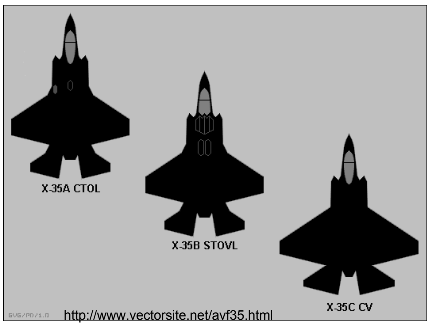
F-35 Joint Strike Fighter

http://www.zap16.com/mil%20fact/lockheed%20X-35.htm

F-35 Geometry



F35-A/B		F35-C	
Surface 1		Surface 1	
x,	y	x	y
0.00	0.00	0.00	0.00
-1.75	-0.75	-1.75	-0.75
-5.00	-0.75	-5.00	-0.75
-6.00	-2.10	-6.00	-2.10
-10.00	-2.10	-9.00	-2.10
-13.10	-7.25	-13.25	-8.75
-15.25	-7.25	-15.00	-8.75
-16.40	-2.50	-16.50	-3.00
-17.25	-2.50	-16.50	0.00
-17.25	0.00		
Surface 2		Surface 2	
x	y	x	y
-17.25	0.00	-16.50	0.00
-17.25	-2.50	-16.50	-3.00
-18.80	-5.00	-18.25	-5.80
-20.00	-5.00	-19.25	-5.80
-20.75	-1.25	-20.25	-1.00
-18.25	-1.00	-18.00	-1.00
-18.25	0.00	-18.00	0.00

Scale: 1 block = 2.45 ft

Acquired geometry data from plan-form pictures.

Plan-form was traced onto grid-line paper

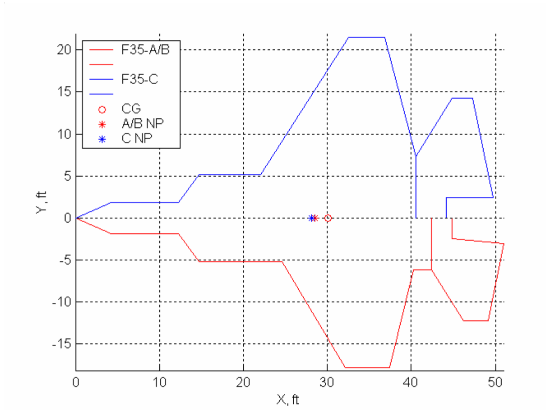
Geometric points were determined by measuring the number of blocks from the nose (for x) and centerline (for y)

Other geometric properties such as wing plan form area, mac, mean geometric chord, aspect ratio, and taper ratio were all found the same way

The semi-span and semi-span of the traced model were then compared to get the scale

Points and properties were adjusted using this scale and then used in VLMpc to get Cl_alpha, Cm_alpha, and the neutral point

Geometry Differences



	F35-A/B	F35-C
Length (ft)	50.85	50.85
b (ft)	34.78	42.98
S (ft^2)	448.05	568.33
c_bar (ft)	15.68	15.12
mac (ft)	15.12	18.44
?	0.244	0.159
AR	2.70	3.25
Np (ft)	28.51	28.15
Cg (ft)	30.05	30.05
Kn	-0.1020	-0.1032

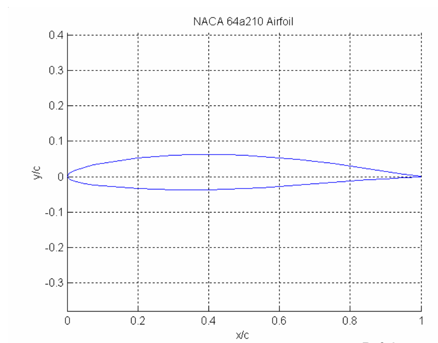
The geometry was plotted using Matlab

Shown is the CG, and neutral points for the respective versions

The CG was found using the assumption that the landing gear fall within 12-15° of the cg. (we used 15°)

Airfoil Selection

- No data on F35 Airfoil
- Selected Airfoil to match F35 desired performance
- Looked at F-15 Airfoil NACA 64A203
  - Supersonic example
- Could not find geometry
- Selected NACA 64A210



Ref. 8

There is no data for the F35 airfoil

We assumed that using an airfoil from another supersonic fighter would be close to what the F35 uses

The F-15 was selected, however the geometry for the F-15 airfoil was not available, so we used a similar configuration the NACA 64A210

VLMPC Goals:

To calculate the $C_{L\alpha}$ and $C_{M\alpha}$

To calculate the span loads of both variants

To extract the drag polars for both variants

VLMPC was used in the analysis to extract lift curve slope, moment slope, and span load of both variants of the F-35. VLMPC was also used to produce a drag polar for the two variations.

