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1475 and 0.1563, interpolated Cl= 0.1510
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0084 and 0.0081, interpolated Cd= 0.0083
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.2378 yields Prandtl-Glauert Cl multiplier of 1.0295 to take Cl to 0.1555
Mach of 0.2378 yields Prandtl-Glauert Cm multiplier of 1.0295 to take Cm to -0.0721
Mach of 0.2378 with a Cl-based drag divergence Mach number of 0.7345 takes the Cd to 0.0083
(increase of 0.0000)

final Cl 0.1555

final Cd 0.0083
final Cm -0.0721

finite-wing lift multiplier of 0.9442 takes cl to 0.1468
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0007
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0007, which takes total Cd to 0.0090
cl multiplied by 1.0000 to get to 0.1468 to lower pressure from Mach 0.9 to 1.3.

lift= $0.1468 * 0.0196 * \text{sqr}(77.0) * 0.7985 / 2 = 6.8 \text{ N}$
drag= $0.0090 * 0.0196 * \text{sqr}(77.0) * 0.7985 / 2 = 0.4 \text{ N}$
momt= $-0.0721 * 0.0196 * \text{sqr}(77.0) * 0.7985 / 2 * 0.19 = -0.6 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 6.8 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 0.4 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -5.2 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -4.4 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.3 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.3 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -5.5 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -4.1 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.26 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and
6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 1.4 N*m, running drag torque on the shaft=
18.4 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -4.1 N,
running thrust normal to the disc in disc coordinate system= 67.4 N

totl force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -5.5 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -4.1 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
-0.26, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
0.03, M PRP on COG acf coord= 0.9)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -4.1 (Z acf=
-2.76, N PRP on COG acf coord= -3.9)

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prop 1, ELEMENT INDEX 4 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
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So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 4.9 m/s BEHIND the disc
propwash speed increment at ring: 2.5 m/s AT the disc after mult by 0.5134, and turb mult
of 0.0312 is added on this as well

total incoming airspeed: 60.9 m/s (geometric inflow + speed increment above + flapping)
 speed due to rotation and translation of disc: 72.8 m/s (from omega of 209.31 rad/sec, rad arm of 0.36, swept arm of 0.00, and sweep of -3.8)
 total element speed: 94.7 m/s
 corkscrew path from disk: 39.9 deg (arctan(60.9/ 72.8))
 element pitch: 38.7 deg (prop= 11.6 + incidence= 27.1 + cyclic)
 blade element geometric angle of attack: -1.2 deg

For Reynolds number: rho= 0.7985, chord= 18.86, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.9085 million at this element

element center at 35% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
 For lookup, alpha airflow= -1.2, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -1.6
 For lookup, alpha airflow= -1.2, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -1.6
 For lo RE using Reynolds number= 0.5000 million
 For hi RE using Reynolds number= 1.0000 million
 mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2587 and 0.2676, interpolated Cl= 0.2659
 mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0076 and 0.0073, interpolated Cd= 0.0074
 mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
 Mach of 0.2927 yields Prandtl-Glauert Cl multiplier of 1.0458 to take Cl to 0.2781
 Mach of 0.2927 yields Prandtl-Glauert Cm multiplier of 1.0458 to take Cm to -0.0732
 Mach of 0.2927 with a Cl-based drag divergence Mach number of 0.7222 takes the Cd to 0.0074 (increase of 0.0000)

final Cl 0.2781
 final Cd 0.0074
 final Cm -0.0732

finite-wing lift multiplier of 0.9440 takes cl to 0.2625
 e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0023
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0023, which takes total Cd to 0.0097
 cl multiplied by 1.0000 to get to 0.2625 to lower pressure from Mach 0.9 to 1.3.

lift= 0.2625 * 0.0207 * sqr(94.7) * 0.7985 / 2 = 19.5 N
 drag= 0.0097 * 0.0207 * sqr(94.7) * 0.7985 / 2 = 0.7 N
 momt=-0.0732 * 0.0207 * sqr(94.7) * 0.7985 / 2 * 0.20= -1.1 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element airflow path: 19.5 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element airflow path: 0.7 N

lift-only force in disc coords lateral right: -12.5 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -14.9 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.6 (positive pulling airplane right or helicopter right)
 drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 drag-only force in disc coords axial aft: 0.5 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -13.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -14.5 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.36 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 4.7 N*m, running drag torque on the shaft= 23.2 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -14.5 N,
running thrust normal to the disc in disc coordinate system= 81.9 N

total force in aircraft coords lateral right: -13.0 (positive pulling the aircraft right)
total force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -14.5 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.39, M PRP on COG acf coord= -4.8)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -14.5 (Z acf= -2.76, N PRP on COG acf coord= -3.9)

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prop 1, ELEMENT INDEX 4 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
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So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 9.0 m/s BEHIND the disc
propwash speed increment at ring: 4.6 m/s AT the disc after mult by 0.5099, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 63.0 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 80.6 m/s (from omega of 209.31 rad/sec, rad arm of 0.36, swept arm of 0.00, and sweep of -3.8)
total element speed: 102.1 m/s
corkscrew path from disk: 38.0 deg (arctan(63.0/ 80.6))
element pitch: 38.7 deg (prop= 11.6 + incidence= 27.1 + cyclic)
blade element geometric angle of attack: 0.6 deg

For Reynolds number: rho= 0.7985, chord= 20.33, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.9792 million at this element

element center at 35% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= 0.6, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= 0.1
For lookup, alpha airflow= 0.6, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= 0.0
For lo RE using Reynolds number= 0.5000 million
For hi RE using Reynolds number= 1.0000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.4047 and 0.4136, interpolated Cl= 0.4132
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0072 and 0.0070, interpolated Cd= 0.0070
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.3155 yields Prandtl-Glauert Cl multiplier of 1.0538 to take Cl to 0.4355
Mach of 0.3155 yields Prandtl-Glauert Cm multiplier of 1.0538 to take Cm to -0.0738
Mach of 0.3155 with a Cl-based drag divergence Mach number of 0.7065 takes the Cd to 0.0070
(increase of 0.0000)

final Cl 0.4355
final Cd 0.0070
final Cm -0.0738

finite-wing lift multiplier of 0.9431 takes cl to 0.4107
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0058
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0058, which takes total Cd to 0.0128
cl multiplied by 1.0000 to get to 0.4107 to lower pressure from Mach 0.9 to 1.3.

lift= 0.4107 * 0.0207 * sqrt(102.1) * 0.7985 / 2 = 35.4 N
drag= 0.0128 * 0.0207 * sqrt(102.1) * 0.7985 / 2 = 1.1 N
momt=-0.0738 * 0.0207 * sqrt(102.1) * 0.7985 / 2 * 0.20= -1.3 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 35.4 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.1 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 21.8 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -27.9 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 0.9 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 22.6 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -27.2 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.36 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 8.2 N*m, running drag torque on the shaft= 31.4 N*m
thrust normal to the disc in disc coordinate system, from the total disc axial force above= -27.2 N,
running thrust normal to the disc in disc coordinate system= 109.1 N

totl force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 22.6 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -27.2 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.36, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= -5.7)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -27.2 (Z acf= -2.76, N PRP on COG acf coord= -13.8)

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prop 1, ELEMENT INDEX 4 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

So final speed is $v_x = 3.3$ $v_y = 4.5$ $v_z = 58.4$ m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 8.5 m/s BEHIND the disc
propwash speed increment at ring: 4.2 m/s AT the disc after mult by 0.4866, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 62.5 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 79.4 m/s (from omega of 209.31 rad/sec, rad arm of 0.36, swept arm of 0.00, and sweep of -3.8)
total element speed: 100.9 m/s
corkscrew path from disk: 38.2 deg ($\arctan(62.5/79.4)$)
element pitch: 38.7 deg (prop= 11.6 + incidence= 27.1 + cyclic)
blade element geometric angle of attack: 0.5 deg

For Reynolds number: $\rho = 0.7985$, chord= 20.09, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), $Re_{meg} = 0.9676$ million at this element

element center at 35% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= 0.5, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -0.1
For lookup, alpha airflow= 0.5, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -0.1
For lo RE using Reynolds number= 0.5000 million
For hi RE using Reynolds number= 1.0000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.3897 and 0.3986, interpolated Cl= 0.3980
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0072 and 0.0070, interpolated Cd= 0.0070
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm= -0.0700
Mach of 0.3117 yields Prandtl-Glauert Cl multiplier of 1.0524 to take Cl to 0.4189
Mach of 0.3117 yields Prandtl-Glauert Cm multiplier of 1.0524 to take Cm to -0.0737
Mach of 0.3117 with a Cl-based drag divergence Mach number of 0.7081 takes the Cd to 0.0070 (increase of 0.0000)

final Cl 0.4189
final Cd 0.0070
final Cm -0.0737

finite-wing lift multiplier of 0.9426 takes cl to 0.3948
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0054
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0054, which takes total Cd to 0.0124
cl multiplied by 1.0000 to get to 0.3948 to lower pressure from Mach 0.9 to 1.3.

lift= $0.3948 * 0.0207 * \sqrt{100.9} * 0.7985 / 2 = 33.2$ N
drag= $0.0124 * 0.0207 * \sqrt{100.9} * 0.7985 / 2 = 1.0$ N
momt= $-0.0737 * 0.0207 * \sqrt{100.9} * 0.7985 / 2 * 0.20 = -1.2$ N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 33.2 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element

airflow path: 1.0 N

lift-only force in disc coords lateral right: 20.5 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -26.1 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.8 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 21.4 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -25.4 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.36 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 7.8 N*m, running drag torque on the shaft= 39.2 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -25.4 N,
running thrust normal to the disc in disc coordinate system= 134.5 N

total force in aircraft coords lateral right: 21.4 (positive pulling the aircraft right)
total force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -25.4 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= -0.33, M PRP on COG acf coord= 2.8)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -25.4 (Z acf= -2.76, N PRP on COG acf coord= -13.8)

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prop 1, ELEMENT INDEX 4 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
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So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 4.4 m/s BEHIND the disc
propwash speed increment at ring: 2.2 m/s AT the disc after mult by 0.4901, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 60.6 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 71.7 m/s (from omega of 209.31 rad/sec, rad arm of 0.36, swept arm of 0.00, and sweep of -3.8)
total element speed: 93.6 m/s
corkscrew path from disk: 40.2 deg (arctan(60.6/ 71.7))
element pitch: 38.7 deg (prop= 11.6 + incidence= 27.1 + cyclic)
blade element geometric angle of attack: -1.5 deg

For Reynolds number: rho= 0.7985, chord= 18.65, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 0.8982 million at this element

element center at 35% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -1.5, alpha zero-lift= -4.5, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -1.9
For lookup, alpha airflow= -1.5, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -1.9
For lo RE using Reynolds number= 0.5000 million
For hi RE using Reynolds number= 1.0000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2339 and 0.2428, interpolated Cl= 0.2410
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0077 and 0.0075, interpolated Cd= 0.0075
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.2894 yields Prandtl-Glauert Cl multiplier of 1.0447 to take Cl to 0.2518
Mach of 0.2894 yields Prandtl-Glauert Cm multiplier of 1.0447 to take Cm to -0.0731
Mach of 0.2894 with a Cl-based drag divergence Mach number of 0.7248 takes the Cd to 0.0075
(increase of 0.0000)

final Cl 0.2518
final Cd 0.0075
final Cm -0.0731

finite-wing lift multiplier of 0.9442 takes cl to 0.2377
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0019
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0019, which takes total Cd to 0.0094
cl multiplied by 1.0000 to get to 0.2377 to lower pressure from Mach 0.9 to 1.3.

lift= $0.2377 * 0.0207 * \sqrt{93.6} * 0.7985 / 2 = 17.2 \text{ N}$
drag= $0.0094 * 0.0207 * \sqrt{93.6} * 0.7985 / 2 = 0.7 \text{ N}$
momt=-0.0731 * 0.0207 * $\sqrt{93.6} * 0.7985 / 2 * 0.20 = -1.1 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 17.2 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 0.7 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -11.1 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -13.2 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.5 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.4 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -11.6 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -12.7 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.36 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and
6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 4.2 N*m, running drag torque on the shaft=
43.4 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -12.7 N,
running thrust normal to the disc in disc coordinate system= 147.2 N

totl force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -11.6 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -12.7 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
-0.36, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
0.03, M PRP on COG acf coord= 2.4)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -12.7 (Z acf=
-2.76, N PRP on COG acf coord= -9.2)

.....
prop 1, ELEMENT INDEX 5 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 5.9 m/s BEHIND the disc
propwash speed increment at ring: 3.0 m/s AT the disc after mult by 0.5172, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 61.4 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 94.6 m/s (from omega of 209.31 rad/sec, rad arm of
0.47, swept arm of 0.00, and sweep of 0.0)
total element speed: 112.8 m/s
corkscrew path from disk: 33.0 deg (arctan(61.4/ 94.6))
element pitch: 31.6 deg (prop= 11.6 + incidence= 20.0 + cyclic)
blade element geometric angle of attack: -1.4 deg

For Reynolds number: rho= 0.7985, chord= 22.92, dynamic_viscosity =16.5794 millionths (divide by
this to get Re min Meg), Re_meg= 1.1037 million at this element

element center at 45% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -1.4, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha
divisor= 1.0000, alpha lookup= -1.8
For lookup, alpha airflow= -1.4, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8880, slat alpha
divisor= 1.0000, alpha lookup= -1.8
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2535 and 0.3067, interpolated Cl= 0.2542
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0074 and 0.0070, interpolated Cd= 0.0074
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.3485 yields Prandtl-Glauert Cl multiplier of 1.0669 to take Cl to 0.2712
Mach of 0.3485 yields Prandtl-Glauert Cm multiplier of 1.0669 to take Cm to -0.0747
Mach of 0.3485 with a Cl-based drag divergence Mach number of 0.7229 takes the Cd to 0.0074
(increase of 0.0000)

final Cl 0.2712
final Cd 0.0074
final Cm -0.0747

finite-wing lift multiplier of 0.9440 takes cl to 0.2560
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0022
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0022, which takes total Cd to 0.0096

cl multiplied by 1.0000 to get to 0.2560 to lower pressure from Mach 0.9 to 1.3.

$$\begin{aligned}\text{lift} &= 0.2560 * 0.0211 * \text{sqr}(112.8) * 0.7985 / 2 &= 27.5 \text{ N} \\ \text{drag} &= 0.0096 * 0.0211 * \text{sqr}(112.8) * 0.7985 / 2 &= 1.0 \text{ N} \\ \text{momt} &= -0.0747 * 0.0211 * \text{sqr}(112.8) * 0.7985 / 2 * 0.20 &= -1.6 \text{ N-m}\end{aligned}$$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 27.5 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.0 N

lift-only force in disc coords lateral right: -14.9 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -23.0 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.9 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -15.8 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -22.5 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.47 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 7.4 N*m, running drag torque on the shaft= 50.8 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -22.5 N,
running thrust normal to the disc in disc coordinate system= 169.7 N

totl force in aircraft coords lateral right: -15.8 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -22.5 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.50, M PRP on COG acf coord= -8.8)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -22.5 (Z acf= -2.76, N PRP on COG acf coord= -9.2)

.....
prop 1, ELEMENT INDEX 5 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 9.4 m/s BEHIND the disc

propwash speed increment at ring: 4.9 m/s AT the disc after mult by 0.5127, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 63.2 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 102.4 m/s (from omega of 209.31 rad/sec, rad arm of 0.47, swept arm of 0.00, and sweep of 0.0)

total element speed: 120.3 m/s

corkscrew path from disk: 31.7 deg (arctan(63.2/ 102.4))

element pitch: 31.6 deg (prop= 11.6 + incidence= 20.0 + cyclic)
blade element geometric angle of attack: -0.1 deg

For Reynolds number: rho= 0.7985, chord= 24.45, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.1776 million at this element

element center at 45% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -0.1, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -0.6

For lookup, alpha airflow= -0.1, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -0.7

For lo RE using Reynolds number= 1.0000 million

For hi RE using Reynolds number= 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.3529 and 0.4060, interpolated Cl= 0.3542

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0071 and 0.0068, interpolated Cd= 0.0071

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.3718 yields Prandtl-Glauert Cl multiplier of 1.0772 to take Cl to 0.3815

Mach of 0.3718 yields Prandtl-Glauert Cm multiplier of 1.0772 to take Cm to -0.0754

Mach of 0.3718 with a Cl-based drag divergence Mach number of 0.7118 takes the Cd to 0.0071 (increase of 0.0000)

final Cl 0.3815

final Cd 0.0071

final Cm -0.0754

finite-wing lift multiplier of 0.9431 takes cl to 0.3598

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0044

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0044, which takes total Cd to 0.0115

cl multiplied by 1.0000 to get to 0.3598 to lower pressure from Mach 0.9 to 1.3.

lift= 0.3598 * 0.0211 * sqrt(120.3) * 0.7985 / 2 = 43.9 N

drag= 0.0115 * 0.0211 * sqrt(120.3) * 0.7985 / 2 = 1.4 N

momt=-0.0754 * 0.0211 * sqrt(120.3) * 0.7985 / 2 * 0.20= -1.9 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 43.9 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.4 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: 23.1 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -37.4 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: 1.2 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: 24.3 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -36.6 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.47 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 11.4 N*m, running drag torque on the shaft= 62.2 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -36.6 N, running thrust normal to the disc in disc coordinate system= 206.3 N

total force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
total force in aircraft coords lateral up: 24.3 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -36.6 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.47, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= -9.9)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -36.6 (Z acf= -2.76, N PRP on COG acf coord= -26.3)

.....
prop 1, ELEMENT INDEX 5 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 9.1 m/s BEHIND the disc
propwash speed increment at ring: 4.4 m/s AT the disc after mult by 0.4828, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 62.8 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 101.2 m/s (from omega of 209.31 rad/sec, rad arm of 0.47, swept arm of 0.00, and sweep of 0.0)
total element speed: 119.1 m/s
corkscrew path from disk: 31.8 deg (arctan(62.8/ 101.2))
element pitch: 31.6 deg (prop= 11.6 + incidence= 20.0 + cyclic)
blade element geometric angle of attack: -0.2 deg

For Reynolds number: rho= 0.7985, chord= 24.20, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.1654 million at this element

element center at 45% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -0.2, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -0.7
For lookup, alpha airflow= -0.2, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -0.8
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.3451 and 0.3983, interpolated Cl= 0.3463
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0071 and 0.0068, interpolated Cd= 0.0071
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.3680 yields Prandtl-Glauert Cl multiplier of 1.0755 to take Cl to 0.3725
Mach of 0.3680 yields Prandtl-Glauert Cm multiplier of 1.0755 to take Cm to -0.0753
Mach of 0.3680 with a Cl-based drag divergence Mach number of 0.7128 takes the Cd to 0.0071

(increase of 0.0000)

final Cl 0.3725
final Cd 0.0071
final Cm -0.0753

finite-wing lift multiplier of 0.9426 takes cl to 0.3511
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0043
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0043, which takes total Cd to 0.0113
cl multiplied by 1.0000 to get to 0.3511 to lower pressure from Mach 0.9 to 1.3.

lift= 0.3511 * 0.0211 * sqrt(119.1) * 0.7985 / 2 = 42.0 N
drag= 0.0113 * 0.0211 * sqrt(119.1) * 0.7985 / 2 = 1.4 N
momt=-0.0753 * 0.0211 * sqrt(119.1) * 0.7985 / 2 * 0.20= -1.8 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 42.0 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.4 N

lift-only force in disc coords lateral right: 22.1 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -35.7 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 1.2 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 23.3 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -35.0 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.47 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 10.9 N*m, running drag torque on the shaft= 73.0 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -35.0 N,
running thrust normal to the disc in disc coordinate system= 241.3 N

totl force in aircraft coords lateral right: 23.3 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -35.0 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= -0.44, M PRP on COG acf coord= 5.4)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -35.0 (Z acf= -2.76, N PRP on COG acf coord= -26.3)

.....
prop 1, ELEMENT INDEX 5 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 5.5 m/s BEHIND the disc
 propwash speed increment at ring: 2.7 m/s AT the disc after mult by 0.4873, and turb mult
 of 0.0312 is added on this as well
 total incoming airspeed: 61.1 m/s (geometric inflow + speed increment above +
 flapping)
 speed due to rotation and translation of disc: 93.4 m/s (from omega of 209.31 rad/sec, rad arm of
 0.47, swept arm of 0.00, and sweep of 0.0)
 total element speed: 111.6 m/s
 corkscrew path from disk: 33.2 deg (arctan(61.1/ 93.4))
 element pitch: 31.6 deg (prop= 11.6 + incidence= 20.0 + cyclic)
 blade element geometric angle of attack: -1.6 deg

For Reynolds number: rho= 0.7985, chord= 22.68, dynamic_viscosity =16.5794 millionths (divide by
 this to get Re min Meg), Re_meg= 1.0926 million at this element

element center at 45% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
 For lookup, alpha airflow= -1.6, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha
 divisor= 1.0000, alpha lookup= -1.9
 For lookup, alpha airflow= -1.6, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8884, slat alpha
 divisor= 1.0000, alpha lookup= -2.0
 For lo RE using Reynolds number= 1.0000 million
 For hi RE using Reynolds number= 8.3000 million
 mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2382 and 0.2915, interpolated Cl= 0.2389
 mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0075 and 0.0070, interpolated Cd= 0.0075
 mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
 Mach of 0.3450 yields Prandtl-Glauert Cl multiplier of 1.0654 to take Cl to 0.2545
 Mach of 0.3450 yields Prandtl-Glauert Cm multiplier of 1.0654 to take Cm to -0.0746
 Mach of 0.3450 with a Cl-based drag divergence Mach number of 0.7246 takes the Cd to 0.0075
 (increase of 0.0000)

final Cl 0.2545
 final Cd 0.0075
 final Cm -0.0746

finite-wing lift multiplier of 0.9442 takes cl to 0.2403
 e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0019
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0019, which takes total Cd to 0.0094
 cl multiplied by 1.0000 to get to 0.2403 to lower pressure from Mach 0.9 to 1.3.

$\text{lift} = 0.2403 * 0.0211 * \text{sqr}(111.6) * 0.7985 / 2 = 25.3 \text{ N}$
 $\text{drag} = 0.0094 * 0.0211 * \text{sqr}(111.6) * 0.7985 / 2 = 1.0 \text{ N}$
 $\text{momt} = -0.0746 * 0.0211 * \text{sqr}(111.6) * 0.7985 / 2 * 0.20 = -1.6 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
 airflow path: 25.3 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
 airflow path: 1.0 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: -13.8 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -21.1 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.8 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.5 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -14.6 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -20.6 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.47 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 6.9 N*m, running drag torque on the shaft= 79.9 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -20.6 N,
running thrust normal to the disc in disc coordinate system= 261.9 N

total force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
total force in aircraft coords lateral up: -14.6 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -20.6 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.47, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 4.8)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -20.6 (Z acf= -2.76, N PRP on COG acf coord= -16.7)

.....
prop 1, ELEMENT INDEX 6 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 5.9 m/s BEHIND the disc
propwash speed increment at ring: 3.0 m/s AT the disc after mult by 0.5210, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 61.4 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 116.4 m/s (from omega of 209.31 rad/sec, rad arm of 0.57, swept arm of 0.00, and sweep of 0.0)
total element speed: 131.6 m/s
corkscrew path from disk: 27.8 deg (arctan(61.4/ 116.4))
element pitch: 25.8 deg (prop= 11.6 + incidence= 14.3 + cyclic)
blade element geometric angle of attack: -2.0 deg

For Reynolds number: rho= 0.7985, chord= 26.74, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.2877 million at this element

element center at 55% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -2.0, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -2.3
For lookup, alpha airflow= -2.0, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8880, slat alpha

divisor= 1.0000, alpha lookup= -2.3

For lo RE using Reynolds number= 1.0000 million

For hi RE using Reynolds number= 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2076 and 0.2609, interpolated Cl= 0.2097

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0077 and 0.0071, interpolated Cd= 0.0077

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.4066 yields Prandtl-Glauert Cl multiplier of 1.0946 to take Cl to 0.2295

Mach of 0.4066 yields Prandtl-Glauert Cm multiplier of 1.0946 to take Cm to -0.0766

Mach of 0.4066 with a Cl-based drag divergence Mach number of 0.7270 takes the Cd to 0.0077

(increase of 0.0000)

final Cl 0.2295

final Cd 0.0077

final Cm -0.0766

finite-wing lift multiplier of 0.9440 takes cl to 0.2167

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0016

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0016, which takes total Cd to 0.0093

cl multiplied by 1.0000 to get to 0.2167 to lower pressure from Mach 0.9 to 1.3.

lift= 0.2167 * 0.0211 * sqrt(131.6) * 0.7985 / 2 = 31.6 N

drag= 0.0093 * 0.0211 * sqrt(131.6) * 0.7985 / 2 = 1.4 N

momt=-0.0766 * 0.0211 * sqrt(131.6) * 0.7985 / 2 * 0.20= -2.3 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 31.6 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.4 N

lift-only force in disc coords lateral right: -14.8 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -28.0 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -1.2 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -16.0 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -27.3 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.57 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 9.1 N*m, running drag torque on the shaft= 89.0 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -27.3 N,
running thrust normal to the disc in disc coordinate system= 289.2 N

totl force in aircraft coords lateral right: -16.0 (positive pulling the aircraft right)

totl force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)

totl force in aircraft coords axial aft: -27.3 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.60, M PRP on COG acf coord= -11.7)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -27.3 (Z acf=

-2.76, N PRP on COG acf coord= -16.7)

.....
prop 1, ELEMENT INDEX 6 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 8.9 m/s BEHIND the disc
propwash speed increment at ring: 4.6 m/s AT the disc after mult by 0.5155, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 63.0 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 124.1 m/s (from omega of 209.31 rad/sec, rad arm of
0.57, swept arm of 0.00, and sweep of 0.0)
total element speed: 139.2 m/s
corkscrew path from disk: 26.9 deg (arctan(63.0/ 124.1))
element pitch: 25.8 deg (prop= 11.6 + incidence= 14.3 + cyclic)
blade element geometric angle of attack: -1.1 deg

For Reynolds number: rho= 0.7985, chord= 28.28, dynamic_viscosity =16.5794 millionths (divide by
this to get Re in Meg), Re_meg= 1.3623 million at this element

element center at 55% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -1.1, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8861, slat alpha
divisor= 1.0000, alpha lookup= -1.5
For lookup, alpha airflow= -1.1, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8861, slat alpha
divisor= 1.0000, alpha lookup= -1.5
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2789 and 0.3321, interpolated Cl= 0.2816
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0073 and 0.0069, interpolated Cd= 0.0073
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.4302 yields Prandtl-Glauert Cl multiplier of 1.1077 to take Cl to 0.3119
Mach of 0.4302 yields Prandtl-Glauert Cm multiplier of 1.1077 to take Cm to -0.0775
Mach of 0.4302 with a Cl-based drag divergence Mach number of 0.7188 takes the Cd to 0.0073
(increase of 0.0000)

final Cl 0.3119
final Cd 0.0073
final Cm -0.0775

finite-wing lift multiplier of 0.9431 takes cl to 0.2941
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0030
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0030, which takes total Cd to 0.0102
cl multiplied by 1.0000 to get to 0.2941 to lower pressure from Mach 0.9 to 1.3.

lift= 0.2941 * 0.0211 * sqr(139.2) * 0.7985 / 2 = 48.1 N
drag= 0.0102 * 0.0211 * sqr(139.2) * 0.7985 / 2 = 1.7 N
momt=-0.0775 * 0.0211 * sqr(139.2) * 0.7985 / 2 * 0.20= -2.6 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 48.1 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.7 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 21.8 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -42.8 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 1.5 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.8 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 23.2 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -42.1 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.57 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 13.3 N*m, running drag torque on the shaft= 102.3 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -42.1 N,
running thrust normal to the disc in disc coordinate system= 331.3 N

total force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
total force in aircraft coords lateral up: 23.2 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -42.1 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
0.57, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
0.03, M PRP on COG acf coord= -13.0)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -42.1 (Z acf=
-2.76, N PRP on COG acf coord= -40.7)

.....
prop 1, ELEMENT INDEX 6 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 8.7 m/s BEHIND the disc
propwash speed increment at ring: 4.2 m/s AT the disc after mult by 0.4790, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 62.6 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 122.9 m/s (from omega of 209.31 rad/sec, rad arm of
0.57, swept arm of 0.00, and sweep of 0.0)
total element speed: 137.9 m/s
corkscrew path from disk: 27.0 deg (arctan(62.6/ 122.9))
element pitch: 25.8 deg (prop= 11.6 + incidence= 14.3 + cyclic)
blade element geometric angle of attack: -1.1 deg

For Reynolds number: rho= 0.7985, chord= 28.03, dynamic_viscosity =16.5794 millionths (divide by
this to get Re in Meg), Re_meg= 1.3501 million at this element

element center at 55% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -1.1, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -1.5

For lookup, alpha airflow= -1.1, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -1.6

For lo RE using Reynolds number= 1.0000 million

For hi RE using Reynolds number= 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2749 and 0.3280, interpolated Cl= 0.2775

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0073 and 0.0069, interpolated Cd= 0.0073

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.4263 yields Prandtl-Glauert Cl multiplier of 1.1055 to take Cl to 0.3067

Mach of 0.4263 yields Prandtl-Glauert Cm multiplier of 1.1055 to take Cm to -0.0774

Mach of 0.4263 with a Cl-based drag divergence Mach number of 0.7193 takes the Cd to 0.0073 (increase of 0.0000)

final Cl 0.3067

final Cd 0.0073

final Cm -0.0774

finite-wing lift multiplier of 0.9426 takes cl to 0.2891

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0029

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0029, which takes total Cd to 0.0102

cl multiplied by 1.0000 to get to 0.2891 to lower pressure from Mach 0.9 to 1.3.

lift= $0.2891 * 0.0211 * \sqrt{137.9} * 0.7985 / 2 = 46.4 \text{ N}$

drag= $0.0102 * 0.0211 * \sqrt{137.9} * 0.7985 / 2 = 1.6 \text{ N}$

momt=-0.0774 * 0.0211 * $\sqrt{137.9} * 0.7985 / 2 * 0.20 = -2.5 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element airflow path: 46.4 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element airflow path: 1.6 N

lift-only force in disc coords lateral right: 21.0 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -41.4 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 1.5 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 22.5 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -40.6 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.57 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 12.9 N*m, running drag torque on the shaft= 115.2 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -40.6 N, running thrust normal to the disc in disc coordinate system= 371.9 N

totl force in aircraft coords lateral right: 22.5 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -40.6 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
-0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
-0.54, M PRP on COG acf coord= 9.0)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -40.6 (Z acf=
-2.76, N PRP on COG acf coord= -40.7)

.....
prop 1, ELEMENT INDEX 6 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 5.6 m/s BEHIND the disc
propwash speed increment at ring: 2.8 m/s AT the disc after mult by 0.4845, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 61.1 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 115.2 m/s (from omega of 209.31 rad/sec, rad arm of
0.57, swept arm of 0.00, and sweep of 0.0)
total element speed: 130.4 m/s
corkscrew path from disk: 27.9 deg (arctan(61.1/ 115.2))
element pitch: 25.8 deg (prop= 11.6 + incidence= 14.3 + cyclic)
blade element geometric angle of attack: -2.1 deg

For Reynolds number: rho= 0.7985, chord= 26.50, dynamic_viscosity =16.5794 millionths (divide by
this to get Re min Meg), Re_meg= 1.2762 million at this element

element center at 55% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -2.1, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -2.4
For lookup, alpha airflow= -2.1, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -2.5
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1985 and 0.2518, interpolated Cl= 0.2005
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0078 and 0.0072, interpolated Cd= 0.0077
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.4030 yields Prandtl-Glauert Cl multiplier of 1.0927 to take Cl to 0.2191
Mach of 0.4030 yields Prandtl-Glauert Cm multiplier of 1.0927 to take Cm to -0.0765
Mach of 0.4030 with a Cl-based drag divergence Mach number of 0.7281 takes the Cd to 0.0077
(increase of 0.0000)

final Cl 0.2191
final Cd 0.0077
final Cm -0.0765

finite-wing lift multiplier of 0.9442 takes c_l to 0.2069
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives C_{di} of 0.0014
ground effect C_{di} multiplier of 1.0000 takes C_{di} to 0.0014, which takes total C_d to 0.0092
 c_l multiplied by 1.0000 to get to 0.2069 to lower pressure from Mach 0.9 to 1.3.

$$\begin{aligned}\text{lift} &= 0.2069 * 0.0211 * \text{sqr}(130.4) * 0.7985 / 2 &= 29.7 \text{ N} \\ \text{drag} &= 0.0092 * 0.0211 * \text{sqr}(130.4) * 0.7985 / 2 &= 1.3 \text{ N} \\ \text{momt} &= -0.0765 * 0.0211 * \text{sqr}(130.4) * 0.7985 / 2 * 0.20 &= -2.2 \text{ N-m}\end{aligned}$$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 29.7 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.3 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -13.9 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -26.2 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -1.2 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -15.1 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -25.6 (positive pulling airplane aft or helicopter down)

radial arm of this element = 0.57 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and
6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock) = 8.6 N*m, running drag torque on the shaft =
123.8 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above = -25.6 N,
running thrust normal to the disc in disc coordinate system = 397.5 N

totl force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -15.1 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -25.6 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf =
-0.57, L PRP on COG acf coord = 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf =
0.03, M PRP on COG acf coord = 8.2)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -25.6 (Z acf =
-2.76, N PRP on COG acf coord = -26.1)

.....
prop 1, ELEMENT INDEX 7 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is $v_x = 3.3$ $v_y = 4.5$ $v_z = 58.4$ m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 5.2 m/s BEHIND the disc
propwash speed increment at ring: 2.7 m/s AT the disc after mult by 0.5248, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 61.1 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 138.1 m/s (from omega of 209.31 rad/sec, rad arm of

0.68, swept arm of 0.00, and sweep of 2.3)
total element speed: 150.9 m/s
corkscrew path from disk: 23.9 deg (arctan(61.1/ 138.1))
element pitch: 21.2 deg (prop= 11.6 + incidence= 9.6 + cyclic)
blade element geometric angle of attack: -2.7 deg

