

## NASA's Exploration Systems Architecture Study



Appendices - Section 6



# Section 6A Overall Launch Options

	 P	350 300 250 200 150 100	364	184.7	207.3	199.1	224.9	224.9 52.2 131.7	228.6 52.7 176.4	286.7 57.8 20.9 20.0 77.53.7	265.6 0.37 120.5 120
					1 Atlas V Heavy	2 Atlas V Heavy	3 Delta IV Heavy	3.1	4 Delta IV Heavy	5 Atlas-Evolved	5.1
Vehicle Name				Shuttle ( Ascent Only)	Crew Human Rated	New Upper Stage Crew Human Rated	Crew Human Rated	Delta IV HLV Crew Human-Rated	New Upper Stage Crew Human Rated	(8-m Core)- Crew	Atlas-Evolved (5 RD 180 & 4 J-2S+) Crew
LV Delivery Orbit 30X160 nmi Payload 28.5 Dea Inc Lift Capability Net Payload	Units mT		50.6 mT	25.0 mT	26.3 mT	33.4 mT	27.5 mT	26.5 mT	31.6 mT	54.3 mT	78.3 mT
Net Payload 51.6 Deg Inc Lift Capability Net Payload	mT mT		n/a	17.6 mT	23.7 mT 19.9 mT 17.9 mT	29.5 mT 26.6 mT	24.8 mT 22.4 mT 20.2 mT	23.9 mT 22.5 mT 20.3 mT	25.5 mT 22.9 mT	48.9 mT 50.8 mT 45.7 mT	70.4 mT 73.7 mT 66.4 mT
Seneral Parameters  Diverall Heicht  Pross Liftoff Mass  Infoff Thrust/Weicht  Second Stage Thrust/Weight  Safety / Reliability  Loss of Mission (LOM) - Mean	ft Ibm a g		363 ft 6.529.784 lbm 1.17 a 1 in 100	185 ft 4.523.142 lbm 1.40 a 0.98 g	207.3 ft 2.170.687 lbm 1.19 a 0.37 g	199.1 ft 2.189.029 lbm 1.18 a 0.57 g	224.9 ft 1.665.725 lbm 1.19 a 0.21 g	224.9 ft 1.663.255 lbm 1.20 a 0.19 g	228.6 ft 1.698.884 lbm 1.17 a 0.59 q	256.7 ft 2.857.475 lbm 1.20 a 0.89 g	265.6 ft 3.577.294 lbm 1.20 a 1.14 g
LOM - Median (50%).  Loss of Crew (LOC) - Mean			1 in 500	1 in 214	1 in 985	1 in 957	1 in 1.174		1 in 1.100	1 in 673	1 in 614
LOC - Median (50%) Strapon Boosters (each) Toos Mass Lenoth Jameter Jasble Procellant Mass Jurnout Mass Jurnout Mass Jurnout Mass Account Thrust @ 100% Account Thrust @ 100% Jurn Time	Ibm ft ft Ibm Ibm Ibm - Ibf sec sec			1.298.565 lbm 150 ft 12.2 ft 1.111.049 lbm 187.516 lbm RSRM 3.139.106 lbf 269 sec 125 sec	680.305 lbm 118.7 ft 16.4 ft 626.341 lbm 53.945 lbm 47.893 lbm 1 / RD-180 933.400 lbf 338.4 sec 229.0 sec	676.137 lbm 119.2 ft 12.5 ft 622.205 lbm 53.913 lbm 47.893 lbm 1 / RD-180 933.400 lbf 338.4 sec 228.0 sec	513.807 lbm 154.1 ft 16.4 ft 451.749 lbm 62,042 lbm 56,031 lbm 1 / RS-68 743.031 lbf 408.8 sec 246.8 sec	513.806 lbm 154.1 ft 16.4 ft 451.749 lbm 62,040 lbm 56,030 lbm 1 / RS-68 743.031 lbf 408.8 sec 247.2 sec	513.824 lbm 154.1 ft 16.4 ft 451.749 lbm 62.058 lbm 56,047 lbm 1 / RS-68 743.031 lbf 408.8 sec 244.0 sec		
irist/Core Stage ross Mass enoth harmonic Mass urmout Mass urmout Mass urmout Mass urmout Mass urmout Mass urmout Mass and Mass urmout Mass acuam Thana (@ 100% acuam Time second Stage	Ibm ft ft Ibm Ibm Ibm Ibm Ibm sec sec		5,049,035 lbm 138.1 ft 33.0 4.746.309 lbm 302,726 lbm 5 /F-1 7,500,000 lbf 265 sec	1,929,506 lbm 166 ft 27,58 (ET) 1,596,995 lbm 338,557 lbm 283,159 lbm 3/SSME bik ll 1,473,450 lbm 453 sec 510 sec	689,767 lbm 120.5 ft 16.4 ft 625,920 lbm 63,828 lbm 57,779 lbm 1 / RD-180 933,400 lbf 338.4 sec 302.0 sec	678,901 lbm 118.5 ft 12.5 ft 622,205 lbm 50,657 lbm 50,657 lbm 1 / RD-180 933,400 lbf 338.4 sec 296.0 sec	518,620 lbm 160,9 ft 164,4 ft 451,749 lbm 66,854 lbm 1,785-68 743,031 lbf 408,8 sec 330,3 sec	518.620 lbm 160.9 ft 16.4 ft 451.749 lbm 66.854 lbm 60.843 lbm 1 / RS-68 743.031 lbf 408.8 sec 331.7 sec	519,558 lbm 163.1 ft 16.4 ft 451.749 lbm 61,782 lbm 1 / RS-68 743,031 lbf 408.8 sec 328.0 sec	1,965,396 lbm 116.0 ft 27.5 ft 17.79 462 lbm 185,783 lbm 180,088 lbm 4 / RD-180 933,400 lbf 338.4 sec 161.0 sec	2,622,069 lbm 120.5 ft 27.5 ft 24.11.799 lbm 210,038 lbm 184,376 lbm 5 / RD-180 933,400 lbf 338.4 sec 174.8 sec
ross Mass nonth sameter sable Propellant Mass shole Propellant Mass surfout Mass roles of the Mass saines (Motor sourm Thrust @ 100% sourm Thrust @ 100% sourm Time hird Stage	Ibm ft ft lbm lbm lbm - lbf sec sec		1.093.921 lbm 81.5 ft 33.0 ft 1.004.545 lbm 89.376 lbm 5 / J-2 200,000 lbf 426.0 sec		53.063 lbm 34.6 ft 10.0 ft 43.840 lbm 9.222 lbm 7.159 lbm 2 / RL-10A-4-2 22.300 lbf 451.0 sec 443.0 sec	75.021 lbm 35.9 ft 16.4 ft 57.113 lbm 17.905 lbm 15.115 lbm 4 / RL-10A-4-2 22.300 lbf 451.0 sec 289.0 sec	49.474 lbm 34.6 ft 16.4 ft 37.490 lbm 11.981 lbm 9.575 lbm 1 / RL-10B-2 24.750 lbf 465.5 sec 705.2 sec	49.263 lbm 34.6 ft 16.4 ft 37.495 lbm 11.766 lbm 9.365 lbm 1R104.4-2 22.300 lbf 451.0 sec 757.5 sec	72.796 lbm 35.9 ft 16.4 ft 57.241 lbm 15.553 lbm 12.889 lbm 4 / RL-10A-4-2 22.300 lbf 451.0 sec 289.0 sec	762.967 lbm 82.9 ft 27.5 ft 669.117 lbm 93,777 lbm 80.824 lbm 3 / J-28 265,000 lbf 436.0 sec 367.0 sec	773.390 lbm 82.9 ft 27.5 ft 666.458 lbm 106.856 lbm 91.222 lbm 4 / J-2S+ 274.500 lbf 451.5 sec 274.1 sec
oss Mass noith ameter ambeler Procellant Mass rroot Mass Wass with Mass count Thrust (g. 100% count Thrust (g.	Ibm ft Ibm Ibm Ibm - Ibf sec sec  ft ft ft Ibm		266.096 lbm 59.3 ft 21.7 ft 239.388 lbm 1/.J-2 200.000 lbf 426.0 sec					9.300 lbm	9,300 lbm		
enoth Jaimeter Jasable Procellant Mass Javanour Mass Javanour Mass Inginies(s)Motor Aracum Thrust @ 100% Aracum Specific Impulse @ 100% sceleration Turn Impur Mass ettison Mass ettison Time	ft ft lbm lbm - lbf sec o's sec lbm sec		9,104 lbm		9,300 lbm 332.0 sec	9,300 lbm 326.0 sec	9,300 lbm 360.3 sec	9,300 lbm 361.7 sec	9,300 lbm 358.0 sec	9,300 lbm 191.0 sec	9.300 lbm 204.8 sec
light Parameters aximum Dvnamic Pressure Time aximum Acceleration Time aximum Altitude During Ascent	g's		437 psf 4.64 g	(STS114) 751 psf 3.2 g/s	292 psf 88.0 sec 4.00 g	424 psf 89.1 sec 4.00 q 212 s 290 s 473,961 ft	368 psf 90.0 sec 4.00 g	370.0 psf 90.0 sec 4.00 g	334 psf 93.0 sec 3.89 g	537 psf 83.2 sec 3.72 g	549 psf 85.8 sec 4.00 g 174.8 sec

	Overall Vehicle Heldrit. ft	350 300 250 200 150 100	184.0	330.1 131.2 330.1 116.0 119.2 119.2	330.1 131.2 82.9 110.0 111	131.2. 131.2.	131.2° 82.9° 334.6° 120.5° 119.2° 27.5° 412.6°	334.6 131.2 82.9 120.5 179.2 27.6 119.2
			5.2	6	7	7.1	7.2	7.3
			Atlas-Evolved for 25 mT - Crew	CSTCA5 - Cargo	Atlas-Evolved (8-m Core) + 4 Atlas V Boosters	Atlas-Evolved (4 RD-180 & 4 J-2S) + 4 AV Boosters Cargo Bik 2	Atlas Evolved (4 RD-180 & 3 J-2S+) + 4 AV Boosters Cargo Blk 2	Atlas-Evolved (4 / RD-180 & 4 J-2S+) + 4 AV Boosters Cargo Blk 2
Vehicle Name  LV Delivery Orbit 30X160 nmi Payload  28.5 Dea Inc  Lift Capability Net Payload	Units mT mT		33.7 mT	85.8 mT 73.0 mT	116.7 mT	118.6 mT 100.8 mT	121.6 mT	124.0 mT
51.6 Deg Inc Lift Capability Net Payload	mT		31.7 mT 28.5 mT	81.0 mT 68.9 mT	111.0 mT 94.3 mT	112.3 mT 95.4 mT	116.5 mT 99.0 mT	117.6 mT 99.9 mT
General Parameters Overall Heicht Gross Liftoff Mass Liftoff ThurstWeicht Second Stage ThrustWeight Safety / Reliability Loss of Mission (LOM) - Mean	ft Ibm a g		184.0 ft 1.850.022 lbm 1.39 a 1.09 g	330.1 ft 4.288.574 libm 1.25 a 0.81 g	330.1 ft 5.715.777 ibm 1.20 a 0.76 g	330.1 ft 5.727.030 lbm 1.20 a 1.00 g	334.6 ft 5.732.103 lbm 1.20 a 0.78 g	334.6 ft 5.722.944 libm 1.20 a 1.03 g
LOM - Median (50%) Loss of Crew (LOC) - Mean LOC - Median (50%)								
Strapon Boosters (each) Cross Mass Leneth Diameter Usable Procellant Mass Burnout Mass Dry Mass Endine(s)Motor Vacuum Thrast @ 100% Vacuum Seecfic Imoulse @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm Ibm - Ibf sec sec			386.506 lbm 119.2 ft 12.5 ft 331.960 lbm 54.527 lbm 48.475 lbm 1 / RD-180 933.400 lbf 338.4 sec 120.0 sec	432 613 lbm 119.2 ft 12.5 ft 378.935 lbm 53.699 lbm 47,607 lbm 1 / RO-180 933.400 lbf 338.4 sec 137.0 sec	432 620 lbm 119.2 ft 12.5 ft 378 935 lbm 53,666 bm 47,614 lbm 1 / RD-180 933.400 lbf 338.4 sec 137.0 sec	432 622 lbm 119.2 ft 12.5 ft 378 935 lbm 53,667 lbm 47,616 lbm 1 / RD-180 933.400 lbf 338.4 sec 137.0 sec	426.991 lbm 119.2 ft 12.5 ft 373.388 lbm 53.664 lbm 47.613 lbm 1 / RD-180 933.400 lbf 338.4 sec 135.3 sec
First/Core Stage Gross Mass Lenoth Diameter Lenoth Diameter Lenoth Diameter Lenoth Len	Ibm ft ft Ibm Ibm Ibm - Ibf sec sec		1,606,226 lbm 78.8 ft 27.5 ft 1,476,742 bm 129,484 lbm 14,599 lbm 3 (RD-180 933,400 lbf 338.4 sec 192.0 sec	2,539,733 lbm 116.0 ft 27.5 ft 231,500 lbm 188,030 lbm 162,291 lbm 4 / RD-180 933,400 lbf 338,4 sec 213,0 sec	2,941,005 lbm 116.0 ft 27.5 ft bbm 16,424 lbm 164,722 lbm 4 / RD-180 933,400 lbf 338.4 sec 249,0 sec	2,940,997 lbm 116.0 ft 27.5 ft 190,436 lbm 190,436 lbm 164.714 lbm 4 / RD-180 933,400 lbf 338.4 sec 249.0 sec	2,945,200 lbm 120.5 ft 27.5 ft 27.5 ft 194,630 lbm 164,937 lbm 4 / RD-180 933,400 lbf 338.4 sec 249.0 sec	2,945,200 lbm 120.5 ft 27.5 ft 27.5 ft 194.630 lbm 168.917 lbm 4 / RD-180 933,400 lbf 338.4 sec 249.3 sec
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm - Ibf sec sec		160.146 lbm 53.0 ft 16.4 ft 132.139 lbm 28.007 lbm 23.468 lbm 1 / J-25 265.000 lbf 435.0 sec 217.0 sec	763.182 lbm 82.9 ft 27.5 ft 667.620 lbm 95.490 lbm 81.049 lbm 3 / J-2S 265,000 lbf 436.0 sec 366.0 sec	763.590 lbm 82.9 ft 27.5 ft 666.146 lbm 97.371 lbm 81.467 lbm 3 / J-28 265.000 lbf 436.0 sec 366.0 sec	770.668 lbm 82.9 ft 27.5 ft 665.905 lbm 104.691 lbm 88.513 lbm 4 / J-2S 265.000 lbf 436.0 sec 274.0 sec	764,946 lbm 82.9 ft 27.5 ft 666.153 lbm 98,717 lbm 82.816 lbm 3 / J-25+ 274,500 lbf 451.5 sec 366.0 sec	772.482 lbm 82.9 ft 27.5 ft 665.888 lbm 106.518 lbm 90.318 lbm 4 / J-25+ 274.500 lbf 451.5 sec 273.8 sec
Third Stage Gross Mass Length Diameter Usable Propolant Mass Burnout Mass Burnout Mass Drv Mass Ernoinels Motor Vacuum Thrust @ 100% Wacuum Seedific Imnuulse @ 100% Burn Time Pavload Shroud Overall Length Outside Diameter Dynamic Length Dynamic Diameter Overall Mass Jettison Mass Jettison Mass Jettison Mass	Ibm ft ft Ibm Ibm Ibm Ibf sec sec ft ft ft Ibm Ibm Ibm Ibm			131.2 ft 27.5 ft 98.4 ft 24.5 ft 23.438 lbm 23.438 lbm 301.0 sec	131.2 ft 27.5 ft 98.4 ft 24.5 ft 23.438 bm 23.438 bm 25.00 sec	131.2 ft 27.5 ft 98.4 ft 24.5 ft 23.438 bm 23.438 bm 334.0 sec	131.2 ft 27.5 ft 98.4 ft 24.5 ft 23.411 bm 23.411 bm 26.0 sec	131.2 ft 27.5 ft 98.4 ft 24.5 ft 23.411 bm 23.411 bm 33.0 sec
Launch Escape System Gross Mass Lenoth Diamete Lenoth Diamete Lenoth Diamete Lenoth Le	Ibf sec o's sec Ibm sec		9.300 lbm 2220 sec	60°-11	600 and	Elft and	506 cud	500 cal
Maximum Dvnamic Pressure Time Maximum Acceleration Time Maximum Altitude During Ascent	g's sec ft		750 psf 70.0 sec 4.00 g	543 osf 84.5 sec 3.21 q	542 osf 84.0 sec 3.02 g	548 osf 84.9 sec 3.00 q 249 sec 476,546 ft	536 osf 84.3 sec 2.98 q 249 sec 546,182 ft	549 osf 84.7 sec 2.95 q

	Overal Vehicle Feient. ft	200	334.6 131.2 120.5 120.5 119.2 27.6 112.5	347.6 62.2 27.5 82.9 120.5 119.2 27.5 119.2 27.5	359.5 131.5' 359.5'	332.3 131.5' 332.3' 117.3' 117.3' 117.3' 117.3'	353.5	205.7	252.9  118.7  118.7  11.7  222.9  10.6  10.6  10.7  10
			7.4	7.5	7.6B	7.6C	8	9	10
Vehicle Name			Atlas-Evolved (8-m Core) + 2 Atlas V Boosters Cargo	Atlas -Evolved (8-m Core) + 2 Atlas V Boosters Crew + Cargo	Atlas Evolved (8-m Core/3RD-171) + 2 Zenit Boosters Cargo Blk 2	Atlas-Evolved (8-m Core/2Rd-171) + 2 Zenit Boosters Cargo Blk 2	Atlas-Evolved (8-m Core) + SRB Boosters	Atlas Phase 2 Crew	Atlas Phase 2 – Cargo
LV Delivery Orbit 30X160 nmi Payload	Units								
Lift Capability Net Payload 51.6 Deg Inc Lift Capability	mT		111.9 mT 95.1 mT 106.1 mT	110.3 mT 93.7 mT 104.2 mT	142.2 mT 120.9 mT 134.5 mT	111.9 mT 95.2 mT 105.9 mT	117.2 mT 99.7 mT 110.9 mT	28.8 mT 25.9 mT 27.3 mT	73.6 mT 62.6 mT 69.5 mT
Net Payload  General Parameters  Overall Heicht Gross Liftoff Mass Liftoff Thrust/Weicht Second Stage Thrust/Weight			90.2 mT 334.6 ft 5.004.575 lbm 1.21 a 1.05 g	88.6 mT 347.6 ft 4.995.071 libm 1.21 a 1.06 g	114.3 mT 359.5 ft 6.563.380 lbm 1.25 a 0.98 g	90.0 mT 332.3 ft 5.397.740 lbf 1.21 a 1.04 q	94.3 mT 353.5 ft 6.594.334 lbm 1.40 a 0.78 g	24.5 mT  205.7 ft 1.409.638 lbm 1.22 a 0.91 g	59.1 mT 252.9 ft 3.811.194 lbf 1.36 a 0.64 q
Safety / Reliability Loss of Mission (LOM) - Mean LOM - Median (50%)			1 in 71	1 in 65			1 in 99	1 in 134	
Loss of Crew (LOC) - Mean LOC - Median (50%)				1 in 536				1 in 939	
Strapon Boosters (each) Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Endine(s) Motor Vacuum Thrust @ 100% Vacuum Seedifc Imoulse @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm - Ibf sec sec		505,948 lbm 119.2 ft 12.5 ft 450,964 lbm 54,965 lbm 48,913 lbm 1 / RD-180 933,400 lbf 338.4 sec 163.5 sec	505,942 lbm 119.2 ft 12.5 ft 450,964 lbm 54,959 lbm 48,907 lbm 1 / RD-180 933,400 lbf 338.4 sec 163.5 sec	760.248 lbm 133.4 ft 12.8 ft 683.476 lbm 76.751 lbm 68.999 lbm 1 / RD-171 1.778.000 lbf 337.0 sec 130.0 sec	759.971 lbm 133.4 ft 12.8 ft 683.476 lbm 76.474 lbm 68,712 lbm 1 / RD-171 1.778.000 lbf 337.0 sec 130.0 sec	1.297.882 lbm 150.0 ft 12.2 ft 1.111.028 lbm 186.854 lbm 2 / 4 seq lSRM 3.139.106 lbf 288.8 sec 125.0 sec		1.144.889 lbm 100.5 ft 17.7 ft 1.054.862 lbm 89.795 lbm 80.231 lbm 2 / RD-180 933.400 lbf 338.4 sec 1912 sec
First/Core Stage Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Engine(s)Motor Vacuum Thrast @ 100% Vacuum Specific Impulse @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm Ibm Ibf sec sec		2,950,951 lbm 120.5 ft 27.5 ft 27.5 ft 27.40,880 lbm 210,038 lbm 184,376 lbm 5 / RD-180 933,400 lbf 338.4 sec 198.7 sec	2,947,670 lbm 120.5 ft 27.5 ft 27.5 gs, 2740,880 lbm 206,757 lbm 181,095 lbm 5 / RD-180 933,400 lbf 338.4 sec 198.7 sec	3,921,961 lbm 145.1 ft 27.5 ft 3,656.821 lbm 284,833 lbm 251,805 lbm 3,780-171 1,778,000 lbf 337.0 sec 230.0 sec	2,825,102 lbm 117.9 ft 27.5 ft 2,616,244 lbm 208,637 lbm 184,038 lbm 2,1 RD-171 1,778,000 lbf 337.0 sec 248.0 sec	2,952,522 lbm 139.4 ft 27.5 ft 2.750,328 lbm 201,961 lbm 176,239 lbm 4 / RD-180 933,400 lbf 338.4 sec 249.0 sec	1,149,874 lbm 96.0 ft 17.7 ft 1,054,852 lbm 94,990 lbm 85,426 lbm 2 / PD-190 933,400 lbf 338.4 sec 194.0 sec	1,148,311 lbm 96.0 ft 17.7 ft 1,054,852 lbm 93,426 lbm 83,862 lbm 2 / RD-180 933,400 lbf 338.4 sec 275.9 sec
Second Stage Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Phidies/Motor Vacuum Thrast @ 100% Vacuum Specific Impulse @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm Ibm sec sec		771,590 lbm 82.9 ft 27.5 ft 666.458 lbm 105.056 lbm 89.422 lbm 4 / J-252 274,500 lbf 451.5 sec 274.1 sec	767.751 lbm 82.9 ft 27.5 ft 666.294 lbm 101,382 lbm 85,585 lbm 4 / J-25+ 274,500 lbf 451.5 sec 274.0 sec	784.246 lbm 82.9 ft 27.5 ft 664.807 lbm 113.364 lbm 102.090 lbm 4 7 J-25+ 274, 500 lbf 451.5 sec 273.0 sec	782.801 lbm 82.9 ft 27.5 ft 666.224 lbm 110.532 lbm 100.635 lbm 4 / J-25+ 274.500 lbf 451.5 sec 274.0 sec	764.165 lbm 82.9 ft 27.5 ft 666.109 lbm 97.983 lbm 82.042 lbm 3 / 1-22 265.000 lbf 436.0 sec 366.0 sec	186.902 lbm 51.0 ft 17.7 ft 158.833 lbm 28.063 lbm 24.174 lbm 4 / Rt.60 60.000 lbf 465.0 sec 388.0 sec	188.542 lbm 47.1 ft 17.7 ft 156.802 lbm 31.734 lbm 25.824 lbm 4 / RL-60 60.000 lbf 465.0 sec 30.5.8 sec
Third Stage Gross Mass Length Diameter Usable Propellant Mass Burnout Mass Dry Mass Encine(s)/Motor Vacuum Thrust @ 100%	Ibm ft ft Ibm Ibm Ibm								
Vacuum Soecific Imoulse @ 100% Burn Time Payload Shroud Overall Length Outside Diameter Dynamic Length Dynamic Length Dynamic Jiameter Overall Mass Jettison Mass Jettison Time	sec sec ft ft ft lbm lbm sec		131.2 ft 27.5 ft 98.4 ft 24.5 ft 23.411 lbm 23.411 lbm 313.5 sec	82.0 ft 27.5 ft 82.0 ft 24.5 ft	131.5 ft 27.5 ft 98.4 ft 24.5 ft 23.113 bm 23.113 bm 308.0 sec	131.5 ft 27.5 ft 98.4 ft 24.5 ft 23.104 lbm 23.104 lbm 306.0 sec	131.2 ft 27.5 ft 98.4 ft 24.5 ft 23.419 lbm 23.419 lbm 241.0 sec		109.8 ft 24.3 ft 82.0 ft 21.3 ft 22.695 bm 22.695 bm 332.0 sec
Launch Escape System Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Craine(s) Motor Vacuum Thrust @ 100% Acceleration Burn Time Burn Time	Ibm ft ft ft Ibm Ibm Ibm - Ibf sec a's sec								
Jettison Mass Jettison Time	lbm sec			9,300 lbm 229.0 sec				9,300 lbm 224.0 sec	
Flight Parameters Maximum Dvnamic Pressure Time Maximum Acceleration Time Maximum Altitude During Ascent	g's		567 psf 85.6 sec 3.73 g 198.7 sec	582 psf 87.0 sec 3.75 g 198.7 sec	635 psf 81.4 sec 3.79 q	574 psf 83.6 sec 3.02 g	595 psf 66.5 sec 3.05 g	532 psf 81.3 4.00 g 385,739 ft	422 psf 78.0 sec 4.14 g

	Overall Vehicle Height, ft	250	252.9 116.3' 40.6' 96.00 17.3' 202.9'	295,7*	290.1 58.7 82.0 290.1 77.7	ehicle 12 Converted to Vehicle 10.1	266.6 50.27 10.6° 10.27 10.27	290.4 105.8	302.7 64.5' 105.8' 102.7' 113.2' 113.2'	267.4 52.7 52.7 52.7 52.7 52.7 52.7 52.7
			10.1  Atlas Phase 2 (4 GEM-60s) –  Cargo	11 Atlas Phase 3A (5-m CBC) Cargo	11.1 Atlas Phase 3A Crew + Cargo	12 Atlas Phase 2 (4 GEM-60s) – Cargo	13 RSRB 4-Segment - Crew	4-Segment SRB with 1 SSME Crew	4-Segment RSRB with 1 SSME Cargo	4-Segment RSRB with 1 J-2S+ Crew
Vehicle Name  LV Delivery Orbit 30X160 nmi Payload  28.5 Dea Inc	Units					Vehicle 12 is same as Vehicle 10.1				
Lift Capability Net Payload 51.6 Deg Inc	mT mT		78.4 mT 66.7 mT	110.4 mT 93.8 mT	106.6 mT 90.6 mT		23.1 mT 20.8 mT	27.2 mT 24.5 mT	26.1 mT 23.4 mT	21.6 mT 19.5 mT
Lift Capability Net Payload	mT mT		74.2mT 63.0 mT	104.4 mT 88.8 mT	100.3 mT 85.3 mT		21.7 mT 19.5 mT	25.4 mT 22.9 mT	24.4 mT 22.0 mT	20.3 mT 18.2 mT
General Parameters Overall Height Gross Liftoff Mass Liftoff Thrust/Weight Sacond Stage Thrust/Weight Safety / Reliability Loss of Mission (LOM) - Mean	ft Ibm a g		252.9 ft 4.119.062 lbf 1.25 a 0.83g	295.7 ft 6.222.816 lbm 1.39 a 0.56 q 1 in 88	290.1 ft 6.195.750 lbm 1.39 a 0.53 q		266.6 ft 1.641.510 lbf 1.49 a 1.13 g	290.4 ft 1.775.385 lbm 1.38 a 1.03 g	302.7 ft 1.772.068 lbm 1.38 a 1.04 g	267.4 ft 1.621.814 lbm 1.51 a 0.85 g
LOM - Median (50%) Loss of Crew (LOC) - Mean				1 11 00	1 in 612			1 in 2021		1 in 1918
LOC - Median (50%) Strapon Boosters (each) Gross Mass Lenoth Diameter Proceilant Mass Usable Proceilant Mass Only Mass Engine(s) Motor Vacuum Thrust @ 100% Vacuum Soedic Imoulse @ 100% Usum Time	Ibm ft ft Ibm Ibm Ibm Ibm Ibf sec sec		1.145.155 lbm 100.5 ft 17.7 ft 1.054.862 lbm 90.261 lbm 80.697 lbm 2 r RD-180 933.400 lbf 338.4 sec 191.2 sec	1.144.160 lbm 88.9 ft 17.7 ft 1.054.862 lbm 98.265 lbm 79.702 lbm 2 / RO-180 933.400 lbf 338.4 sec 197.6 sec	1.144.150 lbm 100.5 ft 17.7 ft 1.054.882 lbm 89.256 lbm 79.692 lbm 2 / RD-180 933.400 lbf 338.4 sec 198.2 sec					
First/Core Stage Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Dry Mass Core Mas	Ibm ft ft Ibm Ibm Ibm Ibm - Ibf sec sec		1.148.814 lbm 96.0 ft 17.7 ft 1.054.852 lbm 93.929 lbm 84.366 lbm 93.3400 lbf 333.400 lbf 338.4 sec 275.9 sec	1.156.803 lbm 96.0 ft 17.7 ft 1.054.852/bm 101.918 lbm 92.354 lbm 2 / RD-180 933.400 lbf 338.4 sec 279.1 sec	1,157,531 lbm 98.0 ft 17.7 ft 1,054,852 lbm 102,646 lbm 93,082 lbm 2 / RD-180 933,400 lbf 338,4 sec 279,4 sec		1.288,047 lbm 13.24 ft 12.2 ft 1.112.256 lbm 175,791 lbm 1 / 4 seq. SRM 3.139,106 lbf 268.8 sec 145.3 sec	1,300,305 lbm 133.2 ft 12.2 ft 1.112.256 lbm 188,049 lbm 1 / 4 seq. SRB 3,139,106 lbf 268.8 sec 145.3 sec	1,300,305 lbm 133.2 ft 12.2 ft 1.112.256 lbm 188,049 lbm 1 / 4 seq. SRM 3,139,106 lbf 268.8 sec 143.3 sec	1.299.827 lbm 133.2 ft 12.2 ft bm 187.572 lbm 17.572 lbm 1/4 seq. SRM 3.139.106 lbf 268.8 sec 145.3 sec
Second Stage Gross Mass Lenoth Usable Procellant Mass Burnout Mass Drv Mass Macun Specific Impulse @ 100% Vacuum Specific Impulse @ 100% Usable Propellant Mass Burnout Mass Evander Mass Burnout Mass Cross Mass Lenoth Diameter Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Vacuum Specific Impulse @ 100% Vacuum Specific Impulse @ 100%	Ibm ft ft Ibm Ibm Ibm sec sec Ibm ft ft ft Ibm Ibm Ibm Ibm Ibm Ibm Ibm Ibm Ibm	1	188.541 lbm 47.1 ft 17.7 ft 156.586 lbm 31.948 lbm 25.825 lbm 47.1.00 47.1.00 48.50 sec 304.5 sec GEM-60 Solid Strapon 73.950 lbm 52.0 ft 5.0 ft 5.800 lbm 8.100 lbm 4 (GEM-60 260.764 lbf @ 1.0 sec 91.0 sec 91.0 sec	186.630 lbm 49.4 ft 17.7 ft 155.198 lbm 31.425 bm 23.932 bm 47.000 48.000 49.000 49.000 50.0000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.0000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.0000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.0000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.0000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.0000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.0000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.000 50.0000 50.000 50.000 50.00000 50.00	186.608 lbm 53.4 ft 17.7 ft 154.747 lbm 31.944 bm 24.004 bm 47.18.00 46.00 sec 465.0 sec 299.9 sec		298.577 lbm 82.0 ft 16.4 ft 254.766 lbm 38.617 lbm 33.627 lbm 41 RL-106 450.0 sec 287.6 sec	405.882 lbm 105.0 ft 16.4 ft 380.519 lbm 45.022 lbm 38.597 lbm 17.53Me Bit III 46.21 sec 333.4 sec	405.882 lbm 105.0 ft 164. ft 380.570 lbm 44.972 bm 38.597 bm 17 SSME BLK II 452.1 sec 333.9 sec	264.989 lbm 82.0 ft 16.4 ft 233.219 lbm 31.444 bm 27.077 bm 17.4 250+ 17.4 250+ 451.5 sec 383.2 sec
Pavload Shroud Overall Lencith Overall Lencith Ordiscle Diameter Drammic Diameter Overall Mass Jettison Mass Jettison Mass Jettison Time  Launch Escape System Gross Mass Lencith Diameter Usable Procellant Mass Burnout Mass Durnout Mass Durnout Mass Vacuum Thinst @ 100% Acceleration	ft ft ft lbm lbm sec lbm ft ft lbm lbm lbm lbm lbm sec sec sec sec		100.6 ft 24.3 ft 22.0 ft 21.3 ft 22.605 bm 22.695 bm 314.5 sec	150.3 ft 27.5 ft 96.4 ft 24.5 ft 93.48 lbm 59.348 lbm 279.1 sec	82.0 R 27.5 R 82.0 R 24.5 R 30.543 bm				24.5 ft 18.4 ft 46.6 ft 13.4 ft 8.441 lbm 8.441 lbm 299.0 sec	
Jettison Mass Jettison Time	lbm sec				9,300 lbm		9,172 lbm 175.3 sec	9,300 lbm 175.3 sec		9,300 lbm 175.3 sec
Flight Parameters Maximum Dvnamic Pressure Time Maximum Acceleration Time Maximum Altitude During Ascent	g's		488 psf 66.5 sec 4.09 q	607 psf 74.6 sec 4.00 g	615 psf 74.2 4.00 g		782 psf 55.5 sec 4.47 g	576 psf 59.2 sec 4.00 g	576 psf 58.5 sec 4.00 g	801 psf 54.4 sec 3.45 g

	Overall Vehicle Height. ft	200	271.3	270.0	309.4 52.2' 16.4' 174.2'	287.9 19.7 19.7 19.7 19.7 19.7 19.7 19.7	311.8 52.2 16.4 85.4 311.8 174.2	99.6 99.6 91.6 91.6 91.6 91.6 91.6 91.6	264.0 52.2* 16.4*	762 0 9) 6 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1
		50	133.2	133.2					122.6	133.2
			14.1	14.3	15	15.1	16	16.1	17	17.1
Vehicle Name			4-Segment SRB with 1 J- 2S+ (5.5-m) Crew	4-Segment SRB with 2 J- 2S+ (7.25 m) Crew	5-Segment SRB with 4 LR-85 Crew	In-line 5-Segment SRM & 4 LR-85 (6 m) - Crew	5-Segment SRB with 1 J-2S+ Crew	5-Segment SRB with 1 J-2S+ (5.5 m) Crew	4-Segment RSRB with 1 J- 2S- Crew	4-Segment RSRB with 1 J- 2S (5.5 m)- Crew
LV Delivery Orbit 30X160 nmi Payload 28.5 Dea Inc										
Lift Capability Net Payload 51.6 Deg Inc	mT		21.1 mT 19.0 mT 19.8 mT	24.4 mT 22.0 mT 22.6 mT	29.9 mT 27.0 mT 28.1 mT	29.8 mT 26.8 mT 27.9 mT	28.7 mT 25.8 mT 27.0 mT	29.0 mT 26.1 mT 27.2 mT	20.3 mT 18.3 mT	18.6 mT 16.8 mT 17.4 mT
Lift Capability Net Payload General Parameters	mT mT		19.8 mT	22.6 mT 20.4 mT	28.1 mT 25.3 mT	27.9 mT 25.1 mT	27.0 mT 24.3 mT	27.2 mT 24.5 mT	18.9 mT 17.0 mT	17.4 ml 15.7 mT
Overall Height Gross Liftoff Mass Liftoff Thrust/Weight Second Stage Thrust/Weight Safety / Reliability Loss of Mission (LOM) - Mean	ft Ibm a g		271.3 ft 1.651.262 lbm 1.48 a 0.78 g	270.0 ft 1.838.553 lbm 1.33 a 1.02 g	309.4 ft 2.029.128 lbm 1.77 a 0.91 g	287.9 ft 2.075.635 lbf 1.73 a 0.79 g	311.8 ft 2.014.084 lbm 1.78 a 0.77 g	315.6 ft 2.047.625 lbm 1.75 a 0.70 g	264.0 ft 1.617.781 lbf 1.51 a 0.80 g	262.9 ft 1.623.852 lbf 1.51 a 0.81 g
LOM - Median (50%) Loss of Crew (LOC) - Mean					1 in 1429		1 in 1,918			
LOC - Median (50%) Strapon Boosters (each) Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Dry Mass Vacuum Threat @ 100% Vacuum Scedic Impulse @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm - Ibf sec sec									
First/Core Stage Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Engine(s)Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm Ibm Ibm sec sec		1,300,464 lbm 133.2 ft 12.2 ft 1.112.256 lbm 188,208 lbm 1./4 seq. SRB 3,139,106 lbf 268.8 sec 145.0 sec	1,304,214 lbm 133.2 ft 12.2 ft 1.112.256 lbm 191,958 lbm 1/4 seq. SRB 3,139,106 lbf 268.8 sec 145.3 sec	1,657,603 lbm 174.2 12.2 ft 1.434.906 lbm 222,697 lbm 1/5 Seq. SRB 3,480,123 lbf 265.4 sec 132.5 sec	1,646,258 lbm 162.0 ft 12.2 ft 1.434,906 lbm 211,352 lbm 1.75 seq SRM 3,480.123 lbf 2654 sec 132.5 sec	1,658,283 lbm 174.2 ft 12.2 ft 14.44.906 lbm 223,377 lbm 1/5 seq. SRB 3,480,123 lbf 265.4 sec 132.5 sec	1,658,870 lbm 174.2 ft 12.2 ft 1.434.906 lbm 223,964 lbm 1 / 5 seg. SRB 3,480,123 lbf 265.4 sec 132.5 sec	1,287,082 lbm 132.4 ft 12.2 ft 1.112.256 lbm 174,827 lbm 1.74 seq. SRM 3,139,106 lbf 268.8 sec 145.3 sec	1,299,385 lbm 133.2 ft 12.2 ft 1.112,256 lbm 187,130 lbm 1.4 seq. SRM 3,139,106 lbf 268.8 sec 145.3 sec
Second Stage Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Engine(s) Motor Vacuum Thrist & 100% Vacuum Thrist & 100% Vacuum Specific Impulse & 100% Burn Time	Ibm ft ft Ibm Ibm Ibm Ibm Ibm - Ibf sec sec		295,060 lbm 78.5 ft 18.0 ft 260.018 lbm 34,712 lbm 29,856 lbm 1 /J-2S+ 274,500 lbf 451.5 sec 428.0 sec	471,270 lbm 77.2 ft 23.8 ft 415.089 lbm 55.833 lbm 48,786 lbm 2 / J-28+ 274,500 lbf 451,5 sec 341.4 sec	296,198 lbm 83.0 ft 16.4 ft 260.093 lbm 36,075 lbm 30.966 lbm 4 / RL-85 rub 85,000 lbf 450.0 sec 344.3 sec	359,593 lbm 73.7 ft 19.7 ft 310.166 lbm 44,253 lbm 38,286 lbm 4 RL-85 Rubbertzed 85,000 lbf 450.0 sec 410.5 sec	283,184 lbm 85.4 ft 16.4 ft 250.193 lbm 32,663 lbm 27,780 lbm 1 / J-28+ 274,500 lbf 451.5 sec 411.1 sec	315,554 lbm 81.8 ft 18.0 ft 280.001 lbm 35,221 lbm 29,979 lbm 1 / J-2S+ 274,500 lbf 451.5 sec 460.6 sec	280,941 lbm 79.4 ft 16.4 ft 244,203 lbm 32,511 lbm 27,916 lbm 1 / J-28 265,000 lbf 436.0 sec 401.8 sec	274.124 lbm 70.1 ft 18.0 ft 240.013 lbm 33.785 lbm 29.214 lbm 17.1-28 265.000 lbf 436.0 sec 394.9 sec
Third Stage Gross Mass Lenath Diameter Usable Propellant Mass Burnout Mass Drv Mass Enaine(s) Mydor Vacuum Thrust @ 100%	Ibm ft ft Ibm Ibm Ibm									
Vacuum Seceffic Imoulse @ 100% Burn Time  Payload Shroud Overall Lenoth Outside Diameter Dynamic Lenoth Dynamic Lenoth Overall Mass Jettison Mass Jettison Time	sec sec ft ft ft ft lbm lbm sec									
Launch Escape System Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Engine(s) Motor Vacuum Thrust @ 100% Vacuum Seedifc Imoulse @ 100% Acceleration	Ibm ft ft Ibm Ibm Ibm Ibm - Ibf sec o's		9,300 lbm	9,300 lbm	9,300 lbm					
Burn Time Jettison Mass Jettison Time	sec Ibm sec		9,300 lbm 175.0 sec	9,300 lbm 175.3 sec	9,300 lbm 162.5	9,172 lbm 163.0 sec	9,300 lbm 162.5 sec	9,300 lbm 162.5 sec	9,172 lbm 175.3 sec	9,300 lbm 175.3 sec
Flight Parameters Maximum Dvnamic Pressure Time Maximum Acceleration Time Maximum Altitude During Ascent	sec g's sec		732 psf 54.4 sec 3.37 g	474 psf 60.9 sec 4.99 g	986 psf 48.4 sec 3.45 g	892 psf 48.3 sec 3.24 g	994 psf 47.3 sec 3.53 g	924 psf 47.2 sec 3.36 g	801 psf 54.0 sec 3.43 g	779 psf 53.9 sec 3.53 g

	Overall Vehicle Height, ft	400 350 300 250 200 150 100	293.1 59.5 110.3 110.3 12.3 12.7 133.2	294.9 16.4' 90.7' 291.9' 12.2'	320.6 52.2 105.0 105	v	183.8	183.8		183.8 183.7 183.7 183.7	
			17.2	18	19		20	21		22	
			4-Segment RSRB with 2 J- 2S (5.5 m)- Crew	-5-Segment RSRB with 1 J-2S - Crew	5-Segment RSRB with 1 SSME - Crew		4-Segment SRM Side- mount SDV	5-Segment SRM Side- mount SDV		Shuttle-Derived Side-mount 4- Seg. SRM & 2 RS-68	
Vehicle Name  LV Delivery Orbit 30X160 nmi Payload	Units										
28.5 Dea Inc  Lift Capability  Net Payload  51.6 Deg Inc	mT mT		25.3 mT 22.8 mT	26.8 mT 24.2 mT	35.9 mT 32.4 mT		78.5 mT 66.7 mT 73.1 mT	93.5 mT 79.5 mT		52.7 mT 44.8 mT	
Lift Capability Net Payload  General Parameters	mT		23.6 mT 21.2 mT	25.1 mT 22.6 mT	33.8 mT 30.4 mT		62.2 mT	87.5 mT 74.4 mT		47.9 mT 40.7 mT	
Overall Height Gross Liftoff Mass Liftoff Thrust/Weight Second Stage Thrust/Weight Safety / Reliability	ft Ibm a g		293.1 ft 1.813.730 lbf 1.35 a 1.03 g	294.9 ft 1.996.987 lbf 1.80 a 0.75 g	320.6 ft 2.152.961 lbf 1.77 a 0.99 g		183.8 ft 4.544.392 lbm 1.52 a 0.97 g	183.8 ft 5.294.308 lbm 1.57 a 0.94 g		183.8 ft 4.492.706 lbf 1.58 a 1.05 g	
Loss of Mission (LOM) - Mean LOM - Median (50%) Loss of Crew (LOC) - Mean					1 in 296 1 in 1359		1 in 173	1 in 172			
LOC - Median (50%) Strapon Boosters (each) Gross Mass Lenoth Diameter Diameter Diameter Down Mass Burnout Mass Burnout Mass Ennine(s)/Motor Vacuum Thrust @ 100% Vacuum Thrust @ 100% Burn Time	Ibm ft ft lbm lbm lbm - lbf sec sec						1,298,467 lbm 149,6 ft 12.2 ft 1,111,019 lbm 186,863 lbm 2 / 4 seg. SRM 3,139,106 lbf 268.8 sec 125.4 sec	1,656,140 lbm 176.7 ft 12.2 ft 1,434,906 lbm 221,234 lbm 2,7 seg. SRM 3,480,123 lbf 265.4 sec 132.5 sec		1.298.467 lbm 149.6 ft 12.2 ft 1.111.019 lbm 186.863 lbm 2./4 seq SRM 3.139.106 lbf 288.8 sec 125.4 sec	
First/Core Stage Gross Mass Lenoth Diameter Usable Procellant Mass Burnout Mass Dry Mass Crayine(5) Motor Vacuum Thrust (2) 100% Vacuum Specific Impulse (2) 100% Burn Time Second Stage	Ibm ft ft Ibm Ibm Ibm Ibm sec sec		1,300,607 lbm 133.2 ft 12.2 ft 1.112.256 lbm 188,351 lbm 1/4 seas. SRM 3,139,106 lbf 288.6 sec 145.3 sec	1,645,215 lbm 162.0 ft 12.2 ft 1.434.906 lbm 210,309 lbm 1 / 5 seq SRM 3,480,123 lbf 2654 sec 132.5 sec	1,658,326 ibm 163.4 ft 12.2 ft 1,434 306 ibm 223,420 ibm 1,5 Sept SRM 3,480,123 bf 265.4 sec	Tankage	1,775,661 lbm 156.9 ft 27.5 ft lbm 159.046 lbm 155,266 lbm 168,364 lbm 3 I SSME lbit I 469,710 lbf 452,2 sec 334.0 sec	1,775,882 lbm 156.9 ft 27.5 ft bm 155.86 636 lbm 155.487 lbm 168.585 lbm 37 SSME lbit II 469,710 lbf 452,2 sec 334.0 sec	Tankage	1,780,793 lbm 156,9 ft 27,5 ft 1,589,533 lbm 161,817 lbm 174,915 lbm 2 / RS-68 746,000 lbf 409,5 sec 476,0 sec	
Gross Mass Lenoth Diameter Lenoth Diameter Lesoth Burnout Mass Burnout Mass Drow Mass Ensinets Whotor Ensinets Whotor Vacuum Specific Impulse @ 100% Burn Time Third Stage Gross Mass	Ibm ft ft Ibm Ibm Ibm Ibm Ibm sec sec		448.063 bm 100.3 ft 18.0 ft 400.007 bm 48.012 lbm 41.248 lbm 2 / 1-25 265.000 lbf 436.0 sec 329.1 sec	283.415 lbm 80.7 ft 16.4 ft 250.010 lbm 33.378 lbm 28.401 lbm 1/J-2S 265.000 lbf 436.0 sec 411.4 sec	406.091 lbm 105.0 ft 16.4 ft 360.115 lbm 45.635 lbm 38.805 lbm 1 SSME BLK II 469.449 lbf 452.1 sec 331.0 sec	Usable Procellant Burnout Mass Drv Mass	1.590.946 lbm 68.297 lbm 59.226 lbm	1.588.636 lbm 68.518 lbm 59.447 lbm	Usable Propellant Burnout Mass Drv Mass	1.589.533 lbm 68.570 lbm 59.499 lbm	
Length Diameter Usable Propellant Mass Burnout Mass Drow Mass Endinels Whotor Vacuum Tinust @ 100% Burn Time	Ibm ft ft Ibm Ibm Ibm Ibm Ibm Sec Sec					Burnout Mass Drv Mass Enaine(s)/Motor Vacuum Thrust Vacuum Specific Impulse	86.969 lbm 109.138 lbm 3 / SSME Blik II 469.710 lbf 452.2 sec	86.969 lbm 109.138 lbm 3 ISSME Blik II 469,710 lbf 452.2 sec	Burnout Mass Drv Mass Endine(s)Motor Vacuum Thrust Vacuum Specific Impulse	93.247 lbm 115.416 lbm 2.7 RS-88 745,000 lbf 409.5 sec	
Pavload Shroud Overall Lendth Outside Diameter Dynamic Lendth Dynamic Biameter Overall Mass Jettison Time Launch Escape System Gross Mass Lendth Diameter Usable Procellant Mass Burnout Mass	ft ft ft lbm lbm sec						125.9 ft 27.0 ft 96.5 ft 24.0 ft 29.295 bm 29.295 bm 237.0 sec	125.9 ft 27.0 ft 96.5 ft 24.0 ft 29.295 bm 29.295 bm 23.30 sec		96.5 ft 24.0 ft 22.225 lbm 263.0 sec	
Dry Mass Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Seedific Imoulse @ 100% Acceleration Burn Time Jettison Mass Jettison Time Flight Parameters	Ibm - Ibf sec a's sec Ibm sec		9,300 lbm 175.3 sec	9,172 lbm 162.5 sec	9,300 lbm 163.0 sec						
Maximum Dvnamic Pressure Time Maximum Acceleration Time Maximum Altitude During Ascent	g's		522 psf 61.0 sec 5.00 g	1.023 psf 47.0 sec 3.61 g	800 psf 54.5 sec 2.93 g		719 psf 53.0 sec 3.00 g	690 psf 60.3 sec 3.00 g		797 psf 52.0 sec 3.06 g	

	Overall Vehicle Height, ft	350 350 300 250 200 183.8 100	Thu. F	315.9 62.2 150.0 150.0 171.7	303.2 31.7 131.7 131.7 131.7 131.7 131.7	155.0 154.7 176.7 176.7	342.3 27.5 131.5 176.7 210.6	31.1 31.1 352.4 352.4 210.8°	
			ļ			1			
		Shuttle-Deri	23 erived Side-	24 4-Segment SRB In-line SDV	25 4-Segment SRB In-line SDV	26 / 5-Segment SRB In-line SDV	27	27.1 5-Segment	
Vehicle Name		mount – 5-Seg	Seg. SRM & 2 RS- 68	4-Segment SRB In-line SDV Crew + Cargo	4-Segment SRB In-line SDV Cargo	5-Segment SRB In-line SDV Crew +Cargo	5-Segment SRB In-line SDV Cargo	SRB In-line SDV - Cargo 8.5-m P/L Envelope Shroud	
LV Delivery Orbit 30X160 nmi Payload 28.5 Dea Inc	Units				_	_			
Lift Capability Net Payload  51.6 Deg Inc Lift Capability	mI	54	64.3 mT 54.6 mT 59.4 mT	82.1 mT 73.9 mT 77.0 mT	86.5 mT 73.5 mT 81.1 mT	107.4 mT 91.3 mT 100.3 mT	113.8 mT 96.7 mT 106.8 mT	112.5 mT 95.6 mT Not Evaluated	ı
Net Payload	mT	50	59.4 mT	69.3 mT	69.0 mT	85.3 mT	106.8 mT	Not Evaluated Not Evaluated	l .
General Parameters Overall Heicht Gross Liftoff Mass Liftoff Thrust/Weicht Second Stage Thrust/Weight	ft Ibm a g	5.234	183.8 ft 234.966 lbf 1.62 a 1.05 g	315.9 ft 4.537.794 lbm 1.54 a 0.96 g	303.2 ft 4.545.168 lbm 1.54 a 0.95 g	355.0 ft 5.984.103 lbm 1.47 a 0.93 g	342.3 ft 5.993.890 lbm 1.46 a 0.93 g	352.4 ft 6.003.780 lbm 1.46 a 0.93 q	
Safety / Reliability Loss of Mission (LOM) - Mean LOM - Median (50%).		_		1 in 164	1 in 176	1 in 124	1 in 133	1 in 133 This vehicle reliability was not run but assumed the same as vehcile 27	ı
Loss of Crew (LOC) - Mean LOC - Median (50%)				1 in 1170		1 in 915		assumed the same at volume 2.	1
Strapon Boosters (each) Gross Mass	lbm ft	1.65	556.140 lbm	1.298.467 lbm	1.298.467 lbm	1.656.140 lbm	1.656.140 lbm	1.656.140 lbm	
Lenath Diameter Usable Procellant Mass	ft Ibm	1 1.434	176.7 ft 12.2 ft I34.906 lbm	150.0 ft 12.2 ft 1.111.019 lbm	150.0 ft 12.2 ft 1.111.028 lbm	176.7 ft 12.2 ft 1.434.906 lbm	176.7 ft 27.5 ft 1.434.906 lbm	176.7 ft 12.2 ft 1.434.906 lbm	
Burnout Mass Dry Mass Engine(s)/Motor	lbm lbm -	2/5	21,234 lbm 5 Seg SRM	186,863 lbm 2 / 4 seg SRM	186,854 lbm 2 / 4 seg SRM	221,234 lbm 2 / 5 seg SRM	221,234 lbm 2 / 5 seg SRM	221,234 lbm 2 / 5 seg SRM	
Vacuum Thrust @ 100% Vacuum Soecific Impulse @ 100% Burn Time	lbf sec sec	3.480 265	480.123 lbf 265.4 sec 132.5 sec	3.139.106 lbf 268.8 sec 123.0 sec	3.139.106 lbf 268.8 sec 125.4 sec	3.480.123 lbf 265.4 sec 132.5 sec	3.480.123 lbf 265.4 sec 132.5 sec	3.480.123 lbf 265.4 sec 132.5 sec	
First/Core Stage Gross Mass	lbm	1,781	781,014 lbm	1,751,697 lbm	1,735,243 lbm	2,425,833 lbm	2,407,408 lbm	2,407,408 lbm	
Length Diameter Usable Propellant Mass	ft ft Ibm	15 2 1.589	156.9 ft 27.5 ft 589.533 lbm	171.7 ft 27.5 ft 1.588.636 lbm	171.7 ft 27.5 ft 1.588.636 lbm	210.8 ft 27.5 ft 2.210.023 lbm	210.8 ft 27.5 ft 2.210.112 lbm	210.8 ft 27.5 ft 2.210.112 lbm	
Burnout Mass Dry Mass Engine(s)/Motor	lbm lbm	162, 175, 2/	32,038 lbm 75,136 lbm 2 / RS-68	160,686 lbm 149,268 lbm 3 / SSME BLK II	144,232 lbm 132,814 lbm 3 / SSME BLK II	215,560 lbm 186,231 lbm 4 / SSME BLK II	197,046 lbm 167,806 lbm 4 / SSME BLK II	197,046 lbm 167,806 lbm 4 / SSME BLK II	
Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Burn Time	lbf sec sec	745 409	45,000 lbf 409.5 sec 470.0 sec	469,449 lbf 452.1 sec 488.0 sec	469,449 lbf 452.1 sec 488.3 sec	469,449 lbf 452.1 sec 509.0 sec	469,449 lbf 452.1 sec 509.5 sec	469,449 lbf 452.1 sec 509.5 sec	
Second Stage			0.0 3.3		33.3	355.0 222	555.4 111	505.2.2.2	
Gross Mass Length Diameter	lbm ft ft	4	589.533 lbm						
Usable Propellant Mass Burnout Mass Drv Mass	lbm lbm lbm	68,7	89.533 lbm 8,791 lbm 9.720 lbm						
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100%	lbf sec								
Burn Time Third Stage	sec								
Gross Mass Length Diameter	lbm ft ft								
Usable Propellant Mass Burnout Mass Drv Mass	Ibm Ibm Ibm	93	3.247 lbm 15.416 lbm						
Dry Mass Engine(s)/Motor Vacuum Thrust @ 100%	Ibm	21	15.416 lbm 2 / RS-68 45,000 lbf						
Vacuum Specific Impulse @ 100% Burn Time	sec sec	40'	409.5 sec						
Payload Shroud Overall Length	ft			82.0 ft	131.5 ft	82.0 ft	131.5 ft	141.6 ft	
Outside Diameter Dynamic Length Dynamic Diameter	ft ft		96.5 ft 24.0 ft	27.5 ft 82.0 ft 24.5 ft	27.5 ft 98.4 ft 24.5 ft	27.5 ft 82.0 ft 24.5 ft	27.5 ft 98.4 ft 24.5 ft	31.1 ft 98.4 ft 28.1 ft	
Overall Mass Jettison Mass Jettison Time	Ibm Ibm sec	29.2	9.295 lbm 297.0 sec		23,419 lbm 23.419 lbm 240.9 sec		23,404 lbm 23,404 lbm 250.0 sec	36.136 lbm 36.136 lbm 250.0 sec	
Launch Escape System Gross Mass	lbm		7.0 000	9,300 lm	#:	9,300 lbm	<del></del>		
Lenath Diameter	ft ft			9,300 IIII		9,300 ווונוו			
Usable Propellant Mass Burnout Mass Dry Mass	lbm lbm lbm								
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100%	lbf sec								
Acceleration Burn Time Jettison Mass	a's sec Ibm			9,300 lbm		9,300 lbm			
Jettison Time Flight Parameters	sec			47.9 sec		162.5 sec			
								500 5	
Maximum Dvnamic Pressure Time Maximum Acceleration	psf sec g's	59	768 psf 59.0 sec 3.00 g	718 psf 58.0 sec 4.00 g	704 psf 55.4 sec 4.00 g	563 psf 70 sec 4.00 g	562 psf 69.5 sec 4.00 g	560 psf 70.5 sec 4.00 g	

	Омеля Vohicle Heinht. ft	350 350 300 250 200 150	342.3 27.5 131.5 342.3 210.8		357.6 77.22 71.5°	368.5 27.5 368.5 3	368.5 27.5 131.5 82.9 368.5	399.7 27.5' 131.5' 139.7' 174.5'
			27.3 Cargo Variant		27.3			
			(Old 27.2)  5-Segment SRBs with 5		(With EDS)  5 SSME Core & 5-Segment SRB + 2	28 4 RS-68 Core + 4 J-2S + &	28.1 4 RS-68 Core + 4 J-2S + 8	29 4-Segment SRBs 3 RS-68
Vehicle Name			SSME Core - Cargo	Deliver Orbit TLI (EDS	J2S+ (EDS) Cargo Blk	2 Deltal IV Boosters Cargo	2 Atlas V Boosters Cargo	& 4 J-2S + Cargo
LV Delivery Orbit 30X160 nmi Payload 28.5 Dea Inc				Suborbital Burn) Gross Payload Net Pavload	60.6 mT 54.6 mT			
Lift Capability Net Payload 51.6 Deg Inc Lift Capability	mI		125.1 mT 106.3 mT Not Evaluated		146.6 mT 124.6 mT	58.2 mT 49.5 mT 54.8 mT	64.1 mT 54.5 mT 60.6 mT	108.2 mT 91.9 mT 102.4 mT
Net Payload  General Parameters	mT		Not Evaluated			46.6 mT	51.5 mT	87.1 mT
Overall Height Gross Liftoff Mass Liftoff Thrust/Weight Second Stage Thrust/Weight Safety / Reliability	ft Ibm a g		342.3 ft 6.027.962 lbm 1.52 a 1.23 g		357.6 ft 6.393.975 lbm 1.43 a 0.84 g	368.5 ft 3.207.626 lbm 1.24 a 1.19 g	368.5 ft 3.601.955 lbm 1.22 a 1.17 g	399.7 ft 5.401.018 lbm 1.44 a 1.09 g
Loss of Mission (LOM) - Mean LOM - Median (50%)			1 in 124		1 in 92	1 in 120		
Loss of Crew (LOC) - Mean LOC - Median (50%)								
Strapon Boosters (each) Gross Mass Length	lbm ft		1,656,140 lbm 176.7 ft		1,656,140 lbm 176.7 ft	430,005 lbm 154.1 ft	620,313 lbm 119.2 ft	1,297,882 lbm 150.0 ft
Diameter Usable Propellant Mass Burnout Mass Dry Mass	ft Ibm Ibm Ibm		12.2 ft 1,434,906 lbm 221,234 lbm		12.2 ft 1,434,906 lbm 221,234 lbm	16.4 ft 368,175 lbm 61,816 lbm 55.806 lbm	12.5 ft 566,207 lbm 54,089 lbm 48,069 lbm	12.2 ft 1,111,028 lbm 186,854 lbm
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Burn Time	Ibf sec sec		2 / 5 Segment SRM 3,480,123 lbf 265.4 sec 132.5 sec		2 / 5 Segment SRM 3,480,123 lbf 265,4 sec 132.5 sec	1 / RS-68 743,031 lbf 408.8 sec 198.7 sec	1 / RD-180 933,400 lbf 338.4 sec 205.0 sec	2 / 4 Segment SRM 3,139,106 lbf 268.8 sec 125.4 sec
First/Core Stage Gross Mass	lbm		2,416,596 lbm		2.430.894 lbm	1,426,174 lbm	1,426,761 lbm	1,774,828 lbm
Lenoth Diameter Usable Propellant Mass	ft ft Ibm		210.8 ft 27.5 ft 2 215 385 lbm		210.8 ft 27.5 ft 2.215.385 lbm	154.1 ft 27.5 ft 1.232.983 lbm	154.1 ft 27.5 ft 1.232.983 lbm	174.5 ft 27.5 ft 1.582.392 lbm
Burnout Mass Dry Mass Engine(s)/Motor	lbm lbm -		200,960 lbm 180,698 lbm 5 / SSME Blk II		215,258 lbm 194,997 lbm 5 / SSME Blk II	193,065 lbm 182,957 lbm 4 / RS-68	193,652 lbm 183,545 lbm 4 / RS-68	192,274 lbm 178,584 lbm 3 / RS-68
Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Burn Time	lbf sec sec		469,449 lbf 452.1 sec 411.5 sec		469,449 lbf 452.1 sec 408.2 sec	743,031 lbf 408.8 sec 239.4 sec	743,031 lbf 408.8 sec 242.0 sec	743,031 lbf 408.8 sec 284.7 sec
Second Stage Gross Mass	lbm				506,576 lbm	769,768 lbm	769,847 lbm	768,710 lbm
Lenoth Diameter Usable Propellant Mass	ft ft Ibm				74.6 ft 27.5 ft 457.884 lbm	82.9 ft 27.5 ft 668.956 lbm	82.9 ft 27.5 ft 668.683 lbm	93.7 ft 27.5 ft 666.690 lbm
Burnout Mass Drv Mass Engine(s)/Motor	-				48,640 lbm 42,645 lbm 2 / J-2S+	100,737 lbm 87,583 lbm 4 / J-2S+	101,089 lbm 87,664 lbm 4 / J-2S+	101,945 lbm 86,542 lbm 4 / J-2S+
Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Burn Time	lbf sec sec				274,500 lbf 451.5 sec 217.7 sec	274,500 lbf 451.5 sec 275.1 sec	274,500 lbf 451.5 sec 275.0 sec	274,500 lbf 451.5 sec 274.1 sec
Third Stage Gross Mass	lbm							
Length Diameter Usable Propellant Mass	ft ft Ibm							
Burnout Mass Drv Mass Engine(s)/Motor	lbm lbm							
Vacuum Thrust @ 100%  Vacuum Specific Impulse @ 100%  Burn Time	sec sec							
Payload Shroud Overall Length	ft		131.5 ft		72.2 ft	131.5 ft	131.5 ft	131.5 ft
Overall Length Outside Diameter Dynamic Length Dynamic Diameter	ft ft ft		131.5 ft 27.5 ft 98.4ft 24.5 ft		72.2 ft 27.5 ft 39.4 ft 24.5 ft	131.5 ft 27.5 ft 98.4ft 24.5 ft	131.5 ft 27.5 ft 98.4 ft 24.5 ft	131.5 ft 27.5 ft 98.4 ft 24.5 ft
Overall Mass Jettison Mass Jettison Time	Ibm Ibm sec		24.5 ft 23,362 lbm 23,362 lbm 293.5 sec		24.5 ft 10,522 lbm 10,522 lbm 447.0 sec	24.5 ft 23,387 lbm 23,387 lbm 336.0 sec	24.5 ft 23,387 lbm 23,387 lbm 338.0 sec	24.5 ft 23,274 lbm 23,274 lbm 269.8 sec
Launch Escape System	sec		203.3 986		997.U SEU	Jou.U SEC	330.U SEC	200.0 380
Gross Mass Lenath Diameter Lenath Propollingt Mass	ft ft							
Usable Propellant Mass Burnout Mass Dry Mass	Ibm Ibm Ibm							
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100%	lbf sec							
Acceleration Burn Time Jettison Mass	a's sec Ibm							
Jettison Time Flight Parameters	sec							
Maximum Dvnamic Pressure Time Maximum Acceleration	g s		661 psf 67.0 sec 4.00 g		561 psf 72.7 sec 2.32 g	175 psf 127.8 sec 4.79 g	196 psf 127.0 sec 4.53 g	610 psf 62.6 sec 3.23 g
Time Maximum Altitude During Ascent						•		
<u> </u>								

		400	399.8	398,3 🛆 🕈 🕈	399.8 △ ↑↑	399.0
	Overall Vehicle Height, ft	350 300 250 200 150 100	27.5'	27.5° • 131.5° • 131.5° • 398.3° • 6.7° • 218.8°	27.5°	62.2 70.0 27.5 4 399.0 176.7
			30	30.1	30.3	31
			5 SSME Core + 4 J-2S+ & 5-Segment SRB Cargo	5 SSME Core + 1 SSME & 5-Segment SRB Cargo Blk 2	5 SSME Core + 2 J-2S+ & 5- Segment SRB Cargo	5 SSME Core + 1 SSME & 5-Segment SRB Crew + Cargo Blk 2
Vehicle Name  LV Delivery Orbit 30X160 nmi Payload	Units					
28.5 Dea Inc Lift Capability Net Payload	mT mT		137.3 mT 116.7 mT	140.9 mT 119.8 mT	141.6 mT 120.3 mT	136.3 mT 115.9 mT
51.6 Deg Inc  Lift Capability  Net Payload	mT		130.4 mT 110.8 mT	134.0 mT 113.9 mT	137.2 mT 116.6 mT	126.8 mT 107.8 mT
General Parameters Overall Heicht Gross Liftoff Mass Liftoff Thrust/Weicht Second Stage Thrust/Weight Safety / Reliability	ft Ibm a g		399.8 ft 6.395.446 lbm 1.43 a 1.78 g	398.3 ft 6.387.809 lbm 1.44 a 0.80 g	399.8 ft 6.390.767 lbm 1.43 a 0.89 g	399.0 ft 6.379.499 lbm 1.44 a 0.78 a
Loss of Mission (LOM) - Mean LOM - Median (50%)			1 in 81	1 in 95		1 in 87
Loss of Crew (LOC) - Mean LOC - Median (50%)						1 in 582
Strapon Boosters (each) Gross Mass	lbm		1,656,140 lbm	1,656,140 lbm	1.656.140 lbm	1.656.140 lbm
Lenath Diameter Usable Propellant Mass	ft ft Ibm		176.7 ft 12.2 ft 1,434,906 lbm	176.7 ft 12.2 ft 1,434,906 lbm	176.7 ft 12.2 ft 1.434.906 lbm	176.7 ft 12.2 ft 1.434.906 lbm
Burnout Mass Dry Mass	lbm lbm		221,234 lbm	221,234 lbm	221,234 lbm	221,234 lbm
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100% Burn Time	lbf sec sec		2 / 5 Segment SRM 3,480,123 lbf 265.4 sec 132.5 sec	2 / 5 Segment SRM 3,480,123 lbf 265.4 sec 132.5 sec	2 / 5 Segment SRM 3.480.123 lbf 265.4 sec 132.5 sec	2 / 5 Segment SRM 3.480.123 lbf 265.4 sec 132.5 sec
First/Core Stage Gross Mass	lbm		2,441,382 lbm	2,440,768 lbm	2,441,382 lbm	2,440,768 lbm
Lenath Diameter	ft		210.8 ft 27.5 ft	210.8 ft 27.5 ft	210.8 ft 27.5 ft 2.215.385 lbm	210.8 ft 27.5 ft 2.215.385 lbm
Usable Propellant Mass Burnout Mass Dry Mass	lbm lbm lbm		2.215.385 lbm 225,745 lbm 205.484 lbm	2.215.385 lbm 225,132 lbm 204.871 lbm	225,745 lbm 205.484 lbm	225,132 lbm 204,871 lbm
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100%	lbf sec		5 / SSME Blk II 469,449 lbf 452.1 sec			
Burn Time	sec		408.2 sec	408.0 sec	408.2 sec	408.2 sec
Second Stage Gross Mass	lbm		315,902 lbm	300,797 lbm	301,740 lbm	316.582 lbm
Lenath Diameter Usable Propellant Mass	ft ft Ibm		57.5 ft 27.5 ft 248.758 lbm	56.0 ft 27.5 ft 248.914 lbm	57.5 ft 27.5 ft 248.856 lbm	56.0 ft 27.5 ft 248.796 lbm
Burnout Mass Drv Mass	lbm lbm		67,115 lbm 55,992 lbm	51,855 lbm 40,964 lbm	52,856 lbm 41,900 lbm	67,757 lbm 56.750 lbm
Engine(s)/Motor Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100%	lbf sec		4 / J-2S+ 274,500 lbf 451.5 sec	1 / SSME Blk II 469,449 lbf 452.1 sec	2 / J-2S+ 274,500 lbf 451.5 sec	1 / SSME Blk II 469,449 lbf 452.1 sec
Burn Time	sec		102.2 sec	229.0 sec	204.4 sec	229.2 sec
Third Stage Gross Mass Length	lbm ft					
Diameter Usable Propellant Mass	ft Ibm					
Burnout Mass Drv Mass Engine(s)/Motor	lbm lbm					
Vacuum Thrust @ 100%	lbf					
Vacuum Specific Impulse @ 100% Burn Time	sec					
Payload Shroud Overall Length	ft		131.5 ft	131.5 ft	131.5 ft	70.0 ft
Outside Diameter Dynamic Length Dynamic Diameter	ft ft		27.5 ft 98.4 ft 24.5 ft	27.5 ft 98.4 ft 24.5 ft	27.5 ft 98.4 ft 24.5 ft	27.5 ft 70.0 ft 24.5 ft
Overall Mass Jettison Mass	lbm lbm		23,287 lbm 23,287 lbm	23,287 lbm 23,287 lbm	23,287 lbm 23,287 lbm	included in 2nd Stage
Jettison Time  Launch Escape System	sec		264.0 sec	249.0 sec	314.5 sec	
Gross Mass Length	lbm ft					
Diameter Usable Propellant Mass	ft Ibm					
Burnout Mass Dry Mass Engine(s)/Motor	lbm lbm					
Vacuum Thrust @ 100% Vacuum Specific Impulse @ 100%	lbf sec					
Acceleration Burn Time	a's sec					
Jettison Mass Jettison Time	lbm sec					9,300 lbm 438.2 sec
Flight Parameters Maximum Dynamic Pressure	psf		549 psf	545 psf	554 psf	569 psf
Time Maximum Acceleration	sec g's		72.0 sec 2.97 g	71.5 sec 2.93 g	72.0 sec 2.92 g	72.8 sec 2.88 g
Time Maximum Altitude During Ascent	sec ft					
<u> </u>						_



Section 6B EDS Options

		160				
		140				
		120				
	Overall Vehicle Height, ft	100				
	icle	80			—⊳  16.4'   <del>↓</del> —	16.4'
	Veh		→ 16.4' ←		1	
	Overall	60		Payload		
		40	62.5'		74.8'	78.1'
		20		53.0'		
			881	88 1	88	88
			N1A	N1B	N2A	N2A.1
Vehicle Name			EDS Only w/ No Suborbital Burn	EDS - No Suborbital Burn w/ Payload	EDS Only w/ No Suborbital Burn	EDS Only w/ No Suborbital Burn
Payload Delivery to Moon	Units		25	25	27	27 + 1 additional SSME
Payload TLI Only					·	
Capability	mT		62.0	32.8	84.1	90.0
Net Payload			55.8	29.5	75.7	81.0
TLI + LOI						
Capability	mT		39.0	24.6	53.9	57.9
Net Payload	<u>m</u> T		35.1	22.2	48.6	52.1
TLI + LOI + PC Capability	<u>m</u> T		30.1	20.7	42.3	45.5
Net Payload			27.1	18.6	38.1	40.9
TLI Delivery w/ On-Orbit Refueling Capability	mT					
Net Payload						
Earth Departure Stage  Gross Mass	lbm_		203,159	15 <u>7,</u> 60 <u>6</u>	260,224	275,725
Length	ft		62.5	53.0	74.8	78.1
Diameter Usable Propellant Mass @ Liftoff	ft Ibm		16.4 173,990	16.4 131,500	16 <u>.4</u> 228,182	16.4 242,756
Usable Propellant Mass @ 160 nmi cir.	lbm		170,407	127,910	223,584	237,889
Burnout Mass	lbm		29,150	26,090	32,016	32,941
Dry Mass	lbm		26,928	24,337	29,213	29,984
Engine(s)/Motor Vacuum Thrust	 lbf		4 / RL-85 85,000	4 / RL-85 85,000	4 / RL-85 85,000	4 / RL-85 85,000
Vacuum Specific Impulse	sec		450.0	450.0	450.0	450.0
Burn Time	sec			ļ	ļ <u>-</u>	
Burn Time Suborbital	<del> </del>			'- — — —   	<u> </u>	<u> </u>
Circ	sec		4.75 225.52	4.75	6.08	6.44
TLI Circ	sec sec		4.75	133.87 4.75	295.93 6.08	314.84 6.44
TLI	sec		191.49	133.87	251.28	267.31
LOI Circ	sec sec		34.03 4.75	23. <del>7</del> 9 4.75	44.65 6.08	47.50 6.44
TLI	sec		4.75 178.35	133.87	234.03	248.98
	sec		31.69	23.79	41.58	44.24
LOI			4E 40			
LOI PC LOI+PC Burn time	sec sec		15.48 47.17	11.62 35.41	20.31 61.89	21.61 65.85
LOI			15.48 47.17 95,845 113%	11.62 35.41 56,896 67%	61.89 125,770 148%	

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	į !	140				
		120				
	ght, f	100			→ 16.4' ←	→ 16.4' ←
	吾				10.4	
	Overall Vehicle Height, ft	80	Payload	Payload		
	ő				86.1'	87.3'
		40	62.2'	63.5'		
		20				
	! !		16.4'	16.4		
	! !		N2B	N2B.1	N3A	N3A.1
Vehicle Name	i i		EDS + Payload w/ No Suborbital	EDS + Payload w/ No Suborbital	EDS Only / No Suborbital Burn	EDS Only / No Suborbital Burn
Payload Delivery to Moon	Units		27	27 + 1 additional SSME	30	30.1
Payload TLI Only			 			
Capability	mT		44.2	47.4	103.9	106.2
	<u> </u>			42.6	93.5	
Net Payload	<u>m</u> T		39.8	42.0	95.5	95.6
TLI + LOI	<u>-</u>				<b>_</b>	<u> </u>
Capability	mT_		33.8	36.3	67.2	68.7
Net Payload TLI + LOI + PC	<u>m</u> T		30.4	32.7	60.5	61.9
<u>Capability</u>			28.7	30.9	53.0	54.3
Net Payload	<u>m</u> T		25.8	27.8	47.7	48.8
TLI Delivery w/ On-Orbit Refueling Capability	mT				<b>_</b>	L
Net Payload						
	<u> </u>		ļ.—.—.—.	<b></b>		
Earth Departure Stage			100.007	207.005	242.000	240.007
Gross Mass Length	lbm ft		196,937 62.2	207,635 63.5	313, <u>696</u> 86.1	319,007 87.3
Diameter	ft		16.4	16.4	16.4	16.4
Usable Propellant Mass @ Liftoff	<u>lbm</u>		168,433	178,472	277,883	283,171
Usable Propellant Mass @ 160 nmi cir.	lbm lbm		163,834 28,485	173,599	272,853	277,536 35,804
Burnout Mass Dry Mass	Ibm		26,324	29,143 26,872	35, <u>781</u> 32,459	35,804 32,427
Engine(s)/Motor	 		4 / RL-85	4 / RL-85	4 / RL-85	4 / RL-85
Vacuum Thrust Vacuum Specific Impulse	<u>lbf</u> sec		85,000 450.0	85,000 450.0	85,000 450.0sec	85,000 450.0sec
Burn Time	sec		Ĭ		450.0300	430.0300
Burn Time	:		<del> </del>	, <del> </del>		
Suborbital Circ	sec		6.08 171.48	6.44 181.69	7.31 359.45	7.46
TLI Circ	sec sec		171.48 6.08	181.69 6.44	359.45 7.31	367.30 7.46
TLI	sec		171.48	181.69	305.21	311.88
LOI Circ	sec sec		30.47 6.08	32.29 6.44	54.23 7.31	55.42 7.46
TLI	sec		171.48	181.69	284.26	290.48
LOI	sec sec		30.47 14.88	32.2 <u>9</u> 15.77	50.51 24.67	51.62 25.21
FU			45.35	10.11 40.0E	75.39	76.82
LOI+PC Burn time	sec		45.55	40.00	13.33	70.02
LOI+PC Burn time 200 sec Stage Thrust 200 sec Power Level	sec lbf %		72,878 86%	48.05 77,220 91%	153,186 180%	70.82 156,101 184%

_						
		160				
		140				
		120				
	!   ⊭	120				
		100				
	Overall Vehicle Height,		→ 16.4'	→ 16.4' ←	16.4 &	
	hicle	80	Payload	Payload	10.00	Payload
	!		1	<b>†</b>		
	i yer	60				
	į				73.7	16.4'
	] :	40	69.1'	69.8'	The state of the s	60.2
	i	20				
	į		\ \ \ \ \ \	$RR \downarrow$	XX ↓	<u>₹₹</u>
	l :			<u> </u>		
	! !		N3B	N3B.1	N4A	N4B
Vehicle Name	İ		EDS + Payload No Suborbital Burn	EDS + Payload Attached / No Suborbital Burn	EDS Only w/ No Suborbital Burn	EDS + Payload w/ No Suborbital Burn
Payload Delivery to Moon	Units		30	30.1	7.4	7.4
Payload	Ullits					
TLI Only	   		<b></b>			
Capability	mT		55.0	56.0	83.0	43.8
Net Payload	<u>m</u> T		49.5	50.4	74.7	39.4
TLI + LOI	 					
Capability	mT		42.4	43.2	53.3	33.5
Net Payload	mT		38.2	38.9	48.0	30.2
TLI + LOI + PC Capability			36.3	37.0	41.9	28.6
Net Payload			32.7	33.3	37.7	25.7
TLI Delivery w/ On-Orbit Refueling	 					
Capability Net Payload						
	<b>[</b>					
Earth Departure Stage	F			·		
Gross Mass Length	lbm ft		234,060 69.1	237,460 69.8	255,011 73.7	<u>192,065</u> 60.2
Diameter	ft		16.4	16.4	16.4	16.4
Usable Propellant Mass @ Liftoff	lbm 		203,285	206,485	219,710	165,048
Usable Propellant Mass @ 160 nmi cir. Burnout Mass	<u>lbm</u> lbm		197,735 30,752	200,847 30,952	na 30,772	<u>na</u> 26,999
Dry Mass Engine(s)/Motor	lbm		28,213	28,378 4 / RL-85	28,020 4 / RL-70	24,884
Vacuum Thrust	lbf		4 / RL-85 85,000	85,000	70,000	4 / RL-70 70,000
Vacuum Specific Impulse Burn Time	sec sec		450.0	450.0	450.0	450.0
Burn Time	<u></u> 				l	
Suborbital Circ	sec		7.31	7.46	7.24	7.24
TLI Circ	sec sec		206.18 7.31	210.20 7.46	353.11 7.24	204.05 7.24
TLI LOI	sec sec		206.18 36.64	210.20 37.35	299.83 53.28	204.05 36.26
Circ TLI	sec		7.31	4.46	7.24	7.24
ILI	sec_		206.18 36.64	210.20 37.35	279.25 49.62	204.05 36.26
LOI	sec					— —
LOI PC LOI+PC Burn time	sec		17.89 54.73	18.24 55.59	24.23 73.85	17.71
LOI PC			17.89		24.23 73.85 123,590 177%	17.71 53.96 71,416 102%