VLMPC

Input:

```
x35ab
2.        1.        4.61      41.6        -9.16
9.        0.0       0.0       0.0
0.        0.        0.        1.
-1.31     -0.56     0.        1.
-3.74     -0.56     0.        1.
-4.49     -1.57     0.        1.
-7.49     -1.57     0.        1.
-9.81     -5.43     0.        1.
-11.41    -5.43     0.        1.
-12.28    -1.87     0.        1.
-12.91    -1.87     0.        1.
-12.91    0.
6.        0.0       0.0       0.0
-12.91    0.        0.        1.
-12.91    -1.87     0.        1.
-14.07    -3.74     0.        1.
-14.97    -3.74     0.        1.
-15.53    -0.94     0.        1.
-13.66    -0.75     0.        1.
-13.66    0.
35.    6. 15.   .4  11.  0.   0.   0.
```

```
x35c
2.        1.        5.62      52.8        -9.16
8.        0.0       0.0       0.0
0.        0.        0.        1.
-1.31     -0.56     0.        1.
-3.74     -0.56     0.        1.
-4.49     -1.57     0.        1.
-6.74     -1.57     0.        1.
-9.92     -6.55     0.        1.
-11.23    -6.55     0.        1.
-12.35    -2.25     0.        1.
-12.35    0.
6.        0.0       0.0       0.0
-12.35    0.        0.        1.
-12.35    -2.25     0.        1.
-13.66    -4.34     0.        1.
-14.41    -4.34     0.        1.
-15.16    -0.75     0.        1.
-13.47    -0.75     0.        1.
-13.47    0.
35.   6.  15.   .4  11.  0.   0.   0.
```

Key values:

C_L = 0.1 to 1

Mach number = 0.4 (low speed)

The first task for this program was to determine the lift curve slope and the moment curve slope. The geometry from previous discussions was used to model the aircraft, the entire airplane was modeled. The speed for this part of the calculations was a Mach number of 0.4, to simulate low speed flight.

Coefficient Output:

X 35 A/B

$C_{L\alpha}$			α			
per rad	per deg	C_L (twist)	at C_L = 0	y cp	C_M/C_L	C_{M0}
3.61389	0.06307	0	0	-0.42331	0.10255	0

X 35 C

$C_{L\alpha}$			α			
per rad	per deg	C_L (twist)	at C_L = 0	y cp	C_M/C_L	C_{M0}
3.90913	0.06823	0	0	-0.42016	0.1043	0

$C_{M\alpha}$ = 0.370606 /rad or 0.006468 /deg for the X 35 A/B
$C_{M\alpha}$ = 0.407722 /rad or 0.007116 /deg for the X 35 C
Static margin = -0.10255 for X 35 A/B
Static margin = -0.1043 for X 35 C

$C_{M\alpha} = (dC_M/dC_L) * (dC_L/d\alpha)$ [VLMPC manual]
Static margin = $-C_M/C_L$ [VLMPC manual]
Static margin = $(h_{np} - h_{cg}) / c_{mac}$

Using a CG at 9.16 m from the tip and the static margin we get:
np = 8.69 m from tip for X 35 A/B
np = 8.58 m from tip for X 35 C

Conclusions:
According to VLMPC, both variants are unstable. This is to be expected from a fighter plane.

Both variants are unstable, as we can see by the negative static margin.

Span Load Output (excluding tail load):

X 35 A/B

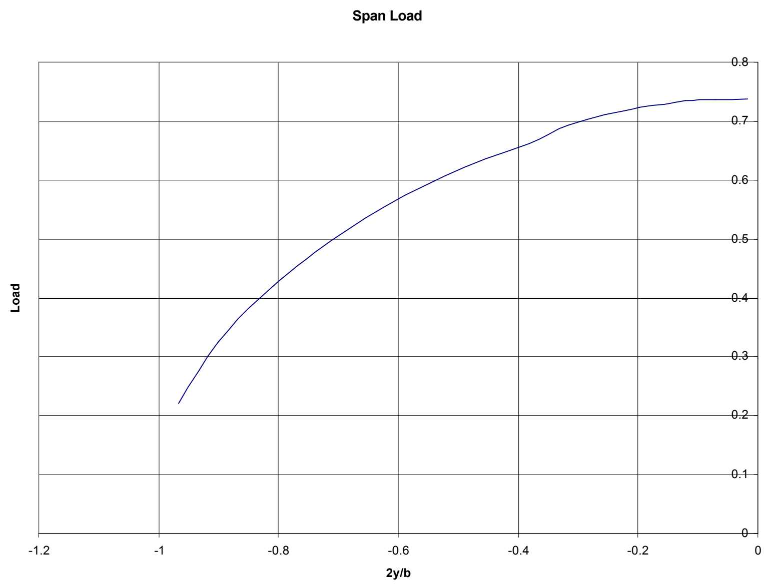
2y/b	$(c_l\, c) / (C_L\, c_{avg})$
-0.967	0.369
-0.9	0.541
-0.833	0.664
-0.767	0.761
-0.711	0.832
-0.655	0.896
-0.589	0.959
-0.522	1.014
-0.455	1.062
-0.383	1.105
-0.317	1.158
-0.256	1.187
-0.198	1.207
-0.156	1.218
-0.121	1.226
-0.07	1.231
-0.018	1.233

