				WING				HORIZONTAL TAIL														
CONVENTIONAL SPECS			Calculate C_root and tip		MAC	Ybar		Tail Arm	Horizontal Tail area		Horizontal Tail root chord n tip		MAC	Ybar								
Span: 124	Tape 0.35		32.99165673		23.9902294	26.02469136		60	Vh=1			19.67006893	14.30329704	11.80204136								
Sref:2076	2066.7		11.54707986						S_H====	793.2897137		6.884524125										
t/c:0.108									AR:4.5		AR:b^2/S											
AR 7.5	Sweep 35								Taper:0.35	HT_Span====	59.74783437											
FUEL WEIGHT	78900								t/c:0.08													
Density [lb/gal]	6.7				2.856401384				sweep	36.5												
D=Mass/Vol					0.350090854			VERTICAL TAIL														
VOLUME==[gal]	11776.1194								V_v : 1.15			Root Chord and tip values		MAC	Ybar							
cubic FEET===	1574.243418								S_V	493.396		21.95067268		17.48972953	10.53632289							
									AR	1.6		12.07286998										
Fuel Tank Sizing	Assume constant thickness throughout the wing								Taper	0.6												
Thick 1	t/c * Croot	3.563098927							sweep	36.5												
Thick 2	t/c * Ctip	1.247084625							b	28.09686103												
Length 0.85 to 0.15		23.09415971					46.5															
WIdth depends on iteration, finding best value such that the volumes match and areas																						
Width for now:::::	28.28																					
									4.8													
						48 initially																
Advanced Tech Spec			Croot	Ctip		Tail arm	Horizontal Tail area			Horizontal Tail root chord n tip		MAC	Ybar		VERTICAL TAIL							
span 122			28.40227088	8.520681265		60	Vh=.86			11.7429461	4.110031134	8.539006482	7.486128136			V_v : 0.111			Root chord&tip		mac	y bar
sref 1800							S_H====	282.7314979								S_V	399.6		19.75435079		16.13271981	9.482088378
ar = 8.5			Cmac	Ybar			AR:4								Values from text	AR	1.6		11.85261047			
Tape 0.3			20.2457213	25.02564103			Taper:0.35	HT_Span====	35.66919877							Taper	0.6					
1751.058824							t/c:0.08									sweep	36.5					
t/c=0.11							sweep	36.5								b	25.28556901					
sweep 35							sweep	36.5														
Fuel weight	66470																					

Conventional CG Calculations

	Weights		
	ToGW	296,000	
	Wing	37000	
	Fuselage	60236	
	landing gear	11840	
	Nacelle+Pylon	8311.68	
	Tails	20720	
	POWER_P	30784	
	Fixed Equip	10360	
	Fuel weight	71905	85313.12
	Passenger:210	37800	
	Cargo	8000	
	Total	296956.68	

12150

Full

Design	Weight	Length	Moment
Wing	37000	176	6512000
Fuselage	60236	156.8	9445004.8
landing gear	11840	176	2083840
Nacelle+Pylon	8311.68	163.9	1362284.352
Tails	20720	236	4889920
POWER_P	30784	163.9	5045497.6
Fixed Equip	10360	176	1823360
Fuel weight	71905	173	12439565
Passenger:210	37800	156.8	5927040
Cargo	8000	180	1440000
	296956.68	1758.4	50968511.75
delta 1/4 MAC	4.363814709		

	274.4	
Nacelle Diam	93.5	7.791666667
Nacelle length	241.45	20.12083333

CG design
171.6361853

Empty	Weight	Length	Moment
Wing	37000	176	6512000
Fuselage	60236	156.8	9445004.8
landing gear	11840	176	2083840
Nacelle+Pylon	8311.68	159.9	1329037.632
Tails	20720	236	4889920
POWER_P	30784	159.9	4922361.6
Fixed Equip	10360	176	1823360
Fuel weight	0	173	0
Passenger:210	0	156.8	0
Cargo	0	180	0
Sum	179251.68		31005524.03
	CG	172.9720136	

Upper And lower Bounds for CG		For	158.4
		AFT	193.6

Ferry	Weight	Length	Moment
Wing	37000	176	6512000
Fuselage	60236	156.8	9445004.8
landing gear	11840	176	2083840
Nacelle+Pylon	8311.68	159.9	1329037.632
Tails	20720	236	4889920
POWER_P	30784	159.9	4922361.6
Fixed Equip	10360	176	1823360
Fuel weight	71905	173	12439565
Passenger:210	0	156.8	0
Cargo	0	180	
Sum	251156.68		43445089.03
	CG	172.980026	

Landing	W	L	Moment
Wing	37000	176	6512000
Fuselage	60236	156.8	9445004.8
landing gear	11840	176	2083840
Nacelle+Pylon	8311.68	159.9	1329037.632
Tails	20720	236	4889920
POWER_P	30784	159.9	4922361.6
Fixed Equip	10360	176	1823360
Fuel weight	0	173	0
Passenger:210	37800	156.8	5927040
Cargo	8000	180	1440000
Sum	225051.68		38372564.03
	CG	170.5055658	