For Reynolds number: rho= 0.7985, chord= 30.20, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.4547 million at this element

element center at 65% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -2.7, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -2.9
For lookup, alpha airflow= -2.7, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -3.0
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1536 and 0.2069, interpolated Cl= 0.1570
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0082 and 0.0074, interpolated Cd= 0.0081
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.4663 yields Prandtl-Glauert Cl multiplier of 1.1304 to take Cl to 0.1774
Mach of 0.4663 yields Prandtl-Glauert Cm multiplier of 1.1304 to take Cm to -0.0791
Mach of 0.4663 with a Cl-based drag divergence Mach number of 0.7323 takes the Cd to 0.0081
(increase of 0.0000)

final Cl 0.1774
final Cd 0.0081
final Cm -0.0791

finite-wing lift multiplier of 0.9440 takes cl to 0.1675
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0009
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0009, which takes total Cd to 0.0091
cl multiplied by 1.0000 to get to 0.1675 to lower pressure from Mach 0.9 to 1.3.

lift= 0.1675 * 0.0208 * sqr(150.9) * 0.7985 / 2 = 31.7 N
drag= 0.0091 * 0.0208 * sqr(150.9) * 0.7985 / 2 = 1.7 N
momt=-0.0791 * 0.0208 * sqr(150.9) * 0.7985 / 2 * 0.20= -3.0 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 31.7 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 1.7 N

lift-only force in disc coords lateral right: -12.8 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -29.0 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -1.6 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -14.4 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -28.3 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.68 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 9.7 N*m, running drag torque on the shaft= 133.5 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -28.3 N, running thrust normal to the disc in disc coordinate system= 425.8 N

total force in aircraft coords lateral right: -14.4 (positive pulling the aircraft right)

total force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)

total force in aircraft coords axial aft: -28.3 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.71, M PRP on COG acf coord= -11.7)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -28.3 (Z acf= -2.76, N PRP on COG acf coord= -26.1)

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prop 1, ELEMENT INDEX 7 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 7.8 m/s BEHIND the disc

propwash speed increment at ring: 4.1 m/s AT the disc after mult by 0.5184, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 62.5 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 145.9 m/s (from omega of 209.31 rad/sec, rad arm of 0.68, swept arm of 0.00, and sweep of 2.3)

total element speed: 158.5 m/s

corkscrew path from disk: 23.2 deg (arctan(62.5/ 145.9))

element pitch: 21.2 deg (prop= 11.6 + incidence= 9.6 + cyclic)

blade element geometric angle of attack: -2.0 deg

For Reynolds number: rho= 0.7985, chord= 31.74, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.5285 million at this element

element center at 65% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -2.0, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -2.3

For lookup, alpha airflow= -2.0, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -2.4

For lo RE using Reynolds number= 1.0000 million

For hi RE using Reynolds number= 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2071 and 0.2603, interpolated Cl= 0.2110

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0077 and 0.0071, interpolated Cd= 0.0077

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.4900 yields Prandtl-Glauert Cl multiplier of 1.1471 to take Cl to 0.2420
Mach of 0.4900 yields Prandtl-Glauert Cm multiplier of 1.1471 to take Cm to -0.0803
Mach of 0.4900 with a Cl-based drag divergence Mach number of 0.7258 takes the Cd to 0.0077
(increase of 0.0000)

final Cl 0.2420
final Cd 0.0077
final Cm -0.0803

finite-wing lift multiplier of 0.9431 takes cl to 0.2282
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0018
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0018, which takes total Cd to 0.0094
cl multiplied by 1.0000 to get to 0.2282 to lower pressure from Mach 0.9 to 1.3.

lift= $0.2282 * 0.0208 * \sqrt{158.5} * 0.7985 / 2 = 47.7 \text{ N}$
drag= $0.0094 * 0.0208 * \sqrt{158.5} * 0.7985 / 2 = 2.0 \text{ N}$
momt= $-0.0803 * 0.0208 * \sqrt{158.5} * 0.7985 / 2 * 0.20 = -3.4 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 47.7 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 2.0 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 18.8 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -43.8 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 1.8 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.8 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 20.6 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -43.0 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.68 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and
6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 13.9 N*m, running drag torque on the shaft=
147.4 N*m
thrust normal to the disc in disc coordinate system, from the total disc axial force above= -43.0 N,
running thrust normal to the disc in disc coordinate system= 468.8 N

totl force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 20.6 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -43.0 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
0.68, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
0.03, M PRP on COG acf coord= -13.1)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -43.0 (Z acf=
-2.76, N PRP on COG acf coord= -55.2)

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prop 1, ELEMENT INDEX 7 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....
So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
 propwash speed increment behind ring: 7.7 m/s BEHIND the disc
 propwash speed increment at ring: 3.7 m/s AT the disc after mult by 0.4752, and turb mult of 0.0312 is added on this as well
 total incoming airspeed: 62.0 m/s (geometric inflow + speed increment above + flapping)
 speed due to rotation and translation of disc: 144.7 m/s (from omega of 209.31 rad/sec, rad arm of 0.68, swept arm of 0.00, and sweep of 2.3)
 total element speed: 157.3 m/s
 corkscrew path from disk: 23.2 deg (arctan(62.0/ 144.7))
 element pitch: 21.2 deg (prop= 11.6 + incidence= 9.6 + cyclic)
 blade element geometric angle of attack: -2.0 deg

For Reynolds number: rho= 0.7985, chord= 31.49, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.5166 million at this element

element center at 65% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
 For lookup, alpha airflow= -2.0, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -2.3
 For lookup, alpha airflow= -2.0, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8853, slat alpha divisor= 1.0000, alpha lookup= -2.4
 For lo RE using Reynolds number= 1.0000 million
 For hi RE using Reynolds number= 8.3000 million
 mid foil: Cl lookups for lo and hi Reynolds numbers: 0.2051 and 0.2582, interpolated Cl= 0.2089
 mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0077 and 0.0071, interpolated Cd= 0.0077
 mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
 Mach of 0.4862 yields Prandtl-Glauert Cl multiplier of 1.1443 to take Cl to 0.2390
 Mach of 0.4862 yields Prandtl-Glauert Cm multiplier of 1.1443 to take Cm to -0.0801
 Mach of 0.4862 with a Cl-based drag divergence Mach number of 0.7261 takes the Cd to 0.0077 (increase of 0.0000)

final Cl 0.2390
 final Cd 0.0077
 final Cm -0.0801

finite-wing lift multiplier of 0.9426 takes cl to 0.2253
 e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0018
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0018, which takes total Cd to 0.0094
 cl multiplied by 1.0000 to get to 0.2253 to lower pressure from Mach 0.9 to 1.3.

lift= 0.2253 * 0.0208 * sqr(157.3) * 0.7985 / 2 = 46.3 N
 drag= 0.0094 * 0.0208 * sqr(157.3) * 0.7985 / 2 = 1.9 N
 momt=-0.0801 * 0.0208 * sqr(157.3) * 0.7985 / 2 * 0.20= -3.3 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
 airflow path: 46.3 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
 airflow path: 1.9 N

lift-only force in disc coords lateral right: 18.2 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -42.6 (positive pulling airplane aft or helicopter down)

 drag-only force in disc coords lateral right: 1.8 (positive pulling airplane right or helicopter right)
 drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
 drag-only force in disc coords axial aft: 0.8 (positive pulling airplane aft or helicopter down)

 total force in disc coords lateral right: 20.0 (positive pulling airplane right or helicopter right)
 total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
 total force in disc coords axial aft: -41.8 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.68 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 13.5 N*m, running drag torque on the shaft= 160.9 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -41.8 N, running thrust normal to the disc in disc coordinate system= 510.6 N

totl force in aircraft coords lateral right: 20.0 (positive pulling the aircraft right)
 totl force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
 totl force in aircraft coords axial aft: -41.8 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.00, L PRP on COG acf coord= 0.0)
 considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= -0.65, M PRP on COG acf coord= 13.9)
 considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -41.8 (Z acf= -2.76, N PRP on COG acf coord= -55.2)

.....
 prop 1, ELEMENT INDEX 7 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
 propwash speed increment behind ring: 5.0 m/s BEHIND the disc
 propwash speed increment at ring: 2.4 m/s AT the disc after mult by 0.4816, and turb mult of 0.0312 is added on this as well
 total incoming airspeed: 60.8 m/s (geometric inflow + speed increment above + flapping)
 speed due to rotation and translation of disc: 136.9 m/s (from omega of 209.31 rad/sec, rad arm of 0.68, swept arm of 0.00, and sweep of 2.3)
 total element speed: 149.7 m/s
 corkscrew path from disk: 23.9 deg (arctan(60.8/ 136.9))
 element pitch: 21.2 deg (prop= 11.6 + incidence= 9.6 + cyclic)
 blade element geometric angle of attack: -2.7 deg

For Reynolds number: rho= 0.7985, chord= 29.97, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 1.4433 million at this element

element center at 65% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -2.7, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= -2.9
 For lookup, alpha airflow= -2.7, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= -3.0
 For lo RE using Reynolds number= 1.0000 million
 For hi RE using Reynolds number= 8.3000 million
 mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1483 and 0.2016, interpolated Cl= 0.1516
 mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0082 and 0.0074, interpolated Cd= 0.0082
 mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
 Mach of 0.4626 yields Prandtl-Glauert Cl multiplier of 1.1280 to take Cl to 0.1710
 Mach of 0.4626 yields Prandtl-Glauert Cm multiplier of 1.1280 to take Cm to -0.0790
 Mach of 0.4626 with a Cl-based drag divergence Mach number of 0.7329 takes the Cd to 0.0082
 (increase of 0.0000)

final Cl 0.1710
 final Cd 0.0082
 final Cm -0.0790

finite-wing lift multiplier of 0.9442 takes cl to 0.1614
 e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0009
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0009, which takes total Cd to 0.0090
 cl multiplied by 1.0000 to get to 0.1614 to lower pressure from Mach 0.9 to 1.3.

lift= $0.1614 * 0.0208 * \sqrt{149.7} * 0.7985 / 2 = 30.0 \text{ N}$
 drag= $0.0090 * 0.0208 * \sqrt{149.7} * 0.7985 / 2 = 1.7 \text{ N}$
 momt=-0.0790 * 0.0208 * $\sqrt{149.7} * 0.7985 / 2 * 0.20 = -2.9 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
 airflow path: 30.0 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
 airflow path: 1.7 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: -12.2 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -27.5 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
 drag-only force in disc coords lateral up: -1.5 (positive pulling airplane up or helicopter aft)
 drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
 total force in disc coords lateral up: -13.7 (positive pulling airplane up or helicopter aft)
 total force in disc coords axial aft: -26.8 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.68 m
 drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 9.3 N*m, running drag torque on the shaft= 170.2 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -26.8 N,
 running thrust normal to the disc in disc coordinate system= 537.4 N

totl force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
 totl force in aircraft coords lateral up: -13.7 (positive pulling the aircraft up)
 totl force in aircraft coords axial aft: -26.8 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
 -0.68, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 13.1)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -26.8 (Z acf= -2.76, N PRP on COG acf coord= -37.1)

.....
prop 1, ELEMENT INDEX 8 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 3.8 m/s BEHIND the disc
propwash speed increment at ring: 2.0 m/s AT the disc after mult by 0.5286, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 60.4 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 159.9 m/s (from omega of 209.31 rad/sec, rad arm of 0.78, swept arm of 0.01, and sweep of 6.4)
total element speed: 169.8 m/s
corkscrew path from disk: 20.7 deg (arctan(60.4/ 159.9))
element pitch: 17.4 deg (prop= 11.6 + incidence= 5.8 + cyclic)
blade element geometric angle of attack: -3.3 deg

For Reynolds number: rho= 0.7985, chord= 31.78, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 1.5307 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -3.3, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -3.5
For lookup, alpha airflow= -3.3, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -3.5
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1028 and 0.1560, interpolated Cl= 0.1066
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0087 and 0.0078, interpolated Cd= 0.0086
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.5249 yields Prandtl-Glauert Cl multiplier of 1.1748 to take Cl to 0.1253
Mach of 0.5249 yields Prandtl-Glauert Cm multiplier of 1.1748 to take Cm to -0.0822
Mach of 0.5249 with a Cl-based drag divergence Mach number of 0.7375 takes the Cd to 0.0086 (increase of 0.0000)

final Cl 0.1253
final Cd 0.0086
final Cm -0.0822

finite-wing lift multiplier of 0.9440 takes cl to 0.1183
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0005
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0005, which takes total Cd to 0.0091
cl multiplied by 1.0000 to get to 0.1183 to lower pressure from Mach 0.9 to 1.3.

lift= 0.1183 * 0.0194 * sqr(169.8) * 0.7985 / 2 = 26.5 N

$$\text{drag} = 0.0091 * 0.0194 * \text{sqr}(169.8) * 0.7985 / 2 = 2.0 \text{ N}$$

$$\text{momt} = -0.0822 * 0.0194 * \text{sqr}(169.8) * 0.7985 / 2 * 0.19 = -3.4 \text{ N-m}$$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
 airflow path: 26.5 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
 airflow path: 2.0 N

lift-only force in disc coords lateral right: -9.4 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -24.8 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -1.9 (positive pulling airplane right or helicopter right)
 drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -11.3 (positive pulling airplane right or helicopter right)
 total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
 total force in disc coords axial aft: -24.0 (positive pulling airplane aft or helicopter down)

radial arm of this element = 0.78 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock) = 8.8 N*m, running drag torque on the shaft = 179.0 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above = -24.0 N, running thrust normal to the disc in disc coordinate system = 561.4 N

total force in aircraft coords lateral right: -11.3 (positive pulling the aircraft right)
 total force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)
 total force in aircraft coords axial aft: -24.0 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf = 0.00, L PRP on COG acf coord = 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf = 0.81, M PRP on COG acf coord = -6.4)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -24.0 (Z acf = -2.76, N PRP on COG acf coord = -37.1)

.....
 prop 1, ELEMENT INDEX 8 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

So final speed is vx = 3.3 vy = 4.5 vz = 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 6.0 m/s BEHIND the disc

propwash speed increment at ring: 3.2 m/s AT the disc after mult by 0.5212, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 61.5 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 167.6 m/s (from omega of 209.31 rad/sec, rad arm of 0.78, swept arm of 0.01, and sweep of 6.4)

total element speed: 177.5 m/s

corkscrew path from disk: 20.2 deg (arctan(61.5/ 167.6))

element pitch: 17.4 deg (prop = 11.6 + incidence = 5.8 + cyclic)

blade element geometric angle of attack: -2.8 deg

For Reynolds number: $\rho = 0.7985$, chord = 33.21, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), $Re_{meg} = 1.5994$ million at this element

element center at 75% span: (wgt root foil = 0.0000, wgt mid foil = 1.0000, wgt tip foil = 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow = -2.8, alpha zero-lift = -4.6, finite wing alpha multiplier = 0.8861, slat alpha divisor = 1.0000, alpha lookup = -3.0

For lookup, alpha airflow = -2.8, alpha zero-lift = -5.3, finite wing alpha multiplier = 0.8861, slat alpha divisor = 1.0000, alpha lookup = -3.1

For lo RE using Reynolds number = 1.0000 million

For hi RE using Reynolds number = 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1452 and 0.1983, interpolated Cl = 0.1495

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0082 and 0.0075, interpolated Cd = 0.0082

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm = -0.0700

Mach of 0.5484 yields Prandtl-Glauert Cl multiplier of 1.1959 to take Cl to 0.1788

Mach of 0.5484 yields Prandtl-Glauert Cm multiplier of 1.1959 to take Cm to -0.0837

Mach of 0.5484 with a Cl-based drag divergence Mach number of 0.7321 takes the Cd to 0.0082 (increase of 0.0000)

final Cl 0.1788

final Cd 0.0082

final Cm -0.0837

finite-wing lift multiplier of 0.9431 takes cl to 0.1686

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0010

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0010, which takes total Cd to 0.0091

cl multiplied by 1.0000 to get to 0.1686 to lower pressure from Mach 0.9 to 1.3.

lift = $0.1686 * 0.0194 * \sqrt{177.5} * 0.7985 / 2 = 41.2 \text{ N}$

drag = $0.0091 * 0.0194 * \sqrt{177.5} * 0.7985 / 2 = 2.2 \text{ N}$

momt = $-0.0837 * 0.0194 * \sqrt{177.5} * 0.7985 / 2 * 0.19 = -3.8 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element airflow path: 41.2 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element airflow path: 2.2 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: 14.2 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -38.7 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: 2.1 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.8 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: 16.3 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -37.9 (positive pulling airplane aft or helicopter down)

radial arm of this element = 0.78 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock) = 12.7 N*m, running drag torque on the shaft = 191.7 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -37.9 N,
running thrust normal to the disc in disc coordinate system= 599.3 N

totl force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 16.3 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -37.9 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
0.78, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
0.03, M PRP on COG acf coord= -7.5)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -37.9 (Z acf=
-2.76, N PRP on COG acf coord= -66.6)

.....
prop 1, ELEMENT INDEX 8 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 5.9 m/s BEHIND the disc
propwash speed increment at ring: 2.8 m/s AT the disc after mult by 0.4714, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 61.2 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 166.5 m/s (from omega of 209.31 rad/sec, rad arm of
0.78, swept arm of 0.01, and sweep of 6.4)
total element speed: 176.2 m/s
corkscrew path from disk: 20.2 deg (arctan(61.2/ 166.5))
element pitch: 17.4 deg (prop= 11.6 + incidence= 5.8 + cyclic)
blade element geometric angle of attack: -2.8 deg

For Reynolds number: rho= 0.7985, chord= 32.98, dynamic_viscosity =16.5794 millionths (divide by
this to get Re in Meg), Re_meg= 1.5885 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -2.8, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8853, slat alpha
divisor= 1.0000, alpha lookup= -3.0
For lookup, alpha airflow= -2.8, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8853, slat alpha
divisor= 1.0000, alpha lookup= -3.1
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.1439 and 0.1970, interpolated Cl= 0.1481
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0082 and 0.0075, interpolated Cd= 0.0082
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.5447 yields Prandtl-Glauert Cl multiplier of 1.1924 to take Cl to 0.1766
Mach of 0.5447 yields Prandtl-Glauert Cm multiplier of 1.1924 to take Cm to -0.0835
Mach of 0.5447 with a Cl-based drag divergence Mach number of 0.7323 takes the Cd to 0.0082
(increase of 0.0000)

final Cl 0.1766

final Cd 0.0082
final Cm -0.0835

finite-wing lift multiplier of 0.9426 takes c_l to 0.1665
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives C_{di} of 0.0010
ground effect C_{di} multiplier of 1.0000 takes C_{di} to 0.0010, which takes total Cd to 0.0091
 c_l multiplied by 1.0000 to get to 0.1665 to lower pressure from Mach 0.9 to 1.3.

lift= $0.1665 * 0.0194 * \sqrt{176.2} * 0.7985 / 2 = 40.1 \text{ N}$
drag= $0.0091 * 0.0194 * \sqrt{176.2} * 0.7985 / 2 = 2.2 \text{ N}$
momt= $-0.0835 * 0.0194 * \sqrt{176.2} * 0.7985 / 2 * 0.19 = -3.8 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 40.1 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 2.2 N

lift-only force in disc coords lateral right: 13.8 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -37.7 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 2.1 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.8 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 15.9 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -36.9 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.78 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 12.4 N*m, running drag torque on the shaft= 204.1 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -36.9 N,
running thrust normal to the disc in disc coordinate system= 636.2 N

totl force in aircraft coords lateral right: 15.9 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -36.9 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= -0.75, M PRP on COG acf coord= 20.1)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -36.9 (Z acf= -2.76, N PRP on COG acf coord= -66.6)

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prop 1, ELEMENT INDEX 8 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is $v_x = 3.3$ $v_y = 4.5$ $v_z = 58.4$ m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 3.7 m/s BEHIND the disc
propwash speed increment at ring: 1.8 m/s AT the disc after mult by 0.4788, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 60.1 m/s (geometric inflow + speed increment above + flapping)
 speed due to rotation and translation of disc: 158.7 m/s (from omega of 209.31 rad/sec, rad arm of 0.78, swept arm of 0.01, and sweep of 6.4)
 total element speed: 168.7 m/s
 corkscrew path from disk: 20.7 deg (arctan(60.1/ 158.7))
 element pitch: 17.4 deg (prop= 11.6 + incidence= 5.8 + cyclic)
 blade element geometric angle of attack: -3.4 deg

For Reynolds number: rho= 0.7985, chord= 31.56, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.5201 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
 For lookup, alpha airflow= -3.4, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= -3.5
 For lookup, alpha airflow= -3.4, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= -3.6
 For lo RE using Reynolds number= 1.0000 million
 For hi RE using Reynolds number= 8.3000 million
 mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0991 and 0.1524, interpolated Cl= 0.1029
 mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0088 and 0.0078, interpolated Cd= 0.0087
 mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
 Mach of 0.5212 yields Prandtl-Glauert Cl multiplier of 1.1718 to take Cl to 0.1206
 Mach of 0.5212 yields Prandtl-Glauert Cm multiplier of 1.1718 to take Cm to -0.0820
 Mach of 0.5212 with a Cl-based drag divergence Mach number of 0.7379 takes the Cd to 0.0087 (increase of 0.0000)

final Cl 0.1206
 final Cd 0.0087
 final Cm -0.0820

finite-wing lift multiplier of 0.9442 takes cl to 0.1139
 e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0004
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0004, which takes total Cd to 0.0091
 cl multiplied by 1.0000 to get to 0.1139 to lower pressure from Mach 0.9 to 1.3.

lift= 0.1139 * 0.0194 * sqr(168.7) * 0.7985 / 2 = 25.1 N
 drag= 0.0091 * 0.0194 * sqr(168.7) * 0.7985 / 2 = 2.0 N
 momt=-0.0820 * 0.0194 * sqr(168.7) * 0.7985 / 2 * 0.19= -3.4 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element airflow path: 25.1 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element airflow path: 2.0 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: -8.9 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -23.5 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
 drag-only force in disc coords lateral up: -1.9 (positive pulling airplane up or helicopter aft)
 drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -10.8 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -22.8 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.78 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 8.4 N*m, running drag torque on the shaft= 212.5 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -22.8 N,
running thrust normal to the disc in disc coordinate system= 659.0 N

total force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
total force in aircraft coords lateral up: -10.8 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -22.8 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.78, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 19.4)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -22.8 (Z acf= -2.76, N PRP on COG acf coord= -48.9)

.....
prop 1, ELEMENT INDEX 9 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 2.1 m/s BEHIND the disc
propwash speed increment at ring: 1.1 m/s AT the disc after mult by 0.5325, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 59.5 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 181.7 m/s (from omega of 209.31 rad/sec, rad arm of 0.88, swept arm of 0.03, and sweep of 10.3)
total element speed: 188.2 m/s
corkscrew path from disk: 18.1 deg (arctan(59.5/ 181.7))
element pitch: 14.2 deg (prop= 11.6 + incidence= 2.6 + cyclic)
blade element geometric angle of attack: -3.9 deg

For Reynolds number: rho= 0.7985, chord= 30.31, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.4597 million at this element

element center at 85% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -3.9, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -4.0
For lookup, alpha airflow= -3.9, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -4.0
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0564 and 0.1097, interpolated Cl= 0.0597
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0093 and 0.0082, interpolated Cd= 0.0093
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.5815 yields Prandtl-Glauert Cl multiplier of 1.2292 to take Cl to 0.0734
Mach of 0.5815 yields Prandtl-Glauert Cm multiplier of 1.2292 to take Cm to -0.0860
Mach of 0.5815 with a Cl-based drag divergence Mach number of 0.7427 takes the Cd to 0.0093
(increase of 0.0000)

final Cl 0.0734
final Cd 0.0093
final Cm -0.0860

finite-wing lift multiplier of 0.9440 takes cl to 0.0693
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0002
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0002, which takes total Cd to 0.0094
cl multiplied by 1.0000 to get to 0.0693 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0693 * 0.0167 * sqrt(188.2) * 0.7985 / 2 = 16.3 N
drag= 0.0094 * 0.0167 * sqrt(188.2) * 0.7985 / 2 = 2.2 N
momt=-0.0860 * 0.0167 * sqrt(188.2) * 0.7985 / 2 * 0.16= -3.3 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 16.3 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 2.2 N

lift-only force in disc coords lateral right: -5.1 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -15.5 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -2.1 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -7.2 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -14.8 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.88 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 6.4 N*m, running drag torque on the shaft= 218.8 N*m
thrust normal to the disc in disc coordinate system, from the total disc axial force above= -14.8 N,
running thrust normal to the disc in disc coordinate system= 673.9 N

totl force in aircraft coords lateral right: -7.2 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -14.8 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.91, M PRP on COG acf coord= 5.9)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -14.8 (Z acf= -2.76, N PRP on COG acf coord= -48.9)

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prop 1, ELEMENT INDEX 9 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS

So final speed is $v_x = 3.3$ $v_y = 4.5$ $v_z = 58.4$ m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 3.8 m/s BEHIND the disc
propwash speed increment at ring: 2.0 m/s AT the disc after mult by 0.5240, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 60.4 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 189.5 m/s (from omega of 209.31 rad/sec, rad arm of 0.88, swept arm of 0.03, and sweep of 10.3)
total element speed: 195.7 m/s
corkscrew path from disk: 17.7 deg ($\arctan(60.4 / 189.5)$)
element pitch: 14.2 deg (prop = 11.6 + incidence = 2.6 + cyclic)
blade element geometric angle of attack: -3.4 deg

For Reynolds number: $\rho = 0.7985$, chord = 31.52, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), $Re_{meg} = 1.5181$ million at this element

element center at 85% span: (wgt root foil = 0.0000, wgt mid foil = 1.0000, wgt tip foil = 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow = -3.4, alpha zero-lift = -4.6, finite wing alpha multiplier = 0.8861, slat alpha divisor = 1.0000, alpha lookup = -3.6
For lookup, alpha airflow = -3.4, alpha zero-lift = -5.3, finite wing alpha multiplier = 0.8861, slat alpha divisor = 1.0000, alpha lookup = -3.6
For lo RE using Reynolds number = 1.0000 million
For hi RE using Reynolds number = 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0922 and 0.1454, interpolated Cl = 0.0960
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0088 and 0.0078, interpolated Cd = 0.0088
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm = -0.0700
Mach of 0.6047 yields Prandtl-Glauert Cl multiplier of 1.2556 to take Cl to 0.1205
Mach of 0.6047 yields Prandtl-Glauert Cm multiplier of 1.2556 to take Cm to -0.0879
Mach of 0.6047 with a Cl-based drag divergence Mach number of 0.7380 takes the Cd to 0.0088 (increase of 0.0000)

final Cl 0.1205
final Cd 0.0088
final Cm -0.0879

finite-wing lift multiplier of 0.9431 takes cl to 0.1136
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0004
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0004, which takes total Cd to 0.0092
cl multiplied by 1.0000 to get to 0.1136 to lower pressure from Mach 0.9 to 1.3.

lift = $0.1136 * 0.0167 * \sqrt{195.7} * 0.7985 / 2 = 29.0$ N
drag = $0.0092 * 0.0167 * \sqrt{195.7} * 0.7985 / 2 = 2.3$ N
momt = $-0.0879 * 0.0167 * \sqrt{195.7} * 0.7985 / 2 * 0.16 = -3.6$ N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 29.0 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element

airflow path: 2.3 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 8.8 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -27.6 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 2.2 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 11.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -26.9 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.88 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 9.8 N*m, running drag torque on the shaft= 228.6 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -26.9 N,
running thrust normal to the disc in disc coordinate system= 700.8 N

total force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
total force in aircraft coords lateral up: 11.0 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -26.9 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.88, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 5.0)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -26.9 (Z acf= -2.76, N PRP on COG acf coord= -72.6)

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prop 1, ELEMENT INDEX 9 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
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So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 3.7 m/s BEHIND the disc
propwash speed increment at ring: 1.8 m/s AT the disc after mult by 0.4675, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 60.1 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 188.3 m/s (from omega of 209.31 rad/sec, rad arm of 0.88, swept arm of 0.03, and sweep of 10.3)
total element speed: 194.5 m/s
corkscrew path from disk: 17.7 deg (arctan(60.1/ 188.3))
element pitch: 14.2 deg (prop= 11.6 + incidence= 2.6 + cyclic)
blade element geometric angle of attack: -3.5 deg

For Reynolds number: rho= 0.7985, chord= 31.33, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.5091 million at this element

element center at 85% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -3.5, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8853, slat alpha
divisor= 1.0000, alpha lookup= -3.6
For lookup, alpha airflow= -3.5, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8853, slat alpha
divisor= 1.0000, alpha lookup= -3.7
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0899 and 0.1430, interpolated Cl= 0.0936
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0089 and 0.0079, interpolated Cd= 0.0088
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.6011 yields Prandtl-Glauert Cl multiplier of 1.2513 to take Cl to 0.1171
Mach of 0.6011 yields Prandtl-Glauert Cm multiplier of 1.2513 to take Cm to -0.0876
Mach of 0.6011 with a Cl-based drag divergence Mach number of 0.7383 takes the Cd to 0.0088
(increase of 0.0000)

final Cl 0.1171
final Cd 0.0088
final Cm -0.0876

finite-wing lift multiplier of 0.9426 takes cl to 0.1104
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0004
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0004, which takes total Cd to 0.0092
cl multiplied by 1.0000 to get to 0.1104 to lower pressure from Mach 0.9 to 1.3.

lift= $0.1104 * 0.0167 * \sqrt{194.5} * 0.7985 / 2 = 27.8 \text{ N}$
drag= $0.0092 * 0.0167 * \sqrt{194.5} * 0.7985 / 2 = 2.3 \text{ N}$
momt= $-0.0876 * 0.0167 * \sqrt{194.5} * 0.7985 / 2 * 0.16 = -3.6 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 27.8 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 2.3 N

lift-only force in disc coords lateral right: 8.5 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -26.5 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 2.2 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 10.7 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -25.8 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.88 m
drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and
6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 9.4 N*m, running drag torque on the shaft=
238.0 N*m
thrust normal to the disc in disc coordinate system, from the total disc axial force above= -25.8 N,
running thrust normal to the disc in disc coordinate system= 726.6 N

totl force in aircraft coords lateral right: 10.7 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -25.8 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf=
-0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf=
-0.85, M PRP on COG acf coord= 27.0)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -25.8 (Z acf=
-2.76, N PRP on COG acf coord= -72.6)

.....
prop 1, ELEMENT INDEX 9 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs,
but not this one)
propwash speed increment behind ring: 2.0 m/s BEHIND the disc
propwash speed increment at ring: 0.9 m/s AT the disc after mult by 0.4760, and turb mult
of 0.0312 is added on this as well
total incoming airspeed: 59.3 m/s (geometric inflow + speed increment above +
flapping)
speed due to rotation and translation of disc: 180.6 m/s (from omega of 209.31 rad/sec, rad arm of
0.88, swept arm of 0.03, and sweep of 10.3)
total element speed: 187.0 m/s
corkscrew path from disk: 18.2 deg (arctan(59.3/ 180.6))
element pitch: 14.2 deg (prop= 11.6 + incidence= 2.6 + cyclic)
blade element geometric angle of attack: -3.9 deg

For Reynolds number: rho= 0.7985, chord= 30.12, dynamic_viscosity =16.5794 millionths (divide by
this to get Re min Meg), Re_meg= 1.4508 million at this element

element center at 85% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -3.9, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -4.0
For lookup, alpha airflow= -3.9, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8884, slat alpha
divisor= 1.0000, alpha lookup= -4.1
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0528 and 0.1061, interpolated Cl= 0.0561
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0094 and 0.0082, interpolated Cd= 0.0093
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.5779 yields Prandtl-Glauert Cl multiplier of 1.2253 to take Cl to 0.0687
Mach of 0.5779 yields Prandtl-Glauert Cm multiplier of 1.2253 to take Cm to -0.0858
Mach of 0.5779 with a Cl-based drag divergence Mach number of 0.7431 takes the Cd to 0.0093
(increase of 0.0000)

final Cl 0.0687
final Cd 0.0093
final Cm -0.0858

finite-wing lift multiplier of 0.9442 takes cl to 0.0649
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0001
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0001, which takes total Cd to 0.0095

cl multiplied by 1.0000 to get to 0.0649 to lower pressure from Mach 0.9 to 1.3.

$$\begin{aligned}\text{lift} &= 0.0649 * 0.0167 * \text{sqr}(187.0) * 0.7985 / 2 &= 15.1 \text{ N} \\ \text{drag} &= 0.0095 * 0.0167 * \text{sqr}(187.0) * 0.7985 / 2 &= 2.2 \text{ N} \\ \text{momt} &= -0.0858 * 0.0167 * \text{sqr}(187.0) * 0.7985 / 2 * 0.16 &= -3.2 \text{ N-m}\end{aligned}$$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 15.1 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 2.2 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: -4.7 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -14.4 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -2.1 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.7 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -6.8 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -13.7 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.88 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 6.0 N*m, running drag torque on the shaft= 244.1 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -13.7 N,
running thrust normal to the disc in disc coordinate system= 740.2 N

totl force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: -6.8 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -13.7 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.88, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 26.6)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -13.7 (Z acf= -2.76, N PRP on COG acf coord= -60.6)

.....
prop 1, ELEMENT INDEX 10 OF 10, BLADE AT 0 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 0.4 m/s BEHIND the disc

propwash speed increment at ring: 0.2 m/s AT the disc after mult by 0.5363, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 58.6 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 203.7 m/s (from omega of 209.31 rad/sec, rad arm of 0.99, swept arm of 0.06, and sweep of 14.6)

total element speed: 205.1 m/s

corkscrew path from disk: 16.0 deg (arctan(58.6/ 203.7))

element pitch: 11.6 deg (prop= 11.6 + incidence= 0.0 + cyclic)
blade element geometric angle of attack: -4.5 deg

For Reynolds number: rho= 0.7985, chord= 24.58, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.1840 million at this element

element center at 95% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -4.5, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -4.5

For lookup, alpha airflow= -4.5, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8880, slat alpha divisor= 1.0000, alpha lookup= -4.5

For lo RE using Reynolds number= 1.0000 million

For hi RE using Reynolds number= 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0123 and 0.0655, interpolated Cl= 0.0136

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0100 and 0.0087, interpolated Cd= 0.0100

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.6340 yields Prandtl-Glauert Cl multiplier of 1.2931 to take Cl to 0.0176