X 35 C

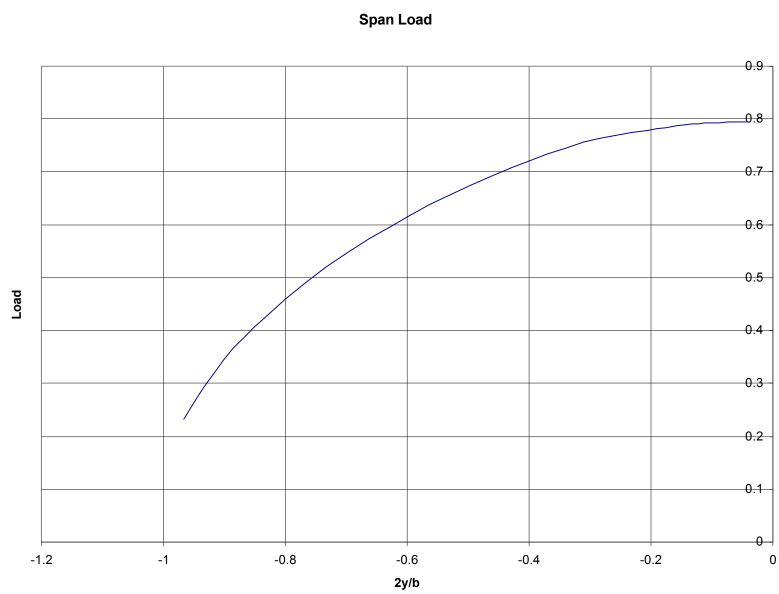
2y/b	$(c_l\, c) / (C_L\, c_{avg})$
-0.967	0.36
-0.9	0.53
-0.833	0.654
-0.767	0.755
-0.698	0.842
-0.629	0.918
-0.563	0.984
-0.496	1.041
-0.429	1.091
-0.37	1.132
-0.31	1.165
-0.258	1.186
-0.206	1.201
-0.144	1.216
-0.1	1.222
-0.043	1.225

This is the table output of the two variants. Showing an increasing load distribution along the half span. Elliptic in shape.

X 35 A/B Span load



X 35 C Span load



Drag Polars

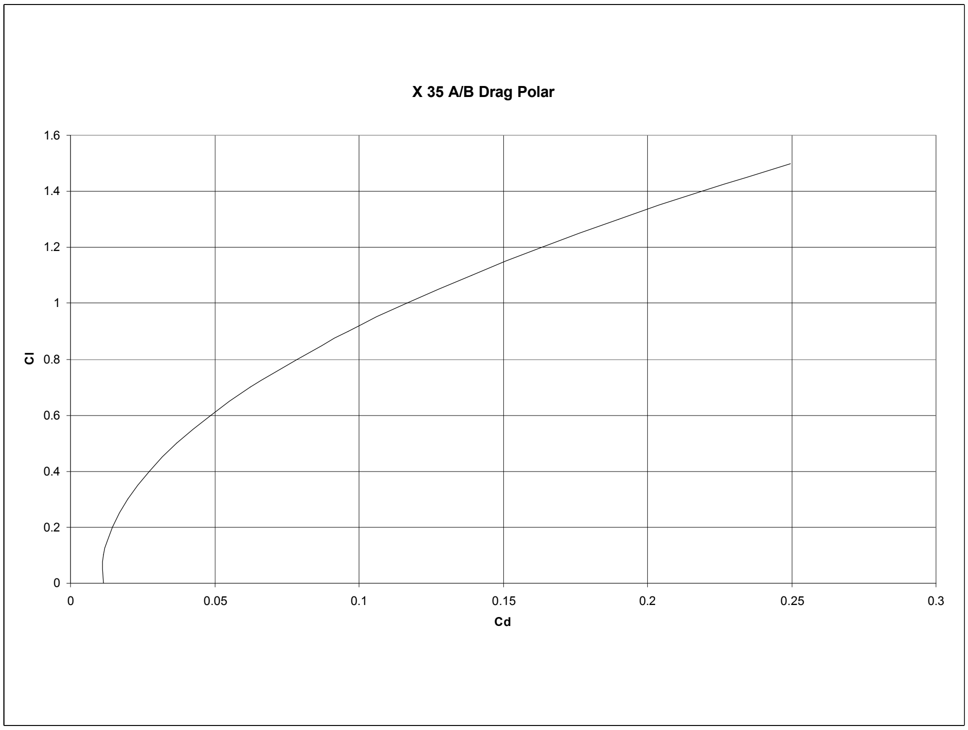
X 35 A/B C_L and Drag

C_L	C_{dTotal}	C_{di}
0	0.011385	0.0011
0.1	0.011385	0.0011
0.2	0.014585	0.0043
0.3	0.019885	0.0096
0.4	0.027285	0.017
0.5	0.036885	0.0266
0.6	0.048585	0.0383
0.7	0.062385	0.0521
0.8	0.078385	0.0681
0.9	0.096385	0.0861
1	0.116685	0.1064
1.1	0.138985	0.1287
1.2	0.163485	0.1532
1.3	0.189985	0.1797
1.4	0.218785	0.2085
1.5	0.249585	0.2393