		1100		1	1	
	į	160				
		140				
		120				
	ř, ft	100			16.4	<b>→</b>   16.4'   <del> </del>
	Overall Vehicle Height, ft	100	16.4'	16.4'		<b>↑</b>
	nicle	80		Payload		
	l Vel					
	Overa	60				
		40	90.2	78.7'	98.8	96.4'
				10.7		
		20				
			88	88	25 25	55 25 ()
			△△→	<i>∆∆</i> +	$\triangle \triangle \rightarrow$	
				2.5		
			S1A EDS Only w/	S1B EDS + Payload w/	S2A EDS Only w/	S2A.1  EDS Only w/ Suborbital
Vehicle Name			Suborbital Burn	Suborbital Burn	Suborbital Burn	Burn
Payload Delivery to Moon	Units		25	25	27	27
Payload TLI Only	 		¦ 			
Capability	mT		74.3	39.0	95.1	99.0
Net Payload			66.9	35.1	85.6	89.1
TLI + LOI						
Capability	mT		46.6	29.2	60.8	63.5
Net Payload			42.0	26.3	54.7	57.2
TLI + LOI + PC Capability			36.0	24.4	47.5	49.8
Net Payload			32.4	21.9	42.7	44.8
TLI Delivery w/ On-Orbit Refueling	mT				124.4	120.2
Capability Net Payload	mT mT				124.4 112.0	120.2 108.2
			ļ.—.—.—. -—.—.—.		 	
Earth Departure Stage  Gross Mass	lbm		330,057	276,326	370,430	359,208
Length Diameter	ft ft		90.2 16.4	78.7 16.4	98.8 16.4	96.4 16.4
Usable Propellant Mass @ Liftoff	lbm_		294,519	243,736	332,401	321,794
Usable Propellant Mass @ 160 nmi cir. Burnout Mass	lbm lbm		204,859 35,505	154,076 32,562	254,623 37,991	262,784 37,378
Dry Mass	lbm		32,012	29,595	34,112	33,607
Engine(s)/Motor Vacuum Thrust	 lbf		4 / RL-70 70,000	4 / RL-70 70,000	4 / RL-70 70,000	4 / RL-70 70,000
Vacuum Specific Impulse Burn Time	sec_sec_		450.0	450.0	450.0	450.0
Burn Time Suborbital	<b></b> 		137.15	137.15	116.55	86.16
Circ TLI	sec_		6.95 329.24	6.95 195.81	8.46 409.23	8.67 422.32
Circ TLI	sec sec		6.95 329.24	6.95 195.81	8.46 347.48	8.67 358.59
LOI Circ	sec sec		49.68 6.95	34.79 6.95	61.74 8.46	63.72 8.67
TLI LOI	sec sec		329.24 49.68	195.81 34.79	323.63 57.51	333.97 59.34
LOI+PC Burn time	sec		22.59 68.86	16.99 51.79	28.08 85.59	28.98 88.32
200 sec Stage Thrust	sec lbf		115233.82	68533.19 98%	143,229 205%	147,811
200 sec Power Level	%		<u>165%</u>	<u> </u>	<u> </u>	211%

	cle Heiaht, ft	160 140 120 100 80	→ 27.5' ←	→ 27.5' ←	Payload	
		60 40 20	79.3'	82.5'	83.6	
			S2A.2  EDS Only w/	S2A.3  EDS w/ Suborbital Burn No	S2B EDS + Payload w/	
Vehicle Name			Suborbital Burn	Payload	Suborbital	
Payload Delivery to Moon Payload	Units		27	27.3	27	
TLI Only			<u> </u>			
Capability	mT_		102.3	114.9	50.1	!
Net Payload	<u> </u>		92.0	103.4	45.1	
TLI + LOI	İ					
I			j;	72.4		
Capability			61.9	72.1	38.2	
Net Payload TLI + LOI + PC	<u>m</u> T		55.7	64.8	34.3	
Capability	mT		46.3	55.5	32.3	
Net Payload	mT_		41.6	49.9	29.1	
TLI Delivery w/ On-Orbit Refueling Capability	mT_		197.1	i — - — - — - —	97.7	i I
Net Payload			177.4		88.0	
			<b>∤</b>	 		
Earth Departure Stage			·			
Gross Mass Length	lbm ft		598,785 79.3	643,873 82.5	299,190 83.6	 
Diameter	ft		79.3	82.5 27.5	16.4	
Usable Propellant Mass @ Liftoff	<u>lbm</u>		532,811	587,884	265,344	
Usable Propellant Mass @ 160 nmi cir. Burnout Mass	lbm lbm		298,162 65,914	316,527 55,922	187,563 33,816	
Dry Mass	<u>lbm</u>		58,983	48,434	30,623	
Engine(s)/Motor Vacuum Thrust	 lbf		4 J-2S+ 274,500	2 / J-2S+ 274,500	4 / RL-70 70,000	
Vacuum Specific Impulse	sec_		451.5	<u>274,500</u> 451.5	450.0	<u> </u>  -
Burn Time	sec		<u> </u>			
Burn Time Suborbital	<sub>-</sub>		93.81	217.68	116.55	
			2.68	5.49 260.29	8.46 238.37	
Circ	sec		122 60	260.20		
Circ TLI Circ	sec sec		2.68 122.60 2.68	260.29 5.49	8.46	<u> </u>
Circ TLI Circ TLI TLI	sec sec sec		2.68 104.07	<u>5.49</u> 220.96	<u>8.46</u> 238.37	<u> </u> 
Circ   TLI   Circ   TLI   Circ   TLI   CIrc   CIr	sec sec sec sec sec		2.68 104.07 18.52	5.49 220.96 39.33	8.46 238.37 42.36	
Circ TLI Circ TLI LOI Circ TLI LOI Circ	sec sec sec sec sec sec		2.68 104.07 18.52	5.49 220.96 39.33 5.49 205.78	8.46 238.37 42.36 8.46 238.37 42.36	
Circ TLI Circ TLI LOI Circ TLI LOI Circ TLI LOI Circ TLI LOI COI PC	sec           sec           sec           sec           sec           sec           sec           sec           sec           sec		2.68 104.07 18.52 2.68 96.92 17.25 8.43	5.49 220.96 39.33	8.46 238.37 42.36 8.46 238.37 42.36 20.68	
Circ   TLI	sec           sec           sec           sec           sec           sec           sec		2.68 104.07 18.52 2.68 96.92 17.25	5.49 220.96 39.33 5.49 205.78	8.46 238.37 42.36 8.46 238.37 42.36	

		160		Ī	
		160			
		140			
		120			
	t, ft		→ 27.5' ←		
	Overall Vehicle Height, ft	100	Payload	→ 27.5' ←	
	ile H			Payload	
	/ehic	80		<b>1</b>	
	rall	60			
	Ove.				
		40		74.6'	
			72.3'	74.5	
		20			
			25 25	52 52	<b>^</b>
			S2B.2	S2B.3	
Vehicle Name			EDS + Payload w/ Suborbital	EDS + Payload w/ Suborbital	
Payload Delivery to Moon	Units		27	27	
Payload					
TLI Only					CEV @ Liftoff
Capability	mT		54.0	60.6	LSAM Payload
Net Payload	mT_		48.6	54.6	CEV Payload
TLI + LOI					Margin Payload Gross Total
Capability	mT			 	Payload
Net Payload	mT			 	Net Total Payload
TLI + LOI + PC Capability	mT_			 	
Net Payload					
TLI Delivery w/ On-Orbit Refueling		_<_(		 	. — - — - — -
Capability Net Payload	mT mT		151.5 136.4		
		=:2:2:			
Earth Departure Stage					
Gross Mass Length	lbm ft		47 <u>9,788</u> 72.3	506,577 74.6	
Diameter	ft		27.5	27.5	· — · — · — · — ·     — · — · — · — ·
Usable Propellant Mass @ Liftoff	<u>lbm</u>		419,610	457,884	
Usable Propellant Mass @ 160 nmi cir. Burnout Mass	lbm lbm		183,339 60,131	186,582 48,640	
Dry Mass Engine(s)/Motor	lbm		54,511 4 / J-2S+	42,645 2 / J-2S+	
Vacuum Thrust	lbf		274,500	274,500	
Vacuum Specific Impulse  Burn Time	secsec		451.5	<u>451.5</u>	
Burn Time			04.49	117.60	
Suborbital  Circ	sec		94.48 2.67	217.68 5.44	
TLI Circ	sec sec		75.38 0.00	153.44 0.00	
TLI LOI	secsec		0.00	0.00 0.00	
Circ TLI	sec		0.00 0.00 0.00	0.00 0.00	
LOI	sec		0.00	0.00	
PC LOI+PC Burn time	sec sec		0.00 0.00	0.00 0.00	
200 sec Stage Thrust 200 sec Power Level	lbf %		103,466 38%	0.00	
				0.00	

		160		1	
		100			
		140			
		120			
	#,				→ 16.4' ←
	eight	100	27.5'	→ 27.5' ←	
	E E	80	Payload	Payload	
	Vehi	80	1	<b>↑</b>	
	Overall Vehicle Height, ft	60			
	Ó				103.2'
	İ	40	76.8'	76.6'	
		20	$\Omega$		
				$\triangle \triangle \downarrow$	$XX \downarrow$
			S2B.4	S2B.5	S3A
Vehicle Name			EDS + Payload w/ Suborbital	EDS + Payload w/ Suborbital	EDS Only w/ Suborbital Burn
Payload Delivery to Moon	Units		27	27	30
Payload TLI Only	: 		20.3 mT	21.8 mT	i
Capability	mT		42.8 mT	44.9 mT	109.8
Net Payload			19.1 mT	20.6 mT	98.8
TLI + LOI			14.3 mT	8.8 mT	
Capability	mT		76.2 mT	74.3 mT	70.8 mT
Net Payload			68.6 mT	66.9 mT	63.7
TLI + LOI + PC Capability				 	55.8
Net Payload					50.2
TLI Delivery w/ On-Orbit Refueling	 			 	
Capability Net Payload					132.3 119.1
	<b></b>		ļ		
Earth Departure Stage				i	· — · — · — · — i · — · — · — · — ·
Gross Mass Length	lbm ft		54 <u>5,924</u> 76.8	541,294 76.6	390,708 103.2
Diameter	ft		27.5	27.5	16.4
Usable Propellant Mass @ Liftoff	lbm		495,128	490,744	351,850
Usable Propellant Mass @ 160 nmi cir.	lbm		223,826	219,443	288,753
Burnout Mass  Dry Mass	lbm lbm		50,74 <u>1</u> 44,314	50,494 44,118	38,8 <u>18</u> 34,743
Engine(s)/Motor	Ibf		2 / J-2S+	2 / J-2S+	4 / RL-70
Vacuum Thrust			274,500	274,500	70,000
Vacuum Specific Impulse Burn Time	secsec		451.5	<u>451.5</u>	450.0
Burn Time	 		217.60		
Suborbital Circ	sec		217.68 5.44	217.68 5.44	85.69 9.24
TLI	sec		5.44 184.04	5.44 180.43	452.74
Circ TLI	sec sec		0.00 0.00	0.00 0.00	9.24 384.43
LÓI	sec		0.00	0.00	68.31
Circ TLI	sec sec		0.00	0.00 0.00	9.24 358.04
LOI	sec		0.00	0.00	63.62
PC LOI+PC Burn time	sec sec		0.00 0.00	0.00 0.00	31.07 97.06
200 sec Stage Thrust	lbf		0.00	0.00	162,423
200 sec Power Level	%		0.00	0.00	232%
i			0.00	0.00	;

		160				
	i I	140				
		120				
	¥		1 1	→ 16.4'		→ 16.4' ←
	ght,	100	16.4'	Payload		rayidau
	Hei			<u></u>		<u> </u>
	Overall Vehicle Height, ft	80				
	Veh					
	ra II	60				
	် လို	00				
		40	92.1'	85.5		83.2'
		40				
		20				
			$\Omega\Omega$	$\Omega$		25 25
			$\triangle \triangle \bot$	$\triangle$ $\triangle$ $\rightarrow$		
			S3A.1	S3B		S3B.1
			EDS Only w/	EDS + Payload w/		EDS + Payload w/
Vehicle Name			Suborbital Burn	Suborbital Burn		Suborbital
Payload Delivery to Moon	Units		30.1	30		27
Payload						
TLI Only					Liftoff Payload	42.8 mT
Capability	mT		107.7	57.7	Docked CEV Payload	19.1mT
Net Payload	mT		96.9	51.9	Net Rq'd TLI Payload	61.9 mT
			30.5		Gross Payload	
TLI + LOI	<u></u>				Capability to TLI Net Payload Capability	72.8 mT
Capability	mT		69.7 mT	44.3 mT	to TLI	65.5 mT
Net Payload	mT		62.8	39.9	Net Margin	3.6 mT
TLI + LOI + PC					 	
<u>Capability</u> Net Payload			55.1 49.6	37 <u>.8</u> 34.0		
			15.0			— - — - — - — - — - — -
TLI Delivery w/ On-Orbit Refueling Capability	<u>m</u> T			100.8	<u> </u>	96.6 mT
Net Payload				90.7		87.0 mT
					Net TLI Margin	25.1 mT
Earth Departure Stage						
Gross Mass	lbm	<i>Z,</i> Z:_::	<u>338,826</u>	307,382		296,350
Length Diameter	ft ft		92.1 16.4	<u>85.5</u> 16.4		83.2 16.4
Usable Propellant Mass @ Liftoff	lbm		302,807	273,051		262,626
Usable Propellant Mass @ 160 nmi cir.	lbm		280,976	209,954		199,530
Burnout Mass	lbm		35,985	34,300	 	33,694
Dry Mass Engine(s)/Motor	<u>lbm</u>		32,406	31,027		30,529 4 / RL-70
Vacuum Thrust	 lbf		4 / RL-70 70,000	4 / RL-70 70,000		70,000
Vacuum Specific Impulse	sec		450.0	450.0		450.0
Burn Time  Burn Time	sec					
Suborbital			25.93	85.69		
Circ TLI	sec sec		25.93 9.16 451.51	9.24 260.61		91. <u>94</u> 9.46
Circ	sec		9.16	9.24		320.66
TLI LOI	sec sec		383.44 68.13	260.61 46.31		0.00
Circ	sec		9.16	9.24		0.00
LOI	sec sec		357.12 63.46	260.61 46.31	i —	0.00 0.00 0.00 0.00
PC	sec		30.99	22.61		0.00
LOI+PC Burn time 200 sec Stage Thrust	sec lbf		94.45 158,050	70.57 93,398		0.00 112,230
200 sec Power Level	lbf %		226%	133%		160%
1 1						

		1400				1
	į	160				
	! !	140				
	ļ !	120				
	ght, ft	100			→ 16.4' ←	
	Overall Vehicle Height, ft	80	<b>→</b> 16.4' <del> </del>		Payload	
	rall Ver	60				16.4' →
	ļ š					86.8'
	! ! !	40	72.0'		74.6'	
	į į	20	88		88	337
	  -  -  -		S3B.3		S3B.2	S4A
Vehicle Name	 		EDS + Payload Attached w/ Suborbital Burn		EDS +Payload Attached w/Suborbital Burn	EDS Only w/ Suborbital Burn
Payload Delivery to Moon	Units		30.1		30.1	7.4
PayloadTLI Only	! 	<del>-</del>	42.8 mT			
Capability	mT	<u> </u>	19.1 mT		56.6	85.8
Net Payload			61.9 mT		50.9	77.3
TLI + LOI	i	<del>-</del>	70.6 mT		7	
Capability	mT	!	63.6 mT		43.7	54.7
Net Payload			1.7 mT		39.3	49.3
TLI + LOI + PC Capability				 	37.3	;
Net Payload					33.6	42.7 38.5
TLI Delivery w/ On-Orbit Refueling	! 			  -		 
Capability Net Payload			<b></b>	-		<u>                                     </u>
	<b>F</b>	<u></u>	<del></del>	-		; — - — - — - —
Earth Departure Stage						ri }
Gross Mass Length	lbm ft		244,468 72.0		<u>256,514</u> 74.6	314,275 86.8
Diameter	ft		16.4	;	16.4	16.4
Usable Propellant Mass @ Liftoff	<u>lbm</u>		213,605		224,985	279,321
Usable Propellant Mass @ 160 nmi cir. Burnout Mass	lbm lbm		191,773 30,839	-	203,152 31,503	<u>230,426</u> 34,922
Dry Mass	lbm		28,190		28,733	31,585
Engine(s)/Motor	 		4 / RL-70	آ	4 / RL-70	4 / RL-70
Vacuum Thrust Vacuum Specific Impulse	lbf sec	<del>-</del>	70,000 450.0	-	70,000 450.0	70,000 450.0
Burn Time	sec		T			[
Burn Time Suborbital	<u>-</u>	· + -	<u> </u>		25.93	70.91
Circ	sec		25.93		9.16	7.67
TLI Circ	sec sec		9.16 308.19		258.21 9.16	370.32 7.67
TLI	sec	+-	0.00		9.16 258.21	314.46
LOI Circ	sec sec		0.00		45.88 9.16	55.88 7.67
TU	sec		0.00		258.21	292.88 52.04
LOI	sec sec	<del> </del> -	0.00		45.88 22.41	25.41
LOI+PC Burn time	sec		0.00		68.29	77.46
200 sec Stage Thrust 200 sec Power Level	lbf %		107,867 154%		90,373 129%	129,613 185%
		_ <del></del> .	r			,

	Overall Vehicle Height, ft	140 120 100	Payload
	Overall	60 40 20	S4B  EDS + Payload w/ Suborbital
Vehicle Name			EBO - 1 ayload w/ Guborbital
Payload Delivery to Moon	Units		7.4
Payload TLI Only	: 		
			45.0
Capability	<u> </u>		45.2
Net Payload	<u>m</u> T		40.7
TLI + LOI	<u> </u> 		
Capability	mT		34.4
Net Payload	mT		31.0
TLI + LOI + PC Capability			29.1
Net Payload	mT		26.2
TLI Delivery w/ On-Orbit Refueling	 		
Capability Net Payload			
Earth Departure Stage			
Gross Mass	<u>lbm</u>		250,128
Length Diameter	ft ft		73.1 16.4
Usable Propellant Mass @ Liftoff	lbm_		218,980
Usable Propellant Mass @ 160 nmi cir.	lbm		170,084
Burnout Mass	lbm lbm		31, <u>123</u> 28,416
Engine(s)/Motor	ibm		28,416 4 / RL-70
Vacuum Thrust	<u>lbf</u>		70,000
Vacuum Specific Impulse Burn Time	secsec		450.0
Burn Time	 		7 7 7 7 7 7 7 7 7 7 7 7 7 7 7 7 7 7 7 7
Suborbital Circ	sec		7.24 353.11
TLI Circ	sec sec		7.24 299.83
TLI	sec		53.28
LOI Circ	sec sec		7.24 279.25
TLI LOI	sec		49.62 24.23
PC	sec sec		73.85
LOI+PC Burn time 200 sec Stage Thrust	sec lbf		57.17 75,656
200 sec Power Level	<u>%</u>		108%
i	<u> </u>		



# Section 6C Launch Vehicle and EDS Performance and Sizing

## **Appendix 6C**

## **Launch Vehicle and EDS Performance and Sizing**

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6C.1	Methodology	/Tools							
6C.2	<b>Ground Rule</b>	s and Assumptions							
6C.3	Launch Vehicle Summaries								
6C.4	EDS Summar	ries							
6C.5	<b>Detailed Lau</b>	nch Vehicle Descriptions							
	6C.5.1 Crew I	Launch Vehicles							
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### **6C.6** Closed Case Trajectory Summaries

#### Launch Vehicle and EDS Performance and Sizing

#### 6C.1 Methodology/Tools

The process used for the preliminary performance and sizing of the Launch Vehicle (LV) and Earth Departure Stage (EDS) concepts is shown in **Figure 6C-1** of this appendix. Based upon the mission requirements for the particular concept under study and within the framework of the Ground Rules and Assumptions (GR&As) established, a preliminary concept is sized using the Mass Estimating Relationships (MERs) in the INTegrated ROcket Sizing Program (INTROS). An initial trajectory is flown of this vehicle in the Program to Optimize Simulated Trajectories (POST) to determine the ascent flight environments (accelerations, dynamic pressure, payload capability, etc.) and then the initial vehicle weights and trajectory outputs are sent for more detailed structural sizing with Launch Vehicle Analysis (LVA). Loads, forces, material properties, and design techniques are all considered within the LVA analysis and new structural weights are calculated for the LV or EDS concept. INTROS then incorporates these new structural element weights and estimates a total injected mass based on the total ideal delta velocity from the previous POST output. POST then determines a new total injected mass and ideal delta velocity. INTROS takes these values from POST and estimates a new value for propellant reserves and continues to iterate with POST until the POST total injected mass is within 0 lb to 300 lb of the INTROS estimated value. The performance and sizing analysis for this concept is then considered closed and a vehicle summary is generated. The vehicle configuration description and mass summary for the vehicle and its elements are then sent to the Cost team for cost analysis. The vehicle configuration description and closed case trajectory summary are also sent to the team for their analysis.

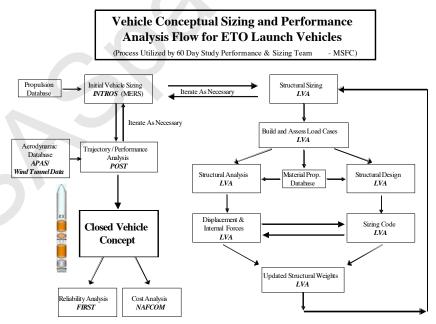


Figure 6C-1. LV and EDS Performance and Sizing Process

A general description of the analysis tools utilized to generate the preliminary LV and EDS concepts is listed below.

**INTROS** is an analytical tool that was developed at MSFC to establish LV designs and sizing. It is written in Visual Basic for Applications computer language and uses the Excel application for all input and output. Launch vehicle design and sizing are based on stage geometry and mass properties. Mass properties are established for selections from a large master list of LV systems, subsystems, propellants, and fluids. Mass calculations are based on MERs that are automatically generated from a large database of MERs that is built into the program. Program mass calculation accuracy for existing and historical LVs has been verified to be well within 5 percent.

LVA is a standalone application written at MSFC in Visual Basic that provides extremely fast LV structural design and analysis. It is important to note this program does not use weight estimating or scaling routines—it supplies detailed analysis by using time-proven engineering methods based on material properties, load factors, aerodynamic loads, stress, elastic stability, deflection, etc. For the fastest turnaround, the program is designed to work with the absolute minimum of input data. The output data is purposely limited to the least possible quantity to prevent the analyst from having to dig through a large amount of data for the necessary information. LVA and its predecessors have been serving NASA for more than 20 years.

Maximum dynamic pressure (max q) and maximum acceleration (max g) are run as the maximum for the class of vehicle. Loads are run as a single combined worst case. Structural analysis is run to within 5–10 percent of closing, the results are these values.

POST3D (Program to Optimize Simulated Trajectories) is a FORTRAN 77-based legacy code developed by NASA Langley for detailed trajectory simulations. Quoting from the introduction in the Utilization Manual: "POST is a generalized point mass, discrete parameter targeting and optimization program. POST provides the capability to target and optimize point mass trajectories for a powered or unpowered vehicle near an arbitrary rotating, oblate planet. POST has been used successfully to solve a wide variety of atmospheric ascent and reentry problems, as well as exoatmospheric orbital transfer problems. The generality of the program is evidenced by its N-phase simulation capability, which features generalized planet and vehicle models. This flexible simulation capability is augmented by an efficient discrete parameter optimization capability that includes equality and inequality constraints."

#### 6C.2 Performance and Sizing Ground Rules and Assumptions (GR&A)

#### 6C.2.1 General GR&A

#### **Payload Definitions**

Payload is defined as the total injected weight minus the burnout weight of the final stage.

#### 6C.2.2 Trajectory (POST)

#### General Trajectory GR&A

Max acceleration = 4.0 g's (3.0 g's for side-mount Shuttle-derived Vehicle (SDV) to avoid External Tank (ET) redesign).

Max dynamic pressure = 800 psf (undispersed), except for certain In-line Crew (ILC) configuration-Solid Rocket Motor (SRM)-In-line cases where the limit was raised to 1,000 psf due to very high accelerations early in the ascent profile.

Max dynamic pressure = 1,000 psf (dispersed), except for certain ILC-SRM-In-line cases where the limit was raised to 1,200 psf due to very high accelerations early in the ascent profile.

Max q-alpha & q-beta =  $\pm 1,000$  psf-deg.

No moment balance.

Launch from Pad 39A: gdlat = 28.6084 deg, long = 279.3959 deg, gdalt = 0 ft.

Launch azimuth optimized.

Standard oblate Earth model (WGS-84).

1963 Patrick Air Force Base (AFB) atmosphere model.

Kennedy Space Center (KSC) mean annual winds (P. 17-19 VIPA-SDV-SM-TR4).

Start simulation at liftoff (all liquid) or Solid Rocket Booster (SRB) ignition (if using solids).

Vehicle hold-down until T/W = 1.0 (solids).

Begin pitch-over at 350 ft altitude. Pitch over ends and gravity turn begins when q = 150 psf.

Alpha and sideslip angles are set to 0 during gravity turn. Gravity turn ends when q = 100 psf.

Optimized pitch profile after gravity turn.

Avoid instantaneous changes in vehicle attitude.

Serial burn staging events are instantaneous unless a coast phase is required for specific analytical purposes.

#### **Orbital Injection**

Inject into 30 x 160 nmi orbit at 28.5 deg inclination. Perigee and apogee are relative to a spherical Earth whose radius equals Earth's mean equatorial radius.

Main Engine Cutoff (MECO) altitude is optimized, but must be  $\geq 57$  nmi.

 $3-\sigma$  (3-sigma) Free Molecular Heating Rate (FMHR) on exposed payload must be  $\leq$  0.1 Btu/ft<sup>2</sup>-sec (if the payload fairing is jettisoned during ascent, this typically constrains MECO altitude to  $\geq$  75 nmi).

Payload Orbital Maneuvering System (OMS) performs circularization at 160 nmi, except for EDS concepts in which the EDS circularizes itself and the payload.

#### **Payload Fairings**

Fairing structural weight determined by structural analysis.

Fairing jettison weight includes: structures, Thermal Protection System (TPS), and acoustic/thermal blankets.

Fairing jettisoned when 3-sigma FMHR =  $0.1 \text{ BTU/ft}^2$ -sec.

- $3-\sigma$  FMHR =  $(1/2 \rho V3)$  (K-factor) = (dynp) (vela) (K-factor) (conv).
- dynp = dynamic pressure; vela = atmospheric relative velocity.
- K-factor = 2.0 (atmospheric density doubled to account for dispersions).
- Conv = 0.00128593 Btu/ft-lb units conversion factor.

#### Launch Escape System (LES)

LES mass = 9,300 lb for vehicles sized under Block 2 analysis. 9,172 lb for vehicles analyzed under Block 1.

LES jettison at 30 sec after the latter of either the last staging event or upper stage ignition.

#### Aerodynamics

SDV-SM: 6-Degrees-of-Freedom (6-DOF) aero and base force (P. 14-15 VIPA-SDV-SM-TR4).

SDV-IL: 3-DOF aero and base force (Magnum wind tunnel data).

Aero data derived from Magnum wind tunnel data.

Modified Magnum base force data.

#### 6C.2.3 Weights & Sizing (W&S) (INTROS)

#### General W&S GR&A

Dry weight for unmodified Super Light Weight Tank (SLWT) = 59,226 lb (Space Transportation System- (STS) 117 TDDP).

SDV Main Propulsion System (MPS) propellant inventory from STS-117 TDDP (P. 20-21 VIPA-SDV-SM-TR4).

Dry mass margins:

- 0% for existing hardware with no modifications,
- 5% for existing hardware with minor modifications,
- 10% for existing hardware with moderate modifications, and
- 15% for new hardware.

#### Propellant density:

- Liquid Oxygen (LOX): 71.14 lbm/ft<sup>3</sup>,
- Liquid Hydrogen (LH2): 4.42 lbm/ ft<sup>3</sup>, and
- Rocket Propellant (RP): 50.50 lbm/ ft<sup>3</sup>.

#### Ullage fraction:

- For all new concepts: 0.02, or
- Existing stages: derived to achieve known propellant inventory.

Vehicle sizing is considered closed when the payload capability is between the target payload and the target payload plus 300 lb

#### **Propellant Allocation:**

Flight Performance Reserves (FPR):

- SDV: MPS propellant inventory from STS-117 TDDP (P. 20-21 VIPA-SDV-SM-TR4).
- New concepts, Atlas, and Delta: amount required to account for 1% ideal ascent delta-V.

#### Fuel bias:

- SDV: MPS propellant inventory from STS-117 TDDP (P. 20-21 VIPA-SDV-SM-TR4).
- New concepts, Atlas, and Delta: 0.0013 \* mixture ratio / 5.29 \* usable propellant.

#### Residuals:

- SDV: MPS propellant inventory from STS-117 TDDP (P. 20-21 VIPA-SDV-SM-TR4).
- New concepts, Atlas, and Delta: 0.0631 \* (usable propellant)<sup>0.8469</sup>.

#### Start propellant:

- SDV: MPS propellant inventory from STS-117 TDDP (P. 20-21 VIPA-SDV-SM-TR4).
- New concepts:
  - o Ground Start Stages: 3.5 x nominal propellant flow rate.
  - o Air-Start Stages: zero start propellant allocated.

#### 6C.2.4 Structures (LVA)

#### General Structural GR&A

LV safety factors for new stages = 1.5 (consistent with NASA-STD-5001).

3-sigma dispersion estimation on flight loads.

#### **Engine Data**

#### **SRB** Data

- Four-segment Shuttle Reusable Solid Rocket Booster (RSRB): 60 deg PMBT with 0.368 burn rate (P. 8-13 VIPA-SDV-SM-TR4),
- Five-segment SRB: Data from ATK Thiokol (FSB\_HT\_266.2\_ADJMASS.txt, FSB\_HT\_3b.doc), and
- Four-segment 90% burn rate RSRB data from SRM prediction with nominal propellant and ignition mass, additional 1,236.6 lb of inert mass overboard (total mass overboard = 1,112,840.0 lb), Point of Contact (POC) Todd Steadman (MSFC).

Block 2 Space Shuttle Main Engine (SSME):

- Engine data provided on P. 6 VIPA-SDV-SM-TR4.
- Uninstalled engine weight = 7,748 lb; engine length = 168 in.
- SDV-SM: 100% at liftoff, throttle-up to 104.5% at 60 fps relative velocity.
- Other concepts: 104.5% at liftoff.

Expendable SSME and Air-Start SSME:

• Assume all performance, size and weight data same as Block 2 SSME.

#### RS-68:

- Two-step throttle settings can be preset within the following ranges: 57–60% and 100–102% power level. Step throttle settings of 57% and 102% should be assumed.
- 102%: Thrust (vacuum (vac)) = 757,800 lb, Specific Impulse (Isp) (vac) = 409.0 sec, Ae = 44.87 ft<sup>2</sup> (proprietary).
- 57%: Thrust (vac) = 425,500 lb, Isp (vac) = 405.1 sec (proprietary).
- Uninstalled engine weight = 14,761 lb, engine length = 204 in.

#### RD-171:

- Continuous throttle: 50–100% power level.
- 100%: Thrust (vac) = 1,778,000 lb, Isp (vac) = 337 sec, Ae = 70.63 ft<sup>2</sup>.
- 70%: Isp (vac) = 337 sec.
- 50%: Isp (vac) = 335 sec.

- Uninstalled engine weight = 26,600 lb (proprietary).
- Engine length = 158.1 in.

#### RD-180:

- Continuous throttle: 47–100% power level.
- 100%: Thrust (vac) = 933,400 lb, Isp (vac) = 338.4 sec, Ae = 35.32 ft<sup>2</sup>.
- 90%: Isp (vac) = 337.8 sec (proprietary).
- 71%: Isp (vac) = 336.6 sec (proprietary).
- 47%: Isp (vac) = 334.6 sec.
- Uninstalled engine weight = 12,225 lb (proprietary).
- Engine length = 146 in.

#### J-2S:

- Continuous throttle: 20–100% power level.
- 100%: Thrust (vac) = 265,000 lb, Isp (vac) = 436.0 sec, Ae = 31.50 ft<sup>2</sup>.
- 40%: Isp (vac) = 436.0 sec.
- 30%: Isp (vac) = 435.6 sec.
- 20%: Isp (vac) = 434.7 sec.
- Uninstalled engine weight = 3,800 lb, engine length = 133 in.

#### J-2S+:

- No throttle capability.
- 100%: Thrust (vac) = 274,500 lb, Isp (vac) = 451.5 sec, Ae = 65.4 ft<sup>2</sup>.
- Uninstalled engine weight = 4,118 lb, engine length = 188 in.

#### RL-10A-4-2:

- No throttle capability.
- 100%: Thrust (vac) = 22,300 lb, Isp (vac) = 451 sec, Ae = 11.29 ft<sup>2</sup> (proprietary).
- Uninstalled engine weight = 386 lb (proprietary).
- Engine length = 91.5 in (proprietary).

#### LR-60:

- Continuous throttle: 83–100% power level.
- 100%: Thrust (vac) = 60,000 lb, Isp (vac) = 465.0 sec, Ae = 44.2 ft<sup>2</sup>.
- Uninstalled engine weight = 1,100 lb, engine length = 87.6 in. (stowed nozzle); 118.5 in. (extended)

#### LR-70:

- Continuous throttle: 50–100% power level.
- 100%: Thrust (vac) = 70,000 lb, Isp (vac) = 450.0 sec, Ae = 22.1 ft<sup>2</sup>.
- 75%: Isp (vac) = 448.0 sec.
- 50%: Isp (vac) = 446.0 sec.
- Uninstalled engine weight = 1,373 lb, engine length = 124 in.

#### LR-85:

- Continuous throttle: 50–100% power level.
- 100%: Thrust (vac) = 85,000 lb, Isp (vac) = 450.0 sec, Ae = 23.2 ft<sup>2</sup>.
- 75%: Isp (vac) = 448.0 sec.
- 50%: Isp (vac) = 446.0 sec.
- Uninstalled engine weight = 1,564 lb, engine length = 126 in.

#### LR-100:

- Continuous throttle: 50–100% power level.
- 100%: Thrust (vac) = 100,000 lb, Isp (vac) = 450.0 sec, Ae = 25.8 ft<sup>2</sup>.
- 75%: Isp (vac) = 448.0 sec.
- 50%: Isp (vac) = 446.0 sec.
- Uninstalled engine weight = 1,759 lb, engine length = 135 in.

#### **6C.3** Launch Vehicle Summaries

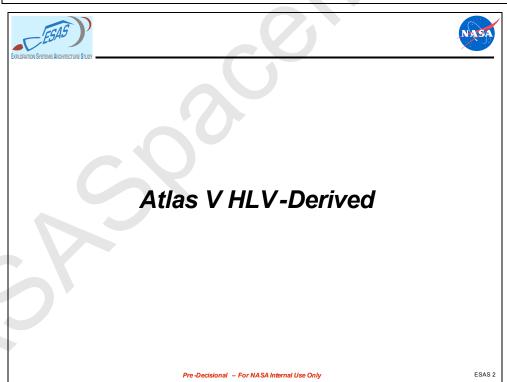
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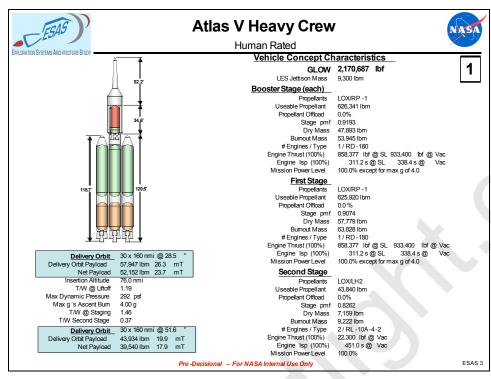


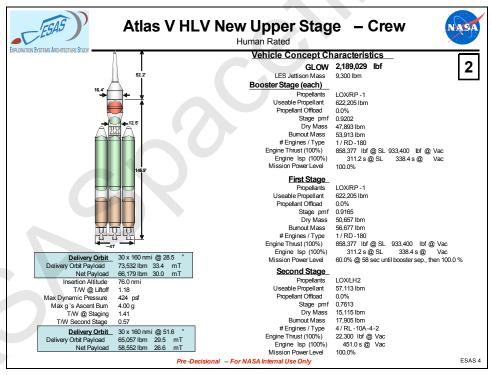
# Launch Vehicle Concept Data Summary

Version 2.8 July 16, 2005

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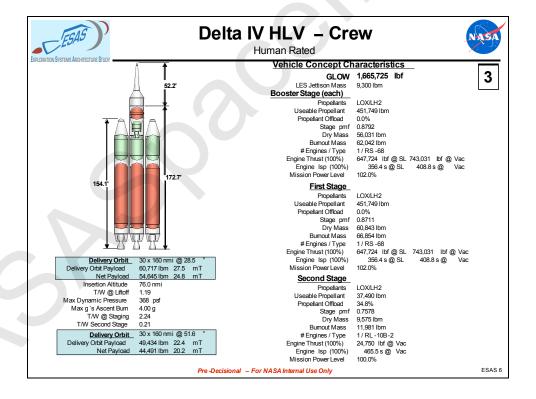


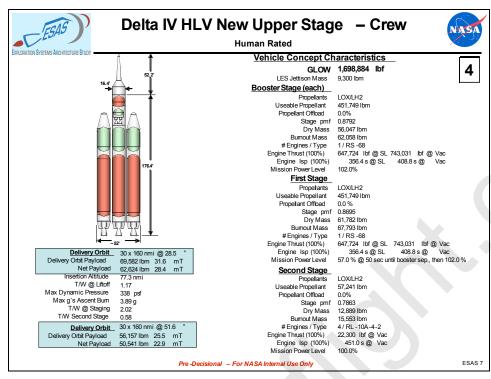


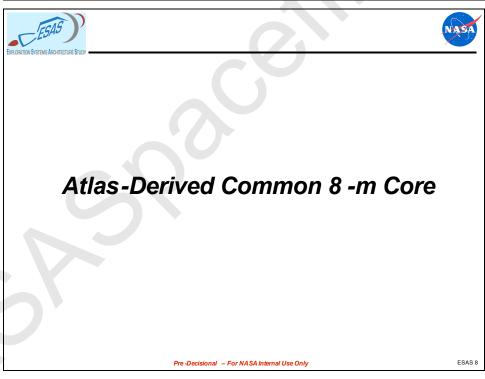
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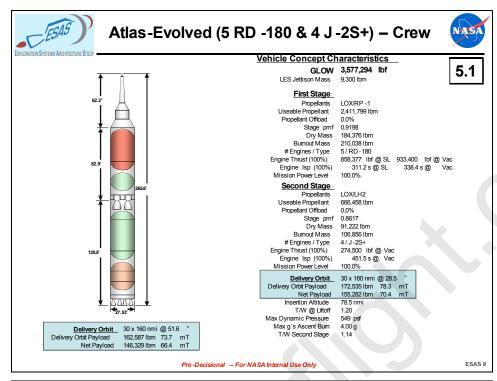
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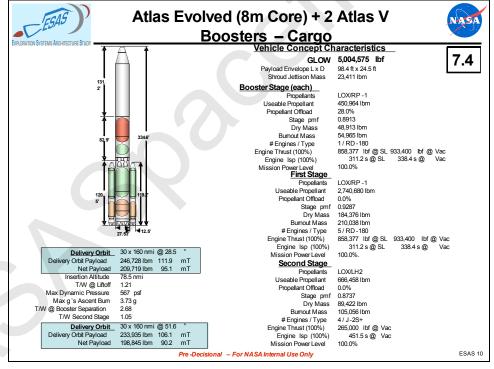
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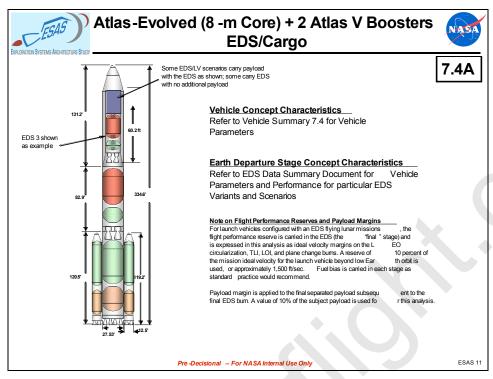


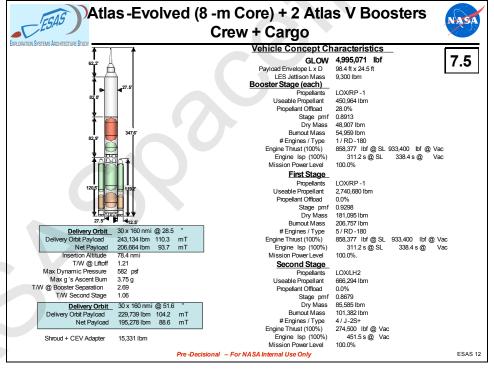


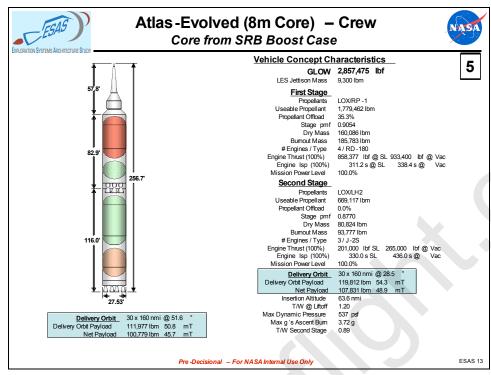


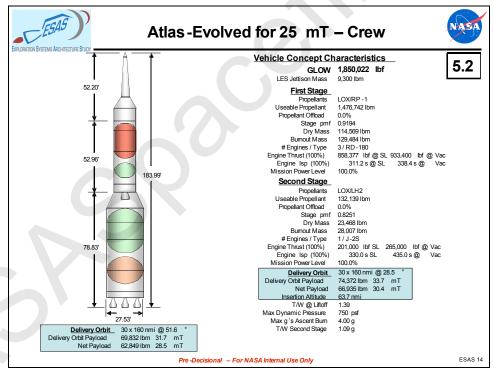


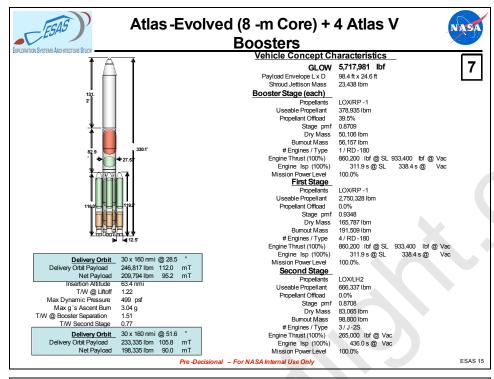


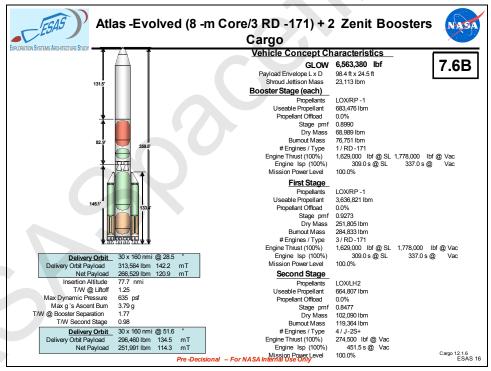


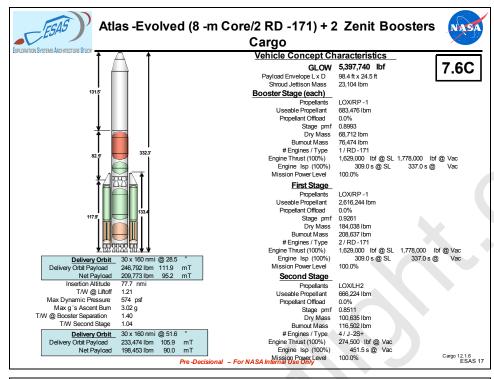


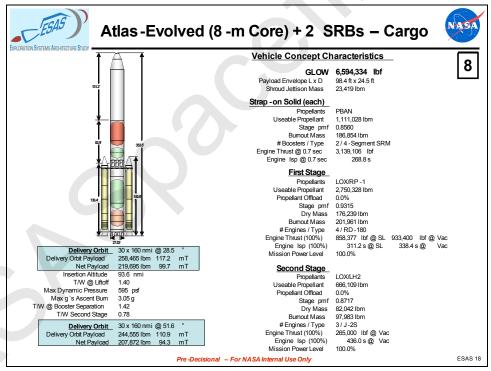


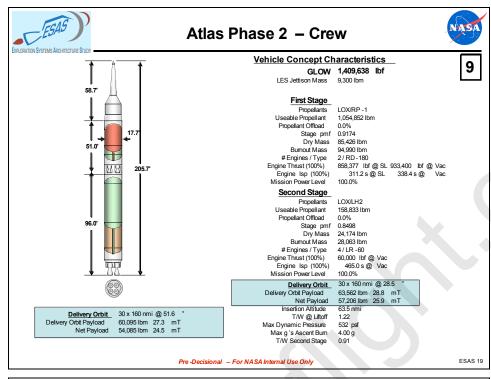


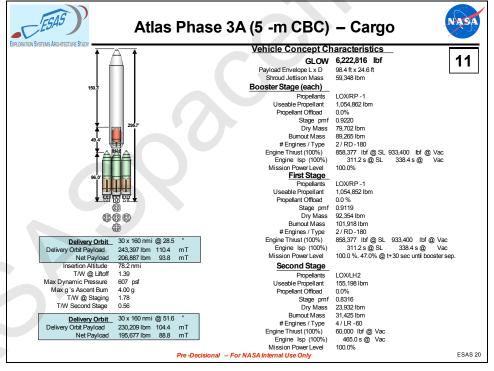


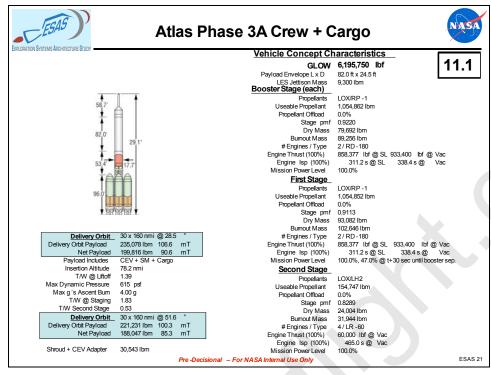


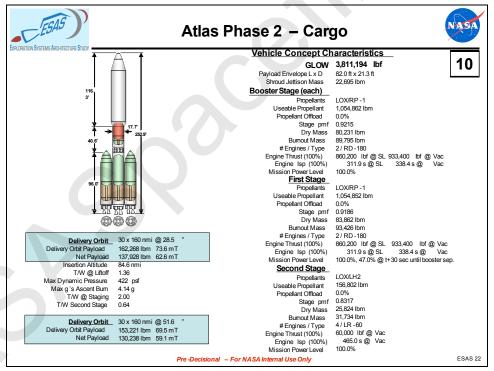


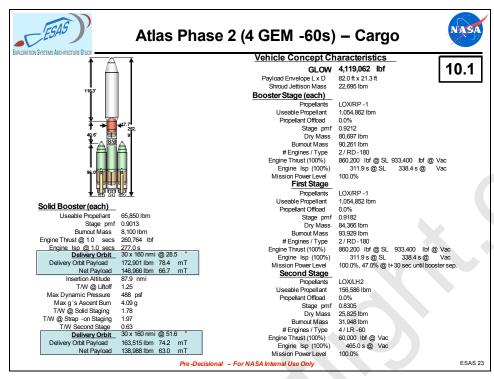




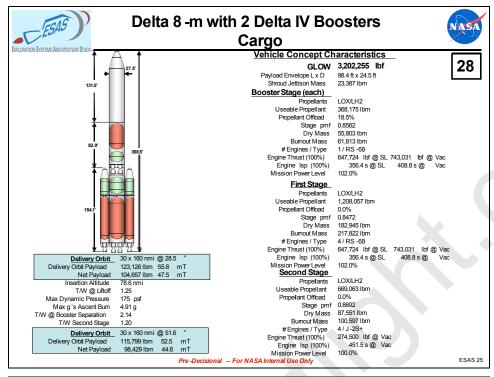


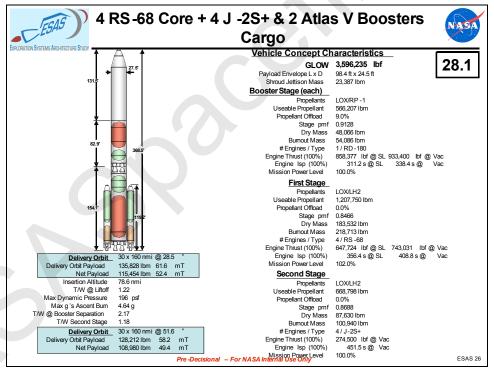


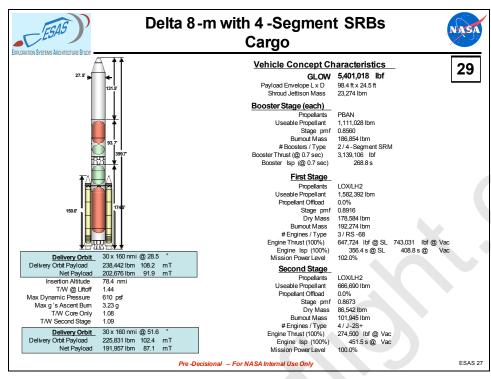




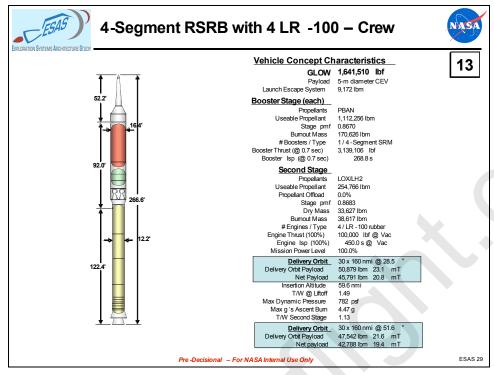


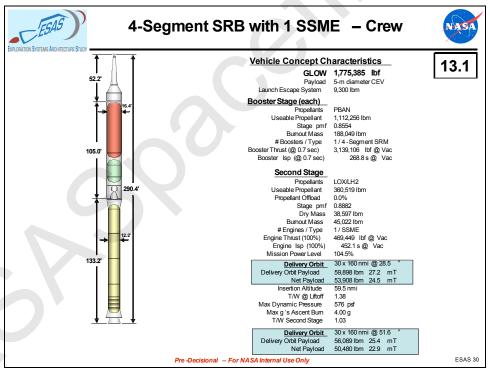


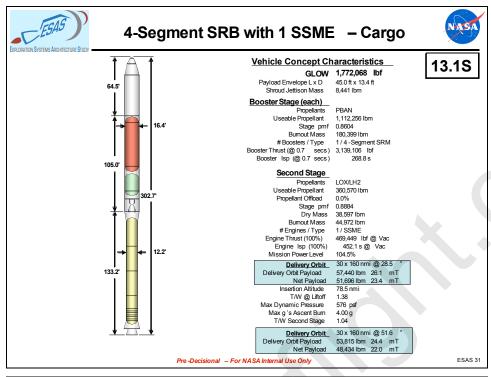


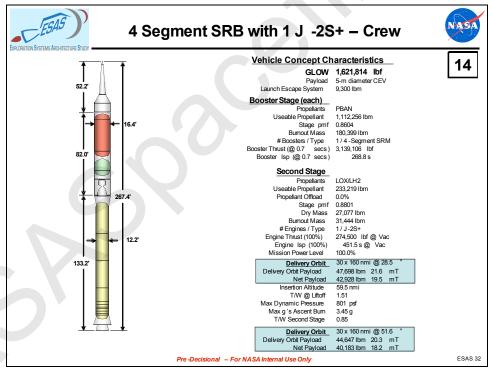


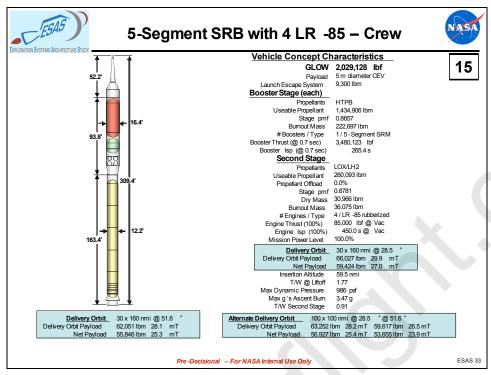


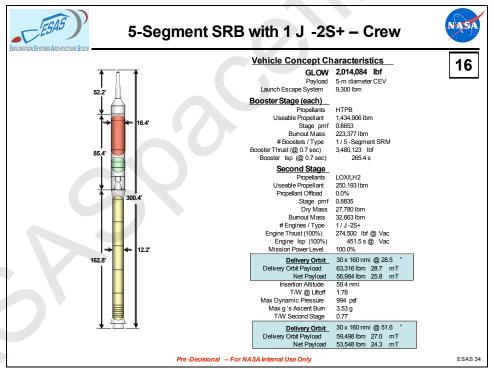


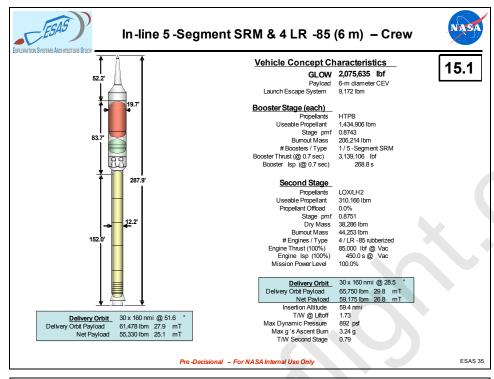


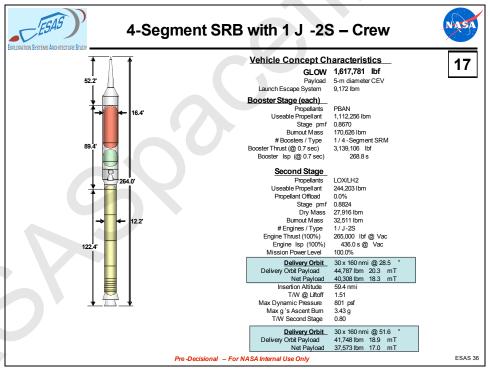


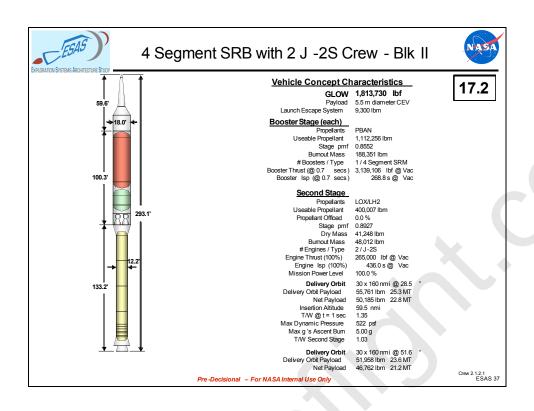


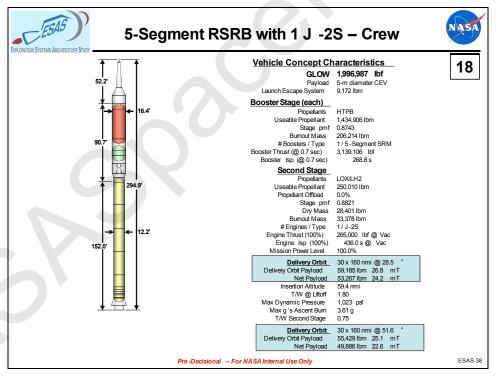


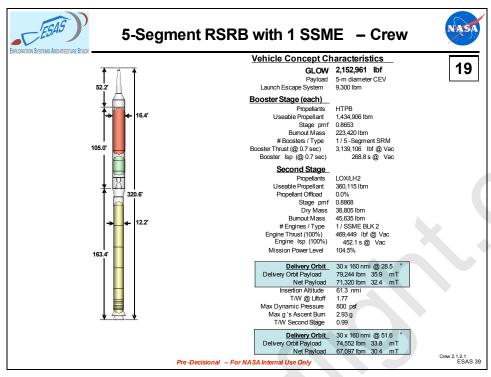


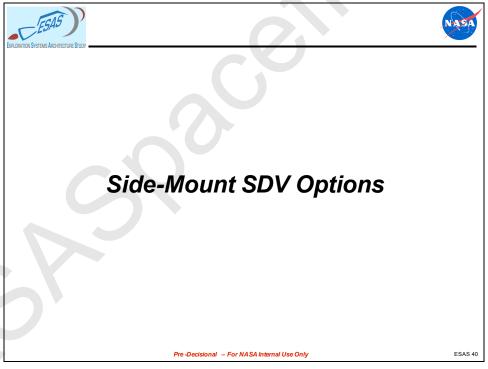


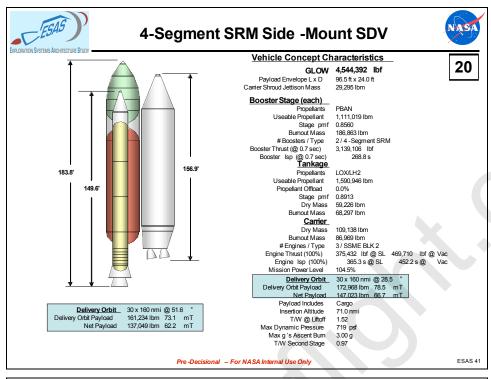


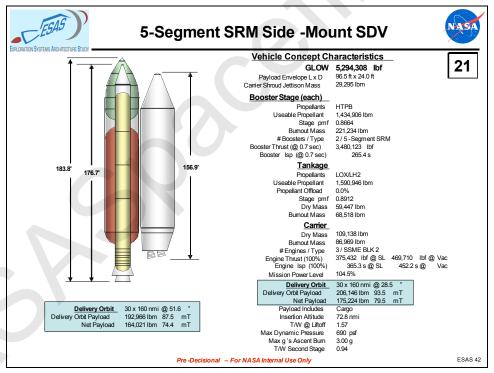


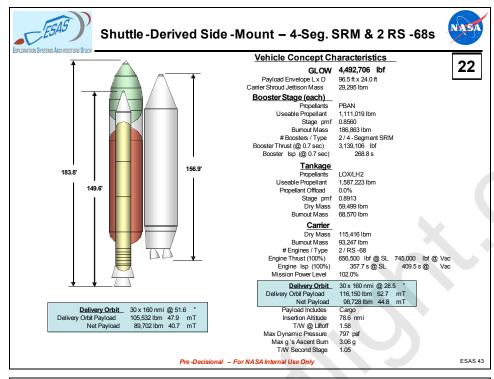


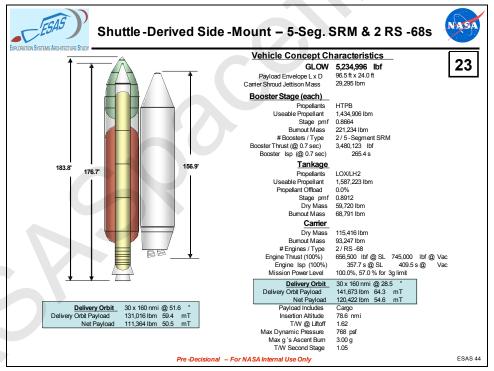












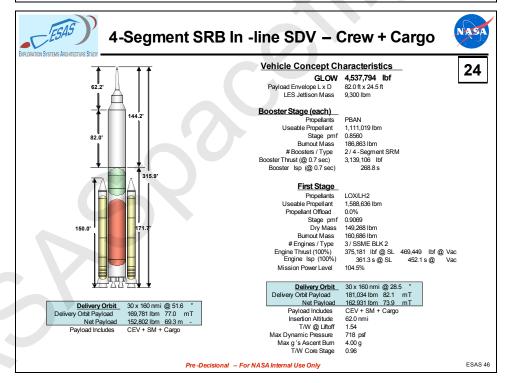


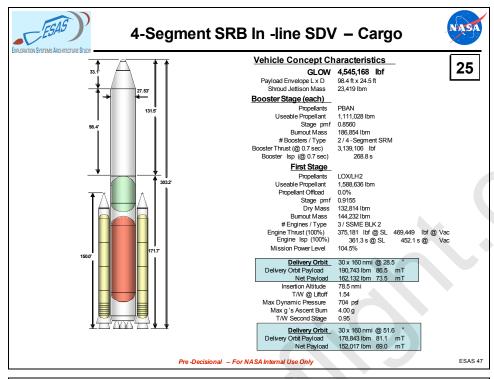


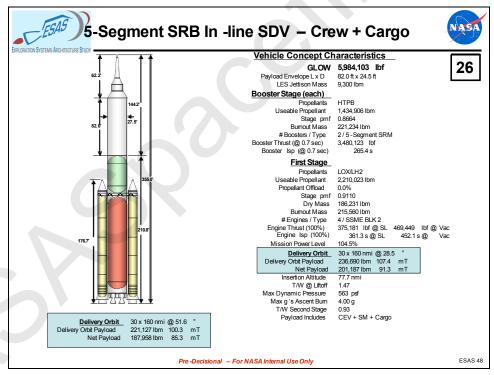
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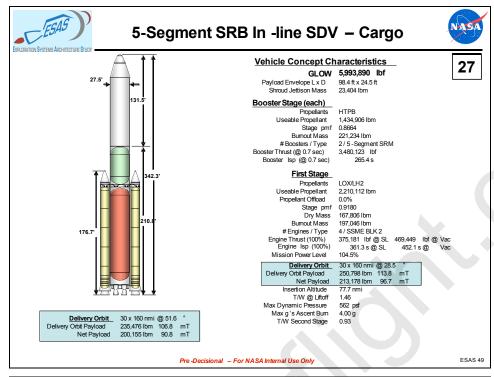
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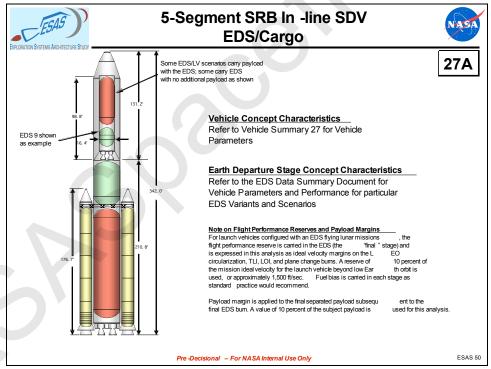
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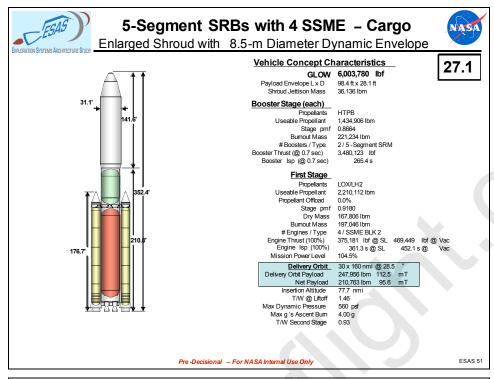


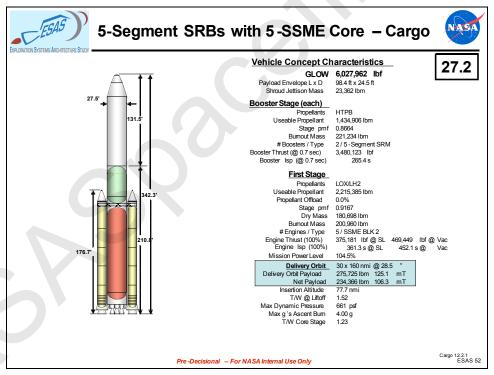


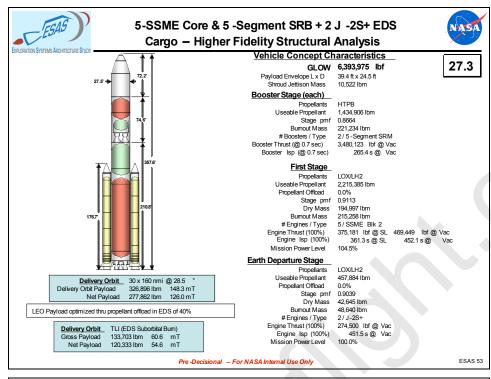


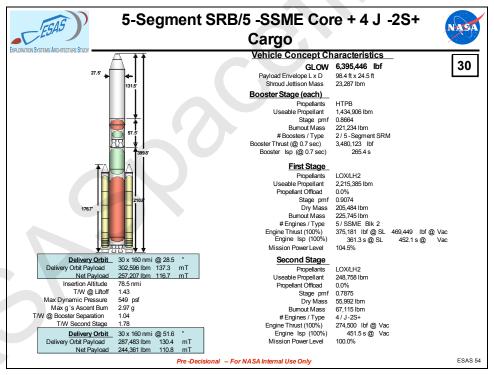


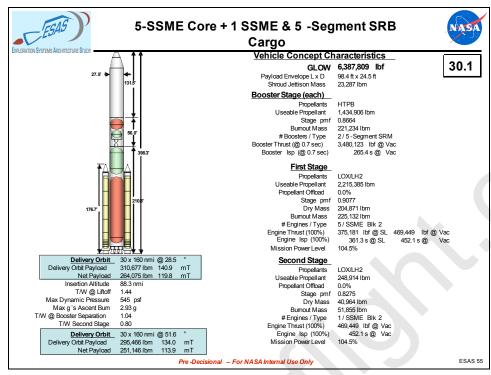


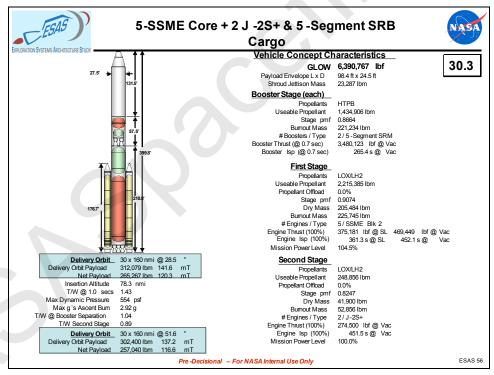


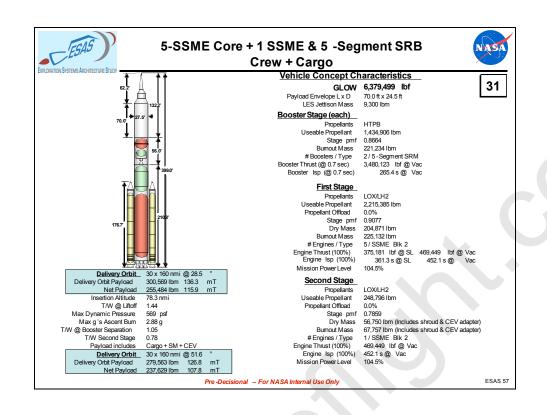






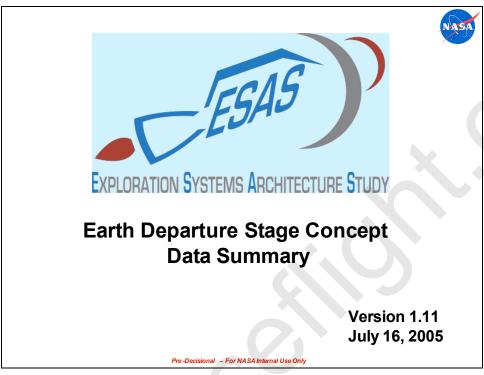


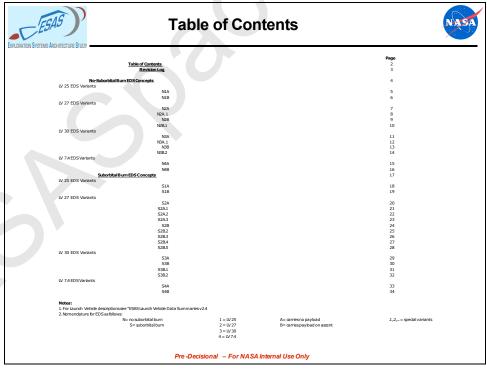




## **6C.4** EDS Summaries

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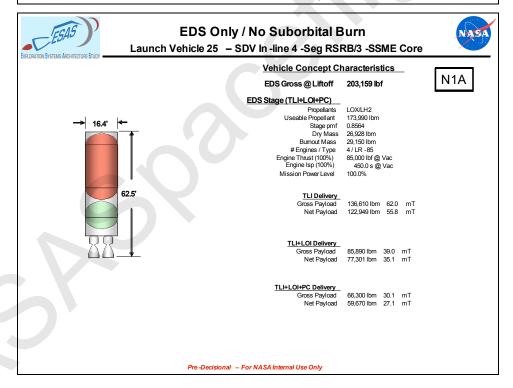


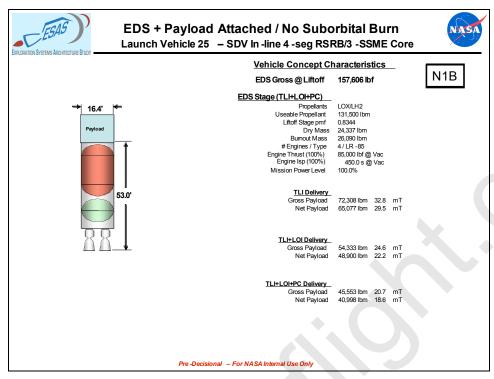


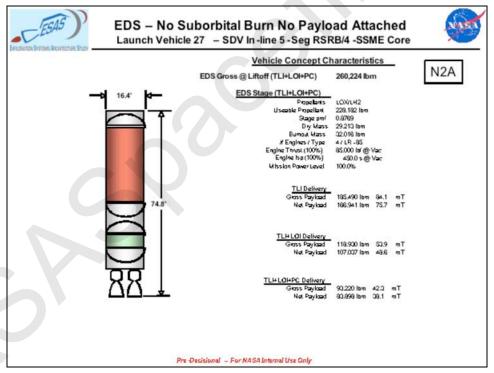


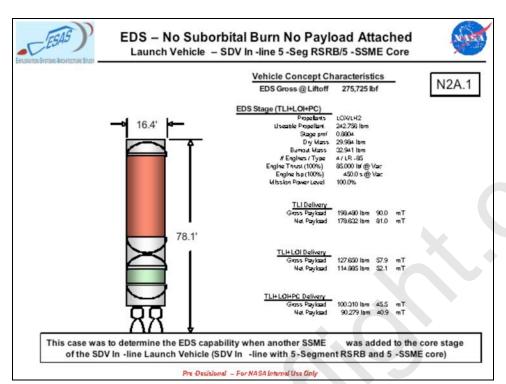
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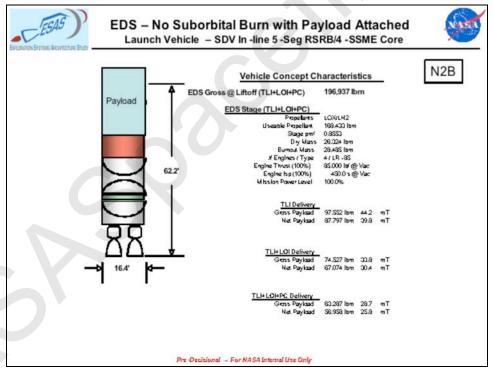
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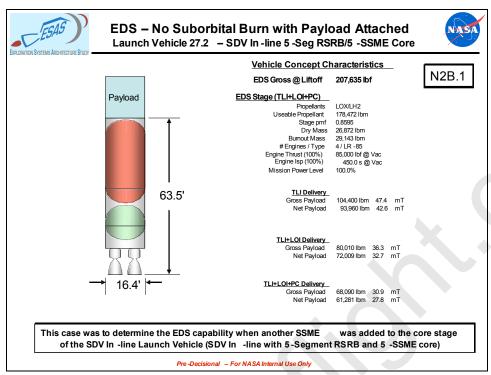


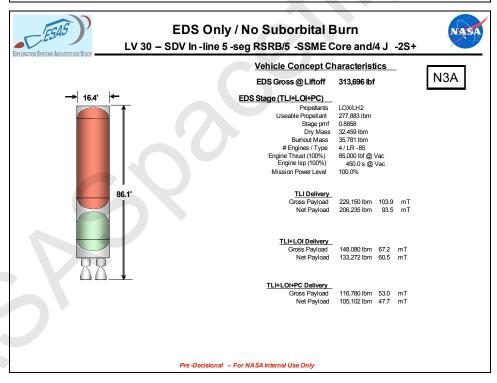


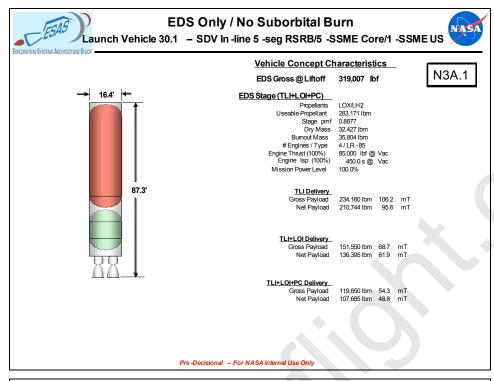


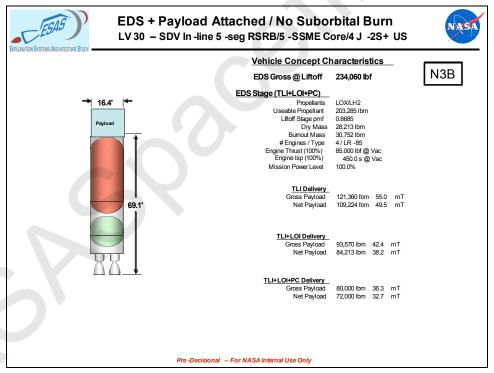


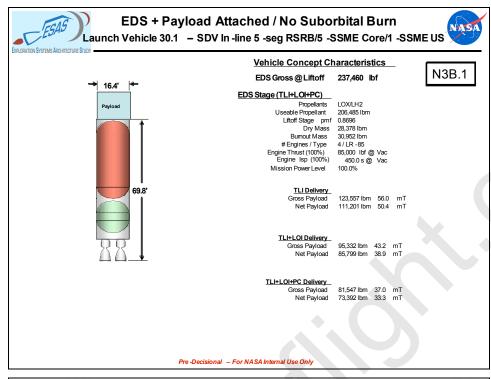


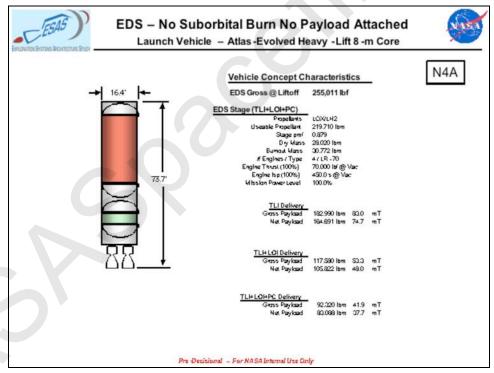


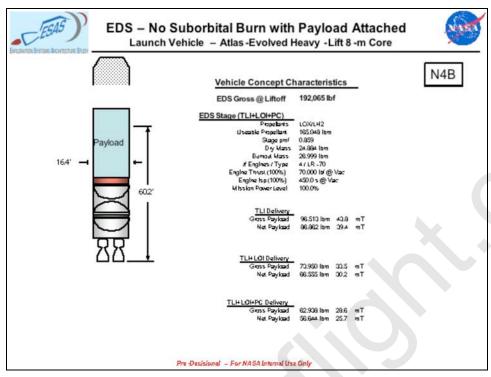


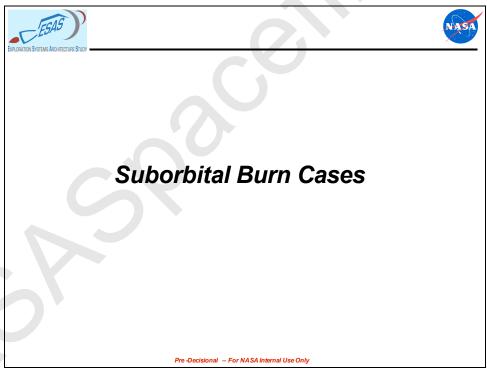


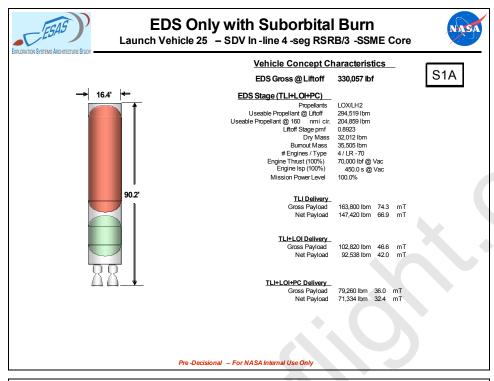


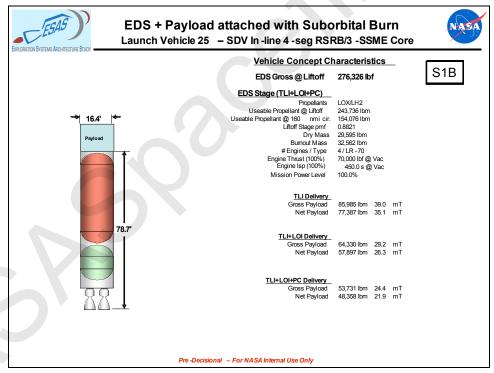


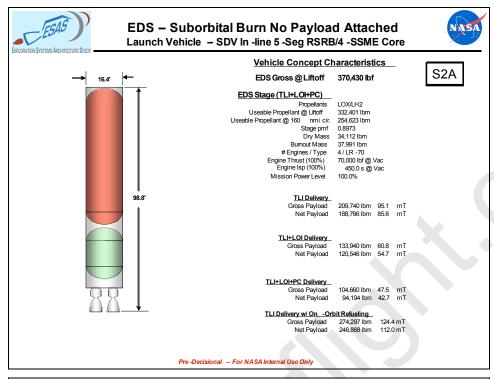


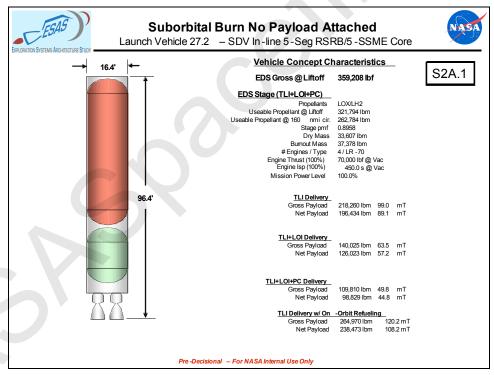


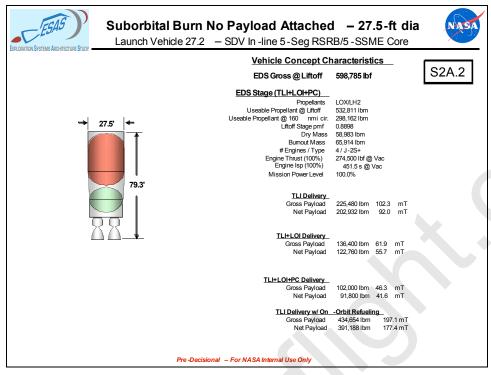


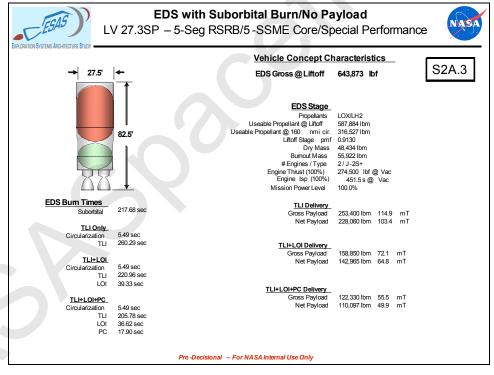


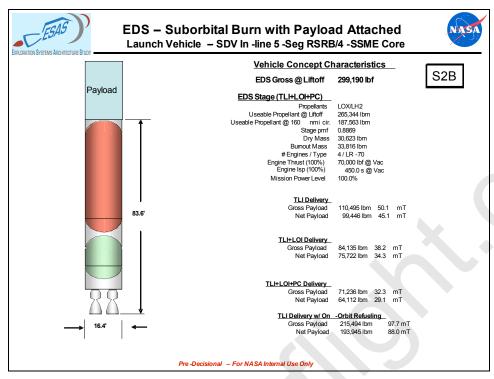


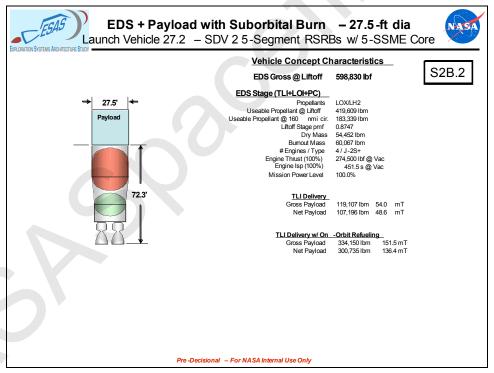


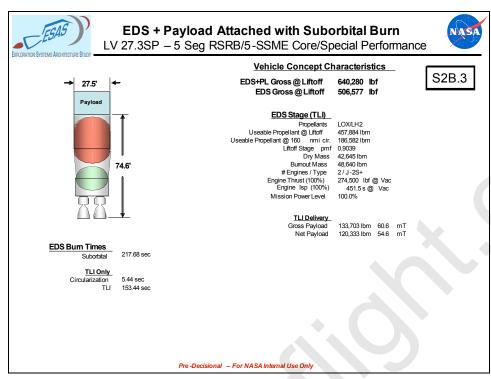


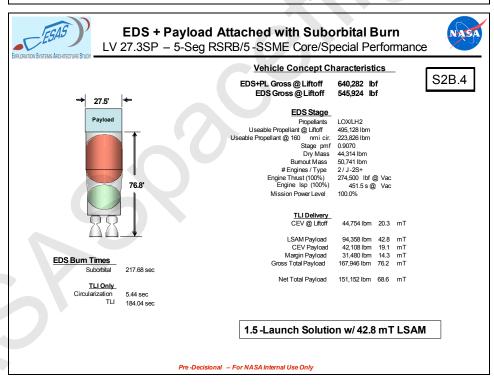


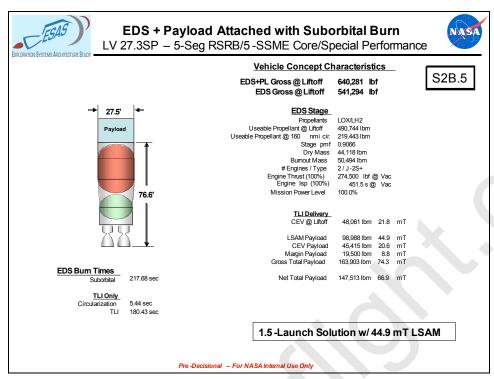


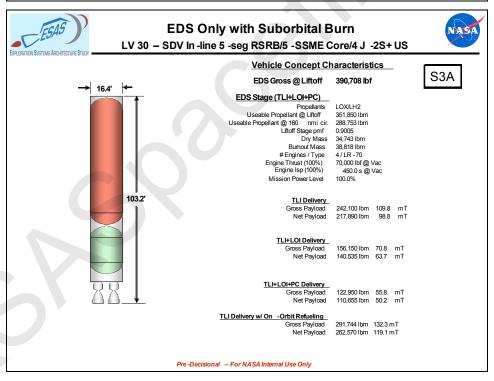


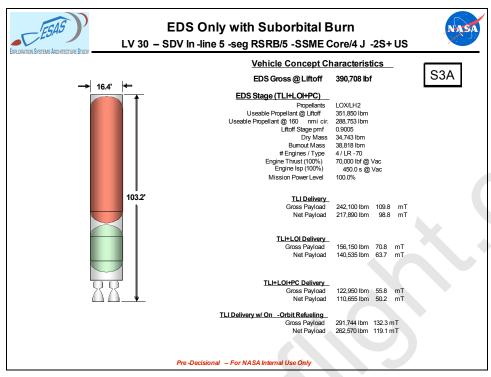


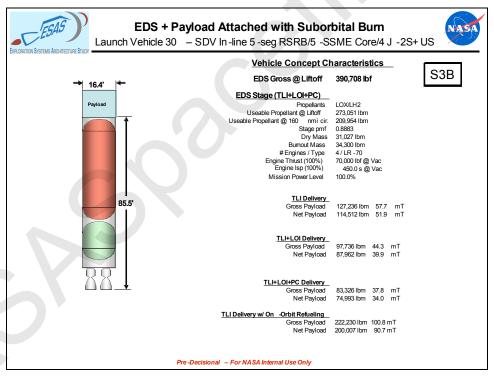


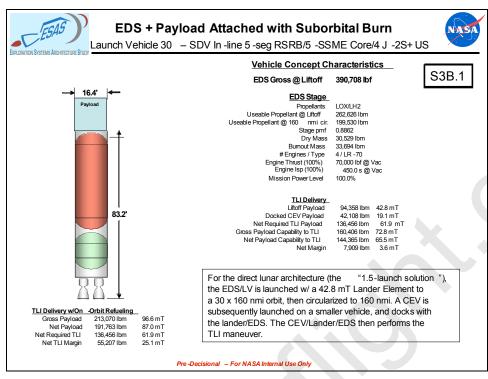


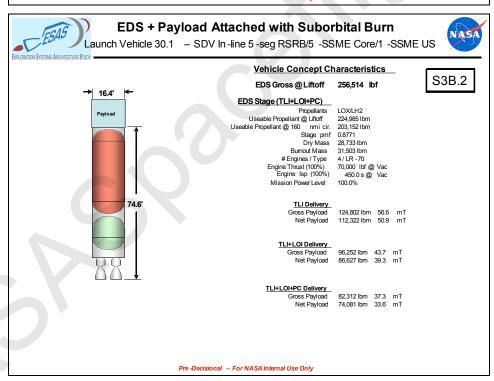


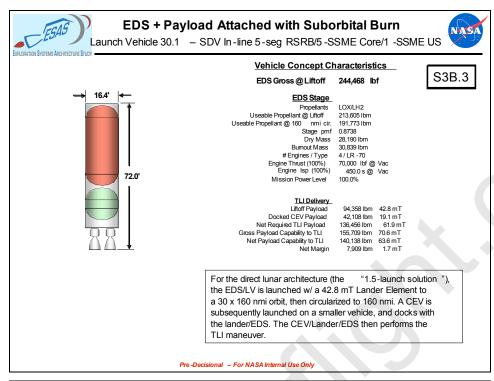


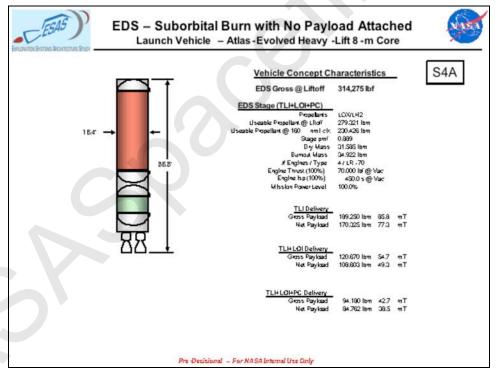


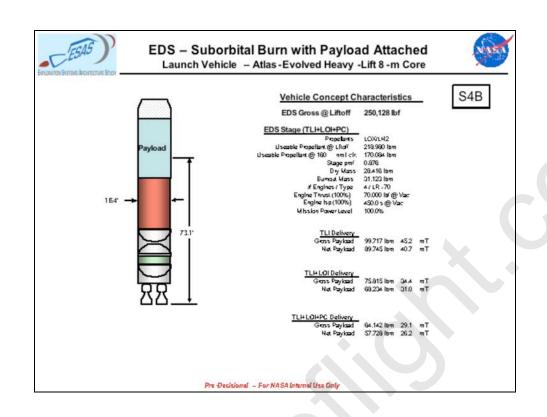












### **6C.5** Detailed Launch Vehicle Descriptions

### 6C.5.1 Crew Launch Vehicles (CLVS)

### **6C.5.1.1** Performance Summary

The CLV Evolved Expendable Launch Vehicle- (EELV-) derived systems performance summary information is included in the following paragraphs. Included are summaries for the following vehicles: LV 2, LV 4, LV 5.1, and LV 9.