Mach of 0.6340 yields Prandtl-Glauert Cm multiplier of 1.2931 to take Cm to -0.0905

Mach of 0.6340 with a Cl-based drag divergence Mach number of 0.7482 takes the Cd to 0.0100 (increase of 0.0000)

final Cl 0.0176

final Cd 0.0100

final Cm -0.0905

finite-wing lift multiplier of 0.9440 takes cl to 0.0166

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.1590, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0100

cl multiplied by 1.0000 to get to 0.0166 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0166 * 0.0122 * sqrt(205.1) * 0.7985 / 2 = 3.4 N

drag= 0.0100 * 0.0122 * sqrt(205.1) * 0.7985 / 2 = 2.1 N

momt=-0.0905 * 0.0122 * sqrt(205.1) * 0.7985 / 2 * 0.12= -2.2 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 3.4 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 2.1 N

lift-only force in disc coords lateral right: -0.9 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -3.3 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -2.0 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -2.9 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: 0.0 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -2.7 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.99 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 2.9 N*m, running drag torque on the shaft= 247.0 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -2.7 N, running thrust normal to the disc in disc coordinate system= 743.0 N

total force in aircraft coords lateral right: -2.9 (positive pulling the aircraft right)
total force in aircraft coords lateral up: 0.0 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -2.7 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.00, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 1.02, M PRP on COG acf coord= 23.9)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -2.7 (Z acf= -2.76, N PRP on COG acf coord= -60.6)

.....
prop 1, ELEMENT INDEX 10 OF 10, BLADE AT 90 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)
propwash speed increment behind ring: 1.5 m/s BEHIND the disc
propwash speed increment at ring: 0.8 m/s AT the disc after mult by 0.5268, and turb mult of 0.0312 is added on this as well
total incoming airspeed: 59.2 m/s (geometric inflow + speed increment above + flapping)
speed due to rotation and translation of disc: 211.5 m/s (from omega of 209.31 rad/sec, rad arm of 0.99, swept arm of 0.06, and sweep of 14.6)
total element speed: 212.5 m/s
corkscrew path from disk: 15.6 deg (arctan(59.2/ 211.5))
element pitch: 11.6 deg (prop= 11.6 + incidence= 0.0 + cyclic)
blade element geometric angle of attack: -4.0 deg

For Reynolds number: rho= 0.7985, chord= 25.47, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.2266 million at this element

element center at 95% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
For lookup, alpha airflow= -4.0, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -4.1
For lookup, alpha airflow= -4.0, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8861, slat alpha divisor= 1.0000, alpha lookup= -4.2
For lo RE using Reynolds number= 1.0000 million
For hi RE using Reynolds number= 8.3000 million
mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0444 and 0.0975, interpolated Cl= 0.0460
mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0095 and 0.0083, interpolated Cd= 0.0095
mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
Mach of 0.6568 yields Prandtl-Glauert Cl multiplier of 1.3261 to take Cl to 0.0610
Mach of 0.6568 yields Prandtl-Glauert Cm multiplier of 1.3261 to take Cm to -0.0928
Mach of 0.6568 with a Cl-based drag divergence Mach number of 0.7439 takes the Cd to 0.0095

(increase of 0.0000)

final Cl 0.0610
final Cd 0.0095
final Cm -0.0928

finite-wing lift multiplier of 0.9431 takes cl to 0.0576
e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.9685, gives Cdi of 0.0001
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0001, which takes total Cd to 0.0096
cl multiplied by 1.0000 to get to 0.0576 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0576 * 0.0122 * \text{sqr}(212.5) * 0.7985 / 2 = 12.7 \text{ N}$
drag= $0.0096 * 0.0122 * \text{sqr}(212.5) * 0.7985 / 2 = 2.1 \text{ N}$
momt= $-0.0928 * 0.0122 * \text{sqr}(212.5) * 0.7985 / 2 * 0.12 = -2.5 \text{ N-m}$

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
airflow path: 12.7 N
with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
airflow path: 2.1 N

lift-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
lift-only force in disc coords lateral up: 3.4 (positive pulling airplane up or helicopter aft)
lift-only force in disc coords axial aft: -12.2 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: 2.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 0.0 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: 5.5 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -11.7 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.99 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 5.4 N*m, running drag torque on the shaft= 252.4 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -11.7 N,
running thrust normal to the disc in disc coordinate system= 754.6 N

totl force in aircraft coords lateral right: 0.0 (positive pulling the aircraft right)
totl force in aircraft coords lateral up: 5.5 (positive pulling the aircraft up)
totl force in aircraft coords axial aft: -11.7 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= 0.99, L PRP on COG acf coord= 0.0)
considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 23.5)
considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -11.7 (Z acf= -2.76, N PRP on COG acf coord= -72.1)

.....
prop 1, ELEMENT INDEX 10 OF 10, BLADE AT 180 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 1.4 m/s BEHIND the disc
 propwash speed increment at ring: 0.7 m/s AT the disc after mult by 0.4637, and turb mult
 of 0.0312 is added on this as well
 total incoming airspeed: 59.0 m/s (geometric inflow + speed increment above +
 flapping)
 speed due to rotation and translation of disc: 210.3 m/s (from omega of 209.31 rad/sec, rad arm of
 0.99, swept arm of 0.06, and sweep of 14.6)
 total element speed: 211.4 m/s
 corkscrew path from disk: 15.7 deg (arctan(59.0/ 210.3))
 element pitch: 11.6 deg (prop= 11.6 + incidence= 0.0 + cyclic)
 blade element geometric angle of attack: -4.1 deg

For Reynolds number: rho= 0.7985, chord= 25.33, dynamic_viscosity =16.5794 millionths (divide by
 this to get Re min Meg), Re_meg= 1.2201 million at this element

element center at 95% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available
 mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available
 For lookup, alpha airflow= -4.1, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8853, slat alpha
 divisor= 1.0000, alpha lookup= -4.1
 For lookup, alpha airflow= -4.1, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8853, slat alpha
 divisor= 1.0000, alpha lookup= -4.2
 For lo RE using Reynolds number= 1.0000 million
 For hi RE using Reynolds number= 8.3000 million
 mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0411 and 0.0943, interpolated Cl= 0.0427
 mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0096 and 0.0083, interpolated Cd= 0.0095
 mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700
 Mach of 0.6533 yields Prandtl-Glauert Cl multiplier of 1.3209 to take Cl to 0.0565
 Mach of 0.6533 yields Prandtl-Glauert Cm multiplier of 1.3209 to take Cm to -0.0925
 Mach of 0.6533 with a Cl-based drag divergence Mach number of 0.7444 takes the Cd to 0.0095
 (increase of 0.0000)

final Cl 0.0565
 final Cd 0.0095
 final Cm -0.0925

finite-wing lift multiplier of 0.9426 takes cl to 0.0532
 e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 9.8847, gives Cdi of 0.0001
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0001, which takes total Cd to 0.0096
 cl multiplied by 1.0000 to get to 0.0532 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0532 * 0.0122 * sqrt(211.4) * 0.7985 / 2 = 11.6 N
 drag= 0.0096 * 0.0122 * sqrt(211.4) * 0.7985 / 2 = 2.1 N
 momt=-0.0925 * 0.0122 * sqrt(211.4) * 0.7985 / 2 * 0.12= -2.4 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element
 airflow path: 11.6 N
 with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element
 airflow path: 2.1 N

lift-only force in disc coords lateral right: 3.1 (positive pulling airplane right or helicopter right)
 lift-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
 lift-only force in disc coords axial aft: -11.2 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: 2.0 (positive pulling airplane right or helicopter right)
drag-only force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: 5.2 (positive pulling airplane right or helicopter right)
total force in disc coords lateral up: -0.0 (positive pulling airplane up or helicopter aft)
total force in disc coords axial aft: -10.6 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.99 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down forces at 3 and 9 o'clock)= 5.1 N*m, running drag torque on the shaft= 257.5 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -10.6 N,
running thrust normal to the disc in disc coordinate system= 765.3 N

total force in aircraft coords lateral right: 5.2 (positive pulling the aircraft right)
total force in aircraft coords lateral up: -0.0 (positive pulling the aircraft up)
total force in aircraft coords axial aft: -10.6 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.00, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= -0.96, M PRP on COG acf coord= 33.7)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -10.6 (Z acf= -2.76, N PRP on COG acf coord= -72.1)

.....
prop 1, ELEMENT INDEX 10 OF 10, BLADE AT 270 DEGREES AS SEEN FROM AFT LOOKING FORWARDS
.....

So final speed is vx= 3.3 vy= 4.5 vz= 58.4 m/s in the aircraft axis coordinate system

speed into the ring, without our propwash: 58.4 m/s (includes propwash from any OTHER discs, but not this one)

propwash speed increment behind ring: 0.2 m/s BEHIND the disc

propwash speed increment at ring: 0.1 m/s AT the disc after mult by 0.4732, and turb mult of 0.0312 is added on this as well

total incoming airspeed: 58.4 m/s (geometric inflow + speed increment above + flapping)

speed due to rotation and translation of disc: 202.6 m/s (from omega of 209.31 rad/sec, rad arm of 0.99, swept arm of 0.06, and sweep of 14.6)

total element speed: 204.0 m/s

corkscrew path from disk: 16.1 deg (arctan(58.4/ 202.6))

element pitch: 11.6 deg (prop= 11.6 + incidence= 0.0 + cyclic)

blade element geometric angle of attack: -4.5 deg

For Reynolds number: rho= 0.7985, chord= 24.45, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.1776 million at this element

element center at 95% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.1000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 0.5000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 1.0000 million available

mid foil: Airfoils/Clark-Y (good propeller).afl, I have Reynolds number 8.3000 million available

For lookup, alpha airflow= -4.5, alpha zero-lift= -4.6, finite wing alpha multiplier= 0.8884, slat alpha divisor= 1.0000, alpha lookup= -4.5

For lookup, alpha airflow= -4.5, alpha zero-lift= -5.3, finite wing alpha multiplier= 0.8884, slat alpha

divisor= 1.0000, alpha lookup= -4.6

For lo RE using Reynolds number= 1.0000 million

For hi RE using Reynolds number= 8.3000 million

mid foil: Cl lookups for lo and hi Reynolds numbers: 0.0079 and 0.0612, interpolated Cl= 0.0092

mid foil: Cd lookups for lo and hi Reynolds numbers: 0.0101 and 0.0087, interpolated Cd= 0.0101

mid foil: Cm lookups for lo and hi Reynolds numbers: -0.0700 and -0.0700, interpolated Cm=-0.0700

Mach of 0.6305 yields Prandtl-Glauert Cl multiplier of 1.2884 to take Cl to 0.0118

Mach of 0.6305 yields Prandtl-Glauert Cm multiplier of 1.2884 to take Cm to -0.0902

Mach of 0.6305 with a Cl-based drag divergence Mach number of 0.7488 takes the Cd to 0.0101

(increase of 0.0000)

final Cl 0.0118

final Cd 0.0101

final Cm -0.0902

finite-wing lift multiplier of 0.9442 takes cl to 0.0111

e of 0.9302, geometric aspect ratio of 11.4494, effective aspect ratio of 10.2020, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0101

cl multiplied by 1.0000 to get to 0.0111 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0111 * 0.0122 * sqrt(204.0) * 0.7985 / 2 = 2.3 N

drag= 0.0101 * 0.0122 * sqrt(204.0) * 0.7985 / 2 = 2.1 N

momt=-0.0902 * 0.0122 * sqrt(204.0) * 0.7985 / 2 * 0.12= -2.2 N-m

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, lift in element

airflow path: 2.3 N

with blade solidity of 1.0000 due to there being 4 blades but 4 positions calculated, drag in element

airflow path: 2.1 N

lift-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)

lift-only force in disc coords lateral up: -0.6 (positive pulling airplane up or helicopter aft)

lift-only force in disc coords axial aft: -2.2 (positive pulling airplane aft or helicopter down)

drag-only force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)

drag-only force in disc coords lateral up: -2.0 (positive pulling airplane up or helicopter aft)

drag-only force in disc coords axial aft: 0.6 (positive pulling airplane aft or helicopter down)

total force in disc coords lateral right: -0.0 (positive pulling airplane right or helicopter right)

total force in disc coords lateral up: -2.6 (positive pulling airplane up or helicopter aft)

total force in disc coords axial aft: -1.6 (positive pulling airplane aft or helicopter down)

radial arm of this element= 0.99 m

drag torque on the shaft, from the total disc lateral forces above (disc lateral left/right forces at 12 and 6 o'clock, disc lateral up/down foreces at 3 and 9 o'clock)= 2.6 N*m, running drag torque on the shaft= 260.0 N*m

thrust normal to the disc in disc coordinate system, from the total disc axial force above= -1.6 N,

running thrust normal to the disc in disc coordinate system= 766.9 N

totl force in aircraft coords lateral right: -0.0 (positive pulling the aircraft right)

totl force in aircraft coords lateral up: -2.6 (positive pulling the aircraft up)

totl force in aircraft coords axial aft: -1.6 (positive pulling the aircraft aft)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral right: 0.0 (X acf= -0.99, L PRP on COG acf coord= 0.0)

considering lift only for flapping or shaft-bending, force in aircraft coords lateral up: 0.0 (Y acf= 0.03, M PRP on COG acf coord= 33.6)

considering lift only for flapping or shaft-bending, force in aircraft coords axial aft: -1.6 (Z acf=

-2.76, N PRP on COG acf coord= -70.5)

L_PRP_on_COG_acf_coord= 0.0 L_prop_gyro= 0.0
M_PRP_on_COG_acf_coord= 33.6 M_prop_gyro= 12.5
N_PRP_on_COG_acf_coord= -70.5 N_prop_gyro= -11.1

---RUNNING PROPULSIVE FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

After propeller:

f_axil= -766.9 N, pos aft, L= 0.0 N-m, pos right
f_side= 57.2 N, pos right, M= 46.1 N-m, pos up
f_nrml= 73.7 N, pos up, N= -81.6 N-m, pos right

.....
...EXHAUST FOR ENGINE # 1.....
.....

engine power= 54745.2 Watt
thrust = 32.6 N
fside = 0.0 N
fnormal = 0.0 N
faxial = -32.6 N

---RUNNING PROPULSIVE FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

After exhaust:

f_axil= -799.5 N, pos aft, L= 0.0 N-m, pos right
f_side= 57.2 N, pos right, M= 45.1 N-m, pos up
f_nrml= 73.7 N, pos up, N= -81.6 N-m, pos right

transmission: locked clutch, if any: transmission=0, engine=0, system_rsc= 209.3, system_trq= 261.5,
system_mi= 7.225
transmission: locked clutch, if any: transmission=0, prop=0, system_rsc= 209.3, system_trq= 1.5,
system_mi= 10.987
transmission: init system_rsc= 209.3, system_trq= 1.5, system_mi= 10.987, xtim.prd_sim= 0.0088
transmission: done system_rsc= 209.3, system_trq= 1.5, system_mi= 10.987

.....
...ENGINE # 1.....
.....

thro_use = 0.2692 ratio
act_omega_rad_engn()= 209.3 rad/sec
engn_driv_TRQ = 260.6 N-m
TRQ_nm = 261.5 N-m
power = 54745.2 W

---RUNNING PROPULSIVE FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

After this engine:

f_axil= -799.5 N, pos aft, L= -260.6 N-m, pos right

f_side= 57.2 N, pos right, M= 45.1 N-m, pos up
f_nrml= 73.7 N, pos up, N= -81.6 N-m, pos right

---Left Wing 1 ELEMENT # 0-----

element x= -0.68 m, pos right from CG
element y= -0.32 m, pos up from CG
element z= 0.05 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.77 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 6.9011:
dz_in_wash= 2.81 m, radii off in z-direction= 2.7014 m, propwash_speed_mult of speed AT the
disc in z-direction= 1.3753

wash AT the ring= 2.5, total rat= 1.3753, so total vz_wash_disc= 3.5

wash vx_disc= -0.6, vy_disc= 1.2, vz_disc= 3.5 m/s (disc coords)
wash vx_acft= -0.6, vy_acft= 1.2, vz_acft= 3.5 m/s (aircraft coords)
So total speed is vx= 2.6, vy= 5.7, vz= 61.8 m/s in (aircraft coords)

So final speed is vx= 2.6 vy= 5.7 vz= 61.8 m/s in the aircraft axis coordinate system

alpha of the wing plane : 5.4 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 1.2 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 6.6 (deg)
v for forces : 60.8

For Reynolds number: rho= 0.7985, chord= 88.73, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 4.2738 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 6.6, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha
divisor= 1.0000, alpha lookup= 5.7

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.6951

mid foil: Cd lookup: 0.0096

mid foil: Cm lookup: -0.0120

Mach of 0.1878 yields Prandtl-Glauert Cl multiplier of 1.0181 to take Cl to 0.7077

Mach of 0.1878 yields Prandtl-Glauert Cm multiplier of 1.0181 to take Cm to -0.0122

Mach of 0.1878 with a Cl-based drag divergence Mach number of 0.7292 takes the Cd to 0.0096
(increase of 0.0000)

final Cl 0.7077

final Cd 0.0096

final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.6639

delta-wing moment addition of -0.0048 takes cm to -0.0170

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0167

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0167, which takes total Cd to 0.0263

cl multiplied by 1.0000 to get to 0.6639 to lower pressure from Mach 0.9 to 1.3.

$\text{lift} = 0.6639 * 0.3992 * \text{sqr}(60.8) * 0.7985 / 2 = 391.0 \text{ N}$
 $\text{drag} = 0.0263 * 0.3992 * \text{sqr}(60.8) * 0.7985 / 2 = 15.5 \text{ N}$
 $\text{momt} = -0.0170 * 0.3992 * \text{sqr}(60.8) * 0.7985 / 2 * 1.46 = -14.6 \text{ N-m}$

Left Wing 1:

local wind x= 2.6 m/s in aircraft coord system pos right.
 local wind y= 5.7 m/s in aircraft coord system pos up.
 local wind z= 61.8 m/s in aircraft coord system pos aft.

lift local wind path= 389.9 N pos up.
 drag local wind path= 15.5 N pos aft.
 side local wind path= 29.5 N pos right.

lift aircraft x y z= 0.0 388.2 -35.6 N
 drag aircraft x y z= 0.7 1.4 15.4 N
 side aircraft x y z= 29.5 -0.1 -1.2 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 1

$f_{\text{axil}} = -21.4 \text{ N}$, pos aft, $L = 257.2 \text{ N-m}$, pos right
 $f_{\text{side}} = 30.1 \text{ N}$, pos right, $M = -26.9 \text{ N-m}$, pos up
 $f_{\text{nrml}} = 389.5 \text{ N}$, pos up, $N = 12.1 \text{ N-m}$, pos right

 ---Left Wing 1 ELEMENT # 1-----

element x= -0.96 m, pos right from CG
 element y= -0.30 m, pos up from CG
 element z= 0.12 m, pos aft from CG

Propwash from prop 1:

radial offset= 1.01 m, prop radius= 1.04 m, radial wash_rat= 0.5510, float ring # 9.0000:
 dz_in_wash= 2.88 m, radii off in z-direction= 2.7671 m, propwash_speed_mult of speed AT the
 disc in z-direction= 1.3628
 wash AT the ring= 0.4, total rat= 0.7510, so total vz_wash_disc= 0.3

wash vx_disc= -0.1, vy_disc= 0.4, vz_disc= 0.3 m/s (disc coords)
 wash vx_acft= -0.1, vy_acft= 0.4, vz_acft= 0.3 m/s (aircraft coords)
 So total speed is vx= 3.1, vy= 4.8, vz= 58.7 m/s in (aircraft coords)

So final speed is vx= 3.1 vy= 4.8 vz= 58.7 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.9 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 1.1 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : 6.0 (deg)
 v for forces : 57.8

For Reynolds number: rho= 0.7985, chord= 78.67, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 3.7892 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available
 For lookup, alpha airflow= 6.0, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha
 divisor= 1.0000, alpha lookup= 5.2
 Using Reynolds number= 3.0000 million
 mid foil: Cl lookup: 0.6416
 mid foil: Cd lookup: 0.0093
 mid foil: Cm lookup: -0.0120
 Mach of 0.1787 yields Prandtl-Glauert Cl multiplier of 1.0164 to take Cl to 0.6521
 Mach of 0.1787 yields Prandtl-Glauert Cm multiplier of 1.0164 to take Cm to -0.0122
 Mach of 0.1787 with a Cl-based drag divergence Mach number of 0.7348 takes the Cd to 0.0093
 (increase of 0.0000)

final Cl 0.6521
 final Cd 0.0093
 final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.6117
 delta-wing moment addition of -0.0044 takes cm to -0.0166
 e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0142
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0142, which takes total Cd to 0.0235
 cl multiplied by 1.0000 to get to 0.6117 to lower pressure from Mach 0.9 to 1.3.

lift= $0.6117 * 0.3721 * \sqrt{57.8} * 0.7985 / 2 = 303.8 \text{ N}$
 drag= $0.0235 * 0.3721 * \sqrt{57.8} * 0.7985 / 2 = 11.7 \text{ N}$
 momt= $-0.0166 * 0.3721 * \sqrt{57.8} * 0.7985 / 2 * 1.36 = -11.2 \text{ N-m}$

Left Wing 1:

local wind x= 3.1 m/s in aircraft coord system pos right.
 local wind y= 4.8 m/s in aircraft coord system pos up.
 local wind z= 58.7 m/s in aircraft coord system pos aft.

lift local wind path= 302.9 N pos up.
 drag local wind path= 11.7 N pos aft.
 side local wind path= 22.5 N pos right.

lift aircraft x y z= 0.0 301.9 -24.9 N
 drag aircraft x y z= 0.6 1.0 11.6 N
 side aircraft x y z= 22.5 -0.1 -1.2 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 1

f_axil= -35.9 N, pos aft, L= 540.3 N-m, pos right
 f_side= 53.2 N, pos right, M= -69.4 N-m, pos up
 f_nrml= 692.3 N, pos up, N= 22.4 N-m, pos right

 ---Right Wing 1 ELEMENT # 0-----

element x= 0.68 m, pos right from CG
 element y= -0.32 m, pos up from CG
 element z= 0.05 m, pos aft from CG

Propwash from prop 1:
radial offset= 0.77 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 6.9011:
dz_in_wash= 2.81 m, radii off in z-direction= 2.7014 m, propwash_speed_mult of speed AT the
disc in z-direction= 1.3753
wash AT the ring= 2.5, total rat= 1.3753, so total vz_wash_disc= 3.5

wash vx_disc= -0.6, vy_disc= -1.2, vz_disc= 3.5 m/s (disc coords)
wash vx_acft= -0.6, vy_acft= -1.2, vz_acft= 3.5 m/s (aircraft coords)
So total speed is vx= 2.6, vy= 3.2, vz= 61.8 m/s in (aircraft coords)

So final speed is vx= 2.6 vy= 3.2 vz= 61.8 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.7 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 1.2 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 3.9 (deg)
v for forces : 59.4

FLAPS #1: crat= 0.1900, def= 13.0 deg: (pos down or extended, neg up)

dx_te = 0.9951
dy_te = 0.0427
al_inc= 1.9728
cl inc= 0.4920
cd inc= 0.0111
cm inc=-0.1425

Effective alpha increased 2.0 deg by all combined lift-incrementation devices up to final geo alpha of 5.9 deg

For Reynolds number: rho= 0.7985, chord= 86.68, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 4.1750 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 5.9, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 5.3

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.6555
mid foil: Cd lookup: 0.0094
mid foil: Cm lookup: -0.0120

Mach of 0.1835 yields Prandtl-Glauert Cl multiplier of 1.0173 to take Cl to 0.6669

Mach of 0.1835 yields Prandtl-Glauert Cm multiplier of 1.0173 to take Cm to -0.0122

Mach of 0.1835 with a Cl-based drag divergence Mach number of 0.7333 takes the Cd to 0.0094 (increase of 0.0000)

final Cl 0.6669
final Cd 0.0094
final Cm -0.0122

adding control-effects cl of 0.4920 takes cl to 1.1588

adding control-effects cd of 0.0111 takes cd to 0.0205

adding control-effects cm of -0.1425 takes cm to -0.1547

finite-wing lift multiplier of 0.9543 takes cl to 1.1059

delta-wing moment addition of -0.0079 takes cm to -0.1626

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0327

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0327, which takes total Cd to 0.0532

cl multiplied by 1.0000 to get to 1.1059 to lower pressure from Mach 0.9 to 1.3.

lift= 1.1059 * 0.3992 * sqrt(59.4) * 0.7985 / 2 = 621.5 N
drag= 0.0532 * 0.3992 * sqrt(59.4) * 0.7985 / 2 = 29.9 N
momt=-0.1626 * 0.3992 * sqrt(59.4) * 0.7985 / 2 * 1.46= -133.4 N-m

Right Wing 1:

local wind x= 2.6 m/s in aircraft coord system pos right.
local wind y= 3.2 m/s in aircraft coord system pos up.
local wind z= 61.8 m/s in aircraft coord system pos aft.

lift local wind path= 619.7 N pos up.
drag local wind path= 29.9 N pos aft.
side local wind path= -46.9 N pos right.

lift aircraft x y z= 0.0 618.9 -31.7 N
drag aircraft x y z= 1.3 1.5 29.8 N
side aircraft x y z= -46.8 0.1 2.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 1

f_axil= -35.8 N, pos aft, L= 129.7 N-m, pos right
f_side= 7.7 N, pos right, M= -233.1 N-m, pos up
f_nrml= 1312.8 N, pos up, N= 34.7 N-m, pos right

---Right Wing 1 ELEMENT # 1-----

element x= 0.96 m, pos right from CG
element y= -0.30 m, pos up from CG
element z= 0.12 m, pos aft from CG

Propwash from prop 1:

radial offset= 1.01 m, prop radius= 1.04 m, radial wash_rat= 0.5510, float ring # 9.0000:
dz_in_wash= 2.88 m, radii off in z-direction= 2.7671 m, propwash_speed_mult of speed AT the
disc in z-direction= 1.3628
wash AT the ring= 0.4, total rat= 0.7510, so total vz_wash_disc= 0.3

wash vx_disc= -0.1, vy_disc= -0.4, vz_disc= 0.3 m/s (disc coords)
wash vx_acft= -0.1, vy_acft= -0.4, vz_acft= 0.3 m/s (aircraft coords)
So total speed is vx= 3.1, vy= 4.0, vz= 58.7 m/s in (aircraft coords)

So final speed is vx= 3.1 vy= 4.0 vz= 58.7 m/s in the aircraft axis coordinate system

alpha of the wing plane : 3.7 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 1.1 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.8 (deg)
v for forces : 56.3

FLAPS #1: crat= 0.2039, def= 13.0 deg: (pos down or extended, neg up)
dx_te = 0.9948

dy_te = 0.0459
al_inc= 2.1178
cl_inc= 0.4898
cd_inc= 0.0110
cm_inc=-0.1419

Effective alpha increased 2.1 deg by all combined lift-incrementation devices up to final geo alpha of 6.9 deg

For Reynolds number: rho= 0.7985, chord= 76.56, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 3.6875 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 6.9, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 6.2

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7474

mid foil: Cd lookup: 0.0100

mid foil: Cm lookup: -0.0120

Mach of 0.1739 yields Prandtl-Glauert Cl multiplier of 1.0155 to take Cl to 0.7589

Mach of 0.1739 yields Prandtl-Glauert Cm multiplier of 1.0155 to take Cm to -0.0122

Mach of 0.1739 with a Cl-based drag divergence Mach number of 0.7241 takes the Cd to 0.0100 (increase of 0.0000)

final Cl 0.7589

final Cd 0.0100

final Cm -0.0122

adding control-effects cl of 0.4898 takes cl to 1.2487

adding control-effects cd of 0.0110 takes cd to 0.0210

adding control-effects cm of -0.1419 takes cm to -0.1541

finite-wing lift multiplier of 0.9543 takes cl to 1.1917

delta-wing moment addition of -0.0085 takes cm to -0.1626

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0380

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0380, which takes total Cd to 0.0590

cl multiplied by 1.0000 to get to 1.1917 to lower pressure from Mach 0.9 to 1.3.

lift= 1.1917 * 0.3721 * sqrt(56.3) * 0.7985 / 2 = 560.4 N

drag= 0.0590 * 0.3721 * sqrt(56.3) * 0.7985 / 2 = 27.7 N

momt=-0.1626 * 0.3721 * sqrt(56.3) * 0.7985 / 2 * 1.36= -104.1 N-m

Right Wing 1:

local wind x= 3.1 m/s in aircraft coord system pos right.

local wind y= 4.0 m/s in aircraft coord system pos up.

local wind z= 58.7 m/s in aircraft coord system pos aft.

lift local wind path= 558.9 N pos up.

drag local wind path= 27.7 N pos aft.

side local wind path= -41.6 N pos right.

lift aircraft x y z= 0.0 557.6 -38.0 N

drag aircraft x y z= 1.5 1.9 27.6 N

side aircraft x y z= -41.5 0.1 2.2 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 1

f_axil= -44.0 N, pos aft, L= -394.3 N-m, pos right
f_side= -32.4 N, pos right, M= -400.2 N-m, pos up
f_nrml= 1872.5 N, pos up, N= 39.3 N-m, pos right

---Left Wing 2 ELEMENT # 0-----

element x= -1.32 m, pos right from CG
element y= -0.27 m, pos up from CG
element z= 0.16 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 1.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 5.6 (deg)
v for forces : 58.6

For Reynolds number: rho= 0.7985, chord= 75.75, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 3.6483 million at this element

element center at 8% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 5.6, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 4.7

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.5985

mid foil: Cd lookup: 0.0091

mid foil: Cm lookup: -0.0120

Mach of 0.1811 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.6086

Mach of 0.1811 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122

Mach of 0.1811 with a Cl-based drag divergence Mach number of 0.7391 takes the Cd to 0.0091 (increase of 0.0000)

final Cl 0.6086

final Cd 0.0091

final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.5709

delta-wing moment addition of -0.0041 takes cm to -0.0163

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0123

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0123, which takes total Cd to 0.0214

cl multiplied by 1.0000 to get to 0.5709 to lower pressure from Mach 0.9 to 1.3.

lift= 0.5709 * 0.5777 * sqrt(58.6) * 0.7985 / 2 = 452.0 N

drag= 0.0214 * 0.5777 * sqrt(58.6) * 0.7985 / 2 = 17.0 N

momt=-0.0163 * 0.5777 * sqrt(58.6) * 0.7985 / 2 * 1.29= -16.7 N-m

Left Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 450.8 N pos up.
drag local wind path= 17.0 N pos aft.
side local wind path= 32.8 N pos right.

lift aircraft x y z= 0.0 449.5 -34.1 N
drag aircraft x y z= 0.9 1.3 16.9 N
side aircraft x y z= 32.8 -0.1 -1.8 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 2

f_axil= -63.0 N, pos aft, L= 190.0 N-m, pos right
f_side= 1.3 N, pos right, M= -483.1 N-m, pos up
f_nrml= 2323.1 N, pos up, N= 57.8 N-m, pos right

---Left Wing 2 ELEMENT # 1-----

element x= -1.76 m, pos right from CG
element y= -0.24 m, pos up from CG
element z= 0.17 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.9 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 5.5 (deg)
v for forces : 58.6

For Reynolds number: rho= 0.7985, chord= 73.65, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 3.5476 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 5.5, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 4.7

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.5890

mid foil: Cd lookup: 0.0090

mid foil: Cm lookup: -0.0120

Mach of 0.1810 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.5989

Mach of 0.1810 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122

Mach of 0.1810 with a Cl-based drag divergence Mach number of 0.7401 takes the Cd to 0.0090 (increase of 0.0000)

final Cl 0.5989

final Cd 0.0090

final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes c_l to 0.5618
delta-wing moment addition of -0.0040 takes c_m to -0.0162
e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives C_{di} of 0.0119
ground effect c_{di} multiplier of 1.0000 takes C_{di} to 0.0119, which takes total C_d to 0.0210
 c_l multiplied by 1.0000 to get to 0.5618 to lower pressure from Mach 0.9 to 1.3.

lift= $0.5618 * 0.5618 * \text{sqr}(58.6) * 0.7985 / 2 = 432.4 \text{ N}$
drag= $0.0210 * 0.5618 * \text{sqr}(58.6) * 0.7985 / 2 = 16.1 \text{ N}$
momt= $-0.0162 * 0.5618 * \text{sqr}(58.6) * 0.7985 / 2 * 1.26 = -15.7 \text{ N-m}$

Left Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 431.4 N pos up.
drag local wind path= 16.1 N pos aft.
side local wind path= 30.5 N pos right.

lift aircraft x y z= 0.0 430.1 -32.6 N
drag aircraft x y z= 0.9 1.2 16.1 N
side aircraft x y z= 30.5 -0.1 -1.7 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 2

f_{axil} = -81.2 N, pos aft, L = 942.4 N-m, pos right
 f_{side} = 32.7 N, pos right, M = -568.2 N-m, pos up
 f_{nrml} = 2754.3 N, pos up, N = 83.5 N-m, pos right

---Left Wing 2 ELEMENT # 2-----

element x= -2.21 m, pos right from CG
element y= -0.21 m, pos up from CG
element z= 0.18 m, pos aft from CG

So final speed is v_x = 3.3 v_y = 4.4 v_z = 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.8 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 5.3 (deg)
v for forces : 58.6

For Reynolds number: ρ = 0.7985, chord= 71.56, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_{meg} = 3.4468 million at this element

element center at 42% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.af, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 5.3, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha

divisor= 1.0000, alpha lookup= 4.6
 Using Reynolds number= 3.0000 million
 mid foil: Cl lookup: 0.5795
 mid foil: Cd lookup: 0.0090
 mid foil: Cm lookup: -0.0120
 Mach of 0.1810 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.5892
 Mach of 0.1810 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122
 Mach of 0.1810 with a Cl-based drag divergence Mach number of 0.7411 takes the Cd to 0.0090
 (increase of 0.0000)

final Cl 0.5892
 final Cd 0.0090
 final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.5527
 delta-wing moment addition of -0.0040 takes cm to -0.0162
 e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0116
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0116, which takes total Cd to 0.0206
 cl multiplied by 1.0000 to get to 0.5527 to lower pressure from Mach 0.9 to 1.3.