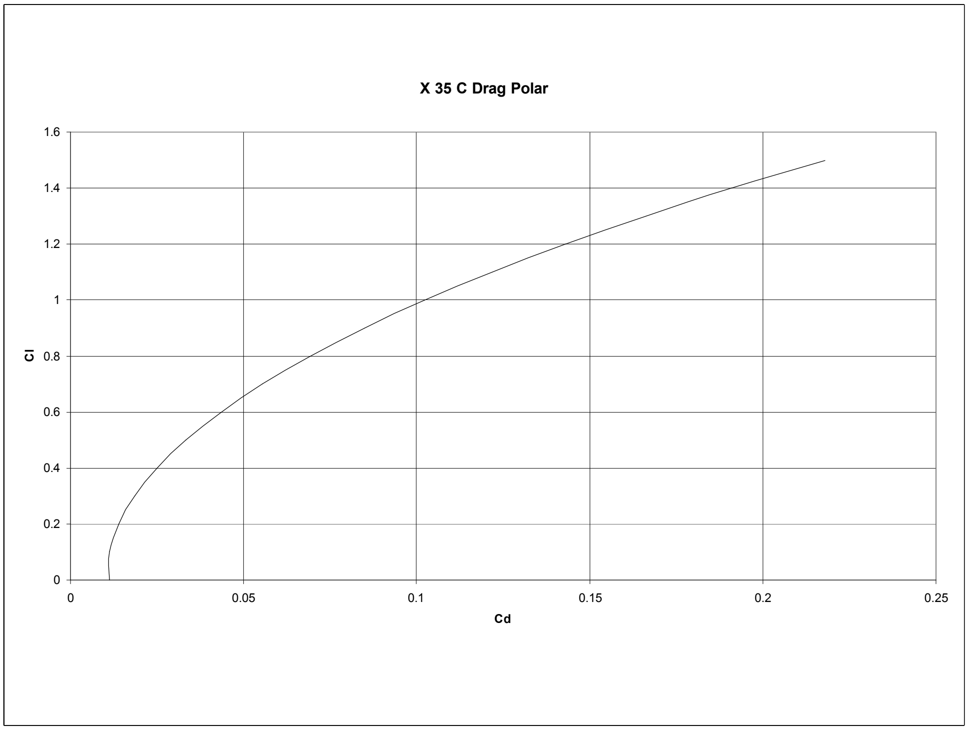
X 35 C C_L and Drag

C_L	C_{dTotal}	C_{di}
0	0.011185	0.0009
0.1	0.011185	0.0009
0.2	0.013985	0.0037
0.3	0.018585	0.0083
0.4	0.025085	0.0148
0.5	0.033385	0.0231
0.6	0.043485	0.0332
0.7	0.055485	0.0452
0.8	0.069285	0.059
0.9	0.084985	0.0747
1	0.102585	0.0923
1.1	0.121885	0.1116
1.2	0.143085	0.1328
1.3	0.166185	0.1559
1.4	0.191085	0.1808
1.5	0.217885	0.2076

These drag polars indicate an expected change of Cd with Cl.



These were graphed using the total drag coefficient against the lift coefficient rather than just the induced drag.


X 35 C Drag Polar
Cl
Cd
1.6
1.4
1.2
1
0.8
0.6
0.4
0.2
0
0.05
0.1
0.15
0.2
0.25

Friction input / output

- INPUT
  - Wetted area
    - Planar surface
    - Body of revolution
  - Mach number
  - Altitude
- OUTPUT
  - Cd_0 total
  - Cd_0 component

http://www.globalsecurity.org/military/systems/aircraft/images/x-35abc.jpg

The friction code uses the aircraft wetted area along with the mach number and the altitude to determine the value of Cdo for the aircraft. The wetted areas can be enter as either a planar surface or as a body of revolution. The friction code outputs the total drag coefficient for the aircraft as well as the drag coefficient produced by each of the components.

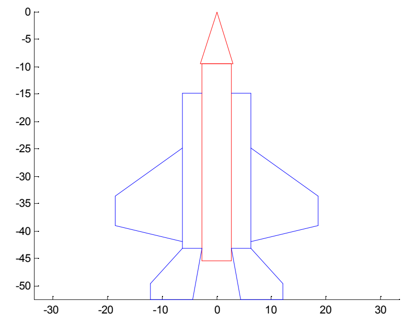
Components

- Nose – Cone
- Fuselage – Cylinder
- Nacelles – Rectangular prism
- Wings – Trapezoidal plate
- Horizontal Tail – Trapezoidal plate
- Vertical Tail – Trapezoidal plate

To determine the surface area of the F-35 models, the aircraft was broken down into six components; the nose, fuselage, nacelles, horizontal tail, vertical tail, and the wings. The friction code allow for objects to be modeled as either a planar surface or a body of revolution. In this case, the fuselage and the nose were modeled as bodies of revolution and the other components were done as planar surfaces.