### 6C.5.1.2 Launch Vehicle 2 (LV 2)

### 6C.5.1.2.1 Vehicle Description

LV 2 (**Figure 6C-2**) is the Atlas V Heavy-Lift Vehicle (HLV) configuration with a new upper stage for CEV. The vehicle consists of two common core, liquid boosters strapped to a center Common Core Booster (CCB). All booster elements are LOX/RP. "Common core" refers to all three booster elements being similar in dimensions, engines, and manufacturing commonality. There are some differences in structural design for the different load paths for the central core and the strap-on boosters, however. A new larger, more powerful upper stage has been added to the configuration to increase the payload capability of the vehicle. The new upper stage is LOX/LH2 with four RL-10A-4-2 engines.

This vehicle concept was flown to  $30 \times 160$  nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 76.0 nmi. The net payload capability of LV 2 is 30.0 mT to a  $30 \times 160$  nmi orbit at a 28.5 deg inclination. The net payload to  $30 \times 160$  nmi at a 51.6 deg inclination is 26.6 mT. No ground rules or constraints were violated for this LV analysis.

A special consideration was required to analyze this vehicle: Additional mass was added to the vehicle for human rating of the ELV. This additional mass was for increasing the structural safety factor for added redundancy for vehicle health monitoring and management.



Figure 6C-2. LV 2 General Configuration

### 6C.5.1.2.2 Vehicle Sizing

The mass properties for LV 2 are shown in **Table 6C-1**. The primary structural mass of the strap-on boosters and the core stage was increased 3.1 percent to provide a 1.4 Factor of Safety (FS). The second stage design margin already included a 1.4 FS.

Table 6C-1. LV 2 INTROS Mass Summary

MASS PROPERTIES ACCOUNTING				
VEHICLE: Atlas V HLV HR (New Upper Stage) Crew-Blk 2 STAGE: Liquid Strap-on Booster (1 RD-180)				
VIII I	MASS SUBTOTALS	MASS TOTALS		
ITEM	Primary			
	lbm	lbm		
Primary Body Structures	22,285			
Secondary Structures	1,301			
Separation Systems	442			

Thermal Protection Systems (TPSs)	132	
Thermal Control Systems (TCSs)	721	
Main Propulsion System	21,066	
Power (Electrical)	726	
Power (Hydraulic)	686	
Avionics	417	
Miscellaneous	117	
STAGE DRY MASS WITH GROW	TH	47,893
Residuals	5,490	
Reserves	431	
In-flight Fluid Losses	100	
STAGE BURNOUT MASS		53,913
Main Ascent Propellant	622,205	
Engine Purge Helium	19	
STAGE GROSS LIFTOFF MASS		676,137
STAGE: Secon	nd Stage (4 RL–10A–	
Primary Body Structures	8,144	
Secondary Structures	152	
Separation Systems	80	
TCSs	530	
Main Propulsion System (MPS)	2,583	
Power (Electrical)	863	b
Power (Hydraulic)	278	
Avionics	813	
Miscellaneous	36	
STAGE DRY MASS WITHOUT G		13,480
Dry Mass Growth Allowance	1,635	-,
STAGE DRY MASS WITH GROW	,	15,115
Residuals	805	20,220
Reserves	1,948	
In-flight Fluid Losses	38	
STAGE BURNOUT MASS	30	17,905
Main Ascent Propellant	57,113	21,500
Engine Purge Helium	2	
STAGE GROSS LIFTOFF MASS	_	75,021
	Stage Core (1 RD–1	
Primary Body Structures	23,541	
Secondary Structures	1,301	
Separation Systems	1,908	
TPSs	132	
TCSs	759	
MPS	21,066	
Power (Electrical)	724	
Power (Hydraulic)	686	
Avionics	417	

Miscellaneous	122	
STAGE DRY MASS WITH GRO	WTH	50,657
Residuals	5,490	
Reserves	431	
In-flight Fluid Losses	100	
STAGE BURNOUT MASS		56,677
Main Ascent Propellant	622,205	
Engine Purge Helium	19	
STAGE GROSS LIFTOFF MASS		678,901
Payload	73,533	
Launch Escape System (LES)	9,300	
Upper Stage(s), Gross	75,021	
Strap-on(s), Gross Liftoff	1,352,274	
VEHICLE GROSS LIFTOFF MA	SS	2,189,029

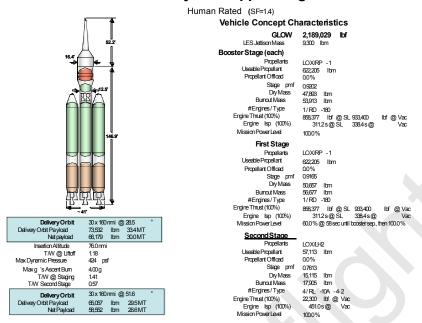
### 6C.5.1.2.3 Structural Analysis

Since this is a derivation of an existing vehicle, an LV analysis was not run. Instead, the structural weights were scaled from the INTROS model.

### 6C.5.1.2.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-3**. Selected trajectory parameters are shown graphically in **Figures 6C-4** through **6C-7**. The vehicle exhibits a 1.18 thrust-to-weight (T/W) ratio at liftoff. Maximum dynamic pressure is 424 psf at 89.8 sec into the flight. The maximum acceleration during the first stage with boosters is 4.00 g's, 4.00 g's without boosters, and is 0.98 g's during the second stage. Booster staging occurs at 228 sec into the flight at an altitude of 191,034 ft and Mach 10.8. Core staging occurs at 296 sec into the flight at an altitude of 280,362 ft and Mach 20.0 for the first stage. The T/W ratio at second-stage ignition is 0.57. Orbital injection occurs at 585 sec at 76.0 nmi.

### Atlas V Heavy New Upper Stage Crew



#### Closed Case Summary Data for Reference Mission (30 x 160 nmi @ 28.5):

```
Liftoff to LRB staging max LRB+1st stg accel = 4.00
                                                                                           After core staging (2nd stage only)
                                                                                                 tstg core = 296 sec
                                                                                                mach@stg = 20.0
dynp@stg = 2.32 psf
alt@stg = 280,362ft
     time of max Q = 89.8
     mach = 1.23
     max Q = 424
                                                                                                 dv2= 22,667 ft/sec
                                                                                                 max 2nd stg accel = 0.976
    After LRB staging (1st + 2nd stages)
     tstg LRBs = 228 sec
mach@stg = 10.8
dynp@stg = 51.1 psf
all@stg = 191,034 ft
                                                                                               Launch Escape System Jettison
                                                                                                 alt@ les jettison = 325,157 ft
      Core throttle = 0.600
                                                                                               2nd Stage MECO
     dv1 = 15,990 ft/sec
                                                                                                 tstg core = 585 sec
      max C ore stg accel = 4.00
                                                                                                 dvt = 29,672 ft/sec
```

Figure 6C-3. LV 2 Summary

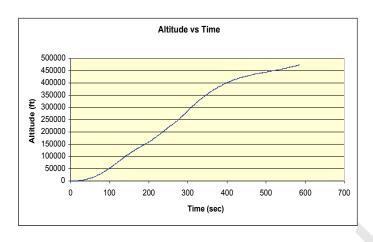


Figure 6C-4. Altitude versus Time

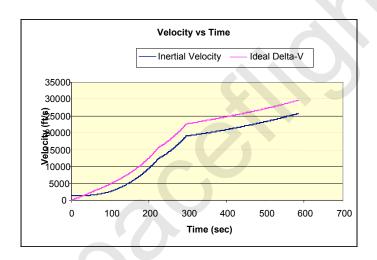


Figure 6C-5. Velocity versus Time

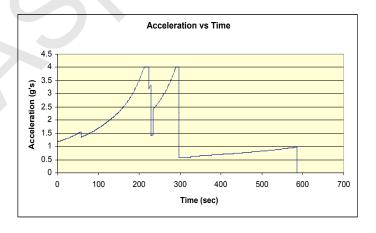


Figure 6C-6. Acceleration versus Time

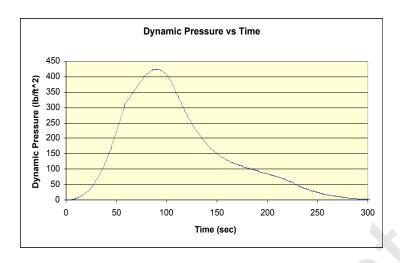


Figure 6C-7. Dynamic Pressure versus Time

### 6C.5.1.3 Launch Vehicle 4 (LV 4)

### 6C.5.1.3.1 Vehicle Description

LV 4 (**Figure 6C-8**) is the Delta IV HLV configuration with a new upper stage for CEV. The vehicle consists of two common core, liquid boosters strapped to a center CCB. All booster elements are LOX/LH2. "Common core" refers to all three booster elements being similar in dimensions, engines, and manufacturing commonality. There are some differences in structural design for the different load paths for the central core and the strap-on boosters. A new larger, more powerful upper stage has been added to the configuration to increase the payload capability of the vehicle. The new upper stage is LOX/LH2 with four RL–10A–4–2 engines.

This vehicle concept was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 77.3 nmi. The net payload capability of LV 4 is 28.4 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 22.9 mT. No ground rules or constraints were violated for this LV analysis.

A special consideration was required to analyze this vehicle: Additional mass was added to the vehicle for human rating of the ELV.



Figure 6C-8. LV 4 General Configuration

### 6C.5.1.3.2 Vehicle Sizing

The mass properties for LV 4 are shown in **Table 6C-2**. The primary structural mass of the strap-on boosters and the core stage was increased 3.1 percent to provide a 1.4 FS. The second-stage design margin already included a 1.4 FS.

Table 6C-2. LV 4 INTROS Mass Summary

MASS PROF	PERTIES ACCOUNTING	G	
VEHICLE: Delta IV HL	,	•	
	on Liquid Booster (1 RS MASS SUBTOTALS	MASS TOTALS	
ITEM	Primary		
	lbm	lbm	
Primary Body Structures	29,843		
Secondary Structures	907		
Separation Systems	509	h	
TPSs	53		
TCSs	1,698		
MPS	21,098		
Power (Electrical)	772		
Power (Hydraulic)	557		
Avionics	417		
Miscellaneous	193		
STAGE DRY MASS WITH GRO	OWTH	56,047	
Residuals	5,252		
Reserves	678		
In-flight Fluid Losses	81		
STAGE BURNOUT MASS		62,058	
Main Ascent Propellant	451,749		
Engine Purge Helium	17		
STAGE GROSS LIFTOFF MAS	S	513,824	
STAGE: Seco	ond Stage (4 RL-10A-4-2	2)	
Primary Body Structures	6,219		
Secondary Structures	152		
Separation Systems	70		
TCSs	530		
MPS	2,583		
Power (Electrical)	863		
Power (Hydraulic)	278		
Avionics	813		
Miscellaneous	36		
STAGE DRY MASS WITHOUT	GROWTH	11,545	
Dry Mass Growth Allowance	1,344		
STAGE DRY MASS WITH GRO	OWTH	12,889	
Residuals	806		
Reserves	1,819		
In-flight Fluid Losses	38		
STAGE BURNOUT MASS		15,553	
Main Ascent Propellant	57,241	,	
Engine Purge Helium	2		

STAGE GROSS LIFTOFF MASS	72,796	
STAGE: C		
Primary Body Structures	33,664	
Secondary Structures	907	
Separation Systems	1,944	
TPSs	53	
TCSs	2,139	
MPS	21,098	
Power (Electrical)	788	
Power (Hydraulic)	557	
Avionics	417	
Miscellaneous	213	
STAGE DRY MASS WITH GROW	61,782	
Residuals	5,252	
Reserves	678	
In-flight Fluid Losses	81	
STAGE BURNOUT MASS	<b>*</b>	67,793
Main Ascent Propellant	451,749	
Engine Purge Helium	17	
STAGE GROSS LIFTOFF MASS	519,558	
Payload	69,582	
LES	9,300	
Upper Stage(s), Gross	72,796	
Strap-on(s), Gross Liftoff	1,027,647	
VEHICLE GROSS LIFTOFF MAS	SS	1,698,884

# 6C.5.1.3.3 Structural Analysis

Since this is an existing vehicle, an LV analysis analysis was not run, except that correlation factors were run for the upper stage. The rest of the structural weights were scaled from the INTROS model. The structural configuration is shown in **Figure 6C-9**, and the results of the structural loads analysis are provided in **Figure 6C-10**.

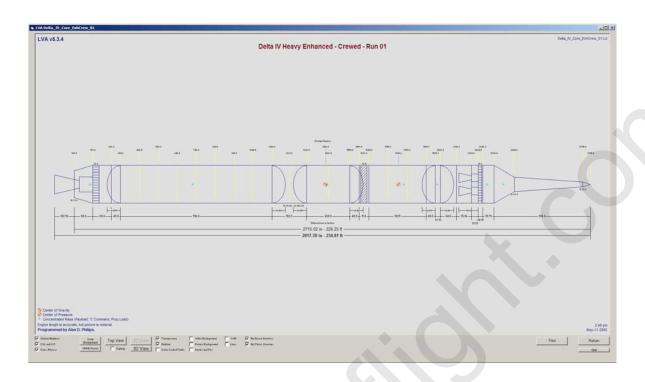


Figure 6C-9. LVA Structural Configuration

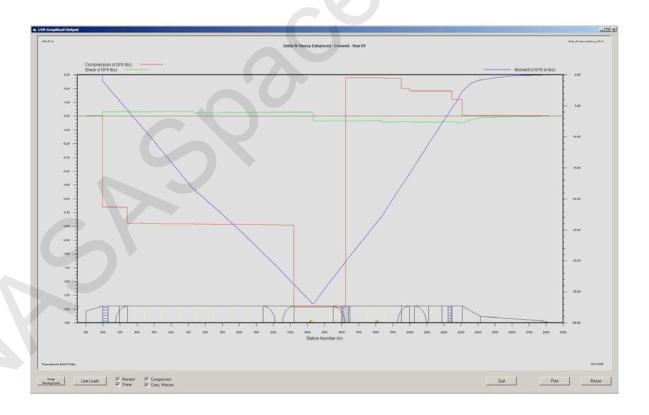
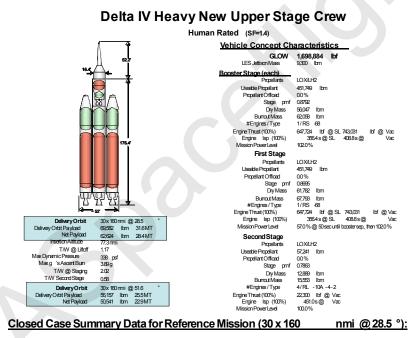


Figure 6C-10. LVA Structural Loads Analysis Results

### 6C.5.1.3.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-11**. Selected trajectory parameters are shown graphically in **Figures 6C-12** through **6C-15**. The vehicle has a 1.17 T/W ratio at liftoff, which is considered acceptable since it is an EELV. Fifty seconds into the flight the RS–68 on the core stage is throttled to 57 percent in order to avoid simultaneous burnout of the core and boosters. Maximum dynamic pressure is 338 psf at 93.7 sec into the flight. The maximum acceleration during the liquid booster burn is 3.89 g's, 3.44 g's during core stage burn after Liquid Rocket Booster (LRB) separation, and 1.03 g's during the upper stage burn. The boosters burn out 244 sec into the flight at an altitude of 237,526 ft and Mach 11.1, during this time the throttled core has burned 903,498 lb of propellant. The core is throttled up at staging and the core burnout occurs at 328 sec into the flight with a corresponding altitude of 353,988 ft and Mach 16.7. The T/W ratio at core ignition is 2.02 and the upper stage is 0.58, orbital injection occurs at 617 sec at 77.3 nmi.



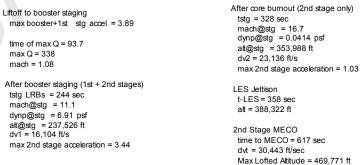


Figure 6C-11. LV 4 Summary

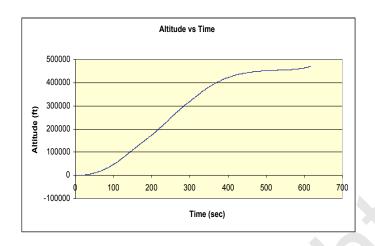


Figure 6C-12. Altitude versus Time

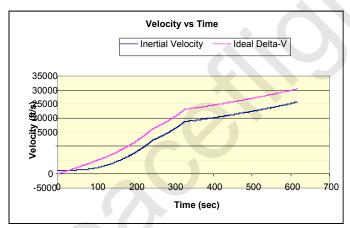


Figure 6C-13. Velocity versus Time

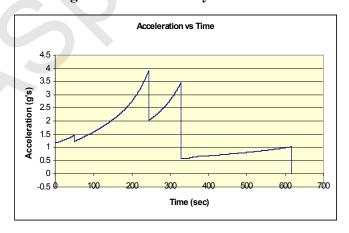


Figure 6C-14. Acceleration versus Time

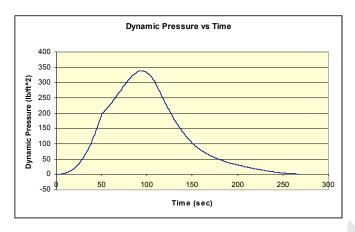


Figure 6C-15. Dynamic Pressure versus Time

### 6C.5.1.4 Launch Vehicle 5.1 (LV 5.1)

### 6C.5.1.4.1 Vehicle Description

LV 5.1 (**Figure 6C-16**) is a two-stage series-burn LV for CEV. The first stage is an ET diameter, LOX/RP stage with five RD–180 engines. The LOX/LH2 second stage is also ET diameter with four J–2S+ engines for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 78.5 nmi. All liquid engines were operated at a 100 percent power level in the analysis.

The net payload capability of LV 5.1 is 70.4 mT to a  $30 \times 160$  nmi orbit at a 28.5 deg inclination. The net payload to  $30 \times 160$  nmi at a 51.6 deg inclination is 66.4 mT. No ground rules or constraints were violated for this LV analysis.

No special considerations were required to analyze this vehicle.

### 6C.5.1.4.2 Vehicle Sizing

The mass properties for LV 5.1 are shown in **Table 6C-3**. The Crew Exploration Vehicle (CEV) adapter mass is included in the second-stage primary body structures mass.



Figure 6C-16. LV 5.1 General Configuration

### Table 6C-3. LV 5.1 INTROS Mass Summary

MASS PROPERTIES ACCOUNTING					
VEHICLE: (5/4+) Atlas Evolved (8-m Core) Crew – Blk 2 STAGE: Second Stage (4 J–2S+)					
	MASS SUBTOTALS				
ITEM	Tertiary	Secondary	Primary	TOTALS	
	lbm	lbm	lbm	lbm	
Primary Body Structures			45,840		
Secondary Structures			2,987		
Separation Systems			2,453		
TPSs			403		
TCSs			2,175	_	

MPS		25,283	
Power (Electrical)		1,775	
Power (Hydraulic)		807	
Avionics		590	
Miscellaneous		194	
STAGE DRY MASS WITHOUT GROW	TH		82,508
Dry Mass Growth Allowance		8,714	, ,
STAGE DRY MASS WITH GROWTH (	mdry)		91,222
Residuals		7,431	
Reserves		8,085	
In-flight Fluid Losses		117	
STAGE BURNOUT MASS (mbo)			106,856
Main Ascent Propellant		666,458	
Engine Purge Helium		75	
STAGE GROSS LIFTOFF MASS (mgro	ss)		773,390
STAGI	E: First Stage (5–RD–180s)		
Primary Body Structures		71,470	
Secondary Structures		4,893	
Separation Systems		3,269	
TPSs		1,473	
TCSs		2,006	
MPS		84,131	
Power (Electrical)		2,017	
Avionics		670	
Miscellaneous		264	
STAGE DRY MASS WITHOUT GROW	TH		170,194
Dry Mass Growth Allowance		14,182	
STAGE DRY MASS WITH GROWTH (	mdry)		184,376
Residuals		23,830	
Reserves		1,832	
Fuel Bias	1,832		
STAGE BURNOUT MASS (mbo)			210,038
Main Ascent Propellant		2,411,799	
Liquid Oxygen	2,003,925		
RP-1	736,755		
Offload Core Stage	-328,882		
Engine Purge Helium		233	
STAGE GROSS LIFTOFF MASS (mgro	ss)		2,622,069
Payload		172,535	
LES		9,300	
Upper Stage(s), Gross		773,390	
VEHICLE GROSS LIFTOFF MASS (mg	gross_veh)		3,577,294

## 6C.5.1.4.3 Structural Analysis

LV analysis for LV 7.4 was applied to LV 5.1 because the core and upper stage are the same for both concepts.

### 6C.5.1.4.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-17**. Selected trajectory parameters are shown graphically in **Figures 6C-18** through **6C-21**. The vehicle exhibits a 1.20 T/W ratio at liftoff. Maximum dynamic pressure is 549 psf at 85.8 sec in the flight. The maximum acceleration is 4.00 g's during the first stage and 3.93 g's during the second stage. Staging occurs at 174.8 sec into the flight at an altitude of 185,516 ft and Mach 7.25. The T/W ratio at second-stage ignition is 1.14. Orbital injection occurs at 448.9 sec at 78.5 nmi.

#### Vehicle Concept Characteristics GLOW 3,577,294 lbf 9300 lbm First Stage Procellants LOX/RP -1 Useable Propellant Propellant Offload 2,411,799 0.0% Stage pmf Dry Mass 0.9198 184,376 Burnout Mass 210,038 210,038 lbm 5/RD -180 #Engines/Type Engine Thrust (100%) Engine Isp (100%) 858,377 lbf @ SL 933,400 311.2 s @ SL 338.4 s @ 100.0%. Mission Power Level Second Stage Propellants IOX/H2 Useable Propellant Propellant Offload 666,458 0.0% Burnout Mass 106.856 lbm #Engines/Type Engine Thrust (100%) Engine Isp (100%) 4/J -2S+ 274,500 lbf 451.5s@ Mission Power Level 100.0% Delivery Orbit ery Orbit Payload 30 x 160 nmi @ 28.5 172,535 Net Payload Insertion Altitude 155,282 lbm 70.4MT T/W @ Liftoff Max Dynamic Pressure 549 nsf Maxg 's Ascent Burn T/W Second Stage Net Payload

Atlas Evolved (5 RD -180 & 4 J -2S+) - Crew

### Closed Case Summary Data for Reference Mission (30 x 160 nmi @ 28.5 °):

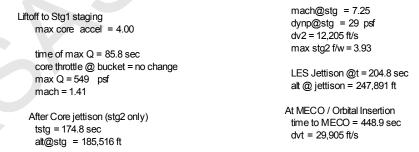


Figure 6C-17. LV 5.1 Summary

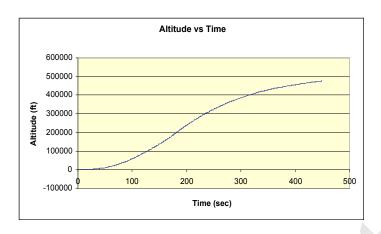


Figure 6C-18. Altitude versus Time

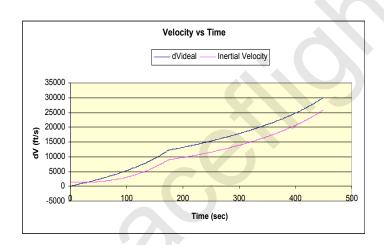


Figure 6C-19. Velocity versus Time

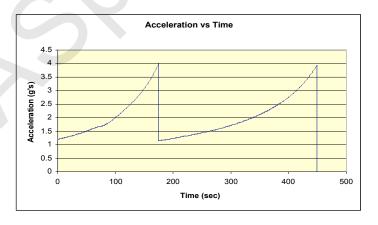


Figure 6C-20. Acceleration versus Time

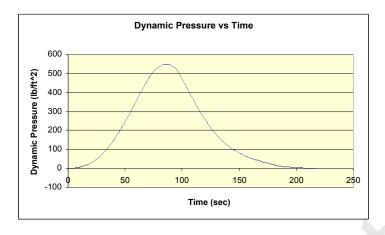


Figure 6C-21. Dynamic Pressure versus Time

### 6C.5.1.5 Launch Vehicle 9 (LV 9)

### 6C.5.1.5.1 Vehicle Description

LV 9 (**Figure 6C-22**) is a two-stage series-burn LV for CEV. The structural configuration is shown in **Figure 6C-23**, and the results of the loads analysis are provided in **Figure 6C-24**. The first stage is a 5.4-m diameter LOX/RP stage with two RD–180 engines. The LOX/LH2 second stage is also 5.4-m diameter with four LR–60 engines for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 63.5 nmi. All liquid engines were operated at a 100 percent power level in the analysis.

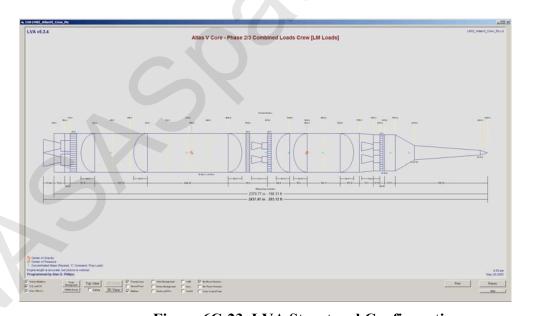


Figure 6C-23. LVA Structural Configuration



Figure 6C-22. LV 9 General Configuration

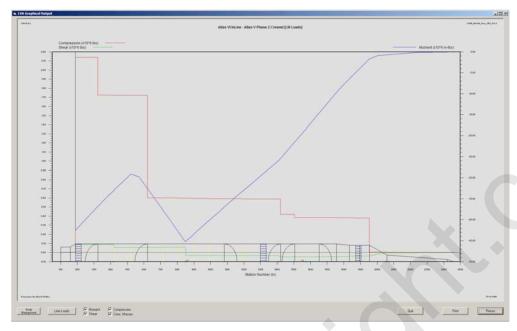


Figure 6C-24. LVA Structural Loads Analysis Results

The net payload capability of LV 9 is 25.9 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 24.5 mT. No ground rules or constraints were violated for this LV analysis.

A special consideration was required to analyze this vehicle: The propellant tanks of both stages were designed with nested domes to be consistent with the contractors design for this concept.

### 6C.5.1.5.2 Vehicle Sizing

The mass properties for LV 9 are shown in **Table 6C-4**. All hardware was considered to be new for this vehicle concept.

Table 6C-4. LV 9 INTROS Mass Summary

	MASS PROPER	TIES ACCOUNTI	NG		
VEHICLE: Atlas Phase 2 Crew – Blk 2 STAGE: Stage 2 - (4 LR–60)					
	N	MASS SUBTOTALS			
ITEM	Tertiary	Secondary	Primary	TOTALS	
	lbm	lbm	lbm	lbm	
Primary Body Structures			11,882		
Secondary Structures			211		
Separation Systems			120		
TPSs			121		
TCSs			452		
MPS			6,828		
Power (Electrical)			934		
Power (Hydraulic)			176		

Avionics	813	
Miscellaneous	57	
STAGE DRY MASS WITHOUT GROWTH	I	21,594
Dry Mass Growth Allowance	2,579	
STAGE DRY MASS WITH GROWTH (md	lry)	24,174
Residuals	1,825	
Reserves	2,039	
In-flight Fluid Losses	26	
STAGE BURNOUT MASS (mbo)		28,063
Main Ascent Propellant	158,833	
Engine Purge Helium	7	
STAGE GROSS LIFTOFF MASS (mgross)		186,902
STAGE:	First Stage (2RD–180)	<b>\</b>
Primary Body Structures	28,267	
Secondary Structures	1,945	
Separation Systems	1,919	
TPSs	188	
TCSs	895	
MPS	41,210	
Power (Electrical)	984	
Power (Hydraulic)	1,373	
Avionics	542	
Miscellaneous	150	
STAGE DRY MASS WITHOUT GROWTH	I	77,473
Dry Mass Growth Allowance	7,953	
STAGE DRY MASS WITH GROWTH (md	(ry)	85,426
Residuals	8,629	
Reserves	735	
In-flight Fluid Losses	199	
STAGE BURNOUT MASS (mbo)		94,990
Main Ascent Propellant	1,054,852	
Engine Purge Helium	32	
STAGE GROSS LIFTOFF MASS (mgross)		1,149,874
Payload	63,562	
Payload Shroud	9,300	
Upper Stage(s), Gross	186,902	
VEHICLE GROSS LIFTOFF MASS (mgro	ss_veh)	1,409,638

### 6C.5.1.5.3 Structural Analysis

The loads plot is a combined worst case including pre-launch, liftoff, maximum dynamic pressure (max q), and maximum acceleration (max g). The tie-down loads are assumed to be carried by the core vehicle. The compression loads show a major jump where the LOX tank, RP tank, and payload loads are integrated into the outside structure. For the purposes of the analysis, all boosters were assumed to introduce axial loads at the aft of the core.

### 6C.5.1.5.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics for the Atlas Phase 2 are shown in **Figure 6C-25**. Selected trajectory parameters are shown graphically in **Figures 6C-26** through **6C-29**. The vehicle lifts off with a T/W ratio of 1.22, quickly achieving the 4.00 g's limit. A maximum dynamic pressure of 532 psf is met along the way at 81.3 sec through the flight. The maximum acceleration during the second stage is 2.62 g's. Staging occurs at 194 sec at an altitude of 218,594 ft and Mach 10.2. The T/W ratio at second stage ignition is 0.91. Orbital injection occurs at 502 sec dropping into orbit at 63.5 nmi.

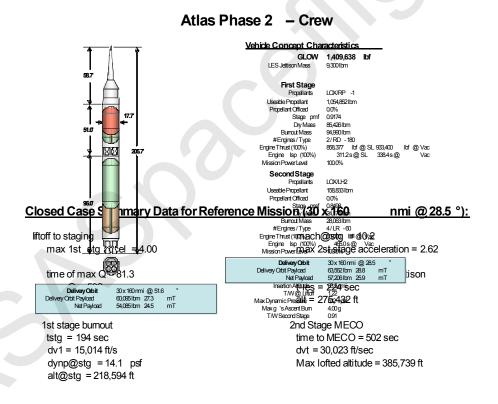


Figure 6C-25. LV 9 Summary

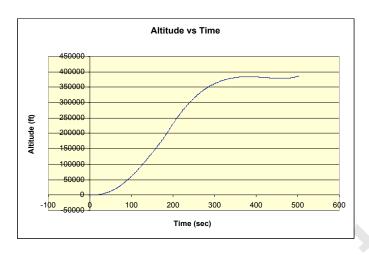


Figure 6C-26. Altitude versus Time

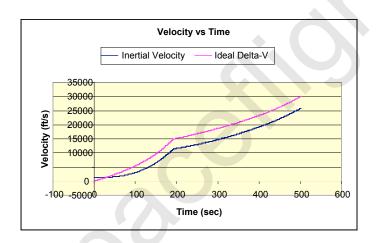


Figure 6C-27. Velocity versus Time

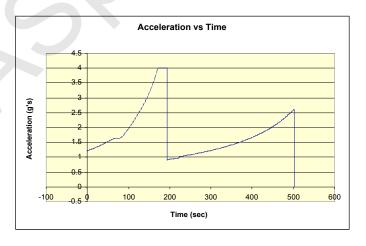


Figure 6C-28. Acceleration versus Time

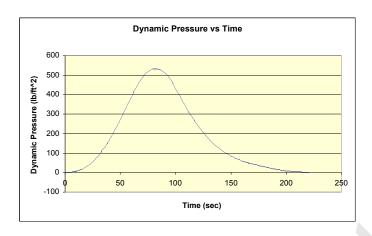


Figure 6C-29. Dynamic Pressure versus Time

### 6C.5.1.6 Cost Analysis EELV Vehicle for Crew (LV 2, LV 4, LV 5.1, LV 9)

The EELV vehicles for crew include a human-rated Delta IV HLV, human-rated Atlas V HLV, and two vehicle configurations that evolved from the Atlas. All vehicles have new upper stages.

### 6C.5.1.6.1 Inputs

### Booster Stage for Delta IV HLV and Atlas V HLV

The booster stage for the Delta IV HLV and the Atlas V HLV crew vehicles is very similar to the existing EELVs in production today.

### Structure and Tanks

Both metallic and composite intertanks, interstages, and thrust structures have been used on various programs. Design and manufacturing capabilities exist today. Material is either 2219 aluminum or Aluminum-Lithium (Al-Li). Shrouds are made of graphite-epoxy panels, based on Titan and Delta IV designs. Structures and tanks are well understood with sufficient manufacturing capability in existence. All structures have similar subsystems (to EELV, Shuttle, or ET). NAFCOM cost estimate assumptions assumed existing structure/tanks with similar subsystems validated in the relevant environment. Greater than 50 percent will require testing and qualification.

### Main Propulsion System—Less Engine

The MPS will take significant heritage from the existing EELV MPS subsystem. However, the existing design will need to accommodate any changes to the subsystem for human rating the RD–180 and the RS–68 engines. NAFCOM cost estimate assumptions assumed an existing design with similar subsystems validated in the relevant environment. Greater than 50 percent will require testing and qualification.

Engine: RD-180

RD-180 is currently in production and is being flown on the Atlas V. However, the production of the RD-180 is presently occurring in Russia. Coproduction in America is desired. Also, design must meet requirements from the program Human Rating Plan.

DDT&E costs assumes minimal hardware modification for human rating, expansion/enhancement of sensor suite, activation of flight redlines, and integrated Fault Detection, Isolation, and Recovery (FDIR).

Engine: RS-68

Bottom-up cost assessment was performed on the RS-68U engine for the RS-68 upgrade for human rating. Engine development assumes regenerative nozzle, main injector, and turbo pump modifications.

### Avionics and Software

The avionics subsystem must support Fail Operational/Fail Safe vehicle fault tolerant requirements. Upon the first failure, the vehicle will keep operating. The second failure will safely recommend an abort. Crew abort failure detection and decision-making capabilities have been demonstrated and are ready for flight. All architectures will meet these requirements, either by adding a modification for instrumentation redundancy for the EELV health management system, or by providing the capabilities through the new design of the avionics for Shuttle-derived configurations.

### EELV Avionics Hardware

EELV avionics was developed for nonhuman flight. To meet the requirements from the program Human Rating Plan, DDT&E costs assume minimal hardware modification for human rating, expansion/enhancement of sensor suite to include redundancy for instrumentation and rate gyro units, addition of an error detection system, and modified flight software. The extent of modification to the existing system is dependent on the level of redundancy required for human rating—single, dual, or triple string redundancy—still under discussion at the time of this report. The avionics hardware suite is essentially the current EELV with these improvements added through the Integrated Vehicle Health Management (IVHM) kit.

### EELV Software

EELV software has been developed for the current vehicles. To meet the new requirements, existing software is used and/or modified. In addition, new software will be required for command and control, database, test, navigation/guidance management, and health management for both ground and flight software. It is also anticipated that a backup Guidance, Navigation, and Control (GN&C) software development will be required.

### Other Booster Subsystems

The remaining booster subsystems all used existing design and technology. Thermal, power, and range safety subsystems are in existence today, and have been validated for the relevant environment. NASA and Air Force Cost Model (NAFCOM) cost estimate assumptions assumed existing structure/tanks with similar subsystems validated in the relevant environment. Minor modifications will be tested and qualified.

### **EELV Upper Stage**

Structure and Tanks

Both metallic and composite intertanks, interstages, and thrust structures have been used on various programs. Design and manufacturing capabilities exist today. The critical elements will be the development of the separation system, a new interstage, and the payload adapter. Material is either 2219 aluminum or Al-Li. Shrouds are made of graphite-epoxy panels, which are based on Titan and Delta IV designs. Structures and tanks are well understood with sufficient manufacturing capability in existence. All structures have similar subsystems (to EELV, Shuttle, or ET). NAFCOM cost estimate assumptions assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

MPS—Less Engine

The MPS will take significant heritage from the existing EELV MPS subsystem. However, a new design is needed to accommodate either an increased number of engines over the current EELV design, or a new upper stage engine. NAFCOM cost estimate assumptions assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

Engine: RL-10s

RL-10 engines are currently being used today. However the engine is not human rated. With this said, the amount of design work, and hence cost, associated with human rating this highly reliable engine is subject to some debate. NASA has completed an independent evaluation on the amount of redesign needed, including increased engine component redundancy, FDIR, and other human-rated and mission requirements.

Engine: J-2S

Two different variants of the J–2S were analyzed for this study. The first assumed a design as close as possible to the original Apollo-era J–2S. The second variant was a J–2S redesign, specifically designed from optimal reliability and low production costs. Once again, cost analysis was performed using a bottom-up approach.

Engine: LR-60

LR-60 is the proposed next generation cryogenic upper stage rocket engine. All major components have been independently tested by the contractor. Once again, cost analysis was performed using a bottom-up approach, with input from the engine contractor and engineering assessments.

Avionics Architecture

The avionics subsystem must support Fail Operational/Fail Safe vehicle fault tolerant requirements. Upon the first failure, the vehicle will keep operating. The second failure will safely recommend an abort. Crew abort failure detection and decision-making capabilities have been demonstrated and are ready for flight. All architectures will meet these requirements, either by adding a modification for instrumentation redundancy for the EELV health management system, or providing the capabilities through the new design of the avionics for Shuttle-derived configurations.

#### EELV Avionics Hardware

EELV avionics was developed for nonhuman flight. To meet the requirements from the program Human Rating Plan, DDT&E costs assume minimal hardware modification for human rating, expansion/enhancement of a sensor suite to include redundancy for instrumentation and rate gyro units, addition of an error detection system, and modified flight software. The extent of modification to the existing system is dependent on the level of redundancy required for human rating—single, dual, or triple string redundancy—still under discussion at the time of this report. The avionics hardware suite is essentially the current EELV with these improvements added through the Integrated Vehicle Health Management (IVHM) kit.

### EELV Software

EELV software has been developed for the current vehicles. To meet the new requirements, existing software is used and/or modified. In addition, new software will be required for command and control, database, test, navigation/guidance management, and health management for both ground and flight software. It is also anticipated that a backup GN&C software development will be required.

### Other Upper Stage Subsystems

The remaining upper stage subsystems all used existing design and technology. Thermal, power, and range safety subsystems are in existence today, and have been validated for the relevant environment. NAFCOM cost estimate assumptions assumed existing structure/tanks with similar subsystems validated in the relevant environment. Minor modifications will be tested and qualified.

### 6C.5.1.6.2 DDT&E

The lowest cost option in this group of vehicles is the Delta IV HLV with a new upper stage. This vehicle is basically designed, and only needs to meet the requirements for human rating. Next is the Atlas V HLV. It is also basically designed, but will require human rating and Americanization of the RD–180 engine. The Phase 2 Atlas is a new design, though it does have some heritage with the existing Atlas. For the Phase 2, the RD–180 will have to be human rated and Americanized, and a new upper stage engine development will be required. The most expensive vehicle in this group is the evolved Atlas with the increased diameter core to accommodate five RD–180 engines. This evolved Atlas will also require a new engine in the upper stage.

### **6C.5.1.6.3 Production**

LV 2, LV 4, and LV 5.1 are ELV-based crew vehicles, derived from either existing Atlas or Delta configurations. LV 2 and LV 4 are both 3-core human-rated versions of the Atlas and Delta vehicles respectively. LV 9 is a 5.4-m Atlas core vehicle with two RD-180 engines and an upper stage with four LR-60 engines, while LV 5.1 is an 8-m Atlas with five RD-180s and no upper stage. The LV 2 configuration is the cheapest to produce followed by the LV 4, LV 9 and LV 5.1 vehicles in ascending order of production cost.

### **6C.5.1.6.4 Operations**

Launch operations for these concepts take place at Cape Canaveral Air Force Station (CCAFS). The lowest annual cost at six flights per year occurs with LV 4 despite the integration of the three cores. The highest cost is for LV 2. Next highest is LV 9.

### **6C.5.1.6.5** Facilities

Facilities costs are for the modifications to the integration facilities, launch pads and Government Supplied Equipment (GSE). The costs are the same for all four concepts, as shown in **Table 6C-5**.

Phase	Relative Cost Position			
Vehicle	2	4	5.1	9
DDT&E	1.23	1.02	2.60	1.74
Production	0.91	0.78	1.43	0.94
Operations	2.55	1.02	1.25	2.01
Facilities	0.92	0.92	0.92	0.92

Table 6C-5. Relative Comparison of ELV Crew Vehicle Costs

### 6C.5.1.7 Safety/Reliability Analysis (EELV-Derived Crew LVs)

The same similarity analysis tool used to estimate the first order Shuttle-derived crew LVs' Loss of Mission (LOM) and Loss of Crew (LOC) estimates was used to estimate the EELV-derived estimates. A complete description of the analyses methodology is provided in **Appendix 6D**, **Risk and Reliability**. Likewise, a complete description of how reliability predictions were developed for the individual LV systems that were used in the similarity analyses is provided in **Appendix 6D**, **Risk and Reliability**. The four EELV-derived LV estimates are shown in **Figures 6C-30** and **6C-31**. Detailed analyses results are provided in **Appendix 6D**, **Risk and Reliability**.

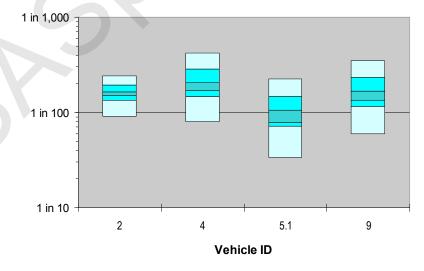


Figure 6C-30 CLV LOM Estimates

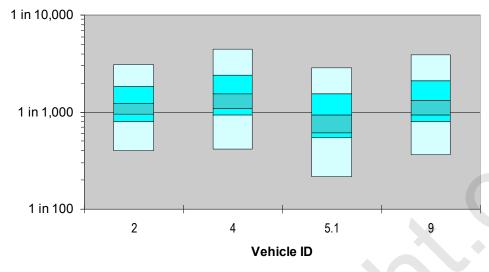


Figure 6C-31 CLV LOC Estimates

**Figure 6C-32** shows the LV subsystem risk contributions. Vehicle reliability was equally dominated by non-catastrophic shutdown risk of all engines on all stages and air-start risk of second-stage engines.

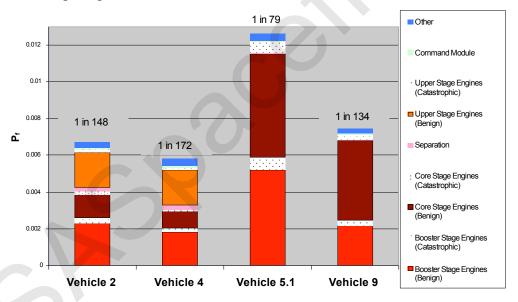


Figure 6C-32. LV Subsystem Risk Contributions

In addition to the similarity analyses, Aerospace Corporation used a "risk decomposition" model that is based on the risk/reliability characteristics of the LV subsystem and major components. For components with earlier heritage the model utilizes a risk estimate based on flight data. Components with minimum heritage utilized a risk estimate based on flight data of components with similar function and design characteristics. The Aerospace LOM for LV 2 (human-rated Atlas V with new upper stage) was 1 in 35, (based on Aerospace's probability of mission success as high

as 0.9711). The LOM estimate for LV 4 (Delta IV HLV with new upper stage) was 1 in 45, (based on a probability of mission success as high as 0.9778). The Aerospace LOC estimates were 1 in 87 for LV 2, and 1 in 296 for LV 4. Note that the Atlas and Delta LVs analyzed by Aerospace were not identical to the vehicle analyzed in the similarity model (FIRST). Aerospace estimates were based on upgrades to the upper stage that were not modeled in NASA's analyses. In addition, Aerospace estimates did not address the abort system reliability. This accounts for the lower LOC estimates. But, Aerospace estimates for both LOC and LOM confirm the comparative ranking of these two vehicles, both analyses methods ranked LV 4 higher than LV 2.

### 6C.5.1.8 Launch Vehicle 15 (LV 15)

### 6C.5.1.8.1 Vehicle Description

LV 15 (**Figure 6C-33**) is a two-stage series-burn LV for CEV. **Figure 6C-34** shows the structural configuration, while **Figure 6C-35** provides the results of the structural loads analysis. The first stage is a five-segment RSRB (Hydroxyl Terminated Poly-Butadiene (HTPB) propellant). The second stage is LOX/LH2 with four LR-85 engines for propulsion. The LR-85 engine would be a new expander cycle engine with 85 klb of thrust. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 59.5 nmi. The LR-85 engines were run at a throttle setting of 100 percent. The purpose of this analysis was to evaluate the performance of the LR-85 engine cluster as an upper stage engine application in comparison to a modified J-2S (J-2S+) engine or SSME.

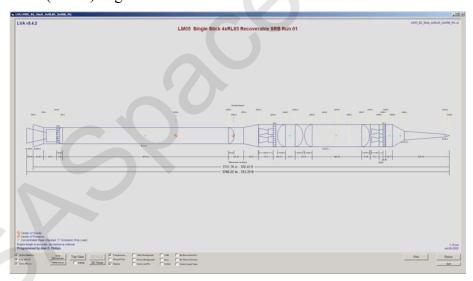


Figure 6C-34. LVA Structural Configuration

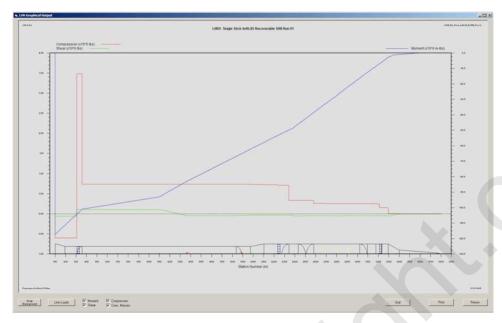


Figure 6C-35. LVA Structural Loads Analysis Results

The net payload capability of LV 15 is 27.0 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 25.3 mT. No ground rules or constraints were violated for this LV analysis.

No special considerations were required to analyze this vehicle.

### 6C.5.1.8.2 Vehicle Sizing

The mass properties for Stage 2 of LV 15 are shown in **Table 6C-6**. The mass properties for the five-segment SRB were supplied by the Solid and Hybrid Propulsion System Branch of the MSFC Engineering Directorate and used as delivered with only two modifications. The current SRB nosecone was removed and an interstage added to complete the vehicle configuration.

**Table 6C-6. LV 15 INTROS Mass Summary** 

MASS PROPERTIES ACCOUNTING					
VEHICLE: In-line Five-Segment SRM with 4 LR-85 Crew - Blk 2					
		nd Stage (4 LR–85			
	N	MASS SUBTOTAL	LS	MASS	
ITEM	Tertiary	Secondary	Primary	TOTALS	
	lbm	lbm	lbm	lbm	
Primary Body Structures			13,477		
Secondary Structures			779		
Separation Systems			99		
TPSs			53		
TCSs			1,284		
MPS			9,766		

Power (Electrical)			1,484	
Power (Hydraulic)			305	
Avionics			813	
Miscellaneous			129	
STAGE DRY MASS WITHOUT	GROWTH			28,188
Dry Mass Growth Allowance			2,778	
STAGE DRY MASS WITH GRO	OWTH (mdry)			30,966
Residuals			2,731	
Reserves			2,328	
In-flight Fluid Losses			50	
STAGE BURNOUT MASS (mbd	p)			36,075
Main Ascent Propellant			260,093	
Engine Purge Helium			29	
STAGE GROSS LIFTOFF MAS	SS (mgross)			296,198
	STAGE: First (Fiv	e-Segment SRB	)	
STAGE BURNOUT MASS (mbd	))			222,697
Main Ascent Propellant			1,434,906	
STAGE GROSS LIFTOFF MAS	SS (mgross)			1,657,603
Net Vehicle Stackup				
Payload			66,027	
LES			9,300	
Upper Stage(s), Gross			296,198	
NET VEHICLE GROSS LIFTO	FF MASS (mgross	veh)		2,029,128

#### 6C.5.1.8.3 Structural Analysis

For the purpose of the LV analysis, only the loads data forward of station number 1972.3 were used. The SRB was assumed to be structurally capable of handling the loads. The loads plot is a combined worst case including pre-launch, liftoff, max q, and max g. The compression loads show a major jump where the LOX tank loads are integrated into the outside structure. The bending moment shows a steady increase from the tip progressing aftward.

#### 6C.5.1.8.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-36**. Selected trajectory parameters are shown graphically in **Figures 6C-37** through **6C-40**. The vehicle exhibits a 1.77 T/W ratio at liftoff. Maximum dynamic pressure is 986 psf at 48.4 sec in the flight. The maximum acceleration during the first stage is 3.47 g's and 3.45 g's during the second stage. Staging occurs at 132.5 sec into the flight at an altitude of 189,843 ft and Mach 7.0. The T/W ratio at second-stage ignition is 0.91. Orbital injection occurs at 476.8 sec at 59.5 nmi.

# 5-Segment SRB with 4 LR -85 - Crew

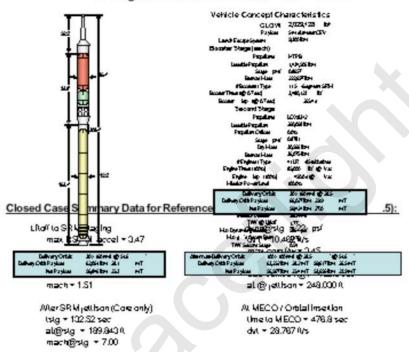


Figure 6C-36. LV 15 Summary

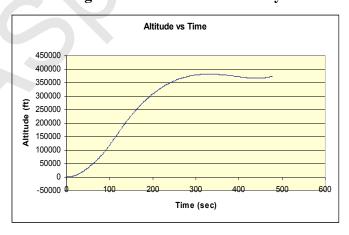


Figure 6C-37. Altitude versus Time

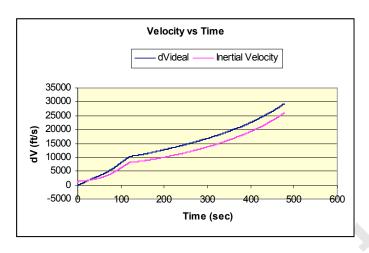


Figure 6C-38. Velocity versus Time

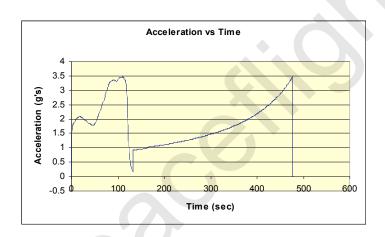


Figure 6C-39. Acceleration versus Time

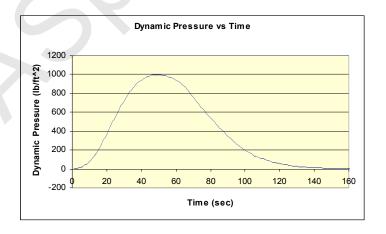
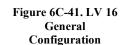


Figure 6C-40. Dynamic Pressure versus Time

# 6C.5.1.9 Launch Vehicle 16 (LV 16)

## 6C.5.1.9.1 Vehicle Description

LV 16 (**Figure 6C-41**) is a two-stage series-burn LV for CEV. The structural configuration is shown in **Figure 6C-42**, with the results of the loads analysis provided in **Figure 6C-43**. The first stage is a five-segment RSRB (HTPB propellant). The second stage is LOX/LH2 with one J-2S+ engine for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 59.4 nmi. The J-2S+ engine was run at a throttle setting of 100 percent. The purpose of this analysis was to evaluate the performance of the J-2S+ as an upper stage engine in comparison to an SSME and cluster of LR-85s.



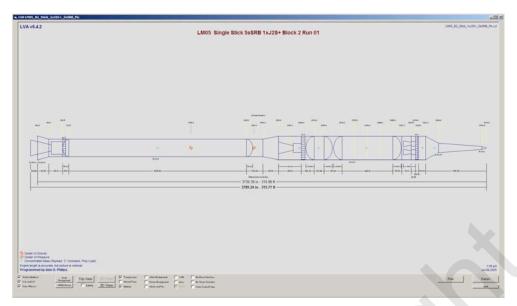


Figure 6C-42. LVA Structural Configuration

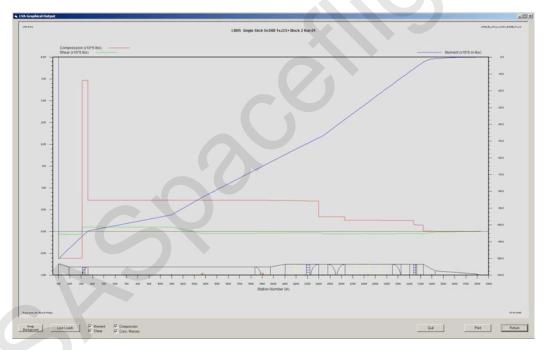


Figure 6C-43. LVA Structural Loads Analysis Results

The net payload capability of LV 16 is 25.8 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 24.3 mT.

For this concept, the Isp of the J–2S+ engine of the upper stage was analyzed at 451.0 sec instead of 451.5 sec, which is the quoted value for this engine. Performance analysis at the corrected Isp value for this engine produced a 0.06 mT payload increase and this result is not considered significant.

No special considerations were required to analyze this vehicle.

## 6C.5.1.9.2 Vehicle Sizing

The mass properties for the second stage of LV 16 are shown in **Table 6C-7**. The mass properties for the five-segment SRB were supplied by the Solid and Hybrid Propulsion System Branch of the MSFC Engineering Directorate and used as delivered with only two modifications. The current SRB nosecone was removed and an interstage added to complete the vehicle configuration.

Table 6C-7. LV 16 INTROS Mass Summary

	MASS PROPERT	TIES ACCOUNTI	ING	
VEHICL	E: Five-Segment S	SRB with 1 J–2S+ nd Stage (1 J-2S+)		
MASS SUBTOTALS				MASS
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
Primary Body Structures			13,643	
Secondary Structures			845	
Separation Systems			95	
TPSs			65	
TCSs			1,291	
MPS			6,333	
Auxiliary Propulsion System			147	
Power (Electrical)			1,743	
Power (Hydraulic)			249	
Avionics			513	
Miscellaneous			132	
STAGE DRY MASS WITHOU	JT GROWTH			24,924
Dry Mass Growth Allowance			2,724	
STAGE DRY MASS WITH G	ROWTH (mdry)			27,780
Residuals			2,646	
Reserves			2,196	
In-flight Fluid Losses			41	
STAGE BURNOUT MASS (m	bo)			32,663
Main Ascent Propellant			250,193	•
Engine Purge Helium			28	
RCS Ascent Propellant			300	
STAGE GROSS LIFTOFF MA	ASS (mgross)			283,184
	STAGE: First (I	Five-Segment SRB	3)	
STAGE BURNOUT MASS (m				223,377
Main Ascent Propellant			1,434,906	
STAGE GROSS LIFTOFF MA	ASS (mgross)			1,658,283
Net Vehicle Stackup				
Payload			63,316	
LES			9,300	
Upper Stage(s), Gross			283,184	
NET VEHICLE GROSS LIFT	OFF MASS (mgro	oss veh)		2,014,083

## 6C.5.1.9.3 Structural Analysis

For the purpose of the LV analysis, only the loads data forward of station number 1972.3 were used. The SRB assessment is included in **Appendix 6H**, **Integrated Vehicle Configuration Definition**. The loads plot is a combined worst case including pre-launch, liftoff, max q, and max g. The compression loads show a major jump where the LOX

tank loads are integrated into the outside structure. The bending moment shows a steady increase from the tip progressing aftwards.

## 6C.5.1.9.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-44**. Selected trajectory parameters are shown graphically in **Figures 6C-45** through **6C-48**. The vehicle exhibits a 1.78 T/W ratio at liftoff. Maximum dynamic pressure is 994 psf at 87.0 sec in the flight. The maximum acceleration during the first stage is 3.53 g's and 2.85 g's during the second stage. Staging occurs at 132.5 sec into the flight at an altitude of 202,406 ft and Mach 7.2. The T/W ratio at second-stage ignition is 0.77. Orbital injection occurs at 543.6 sec at 59.4 nmi.

# 5-Segment SRB with 1 J -2S+ - Crew Vehicle Concept Characteristics GLOW ZOIGES BY Lauble Propelling SHUTTER PECALIFIE nmi @ 28.5 °): occel = 0.50 TW PLES Q = 47.3 sec Hong 'admiritar B9Tetthan (8) = 162,5 sec after SR Mijetthan (care anly) MECO / orbital Insettion Islg - 102,52 sec time to MECO = \$40.6 sec ali@stg = 202,406 ft

Figure 6C-44. LV 16 Summary

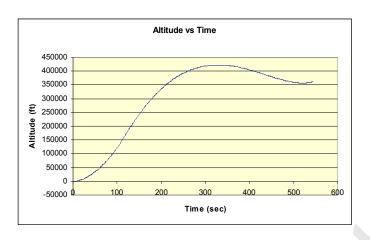


Figure 6C-45. Altitude versus Time

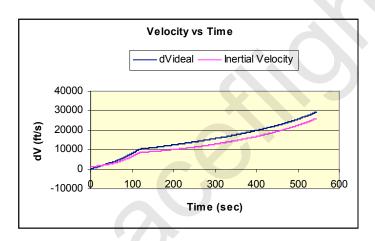


Figure 6C-46. Velocity versus Time

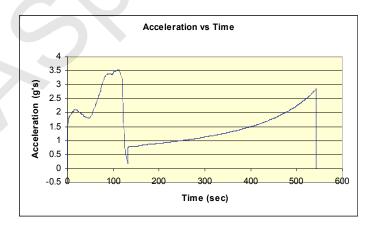


Figure 6C-47. Acceleration versus Time

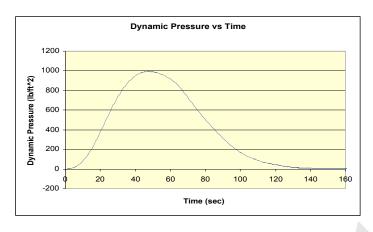


Figure 6C-48. Dynamic Pressure versus Time

# 6C.5.1.10 Cost Analysis for Crew Vehicles (LV 13.1, LV 15, LV 16)

## 6C.5.1.10.1 Inputs

The booster stage for these crew vehicles is either a four-segment RSRB or a five-segment RSRB. The four-segment RSRB is in production today. While the five-segment will draw heavily from the four-segment, some DDT&E will be needed.

Upper stages are used to deliver the payload to the desired orbit. In general, all of the upper stages are considered new designs using existing technology.

#### Structure and Tanks

Both metallic and composite intertanks, interstages, and thrust structures have been used on various programs. Design and manufacturing capabilities exist today. The critical elements will be the development of the separation system, a new interstage, and the payload adapter. Material is either 2219 aluminum or Al-Li. Shrouds are made of graphite-epoxy panels, which are based on Titan and Delta IV designs. Structures and tanks are well understood with sufficient manufacturing capability in existence. All structures have similar subsystems (to EELV, Shuttle, or ET). NAFCOM cost estimate assumptions assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

#### **MPS—Less Engine**

The MPS will take significant heritage from the existing SSME MPS subsystem. However, a new design is needed to accommodate one SSME. NAFCOM cost estimate assumptions assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

Both the J–2S+ and LR–85 engines are equivalent to new engines, due to the length of time that has passed since each was in production. Each will take heritage from the previously existing engine, but the MPS on the upper stage will be new. NAFCOM cost estimate assumptions assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

#### **Engine - SSME**

#### Altitude Start SSME

A 1993 study (NAS8-39211) and a 2004 MSFC study examined the Block 2 engines for altitude start. Both studies determined altitude start will require minor changes, but is considered straightforward. Specialized testing for certification to the environment will be required. Development and certification of altitude start for the Block 2 RS–25d engine is needed. The cost estimate is based on contractor-provided information, which included SSME historical actuals, vendor quotes, and estimates. It also assumes the Shuttle Program continues to pay the fixed cost of infrastructure through Shuttle termination

#### Current Inventory SSME

At the conclusion of the STS Program, there will be 12 Block 2 (RS–25d) engines in inventory if the 28-flight manifest occurs, or 14 engines in inventory with a 16-flight manifest. In either case, the program plans to use at lease 12 of the existing Block 2 assets for the early flights. Assembly, handling, and refurbishment of the existing engines, and conversion of the reusable engine for upper stage use will be needed. Excluded from these costs are any sustaining engineering or Space Shuttle Program (SSP) hardware refurbishment. These early flights will incur some operations costs, which is yet to be determined.

#### Minimal Changes for Expendable Applications SSME

In addition to the minor changes required to altitude start the SSME (RS–25d), it is desirable to make some engine improvements to lower the unit cost and improve producibility. Suggested improvements include low pressure turbomachinery simplifications; a new controller; a Hot Isostatic Press (HIP) bonded main combustion chamber; flex hoses to replace flex joints on four ducts; and simplified nozzle processing. In addition, process changes would be incorporated to eliminate inspections for reuse and accommodate obsolescence of the controller. Development and certification of these minimal changes is designated SSME RS–25e. The estimate is based on contractor-provided information, which included SSME historical actuals, vendor quotes, and estimates

#### **Engine: J-2S**

Two different variants of the J–2S were analyzed for this study. The first assumed a design as close as possible to the original Apollo-era J–2S. The second variant was a J–2S redesign, specifically designed from optimal reliability and low production costs. Once again, cost analysis was performed using a bottom-up approach. All production costs were derived assuming a manufacturing rate of six engines per year.

#### **Engine: LR-85**

LR-85 is a conceptual design engineered to meet derived requirements from the program Human Rating Plan. Production of the LR-85 was assumed to utilize domestic production capabilities. Parametric analysis was performed on the engine using the Liquid Rocket Engine Cost Model (LRECM). Major cost drivers to this model are the Isp and thrust. Options are available to include heritage from older engines.

Appropriate rate curves were applied to both manufacturing and refurbishment to reflect dynamics of the engine production rates with respect to the largely fixed nature of the costs. Theoretical First Unit (TFU) costs from NAFCOM or vendor data were used as a baseline point in the analysis. Historic RS–68, RL–10, and SSME data was also used to help generate Productivity Rate Curves (PRCs).

#### **Avionics and Software**

The avionics subsystem must support Fail Operational/Fail Safe vehicle fault tolerant requirements. Upon the first failure, the vehicle will keep operating. The second failure will safely recommend an abort. Crew abort failure detection and decision-making capabilities have been demonstrated and are ready for flight. All architectures will meet these requirements, either by adding a modification for instrumentation redundancy for the EELV health management system, or providing the capabilities through the new design of the avionics for Shuttle-derived configurations.

Avionics hardware is divided into GN&C, and Command, Control, and Data Handling (CCDH). GN&C provided for attitude control, attitude determination, and attitude stabilization. CCDH provides all the equipment necessary to transfer and process data; communication for personnel, as well as spacecraft operations/telemetry data; and instrumentation for monitoring the vehicle and its performance. Both systems are tied together through the LV software system. LV hardware requirements are well understood.

During the benchmarking activity for NAFCOM, it was discovered that the Cost Estimating Relationships (CERs) for avionics were significantly different from the contractors'. This difference led to NAFCOM developers reviewing the database and statistical analysis of the avionics CERs. One result of this exercise was to drop very old avionics data points as unrepresentative of modern avionics. In addition to the CER adjustment, the avionics Mass Estimating Relationships (MERs) used in the INTROS LV sizing program were revised. Previous MERs were derived from STS data, Centaur stage data, Shuttle C, Heavy-Lift Launch Vehicle (HLLV), and other studies, leading to a much heavier weight input into NAFCOM than would be expected with modern electronics. In recent years avionics have changed considerably due to such things as electronics miniaturization and function integration. State-of-the-art avionics masses are considerably less than what was previously used in INTROS. Revised MERs were provided by MSFC's Avionics Department for GN&C, Actuator Control, Radio Frequency (RF) Communications, Instrumentation, Data Management/Handling, and Range Safety. The revised MERs were used within NAFCOM as one input into the multivariate CERs.

The core booster does not guide and control the ascent. This function is in the upper stage. Core booster avionics includes translators, controllers, Analog-to-Digitial (AD) converters, actuator control, electronics, and sufficient CCDH hardware to interface with the upper stage. The upper stage avionics controls ascent, separations, and flight. Upper stage avionics hardware includes the Inertial Measuring Unit (IMU), processors, communication, telemetry, and instrumentation. Software provides the separation commands, and the software for general flight, mission specific flight algorithms, and launch-date-specific software.

Software provides the commands that control the vehicle, viewed as one entity for the LV. As such, the software estimate is not divided between the core and upper stage. Software is normally located on the upper stage since it is the upper stage that controls the ascent of the LV. The software estimate for the LVs is based on a detailed breakdown of the functional requirements.

#### Table 6C-8. Functional Breakout of Lines of Codes Estimates

#### Events Manager (50 Hz) (approximately 500 to 1,000 SLOC

Manage Events Sequencer

Manage Events Updates

#### Navigation Manager (50 Hz) (approximately 8,000 to 15,000 SLOC)

Provide Translational Navigation Estimates

Provide Rotational Navigation Estimates

#### Guidance Manager (1 Hz) (approximately 15,000 to 25,000 SLOC)

## Ascent Mode

Provide Open-Loop Guidance

Provide Closed-Loop Guidance

Provide Circularization Guidance

#### **Abort Mode**

Provide Ascent Abort (IIP) (50 Hz) (flight planning for avoiding undesirable landing areas using reduced capability)

Note: This could contain added capability; currently no defined requirements.

#### Control Manager (50 Hz) (approximately 8,000 to 15,000 SLOC)

Manage Stage Separation Control

Manage Ascent Vehicle Control

Manage RCS Control

#### Command and Data Manager (50 Hz) (approximately 28,000 to 40,000 SLOC)

Initialize Software

Initialize Hardware

Provide Payload Interface

Provide Sensor Interface (GPS, INS, Gyro)

Provide Telemetry Data

Provide Ground Interface

Provide Engine Controller Interface

Provide Upper Stage Controller Interface

Provide Booster Interface Unit Interface

Provide TVC Controller Interface

Provide Flight Termination System Interface

*Note*: This assumes a limited fault detection and notification/recovery capability.

#### Time Manager (50 Hz) (approximately 1,500 to 2,000 SLOC)

Provide Time

#### Power Manager (25 Hz) (approximately 2,500 to 4,000 SLOC)

Provide Power System Management

#### Vehicle Management Software (110K SLOC $\pm$ 50%)

Abort Management System (70K SLOC ± 50%)

Trajectory Replan Requests (10K SLOC)

- · Engine Operation
- · Stage Separation

Status Payload (10K SLOC)

- · Abort Conditions
- · Health Indications

Determination of Proper Scenario (50K SLOC)

· Burn Remaining Engines Longer

· Separate Upper Stage Early

Launch Pad Interface (15K SLOC  $\pm$  50%)

**Data Gathering** 

Communication with Launch Pad—ability to diagnose health of engine

Fault Identification on Vehicle

Onboard FTS Tracking  $(25K \text{ SLOC} \pm 50\%)$ 

Trajectory Following

**RT Position Monitoring** 

Compare Position Monitoring

Abort Scenario Updates

- · Trajectory Modifications
- · Flight Termination Delay

Communication with Range Safety to Request Flight Termination

Total Flight Software SLOC estimate: 48,500 to 102,000 Vehicle Management included: 55,000 to 165,000 Total: 103,000 to 267,000

Software estimates are based on the above maximum lines of code, using the SEER-SEM tool for software estimation, planning, and project control. SEER-SEM is a recognized software estimation tool developed by Galorath Incorporated for use in industry and government.

#### **Shuttle-Derived Avionics Hardware**

The GN&C and CCDH subsystems for Shuttle-derived LVs are considered new designs. Because the subsystems and software are new, integrated health management and human-rating requirements are incorporated from the start. The avionics hardware assumed a new design with existing technology.

#### **Shuttle-Derived Software**

All Shuttle-derived software is considered a new software development, incorporating the functions identified above. The maximum SLOC estimate was used with the SEER-SEM model to arrive at a deterministic software estimate.

#### **Other Subsystems**

The basic thermal systems are ½ to 1 inch SOFI, with cold plates and insulation for passive cooling of equipment and avionics. No new technology is planned. Heritage has normally been given to the thermal subsystem because it is well understood and used on existing systems today.

Electrical power is provided by silver-zinc batteries with a redundancy of two. Conversion, distribution, and circuitry are considered new designs with state-of-the-art technology. Hydraulic power is hydrazine fueled, used in LVs today.

Reaction Control Systems (RCSs), when used, are the same type as used in the Shuttle today. Range safety will require modifications to the Flight Termination System (FTS) to add a time delay for abort. Human-rating requirements may require the removal of the autodestruct capability. All of these subsystems are similar to those already in existence, either on EELVs or Shuttle, and have been validated in the relevant environment. Full qualification and testing is estimated for all crew and cargo vehicles.

#### 6C.5.1.10.2 DDT&E

The lowest cost option uses the existing four-segment RSRB and the modified SSME. Of the two five-segment configurations, the vehicle that uses only one J–2S+ engine is cheaper than the vehicle that requires four LR–85s.

#### **6C.5.1.10.3** Production

LV 13.1, LV 15, and LV 16 are single SRB-based crew vehicles, with either a four- or five-segment booster modified from the current Shuttle SRBs. As described above, the modifications will enable the integration of the booster with an upper stage. The recurring production costs of these three concepts are very close and within the accuracy of the model. Although the four-segment SRM is slightly cheaper to refurbish than the five-segment version (the cost of refurbishing and reloading a single motor segment is relatively small), the cost of the Expendable Space Shuttle Main Engine (eSSME) equipped upper stage more than offsets this savings, so that LV 13.1 has the highest recurring production cost. LV 16, with the single J–2S+ upper stage has the lowest production cost, while LV 15 with four LR–85 engines is slightly higher.

#### 6C.5.1.10.4 Launch Operations

All of these concepts require the stacking of either a four- or five-segment SRB with modified forward skirt and the interface to the interstage. The SRM segments are refurbished in the same manner as in the current Shuttle operation (described previously in the Production section). A portion of the interstage is also a refurbished item. The upper stage, upper stage engine, and part of the interstage are newly manufactured hardware. The launch operations activities include receipt, checkout, stacking and integration, testing, transport to the launch pad, pad operations, and launch. The cost of launch operations is lowest for LV 16 and greatest for LV 13.1. However, the difference at six flights per year is slight.

#### **6C.5.1.10.5** Facilities

The facilities costs include modifications to Mobile Launch Platform (MLP), Vehicle Assembly Building (VAB), and launch pad to accommodate the different profile and footprint of the in-line SRB configuration. The cost is the same for all three concepts, as shown in **Table 6C-9**.