ADVANCED TECH CG calc

Section	Weights	
TOGW	263,650	274500
Wing	30319.75	
Fuselage	53520.95	
landing gear	10546	
Nacelle+Pylon	7403.292	
Tails	23728.5	
POWER_P	27705	
Fixed Equip	9227.75	
Fuel weight	66250	
Passenger:210	37800	
Cargo	8000	
TOTAL	274501.242	

Full	Weight	Moment arm	Moment	CG
Wing	30319.75	172	5214997	170.2991501
Fuselage	53520.95	156.8	8392084.96	
landing gear	10546	186	1961556	
Nacelle+Pylon	7403.292	155	1147510.26	
Tails	23728.5	232	5505012	
POWER_P	27705	155	4294275	
Fixed Equip	9227.75	172	1587173	
Fuel weight	66250	168	11130000	
Passenger:210	37800	162.4	6138720	
Cargo	8000	172	1376000	
	274501.242	sum	46747328.22	
	CG			

CG tolerance 10% fore and Aft A/C CG taken at .25Chord MAC

Fore	154.8
Aft	189.2

Nacelle Diam	96.8	8.066666667
Length	239.36	19.94666667

Empty	Weight	Length	Moment	CG
Wing	30319.75	170	5154357.5	172.5361919
Fuselage	53520.95	156.8	8392084.96	
landing gear	10546	186	1961556	
Nacelle+Pylon	7403.292	156.5	1158615.198	
Tails	23728.5	230	5457555	
POWER_P	27705	156.5	4335832.5	
Fixed Equip	9227.75	170	1568717.5	
Fuel weight	0	166	0	
Passenger:210	0	162.4	0	
Cargo	0	170	0	
	162451.242		28028718.66	

Ferry	Weight	Moment arm	Moment	CG
Wing	30319.75	170	5154357.5	170.6427928
Fuselage	53520.95	156.8	8392084.96	
landing gear	10546	186	1961556	
Nacelle+Pylon	7403.292	156.5	1158615.198	
Tails	23728.5	230	5457555	
POWER_P	27705	156.5	4335832.5	
Fixed Equip	9227.75	170	1568717.5	
Fuel weight	66250	166	10997500	
Passenger:210	0	162.4	0	
Cargo	0	170	0	
	228701.242		39026218.66	

LANDING	Weight	Moment arm	Moment	CG	170.5989281
Wing	30319.75	170	5154357.5		
Fuselage	53520.95	156.8	8392084.96		
landing gear	10546	186	1961556		
Nacelle+Pylon	7403.292	156.5	1158615.198		
Tails	23728.5	230	5457555		
POWER_P	27705	156.5	4335832.5		
Fixed Equip	9227.75	170	1568717.5		
Fuel weight	0	166	0		
Passenger:210	37800	162.4	6138720		
Cargo	8000	170	1360000		
	208251.242		35527438.66		