F-35 ab component breakdown

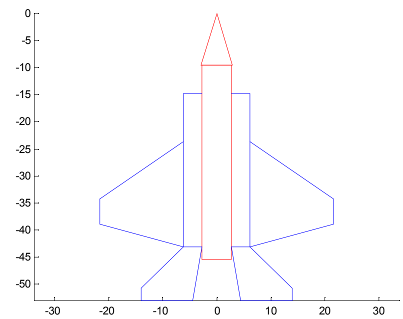
- Body of revolution in red
- Planar surface in blue



The divisions used for the various components can be seen in following Matlab plots. In each of these plots the blue sections represent the planar surfaces while the red lines represent the bodies of revolution.

F-35 c component breakdown

- Body of revolution in red
- Planar surface in blue



The divisions used for the various components can be seen in following Matlab plots. In each of these plots the blue sections represent the planar surfaces while the red lines represent the bodies of revolution. As seen by the plot, the F-35 c model has larger wings and a larger horizontal tail.

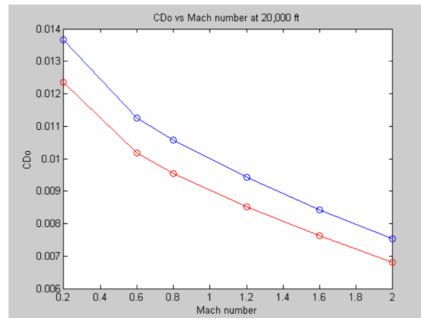
Calculated wetted area

Component	F-35 a-b (Ft^2)	F-35 c (Ft^2)
Nose	91.7	91.7
Fuselage	388.6	388.6
Wings	572.3	764.4
Horizontal Tail	239.1	283.7
Vertical Tail	210.9	210.9
Nacelles	693.4	693.4

Once the aircraft had been broken down into its six components measurements were taken to determine the geometry of the aircraft. Once the geometry of the aircraft was known, calculations were performed to determine the wetted area of each section of the aircraft. To estimate the wetted area of the nose, the nose was modeled as a cone to determine its surface area. The fuselage was modeled as a cylinder minus the area of sides covered up by the attached nacelles. The fuselage wetted area calculation also included the area of back end of the cylinder where the engine exhaust is. The wings, horizontal tails, and vertical tails were all modeled as trapezoids along with a frontal thickness. The nacelles were modeled as rectangular prisms without the sides that were against the fuselage. The estimated wetted surfaces areas for each of the six components for both models of the F-35 can be seen in the following table.

Friction output Cd_0

Mach number	F-35 a-b	F-35 c
0.2	0.01365	0.01234
0.6	0.01126	0.01018
0.8	0.01057	0.00955
1.2	0.00943	0.00852
1.6	0.00843	0.00762
2.0	0.00754	0.00681



F-35 ab in blue, F-35c in red

These are the results from the friction code for both configurations of the F-35. These calculations were performed at and altitude of 20,000 ft with a mach number varying from 0.2 to 2.0. The results are plotted with the ab model in blue, and the c model in red. From the friction output it can be noted that the friction drag decreases with an increasing velocity. An interesting note to this is that the c model had a larger surface area than the ab model, but had a lower friction drag.

L/D_{max} Calculation

- $K = 1 / (p\,ARe)$
- $(L/D)_{max} = 1/[2v(Cd_0K)]$

	F-35 a-b	F-35 c
AR	2.21	2.54
E	0.9734	1.0693
Cd_0	0.01077	0.00973
K	0.1479	0.1174
L/D_{max}	12.53	14.80

The L/D max calculations were done based on the shown equations. The driving factors for the L/D max calculation were aspect ratio, the span efficiency, and the value of Cdo. The results can be seen in the table and show the ab model to have a L/D max of 12.53 and the c model with a L/D max of 14.80. The increase in L/D max for the c model can mostly be attributed to its larger wingspan.

Cruise altitude estimation

- Use C_{Lmd} for cruise
- C_{Lmd} = v(Cd_0/K)
- L = CL_{md}*1/2*?*V^2*s
- Assumed:
  - Mach 0.85
  - 10,000 lbs fuel
  - 10,000 lbs payload

	F-35 a-b	**F-35 c**
W_{emptey}	23,000 lbs	24,000 lbs
W_{total}	43,000 lbs	44,000lbs
K	0.1479	0.1174
Cd_0	0.01077	0.00973
?	0.0013 slug/ft^3	0.00092 slug/ft^3
altitude	19,000 ft	29,000 ft

The cruise altitude estimations were done using minimum drag calculations for the lift coefficient. For these calculations a cruise velocity of mach 0.85 was assumed, along with a two thirds loading, which lead to about 10,000 lbs of fuel and 10,000 lbs payload. Performing an iterative calculation setting the lift equal to the weight, minimum density could be determined. This density was then used to determine the cruise altitude for min drag. It was determined the the a model would cruise at 19,000 ft and the c model would cruise at 29,000 ft.