Table 6C-9. Relative	Comparison of SDV	Crew Vehicle Costs

Phase	Relative Cost Position			
Vehicle	13.1	13.1 15		
DDT&E	1.00	1.38	1.32	
Production	1.00	0.92	0.93	
Operations	1.00	1.03	0.85	
Facilities	1.00	1.00	1.00	

6C.5.2 Cargo Launch Vehicles

6C.5.2.1 Launch Vehicle 20 (LV 20)

6C.5.2.1.1 Vehicle Description

LV 20 (**Figure 6C-49**) is a 1.5-stage parallel burn LV for cargo. This sidemount Shuttle-derived LV concept (Carrier B) was developed at MSFC. The general configuration is two solid strap-on boosters connected to an ET with a payload carrier mounted on the side. The propulsion module for this concept is attached to the back end of the payload carrier and contains three SSMEs. The two solid strap-on boosters are four-segment RSRBs (Polybutadiene Acrylonitride (PBAN) propellant). A portion of the payload carrier is jettisoned during ascent according to the FMHR ground rule for shroud jettison conditions. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 71.0 nmi. The SSMEs were run at a throttle setting of 104.5 percent after liftoff.

The net payload capability of LV 20 is 66.7 mT to a  $30 \times 160 \text{ nmi}$  orbit at a 28.5 deg inclination. The net payload to  $30 \times 160 \text{ nmi}$  at a 51.6 deg inclination is 62.2 mT. No ground rules or constraints were violated for this LV analysis.

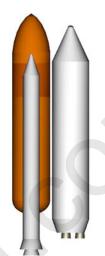


Figure 6C-49. LV 20 General Configuration

Special considerations required to analyze this vehicle were:

- Maximum allowed acceleration was 3.00 g's, based on Shuttle ET structural limits,
- The vehicle was flown inverted to more closely mimic the Shuttle trajectory,
- Liftoff with SSMEs at 100% power level,
- Throttle-up to 104.5% at 60 fps relative velocity,
- Gravity turn was maintained until 1 sec after RSRB jettison, and
- Used average Shuttle SSME cant angles.

#### 6C.5.2.1.2 Vehicle Sizing

The mass properties for LV 20 are shown in **Table 6C-10**. These properties were supplied by the MSFC Engineering Directorate and used as delivered.

Table 6C-10. LV 20 INTROS Mass Summary

MASS PROPERTIES ACCOUNTING					
VEHICLE: Side-mounted Shuttle-derived Four-segment RSRB					
	STAGE: ET/F	Payload Carrier B			
		MASS SUBTOTA	ALS	MASS	
ITEM	Tertiary	Secondary	Primary	TOTALS	
	lbm	lbm	lbm	lbm	
External Tank					
Primary Structures			50,062		
Secondary Structures			1,502		
Separation Systems			800		
MPS			2,167		
TPS/TCS			3,128		
Electrical Systems			303		

Delta from STS-117TDDP	1,263	
EXTERNAL TANK DRY MASS (mdry_ET)	)	59,226
Residual ET Propellant	895	
Pressurization Gases	3,809	
Reserves	4,367	
EXTERNAL TANK MASS AT MECO (mjet	tt_ET)	68,297
Payload Carrier		
Primary Structures	47,541	
Secondary Structures	13,005	
Separation Systems	3,510	
TPS	4,868	
TCS	1,539	
MPS	32,302	
Auxiliary Propulsion Systems	467	
Power (Hydraulic)	873	
Power (Electrical)	850	
Avionics (No Breakout Available)	922	
Cabling (Electrical Power/Avionics)	2,591	
Miscellaneous	670	
PAYLOAD CARRIER DRY MASS WITH O	GROWTH (mdry_orb)	109,138
Payload Carrier Main Propulsion Residual	2,139	
Subsystem Residuals	4	
Payload Carrier Reserve Fluids	842	
RCS On-Orbit Propellant	4,142	
Less Jettisoned Payload Carrier Shroud	-29,295	
PAYLOAD CARRIER BURNOUT MASS (1	mbo, mbo orb)	86,969
Main Ascent Propellant	1,588,636	
In-flight Fluid Losses	89	
Engine Purge Helium	65	
Shutdown Propellant	2,310	
Payload Carrier Shroud	29,295	
ET/PAYLOAD CARRIER GROSS LIFTOF	F MASS (mgross)	1,775,661
Payload	172,968	
Strap-on Boosters, Gross Liftoff	2,595,763	
VEHICLE GROSS LIFTOFF MASS (mgros	ss veh)	4,544,392

6C.5.2.1.3 Structural Analysis

The structural analysis for this condition of the structural analysis for the structural analysis for the structural analysis for the structural analysis for the structural analysis for the structural analysis for the structural analysis for the structural analysis for the structural analysis for the structural analysis. The structural analysis for this concept was performed by the MSFC Engineering Directorate and was used as delivered. There was no LV analysis run for this vehicle.

# 6C.5.2.1.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-50**. Selected trajectory parameters are shown graphically in **Figures 6C-51** through **6C-54**. The vehicle exhibits a 1.52 T/W ratio at liftoff. Maximum dynamic pressure is 719 psf at 53.0 sec into the flight. The maximum acceleration is 2.42 g's during the RSRB burn and 3.00 g's during the first-stage burn after RSRB separation. RSRB jettison occurs at 125.4 sec into the flight at an altitude of 184,910 ft and Mach 4.2. The T/W ratio after Reusable Solid Rocket Motor (RSRM) jettison is 0.97. Orbital injection occurs at 499.0 sec at 71.0 nmi.

#### 4 Segment SRM Side Mount SDV Vehicle Concept Characteristics GLOW 4,544,392 lbf 29 295 lbm Booster Stage (each) Useable Propellant 1.111.019 lbm BumoutMass 186,863 lbm #Boosters/Type 2 / 4 Segment SRM BoosterThrust (@ 0.7 secs ) 3,139,106 lbf @ 0.7 secs Tankage 268.8 s Propellants Useable Propellant LOX/LH2 1.590.946 lbm Propellant Officed Stage pm Dry Mass Burnout Mass 0.0 % 0.8913 Carrier Dry Mass 109,138 lbm Bumout Mass # Engines / Type agine Thrust (100%) Engine Isp (100%) Mission PowerLevel 86,969 lbm 3/SSMEBLKII 375,432 lbf @ SL 469,710 lbf @ Vac 365.3 s@ SL 4522 s@ Vac Delivery Orbit ery Orbit Payload Net Payload 30 x160 rmi @ 28.5 ° 172,968 lbm 78.5 MT 147,023 lbm 66.7 MT Payload Indudes Cargo 71.0 nmi ertion Altitude T/W @ Liftoff Max Dynamic Pressure 719 psf Maxg 's Ascent Burn T/W Second Stage

#### Closed Case Summary Data for Reference Mission (30 x 160 nmi @ 28.5 °): Liftoff to SRM staging dynp@stg = 9.80 psf dv1 = 8,427 ft/smax RSRM accel = 2.42 max core f/w = 3.00time of max Q = 53.0 sec max Q = 719 psf Shroud Jettison @t = 237 sec throttle @ bucket = no change alt @ jettison = 357,980 ft After RSRM jettison time to MECO = 499 sec (ET+Carrier) dvt = 30,252 ft/ststg = 125.42 sec alt@stg = 184,910 ft mach@stg = 4.20

Figure 6C-50. LV 20 Summary

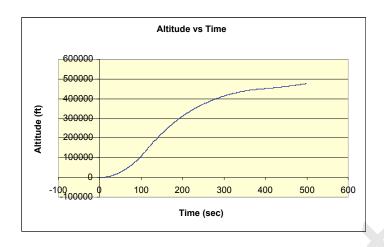


Figure 6C-51. Altitude versus Time

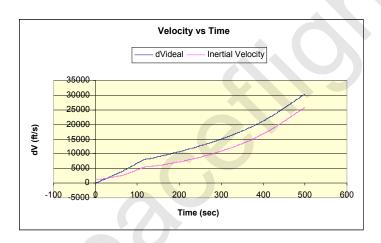


Figure 6C-52. Velocity versus Time

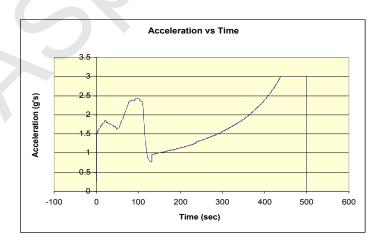


Figure 6C-53. Acceleration versus Time

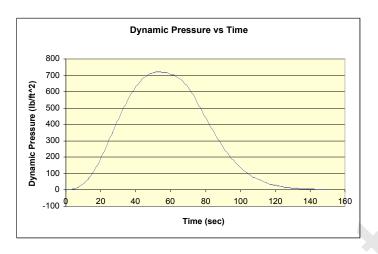


Figure 6C-54. Dynamic Pressure versus Time

#### 6C.5.2.2 Launch Vehicle 21 (LV 21)

#### 6C.5.2.2.1 Vehicle Description

LV 21 (**Figure 6C-55**) is a 1.5-stage parallel burn LV for cargo. This sidemount Shuttle-derived LV concept (Carrier B) was developed at MSFC. The general configuration is two solid strap-on boosters connected to an ET with a payload carrier mounted on the side. The propulsion module for this concept is attached to the back end of the payload carrier and contains three SSMEs. The two solid strap-on boosters are five-segment RSRBs (HTPB propellant). A portion of the payload carrier is jettisoned during ascent according to the FMHR ground rule for shroud jettison conditions. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 72.8 nmi. The SSMEs were run at a throttle setting of 104.5 percent after liftoff.

The net payload capability of LV 21 is 79.5 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 74.4 mT.

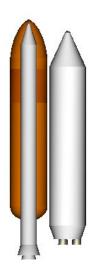


Figure 6C-55. LV 21 General Configuration

In final data review of this concept, it was determined that the useable propellant load in the ET of this concept was 1,413 lb less than the useable propellant load of LV 20. Because of time constraints this case was not reanalyzed, however, this was not considered significant.

Special considerations required to analyze this vehicle were:

- Max allowed acceleration was 3.00 g's, based on Shuttle ET structural limits,
- The vehicle was flown inverted to more closely mimic the Shuttle trajectory,
- Liftoff with SSMEs at 100 percent power level,
- Throttle-up to 104.5 percent at 60 fps relative velocity,
- Gravity turn was maintained until 1 sec after RSRB jettison, and

• Used average Shuttle SSME cant angles.

## 6C.5.2.2.2 Vehicle Sizing

The mass properties for LV 21 are shown in **Table 6C-11**. These properties were supplied by the MSFC Engineering Directorate and used as delivered.

Table 6C-11. LV 21 INTROS Mass Summary

MAS	SS PROPERT	TIES ACCOUNTII	NG	
		uttle-Derived Five Payload Carrier B	-Segment SRB	
	MASS SUBTOTALS			
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
External Tank				
Primary Structures			50,063	
Secondary Structures			1,502	
Separation Systems			1,020	
MPS			2,167	
TPS/TCS			3,128	
Electrical Systems			303	
Delta from STS-117TDDP			1,263	
EXTERNAL TANK DRY MASS (n	ndry_ET)			59,447
Residual ET Propellant			895	
Pressurization Gases			3,809	
Reserves			4,367	
EXTERNAL TANK MASS AT ME	CO (mjett_E	Τ)		68,518
Payload Carrier				
Primary Structures			47,541	
Secondary Structures			13,005	
Separation Systems			3,510	
TPS			4,868	
TCS			1,539	
MPS			32,302	
Auxiliary Propulsion Systems			467	
Power (Hydraulic)			873	
Power (Electrical)			850	
Avionics (No Breakout Available)			922	
Cabling (Electrical Power/Avionics)			2,591	
Miscellaneous			670	
PAYLOAD CARRIER DRY MASS	WITH GRO	WTH (mdry_orb)		109,138
Payload Carrier Main Propulsion Residual			2,139	
Subsystem Residuals			4	
Payload Carrier Reserve Fluids			842	

RCS On-Orbit Propellant		4,142	
Less Jettisoned Payload Carrier Shroud		-29,295	
PAYLOAD CARRIER BURNO	UT MASS (mbo, mbo_orb)		86,969
Main Ascent Propellant		1,588,636	
In-flight Fluid Losses		89	
Engine Purge Helium		65	
Shutdown Propellant		2,310	
Payload Carrier Shroud		29,295	
ET/PAYLOAD CARRIER GROSS LIFTOFF MASS (mgross)			1,775,882
Payload		206,146	
Strap-on Boosters, Gross Liftoff		3,312,279	
VEHICLE GROSS LIFTOFF M	IASS (mgross_veh)		5,294,308

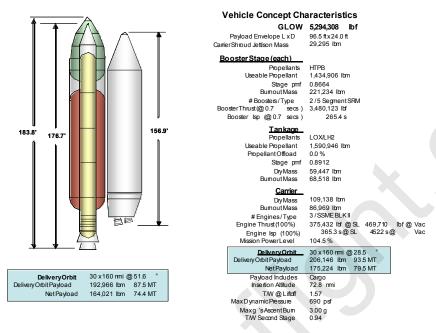
#### 6C.5.2.2.3 Structural Analysis

The structural analysis for this concept was performed by the MSFC Engineering Directorate and was used as delivered. There was no LV analysis run for this vehicle.

#### 6C.5.2.2.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-56**. Selected trajectory parameters are shown graphically in **Figures 6C-57** through **6C-60**. The vehicle exhibits a 1.57 T/W ratio at liftoff. Maximum dynamic pressure is 690 psf at 60.3 sec into the flight. The maximum acceleration is 2.74 g's during the RSRB burn and 3.00 g's during the first-stage burn after RSRB separation. RSRB jettison occurs at 132.5 sec into the flight at an altitude of 183,757 ft and Mach 5.0. The T/W ratio after RSRB jettison is 0.94. Orbital injection occurs at 494.2 sec at 72.8 nmi.

## **5 Segment SRM Side Mount SDV**



#### Closed Case Summary Data for Reference Mission (30 x 160 nmi @ 28.5 °):

Liftoff to SRM staging max RSRM accel = 2.74

time of max Q = 60.3 sec max Q = 690 psf throttle @ bucket = no change

After RSRM jettison (ET+Carrier) tstg = 132.52 sec att@stg = 183,757 ft mach@stg = 5.02 dynp@stg = 14.7 psf dv1 = 9,118 ft/s max core f/w = 3.00

Shroud Jettison @t = 233 sec alt @ jettison = 364,678 ft

time to MECO = 494 secdvt = 29,860 ft/s

Figure 6C-56. LV 21 Summary

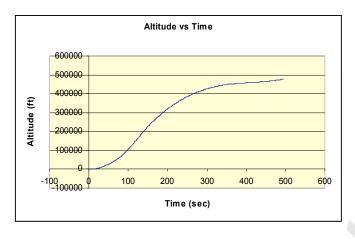


Figure 6C-57. Altitude versus Time

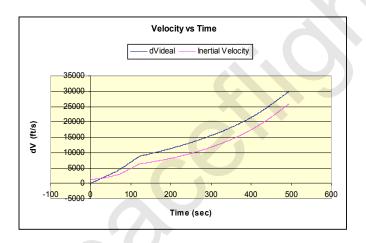


Figure 6C-58. Velocity versus Time

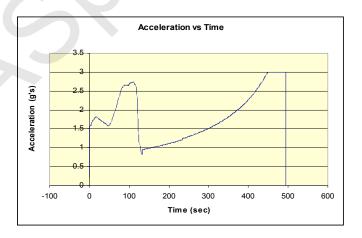


Figure 6C-59. Acceleration versus Time

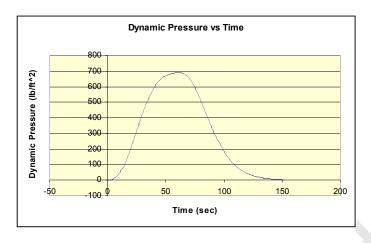


Figure 6C-60. Dynamic Pressure versus Time

#### 6C.5.2.3 Launch Vehicle 24 (LV 24)

## 6C.5.2.3.1 Vehicle Description

LV 24 (**Figure 6C-61**) is a 1.5-stage parallel burn LV for crew and cargo based on an inline Shuttle-derived design and uses ET diameter tanks and structure in the core. This LV concept is the same as LV 25, except it carries a CEV above the payload shroud. The general configuration, shown in **Figure 6C-62**, is two solid strap-on boosters connected to a LOX/LH2 core stage with the payload contained in a shroud above the core stage, with a CEV attached above the shroud. The two solid strap-on boosters are four-segment RSRBs (PBAN propellant). The LOX/LH2 core stage utilizes three SSMEs for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 62.0 nmi. The SSMEs were run at a throttle setting of 104.5 percent. The results of the structural

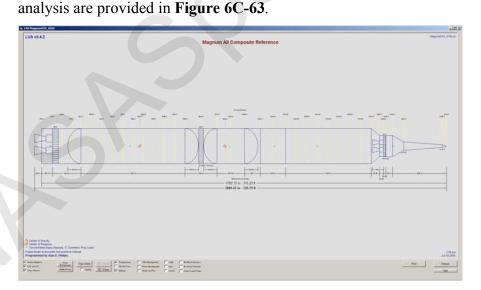


Figure 6C-61. LV 24 General Configuration

Figure 6C-62. LVA Structural Configuration

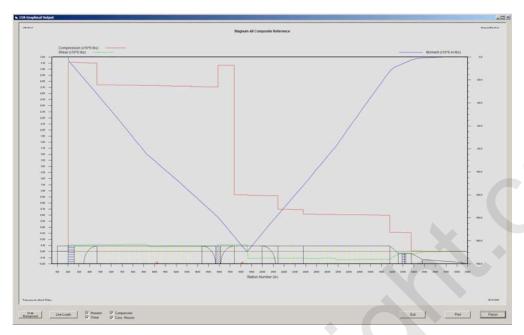


Figure 6C-63. LVA Structural Loads Analysis Results

The net payload capability of LV 24 is 73.9 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. This net payload would constitute the CEV mass, SM mass, and payload mass contained in the cylindrical shroud. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 69.3 mT.

For this concept, the LES was jettisoned 47 sec after liftoff. This is a deviation from the nominal ground rule of LES jettison at 30 sec after RSRB separation for this type of vehicle configuration. Performance analysis at nominal LES jettison conditions was later checked and determined that this resulted in a 0.3 mT payload increase and is not considered significant for this concept.

For this concept, the RSRBs were jettisoned 2.4 sec early. This error was due to a slightly smaller propellant mass than was standard. Performance analysis with the correct propellant loadings was later performed and determined that this resulted in a 0.2 mT payload decrease and is not considered significant for this concept.

No special considerations were required to analyze this vehicle.

#### 6C.5.2.3.2 Vehicle Sizing

The mass properties for the core stage of LV 24 are shown in **Table 6C-12**. The mass properties for the four-segment RSRB were supplied by the Solid and Hybrid Propulsion System Branch of the MSFC Engineering Directorate and used as delivered. The shroud cylindrical shell and the CEV adapter were both included in the core mass properties accounting.

Table 6C-12. LV 24 INTROS Mass Summary

	MASS PROPERT	TIES ACCOUNTI	NG	
	_		rew + Cargo – Blk	II
STAG	E: Strap-on Solid			
	N	IASS SUBTOTAI		MASS
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
STAGE BURNOUT MASS (m	bo)			186,863
Main Ascent Propellant			1,111,019	
STAGE GROSS LIFTOFF MA	ASS (mgross)			1,297,882
	STA	GE: Core		
Primary Body Structures			88,249	
Secondary Structures			3,409	
Separation Systems			1,818	
TPS			427	
TCS			3,574	
MPS			34,810	
Power (Electrical)			2,492	
Power (Hydraulic)			1,082	
Avionics			713	
Miscellaneous			381	
STAGE DRY MASS WITHOU	JT GROWTH			136,955
Dry Mass Growth Allowance			12,313	
STAGE DRY MASS WITH G	ROWTH (mdry)			149,268
Residuals			6,870	
Reserves			4,391	
In-flight Fluid Losses			157	
STAGE BURNOUT MASS (m	bo)			160,686
Main Ascent Propellant			1,588,636	
Shutdown Propellant			2,310	
Engine Purge Helium			65	
STAGE GROSS LIFTOFF MA	ASS (mgross)			1,751,697
Payload			181,034	
LES			9,300	
Strap-on(s), Gross Liftoff			2,595,763	
VEHICLE GROSS LIFTOFF	MASS (mgross ve	eh)		4,537,794

## 6C.5.2.3.3 Structural Analysis

The loads plot is a combined worst case including liftoff, max q, and max g. The tie-down loads are assumed to be carried by the SRBs, as they do with the STS. The compression loads show a major jump where the LOX tank loads are integrated into the outside structure with a quick reduction of the loads where the introduced SRB loads counteract the compression. The bending moment shows a steady increase from the tip of

the vehicle to the liftoff Center of Gravity (CG), then a steady decrease back to zero, as expected from an in-flight case.

## 6C.5.2.3.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-64**. Selected trajectory parameters are shown graphically in **Figures 6C-65** through **6C-68**. The vehicle exhibits a 1.54 T/W ratio at liftoff. Maximum dynamic pressure is 718 psf at 58.0 sec into the flight. The maximum acceleration is 2.45 g's during the RSRB burn and 4.00 g's during the first-stage burn after RSRB separation. RSRB jettison occurs at 123 sec into the flight at an altitude of 151,458 ft and Mach 4.3. The T/W ratio of the core stage after RSRB separation is 0.96. Orbital injection occurs at 488 sec at 62 nmi.

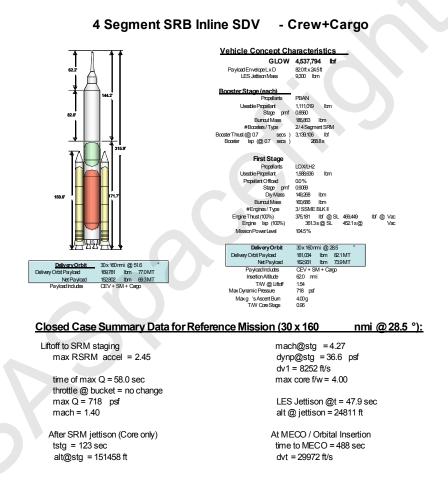


Figure 6C-64. LV 24 Summary

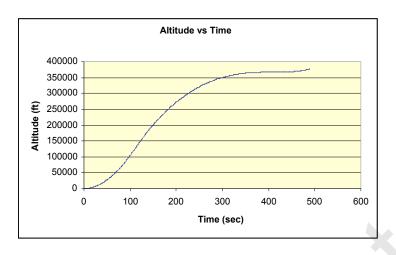


Figure 6C-65. Altitude versus Time

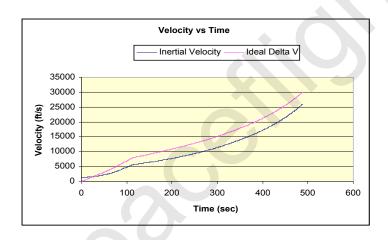


Figure 6C-66. Velocity versus Time

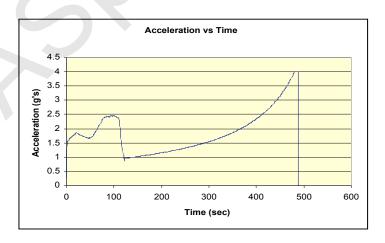


Figure 6C-67. Acceleration versus Time

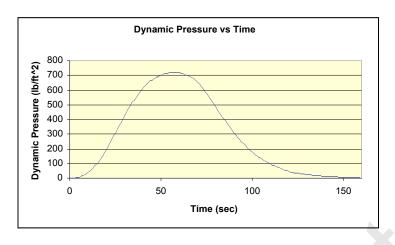


Figure 6C-68. Dynamic Pressure versus Time

#### 6C.5.2.4 Launch Vehicle 25 (LV 25)

#### 6C.5.2.4.1 Vehicle Description

LV 25 (**Figure 6C-69**) is a 1.5-stage parallel burn LV for cargo and uses ET diameter tanks and structure in the core. The general configuration, shown in **Figure 6C-70**, is two solid strap-on boosters connected to a LOX/LH2 core stage with the payload contained in a shroud above the core stage. The two solid strap-on boosters are four-segment RSRBs (PBAN propellant). The LOX/LH2 core stage utilizes three SSMEs for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 78.5 nmi. The SSMEs were run at a throttle setting of 104.5 percent. Loads analysis results for LV 25 are provided in **Figure 6C-71**.



Figure 6C-69. LV 25 General Configuration

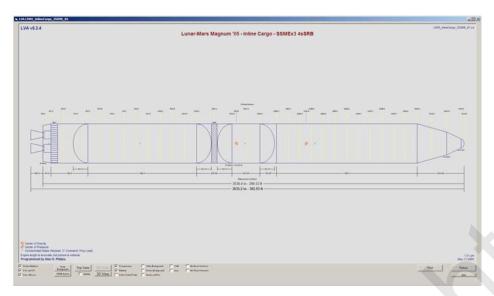


Figure 6C-70. LVA Structural Configuration



Figure 6C-71. LVA Structural Loads Analysis Results

The net payload capability of LV 25 is 73.5 mT to a 30 x 160 nmiorbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 69.0 mT. No ground rules or constraints were violated for this LV analysis.

No special considerations were required to analyze this vehicle.

This concept was also flown with EDS to determine the lunar payload capability for this vehicle. Four different EDS scenarios were analyzed: (1) No suborbital burn EDS only (no payload attached at launch), (2) No suborbital burn of EDS with payload attached, (3) Suborbital burn of the EDS only (no payload attached at launch), and (4) Suborbital burn of the EDS with payload attached. Payloads to three different lunar orbits were calculated

for each scenario: Trans-Lunar Injection (TLI), TLI + Lunar Orbit Insertion (LOI), and TLI + LOI + Plane Change.

# 6C.5.2.4.2 Vehicle Sizing

The mass properties for the core stage of LV 25 are shown in **Table 6C-13**. The mass properties for the four-segment RSRB were supplied by the Solid and Hybrid Propulsion System Branch of the MSFC Engineering Directorate and used as delivered.

Table 6C-13. LV 25 INTROS Mass Summary

	MASS PROPER	TIES ACCOUNTI	NG	
	In-line 3-SSME & E: Strap-on Solid	_	_	U
	N	IASS SUBTOTAL	LS .	MASS
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
STAGE BURNOUT MASS (m	ibo)			186,854
Main Ascent Propellant			1,111,028	
STAGE GROSS LIFTOFF M	ASS (mgross)			1,297,882
	STAGE: Cor	e Stage (3 SSME)		
Primary Body Structures			74,173	
Secondary Structures			3,409	
Separation Systems			1,838	
TPS			427	
TCS			3,542	
MPS			34,810	
Power (Electrical)			2,492	
Power (Hydraulic)		>	1,082	
Avionics			510	
Miscellaneous			381	
STAGE DRY MASS WITHO	UT GROWTH			122,663
Dry Mass Growth Allowance			10,151	
STAGE DRY MASS WITH G	ROWTH (mdry)			132,814
Residuals			6,870	
Reserves			4,391	
In-flight Fluid Losses			157	
STAGE BURNOUT MASS (m	ıbo)			144,232
Main Ascent Propellant			1,588,636	
Shutdown Propellant			2,310	
Engine Purge Helium			65	
STAGE GROSS LIFTOFF M	ASS (mgross)			1,735,243
Payload			190,743	
Payload Shroud			23,419	
Strap-on(s), Gross Liftoff			2,595,763	
VEHICLE GROSS LIFTOFF	MASS (mgross_ve	eh)		4,545,168

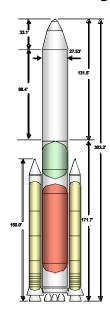
#### 6C.5.2.4.3 Structural Analysis

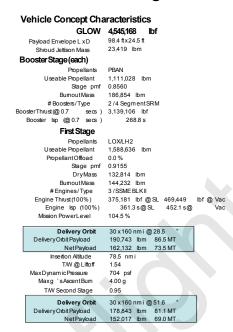
The loads plot is a combined worst case including liftoff, max q, and max g. The tie-down loads are assumed to be carried by the SRBs, as they do with the STS. The compression loads show a major jump where the LOX tank loads are integrated into the outside structure with a quick reduction of the loads where the introduced SRB loads counteract the compression. The bending moment shows a steady increase from the tip of the vehicle to the liftoff CG, then a steady decrease back to zero, as expected from an inflight case.

## 6C.5.2.4.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-72**. Selected trajectory parameters are shown graphically in **Figures 6C-73** through **6C-76**. The vehicle exhibits a 1.54 T/W ratio at liftoff. Maximum dynamic pressure is 704 psf at 55.4 sec into the flight. The maximum acceleration is 2.47 g's during the RSRB burn and 4.00 g's during the first-stage burn after RSRB separation. RSRB jettison occurs at 125.4 sec into the flight at an altitude of 167,999 ft and Mach 4.2. The T/W ratio of the core stage after RSRB separation is 0.95. Orbital injection occurs at 488.3 sec at 78.5 nmi. **Figures 6C-77** through **6C-80** provide the results of the analysis of the four separate EDS scenarios.

## 4 Segment SRB Inline SDV - Cargo





#### Closed Case Summary Data for Reference Mission (30 x 160

nmi @ 28.5 °):

Liftoff to SRM staging max RSRM accel = 2.47

> time of max Q = 55.40 sec max Q = 704 psf mach = 1.32

After SRM jettison (Core only) tstg = 125.42 sec alt@stg = 167,999 ft mach@stg = 4.21 dynp@stg = 19 psf dv1 = 8,277 ft/s max core f/w = 4.00

Shroud Jettison @t = 240.9 sec alt @ jettison = 361,469 ft

At MECO / Orbital Insertion time to MECO = 488.3 sec dvt = 30,082 ft/s

Figure 6C-72. LV 25 Summary

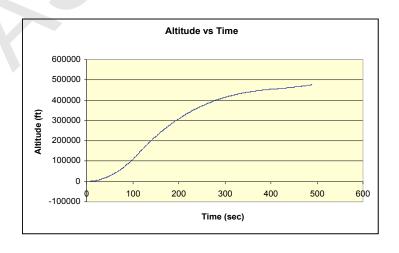


Figure 6C-73. Altitude versus Time

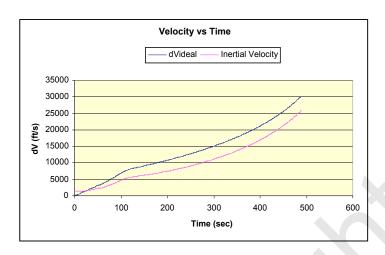


Figure 6C-74. Velocity versus Time

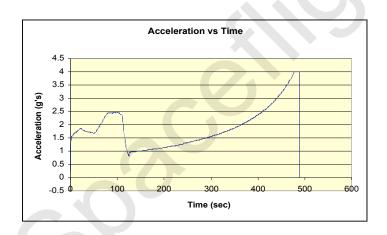
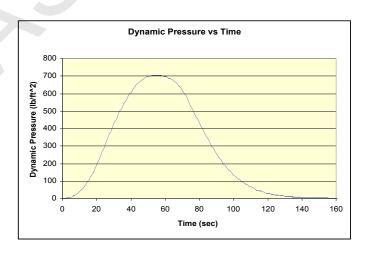


Figure 6C-75. Acceleration versus Time



#### Figure 6C-76. Dynamic Pressure versus Time

#### EDS Only / No Suborbital Burn Launch Vehicle 25 - SDV In-line 4-Seg RSRB/3-SSME Core Vehicle Concept Characteristics N<sub>1</sub>A EDS Gross @ Liftoff 203,159 lbf EDS Stage (TLI+LOI+PC) Propellants Useable Propellant LOX/LH2 173.990 lbm → 16.4' < Stage pmf 0.8564 Dry Mass 26,928 lbm Burnout Mass 29,150 lbm # Engines / Type 4 / LR-85 Engine Thrust (100%) 85,000 lbf @ Vac Engine Isp (100%) 450.0 s @ Vac 100.0% Mission Power Level TLI Delivery Gross Payload Net Payload 136,610 lbm 62.0 mT 122,949 lbm 55.8 mT TLI+LOI Delivery Gross Payload 85,890 lbm 39.0 mT Net Payload 77,301 lbm 35.1 mT TLI+LOI+PC Delivery Gross Payload 66,300 lbm 30.1 mT Net Payload 59,670 lbm 27.1 mT

Figure 6C-77. EDS with No Payload and No Suborbital Burn

#### EDS + Payload Attached / No Suborbital Burn Launch Vehicle 25 - SDV In-line 4-seg RSRB/3-SSME Core Vehicle Concept Characteristics N<sub>1</sub>B EDS Gross @ Liftoff 157,606 lbf EDS Stage (TLI+LOI+PC) LOX/LH2 Propellants 16.4' Useable Propellant 131,500 lbm 0.8344 Liftoff Stage pmf Dry Mass 24,337 lbm Burnout Mass 26,090 lbm # Engines / Type 4 / LR-85 Engine Thrust (100%) 85,000 lbf @ Vac Engine Isp (100%) Mission Power Level 450.0 s @ Vac 100.0% TLI Delivery Gross Payload 53.0 72,308 lbm 32.8 mT Net Payload 65,077 lbm 29.5 mT TLI+LOI Delivery Gross Payload 54,333 lbm 24.6 mT Net Payload 48,900 lbm 22.2 mT 54,333 lbm 24.6 mT TLI+LOI+PC Delivery Gross Payload 45,553 lbm 20.7 mT Net Payload 40,998 lbm

Figure 6C-78. EDS with Payload and No Suborbital Burn

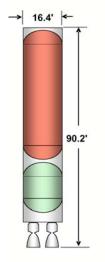
# **EDS Only with Suborbital Burn**

Launch Vehicle 25 - SDV In-line 4-seg RSRB/3-SSME Core

#### **Vehicle Concept Characteristics**

EDS Gross @ Liftoff 330,057 lbf





EDS Stage (TLI+LOI+PC)	
Propellants	LOX/LH2
Useable Propellant @ Liftoff	294,519 lbm
Useable Propellant @ 160 nmi cir.	204,859 lbm
Liftoff Stage pmf	0.8923
Dry Mass	32,012 lbm
Burnout Mass	35,505 lbm
# Engines / Type	4 / LR-70
Engine Thrust (100%)	70,000 lbf @ Vac
Engine Isp (100%)	450.0 s @ Vac
Mission Power Level	100.0%

TLI Delivery		
Gross Payload	163,800 lbm	74.3 mT
Net Payload	147,420 lbm	66.9 mT

TLI+LOI Delivery		
Gross Payload	102,820 lbm	46.6 mT
Not Payload	92 538 lbm	42 0 mT

TLI+LOI+PC Delivery		
Gross Payload	79,260 lbm	36.0 m
Not Payload	71 334 lbm	32 4 m

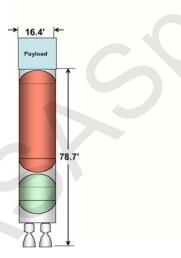
Figure 6C-79. EDS with Suborbital Burn

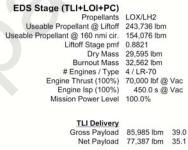
#### EDS + Payload attached with Suborbital Burn Launch Vehicle 25 - SDV In-line 4-seg RSRB/3-SSME Core

**Vehicle Concept Characteristics** 

EDS Gross @ Liftoff 276,326 lbf

S<sub>1</sub>B





Gross Payload	85,985 lbm	39.0 m
Net Payload	77,387 lbm	35.1 m

TLI+LOI Delivery		
Gross Payload	64,330 lbm	29.2 m
Net Payload	57,897 lbm	26.3 m

TLI+LOI+PC Delivery		
Gross Payload	53,731 lbm	24.4 m
Net Payload	48,358 lbm	21.9 m

Figure 6C-80. EDS with Payload and Suborbital Burn

Launch Vehicle 26 (LV 26) 6C.5.2.5

6C.5.2.5.1 Vehicle Description



LV 26 (**Figure 6C-81**) is a 1.5-stage parallel burn LV for crew and cargo based on an inline Shuttle-derived design and uses ET diameter tanks and structure in the core. The structural configuration is shown in **Figure 6C-82**. This LV concept is the same as LV 27, except it carries a CEV above the payload shroud. The general configuration is two solid strap-on boosters connected to a LOX/LH2 core stage with the payload contained in a shroud above the core stage with a CEV attached above the shroud. The two solid strap-on boosters are five-segment SRBs (HTPB propellant). The LOX/LH2 core stage utilizes four SSMEs for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 77.7 nmi. The SSMEs were run at a throttle setting of 104.5 percent. The results of the structural loads analysis are provided in **Figure 6C-83**.

The net payload capability of LV 26 is 91.3 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. This net payload would constitute the CEV mass, SM mass, and payload mass contained in the cylindrical shroud. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 85.3 mT. No ground rules or constraints were violated for this LV analysis.

No special considerations were required to analyze this vehicle.



Figure 6C-82. LVA Structural Configuration



Figure 6C-83. LVA Structural Loads Analysis Results

#### 6C.5.2.5.2 Vehicle Sizing

The mass properties for the core stage of LV 26 are shown in **Table 6C-14**. The mass properties for the five-segment SRB were supplied by the Solid and Hybrid Propulsion System Branch of the MSFC Engineering Directorate and used as delivered. The shroud cylindrical shell and the CEV adapter were both included in the core mass properties accounting.

Table 6C-14. LV 26 INTROS Mass Summary

MASS PROPERTIES ACCOUNTING				
VEHICLE: In-line				k 2
STAGE:	-	Booster (Five-Seg	ŕ	
	N	MASS SUBTOTAL	LS	MASS
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
STAGE BURNOUT MASS (mbd	0)			221,234
Main Ascent Propellant			1,434,906	
STAGE GROSS LIFTOFF MAS	SS (mgross)			1,656,140
	STAGE: Cor	e Stage (4 SSME)		
Primary Body Structures			108,865	
Secondary Structures			3,943	
Separation Systems			2,152	
TPS			514	
TCS			3,999	
MPS			46,412	
Power (Electrical)			2,719	

Power (Hydraulic)		1,443	
Avionics		813	
Miscellaneous		472	
STAGE DRY MASS WITHOU	T GROWTH		171,332
Dry Mass Growth Allowance		14,900	
STAGE DRY MASS WITH GR	OWTH (mdry)		186,231
Residuals		16,643	
Reserves		12,477	
In-flight Fluid Losses		210	
STAGE BURNOUT MASS (ml	00)		215,560
Main Ascent Propellant		2,210,023	
Engine Purge Helium		251	
STAGE GROSS LIFTOFF MA	SS (mgross)		2,425,833
Payload		236,690	
Payload Shroud		9,300	
Strap-on(s), Gross Liftoff		3,312,279	
VEHICLE GROSS LIFTOFF	MASS (mgross_veh)		5,984,103

#### 6C.5.2.5.3 Structural Analysis

The loads plot is a combined worst case including liftoff, max q, and max g. The tie-down loads are assumed to be carried by the SRBs, as they do with the STS. The compression loads show a major jump where the LOX tank loads are integrated into the outside structure with a quick reduction of the loads where the introduced SRB loads counteract the compression. The bending moment shows a steady increase from the tip of the vehicle to the liftoff CG, then a steady decrease back to zero, as expected from an inflight case.

#### 6C.5.2.5.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-84**. Selected trajectory parameters are shown graphically in **Figures 6C-85** through **6C-88**. The vehicle exhibits a 1.47 T/W ratio at liftoff. Maximum dynamic pressure is 563 psf at 70.0 sec into the flight. The maximum acceleration is 2.39 g's during the SRB burn and 4.00 g's during the first-stage burn after SRB separation. SRB jettison occurs at 132.5 sec into the flight at an altitude of 165,206 ft and Mach 4.0. The T/W ratio of the core stage after SRB separation is 0.93. Orbital injection occurs at 509.1 sec at 77.7 nmi.

# 5-Segment SRB In -line SDV - Crew + Cargo

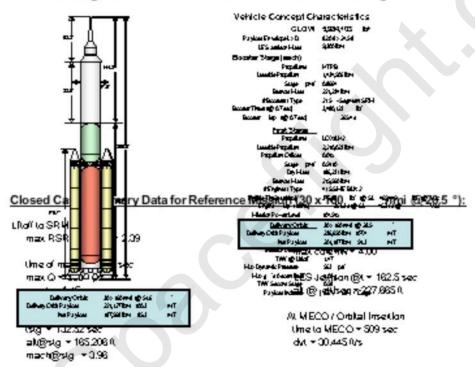


Figure 6C-84. LV 26 Summary

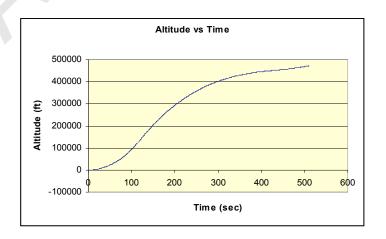


Figure 6C-85. Altitude versus Time

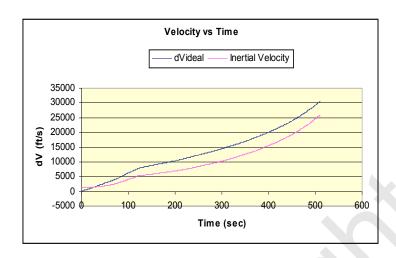


Figure 6C-86. Velocity versus Time

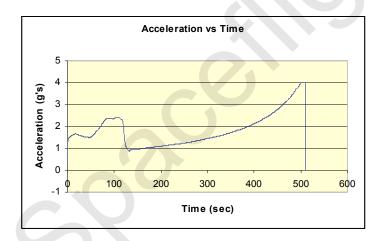
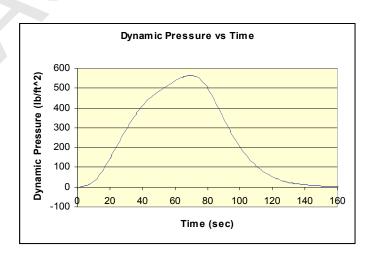


Figure 6C-87. Acceleration versus Time



#### Figure 6C-88. Dynamic Pressure versus Time

### 6C.5.2.6 Launch Vehicle 27 (LV27)

#### 6C.5.2.6.1 Vehicle Description

LV 27 (**Figure 6C-89**) is a 1.5-stage parallel burn LV for cargo. This is an in-line Shuttle-derived concept. The general configuration is two solid strap-on boosters connected to a LOX/LH2 core stage with the payload contained in a shroud above the core stage. The structural configuration of LV 27 is shown in **Figure 6C-90**, while the results of the loads analysis are provided in **Figure 6C-91**. The two solid strap-on boosters are five-segment SRBs (HTPB propellant). The LOX/LH2 core stage utilizes four SSMEs for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 77.7 nmi. The SSMEs were run at a throttle setting of 104.5 percent.

The net payload capability of LV 25 is 96.7 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 90.8 mT. No ground rules or constraints were violated for this LV analysis.

No special considerations were required to analyze this vehicle.

This concept was also flown with EDS to determine the lunar payload capability for this vehicle. Four different EDS scenarios were analyzed: (1)

No suborbital burn EDS only (no payload attached at launch), (2) No suborbital burn of EDS with payload attached, (3) Suborbital burn of the EDS only (no payload attached at launch), and (4) Suborbital burn of the EDS with payload attached. Payloads to three different lunar orbits were calculated for each scenario: TLI, TLI + LOI, and TLI + LOI + Plane Change.

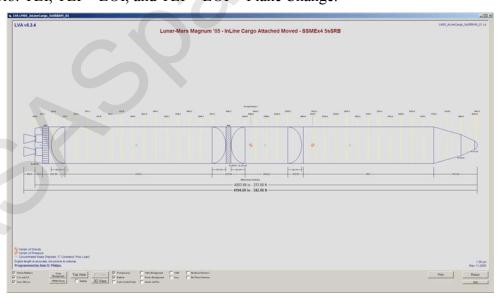


Figure 6C-90. LVA Structural Configuration



Figure 6C-89. LV 27 General Configuration

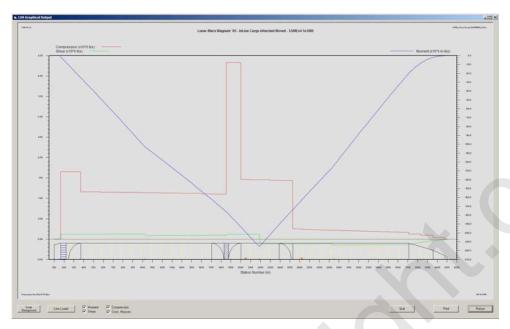


Figure 6C-91. LVA Structural Loads Analysis Results

# 6C.5.2.6.2 Vehicle Sizing

The mass properties for the core stage of LV 27 are shown in **Table 6C-15**. The mass properties for the five-segment SRB were supplied by the Solid and Hybrid Propulsion System Branch of the MSFC Engineering Directorate and used as delivered.

Table 6C-15. LV 27 INTROS Mass Summary

MASS PROPERTIES ACCOUNTING				
VEHICLE: Five-Segment SRB & 4-SSME Cargo – Blk 2 STAGE: Strap-on Solid Booster (Five-Segment SRB)				
	N.	IASS SUBTOTAI	LS	MASS
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
STAGE BURNOUT MASS (mb	0)			221,234
Main Ascent Propellant			1,434,906	
STAGE GROSS LIFTOFF MAS	SS (mgross)			1,656,140
	STAGE: Core	Stage (4-SSME)		
Primary Body Structures			93,061	
Secondary Structures			3,943	
Separation Systems			2,181	
TPS			514	
TCS			3,963	
MPS			46,412	

Power (Electrical)			2,719	
Power (Hydraulic)			1,443	
Avionics			590	
Miscellaneous			472	
STAGE DRY MASS WITHOU	T GROWTH			155,299
Dry Mass Growth Allowance			12,508	
STAGE DRY MASS WITH GR	ROWTH (mdry)			167,806
Residuals			16,643	
Reserves			12,387	
In-flight Fluid Losses			210	
STAGE BURNOUT MASS (ml	00)			197,046
Main Ascent Propellant			2,210,112	
Engine Purge Helium			251	
STAGE GROSS LIFTOFF MA	SS (mgross)			2,407,408
Payload			250,798	
Payload Shroud			23,404	
Strap-on(s), Gross Liftoff		<b>\langle</b>	3,312,279	
VEHICLE GROSS LIFTOFF	MASS (mgross_ve	eh)		5,993,890

#### 6C.5.2.6.3 Structural Analysis

The loads plot is a combined worst case including liftoff, max q, and max g. The tie-down loads are assumed to be carried by the SRBs, as they do with the STS. The compression loads show a major jump where the LOX tank loads are integrated into the outside structure with a quick reduction of the loads where the introduced SRB loads counteract the compression. The bending moment shows a steady increase from the tip of the vehicle to the liftoff CG, then a steady decrease back to zero, as expected from an inflight case.

#### 6C.5.2.6.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-92**. Selected trajectory parameters are shown graphically in **Figures 6C-93** through **6C-96**. The vehicle exhibits a 1.46 T/W ratio at liftoff. Maximum dynamic pressure is 562 psf at 69.5 sec into the flight. The maximum acceleration is 2.38 g's during the SRB burn and 4.00 g's during the first-stage burn after SRB separation. SRB jettison occurs at 132.5 sec into the flight at an altitude of 164,634 ft and Mach 3.9. The T/W ratio of the core stage after SRB separation is 0.93. Orbital injection occurs at 509.5 sec at 77.7 nmi. **Figures 6C-97** through **6C-100** provide the results of the analysis of the four separate EDS scenarios.

# Vehicle Concept Characteristics 2.07 422229 be Palantinaper 10 2015 A24 See allow Him Palantinaper 10 2015 A24 See allow Him Section Theorem 10 2015 A24 See allow Him Section Theorem 10 2015 A24 See allow Him 2015 A24 See allow

ÄÄÄECO/Orblatinseklon UmetoMECO + 309.5 sec

du = 30,494 fbs

5-Segment SRB In -line SDV - Cargo

Figure 6C-92. LV 27 Summary

⊒l@গg + 164.6341\ mach@গg + 3.92

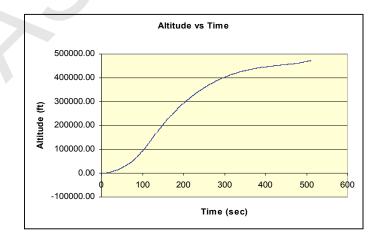


Figure 6C-93. Altitude versus Time

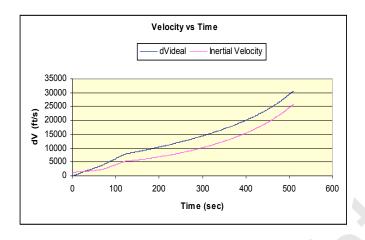


Figure 6C-94. Velocity versus Time

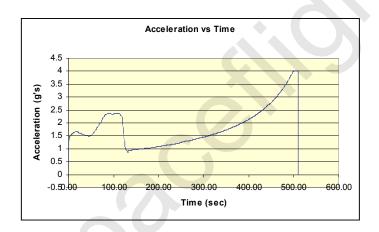


Figure 6C-95. Acceleration versus Time

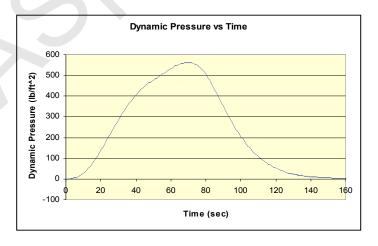


Figure 6C-96. Dynamic Pressure versus Time

#### EDS – No Suborbital Burn No Payload Attached Launch Vehicle 27 – SDV In-line 5-Seg RSRB/4-SSME Core

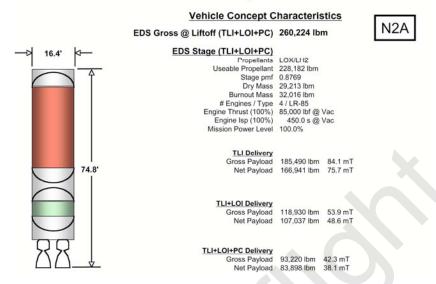


Figure 6C-97. EDS with No Payload and No Suborbital Burn

#### EDS – No Suborbital Burn with Payload Attached Launch Vehicle – SDV In-line 5-Seg RSRB/4-SSME Core

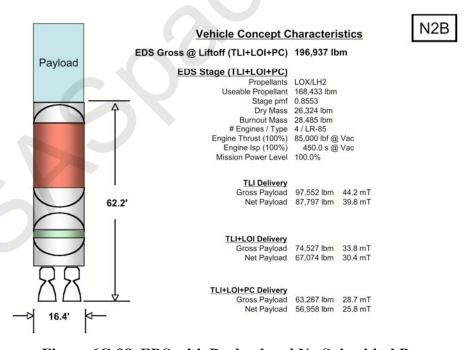


Figure 6C-98. EDS with Payload and No Suborbital Burn

#### EDS – Suborbital Burn No Payload Attached Launch Vehicle – SDV In-line 5-Seg RSRB/4-SSME Core

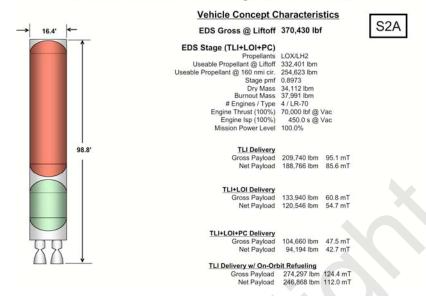


Figure 6C-99. EDS with No Payload and Suborbital Burn

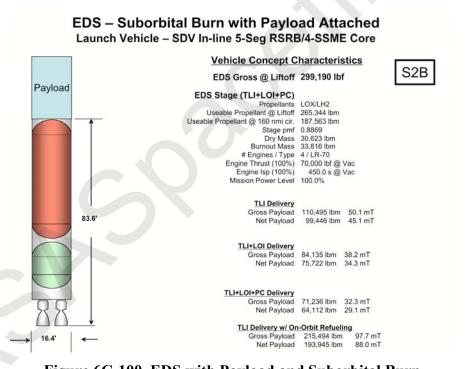


Figure 6C-100. EDS with Payload and Suborbital Burn

The Lunar Crew/Cargo Vehicle EELV-derived systems performance summary information is included in the following paragraphs. Included are summaries for the following vehicles: LV 7.4, LV 7.5, LV 11, and LV 11.1.

#### 6C.5.2.7 Launch Vehicle 7.4 (LV 7.4)



#### 6C.5.2.7.1 Vehicle Description

LV 7.4 (**Figure 6C-101**) is a two-and-one-half-stage LV for cargo. The first stage is an ET diameter, LOX/RP stage with five RD–180 engines. The LOX/LH2 second stage is also ET diameter with four J–2S+ engines for propulsion. The structural configuration is shown in **Figure 6C-102**. This vehicle concept utilizes two Atlas V strap-on liquid rocket boosters that are attached to the core stage. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 78.5 nmi. All liquid engines were operated at a 100 percent power level in the analysis. Complete results of the structural loads analysis are provided in **Figure 6C-103**.

The net payload capability of LV 7.4 is 95.1 mT to a  $30 \times 160 \text{ nmi}$  orbit at a 28.5 deg inclination. The net payload to  $30 \times 160 \text{ nmi}$  at a 51.6 deg inclination is 90.2 mT. No ground rules or constraints were violated for this LV analysis.

A special consideration was required to analyze this vehicle: 28 percent offloaded propellant on the Atlas V boosters.

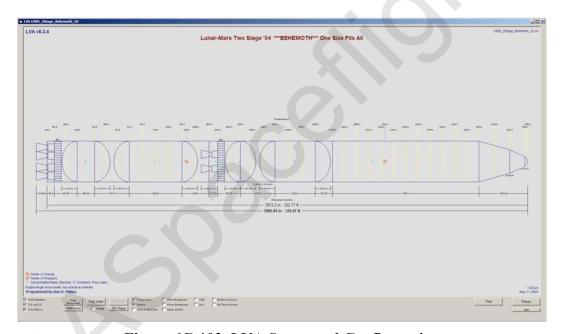


Figure 6C-102. LVA Structural Configuration

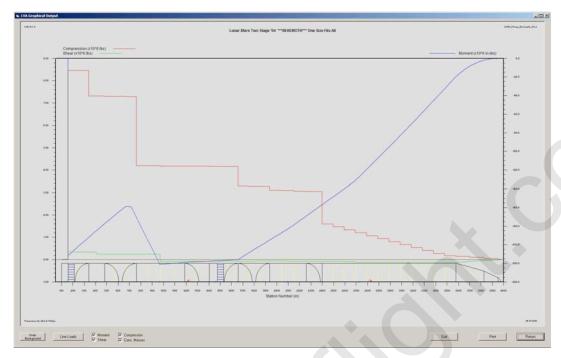


Figure 6C-103. LVA Structural Loads Analysis Results

This concept was also flown with EDS to determine the lunar payload capability for this vehicle. Four different EDS scenarios were analyzed: (1) No suborbital burn EDS only (no payload attached at launch), (2) No suborbital burn of EDS with payload attached, (3) Suborbital burn of the EDS only (no payload attached at launch), and (4) Suborbital burn of the EDS with payload attached. Payloads to three different lunar orbits were calculated for each scenario: TLI,; TLI + LOI; TLI + LOI + Plane Change.

# 6C.5.2.7.2 Vehicle Sizing

The mass properties for LV 7.4 are shown in **Table 6C-16**. No growth allowance was applied to the Atlas V booster flight hardware.

Table 6C-16. LV 7.4 INTROS Mass Summary

MASS PROPERTIES ACCOUNTING  VEHICLE: (5/4+) Atlas-Evolved 8-m Core + 2 AV Boosters Cargo – Blk 2  STAGE: Liquid Booster (1 RD–180)				
ITEM	Tertiary	Secondary	Primary	<b>TOTALS</b>
	lbm	lbm	lbm	lbm
Primary Body Structures			23,467	
Secondary Structures			1,301	
Separation Systems			451	
TPS			132	
TCS			697	
MPS			21,066	
Power (Electrical)			726	

STAGE DRY MASS WITH GRO	OWTH (mdry)		184,376
Dry Mass Growth Allowance		14,182	
STAGE DRY MASS WITHOUT	GROWTH		170,194
Miscellaneous		264	
Avionics		670	
Power (Electrical)		2,017	
MPS		84,131	
TCS		2,006	
TPS		1,473	
Separation Systems		3,269	
Secondary Structures		4,893	
Primary Body Structures		71,470	
	STAGE: First St	rage (5 RD-180s)	
STAGE GROSS LIFTOFF MAS	SS (mgross)		771,590
Engine Purge Helium		75	
Main Ascent Propellant		666,458	
STAGE BURNOUT MASS (mbd	0)		105,056
In-flight Fluid Losses		117	
Reserves		8,085	
Residuals		7,431	
STAGE DRY MASS WITH GRO	OWTH (mdry)		89,422
Dry Mass Growth Allowance		8,714	
STAGE DRY MASS WITHOUT	GROWTH		80,708
Miscellaneous		194	
Avionics		590	
Power (Hydraulic)		807	
Power (Electrical)		1,775	
MPS		25,283	
TCS		2,175	
TPS		403	
Separation Systems		2,453	
Secondary Structures		2,987	
Primary Body Structures		44,040	<b>\</b>
		1 Stage (4 J–2S+)	
STAGE GROSS LIFTOFF MAS	SS (mgross)		505,948
Engine Purge Helium		19	
Main Ascent Propellant	,	450,964	34,700
STAGE BURNOUT MASS (mbd	)	100	54,965
In-flight Fluid Losses		100	
Reserves		3,318	
Residuals	(mury)	5,518	40,913
Miscellaneous  STAGE DRY MASS WITH GRO	OWTU (mdm)	117	48,913
Avionics		270	
Power (Hydraulic)		686	

Residuals		23,830	
Reserves		1,832	
STAGE BURNOUT MASS (m		210,038	
Main Ascent Propellant	2	2,740,680	
Engine Purge Helium		233	
STAGE GROSS LIFTOFF MA	2,950,951		
Payload		246,728	
Payload Shroud		23,411	
Upper Stage(s), Gross		771,590	
Strap-on(s), Gross Liftoff	1	,011,896	
VEHICLE GROSS LIFTOFF MASS (mgross_veh)			5,004,575

#### 6C.5.2.7.3 Structural Analysis

The loads plot is a combined worst case including pre-launch, liftoff, max q, and max g. The tie-down loads are assumed to be carried by the core vehicle. The compression loads show a major jump where the LOX tank, RP tank, and payload loads are integrated into the outside structure. For the purposes of the analysis, all boosters were assumed to introduce axial loads at the aft of the core.

#### 6C.5.2.7.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results are shown in **Figure 6C-104**. Selected trajectory parameters are shown graphically in **Figures 6C-105** through **6C-107**. The vehicle exhibits a 1.21 T/W ratio at liftoff. Maximum dynamic pressure is 567 psf at 85.6 sec in the flight. The maximum acceleration is 3.53 g's during the LRB burn, 3.73 g's during the first-stage burn after LRB separation, and 3.12 g's during the second-stage burn. LRB jettison occurs at 163.5 sec into the flight at an altitude of 164,902 ft and Mach 6.03. Stage 1 jettison occurs at 198.7 sec into the flight at an altitude of 236,435 ft and Mach 10.2. The T/W ratio at second-stage ignition is 1.05. Orbital injection occurs at 472.8 sec at 78.5 nmi. The analysis of the four EDS scenarios are shown in **Figures 6C-108** through **6C-111**.

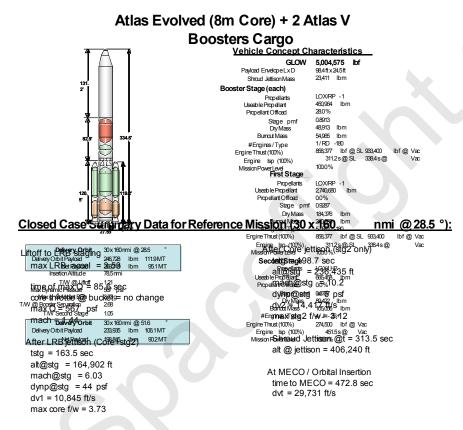


Figure 6C-104. LV 7.4 Summary

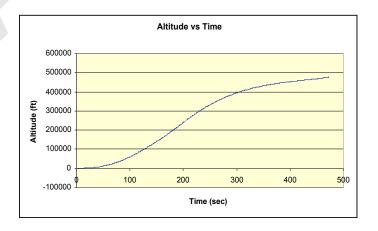


Figure 6C-105. Altitude versus Time

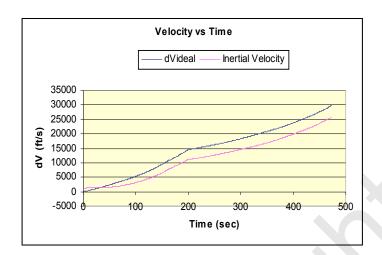


Figure 6C-106. Velocity versus Time

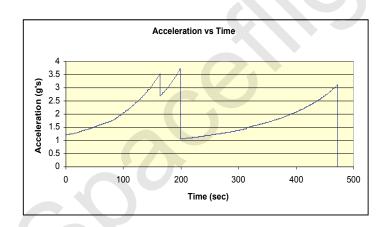


Figure 6C-107. Acceleration versus Time

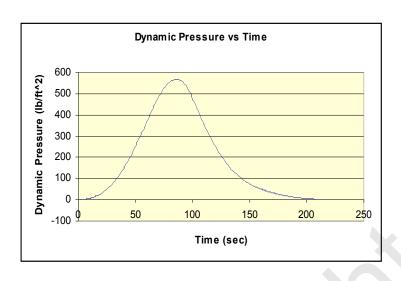


Figure 6C-108. Dynamic Pressure versus Time

# EDS – No Suborbital Burn No Payload Attached Launch Vehicle – Atlas-Evolved Heavy-Lift 8-m Core

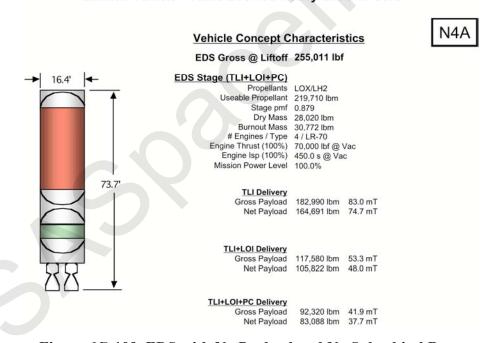


Figure 6C-109. EDS with No Payload and No Suborbital Burn

#### EDS – No Suborbital Burn with Payload Attached Launch Vehicle – Atlas-Evolved Heavy-Lift 8-m Core

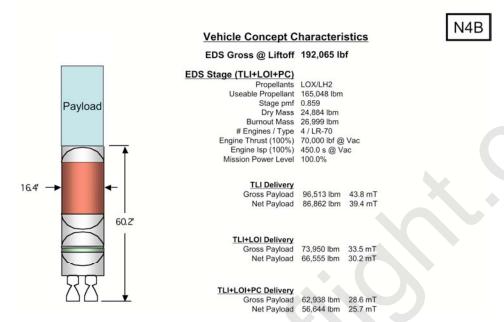


Figure 6C-110. EDS with Payload and No Suborbital Burn

#### EDS – Suborbital Burn with No Payload Attached Launch Vehicle – Atlas-Evolved Heavy-Lift 8-m Core



Figure 6C-111. EDS with No Payload and Suborbital Burn

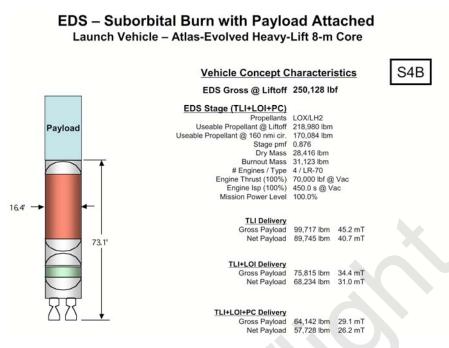


Figure 6C-112. EDS with Payload and Suborbital Burn

#### 6C.5.2.8 Launch Vehicle 7.5 (LV 7.5)

#### 6C.5.2.8.1 Vehicle Description

LV 7.5 (**Figure 6C-113**) is a two-and-one-half-stage LV for CEV plus cargo. The first stage is an ET diameter, LOX/RP stage with five RD–180 engines. The structural configuration is shown in **Figure 6C-113**. The LOX/LH2 second stage is also ET diameter with four J–2S+ engines for propulsion. This vehicle concept utilizes two Atlas V strap-on liquid rocket boosters that are attached to the core stage. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 78.4 nmi. All liquid engines were operated at a 100 percent power level in the analysis. Results of the loads analysis are provided in **Figure 6C-114**.

The net payload capability of LV 7.5 is 93.7 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. This net payload would constitute the CEV mass, SM mass, and payload mass contained in the cylindrical shroud. The net payload to  $30 \times 160$  nmi at a 51.6 deg inclination is 88.6 mT. No ground rules or constraints were violated for this LV analysis.

A special consideration was required to analyze this vehicle: The propellant in the Atlas V strap-on boosters was 28 percent to enable the LV to lift off with a T/W ratio greater than 1.2.



Figure 6C-113. LV 7.5 General Configuration

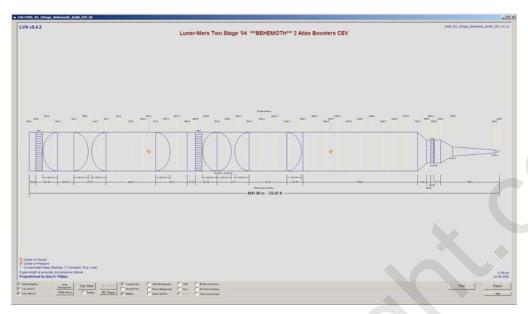


Figure 6C-114. LVA Structural Configuration



Figure 6C-115. LVA Structural Loads Analysis Results

# 6C.5.2.8.2 Vehicle Sizing

The mass properties for LV 7.5 are shown in **Table 6C-17**. No growth allowance was applied to the Atlas V booster flight hardware. The CEV adapter mass and cylindrical shroud shell mass were not included in the second stage primary body structures mass.

# Table 6C-17. LV 7.5 INTROS Mass Summary

1	MASS PROPERT	TES ACCOUNTIN	G	
VEHICLE: Atlas-E		) + 2 AV Boosters ( Booster (1 RD–180	_	lk 2
	M	ASS SUBTOTALS	8	MASS
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
Primary Body Structures			23,462	
Secondary Structures			1,301	
Separation Systems			451	
TPS			132	
TCS			697	
MPS			21,066	
Power (Electrical)			726	
Power (Hydraulic)			686	
Avionics			270	
Miscellaneous			117	
STAGE DRY MASS WITH GR	OWTH (mdry)			48,907
Residuals			5,518	
Reserves			434	
In-flight Fluid Losses			100	
STAGE BURNOUT MASS (mb	0)			54,959
Main Ascent Propellant			450,964	
Liquid Oxygen		457,967		
RP-1		168,372		
Offload Propellant 39.5%		-175,375		
Engine Purge Helium			19	
STAGE GROSS LIFTOFF MA	SS (mgross)			505,942
	STAGE: Secon	d Stage (4 J–2S+)		
Primary Body Structures			40,699	
Secondary Structures			2,987	
Separation Systems			2,457	
TPS			403	
TCS			2,175	
MPS			25,283	
Power (Electrical)			1,775	
Power (Hydraulic)			807	
Avionics			590	
Miscellaneous			194	
STAGE DRY MASS WITHOU'	T GROWTH			77,371
Dry Mass Growth Allowance			8,214	
STAGE DRY MASS WITH GR	OWTH (mdry)			85,585
Residuals			7,430	
Reserves			8,249	
In-flight Fluid Losses			117	

STAGE BURNOUT MASS (mbo)	101,382	
Main Ascent Propellant	666,294	
Engine Purge Helium	75	
STAGE GROSS LIFTOFF MASS (mgross	)	767,751
STAGE:	First Stage (5 RD–180s)	
Primary Body Structures	68,586	
Secondary Structures	4,893	
Separation Systems	3,246	
TPS	1,473	
TCS	2,006	
MPS	84,131	
Power (Electrical)	2,017	
Avionics	670	
Miscellaneous	264	
STAGE DRY MASS WITHOUT GROWT	Н	167,288
Dry Mass Growth Allowance	13,807	
STAGE DRY MASS WITH GROWTH (m	dry)	181,095
Residuals	23,830	
Reserves	1,832	
STAGE BURNOUT MASS (mbo)		206,757
Main Ascent Propellant	2,740,680	
Engine Purge Helium	233	
STAGE GROSS LIFTOFF MASS (mgross	2,947,670	
Payload	243,134	
Payload Shroud Cylindrical Shell	15,331	
LES	9,300	
Upper Stage(s), Gross	767,751	
Strap-on(s), Gross Liftoff	1,011,885	
VEHICLE GROSS LIFTOFF MASS (mgr	4,995,071	

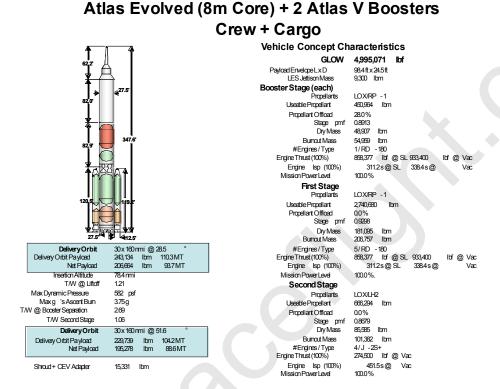
#### 6C.5.2.8.3 Structural Analysis

The loads plot is a combined worst case including pre-launch, liftoff, max q, and max g. The tie-down loads are assumed to be carried by the core vehicle. The compression loads show a major jump where the LOX tank, RP tank, and payload loads are integrated into the outside structure. For the purposes of the analysis, all boosters were assumed to introduce axial loads at the aft of the core.

#### 6C.5.2.8.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-116**. Selected trajectory parameters are shown graphically in **Figures 6C-117** through **6C-120**. The vehicle exhibits a 1.21 T/W ratio at liftoff. The maximum dynamic pressure was 582 psf at 87.0 sec into the flight. The maximum acceleration is 3.55 g's during the LRB burn, 3.75 g's during the first-stage burn after LRB separation, and 3.05 g's during

the second-stage burn. LRB jettison occurs at 163.5 sec into the flight at an altitude of 157,249 ft and Mach 6.13. Stage 1 jettison occurs at 198.7 sec into the flight at an altitude of 222,801 ft and Mach 10.1. The T/W ratio at second-stage ignition is 1.06. Orbital injection occurs at 472.7 sec at 78.4 nmi.



#### Closed Case Summary Data for Reference Mission (30 x 160 nmi @ 28.5 °):

```
Liftoff to LRB staging
                                                                After Core jettison (stg2 only)
   max LRB accel = 3.55
                                                                   tstg = 198.7 sec
                                                                   alt@stg = 222,801 ft
   time of max Q = 87.0 sec
                                                                   mach@stg = 10.1
   core throttle @ bucket = no change
                                                                   dynp@stg = 11 psf
   max Q = 582 psf
                                                                   dv2 = 14,476 \text{ ft/s}
   mach = 1.48
                                                                   max stg2 f/w = 3.05
 After LRB jettison (Core+stg2)
                                                                  Launch Escape System Jettison
   tstg = 163.5 sec
                                                                   t-1 FS = 229 sec
   alt@stg = 157,249 ft
                                                                   alt @ LES jettison = 281,597 ft
   mach@stg = 6.13
   dynp@stg = 61 psf
                                                                  At MECO / Orbital Insertion
   dv1 = 10.881 \text{ ft/s}
                                                                   time to MECO = 472.7 sec
   max core f/w = 3.75
                                                                   dvt = 29,686 \text{ ft/s}
```

Figure 6C-116. LV 7.5 Summary

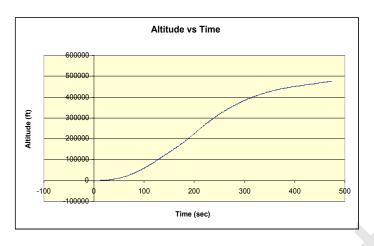


Figure 6C-117. Altitude versus Time

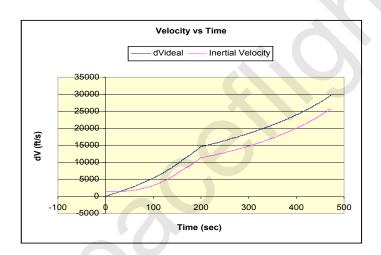


Figure 6C-118. Velocity versus Time

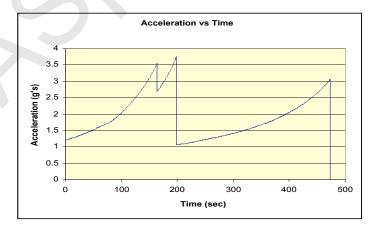


Figure 6C-119. Acceleration versus Time

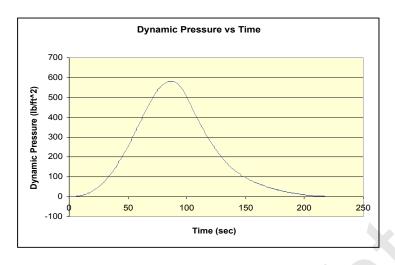


Figure 6C-120. Dynamic Pressure versus Time

#### 6C.5.2.3.10 Launch Vehicle 11 (LV 11)

#### 6C.5.2.3.10.1 Vehicle Description

LV 11 (**Figure 6C-121**) is a two-and-one-half-stage LV for cargo based on a common booster design. The core stage is a 5.4-m diameter LOX/RP stage with two RD–180 engines. The structural configuration is shown in **Figure 6C-122**. This vehicle concept utilizes four LRBs that are attached to the core stage. These strap-on LRBs are common with the core stage in dimensions, propulsion, and propellant load, with only some differences in structural design for the different load paths. The LOX/LH2 second stage is also a 5.4-m diameter with four LR–60 engines for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 78.2 nmi. The booster and upper stage engines were operated at a 100 percent power level in the analysis. The core stage engines lifted off at 100 percent and then throttled down to 47 percent after 30 sec and remained at that power level until booster separation, then powered back up to 100 percent. The structural loads analysis results are provided in **Figure 6C-123**.

The net payload capability of LV 11 is 93.8 mT to a 30 x 160 nmi orbit at a 28.5 deg inclination. The net payload to 30 x 160 nmi at a 51.6 deg inclination is 88.8 mT. No ground rules or constraints were violated for this LV analysis.



Figure 6C-121. LV 11 General Configuration



Figure 6C-122. LVA Structural Configuration



Figure 6C-123. LVA Structural Loads Analysis Results

Special considerations required to analyze this vehicle were:

• Original concept used four RL-10A-4-2 engines for the second stage. This provided a second stage T/W that was insufficient to control the second stage orbital insertion. The engines were subsequently upgraded to four LR-60s.

- The second stage is included in the shroud volume. Because of this, the shroud must be jettisoned before first stage jettison and second stage ignition. At the time of first stage jettison, the FMHR is 14.45 Btu/s/ft², which is 144 times greater than the ground ruled limit of 0.1 Btu/s/ft².
- The propellant tanks of all stages were designed with nested domes to be consistent with the contractors' design for this concept.

# 6C.5.2.9.2 Vehicle Sizing

The mass properties for LV 11 are shown in **Table 6C-18**. All hardware was considered to be new for this vehicle concept.

Table 6C-18. LV 11 INTROS Mass Summary

	MASS PROPERT	TIES ACCOUNTI	NG	
		Phase 3A Cargo – I p-on Booster (2 Rl		
STA				
ITEM 4	MASS SUBTOTALS			MASS TOTALS
ITEM	Tertiary	Secondary	Primary	
Drive and Dada Charachana	lbm	lbm	lbm	lbm
Primary Body Structures			24,459	
Secondary Structures			2,335 732	
Separation Systems TPS			297	
TCS			618	
MPS				
			41,210	
Power (Electrical)			964	
Power (Hydraulic)	(/)		1,373	
Avionics			370	
Miscellaneous	T CDOWTH		137	72.405
Dry Mass Growth Allowance	I GROWTH	1	7,207	72,495
STAGE DRY MASS WITH GR	OWTH (mdm)		7,207	79,702
Residuals	(mury)		8,629	13,102
Reserves			735	
In-flight Fluid Losses			199	
STAGE BURNOUT MASS (mb			199	89,265
Main Ascent Propellant			1,054,862	67,203
Engine Purge Helium			32	
STAGE GROSS LIFTOFF MA	SS (mgross)		32	1,144,160
STAGE GROSS EIT TOFF WA	( 0 /	ge 2- (4 LR-60)		1,144,100
Primary Body Structures	STITUE. Su		11,882	
Secondary Structures			211	
Separation Systems			133	
TPS			121	
TCS			452	
MPS			6,828	

Power (Electrical)	934	
Power (Hydraulic)	176	
Avionics	590	
Miscellaneous	57	
STAGE DRY MASS WITHOUT GROWTH		21,385
Dry Mass Growth Allowance	2,548	
STAGE DRY MASS WITH GROWTH (mdry)		23,932
Residuals	1,794	
Reserves	5,673	
In-flight Fluid Losses	26	
STAGE BURNOUT MASS (mbo)		31,425
Main Ascent Propellant	155,198	
Engine Purge Helium	7	<b>\</b>
STAGE GROSS LIFTOFF MASS (mgross)		186,630
STAGE: First Stag	e (2 RD–180)	
Primary Body Structures	32,990	·
Secondary Structures	1,945	
Separation Systems	3,420	
TPS	188	
TCS	868	
MPS	41,210	
Power (Electrical)	984	
Power (Hydraulic)	1,373	
Avionics	370	
Miscellaneous	150	
STAGE DRY MASS WITHOUT GROWTH		83,497
Dry Mass Growth Allowance	8,857	,
STAGE DRY MASS WITH GROWTH (mdry)		92,354
Residuals	8,629	
Reserves	735	
In-flight Fluid Losses	199	
STAGE BURNOUT MASS (mbo)		101,918
Main Ascent Propellant	1,054,852	
Engine Purge Helium	32	
STAGE GROSS LIFTOFF MASS (mgross)		1,156,803
Payload	243,397	
Payload Shroud	59,348	
Upper Stage(s), Gross	186,630	
Strap-on(s), Gross Liftoff	4,576,639	
VEHICLE GROSS LIFTOFF MASS (mgross veh)		6,222,816

# 6C.5.2.9.3 Structural Analysis

The loads plot is a combined worst case including pre-launch, liftoff, max q, and max g. The tie-down loads are assumed to be carried by the core vehicle. The compression loads show a major jump where the LOX tank, RP tank, and payload loads are integrated into the outside structure. For the purposes of the analysis, all boosters were assumed to introduce axial loads aft of the core.

#### 6C.5.2.9.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-124**. Selected trajectory parameters are shown graphically in **Figures 6C-125** through **6C-128**. The vehicle exhibits a 1.39 T/W ratio at liftoff. Maximum dynamic pressure is 607 psf at 74.6 sec into the flight. The maximum acceleration is 4.00 g's during the LRB burn, 3.16 g's during the first-stage burn after LRB separation, and 0.873 g's during the second-stage burn. LRB jettison occurs at 197.6 sec into the flight at an altitude of 205,457 ft and Mach 11.9. First stage jettison occurs at 279.1 sec into the flight at an altitude of 315,886 ft and Mach 19.3. The T/W ratio at second stage ignition is 0.56. Orbital injection occurs at 579.8 sec at 78.2 nmi.