MATLAB CODE
STEVEN MURILLO

```matlab
%% MAE Midsizing Report CODE
clear
clc
%close all
%%%%NAME AND COMMENT EACH OF THESE TO debug
load('DeltaMdivCritical'); %  critical Mach number variations - digitized plots
load('MdivCO.mat'); %  Mach number division interpolation function
load('CritDeltaMdivV3.mat'); %  updated critical Mach number division data
load('tc_curve.mat'); %  thickness-to-chord ratio data for various airfoils
load('tcConv.mat'); %  thickness-to-chord ratios for conventional airfoils
load('tcCrti.mat'); %  thickness-to-chord ratios for supercritical airfoils
load('tcCrit.mat'); %  additional supercritical airfoil data
load('tcCritt.mat'); %  more supercritical airfoil data
load('ClMaxLand.mat'); %  maximum lift coefficient data for landing
configurations
load('ClMTakeoff.mat'); %  maximum lift coefficient data for takeoff
configurations
load('CLMaxClean.mat'); %  maximum clean configuration lift coefficients
load("ClClean0Sweep.mat"); %  clean configuration lift coefficient for 0-degree
sweep wings
load('ClClean35Sweep.mat'); %  clean configuration lift coefficient for
35-degree sweep wings
load('ClCleanSweep15.mat'); %  clean configuration lift coefficient for
15-degree sweep wings
load('FuelFractionJT8D.mat'); %  fuel fraction data for JT8D engine
configurations
load('Thrust.mat'); %  engine thrust capabilities data
load('MaxThrustSL.mat'); %  maximum sea level thrust
load('MaxDryThrustSealevel'); %  maximum dry thrust data at sea level
load('SLTHRUST.mat'); %  sea level thrust
load('KFactorNonWing.mat'); %  non-wing aerodynamic factors for drag
calculation
load('Cfcoef.mat'); %  skin friction coefficient
load('CRUISE.mat'); %  cruise condition parameters
load("CDPChangeLand.mat"); %  parasitic drag coefficient upon landing
load("CDChange.mat"); %  drag coefficient change data
load('TakeOffCdpdelta.mat'); %  change in drag coefficient during takeoff
%% SPECS/Variation
wingtype=0; % 0 for critical wingtype, 1 for conventional
AdvanceEngines=1;
AluminummMat=0;
CompositeMat=1;
if CompositeMat==1
  WCM=.7; % Weight correction factor for wings
  FCM=.85; % Fuselage correction factor
  FECM=.9; % Fixed equipment correction factor
  NacelePylonCM=.8; % Nacelle and pylon correction
```

```
  c=1.2; % General correction coefficient
else
  WCM=1;
  FCM=1;
  FECM=1;
  NacelePylonCM=1;
end
if AluminummMat==1
  wingAL=.94; % Correction factor for wing weight aluminumm
  FAL=.954; % Correction factor for fuselage weight
  c=1.356; %
else
  wingAL=1;
  FAL=1;
end
if AdvanceEngines==0
  ATSFC=1; % Standard thrust-specific fuel consumption
  ATENGINE=1;
else
  ATSFC=.9; % Reduced fuel consumption for advanced engines
  ATENGINE=1.1; % Increased performance for advanced engines
end
M=.80; % Cruise Mach number
Range=3500; % Range in nautical miles
TOFL=6900; % Takeoff field length in feet
V=140; % Stall speed in knots
pax=210; % Number of passengers
NA=7; % Number of attendants
NAS=2; % Number of auxiliary staff
Wcargo=8000; % Weight of cargo in pounds
hcl=35000; % Initial cruise altitude in feet
q=1; % Indicates wing-mounted engines
xyz=0.45; % Fraction of fuel used before landing
Z=2; % Number of engines
Cl=.54; % Initial guess for lift coefficient
sweep=35; % Wing sweep angle in degrees
AR=8; % Aspect ratio
TR=.35; % Taper ratio
rho=.0007382; % Air density at cruise height
rhosl=.0023769; % Air density at sea level
sigma=rho/rhosl; % Density ratio SL
sigmaLand=.953; % Density ratio at landing
Talt=394.08; % Temperature at altitude
RQ=1717; % Universal gas constant for air
gam=1.4; % Heat capacity ratio
p=499.34; % Pressure at altitude
pr=p/2116.2; % Pressure ratio
Wfraction=0.32; %Typical Fuel Fraction for a commercial airliner of these specs
if wingtype==1
```

```
  jq=4; % Correction factor based on wing type
  ii=7; % Iteration parameter based on wing type
else
  jq=6;
  ii=6;
end
for j=1:4
  sweep=15+5*j;
  sweepcheck(1,j)=sweep;
end
for i=1:ii
  AR=jq+i;
  ARgraph(j,i)=AR;
  adjust=-.018;
  aj=.3;
   if sweep==35
      aj=.8;
   end
end
while 1