lift= $0.5527 * 0.5460 * \text{sqr}(58.6) * 0.7985 / 2 = 413.3 \text{ N}$
 drag= $0.0206 * 0.5460 * \text{sqr}(58.6) * 0.7985 / 2 = 15.4 \text{ N}$
 momt= $-0.0162 * 0.5460 * \text{sqr}(58.6) * 0.7985 / 2 * 1.22 = -14.8 \text{ N-m}$

Left Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.
 local wind y= 4.4 m/s in aircraft coord system pos up.
 local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 412.3 N pos up.
 drag local wind path= 15.4 N pos aft.
 side local wind path= 28.4 N pos right.

lift aircraft x y z= 0.0 411.2 -31.2 N
 drag aircraft x y z= 0.9 1.2 15.3 N
 side aircraft x y z= 28.3 -0.1 -1.6 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 2

f_axil= -98.7 N, pos aft, L= 1846.4 N-m, pos right
 f_side= 61.9 N, pos right, M= -654.9 N-m, pos up
 f_nrml= 3166.5 N, pos up, N= 115.7 N-m, pos right

 ---Left Wing 2 ELEMENT # 3-----

element x= -2.65 m, pos right from CG
 element y= -0.18 m, pos up from CG
 element z= 0.20 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-

incidence plane)

element incidence : 0.7 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 5.2 (deg)
v for forces : 58.6

For Reynolds number: rho= 0.7985, chord= 69.47, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 3.3462 million at this element

element center at 58% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 5.2, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 4.5

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.5699

mid foil: Cd lookup: 0.0089

mid foil: Cm lookup: -0.0120

Mach of 0.1810 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.5795

Mach of 0.1810 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122

Mach of 0.1810 with a Cl-based drag divergence Mach number of 0.7420 takes the Cd to 0.0089 (increase of 0.0000)

final Cl 0.5795

final Cd 0.0089

final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.5436

delta-wing moment addition of -0.0039 takes cm to -0.0161

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0112

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0112, which takes total Cd to 0.0201

cl multiplied by 1.0000 to get to 0.5436 to lower pressure from Mach 0.9 to 1.3.

lift= 0.5436 * 0.5301 * sqrt(58.6) * 0.7985 / 2 = 394.6 N

drag= 0.0201 * 0.5301 * sqrt(58.6) * 0.7985 / 2 = 14.6 N

momt=-0.0161 * 0.5301 * sqrt(58.6) * 0.7985 / 2 * 1.19= -13.9 N-m

Left Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 393.7 N pos up.

drag local wind path= 14.6 N pos aft.

side local wind path= 26.3 N pos right.

lift aircraft x y z= 0.0 392.6 -29.8 N

drag aircraft x y z= 0.8 1.1 14.5 N

side aircraft x y z= 26.3 -0.1 -1.5 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 2

f_axil= -115.4 N, pos aft, L= 2885.9 N-m, pos right

f_side= 89.0 N, pos right, M= -742.9 N-m, pos up

f_nrml= 3560.1 N, pos up, N= 153.8 N-m, pos right

---Left Wing 2 ELEMENT # 4-----

element x= -3.10 m, pos right from CG

element y= -0.15 m, pos up from CG

element z= 0.21 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.6 (deg)

incidence increase : 0.0 (deg)

geometric alpha : 5.1 (deg)

v for forces : 58.6

For Reynolds number: rho= 0.7985, chord= 67.38, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 3.2455 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 5.1, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 4.4

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.5604

mid foil: Cd lookup: 0.0089

mid foil: Cm lookup: -0.0120

Mach of 0.1810 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.5698

Mach of 0.1810 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122

Mach of 0.1810 with a Cl-based drag divergence Mach number of 0.7430 takes the Cd to 0.0089 (increase of 0.0000)

final Cl 0.5698

final Cd 0.0089

final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.5345

delta-wing moment addition of -0.0038 takes cm to -0.0160

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0108

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0108, which takes total Cd to 0.0197

cl multiplied by 1.0000 to get to 0.5345 to lower pressure from Mach 0.9 to 1.3.

lift= 0.5345 * 0.5142 * sqrt(58.6) * 0.7985 / 2 = 376.3 N

drag= 0.0197 * 0.5142 * sqrt(58.6) * 0.7985 / 2 = 13.9 N

momt=-0.0160 * 0.5142 * sqrt(58.6) * 0.7985 / 2 * 1.15= -13.0 N-m

Left Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 375.5 N pos up.

drag local wind path= 13.9 N pos aft.

side local wind path= 24.3 N pos right.

lift aircraft x y z= 0.0 374.4 -28.4 N
drag aircraft x y z= 0.8 1.0 13.8 N
side aircraft x y z= 24.3 -0.1 -1.3 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 2

f_axil= -131.4 N, pos aft, L= 4045.2 N-m, pos right
f_side= 114.0 N, pos right, M= -831.8 N-m, pos up
f_nrml= 3935.5 N, pos up, N= 197.2 N-m, pos right

---Left Wing 2 ELEMENT # 5-----

element x= -3.54 m, pos right from CG
element y= -0.12 m, pos up from CG
element z= 0.22 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.5 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 5.0 (deg)
v for forces : 58.6

For Reynolds number: rho= 0.7985, chord= 65.29, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 3.1448 million at this element

element center at 92% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 5.0, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 4.3

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.5509

mid foil: Cd lookup: 0.0089

mid foil: Cm lookup: -0.0120

Mach of 0.1809 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.5601

Mach of 0.1809 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122

Mach of 0.1809 with a Cl-based drag divergence Mach number of 0.7440 takes the Cd to 0.0089 (increase of 0.0000)

final Cl 0.5601

final Cd 0.0089

final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.5254

delta-wing moment addition of -0.0038 takes cm to -0.0160

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0104

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0104, which takes total Cd to 0.0193

cl multiplied by 1.0000 to get to 0.5254 to lower pressure from Mach 0.9 to 1.3.

lift= 0.5254 * 0.4983 * sqrt(58.6) * 0.7985 / 2 = 358.3 N
drag= 0.0193 * 0.4983 * sqrt(58.6) * 0.7985 / 2 = 13.2 N
momt=-0.0160 * 0.4983 * sqrt(58.6) * 0.7985 / 2 * 1.12= -12.1 N-m

Left Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 357.6 N pos up.
drag local wind path= 13.2 N pos aft.
side local wind path= 22.5 N pos right.

lift aircraft x y z= 0.0 356.6 -27.1 N
drag aircraft x y z= 0.7 1.0 13.1 N
side aircraft x y z= 22.4 -0.1 -1.2 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 2

f_axil= -146.6 N, pos aft, L= 5309.5 N-m, pos right
f_side= 137.2 N, pos right, M= -921.1 N-m, pos up
f_nrml= 4293.0 N, pos up, N= 245.3 N-m, pos right

---Right Wing 2 ELEMENT # 0-----

element x= 1.32 m, pos right from CG
element y= -0.27 m, pos up from CG
element z= 0.16 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 1.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 5.1 (deg)
v for forces : 58.4

FLAPS #2: crat= 0.2200, def= 12.0 deg: (pos down or extended, neg up)

dx_te = 0.9952
dy_te = 0.0457
al_inc= 2.1095
cl inc= 0.5174
cd inc= 0.0118
cm inc=-0.1497

Effective alpha increased 2.1 deg by all combined lift-incrementation devices up to final geo alpha of 7.2 deg

For Reynolds number: rho= 0.7985, chord= 75.52, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 3.6374 million at this element

element center at 8% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 7.2, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 6.4

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7768

mid foil: Cd lookup: 0.0102

mid foil: Cm lookup: -0.0120

Mach of 0.1805 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.7898

Mach of 0.1805 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122

Mach of 0.1805 with a Cl-based drag divergence Mach number of 0.7210 takes the Cd to 0.0102 (increase of 0.0000)

final Cl 0.7898

final Cd 0.0102

final Cm -0.0122

adding control-effects cl of 0.5174 takes cl to 1.3072

adding control-effects cd of 0.0118 takes cd to 0.0220

adding control-effects cm of -0.1497 takes cm to -0.1619

finite-wing lift multiplier of 0.9543 takes cl to 1.2475

delta-wing moment addition of -0.0089 takes cm to -0.1708

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0416

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0416, which takes total Cd to 0.0636

cl multiplied by 1.0000 to get to 1.2475 to lower pressure from Mach 0.9 to 1.3.

lift= 1.2475 * 0.5777 * sqrt(58.4) * 0.7985 / 2 = 981.8 N

drag= 0.0636 * 0.5777 * sqrt(58.4) * 0.7985 / 2 = 50.0 N

momt=-0.1708 * 0.5777 * sqrt(58.4) * 0.7985 / 2 * 1.29= -173.8 N-m

Right Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 979.2 N pos up.

drag local wind path= 50.0 N pos aft.

side local wind path= -71.3 N pos right.

lift aircraft x y z= 0.0 976.4 -73.8 N

drag aircraft x y z= 2.8 3.8 49.8 N

side aircraft x y z= -71.2 0.3 3.9 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 2

f_axil= -166.7 N, pos aft, L= 4037.0 N-m, pos right

f_side= 68.8 N, pos right, M= -1244.5 N-m, pos up

f_nrml= 5273.4 N, pos up, N= 242.4 N-m, pos right

---Right Wing 2 ELEMENT # 1-----

element x= 1.76 m, pos right from CG
element y= -0.24 m, pos up from CG
element z= 0.17 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.9 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 5.0 (deg)
v for forces : 58.4

FLAPS #2: crat= 0.2243, def= 12.0 deg: (pos down or extended, neg up)

dx_te = 0.9951
dy_te = 0.0466
al_inc= 2.1508
cl inc= 0.5177
cd inc= 0.0118
cm inc=-0.1498

Effective alpha increased 2.2 deg by all combined lift-incrementation devices up to final geo alpha of 7.1 deg

For Reynolds number: rho= 0.7985, chord= 73.45, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 3.5378 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 7.1, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 6.4

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7717
mid foil: Cd lookup: 0.0101
mid foil: Cm lookup: -0.0120

Mach of 0.1805 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.7845

Mach of 0.1805 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122

Mach of 0.1805 with a Cl-based drag divergence Mach number of 0.7215 takes the Cd to 0.0101 (increase of 0.0000)

final Cl 0.7845
final Cd 0.0101
final Cm -0.0122

adding control-effects cl of 0.5177 takes cl to 1.3023

adding control-effects cd of 0.0118 takes cd to 0.0219

adding control-effects cm of -0.1498 takes cm to -0.1620

finite-wing lift multiplier of 0.9543 takes cl to 1.2428

delta-wing moment addition of -0.0089 takes cm to -0.1709

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0413

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0413, which takes total Cd to 0.0632

cl multiplied by 1.0000 to get to 1.2428 to lower pressure from Mach 0.9 to 1.3.

lift= 1.2428 * 0.5618 * sqr(58.4) * 0.7985 / 2 = 951.4 N

drag= 0.0632 * 0.5618 * sqr(58.4) * 0.7985 / 2 = 48.4 N

momt=-0.1709 * 0.5618 * sqr(58.4) * 0.7985 / 2 * 1.26= -164.5 N-m

Right Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 949.0 N pos up.
drag local wind path= 48.4 N pos aft.
side local wind path= -67.2 N pos right.

lift aircraft x y z= 0.0 946.4 -71.5 N
drag aircraft x y z= 2.7 3.6 48.2 N
side aircraft x y z= -67.1 0.3 3.7 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 2

f_axil= -186.2 N, pos aft, L= 2377.7 N-m, pos right
f_side= 4.4 N, pos right, M= -1566.5 N-m, pos up
f_nrml= 6223.7 N, pos up, N= 230.5 N-m, pos right

---Right Wing 2 ELEMENT # 2-----

element x= 2.21 m, pos right from CG
element y= -0.21 m, pos up from CG
element z= 0.18 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.8 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.9 (deg)
v for forces : 58.4

FLAPS #2: crat= 0.2289, def= 12.0 deg: (pos down or extended, neg up)

dx_te = 0.9950

dy_te = 0.0476

al_inc= 2.1945

cl inc= 0.5180

cd inc= 0.0118

cm inc=-0.1499

Effective alpha increased 2.2 deg by all combined lift-incrementation devices up to final geo alpha of 7.1 deg

For Reynolds number: rho= 0.7985, chord= 71.39, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 3.4383 million at this element

element center at 42% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 7.1, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha

divisor= 1.0000, alpha lookup= 6.3
 Using Reynolds number= 3.0000 million
 mid foil: Cl lookup: 0.7667
 mid foil: Cd lookup: 0.0101
 mid foil: Cm lookup: -0.0120
 Mach of 0.1806 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.7795
 Mach of 0.1806 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122
 Mach of 0.1806 with a Cl-based drag divergence Mach number of 0.7220 takes the Cd to 0.0101
 (increase of 0.0000)

final Cl 0.7795
 final Cd 0.0101
 final Cm -0.0122

adding control-effects cl of 0.5180 takes cl to 1.2976
 adding control-effects cd of 0.0118 takes cd to 0.0219
 adding control-effects cm of -0.1499 takes cm to -0.1621
 finite-wing lift multiplier of 0.9543 takes cl to 1.2383
 delta-wing moment addition of -0.0089 takes cm to -0.1710
 e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0410
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0410, which takes total Cd to 0.0629
 cl multiplied by 1.0000 to get to 1.2383 to lower pressure from Mach 0.9 to 1.3.

lift= $1.2383 * 0.5460 * \sqrt{58.4} * 0.7985 / 2 = 921.4 \text{ N}$
 drag= $0.0629 * 0.5460 * \sqrt{58.4} * 0.7985 / 2 = 46.8 \text{ N}$
 momt= $-0.1710 * 0.5460 * \sqrt{58.4} * 0.7985 / 2 * 1.22 = -155.4 \text{ N-m}$

Right Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.
 local wind y= 4.4 m/s in aircraft coord system pos up.
 local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 919.2 N pos up.
 drag local wind path= 46.8 N pos aft.
 side local wind path= -63.3 N pos right.

lift aircraft x y z= 0.0 916.6 -69.2 N
 drag aircraft x y z= 2.6 3.5 46.6 N
 side aircraft x y z= -63.2 0.3 3.5 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 2

f_axil= -205.3 N, pos aft, L= 358.3 N-m, pos right
 f_side= -56.2 N, pos right, M= -1886.5 N-m, pos up
 f_nrml= 7144.1 N, pos up, N= 210.1 N-m, pos right

 ---Right Wing 2 ELEMENT # 3-----

element x= 2.65 m, pos right from CG
 element y= -0.18 m, pos up from CG
 element z= 0.20 m, pos aft from CG

So final speed is $v_x = 3.3$ $v_y = 4.4$ $v_z = 58.4$ m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.7 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.8 (deg)
v for forces : 58.4

FLAPS #2: crat= 0.2337, def= 12.0 deg: (pos down or extended, neg up)

dx_te = 0.9949

dy_te = 0.0486

al_inc= 2.2407

cl inc= 0.5183

cd inc= 0.0118

cm inc=-0.1500

Effective alpha increased 2.2 deg by all combined lift-incrementation devices up to final geo alpha of 7.0 deg

For Reynolds number: rho= 0.7985, chord= 69.32, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 3.3387 million at this element

element center at 58% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 7.0, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 6.3

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7620

mid foil: Cd lookup: 0.0101

mid foil: Cm lookup: -0.0120

Mach of 0.1806 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.7747

Mach of 0.1806 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122

Mach of 0.1806 with a Cl-based drag divergence Mach number of 0.7225 takes the Cd to 0.0101 (increase of 0.0000)

final Cl 0.7747

final Cd 0.0101

final Cm -0.0122

adding control-effects cl of 0.5183 takes cl to 1.2931

adding control-effects cd of 0.0118 takes cd to 0.0219

adding control-effects cm of -0.1500 takes cm to -0.1622

finite-wing lift multiplier of 0.9543 takes cl to 1.2340

delta-wing moment addition of -0.0088 takes cm to -0.1710

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0407

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0407, which takes total Cd to 0.0626

cl multiplied by 1.0000 to get to 1.2340 to lower pressure from Mach 0.9 to 1.3.

lift= $1.2340 * 0.5301 * \sqrt{58.4} * 0.7985 / 2 = 891.7$ N

drag= $0.0626 * 0.5301 * \sqrt{58.4} * 0.7985 / 2 = 45.2$ N

momt= $-0.1710 * 0.5301 * \sqrt{58.4} * 0.7985 / 2 * 1.19 = -146.6$ N-m

Right Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 889.8 N pos up.
drag local wind path= 45.2 N pos aft.
side local wind path= -59.5 N pos right.

lift aircraft x y z= 0.0 887.2 -66.9 N
drag aircraft x y z= 2.5 3.4 45.0 N
side aircraft x y z= -59.4 0.2 3.3 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 2

f_axil= -224.0 N, pos aft, L= -1995.4 N-m, pos right
f_side= -113.0 N, pos right, M= -2204.1 N-m, pos up
f_nrml= 8035.0 N, pos up, N= 181.6 N-m, pos right

---Right Wing 2 ELEMENT # 4-----

element x= 3.10 m, pos right from CG
element y= -0.15 m, pos up from CG
element z= 0.21 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.6 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.7 (deg)
v for forces : 58.4

FLAPS #2: crat= 0.2388, def= 12.0 deg: (pos down or extended, neg up)

dx_te = 0.9948
dy_te = 0.0496
al_inc= 2.2898
cl inc= 0.5186
cd inc= 0.0118
cm inc=-0.1500

Effective alpha increased 2.3 deg by all combined lift-incrementation devices up to final geo alpha of 7.0 deg

For Reynolds number: rho= 0.7985, chord= 67.25, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 3.2390 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 7.0, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 6.3

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7576
mid foil: Cd lookup: 0.0100
mid foil: Cm lookup: -0.0120

Mach of 0.1806 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.7702
Mach of 0.1806 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122
Mach of 0.1806 with a Cl-based drag divergence Mach number of 0.7230 takes the Cd to 0.0100
(increase of 0.0000)

final Cl 0.7702
final Cd 0.0100
final Cm -0.0122

adding control-effects cl of 0.5186 takes cl to 1.2889
adding control-effects cd of 0.0118 takes cd to 0.0219
adding control-effects cm of -0.1500 takes cm to -0.1622
finite-wing lift multiplier of 0.9543 takes cl to 1.2300
delta-wing moment addition of -0.0088 takes cm to -0.1711
e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0404
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0404, which takes total Cd to 0.0623
cl multiplied by 1.0000 to get to 1.2300 to lower pressure from Mach 0.9 to 1.3.

lift= 1.2300 * 0.5142 * sqrt(58.4) * 0.7985 / 2 = 862.4 N
drag= 0.0623 * 0.5142 * sqrt(58.4) * 0.7985 / 2 = 43.7 N
momt=-0.1711 * 0.5142 * sqrt(58.4) * 0.7985 / 2 * 1.15= -138.0 N-m

Right Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 860.6 N pos up.
drag local wind path= 43.7 N pos aft.
side local wind path= -55.8 N pos right.

lift aircraft x y z= 0.0 858.2 -64.7 N
drag aircraft x y z= 2.4 3.3 43.5 N
side aircraft x y z= -55.7 0.2 3.1 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 2

f_axil= -242.1 N, pos aft, L= -4657.6 N-m, pos right
f_side= -166.3 N, pos right, M= -2518.9 N-m, pos up
f_nrml= 8896.7 N, pos up, N= 145.4 N-m, pos right

---Right Wing 2 ELEMENT # 5-----

element x= 3.54 m, pos right from CG
element y= -0.12 m, pos up from CG
element z= 0.22 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)
element incidence : 0.5 (deg)

incidence increase : 0.0 (deg)
geometric alpha : 4.6 (deg)
v for forces : 58.4

FLAPS #2: crat= 0.2442, def= 12.0 deg: (pos down or extended, neg up)

dx_te = 0.9947

dy_te = 0.0508

al_inc= 2.3420

cl_inc= 0.5189

cd_inc= 0.0118

cm_inc=-0.1501

Effective alpha increased 2.3 deg by all combined lift-incrementation devices up to final geo alpha of 6.9 deg

For Reynolds number: rho= 0.7985, chord= 65.18, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 3.1394 million at this element

element center at 92% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 6.9, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 6.2

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7534

mid foil: Cd lookup: 0.0100

mid foil: Cm lookup: -0.0120

Mach of 0.1806 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.7660

Mach of 0.1806 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122

Mach of 0.1806 with a Cl-based drag divergence Mach number of 0.7234 takes the Cd to 0.0100 (increase of 0.0000)

final Cl 0.7660

final Cd 0.0100

final Cm -0.0122

adding control-effects cl of 0.5189 takes cl to 1.2849

adding control-effects cd of 0.0118 takes cd to 0.0218

adding control-effects cm of -0.1501 takes cm to -0.1623

finite-wing lift multiplier of 0.9543 takes cl to 1.2263

delta-wing moment addition of -0.0088 takes cm to -0.1711

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0402

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0402, which takes total Cd to 0.0620

cl multiplied by 1.0000 to get to 1.2263 to lower pressure from Mach 0.9 to 1.3.

lift= 1.2263 * 0.4983 * sqrt(58.4) * 0.7985 / 2 = 833.5 N

drag= 0.0620 * 0.4983 * sqrt(58.4) * 0.7985 / 2 = 42.2 N

momt=-0.1711 * 0.4983 * sqrt(58.4) * 0.7985 / 2 * 1.12= -129.7 N-m

Right Wing 2:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 831.8 N pos up.

drag local wind path= 42.2 N pos aft.

side local wind path= -52.3 N pos right.

lift aircraft x y z=	0.0	829.5	-62.5 N
drag aircraft x y z=	2.3	3.2	42.0 N
side aircraft x y z=	-52.2	0.2	2.9 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

 Right Wing 2

f_axil= -259.8 N, pos aft, L= -7603.4 N-m, pos right
 f_side= -216.2 N, pos right, M= -2830.3 N-m, pos up
 f_nrml= 9729.6 N, pos up, N= 102.1 N-m, pos right

 ---Left Wing 3 ELEMENT # 0-----

element x= -3.96 m, pos right from CG
 element y= -0.10 m, pos up from CG
 element z= 0.23 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.4 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : 4.9 (deg)
 v for forces : 58.5

AILERON #1: crat= 0.2000, def= 13.0 deg: (pos down or extended, neg up)

dx_te = 0.9949
 dy_te = 0.0450
 al_inc= 2.0768
 cl inc= 0.2007
 cd inc= 0.0005
 cm inc=-0.1304

Effective alpha increased 2.1 deg by all combined lift-incrementation devices up to final geo alpha of 7.0 deg

For Reynolds number: rho= 0.7985, chord= 62.07, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.9898 million at this element

element center at 12% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 7.0, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 6.0

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7324

mid foil: Cd lookup: 0.0099

mid foil: Cm lookup: -0.0120

Mach of 0.1809 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.7447

Mach of 0.1809 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122

Mach of 0.1809 with a Cl-based drag divergence Mach number of 0.7255 takes the Cd to 0.0099 (increase of 0.0000)

final Cl 0.7447
final Cd 0.0099
final Cm -0.0122

adding control-effects cl of 0.2007 takes cl to 0.9453
adding control-effects cd of 0.0005 takes cd to 0.0104
adding control-effects cm of -0.1304 takes cm to -0.1426
finite-wing lift multiplier of 0.9381 takes cl to 0.8868
delta-wing moment addition of -0.0064 takes cm to -0.1490
e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0298
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0298, which takes total Cd to 0.0401
cl multiplied by 1.0000 to get to 0.8868 to lower pressure from Mach 0.9 to 1.3.

lift= $0.8868 * 0.4176 * \sqrt{58.5} * 0.7985 / 2 = 506.9 \text{ N}$
drag= $0.0401 * 0.4176 * \sqrt{58.5} * 0.7985 / 2 = 22.9 \text{ N}$
momt= $-0.1490 * 0.4176 * \sqrt{58.5} * 0.7985 / 2 * 1.06 = -90.3 \text{ N-m}$

Left Wing 3:

local wind x= 3.3 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 505.9 N pos up.
drag local wind path= 22.9 N pos aft.
side local wind path= 30.8 N pos right.

lift aircraft x y z= 0.0 504.5 -38.3 N
drag aircraft x y z= 1.3 1.7 22.8 N
side aircraft x y z= 30.8 -0.1 -1.7 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 3

f_axil= -277.0 N, pos aft, L= -5600.7 N-m, pos right
f_side= -184.1 N, pos right, M= -3037.4 N-m, pos up
f_nrml= 10235.6 N, pos up, N= 157.3 N-m, pos right

---Left Wing 3 ELEMENT # 1-----

element x= -4.36 m, pos right from CG
element y= -0.07 m, pos up from CG
element z= 0.25 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.3 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.8 (deg)
v for forces : 58.5

AILERON #1: crat= 0.1923, def= 13.0 deg: (pos down or extended, neg up)

dx_te = 0.9951
dy_te = 0.0433
al_inc= 1.9968
cl inc= 0.1931
cd inc= 0.0005
cm inc=-0.1262

Effective alpha increased 2.0 deg by all combined lift-incrementation devices up to final geo alpha of 6.8 deg

For Reynolds number: rho= 0.7985, chord= 57.05, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.7478 million at this element

element center at 38% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 6.8, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 5.9

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7155

mid foil: Cd lookup: 0.0098

mid foil: Cm lookup: -0.0120

Mach of 0.1809 yields Prandtl-Glauert Cl multiplier of 1.0168 to take Cl to 0.7275

Mach of 0.1809 yields Prandtl-Glauert Cm multiplier of 1.0168 to take Cm to -0.0122

Mach of 0.1809 with a Cl-based drag divergence Mach number of 0.7272 takes the Cd to 0.0098 (increase of 0.0000)

final Cl 0.7275

final Cd 0.0098

final Cm -0.0122

adding control-effects cl of 0.1931 takes cl to 0.9206

adding control-effects cd of 0.0005 takes cd to 0.0102

adding control-effects cm of -0.1262 takes cm to -0.1384

finite-wing lift multiplier of 0.9381 takes cl to 0.8635

delta-wing moment addition of -0.0062 takes cm to -0.1446

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0282

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0282, which takes total Cd to 0.0384

cl multiplied by 1.0000 to get to 0.8635 to lower pressure from Mach 0.9 to 1.3.

lift= 0.8635 * 0.3837 * sqrt(58.5) * 0.7985 / 2 = 453.4 N

drag= 0.0384 * 0.3837 * sqrt(58.5) * 0.7985 / 2 = 20.2 N

momt=-0.1446 * 0.3837 * sqrt(58.5) * 0.7985 / 2 * 0.97= -74.0 N-m

Left Wing 3:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 452.6 N pos up.

drag local wind path= 20.2 N pos aft.

side local wind path= 26.8 N pos right.

lift aircraft x y z= 0.0 451.3 -34.3 N

drag aircraft x y z= 1.1 1.5 20.1 N

side aircraft x y z= 26.7 -0.1 -1.5 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 3

f_axil= -292.7 N, pos aft, L= -3630.5 N-m, pos right
f_side= -156.2 N, pos right, M= -3225.4 N-m, pos up
f_nrml= 10688.3 N, pos up, N= 214.3 N-m, pos right

---Left Wing 3 ELEMENT # 2-----

element x= -4.75 m, pos right from CG
element y= -0.05 m, pos up from CG
element z= 0.29 m, pos aft from CG

So final speed is vx= 3.2 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.2 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.7 (deg)
v for forces : 58.4

AILERON #1: crat= 0.1878, def= 13.0 deg: (pos down or extended, neg up)

dx_te = 0.9952
dy_te = 0.0422
al_inc= 1.9499
cl inc= 0.1886
cd inc= 0.0005
cm inc=-0.1238

Effective alpha increased 1.9 deg by all combined lift-incrementation devices up to final geo alpha of 6.7 deg

For Reynolds number: rho= 0.7985, chord= 50.33, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.4244 million at this element

element center at 62% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 6.7, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 5.7

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.7016
mid foil: Cd lookup: 0.0097
mid foil: Cm lookup: -0.0120

Mach of 0.1804 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.7133

Mach of 0.1804 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122

Mach of 0.1804 with a Cl-based drag divergence Mach number of 0.7287 takes the Cd to 0.0097 (increase of 0.0000)

final Cl 0.7133
final Cd 0.0097
final Cm -0.0122

adding control-effects cl of 0.1886 takes cl to 0.9020

adding control-effects cd of 0.0005 takes cd to 0.0101
 adding control-effects cm of -0.1238 takes cm to -0.1360
 finite-wing lift multiplier of 0.9381 takes cl to 0.8461
 delta-wing moment addition of -0.0061 takes cm to -0.1420
 e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0271
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0271, which takes total Cd to 0.0372
 cl multiplied by 1.0000 to get to 0.8461 to lower pressure from Mach 0.9 to 1.3.

lift= $0.8461 * 0.3392 * \sqrt{58.4} * 0.7985 / 2 = 390.3 \text{ N}$
 drag= $0.0372 * 0.3392 * \sqrt{58.4} * 0.7985 / 2 = 17.2 \text{ N}$
 momt=-0.1420 * 0.3392 * $\sqrt{58.4} * 0.7985 / 2 * 0.86 = -56.5 \text{ N-m}$

Left Wing 3:

local wind x= 3.2 m/s in aircraft coord system pos right.
 local wind y= 4.4 m/s in aircraft coord system pos up.
 local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 389.7 N pos up.
 drag local wind path= 17.2 N pos aft.
 side local wind path= 22.4 N pos right.

lift aircraft x y z= 0.0 388.6 -29.6 N
 drag aircraft x y z= 1.0 1.3 17.1 N
 side aircraft x y z= 22.4 -0.1 -1.2 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 3

f_axil= -306.4 N, pos aft, L= -1780.5 N-m, pos right
 f_side= -132.9 N, pos right, M= -3396.1 N-m, pos up
 f_nrml= 11078.1 N, pos up, N= 269.2 N-m, pos right

---Left Wing 3 ELEMENT # 3-----

element x= -5.14 m, pos right from CG
 element y= -0.03 m, pos up from CG
 element z= 0.36 m, pos aft from CG

So final speed is vx= 3.2 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.1 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : 4.6 (deg)
 v for forces : 58.0

AILERON #1: crat= 0.1894, def= 13.0 deg: (pos down or extended, neg up)

dx_te = 0.9951
 dy_te = 0.0426
 al_inc= 1.9665
 cl inc= 0.1904
 cd inc= 0.0005

cm inc=-0.1247

Effective alpha increased 2.0 deg by all combined lift-incrementation devices up to final geo alpha of 6.6 deg

For Reynolds number: rho= 0.7985, chord= 41.85, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.0158 million at this element

element center at 88% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 6.6, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 5.7

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.6936

mid foil: Cd lookup: 0.0096

mid foil: Cm lookup: -0.0120

Mach of 0.1793 yields Prandtl-Glauert Cl multiplier of 1.0165 to take Cl to 0.7050

Mach of 0.1793 yields Prandtl-Glauert Cm multiplier of 1.0165 to take Cm to -0.0122

Mach of 0.1793 with a Cl-based drag divergence Mach number of 0.7295 takes the Cd to 0.0096 (increase of 0.0000)

final Cl 0.7050

final Cd 0.0096

final Cm -0.0122

adding control-effects cl of 0.1904 takes cl to 0.8954

adding control-effects cd of 0.0005 takes cd to 0.0101

adding control-effects cm of -0.1247 takes cm to -0.1369

finite-wing lift multiplier of 0.9381 takes cl to 0.8399

delta-wing moment addition of -0.0060 takes cm to -0.1430

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0267

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0267, which takes total Cd to 0.0368

cl multiplied by 1.0000 to get to 0.8399 to lower pressure from Mach 0.9 to 1.3.

lift= 0.8399 * 0.2831 * sqrt(58.0) * 0.7985 / 2 = 319.7 N

drag= 0.0368 * 0.2831 * sqrt(58.0) * 0.7985 / 2 = 14.0 N

momt=-0.1430 * 0.2831 * sqrt(58.0) * 0.7985 / 2 * 0.72= -39.2 N-m

Left Wing 3:

local wind x= 3.2 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 319.2 N pos up.

drag local wind path= 14.0 N pos aft.

side local wind path= 17.8 N pos right.

lift aircraft x y z= 0.0 318.3 -24.2 N

drag aircraft x y z= 0.8 1.1 13.9 N

side aircraft x y z= 17.7 -0.1 -1.0 N

--RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 3

f_axil= -317.6 N, pos aft, L= -139.5 N-m, pos right

f_side= -114.4 N, pos right, M= -3549.2 N-m, pos up
f_nrml= 11397.4 N, pos up, N= 318.3 N-m, pos right

---Right Wing 3 ELEMENT # 0---

element x= 3.96 m, pos right from CG
element y= -0.10 m, pos up from CG
element z= 0.23 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.4 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.5 (deg)
v for forces : 58.4

AILERON #1: crat= 0.2000, def= -17.0 deg: (pos down or extended, neg up)

dx_te = 0.9913
dy_te = -0.0585
al_inc = -2.7128
cl_inc = -0.2519
cd_inc = 0.0009
cm_inc = 0.1637

Effective alpha increased -2.7 deg by all combined lift-incrementation devices up to final geo alpha of 1.8 deg

For Reynolds number: rho= 0.7985, chord= 61.97, dynamic_viscosity = 16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.9846 million at this element

element center at 12% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 1.8, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 1.5

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.2620
mid foil: Cd lookup: 0.0081
mid foil: Cm lookup: -0.0120

Mach of 0.1806 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to 0.2664

Mach of 0.1806 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to -0.0122

Mach of 0.1806 with a Cl-based drag divergence Mach number of 0.7734 takes the Cd to 0.0081 (increase of 0.0000)

final Cl 0.2664
final Cd 0.0081
final Cm -0.0122

adding control-effects cl of -0.2519 takes cl to 0.0145

adding control-effects cd of 0.0009 takes cd to 0.0089

adding control-effects cm of 0.1637 takes cm to 0.1515

finite-wing lift multiplier of 0.9543 takes cl to 0.0139

delta-wing moment addition of -0.0001 takes cm to 0.1514

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0089
cl multiplied by 1.0000 to get to 0.0139 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0139 * 0.4176 * \text{sqr}(58.4) * 0.7985 / 2$ = 7.9 N
drag= $0.0089 * 0.4176 * \text{sqr}(58.4) * 0.7985 / 2$ = 5.1 N
momt= $0.1514 * 0.4176 * \text{sqr}(58.4) * 0.7985 / 2 * 1.06$ = 91.4 N-m