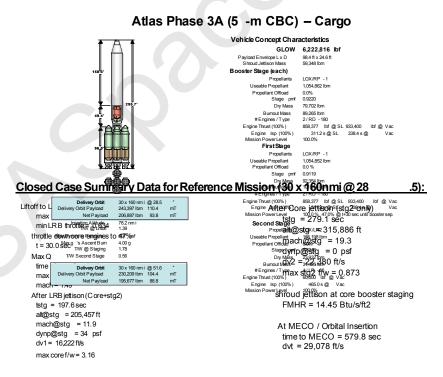


Figure 6C-124. LV 11 Summary

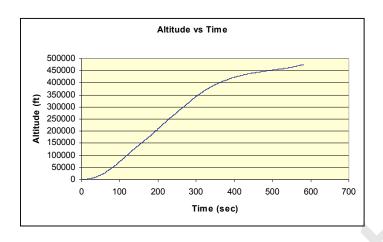


Figure 6C-125. Altitude versus Time

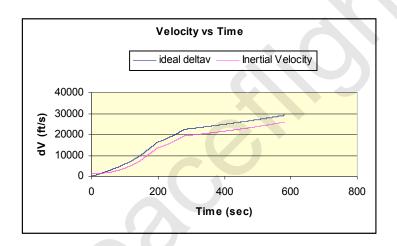


Figure 6C-126. Velocity versus Time

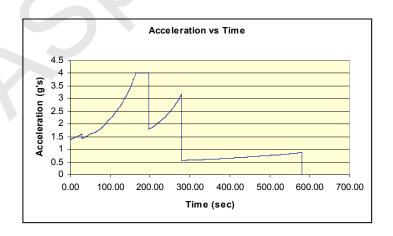


Figure 6C-127. Acceleration versus Time

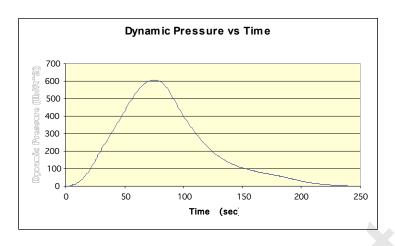


Figure 6C-128. Dynamic Pressure versus Time

#### 6C.5.2.10 Launch Vehicle 11.1 (LV 11.1)

#### 6C.5.2.10.1 Vehicle Description

LV 11.1 (Figure 6C-129) is a two-and-one-half-stage LV for crew and cargo based on a common booster design. This LV concept is the same as LV 11. except it carries a CEV above the payload shroud and the upper stage is located below the shroud. The structural configuration of LV 11.1 is shown in Figure **6C-130**. The core stage is a 5.4-m diameter LOX/RP stage with two RD-180 engines. This vehicle concept utilizes four LRBs that are attached to the core stage. These strap-on LRBs are common with the core stage in dimensions, propulsion, and propellant load, with only some differences in structural design for the different load paths. The LOX/LH2 second stage is also a 5.4-m diameter with four LR-60 engines for propulsion. This vehicle was flown to 30 x 160 nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 78.2 nmi. The booster and upper stage engines were operated at a 100 percent power level in the analysis. The core stage engines lifted off at 100 percent and then throttled down to 47 percent after 30 sec and remained at that power level until booster separation, then powered back up to 100 percent. Complete loads analysis results are provided in Figure 6C-131.



Figure 6C-129. LV 11.1 General Configuration

The net payload capability of LV 11.1 is 90.6 mT to a  $30 \times 160 \text{ nmi}$  orbit at a 28.5 deg inclination. This net payload would constitute the CEV mass, SM mass, and payload mass contained in the cylindrical shroud. The net payload to  $30 \times 160 \text{ nmi}$  at a 51.6 deg inclination is 85.3 mT.

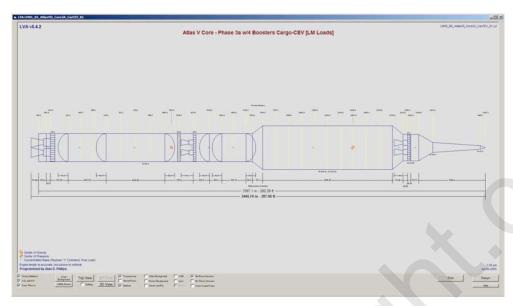


Figure 6C-130. LVA Structural Configuration



Figure 6C-131. LVA Structural Loads Analysis Results

For this concept, the LES was jettisoned at core stage burnout. This is a deviation from the nominal ground rule of LES jettison at 30 sec after upper stage ignition for this type of vehicle configuration. Performance analysis at nominal conditions was later checked and determined that this resulted in a 0.1 mT payload reduction and is not considered significant.

# 6C.5.2.10.2 Vehicle Sizing

The mass properties for LV 11.1 are shown in **Table 6C-19**. All hardware was considered to be new for this vehicle concept. The cylindrical shroud shell and CEV adapter were not included in the second stage accounting.

Table 6C-19. LV 11.1 INTROS Mass Summary

	MASS PROPERT	IES ACCOUNTI	NG	
	CLE: Atlas Phase AGE: Liquid Strap			
	M	MASS SUBTOTALS		
ITEM	Tertiary	Secondary	Primary	TOTALS
	lbm	lbm	lbm	lbm
Primary Body Structures			24,451	<u> </u>
Secondary Structures			2,335	
Separation Systems			732	
TPS			297	
TCS			618	
MPS			41,210	
Power (Electrical)			964	
Power (Hydraulic)			1,373	
Avionics			370	
Miscellaneous			137	
STAGE DRY MASS WITHOU	T GROWTH			72,487
Dry Mass Growth Allowance			7,206	
STAGE DRY MASS WITH GR	ROWTH (mdry)			79,692
Residuals			8,629	
Reserves			735	
In-flight Fluid Losses			199	
STAGE BURNOUT MASS (mb	00)			89,256
Main Ascent Propellant			1,054,862	
Engine Purge Helium			32	
STAGE GROSS LIFTOFF MA	SS (mgross)			1,144,150
	STAGE: Stag	ge 2 (4 LR–60)		
Primary Body Structures			11,943	
Secondary Structures			211	
Separation Systems			135	
TPS			121	
TCS			452	
MPS			6,828	
Power (Electrical)			934	
Power (Hydraulic)			176	
Avionics			590	
Miscellaneous			57	
STAGE DRY MASS WITHOUT GROWTH				21,447
Dry Mass Growth Allowance			2,557	

STAGE DRY MASS WITH GROWTH	24,004
Residuals	1,790
Reserves	6,125
In-flight Fluid Losses	26
STAGE BURNOUT MASS (mbo)	31,944
Main Ascent Propellant	154,747
Engine Purge Helium	7
STAGE GROSS LIFTOFF MASS (mg	186,698
STA	rst Stage (2 RD–180)
Primary Body Structures	33,616
Secondary Structures	1,945
Separation Systems	3,427
TPS	188
TCS	868
MPS	41,210
Power (Electrical)	984
Power (Hydraulic)	1,373
Avionics	370
Miscellaneous	150
STAGE DRY MASS WITHOUT GRO	84,130
Dry Mass Growth Allowance	8,952
STAGE DRY MASS WITH GROWTH	93,082
Residuals	8,629
Reserves	735
In-flight Fluid Losses	199
STAGE BURNOUT MASS (mbo)	102,646
Main Ascent Propellant	1,054,852
Engine Purge Helium	32
STAGE GROSS LIFTOFF MASS (mg	1,157,531
Payload	235,078
Payload Cylindrical Shroud Shell + CEV Adapter	30,543
Payload Shroud	9,300
Upper Stage(s), Gross	186,698
Strap-on(s), Gross Liftoff	4,576,601
VEHICLE GROSS LIFTOFF MASS (	_veh) 6,195,750

# 6C.5.2.10.3 Structural Analysis

The loads plot is a combined worst case including pre-launch, liftoff, max q, and max g. The tie-down loads are assumed to be carried by the boosters. The compression loads show a major jump where the LOX tank, RP tank, and payload loads are integrated into the outside structure. For the purposes of the analysis, all boosters were assumed to introduce axial loads at the interstage section of the core and the compression loads reflect such.

# 6C.5.2.10.4 Flight Performance Analysis and Trajectory Design

The closed case trajectory summary results and LV characteristics are shown in **Figure 6C-132**. Selected trajectory parameters are shown graphically in **Figures 6C-133** through **6C-136**. The vehicle exhibits a 1.39 T/W ratio at liftoff. The maximum dynamic pressure is 615 psf at 74.2 sec in the flight. The maximum acceleration prior to booster separation is 4.0 g's. Maximum core stage acceleration is 3.31 g's and maximum acceleration during stage 2 is 0.81 g's. LRB jettison occurs at 198.2 sec into the flight at an altitude of 204,512 ft and Mach 12.1. Core staging occurs at 279.4 sec into the flight at an altitude of 310,963 ft and Mach 20.0. The T/W ratio at second stage ignition is 0.53. Orbital injection occurs at 579.2 sec at 78.2 nmi.

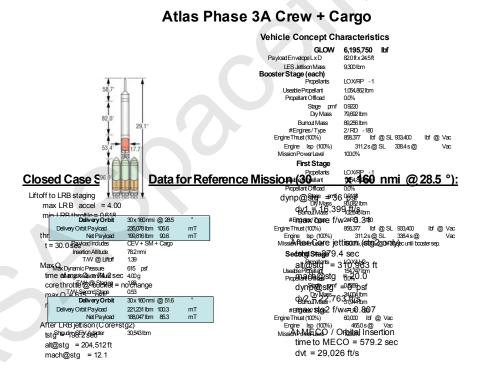


Figure 6C-132. LV 11.1 Summary

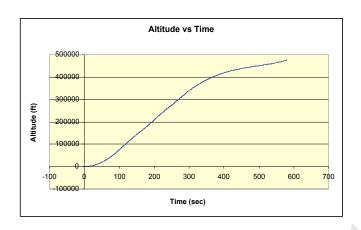


Figure 6C-133. Altitude versus Time

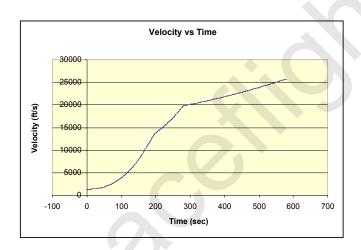


Figure 6C-134. Velocity versus Time

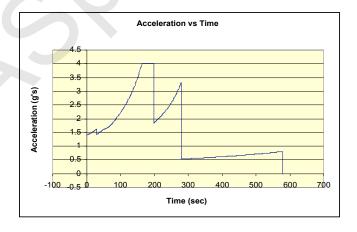


Figure 6C-135. Acceleration versus Time

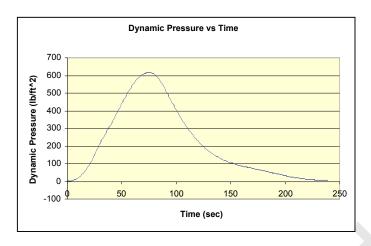


Figure 6C-136. Dynamic Pressure versus Time

# 6C.5.2.11 Cost Analysis for EELV for Cargo (LV 7.4, LV 7.5, LV 11, LV 11.1)

# **6C.5.2.11.1 Inputs – Core Stage**

## Structure and Tanks

All structures and tanks are considered a new design but with no new technology. Material is either 2219 aluminum or Al-Li. Shrouds are made of graphite-epoxy panels, which are based on Titan and Delta IV designs. Structures and tanks are well understood with sufficient manufacturing capability in existence. All structures are similar to EELV and ET, and have been validated in the relevant environment. All vehicles will, however, require full testing and qualification.

# **MPS—Less Engine**

The MPS will take significant heritage from the existing EELV MPS subsystem. However, the existing design will need to accommodate any changes to the subsystem for human rating the RD–180 and the RS–68 engines. NAFCOM cost estimate assumptions assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification is needed.

# Engine: RD-180

RD-180 is currently in production and is being flown on the Atlas V. However, the RD-180 is presently being produced in Russia. Coproduction in America is required. Also, design must meet requirements from the program Human Rating Plan. DDT&E costs assume minimal hardware modification for human rating, expansion/enhancement of the sensor suite, activation of flight redlines, and integrated FDIR.

## **Avionics and Software**

The avionics subsystem must support Fail Operational/Fail Safe vehicle fault tolerant requirements. Upon the first failure, the vehicle will keep operating. The second failure will safely recommend an abort. Crew abort failure detection and decision-making capabilities have been demonstrated and are ready for flight. All architectures will meet these requirements, either by adding a modification for instrumentation redundancy for

the EELV health management system, or providing the capabilities through the new design of the avionics for Shuttle-derived configurations.

# **EELV Avionics Hardware**

The GN&C and CCDH subsystems for cargo EELV vehicles are considered new designs. Because the subsystems and software are new, integrated health management and human-rating requirements are incorporated from the start. The avionics hardware assumed a new design with existing technology.

#### **EELV Software**

All software is considered a new software development, incorporating the functions identified above. The maximum SLOC estimate was used with the SEER-SEM model to arrive at a deterministic software estimate.

# **Other Booster Subsystems**

The remaining booster subsystems all used existing design and technology. Thermal, power, and range safety subsystems are in existence today, and have been validated for the relevant environment. NAFCOM cost estimate assumptions assumed existing structure/tanks with similar subsystems validated in the relevant environment. Minor modifications will be tested and qualified.

#### 6C.5.2.11.2 DDT&E

The Atlas Phase 3A 5-m common booster core is the least expensive of this group of vehicles. It can take more credit for heritage from the existing Atlas EELV. The other two configurations have more new design, since the core is stretched to 8 m.

# 6C.5.2.11.3 Production

LV 7.4/7.5 and LV 11/11.1 are heavy-lift crew/cargo configurations derived from the current commercial Atlas vehicle. The LV 7.4/7.5 vehicles are more costly to produce than the LV 11/11.1 vehicles.

# **6C.5.2.11.4** Operations

Except for the increased size, the configurations are essentially the same as the current vehicles. Operations would be essentially the same. The LV 11 configurations are more costly to process and launch than the LV 7.4/7.5 configurations.

# **6C.5.2.11.5** Facilities

The costs, provided in **Table 6C-20**, include both modifications to the vehicle processing and launch facilities and GSE and the provision of new lunar mission hardware facilities. The LV 11 vehicle facilities cost is the higher of the two by a significant margin.

Phase	Relative Cost Position		
Vehicle	7.4	7.5	11
DDT&E	1.43	1.43	1.00
Production	1.08	1.08	0.74
Operations	2.39	2.39	2.58
Facilities	1.12	1.12	1.56

# 6C.5.2.12 Safety/Reliability Analysis (Lunar Crew/Cargo EELV-Derived LVs)

The last set of LVs, the EELV-derived lunar crew/cargo vehicles, was analyzed using the same methodology as the previous vehicles. As before, the complete description of the analyses methodology is provided in **Appendix 6D**, **Risk and Reliability**, and the description of the reliability for LV systems is provided in **Section 6.8** of the **ESAS Final Report**. The four EELV-derived lunar crew/cargo LV estimates are shown in **Figures 6C-137** and **6C-138**. LOC estimates are only provided for the two crew versions. A 10 percent reduction in risk was applied to cargo vehicles to reflect the removal of redlines. This reduction of risk is discussed in detail in **Section 6.8** of the **ESAS Final Report**. Detailed LV analysis results are provided in **Appendix 6D**, **Risk and Reliability**.

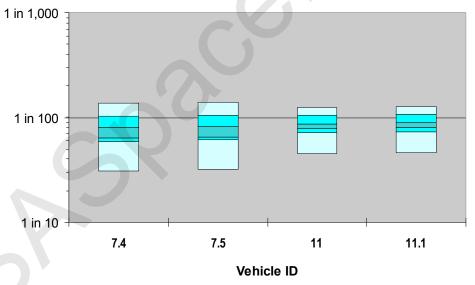


Figure 6C-137. Crew/Cargo LV LOM Estimates

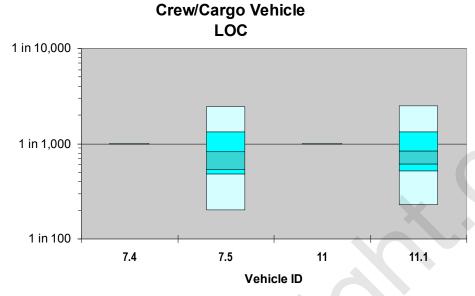
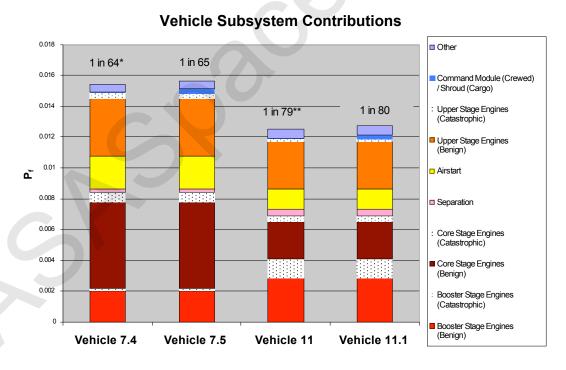


Figure 6C-138. LV LOC Estimates

**Figure 6C-139** shows the LV subsystem risk contributions. The LOM risk is approximately equal for all four vehicles and the differences in risk contributors are in the number of core stage engines.



<sup>\* 1</sup> in 71 with 10% adjustment for removal of redline inhibit

Figure 6C-139. Launch Vehicle Subsystem Risk Contributions

<sup>\*\* 1</sup> in 88 with 10% adjustment for removal of redline inhibit

# 6C.5.2.13 Schedule Assessment

There were no detailed development schedules generated for the cargo launch system options. The consensus was that the more clean-sheet EELV-derived design would require a longer development time than the Shuttle-derived solutions due to using well-characterized heritage systems (SRB, SSME). Given the traffic model of the first flight in 2017, the development would likely be 6–8 yrs depending on the option. The shorter development schedule allows the expenditure of most funds to be applied later, which helps in fitting the cost curve within the budget curve.

# **6C.6** Closed Case Trajectory Summaries

The complete results of all trajectory analysis are provided separately.



# Section 6D Safety and Reliability

# Appendix 6D.

# Safety and Reliability

# Contents

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6D.6 Results for Each Concept (LOM/LOC/Ranges)

# Appendix 6D Safety and Reliability

# 6D.1 Methodology/Tools

Two methodologies were used during this study. The FIRST tool, described in the following paragraphs, was the primary tool. A second tool was used to check the FIRST results as described in **Section 6D.1.2**, **Simple Reliability Model**.

FIRST, the Flight-oriented Integrated Reliability and Safety Tool, is a software application designed to calculate safety and risk Figures of Merit (FOMs) for manned and unmanned Launch Vehicles (LVs). The risk assessment methodology employed by FIRST builds on conventional, well-known techniques used to evaluate risk in complex systems, such as fault-tree and event-tree analysis. FIRST extends the accuracy and applicability of these techniques to LVs through the use of physics-based algorithms to estimate the probability of failure of vehicles based on their operational characteristics.

FIRST has been designed to facilitate rapid, consistent risk assessment of LVs in the early, pre-decisional, design phase. It is ideally suited for the comparison of diverse vehicle concepts and enables the direct comparison of LV safety and risk measures.

# 6D.1.1 Main Propulsion System (MPS) Modeling

The simplified MPS model used for launch vehicle reliability Loss of Mission (LOM) and Loss of Crew (LOC) predictions in this study is part of a launch vehicle analysis tool with models for all other subsystems. Although the incorporated model did not support MPS trade studies like the general MSFC MPS model described in **Section 6.8.2.5.2**, it did allow the rapid assessment of many complex launch vehicles against a uniform standard using strictly standardized methodologies to support objective reliability comparisons at the vehicle level.

# 6D.1.1.1 FIRST Launch Vehicle MPS Model

The fundamental methodology employed by FIRST in the analysis and risk assessment of LVs satisfies the standards and methodologies laid out in the guidelines on Probabilistic Risk Assessment (PRA) provided in the NASA publication, "Probabilistic Risk Assessment Procedures Guide for NASA Managers and Practitioners". However, FIRST extends standard methods of risk analysis using phenomenological models to dynamically calculate failure probabilities, more accurately reflecting differences between vehicles.

FIRST operates by taking detailed information about a launch vehicle, including the vehicle's configuration, subsystem technologies, and performance parameters such as weight, drag, engine per-second thrust values, and power level profiles, and processes

<sup>&</sup>lt;sup>1</sup> Stamatelatos, Michael *et al*, *Probabilistic Risk Assessment Procedures Guide for NASA Managers and Practitioners*, Version 1.0, NASA, March 31, 2002

this information using physics-based algorithms to calculate probabilistic risk estimates in the form of FOM distributions. The probability distributions are built up from individual data points via a Monte Carlo analysis, which is typically carried out using 10,000 trials. After a vehicle has been analyzed, an output file of the results is generated containing the system and vehicle level FOMs probability distributions reported at the median, mean, 5th, 25th, 75th, and 95th percentiles.

Vehicle information is passed to FIRST either automatically via an input file, or manually, by direct input via FIRST's Graphical User Interface (GUI) (**Figure 6D-1**). System level failure probabilities are stored in a database that is part of the FIRST executable. Baseline failure probabilities for subsystems are estimated through off-line analysis prior to running the tool. The failure rates used by FIRST are stored as mean failure probabilities along with associated Error Factors (EF) from which the uncertainty distributions are derived; lognormal distributions are assumed for most of the systems.

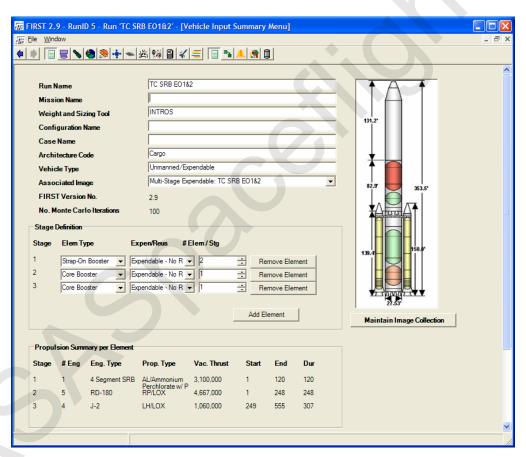


Figure 6D-1. Screenshot of FIRST

Contained within FIRST's database of subsystems is reliability data on specific subsystem technologies such as Auziliary Power Units (APUs), separation systems, Solid Rocket Boosters (SRBs), and liquid engines. The reliability data is generally stored as failure rates or failure probabilities, and in some cases (benign failure probability for

liquid engines) calculated dynamically based on the operational specifics of the mission (the engine burn time, power level, etc.). Reliability estimates for the subsystem technologies contained in FIRST are produced via off-line analyses by reliability analysts. Typical analyses use vehicle and systems heritage data, similarity analysis between existing (state-of-the-art) systems and advanced technologies, and expert solicitation and engineering judgment to model the reliability of the subsystems. For the 60-day study, reliability estimates for the liquid engines and payload shroud were updated by MSFC.

# **6D.1.1.1.1** Immediate Catastrophic Failure of Engine

Immediate Catastrophic Failure (ICF) of the engine subsystem is defined as any failure that would cause rapid disassembly of the engine and would lead immediately to breakup of the vehicle.

The SSME Block 2 is used as a baseline for the LH22 engines in the FIRST database. The Space Shuttle Main Engine (SSME) per-second failure rate is derived using Rocketdyne's 1999 three-engine nominal mission mean estimate of 1 in 1,181, which is also the mean reported in the QRAS 2000 Space Shuttle PRA results. The nominal mission is assumed to be 515 sec long, therefore the total per-second catastrophic risk per engine for the SSME at a nominal 104.5% power level is:

Equation 1 
$$P_{ICF}(SSME) = \frac{1/1181}{3/515} = 5.48E-07$$

To estimate the remaining engines, a Reliability Growth exercise was conducted by MSFC using the SSME Block 2 engine as a baseline. Details about their process can be found in **Section 6.8.2** of the **ESAS Final Report**. The results of their study for the remaining catastrophic failure estimates per second can be found in **Table 6-56**.

# 6D.1.1.1.2 Rubberized Engine Failure Probability Adjustment

To model engines that are identical in design, but of a different nominal thrust than an engine already in the FIRST database, a methodology was developed that allows an appropriate scaling of the risk. An equation was developed, Equation 2, based on the estimated catastrophic failure probability of the RD–180 and NK–33 engines, which are very similarly designed engines, but with significant vacuum thrust differences (the RD–180 is a 933,000 lbf (vac) thrust engine, while the NK–33 is a 367,600 lbf (vac) thrust engine). Equation 2 is assumed to hold for all types of engines in the FIRST database, irrespective of engine cycle or fuel.

Equation 2 
$$P_F(ENG_{RE}) = P_F(ENG_{BL}) \left(\frac{TR_{RE}}{TR_{BL}}\right)^{0.1017}$$

Where:

 $P_F$  (*ENG<sub>RE</sub>*) is the failure rate of the rubberized engine at 100% power level,  $P_F$  (*ENG<sub>BL</sub>*) is the failure rate of the baseline engine at 100% power level,

TR<sub>RE</sub> is the 100% power level vacuum thrust of the rubberized engine,

TR<sub>BL</sub> is the 100% power level baseline vacuum thrust, and

The exponent, 0.1017, was calculated by curve fitting the two NK-33 and RD-180 failure rate points

In application, the rubberized engine should be nearly identical in design as the baseline engine for Equation 2 to apply.

# 6D.1.1.1.3 Adjustment of Engine Failure Rate as a Function of Throttle Setting

Using an algorithm derived from SSME test history at Rocketdyne, an engine at 90 percent of its rated power level has about a 25 percent reduction in its failure probability. Additionally, an engine at 105 percent of its rated power level has about a 60 percent increase in its failure probability. Variations in the failure probabilities due to the changing power level are calculated from analysis of the SSME and are assumed to be the same for other engines. Equation 3 is used in FIRST to adjust the catastrophic and benign failure rates as a function of the throttle setting throughout the flight, and is calculated on a second-by-second basis.

Equation 3 
$$P_F(ENG_{PL}) = P_F(ENG_{NM})[TIFF + (1 - TIFF) \cdot e^{\left(\frac{\Delta PL}{RF}\right)}]$$

where,

 $P_F$  (ENG<sub>PL</sub>) is the failure rate at the operational throttle power level, i.e., the failure rate at 80% throttle is designated by  $P_{ICF}$  (ENG<sub>80</sub>),

 $P_F(ENG_{NM})$  is the failure rate at the nominal throttle setting (100% normally),

TIFF is the Throttle Insensitivity Failure Fraction,

 $\Delta PL$  is the change in throttle from nominal expressed as an integer (10% is 10), and RF is the reliability factor.

The RF is an exponential constant in the equation that defines normalized probability of failure as a function of throttle level. It is a calculated value using known throttle levels for an engine type and their corresponding probabilities of failure. Two different RFs are defined for each engine ( $RF_B$  and  $RF_A$ ) to denote the calculated reliability factors *below* and *above* a 100 percent throttle level. For RP-fueled engines, the calculated  $RF_B$  is 17.3 and the  $RF_A$  is 20. These numbers are based on data from the RD–180, and are assumed to be identical for all RP-fueled engines available in FIRST. For LH2-fueled engines, the calculated  $RF_B$  and  $RF_A$  values are very close and thus are considered as the same value of 8.29. The RF of 8.29 was calculated using known SSME data, and is also assumed to be identical for all LH2 fueled engines in FIRST.

The Throttle-Insensitivity Failure Fraction (TIFF) for liquid hydrogen engines is estimated to be 65 percent. This means that only 35 percent of the failure probability is sensitive to throttle variability. This estimate was obtained through MSFC in 2002 with support from Rocketdyne. The TIFF is assumed to remain constant regardless of the burn

time of the engine. Using test data from the RD-180 the TIFF for RP fueled engines is estimated at 80 percent.

**Table 6D-1** provides the failure rates for the engines used in this study. The per-second risk estimates shown in the table are nominal, i.e., they are the baseline values used prior to any adjustments made for operational throttle setting, or vacuum thrust adjustments.

# 6D.1.1.1.4 Benign Engine Failure

Benign engine shutdown is caused either by off-nominal operation, which is detected by a sensor, or by erroneous sensor readings that trigger an engine cutoff. A benign shutdown usually leads to an Loss of Mission (LOM) event for vehicles that do not have engine-out capability.

The statistical analysis of benign engine failure probability as a function of burn time follows a Weibull distribution, and is therefore not linear with engine burn time. Rather, the failure probability is skewed towards engine start<sup>2</sup> as shown in **Figure 6D-2**. The Weibull equation, given in Equation 4, is determined by two parameters, the scale factor,  $\eta$ , and the shape factor,  $\beta$ . The MSFC team recommended that the shape factor be set to  $\beta$  = 0.5, for all engines. The varying values of  $\eta$  for each engine are listed in **Table 6D-1** 

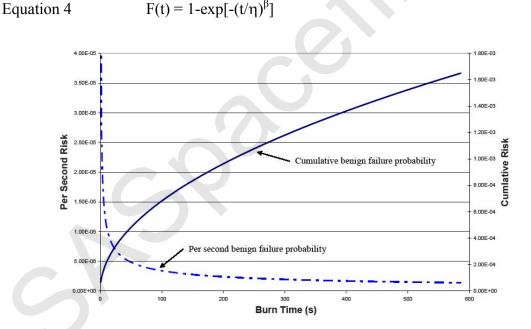


Figure 6D-2. Plot Showing Engine Benign Failure Probability Strongly Skewed
Towards Engine Start Time

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<sup>&</sup>lt;sup>2</sup> Joseph R. Fragola, Gaspare Maggio, et al. *Probabilistic Risk Assessment of the Space Shuttl:*, A Study of the Potential of Losing the Vehicle during Nominal Operation, Volume I: Final Report. SAIC, February 28, 1995.

# **6D.1.1.1.5 Eta (η) Calculations for Weibull Distribution**

The following discussion explains the calculation of the  $\eta$  (eta) values used in the Weibull Distribution for the benign engine shutdown failure probability. This explanation is essential since the MSFC team did not provide direct failure probabilities for benign engine shutdown. Instead, the team provided per-second catastrophic failure rates through estimating reliability growth, and Catastrophic Failure Fractions (CFFs) through expert opinion. These values are in **Table 6D-1** with the resulting  $\eta$  values.

The provided CFF and per-second catastrophic failure rate is used to find the corresponding benign failure probability. Since benign failure probabilities are not calculated in a per-second unit, a generic mission burn time of 515 sec was selected in order to apply the CFF to derive consistent failure probabilities. Therefore, the catastrophic failure rate per second is converted to an overall mission failure probability for an assumed engine burn time of 515 sec.

Equation 5 
$$P_{ICF}(ENG_M) = P_{ICF}(ENG_{PerSec}) \cdot 515$$

The resulting benign failure probability for 515 sec is then calculated using the CFF ratio, as shown in Equation 6.

Equation 6 
$$P_{BGN}(ENG_{515}) = \frac{P_{ICF}(ENG_{515})}{CFF} - P_{ICF}(ENG_{515})$$

From the mission benign failure probability, a corresponding  $\eta$  value is calculated using a Weibull equation that has been solved for  $\eta$ . Equation 7 gives the proper form for deriving  $\eta$ .

Equation 7 
$$\eta = \frac{515}{\left[-\ln(1 - P_{BGN}(ENG_{515}))\right]^{\frac{1}{\beta}}}$$

# **6D.1.1.1.6 Delayed Catastrophic Failure**

The outcomes of engine shutdowns are a function of the state of the vehicle (i.e., velocity, Thrust-to-Weight (T/W), etc.) and the ability of the vehicle's remaining engines to compensate for the lost thrust. FIRST handles two types of engine shutdown initiated vehicle outcomes: Delayed Catastrophic Failure (DCF) and Benign Vehicle Failure (BGN). For expendable vehicles DCF and BGN failures both lead to LOM. The difference between the two is that delayed catastrophic failure implies an imminent loss of control of the vehicle, while a benign vehicle failure implies that a vehicle maintains sufficient thrust even though the remaining impulse is not sufficient to make the mission.

For single engine core or upper stages, an engine shutdown would be catastrophic since there would be total loss of thrust and an implied loss of control of the vehicle. In this case the probability of DCF is equivalent to that of the BGN failure probability (and if the stage is air-started, the startup failure probability). For multi-engine stages, the probability of DCF decreases with the number of engines.

The algorithm in FIRST that is used to estimate the probability of a delayed catastrophic failure utilizes the T/W and the throttle-up ability of the vehicle to determine the

consequences of engine shutdowns. Ascent is broken into multiple regions depending on the number of stages as:

Region 1: Liftoff to clear tower

Region 2: Clear tower to 1<sup>st</sup> stage separation

Region 3: 1st stage separation to 2nd stage separation

Region 4: 2<sup>nd</sup> stage separation to 3<sup>rd</sup> stage separation

Region m: (n-1)<sup>th</sup> stage separation to n<sup>th</sup> stage Main Engine Cutoff (MECO)

The vehicle must maintain critical T/W ratios after engine shutdowns or it will suffer a DCF. The basic trend is a decrease in the required T/W ratio in each region as the vehicle gains momentum and loses weight.

The nominal T/W ratio profile is calculated within FIRST as:

Equation 8 
$$T/W(t) = [VTP(t) - DRAG(t)]/GWP(t)$$

Where:

T/W(t) is the T/W ratio of the vehicle,

VTP(t) is the vehicle thrust profile,

DRAG(t) is the drag profile of the vehicle, and

GWP(t) is the gross weight profile of the vehicle.

All variables are time-dependent in this equation. Angle of attack effects are assumed negligible.

The maximum T/W is calculated for each engine shutdown scenario. The scenarios are calculated for combinations of engine shutdowns per stage. The maximum T/W that a vehicle can attain after losing the thrust of one or more engines on a stage is the instantaneous T/W of the vehicle times the ratio of the maximum thrust available to the vehicle relative to the instantaneous thrust of the vehicle. This method assumes that the engines ramp up to their maximum power level as soon as an engine shutdown is detected.

# 6D.1.1.1.7 Maximum Thrust-to-Weight Available after Engine Shutdown

The general equation describing the maximum T/W available to a vehicle after an engine shutdown is:

Equation 9 
$$T/W(t)_{max} = T/W(t) [Max/Nom]$$

Where Max and Nom are defined by:

$$Max = \sum_{i} [VT_{i}(MPL_{i})(E_{i}-EO_{i})(NE_{i})]$$

$$=VT_1(MPL_1)(E_1\text{-}EO_1)(NE_1)+VT_2(MPL_2)(E_2\text{-}EO_2)(NE_2)+\ldots+VT_n(MPL_n)(E_n\text{-}EO_n)(NE_n)$$

$$Nom = \sum_{i} [VT_{i}(PL_{i})(E_{i})(NE_{i})]$$

 $= VT_1(PL_1)(E_1)(NE_1) + VT_2(PL_2)(E_2)(NE_2) + ... + VT_n(PL_n)(E_n)(NE_n)$ 

and where,

(T/W)(t) = Instantaneous thrust to weight at time t,

 $VT_i$  = Vacuum thrust per engine on the  $i^{th}$  stage, with i = 1, 2, ..., n,

 $MPL_i$  = Maximum power level per engine for the  $i^{th}$  stage (assumed to be 109% for the SSME and 100% for all other engines),

 $PL_i$  = Instantaneous power level of the engines on the  $i^{th}$  stage,

 $E_i$  = Number of engines on the  $i^{th}$  stage,

 $EO_i$  = Number of engine shutdowns on the  $i^{th}$  stage, and

 $NE_i$  = Number of thrusting elements on the  $i^{th}$  stage.

Equation 12 is applicable to both parallel burn stage combinations as well as series burn stages. In the series burn case, the index i is set equal to the stage in question and the sums for the Max and Nom terms in Equation 12 contain only one term.

Once the maximum T/W with engine losses is calculated for each shutdown scenario, the various scenarios can then be compared to the T/W limits (described in the following paragraphs) and the times at which the vehicle can sustain engine shutdowns determined.

# 6D.1.1.1.8 Thrust-to-Weight Ratios

Once the maximum T/W profile is calculated, the following assumptions are used to calculate the time at which particular engine shutdown combinations lead to a DCF.

Assumptions driving the DCF criteria:

The vehicle takes 10 seconds to clear the tower.

Additional required throttle setting is available instantaneously.

From liftoff to clear tower:

With fall away towers or no tower:  $T/W_{max} < 1.0$  results in DCF Without fall away towers:

 $T/W_{max} > 1.2$  results in 0% DCF

 $T/W_{max} < 1.0$  results in 100% DCF

 $1.0 < T/W_{max} < 1.2$  DCF follows a linear relationship ranging between 33.3%-0%

From clear tower to  $1^{st}$  stage separation  $T/W_{max} < 1.0$  results in DCF.

For all subsequent separations:  $T/W_{max} < (0.6) T/W_{nom}$  after staging results in DCF.

The majority of risk occurs in the first 10 to 30 seconds of flight and is due to many factors. The T/W limit is the most stringent at liftoff due mainly to the fact that any translation of the vehicle may cause contact with the tower. Since benign engine failure is skewed towards engine start (see **Figure 6D-2** in **Section 6D.1.1.1.4**, **Benign Engine Failure**), the probability of delayed catastrophic failure due to contained engine shutdowns starts at its highest value at liftoff and decreases during the vehicle's ascent.

# **6D.1.1.1.9** Engine Startup Failures

# **Ground-Start Failure Probability**

Hold-down failures are failures that occur while the launch vehicle is still on the ground and do not contribute to FOMs in FIRST. Hold-down failures occur when liquid engines or Solid Rocket Motors (SRMs) either fail to ignite or, in the case of liquid engines, are shutdown due to reaching the redline limits. For vehicles with liquid engines, it is assumed that if any engine fails to start, the remaining engines in the cluster are shut down and the launch is scrubbed. On vehicles with multiple Shuttle-type SRBs, failure of any of the SRBs to ignite causes an immediate catastrophic event and, in the case of crewed vehicles, necessitates an abort. In the case of in-line Shuttle-Derived Vehicles (SDVs) that use a single SRB, failure of the SRB to ignite is considered a hold-down failure and does not contribute to the FOMs.

# **Air-Start Failure Probability**

For series burn vehicles, there is an inherent risk of an upper stage engine, or engine cluster failing to start. Air-start failure probabilities for the ESAS were based on expert opinion and the limited historic data, listed in **Table 6D-1**. Aside from the specific startup failure probabilities, the following assumptions were made regarding air-started stages:

- For an engine cluster (two or more engines) there is a 10 percent chance that any single engine startup failure will cause the entire cluster to fail to start (i.e., the probability of the entire cluster failing to start is 10 percent of the startup failure probability);
- The startup period, from ignition to full stable thrust, is instantaneous (no transients);
- The cluster startup failure is independent of:
  - o Manifold complexity changes for different number of engines in a cluster,
  - o Overall engine size, and
  - Nominal thrust level of the engine.
- The probability of catastrophic (uncontained) failure during engine startup is negligible.

Table 6D-1. Engine Data

	Catastrophic Per Sec. Risk	Probability (515 sec)	η	Startup Risk	CFF
SSME Block 2	5.480E-07	1.482E-03	2.342E+08	3.000E-04	16.0%
RL-10A-4-2	3.360E-08	2.711E-04	7.006E+09	3.000E-04	6.0%
LR-60	1.026E-07	1.004E-03	5.103E+08	3.318E-04	5.0%
LR-85	1.049E-07	1.026E-03	4.885E+08	3.438E-04	5.0%
RS-68	3.921E-07	1.351E-03	2.816E+08	<b>&gt;</b>	13.0%
J-2S	3.301E-07	1.247E-03	3.310E+08	5.280E-04	12.0%
RD-180	8.104E-07	1.779E-03	1.039E+08		15.8%

None of the vehicles analyzed during the 60-day study had engine-out to mission continuance capability; hence any air-start failure led to an LOM event. For a single-engine upper stage, failure to start the engine would lead to loss of all thrust of the upper stage, and subsequent loss of the mission. A startup failure on a single-engine upper stage is categorized as a DCF (P<sub>DCF</sub>); see **Appendix 6D.1.1.6**, **Delayed Catastrophic Failure**, for a description of DCF.

P<sub>DCF</sub> is calculated for a single engine as:

Equation 1

$$P_{DCF} = P_{SUF}$$

Where:

P<sub>DCF</sub> is the Probability of Delayed Catastrophic Failure

 $P_{\text{SUF}}$  is the Probability of startup failure

For multi-engine vehicles without engine-out to mission continuance capability:

Equation 2

$$P_{DCF} = [1-(1-P_{SUF})^{\#ENG}] \cdot 0.1$$

and

Equation 3

$$P_{BEN} = [1-(1-P_{SUF})^{\#ENG}] \cdot 0.9$$

Where:

P<sub>BEN</sub> is the Probability of Benign Failure,

P<sub>DCF</sub> is the Probability of Delayed Catastrophic Failure,

P<sub>SUF</sub> is the Probability of a single engine failing to start, and

#ENG is the number of engines on the stage.

Equation 2 reflects the assumption (based on engineering judgment) that there is a 10 percent probability that a startup failure in one engine on a stage results in the (common cause) failure of all engines on that stage, which leads to a delayed catastrophic failure of the vehicle. The remainder of the time it is assumed that, according to Equation 3, the remaining air-start failures lead to benign vehicle failures (P<sub>BGN</sub>), which still necessitate an abort, but under more favorable conditions (implying a CEEF (Crew Escape Effectiveness Factor) of 90 percent, rather than 80 percent as in the case of the DCF).

The startup failure probability for each engine type can be found in previous **Table 6D-1**.

The sources for the startup failure probabilities used in this study are the following:

SSME Block 2—Test data from a March 2003 Rocketdyne report (Biggs) was updated assuming a 99 percent fix factor for startup problems resulting in a failure probability per engine of 1 in 661. Based on an SAIC analysis of redline data from the SSME it was determined that 79 percent of all redlines are preventable or catastrophic failures which, when applied to an air-started SSME, would adjust the 1 in 661 ground-start failure probability to 1.193E-03, or 1 in 837. Further, it was assumed that a rigorous test program could reduce the SSME startup risk from its current value by 75 percent. This assumption led to a final SSME air-start estimate of 3.0E-04, or 1 in 3,333.

RL-10A-4-2—Startup failure probability was determined via a parametric analysis with the RLX concept engine, since the RLX was essentially a scaled up version of the RL-10 design.

LR-60—Based on the RL-10A-4-2 startup failure probability and scaled for thrust.

LR-85—Based on the RL-10A-4-2 startup failure probability and scaled for thrust.

J–2S—Startup risk was calculated using actual flight data of the Ariane 4 launch vehicle (Viking gas generator engine), which shows no startup failures. The Viking engine was chosen since there was a significant flight history for this engine. At the time of the analysis, there had been 115 launches of the Ariane 4 and 567 startup successes with no failures, so the statistical one-third rule was applied.

# 6D.1.1.1.10 Loss of Thrust Vector Control (TVC)

A fault tree developed for the Shuttle 1995 PRA<sup>3</sup> is used to estimate the risk of TVC failure and is adjusted for the number of engine TVCs that can be lost before a catastrophic event occurs (ignoring common cause). The failure probabilities were calculated for different sequences defined by the number of engines and the number of failures required for loss of control and subsequent breakup of the vehicle using the TVC failure model in the Shuttle 1995 PRA as a basis. **Table 6D-2** shows the LOV probability for different combinations of number of engines and number of TVC failures that lead to catastrophic vehicle failures.

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<sup>&</sup>lt;sup>3</sup> Joseph R. Fragola, Gaspare Maggio, et al. *Probabilistic Risk Assessment of the Space Shuttl:*, A Study of the Potential of Losing the Vehicle during Nominal Operation, Volume I: Final Report. SAIC, February 28, 1995.

As the number of engines increases, the number of TVC failures required before an LOV event also increases with an associated decrease in LOV risk with number of TVC actuators. Computation of the probability of MPS LOV due to TVC failures is based on the number of engines and the number of allowed TVC failures. Expert opinion was solicited to determine the number of TVC failures that could occur before the vehicle would lose control authority. It was ascertained that loss of more than one-third of total TVC control would cause the vehicle to be subject to irrecoverable aerodynamic forces leading to catastrophic failure of the vehicle.

1 Engine TVC Lost **Number of Engines** 2 Engine TVCs Lost 3.68E-05 2 7.36E-05 1.36E-09 3 1.10E-04 4.06E-09 1.47E-04 8.13E-08 4 1.84E-04 1.36E-08 5

Table 6D-2: Probability of LOV Due to TVC Failure

# 6D.1.1.1.11 Loss of Stable Propellant Feed

The Propulsion Management System (PMS) transports the propellants from the tanks and delivers the propellants to the main engines. The PMS includes all connections, piping, and purging systems. The main failure modes are leaks leading to mixing of oxidizer and fuel, and leaks sufficient to affect the performance of the engines. The PMS risk was assumed to be similar to the Space Shuttle's PMS risk and was estimated by combination of the ascent time and the stages' engine burn time. The probability of failure of the MPS due to the PMS failure is given by:

Equation 13 
$$P_{ICF} (PMS) = (F_{PMS})(\#ENG)(t_{BT})$$

where

 $F_{PMS}$  = is the probability of the PMS failing per engine per second, and is

1.15E-08 based on the 1995 Space Shuttle PRA<sup>4</sup>,

#ENG = number of engines for the element under consideration, and

 $t_{BT}$  = engine burn time in seconds.

# **6D.1.1.1.12** Engine Error Factors

The baseline EFs used for the SSME and RD–180 are based upon the cluster error factor from the Block 2 SSME QRAS 2000 Study. The SSME EF was increased by 50 percent, based on engineering judgment, and rounded for the effect of incorporating the engine

<sup>&</sup>lt;sup>4</sup> Joseph R. Fragola, Gaspare Maggio, et al. *Probabilistic Risk Assessment of the Space Shuttl:*, A Study of the Potential of Losing the Vehicle during Nominal Operation, Volume I: Final Report. SAIC, February 28, 1995.

into a new launch vehicle design. The result is an EF of 2.6. The EFs for other advanced LH22 engines are based upon an SAIC RS–2100 heritage assessment. This EF is 9.7 and is applied to all LH22 engines such as the RS–68 and J–2S. The RL–10 and its derivatives were assigned an error factor of 5.0 that was assumed based on engineering judgment

The error factors for TVC and PMS are both assumed to be 10.

# 6D.1.1.1.13 Reusable Solid Rocket Booster (RSRB)

Table 6D-3. SRB and RSRM Failure Probabilities and Error Factors

		SRB		SRM	
Booster Type	Propellant Type	P <sub>ICF</sub>	EF	P <sub>ICF</sub>	EF
4-Segment	Aluminum (AL) w/ PBAN	2.026E-04	2.15	9.446E-05	1.84
5-Segment	AL w/ PBAN	2.026E-04	2.15	9.848E-05	1.84
4-Segment	AL w/ HTPB	2.026E-04	2.15	9.331E-05	3.68
5-Segment	AL w/ HTPB	2.026E-04	2.15	9.708E-05	3.68

Table 6D-4. RSRM Components in QRAS

Description	Mean (2 Motors)	Mean (1 Motor)
Nozzle-to-Case Joint	4.067E-05	2.034E-05
Internal Nozzle Joints	2.978E-05	1.489E-05
Case Field Joints (per 3)	1.306E-05	6.532E-06
Case Factory Joints *	1.614E-08	8.070E-09
Nozzle Liners	3.061E-05	1.530E-05
Case Membrane Burst	8.135E-07	4.068E-07
Head End Joint Seal Integrity	1.968E-05	9.840E-06
Propellant Energy	4.601E-06	2.300E-06
Internal Insulation Acreage	6.504E-06	3.252E-06
Foreign Object from RSRM Vehicle Damage	4.896E-06	2.448E-06
Exit Cone Premature Severance	8.131E-07	4.065E-07
Case/Nozzle Ring, Bracket, & Shell Structural	8.182E-07	4.091E-07

Flex Bearing Structural/Thermal	4.882E-06	2.441E-06
Igniter Structural	8.134E-07	4.067E-07
Igniter Thermal	5.417E-06	2.709E-06
Igniter and Main Propellant Ignition	2.556E-05	1.278E-05

Table 6D-5. SRB Components in QRAS

Description	Mean (2 Motors)	Mean (1 Motor)
Separation System	1.165E-04	5.825E-05
APU	9.879E-05	4.940E-05
BSM	6.251E-05	3.126E-05
FIV	3.819E-05	1.910E-05
Structures System	4.704E-05	2.352E-05
Servoactuator	4.611E-05	2.306E-05
Thermal Protection System	3.631E-05	1.816E-05
TVC Plumbing	5.822E-06	2.911E-06
Check Valve and Filter Assembly	3.614E-06	1.807E-06
Fluid Manifold	3.352E-06	1.676E-06
Hydraulic Pump	3.316E-06	1.658E-06
Hydraulic Accumulator	3.162E-06	1.581E-06
Hydraulic Bootstrap Reservoir	2.145E-06	1.073E-06
Manual Shut-off Valve	1.686E-06	8.430E-07
Hold-down System	9.684E-07	4.842E-07
Quick Disconnect Cap Assembly	6.629E-07	3.315E-07
Fuel Supply Module	5.764E-07	2.882E-07
Hydraulic Pressure Block	8.410E-07	4.205E-07
Hydraulic Accumulator/Alternate	5.252E-07	2.626E-07

Fuel Filter	3.745E-07	1.873E-07
Other Risks	1.118E-04	5.590E-05

# **6D.1.1.2** Simple Reliability Model

An independent reliability model was constructed to check the results of the similarity model (FIRST) used to estimate the launch vehicle LOM and LOC. This model is a very simple reliability model that "ands" and "ors" mean point estimates for the launch vehicle primary risk contributors, the propulsion elements. **Figure 6D-3** shows the LV 13.1 case. **Figures 6D-4** and **6D-5** show the comparison of this simple model to the similarity model (FIRST) for all the vehicles studied. The similarity in LOM and LOC estimates, especially the comparative ranking establishes the credibility of the similarity model.

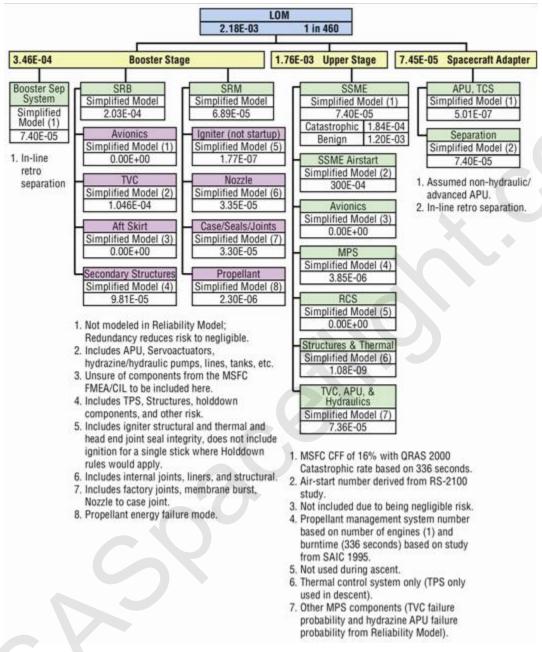


Figure 6D-3. Simple Reliability Model for LV 13.1

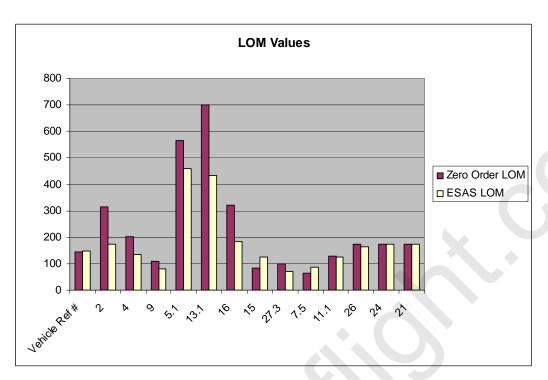


Figure 6D-4. LOM Estimates Comparison

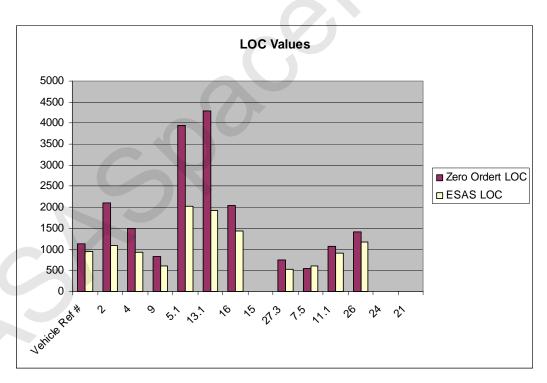


Figure 6D-5. LOC Estimates Comparison

## **6D.1.1.3** General MPS Model

In support of the ESAS, a liquid propulsion system reliability model was developed. The model was used to predict the reliability of selected stages including launch vehicle stages, Earth Departure Stages (EDSs), and in-space stages. Reliability trades on engine cycle, number of engines, and engine-out scenarios were performed.

# 6D.1.1.3.1 System Description and Reliability Logic Model

The liquid propulsion system reliability model reflects a systems approach to reliability modeling, i.e., the model simulates an engine in a propulsion system that includes main propulsion system elements and avionics elements. **Figure 6D-6** shows the modeled liquid propulsion system. The model reflects those physical elements that would have a significant contribution to stage reliability. For example, an engine purge system is indicated, because of the potential requirement for restart. However, while a fill-and-drain system would be present physically, such a system would be verified and latched prior to launch commit. Note that since the engine interface requirements are not known, the avionics, pneumatics, and hydraulic subsystems are modeled as combined elements.

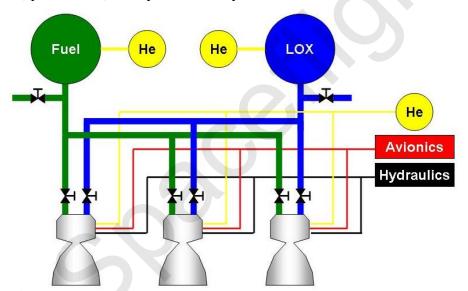


Figure 6D-6. Liquid Propulsion System Schematic

The liquid propulsion system reliability model described here is an event driven, Monte Carlo simulation of the schematic shown in **Figure 6D-6**. For each event, the cumulative failure distribution is randomly sampled to obtain a time-to-failure. The time-to-failure is compared to mission burn time. If the time-to-failure is less than the burn time, a failure is recorded. **Figure 6D-7** shows the top event logic for the reliability model. Note that parallel events indicate that a failure in any one path is a system failure. **Figure 6D-7a** shows the top-level events where the engine cluster is modeled in parallel with failures in the purge system and external leakage events. **Figure 6D-7b** shows the further breakdown of the cluster where each engine is modeled along with support systems. **Figure 6D-7c** shows the further breakdown of the engine support systems to include the pneumatics, hydraulics, and avionics provided to the engines. **Figure 6D-7d** shows the

sequence of events modeled at the individual engine level to include isolation valve failures and engine start and main stage failures.

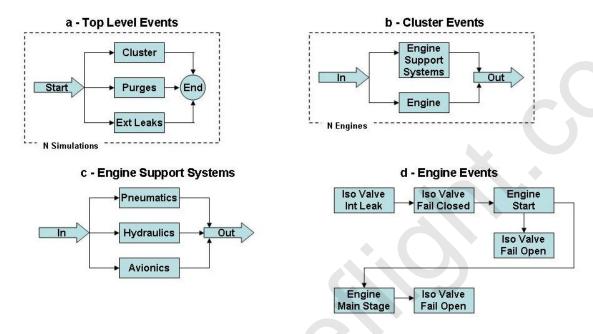


Figure 6D-7. Event Logic Model

For engine-out cases, if a first benign failure is recorded, then the burn time is scaled by the ratio of the original number of engines divided by the number of remaining operational engines. The time-to-failures for the remaining operational engines are compared to this new extended burn time. If the time-to-failure of any one of the remaining operational engines is less than the new extended burn time, then a stage failure is recorded.

# **6D.1.1.3.2** Data Sources and Event Quantification

The data source for quantifying the non-engine events is the current version of the Probabilistic Risk Assessment for the Space Shuttle Orbiter main propulsion system provided by Johnson Space Center. The one exception is that the avionics failure rates for the Space Shuttle Orbiter were not available. The engine controller failure rates for the SSME were used instead. **Table 6D-6** shows the failure parameters that were used for quantifying the non-engine failure events. Using Space Shuttle data to quantify event probabilities means that Space Shuttle design and operational philosophies are inherently assumed.

**Table 6D-6. Non-engine Failure Event Parameters** 

Event	Number Per Engine	Distribution Type	Distribution Parameters
Purge Valve Failure	2	Weibull	Shape = $0.5$ Scale = $8.02 \times 10^{12}$
External Leakage	6	Weibull	Shape = $0.5$ Scale = $1.73 \times 10^{12}$
Pneumatic System Failure	1	Weibull	Shape = $0.5$ Scale = $5.12 \times 10^{18}$
Hydraulic System Failure	1	Weibull	Shape = $0.5$ Scale = $5.12 \times 10^{18}$
Avionics System Failure	1	Weibull	Shape = $0.5$ Scale = $1.14 \times 10^{11}$
Isolation Valve – Internal	2	Demand	Mean = $3.15 \times 10^{-6}$
Isolation Valve – Fail Open	2	Demand	Mean = $3.88 \times 10^{-4}$
Isolation Valve – Fail Closed	2	Demand	Mean = $2.23 \times 10^{-4}$

For pump-fed engine cycles, a similarity analysis using SSME as the baseline was performed to obtain main stage engine failure rates. The similarity analysis is described in **Section 6.8.2.2** of the **ESAS Final Report**, provided engine main stage catastrophic failure probability per second and the CFF. For pressure-fed engine cycles, the Space Shuttle Orbital Maneuvering System (OMS), a pressure-fed system, was used as a baseline. Failure rates for the Space Shuttle OMS were also provided by Johnson Space Center. For a single OMS thruster, a catastrophic failure probability of  $1.03 \times 10^{-6}$  is predicted for a typical four-burn mission. Each burn was assumed to be 200 sec; this results in a per-second catastrophic failure probability of  $9.72 \times 10^{-9}$ . **Table 6D-7** shows the engine failure parameters used for this study.

**Table 6D-7. Engine Failure Parameters** 

Engine	P <sub>start</sub>	P <sub>cat</sub> / s (1 <sup>st</sup> Launch)	P <sub>cat</sub> / s (Mature)	CFF
Launch Vehicle Stages				
SSME	0.0005	5.48E-07	5.48E-07	0.16

RS-68	0.0001	2.73E-07	2.73E-07	0.13
RD-180	0.0001	6.49E-07	6.49E-07	0.16
J-2S+	0.0001	3.30E-07	2.27E-07	0.12
RL-10	0.0001	3.36E-08	3.36E-08	0.06
LR-85	0.0001	6.91E-08	6.91E-08	0.05
In Space Stages				
LH2-10K*	0.0001		1.89E-07	0.05
LH2-15K	0.0001		1.97E-07	0.05
LH2-20K	0.0001		2.03E-07	0.05
LM-10K Pump	0.0005		1.89E-07	0.05
LM-15K Pump	0.0005	-	1.97E-07	0.05
LM-20K Pump	0.0005		2.03E-07	0.05
LM-XK Pressure Fed**	0.0005		9.72E-09	0.25

<sup>\*</sup> All LOX/LH22 engines are pump fed.

# **6D.1.1.3.3** Reliability Growth Modeling

# Failure Data

In order to make the appropriate adjustments, it was necessary to analyze historical data to determine the reliability growth profile experienced over the life of the SSME Program.

A data file was obtained that provided the chronological list of the test and operations data for SSME through the last Shuttle flight and tests completed subsequent to that flight. The total accumulated time was approximately 1,000,225 sec. The initial estimate of the parameters of the SSME reliability growth model included all failures indicated in the file. At the time, it was known that some of the failures were properly categorized as "Facility Failures," however the data did not clearly distinguish between which failures resulted from facility issues and which were attributable to the SSME.

In order to properly estimate the reliability growth of the SSME, it was necessary to identify failures attributable to the SSME and to further distinguish between failures that would result in Loss of Mission (LOM) (Criticality 2 failures) and those that would result in Loss of Crew (LOC) (Criticality 1 failures). In searching for the appropriate data, it was determined that numerous versions of lists existed identifying the specific SSME failures as a subset of the total failures. The team settled on a list of 76 failures accepted

<sup>\*\*</sup> No relationship available for scaling pressure-fed reliability based on thrust.

by the SSME Project Office at MSFC as the basis for estimating the parameters of the SSME Reliability Growth Model. These failures were entered into the test/operation history time line as either Criticality 1 or 2 failures, allowing estimation of the parameters of the SSME Reliability Growth Model considering Criticality 1 failures only, Criticality 2 failures only, and the combined Criticality 1 and 2 failures.

The parameters were estimated by performing linear regression on the data points in the plot of LN(Cumulative Test/Operation Time) versus LN(Cumulative Failures), where Cumulative Failures were either Criticality 1, Criticality 2, or both.

# **Reliability Growth Model**

Various reliability growth models are available to use in describing the growth experienced by SSME. Because the Army Material System Analysis Activity (AMSAA) Reliability Growth Model was applied to some component-level SSME data, it was selected as the model for describing the engine growth. The AMSAA model takes the form

```
\theta i(t)=(\alpha/\beta)(t/\alpha)1-\beta where \theta i(t) = the instantaneous Mean Time Between Failure (MTBF) at time t \beta = the shape parameter of the AMSAA model \alpha = the scale parameter of the AMSAA model
```

In order to obtain estimates of the parameters of the AMSAA model, chronological data on cumulative test/operations time and failures experienced was used. The number of failures during any cumulative period of time, t, is given by the equation

$$N(t) = (t/\alpha)\beta$$

This is converted to a linear equation by taking the natural logarithm of each side to yield  $ln[N(t)] = \beta ln(t) - \beta ln(\alpha)$ 

The parameters are then estimated by plotting the natural logarithm of cumulative test/operation time (on the x-axis) versus the natural logarithm of cumulative failures (on the y-axis). The slope of the best fit line through these data points provides an estimate of the value of  $\beta$ . The value of  $\alpha$  is estimated by setting the y intercept of the best fit line equal to  $-\beta \ln(\alpha)$ . The value of  $\alpha$  is then calculated using the formula

$$A = e-y-intercept/\beta$$

Since the similarity analyses were based on QRAS 2000 Criticality 1 values, the Criticality 1 Reliability Growth Model was used. The parameters obtained were:

```
\alpha = 16200.89
\beta = 0.8191
y intercept = -7.9393
```

**Figure 6D-8** below shows the regression analysis performed on the data to obtain these values

# Engine Reliability Growth Curves - Cumulative Failures by Test Seconds and Number of Tests

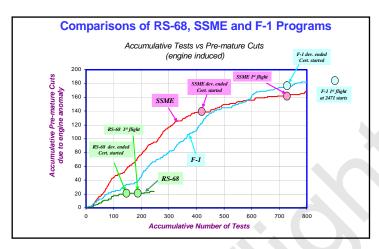


Figure 6D-8. Regression Analysis

# **Reliability Growth Model Applied to Other Engines:**

In order to use the AMSAA model obtained from the SSME to compensate for the immaturity of the engines, find a parameter c, such that  $\Theta_{newengine}$  (t) = c  $\Theta_{SSMEe}$  (t). This is accomplished by using the similarity analysis values and taking the fraction  $\Theta_{newengine}$  (t) /  $\Theta_{SSME}$  (t) = c. Since the instantaneous MTBF of a mature SSME is given by:  $\Theta_{SSME}$  (t) =  $(\alpha/\beta)(t/\alpha)^{1-\beta}$ , where t = 1,000,225 sec, the instantaneous MTBF of a mature new engine is:

$$\Theta_{newengine}\left(t\right)=c\ \Theta_{SSMEe}\left(t\right)=c\Theta_{SSME}\left(t\right)=c(\alpha/\beta)(t/\alpha)^{1-\beta}$$

These calculations provide the function relating the new engine to the MTBF of the SSME with respect to time.

Next, to find the corresponding "immature" value for the new engine replace t = 1,000,225 sec with  $t = t_{new}$ , where  $t_{new}$  is the maximum test time of the new engine, and where  $t_{new} \le t$ . This produces:

$$\Theta_{newengine} (t_{new}) = c(\alpha/\beta)(t_{new}/\alpha)^{1-\beta},$$

and provides an estimate of the compensation for the immaturity of the new engine. In actuality, the parameter c is also a function of time c(t), but in the interest of time and due to the lack of data, the constant c provides a good estimate, and as long as the same methodology is applied to each case, the results should be consistent.

# **Uncertainty in Reliability Growth**

The estimate of the uncertainty in the reliability growth model comes directly from the uncertainty in the parameters which, in turn, comes from the uncertainty in the regression

analysis applied to the data. The data shown below provides the uncertainty on the y intercept and the parameter  $\beta$ , or X Variable 1, as shown in **Figure 6D-9**. This uncertainty has a Gaussian distribution associated with it, consistent with the assumptions for the regression analysis.

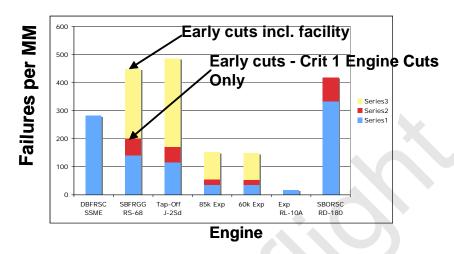


Figure 6D-9. Regression Analysis

Other sources of uncertainty in the final number come from the similarity analysis values found through the expert opinion of the engine leads, as well as the uncertainty associated with the QRAS values for the SSME.

# 6D.1.2 Historical Launch Vehicle Risk Contributors/Justification

Use of propulsion elements as the primary driver for launch vehicle reliability estimates is based on historical data. The historical data shows that in the early years of rocketry, avionics/software contributed approximately 15 percent to launch failures. From 1958 to 1997, propulsion systems (both liquid and solid) steadily came to dominate launch failures with up to 90 percent of launch failures due to propulsion systems from 1988–1997 for Atlas, Delta, and Titan vehicles. Most launch vehicle reliability assessments are based on this data, concentrated on estimating propulsion system failures as an indication of launch vehicle failure probability. The historic data also shows that the trend has sharply transitioned since 1980 with avionics/software and structures contributing approximately 70 percent to launch failures for U.S. launch vehicles. The reason for this sharp reversal is unclear, as is the possibility that this trend will continue in the next generation of expendable launch vehicles; however, all three of the four known avionics failures occurred within a 9-month period from 1998–1999 and none have occurred since in U.S. launch vehicles. Also, it was assumed that every launch vehicle would have two-fault avionics to meet NASA safety requirements.

# 6D.1.2.1 MSFC Historic Database 1980–2005 Tabular Data

Historic data from the years 1980–2005 for a variety of worldwide launch vehicles was assessed. The data was broken into unmanned and manned categories to determine subsystem failure percentages.

# **6D.1.2.1.1** Unmanned Historic Data 1980–2005

Fourteen launch vehicles were considered in this assessment, as shown in **Table 6D-8**.

Table 6D-8. Unmanned Launch Vehicles Assessed

Ariane	Athena	Atlas
Delta	H-Series	Long March
Pegasus	Proton	Soyuz/Molniya
Taurus	Titan	Tsiklon/Dnepr
Zenit		•

During the time period of January 1980 thru May 2005, there were 1,930 launch attempts of these vehicles with 92 failures. Only failures which were attributable to the launch vehicle and caused complete loss of mission objectives were counted.

The Proton and Soyuz/Molniya launch vehicle failures account for 38 percent of the total launch vehicle failures during this time period, with a combined total of 35 failures. The Ariane launch vehicle failures account for 11 percent of the total, having 10 failures during this time period. All other launch vehicles each account for less than 10 percent of the total launch vehicle failures. **Table 6D-9** shows, for each launch vehicle, the number of attempts, number of failures, percentage of the total failures, and demonstrated reliability during this time period.

Table 6D-9. Unmanned LV Historic Data 1980–2005

Launch Vehicle	Number of Attempts	Number of Failures	% of Total Failures	Reliability
Ariane	164	10	10.9%	0.9390
Athena	7	2	2.2%	0.7143
Atlas	135	8	8.7%	0.9407
Delta	161	5	5.4%	0.9689
H-Series	37	2	2.2%	0.9459
Long March	71	6	6.5%	0.9155
Pegasus	35	4	4.3%	0.8857

Proton	232	15	16.3%	0.9353
Soyuz/Molniya	765	20	21.7%	0.9739
Taurus	7	1	1.1%	0.8571
Titan	87	7	7.6%	0.9195
Tsiklon/Dnepr	178	6	6.5%	0.9663
Zenit	51	6	6.5%	0.8824
Total	1930	92		0.9523

The failures were then binned into five subsystem groups. The five groupings are:

- Liquid Propulsion,
- Solid Propulsion,
- Avionics/Software,
- Structure, and
- Unknown.

Liquid propulsion failures dominate the total failures during this time period, accounting for 46 percent of the total. Unknown failure causes make up 22 percent of the total failures. The majority of the unknown causes are from the Soyuz/Molniya and Tsiklon/Dnepr launch vehicles. The percentages for each failure grouping are shown in **Table 6D-10**.

Table 6D-10. Unmanned LV Historic Data 1980-2005

	Number of failures	% of total failures	
Liquid Propulsion	42	45.7%	
Unknown	20	21.7%	
Structure	14	15.2%	
Avionics/Software	12	13.0%	
Solid Propulsion	4	4.3%	
Total	92		

The failure descriptions for each vehicle are provided in **Tables 6D-11** through **6D-23**.

Table 6D-11. Arianne Failures

Date	Description	Vehicle Model	Category of Failure
5/23/1980	High frequency combustion instability at H+4s and H+28s degraded the injector of one of the first-stage engines. At H+64s the chamber pressure dropped, reducing thrust, and causing the vehicle to begin to roll. This affected the propellant feed system of the other engines, causing them to shut down between H+104 and 108 sec. The vehicle broke up, triggering auto-destruct at H+108.	Ariane 1	Propulsion
9/9/1982	A failure of the the third-stage turbopump gear caused pump speed to graually drop, beginning at H+560s until complete shutdown of the engine at H+610, halfway through the planned burn. The vehicle failed to reach orbit.	Ariane 1	Propulsion
9/12/1985	Third stage engine failed to ignite properly and shut down at H+276s because of a leaking hydrogen injector valve. The vehicle was destroyed by a range safety officer.	Ariane 3	Propulsion
5/31/1986	Third-stage igniter failed.	Ariane 2	Propulsion
2/22/1990	First-stage engine feedline blocked by rag, causing vehicle destruction 100 sec after launch.	Ariane 4 (44L)	Propulsion
1/24/1994	Third-stage turbopump bearing overheated due to insufficient precooling, causing stage shutdown 80 sec after ignition.	Ariane 4 (44LP+)	Propulsion
12/1/1994	Third-stage propellant impurity clogged oxygen line to gas generator causing insufficient thrust.	Ariane 4 (42P-3)	Propulsion
6/4/1996	Software error shut down redundant inertial reference systems, resulting in loss of control and aerodynamic breakup.	Ariane 5	Software
7/12/2001	Defect in the propellant on the last stage; this upper stage problem resulted in satellites being left in low orbit and a useless state.	Ariane 5	Propulsion
12/11/2002	Cracks in the cooling passages of the Vulcain 2 nozzle caused loss of coolant. The nozzle deformed from overheating causing assymetric thrust leading to flight control difficulties and a complete loss of control when fairing separation occurred.	Ariane 5	Propulsion

Table 6D-12. Athena Failures

Date	Description	Vehicle Model	Category of Failure
8/15/1995	Expended hydraulic fluid burned in 1st stage aft section, damaging nozzle feedback cables causing loss of gimbal control and tumbling. Separately, arcing in the IMU high-voltage power supply caused loss of attitude refrence. Flight terminated by range safety officer at T+160s.	Athena I	Propulsion
4/27/1999	Payload fairing failed to separate, and the extra weight prevented the vehicle from reaching orbit. During the fairing separation event, the shock of the circumferential ordnance firing disconnected the cable carrying the signal to fire the longitudinal ordnance.	Athena II	Shroud

Table 6D-13. Atlas Failures

Date	Description	Vehicle Model	Category of Failure
12/9/1980	Booster lube oil flow loss.	Atlas E	Propulsion
8/6/1981	Satellite reached GEO but did not become operational because of damage caused by shroud during launch.	Atlas SLV 3D	Shroud
12/19/1981	Booster engine gas generator cooling plugged.	Atlas E	Propulsion
6/9/1984	A leak in the Centaur LOX tank, which started at first/second stage separation resulted in a pressure differential across the common tank bulkhead that caused it to collapse before the second burn, stranding the satellite in wrong orbit.	Atlas G	Propulsion
3/26/1987	Vehicle was struck by lightning at T+48s. Resulting electrical transients in guidance system caused vehicle to yaw and lose control, resulting in destruction by range safety officer.	Atlas G (older nomen- clature)	Lightning
4/18/1991	Following ground prechilling, air entered Centaur C–1 engine through stuck check valve and froze in LH22 turbopump and gearbox. At Centaur ignition, engine did not achieve full thrust, causing stage to tumble. Vehicle was destroyed by range safety officer. Fault was not properly diagnosed until flight AC71. (8/22/92 failure)	Atlas I	Propulsion
8/22/1992	Following ground prechilling, air entered Centaur C-1 engine through stuck check valve and froze in LH22 turbopump and gearbox. At Centaur ignition, engine did not achieve full thrust, causing stage to tumble. Vehicle was destroyed by range safety officer.	Atlas I	Propulsion

3/25/1993	Inadequately torqued set screw in first-stage engine precision regulator resulted in reduced oxygen flow to gas generator. Engine suffered reduced power and early shutdown, stranding satellite in low orbit.	Atlas I	Propulsion
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### Table 6D-14. Delta Failures

Date	Description	Vehicle Model	Category of Failure
5/3/1986	First-stage engine suffered a premature shutdown at T+71s because of an electrical short. Destroyed by range safety at T+91s.	Delta 3914	Propulsion
8/5/1995	One of the air-ignited SRM GEMs failed to separate, because of overheated explosive lines in the separation system. The extra mass caused the launch vehicle to deliver the payload to a lower than planned orbit.	Delta 7925 (II)	Solid Propulsion
1/17/1997	At T+12s the No. 2 GEM strap-on motor suffered a structural failure of its composite case (not a motor burn through), resulting in a long vertical crack along the side of the motor. The vehicle self-destructed.	Delta 7925 (II)	Solid Propulsion
8/27/1998	At T+55s the rocket began a normal 4 Hz roll oscillation. Because the control software design had not accounted for the oscillation, the vehicle used up all the hydraulic fluid in the strap-on booster nozzle TVC system attempting to correct the roll. Once the hydraulic fluid was exhausted, attitude control was lost, the vehicle pitched over at T+72s and began to break up because of aerodynamic forces, cauing auto-destruct.	Delta 8930 (III)	Software
5/5/1999	Second-stage RL—10 engine shut down immediately after start of second burn due to structural failure of the combustion chamber, stranding spacecraft in low orbit. Structural failure occurred because of poor brazing process in combustion chamber fabrication.	Delta 8930 (III)	Propulsion

Table 6D-15. H-II Failures

Date	Description	Vehicle Model	Category of Failure
11/15/1999	Vehicle went out of the planned flight path due to the abnormal stoppage of the combustion of the 1st stage engine, and the command for destruction was sent. (faulty turbine—actual failure cause not found in literature)	H-II	Propulsion
11/29/2003	A hot gas leak on the SRB-A motor destroyed its separation system. The strap-on did not separate as planned, and the weight of the spent motor prevented the vehicle from achieving its planned velocity.	Н-ПА	Separation system (boost)

# Table 6D-16. LM Failures

Date	Description	Vehicle Model	Category of Failure
1/29/1984	Third stage failed 4 sec after restart for GTO insertion of the satellite, because of incorrect mixture ratio in the engine gas generator, which caused high temperatures and burned out the turbine shell.	LM3	Propulsion
12/28/1991	Third-stage shut down too early during the second burn for GTO insertion, because of loss of pressure in the high-pressure helium gas supply used for engine control.	LM3	Propulsion
12/21/1992	At T+48s at an altitude of 7,000 m an explosion occurred in the payload fairing, destroying both the payload and the fairing. The payload was scattered downrange, but the rest of the launch vehicle went on to achieve orbit with the intended accuracy.	LM2E	Shroud
1/25/1995	At T+51s an explosion destroyed the forward portion of the vehicle. Six people were killed by falling debris. Hughes concluded that the fairing longitudinal split line opened because of high aerodynamic loads from buffeting and wind shear. As the fairing collapsed, it damaged the spacecraft propellant tanks, causing the fire that destroyed the spacecraft and forward end of the launch vehicle.	LM2E	Shroud
2/14/1996	The vehicle pitched over immediately after liftoff, impacting and exploding at T+22s near a village close to the launch site. At least six people were killed. The fault was traced to a lack of output from the power module for the servo-loop in the follow-up frame of the inertial platform. This caused a faulty inertial reference, which	LM3B	GN&C

	made the launch vehicle steer incorrectly.		
8/18/1996	The third-stage engine shut down roughly 40 sec earlier than planned because of a fire in the LH22 injector of the gas generator. Insufficient purging had permitted oxygen to freeze in the gas generator during flight.	LM3	Propulsion

## Table 6D-17. Pegasus Failures

Date	Description	Vehicle Model	Category of Failure
7/17/1991	Malfunction of pyrotechnic separation system caused the launch vehicle to veer off course following first stage separation. Course corrections during subsequent stage burns allowed launch vehicle to reach orbit, but at a much lower altitude than planned.	Pegasus H	Separation System (boost)
6/27/1994	At T+39s vehicle lost control and was destroyed by range safety after first stage burn out. Fault was traced to improper aerodynamics model used in control system autopilot design.	Pegasus XL	Software
6/22/1995	The inter-stage ring between the first and second stages failed to separate, constraining the second-stage nozzle gimbal and reducing control authority; the vehicle began to tumble out of control during second-stage flight, and was destroyed by the range safety officer.	Pegasus XL	Separation System (boost)
11/4/1996	Satellites were delivered to the correct orbit, but the launch vehicle separation system failed to deploy them. Failure to separate was due to a rapid decrease in voltage from the transient battery prior to the payload separation pyro event. A defective batery exposed to flight staging environments most likely was the failure mechanism.	Pegasus XL	Separation System (payload)

## **Table 6D-18. Proton Failures**

Date	Description	Vehicle Model	Category of Failure
7/22/1982	First-stage engine number 5 suffered failure of hydraulic gimbal actuator because of dynamic excitation at T+45s. Automatic flight shutdown commanded.	Proton K DM	Propulsion
12/24/1982	Second-stage failure due to high frequency vibration.	Proton K	Structural

		DM	
11/29/1986	Second-stage failure.	Proton K	Unknown
1/30/1987	Fourth stage failed to start because of control system component failure.	Proton K DM	GN&C
4/24/1987	Fourth stage shut down early and failed to restart. Failure occurred in control system because of manufacturing defect in instrument.	Proton K DM	GN&C
1/18/1988	Third-stage engine failure caused by destruction of fuel line leading to mixer.	Proton K DM	Propulsion
2/17/1988	Fourth-stage engine failure because of high combustion chamber temperatures caused by foreign particles from propellant tank.	Proton K DM	Propulsion
8/9/1990	Second-stage engine shut off because of termination of oxidizer supply due to fuel line being clogged by a wiping rag.	Proton K DM	Propulsion
5/27/1993	Second- and third-stage engines suffered multiple burn throughs of combustion chambers because of propellant contamination.	Proton K DM	Propulsion
2/19/1996	Block DM-2 stage failed at ignition for second burn. Suspected causes were failure of a tube joint, which could cause a propellant leak, or possible contamination of hypergolic start system.	Proton K DM	Propulsion
11/16/1996	Block D-2 fourth-stage kick engine failed to reignite to boost spacecraft into desired transfer orbit; injection burn did not propel spacecraft out of Earth orbit. Spacecraft and upper stage reentered after a few hours.	Proton K DM	Propulsion
12/24/1997	Block DM shut down early because of improperly coated turbopump seal, leaving spacecraft in high-inclination geosynchrononous transfer orbit. Customer declared spacecraft a total loss and collected insurance payment.	Proton K DM	Propulsion
7/5/1999	Second-stage engine failure due to faulty weld which gave way and triggered an explosion during flight.	Proton K DM	Propulsion
10/27/1999	Second-stage engine failure triggered an explosion during flight.	Proton K DM	Propulsion
11/25/2002	A failed valve caused excess fuel to collect in the Block DM main engine during the parking orbit coast after the first burn. The engine was destroyed, and the Astra–1K satellite was separated into the parking orbit.	Proton K DM	Propulsion

Table 6D-19. Molniya and Soyuz Failures

Date	Description	Vehicle Model	Category of Failure
4/18/1980	Fourth-stage engine ignited but shut down prematurely.	Molniya M	Propulsion
3/28/1981	Failure to orbit.	Soyuz U	Unknown
9/11/1981	Fourth-stage engine ignited exploding part way through the burn.	Molniya M	Propulsion
5/15/1982	Failure to orbit.	Soyuz U	Unknown
6/12/1982	Failure to orbit.	Soyuz U	Unknown
12/8/1982	Fourth-stage engine ignited exploding part way through the burn.	Molniya M	Propulsion
7/8/1983	Upper stage exploded.	Molniya 2BL	Unknown
3/26/1986	Failure to orbit.	Soyuz U	Unknown
10/3/1986	Fourth-stage engine ignited, but shut down prematurely.	Molniya	Propulsion
6/18/1987	Failure to orbit.	Soyuz U	Unknown
7/9/1988	Failure to orbit.	Soyuz U	Unknown
7/27/1988	Failure to orbit.	Soyuz U	Unknown
11/11/1988	Failure to orbit.	Soyuz U	Unknown
4/3/1990	Failure to orbit.	Soyuz U	Unknown
6/21/1990	Fourth stage failure.	Molniya	Unknown
7/3/1990	Failure to orbit.	Soyuz U	Unknown
4/27/1993	Spacecraft was delivered to planned orbit, but at the intended separation time an explosion occurred in the upper stage, damaging the spacecraft.	Soyuz U	Separation System (payload)
5/14/1996	Payload shroud failed 49 sec into flight. The flight continued until separation of the strap-on boosters, when the vehicle veered off course causing the main engines to automatically shut down and the vehicle to crash.	Soyuz U	Shroud

6/20/1996	Payload shroud failed 50 sec into flight, causing flight termination and impact 8 km from pad. Investigation concluded that this and the previous failure were due to a defective manufacturing process that resulted in weaker glue bonds between layers of glass-reinforced-plastic in the fairing structure.	Soyuz U	Shroud
10/15/2002	Contamination in the rocket's hydrogen peroxide system caused an engine failure 8–9 sec after liftoff. The booster broke away from the vehicle and crashed near the pad. The safety system shut down the remaining engines at T+29s and the vehicle crashed about 1 km from the pad. One soldier was killed and eight others injured by a subsequent explosion when they were sent to extinguish fires.	Soyuz U	Propulsion

## Table 6D-20. Taurus Failures

Date	Description	Vehicle Model	Category of Failure
9/26/2001	A problem with the second stage caused the Taurus to release the satellites in a lower orbit making them unusable.	Taurus	Unknown

## Table 6D-21. Titan Failures

Date	Description	Vehicle Model	Category of Failure
8/28/1985	Destroyed by range safety officer after first-stage propellant feed system leaked and one engine turbopump failed.	Titan 34D	Propulsion
4/18/1986	Ignition pressure caused SRM case insulation to debond between segments, resulting in case burn through and vehicle explosion 8.5 sec after liftoff, damaging launch pad.	Titan 34D	Solid Propulsion
9/2/1988	Damage to Transtage resulted in hydrazine and helium leaks, which prevented ignition of the second burn for injection into GEO.	Titan 34D	Propulsion
3/14/1990	Second stage reached correct orbit but failed to deploy payload because of incorrect interface wiring; Intelsat 603 separated itself from its kick stage and was rescued and reboosted by astronauts on STS-49.	Titan III	Separation System (payload)
8/2/1993	A radial cut in the propellant of one SRM segment during repairs permitted combustion propagation to the motor	Titan IVA	Solid Propulsion

	case, causing the motor to explode 101 sec after liftoff, destroying the vehicle.		
8/12/1998	Intermittent power shorts, possibly caused by a damaged cable, caused the inertial guidance unit to lose its reference attitude and begin generating improper steering commands; vehicle pitched over 40 sec into flight and was destroyed by aerodynamic forces.	Titan IVA	GN&C
4/30/1999	Centaur attitude control propellant depleted prematurely, causing deployment of payload in incorrect low orbit. Fault traced to incorrect roll rate parameter in Centaur flight software, a decimal point misplaced by human error during manual data entry.	Titan IVB	Software

## Table 6D-22. Tsiklon Failures

Date	Description	Vehicle Model	Category of Failure
1/23/1981	Payload shroud failed to separate.	Tsiklon 3	Shroud
11/27/1984	Stage 3 failed to reignite.	Tsiklon 3	Unknown
10/15/1986	First stage failure.	Tsiklon 3	Unknown
6/6/1989	Third stage failed to restart.	Tsiklon 3	Unknown
5/25/1994	Control system aboard the satellite failed to correctly transmit the command for the second and third stages to separate.	Tsiklon 3	GN&C
12/26/2000	Booster failed in third stage resulting in the loss of six satellites.	Tsiklon 3	Unknown

### Table 6D-23. Zenit Failures

Date	Description	Vehicle Model	Category of Failure
10/4/1990	First-stage failure RD–171 caught fire. Failure attributed to contamination by traces of lubricating oil in the oxygen manifold after testing.	Zenit 2	Propulsion
8/30/1991	Second-stage RD–120 engine oxidizer turbopump caught fire causing failure to reach orbit.	Zenit 2	Propulsion

2/5/1992	Second-stage RD–120 engine oxidizer turbopump caught fire causing failure to reach orbit. Failures traced to material change in oxidizer turbopump component, which caused sparks during engine start. Problem had not been detected following previous flight because normal engine tests are performed horizontally, but failure only occurred in the vertical position as in flight.	Zenit 2	Propulsion
5/20/1997	Structural failure of strut in first-stage RD–171 engine occurred causing failure of the engine and resulting loss of vehicle. Failure traced to undetected damage suffered due to high vibration during engine acceptance test.	Zenit 2	Propulsion
9/9/1998	Two of three redundant guidance computer channels became unsynchronized and were voted out. As designed, remaining channel ordered thrust termination for flight safety reasons, causing vehicle to impact in Siberia.	Zenit 2	GN&C
3/12/2000	A ground software error resulted from mission-specific changes for this launch that required a 1 sec launch window. Software failed to command a valve in the second-stage pneumatic system to close after liftoff. The pneumatic system is used for several functions, including operation and actuation of the RD–8 steering engine. The system lost more the 60% of its pressure in flight, reducing the control capability of the engine, and ultimately causing loss of attitude control. The automatic flight termination system shutdown the vehicle 8 minutes after liftoff.	Zenit 3SL	Software

**Figure 6D-10** breaks out the 92 unmanned launch vehicle failures by failure grouping for each year.

## Unmanned Vehicles (1980-2005) Worldwide 1,930 Launches, 92 Failures

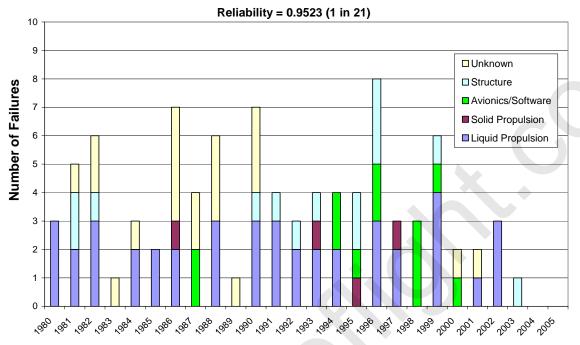


Figure 6D-10. Failure Breakdown by Year for Unmanned Launch Vehicles (1980–2005)

#### **6D.1.2.1.2 Manned Historic Data 1980–2005**

The three manned launch vehicles considered in this assessment were Long March, Soyuz, and Space Shuttle.