while 1
while 1

if wingtype==1
MdivDelta=MdivCO(Cl)*1.02;
elseif wingtype==0
MdivDelta=CritDeltaMdivV3(Cl);
end
Mdiv=(M+.004)-MdivDelta;
if wingtype==1 %%%%%finding tc loop
tc=tovercConv(Mdiv, sweep);
   if sweep==20
    tc=(((sweep-20)/(25-20)*(-17.0631*Mdiv^3 + 43.9452*Mdiv^2 - 38.1861*Mdiv +
...
           11.2626-(-14.7915*Mdiv^3 + 37.4607*Mdiv^2 - 32.0814*Mdiv +
9.3546))+(-14.7915*Mdiv^3 +37.4607*Mdiv^2 - 32.0814*Mdiv + 9.3546)))*.93;
   elseif sweep==25
    tc=((sweep-25)/(30-25)*(-16.4307*Mdiv^3 + 44.1728*Mdiv^2 - 40.0666*Mdiv
+12.3264- ...
           (-18.2575*Mdiv^3 + 47.0214*Mdiv^2 - 40.8591*Mdiv +
12.0510))+(-18.2575*Mdiv^3 + 47.0214*Mdiv^2 - 40.8591*Mdiv + 12.0510))*.9;
   end
% Define sweep range adjustments
SWEEP_ADJUSTMENT_FACTOR = 0.95;
sweep_ranges = [20, 25; 25, 30; 30, 35]; % Define ranges in a matrix
correction_factors = [-12.8960, 32.6167, -27.9454, 8.1774; -15.4403, 39.6188,
-34.3375, 10.1208];
% Calculate thickness to chord ratio based on sweep range
if wingtype == 0
```

```
    for idx = 1:size(sweep_ranges, 1)
        if sweep >= sweep_ranges(idx, 1) && sweep < sweep_ranges(idx, 2)
            range_factor = (sweep_ranges(idx, 2) - sweep) / (sweep_ranges(idx,
2) - sweep_ranges(idx, 1));
            tc = sum(correction_factors(idx, :) .* [Mdiv^3, Mdiv^2, Mdiv, 1]) *
(1 - range_factor) * SWEEP_ADJUSTMENT_FACTOR;
            break;
        end
    end
end
weightLoadng=(cos(sweep*pi/180))^2*tc^2*AR; % Calc wing loading
CLmaxT=ClMTakeoff(weightLoadng); % Max takeoff lift coef.
CLmaxL=ClMLand(weightLoadng); % Max landing lift coef.
WSL=(V/1.3)^2*sigmaLand*CLmaxL/296; % Stall wing loading
Vcruise=M*sqrt(RQ*Talt*gam)*.5924; % Compute cruise speed
RangeAO=Range+200+.75*Vcruise; % Adjust nominal range
WSto=WSL/(1-xyz*Wfration); % Takeoff wing loading
WIC=.965*WSto; % Weight index coeff.
Clfinal=WIC/(1481*pr*M^2); % Final lift coef.
if abs(Clfinal-Cl)<.001
    break
else
    if Cl>Clfinal
        Cl=Cl-.0001;
    else
        Cl=Cl+.0001;
    end
end
end
if Z==2
  TOFLVAR=0.0285*TOFL - 10.893;
elseif Z==3
  TOFLVAR=0.0316*TOFL - 8.8235;
elseif Z==4
  TOFLVAR=0.0327*TOFL - 0.8809;
end
WeightoverT70V=TOFLVAR/WSto*(sigmaLand*CLmaxT);
Vlo=1.2*(296*WSto/sigmaLand/CLmaxT)^.5;
Mlo=Vlo/661/sqrt(sigmaLand);
Mlo7Vl=.7*Mlo;
y=[45500 39120 34820 317501];
xq=[0 .15 .30 .45]; %AR correction
TMlo7Vl=interp1(xq,y,Mlo7Vl);
TSLST=interp1(xq,y,0);
WoverTy=WeightoverT70V*TMlo7Vl/TSLST-aj;
%% WEIGHT CALCULATIONS
% Define load factor for design safety and structural integrity considerations
n=1.5*2.5; % load factor
% Determine weight correction factors based on engine configuration
```

```
if q==1
   Kw=1.0; % Weight factor for wing-mounted engines
   Kts=.17; % Tail size factor for wing-mounted engines
end
%  fuselage weight based on passenger numbers and attendants
lfus=(3.76*pax/NA+33.2); % Length of fuselage adjusted by passenger count
dfus=(1.75*NA+1.58*NAS+1); % Fuselage diameter adjustment based on crew and
service staff
% thickness chord ratio
tca=tc+.03; % Adjusted thickness-to-chord ratio for airfoil
% Wing weight calculation
Ww=.00945*(AR)^.8*(1+TR)^.25/(tca^.4*cosd(sweep)*WSto^.695)*Kw*n^.5;
% Fuselage weight calculation incorporating fuselage length and diameter
Wfus=.6727*11.5*lfus^.6*dfus^.72*n^.3;
% Weight of landing gear assumed constant
WLG=.040; % Constant landing gear weight factor
% Nacelle and pylon weight calculated based on total weight over thrust ratio
Wnp=.0555/WoverTy*NacelePylonCM; % Weight of nacelles and pylons adjusted by
thrust factor and material
% Tail weight determined by tail size factor and wing weight
Wtail=(Kts)*Ww;
if roff==1
   Wtail=(Kts+.08/3)*Ww; % Additional tail weight adjustment for profile 1
end
% Weight of propulsion for using advanced specific fuel consumption factor
Wpp=1/(3.58*WoverTy)*ATSFC;
% Fuel weight calculation w/ fuel fraction
Wfuel=1.0275*Wfration;
% Payload weight sum of passenger and cargo weights
Wpayload=215*pax+Wcargo;
% Fixed equipment weight considering passenger capacity, number of engines, and