LamDes Goals:

To determine the trimmed drag of the F-35 variants

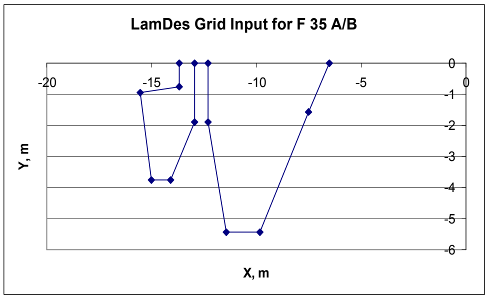
To determine the load split between wing and tail

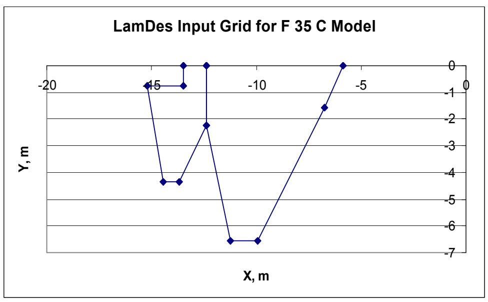
To determine the optimum CG location

Then to determine the optimum wing twist for minimum drag

One problem found with the LamDes program is the sensitivity of the program to the proximity of the two lifting surfaces. To achieve meaningful results from the program, it was necessary to vertically offset the tail of the aircraft by a distance of 0.5 m below the main wing. At vertical offsets of less than 0.5 m, the program predicted unreasonable values of twist and Cd. Twist values of ± 75º and Cd values of 2 or 3 were recorded for offsets less than 0.5 m. For vertical separations greater than 0.5 m, the change in results was small.

Input Geometry for LamDes





Input Variables:

M # = 0.9
h = 30,000 ft
W_{gross} = 50,000 lb [Ref. 2]
0 body moment constraint
0.0006 convergence criteria
0.03 under relaxation factor
Drag polar from VLMPC
C_{D0} from Friction
CG at 9.16 m from nose

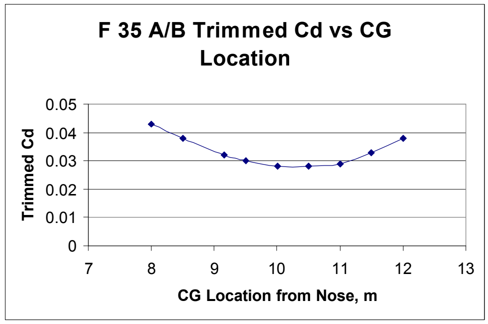
These values resulted in cruise C_L's of 0.31 for the A/B variant and 0.24 for the C variant

These figures show the geometry inputs for the LamDes program. As shown in the figures, only the lifting surfaces were modeled in LamDes, not the fuselage or aircraft nose.

The above geometry was entered into the LamDes input file for each variant. The reference areas and mean aerodynamic chords used were the same as described for the VLMpc input. The aircraft were analyzed at a cruise Mach of 0.9 at an assumed altitude of 30,000 ft. Given an estimated gross weight for both craft of 50,000 lbs#.2, this resulted in a cruise Cl of 0.31 for the A and B variants, and 0.24 for the C version. For both craft, the estimated CG location of 9.16 m behind the aircraft nose was originally set as the moment reference point. From this datum, the CG location was adjusted forward and backward to determine the location for minimum trimmed drag.

X 45 A/B Results

CG location	Cd trim	CL	e	CL Wing	Cl Tail
12	0.038	0.31	0.9	0.125	0.185
11.5	0.033	0.31	1.04	0.159	0.152
11	0.029	0.31	1.15	0.192	0.118
10.5	0.028	0.31	1.2	0.226	0.085
10	0.028	0.31	1.17	0.258	0.051
9.5	0.03	0.31	1.07	0.292	0.0184
9.16	0.032	0.31	0.98	0.315	-0.0046
8.5	0.038	0.31	0.8	0.36	-0.05
8	0.043	0.31	0.67	0.393	-0.083



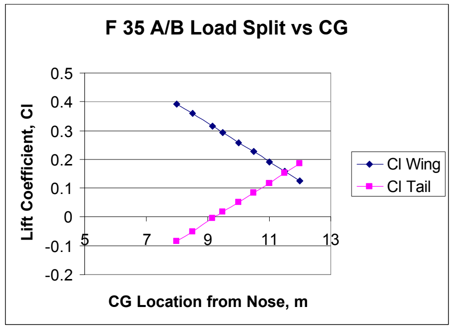
The minimum drag occurs with the CG between 10 m and 10.5 m from the nose

LamDes predicts a 28.4% instability, but LamDes take only the wing and tail into consideration, while VLMPC uses the entire airframe which also contributes to the lift

Because of this LamDes calculated a neutral point that is further aft than the one predicted by VLMPC

This figure clearly shows a minimum in trimmed drag for the A/B models occurs for a CG location of 10 or 10.5 m from the nose of the aircraft. Since the curve flattens off between 10 and 10.5 m, the team chose 10 m to be the desired location of the Cg due to the reduction in percent instability resulting from the 0.5 m forward movement of the CG. This position of CG only differs by 0.84 m from the position of 9.16 m estimated from the landing gear positioning.