Right Wing 3:

local wind x= 3.3 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 7.9 N pos up.
drag local wind path= 5.1 N pos aft.
side local wind path= -0.5 N pos right.

lift aircraft x y z= 0.0 7.9 -0.6 N
drag aircraft x y z= 0.3 0.4 5.1 N
side aircraft x y z= -0.5 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 3

f_axil= -313.1 N, pos aft, L= -172.1 N-m, pos right
f_side= -114.6 N, pos right, M= -3460.3 N-m, pos up
f_nrml= 11405.6 N, pos up, N= 330.6 N-m, pos right

---Right Wing 3 ELEMENT # 1-----

element x= 4.36 m, pos right from CG
element y= -0.07 m, pos up from CG
element z= 0.25 m, pos aft from CG

So final speed is vx= 3.3 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.3 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.4 (deg)
v for forces : 58.2

AILERON #1: crat= 0.1923, def= -17.0 deg: (pos down or extended, neg up)

dx_te = 0.9916
dy_te = -0.0562
al_inc = -2.6082
cl inc = -0.2423
cd inc = 0.0008
cm inc = 0.1584

Effective alpha increased -2.6 deg by all combined lift-incrementation devices up to final geo alpha of 1.8 deg

For Reynolds number: rho= 0.7985, chord= 56.70, dynamic_viscosity = 16.5794 millionths (divide by

this to get Re min Meg), Re_meg= 2.7308 million at this element

element center at 38% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 1.8, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 1.6

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.2628

mid foil: Cd lookup: 0.0081

mid foil: Cm lookup: -0.0120

Mach of 0.1798 yields Prandtl-Glauert Cl multiplier of 1.0166 to take Cl to 0.2672

Mach of 0.1798 yields Prandtl-Glauert Cm multiplier of 1.0166 to take Cm to -0.0122

Mach of 0.1798 with a Cl-based drag divergence Mach number of 0.7733 takes the Cd to 0.0081 (increase of 0.0000)

final Cl 0.2672

final Cd 0.0081

final Cm -0.0122

adding control-effects cl of -0.2423 takes cl to 0.0249

adding control-effects cd of 0.0008 takes cd to 0.0089

adding control-effects cm of 0.1584 takes cm to 0.1462

finite-wing lift multiplier of 0.9543 takes cl to 0.0237

delta-wing moment addition of -0.0002 takes cm to 0.1461

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0089

cl multiplied by 1.0000 to get to 0.0237 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0237 * 0.3837 * \sqrt{58.2} * 0.7985 / 2 = 12.3 \text{ N}$

drag= $0.0089 * 0.3837 * \sqrt{58.2} * 0.7985 / 2 = 4.6 \text{ N}$

momt= $0.1461 * 0.3837 * \sqrt{58.2} * 0.7985 / 2 * 0.97 = 73.8 \text{ N-m}$

Right Wing 3:

local wind x= 3.3 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 12.3 N pos up.

drag local wind path= 4.6 N pos aft.

side local wind path= -0.7 N pos right.

lift aircraft x y z= 0.0 12.3 -0.9 N

drag aircraft x y z= 0.3 0.3 4.6 N

side aircraft x y z= -0.7 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 3

f_axil= -309.4 N, pos aft, L= -227.0 N-m, pos right

f_side= -115.1 N, pos right, M= -3390.1 N-m, pos up

f_nrml= 11418.3 N, pos up, N= 342.6 N-m, pos right

---Right Wing 3 ELEMENT # 2-----

element x= 4.75 m, pos right from CG
element y= -0.05 m, pos up from CG
element z= 0.29 m, pos aft from CG

So final speed is vx= 3.2 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.2 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.3 (deg)
v for forces : 57.6

AILERON #1: crat= 0.1878, def= -17.0 deg: (pos down or extended, neg up)

dx_te = 0.9918

dy_te = -0.0549

al_inc = -2.5468

cl inc = -0.2367

cd inc = 0.0008

cm inc = 0.1553

Effective alpha increased -2.5 deg by all combined lift-incrementation devices up to final geo alpha of 1.8 deg

For Reynolds number: rho= 0.7985, chord= 49.70, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.3940 million at this element

element center at 62% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 1.8, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 1.5

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.2595

mid foil: Cd lookup: 0.0081

mid foil: Cm lookup: -0.0120

Mach of 0.1781 yields Prandtl-Glauert Cl multiplier of 1.0162 to take Cl to 0.2637

Mach of 0.1781 yields Prandtl-Glauert Cm multiplier of 1.0162 to take Cm to -0.0122

Mach of 0.1781 with a Cl-based drag divergence Mach number of 0.7736 takes the Cd to 0.0081 (increase of 0.0000)

final Cl 0.2637

final Cd 0.0081

final Cm -0.0122

adding control-effects cl of -0.2367 takes cl to 0.0269

adding control-effects cd of 0.0008 takes cd to 0.0089

adding control-effects cm of 0.1553 takes cm to 0.1431

finite-wing lift multiplier of 0.9543 takes cl to 0.0257

delta-wing moment addition of -0.0002 takes cm to 0.1429

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0089

cl multiplied by 1.0000 to get to 0.0257 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0257 * 0.3392 * sqrt(57.6) * 0.7985 / 2 = 11.6 N
drag= 0.0089 * 0.3392 * sqrt(57.6) * 0.7985 / 2 = 4.0 N

$$\text{momt} = 0.1429 * 0.3392 * \text{sqr}(57.6) * 0.7985 / 2 * 0.86 = 55.5 \text{ N-m}$$

Right Wing 3:

local wind x= 3.2 m/s in aircraft coord system pos right.
 local wind y= 4.4 m/s in aircraft coord system pos up.
 local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 11.5 N pos up.
 drag local wind path= 4.0 N pos aft.
 side local wind path= -0.7 N pos right.

lift aircraft x y z= 0.0 11.5 -0.9 N
 drag aircraft x y z= 0.2 0.3 4.0 N
 side aircraft x y z= -0.7 0.0 0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 3

f_axil= -306.3 N, pos aft, L= -283.1 N-m, pos right
 f_side= -115.5 N, pos right, M= -3338.3 N-m, pos up
 f_nrml= 11430.1 N, pos up, N= 354.6 N-m, pos right

 ---Right Wing 3 ELEMENT # 3-----

element x= 5.14 m, pos right from CG
 element y= -0.03 m, pos up from CG
 element z= 0.36 m, pos aft from CG

So final speed is vx= 3.2 vy= 4.4 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.1 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : 4.2 (deg)
 v for forces : 57.0

AILERON #1: crat= 0.1894, def= -17.0 deg: (pos down or extended, neg up)

dx_te = 0.9917
 dy_te = -0.0554
 al_inc = -2.5686
 cl_inc = -0.2389
 cd_inc = 0.0008
 cm_inc = 0.1565

Effective alpha increased -2.6 deg by all combined lift-incrementation devices up to final geo alpha of 1.6 deg

For Reynolds number: rho= 0.7985, chord= 41.09, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.9791 million at this element

element center at 88% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 1.6, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 1.4
 Using Reynolds number= 3.0000 million
 mid foil: Cl lookup: 0.2482
 mid foil: Cd lookup: 0.0081
 mid foil: Cm lookup: -0.0120
 Mach of 0.1761 yields Prandtl-Glauert Cl multiplier of 1.0159 to take Cl to 0.2521
 Mach of 0.1761 yields Prandtl-Glauert Cm multiplier of 1.0159 to take Cm to -0.0122
 Mach of 0.1761 with a Cl-based drag divergence Mach number of 0.7748 takes the Cd to 0.0081
 (increase of 0.0000)

final Cl 0.2521
 final Cd 0.0081
 final Cm -0.0122

adding control-effects cl of -0.2389 takes cl to 0.0132
 adding control-effects cd of 0.0008 takes cd to 0.0089
 adding control-effects cm of 0.1565 takes cm to 0.1443
 finite-wing lift multiplier of 0.9543 takes cl to 0.0126
 delta-wing moment addition of -0.0001 takes cm to 0.1442
 e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0000
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0089
 cl multiplied by 1.0000 to get to 0.0126 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0126 * 0.2831 * \sqrt{57.0} * 0.7985 / 2 = 4.6 \text{ N}$
 drag= $0.0089 * 0.2831 * \sqrt{57.0} * 0.7985 / 2 = 3.3 \text{ N}$
 momt= $0.1442 * 0.2831 * \sqrt{57.0} * 0.7985 / 2 * 0.72 = 38.2 \text{ N-m}$

Right Wing 3:

local wind x= 3.2 m/s in aircraft coord system pos right.
 local wind y= 4.4 m/s in aircraft coord system pos up.
 local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= 4.6 N pos up.
 drag local wind path= 3.3 N pos aft.
 side local wind path= -0.3 N pos right.

lift aircraft x y z= 0.0 4.6 -0.3 N
 drag aircraft x y z= 0.2 0.2 3.2 N
 side aircraft x y z= -0.3 0.0 0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 3

f_axil= -303.3 N, pos aft, L= -308.1 N-m, pos right
 f_side= -115.6 N, pos right, M= -3302.0 N-m, pos up
 f_nrml= 11434.9 N, pos up, N= 367.5 N-m, pos right

 ---Left Wing 4 ELEMENT # 0-----

element x= -5.40 m, pos right from CG
 element y= -0.01 m, pos up from CG
 element z= 0.43 m, pos aft from CG

So final speed is $v_x = 3.2$ $v_y = 4.4$ $v_z = 58.3$ m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.1 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.6 (deg)
v for forces : 52.6

For Reynolds number: $\rho = 0.7985$, chord = 30.74, dynamic_viscosity = 16.5794 millionths (divide by this to get Re in Meg), $Re_{meg} = 1.4808$ million at this element

element center at 25% span: (wgt root foil = 0.0000, wgt mid foil = 1.0000, wgt tip foil = 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow = 4.6, alpha zero-lift = -1.0, finite wing alpha multiplier = 0.8761, slat alpha divisor = 1.0000, alpha lookup = 3.9

Using Reynolds number = 3.0000 million

mid foil: Cl lookup: 0.5124

mid foil: Cd lookup: 0.0087

mid foil: Cm lookup: -0.0120

Mach of 0.1627 yields Prandtl-Glauert Cl multiplier of 1.0135 to take Cl to 0.5193

Mach of 0.1627 yields Prandtl-Glauert Cm multiplier of 1.0135 to take Cm to -0.0122

Mach of 0.1627 with a Cl-based drag divergence Mach number of 0.7481 takes the Cd to 0.0087 (increase of 0.0000)

final Cl 0.5193

final Cd 0.0087

final Cm -0.0122

finite-wing lift multiplier of 0.9381 takes cl to 0.4872

delta-wing moment addition of -0.0035 takes cm to -0.0157

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0090

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0090, which takes total Cd to 0.0177

cl multiplied by 1.0000 to get to 0.4872 to lower pressure from Mach 0.9 to 1.3.

lift = $0.4872 * 0.0748 * \sqrt{52.6} * 0.7985 / 2 = 40.3$ N

drag = $0.0177 * 0.0748 * \sqrt{52.6} * 0.7985 / 2 = 1.5$ N

momt = $-0.0157 * 0.0748 * \sqrt{52.6} * 0.7985 / 2 * 0.58 = -0.8$ N-m

Left Wing 4:

local wind x = 3.2 m/s in aircraft coord system pos right.

local wind y = 4.4 m/s in aircraft coord system pos up.

local wind z = 58.3 m/s in aircraft coord system pos aft.

lift local wind path = 40.2 N pos up.

drag local wind path = 1.5 N pos aft.

side local wind path = 2.2 N pos right.

lift aircraft x y z = 0.0 40.1 -3.1 N

drag aircraft x y z = 0.1 0.1 1.5 N

side aircraft x y z = 2.2 -0.0 -0.1 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 4

f_axil= -305.1 N, pos aft, L= -90.8 N-m, pos right
f_side= -113.3 N, pos right, M= -3320.0 N-m, pos up
f_nrml= 11475.1 N, pos up, N= 375.7 N-m, pos right

---Left Wing 4 ELEMENT # 1-----

element x= -5.53 m, pos right from CG
element y= -0.00 m, pos up from CG
element z= 0.54 m, pos aft from CG

So final speed is vx= 3.2 vy= 4.4 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.5 (deg)
v for forces : 41.2

For Reynolds number: rho= 0.7985, chord= 17.47, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 0.8416 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 4.5, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.8761, slat alpha divisor= 1.0000, alpha lookup= 3.8

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.5030

mid foil: Cd lookup: 0.0087

mid foil: Cm lookup: -0.0120

Mach of 0.1273 yields Prandtl-Glauert Cl multiplier of 1.0082 to take Cl to 0.5071

Mach of 0.1273 yields Prandtl-Glauert Cm multiplier of 1.0082 to take Cm to -0.0121

Mach of 0.1273 with a Cl-based drag divergence Mach number of 0.7493 takes the Cd to 0.0087 (increase of 0.0000)

final Cl 0.5071

final Cd 0.0087

final Cm -0.0121

finite-wing lift multiplier of 0.9381 takes cl to 0.4757

delta-wing moment addition of -0.0034 takes cm to -0.0155

e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 8.6695, gives Cdi of 0.0086

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0086, which takes total Cd to 0.0172

cl multiplied by 1.0000 to get to 0.4757 to lower pressure from Mach 0.9 to 1.3.

lift= 0.4757 * 0.0532 * sqrt(41.2) * 0.7985 / 2 = 17.1 N

drag= 0.0172 * 0.0532 * sqrt(41.2) * 0.7985 / 2 = 0.6 N

momt=-0.0155 * 0.0532 * sqrt(41.2) * 0.7985 / 2 * 0.42= -0.2 N-m

Left Wing 4:

local wind x= 3.2 m/s in aircraft coord system pos right.

local wind y= 4.4 m/s in aircraft coord system pos up.

local wind $z = 58.3$ m/s in aircraft coord system pos aft.

lift local wind path= 17.1 N pos up.
drag local wind path= 0.6 N pos aft.
side local wind path= 0.9 N pos right.

lift aircraft x y z= 0.0 17.1 -1.3 N
drag aircraft x y z= 0.0 0.0 0.6 N
side aircraft x y z= 0.9 -0.0 -0.1 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Wing 4

$f_{axil} = -305.8$ N, pos aft, $L = 3.9$ N-m, pos right
 $f_{side} = -112.4$ N, pos right, $M = -3329.4$ N-m, pos up
 $f_{nrml} = 11492.3$ N, pos up, $N = 379.3$ N-m, pos right

---Right Wing 4 ELEMENT # 0-----

element x= 5.40 m, pos right from CG
element y= -0.01 m, pos up from CG
element z= 0.43 m, pos aft from CG

So final speed is $v_x = 3.2$ $v_y = 4.4$ $v_z = 58.5$ m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.1 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 4.2 (deg)
v for forces : 49.6

For Reynolds number: $\rho = 0.7985$, chord= 28.98, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), $Re_{meg} = 1.3959$ million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 4.2, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 3.7

Using Reynolds number= 3.0000 million

mid foil: Cl lookup: 0.4934

mid foil: Cd lookup: 0.0086

mid foil: Cm lookup: -0.0120

Mach of 0.1533 yields Prandtl-Glauert Cl multiplier of 1.0120 to take Cl to 0.4993

Mach of 0.1533 yields Prandtl-Glauert Cm multiplier of 1.0120 to take Cm to -0.0121

Mach of 0.1533 with a Cl-based drag divergence Mach number of 0.7501 takes the Cd to 0.0086 (increase of 0.0000)

final Cl 0.4993
final Cd 0.0086
final Cm -0.0121

finite-wing lift multiplier of 0.9543 takes c_l to 0.4765
 delta-wing moment addition of -0.0034 takes c_m to -0.0156
 e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives C_{di} of 0.0061
 ground effect c_{di} multiplier of 1.0000 takes C_{di} to 0.0061, which takes total C_d to 0.0147
 c_l multiplied by 1.0000 to get to 0.4765 to lower pressure from Mach 0.9 to 1.3.

lift= $0.4765 * 0.0748 * \text{sqr}(49.6) * 0.7985 / 2 = 35.0 \text{ N}$
 drag= $0.0147 * 0.0748 * \text{sqr}(49.6) * 0.7985 / 2 = 1.1 \text{ N}$
 momt= $-0.0156 * 0.0748 * \text{sqr}(49.6) * 0.7985 / 2 * 0.58 = -0.7 \text{ N-m}$

Right Wing 4:

local wind x= 3.2 m/s in aircraft coord system pos right.
 local wind y= 4.4 m/s in aircraft coord system pos up.
 local wind z= 58.5 m/s in aircraft coord system pos aft.

lift local wind path= 35.0 N pos up.
 drag local wind path= 1.1 N pos aft.
 side local wind path= -1.9 N pos right.

lift aircraft x y z= 0.0 34.9 -2.6 N
 drag aircraft x y z= 0.1 0.1 1.1 N
 side aircraft x y z= -1.9 0.0 0.1 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 4

f_{axil} = -307.2 N, pos aft, L = -185.0 N-m, pos right
 f_{side} = -114.2 N, pos right, M = -3345.1 N-m, pos up
 f_{nrml} = 11527.2 N, pos up, N = 372.3 N-m, pos right

 ---Right Wing 4 ELEMENT # 1-----

element x= 5.53 m, pos right from CG
 element y= -0.00 m, pos up from CG
 element z= 0.54 m, pos aft from CG

So final speed is v_x = 3.2 v_y = 4.4 v_z = 58.5 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : 4.1 (deg)
 v for forces : 36.4

For Reynolds number: ρ = 0.7985, chord= 15.46, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_{meg} = 0.7446 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 23016.afl, I have Reynolds number 3.0000 million available

For lookup, alpha airflow= 4.1, alpha zero-lift= -1.0, finite wing alpha multiplier= 0.9087, slat alpha divisor= 1.0000, alpha lookup= 3.7

Using Reynolds number= 3.0000 million
mid foil: Cl lookup: 0.4839
mid foil: Cd lookup: 0.0086
mid foil: Cm lookup: -0.0120
Mach of 0.1126 yields Prandtl-Glauert Cl multiplier of 1.0064 to take Cl to 0.4869
Mach of 0.1126 yields Prandtl-Glauert Cm multiplier of 1.0064 to take Cm to -0.0121
Mach of 0.1126 with a Cl-based drag divergence Mach number of 0.7513 takes the Cd to 0.0086
(increase of 0.0000)

final Cl 0.4869
final Cd 0.0086
final Cm -0.0121

finite-wing lift multiplier of 0.9543 takes cl to 0.4647
delta-wing moment addition of -0.0033 takes cm to -0.0154
e of 0.9704, geometric aspect ratio of 9.2364, effective aspect ratio of 12.2709, gives Cdi of 0.0058
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0058, which takes total Cd to 0.0144
cl multiplied by 1.0000 to get to 0.4647 to lower pressure from Mach 0.9 to 1.3.

lift= 0.4647 * 0.0532 * sqrt(36.4) * 0.7985 / 2 = 13.1 N
drag= 0.0144 * 0.0532 * sqrt(36.4) * 0.7985 / 2 = 0.4 N
momt=-0.0154 * 0.0532 * sqrt(36.4) * 0.7985 / 2 * 0.42= -0.2 N-m

Right Wing 4:

local wind x= 3.2 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 58.5 m/s in aircraft coord system pos aft.

lift local wind path= 13.1 N pos up.
drag local wind path= 0.4 N pos aft.
side local wind path= -0.7 N pos right.

lift aircraft x y z=	0.0	13.1	-1.0 N
drag aircraft x y z=	0.0	0.0	0.4 N
side aircraft x y z=	-0.7	0.0	0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Wing 4

f_axil= -307.8 N, pos aft, L= -257.4 N-m, pos right
f_side= -114.9 N, pos right, M= -3352.3 N-m, pos up
f_nrml= 11540.3 N, pos up, N= 369.7 N-m, pos right

---Left Stab ELEMENT # 0-----

element x= -0.10 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.68 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.34 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 2.7308:
dz_in_wash= 7.44 m, radii off in z-direction= 7.1543 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7418

wash AT the ring= 3.1, total rat= 0.7418, so total vz_wash_disc= 2.3

wash vx_disc= 2.1, vy_disc= 0.7, vz_disc= 2.3 m/s (disc coords)
wash vx_acft= 2.1, vy_acft= 0.7, vz_acft= 2.3 m/s (aircraft coords)
So total speed is vx= 5.3, vy= 5.0, vz= 60.7 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

BODY SHADOWING with Cd of 0.1511 multiplies V by 0.9213

So speed is changed to vx= 4.9 vy= 4.6 vz= 55.9 m/s

So final speed is vx= 4.9 vy= 4.6 vz= 55.9 m/s in the aircraft axis coordinate system

alpha of the wing plane : 4.7 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)

incidence increase : 0.0 (deg)

geometric alpha : 1.7 (deg)

v for forces : 56.3

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0358

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of 2.0 deg

For Reynolds number: rho= 0.7985, chord= 57.23, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.7564 million at this element

element center at 5% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= 2.0, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= 1.7

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: 0.1825

mid foil: Cd lookup: 0.0053

mid foil: Cm lookup: 0.0000

Mach of 0.1740 yields Prandtl-Glauert Cl multiplier of 1.0155 to take Cl to 0.1853

Mach of 0.1740 yields Prandtl-Glauert Cm multiplier of 1.0155 to take Cm to 0.0000

Mach of 0.1740 with a Cl-based drag divergence Mach number of 0.8315 takes the Cd to 0.0053 (increase of 0.0000)

final Cl 0.1853

final Cd 0.0053

final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to 0.2211

adding control-effects cd of 0.0000 takes cd to 0.0053

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.9265 takes cl to 0.2049

delta-wing moment addition of -0.0067 takes cm to -0.0273

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0019

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0019, which takes total Cd to 0.0072

cl multiplied by 1.0000 to get to 0.2049 to lower pressure from Mach 0.9 to 1.3.

$\text{lift} = 0.2049 * 0.2069 * \text{sqr}(56.3) * 0.7985 / 2 = 53.6 \text{ N}$
 $\text{drag} = 0.0072 * 0.2069 * \text{sqr}(56.3) * 0.7985 / 2 = 1.9 \text{ N}$
 $\text{momt} = -0.0273 * 0.2069 * \text{sqr}(56.3) * 0.7985 / 2 * 1.02 = -7.3 \text{ N-m}$

Left Stab:

local wind x= 4.9 m/s in aircraft coord system pos right.
 local wind y= 4.6 m/s in aircraft coord system pos up.
 local wind z= 55.9 m/s in aircraft coord system pos aft.

lift local wind path= 53.6 N pos up.
 drag local wind path= 1.9 N pos aft.
 side local wind path= -0.0 N pos right.

lift aircraft x y z= 0.0 53.5 -4.4 N
 drag aircraft x y z= 0.2 0.2 1.9 N
 side aircraft x y z= -0.0 0.0 0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

$f_{\text{axil}} = -310.3 \text{ N}$, pos aft, $L = -251.9 \text{ N-m}$, pos right
 $f_{\text{side}} = -114.8 \text{ N}$, pos right, $M = -3611.2 \text{ N-m}$, pos up
 $f_{\text{nrml}} = 11593.9 \text{ N}$, pos up, $N = 369.3 \text{ N-m}$, pos right

 ---Left Stab ELEMENT # 1-----

element x= -0.31 m, pos right from CG
 element y= 0.35 m, pos up from CG
 element z= 4.69 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.44 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 3.7544:
 dz_in_wash= 7.45 m, radii off in z-direction= 7.1698 m, propwash_speed_mult of speed AT the
 disc in z-direction= 0.7402
 wash AT the ring= 3.7, total rat= 0.7402, so total vz_wash_disc= 2.7

wash vx_disc= 1.5, vy_disc= 1.4, vz_disc= 2.7 m/s (disc coords)
 wash vx_acft= 1.5, vy_acft= 1.4, vz_acft= 2.7 m/s (aircraft coords)
 So total speed is vx= 4.7, vy= 5.8, vz= 61.1 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

BODY SHADOWING with Cd of 0.1511 multiplies V by 0.9213

So speed is changed to vx= 4.3 vy= 5.3 vz= 56.3 m/s

So final speed is vx= 4.3 vy= 5.3 vz= 56.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 5.4 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : 2.4 (deg)
 v for forces : 56.7

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0358

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of 2.7 deg

For Reynolds number: rho= 0.7985, chord= 55.39, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.6676 million at this element

element center at 15% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= 2.7, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= 2.3

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: 0.2420

mid foil: Cd lookup: 0.0054

mid foil: Cm lookup: 0.0000

Mach of 0.1752 yields Prandtl-Glauert Cl multiplier of 1.0157 to take Cl to 0.2458

Mach of 0.1752 yields Prandtl-Glauert Cm multiplier of 1.0157 to take Cm to 0.0000

Mach of 0.1752 with a Cl-based drag divergence Mach number of 0.8254 takes the Cd to 0.0054 (increase of 0.0000)

final Cl 0.2458

final Cd 0.0054

final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to 0.2815

adding control-effects cd of 0.0000 takes cd to 0.0054

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.9265 takes cl to 0.2608

delta-wing moment addition of -0.0085 takes cm to -0.0291

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0032

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0032, which takes total Cd to 0.0085

cl multiplied by 1.0000 to get to 0.2608 to lower pressure from Mach 0.9 to 1.3.

lift= 0.2608 * 0.1989 * sqr(56.7) * 0.7985 / 2 = 66.6 N

drag= 0.0085 * 0.1989 * sqr(56.7) * 0.7985 / 2 = 2.2 N

momt=-0.0291 * 0.1989 * sqr(56.7) * 0.7985 / 2 * 0.98= -7.3 N-m

Left Stab:

local wind x= 4.3 m/s in aircraft coord system pos right.

local wind y= 5.3 m/s in aircraft coord system pos up.

local wind z= 56.3 m/s in aircraft coord system pos aft.

lift local wind path= 66.6 N pos up.

drag local wind path= 2.2 N pos aft.

side local wind path= -0.1 N pos right.

lift aircraft x y z= 0.0 66.3 -6.3 N

drag aircraft x y z= 0.2 0.2 2.2 N

side aircraft x y z= -0.1 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_axil= -314.4 N, pos aft, L= -231.6 N-m, pos right
f_side= -114.7 N, pos right, M= -3931.9 N-m, pos up
f_nrml= 11660.4 N, pos up, N= 370.2 N-m, pos right

---Left Stab ELEMENT # 2-----

element x= -0.51 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.71 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.60 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 5.2814:
dz_in_wash= 7.47 m, radii off in z-direction= 7.1852 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7387

wash AT the ring= 3.5, total rat= 0.7387, so total vz_wash_disc= 2.6

wash vx_disc= 0.9, vy_disc= 1.5, vz_disc= 2.6 m/s (disc coords)
wash vx_acft= 0.9, vy_acft= 1.5, vz_acft= 2.6 m/s (aircraft coords)
So total speed is vx= 4.1, vy= 5.9, vz= 61.0 m/s in (aircraft coords)

So final speed is vx= 4.1 vy= 5.9 vz= 61.0 m/s in the aircraft axis coordinate system

alpha of the wing plane : 5.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 2.5 (deg)
v for forces : 61.4

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0358

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of 2.8 deg

For Reynolds number: rho= 0.7985, chord= 57.55, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.7718 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= 2.8, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= 2.4

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: 0.2487

mid foil: Cd lookup: 0.0054

mid foil: C_m lookup: 0.0000
Mach of 0.1897 yields Prandtl-Glauert C_l multiplier of 1.0185 to take C_l to 0.2533
Mach of 0.1897 yields Prandtl-Glauert C_m multiplier of 1.0185 to take C_m to 0.0000
Mach of 0.1897 with a C_l -based drag divergence Mach number of 0.8247 takes the C_d to 0.0054
(increase of 0.0000)

final C_l 0.2533
final C_d 0.0054
final C_m 0.0000

adding control-effects c_l of 0.0358 takes c_l to 0.2890
adding control-effects c_d of 0.0000 takes c_d to 0.0054
adding control-effects c_m of -0.0206 takes c_m to -0.0206
finite-wing lift multiplier of 0.9265 takes c_l to 0.2678
delta-wing moment addition of -0.0088 takes c_m to -0.0293
e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives C_{di} of 0.0033
ground effect c_{di} multiplier of 1.0000 takes C_{di} to 0.0033, which takes total C_d to 0.0087
 c_l multiplied by 1.0000 to get to 0.2678 to lower pressure from Mach 0.9 to 1.3.

lift = $0.2678 * 0.1908 * \sqrt{61.4} * 0.7985 / 2 = 76.9 \text{ N}$
drag = $0.0087 * 0.1908 * \sqrt{61.4} * 0.7985 / 2 = 2.5 \text{ N}$
momt = $-0.0293 * 0.1908 * \sqrt{61.4} * 0.7985 / 2 * 0.94 = -7.9 \text{ N-m}$

Left Stab:

local wind x = 4.1 m/s in aircraft coord system pos right.
local wind y = 5.9 m/s in aircraft coord system pos up.
local wind z = 61.0 m/s in aircraft coord system pos aft.

lift local wind path = 76.9 N pos up.
drag local wind path = 2.5 N pos aft.
side local wind path = -0.2 N pos right.

lift aircraft x y z = 0.0 76.5 -7.4 N
drag aircraft x y z = 0.2 0.2 2.5 N
side aircraft x y z = -0.2 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_{axil} = -319.3 N, pos aft, L = -192.5 N-m, pos right
 f_{side} = -114.7 N, pos right, M = -4303.1 N-m, pos up
 f_{nrml} = 11737.2 N, pos up, N = 372.8 N-m, pos right

---Left Stab ELEMENT # 3-----

element x = -0.71 m, pos right from CG
element y = 0.35 m, pos up from CG
element z = 4.73 m, pos aft from CG

Propwash from prop 1:

radial offset = 0.78 m, prop radius = 1.04 m, radial wash_rat = 0.9978, float ring # 7.0108:
dz_in_wash = 7.48 m, radii off in z-direction = 7.2006 m, propwash_speed_mult of speed AT the
disc in z-direction = 0.7371

wash AT the ring= 2.4, total rat= 0.7355, so total vz_wash_disc= 1.8

wash vx_disc= 0.6, vy_disc= 1.2, vz_disc= 1.8 m/s (disc coords)
wash vx_acft= 0.6, vy_acft= 1.2, vz_acft= 1.8 m/s (aircraft coords)
So total speed is vx= 3.7, vy= 5.6, vz= 60.1 m/s in (aircraft coords)

DOWNWASH from Left Wing 1 el #0 is 1.8 deg, at v= 60.5 m/s for vertical delta= -1.92 m/s
So speed is changed to vx= 3.7 vy= 3.7 vz= 60.1 m/s
So final speed is vx= 3.7 vy= 3.7 vz= 60.1 m/s in the aircraft axis coordinate system

alpha of the wing plane : 3.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 0.5 (deg)
v for forces : 60.4

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
dy_te = 0.0064
al_inc= 0.2922
cl inc= 0.0359
cd inc= 0.0000
cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of 0.8 deg

For Reynolds number: rho= 0.7985, chord= 54.20, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.6103 million at this element

element center at 35% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= 0.8, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= 0.7

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: 0.0707
mid foil: Cd lookup: 0.0052
mid foil: Cm lookup: 0.0000

Mach of 0.1866 yields Prandtl-Glauert Cl multiplier of 1.0179 to take Cl to 0.0719

Mach of 0.1866 yields Prandtl-Glauert Cm multiplier of 1.0179 to take Cm to 0.0000

Mach of 0.1866 with a Cl-based drag divergence Mach number of 0.8428 takes the Cd to 0.0052 (increase of 0.0000)

final Cl 0.0719
final Cd 0.0052
final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.1078

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.9265 takes cl to 0.0999

delta-wing moment addition of -0.0033 takes cm to -0.0239

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0005

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0005, which takes total Cd to 0.0057

cl multiplied by 1.0000 to get to 0.0999 to lower pressure from Mach 0.9 to 1.3.