During the time period of January 1980 through May 2005, there were 174 manned launch attempts of these vehicles with 3 failures. Only failures which were attributable to the launch vehicle and caused complete loss of mission objectives were counted.

The Space Shuttle had two of the three failures, while Soyuz had the other. The Long March only had one flight during this time period. **Table 6D-24** shows, for each launch vehicle, the number of attempts, number of failures, percentage of the total failures, and demonstrated reliability during this time period.

Launch Vehicle	Number of attempts		% of total failures	Reliability	
Long March	1	0	0.0%	1	
Soyuz	60	1	33.3%	0.9833	

Table 6D-24. Manned LV Historic Data 1980–2005

Space Shuttle	113	2	66.7%	0.9823
Total	174	3	100%	0.9828

The three failures were then binned into subsystem groups:

- Solid Propulsion,
- Structure (TPS), and
- Unknown.

The percentages for each failure grouping are shown in Table 6D-25.

Table 6D-25. Manned LV Historic Data 1980-2005

	Number of failures	% of total failures	Rate
Solid Propulsion	1	33.3%	1 in 174
Structure (TPS)	1	33.3%	1 in 174
Unknown	1	33.3%	1 in 174
Total	3		1 in 58

The failure descriptions for each vehicle are provided in Table 6D-26.

**Table 6D-26. Manned LV Failures** 

Date	Description	Vehicle Model	Category of Failure		
9/26/1983	Fire started at the base of the vehicle, but the cosmonauts were able to escape before the vehicle exploded by firing the escape tower.	Soyuz U	Unknown		
1/28/1986	At T+70s, a burn through of a SRB O-ring resulted in the rupturing of the external tank and the subsequent breakup of the orbiter.	Space Shuttle	Solid Propulsion		
1/16/2003	A piece of foam falling from the external tank damaged the thermal protection system on the leading edge of the orbiter's left wing. During reentry, hot gas penetrated the wing, causing destruction of the orbiter.	Space Shuttle	TPS		

**Figure 6D-11** lists the three manned launch vehicle failures by failure grouping for each year.

#### Manned Vehicles (Worldwide) 1980-2005 174 Launches, 3 Failures Reliability = 0.9828 (1 in 58)

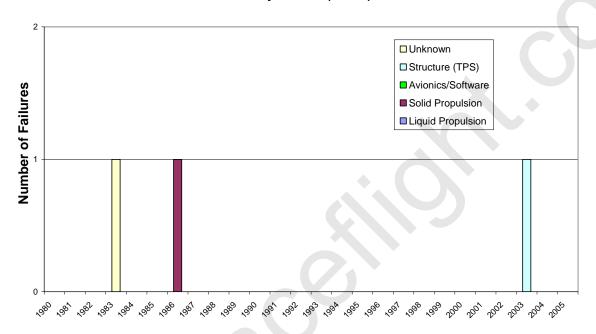


Figure 6D-11. Failure Breakdown by Year for Manned Launch Vehicles (1980–2005)

#### **6D.1.2.2** Conclusions

An overview of launch vehicle historical data has been presented in this section. The worldwide unmanned data from January 1980 though May 2005 clearly shows that liquid propulsion failures dominated the total launch vehicle failures making up 46 percent of the total (42 out of 92 total failures). A significant percentage of failures are unknown at 22 percent (20 out of 92 total failures). The majority of the unknown failures (80 percent, or 16 out of 20) occurred on the Soyuz/Molniya and Tsiklon/Dnepr vehicles. Stuctural failures make up 15 percent (14 out 92) of the total failures. Structures included shroud and separation system failures in addition to general structural failures. The avionics failures constitute 13 percent (12 out of 92) of the total failures during this time period, with the majority (67 percent, or 8 out of 12) occurring on non-U.S. launch vehicles. Solid propulsion failures make up the least amount of the total at only 4 percent (4 out of 92 total failures).

There is significantly less historical data available for worldwide manned launch vehicles during the January 1980 through May 2005 time period. The three failures that occurred were due to solid propulsion, structure (TPS), and an unknown cause. Due to such a small

number of failures, no conclusions can be drawn about which subsystem failures will likely contribute more to actual manned launch vehicle's risk. Probabilistic risk assessment methods are necessary.

Early launch vehicle history was discussed relative to the Aerospace Corporation's report from 1987, which included data from 1957–1987.

#### 6D.2 Ground Rules and Assumptions

Safety and reliability FOMs, LOM, and LOC, were estimated only for the ascent phase of the mission. The ascent phase was defined as being the duration between main engine ignition and orbital insertion; no ground safety or reliability was assessed.

#### **6D.2.1 Crew Safety**

Due to the focus of this study and the time constraints, detailed abort assessments were not carried out, nor were detailed Crew Escape System (CES) subsystem designs assessed or reliability estimates performed. Instead, crew safety was estimated using top-level assumptions as to the abort effectiveness of a crew escape system. For catastrophic failures of a crewed launch vehicle the CES was assumed to be 80 percent effective, while for non-catastrophic failures that necessitated an abort (such as a premature engine shutdown on a vehicle without engine-out capability) the CES was assumed to be 90 percent effective. The LOC figure of merit was mitigated by the success probability of the CES.

#### 6D.3 Input Parameters for Each Concept

Data input to the reliability analysis includes historic launch vehicle data (see **Appendix 6D.1.2**); trajectory and vehicle configuration data; and system reliability data (see **Section 6.8** of the **ESAS Final Report**).

#### **6D.4** LV 13.1 Sensitivity Analysis

There exists large uncertainties in the air-start failure probability ( $P_{SU}$ ) of the SSME second stage engine in LV 13.1, and the Crew Escape Effectiveness Factors applied to DCF (CEEF<sub>DCF</sub>) events for this vehicle. (DCF events for this vehicle are engine shutdown and air-start failure.) An analysis of LV 13.1 Mean LOC sensitivity to variation in  $P_{SU}$  and CEEF<sub>DCF</sub> follows. The analysis shows both graphically and in tabular data how the LOC estimates may vary from 1 in 900 to 1 in 2,908 within the bounds of reasonable selections of CEEF and air-start reliability. **Table 6D-27** is tabular combinations of  $P_{SU}$  and CEEF<sub>DCF</sub>, with the resulting Mean LOC for LV 13.1, and **Figure 6D-12** is a surface plot of the same data.

Table 6D-27. Tabular Combination of P<sub>SU</sub> and CEEF<sub>DCF</sub>

	CEEF <sub>DCF</sub>										
P <sub>su</sub>	80.00%	81.00%	82.00%	83.00%	84.00%	85.00%	86.00%	87.00%	88.00%	89.00%	90.00%
1 in 300	1 in 908	1 in 947	1 in 989	1 in 1,036	1 in 1,087	1 in 1,143	1 in 1,205	1 in 1,275	1 in 1,353	1 in 1,441	1 in 1,542
1 in 400	1 in 1,070	1 in 1,114	1 in 1,162	1 in 1,214	1 in 1,271	1 in 1,333	1 in 1,402	1 in 1,479	1 in 1,564	1 in 1,660	1 in 1,769
1 in 500	1 in 1,198	1 in 1,246	1 in 1,297	1 in 1,353	1 in 1,414	1 in 1,481	1 in 1,555	1 in 1,636	1 in 1,726	1 in 1,827	1 in 1,941
1 in 600	1 in 1,302	1 in 1,352	1 in 1,407	1 in 1,466	1 in 1,530	1 in 1,600	1 in 1,677	1 in 1,761	1 in 1,855	1 in 1,958	1 in 2,075
1 in 700	1 in 1,388	1 in 1,440	1 in 1,497	1 in 1,558	1 in 1,624	1 in 1,697	1 in 1,776	1 in 1,863	1 in 1,958	1 in 2,064	1 in 2,183
1 in 800	1 in 1,460	1 in 1,514	1 in 1,573	1 in 1,635	1 in 1,704	1 in 1,778	1 in 1,858	1 in 1,947	1 in 2,044	1 in 2,152	1 in 2,271
1 in 900	1 in 1,522	1 in 1,577	1 in 1,637	1 in 1,701	1 in 1,771	1 in 1,846	1 in 1,928	1 in 2,018	1 in 2,116	1 in 2,225	1 in 2,345
1 in 1,000	1 in 1,575	1 in 1,632	1 in 1,692	1 in 1,758	1 in 1,828	1 in 1,905	1 in 1,988	1 in 2,078	1 in 2,178	1 in 2,287	1 in 2,408
1 in 1,100	1 in 1,622	1 in 1,679	1 in 1,740	1 in 1,807	1 in 1,878	1 in 1,955	1 in 2,039	1 in 2,131	1 in 2,231	1 in 2,340	1 in 2,462
1 in 1,200	1 in 1,663	1 in 1,721	1 in 1,783	1 in 1,850	1 in 1,922	1 in 2,000	1 in 2,084	1 in 2,176	1 in 2,277	1 in 2,387	1 in 2,508
1 in 1,300	1 in 1,699	1 in 1,757	1 in 1,820	1 in 1,888	1 in 1,960	1 in 2,039	1 in 2,124	1 in 2,216	1 in 2,317	1 in 2,428	1 in 2,549
1 in 1,400	1 in 1,731	1 in 1,790	1 in 1,854	1 in 1,922	1 in 1,995	1 in 2,074	1 in 2,159	1 in 2,252	1 in 2,353	1 in 2,464	1 in 2,586
1 in 1,500	1 in 1,760	1 in 1,820	1 in 1,884	1 in 1,952	1 in 2,026	1 in 2,105	1 in 2,191	1 in 2,284	1 in 2,385	1 in 2,496	1 in 2,618
1 in 1,600	1 in 1,786	1 in 1,846	1 in 1,911	1 in 1,979	1 in 2,053	1 in 2,133	1 in 2,219	1 in 2,313	1 in 2,414	1 in 2,525	1 in 2,647
1 in 1,700	1 in 1,810	1 in 1,870	1 in 1,935	1 in 2,004	1 in 2,078	1 in 2,158	1 in 2,245	1 in 2,338	1 in 2,440	1 in 2,551	1 in 2,673
1 in 1,800	1 in 1,832	1 in 1,892	1 in 1,957	1 in 2,027	1 in 2,101	1 in 2,182	1 in 2,268	1 in 2,362	1 in 2,464	1 in 2,575	1 in 2,696
1 in 1,900	1 in 1,852	1 in 1,913	1 in 1,978	1 in 2,047	1 in 2,122	1 in 2,203	1 in 2,289	1 in 2,383	1 in 2,485	1 in 2,596	1 in 2,718
1 in 2,000	1 in 1,870	1 in 1,931	1 in 1,996	1 in 2,066	1 in 2,141	1 in 2,222	1 in 2,309	1 in 2,403	1 in 2,505	1 in 2,616	1 in 2,737
1 in 2,100	1 in 1,887	1 in 1,948	1 in 2,014	1 in 2,084	1 in 2,159	1 in 2,240	1 in 2,327	1 in 2,421	1 in 2,523	1 in 2,634	1 in 2,755
1 in 2,200	1 in 1,902	1 in 1,964	1 in 2,030	1 in 2,100	1 in 2,175	1 in 2,256	1 in 2,343	1 in 2,438	1 in 2,540	1 in 2,651	1 in 2,772
1 in 2,300	1 in 1,917	1 in 1,978	1 in 2,044	1 in 2,115	1 in 2,190	1 in 2,271	1 in 2,359	1 in 2,453	1 in 2,555	1 in 2,666	1 in 2,787
1 in 2,400	1 in 1,930	1 in 1,992	1 in 2,058	1 in 2,129	1 in 2,204	1 in 2,285	1 in 2,373	1 in 2,467	1 in 2,569	1 in 2,680	1 in 2,801
1 in 2,500	1 in 1,942	1 in 2,005	1 in 2,071	1 in 2,142	1 in 2,217	1 in 2,299	1 in 2,386	1 in 2,480	1 in 2,583	1 in 2,694	1 in 2,814
1 in 2,600	1 in 1,954	1 in 2,016	1 in 2,083	1 in 2,154	1 in 2,229	1 in 2,311	1 in 2,398	1 in 2,493	1 in 2,595	1 in 2,706	1 in 2,827
1 in 2,700	1 in 1,965	1 in 2,027	1 in 2,094	1 in 2,165	1 in 2,241	1 in 2,322	1 in 2,410	1 in 2,504	1 in 2,606	1 in 2,717	1 in 2,838
1 in 2,800	1 in 1,975	1 in 2,038	1 in 2,104	1 in 2,175	1 in 2,251	1 in 2,333	1 in 2,421	1 in 2,515	1 in 2,617	1 in 2,728	1 in 2,849
1 in 2,900	1 in 1,985	1 in 2,048	1 in 2,114	1 in 2,185	1 in 2,261	1 in 2,343	1 in 2,431	1 in 2,525	1 in 2,627	1 in 2,738	1 in 2,859
1 in 3,000	1 in 1,994	1 in 2,057	1 in 2,124	1 in 2,195	1 in 2,271	1 in 2,353	1 in 2,440	1 in 2,535	1 in 2,637	1 in 2,748	1 in 2,868
1 in 3,100	1 in 2,003	1 in 2,065	1 in 2,132	1 in 2,204	1 in 2,280	1 in 2,362	1 in 2,449	1 in 2,544	1 in 2,646	1 in 2,757	1 in 2,877
1 in 3,200	1 in 2,011	1 in 2,074	1 in 2,141	1 in 2,212	1 in 2,288	1 in 2,370	1 in 2,458	1 in 2,552	1 in 2,655	1 in 2,765	1 in 2,885
1 in 3,300	1 in 2,018	1 in 2,081	1 in 2,148	1 in 2,220	1 in 2,296	1 in 2,378	1 in 2,466	1 in 2,560	1 in 2,663	1 in 2,773	1 in 2,893
1 in 3,400	1 in 2,026	1 in 2,089	1 in 2,156	1 in 2,227	1 in 2,304	1 in 2,386	1 in 2,473	1 in 2,568	1 in 2,670	1 in 2,781	1 in 2,901
1 in 3,500	1 in 2,033	1 in 2,096	1 in 2,163	1 in 2,234	1 in 2,311	1 in 2,393	1 in 2,481	1 in 2,575	1 in 2,677	1 in 2,788	1 in 2,908

The tabular data shows that with variations of  $P_{SU}$  from 1 in 300 to 1 in 3,500, and CEEF<sub>DCF</sub> from 80% to 90%, Mean LOC for LV 13.1 can vary from 1 in 908 to 1 in 2,908. The red cell in **Table 6D-27** approximates the current LV 13.1 estimate. It is notable that with this  $P_{SU}$ , a Mean LOC of almost 1 in 3,000 is possible with the highest CEEF. Also significant is that even when a conservative  $P_{SU}$ , similar to the current SSME ground start failure probability (1 in 661), is selected, a Mean LOC of 1 in 2,000 may still be achieved with a CEEF<sub>DCF</sub> of 89%.

The blue cells in the table represent the selections of  $P_{SU}$  and  $CEEF_{DCF}$  that most closely provide a Mean LOC of 1 in 2,000.

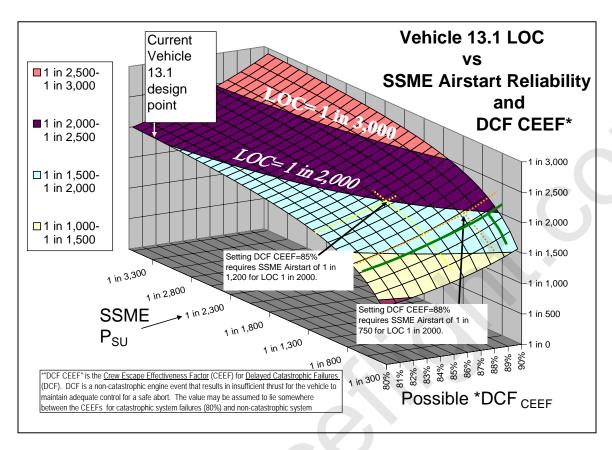


Figure 6D-12. Combination of P<sub>SU</sub> and CEEF<sub>DCF</sub>

**Figure 6D-12** graphically depicts the data in **Table 6D-28**. This plot shows that for lower  $P_{SU}$  and  $CEEF_{DCF}$ , Mean LOC is responds rapidly to improvements in both. However, after moderate improvements in  $P_{SU}$ , further improvements yield diminishing returns. (This is due to engine shutdown becoming the dominant risk contributor as airstart failure becomes more reliable.) Because  $CEEF_{DCF}$  mitigates both airstart failures and engine shutdowns for LOC risk, increases in  $CEEF_{DCF}$  positively affect Mean LOC even when  $P_{SU}$  is no longer the dominant risk contributor.

**Figure 6D-13** depicts a specific test case in which a very conservative value for the SSME air-start failure probability was selected, combined with more optimistic CEEF values for Command Module failures (CEEF<sub>CM</sub>) and Delayed Catastrophic Failures (CEEF<sub>DCF</sub>).

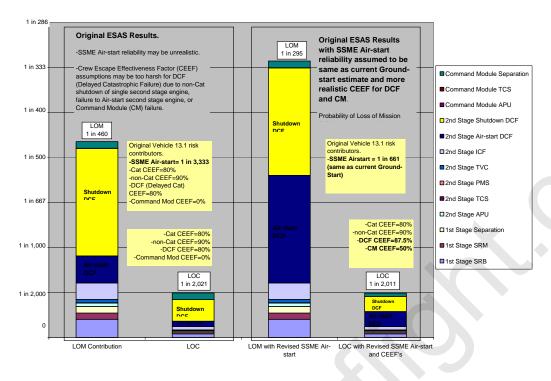


Figure 6D-13. Specific Test Case

As more details for the planned SSME air-start modifications become available, the uncertainty in the SSME air-start reliability model can be reduced and vehicle reliability predictions can become more certain. The simplistic CEEF model necessarily employed in ESAS for analyzing a large number of vehicles in a brief period should be replaced by a dynamic abort reliability analysis. This analysis should combine per-second subsystem/failure mode probabilities with dynamic of abortability and survivability models (using mission-specific trajectory/aerodynamic data along with detailed vehicle characteristics) to yield accurate LOC predictions. This type of analysis may also yield valuable feedback for vehicle designers by highlighting critical vulnerabilities.

#### **6D.5** Future Work

The ESAS reliability model applied to launch vehicles for this study was a general model, intended to measure the reliability of a wide variety of vehicle configurations using a repeatable standard. Specific launch vehicle analysis, tailored to a single vehicle design, would more accurately capture the spectrum of engine and subsystem failure severities and the temporal distributions of failure probabilities in the mission timeline. This analysis should combine per-second subsystem/failure mode probabilities with dynamic of abortability and survivability models (using mission-specific trajectory/aerodynamic data along with detailed vehicle characteristics) to yield accurate LOC predictions. This type of analysis may also yield valuable feedback for vehicle designers by highlighting critical vulnerabilities.

Based on the sensitivity of the LOC FOM to the abort effectiveness, it is recommended that follow-on studies of the ESAS proposed launch vehicles with greater design resolution use detailed abort modeling rather than point estimates for the reliability of the crew escape systems. Detailed abort effectiveness analysis can be done with existing

tools such as the Dynamic Abort Risk Evaluator (DARE), which has been used by the Shuttle Program Office to assist in determining the least risky manner of operating the Space Shuttle if an abort event occurs. DARE has been extended to model the abort risk characteristics of certain types of Shuttle-derived vehicles, and will be developed further to encompass different launch vehicle configurations. A notional presentation of this type of analysis is shown in **Figure 6D-14**.

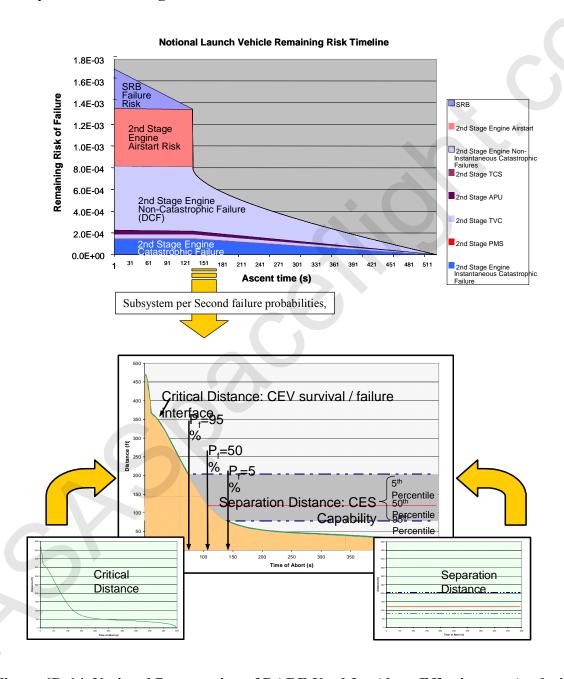


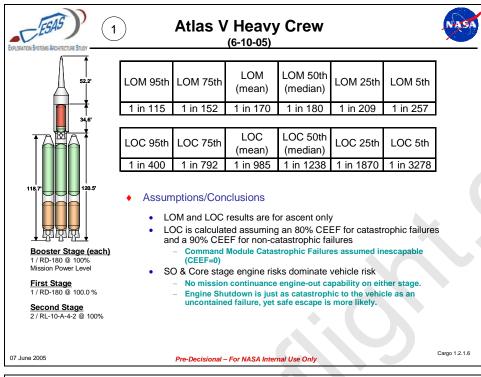
Figure 6D-14. Notional Presentation of DARE Used for Abort Effectiveness Analysis

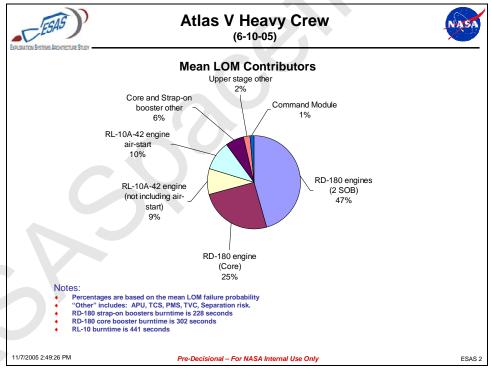
### 6D.6 Results for Each Concept (LOM/LOC/Ranges)

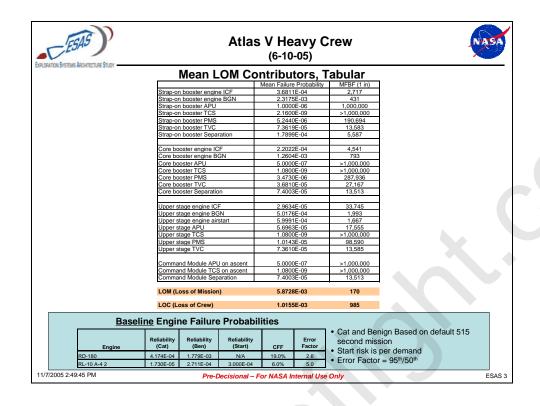
**Table 6D-29** lists all the launch vehicles assessed during this study. Detailed analysis results are included in this section.

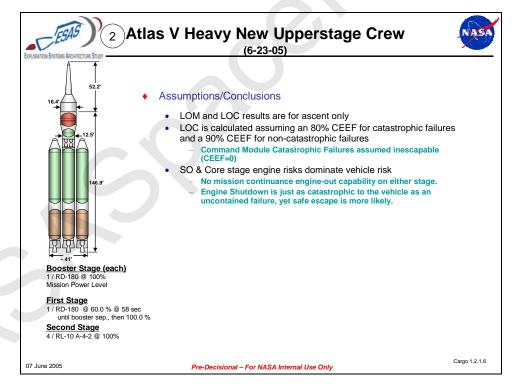
Table 6D-29. Launch Vehicles Assessed

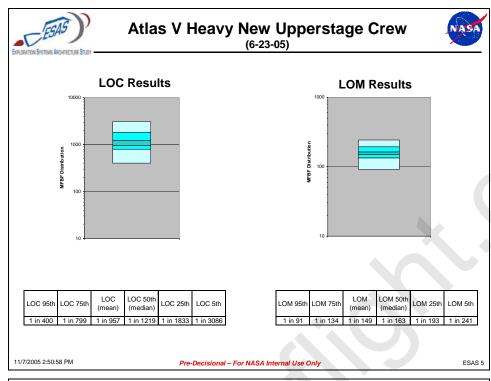
Vehicle ID		LOM Mean	LOC Mean
1	Crew	170	985
2	Crew	149	957
3	Crew	220	1174
4	Crew	172	1100
5	Crew	95	673
5.1	Crew	79	614
6	Cargo	67	N/A
7	Cargo	53	N/A
7.4	Cargo	71	N/A
7.5	Crew	65	536
8	Cargo	110	N/A
9	Crew	134	939
11	Cargo	88	N/A
11.1	Crew	80	612
13.1	Crew	460	2021
14	Crew	444	1958
15	Crew	182	1429
16	Crew	433	1918
19	Crew	296	1359
20	Cargo	173	N/A
21	Cargo	172	N/A
24	Crew	164	1170
25	Cargo	176	N/A
26	Crew	124	915
27	Cargo	133	N/A
27.2	Cargo	124	N/A
27.2	Crew	115	869
27.3	Cargo	92	N/A
28	Cargo	133	N/A
30	Cargo	81	N/A
30.1	Cargo	95	N/A
31	Crew	87	582

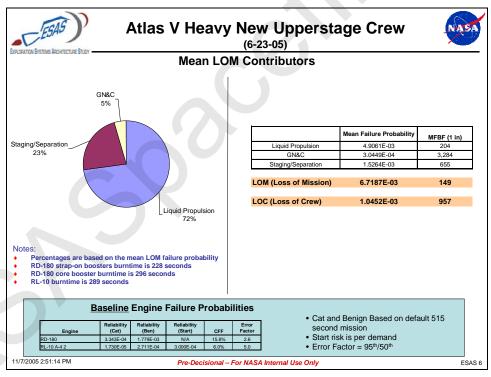


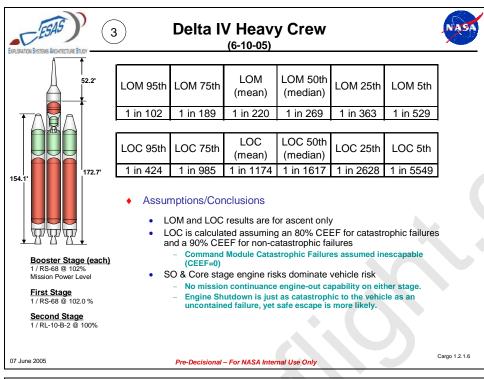


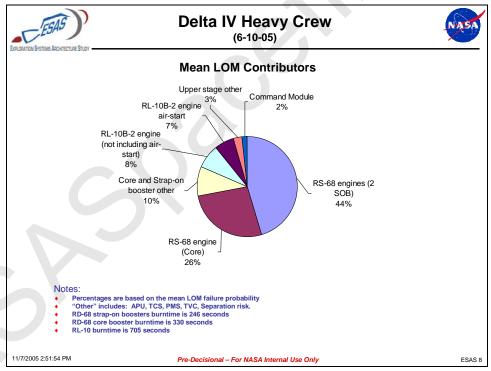














#### **Delta IV Heavy Crew** (6-10-05)



Mean LOM Contributors, Tabular

	Mean Failure Probability	MFBF (1 in)
Strap-on booster engine ICF	1.9290E-04	5,184
Strap-on booster engine BGN	1.8692E-03	535
Strap-on booster APU	3.7720E-05	26,511
Strap-on booster TCS	2.1600E-09	>1,000,000
Strap-on booster PMS	5.6580E-06	176,741
Strap-on booster TVC	7.3619E-05	13,583
Strap-on booster Separation	1.7899E-04	5,587
Core booster engine ICF	1.2939E-04	7,728
Core booster engine BGN	1.0827E-03	924
Core booster APU	2.5300E-05	39,526
Core booster TCS	1.0800E-09	>1,000,000
Core booster PMS	3.7950E-06	263,505
Core booster TVC	3.6810E-05	27,167
Core booster Separation	7.4003E-05	13,513
Upper stage engine ICF	2.3939E-05	41,773
Upper stage engine BGN	3.2054E-04	3,120
Upper stage engine airstart	3.0000E-04	3,333
Upper stage APU	7.9350E-05	12,602
Upper stage TCS	1.0800E-09	>1,000,000
Upper stage PMS	8.1075E-06	123,343
Upper stage TVC	3.6810E-05	27,167
Command Module APU on ascent	5.0000E-07	>1,000,000
Command Module TCS on ascent	1.0800E-09	>1,000,000
Command Module Separation	7.4003E-05	13,513

LOM (Loss of Mission) LOC (Loss of Crew) 8.5153E-04

Baseline Engine Failure Probabilities						
Engine	Reliability (Cat)	Reliability (Ben)	Reliability (Start)	CFF	Error Factor	
	2.019E-04	, , , , , , , , , , , , , , , , , , , ,				

- Cat and Benign Based on default 515 second mission

1174

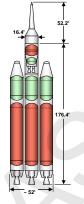
- Start risk is per demand
   Error Factor = 95th/50th

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RS-68 RL-10 B-



## **Delta IV Heavy New Upperstage Crew** (6-10-05)



	LOM 95th	LOM 75th	LOM (mean)	LOM 50th (median)	LOM 25th	LOM 5th
١	1 in 80	1 in 146	1 in 172	1 in 207	1 in 285	1 in 422

LOC 95th	LOC 75th	LOC (mean)	LOC 50th (median)	LOC 25th	LOC 5th
1 in 415	1 in 933	1 in 1100	1 in 1535	1 in 2394	1 in 4427

- Assumptions/Conclusions
  - . LOM and LOC results are for ascent only
  - LOC is calculated assuming an 80% CEEF for catastrophic failures and a 90% CEEF for non-catastrophic failures
    - Command Module Catastrophic Failures assumed inescapable (CEEF=0)
  - SO & Core stage engine risks dominate vehicle risk
    - No mission continuance engine-out capability on either stage.
    - Engine Shutdown is just as catastrophic to the vehicle as an uncontained failure, yet safe escape is more likely.

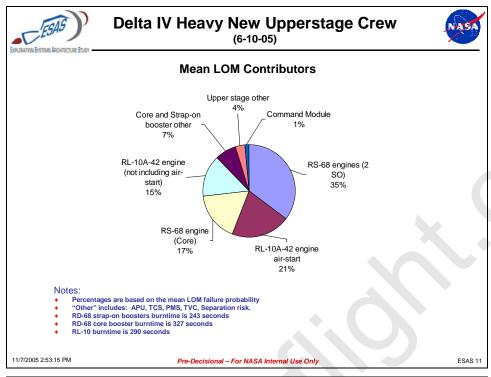
Booster Stage (each)
1 / RS-68 @ 102%
Mission Power Level

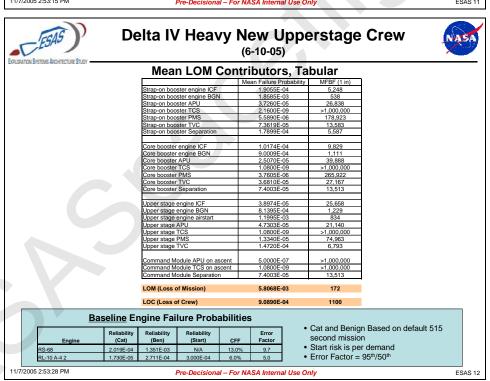
First Stage 1 / RS-68 @ 57.0 % @ 50 sec until booster sep., then 102.0 %

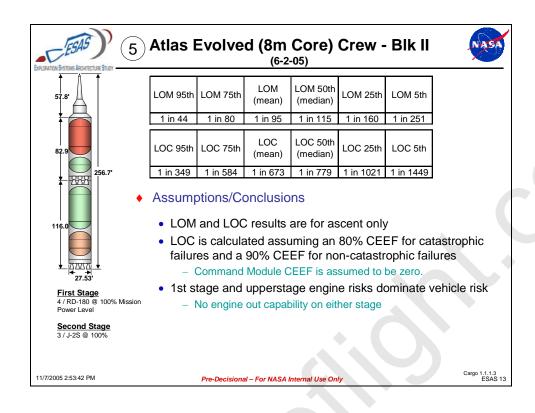
Second Stage 4 / RL-10-A-4-2 @ 100%

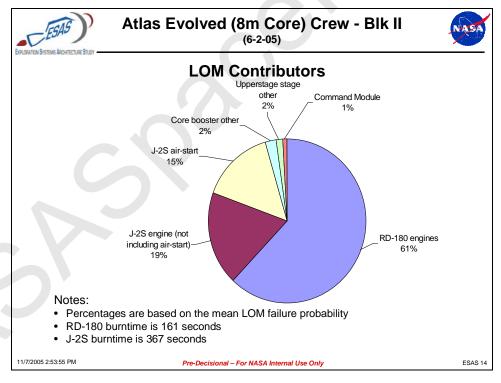
Pre-Decisional – For NASA Internal Use Only

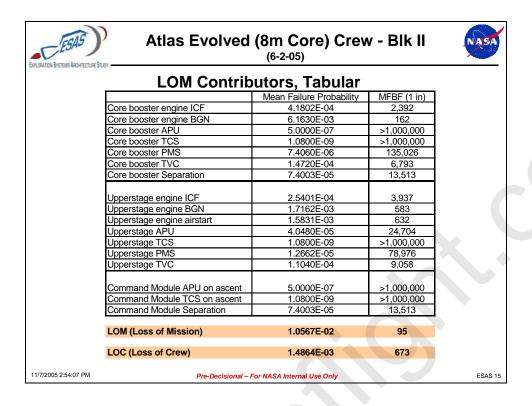
Cargo 1.2.1.6

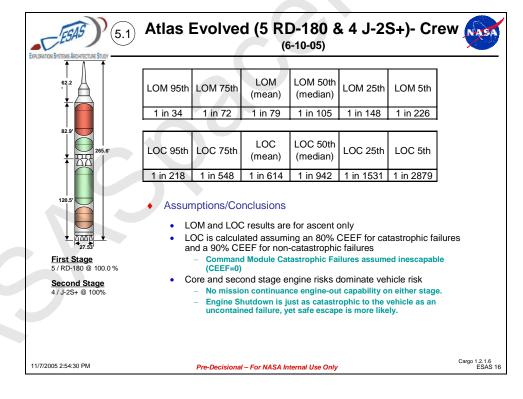


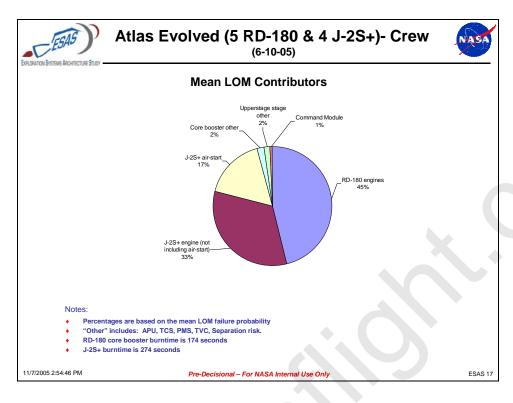




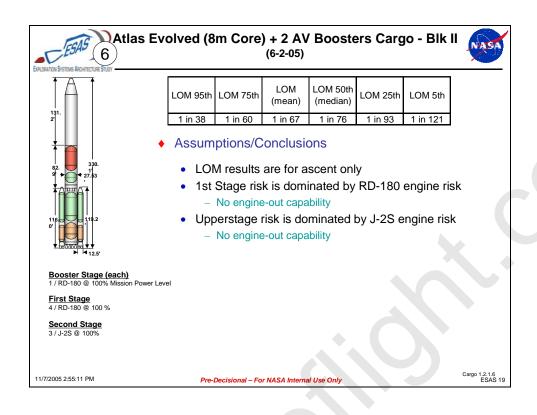


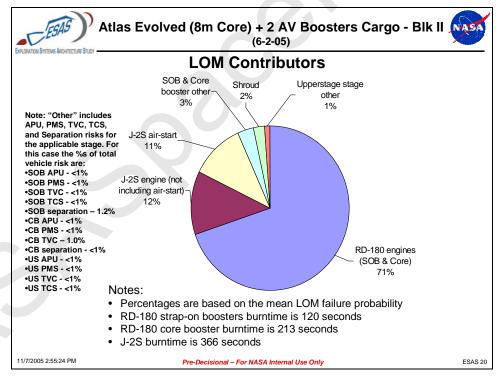


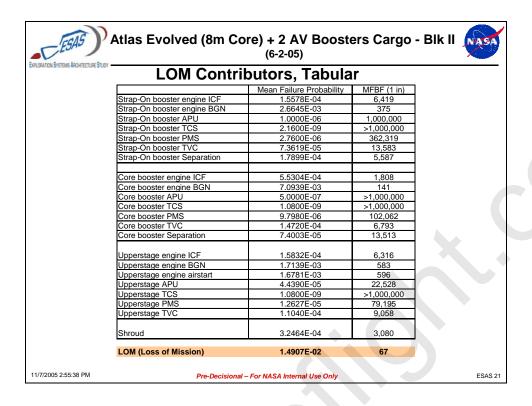


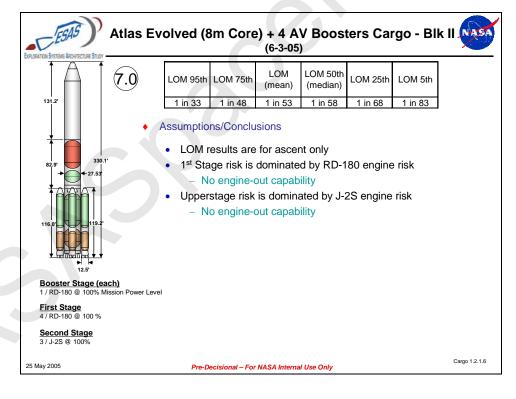


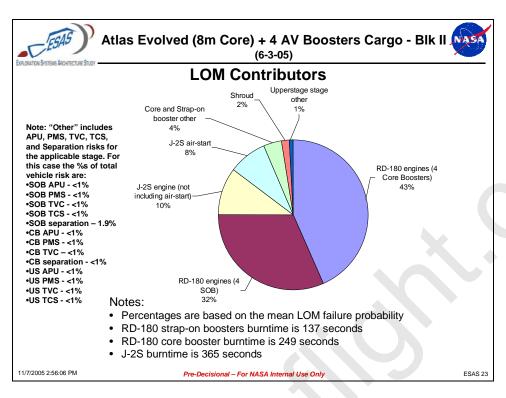


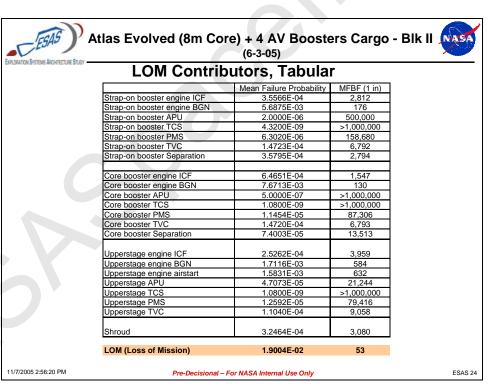


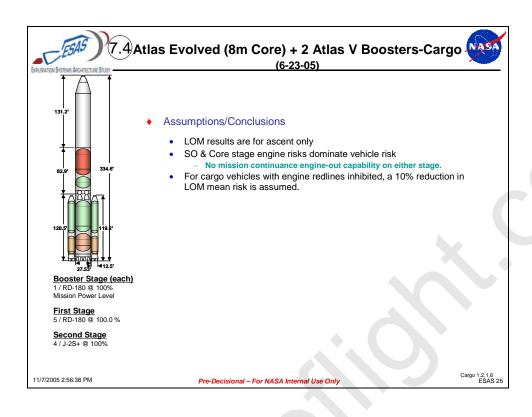


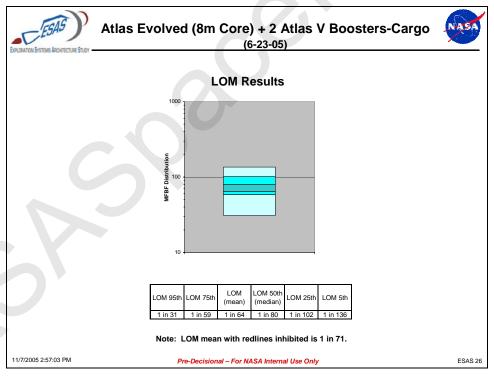


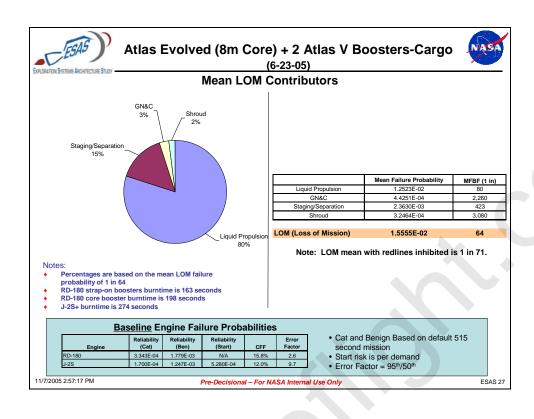


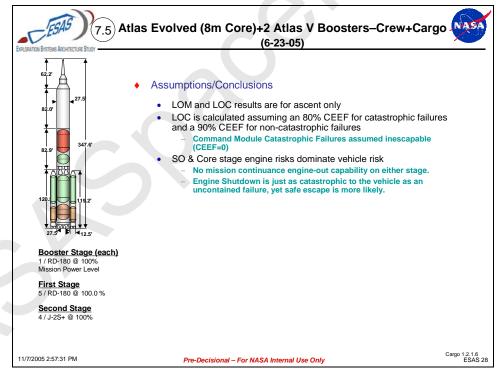


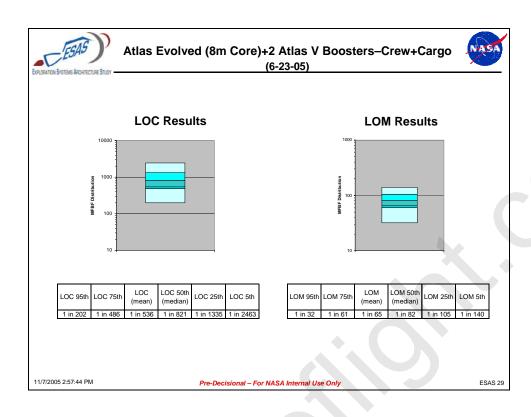


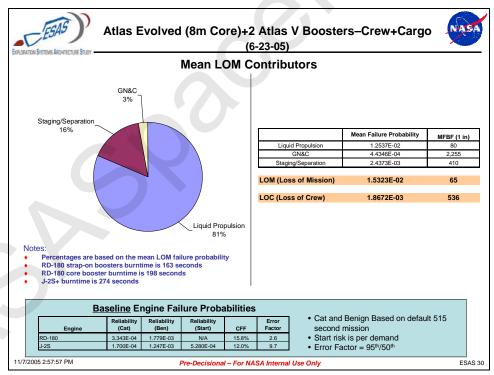


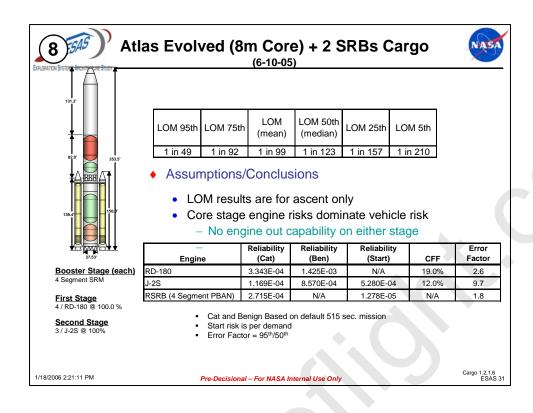


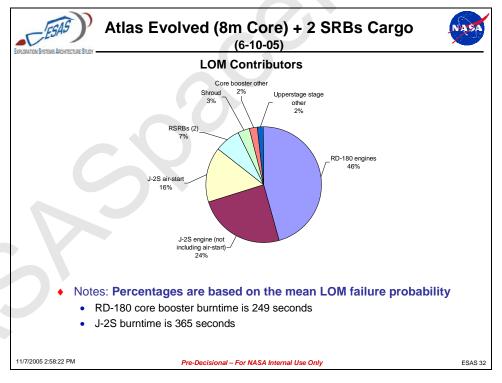














## Atlas Evolved (8m Core) + 2 SRBs Cargo (6-10-05)



LOM Contributors, Tabular

zem cemmatere, rabaia.					
	Mean Failure Probability	MFBF (1 in)			
RSRB (2)	5.6853E-04	1759			
RSRB Separation	1.7899E-04	5587			
Core booster engine ICF	6.4651E-04	1,547			
Core booster engine BGN	3.9790E-03	251			
Core booster APU	5.0000E-07	>1,000,000			
Core booster TCS	1.0800E-09	>1,000,000			
Core booster PMS	1.1454E-05	87,306			
Core booster TVC	1.4720E-04	6,793			
Core booster Separation	7.4003E-05	13,513			
Upperstage engine ICF	2.5262E-04	3,959			
Upperstage engine BGN	2.2025E-03	454			
Upperstage engine airstart	1.5831E-03	632			
Upperstage APU	4.7073E-05	21,244			
Upperstage TCS	1.0800E-09	>1,000,000			
Upperstage PMS	1.2592E-05	79,416			
Upperstage TVC	1.0400E-04	9,615			
Shroud	3.2464E-04	3,080			

LOM (Loss of Mission)

1.0096E-02

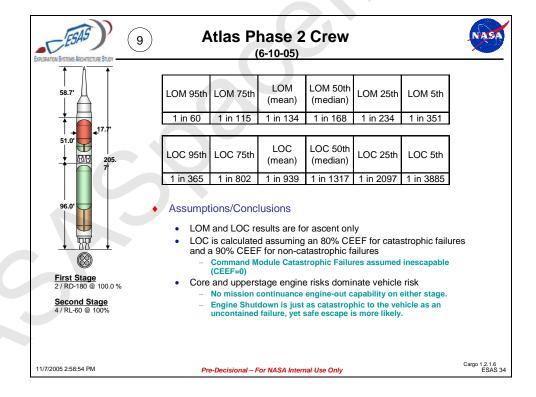
99

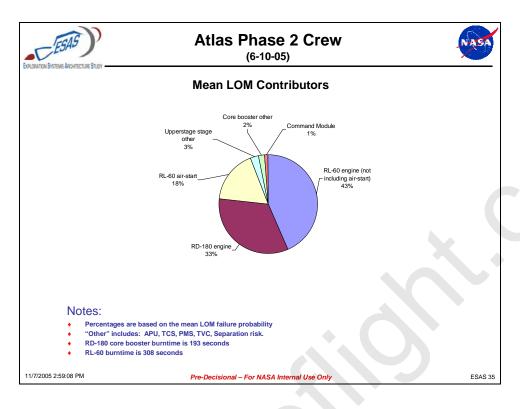
Note: LOM mean is 1 in 110 assuming 10% reduction due to inhibiting redlines.

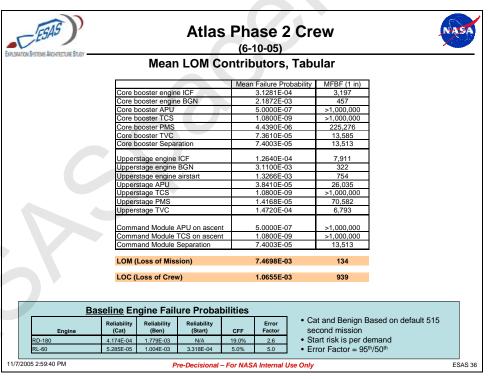
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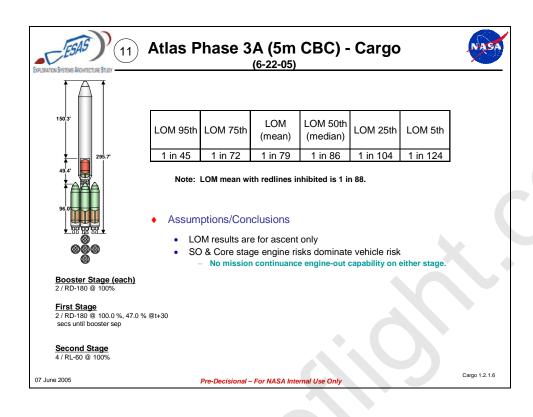
Pre-Decisional - For NASA Internal Use Only

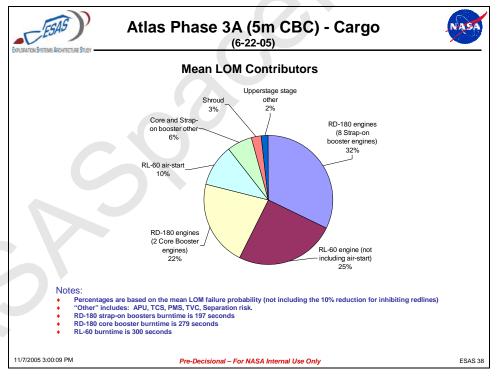
ESAS 3

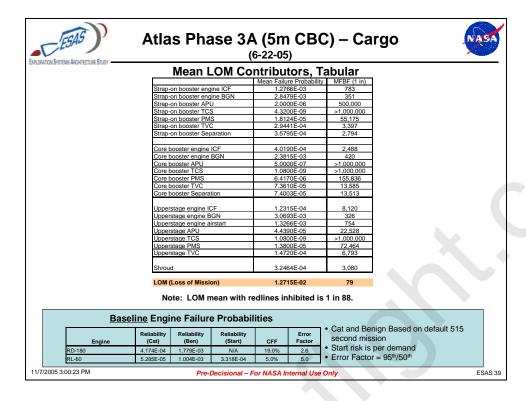


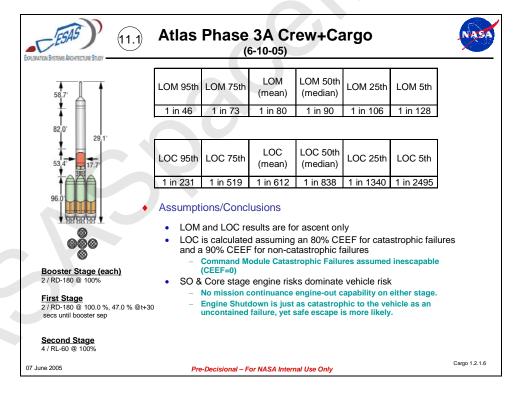


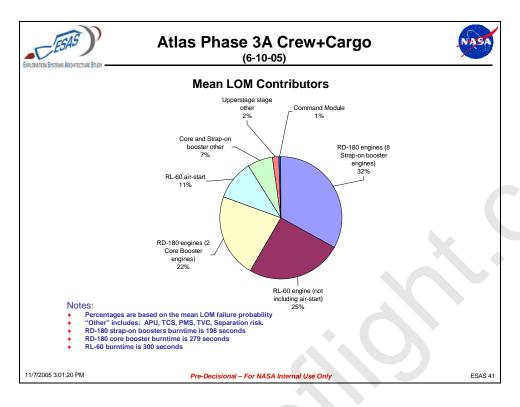


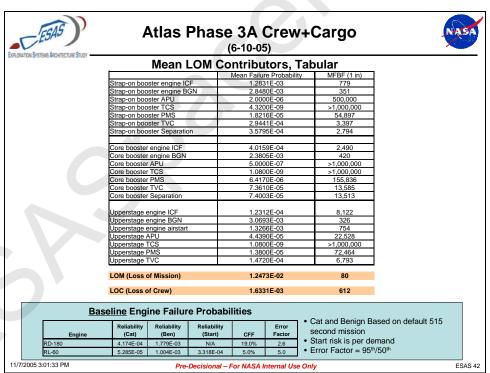


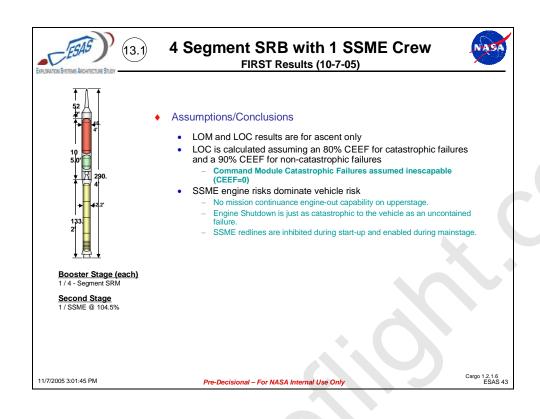


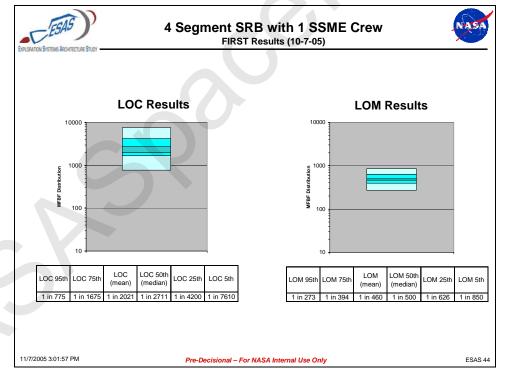


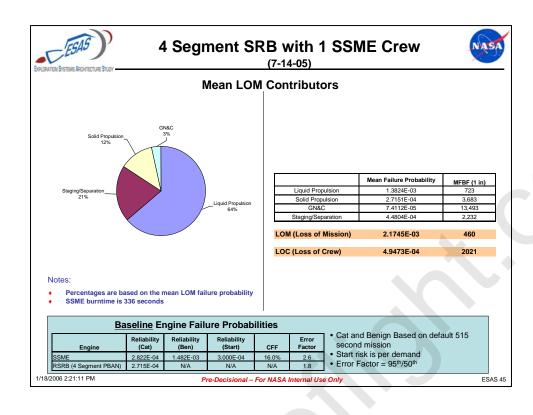


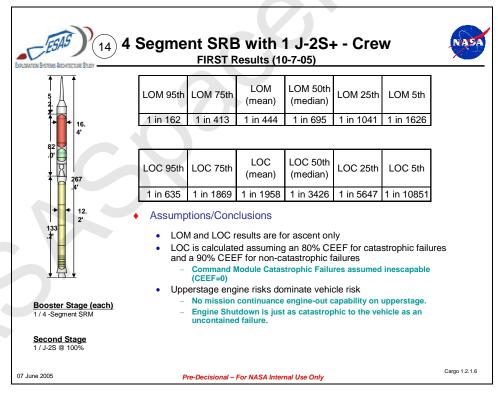


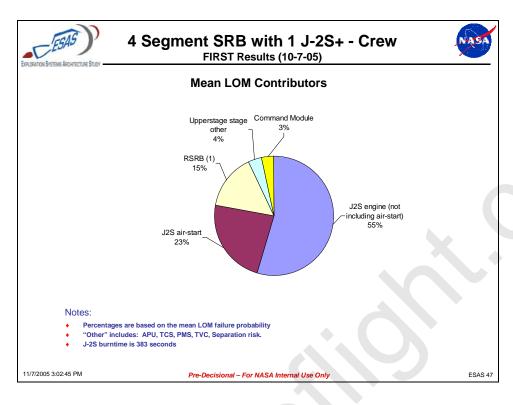


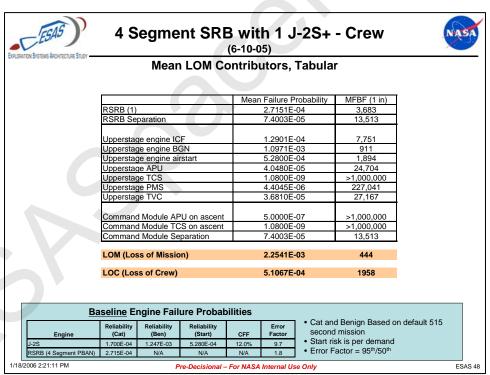


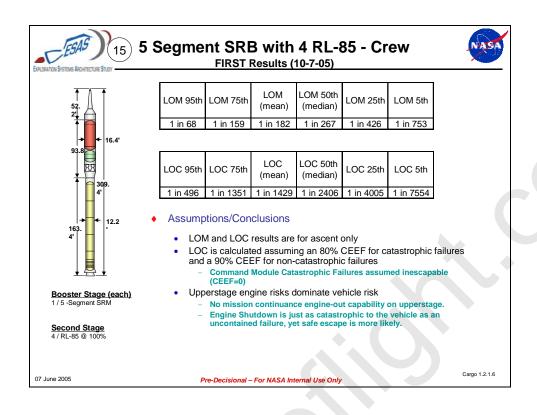


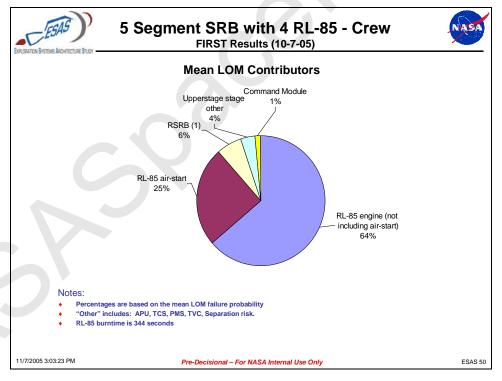


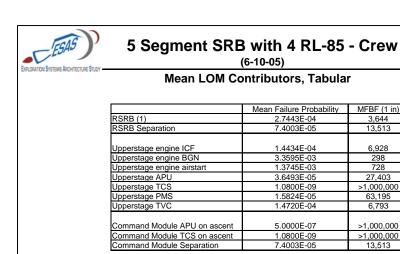












<u>baseline</u> Engine Fallure Probabilities					
Engine	Reliability (Start)	CFF	Error Factor		
RL-85	5.401E-05	1.026E-03	3.438E-04	5.0%	5.0
RSRB (5 Segment HTPB)	2.744F-04	N/A	N/A	N/A	2.0

Basalina Engina Failura Brahabilitias

LOM (Loss of Mission)

LOC (Loss of Crew)

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• Cat and Benign Based on default 515 second mission

MFBF (1 in)

13.513

6,928

298

728

27,403

>1,000,000

63,195

6,793

>1,000,000

>1,000,000

1429

NASA

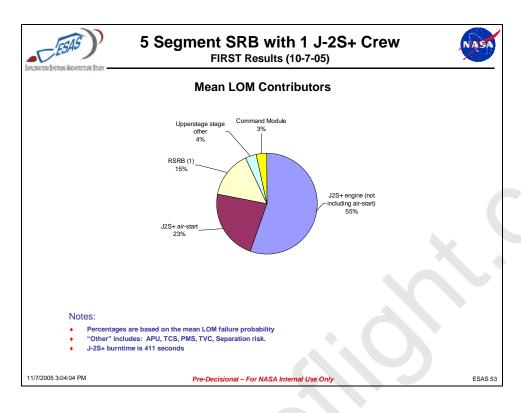
ESAS 51

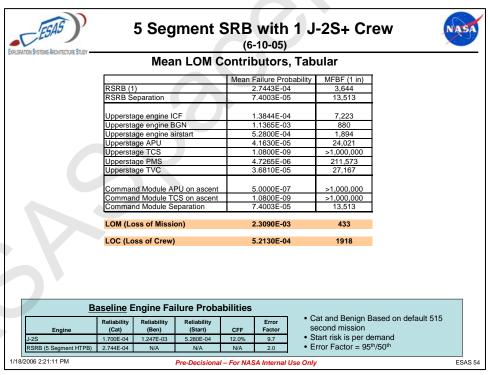
- · Start risk is per demand
- Error Factor = 95th/50th

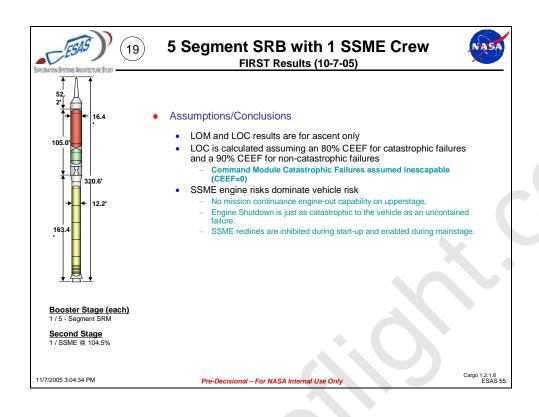
5.4922E-03

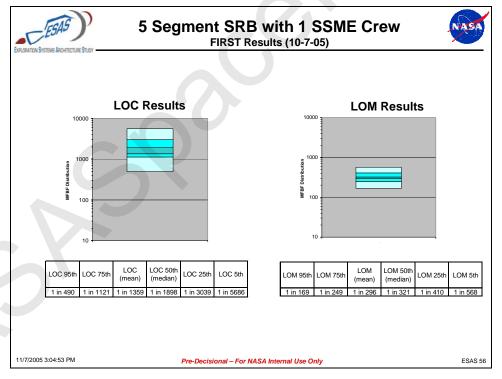
6.9964E-04

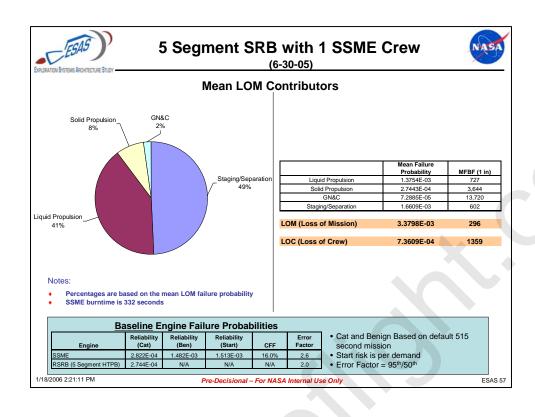
5 Segment SRB with 1 J-2S+ Crew 16 FIRST Results (10-7-05) LOM LOM 50th LOM 75th LOM 25th LOM 95th LOM 5th (median) (mean) 1 in 158 1 in 403 1 in 433 LOC LOC 50th LOC 75th LOC 25th LOC 5th LOC 95th (median) (mean) 1 in 1833 1 in 1918 1 in 3380 1 in 5592 1 in 10807 Assumptions/Conclusions LOM and LOC results are for ascent only LOC is calculated assuming an 80% CEEF for catastrophic failures and a 90% CEEF for non-catastrophic failures **Command Module Catastrophic Failures assumed inescapable** (CEEF=0) Booster Stage (each) 1/5 – Segment SRM Upperstage engine risks dominate vehicle risk No mission continuance engine-out capability on upperstage. Second Stage Engine Shutdown is just as catastrophic to the vehicle as an uncontained failure Cargo 1.2.1.6 07 June 2005 Pre-Decisional - For NASA Internal Use Only

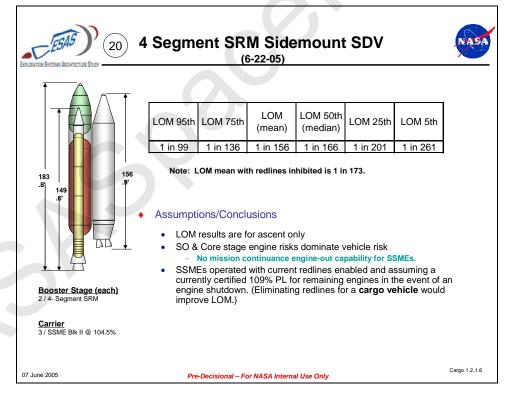


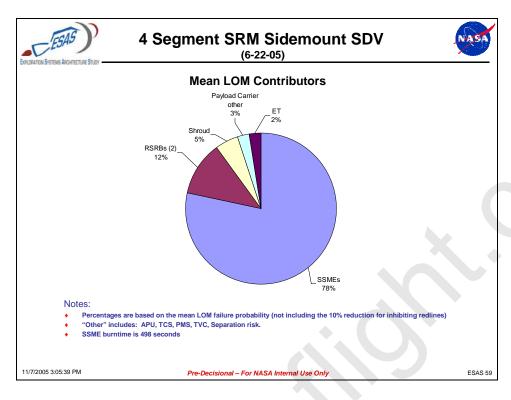


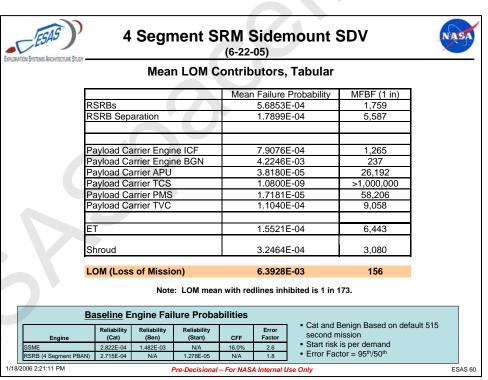


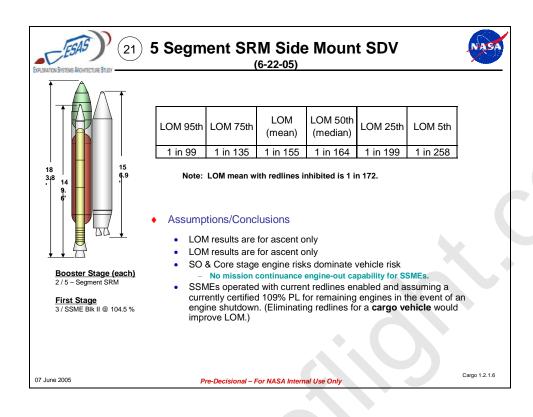


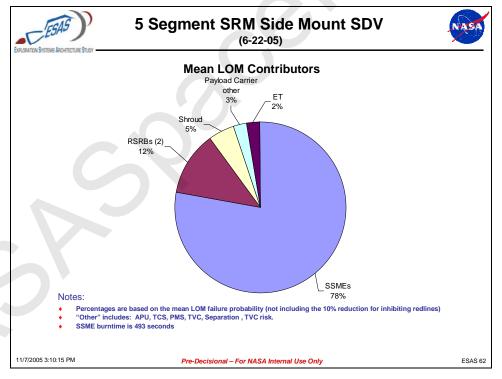














## **5 Segment SRM Side Mount SDV**



(6-22-05)

### Mean LOM Contributors, Tabular

	Mean Failure Probability	MFBF (1 in)
RSRBs	5.7437E-04	1,741
RSRB Separation	2.1219E-04	4,713
Payload Carrier Engine ICF	7.9093E-04	1,264
Payload Carrier Engine BGN	4.2522E-03	235
Payload Carrier APU	3.7797E-05	26,457
Payload Carrier TCS	1.0800E-09	>1,000,000
Payload Carrier PMS	1.7009E-05	58,794
Payload Carrier TVC	1.1040E-04	9,058
ET	1.5521E-04	6,443
Shroud	3.2464E-04	3,080
LOM (Loss of Mission)	£ 4599E 03	155

Note: LOM mean with redlines inhibited is 1 in 172.

Ba	<u>iseline</u> E	ingine Fai	ilure Proba	abilities	
	Reliability	Reliability	Reliability		

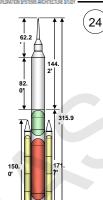
- Cat and Benign Based on default 515 second mission
- Start risk is per demand

• Error Factor = 95th/50th



## 4 Segment SRB Inline SDV - Crew+Cargo (6-10-05)





LOM 95th	LOM 75th	LOM (mean)	LOM 50th (median)	LOM 25th	LOM 5th
1 in 103	1 in 142	1 in 164	1 in 175	1 in 213	1 in 278

LOC 95th	LOC 75th	LOC (mean)	LOC 50th (median)	LOC 25th	LOC 5th
1 in 785	1 in 1035	1 in 1170	1 in 1223	1 in 1430	1 in 1773

- Assumptions/Conclusions
  - LOM and LOC results are for ascent only
  - LOC is calculated assuming an 80% CEEF for catastrophic failures and a 90% CEEF for non-catastrophic failures
    - Command Module Catastrophic Failures assumed inescapable (CEEF=0)
  - First stage engine risks dominate vehicle risk
    - No mission continuance engine-out capability on either stage.
    - Engine Shutdown is just as catastrophic to the vehicle as an uncontained failure, yet safe escape is more likely.
  - SSMEs operated with current redlines enabled and assuming a currently certified 109% PL for remaining engines in the event of an engine shutdown. (Eliminating redlines for a cargo vehicle would improve LOM.)

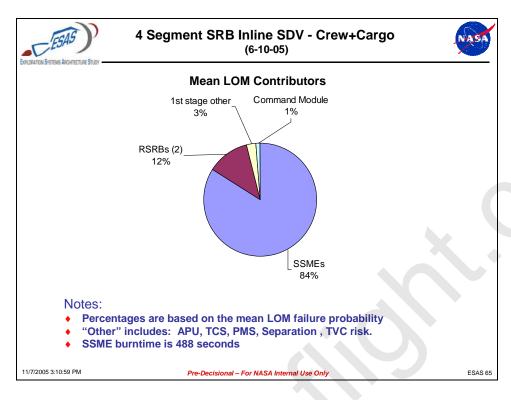
Pre-Decisional – For NASA Internal Use Only

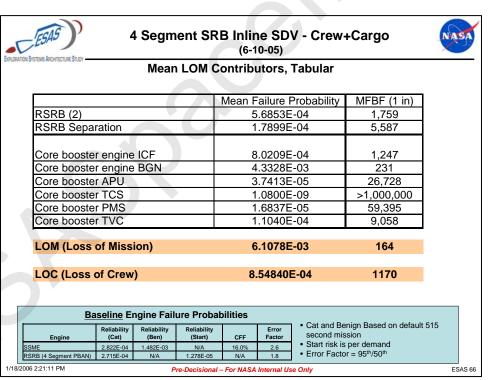
Cargo 1.2.1.6

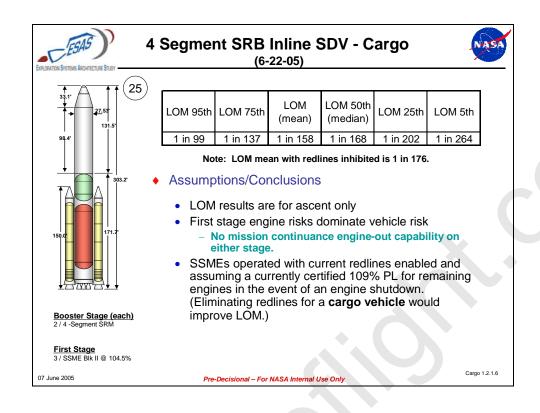
Booster Stage (each) 2/4-Segment SRM

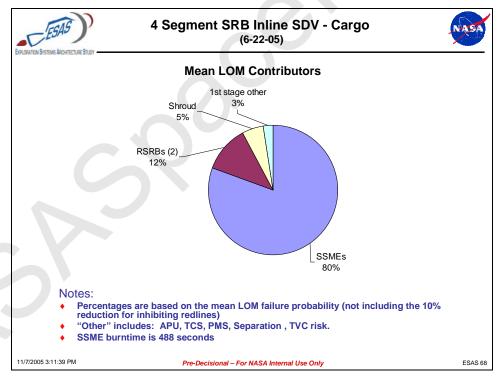
First Stage 3 / SSME Blk II @ 104.5%

07 June 2005











### 4 Segment SRB Inline SDV - Cargo (6-22-05)



### Mean LOM Contributors, Tabular

	Mean Failure Probability	MFBF (1 in)
RSRB (2)	5.6853E-04	1,759
RSRB Separation	1.7899E-04	5,587
Core booster engine ICF	7.9982E-04	1,250
Core booster engine BGN	4.3266E-03	231
Core booster APU	3.7413E-05	26,728
Core booster TCS	1.0800E-09	>1,000,000
Core booster PMS	1.6837E-05	59,395
Core booster TVC	1.1040E-04	9,058
Shroud	3.2464E-04	3,080

LOM (Loss of Mission)

6.3481E-03

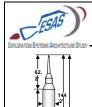
158

Note: LOM mean with redlines inhibited is 1 in 176.