equipment factor
Wfe=(132*pax+300*Z*ATENGINE+260*2+170*ceil(pax/50))*FECM;
% Total weight equation to be solved using iterative root finding method
WEIGHTequation = @(wto) (Ww+Wtail)*WCM*wingAL * wto.^1.195 + (Wfus)*FCM*FAL *
wto.^0.235+(WLG+Wnp+Wpp+Wfuel+.035*FECM-1)* wto + Wpayload+Wfe;
% Initial guess for the total weight optimization
initialGuess = 350000;
% Solve the total weight equation using fzero to find optimal weight
wo = fzero(WEIGHTequation, initialGuess);
% wing area based on optimized weight
St=wo/WSto;
Bw=(AR*St)^.5;
mac=St/Bw;
% Thrust requried
Thrust=wo/WoverTy;
% Calculate thrust per engine
ThrustE=Thrust/Z;
%% DRAG
```

```
RnL=(.5*994.85)/.00034884;
maco=mac*1.3;
%%Wings CFI== 0.0826*(Rn/L*L)^-0.196
Mo=.5;
Zm=(2-Mo^2)*cosd(sweep)/sqrt(1-Mo^2*(cosd(sweep)^2));
Kwing=1+Zm*tc+100*tc^4;
CfiW= 0.0826*(RnL*mac)^-0.196;
SwetWing=2*(St-dfus*maco)*1.02; %%fix correction factor to
Fwing=Kwing*CfiW*SwetWing;
%%Fuseluge
Swetfus=.9*pi*lfus*dfus;
cfifus=0.0826*(RnL*lfus)^-0.196;
FFfus=lfus/dfus;
Kfus=KFactorNonWing(FFfus);
Ffus=cfifus*Swetfus*Kfus;
%%Tail
Ftail=.38*Fwing;
%Nacelles
SwetNAC=2.1*(ThrustE)^.5*Z;
cfinac= 0.0826*(RnL*mac)^-0.196;
Knac=1.25;
Fnac=Knac*cfinac*SwetNAC;
%Pylons
Fpylon=.20*Fnac;
Ftotalsum=(Fwing+Ffus+Ftail+Fnac+Fpylon)*1.06;
Cdop=Ftotalsum/St;
eez=1/(1.035+.38*Cdop*pi*AR);
%% CLIMB
% air density ratio for climb phase
sigma=.53317;
% average weight
Wcl=(1+.965)/2*wo;
% Compute the climb speed (VcCl) using the drag (Ftotal) and Oswald efficiency
number (e)
VcCl=1.3*12.9/(Ftotalsum*eez)^.25*(Wcl/(sigma*Bw))^0.5;
% Mach number during climb
Mcll=VcCl/614.3464;  % Converts climb speed to Mach number assuming standard
sea level speed of sound= knots
% Required thrust for climb (Trcl) based onspeed, weight, and drag
Trcl=sigma*Ftotalsum*VcCl^2/296+94.1*(Wcl/Bw)^2/(sigma*eez*VcCl^2);
TaG=(-9993.8*Mcll^3 + 23895*Mcll^2 - 25308*Mcll + 27030 -16043*Mcll^4 +
35397*Mcll^3 - 24488*Mcll^2 + 3521.4*Mcll + 15536)/2; F% Combined equation for
available thrust
% Calculate specific fuel consumption (SFC) for the climb phase, averaged for
different Mach numbers
SFC=(0.3664*Mcll+0.344+0.4238*Mcll+0.3235)/2; % Average SFC acrossthe bored
% Calculate available thrust (Ta) adjusted by actual thrust and available
gradient
Ta=Thrust/45500*TaG;  % Normalize thrust factor
```

```
RC=101*(Z*Ta-Trcl)*VcCl/Wcl;  % RATE OF CLIMB Factor 101 converts to
appropriate units, Z is number of engines
%time required to reach cruise altitude (hcl)
TimeCL=hcl/RC;  % hcl is cruise altitude in feet
% Convert climb time to range covered during climb
RangeCL=VcCl*TimeCL/60;  % Convert time to hours and multiply by climb speed
% fuel used during climb
WfCL=Z*Ta*SFC*TimeCL/60;  % Calculate total fuel consumption based on time,
thrust, and SFC
%% Total Range
wo=wo-WfCL;
w1=(1-Wfration)*wo;
CLavg=(wo+w1)/(2*St)/(1481*.2360*M^2);
Cdi=CLavg^2/(pi*AR*eez);
CDR=Cdi+Cdop+.0010;
LoD=CLavg/CDR;
TrR=(wo+w1)/2/LoD;
TrJ9=TrR*45500/ThrustE/Z;
SFCR=CatCruiseH(TrJ9)*ATSFC;
Rcruise=Vcruise/SFCR*LoD*log(wo/w1);
R=RangeCL+Rcruise;
if abs(R-RangeAO)>30 %%ADJUSTMENTS TO MAKE GRAPHS WORK<
  if R>RangeAO
      adjust=adjust-.0005;
  elseif RangeAO>R
      adjust=adjust+.0005;
  end
else
end
end
% Thrust Check
CLIC=wo/St/1481/.2360/M^2;
CdiC=CLIC^2/pi/8/eez;
CDC=CdiC+Cdop+.0010;
LoDC=CLIC/CDC;
TreqC=Wcl/LoDC/Z;
TreqCJ9=TreqC*45500/ThrustE;
% 1 Climb Gradient
CL1=CLmaxT/(1.2)^2;
CLtoOCLmaxT1=1/1.2^2;
deltaCDo1=TakeOffCdpdelta(CLtoOCLmaxT1);
CD1=2*Cdop+deltaCDo1+CL1^2/(pi*AR*eez);
LoD1=CL1/CD1;
Treq1=wo/LoD1;
Ta1eng=ThrustE/45500*ThrustSL(Mlo);
Grad1=(((Z-1)*Ta1eng-Treq1)/wo)*100;
a=0;
%Second Gradient
CD2=CD1-Cdop;
```

```
LoD2=CL1/CD2;
Treq2=wo/LoD2;
Grad2=((Z-1)*Ta1eng-Treq2)/wo*100;
b=2.4;
%3rd Gradient
% if sweep>=0 &&sweep<15
%
CLClean=ClClean0Sweep(tc)*((15-sweep)/(15-0))+ClCleanSweep15(tc)*(1-(15-sweep)/
(15-0));
% elseif sweep>=15 &&sweep<35
%
CLClean=ClCleanSweep15(tc)*((35-sweep)/(35-15))+ClClean35Sweep(tc)*(1-(35-sweep
)/(35-15));
% elseif sweep==35
% CLClean=ClClean35Sweep(tc);
% end