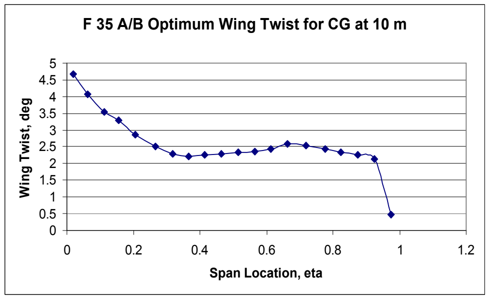
From the VLMpc analysis presented earlier, we know that the neutral point of the A/B models occurs at approximately 8.69 m from the nose. This results in a 28.4 % instability based on the LamDes analysis. However, it should be noted that the LamDes analysis considers only the lifting surfaces of the aircraft, while the VLMpc analysis considered the entire airframe. Consequently, the LamDes program ignores the lift produced by the fuselage of the airplane and estimates the neutral point of the aircraft to be further aft than the VLMpc method.



At the most forward position the lift on the wing is positive and the lift on the tail is negative as expect

As the CG is moved back and lift on the wing decreases and the lift on the tail increases and eventually becomes positive, and makes the airplane unstable

This figure shows the lift distribution on both the wing and tail for the F 35 A/B over a range of CG locations. As shown in the picture, when the CG is placed at a forward position (smaller number) the Cl on the wing becomes higher, while the Cl on the tail becomes small or negative to maintain moment balance. As the CG is moved backward and the airplane becomes unstable, the tail creates positive lift. By creating positive lift on the tail, the airplane reduces its induced drag. However, as shown in the chart and figure from above, there is a limit to how much drag can be reduced.



This twist analysis is for the minimum drag CG location of 10 m from the nose

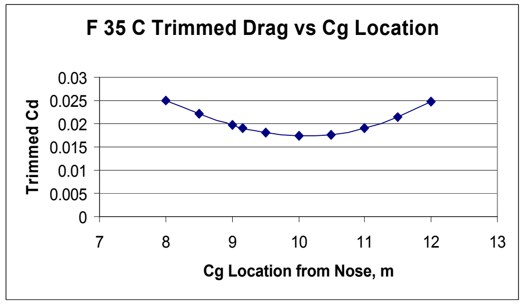
The twist decreases across the span, having washout at the tip

This figure shows the wing twist for the F 35 A/b, which LamDes solved for. This twist analysis is for the minimum drag CG location of 10 m from the nose. As shown, the twist decreases across the span, having washout at the tip.

X 35 C Results

CG Location	Cd trim	CL	E	CL Wing	CL Tail
12	0.0247	0.24	0.88	0.0778	0.1624
11.5	0.02141	0.24	1.056	0.104	0.1362
11	0.01895	0.24	1.2233	0.13048	0.1096
10.5	0.01773	0.24	1.328	0.1576	0.0824
10	0.01743	0.24	1.307	0.18382	0.0562
9.5	0.01799	0.24	1.182	0.21	0.0301
9.16	0.01914	0.24	1.069	0.2284	0.0117
9	0.01982	0.24	1.012	0.2372	0.0029
8.5	0.0222	0.24	0.8314	0.2642	-0.0242
8	0.02496	0.24	0.6717	0.2908	-0.0507

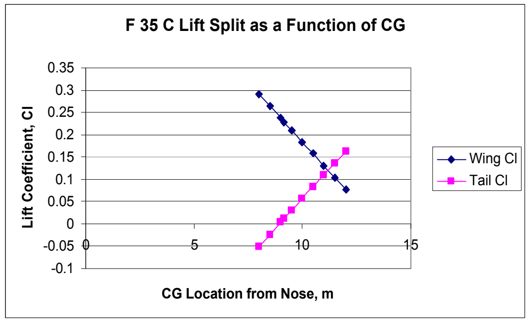
The data in this table is represented in above figures. The next figure shows that the C model of the F-35 experiences a minimum in trimmed drag for a CG location of 10 m behind the aircraft nose. The C model experiences a minimum drag coefficient of 0.017, while the table for the A/B shows that the A/B model experiences a minimum drag coefficient of 0.028. However, these coefficients are referenced to different areas.