Daseille Eligille Fallure Frobabilities					
Engine	Reliability (Cat)	Reliability (Ben)	Reliability (Start)	CFF	Error Factor
SSME	2.822E-04	1.482E-03	N/A	16.0%	2.6
DCDD (4 Segment DDANI)	2.7155.04	NI/A	1 270E 0E	NI/A	1.0

Rasolino Engino Egiluro Probabilitios

- Cat and Benign Based on default 515 second mission
- Start risk is per demand
   Error Factor = 95<sup>th</sup>/50<sup>th</sup>



## 5 Segment SRB Inline SDV - Crew+Cargo (6-23-05)





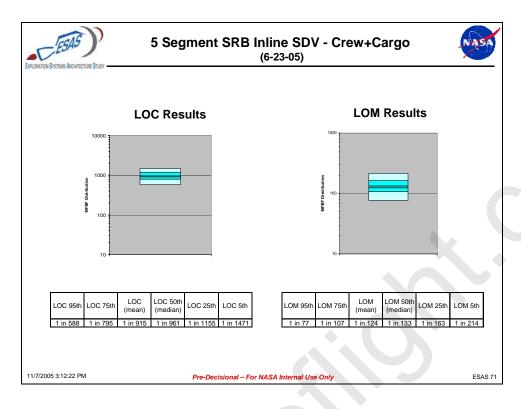
# Assumptions/Conclusions

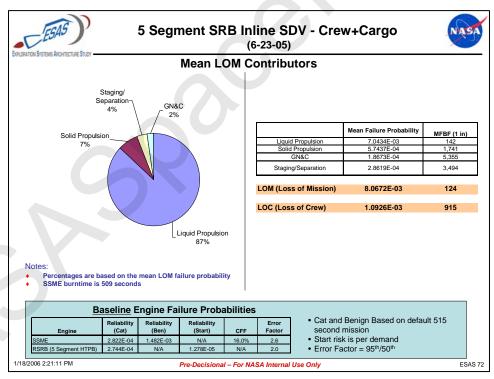
- LOM and LOC results are for ascent only
- LOC is calculated assuming an 80% CEEF for catastrophic failures and a 90% CEEF for non-catastrophic failures
  - Command Module Catastrophic Failures assumed inescapable (CEEF=0)
- First stage engine risks dominate vehicle risk
  - No mission continuance engine-out capability on either stage.
  - Engine Shutdown is just as catastrophic to the vehicle as an uncontained failure, yet safe escape is more likely.

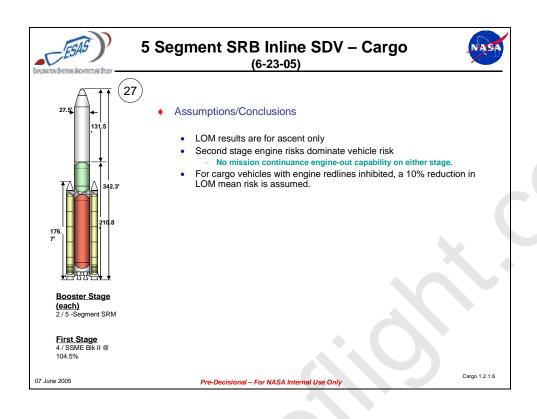
Booster Stage (each) 2 / 5 -Segment SRM

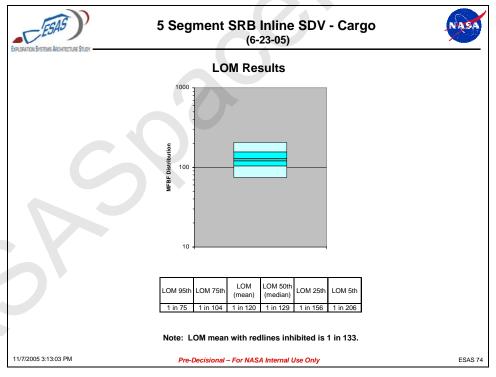
First Stage 4 / SSME Blk II @ 104.5%

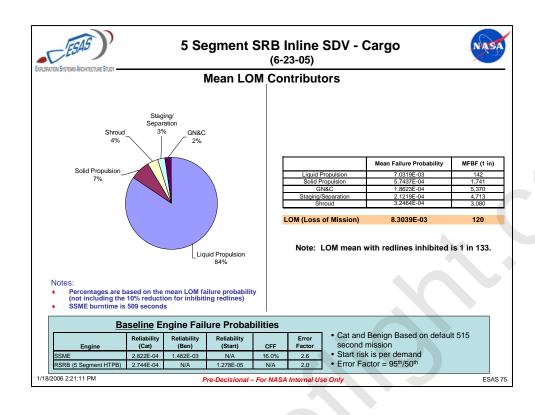
07 June 2005 Pre-Decisional - For NASA Internal Use Only Cargo 1.2.1.6

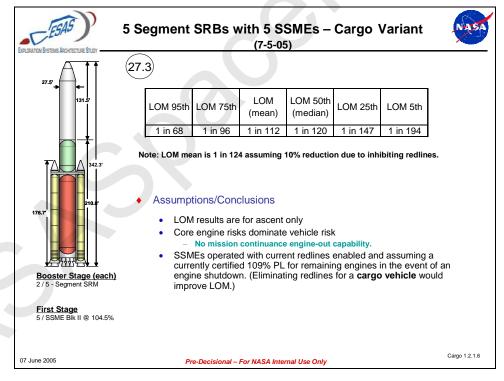


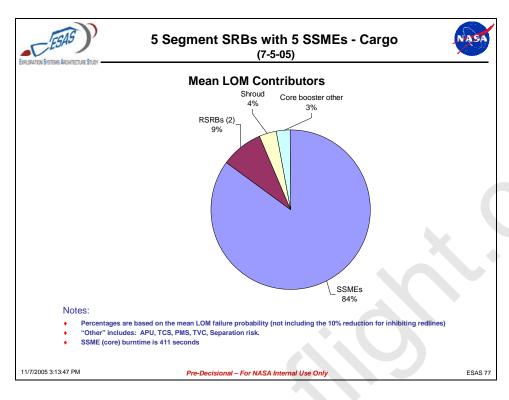


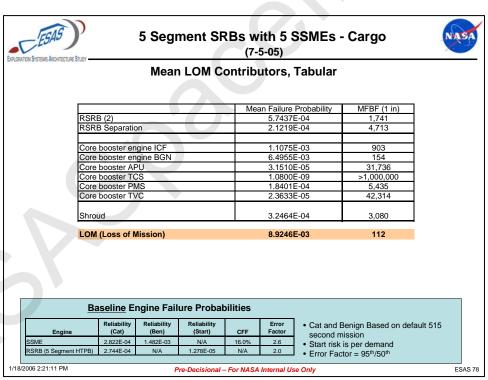


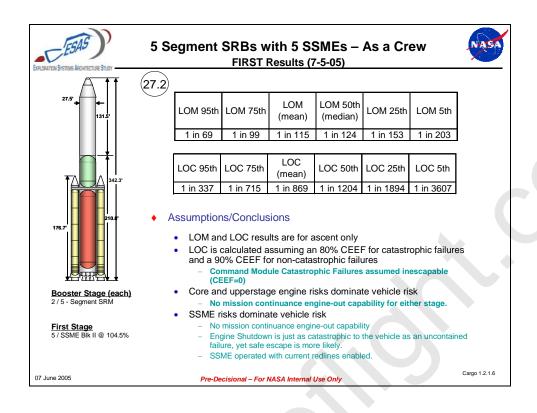


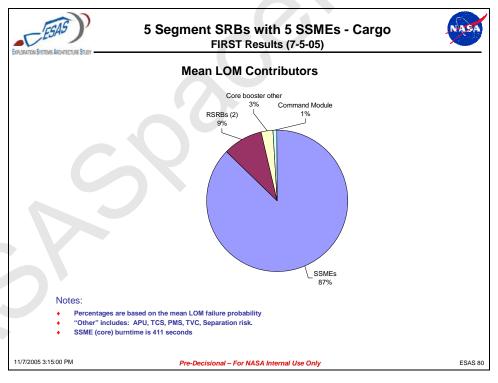














# 5 Segment SRBs with 5 SSMEs - Cargo FIRST Results (7-5-05)



### Mean LOM Contributors, Tabular

	Mean Failure Probability	MFBF (1 in)
RSRB (2)	5.7437E-04	1,741
RSRB Separation	2.1219E-04	4,713
Core booster engine ICF	1.1075E-03	903
Core booster engine BGN	6.4955E-03	154
Core booster APU	3.1510E-05	31,736
Core booster TCS	1.0800E-09	>1,000,000
Core booster PMS	1.8401E-04	5,435
Core booster TVC	2.3633E-05	42,314
Command Module APU on ascent	5.0000E-07	> 1.000.000
Command Module TCS on ascent	1.0800E-09	> 1,000,000
Command Module Separation	7.4003E-05	13,513

LOM (Loss of Mission)	8.6765E-03	115
LOC (Loss of Crew)	1.1501E-03	869

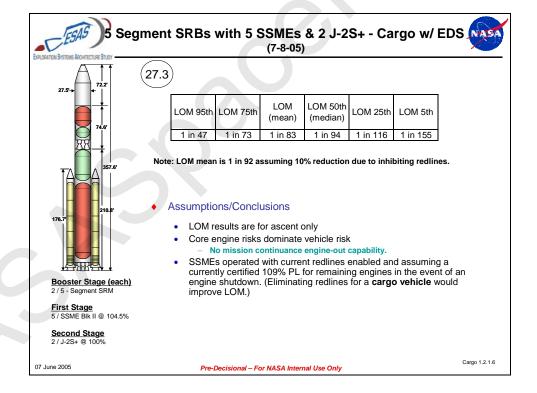
#### Baseline Engine Failure Probabilities

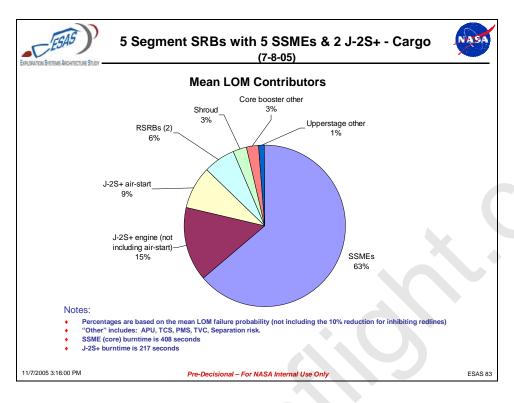
Engine	Reliability (Cat)	Reliability (Ben)	Reliability (Start)	CFF	Error Factor	Cat and Benign Based on default 515 second mission
SSME	2.822E-04	1.482E-03	N/A	16.0%	2.6	Start risk is per demand
RSRB (5 Segment HTPB)	3.484E-04	N/A	N/A	N/A	1.8	Error Factor = 95 <sup>th</sup> /50 <sup>th</sup>
RSRB (5 Segment HTPB)	3.484E-04	N/A	N/A	N/A	1.8	• Error Factor = 95 <sup>th</sup> /50 <sup>th</sup>

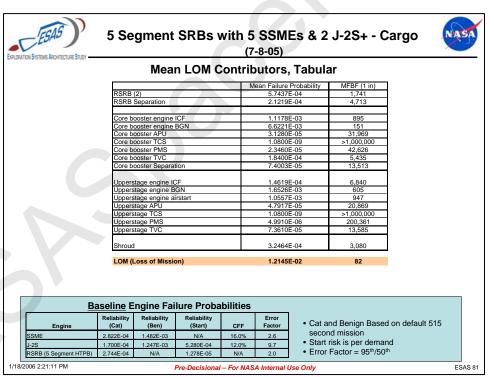
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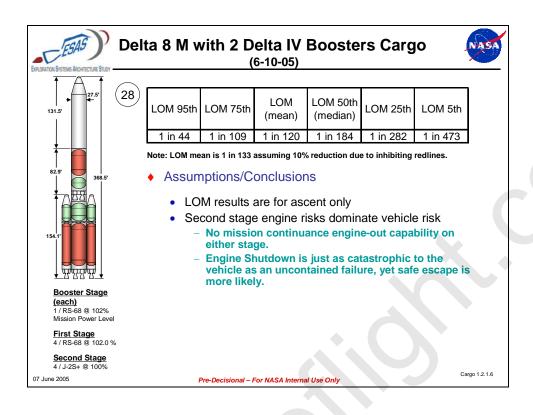
Pre-Decisional - For NASA Internal Use Only

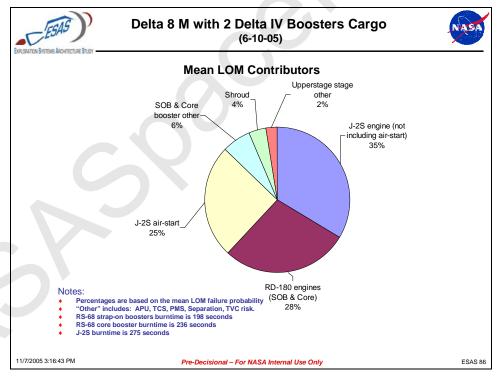
ESAS 8

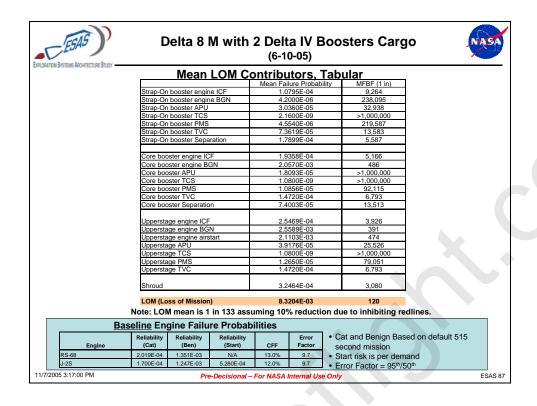


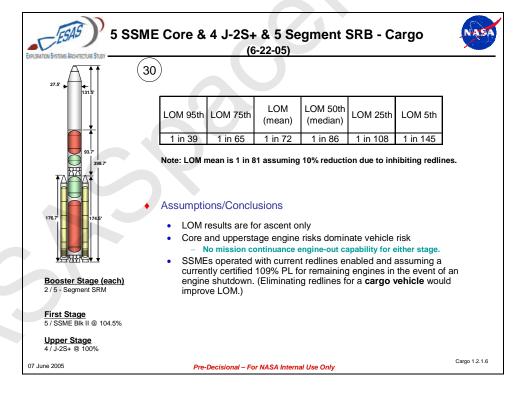


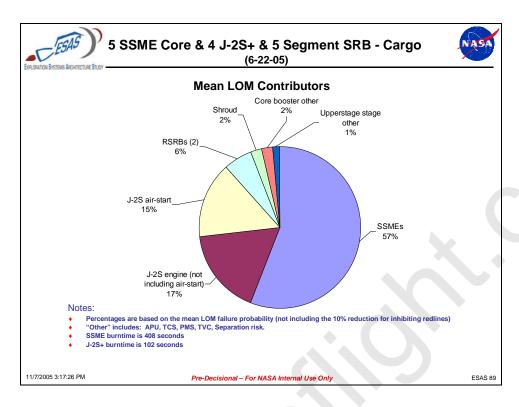


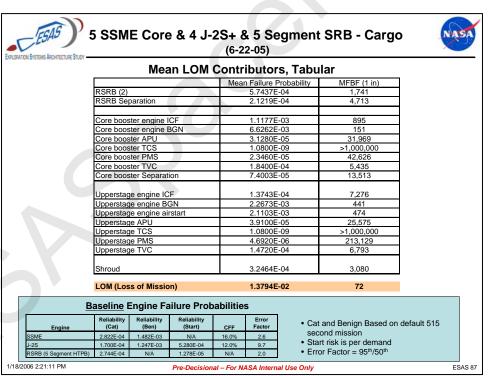


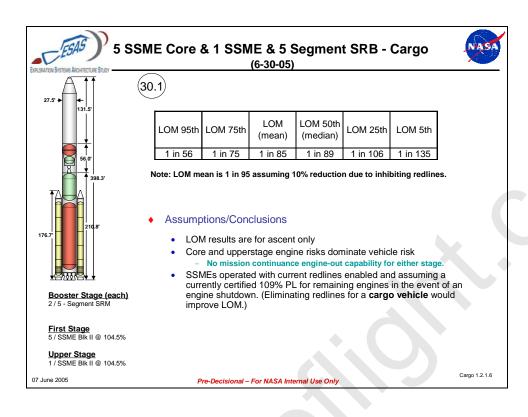


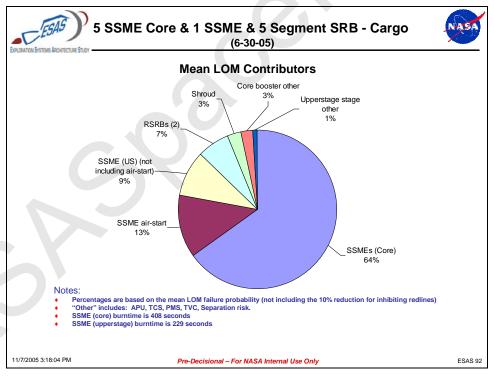


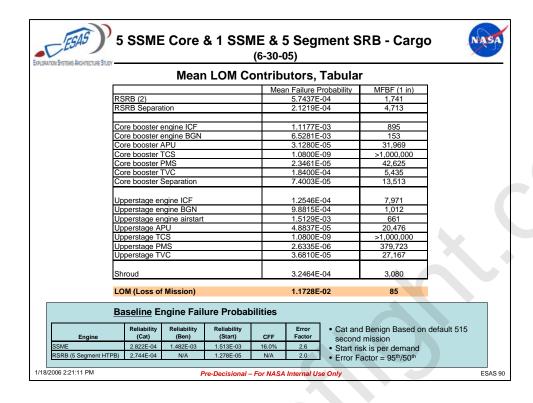


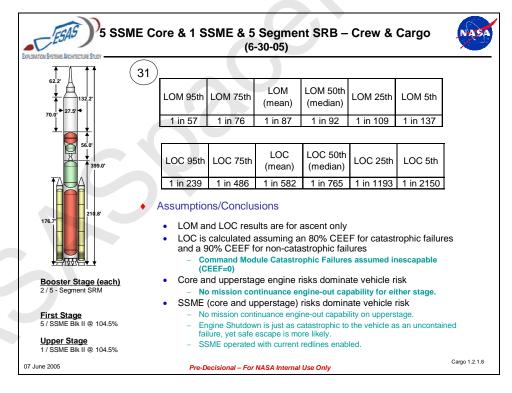


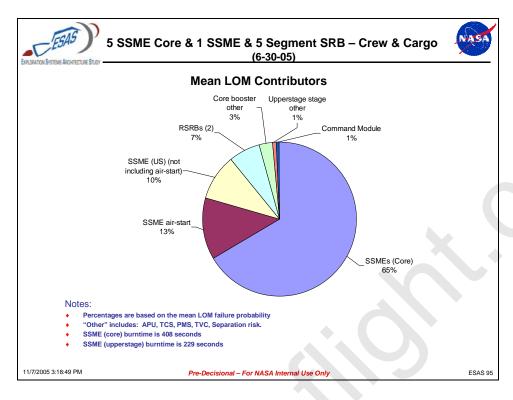


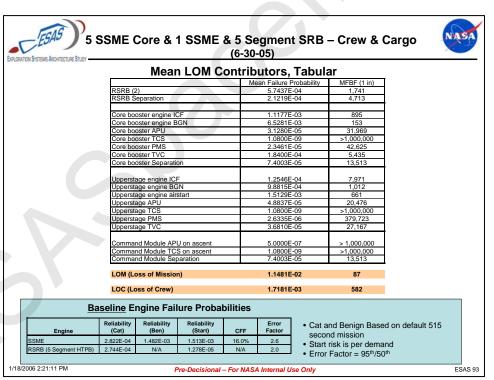














# Section 6E Cost Estimation

# Appendix 6E

# **Cost Estimation**

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	6E.1.1 Benchmarking
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	6E.1.3 Cost Estimating Methodology and Ground Rules and Assumption
6E.2	Recurring Production Costs
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6E.5	Facilities and GSE Acquisition Costs

### **6E.1 NAFCOM Overview and Verification**

The NASA and Air Force Cost Model (NAFCOM) is a parametric cost model based on 122 NASA and Air Force space flight hardware projects. The database includes launch vehicles, robotic satellites, human-rated spacecraft, and the Space Shuttle. Resume content includes mission description, major changes or unusual events, and Work Breakdown Structure (WBS) subsystem descriptions that allow an analyst to put the cost, technical, and programmatic data in perspective. Recent updates to the model include benchmarking activity with contractors and an internal assessment of the statistical fit of the historical data. The model will generally do a good job of estimating, unless the input variables are significantly in error and/or the project experiences major cost drivers not modeled by the cost estimating relationships.

### **6E.1.1 Benchmarking**

MSFC Engineering Cost Group and SAIC, MSFC's NAFCOM development contractor, worked with Lockheed, Boeing, and Northrop Grumman to benchmark NAFCOM with relevant and recently completed missions. Using Atlas V, Delta IV, RS-68, EOS-Aqua, Genesis, and Geosynchronous Operations Environmental Satellite (GOES), benchmarking was used to identify potential areas of improvement, and to measure the model improvements incorporated due to the benchmarking activities. Results of the final phase of benchmarking showed close comparisons to actual costs at the top level, except for Atlas V, which is attributable to the RD–180 engine estimate.

### **6E.1.2 NAFCOM Assessment and Improvements**

NAFCOM's Cost Estimating Relationships (CERs) were developed using manned and unmanned spacecraft and launch vehicles. Each subsystem is analyzed by mission class, weight class, launch year, etc., comparing actuals to CER estimates at the total and subsystem level. In addition to the benchmarking activities, recent improvements included a review of all subsystem CERs. Some modifications include the addition of new cost drivers (e.g., weight/volume ratio for ECLS), stratification of data according to mission type, incorporation of launch year for improvements based on "year of technology," and removal of pre-1976 missions for some subsystems (e.g., Command, Control, and Data Handling (CCDH)). The combined changes improved the goodness-of-fit statistics on the average of 8 percent for correlation coefficients, and 22 percent for standard error. MSFC continually improves the model to increase estimating capability and credibility.

### 6E.1.3 Cost Estimating Methodology and Ground Rules and Assumptions

The NAFCOM model was used to estimate prime contractor Design, Development, Test, and Evaluate (DDT&E) and Theoretical First Unit (TFU) costs using engineering assessments for subsystem input parameters. (A TFU cost is defined as the cost to produce one unit at a rate of one per year.) The total vehicle was estimated, except for the crew Launch Escape System (LES). The LES was estimated by the Crew Exploration Vehicle (CEV) team. Shuttle Program and contractor inputs for Shuttle elements and engines were used where applicable, after verification and adjustment for content by program and engineering assessments. EELV inputs were based on extensive discussion and coordination with EELV contractors, and an engineering team's assessment of

modifications required for human rating the EELVs. All launch vehicle costs include a Structural Test Article (STA) and main propulsion test article. DDT&E also includes three test flights for crewed vehicles and one test flight for cargo vehicles and facilities, which are estimated outside of NAFCOM, as discussed below. Software estimates were developed using the SEER model, with results entered into the appropriate NAFCOM WBS. SEER inputs were based on an assessment of the functional requirements for launch vehicles by the software engineering team at MSFC. Vehicle physical integration of stages into a complete launch vehicle was an additional 4 percent of DDT&E, based on NASA experience. A standard fee of 10 percent was used, and a 25 percent reserve was added to each vehicle estimate. U.S. Government oversight of 25 percent was included as a full cost accounting factor. The full cost accounting factor includes civil service salaries, travel, Infrastructure upkeep, utilities, security, cost of facilities, and corporate General and Administrative (G&A). Facilities costs are based on engineering assessments of infrastructure requirements. When contractor inputs were available, Government estimates were compared and reconciled with those inputs.

Standard NAFCOM inputs across all vehicles are listed in **Table 6E-1**.

NAFC OM Parameter Cargo (2) Significant Use of Advanced (2) Significant Use of Advanced Manufacturing Methods Methods Methods Range (2) to (4) Few to Significant Range (2) to (4) Few to Significant Engineering Management Changes Changes Range (2) to (6) Exisitng Design to Range (2) to 6) Exisitng Design to New De sign New Design New Design Funding Availability (1) Funding is Assured (1) Funding is Assured Test Approach (3) Maximum Testing (3) Maximum Testing (3) Maximum Interfaces/Complexities Integration Complexity (3) Maximum Interfaces/Complexities (1) Two or more studies prior to start (1) Two or more studies prior to start of D&D of D&D Pre-Development Study

**Table 6E-1. Standard NAFCOM Inputs** 

### **6E.2** Recurring Production Costs

The Operations Cost Model (OCM) is an Excel-based, parametric model developed for the estimation of space launch systems operations costs. For the purpose of modeling in OCM, launch system operations are defined as those activities that are required to deliver a payload from a launch site on the Earth's surface to Low Earth Orbit (LEO). The OCM WBS cost elements represent the full complement of products and services potentially required to operate a launch vehicle. The cost elements are arranged into four segments: Program (P), Vehicle (V), Launch Operations (L), and Flight Operations (F). The individual WBS cost elements are assigned to one of these four segments. Estimating cost for every WBS cost element is not required, nor is it necessarily expected. For instance, an unmanned vehicle would not be expected to have costs for F7 Crew Operations or V2 Reusable Hardware Refurbishment. **Figure 6E-1** below shows the WBS cost element arrangement.

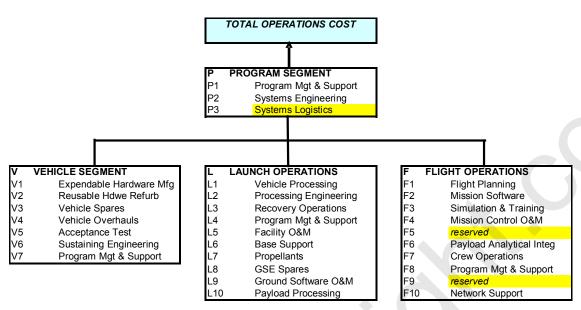


Figure 6E-1. OCM Cost Element Work Breakdown Structure

The WBS structure is based generally on the Shuttle, Delta, and Atlas program organizations. The Program Segment elements represent top-level program management and systems engineering activities. The Vehicle Segment reflects all the activities and tasks required to support the launch system with the provision of flight hardware, either through the refurbishment of reusable items or the manufacture of expendable items. With the Vehicle Segment, production costs can be estimated for each element in the vehicle WBS.

The Launch Operations Segment generally includes those products and services provided at the launch site (e.g., Kennedy Space Center (KSC) or Cape Canaveral Air Force Station (CCAFS)), while Flight Operations contains the products and services associated with Johnson Space Center (JSC) or the center of engineering operations.

There are different individual cost elements for Program Management and Support provided within each segment (P1, V7, L4, and F8). This provides the flexibility to model different levels of program management for different vehicle elements, contractual elements, configuration items, or other types of hierarchical program structures, which might have multiple levels of significant program management efforts. Some, all, or none of the elements can be removed from the estimate, if desired.

WBS elements highlighted in the above figure are not currently in use in OCM. Elements F5 and F9 are blank and may be employed in a subsequent model version. P3 System Logistics is included as a placeholder for future incorporation of aircraft-like operations estimating capabilities within the model.

For this analysis only the Vehicle segment of OCM was used to estimate the recurring production costs of flight hardware elements. Launch operations costs, as defined above, were estimated by KSC personnel while flight operations costs were estimated by JSC personnel. Program segment costs and full cost accounting were included by adding 25

percent wraps and 10 percent reserve to the other operations cost estimates. The necessary detailed cost estimates for new or modified launch facilities and Government Supplied Equipment (GSE) were prepared also by KSC personnel.

OCM is generally used to estimate the fixed and variable cost for each WBS element. The fixed and variable costs are derived from a linear approximation of production costs estimated at different production rates using rate curves and TFU costs derived from historical data or from NAFCOM. An example, using eSSME production, is shown below in **Figure 6E-2**. Production costs for one engine per year through 18 per year are estimated using a 64 percent rate curve and a TFU of \$142.6M. A "best fit" linear approximation is derived for the expected production range. In the example, this was assumed to be between 1 and 10 engines per year.

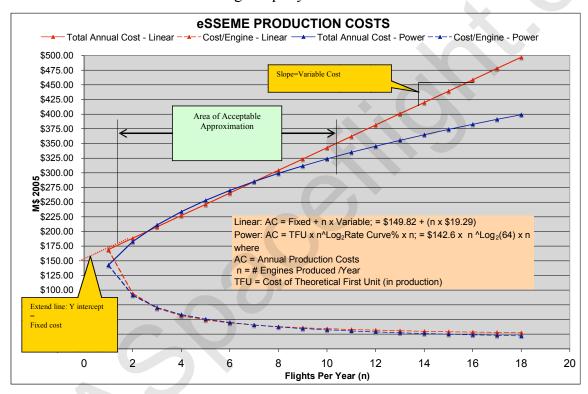


Figure 6E-2. Hardware Production Costs

If a greater production range were used, for example, if the production capability is assumed to be 16 engines per year, the "Area of Acceptable Approximation" in the above figure would stretch to the right and the line representing the linear approximation would flatten out as it passed through the power curve at close to 16 engines rather than close to 10. The Y-intercept, or fixed cost, would increase and the slope, or variable cost, would decrease. This is consistent with having to increase plant production capacity by adding equipment and staff (fixed cost) and decreasing the cost of materials (variable cost) by buying in greater quantities. At the same time the gap, or variance, between the blue rate curve and the red linear approximation would be greater in the midrange of production than seen above, indicating that engine cost estimates in the midrange of production would tend to be lower than actual costs.

Fixed and variable costs may be aggregated at the segment level if desired. The annual cost for any production or flight rate for any hardware element can be estimated from the fixed and variable costs using the above equation. This allows the analyst to estimate annual operating costs in the face of variations in production or flight rates from year to year.

OCM is predicated on operations as an "ongoing concern," that is to say, while there may be variations in output from year to year, such as more or fewer flights than a nominal number, operation is continuous with full fixed costs incurred, i.e., no staff or capacity reductions, even in periods when there is little activity. Otherwise, some means of estimating fluctuations in the fixed costs would be necessary, requiring a very detailed level of information on manufacturing practices.

#### **6E.3** General Assumptions and Ground Rules

In general, a conservative approach was adopted. Production of hardware for all architectural configurations was estimated as for manned systems, whether the vehicle was designated as "Crew" or "Cargo." For "shadow" estimates of Program, Launch, and Flight operations, given the high degree of uncertainty in current expendable systems programs (for example, the effects of a Boeing/Lockheed Martin merger into the United Launch Alliance (ULA)), OCM factors for manned and reusable configurations were also applied to the expendable concepts. The "Manned/Reusable" setting in OCM yields the highest cost for all operations elements while the "Unmanned/Expendable" setting results in the lowest. This rule was applied to simplify the analyses and account for at least some of the uncertainties in a conservative manner. For all except the primary operations element for hardware element production (expendable hardware production (V1) or reusable hardware refurbishment (V2) – see **Figure 6E-1**), ratio analysis rather than parametrics was used. In ratio analysis, the costs for V3 through V7 are ratios of V1 and/or V2 rather than being based on pertinent parameters. The exception to this is the reusable SRB and SRM elements, for which more detailed production data was available.

#### **6E.4** Recurring Launch Operations Costs

The ESAS used a data-driven methodology for recurring yearly launch operations. The ESAS focus on closing the gap between the retirement of the Shuttle orbiters and the development of the capability to provide crewed launches for exploration had the direct effect of reducing the trade-space provided for operations analysis to elements of architectures that already exist and are operational (e.g., Solid Rocket Boosters (SRBs), External Tanks (ETs), Space Shuttle Main Engines (SSMEs), Atlas or Delta size (diameter) tanks and core boosters). The differences in data quality between elements were driven by the following:

• Shuttle systems have a relative wealth of operations data; the principal issue of data quality is that amidst a wealth of data no standard set exists for the conditions required to support cost estimation, and no basic tool, model, or sets of estimating relationships exist with which diverse experts might still consistently use such data and arrive at similar conclusions.

- For ESAS estimation the Shuttle data used was in FY 05 steady-state dollars, but did not include upgrades expenditures or Post Columbia return-to-flight effects.
- Expendable Launch Vehicles (ELVs) have a lack of public openness as to data as
  well as a relatively short lived history for what data is available; this has created
  few opportunities for the operations community to analyze, research, publish, peer
  review, or develop community recognized insights into these new systems
  operations.
  - For ESAS estimation the ELV data used was in FY 05 steady state dollars, based principally on Air Force budget data, knowledge of renegotiations in work to have the Government cover launch operations fixed infrastructure costs, and knowledge of flight rates being planned or purchased.

In all cases, recurring cost estimation includes:

Civil Service and Contractor (prime and subcontractors) for

- 1) Logistics & GSE,
- 2) Propellant,
- 3) Launch operations inclusive of:
  - processing,
  - systems engineering support,
  - facility O&M,
  - command, control and checkout center Operations & Maintenance, inclusive of instrumentation,
  - modifications (as an annual allotment, used as required),
  - sustaining,
  - program support (procurement, etc),
  - communications,
  - base operations support/O&M,
  - weather support, and
  - payload integration.
- 4) Payload processing and Multi-Element Integrated Test (MEIT).

Additionally, as with infrastructure estimates, the launch site operations estimate involved:

• Inclusion of factors above and beyond the procurement such as to provide reserve (10%), as addressing the level of uncertainty involved, and

• Inclusion of factors above and beyond the procurement such as to cover center overheads (e.g., PMS/Network, 5%).

The launch site recurring costs estimation methodology maps are as shown in **Figures 6E-3** and **6E-4**.

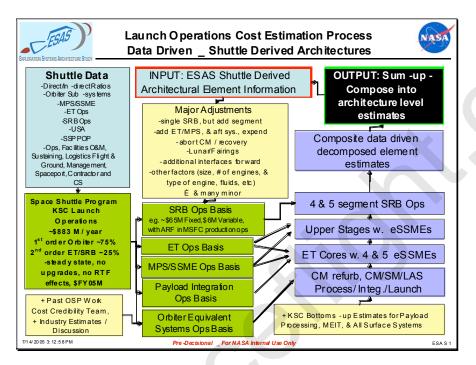
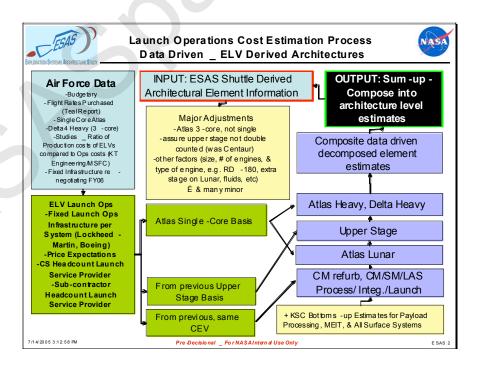


Figure 6E-3. Launch Site Operations Cost Estimating Methodology for Shuttle-Derived ESAS Architectures



#### Figure 6E-4. Launch Site Operations Cost Estimating Methodology for ELV-Derived ESAS Architectures

#### **6E.5** Facilities and GSE Acquisition Costs

#### **Launch Site Infrastructure**

The methodology used to derive non-recurring launch site infrastructure cost estimates involved:

- Development of the concept for the operation;
- Architectural level analysis of the costs to modify or build a facility within the concept (i.e., the acquisition cost of the facility), and analysis of the costs to outfit the facility, (i.e., to acquire all the ground support equipment required to process the launch vehicle or spacecraft element);
- Inclusion of factors above and beyond the procurement, such as Government engineering, technical skills, and management brought to bear to assure a successful acquisition (varying by phasing year);
- Inclusion of factors above and beyond the procurement, such as to provide reserve (25%), as addressing the level of uncertainty involved; and
- Inclusion of factors above and beyond the procurement, such as to cover center overheads (e.g., PMS/Network, 5%).

The ground infrastructure concepts developed focused on the Crew Exploration Vehicle (CEV), the Crew Launch Vehicle (CLV), and the heavy-lift or Lunar Launch Vehicle (LLV). Additionally, Risk-Reduction Flight (RRF) concepts were developed. Deciding factors in infrastructure choice were:

- Costs:
- Responsiveness to fulfill planned manifest requirements;
- Best merger of the near-term CLV infrastructure backwards to any Shuttle assets required in the concept;
- Best merger of the near-term CLV infrastructure forward with future LLV assets required of those concepts;
- Hazards, such as with SRBs or Launch Abort Systems (LASs);
- Overall flow efficiency, as in number of steps; and
- Overall flow effectiveness, such as in determining the number of footprints, station-sets, or facilities required to meet flight rate needs.

One example concept, associated costs, and basis of estimate developed is as shown in **Figure 6E-5**.



Figure 6E-5. Crew Exploration Vehicle Ground Operations/Processing Concept, Associated Costs, and Basis of Estimate



# Section 6F EELV Modifications for Human Rating

## **Appendix 6F**

## **EELV Modifications for Human Rating**

#### Contents

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  - 6F.2.1.2 First Stage Main Propulsion
  - 6F.2.1.3 Second Stage Main Propulsion
  - 6F.2.1.4 Structure
- 6F.2.2 Atlas V Heavy with New Upper stage
- 6F.2.3 Delta IV with New Upper Stage
  - 6F.2.3.1 Avionics and Software
  - 6F.2.3.2 Delta IV Booster MPS
  - 6F.2.3.3 Upper Stage MPS
  - 6F.2.3.4 Structure

## Appendix 6F

## **EELV Modifications for Human Rating**

The Evolved Expendable Launch Vehicle (EELV) program was intended to provide for a reliable access for commercial and military payloads, hence, considerations for flying crew were never factored into the original design of the vehicles. The Mercury and Gemini Program used vehicles originally designed for other purposes for launching crews to orbit. To do so, modifications were performed to provide for increased reliability, redundancy, failure detection and warning, and removing hardware not necessary for the crew launch mission. The same considerations would be required in order to utilize the EELV fleet to launch crew to and from Earth orbit.

#### **6F.1 Human-Rating Requirement Drivers**

The technical requirements for human rating are in NPR 8705.2a, Human Rating Requirements for Space Systems. The document applies human rating at the "system" level, identifying the system as Launch Vehicle (LV), and spacecraft. Allocation between the LV and spacecraft is provided for in subsequent system requirements documents for the elements. For this study NPR 8705.2a is the basis for evaluating the EELV fleet to ascertain the modifications necessary for carrying crew to Earth orbit.

The main requirement drivers from NPR 8705.2a are:

- Specifications and Standards,
- Two-Fault Tolerant Systems,
- Crew-System Interactions,
- Pad-Emergency Egress,
- Abort Modes,
- Software Common Cause Failures,
- Manual Control on Ascent, and
- Flight Termination System.

The EELV fleet was built primarily to company standards and processes. The EELV was developed to "high level" system requirements and no aerospace industry design practices or standards were imposed. At the time the Program was implemented, high reliability was to be demonstrated with multiple commercial launches before committing Government payloads. In response to the commercial launch market collapse (and resulting loss of demonstrated and envisioned reliability gains) Government mission assurance was ramped up with support from the Aerospace Corporation and the National Reconnaissance Office (NRO). The New CY2005 Buy III will now include Government Mission Assurance requirements and standards. For the EELV, these standards would need thorough evaluation and approval against NASA standards and processes, in order to be used for flying crewed missions, with changes and additions implemented to close known gaps in requirements.

One of the most important requirement drivers is the requirement for two-fault tolerance to loss of life or permanent disability. NPR 8705.2a also states that abort cannot be used

in response to the first failure. This implies that the LV must be at least single fault tolerant, and for subsystems that are required for abort, it must be two-fault tolerant. EELV will require upgrades in certain areas to achieve single fault tolerance.

In order to fly crew for any launch system, the crew must have certain situational awareness and be able to react to contingencies based on that awareness. As such, NPR 8795.2a contains many requirements that deal with the crew's ability to monitor health and status and take appropriate actions as a result of that status, if required. This will require upgrades in the EELV avionics architecture to accommodate an interface with the spacecraft as well as be able to accept certain commands from the crew. For the LV, these commands will primarily be for contingency situations and will be for things such as abort initiation, retargeting (Abort-to-Orbit (ATO)), and responses to other contingencies. Manual control is also a response to a contingency, although its use would primarily be limited to second stage where structural and thermal margins allow manual control. The form of manual control would be the subject of future trade studies and could be classical "yoke" control, to a series of discrete commands to allow retargeting and ATO scenarios.

Another important requirement is the requirement to provide for successful abort modes from the pad through the entire ascent profile. This will require the EELV to be modified to provide the data necessary for abort decision-making. It may also require computer and software for making the decision, or a system that may reside with the spacecraft; this is a subject of a future trade study. Regardless, of the outcome of that trade, significant effort on the LV will be required for health management and abort decision-making.

Other requirements such as protection against software common cause failures, and Flight Termination Systems (FTS) are not extensive as those discussed but require some effort on the LV to implement. Protection against common cause software failures can take several forms and is discussed in NPR 8705.2a. In the case of FTS, the EELVs utilize autodestruct with lanyard pull devices to initiate an FTS event. Human spaceflight has never utilized autodestruct and the utility of using these devices needs to be examined. Lanyard pulls allow the booster (first stage) to not have a dedicated receiver and command decoder unit, being able to accept the commands from the second stage and being capable of autodestruct in the event of inadvertent separation. Removal of the autodestruct may require addition of the dedicated receiver and command decoder unit on the first stage.

#### **6F.2 EELV Modifications for Human Rating**

For the Design Reference Mission (DRM), four vehicles were examined. (See Section 6, Launch Vehicles and Earth Departure Stages, of the ESAS Final Report.) Only the Atlas V Heavy-Lift Vehicle (HLV) with the existing upper stage and new upper stage, and Delta IV HLV with a new upper stage were considered for assessing modifications for flying crew. In some cases, detailed assessments were possible; in other cases, only the type of issues and resultant potential modifications were identified, depending upon the fidelity of data available from the commercial launch provider. In either case, the goal of the analysis was to make reasonable judgments concerning the NAFCOM model inputs to provide valid cost assessments and ascertain potential schedule issues.

#### 6F.2.1 Atlas V HLV with Existing Upper Stage

#### 6F.2.1.1 Avionics and Software

The current avionics architecture used for the Atlas V vehicle is single string in most all areas. Current plans call for the Atlas to be upgraded to single-fault tolerant starting at flight AV010. **Figure 6F-1** shows the upgraded architecture (Block 2)

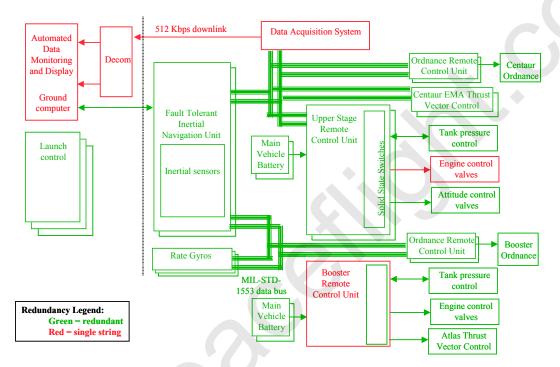


Figure 6F-1. Atlas V Avionics Block 2 Upgrade

Almost the entire avionics suite is upgraded in Block 2. Remaining items that would have to be addressed are the Booster Remote Control Unit (BRCU) and preventing software common cause failures. A BRCU upgrade is currently in the study phase.

In addition the system must be modified to provide Launch Vehicle Health Management (LVHM) functions. This is envisioned to consist of the core avionics suite along with a LVHM "kit" to provide the functionality needed. **Figure 6F-2** shows the Atlas version of this.

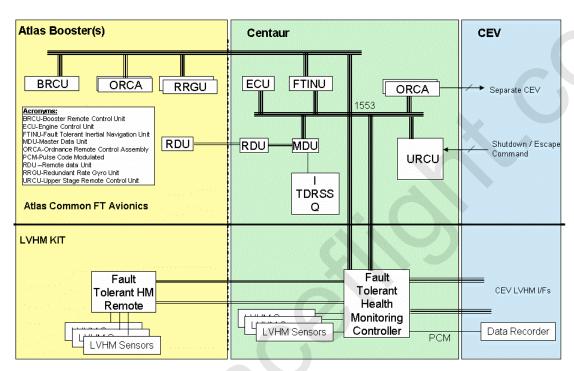


Figure 6F-2. Atlas LVHM Implementation

An alternate approach was defined by the study team which more fully integrated the Integrated Vehicle Health Management (IVHM) capabilities into the core avionics architecture (**Figure 6F-3**). This would require more extensive changes to the current flying architecture and was used to define an upper bound for costing purposes.

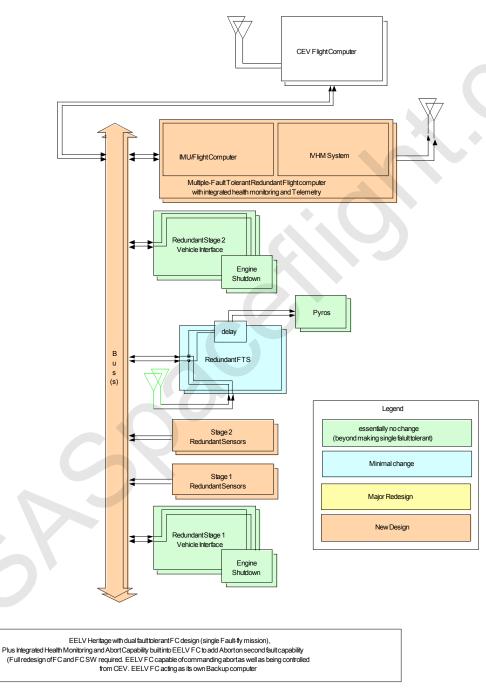


Figure 6F-3. Generic LVHM Implementation

In either case the core avionics meets the minimum single-fault tolerant requirement. Those elements needed for abort are two-fault tolerant. In the case of the integrated LVHM option, the core functions also get enhanced redundancy over the "kit" approach.

The Software Lines of Code (SLOC) for a new build avionics system was estimated as follows:

- Events Manager (50 Hz) 500 to 1,000 SLOC
- Navigation Manager (50 Hz) 8 to 15 KSLOC
- Guidance Manager (1 Hz) 15 to 25 KSLOC (both ascent and abort modes)
- Control Manager (50 Hz) 8 to 15 KSLOC
- Command and Data Manager (50 Hz) 25 to 40 KSLOC
- Time Manager (50 Hz) 1,500 to 2,000 SLOC
- Power Manager (25 Hz) 2,500 to 4,000 SLOC
- Vehicle Management Software (55 to 165 KSLOC)
- Total SLOC = 103 to 267 KSLOC

The large range in values is due to the vehicle management software that incorporates the LVHM, Fault Detection, Isolation and Recovery (FDIR), and abort decision-making. At present there is a large uncertainty concerning the extent of LVHM that will be required and will be the subject of future trade studies.

The estimates for software were lowered for the avionics approach of **Figure 6F-2** where much of the core avionics is kept intact and thus much of the flight software. The software estimate for this case was 103 to 165 KSLOC. For the avionics approach of **Figure 6F-3** the approach was to assume primarily a totally new software load for the estimate as that approach, although using some of the GN&C and electrical power existing system, it results in a new flight computer with LVHM functionally integrated with the system.

#### **6F.2.1.2** First Stage Main Propulsion

The primary focus of the effort was to examine changes required to the RD–180 for use in a human-rated system. As a minimum, the RD–180 was required to be built with U.S. production capability. Data was used from Lockheed Martin to provide the cost profile for this effort. Other potential upgrades include the following.

Reliability enhancement program:

- New Turbine Stator Blade Profile,
- Revised Design for turbine exit manifold flow straightening device,
- Modified pre-burner injector head faceplate.
- Introduce Nickel coating on projecting chamber elements,
- Decreased installation stresses in small tubing, and
- Enhanced health check using upgraded pressure and new optical sensors.

Other human-rating considerations for the engine:

• Add safe in-flight shutdown mode, and valve lockup to enable abort,

- Potential redesigns for controls redundancy, FDIR capability and health management, and
- Potential additional RP–1 tap for additional Thrust Vector Control (TVC) hydraulic redundancy.

Other changes to the MPS are required to provide additional redundancy. The identified changes were:

- Incorporate engine pre-valves to enhance pre-launch pad safety,
- Upgrade valve actuators for redundancy,
- Increase instrumentation for health management purposes,
- Evaluate component for margins (modify as required),
- Upgrade fill-and-drain valves for redundancy; current designs have single poppet seat, and
- Redesign Pogo Suppressor System to allow for the deletion of pyros, tubing, and elimination of baffle spot weld cracking.

#### **6F.2.1.3 Second Stage Main Propulsion**

The primary modifications required for the second stage are with the RL-10A-4-2. The following modifications were identified for the RL-10A-4-2 engine:

- Engine modifications required for use on a human-rated system:
  - Valve actuation redundancy;
  - o Turbo-pump enhancements;
  - o Ignition enhancements;
  - o Increase interface loads margins;
  - Re-design for controls redundancy, full FDIR capability, and health monitoring development;
  - Add controller channel redundancy and cross-strapping (vehicle or on-engine);
  - Add safe in-flight shutdown mode and valve lock-up mode to enable abort;
  - O Different operating point (75%) assess for performance, dynamic and throttling issues (i.e., engine-out philosophy); and
  - Design changes from structural assessments (e.g., fracture control, strength).

Other modifications associated with the second stage MPS include increased valve and valve actuator redundancy, pneumatic system upgrades to eliminate single point failures, and increase instrumentation for health management.

#### 6F.2.1.4 Structure

NPR 8705.2a imposes as an applicable document NASA-STD-5001, Structural Design and Test Factors of Safety for Spaceflight Hardware. This standard requires all structural Factors of Safety (FSs) for tested structure to be greater than 1.4. The commercial EELVs were designed to structural FSs of 1.25. NASA has taken exception to NASA-STD-5001 for FSs of less than 1.4 for well-defined loads. The process involves looking at the load contribution (static versus dynamic) in assessing the required FS. For the purposes of

bounding the problem in assessing costs for modification of structure, the criteria was used that for a structure with margins of less than 0.05 for an FS of 1.25, redesign would be required for EELV. Margins were assessed for actual flight loads. Since the Atlas has not flown in the heavy configuration the 552 (5 m core with 5 solids) configuration was used for this assessment. **Table 6F-1** summarizes the results of the assessments.

Table 6F-1. Assessment of Costs for Structure Modification

		Subsystem: Structures –	Minimum Change	11
Requirement: Spec	and Standards: NA	ASA STD-5001 requires s	tructural factor of safety of 1.4; Atlas	s V structures
		ns based on 1.25 factor of		
Hardware	Modification	Comments	Change Description for Costing	Benefit
Payload Fairing	New system	PLF replaced by	PLF eliminated; Requires	Cost savings.
(PLF)		CEV.	redesign to accommodate Centaur and CEV.	
PLF Separation	New System	PLF replaced by CEV.	PLF eliminated; Requires redesign to accommodate Centaur and CEV.	Cost savings.
Centaur/Payload Adapter	New System	Requires redesign to accommodate CEV interface. Use 1.4 FS.	New design with 1.4 FS; Qual. test.	Provides interface to CEV.
Centaur US Tank – Cylindrical tanks, and forward, intermediate, and aft bulkheads.	Moderate	Beef up tank structure to increase margin. Consider aluminum instead of steel for non-pressure- stabilized tank.	Qual Test. Additional Quality Assurance (QA) measures, such as KPP tracking, tag end testing, additional inspection, and higher- level proof test to accommodate new loads and environments.	Increase shell thickness.
Interstage Adapter	Minor	Accept 1.25 FS. Assume analysis will show positive margins for validated new loads and heating environments.	Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and proof test to accommodate new loads and environments.	Avoids redesigning and requalification.
Forward LOX Skirt	Moderate	Beef up structure to increase margin.	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Increase shell thickness.
CCB LOX Tank	Moderate	Beef up tank structure to increase margin. Questionable FS applied to pressure plus ext loads combination.	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Increase shell thickness.
CCB RP-1 Tank	Moderate	Beef up tank structure to increase margin. Questionable FS applied to pressure plus ext loads combination.	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Increase shell thickness.
LOX/RP-1 Intertank	Moderate	Beef up structure to increase margin.	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Increase shell thickness.
Aft RP Skirt	Moderate	Accept 1.25 FS. Assume analysis will show positive	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional	Avoids redesigning and requalification

A TEC	Malant	margins for validated new loads and heating environments.	inspection, and higher-level proof test to accommodate new loads and environments.	1.11
ATS	Moderate	Beef up structure to increase margin	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Increase shell thickness.
Solid Rocket Motors	Minor	Accept 1.25 FS. Assume analysis will show positive margins for validated new loads and heating environments.	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Avoids redesigning and requalification.
Press Vessels	Minor	Accept design per MIL-STD-1522A. Assume analysis will show positive margins for validated new loads and heating environments.	Qual Test. Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Avoids redesigning and requalification.

As can be seen, much of the structure shows low margins and was assumed to need redesign for use in a human-rated system. Even where the structure was accepted, some effort for requalification was assumed to provide for the analysis to assess and prove adequate margin exists.

#### 6F.2.2 Atlas V HLV with New Upper Stage

The modifications for the Atlas V with a new upper stage are similar to those for the Atlas V HLV with existing upper stage. The booster modifications for MPS and structure are identical to those previously described. The avionics was assumed to be more in line with **Figure 6F-3**. Thus, the avionics uses much of the existing GN&C capability but integrates the LVHM system into the avionics system as opposed to the mission kit approach of **Figure 6F-2**. The SLOC estimate was taken to be more of a new system approach of about 103 to 267 KSLOC. The structural modifications for the booster were identical to those defined in **Section 6F.2.1.4**, **Structure**. The upper stage MPS and structure were assumed new. RL-10A-4-2 modifications are as described in **Section 6F.2.1.3**, **Second Stage Main Propulsion**.

#### **6F.2.3 Delta IV with New Upper Stage**

#### **6F.2.3.1** Avionics and Software

The current Boeing Delta IV avionics subsystem is shown in **Figure 6F-4**.

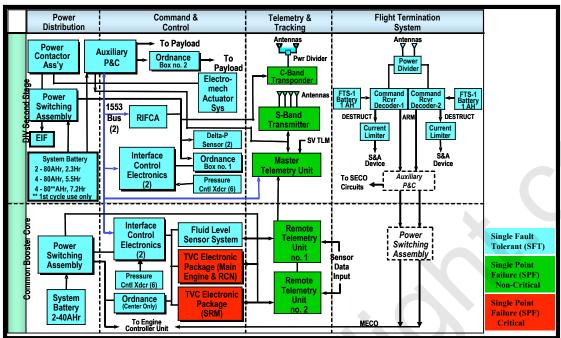


Figure 6F-4. Boeing Delta IV Avionics

The system is primarily single-fault tolerant already. One critical single-point failure is associated with the Booster TVC electronic package. An additional rate gyro electronics assembly on the booster was assumed to be required, although failure may not be critical, but could result in loads being exceeded because the rate gyro is used for load relief.

LVHM implementation was similar to the approaches previously discussed for the Atlas with new upper stage vehicle where the LVHM function was integrated into the LV avionics (**Figure 6F-3**). For the purposes of cost estimation SLOC estimates were considered the same as for the Atlas case with new upper stage.

#### 6F.2.3.2 Delta IV Booster MPS

The primary consideration for the Delta IV booster MPS was the upgrades for the RS–68 engine. The following upgrades were assumed:

- Engine Control Unit Upgrade:
  - o Redlines,
  - o In-flight lockup, and
  - o Health management.
- Redundant Main Chamber radial outward firing ignitor,
- Turbopump blade enhancements to eliminate cracking,
- GG igniter redesign to eliminate debris sources,
- Turbopump bearing tolerance modification,
- LOX post cracking.
- MCC liner cracking,
- Turbopump and start transient changes to reduce/eliminate excessive hydrogen fire on launch pad, and
- Redundant valve actuators.

The following enhancements were identified for the MPS:

- Incorporate pre-valves to allow termination of propellant flow on pad currently no pre-valves on Delta IV and any pad fire would continue until tank drain is complete (~120 minute).
- Redesign valve actuators and position sensors for redundancy current systems have various SPFs that must be eliminated.
- Incorporate redundant attach point separation nut or alternative current system employs single separation nut with two initiators with likely catastrophic failure modes.
- Incorporate redundant pneumatics for purges and actuation currently ground-based engine pneumatics has SPFs, which could be catastrophic during tanking operations; must also evaluate in-flight redundancy requirements.
- Reroute fuel bleed flows to reduce thermal effects on heat shield/ insulation – current fuel bleed lines are dumped overboard through the fuel bleed drain and contributes to excess burning of insulation on the heat shield.
- Upgrade valves/actuators for redundancy current systems have various SPFs that must be eliminated.
- Increase instrumentation required to incorporate health management suite current system has limited instrumentation.
- Evaluate component and subsystems for structural margins Increase structural margins to meet 1.4 FS requirement.

#### **6F.2.3.3 Upper Stage MPS**

The upper stage MPS was assumed to be a new design utilizing RL-10A-4-2 modified as discussed in Section 6F.2.1.3, Second Stage Main Propulsion.

#### 6F.2.3.4 Structure

The Delta IV structure was evaluated using the same procedure as described in **Section 6F.2.1.4**, **Structure**. As flown margins of the Delta IV heavy booster were used for this assessment. The upper stage structure was all assumed new. **Table 6F-2** summarizes the results of the structural evaluation.

**Table 6F-2. Delta IV Structural Assessment** 

Subsystem: Structures – Minimum Change								
<b>Requirement</b> : Spec and Standards: NASA STD-5001 requires structural factor of safety of 1.4; Delta IV designs based on								
1.25 FS requireme	1.25 FS requirement.							
Hardware	Modification Comments Change Description for Costing Ben							
First Stage Propellant Tanks	Minor	Accept 1.25 FS. Assume analysis will show positive margins for validated new loads and heating environments.	Additional Quality Assurance (QA) measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Avoids redesigning and requalification.				
First Stage Intertank	Moderate	Beef up intertank structure to increase	Additional QA measures, such as KPP tracking, tag end testing,	Increase shell thickness.				

Structures		margin.	additional inspection, and higher- level proof test to accommodate new loads and environments.	
LOX Skirt	Moderate	Beef up structure to increase margin.	Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Increase shell thickness.
LH2 Skirt	Minor	Accept 1.25 FS. Assume analysis will show positive margins for validated new loads and heating environments.	Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Avoids redesigning and requalification.
Engine Supports	Minor	Accept 1.25 FS. Assume analysis will show positive margins for validated new loads and heating environments.	Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Avoids redesigning and requalification.
Solid Rocket Motors	Minor	Accept 1.25 FS. Assume analysis will show positive margins for validated new loads and heating environments.	Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Avoids redesigning and requalification.
Pressure Vessels	Minor	Accept design per MIL-STD-1522a. Assume analysis will show positive margins for validated new loads and heating environments.	Additional QA measures, such as KPP tracking, tag end testing, additional inspection, and higher-level proof test to accommodate new loads and environments.	Avoids redesigning and requalification.

For the booster stage only the LOX skirt was shown to have low margins and need redesign, based on the acceptance criteria of margins greater than 0.05 for an FS of 1.25.



# Section 6G Candidate Vehicle Subsystems

## Appendix 6G

## **Candidate Vehicle Subsystems**

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#### **Booster and Upper Stage Propulsion Assessment Summary**

#### 6G.1 Introduction

This appendix serves as an adjunct to the Crew Launch Vehicle (CLV) Recommendation section, which discusses the four-segment Solid Rocket Booster (SRB) with the RS–25 Space Shuttle Main Engine (SSME) as the upper stage engine (designated as LV 13.1). Described below are propulsion subsystems for potential boosters and upper stages based on analysis parameters, such as the ability to meet performance requirements (including Factors of Safety (FS)) and the mandate to launch the first Crew Exploration Vehicle (CEV) mission to the International Space Station (ISS) no later than 2011.

## **6G.2** Evolved Expendable Launch Vehicle (EELV) Booster and Upper Stage Assessments

#### 6G.2.1 Atlas V Booster Main Propulsion System (MPS)

#### Characteristics are:

- Common booster core with 1 RD–180 engine,
- Provides 850k pounds of force (lbf) thrust times 3,
- Common Booster Core (CBC) MPSs are identical, and
- Cores throttled to 60% for outer core separation, then center core throttled to 100%.

#### Human-rating enhancements include the following:

- Currently, there are no pre-valves on the Atlas V, thus any pad fire would continue until tank is drained. Incorporate pre-valves to allow termination of propellant flow on pad.
- Current systems have various Single-Point Failures (SPFs) that must be eliminated. Upgrade valves and actuators for redundancy.
- Current system has limited instrumentation, required increase instrumentation to incorporate health management suite.
- Evaluate component and subsystems for structural margins; increase margins to meet 1.4 FS requirement.
- Upgrade fill-and-drain valves for redundancy. Current design has a single poppet seat. Due to large size, seat leak or failure could cause inability to meet performance requirements and/or explosion risk from external leakage.
- Redesign pogo suppressor system to allow for the deletion of pyrotechnics and tubing, and for the elimination of baffle spot-weld cracking issues. This reduces system complexity and failure modes, and could better protect harnesses and tubing.

1

#### **6G.2.2** Modified Existing Upper Stage

Characteristics: Utilizes RL-10A-4-2 or RL-10B-2.

Safety and reliability issues:

- Documentation required to meet human-rating certification, including updated Failure Modes and Effects Analysis (FMEA), Critical Items List (CIL), fracture control plan, etc.
- NASA Procedure and Guideline (NPG) 8705.2A requires system health management above traditional redline monitoring.
- Incorporate Air Force reliability enhancements along with human-rating modifications identified to meet NPG 8705.2A.
- Potential issues with structural margin requirements, with a 1.4 FS.

Development path and issues:

- Time-consuming development path for elimination of the SPFs.
- Additional reliability improvements may be required beyond the human-rating minimum.

Production and obsolescence: Technology is mature for human-rating and reliability upgrades.

#### Risks:

- Requalification required for human-rating and reliability upgrades.
- Potential for interference with current EELV schedules during development, qualification, and certification.

The following human-rating enhancements are needed (applies to both Centaur and Delta IV upper stage):

- Currently, there are no pre-valves on the Centaur upper stage and certain failure modes could warrant incorporating pre-valves.
- Current systems have SPFs that must be eliminated. Upgrade valves and actuators, press bottle, manifolding, and valves and components for redundancy.
- Potential pneumatic system upgrades.
- Upgrades for redundancy in Reaction Control System (RCS).
- Current system has limited instrumentation. Increase instrumentation required to incorporate health management suite.
- Evaluate component and subsystems for structural margins. Increase margins to meet 1.4 FS requirements.

#### 6G.2.3 Delta IV Booster MPS

#### Characteristics:

- Has common core booster with 1 RS-68 engine.
- Provides 650k lbf thrust times 3.
- Common core booster MPSs are identical.
- Center core is throttled for outer core early staging.

#### Safety and reliability issues:

- Documentation required to meet human-rating certification, including updated FMEA, CIL, fracture control plan, etc.
- NPG 8705.2A requires system health management above traditional redline monitoring.
- Incorporate Air Force reliability enhancements along with human-rating modifications identified to meet NPG 8705.2A.
- Potential issues with structural margin requirements, with a 1.4 FS.

#### Development path and issues:

- Estimated 4–5 year development path for elimination of SPFs.
- Additional reliability improvements may be required beyond the human-rating minimum

#### Production and obsolescence:

- The vendor base for development of large-scale propellant pre-valves has diminished.
- The current Shuttle design could be leveraged.
- Technology is mature for human-rating and reliability upgrades.

#### Risks:

- Development time for large-scale pre-valves may not meet required milestones.
- Requalification required for human-rating and reliability upgrades.
- Potential for interference with current EELV schedules during development, qualification, and certification.

#### 6G.3 Assessment of EELV Schedule and Schedule Risks

#### 6G.3.1 Delta IV HLV

**Table 6G-1** summarizes the schedule and risks associated with the use of a Delta IV heavy-lift CLV. The acquisition approach for this vehicle will consist of a new Request for Proposal (RFP) with Government-led integration for startup.

This vehicle will utilize RL-10 engines for a new upper stage and RS-68 engines for the boost stage. The critical path item for this vehicle is the development of a new MPS for the upper stage and human-rating of the engines. The Delta IV vehicle has flown; therefore, flight environment information can be utilized in the new design. The development schedule for this vehicle predicts approximately 5 years from contract award to the first human flight launch in early 2012.

Table 6G-1. Schedule Risk to Human Flight for Delta IV HLV with 4 RL-10 Engines

Area	Risk Level	Comments/Action
Acquisition	High	Government-led integration for startup and utilization of non-human-rated design process, practices, and culture.  New RFP required, with major obstacles for proprietary/data rights for application to human rating.
Risk Adverse Culture	High	Assumed development program would proceed immediately after failure without delay, otherwise add at least 12 months minimum to schedule for investigations.
Vehicle Human Rating	Medium	Non-human-rated design process, practices, and culture.  Structural assessment indicates that most parts are greater than 1.3 FS (1.4 FS required).  Delta IV HLV has flown, so environments are better characterized.
RD-68 Human Rating	Low	Limited Government knowledge of detailed engine design and processes.
New 4 RL—10 Upper Stage and RL—10 Human-Rating Upgrades.	Low	Relatively minor modifications required for human rating.  New MPS required for 4-cluster upper stage.

#### 6G.3.2 Atlas V HLV

**Table 6G-2** summarizes the schedule and risks associated with the use of an Atlas V heavy-lift CLV. The acquisition approach for this vehicle will also consist of a new RFP with Government-led integration for startup. This vehicle will utilize RL-10 engines for the upper stage and Americanized RD-180 Russian engines for the boost stage. Since the RD-180 engine is a Russian-made engine, International Traffic in Arms Regulation concerns must also be addressed as part of the acquisition process for this engine.

The critical path item for this vehicle is the development, Americanization, and certification of the RD–180 engine. Since the RD–180 is a foreign-made engine, there is limited Government knowledge of the detailed design and processes for this engine. New production facilities for the Americanized version of this engine will also have to be evaluated. The development time for this engine is approximately 5 years. The Altas V

vehicle is still in the design process and has not flown. The development schedule for this vehicle predicts approximately 6 years from contract award to the first human flight launch in late 2012.

Table 6G-2. Schedule Risk to Human Flight for Atlas V

Area	Risk Level	Comments/Action
Acquisition	High	Government-led integration required for startup and utilization of non-human-rated design process, practices, and culture.
		Utilization of foreign-produced engine with new RFP required and major obstacles for proprietary, International Traffic and Arms Regulations (ITAR), and licensing agreements for human-rating application.
Vehicle Human Rating	High	Non-human-rated design process, practices, and culture.
		Structural assessment indicates that most parts are not greater than 1.3 FS (1.4 FS required).
RD-180 Americanization	High	New production facilities with large-scale hardware production with new materials and processes, coupled with application of human-rating requirements.
		Limited Government knowledge of detailed engine design and processes.
Risk Adverse Culture	High	Assumed development program would proceed immediately after failure without delay, otherwise add at least 12 months minimum to schedule for investigations.
Atlas V HLV Systems Engineering and Integration	High	3-core heavy configuration design is incomplete and has never flown.
RD–180 Initial Engine Production Rate	High	Currently no production, with projections at 6 units per year after co-production transition is complete.
Certification Flight Conducted With Non- Human-Rated Russian Engine	Medium	Using Russian engine decreases schedule to flight certification, which mitigates risk for overall vehicle, but engine/vehicle avionics are not certified.

Some common items of note regardless of which vehicle is utilized include that effort must be put in place to human rate the launch system (i.e., changes in design processes, practices, and culture). Also, design and testing failures must be investigated and solved in a swift manner to avoid delays in the program schedules.

#### 6G.4 Booster and Upper Stage Propulsion Options

#### **6G.4.1 Booster Stage Propulsion Summary**

Eight booster stage engine options were studied, including:

- RSRM-5,
- RSRM-4,
- RS-68U,
- RS–68,
- RD-180,
- RS-25d (Block 2 SSME),
- RS–25e (SSME), and
- RS-25f (SSME).

**Table 6G-3** gives a brief description of each booster stage propulsion option studied, along with the development time estimate and projected nonrecurring and recurring costs.

Table 6G-3. Booster Stage Propulsion Option Comparisons

Boost Propulsion	RSRM -5	RSRM-4	RS-68U	RS-68	RD-180	RS-25d	RS-25e	RS-25f
Description & Status (Thrust/ Specific Impulse (Isp))	Solid 3.5M lbf 264 sec	Solid 3M lbf 268 sec	Liquid Oxygen/ Liquid Hydrogen (LOX/LH2) Gas Generator (GG) 825k lbf 442 sec	LOX/L H2 GG 758 lbf 409 sec	LOX/RP-1 Oxygen- Rich Staged Combustion (ORSC) 869k lbf 338 sec	LOX/LH2 Fuel-Rich Staged Combustion (FRSC) 512k lbf 452 sec	LOX/LH2 Minimal Mods	LOX/LH2 Major Mods Expendable
Development Time	4 yr	Existing	4–5 yr	3–4 yr	5 yr	Existing	5 yr	7 yr
Nonrecurring Costs	\$800M	\$150M	\$250M - \$350M	\$150M - \$300M	\$700M	Existing	\$191M upgrades	\$292M upgrades
Recurring Costs	\$82M	\$80M	TBD	\$14M	\$25M — \$30M	\$2.9M Existing	\$53M	\$46M

## **6G.4.2** Upper Stage Propulsion Summary

A number of upper stage engine options were evaluated, including:

- RL-10,
- LR-60,
- LR-85,

- LR-120,
- J–2Sd,
- J-2+,
- RS-25d,
- RS–25e,
- RS-25f, and
- MB-60.

The key tradeoffs for the second stage and Earth Departure Stage (EDS) propulsion were primary configuration decisions that affect the vehicle stage propulsion system definition, hence, define the development cost, schedule, and risk. Trades included the following:

- Engine Trades:
  - o Single Engine:
    - **230k** to 265k,
    - 400k, and
    - 490k.
  - o Multiple Engines:
    - 22k to 25k,
    - 60k,
    - 85k, and
    - 120k.
- Control and Fault Detection, Isolation, and Recovery (FDIR) System:
  - o Vehicle propulsion controller, and
  - o Engine controllers.
- Repressurization:
  - Autogenous, and
  - o Heated Helium.
- Thrust Vector Control (TVC):
  - o Hydraulic,
  - o Pneumatic,
  - o Electromechanical actuators,
  - o Electrohydraulic actuators, and
  - o Electropneumatic actuators.

Within upper stage main engines, the key trade is use of a single engine versus multiple engines, and within multiple engines it is if and how engine-out is incorporated. All thrust levels identified do not necessarily apply to all vehicles; the Launch Vehicle (LV) descriptions show the number of engines and thrust levels for each vehicle configuration concept. For example, the multiple 22k to 25k thrust engine configurations apply only to new EELV (Atlas V and Delta IV) upper stages, and the higher thrust engines apply only to the Shuttle-derived concepts.

No formal tradeoff studies were performed for the Control and FDIR System, Repressurization, and TVC subsystems. For the LV configurations under consideration, these trades are not significant discriminators and, therefore, are not discussed.

The Reusable Solid Rocket Motor- (RSRM-) based CLV concepts were generated with derivatives of the J–2 and RS–25 engines. The J-2-based engine can be divided into two categories—Gas Generator (GG) engines (J–2 as flown on the Saturn S-II and S-IVB stages) and tap-off cycle engines (J–2S as was in development at the end of the Apollo Program). Use of the original J–2 engine, recreating the engine as it existed for flight in 1970, with modifications only as absolutely required due to electronic obsolescence, was determined to have insufficient performance and known problems with its pressurized gas system for on-orbit restart. The engine team evaluated creation of a new GG engine, based upon the turbomachinery designed for the J–2S engine and most recently used in a GG cycle for the X–33 linear aerospike engine in the 1994 to 2001 time period. Such a configuration would have higher thrust and specific impulse than the original J–2 engine, but would require new designs for the main injector, thrust chamber assembly, nozzle, and integrated engine system.

#### 6G.5 RD-180 Booster Engine Assessment

The RD–180 (see **Figure 6G-1**) is an ORSC cycle, 930k lbf thrust engine that burns Liquid Oxygen (LOX) and kerosene propellants. It is a two thrust chamber derivative of the four-chamber RD–170 and is currently being flown on the Lockheed Martin Atlas III and Atlas V LVs. It was developed by NPO Energomash (NPOE) as part of a joint venture with Pratt & Whitney, known as RD Amross. It packages the high-performance, operability, and reusability features of the RD–170 in a thrust class more compatible with U.S. booster propulsion needs.



Figure 6G-1. RD-180 Booster Engine

The RD–180 is a self-contained propulsion unit that permits simplified engine/vehicle interfaces—with hydraulics for control valve actuation and thrust vector gimbaling, pneumatics for valve actuation and system purging, and a thrust frame to distribute loads, all located within the engine envelope. The ORSC cycle used by the engine delivers a 10 percent performance increase over current operational U.S. booster engines and provide clean reusable (certified for 7 starts) operation. It is capable of thrust (40 percent to 100 percent continuous throttling) and mixture ratio control. The throttling capability enables real-time trajectory matching and on-pad engine checkout before launch commit. Control of the mixture ratio permits ascent trajectory optimization and minimization of residual propellants at engine cut-off.

#### 6G.5.1 Assessment of the RD-180 as a Human-Rated Booster Engine

The RD-180 can be human rated and Americanized (i.e., coproduced) for crewed launches on the heavy-lift launch configuration of the Atlas V booster system utilizing three common core boosters. Pratt & Whitney can begin flight engine production as early as January 2008 for delivery in mid-2010, with risk acceptance of successful completion of Module 3 certification (explained in Section 6G.5.3, RD-180 Coproduction Requirement Summary) if authority to proceed is received by August 2005.

With regard to coproduction, a proposal for Americanization of the RD–180 has been submitted to Lockheed Martin by Pratt & Whitney (approximately \$500M and 54 months to complete U.S. coproduction certification). Certification would be for Pratt & Whitney to produce the Atlas V flight-certified RD–180 system only. A human-rated RD–180 derivative could be developed by Pratt & Whitney in parallel with the baseline coproduction effort, provided all licensing agreements with the government of Russia and NPOE are covered. More extensive modifications will require engine recertification and will degrade the experience base with the baseline engine configuration. The current Rough Order of Magnitude (ROM) recurring cost for a human-rated RD–180 derivative is \$25M to \$30M. The proposed 54-month coproduction schedule is constrained by the critical path at medium risk and cannot be effectively compressed without additional risk. Current Pratt & Whitney production is projected at 6 units per year, which can be expanded as needed to support the planned launch manifest, to as many as 12 to 18 units per year.

#### 6G.5.2 RD-180 Development Path and Issues

The RD–180 has several development path issues including obstacles to human-rating-driven design changes. Since NPOE is the current design agent, it has final approval authority for proposed design changes. Export constraints prevent Pratt & Whitney and NPOE from fully collaborating on design changes. As stated earlier, Pratt & Whitney could spin off a human-rated RD–180 derivative in parallel to the coproduction effort, provided all licensing agreements with the government of Russia and NPOE are covered.

Near-term strategies for resolving these obstacles include directing NPOE to implement hardware reliability enhancements and providing NPOE with specific requirements for a human-rated RD–180. In addition, Pratt & Whitney could develop a passive background health monitoring system using development and production acceptance and flight data and component FMEA.

#### 6G.5.3 RD-180 Coproduction Requirement Summary

A major requirement that would be imposed on the RD–180 is for American coproduction capability in addition to that of NPOE. This is a requirement levied by the National Reconnaissance Office (NRO) to support launches of national security payloads. The objective of coproduction would be to certify Pratt & Whitney to produce the RD–180 system certified for use on the Lockheed Martin Atlas V LV. This would be organized into three modules:

- Module 1: Receipt and translation of all RD–180 design data, which has been completed.
- Module 2: Demonstration of manufacture of critical components utilizing the transferred data and demonstrating capability to coproduce hardware, which is in work. Module 2 success criteria include fabrication of an RD–180 preburner/stator assembly and successful hot-fire testing on an RD–180 production unit in Russia.
- Module 3: Complete establishment of full U.S. coproduction with Pratt & Whitney as the qualified domestic source, for which a proposal has been submitted to the customer (Lockheed Martin). Module 3 success criteria includes performing 10 development engine tests, including 3 to 5 stand-correlation tests plus 5 to 7 tests swapping out coproduced components as they are fabricated. Two-engine qualification would be at a rate of 5 to 7 tests per engine, with an initial capability of 6 engines per year.

#### 6G.5.4 RD-180 Production and Obsolescence

The RD–180 engine leverages from RD-170 ancestry with an infusion of current technology. Currently, the RD–180 is produced in Russia and represents the current state-of-the-art in ORSC-cycle LOX/kerosene engine technology. A total of 28 production engines have been built (26 delivered). NPOE current production rate is 8 per year and is expandable to 12 per year. Recurring engine cost is approximately \$10M per unit. Pratt & Whitney planned production is 6 per year with a recurring engine cost of approximately \$25 million per unit.

#### 6G.5.5 RD-180 Risks, Opportunities, and Watches

Risks, Opportunities, and Watches (ROW) for the RD–180 engine are provided in **Table 6G-4**.

Area	ROW	Notes
Human-Rating Design Modifications	Risk	NPOE has final approval authority for proposed design changes. All reliability improvements for human-rating certification will have to be performed by NPOE.
Beginning Pre- Certification Production	Risk	Achieving the 2011 first flight goal would require insertion of the human-rated derivative RD–180 into the fabrication pipeline prior to completion of the Modification 3

Table 6G-4. RD-180 Risk Summary

		certification.
Data Access	Watch	Shallow engine system technical penetration by NASA; high analysis uncertainty and coarse model fidelity.
Data Access	Watch	No NASA insight into ongoing coproduction effort. May enable possible process escape with critical consequences to human-rating effort.
Data Access	Watch	No NASA insight into human-rating modifications or certification implementation.
Data Access	Watch	Currently, all testing being conducted in Russia. No insight into test and acceptance procedures.
Partial Coproduction	Opportunity	Possibility exists for saving schedule and budget by completing coproduction on critical and/or long-lead components and utilizing existing noncritical elements (i.e., fasteners, brackets, etc.)