CLClean=CLMaxClean(tc,sweep);
V3=1.2*(296*WSto/.925/CLClean)^.5;
M3=V3/659;
CL3=CLClean/1.2^2;
CD3=Cdop+CL3^2/pi/AR/eez;
LoD3=CL3/CD3;
Treq3=wo/LoD3;
Taeng3=ThrustE/45500*MaxThrustSL(M3);
Grad3=((Z-1)*Taeng3-Treq3)/wo*100;
% Approach Gradient
CL4=CLmaxT/1.3^2;
ClAp_Clmax=1/1.3^2;
deltaCDp4=TakeOffCdpdelta(ClAp_Clmax);
CD4=deltaCDp4+Cdop+CL4^2/pi/AR/eez;
LoD4=CL4/CD4;
Treq4=Wlanding4/LoD4;
V4=(296*WSL/(.953*CL4))^.5;
M4=V4*1.6878/sqrt(gam*1717*543.67);
Ta4=ThrustE/45500*MaxThrustSL(M4);
Grad4=((Z-1)*Ta4-Treq4)/Wlanding4*100;
d=2.1;
%Landing Grad
CL5=CLmaxL/1.3^2;
CL_CLmax5=1/1.3^2;
deltaCDp5=CDPChangeLandingN(CL_CLmax5);
CD5=deltaCDp5+2*Cdop+CL5^2/(pi*AR*eez);
LoD5=CL5/CD5;
Treq5=Wlanding4/LoD5;
V5=140;
M5=V5*1.6878/sqrt(gam*1717*543.67);
Ta5=ThrustE/45500*ThrustSL(M5);
Grad5=((Z)*Ta5-Treq5)/Wlanding4*100;
eez=3.2;
```

```
c=1.2;
%% Direct Operating COSTS
  D=Range*1.15;
  Tgm=.25;
  Tcl=.18;
  Td=0;
  Tam=.10;
  Tcr=(D*1.02+20-RangeCL*1.15)/Vcruise/1.15;
  TB=(Tgm+Tcl+Td+Tcr+Tam);
  VB=D/TB;
  Fcl=WfCL;
  Fcr_Fam=TrR*SFCR*(Tcr+Tam);
  FB=Fcl+Fcr_Fam;
%Flight Crew
  P=Wpayload/2000;
  dollarbhour=17.849*(Vcruise*1.15*wo/10^5)^.3+40.83;
  Ctma=dollarbhour/(VB*P);
%Fuel and Oil
  CF=.28*(1/6.4);
  Cott=2.15;
  Ctmb=(1.02*FB*CF+Z*Cott*TB*.135)/(D*P);
%Hull Insurance
  Wa=wo*(1-Wfration)-Wpayload-Wpp*wo;
  Ca=2.4E6+87.5*Wa;
  Ce=590000+16*ThrustE;
  CT=Ca+Z*Ce;
  IRa=.01;
  U=630+4000/(1+1/(TB+.5));
  Ctmc=IRa*CT/(U*VB*P);
 %Maintence
  Kfha=4.9169*log10(Wa/10^3)-6.425;
  Kfca=.21257*(log10(Wa/10^3))^3.7375;
  Tf=TB-Tgm;
  Rl=8.60;
  Ctmd=(Kfha*Tf+Kfca)*Rl/(VB*TB*P);
 %Airframe Mat
  Cfha=1.5994*Ca/10^6+3.4263;
  Cfca=1.9229*Ca/10^6+2.2504;
  Ctme=(Cfha*Tf+Cfca)/(VB*TB*P);
 %Labor
  Kfhe=Z*(ThrustE/10^3)/(.82715*(ThrustE/10^3)+13.639);
  Kfce=.20*Z;
  Ctmf=(Kfhe*Tf+Kfce)*Rl/(VB*TB*P);
 %Engine mats
  Cfhe=(28.2352*Ce/10^6-6.5716)*Z;
  Cfce=(3.6698*Ce/10^6+1.3685)*Z;
  Ctmg=(Cfhe*Tf+Cfce)/(VB*TB*P)*ATENGINE;
 % Maintence sum
  CtmT=(Ctmg+Ctmf+Ctme+Ctmd)*2;
```

```
  %%Depreciation
  CtmD=(CT+.06*(CT-Z*Ce)+.3*Z*Ce)/(14*U*VB*P);
  %%%Total DOC
  CtmTot=(CtmT+Ctma+Ctmb+Ctmc+CtmD)*P/pax;
  DOC1(j,i)=CtmTot;
  MaxW(j,i)=wo;
if wingtype==1
  if sweep==25 && AR==8
      finalWeight=wo;
      LCFinal=LoD;
      FinalVelocity=Vcruise;
      Cruise_SFC=SFCR;
      Final_FuelFraction=Wfration;
      Final_PayloadWeight=61750;
      OEW=wo-Wpayload;
  end
elseif wingtype==0
      if sweep==25 && AR==9
      finalWeight=wo;
      LCFinal=LoD;
      FinalVelocity=Vcruise;
      Cruise_SFC=SFCR;
      Final_FuelFraction=Wfration;
      Final_PayloadWeight=61750;
      OEW=wo-Wpayload;
      end
end
end
end
%% Graphs
% Constants for display
SWEEP_INCREMENT = 5;
BASE_SWEEP = 15;
NUMBER_OF_SWEEPS = 4;
% Plotting Direct Operating Costs
figure;
hold on;
for sweepIndex = 1:NUMBER_OF_SWEEPS
   currentSweep = BASE_SWEEP + SWEEP_INCREMENT * sweepIndex;
   sweepLabel = ['Sweep Angle = ', num2str(currentSweep), ' degrees'];
   plot(ARgraph(sweepIndex, :), DOC1(sweepIndex, :), 'DisplayName',
sweepLabel);
end
xlabel('Aspect Ratio (AR)');
ylabel('Direct Operating Cost ($/ton mile)');
airplaneType = 'Traditional Airplane';
if wingtype == 0 && AdvanceEngines == 1 && CompositeMat == 1
   airplaneType = 'Advanced Airplane';
end
```

```
title(airplaneType);
legend show;
hold off;
% Plotting Take-Off Weight
figure;
hold on;
for sweepIndex = 1:NUMBER_OF_SWEEPS
   currentSweep = BASE_SWEEP + SWEEP_INCREMENT * sweepIndex;
   sweepLabel = ['Sweep Angle = ', num2str(currentSweep), ' degrees'];
   plot(ARgraph(sweepIndex, :), MaxW(sweepIndex, :), 'DisplayName',
sweepLabel);
end
xlabel('Aspect Ratio (AR)');
ylabel('Take-Off Weight (lbs)');
title(airplaneType);
legend show;
hold off;
% Displaying configuration and results
configurationInfo = ['Using correction factor x=', num2str(xyz), ...
                     '. Take Off Weight= ', num2str(wo), ' lbs.'];
disp(configurationInfo);
engineType = 'Regular Engines Used';
if AdvanceEngines == 1
   engineType = 'Advanced Engines Used';
end
disp(engineType);
```