$\text{lift} = 0.0999 * 0.1827 * \text{sqr}(60.4) * 0.7985 / 2 = 26.6 \text{ N}$
 $\text{drag} = 0.0057 * 0.1827 * \text{sqr}(60.4) * 0.7985 / 2 = 1.5 \text{ N}$
 $\text{momt} = -0.0239 * 0.1827 * \text{sqr}(60.4) * 0.7985 / 2 * 0.90 = -5.7 \text{ N-m}$

Left Stab:

local wind x= 3.7 m/s in aircraft coord system pos right.
 local wind y= 3.7 m/s in aircraft coord system pos up.
 local wind z= 60.1 m/s in aircraft coord system pos aft.

lift local wind path= 26.6 N pos up.
 drag local wind path= 1.5 N pos aft.
 side local wind path= -0.1 N pos right.

lift aircraft x y z= 0.0 26.5 -1.6 N
 drag aircraft x y z= 0.1 0.1 1.5 N
 side aircraft x y z= -0.1 0.0 0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

$f_{\text{axil}} = -319.4 \text{ N}$, pos aft, $L = -173.6 \text{ N-m}$, pos right
 $f_{\text{side}} = -114.7 \text{ N}$, pos right, $M = -4434.6 \text{ N-m}$, pos up
 $f_{\text{nrml}} = 11763.8 \text{ N}$, pos up, $N = 372.8 \text{ N-m}$, pos right

 ---Left Stab ELEMENT # 4-----

element x= -0.92 m, pos right from CG
 element y= 0.35 m, pos up from CG
 element z= 4.74 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.97 m, prop radius= 1.04 m, radial wash_rat= 0.6339, float ring # 8.8307:
 dz_in_wash= 7.50 m, radii off in z-direction= 7.2160 m, propwash_speed_mult of speed AT the
 disc in z-direction= 0.7355
 wash AT the ring= 0.6, total rat= 0.4662, so total vz_wash_disc= 0.3

wash vx_disc= 0.2, vy_disc= 0.7, vz_disc= 0.3 m/s (disc coords)
 wash vx_acft= 0.2, vy_acft= 0.7, vz_acft= 0.3 m/s (aircraft coords)
 So total speed is vx= 3.4, vy= 5.0, vz= 58.6 m/s in (aircraft coords)

DOWNWASH from Left Wing 1 el #1 is 1.7 deg, at v= 59.0 m/s for vertical delta= -1.76 m/s
 So speed is changed to vx= 3.4 vy= 3.3 vz= 58.6 m/s
 So final speed is vx= 3.4 vy= 3.3 vz= 58.6 m/s in the aircraft axis coordinate system

alpha of the wing plane : 3.2 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : 0.2 (deg)
 v for forces : 58.8

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)
 dx_te = 0.9999

dy_te = 0.0064
al_inc= 0.2922
cl inc= 0.0359
cd inc= 0.0000
cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of 0.5 deg

For Reynolds number: rho= 0.7985, chord= 50.48, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.4311 million at this element

element center at 45% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= 0.5, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= 0.4

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: 0.0422

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1818 yields Prandtl-Glauert Cl multiplier of 1.0169 to take Cl to 0.0430

Mach of 0.1818 yields Prandtl-Glauert Cm multiplier of 1.0169 to take Cm to 0.0000

Mach of 0.1818 with a Cl-based drag divergence Mach number of 0.8457 takes the Cd to 0.0052 (increase of 0.0000)

final Cl 0.0430

final Cd 0.0052

final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0788

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.9265 takes cl to 0.0730

delta-wing moment addition of -0.0024 takes cm to -0.0230

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0002

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0002, which takes total Cd to 0.0055

cl multiplied by 1.0000 to get to 0.0730 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0730 * 0.1747 * sqrt(58.8) * 0.7985 / 2 = 17.6 N

drag= 0.0055 * 0.1747 * sqrt(58.8) * 0.7985 / 2 = 1.3 N

momt=-0.0230 * 0.1747 * sqrt(58.8) * 0.7985 / 2 * 0.86= -4.8 N-m

Left Stab:

local wind x= 3.4 m/s in aircraft coord system pos right.

local wind y= 3.3 m/s in aircraft coord system pos up.

local wind z= 58.6 m/s in aircraft coord system pos aft.

lift local wind path= 17.6 N pos up.

drag local wind path= 1.3 N pos aft.

side local wind path= -0.1 N pos right.

lift aircraft x y z= 0.0 17.6 -1.0 N

drag aircraft x y z= 0.1 0.1 1.3 N

side aircraft x y z= -0.1 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_axil= -319.1 N, pos aft, L= -157.4 N-m, pos right
f_side= -114.7 N, pos right, M= -4523.0 N-m, pos up
f_nrml= 11781.4 N, pos up, N= 372.5 N-m, pos right

---Left Stab ELEMENT # 5-----

element x= -1.12 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.76 m, pos aft from CG

Propwash from prop 1:

radial offset= 1.16 m, prop radius= 1.04 m, radial wash_rat= 0.2606, float ring # 9.0000:
dz_in_wash= 7.52 m, radii off in z-direction= 7.2314 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7339
wash AT the ring= 0.4, total rat= 0.1913, so total vz_wash_disc= 0.1

wash vx_disc= 0.0, vy_disc= 0.0, vz_disc= 0.1 m/s (disc coords)
wash vx_acft= 0.0, vy_acft= 0.0, vz_acft= 0.1 m/s (aircraft coords)
So total speed is vx= 3.2, vy= 4.4, vz= 58.4 m/s in (aircraft coords)

DOWNWASH from Left Wing 2 el #0 is 1.6 deg, at v= 58.7 m/s for vertical delta= -1.66 m/s
So speed is changed to vx= 3.2 vy= 2.7 vz= 58.4 m/s
So final speed is vx= 3.2 vy= 2.7 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -0.4 (deg)
v for forces : 58.6

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
dy_te = 0.0064
al_inc= 0.2922
cl inc= 0.0359
cd inc= 0.0000
cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -0.1 deg

For Reynolds number: rho= 0.7985, chord= 47.94, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.3091 million at this element

element center at 55% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -0.1, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= -0.1

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.0072
mid foil: Cd lookup: 0.0052

mid foil: C_m lookup: 0.0000
Mach of 0.1810 yields Prandtl-Glauert C_l multiplier of 1.0168 to take C_l to -0.0073
Mach of 0.1810 yields Prandtl-Glauert C_m multiplier of 1.0168 to take C_m to 0.0000
Mach of 0.1810 with a C_l -based drag divergence Mach number of 0.8493 takes the C_d to 0.0052
(increase of 0.0000)

final C_l -0.0073
final C_d 0.0052
final C_m 0.0000

adding control-effects c_l of 0.0359 takes c_l to 0.0286
adding control-effects c_d of 0.0000 takes c_d to 0.0052
adding control-effects c_m of -0.0206 takes c_m to -0.0206
finite-wing lift multiplier of 0.9265 takes c_l to 0.0265
delta-wing moment addition of -0.0009 takes c_m to -0.0215
e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives C_{di} of 0.0000
ground effect c_{di} multiplier of 1.0000 takes C_{di} to 0.0000, which takes total C_d to 0.0052
 c_l multiplied by 1.0000 to get to 0.0265 to lower pressure from Mach 0.9 to 1.3.

lift = $0.0265 * 0.1666 * \sqrt{58.6} * 0.7985 / 2 = 6.0 \text{ N}$
drag = $0.0052 * 0.1666 * \sqrt{58.6} * 0.7985 / 2 = 1.2 \text{ N}$
momt = $-0.0215 * 0.1666 * \sqrt{58.6} * 0.7985 / 2 * 0.82 = -4.0 \text{ N-m}$

Left Stab:

local wind x = 3.2 m/s in aircraft coord system pos right.
local wind y = 2.7 m/s in aircraft coord system pos up.
local wind z = 58.4 m/s in aircraft coord system pos aft.

lift local wind path = 6.0 N pos up.
drag local wind path = 1.2 N pos aft.
side local wind path = -0.0 N pos right.

lift aircraft x y z =	0.0	6.0	-0.3 N
drag aircraft x y z =	0.1	0.1	1.2 N
side aircraft x y z =	-0.0	0.0	0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_{axil} = -318.2 N, pos aft, L = -150.5 N-m, pos right
 f_{side} = -114.6 N, pos right, M = -4555.7 N-m, pos up
 f_{nrml} = 11787.5 N, pos up, N = 371.3 N-m, pos right

---Left Stab ELEMENT # 6-----

element x = -1.32 m, pos right from CG
element y = 0.35 m, pos up from CG
element z = 4.77 m, pos aft from CG

DOWNWASH from Left Wing 2 el #0 is 1.6 deg, at v = 58.6 m/s for vertical delta = -1.65 m/s
So speed is changed to v_x = 3.2 v_y = 2.7 v_z = 58.3 m/s
So final speed is v_x = 3.2 v_y = 2.7 v_z = 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -0.4 (deg)
v for forces : 58.5

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0359

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -0.1 deg

For Reynolds number: rho= 0.7985, chord= 45.55, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.1941 million at this element

element center at 65% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -0.1, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= -0.1

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.0066

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1807 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to -0.0067

Mach of 0.1807 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to 0.0000

Mach of 0.1807 with a Cl-based drag divergence Mach number of 0.8493 takes the Cd to 0.0052 (increase of 0.0000)

final Cl -0.0067

final Cd 0.0052

final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0292

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.9265 takes cl to 0.0270

delta-wing moment addition of -0.0009 takes cm to -0.0215

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0052

cl multiplied by 1.0000 to get to 0.0270 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0270 * 0.1585 * \sqrt{58.5} * 0.7985 / 2 = 5.9 \text{ N}$

drag= $0.0052 * 0.1585 * \sqrt{58.5} * 0.7985 / 2 = 1.1 \text{ N}$

momt= $-0.0215 * 0.1585 * \sqrt{58.5} * 0.7985 / 2 * 0.78 = -3.6 \text{ N-m}$

Left Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.

local wind y= 2.7 m/s in aircraft coord system pos up.

local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 5.9 N pos up.

drag local wind path= 1.1 N pos aft.
side local wind path= -0.0 N pos right.

lift aircraft x y z= 0.0 5.8 -0.3 N
drag aircraft x y z= 0.1 0.1 1.1 N
side aircraft x y z= -0.0 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_axil= -317.3 N, pos aft, L= -142.7 N-m, pos right
f_side= -114.6 N, pos right, M= -4587.2 N-m, pos up
f_nrml= 11793.4 N, pos up, N= 370.1 N-m, pos right

---Left Stab ELEMENT # 7-----

element x= -1.53 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.79 m, pos aft from CG

DOWNWASH from Left Wing 2 el #0 is 1.6 deg, at v= 58.6 m/s for vertical delta= -1.65 m/s
So speed is changed to vx= 3.2 vy= 2.7 vz= 58.3 m/s
So final speed is vx= 3.2 vy= 2.7 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -0.4 (deg)
v for forces : 58.5

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
dy_te = 0.0064
al_inc= 0.2922
cl_inc= 0.0359
cd_inc= 0.0000
cm_inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -0.1 deg

For Reynolds number: rho= 0.7985, chord= 43.23, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.0824 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -0.1, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= -0.1

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.0066
mid foil: Cd lookup: 0.0052
mid foil: Cm lookup: 0.0000

Mach of 0.1807 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to -0.0067
Mach of 0.1807 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to 0.0000
Mach of 0.1807 with a Cl-based drag divergence Mach number of 0.8493 takes the Cd to 0.0052
(increase of 0.0000)

final Cl -0.0067
final Cd 0.0052
final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0292
adding control-effects cd of 0.0000 takes cd to 0.0052
adding control-effects cm of -0.0206 takes cm to -0.0206
finite-wing lift multiplier of 0.9265 takes cl to 0.0270
delta-wing moment addition of -0.0009 takes cm to -0.0215
e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0000
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0052
cl multiplied by 1.0000 to get to 0.0270 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0270 * 0.1505 * sqrt(58.5) * 0.7985 / 2 = 5.6 N
drag= 0.0052 * 0.1505 * sqrt(58.5) * 0.7985 / 2 = 1.1 N
momt=-0.0215 * 0.1505 * sqrt(58.5) * 0.7985 / 2 * 0.74= -3.3 N-m

Left Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.
local wind y= 2.7 m/s in aircraft coord system pos up.
local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 5.6 N pos up.
drag local wind path= 1.1 N pos aft.
side local wind path= -0.0 N pos right.

lift aircraft x y z= 0.0 5.6 -0.3 N
drag aircraft x y z= 0.1 0.0 1.1 N
side aircraft x y z= -0.0 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_axil= -316.5 N, pos aft, L= -134.2 N-m, pos right
f_side= -114.6 N, pos right, M= -4617.0 N-m, pos up
f_nrml= 11799.0 N, pos up, N= 368.8 N-m, pos right

---Left Stab ELEMENT # 8-----

element x= -1.73 m, pos right from CG
element y= 0.34 m, pos up from CG
element z= 4.81 m, pos aft from CG

DOWNWASH from Left Wing 2 el #1 is 1.6 deg, at v= 58.6 m/s for vertical delta= -1.64 m/s
So speed is changed to vx= 3.2 vy= 2.7 vz= 58.3 m/s
So final speed is vx= 3.2 vy= 2.7 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-

incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -0.4 (deg)
v for forces : 58.5

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0359

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -0.1 deg

For Reynolds number: rho= 0.7985, chord= 40.92, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 1.9707 million at this element

element center at 85% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -0.1, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= -0.1

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.0061

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1807 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to -0.0062

Mach of 0.1807 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to 0.0000

Mach of 0.1807 with a Cl-based drag divergence Mach number of 0.8494 takes the Cd to 0.0052 (increase of 0.0000)

final Cl -0.0062

final Cd 0.0052

final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0297

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.9265 takes cl to 0.0275

delta-wing moment addition of -0.0009 takes cm to -0.0215

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0052

cl multiplied by 1.0000 to get to 0.0275 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0275 * 0.1424 * \sqrt{58.5} * 0.7985 / 2 = 5.3 \text{ N}$

drag= $0.0052 * 0.1424 * \sqrt{58.5} * 0.7985 / 2 = 1.0 \text{ N}$

momt=-0.0215 * 0.1424 * $\sqrt{58.5} * 0.7985 / 2 * 0.70 = -2.9 \text{ N-m}$

Left Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.

local wind y= 2.7 m/s in aircraft coord system pos up.

local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 5.3 N pos up.

drag local wind path= 1.0 N pos aft.

side local wind path= -0.0 N pos right.

lift aircraft x y z= 0.0 5.3 -0.2 N
drag aircraft x y z= 0.1 0.0 1.0 N
side aircraft x y z= -0.0 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_axil= -315.7 N, pos aft, L= -124.8 N-m, pos right
f_side= -114.6 N, pos right, M= -4645.6 N-m, pos up
f_nrml= 11804.4 N, pos up, N= 367.4 N-m, pos right

---Left Stab ELEMENT # 9-----

element x= -1.93 m, pos right from CG
element y= 0.34 m, pos up from CG
element z= 4.82 m, pos aft from CG

DOWNWASH from Left Wing 2 el #1 is 1.6 deg, at v= 58.6 m/s for vertical delta= -1.64 m/s
So speed is changed to vx= 3.2 vy= 2.7 vz= 58.3 m/s
So final speed is vx= 3.2 vy= 2.7 vz= 58.3 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -0.4 (deg)
v for forces : 58.5

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
dy_te = 0.0064
al_inc= 0.2922
cl inc= 0.0359
cd inc= 0.0000
cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -0.1 deg

For Reynolds number: rho= 0.7985, chord= 38.60, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 1.8590 million at this element

element center at 95% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -0.1, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.8530, slat alpha divisor= 1.0000, alpha lookup= -0.1

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.0061

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1807 yields Prandtl-Glauert Cl multiplier of 1.0167 to take Cl to -0.0062

Mach of 0.1807 yields Prandtl-Glauert Cm multiplier of 1.0167 to take Cm to 0.0000
Mach of 0.1807 with a Cl-based drag divergence Mach number of 0.8494 takes the Cd to 0.0052
(increase of 0.0000)

final Cl -0.0062
final Cd 0.0052
final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0296
adding control-effects cd of 0.0000 takes cd to 0.0052
adding control-effects cm of -0.0206 takes cm to -0.0206
finite-wing lift multiplier of 0.9265 takes cl to 0.0275
delta-wing moment addition of -0.0009 takes cm to -0.0215
e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 7.0005, gives Cdi of 0.0000
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0052
cl multiplied by 1.0000 to get to 0.0275 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0275 * 0.1343 * \sqrt{58.5} * 0.7985 / 2 = 5.0 \text{ N}$
drag= $0.0052 * 0.1343 * \sqrt{58.5} * 0.7985 / 2 = 1.0 \text{ N}$
momt= $-0.0215 * 0.1343 * \sqrt{58.5} * 0.7985 / 2 * 0.66 = -2.6 \text{ N-m}$

Left Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.
local wind y= 2.7 m/s in aircraft coord system pos up.
local wind z= 58.3 m/s in aircraft coord system pos aft.

lift local wind path= 5.0 N pos up.
drag local wind path= 1.0 N pos aft.
side local wind path= -0.0 N pos right.

lift aircraft x y z= 0.0 5.0 -0.2 N
drag aircraft x y z= 0.1 0.0 1.0 N
side aircraft x y z= -0.0 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Left Stab

f_axil= -315.0 N, pos aft, L= -115.0 N-m, pos right
f_side= -114.6 N, pos right, M= -4672.4 N-m, pos up
f_nrml= 11809.5 N, pos up, N= 366.0 N-m, pos right

---Right Stab ELEMENT # 0-----

element x= 0.10 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.68 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.34 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 2.7308:
dz_in_wash= 7.44 m, radii off in z-direction= 7.1543 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7418
wash AT the ring= 3.1, total rat= 0.7418, so total vz_wash_disc= 2.3

wash vx_disc= 2.1, vy_disc= -0.7, vz_disc= 2.3 m/s (disc coords)
wash vx_acft= 2.1, vy_acft= -0.7, vz_acft= 2.3 m/s (aircraft coords)
So total speed is vx= 5.3, vy= 3.7, vz= 60.7 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2
BODY SHADOWING with Cd of 0.1511 multiplies V by 0.9213
So speed is changed to vx= 4.9 vy= 3.4 vz= 55.9 m/s
So final speed is vx= 4.9 vy= 3.4 vz= 55.9 m/s in the aircraft axis coordinate system

alpha of the wing plane : 3.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)
element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : 0.5 (deg)
v for forces : 55.4

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0359

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of 0.8 deg

For Reynolds number: rho= 0.7985, chord= 56.36, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.7145 million at this element

element center at 5% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= 0.8, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha divisor= 1.0000, alpha lookup= 0.6

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: 0.0623

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1713 yields Prandtl-Glauert Cl multiplier of 1.0150 to take Cl to 0.0632

Mach of 0.1713 yields Prandtl-Glauert Cm multiplier of 1.0150 to take Cm to 0.0000

Mach of 0.1713 with a Cl-based drag divergence Mach number of 0.8437 takes the Cd to 0.0052 (increase of 0.0000)

final Cl 0.0632

final Cd 0.0052

final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0991

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.8924 takes cl to 0.0884

delta-wing moment addition of -0.0029 takes cm to -0.0235

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0006

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0006, which takes total Cd to 0.0058

cl multiplied by 1.0000 to get to 0.0884 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0884 * 0.2069 * sqr(55.4) * 0.7985 / 2 = 22.5 N

drag= $0.0058 * 0.2069 * \text{sqr}(55.4) * 0.7985 / 2$ = 1.5 N
momt=- $0.0235 * 0.2069 * \text{sqr}(55.4) * 0.7985 / 2 * 1.02$ = -6.1 N-m

Right Stab:

local wind x= 4.9 m/s in aircraft coord system pos right.
local wind y= 3.4 m/s in aircraft coord system pos up.
local wind z= 55.9 m/s in aircraft coord system pos aft.

lift local wind path= 22.5 N pos up.
drag local wind path= 1.5 N pos aft.
side local wind path= 0.0 N pos right.

lift aircraft x y z= 0.0 22.4 -1.4 N
drag aircraft x y z= 0.1 0.1 1.5 N
side aircraft x y z= 0.0 -0.0 -0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -314.9 N, pos aft, L= -117.2 N-m, pos right
f_side= -114.4 N, pos right, M= -4783.7 N-m, pos up
f_nrml= 11832.0 N, pos up, N= 365.3 N-m, pos right

---Right Stab ELEMENT # 1-----

element x= 0.31 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.69 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.44 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 3.7544:
dz_in_wash= 7.45 m, radii off in z-direction= 7.1698 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7402
wash AT the ring= 3.7, total rat= 0.7402, so total vz_wash_disc= 2.7

wash vx_disc= 1.5, vy_disc= -1.4, vz_disc= 2.7 m/s (disc coords)
wash vx_acft= 1.5, vy_acft= -1.4, vz_acft= 2.7 m/s (aircraft coords)
So total speed is vx= 4.7, vy= 2.9, vz= 61.1 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

check-point is in the wake of Body 22 (serpra e) which has Cd=0.5

BODY SHADOWING with Cd of 0.6513 multiplies V by 0.5905

So speed is changed to vx= 2.8 vy= 1.7 vz= 36.1 m/s

So final speed is vx= 2.8 vy= 1.7 vz= 36.1 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.7 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -0.3 (deg)
v for forces : 35.8

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
dy_te = 0.0064
al_inc= 0.2922
cl_inc= 0.0359
cd_inc= 0.0000
cm_inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of 0.0 deg

For Reynolds number: rho= 0.7985, chord= 34.96, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.6840 million at this element

element center at 15% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= 0.0, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha divisor= 1.0000, alpha lookup= 0.0

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: 0.0026

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1106 yields Prandtl-Glauert Cl multiplier of 1.0062 to take Cl to 0.0027

Mach of 0.1106 yields Prandtl-Glauert Cm multiplier of 1.0062 to take Cm to 0.0000

Mach of 0.1106 with a Cl-based drag divergence Mach number of 0.8497 takes the Cd to 0.0052 (increase of 0.0000)

final Cl 0.0027

final Cd 0.0052

final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0385

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.8924 takes cl to 0.0344

delta-wing moment addition of -0.0011 takes cm to -0.0218

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0001

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0001, which takes total Cd to 0.0053

cl multiplied by 1.0000 to get to 0.0344 to lower pressure from Mach 0.9 to 1.3.

lift= 0.0344 * 0.1989 * sqrt(35.8) * 0.7985 / 2 = 3.5 N

drag= 0.0053 * 0.1989 * sqrt(35.8) * 0.7985 / 2 = 0.5 N

momt=-0.0218 * 0.1989 * sqrt(35.8) * 0.7985 / 2 * 0.98= -2.2 N-m

Right Stab:

local wind x= 2.8 m/s in aircraft coord system pos right.

local wind y= 1.7 m/s in aircraft coord system pos up.

local wind z= 36.1 m/s in aircraft coord system pos aft.

lift local wind path= 3.5 N pos up.

drag local wind path= 0.5 N pos aft.

side local wind path= 0.0 N pos right.

lift aircraft x y z= 0.0 3.5 -0.2 N

drag aircraft x y z= 0.0 0.0 0.5 N

side aircraft x y z= 0.0 -0.0 -0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -314.5 N, pos aft, L= -118.3 N-m, pos right
f_side= -114.4 N, pos right, M= -4802.2 N-m, pos up
f_nrml= 11835.5 N, pos up, N= 365.2 N-m, pos right

---Right Stab ELEMENT # 2-----

element x= 0.51 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.71 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.60 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 5.2814:
dz_in_wash= 7.47 m, radii off in z-direction= 7.1852 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7387

wash AT the ring= 3.5, total rat= 0.7387, so total vz_wash_disc= 2.6

wash vx_disc= 0.9, vy_disc= -1.5, vz_disc= 2.6 m/s (disc coords)
wash vx_acft= 0.9, vy_acft= -1.5, vz_acft= 2.6 m/s (aircraft coords)
So total speed is vx= 4.1, vy= 2.8, vz= 61.0 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

check-point is in the wake of Body 22 (serpentine) which has Cd=0.5

BODY SHADOWING with Cd of 0.6513 multiplies V by 0.5905

So speed is changed to vx= 2.4 vy= 1.7 vz= 36.0 m/s

So final speed is vx= 2.4 vy= 1.7 vz= 36.0 m/s in the aircraft axis coordinate system

alpha of the wing plane : 2.7 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -0.3 (deg)
v for forces : 35.8

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0359

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -0.0 deg

For Reynolds number: rho= 0.7985, chord= 33.52, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.6143 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -0.0, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha divisor= 1.0000, alpha lookup= -0.0

Using Reynolds number= 6.0000 million
mid foil: Cl lookup: -0.0025
mid foil: Cd lookup: 0.0052
mid foil: Cm lookup: 0.0000
Mach of 0.1105 yields Prandtl-Glauert Cl multiplier of 1.0062 to take Cl to -0.0025
Mach of 0.1105 yields Prandtl-Glauert Cm multiplier of 1.0062 to take Cm to 0.0000
Mach of 0.1105 with a Cl-based drag divergence Mach number of 0.8497 takes the Cd to 0.0052
(increase of 0.0000)

final Cl -0.0025
final Cd 0.0052
final Cm 0.0000

adding control-effects cl of 0.0359 takes cl to 0.0334
adding control-effects cd of 0.0000 takes cd to 0.0052
adding control-effects cm of -0.0206 takes cm to -0.0206
finite-wing lift multiplier of 0.8924 takes cl to 0.0298
delta-wing moment addition of -0.0010 takes cm to -0.0216
e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0001
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0001, which takes total Cd to 0.0053
cl multiplied by 1.0000 to get to 0.0298 to lower pressure from Mach 0.9 to 1.3.

lift= $0.0298 * 0.1908 * \sqrt{35.8} * 0.7985 / 2 = 2.9 \text{ N}$
drag= $0.0053 * 0.1908 * \sqrt{35.8} * 0.7985 / 2 = 0.5 \text{ N}$
momt= $-0.0216 * 0.1908 * \sqrt{35.8} * 0.7985 / 2 * 0.94 = -2.0 \text{ N-m}$

Right Stab:

local wind x= 2.4 m/s in aircraft coord system pos right.
local wind y= 1.7 m/s in aircraft coord system pos up.
local wind z= 36.0 m/s in aircraft coord system pos aft.

lift local wind path= 2.9 N pos up.
drag local wind path= 0.5 N pos aft.
side local wind path= 0.0 N pos right.

lift aircraft x y z= 0.0 2.9 -0.1 N
drag aircraft x y z= 0.0 0.0 0.5 N
side aircraft x y z= 0.0 -0.0 -0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -314.1 N, pos aft, L= -119.8 N-m, pos right
f_side= -114.3 N, pos right, M= -4817.8 N-m, pos up
f_nrml= 11838.4 N, pos up, N= 365.2 N-m, pos right

---Right Stab ELEMENT # 3-----

element x= 0.71 m, pos right from CG
element y= 0.35 m, pos up from CG
element z= 4.73 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.78 m, prop radius= 1.04 m, radial wash_rat= 0.9978, float ring # 7.0108:
dz_in_wash= 7.48 m, radii off in z-direction= 7.2006 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7371

wash AT the ring= 2.4, total rat= 0.7355, so total vz_wash_disc= 1.8

wash vx_disc= 0.6, vy_disc= -1.2, vz_disc= 1.8 m/s (disc coords)
wash vx_acft= 0.6, vy_acft= -1.2, vz_acft= 1.8 m/s (aircraft coords)
So total speed is vx= 3.7, vy= 3.1, vz= 60.2 m/s in (aircraft coords)

DOWNWASH from Right Wing 1 el #0 is 2.9 deg, at v= 60.4 m/s for vertical delta= -3.02 m/s
So speed is changed to vx= 3.7 vy= 0.1 vz= 60.2 m/s
check-point is in the wake of Body 1 (fuselage) which has Cd=0.2
BODY SHADOWING with Cd of 0.1511 multiplies V by 0.9213
So speed is changed to vx= 3.4 vy= 0.1 vz= 55.4 m/s
So final speed is vx= 3.4 vy= 0.1 vz= 55.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 0.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-
incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -2.9 (deg)
v for forces : 55.0

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
dy_te = 0.0064
al_inc= 0.2922
cl inc= 0.0357
cd inc= 0.0000
cm inc=-0.0205

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of
-2.6 deg

For Reynolds number: rho= 0.7985, chord= 49.37, dynamic_viscosity =16.5794 millionths (divide by
this to get Re in Meg), Re_meg= 2.3780 million at this element

element center at 35% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available
For lookup, alpha airflow= -2.6, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha
divisor= 1.0000, alpha lookup= -2.1
Using Reynolds number= 6.0000 million
mid foil: Cl lookup: -0.2155
mid foil: Cd lookup: 0.0053
mid foil: Cm lookup: 0.0000
Mach of 0.1700 yields Prandtl-Glauert Cl multiplier of 1.0148 to take Cl to -0.2187
Mach of 0.1700 yields Prandtl-Glauert Cm multiplier of 1.0148 to take Cm to 0.0000
Mach of 0.1700 with a Cl-based drag divergence Mach number of 0.8281 takes the Cd to 0.0053
(increase of 0.0000)

final Cl -0.2187
final Cd 0.0053
final Cm 0.0000

adding control-effects cl of 0.0357 takes cl to -0.1830
adding control-effects cd of 0.0000 takes cd to 0.0053
adding control-effects cm of -0.0205 takes cm to -0.0205

finite-wing lift multiplier of 0.8924 takes c_l to -0.1633
delta-wing moment addition of 0.0054 takes c_m to -0.0152
e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives C_{di} of 0.0020
ground effect c_{di} multiplier of 1.0000 takes C_{di} to 0.0020, which takes total C_d to 0.0073
 c_l multiplied by 1.0000 to get to -0.1633 to lower pressure from Mach 0.9 to 1.3.

lift = $-0.1633 * 0.1827 * \sqrt{55.0} * 0.7985 / 2 = -36.0 \text{ N}$
drag = $0.0073 * 0.1827 * \sqrt{55.0} * 0.7985 / 2 = 1.6 \text{ N}$
momt = $-0.0152 * 0.1827 * \sqrt{55.0} * 0.7985 / 2 * 0.90 = -3.0 \text{ N-m}$

Right Stab:

local wind x = 3.4 m/s in aircraft coord system pos right.
local wind y = 0.1 m/s in aircraft coord system pos up.
local wind z = 55.4 m/s in aircraft coord system pos aft.

lift local wind path = -36.0 N pos up.
drag local wind path = 1.6 N pos aft.
side local wind path = -0.1 N pos right.

lift aircraft x y z = -0.0 -36.0 0.1 N
drag aircraft x y z = 0.1 0.0 1.6 N
side aircraft x y z = -0.1 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

$f_{axil} = -312.5 \text{ N}$, pos aft, $L = -94.1 \text{ N-m}$, pos right
 $f_{side} = -114.4 \text{ N}$, pos right, $M = -4650.0 \text{ N-m}$, pos up
 $f_{nrml} = 11802.4 \text{ N}$, pos up, $N = 366.5 \text{ N-m}$, pos right

---Right Stab ELEMENT # 4-----

element x = 0.92 m, pos right from CG
element y = 0.35 m, pos up from CG
element z = 4.74 m, pos aft from CG

Propwash from prop 1:

radial offset = 0.97 m, prop radius = 1.04 m, radial wash_rat = 0.6339, float ring # 8.8307:
dz_in_wash = 7.50 m, radii off in z-direction = 7.2160 m, propwash_speed_mult of speed AT the
disc in z-direction = 0.7355
wash AT the ring = 0.6, total rat = 0.4662, so total $v_{z_wash_disc} = 0.3$

wash $v_{x_disc} = 0.2$, $v_{y_disc} = -0.7$, $v_{z_disc} = 0.3 \text{ m/s}$ (disc coords)
wash $v_{x_acft} = 0.2$, $v_{y_acft} = -0.7$, $v_{z_acft} = 0.3 \text{ m/s}$ (aircraft coords)
So total speed is $v_x = 3.4$, $v_y = 3.7$, $v_z = 58.7 \text{ m/s}$ in (aircraft coords)

DOWNWASH from Right Wing 1 el #1 is 3.1 deg, at $v = 58.9 \text{ m/s}$ for vertical delta = -3.18 m/s
So speed is changed to $v_x = 3.4$ $v_y = 0.5$ $v_z = 58.7 \text{ m/s}$
So final speed is $v_x = 3.4$ $v_y = 0.5$ $v_z = 58.7 \text{ m/s}$ in the aircraft axis coordinate system

alpha of the wing plane : 0.5 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)
element incidence : -3.0 (deg)

incidence increase : 0.0 (deg)
geometric alpha : -2.5 (deg)
v for forces : 58.2

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl_inc= 0.0358

cd_inc= 0.0000

cm_inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -2.2 deg

For Reynolds number: rho= 0.7985, chord= 49.96, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.4065 million at this element

element center at 45% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -2.2, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha divisor= 1.0000, alpha lookup= -1.7

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.1820

mid foil: Cd lookup: 0.0053

mid foil: Cm lookup: 0.0000

Mach of 0.1799 yields Prandtl-Glauert Cl multiplier of 1.0166 to take Cl to -0.1850

Mach of 0.1799 yields Prandtl-Glauert Cm multiplier of 1.0166 to take Cm to 0.0000

Mach of 0.1799 with a Cl-based drag divergence Mach number of 0.8315 takes the Cd to 0.0053 (increase of 0.0000)

final Cl -0.1850

final Cd 0.0053

final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to -0.1492

adding control-effects cd of 0.0000 takes cd to 0.0053

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.8924 takes cl to -0.1332

delta-wing moment addition of 0.0044 takes cm to -0.0162

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0013

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0013, which takes total Cd to 0.0066

cl multiplied by 1.0000 to get to -0.1332 to lower pressure from Mach 0.9 to 1.3.

lift=-0.1332 * 0.1747 * sqrt(58.2) * 0.7985 / 2 = -31.5 N

drag= 0.0066 * 0.1747 * sqrt(58.2) * 0.7985 / 2 = 1.6 N

momt=-0.0162 * 0.1747 * sqrt(58.2) * 0.7985 / 2 * 0.86= -3.3 N-m

Right Stab:

local wind x= 3.4 m/s in aircraft coord system pos right.

local wind y= 0.5 m/s in aircraft coord system pos up.

local wind z= 58.7 m/s in aircraft coord system pos aft.

lift local wind path= -31.5 N pos up.

drag local wind path= 1.6 N pos aft.

side local wind path= -0.1 N pos right.

lift aircraft x y z=	-0.0	-31.5	0.3 N
drag aircraft x y z=	0.1	0.0	1.6 N
side aircraft x y z=	-0.1	0.0	0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

 Right Stab

f_axil= -310.6 N, pos aft, L= -65.3 N-m, pos right
 f_side= -114.4 N, pos right, M= -4503.4 N-m, pos up
 f_nrml= 11770.9 N, pos up, N= 368.3 N-m, pos right

 ---Right Stab ELEMENT # 5-----

element x= 1.12 m, pos right from CG
 element y= 0.35 m, pos up from CG
 element z= 4.76 m, pos aft from CG

Propwash from prop 1:

radial offset= 1.16 m, prop radius= 1.04 m, radial wash_rat= 0.2606, float ring # 9.0000:
 dz_in_wash= 7.52 m, radii off in z-direction= 7.2314 m, propwash_speed_mult of speed AT the
 disc in z-direction= 0.7339
 wash AT the ring= 0.4, total rat= 0.1913, so total vz_wash_disc= 0.1

wash vx_disc= 0.0, vy_disc= -0.0, vz_disc= 0.1 m/s (disc coords)
 wash vx_acft= 0.0, vy_acft= 0.0, vz_acft= 0.1 m/s (aircraft coords)
 So total speed is vx= 3.2, vy= 4.3, vz= 58.5 m/s in (aircraft coords)

DOWNWASH from Right Wing 2 el #0 is 3.3 deg, at v= 58.7 m/s for vertical delta= -3.34 m/s
 So speed is changed to vx= 3.2 vy= 1.0 vz= 58.5 m/s
 So final speed is vx= 3.2 vy= 1.0 vz= 58.5 m/s in the aircraft axis coordinate system

alpha of the wing plane : 1.0 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-
 incidence plane)
 element incidence : -3.0 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : -2.0 (deg)
 v for forces : 58.1

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
 dy_te = 0.0064
 al_inc= 0.2922
 cl inc= 0.0358
 cd inc= 0.0000
 cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of
 -1.7 deg