The minimum drag again occurs with the CG at 10 m

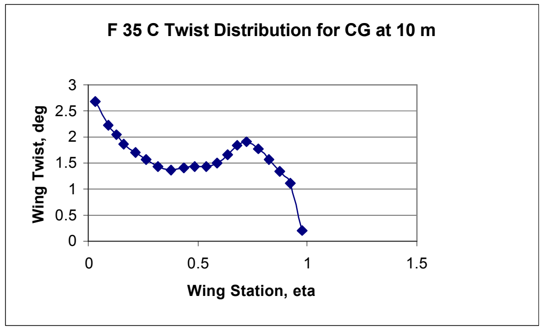
This location represents a 25% unstable aircraft, but the same problem occurs here as in the other variant, in that LamDes did not take into account the fuselage

As with the A/B model, a CG location of 10 m represents a difference of 0.84 m from the predicted value of 9.16 m. From the VLMpc estimation of the neutral point of the C version at 8.58 m behind the nose, the 10 m CG location would result in a 25% unstable aircraft. However, the discussion about differences in VLMpc and LamDes is again applicable to the C model aircraft. So, the actual airplane is most likely not as unstable as these results indicate.



Again we can see that the lift is a linear function of the CG location

This figure shows the distribution of lift between the wing and tail for the C model as a function of the CG location. As expected, the lift on either surface is a linear function of the CG location.



This variant has the same general twist distribution, although due to a lower required C_L the twist distribution is less

Assuming a design CG location of 10 m, LamDes was used to solve for the optimum wing twist. The twist distribution is shown in this figure. As with the A/B model, the wing has a significant amount of washout in it. However, due to the lower Cl values needed for the Navy craft assuming both variants to have a gross weight of 50,000 lbs, the C version requires less twist than the A/B, although the overall shape of the distribution is the same.

Transonic Airfoil Performance

- TSFOIL INPUT
  - Free Stream Mach number
  - Angle of Attack (AOA)
  - Airfoil Geometry
- Test cases
  - 2 angles of attack
  - 2 free stream mach numbers
- Output
  - CL, CD and pressure distribution

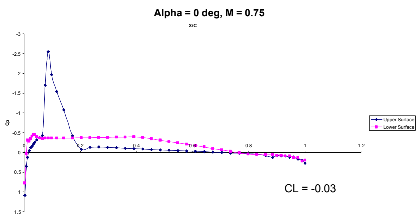
Free stream Mach #	AOA (deg)	C_L	C_D
0.75	0	-0.031	0.00647
0.95	0	0.087	0.05523
0.75	4	1.566	0.79486
0.95	4	0.588	0.08673

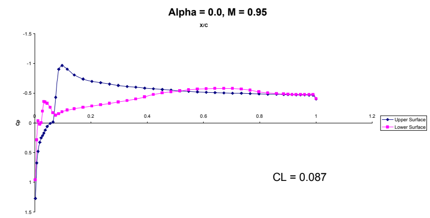
4 cases were run in TSFOIL with 2 different angles of attack and 2 different free stream mach numbers

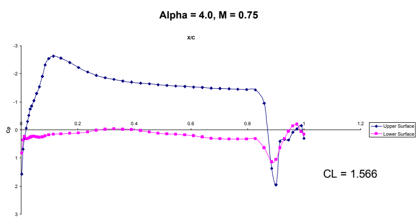
The plots shown on the subsequent slide show how the pressure distribution changes with the above two variables

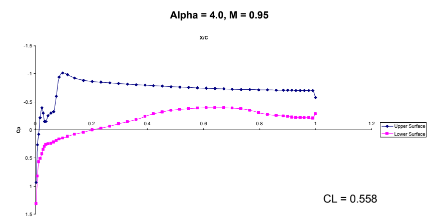
As the mach number increases the shock moves further back on the airfoil, and as the angle of attack increases the pressure on the upper surface decreases and maintains a good amount of lift. The pressure on the lower surface is slightly increased.

Pressure Distribution









References:

1. Lamar and Mason, *LamDes Optimization Program,* available at: http://www.aoe.vt.edu/~mason/Mason_f/MRsoft.html
2. Aerospaceweb.org, *X 35,* available at: http://www.aerospaceweb.org/aircraft/research/x35/index.shtml
3. Mason, W. *Skin fritcion and form drag program,* available at http://www.aoe.vt.edu/~mason/Mason_f/friction.exe
4. Lamar, J.E and Gloss, B.B *Vortex Lattice Method Program*,available at http://www.aoe.vt.edu/~mason/Mason_f/VLMpc.exe
5. Murman, Earll *Transonic Airfoil Program (TSFOIL2)* available at http://www.aoe.vt.edu/~mason/Mason_f/tsfoil2.exe
6. Marchman, James F. *Introductory Aircraft Performance*, 1996
7. http://www.globalsecurity.org/military/systems/aircraft/x-35-pics.htm
8. Airfoil Geometry http://www.aae.uiuc.edu/m-selig/ads/coord_database.html#N