MAE 164 HW2

$M = 0.8$ @ 35,000 ft $R = 3500$ mi

Conventional Airfoil

$\Lambda = 35°$ $AR = 8$

Assume $C_L = 0.5$ Fig 2 $\rightarrow \Delta M_{Div} = 0.016$

$M_{div} = M + .004 - .016 = .806$

Figure 1a)

$\downarrow$

$t/c = .108$

$\cos^2(35)(.108)^2(8) = .0626$

$C_{L_{max_{TO}}} = 1.86$ $C_{L_{max_{Land}}} = 2.78$

$$W/S_{Land} = \left(\frac{V_{AP}}{1.3}\right)^2 \frac{\sigma C_{L_{max}}}{296} = \left(\frac{140}{1.3}\right)^2 \frac{.953 \cdot 2.78}{296} = 104$$

$V_{cruise} = 0.82 \cdot 576.4 = 473$

$R_{a/o} = 3500 + 200 + .75 \cdot 473 = 4054.8$

Figure 4 $\quad W_S (W_{T/O}) = .48$

$W_S / W_{TO} = .48 \cdot \frac{.61}{.78} = 3.8 \approx 3.9$

$\frac{W_S}{T_O} = \frac{164}{1 - .75 \cdot .39} = 147 \frac{\text{lbs}}{F_{T^2}}$

$(W_S)_{IC} = .965 \cdot 147 = 142$

$\frac{C_L}{I_C} = \frac{142}{1481 \cdot 2360 \cdot .82^2} = .604 \neq .5$

Try $C_L = .58$ $\quad$ Fig $\quad \Delta M_{Div} = -0.1$

$M_{Div} = .804 - (-0.01) = .814$

$t/c = .096$

$\cos^2(35)(.096)^2(8) = .516$

$\downarrow$

$C_{L\,max\,T/O} = 1.76 \quad C_{L\,DO} = 2.68$

$W/S_{LDG} = 100$ $\quad \frac{W/S}{T/O} = 141$ $\quad \frac{W/S}{IC} = 136$

$$C_{L_{IC}} = \frac{136}{1481} \cdot .2360 \cdot .52^2$$

$$= 0.5 \checkmark$$

3 Engines $\quad$ TOFL = 9000

$$\left(W/T\right)_{.7V_{20}} = \frac{274}{141} \cdot .953 \cdot 1.76 = 3.26$$

$$V_{L/O} = 189 \text{ kts}$$

$$M_{L/O} = \frac{189}{661 \cdot .953^{1/2}} = .29 \quad .7M_{L/O} = .21$$

$T_{S\,LST} = 45{,}500$

$T_M = .21 = 37{,}200$

$$3.26 \cdot \frac{37{,}200}{45{,}500} = 2.67$$

HW 4

Range

$W_0 = W_{T/O} - W_{CR} = (658 - 9.642)K = 648376$ pounds

$W_1 = (1 - W_0/W_{T/O})W_{T/O} = (1 - .39)658{,}000 = 401350$

$C_{L_{WO}} = \frac{(W_0 + W_1)/2S}{14813} = .479$ $\quad C_{D_L} = \frac{C_L^2}{\pi ARe} = \frac{.479^2}{\pi \cdot 8 \cdot .852} = .0107$