For Reynolds number: rho= 0.7985, chord= 47.52, dynamic_viscosity =16.5794 millionths (divide by
 this to get Re min Meg), Re_meg= 2.2887 million at this element

element center at 55% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available
 For lookup, alpha airflow= -1.7, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha
 divisor= 1.0000, alpha lookup= -1.3
 Using Reynolds number= 6.0000 million
 mid foil: Cl lookup: -0.1403
 mid foil: Cd lookup: 0.0052
 mid foil: Cm lookup: 0.0000
 Mach of 0.1794 yields Prandtl-Glauert Cl multiplier of 1.0165 to take Cl to -0.1426
 Mach of 0.1794 yields Prandtl-Glauert Cm multiplier of 1.0165 to take Cm to 0.0000
 Mach of 0.1794 with a Cl-based drag divergence Mach number of 0.8357 takes the Cd to 0.0052
 (increase of 0.0000)

final Cl -0.1426
 final Cd 0.0052
 final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to -0.1068
 adding control-effects cd of 0.0000 takes cd to 0.0052
 adding control-effects cm of -0.0206 takes cm to -0.0206
 finite-wing lift multiplier of 0.8924 takes cl to -0.0953
 delta-wing moment addition of 0.0031 takes cm to -0.0175
 e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0007
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0007, which takes total Cd to 0.0059
 cl multiplied by 1.0000 to get to -0.0953 to lower pressure from Mach 0.9 to 1.3.

lift=-0.0953 * 0.1666 * sqrt(58.1) * 0.7985 / 2 = -21.4 N
 drag= 0.0059 * 0.1666 * sqrt(58.1) * 0.7985 / 2 = 1.3 N
 momt=-0.0175 * 0.1666 * sqrt(58.1) * 0.7985 / 2 * 0.82= -3.2 N-m

Right Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.
 local wind y= 1.0 m/s in aircraft coord system pos up.
 local wind z= 58.5 m/s in aircraft coord system pos aft.

lift local wind path= -21.4 N pos up.
 drag local wind path= 1.3 N pos aft.
 side local wind path= -0.1 N pos right.

lift aircraft x y z= -0.0 -21.4 0.4 N
 drag aircraft x y z= 0.1 0.0 1.3 N
 side aircraft x y z= -0.1 0.0 0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -308.9 N, pos aft, L= -41.4 N-m, pos right
 f_side= -114.4 N, pos right, M= -4404.5 N-m, pos up
 f_nrml= 11749.6 N, pos up, N= 370.4 N-m, pos right

 ---Right Stab ELEMENT # 6-----

element x= 1.32 m, pos right from CG
 element y= 0.35 m, pos up from CG

element z= 4.77 m, pos aft from CG

DOWNWASH from Right Wing 2 el #0 is 3.2 deg, at v= 58.6 m/s for vertical delta= -3.33 m/s

So speed is changed to vx= 3.2 vy= 1.0 vz= 58.4 m/s

So final speed is vx= 3.2 vy= 1.0 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 1.0 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)

incidence increase : 0.0 (deg)

geometric alpha : -2.0 (deg)

v for forces : 58.0

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0358

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -1.7 deg

For Reynolds number: rho= 0.7985, chord= 45.16, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.1749 million at this element

element center at 65% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -1.7, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha divisor= 1.0000, alpha lookup= -1.3

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.1392

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1792 yields Prandtl-Glauert Cl multiplier of 1.0164 to take Cl to -0.1414

Mach of 0.1792 yields Prandtl-Glauert Cm multiplier of 1.0164 to take Cm to 0.0000

Mach of 0.1792 with a Cl-based drag divergence Mach number of 0.8359 takes the Cd to 0.0052 (increase of 0.0000)

final Cl -0.1414

final Cd 0.0052

final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to -0.1056

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.8924 takes cl to -0.0943

delta-wing moment addition of 0.0031 takes cm to -0.0175

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0007

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0007, which takes total Cd to 0.0059

cl multiplied by 1.0000 to get to -0.0943 to lower pressure from Mach 0.9 to 1.3.

lift=-0.0943 * 0.1585 * sqr(58.0) * 0.7985 / 2 = -20.1 N

drag= 0.0059 * 0.1585 * sqr(58.0) * 0.7985 / 2 = 1.3 N

momt=-0.0175 * 0.1585 * sqr(58.0) * 0.7985 / 2 * 0.78= -2.9 N-m

Right Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.

local wind y= 1.0 m/s in aircraft coord system pos up.

local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= -20.1 N pos up.

drag local wind path= 1.3 N pos aft.

side local wind path= -0.1 N pos right.

lift aircraft x y z= -0.0 -20.1 0.4 N

drag aircraft x y z= 0.1 0.0 1.3 N

side aircraft x y z= -0.1 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -307.3 N, pos aft, L= -14.9 N-m, pos right

f_side= -114.5 N, pos right, M= -4311.2 N-m, pos up

f_nrml= 11729.6 N, pos up, N= 372.7 N-m, pos right

---Right Stab ELEMENT # 7-----

element x= 1.53 m, pos right from CG

element y= 0.35 m, pos up from CG

element z= 4.79 m, pos aft from CG

DOWNWASH from Right Wing 2 el #0 is 3.2 deg, at v= 58.6 m/s for vertical delta= -3.32 m/s

So speed is changed to vx= 3.2 vy= 1.0 vz= 58.4 m/s

So final speed is vx= 3.2 vy= 1.0 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 1.0 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)

incidence increase : 0.0 (deg)

geometric alpha : -2.0 (deg)

v for forces : 58.0

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999

dy_te = 0.0064

al_inc= 0.2922

cl inc= 0.0358

cd inc= 0.0000

cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -1.7 deg

For Reynolds number: rho= 0.7985, chord= 42.86, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.0644 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -1.7, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha divisor= 1.0000, alpha lookup= -1.3
 Using Reynolds number= 6.0000 million
 mid foil: Cl lookup: -0.1386
 mid foil: Cd lookup: 0.0052
 mid foil: Cm lookup: 0.0000
 Mach of 0.1792 yields Prandtl-Glauert Cl multiplier of 1.0164 to take Cl to -0.1408
 Mach of 0.1792 yields Prandtl-Glauert Cm multiplier of 1.0164 to take Cm to 0.0000
 Mach of 0.1792 with a Cl-based drag divergence Mach number of 0.8359 takes the Cd to 0.0052
 (increase of 0.0000)

final Cl -0.1408
 final Cd 0.0052
 final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to -0.1050
 adding control-effects cd of 0.0000 takes cd to 0.0052
 adding control-effects cm of -0.0206 takes cm to -0.0206
 finite-wing lift multiplier of 0.8924 takes cl to -0.0937
 delta-wing moment addition of 0.0031 takes cm to -0.0175
 e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0007
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0007, which takes total Cd to 0.0059
 cl multiplied by 1.0000 to get to -0.0937 to lower pressure from Mach 0.9 to 1.3.

lift=-0.0937 * 0.1505 * sqrt(58.0) * 0.7985 / 2 = -18.9 N
 drag= 0.0059 * 0.1505 * sqrt(58.0) * 0.7985 / 2 = 1.2 N
 momt=-0.0175 * 0.1505 * sqrt(58.0) * 0.7985 / 2 * 0.74= -2.6 N-m

Right Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.
 local wind y= 1.0 m/s in aircraft coord system pos up.
 local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= -18.9 N pos up.
 drag local wind path= 1.2 N pos aft.
 side local wind path= -0.1 N pos right.

lift aircraft x y z= -0.0 -18.9 0.3 N
 drag aircraft x y z= 0.1 0.0 1.2 N
 side aircraft x y z= -0.1 0.0 0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -305.8 N, pos aft, L= 13.9 N-m, pos right
 f_side= -114.5 N, pos right, M= -4222.8 N-m, pos up
 f_nrml= 11710.7 N, pos up, N= 375.3 N-m, pos right

 ---Right Stab ELEMENT # 8-----

element x= 1.73 m, pos right from CG
 element y= 0.34 m, pos up from CG
 element z= 4.81 m, pos aft from CG

DOWNWASH from Right Wing 2 el #1 is 3.3 deg, at $v = 58.6$ m/s for vertical $\Delta = -3.34$ m/s
So speed is changed to $v_x = 3.2$ $v_y = 1.0$ $v_z = 58.4$ m/s
So final speed is $v_x = 3.2$ $v_y = 1.0$ $v_z = 58.4$ m/s in the aircraft axis coordinate system

alpha of the wing plane : 1.0 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -2.0 (deg)
v for forces : 58.0

ELEVATOR #1: $crat = 0.3500$, $def = 1.0$ deg: (pos down or extended, neg up)

$dx_{te} = 0.9999$

$dy_{te} = 0.0064$

$al_{inc} = 0.2922$

$cl_{inc} = 0.0358$

$cd_{inc} = 0.0000$

$cm_{inc} = -0.0206$

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -1.7 deg

For Reynolds number: $\rho = 0.7985$, $chord = 40.57$, $dynamic_viscosity = 16.5794$ millionths (divide by this to get Re in Meg), $Re_{meg} = 1.9539$ million at this element

element center at 85% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).af, I have Reynolds number 6.0000 million available

For lookup, $\alpha_{airflow} = -1.7$, $\alpha_{zero-lift} = -0.0$, finite wing alpha multiplier= 0.7848, slat alpha divisor= 1.0000, alpha lookup= -1.3

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.1402

mid foil: Cd lookup: 0.0052

mid foil: Cm lookup: 0.0000

Mach of 0.1792 yields Prandtl-Glauert Cl multiplier of 1.0164 to take Cl to -0.1425

Mach of 0.1792 yields Prandtl-Glauert Cm multiplier of 1.0164 to take Cm to 0.0000

Mach of 0.1792 with a Cl -based drag divergence Mach number of 0.8358 takes the Cd to 0.0052 (increase of 0.0000)

final Cl -0.1425

final Cd 0.0052

final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to -0.1067

adding control-effects cd of 0.0000 takes cd to 0.0052

adding control-effects cm of -0.0206 takes cm to -0.0206

finite-wing lift multiplier of 0.8924 takes cl to -0.0952

delta-wing moment addition of 0.0031 takes cm to -0.0175

e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0007

ground effect Cdi multiplier of 1.0000 takes Cdi to 0.0007, which takes total Cd to 0.0059

cl multiplied by 1.0000 to get to -0.0952 to lower pressure from Mach 0.9 to 1.3.

$lift = -0.0952 * 0.1424 * \sqrt{58.0} * 0.7985 / 2 = -18.2$ N

$drag = 0.0059 * 0.1424 * \sqrt{58.0} * 0.7985 / 2 = 1.1$ N

$momt = -0.0175 * 0.1424 * \sqrt{58.0} * 0.7985 / 2 * 0.70 = -2.3$ N-m

Right Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.
local wind y= 1.0 m/s in aircraft coord system pos up.
local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= -18.2 N pos up.
drag local wind path= 1.1 N pos aft.
side local wind path= -0.1 N pos right.

lift aircraft x y z= -0.0 -18.2 0.3 N
drag aircraft x y z= 0.1 0.0 1.1 N
side aircraft x y z= -0.1 0.0 0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -304.3 N, pos aft, L= 45.3 N-m, pos right
f_side= -114.6 N, pos right, M= -4137.3 N-m, pos up
f_nrml= 11692.5 N, pos up, N= 378.2 N-m, pos right

---Right Stab ELEMENT # 9-----

element x= 1.93 m, pos right from CG
element y= 0.34 m, pos up from CG
element z= 4.82 m, pos aft from CG

DOWNWASH from Right Wing 2 el #1 is 3.3 deg, at v= 58.6 m/s for vertical delta= -3.34 m/s
So speed is changed to vx= 3.2 vy= 1.0 vz= 58.4 m/s
So final speed is vx= 3.2 vy= 1.0 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : 1.0 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : -3.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -2.0 (deg)
v for forces : 58.0

ELEVATOR #1: crat= 0.3500, def= 1.0 deg: (pos down or extended, neg up)

dx_te = 0.9999
dy_te = 0.0064
al_inc= 0.2922
cl inc= 0.0358
cd inc= 0.0000
cm inc=-0.0206

Effective alpha increased 0.3 deg by all combined lift-incrementation devices up to final geo alpha of -1.7 deg

For Reynolds number: rho= 0.7985, chord= 38.27, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 1.8434 million at this element

element center at 95% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available
For lookup, alpha airflow= -1.7, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7848, slat alpha

divisor= 1.0000, alpha lookup= -1.3
 Using Reynolds number= 6.0000 million
 mid foil: Cl lookup: -0.1396
 mid foil: Cd lookup: 0.0052
 mid foil: Cm lookup: 0.0000
 Mach of 0.1792 yields Prandtl-Glauert Cl multiplier of 1.0165 to take Cl to -0.1419
 Mach of 0.1792 yields Prandtl-Glauert Cm multiplier of 1.0165 to take Cm to 0.0000
 Mach of 0.1792 with a Cl-based drag divergence Mach number of 0.8358 takes the Cd to 0.0052
 (increase of 0.0000)

final Cl -0.1419
 final Cd 0.0052
 final Cm 0.0000

adding control-effects cl of 0.0358 takes cl to -0.1061
 adding control-effects cd of 0.0000 takes cd to 0.0052
 adding control-effects cm of -0.0206 takes cm to -0.0206
 finite-wing lift multiplier of 0.8924 takes cl to -0.0947
 delta-wing moment addition of 0.0031 takes cm to -0.0175
 e of 0.9819, geometric aspect ratio of 4.8577, effective aspect ratio of 4.3408, gives Cdi of 0.0007
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0007, which takes total Cd to 0.0059
 cl multiplied by 1.0000 to get to -0.0947 to lower pressure from Mach 0.9 to 1.3.

lift=-0.0947 * 0.1343 * sqrt(58.0) * 0.7985 / 2 = -17.1 N
 drag= 0.0059 * 0.1343 * sqrt(58.0) * 0.7985 / 2 = 1.1 N
 momt=-0.0175 * 0.1343 * sqrt(58.0) * 0.7985 / 2 * 0.66= -2.1 N-m

Right Stab:

local wind x= 3.2 m/s in aircraft coord system pos right.
 local wind y= 1.0 m/s in aircraft coord system pos up.
 local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= -17.1 N pos up.
 drag local wind path= 1.1 N pos aft.
 side local wind path= -0.1 N pos right.

lift aircraft x y z= -0.0 -17.1 0.3 N
 drag aircraft x y z= 0.1 0.0 1.1 N
 side aircraft x y z= -0.1 0.0 0.0 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Right Stab

f_axil= -303.0 N, pos aft, L= 78.3 N-m, pos right
 f_side= -114.7 N, pos right, M= -4056.7 N-m, pos up
 f_nrml= 11675.4 N, pos up, N= 381.3 N-m, pos right

 ---Vert Stab 1 ELEMENT # 0-----

element x= -0.00 m, pos right from CG
 element y= 0.12 m, pos up from CG
 element z= 4.44 m, pos aft from CG

Propwash from prop 1:
radial offset= 0.09 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 0.3267:
dz_in_wash= 7.20 m, radii off in z-direction= 6.9299 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7653
wash AT the ring= 0.7, total rat= 0.7653, so total vz_wash_disc= 0.5

wash vx_disc= 2.6, vy_disc= 0.0, vz_disc= 0.5 m/s (disc coords)
wash vx_acft= 2.6, vy_acft= 0.0, vz_acft= 0.5 m/s (aircraft coords)
So total speed is vx= 5.8, vy= 4.4, vz= 58.9 m/s in (aircraft coords)

check-point is buried inside of Body 1 (fuselage)
BODY BURYING multiplies V by 0.7071
So speed is changed to vx= 4.1 vy= 3.1 vz= 41.7 m/s
So final speed is vx= 4.1 vy= 3.1 vz= 41.7 m/s in the aircraft axis coordinate system

alpha of the wing plane : -5.6 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-
incidence plane)
element incidence : 0.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -5.6 (deg)
v for forces : 33.6

RUDDER #1: crat= 0.2600, def= 5.3 deg: (pos down or extended, neg up)
dx_te = 0.9989
dy_te = 0.0239
al_inc= 1.0965
cl inc= 0.1226
cd inc= 0.0001
cm inc=-0.0760
Effective alpha increased 1.1 deg by all combined lift-incrementation devices up to final geo alpha of
-4.5 deg

For Reynolds number: rho= 0.7985, chord= 54.57, dynamic_viscosity =16.5794 millionths (divide by
this to get Re min Meg), Re_meg= 2.6285 million at this element

element center at 5% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available
For lookup, alpha airflow= -4.5, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha
divisor= 1.0000, alpha lookup= -3.3
Using Reynolds number= 6.0000 million
mid foil: Cl lookup: -0.3506
mid foil: Cd lookup: 0.0056
mid foil: Cm lookup: 0.0000
Mach of 0.1040 yields Prandtl-Glauert Cl multiplier of 1.0054 to take Cl to -0.3525
Mach of 0.1040 yields Prandtl-Glauert Cm multiplier of 1.0054 to take Cm to 0.0000
Mach of 0.1040 with a Cl-based drag divergence Mach number of 0.8147 takes the Cd to 0.0056
(increase of 0.0000)

final Cl -0.3525
final Cd 0.0056
final Cm 0.0000

adding control-effects cl of 0.1226 takes cl to -0.2299
adding control-effects cd of 0.0001 takes cd to 0.0057
adding control-effects cm of -0.0760 takes cm to -0.0760
finite-wing lift multiplier of 0.8680 takes cl to -0.1995

delta-wing moment addition of 0.0182 takes cm to -0.0578
 e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0039
 ground effect cdi multiplier of 1.0000 takes Cdi to 0.0039, which takes total Cd to 0.0097
 cl multiplied by 1.0000 to get to -0.1995 to lower pressure from Mach 0.9 to 1.3.

lift = $-0.1995 * 0.2787 * \sqrt{33.6} * 0.7985 / 2 = -25.1 \text{ N}$
 drag = $0.0097 * 0.2787 * \sqrt{33.6} * 0.7985 / 2 = 1.2 \text{ N}$
 momt = $-0.0578 * 0.2787 * \sqrt{33.6} * 0.7985 / 2 * 1.62 = -11.8 \text{ N-m}$

Vert Stab 1:

local wind x = 4.1 m/s in aircraft coord system pos right.
 local wind y = 3.1 m/s in aircraft coord system pos up.
 local wind z = 41.7 m/s in aircraft coord system pos aft.

lift local wind path = -0.0 N pos up.
 drag local wind path = 1.2 N pos aft.
 side local wind path = 25.1 N pos right.

lift aircraft x y z = -0.0 -0.0 0.0 N
 drag aircraft x y z = 0.1 0.1 1.2 N
 side aircraft x y z = 25.0 -0.2 -2.5 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil = -304.2 N, pos aft, L = 81.2 N-m, pos right
 f_side = -89.6 N, pos right, M = -4056.4 N-m, pos up
 f_nrml = 11675.4 N, pos up, N = 281.4 N-m, pos right

 ---Vert Stab 1 ELEMENT # 1-----

element x = -0.00 m, pos right from CG
 element y = 0.29 m, pos up from CG
 element z = 4.55 m, pos aft from CG

Propwash from prop 1:

radial offset = 0.26 m, prop radius = 1.04 m, radial wash_rat = 1.0000, float ring # 1.9801:
 dz_in_wash = 7.31 m, radii off in z-direction = 7.0361 m, propwash_speed_mult of speed AT the
 disc in z-direction = 0.7541
 wash AT the ring = 2.3, total rat = 0.7541, so total vz_wash_disc = 1.8

wash vx_disc = 2.4, vy_disc = 0.0, vz_disc = 1.8 m/s (disc coords)
 wash vx_acft = 2.4, vy_acft = 0.0, vz_acft = 1.8 m/s (aircraft coords)
 So total speed is vx = 5.6, vy = 4.4, vz = 60.1 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

BODY SHADOWING with Cd of 0.1511 multiplies V by 0.9213

So speed is changed to vx = 5.1 vy = 4.0 vz = 55.4 m/s

So final speed is vx = 5.1 vy = 4.0 vz = 55.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : -5.3 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)

incidence increase : 0.0 (deg)
geometric alpha : -5.3 (deg)
v for forces : 44.7

RUDDER #1: crat= 0.2620, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9989

dy_te = 0.0241

al_inc= 1.1047

cl inc= 0.1238

cd inc= 0.0001

cm inc=-0.0767

Effective alpha increased 1.1 deg by all combined lift-incrementation devices up to final geo alpha of -4.2 deg

For Reynolds number: rho= 0.7985, chord= 67.67, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 3.2595 million at this element

element center at 15% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -4.2, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -3.1

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.3230

mid foil: Cd lookup: 0.0055

mid foil: Cm lookup: 0.0000

Mach of 0.1383 yields Prandtl-Glauert Cl multiplier of 1.0097 to take Cl to -0.3261

Mach of 0.1383 yields Prandtl-Glauert Cm multiplier of 1.0097 to take Cm to 0.0000

Mach of 0.1383 with a Cl-based drag divergence Mach number of 0.8174 takes the Cd to 0.0055 (increase of 0.0000)

final Cl -0.3261

final Cd 0.0055

final Cm 0.0000

adding control-effects cl of 0.1238 takes cl to -0.2023

adding control-effects cd of 0.0001 takes cd to 0.0057

adding control-effects cm of -0.0767 takes cm to -0.0767

finite-wing lift multiplier of 0.8680 takes cl to -0.1756

delta-wing moment addition of 0.0160 takes cm to -0.0606

e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0030

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0030, which takes total Cd to 0.0087

cl multiplied by 1.0000 to get to -0.1756 to lower pressure from Mach 0.9 to 1.3.

lift=-0.1756 * 0.2598 * sqrt(44.7) * 0.7985 / 2 = -36.5 N

drag= 0.0087 * 0.2598 * sqrt(44.7) * 0.7985 / 2 = 1.8 N

momt=-0.0606 * 0.2598 * sqrt(44.7) * 0.7985 / 2 * 1.51= -19.0 N-m

Vert Stab 1:

local wind x= 5.1 m/s in aircraft coord system pos right.

local wind y= 4.0 m/s in aircraft coord system pos up.

local wind z= 55.4 m/s in aircraft coord system pos aft.

lift local wind path= -0.0 N pos up.

drag local wind path= 1.8 N pos aft.

side local wind path= 36.5 N pos right.

lift aircraft x y z=	-0.0	-0.0	0.0 N
drag aircraft x y z=	0.2	0.1	1.8 N
side aircraft x y z=	36.3	-0.2	-3.3 N

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

 Vert Stab 1

f_axil= -305.8 N, pos aft, L= 91.7 N-m, pos right
 f_side= -53.1 N, pos right, M= -4056.4 N-m, pos up
 f_nrml= 11675.2 N, pos up, N= 134.3 N-m, pos right

 ---Vert Stab 1 ELEMENT # 2-----

element x= -0.00 m, pos right from CG
 element y= 0.46 m, pos up from CG
 element z= 4.66 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.43 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 3.6335:
 dz_in_wash= 7.42 m, radii off in z-direction= 7.1422 m, propwash_speed_mult of speed AT the
 disc in z-direction= 0.7431
 wash AT the ring= 3.6, total rat= 0.7431, so total vz_wash_disc= 2.7

wash vx_disc= 2.1, vy_disc= 0.0, vz_disc= 2.7 m/s (disc coords)
 wash vx_acft= 2.1, vy_acft= 0.0, vz_acft= 2.7 m/s (aircraft coords)
 So total speed is vx= 5.3, vy= 4.4, vz= 61.1 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

check-point is in the wake of Body 22 (serpentine) which has Cd=0.5

BODY SHADOWING with Cd of 0.6513 multiplies V by 0.5905

So speed is changed to vx= 3.1 vy= 2.6 vz= 36.1 m/s

So final speed is vx= 3.1 vy= 2.6 vz= 36.1 m/s in the aircraft axis coordinate system

alpha of the wing plane : -4.9 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
 incidence increase : 0.0 (deg)
 geometric alpha : -4.9 (deg)
 v for forces : 29.1

RUDDER #1: crat= 0.2642, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9989

dy_te = 0.0243

al_inc= 1.1142

cl inc= 0.1252

cd inc= 0.0001

cm inc=-0.0773

Effective alpha increased 1.1 deg by all combined lift-incrementation devices up to final geo alpha of -3.8 deg

For Reynolds number: rho= 0.7985, chord= 40.85, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 1.9675 million at this element

element center at 25% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -3.8, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -2.8

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.2960

mid foil: Cd lookup: 0.0055

mid foil: Cm lookup: 0.0000

Mach of 0.0900 yields Prandtl-Glauert Cl multiplier of 1.0041 to take Cl to -0.2972

Mach of 0.0900 yields Prandtl-Glauert Cm multiplier of 1.0041 to take Cm to 0.0000

Mach of 0.0900 with a Cl-based drag divergence Mach number of 0.8203 takes the Cd to 0.0055 (increase of 0.0000)

final Cl -0.2972

final Cd 0.0055

final Cm 0.0000

adding control-effects cl of 0.1252 takes cl to -0.1721

adding control-effects cd of 0.0001 takes cd to 0.0056

adding control-effects cm of -0.0773 takes cm to -0.0773

finite-wing lift multiplier of 0.8680 takes cl to -0.1493

delta-wing moment addition of 0.0136 takes cm to -0.0637

e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0022

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0022, which takes total Cd to 0.0078

cl multiplied by 1.0000 to get to -0.1493 to lower pressure from Mach 0.9 to 1.3.

lift=-0.1493 * 0.2409 * sqrt(29.1) * 0.7985 / 2 = -12.2 N

drag= 0.0078 * 0.2409 * sqrt(29.1) * 0.7985 / 2 = 0.6 N

momt=-0.0637 * 0.2409 * sqrt(29.1) * 0.7985 / 2 * 1.40= -7.3 N-m

Vert Stab 1:

local wind x= 3.1 m/s in aircraft coord system pos right.

local wind y= 2.6 m/s in aircraft coord system pos up.

local wind z= 36.1 m/s in aircraft coord system pos aft.

lift local wind path= -0.0 N pos up.

drag local wind path= 0.6 N pos aft.

side local wind path= 12.2 N pos right.

lift aircraft x y z= -0.0 -0.0 0.0 N

drag aircraft x y z= 0.1 0.0 0.6 N

side aircraft x y z= 12.1 -0.1 -1.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil= -306.2 N, pos aft, L= 97.3 N-m, pos right

f_side= -40.9 N, pos right, M= -4056.4 N-m, pos up

f_nrml= 11675.2 N, pos up, N= 84.8 N-m, pos right

---Vert Stab 1 ELEMENT # 3-----

element x= -0.00 m, pos right from CG
element y= 0.63 m, pos up from CG
element z= 4.78 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.60 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 5.2869:
dz_in_wash= 7.53 m, radii off in z-direction= 7.2483 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7322

wash AT the ring= 3.5, total rat= 0.7322, so total vz_wash_disc= 2.6

wash vx_disc= 1.8, vy_disc= 0.0, vz_disc= 2.6 m/s (disc coords)
wash vx_acft= 1.8, vy_acft= 0.0, vz_acft= 2.6 m/s (aircraft coords)
So total speed is vx= 5.0, vy= 4.4, vz= 61.0 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

check-point is in the wake of Body 22 (serpentine) which has Cd=0.5

BODY SHADOWING with Cd of 0.6513 multiplies V by 0.5905

So speed is changed to vx= 2.9 vy= 2.6 vz= 36.0 m/s

So final speed is vx= 2.9 vy= 2.6 vz= 36.0 m/s in the aircraft axis coordinate system

alpha of the wing plane : -4.7 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)

incidence increase : 0.0 (deg)

geometric alpha : -4.7 (deg)

v for forces : 29.1

RUDDER #1: crat= 0.2668, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9989

dy_te = 0.0245

al_inc= 1.1252

cl_inc= 0.1266

cd_inc= 0.0001

cm_inc=-0.0781

Effective alpha increased 1.1 deg by all combined lift-incrementation devices up to final geo alpha of -3.5 deg

For Reynolds number: rho= 0.7985, chord= 37.57, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.8095 million at this element

element center at 35% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -3.5, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -2.6

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.2726

mid foil: Cd lookup: 0.0054

mid foil: Cm lookup: 0.0000

Mach of 0.0898 yields Prandtl-Glauert Cl multiplier of 1.0041 to take Cl to -0.2737

Mach of 0.0898 yields Prandtl-Glauert Cm multiplier of 1.0041 to take Cm to 0.0000

Mach of 0.0898 with a Cl-based drag divergence Mach number of 0.8226 takes the Cd to 0.0054 (increase of 0.0000)

final Cl -0.2737

final Cd 0.0054

final Cm 0.0000

adding control-effects c_l of 0.1266 takes c_l to -0.1471
 adding control-effects c_d of 0.0001 takes c_d to 0.0055
 adding control-effects c_m of -0.0781 takes c_m to -0.0781
 finite-wing lift multiplier of 0.8680 takes c_l to -0.1277
 delta-wing moment addition of 0.0116 takes c_m to -0.0664
 e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives C_{di} of 0.0016
 ground effect c_{di} multiplier of 1.0000 takes C_{di} to 0.0016, which takes total C_d to 0.0071
 c_l multiplied by 1.0000 to get to -0.1277 to lower pressure from Mach 0.9 to 1.3.

$\text{lift} = -0.1277 * 0.2221 * \text{sqr}(29.1) * 0.7985 / 2 = -9.6 \text{ N}$
 $\text{drag} = 0.0071 * 0.2221 * \text{sqr}(29.1) * 0.7985 / 2 = 0.5 \text{ N}$
 $\text{momt} = -0.0664 * 0.2221 * \text{sqr}(29.1) * 0.7985 / 2 * 1.29 = -6.4 \text{ N-m}$

Vert Stab 1:

local wind $x = 2.9 \text{ m/s}$ in aircraft coord system pos right.
 local wind $y = 2.6 \text{ m/s}$ in aircraft coord system pos up.
 local wind $z = 36.0 \text{ m/s}$ in aircraft coord system pos aft.

lift local wind path = -0.0 N pos up.
 drag local wind path = 0.5 N pos aft.
 side local wind path = 9.6 N pos right.

lift aircraft $x \ y \ z = -0.0 \ -0.0 \ 0.0 \text{ N}$
 drag aircraft $x \ y \ z = 0.0 \ 0.0 \ 0.5 \text{ N}$
 side aircraft $x \ y \ z = 9.5 \ -0.1 \ -0.8 \text{ N}$

 ---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

$f_{\text{axil}} = -306.4 \text{ N}$, pos aft, $L = 103.4 \text{ N-m}$, pos right
 $f_{\text{side}} = -31.4 \text{ N}$, pos right, $M = -4056.5 \text{ N-m}$, pos up
 $f_{\text{nrml}} = 11675.2 \text{ N}$, pos up, $N = 45.5 \text{ N-m}$, pos right

 ---Vert Stab 1 ELEMENT # 4-----

element $x = -0.00 \text{ m}$, pos right from CG
 element $y = 0.80 \text{ m}$, pos up from CG
 element $z = 4.89 \text{ m}$, pos aft from CG

Propwash from prop 1:

radial offset = 0.77 m, prop radius = 1.04 m, radial wash_rat = 1.0000, float ring # 6.9403:
 $dz_{\text{in_wash}} = 7.64 \text{ m}$, radii off in z -direction = 7.3545 m, propwash_speed_mult of speed AT the
 disc in z -direction = 0.7215
 wash AT the ring = 2.5, total rat = 0.7215, so total $vz_{\text{wash_disc}} = 1.8$

wash $vx_{\text{disc}} = 1.4$, $vy_{\text{disc}} = 0.0$, $vz_{\text{disc}} = 1.8 \text{ m/s}$ (disc coords)
 wash $vx_{\text{acft}} = 1.4$, $vy_{\text{acft}} = 0.0$, $vz_{\text{acft}} = 1.8 \text{ m/s}$ (aircraft coords)
 So total speed is $vx = 4.6$, $vy = 4.4$, $vz = 60.2 \text{ m/s}$ in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has $C_d = 0.2$
 BODY SHADOWING with C_d of 0.1511 multiplies V by 0.9213
 So speed is changed to $vx = 4.2$, $vy = 4.0$, $vz = 55.4 \text{ m/s}$

So final speed is $v_x = 4.2$ $v_y = 4.0$ $v_z = 55.4$ m/s in the aircraft axis coordinate system

alpha of the wing plane : -4.3 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -4.3 (deg)
v for forces : 44.7

RUDDER #1: crat= 0.2699, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9989

dy_te = 0.0248

al_inc= 1.1382

cl inc= 0.1283

cd inc= 0.0001

cm inc=-0.0789

Effective alpha increased 1.1 deg by all combined lift-incrementation devices up to final geo alpha of -3.2 deg

For Reynolds number: rho= 0.7985, chord= 52.88, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.5471 million at this element

element center at 45% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -3.2, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -2.4

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.2475

mid foil: Cd lookup: 0.0054

mid foil: Cm lookup: 0.0000

Mach of 0.1381 yields Prandtl-Glauert Cl multiplier of 1.0097 to take Cl to -0.2499

Mach of 0.1381 yields Prandtl-Glauert Cm multiplier of 1.0097 to take Cm to 0.0000

Mach of 0.1381 with a Cl-based drag divergence Mach number of 0.8250 takes the Cd to 0.0054 (increase of 0.0000)

final Cl -0.2499

final Cd 0.0054

final Cm 0.0000

adding control-effects cl of 0.1283 takes cl to -0.1216

adding control-effects cd of 0.0001 takes cd to 0.0055

adding control-effects cm of -0.0789 takes cm to -0.0789

finite-wing lift multiplier of 0.8680 takes cl to -0.1055

delta-wing moment addition of 0.0096 takes cm to -0.0693

e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0011

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0011, which takes total Cd to 0.0066

cl multiplied by 1.0000 to get to -0.1055 to lower pressure from Mach 0.9 to 1.3.

lift=-0.1055 * 0.2032 * sqrt(44.7) * 0.7985 / 2 = -17.1 N

drag= 0.0066 * 0.2032 * sqrt(44.7) * 0.7985 / 2 = 1.1 N

momt=-0.0693 * 0.2032 * sqrt(44.7) * 0.7985 / 2 * 1.18= -13.3 N-m

Vert Stab 1:

local wind x= 4.2 m/s in aircraft coord system pos right.

local wind y= 4.0 m/s in aircraft coord system pos up.

local wind z= 55.4 m/s in aircraft coord system pos aft.