$C_{D_L} = \frac{C_L^2}{\pi ARe} = .479^2 / \pi \cdot 8 \cdot .852 = 0.0107$

$C_D = C_{D0} + C_{Di} + \Delta C_{DC} = 0.0262$ $\quad L/D = \frac{.479}{0.0262} = 18.27$

$T_R = \frac{W_0 + W_1}{2} / L/D = 28729$ $\quad T_{TAD}$: 5304 / Engine

$R_{cruise} = \frac{V}{C} L/D \log_e(W_0/W_1) = \frac{473}{.63} \cdot 18.27 \log_e\left(\frac{648376}{401350}\right) = 6578$ n mi

$R = R_{CL} + R_{CR} = 83 + 6578 = 6661.$ $\quad R_{T/O} = 6555$ Req

Top of Climb

$C_{L_{IC}} = .59$ $\quad C_{Di} = 0.0163$ $\quad C_D = 0.0318$ $\rightarrow L/D = .59 / .0318 = 15.05$

$T_{Req} = \frac{648576}{15.05} = 43094 = 14365$ $\quad J_{TAD} = 79560$

$T_{Avail} = 10000$ OK

Climb Grad

I: $\quad C_{L_{max}\,T/O} / 1.2^2 = 1.22$ $\quad \frac{C_{L\,T/O}}{C_{L_{max}\,T/O}} = .694 \rightarrow$ Figure 6 $\quad \Delta C_{D_0} = 0.014$

$C_D = .0145 + .0140 + 0.0145 + \frac{1.22^2}{\pi \cdot 8 \cdot 8} = .1125$ $\quad \frac{L}{D})_{T/O} = 1.22 / .1125 = 10.84$

$T_{Req} = 658000 / 10.84 = 60681$ # $\quad \frac{T_a}{Eng} = \frac{82147}{24800} \cdot 34800 = 62287$

$Grad\ 2 = \frac{2 \cdot 62287 - 60681}{658} \cdot 100 = 9.7\%$ $\quad$ .3% Req

2:

$C_D = C_{D_0} + \Delta C_{D_0} + C_L^2/\pi A R e = 0.0980$

$L/D = 1.22/.0980 = 12.45$ $\quad T_{REQ} = 658000/12.45$
$= 52851$

$= 2 \cdot \frac{62287 - 52851}{658000} \cdot 100 = 10.9\%$ $\quad$ 2.7% REQ

Landing

$C_L = \frac{C_L}{1.3} = \frac{2.68}{1.32^2} = 1.59$

$\frac{C_L}{C_{L_{max}}} = \frac{1}{1.32} = .592 \rightarrow \Delta C_{D_0} = .0198$

$C_D = .0145 + .0198 + .0145 + \frac{1.59^2}{\pi \cdot 8 \cdot .852} = .1662$

$L/D = 1.59/.1662 = 9.56$

$T_{REQ} = 466{,}700/9.56 = 48795$ $\quad V = 140 \text{ KTS} = M = .21$

$T_a = \frac{82147}{45500} \cdot 37200 = 67162$ $\quad GRAD)_{LIG} = \frac{3 \cdot 67162 - 48795}{466700}$

$= 32.72\%$

DOC

$$V_B = \frac{6900}{.25 + .18 + 12.8 + 1} = 518\ mph$$

$T_B = .25 + .18 + 12.8 + .1 = 13.33$

Block Fuel

$F_B = \overset{70}{\cancel{F_{Am}}} + F_{CL} + \overset{70}{\cancel{F_D}} + F_{CR} + F_{Am}$

$F_{CL} = 9624$ $\quad F_{CR} + F_{Am} = T_{Req} \cdot c(T_{CR} + T_{Am})$

$= 233,481$

$F_B = 9624 + 233481 = 243,105$

Flight Crew

$P = 71125/2000 = 35.56$ TONS

Fuel & oil

$C_{FM} = (1.02\ F_B C_F + N_e C_{oT} \cdot T_D \cdot 0.135) / D \cdot P$

$C_F = .4/Gal \cdot \frac{1}{6.4}/6.4 = .0625/lb$

$C_{oT} = 2.15/lb$

$$C_{FM} = \frac{1.02 \cdot 243105 \cdot 0.0625 + 3 \cdot 2.15 \cdot 13.33 \cdot .135}{6900 \cdot 35.56} = 0.0632$$

Hull Insurance

$C_T = C_a + N_e$ $\quad C_a = 2.4 \cdot 10^6 + 87.5$

$W_a = W_0 - W_F - W_{F/L} - W_e$

$= 658(1 - .39) - 21.125 - .1046 \cdot 658$

$= 254,322$ #

$C_a = 24653158$

$C_e = 1344446$

$C_T = 28867974$

$C_{TM} = 0.0036$ $C_{TM} = 0.0027$ $C_{TM} = 0.0028$

$C_{TM} = 0.0014$ $C_{TM} = 0.005$ $C_{TM} = 0.0232$

$.0287 = .1313$ /ton mile

$$\frac{.1313 \cdot 35.576}{275} = 0.0170 \text{ /PAX mile}$$