lift local wind path= -0.0 N pos up.
drag local wind path= 1.1 N pos aft.
side local wind path= 17.1 N pos right.

lift aircraft x y z= -0.0 -0.0 0.0 N
drag aircraft x y z= 0.1 0.1 1.1 N
side aircraft x y z= 17.1 -0.1 -1.3 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil= -306.6 N, pos aft, L= 117.2 N-m, pos right
f_side= -14.2 N, pos right, M= -4056.6 N-m, pos up
f_nrml= 11675.2 N, pos up, N= -24.9 N-m, pos right

---Vert Stab 1 ELEMENT # 5-----

element x= -0.00 m, pos right from CG
element y= 0.98 m, pos up from CG
element z= 5.00 m, pos aft from CG

Propwash from prop 1:

radial offset= 0.95 m, prop radius= 1.04 m, radial wash_rat= 0.6813, float ring # 8.5937:
dz_in_wash= 7.75 m, radii off in z-direction= 7.4606 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7110
wash AT the ring= 0.9, total rat= 0.4844, so total vz_wash_disc= 0.4

wash vx_disc= 0.8, vy_disc= 0.0, vz_disc= 0.4 m/s (disc coords)
wash vx_acft= 0.8, vy_acft= 0.0, vz_acft= 0.4 m/s (aircraft coords)
So total speed is vx= 4.0, vy= 4.3, vz= 58.8 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

BODY SHADOWING with Cd of 0.1511 multiplies V by 0.9213

So speed is changed to vx= 3.7 vy= 4.0 vz= 54.2 m/s

So final speed is vx= 3.7 vy= 4.0 vz= 54.2 m/s in the aircraft axis coordinate system

alpha of the wing plane : -3.9 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -3.9 (deg)
v for forces : 43.6

RUDDER #1: crat= 0.2736, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9988

dy_te = 0.0251

al_inc= 1.1537

cl_inc= 0.1303

cd_inc= 0.0001

cm_inc=-0.0799

Effective alpha increased 1.2 deg by all combined lift-incrementation devices up to final geo alpha of
-2.7 deg

For Reynolds number: rho= 0.7985, chord= 46.79, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 2.2537 million at this element

element center at 55% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available
For lookup, alpha airflow= -2.7, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -2.0

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.2120

mid foil: Cd lookup: 0.0053

mid foil: Cm lookup: 0.0000

Mach of 0.1347 yields Prandtl-Glauert Cl multiplier of 1.0092 to take Cl to -0.2139

Mach of 0.1347 yields Prandtl-Glauert Cm multiplier of 1.0092 to take Cm to 0.0000

Mach of 0.1347 with a Cl-based drag divergence Mach number of 0.8286 takes the Cd to 0.0053 (increase of 0.0000)

final Cl -0.2139

final Cd 0.0053

final Cm 0.0000

adding control-effects cl of 0.1303 takes cl to -0.0836

adding control-effects cd of 0.0001 takes cd to 0.0054

adding control-effects cm of -0.0799 takes cm to -0.0799

finite-wing lift multiplier of 0.8680 takes cl to -0.0726

delta-wing moment addition of 0.0066 takes cm to -0.0733

e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0005

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0005, which takes total Cd to 0.0060

cl multiplied by 1.0000 to get to -0.0726 to lower pressure from Mach 0.9 to 1.3.

lift=-0.0726 * 0.1844 * sqrt(43.6) * 0.7985 / 2 = -10.1 N

drag= 0.0060 * 0.1844 * sqrt(43.6) * 0.7985 / 2 = 0.8 N

momt=-0.0733 * 0.1844 * sqrt(43.6) * 0.7985 / 2 * 1.07= -11.0 N-m

Vert Stab 1:

local wind x= 3.7 m/s in aircraft coord system pos right.

local wind y= 4.0 m/s in aircraft coord system pos up.

local wind z= 54.2 m/s in aircraft coord system pos aft.

lift local wind path= -0.0 N pos up.

drag local wind path= 0.8 N pos aft.

side local wind path= 10.1 N pos right.

lift aircraft x y z= -0.0 -0.0 0.0 N

drag aircraft x y z= 0.1 0.1 0.8 N

side aircraft x y z= 10.1 -0.1 -0.7 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil= -306.5 N, pos aft, L= 127.1 N-m, pos right

f_side= -4.0 N, pos right, M= -4056.5 N-m, pos up

f_nrml= 11675.2 N, pos up, N= -64.7 N-m, pos right

---Vert Stab 1 ELEMENT # 6-----

element x= -0.00 m, pos right from CG
element y= 1.15 m, pos up from CG
element z= 5.11 m, pos aft from CG

Propwash from prop 1:

radial offset= 1.12 m, prop radius= 1.04 m, radial wash_rat= 0.3506, float ring # 9.0000:
dz_in_wash= 7.86 m, radii off in z-direction= 7.5668 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.7006
wash AT the ring= 0.4, total rat= 0.2456, so total vz_wash_disc= 0.1

wash vx_disc= 0.0, vy_disc= 0.0, vz_disc= 0.1 m/s (disc coords)
wash vx_acft= 0.0, vy_acft= 0.0, vz_acft= 0.1 m/s (aircraft coords)
So total speed is vx= 3.2, vy= 4.3, vz= 58.5 m/s in (aircraft coords)

check-point is in the wake of Body 1 (fuselage) which has Cd=0.2

BODY SHADOWING with Cd of 0.1511 multiplies V by 0.9213

So speed is changed to vx= 2.9 vy= 4.0 vz= 53.9 m/s

So final speed is vx= 2.9 vy= 4.0 vz= 53.9 m/s in the aircraft axis coordinate system

alpha of the wing plane : -3.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -3.1 (deg)
v for forces : 43.3

RUDDER #1: crat= 0.2780, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9988

dy_te = 0.0255

al_inc= 1.1726

cl inc= 0.1329

cd inc= 0.0001

cm inc=-0.0812

Effective alpha increased 1.2 deg by all combined lift-incrementation devices up to final geo alpha of -1.9 deg

For Reynolds number: rho= 0.7985, chord= 41.73, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 2.0101 million at this element

element center at 65% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -1.9, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -1.4

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.1498

mid foil: Cd lookup: 0.0053

mid foil: Cm lookup: 0.0000

Mach of 0.1338 yields Prandtl-Glauert Cl multiplier of 1.0091 to take Cl to -0.1511

Mach of 0.1338 yields Prandtl-Glauert Cm multiplier of 1.0091 to take Cm to 0.0000

Mach of 0.1338 with a Cl-based drag divergence Mach number of 0.8349 takes the Cd to 0.0053 (increase of 0.0000)

final Cl -0.1511
final Cd 0.0053
final Cm 0.0000

adding control-effects cl of 0.1329 takes cl to -0.0182
adding control-effects cd of 0.0001 takes cd to 0.0054
adding control-effects cm of -0.0812 takes cm to -0.0812
finite-wing lift multiplier of 0.8680 takes cl to -0.0158
delta-wing moment addition of 0.0014 takes cm to -0.0798
e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0000
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0054
cl multiplied by 1.0000 to get to -0.0158 to lower pressure from Mach 0.9 to 1.3.

lift=-0.0158 * 0.1655 * sqrt(43.3) * 0.7985 / 2 = -2.0 N
drag= 0.0054 * 0.1655 * sqrt(43.3) * 0.7985 / 2 = 0.7 N
momt=-0.0798 * 0.1655 * sqrt(43.3) * 0.7985 / 2 * 0.96= -9.5 N-m

Vert Stab 1:

local wind x= 2.9 m/s in aircraft coord system pos right.
local wind y= 4.0 m/s in aircraft coord system pos up.
local wind z= 53.9 m/s in aircraft coord system pos aft.

lift local wind path= -0.0 N pos up.
drag local wind path= 0.7 N pos aft.
side local wind path= 2.0 N pos right.

lift aircraft x y z= -0.0 -0.0 0.0 N
drag aircraft x y z= 0.0 0.0 0.7 N
side aircraft x y z= 2.0 -0.0 -0.1 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil= -305.9 N, pos aft, L= 129.4 N-m, pos right
f_side= -2.1 N, pos right, M= -4056.1 N-m, pos up
f_nrml= 11675.2 N, pos up, N= -65.4 N-m, pos right

---Vert Stab 1 ELEMENT # 7-----

element x= -0.00 m, pos right from CG
element y= 1.32 m, pos up from CG
element z= 5.22 m, pos aft from CG

Propwash from prop 1:

radial offset= 1.29 m, prop radius= 1.04 m, radial wash_rat= 0.0199, float ring # 9.0000:
dz_in_wash= 7.98 m, radii off in z-direction= 7.6729 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.6904

wash AT the ring= 0.4, total rat= 0.0137, so total vz_wash_disc= 0.0

wash vx_disc= 0.0, vy_disc= 0.0, vz_disc= 0.0 m/s (disc coords)
wash vx_acft= 0.0, vy_acft= 0.0, vz_acft= 0.0 m/s (aircraft coords)
So total speed is vx= 3.2, vy= 4.3, vz= 58.4 m/s in (aircraft coords)

So final speed is $v_x = 3.2$ $v_y = 4.3$ $v_z = 58.4$ m/s in the aircraft axis coordinate system

alpha of the wing plane : -3.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -3.1 (deg)
v for forces : 46.9

RUDDER #1: crat= 0.2836, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9988

dy_te = 0.0261

al_inc= 1.1960

cl inc= 0.1356

cd inc= 0.0001

cm inc=-0.0825

Effective alpha increased 1.2 deg by all combined lift-incrementation devices up to final geo alpha of -1.9 deg

For Reynolds number: rho= 0.7985, chord= 40.08, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.9303 million at this element

element center at 75% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -1.9, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -1.4

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.1483

mid foil: Cd lookup: 0.0053

mid foil: Cm lookup: 0.0000

Mach of 0.1449 yields Prandtl-Glauert Cl multiplier of 1.0107 to take Cl to -0.1499

Mach of 0.1449 yields Prandtl-Glauert Cm multiplier of 1.0107 to take Cm to 0.0000

Mach of 0.1449 with a Cl-based drag divergence Mach number of 0.8350 takes the Cd to 0.0053 (increase of 0.0000)

final Cl -0.1499

final Cd 0.0053

final Cm 0.0000

adding control-effects cl of 0.1356 takes cl to -0.0143

adding control-effects cd of 0.0001 takes cd to 0.0054

adding control-effects cm of -0.0825 takes cm to -0.0825

finite-wing lift multiplier of 0.8680 takes cl to -0.0124

delta-wing moment addition of 0.0011 takes cm to -0.0813

e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0000

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0054

cl multiplied by 1.0000 to get to -0.0124 to lower pressure from Mach 0.9 to 1.3.

lift=-0.0124 * 0.1467 * sqrt(46.9) * 0.7985 / 2 = -1.6 N

drag= 0.0054 * 0.1467 * sqrt(46.9) * 0.7985 / 2 = 0.7 N

momt=-0.0813 * 0.1467 * sqrt(46.9) * 0.7985 / 2 * 0.85= -9.0 N-m

Vert Stab 1:

local wind x= 3.2 m/s in aircraft coord system pos right.

local wind y= 4.3 m/s in aircraft coord system pos up.

local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= -0.0 N pos up.
drag local wind path= 0.7 N pos aft.
side local wind path= 1.6 N pos right.

lift aircraft x y z= -0.0 -0.0 0.0 N
drag aircraft x y z= 0.0 0.1 0.7 N
side aircraft x y z= 1.6 -0.0 -0.1 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil= -305.3 N, pos aft, L= 131.5 N-m, pos right
f_side= -0.4 N, pos right, M= -4055.5 N-m, pos up
f_nrml= 11675.3 N, pos up, N= -64.9 N-m, pos right

---Vert Stab 1 ELEMENT # 8-----

element x= -0.00 m, pos right from CG
element y= 1.49 m, pos up from CG
element z= 5.33 m, pos aft from CG

So final speed is vx= 3.2 vy= 4.3 vz= 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : -3.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)

element incidence : 0.0 (deg)
incidence increase : 0.0 (deg)
geometric alpha : -3.1 (deg)
v for forces : 46.9

RUDDER #1: crat= 0.2907, def= 5.3 deg: (pos down or extended, neg up)

dx_te = 0.9988
dy_te = 0.0267
al_inc= 1.2258
cl inc= 0.1390
cd inc= 0.0001
cm inc=-0.0840

Effective alpha increased 1.2 deg by all combined lift-incrementation devices up to final geo alpha of -1.9 deg

For Reynolds number: rho= 0.7985, chord= 34.94, dynamic_viscosity =16.5794 millionths (divide by this to get Re min Meg), Re_meg= 1.6828 million at this element

element center at 85% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available

For lookup, alpha airflow= -1.9, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -1.4

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.1459
mid foil: Cd lookup: 0.0052
mid foil: Cm lookup: 0.0000

Mach of 0.1449 yields Prandtl-Glauert Cl multiplier of 1.0107 to take Cl to -0.1475
Mach of 0.1449 yields Prandtl-Glauert Cm multiplier of 1.0107 to take Cm to 0.0000
Mach of 0.1449 with a Cl-based drag divergence Mach number of 0.8353 takes the Cd to 0.0052
(increase of 0.0000)

final Cl -0.1475
final Cd 0.0052
final Cm 0.0000

adding control-effects cl of 0.1390 takes cl to -0.0085
adding control-effects cd of 0.0001 takes cd to 0.0054
adding control-effects cm of -0.0840 takes cm to -0.0840
finite-wing lift multiplier of 0.8680 takes cl to -0.0074
delta-wing moment addition of 0.0007 takes cm to -0.0834
e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0000
ground effect cdi multiplier of 1.0000 takes Cdi to 0.0000, which takes total Cd to 0.0054
cl multiplied by 1.0000 to get to -0.0074 to lower pressure from Mach 0.9 to 1.3.

lift = $-0.0074 * 0.1278 * \sqrt{46.9} * 0.7985 / 2 = -0.8 \text{ N}$
drag = $0.0054 * 0.1278 * \sqrt{46.9} * 0.7985 / 2 = 0.6 \text{ N}$
momt = $-0.0834 * 0.1278 * \sqrt{46.9} * 0.7985 / 2 * 0.75 = -7.0 \text{ N-m}$

Vert Stab 1:

local wind x = 3.2 m/s in aircraft coord system pos right.
local wind y = 4.3 m/s in aircraft coord system pos up.
local wind z = 58.4 m/s in aircraft coord system pos aft.

lift local wind path = -0.0 N pos up.
drag local wind path = 0.6 N pos aft.
side local wind path = 0.8 N pos right.

lift aircraft x y z = -0.0 -0.0 0.0 N
drag aircraft x y z = 0.0 0.0 0.6 N
side aircraft x y z = 0.8 -0.0 -0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil = -304.8 N, pos aft, L = 132.8 N-m, pos right
f_side = 0.4 N, pos right, M = -4054.9 N-m, pos up
f_nrml = 11675.3 N, pos up, N = -62.5 N-m, pos right

---Vert Stab 1 ELEMENT # 9-----

element x = -0.00 m, pos right from CG
element y = 1.66 m, pos up from CG
element z = 5.44 m, pos aft from CG

So final speed is vx = 3.2 vy = 4.3 vz = 58.4 m/s in the aircraft axis coordinate system

alpha of the wing plane : -3.1 (deg, if there is 0 incidence on the wing. this is aero alpha on the 0-incidence plane)
element incidence : 0.0 (deg)

incidence increase : 0.0 (deg)
geometric alpha : -3.1 (deg)
v for forces : 46.9

For Reynolds number: rho= 0.7985, chord= 29.81, dynamic_viscosity =16.5794 millionths (divide by this to get Re in Meg), Re_meg= 1.4356 million at this element

element center at 95% span: (wgt root foil= 0.0000, wgt mid foil= 1.0000, wgt tip foil= 0.0000)

mid foil: Airfoils/NACA 0009 (symmetrical).afl, I have Reynolds number 6.0000 million available
For lookup, alpha airflow= -3.1, alpha zero-lift= -0.0, finite wing alpha multiplier= 0.7359, slat alpha divisor= 1.0000, alpha lookup= -2.3

Using Reynolds number= 6.0000 million

mid foil: Cl lookup: -0.2405

mid foil: Cd lookup: 0.0054

mid foil: Cm lookup: 0.0000

Mach of 0.1449 yields Prandtl-Glauert Cl multiplier of 1.0107 to take Cl to -0.2431

Mach of 0.1449 yields Prandtl-Glauert Cm multiplier of 1.0107 to take Cm to 0.0000

Mach of 0.1449 with a Cl-based drag divergence Mach number of 0.8257 takes the Cd to 0.0054 (increase of 0.0000)

final Cl -0.2431

final Cd 0.0054

final Cm 0.0000

finite-wing lift multiplier of 0.8680 takes cl to -0.2110

delta-wing moment addition of 0.0192 takes cm to 0.0192

e of 0.9995, geometric aspect ratio of 3.0476, effective aspect ratio of 3.2270, gives Cdi of 0.0044

ground effect cdi multiplier of 1.0000 takes Cdi to 0.0044, which takes total Cd to 0.0098

cl multiplied by 1.0000 to get to -0.2110 to lower pressure from Mach 0.9 to 1.3.

lift=-0.2110 * 0.1090 * sqrt(46.9) * 0.7985 / 2 = -20.2 N

drag= 0.0098 * 0.1090 * sqrt(46.9) * 0.7985 / 2 = 0.9 N

momt= 0.0192 * 0.1090 * sqrt(46.9) * 0.7985 / 2 * 0.64= 1.2 N-m

Vert Stab 1:

local wind x= 3.2 m/s in aircraft coord system pos right.

local wind y= 4.3 m/s in aircraft coord system pos up.

local wind z= 58.4 m/s in aircraft coord system pos aft.

lift local wind path= -0.0 N pos up.

drag local wind path= 0.9 N pos aft.

side local wind path= 20.2 N pos right.

lift aircraft x y z= -0.0 -0.0 0.0 N

drag aircraft x y z= 0.1 0.1 0.9 N

side aircraft x y z= 20.2 -0.1 -1.1 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Vert Stab 1

f_axil= -304.9 N, pos aft, L= 166.4 N-m, pos right

f_side= 20.6 N, pos right, M= -4055.1 N-m, pos up

f_nrml= 11675.3 N, pos up, N= -173.6 N-m, pos right

Body 1 (fuselage):

Propwash from prop 1:

radial offset= 0.03 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 0.0000:
dz_in_wash= 1.59 m, radii off in z-direction= 1.5279 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.0000
wash AT the ring= -0.4, total rat= 0.0000, so total vz_wash_disc= -0.0

wash vx_disc= 2.7, vy_disc= -0.0, vz_disc= -0.0 m/s (disc coords)
wash vx_acft= 2.7, vy_acft= 0.0, vz_acft= -0.0 m/s (aircraft coords)
So total speed is vx= 6.0, vy= 4.4, vz= 58.4 m/s in (aircraft coords)

check-point is in the wake of Body 22 (serpra e) which has Cd=0.5

BODY SHADOWING with Cd of 0.5002 multiplies V by 0.7070

So speed is changed to vx= 4.2 vy= 3.1 vz= 41.3 m/s

So final speed is vx= 4.2 vy= 3.1 vz= 41.3 m/s in the aircraft axis coordinate system

body psi= 0.0 deg

body the= 0.0 deg

body phi= 0.0 deg

wind x= 4.2 m/s in body coordinate system

wind y= 3.1 m/s in body coordinate system

wind z= 41.3 m/s in body coordinate system

TOP area reference: lift slope slender ellipsoid body= 0.0040, lift slope MIT= 0.0031, lift slope use= 0.0036 deg, alph deg= 4.3, small angle rat= 0.9683, cl vert=0.0150

SIDE area reference: lift slope slender ellipsoid body= 0.0033, lift slope MIT= 0.0031, lift slope use= 0.0054 deg, beta deg= 5.8, small angle rat= 0.9683, cl side=0.0304

FRONT area reference: Cd body at zero-slip= 0.1500, Cd of cylinder at 90 slip= 4.5516, slip= 7.3 deg, Cd= 0.1511

SUBSONIC lift= 67.1 N

SUBSONIC drag= 165.0 N

SUBSONIC side= 166.0 N

body_Mach= 0.1286, but body thickness is 0.1708, and body_cl=0.0339, so the supersonic ratio= 0.0000, so:

SUBSONIC/SUPERSONIC INTERPOLATED lift force= 67.1 N in wind coordinate system (is Cl of 0.0150 based on top area)

SUBSONIC/SUPERSONIC INTERPOLATED drag force= 165.0 N in wind coordinate system (is Cd of 0.1511 based on frontal area), which is a Cd= 0.1511

SUBSONIC/SUPERSONIC INTERPOLATED side force= 166.0 N in wind coordinate system (is Cl of 0.0304 based on side area)

Now adding lift, side, and drag forces at the centroid:

local wind x= 4.2 m/s in aircraft coord system pos right.

local wind y= 3.1 m/s in aircraft coord system pos up.

local wind z= 41.3 m/s in aircraft coord system pos aft.

lift local wind path= 67.1 N pos up.

drag local wind path= 165.0 N pos aft.

side local wind path= 166.0 N pos right.

lift aircraft x y z= 0.0 66.9 -5.1 N

drag aircraft x y z= 16.8 12.4 163.7 N

side aircraft x y z= 165.2 -1.3 -16.8 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Now adding lift, side, and drag forces at the centroid

f_axil= -163.2 N, pos aft, L= 177.4 N-m, pos right
f_side= 202.6 N, pos right, M= -3968.2 N-m, pos up
f_nrml= 11753.4 N, pos up, N= 9.0 N-m, pos right

Body 2 (engine to body fairing):

Propwash from prop 1:

radial offset= 0.43 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 3.6670:
dz_in_wash= 1.09 m, radii off in z-direction= 1.0440 m, propwash_speed_mult of speed AT the
disc in z-direction= 1.7305
wash AT the ring= 3.6, total rat= 1.7305, so total vz_wash_disc= 6.3

wash vx_disc= -2.1, vy_disc= -0.0, vz_disc= 6.3 m/s (disc coords)
wash vx_acft= -2.1, vy_acft= 0.0, vz_acft= 6.3 m/s (aircraft coords)
So total speed is vx= 1.2, vy= 4.4, vz= 64.6 m/s in (aircraft coords)

check-point is buried inside of Body 1 (fuselage)

BODY BURYING multiplies V by 0.7071

So speed is changed to vx= 0.8 vy= 3.1 vz= 45.7 m/s

So final speed is vx= 0.8 vy= 3.1 vz= 45.7 m/s in the aircraft axis coordinate system

body psi= 0.0 deg
body the= 0.0 deg
body phi= 0.0 deg
wind x= 0.8 m/s in body coordinate system
wind y= 3.1 m/s in body coordinate system
wind z= 45.7 m/s in body coordinate system

TOP area reference: lift slope slender ellipsoid body= 0.0018, lift slope MIT= 0.0031, lift slope use= 0.0098 deg, alph deg= 3.9, small angle rat= 0.9900, cl vert=0.0380

SIDE area reference: lift slope slender ellipsoid body= 0.0031, lift slope MIT= 0.0031, lift slope use= 0.0031 deg, beta deg= 1.0, small angle rat= 0.9900, cl side=0.0032

FRONT area reference: Cd body at zero-slip= 0.2000, Cd of cylinder at 90 slip= 7.1493, slip= 4.1 deg, Cd= 0.2002

SUBSONIC lift= 37.7 N
SUBSONIC drag= 22.1 N
SUBSONIC side= 1.9 N

body_Mach= 0.1416, but body thickness is 0.1680, and body_cl=0.0382, so the supersonic ratio= 0.0000, so:

SUBSONIC/SUPERSONIC INTERPOLATED lift force= 37.7 N in wind coordinate system (is Cl of 0.0380 based on top area)
SUBSONIC/SUPERSONIC INTERPOLATED drag force= 22.1 N in wind coordinate system (is Cd of 0.2002 based on frontal area), which is a Cd= 0.2002
SUBSONIC/SUPERSONIC INTERPOLATED side force= 1.9 N in wind coordinate system (is Cl of 0.0032 based on side area)

Now adding lift, side, and drag forces at the centroid:

local wind x= 0.8 m/s in aircraft coord system pos right.

local wind y= 3.1 m/s in aircraft coord system pos up.
local wind z= 45.7 m/s in aircraft coord system pos aft.

lift local wind path= 37.7 N pos up.
drag local wind path= 22.1 N pos aft.
side local wind path= 1.9 N pos right.

lift aircraft x y z= 0.0 37.6 -2.6 N
drag aircraft x y z= 0.4 1.5 22.1 N
side aircraft x y z= 1.9 -0.0 -0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Now adding lift, side, and drag forces at the centroid
f_axil= -143.7 N, pos aft, L= 176.4 N-m, pos right
f_side= 204.9 N, pos right, M= -3911.6 N-m, pos up
f_nrml= 11792.5 N, pos up, N= 12.8 N-m, pos right

Body 22 (serpra e):

Propwash from prop 1:
radial offset= 0.00 m, prop radius= 1.04 m, radial wash_rat= 1.0000, float ring # 0.0000:
dz_in_wash= 0.32 m, radii off in z-direction= 0.3033 m, propwash_speed_mult of speed AT the
disc in z-direction= 0.6680
wash AT the ring= -0.4, total rat= 0.6680, so total vz_wash_disc= -0.3

wash vx_disc= 2.7, vy_disc= -0.0, vz_disc= -0.3 m/s (disc coords)
wash vx_acft= 2.7, vy_acft= 0.0, vz_acft= -0.3 m/s (aircraft coords)
So total speed is vx= 6.0, vy= 4.5, vz= 58.1 m/s in (aircraft coords)

check-point is buried inside of Body 1 (fuselage)
BODY BURYING multiplies V by 0.7071
So speed is changed to vx= 4.3 vy= 3.1 vz= 41.1 m/s
So final speed is vx= 4.3 vy= 3.1 vz= 41.1 m/s in the aircraft axis coordinate system

body psi= -0.0 deg
body the= -0.0 deg
body phi= -0.0 deg
wind x= 4.3 m/s in body coordinate system
wind y= 3.1 m/s in body coordinate system
wind z= 41.1 m/s in body coordinate system

TOP area reference: lift slope slender ellipsoid body= 0.0091, lift slope MIT= 0.0031, lift slope use=
0.0486 deg, alph deg= 4.4, small angle rat= 0.9675, cl vert=0.2062
SIDE area reference: lift slope slender ellipsoid body= 0.0489, lift slope MIT= 0.0031, lift slope use=
0.0260 deg, beta deg= 5.9, small angle rat= 0.9675, cl side=0.1487
FRONT area reference: Cd body at zero-slip= 0.5000, Cd of cylinder at 90 slip= 1.0774, slip= 7.4
deg, Cd= 0.5002
SUBSONIC lift= 66.9 N
SUBSONIC drag= 89.3 N
SUBSONIC side= 9.0 N

body_Mach= 0.1280, but body thickness is 1.0000, and body_cl=0.2542, so the supersonic ratio=

0.0000, so:

SUBSONIC/SUPERSONIC INTERPOLATED lift force= 66.9 N in wind coordinate system (is C_l of 0.2062 based on top area)
SUBSONIC/SUPERSONIC INTERPOLATED drag force= 89.3 N in wind coordinate system (is C_d of 0.5002 based on frontal area), which is a C_d = 0.5002
SUBSONIC/SUPERSONIC INTERPOLATED side force= 9.0 N in wind coordinate system (is C_l of 0.1487 based on side area)

Now adding lift, side, and drag forces at the centroid:

local wind x= 4.3 m/s in aircraft coord system pos right.
local wind y= 3.1 m/s in aircraft coord system pos up.
local wind z= 41.1 m/s in aircraft coord system pos aft.

lift local wind path= 66.9 N pos up.
drag local wind path= 89.3 N pos aft.
side local wind path= 9.0 N pos right.

lift aircraft x y z= 0.0 66.7 -5.1 N
drag aircraft x y z= 9.2 6.8 88.6 N
side aircraft x y z= 8.9 -0.1 -0.9 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

Now adding lift, side, and drag forces at the centroid

f_{axil} = -61.2 N, pos aft, L = 177.0 N-m, pos right
 f_{side} = 223.0 N, pos right, M = -3730.2 N-m, pos up
 f_{nrml} = 11866.0 N, pos up, N = 56.9 N-m, pos right

radiator drag:

local wind x= 3.3 m/s in aircraft coord system pos right.
local wind y= 4.4 m/s in aircraft coord system pos up.
local wind z= 63.0 m/s in aircraft coord system pos aft.

lift local wind path= 0.0 N pos up.
drag local wind path= 55.6 N pos aft.
side local wind path= 0.0 N pos right.

lift aircraft x y z= 0.0 0.0 -0.0 N
drag aircraft x y z= 2.9 3.9 55.4 N
side aircraft x y z= 0.0 -0.0 -0.0 N

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

radiator drag

f_{axil} = -5.8 N, pos aft, L = 177.1 N-m, pos right
 f_{side} = 225.9 N, pos right, M = -3717.8 N-m, pos up
 f_{nrml} = 11869.8 N, pos up, N = 64.8 N-m, pos right

.....
Landing gear tire contact with ground #:0
.....

.....
Landing gear tire contact with ground #:1
.....

.....
Landing gear tire contact with ground #:2
.....

---RUNNING AERODYNAMIC FORCE AND TOTAL MOMENT TOTALS, AIRCRAFT AXIS SYSTEM-----

After water skimming forces:

f_axial= -5.8 N, pos aft, L= 177.1 N-m, pos right
f_side= 225.9 N, pos right, M= -3717.8 N-m, pos up
f_nrml= 11869.8 N, pos up, N= 64.8 N-m, pos right

I am adding mass for the ! Adding 26.6 kg, at long= -4.53 lat= 0.00 vert= 0.03, and changing the moments to:

L= -0.2 N-m, pos right
M= -1070.9 N-m, pos up
N= -25.5 N-m, pos right

I am adding mass for the ! Adding 102.7 kg, at long= 4.53 lat= 0.00 vert= 0.03, and changing the moments to:

L= -0.8 N-m, pos right
M= 3062.5 N-m, pos up
N= 72.7 N-m, pos right

I am adding mass for the fuel! Adding 63.0 kg, at long= 0.58 lat= -2.44 vert= -0.23, and changing the moments to:

L= -1361.9 N-m, pos right
M= 3377.9 N-m, pos up
N= -11.6 N-m, pos right

I am adding mass for the fuel! Adding 64.0 kg, at long= 0.58 lat= 2.44 vert= -0.23, and changing the moments to:

L= 26.9 N-m, pos right
M= 3698.2 N-m, pos up
N= 89.7 N-m, pos right

---FINAL PROPULSIVE (ENGINE) FORCES, ALONG AIRCRAFT AXIS (pos aft/right/up)-----

f_side = 57.2 N, pos right
f_vertical= 73.7 N, pos up
f_axial = -799.5 N, pos aft
L = -260.6 N-m, pos right
M = 45.1 N-m, pos up
N = -81.6 N-m, pos right

---FINAL AERODYNAMIC FORCES, ALONG AIRCRAFT AXIS (pos aft/right/up)-----

f_side = 225.9 N, pos right

f_vertical= 11869.8 N, pos up
f_axial = -5.8 N, pos aft
L = 177.1 N-m, pos right
M = -3717.8 N-m, pos up
N = 64.8 N-m, pos right

---FINAL GEAR FORCES, ALONG AIRCRAFT AXIS (pos aft/right/up)-----

f_side = 0.0 N, pos right
f_vertical= 0.0 N, pos up
f_axial = 0.0 N, pos aft
L = 0.0 N-m, pos right
M = 0.0 N-m, pos up
N = 0.0 N-m, pos right

---FINAL MASS-OFFSET-FROM-CG FORCES, ALONG AIRCRAFT AXIS (pos aft/right/up)-----

L = 26.9 N-m, pos right
M = 3698.2 N-m, pos up
N = 89.7 N-m, pos right

---FINAL FORCES BEFORE ASSYMETRIC LOADING EFFECTS, ALONG AIRCRAFT AXIS (pos aft/right/up)-----

f_side = 283.1 N, pos right
f_vertical= 11943.6 N, pos up
f_axial = -805.3 N, pos aft
L = -56.5 N-m, pos right
M = 25.5 N-m, pos up
N = 72.9 N-m, pos right

---FINAL FORCES AFTER ASSYMETRIC LOADING EFFECTS, ALONG AIRCRAFT AXIS (pos aft/right/up)-----

f_side = 283.1 N, pos right
f_vertical= 11943.6 N, pos up
f_axial = -805.3 N, pos aft
L = -56.5 N-m, pos right
M = 25.5 N-m, pos up
N = 72.9 N-m, pos right

---FINAL G-LOADS ON AIRCRAFT AXIS-----

mass= 1344.9 kg (ice accretion will change mass and MI if present!)
g_side= 0.0215 G
g_nrml= 0.9064 G
g_axil=-0.0611 G

---FINAL TOTAL LINEAR ACCELS, ALONG GLOBAL AXIS (pos south/east/up)-----

a_x= 0.1 m/ss

a_y= -0.9 m/ss
a_z= 0.2 m/ss

---FINAL TOTAL ANGULAR ACCELS, ABOUT AIRCRAFT AXIS (pos right/up/right)-----

roll MI = 2169.8 kg-m-m... M/mi=accel in RADIANS per sec
pitch MI = 5727.6 kg-m-m... M/mi=accel in RADIANS per sec
yaw MI = 7601.6 kg-m-m... M/mi=accel in RADIANS per sec
Pdot (roll) = -0.0 deg/ss
Qdot (pitch)= 0.0 deg/ss
Rdot (yaw) = 0.0 deg/ss

---FINAL TOTAL LINEAR VELOCITIES, ALONG GLOBAL AXIS (pos south/east/up)-----

v_x= -55.5 m/s
v_y= -0.3 m/s
v_z= 18.7 m/s

---FINAL TOTAL ANGULAR VELOCITIES, ABOUT AIRCRAFT AXIS (pos right/up/right)-----

Prad= -0.0 rad/s
Qrad= -0.0 rad/s
Rrad= -0.0 rad/s

---FINAL AERODYNAMIC FORCES, ALONG FLIGHT PATH AXIS (pos aft/right/up)-----

f_side= 225.8 N, pos right
f_lift= 11836.4 N, pos up
f_drag= 903.0 N, pos aft