#### 6G.6 RS-68 Booster Engine Assessment

This section discusses the RS–68 engine assessment, including human-rating considerations. The RS–68, seen in **Figure 6G-2**, was developed between 1995 and 2001 for the Delta IV EELV. The primary objective was to minimize development cost. Most of the original design team is still available to reduce modification risk. Development and certification is based on 183 starts and 18,945 sec. There have been no catastrophic test failures and no flight failures for six engines on four flights.



Figure 6G-2. RS-68 Booster Engine

Considerations for use in a human-rated system include safety factors and design margins. Redundancy issues include the engine controller and valve actuation. FDIR and health management system instrumentation and algorithms must be assessed. Cracking issues with turbine blades, Main Combustion Chamber (MCC) liner, and LOX posts have been noted and need to be assessed. In-flight safe shutdown would be by hydraulic lockup and pneumatic shutdown or redundant actuators. The GG igniter debris SPF also is a factor. The start transient would have to be modified and fuel pumps redesigned to significantly reduce the free hydrogen fires at engine start. Seals would also have to be added to the hydrogen system, including valves and actuators.

#### 6G.7 J-2S Upper Stage Engine Assessment

While the J–2S was one of the engine options considered, it was not recommended as a viable option for the following reasons:

- Flight engine delivery was no better than other more producible, lower-risk J–2 engine-based options due to the need for design cycles prior to production and antiquated component design, especially the tube wall chamber and nozzle.
- The predicted cost, while lower than other options, was a high-risk area due to the lack of reliable design definition.

#### 6G.7.1 History of the J-2 Engine

The original J–2 engine was developed in the early 1960s for the Saturn IB (S-IVB Stage one engine, flown nine times) and Saturn V (S-II Stage five engines and S-IVB Stage one engines – flown 13 times) vehicles. The Liquid Hydrogen (LH2) engine, developed by Boeing's Rocketdyne division, had the following design goals:

- High reliability,
- Efficient packaging of component,
- Restart capability at altitude,
- LOX tank pressure maintained by heat exchanger on Oxygen (O2) pump exhaust duct.
- 200k to 230k lbf thrust,
- 5.5:1 oxidizer/fuel ratio, and
- Vac Isp: 423 to 427 sec.

The J–2 (see **Figure 6G-3**) was tested at the Santa Susana and the Army's Arnold Engineering Development Center (AEDC) J–4 test sites, with a total of 500 sec of burn time and 87 engines flown.

Figure 6G-3. J-2 Engine Schematic

#### 6G.7.2 J-2S (Simplified) History

Rocketdyne developed a simplified version of the J–2 (identified as the J–2S) later in the J–2 Program Phase as a sustaining engineering effort. The J–2S was also a LH2 engine with the design goals of:

- An open-loop, throttle 6:1 ratio,
- 5,000 lb idle mode,
- Reduce thermal conditioning,
- Eliminate propellant bleed, sub-cooled propellant requirements, and a start transient,
- Increase thrust and Specific Impulse (Isp),
- Improve component accessibility, and
- Same interface as the J–2.

The J–2S (depicted in **Figure 6G-4**) was a totally different engine than the J-2, designed with an almost-common vehicle interface. Six development J–2S engines were produced in the late 1960s, three with water-cooled MCC and 2:1 nozzle, and three with full 40:1 nozzles. The design featured a unique tap-off cycle, where pressure is tapped off the MCC. These gases drive the turbines, and the need for a GG is eliminated. The J–2Swas tested at Santa Susana and AEDC J–4 test sites, with a total of 21,000 sec at main stage and 6,500 sec at idle mode. While the tap-off cycle engine (the only one of its kind ever built) had its benefits, it also had design problems that were identified and addressed, specifically tap-off lip erosion, tap-off manifold burnout, idle mode problems, and high-frequency MCC instability. The engine was never flown.

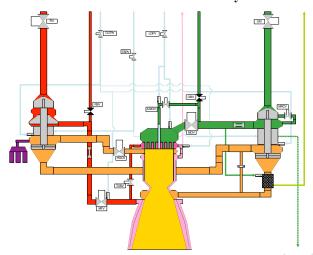


Figure 6G-4: J-2S Engine Schematic

The 60-Day Study team reviewed multiple documents in its evaluation of the J–2S engine. Chief among those was a study conducted by Rocketdyne in 1993 that considered the possibility of restarting the J–2S Program. The Study Report, NAS8-39210 (April 1993), stated a concern that, among other things, the tap-off port had production and durability concerns, and the high-frequency combustion chamber oscillations (4,400 Hz) that did not occur on all engines or tests. The latter, which was determined to be a problem that may have been completely characterized and the cause accurately identified, may not have been totally resolved from the design perspective. In addition to the 1993 Report, the 60-Day Study team also reviewed AEDC Test Reports and other original J–2S documentation that support these findings.

The Study team also found the issue of incomplete drawings and documentation to be a primary concern. The 1993 Rocketdyne Study reported missing valve drawings for the original J–2S design. Subsequently, the fact that those drawings have still not been located as of 2005 poses a major problem as well as the fact that configuration management data on existing drawings does not exist. Tooling drawings, if they exist, and component design drawings would possibly contain antiquated processes and design. All of this leads to the conclusion that an engine design cycle would be necessary before production could start. The only time saved would be that of system-level definition and taking pump development of critical path. The fact that a current pump design exists, however, significantly reduces development cost and risk.

## 6G.7.3 Other J-2 Configurations Considered: J-2Sd and J-2+

Two additional variations on the J–2S design (i.e., the J–2Sd and J–2+) also were considered as engine options for the 60-Day Study. Each could be equipped with nozzle extensions to increase Isp to 450 sec. At this point, these variations are both strictly theoretical designs except for the actual pumps that would be used for each.

The J–2Sd is a derived J–2S engine concept based on J–2S cycle legacy and existing J–2S Mk29 fuel and oxidizer pumps. The J–2Sd can use the pumps as either a GG or a tap-off cycle engine. It has the following characteristics:

- New tap-off design with Mk29 pumps, new state-of-the-art injector, chamber, and nozzle,
- Cycle: Tap-off,
- Thrust Class: 250k lbf, and
- Vac Isp: 436 to 450.

Potential problems with this engine design include: low Technology Readiness Levels (TRLs) in the state-of-the-art Thrust Chamber Assembly (TCA) and the implied schedule problems associated with low TRLs; multiple test stands; and the fact that the engine would be heavier than the original tube wall design.

This engine design ultimately was not chosen for near-term development for the following reasons:

- Not a viable schedule option for support of the 2011 launch date,
- Schedule and cost were comparable to the LR-85, which had lower catastrophic failure rate and higher performance (given weight and envelope constraints), and
- Tap-off port erosion concerns.

The J–2+ engine is a new GG design using existing Mk29 pumps and a new state-of-theart injector, chamber, and nozzle, and leveraging linear aerospike engine technology from the XRS2200. Specific characteristics include:

- Cycle: Gas Generator,
- Thrust Class: 250k lbf, and
- Vac Isp: 430 to 450.

Potential problems with this engine design are the same as those associated with the J-2Sd: (1) low TRLs in the state-of-the-art TCA, and the implied schedule problems associated with low TRLs; (2) multiple test stands; and (3) the engine would be heavier than the tube wall. This engine design ultimately was not chosen for near-term development for the following reasons:

- Not a viable schedule option for support of the 2011 launch date, and
- Schedule and cost were comparable to the LR-85, which had a lower catastrophic failure rate and higher performance (given weight and envelope constraints).

The J–2+ plots shown in **Figures 6G-5** through **6G-7** depict an engine with a constant chamber design and power-head. Area ratio changes are a function of nozzle size only.

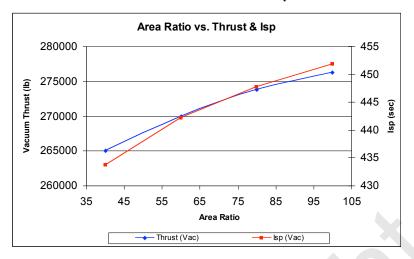


Figure 6G-5. J-2+ Area Ratio versus Thrust and Specific Impulse



Figure 6G-6. J-2+ Area Ratio versus Weight and Thrust-to-Weight (T/W) Ratio

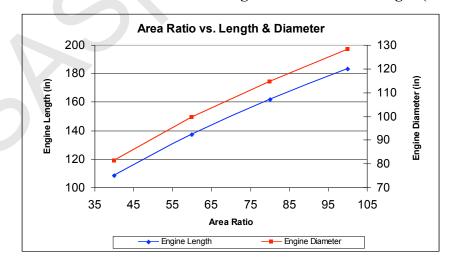


Figure 6G-7. J-2+ Area Ratio versus Length and Diameter

The J–2S plots in **Figures 6G-8** through **6G-10** depict an engine with a constant chamber design and power-head. Area ratio changes are a function of nozzle size only.

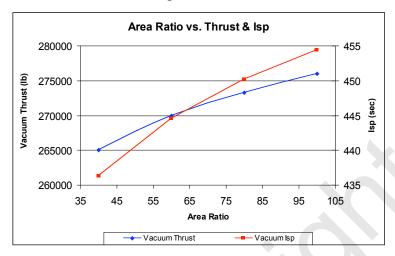


Figure 6G-8. J-2S Area Ratio versus Thrust and Specific Impulse



Figure 6G-9. J-2S Area Ratio versus Weight and T/W

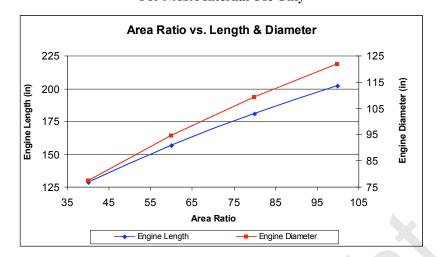


Figure 6G-10. J-2S Area Ratio versus Length and Diameter

## 6G.8 RL-10A-4-2 Upper Stage Engine Assessment

The RL-10A-4 was the first of the RL-10A-4-derivative engines to be qualified. The RL-10-4 was qualified for flight in May 1991. This engine's reliability factor was 100 percent after 180 engines and 296 firings in space. The first in-flight failure occurred on AC-70.

Subsequently, the RL-10A-4-1 derivative was qualified for flight in 1994, while the RL-10A-4-2 derivative was qualified for flight in 2000.

Major RL-10 accomplishments through November 2004 include:

- 190 missions (3 failures),
- 368 engines in space,
- 693 firings in space,
- 14,221 total hot-firings,
- Over 2.1M sec of hot-fire time,
- 10 qualifications completed, and
- Demonstrated reliability of 0.9981.

The RL-10-4-2 upper stage engine is shown in **Figure 6G-11** and performance data is shown in **Table 6G-5**.



Figure 6G-11. RL-10-4-2 Upper Stage Engine

Table 6G-5. RL-10-4-2 Performance Data

Thrust <sub>vac</sub>	22,300 lbf
Chamber Pressure	610 psia
Isp	451 (MR = 5.5)
Weight	375 lbm
Propellants	LOX/LH2
Nozzle Area Ratio	84:1
T/W	59
Mixture Ratio Control	4.8 to 5.8
Throttle	100% ONLY
Length	90 in
Diameter	46 in
Reliability	≈ 0.998 (90% Confidence)

Engine health and status data requirements was an area of consideration that was identified during the Pratt & Whitney and Air Force Reliability Enhancement Program, which addresses issues such as:

- Long-duration, multiple burns while in extreme environments (severe radiation, temperatures, etc.),
- Dormant component failures during extended engine-off conditions,
- Electrical power management, and
- Throttling (and/or deep throttling).

Structural assessments also must be made to human-rate the RL-10-4-2. The position of Government experts is that engines must meet more severe structural requirements for human rating, such as:

- NASA-STD-5012 (Strength and Life),
- NASA-STD-5007 (Fracture Control), and
- MSFC-RQMT-3019 (Launch Vehicle Qualification Requirements).

## 6G.8.1 RL-10A-4-2 Development Path and Issues

The major issue with using the RL-10A-4-2 arises from the perspective of engine modifications for use on a human-rated system. These modifications include:

- Valve actuation redundancy,
- Turbopump enhancements,
- Ignition enhancements,
- Enhanced thrust chamber cooling,
- Redesign for controls redundancy, full FDIR capability, and health monitoring development:
  - Add controller channel redundancy and cross-strapping (vehicle- or engine-mounted), and
  - Assess sensor needs for full FDIR and health monitoring development,
- Safe in-flight shutdown mode and valve lock-up mode to enable abort,
- Different operating point—Assess for performance, dynamic, and throttling issues (e.g., engine-out philosophy), and
- Design changes required as a result of the structural assessments (e.g., fracture control, strength, etc.).

#### 6G.8.2 RL-10A-4-2 Production and Obsolescence

Currently there are no Bill of Material production or obsolescence issues. One possible exception is that the production facility is limited to producing no more than 50 engines per year. The limiting factor is the TCA braze furnace cycle time. Pratt & Whitney has

determined that the optimum number of engines per year is 25. The actual maximum was 33 engines in a single year during the mid-1990s.

## 6G.8.3 RL-10A-4-2 Test Issues

Two major issues that require consideration are test-stand modifications and an integrated stage test capability. Test-stand modifications would be required for the upgraded RL–10A–4–2, as well as the engine processing modifications. These modifications are driven mainly by human-rating requirements. It should be noted, however, that program and mission requirements may be drivers to the overall set of modifications (including those of the engine).

The integrated stage testing capability requires further assessment. For example, in the context of the selected stage configuration, the existing capabilities must be understood, along with the modifications required. These then feed into cost, schedule, and requirements.

#### 6G.8.4 RL-10A-4-2 Cost and Schedule Data

Given below are Government estimates of a 52–60 month schedule from authority to proceed, with a cost of \$80M to \$100M, based in part on Pratt & Whitney and Lockheed Martin cost and schedule data and on the following RL–10A–4–2 modifications:

- All modifications identified by Pratt & Whitney and Lockheed Martin.
   Exceptions exist in cases where this list identifies more stringent redesign of the system.
- Redesigns for additional controls redundancy; for example, actuation redundancy for all control valves on the RL-10A-4-2.
- FDIR, at least to the extent required by Program, Mission, and System Requirements:
  - Controller channel redundancy and cross-strapping (vehicle or engine trade), and
  - Additional sensor and instrumentation needs.
- Health Monitoring:
  - Controller channel redundancy and cross-strapping (vehicle or engine trade), and
  - o Advanced instrumentation using full redundancy.
- Engine redlines.
- Safe in-flight shutdown modes.
- Valve lock-up mode to enable abort.
- Assess throttling (e.g., engine-out, de-rated).
- Assess upgrading to smooth-wall, copper-based MCC, which is necessary if required burn durations are significantly greater than current qualification level.

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- Design changes required due to structural assessments (e.g., fracture control, strength).
- Generically: All redesigns due to program, mission, and system requirements.

It should be noted that these cost and schedule estimates do not include integrated stage testing (i.e., hot-firing at the system level) or the associated analyses for such testing. This effort is estimated at \$190M ( $\Delta = $44 \text{ M}$ ).

## 6G.9 Expander Cycle Engine Development MB-60/LR-60 Assessment

The goal of expander cycle engine development with the MB–60/LR–60 is to take advantage of emerging engines (see **Figure 6G-12**). For the MB–60, Rocketdyne has produced the thrust chamber assembly (injector, MCC, and igniter) and successfully hot-fire tested it at full power. Demonstration assembly is planned for fall 2005. Pratt & Whitney developed the LR–50 in 1999, followed by the LR–60 demonstrator in 2001. The LR–60 injector has been successfully tested to 65k lbf. Performance data is provided in **Table 6G-6** below.



Figure 6G-12. MB-60/LR-60 Engines

Table 6G-6. MB-60/LR-60 Performance Data

Cycle	Open Expander
Propellants	LOX/LH2
Thrust (lbf)	90k

Isp, vacuum (sec)	467
Pressure Chamber (psia)	1979
Mixture Ratio	5.4
Area Ratio	454
Weight (lbm)	1,700
Throttle	Fixed
Restartable	Y (3+)
Reliability	Greater than 0.998
Design Starts	64
Design Sec	10k
Length (in)	155
Exit Diameter (in)	91

ROM cost estimates for Design, Develop, Test, and Evaluate (DDT&E) are \$250M to \$300M, which does not reflect human rating. Recurring costs are \$6M to \$10M.

Development issues include the following:

- Prototype engines with facility-type hardware,
- Engine was not originally designed to NASA's current standards,
- Failure/fault tolerance standards,
- Engine control system requirements,
- FDIR, and
- Engine health and status data requirements.

Overall, development risk is low for the MB-60/LR-60.

## 6G.10 MB-XX Engine Class (60k and 200k) Assessment

The MB–XX is a prototype developed by Rocketdyne/MHI. The thrust chamber assembly has been successfully hot-fired at full power. Fuel Turbopump (FTP) assembly is complete and ready for testing. A demonstration engine assembly is complete using the FTP simulator. The FTP export license is pending. The demonstration engine utilizes MHI-provided slave Oxygen Turbopump (OTP). Demonstration engine hot-fire testing is scheduled for summer 2005 at the Tashiro test facility in Japan. OTP design is complete

and long-lead orders will be placed in 2005. The complete OTP fabrication and component-level testing is scheduled for 2006. A flight-like demonstration engine will be assembled following OTP completion. Performance specifications are provided in **Table 6G-7**.

Table 6G-7. MB-XX Engine Performance Specifications

Cycle	Open Expander
Propellants	LOX/LH2
Thrust, v (lb)	60k
Isp, v (sec)	467
Pressure Chamber (psia)	1,979
Mixture Ratio	5.4
Area Ratio	300
Weight (lbm)	1,300
Throttle (upgradeable, idle mode)	Baseline Fixed
Restartable	Y (3+)
Reliability (Cat.)	0.99923
Length (in)	130
Exit Diameter (in)	74

Recurring cost estimates (in constant 2004 dollars) for the MB–60 are \$10M, based on 12 production units per year, with nonrecurring costs of \$385M. For the MB–200, recurring cost estimates are \$16M based on 6 production units per year, with nonrecurring costs of \$406M.

It should be noted that approximately 50 percent of engine components would be produced by MHI in Japan, with assembly and hot-fire testing conducted in the U.S.

## 6G.11 Upper Stage Development Drivers: LR-85 Upper Stage Engine

The LR-85 is a liquid rocket engine designed to deliver 85k thrust. This clean-sheet design does not rely on any existing hardware. The requirements and concept development needed to generate the RFP for this upper-stage option would be Government led. Additionally, an advanced development program initiated by the Government would lead to risk reduction for the contractor.

Subscale development would be used to anchor a full-scale design prior to the fabrication of certification engine hardware. Engine system-level testing would include the following engines and schedule:

- 4 development engines: 12 tests each at 2 to 3 tests per month,
- 2 qualification engines: 12 tests each at 4 tests per month (each), and
- 2 certification engines: 10 tests each at 6 tests per month (each).

The schedule dictates the need for access to two test stands, both with vacuum simulated capability. A total of 16 engines would be required from the onset of the Main Propulsion Test Article (MPTA) testing to the completion of the first human flight.

## 6G.11.1 LR-85 Description and Characteristics

The LR-85 is an expander cycle engine using LOX/LH2 as propellants. It would generate 85k pounds of thrust at altitude and a vacuum Isp of 450 sec (depending on the flight envelope). At nominal power level, the LR-85 nozzle stagnation chamber pressure is 1,103 psi.

As a new clean-sheet engine, the LR-85 would be designed to meet all upper stage and/or Earth Departure Stage (EDS) requirements. Component design-level risk mitigation would be drawn from contractor independent research and development and/or Air Force Research Laboratory Upper Stage Engine Technology Development.

An additional benefit to a clean-sheet design is direct application to a human-rated system. The LR-85 would be designed and developed from the start with human rating in mind. Thus, compliance with NPR 8705.2A would come without any additional costs.

By definition, the expander cycle is a relatively simple design. Reliability for this particular expander would be enhanced by the lack of boost pumps, the use of series turbines, and its full-flow design.

## 6G.11.2 LR-85 Issues and Risks

Accelerating the development schedule would require access to two test facilities: one being a simulated altitude test stand for the start transient, and the second being a diffuser-equipped stand for steady state testing. Examples of the types of facilities required include an E8-type test stand, along with Plumbrook (B2) or AEDC (J–4).

The primary technology challenge is the MCC (see **Figure 6G-13**). The LR–85 thrust level is three to four times greater than the previous experience base for an expander engine. Chamber walls with enhanced surface areas will be required to extract sufficient heat to drive the turbines, while still providing adequate cooling.

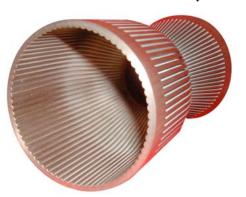


Figure 6G-13. Advanced Combustion Chamber with Enhanced Surface Area

A systematic approach and validation of analysis tools will be required for turbopump development. As with any new engine development, schedule is always a risk. This is compounded by the lack of clearly defined baseline requirements. On the positive side, the LR–85 design would benefit from the ability to utilize domestic production capabilities.

**Table 6G-8** captures four key risk areas and places them into ROW categories. Schedule stands out as the risk in greatest need of mitigation.

Area	ROW	Notes
Operating Point is 3–4 times Greater Than Experience	Watch	Advances in combustion chamber design and turbopump design analysis tools provide basis for design.
Weight	Watch	Engine concept weight models are inaccurate and engine weight is subject to component design decisions.
Schedule	Risk	Engine system fabrication in parallel with component testing prevents test results from being incorporated into design.
Requirement Definition	Watch	Early over-specification of requirements can increase complexity, leading to increased cost and decreased reliability.

Table 6G-8. LR-85 Key Risk Areas

## 6G.11.3 LR-85 Performance Data

Top-level performance data for the LR-85 is shown in **Table 6G-9**. This data was generated by PSTAR, a Marshall developed conceptual engine sizing and performance model.

Table 6G-9. LR-85 Top-Level Performance Data

Cycle	Expander
Propellants	LOX/LH2
Thrust, v (lb)	85k
Isp, v (sec)	450
Chamber Pressure (psia)	1,100
Mixture Ratio (nom)	5.5:1
Area Ratio	87:1
Engine Weight (lb)	1,500 ±150
T/W-SL	52-63
Throttling	TBD

Though the LR-85 is specified here with an area ratio of 87:1 and a vacuum Isp of 450 sec, more performance can be realized. The plot shown in **Figure 6G-15** shows how Isp on the order of 460 sec could be achieved with area ratios around 160.

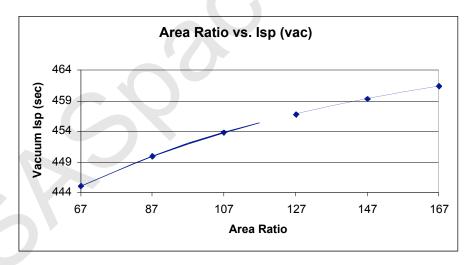


Figure 6G-15. LR-85 Area Ratios versus Calculated Specific Impulses

**Figures 6G-16** shows performance data for the LR-85 closed expander. Each data point represents a different design point. Thrust is held constant for all design points.

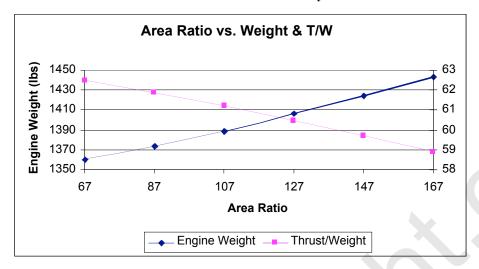


Figure 6G-16. LR-85 Closed Expander Area Ratio versus Weight and T/W

## 6G.11.4 LR-85 Cost Estimates

The total DDT&E cost estimate (in Fiscal Year 2005 dollars) is \$550M plus/or minus \$100k. The time estimated from authority to proceed to engine certification is approximately 5 years. Average unit cost is projected to be \$10M plus/or minus \$2M, based upon a production rate of 16 engines per year.

## 6G.12 Upper Stage Clean-Sheet Main Propulsion System

A clean-sheet upper stage design inherently carries more risk than utilizing a modified design. There are several key drivers for this increased risk. New systems, especially those that will be human rated, require extensive DDT&E prior to flight certification. Additionally, the existence of qualified hardware, as well as vendors to produce human-rated hardware, is limited.

Most current flight hardware being produced supports non-human-rated, expendable launch systems. While designs for human-rated components and subsystems exist, these primarily represent designs for reusable Shuttle systems and are not necessarily applicable for a clean-sheet expendable system. Furthermore, many of the original hardware vendors have been displaced or retired, the designs and fabrication rights have been bought and sold, and the design drawings are not currently up to date. Reconstitution of the vendor base for the production of human-rated designs is required.

However, a clean-sheet approach has many advantages. It can be designed for increased reliability as would be necessary to meet the human-rating requirements imposed by NPG 8705.2A. Extensibility could be built in to allow for commonality/growth without major redesign. State-of-the-art materials, hardware, and design, fabrication, and test techniques and processes would be incorporated facilitating a potentially better, more reliable system. A drawing of a clean-sheet upper stage design is shown in **Figure 6G-18**.

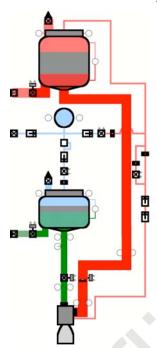


Figure 6G-18. Clean-Sheet Upper Stage Drawing

Clean-sheet development issues include: interstage space issues for multiple-engine configurations, structural tankage (bulk and structure of the stage), and pressure control of tankage (autogenous, onboard gas, etc.). Propellant conditioning requirements at the engine interface will drive increased complexity in the system.

The MPS represents the critical path in the development of a new upper stage system. The major MPS development path issues are feed lines and pre-valves, driven by system requirements, design cycle, long-lead procurement times, and lengthy fabrication and qualification times. System integration issues are also significant with a clean-sheet upper stage system. These include integration with both the booster stage and the payload/CEV. LV requirements that must be decomposed and allocated to the upper stage system drive the DDT&E of the upper stage system. Ability to meet upper stage and EDS goals introduces additional development risk. Based on upper stage selection, potential issues may be present for concurrent production to meet both CLV and EELV milestones.

Test and evaluation issues include the timely and costly test-stand modifications required to support integrated propulsion test article development, qualification, and certification. Limited availability of test positions may require series operations, increasing development time. Human-rated upper stage systems must be acceptance tested prior to delivery to the launch site for vehicle integration. This may highlight potential facility constraints and must be considered.

## 6G.13 Space Shuttle Solid Rocket Booster (SRB), Five-Segment SRB Derivative

More than 200 four-segment SRBs have been flown on the Space Shuttle Program. A total of 42 Shuttle Solid Rocket Motor (SRM) static test firings have been conducted, with 18 Reusable Solid Rocket Motors (RSRMs) and a single five-segment RSRM

design. Production is ongoing, and reusable assets are available for flight past 2016 with the current four-segment configuration.

The five-segment RSRM margin test (ETM-3) demonstrated DDT&E upgrade capability. Other enhancements may be required as intermediate block upgrades, but are not included in the cost assessments. For example, motor material (insulation) obsolescence and upgrading the TVC auxiliary propulsion unit system, which is currently 1970's vintage.

The development path for the five-segment SRB includes a 57-month time line projection from start to first flight hardware delivered leverages boosters from Shuttle heritage with infusion of current capability. Minimal reliability improvements are envisioned, with incorporation of a health monitoring capability only as part of avionics upgrades. However, only one test stand has five-segment SRB capability and only three tests may be conducted each year, per the Environmental Protection Agency (EPA).

Currently the four-segment SRB is in production at ATK Thiokol, and Nasa's Kennedy Space Center (KSC) has supported up to 19 motors per year (8 flight sets and 3 static tests). In addition, 6 production RSRMs have been built, with 23 additional available in the current contract scope. SRB hardware deliveries are set until Shuttle retirement in 2010, but can be extended. The five-segment SRB configuration would require near-term notification of vendor expectations. This approach requires minimal "keep alive" activities.

## Five-Segment Enhanced Performance SRB Overview

The goal of enhancing SRB performance with a five-segment configuration is to take advantage of an existing booster with added performance. **Figure 6G-19** shows a cross-section of a five-segment SRB with performance modifications and **Figure 6G-20** shows the resulting five-segment SRB performance.

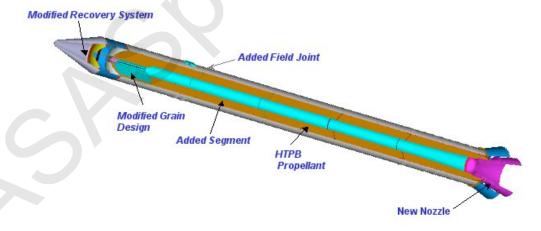


Figure 6G-19. Cross-section of a Five-Segment SRB with Performance Modifications

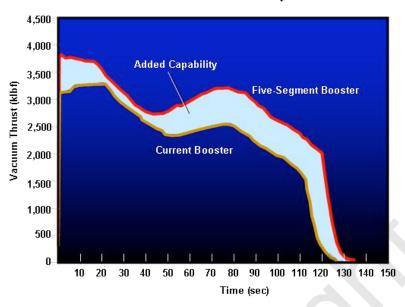


Figure 6G-20. Resulting Five-segment SRB Performance

Overall, development risk is low, based on utilization of existing assets and experience. Risk comes primarily from an aggressive DDT&E schedule. **Table 6G-9** shows additional ROWs for this approach.

Table 6G-9. Five-segment SRB Enhanced Performance ROW Chart

Component	ROW	Notes
System	Opportunity	Mature design, experienced staff, and existing test stands with 150+ 4-segment firings and 5-segment margin successfully completed.
Structures/Joints	Watch	Preliminary assessment allows for hardware migration based on margin.
Propellant/Ballistics/ Insulation	Watch	New propellant, HTPB, well-characterized (non-Shuttle Programs) with historical basis for grain design and validated models.  Chrysotile replacement may require additional testing for 5-segment certification.
Separation System	Watch	Currently qualifying ATK as new source for booster separation motor – design change to be determined.
Nozzle	Risk (Low)	Minimal delta design from current RSRM with development program in place for obsolescence if required (Shuttle).
Avionics	Risk (Low)	Replacement/upgrade of outdated parts and

		modification of systems tunnel, integrated electronic avionics, and ground support equipment.
Recovery System	Risk (Low)	Delta design with modern materials based on proven design techniques and testing.
TVC	Opportunity	Potential mitigation. Upgrades improve the flight safety, reliability, and operability over hydrazine.

## **6G.14 Engine Failure Estimates**

The percentages of risk reduction or increase were rolled up into an overall catastrophic reliability probability. The CFF was also generated from the expert comparative approach and presented in **Tables 6G10** through **6G-16**.

Table 6G-10. RD-180 Assessment

Component	% of Cat Risk Compar ed to SSME	Rationale	% of Catastrophi c Failures	% of Benign Failure s	Rationale
Actuators	200	Actuators do not have the redundancy that SSME actuators have.	10	90	Less redundancy.
Anti-Flood Valve (AFV)	0	This system does not have an AFV or equivalent. Set risk percentage to 0%.	N/A	N/A	
Fuel Preburner	0	Does not have fuel preburner.	N/A	N/A	
Fuel/Hot Gas System	100	Assume similar.	13	87	
Heat Exchanger	0	A heat exchanger failure is benign because it is GOX heating GHe.	1	99	Uses helium and tap- off gases.
HPFTP/AT	50	Recommend reducing to 40-50% because in addition to the "RP vs. H2" rationale, the single-shaft configuration prevents pump runaway and provides thrust balancing between the fuel and oxidizer pumps.	15	85	RP vs. H2
HPOTP/AT	50	Reduce by 50% since RD-180 has single shaft TPA and does not have the runaway LOX pump failure modes.	15	85	Pump runaway prevented; thrust balancing.
Igniters	10	Hypergolic igniters should be expected to be more reliable than ASIs.	1	99	Hypergol - no backflow issues.
LPFTP	100	Keep same, but no logic for doing so; doubt that a failure of the low pressure fuel boost pump could result in a catastrophic failure.	5	95	Not enough info.
LPOTP	150	GOX-drive low-pressure LOX boost pump might be more risk than SSME LPOTP.	61	39	Not enough info.
LTMCC	200	Double since the RD-180 has 2 thrust chamber assemblies.	33	67	Dual chambers with interactions.
Main Injector	200	Double since the RD-180 has 2 thrust chamber assemblies.	15	85	Dual injectors with interactions.
Nozzle	200	Recommend increasing to 200% because the RD-180 has two nozzles.	21	79	Dual nozzles with interactions.
Oxidizer Preburner (OPB)	100	Same	11	89	1 OX-rich preburner
Oxidizer System	100	Same	4	96	
Pneumatic System	100	Same	33	67	
Powerhead	0	This system does not have a power- head or equivalent. The structural backbone function that the SSME powerhead serves is performed on the RD-180 by a simple low-risk structural	N/A	N/A	

		framework connecting the thrust chambers to the vehicle. Set risk percentage to 0% to reflect lack of powerhead on RD-180, or at least 10- 20% to reflect simpler structural			
		configuration.			
Valves	20	Fewer, simpler valves.	16	84	
Other Risk	100	Same	23	77	

## Table 6G-11. J-2S Assessment

Component	% of Cat Risk Compare d to SSME	Rationale	% of Catastrophi c Failures	% of Benign Failures	Rationale
Actuators	100	Assume a redeveloped J-2Sd would have SSME-type actuators.	1	99	SSME-like redundancy.
Anti-Flood Valve (AFV)	100	Same	0	100	
Fuel Preburner	0	J-2S is a tap-off cycle engine and does not have a preburner.	N/A	N/A	
Fuel/Hot Gas System	200	Tap-off system has much greater risk than SSME.	25	75	Tap-off more critical 1's.
Heat Exchanger	100	Assume heritage HEX design, therefore similar to SSME.	9	91	
HPFTP/AT	50	Reduction due to lower operating conditions and XRS-2200 heritage.	10	90	Lower temperatures/ pressures.
HPOTP/AT	25	Reduction due to lower operating conditions and XRS-2200 heritage.	10	90	
Igniters	100	No change in risk.	16	84	
LPFTP	0	Eliminated through design; No boost pumps required with Gas Generator (GG).	N/A	N/A	
LPOTP	0		N/A	N/A	
LTMCC	50	J-2Sd would also reduce welds and has lower operating pressure and temperature operating conditions.	60	40	Tap-off duct critical.
Main Injector	75		10	90	
Nozzle	50	Nozzle failures would tend to be more benign than SSME.	9	91	Nozzle failures more benign.
Oxidizer Preburner (OPB)	0	No OPB.	N/A	N/A	
Oxidizer System	25	Lower pressure and simpler system.	2	98	Simpler system.
Pneumatic System	25		33	67	
Powerhead	0	No powerhead equivalent to SSME.	N/A	N/A	
Valves	25	Fewer valves and simpler design at lower pressure.	10	90	Lower pressure.
Other Risk	10	Lower part count and touch labor.	23	77	

Table 6G-12. RS-68 Assessment

Component	% of Cat Risk Compared to SSME	Rationale	% of Catastrophi c Failures	% of Benign Failures	Rationale
Actuators	200	RS-68 actuators have less redundancy than SSME.	10	90	Less redundancy.
Anti-Flood Valve (AFV)	0	RS-68 does not have an AFV.	N/A	N/A	
Fuel Preburner	100	Assume similar failure modes.	4	96	
Fuel/Hot Gas System	75	GG combustion is assumed more benign than Stage Combustion cycle.	10	90	GG cycle more benign.
Heat Exchanger	10	RS-68 has an external heat exchanger and does not have the same failure modes as SSME HEX.	5	95	Heat exchanger external.
HPFTP/AT	75	RS-68 turbine blisk versus SSME blades on a hub, and lower operating conditions.	15	85	Blisk and lower operating conditions.
HPOTP/AT	50	RS-68 turbine blisk versus SSME blades on a hub, and lower operating conditions.	15	85	Blisk and lower operating conditions.
Igniters	100	No change in risk.	16	84	
LPFTP	0	Eliminated through design; No boost pumps required with GG.	N/A	N/A	
LPOTP	0	Eliminated through design; No boost pumps required with GG.	N/A	N/A	
LTMCC	75	RS-68 reduces welds from SSME.	20	80	Reduction in welds.
Main Injector	25	Greatly simplified and significantly lower temperature of fuel entering main injector. Wood, B.K., Propulsion for the 21 <sup>st</sup> Century – RS-68, AIAA 2002-4324.	5	95	Lower temperature of fuel entering injector.
Nozzle	10	Ablative Nozzle Design. Wood, B.K., Propulsion for the 21 <sup>st</sup> Century – RS-68, AIAA 2002- 4324.	5	95	Ablative nozzle.
Oxidizer Preburner (OPB)	0	No OPB required with GG.	N/A	N/A	
Oxidizer System	50	Lower pressure and simpler system.	4	96	
Pneumatic System	25	No change in risk.	33	67	
Powerhead	0	No powerhead equivalent to SSME.	N/A	N/A	
Valves	100	No change in risk.	16	84	
Other Risk	10	Parts count reduced by 80%, touch labor reduced by 92%. Wood, B.K., Propulsion for the 21 <sup>st</sup> Century – RS-68, AIAA 2002-4324.	23	77	

Table 6G-13. RL-10 Assessment

Component	% of Cat Risk Compare d to SSME	Rationale	% of Catastrophi c Failures	% of Benign Failures	Rationale
Actuators	10	Pneumatic actuators instead of hydraulic as on SSME, but failure of actuators are benign failures.	10	90	Less redundancy than SSME.
Anti-Flood Valve (AFV)	0	Does not have an AFV.	N/A	N/A	Remove.
Fuel Preburner	0	Does not have this component.	N/A	N/A	Remove.
Fuel/Hot Gas System	5	Reduced due to fail-safe hot gas system.	5	95	Lower energy than SSME.
Heat Exchanger	0	Does not have this component.	N/A	N/A	Remove.
HPFTP/AT	10	Significantly lower temperatures.	5	95	One turbine for both pumps.
HPOTP/AT	10	Significantly lower temperatures.	5	95	Lower speed, lower energy, series with FTPT.
Igniters	10	Igniters exist, however failure modes are much more benign.	10	90	Redundant igniter.
LPFTP	0	Eliminated due to design.	N/A	N/A	Remove.
LPOTP	0	Eliminated due to design.	N/A	N/A	Remove.
LTMCC	10	Has tube wall chamber, but failure modes are more benign, unless in multiple engine application.	5	95	More benign.
Main Injector	10	Failure modes are more benign, unless in multiple engine application.	5	95	More benign.
Nozzle	0	Does not have a separate nozzle.	N/A	N/A	Remove.
Oxidizer Preburner (OPB)	0	Does not have an OX preburner.	N/A	N/A	Remove.
Oxidizer System	10	Failure modes are more benign, unless in multiple engine application.	4	96	
Pneumatic System	10	Failure modes are more benign, unless in multiple engine application.	10	90	More benign.
Powerhead	10	Failure modes are more benign, unless in multiple engine application.	N/A	N/A	Remove.
Valves	10	Failure modes are more benign, unless in multiple engine application.	5	95	More benign.
Other Risk	10	Failure modes are more benign, unless in multiple engine application.	15	85	More benign.

Table 6G-14. LR-60 Assessment

Component	% of Cat Risk Compare d to SSME	Rationale	% of Catastrophi c Failures	% of Benign Failure s	Rationale
Actuators	10	Pneumatic actuators instead of hydraulic as on SSME, but failure of actuators are benign failures.	10	90	Less redundancy than SSME.
Anti-Flood Valve (AFV)	0	Does not have an AFV.	N/A	N/A	Remove.
Fuel Preburner	0	Does not have this component.	N/A	N/A	Remove.
Fuel/Hot Gas System	0	Reduced due to fail-safe hot gas system.	1	99	Lower energy than SSME.
Heat Exchanger	0	Does not have this component.	N/A	N/A	Remove.
HPFTP/AT	25	Significantly lower temperatures.	5	95	One turbine for both pumps.
HPOTP/AT	25	Significantly lower temperatures.	5	95	Lower speed, lower energy, series with FTPT.
Igniters	10	Igniters exist, however failure modes are much more benign.	10	90	Redundant igniter.
LPFTP	10	MB-60, LR-60, and LR-100 are defined with low-pressure boost pumps, but the LR-85 is not.	5	95	
LPOTP	10		61	39	
LTMCC	10	Channel wall MCC and failure modes are more benign.	5	95	Failure modes more benign.
Main Injector	10	Failure modes are more benign, unless in multiple engine application.	5	95	Lower pressure.
Nozzle	10	Currently defined with a separate regen nozzle.	5	95	More benign.
Oxidizer Preburner (OPB)	0	Does not have an OPB.	N/A	N/A	Remove.
Oxidizer System	10	Failure modes are more benign, unless in multiple engine application.	4	96	No change.
Pneumatic System	10	Failure modes are more benign, unless in multiple engine application.	10	90	More benign.
Powerhead	10	Failure modes are more benign, unless in multiple engine application.	N/A	N/A	Remove.
Valves	10	Failure modes are more benign, unless in multiple engine application.	5	95	More benign.
Other Risk	10	Failure modes are more benign, unless in multiple engine application.	15	85	More benign.

Table 6G-15. LR-85 Assessment

Component	% of Cat Risk Compare d to SSME	Rationale	% of Catastrophi c Failures	% of Benign Failure s	Rationale
Actuators	10	Pneumatic actuators instead of hydraulic as on SSME, but failure of actuators are benign failures.	10	90	Less redundancy than SSME.
Anti-Flood Valve (AFV)	0	Does not have an AFV.	N/A	N/A	Remove.
Fuel Preburner	0	Does not have this component.	N/A	N/A	Remove.
Fuel/Hot Gas System	0	Reduced due to fail-safe hot gas system.	1	99	Lower energy than SSME.
Heat Exchanger	0	Does not have this component.	N/A	N/A	Remove.
HPFTP/AT	25	Significantly lower temperatures.	5	95	85k – blisk not blades.
HPOTP/AT	25	Significantly lower temperatures.	5	95	85k – in series with FTPT.
Igniters	10	Igniters exist, however failure modes are much more benign.	10	90	Redundant igniter.
LPFTP	0	MB-60, LR-60, and LR-100 are defined with low-pressure boost pumps, but the LR-85 is not.	N/A	N/A	Remove.
LPOTP	0	Eliminated due to design.	N/A	N/A	Remove.
LTMCC	10	Channel wall MCC and failure modes are more benign.	5	95	Failures modes more benign.
Main Injector	10	Failure modes are more benign, unless in multiple engine application.	5	95	Lower pressure.
Nozzle	10	Currently defined with a separate regen nozzle.	5	95	More benign.
Oxidizer Preburner (OPB)	0	Does not have an OPB.	N/A	N/A	Remove.
Oxidizer System	10	Failure modes are more benign, unless in multiple engine application.	4	96	No change.
Pneumatic System	10	Failure modes are more benign, unless in multiple engine application.	10	90	More benign.
Powerhead	10	Failure modes are more benign, unless in multiple engine application.	N/A	N/A	Remove.
Valves	10	Failure modes are more benign, unless in multiple engine application.	5	95	More benign.
Other Risk	10	Failure modes are more benign, unless in multiple engine application.	15	85	More benign.

Table 6G-16. LR-100 Assessment

Component	% of Cat Risk Compare d to SSME	Rationale	% of Catastro phic Failures	% of Benign Failure s	Rationale
Actuators	10	Pneumatic actuators instead of hydraulic as on SSME, but failure of actuators are benign failures.	10	90	Less redundancy than SSME.
Anti-Flood Valve (AFV)	0	Does not have an AFV.	N/A	N/A	Remove.
Fuel Preburner	0	Does not have this component.	N/A	N/A	Remove.
Fuel/Hot Gas System	0	Reduced due to fail-safe hot gas system.	1	99	Lower energy than SSME.
Heat Exchanger	0	Does not have this component.	N/A	N/A	Remove.
HPFTP/AT	50	Significantly lower temperatures.	10	90	85k – blisk not blades.
HPOTP/AT	50	Significantly lower temperatures.	10	90	85k – in series with FTPT.
Igniters	10	Igniters exist, however failure modes are much more benign.	10	90	Redundant igniter.
LPFTP	10	MB-60, LR-60, and LR-100 are defined with low-pressure boost pumps, but the LR-85 is not.	5	95	MB-60, LR-60, and LR-100 are defined with low-pressure boost pumps, but the LR-85 is not.
LPOTP	10		61	39	
LTMCC	15	Channel wall MCC and failure modes are more benign.	10	90	Failure modes more benign.
Main Injector	10	Failure modes are more benign, unless in multiple engine application.	10	90	Lower pressure.
Nozzle	10	Currently defined with a separate regn nozzle.	10	90	More benign.
Oxidizer Preburner (OPB)	0	Does not have an OPB.	N/A	N/A	Remove.
Oxidizer System	10	Failure modes are more benign, unless in multiple engine application.	4	96	No change.
Pneumatic System	10	Failure modes are more benign, unless in multiple engine application.	10	90	More benign.
Powerhead	10	Failure modes are more benign, unless in multiple engine application.	N/A	N/A	Remove.
Valves	10	Failure modes are more benign, unless in multiple engine application.	5	95	More benign.
Other Risk	10	Failure modes are more benign, unless in multiple engine application.	15	85	More benign.



# Section 6H CLV Integrated Vehicle Configuration Definition

## **APPENDIX 6H**

## **CLV INTEGRATED VEHICLE CONFIGURATION DEFINITION**

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- 5.7 STRUCTURAL LOADS AND DYNAMICS
  - 5.7.1 Study Objective
  - 5.7.2 Point-of-Departure Vehicle and Previous Studies
  - 5.7.3 Loads and Dynamics Assessments
    - 5.7.3.1 Structural Modeling
    - 5.7.3.2 Sizing Cycle
    - 5.7.3.3 Pre-Launch Ground Winds
    - 5.7.3.4 Ascent Flight
    - 5.7.3.5 SRB Actuator Hard Over Failure
    - 5.7.3.6 Liftoff
    - 5.7.3.7 SSME/RS-25 Loads, Dynamics, Life Comments
    - 5.7.3.8 SRB Acoustic Mode Thrust Oscillations
  - 5.7.4 Significant Findings
  - 5.7.5 References

## 1.0 INTRODUCTION

This final report describes the design and analysis results developed as part of the "60 Day Study" to determine an optimal design for the Crew Launch Vehicle (CLV) System derived from the Shuttle RSRM. The CLV System consists of a crew rescue capability; a crew transfer capability; and the necessary flight and ground support to perform these capabilities. The CEV spacecraft, for the ISS crew vehicle and the Lunar crew vehicle, are to be launched on a 4 Segment SRB with 1 SSME and required LH2 and LO2 systems for the upper stage. The overall hierarchy of the Vision for Space is shown in Figure 1.0-1. Programmatic requirements are not captured in this document.

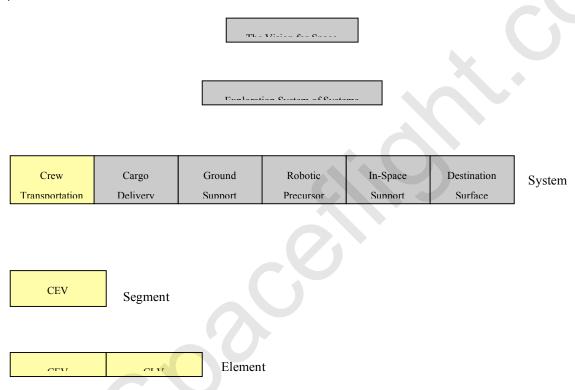


Figure 1.0-1: CTS System Hierarchy

## 1.1 PURPOSE

Marshall Space Flight Center (MSFC) performed a conceptual definition and evaluation of vehicles based on existing engines and systems technologies called the "60 Day Study" using Space Shuttle Derived expendable approaches. As part of the Shuttle-derived approach the Vehicle Integrated Performance Analysis (VIPA) team, composed of multi-disciplinary subject experts from the Engineering and Space Transportation Directorates, performed a conceptual design and evaluation of the 4-segment SRB with one SSME – Block II Configuration.

## 1.2 GROUNDRULES AND ASSUMPTIONS

The groundrules and assumptions for this document are presented in this section. These groundrules and assumptions provided constraints that the conceptual design had to meet. The groundrules and assumptions applied to this study were:

- The launch vehicle shall deliver the CEV spacecraft to an elliptical delivery orbit of 30 x 160 nautical miles
- The final stage of the launch vehicle will be safely disposed of by re-entry into the Earth's atmosphere
- The concept definition shall, at a minimum, be to the subsystem level and may be extended to the component level as the need is justified
- Technical risk shall be assessed for the following:
  - Aerodynamics
  - o Avionics/Power
  - Controls/Stability
  - o Propulsion
  - o Structures/Materials
  - Thermal/TPS
  - o Trajectory/Flight Mechanics
  - Lift Capability
- Loads are assessed for off-nominal flight conditions for the following cases
  - Pre-launch
  - Liftoff
  - Maximum dynamic pressure
  - Maximum axial acceleration
- For Shuttle derived elements, the safety factors shall be those currently in use by the parent vehicles
- Launch vehicle will provide 3-sigma flight performance reserve dispersion coverage
- The launch vehicle shall use known fuel bias values for existing propulsion systems

## 1.3 ORGANIZATION OF DOCUMENT

The document will describe the approach used for the "60 day study", and an overview of the conceptual design of the 4 segment SRB with 1 SSME CLV configuration along with descriptions of each of the elements. Detailed design information will be in Section 4.0, and Integrated Vehicle Analyses and Trades in Section 5.0.

## 2.0 CLV SE&I ANALYSIS APPROACH

The analysis approach used for this study consisted of applying existing engineering tools, models and processes to this new vehicle configuration. Data was obtained from the Space Shuttle Program and Project Offices for existing unmodified elements (e.g. RSRM, SSME). New designs and models were developed for the additional new elements. Once the specific discipline models and tools were developed, the analyses inputs were carefully examined to assure consistency across the discipline analyses creating an integrated analysis package.

The Systems Engineering and Integration (SE&I) function has the primary responsibility for systems engineering and technical integration across the system definition teams. SE&I works with the Business Management team, as necessary, to support their costing, business management, and business case definition tasks.

The systems engineering approach is based on the systems engineering process shown below in Figure 2.1.

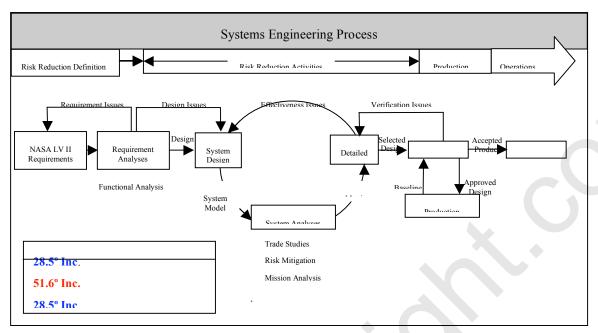
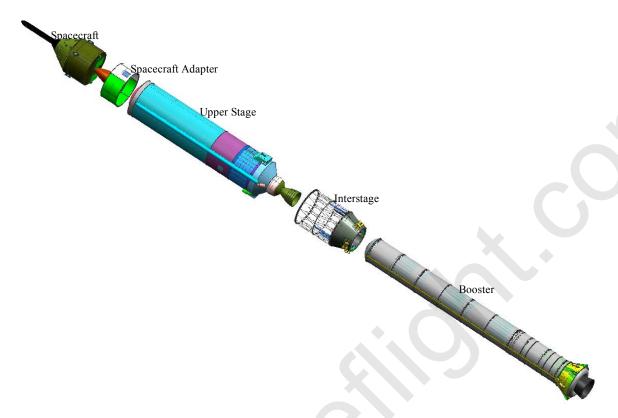


Figure 2.1 Systems Engineering Process

# 3.0 4 SEGMENT SRB WITH 1 SSME CREW – BLOCK II CONFIGURATION

The CLV launch system consists of 5 elements shown in Figure 3.1: The Spacecraft with the Launch Escape System, the Spacecraft Adapter, the Upper Stage, the Interstage, and the Booster. Overall dimensions and coordinate systems are shown in Figure 3.2. These elements are as the assembly structure exists at the time of the POD, and do not reflect any WBS.



**Figure 3.1 CLV Elements** 

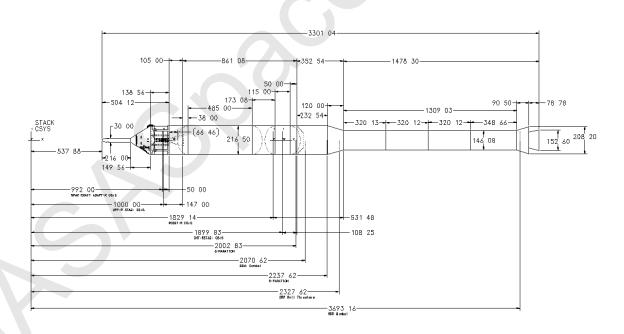


Figure 3.2 CLV Dimensions and Coordinates

# 3.1 VIPA VEHICLE DESIGN PROCESS FOR REFERENCE GEOMETRY

The VIPA Vehicle Design (VVD) team is one of the VIPA discipline teams. VVD maintains the reference geometry of the vehicle, with data from a wide variety of sources, and distills that data into information directly usable by the rest of the VIPA discipline teams.

## 3.1.1 Design Modeling Process

The basis of VVD reference geometry modeling is the Top-Down, Layout-Based modeling approach using UGS/NX and managing all data in UGS/Teamcenter. The basic modeling approach is shown in Figure 3.3. The VIPA process is based on NX best practices for WAVE Systems Engineering, with slight modifications made specifically to accommodate VIPA's needs.

In VVD's top-down modeling approach, each stage element has a Stage Layout that has sketches, surfaces or bodies as needed to define major interfaces between lower level components such as tanks. Each lower level component has its own Component Layout which is linked to the geometric objects of the Stage Layout and adds additional detail for that specific component, such as ring locations and dimensions. From the Component Layout the "Create Linked Part" (CLP) function is used to link required information to the individual part files. The parts are then modeled starting with the referencing geometry. CLP is used again from the Component Layout to the Component Assembly to provide mating data for any non-VVD generated parts that are being used. The individual parts are assembled in the stage absolute coordinate system and mating conditions are generally not needed. The Component Assemblies are then added to the Stage Assembly.

The several Stage Layouts that make up the vehicle stack are assembled into a Stack Layout file. The layouts are mated to each other by aligning datums and curves. The datums from each Stage Layout coordinate system are then geometrically linked, or WAVEd using the NX WAVE geometry linker, from the Stage Layouts to the Stack Layout. CLP is used to WAVE the coordinate system datums into the Stack Assembly, allowing the Stage Assemblies to be robustly mated into the Stack Assembly by mating coordinate systems. Mating of coordinate systems is more robust then mating elements of the parts themselves because the mated objects may be changed in ways that could invalidate conventional mating conditions.

This approach allows control of very large assemblies with a small handful of layout files that act as 3D interface control documents. The structure of the files also allows for each component to travel as a unit so it can be easily cloned and reused in different assemblies by reassembling and redirecting the handful of curves that define its major interfaces.

The level of detail achieved on the 60 day study would not have been possible if this process had not been used in previous VIPA studies. Many elements of this vehicle were reused from a previous Exploration Office heavy lift launcher study, which in turn had used many elements from a VIPA validation exercise using the Saturn V.

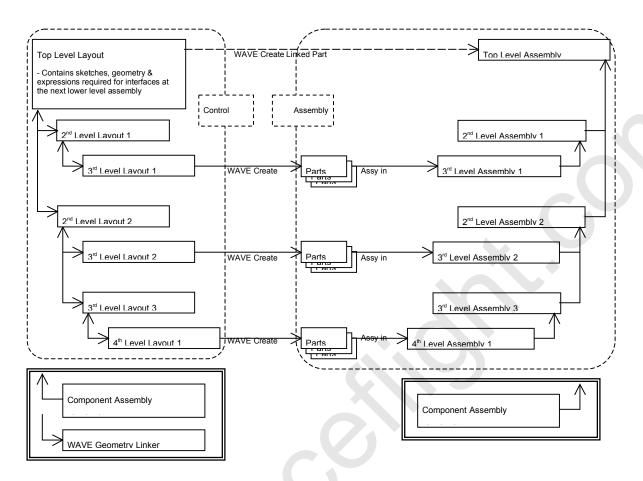


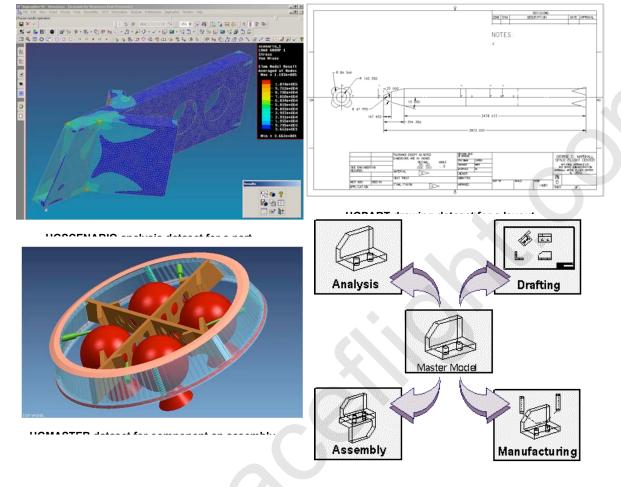
Figure 3.3 VVD Top-Down, Layout-Based Modeling

## 3.1.2 VVD Interaction with other VIPA Teams

#### 3.1.2.1 Master Model

VVD's second major function is the extraction of data from the reference geometry for the various analysis teams. To perform this function, VVD relies on the Master Model approach, in conjunction with Top-Down modeling, all managed and access by the Teamcenter Product Data Manager.

A notional view of Master Modeling is shown in Figure 3.4. In this approach there is a master product definition file that defines the part of interest. Referencing files start with the master product definition and add, extract or simplify information as needed. Separate referencing files are used for drawings, visualization, analysis model simplification/translation, analysis, etc. The representation of objects within a Teamcenter "Item" (or part number), and "ItemRevision" (specific revision of a given part) is shown in Figure 3.5



**Figure 3.4 Master Model Concept** 

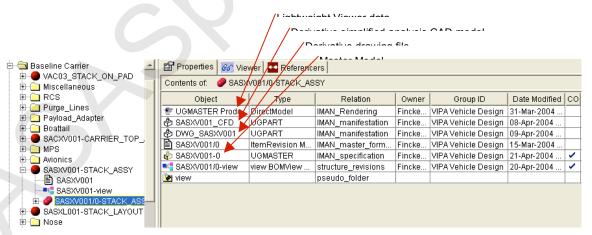


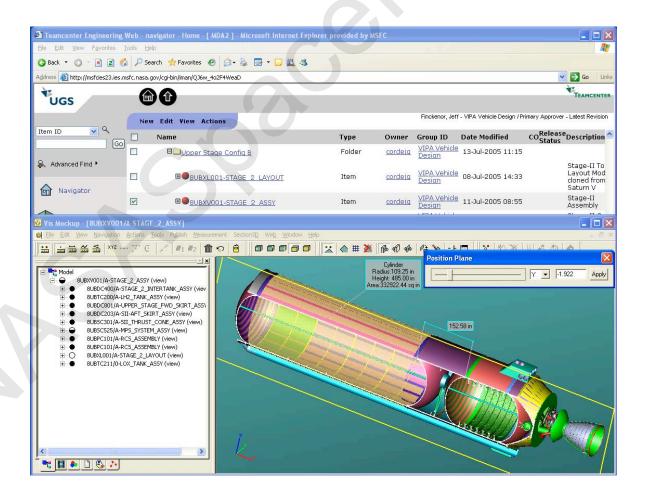
Figure 3.5 Teamcenter Data Structure and Master Model

# 3.1.2.2 VVD Delivered Analytical Data

The first important data delivered to analysts is the Stage Layout drawing, already shown in Figure 3.2. The Top-Down modeling approach has proven to be extremely valuable to concurrent engineering as practiced by the VIPA Team. The layout is the first piece of information required by the geometric modeling process. It also contains most of the up front information required for analysts to start building their discipline models almost immediately.

Simplified geometric models, such as those needed for early stress, thermal, or simulation modeling can also be rapidly generated from the layout as referencing part files. As needed, translations in the appropriate formats are generated and automatically stored with the appropriate CAD part. Delivered formats include parasolid, STEP, IGES and STL. The objective of VVD is to provide associative mesh ready models for each discipline that uses geometric inputs.

Analysts are also able to access lightweight models and discipline specific data directly using Teamcenter's web interface, as shown in Figure 3.6. The lightweight models are generated automatically on save, and since they have the same underlying parasolid kernel as NX are mathematically exact representations of the CAD parts. Having live versions of the models immediately available to all users allows anyone to find and measure desired data without having to wait on the availability of a CAD user. Productivity is significantly increased by freeing the designer from the task of delivering ever changing 2D representations of their work. In addition'the Teamcenter



#### Figure 3.6 Teamcenter Web Access and Visualization

visualization tools include the ability to create a STEP translation allowing analysts to generate their own import data without needing to wait on a CAD operator.

The last important data delivered in a sizing iteration, after all the discipline models have been prepared, is the mass properties report. VVD generates the nominal, dry mass properties for the Systems Modeling team. The Systems Modeling team is responsible for managing margins on top of the nominal design, and generating sequenced mass properties for use by Loads & Dynamics, trajectories, simulation, GN&C, etc. VVD uses the NX "Assembly Weight Management" function which exports all the 6DOF mass properties data directly to an Excel spreadsheet, shown in Figure 3.7. 6DOF properties for every piece part component are included, and rolled up into assemblies. It is important that the piece part information is part of the spreadsheet since the Systems Modeling team often uses a system oriented product breakdown rather then the VVD assembly oriented breakdown. Inclusion of the piece part data enables direct manipulation and sorting via spreadsheet.

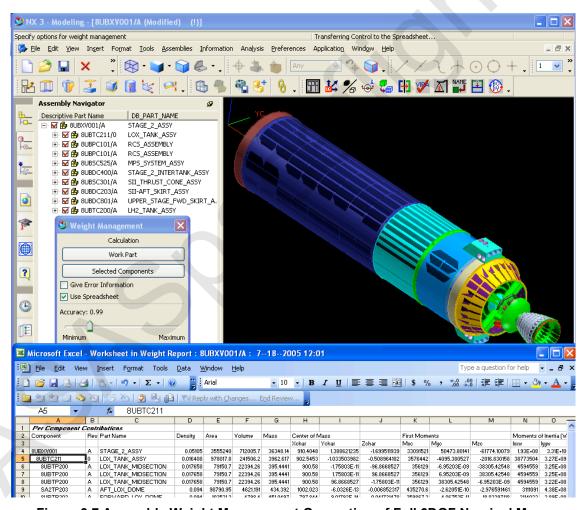


Figure 3.7 Assembly Weight Management Generation of Full 6DOF Nominal Mass Properties

## 3.1.2.3 Data Organization

Folders within Teamcenter are organized to provide a familiar format for users to find information. Access control is handled with a series of Teamcenter "groups". Each discipline has its own group, and only other members of the same group can edit objects created by that group. Data generated by any VIPA group is made visible to users in any other VIPA group. In addition, there is a "VIPA Team" group that all VIPA team members are placed in and all can share and edit data. This is used for shared documentation and collaboration across disciplines.

## 3.2 SYSTEMS INTEGRATOR ROLE

# 3.2.1 CAD and CAD Management Compatibility

The VIPA team originated in 2001 to provide the technical aspects of Systems Engineering, or Analytical Integration, for the Space Launch Initiative program. VIPA Vehicle Design has established and demonstrated the ability to deal with disparate data so that MSFC can be leaders in Integration. Relevant to this is an understanding of what tools we need to interface with. Figure 3.8 shows the current understanding of the CAD and CAD Management landscape across NASA and most relevant Aerospace companies. This is a simplified presentation of a very complex integration problem, but a couple messages are clear. First, the most common CAD in aerospace is Dassault's Catia (including a mix of both Catia V4 and V5 which are completely different CAD packages). Second is UGS's NX (including both UGNX and IDEAS-NX which have converged over the last several years into a single product). VVD's use of NX is as consistent with industry CAD usage as possible. CAD Management in aerospace is clearly Teamcenter dominated. VIPA's use of Teamcenter will simplify integration with these contractors. Note that PTC/Intralink is a recently discontinued product which will be supported only through 2008. The Intralink name is maintained, but is now in a PTC-Windchill database requiring a full migration of existing Intralink data.

	CAD Package					CAD Management				
	PTC Pro-E	UGS-NX (parasolid)	Dassault- Solidworks (parasolid)	Dassault- Catia	Bently- Microstation (parasolid)	PTC- Windchill	PTC- Intralink	UGS- Team center	Dassault- Enovia	Matrix
NASA										
JSC	X	X					X			
KSC	X			Х	х					
Stennis	X									
MSFC	X	X			Х			X		
GSFC	×	х	x							
JPL		X						X		
GRC	X	Х	Х							Х
_aRC	X	х	х							
Ames	Х		X							
ndustry										
_ockheed/Martin	V									
Corporate	X	х		х				X		
JSF				X				X		
MAF-ET				X				X		
Atlas-V	X					X	X	X		
Boeing	/									
Corporate		X		X				X	Х	
Orbiter				X						
Delta IV		X						X		
Jnited Technologies										
Pratt and Whitney		Х						X		
Rocketdyne	X						X	Х		
JSA	Х	Х		Х	Х					
ATK/Thiokol		X						X		
Northrup Grumman		X		X				X	Х	

Figure 3.8 Aerospace CAD and CAD Management Landscape (best available data)

# 3.2.2 Multi-CAD Integration

Since there is a variety of CAD in use, VIPA has focused on being able to manage and manipulate this data as required. A previous VIPA study demonstrated the use of Dassault/Catia and PTC/ProE CAD models, managed directly in Teamcenter, and integrated directly into the VIPA geometry definition via Managed Translations. Figure 3.9 shows the use of these models in a heterogeneous product structure. The basic VIPA modeling structure is maintained, but provided models from other systems are linked into the control and assembly structure. This example shows all data flowing through the Stage Layout, but foreign subassemblies can also be plugged directly into the assembly as needed. Figure 3.10 shows the Teamcenter thick client view of a multi-CAD part with Managed Translations as separate referencing objects to the master definition in the originating CAD.

This same capability was used in the 60 day study for managing and using ProE data of the CEV and old trade study models of the SRB aft skirt. These models are shown in Figure 3.11. Highlighting the diversity of provided data is the VVD CLV on pad assembly. The CLV on pad assembly shown in Figure 3.12 includes:

- NX/Teamcenter developed Spacecraft Adapter, Upper Stage, Interstage
- ProE/Teamcenter managed CEV and SRB Aft Skirt
- Catia and ProE translated SSME
- Ideas translated SRM, and
- Microstation translated pad

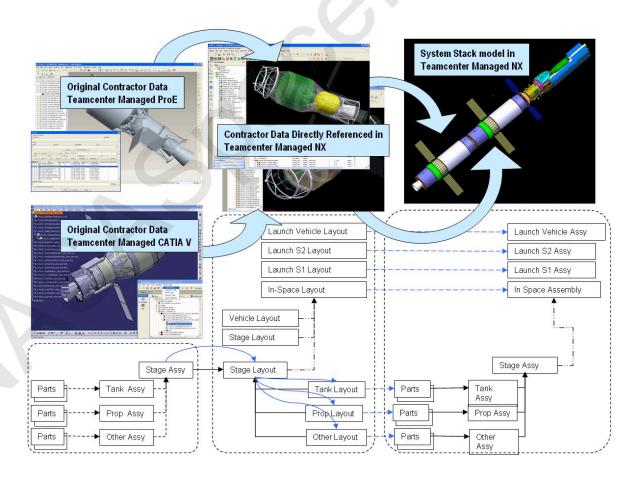


Figure 3.9 Multi-CAD Management & Integration

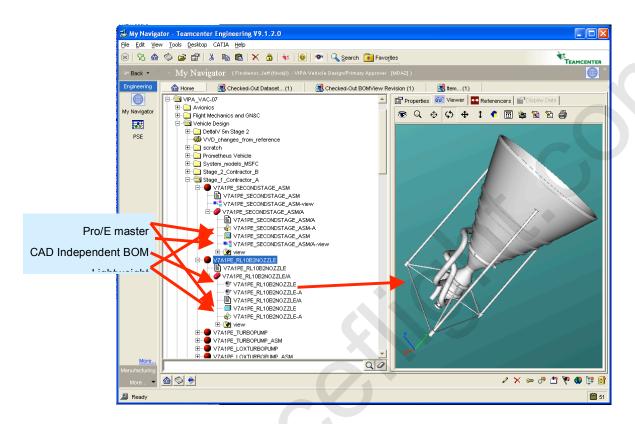


Figure 3.10 Multi-CAD Management Heterogeneous BOM



Figure 3.11 ProE Assemblies Managed in Teamcenter

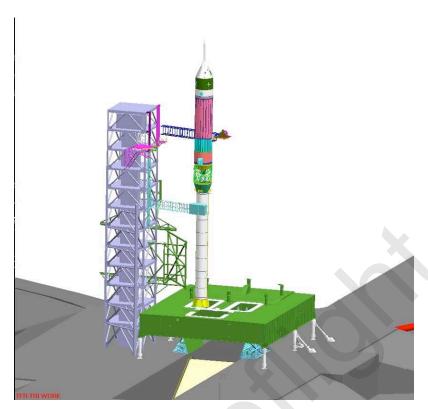


Figure 3.12 Multi-CAD CLV on Pad Assembly

## 3.3 HUMAN FACTORS

# 3.3.1 Human Factors Engineering

Human Factors Engineering (HFE) is a Systems Engineering and Integration function that will systematically apply HFE principles, data, and tools in the design and analysis of the launch vehicle. HFE technical and analytical input will contribute to the definition, identification, analysis, cost and efficiency estimation of the human interfaces associated with three primary areas in launch vehicle design. The areas include definition of line replaceable units (LRUs), definition of human ground interfaces to the vehicle, and vehicle component manufacturability.

HFE defines, reviews, and approves the requirements associated with identification of the LRUs and identifies the associated worksites on the vehicle. This effort is accomplished in collaboration with other systems engineering disciplines, including, primarily, reliability. The effort during the concept development phase of identifying LRUs will help discern operations cost drivers early in the program. This result should be used to perform programmatic trades (with HFE participation) to identify those components for which reliability must be improved versus those for which NASA is willing to accept the cost of ground processing as nominal.

The next area of focus for HFE support is in ground operations where HFE considerations impact the design of the launch vehicle's infrastructure for ground support. A function for HFE is to apply task analyses and the associated requirements (generated above) to identify the human interfaces to be accessed during ground operations. For example, some components (LRUs) will require maintenance during launch processing. The tasks associated with these are analyzed by

human engineers to identify the applicable requirements. One of these will be that the LRU should be accessible so that maintainers are not required to remove other components to reach it, nor should maintainers be required to enter the vehicle to gain access to it. Thereby, the risk of damage to other components (e.g., cabling) is reduced; logically, the cost and schedule impacts associated with any damage are reduced. Throughout the design phase, HFE will participate in trades to identify such issues with the ultimate objective of reducing operations costs.

Later in the design phase, HFE can define, review, and approve requirements associated with the vehicle manufacturability that affect cost, efficiency, and human safety. By using worksite analysis (modeling) tools, HFE will support trade studies addressing tooling, robotic versus human assembly, and plant configuration. HFE will have input into micro- and macro-analyses addressing individual work cells through factory flow terminating with vehicle assembly.

### 3.3.2 Vibration Effects on Humans

The RSRM has a longitudinal acoustic mode that causes an oscillating axial load. The force is A sin f where A is as high as 80 Klbf, i.e., 160 Klbf peak to peak. For the 4-segment booster f is 15 Hz and for the 5 segment booster f is 11 Hz. The effect is roughly equivalent to being accelerated at 2 g, with an oscillation about 2 g of 0.1-0.2 g, and a frequency of 15 Hz. Human performance is known to be affected by frequencies in the 1-30 Hz range. Frequencies above 30 - 60 Hz are not normally a problem for humans, and can be damped by cushioned couches. Vibration below 1 Hz is not generally known to affect performance or body resonances, rather is associated with motion sickness. "Moderate" vibration affects human performance in the following ways:

- · Loss of tracking performance, for example following a target with a hand controller
- Typically there is no loss in reaction times, but there are exceptions
- Visual impairment, (trouble reading, for example) but very sensitive to the type of vibration and task, also dependent on posture and illumination levels

Early in the space program, one of the rockets had a strong pogo resonance at 11 Hz along the long axis of the rocket due to the rocket engine. The crew complained of vibration which interfered with their vision. A modification to headrest padding gave a more subjectively acceptable vibration level.

Currently the team believes that this is not a major issue, based on what is known about the duration (about 2 minutes), crew tasks (monitoring and choosing abort if necessary), crew orientation (on their backs), acceleration and vibration intensity, frequency, and direction (opinion from George Hamilton/EV12). As the design matures, this is an integration issue that should be worked for astronaut performance during the first stage.

## 4.0 CLV BASELINE SYSTEM DESCRIPTION

## 4.1 UPPER STAGE

An expanded view of the Upper Stage is shown in Figure 4.1.

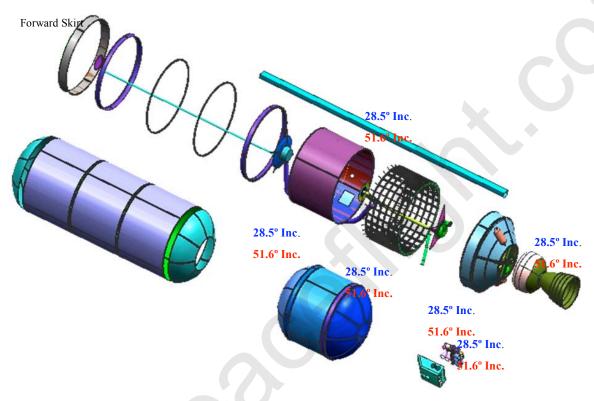


Figure 4.1 Upper Stage Expanded View

# 4.1.1 General Design Assumptions

There are a number of assumptions that apply across many elements of the baseline vehicle, though many of these assumptions are open to trades. The assumptions are:

- The nominal structural diameter is 216.5" (5.5m). This dimension is to the outer surface of the component skins. TPS and flanges are external to the nominal diameter.
- All primary structure is Friction Stir Welded Aluminum-Lithium 2195 with a maximum stock thickness of 1.85" based on material currently available for the Shuttle External Tank.
- Maximum barrel lengths of 14 feet to increase availability of existing tooling.
- All barrels and rings consist of 4 radial segments
- Tank domes are in 8 radial gores, limited by stretch forming capabilities, with a welded end cap.
- Domes are Sqrt(2) tangent ellipses
- Axial segment/gore interfaces are staggered
- Segment/gore edges and weld lands assumed 1.5" wide on the edges of each part.
- Stage component interfaces are external flanged joints with a radius 2" greater then the nominal structural radius.

- Where needed, system tunnel attachments along the length will be by external blind holes at the structural grid nodes.
- Baseline Thermal Protection is assumed to be NCFI 1.0" thick on barrels, and BX-261 0.5" thick on the domes.

# 4.1.2 Typical Joints

Figure 4.2 shows the typical joints used in the Upper Stage.

# 4.1.3 Upper Stage Component Specifics

#### 4.1.3.1 Forward Skirt

The forward skirt is 26" long. It provides the interface between the Upper Stage and the Spacecraft Adapter. The system tunnel starts at the forward skirt with lines for the Spacecraft Adapter systems and the LH2 tank press/vent lines.

## 4.1.3.2 LH2 Tank

The LH2 Tank has a barrel length of 485". The feedline connects to a sump on the bottom dome. A vortex baffle is mounted in the aft dome, but slosh baffling is assumed to be unnecessary.

#### 4.1.3.3 Intertank

The intertank is 173" long. The baseline vehicle definition has separate tanks with an intertank in between. The intertank has a purge umbilical and fill/drain umbilicals for both tanks. An access panel is also provided.

## 4.1.3.4 LOX Tank

The LOX tank has a barrel length of 115". A vortex baffle is mounted in the aft dome and slosh baffle rings are mounted along the length of the barrel. Since the LOX fill/drain line is forward of the tank, an internally mounted fiil/drain line runs down the inside of the tank.

#### 4.1.3.5 Thrust Structure

The Thrust structure consists of a cylinder 38.75" long and a cone 52.5" long. The cylinder interfaces with the Y-ring of the LOX tank and provides clearance for the feedline to enter the thrust structure, as well as lines from the systems tunnel. It provides a purge umbilical to the ground. The interface to the Interstage is immediately aft of the cylinder. The engine is mounted on cross beams at the bottom of the cone. The beam loads are transmitted into the cone by tapered longerons. Mounted on the thrust cone are the APU's for engine and TVC power, and a pressurized He tank for the SSME. A thermal curtain is connected between the top of the RD25 nozzle (perhaps using the engine mounted heat shield) and the thrust cone. This provides a closed environment for purge and hazardous gas detection, as well as prevention of external contaminants on the pad and recirculation gasses on ascent.

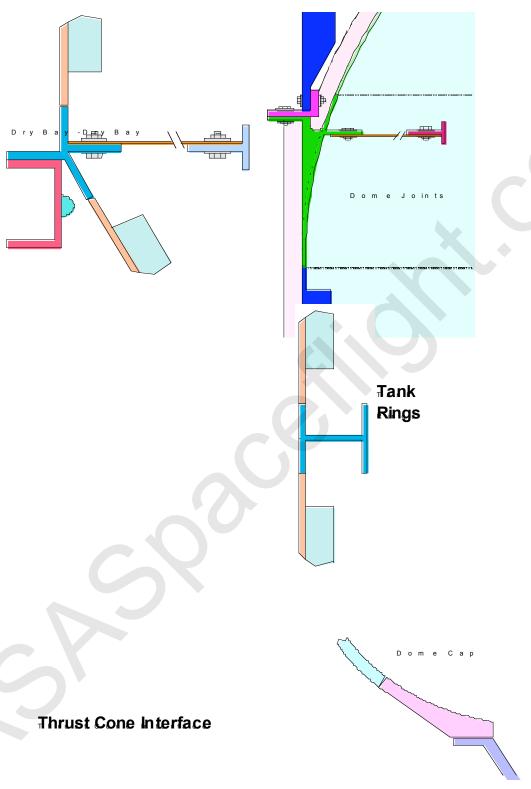


Figure 4.2 Baseline Joints

## 4.1.3.6 Main Propulsion

The Upper Stage engine is the RS-25 (SSME). The engine and TVC actuators are mounted on the center cross beam of the thrust structure. The engine provides tank ullage pressure. Feedlines to the engines are 12" ID. The baseline APU's are orbiter based and use hydrazine. The hydrazine portions will be in contained line replaceable pods.

# 4.1.3.7 Reaction Control System (RCS)

The RCS system is a He pressurized bipropellant hydrazine system. 2 RCS line replaceable pods are mounted on the aft skirt. Each pod includes He and propellant tanks with 3 1000lb aft pointing thrusters and 6 100# thrusters for pitch roll and yaw.

# 4.1.3.8 Upper Stage Trades

In the course of developing the baseline vehicle definition a number of trades have been identified.

#### Material and Construction Trade

The baseline material and construction method is AL-LI isogrid. For the Upper Stage, Spacecraft Adapter and Interstage, the baseline construction approach needs to be traded against:

All major components:

- AL isogrid
- AL-LI orthogrid
- AL orthogrid

Dry bay components:

- AL/AL core honeycomb
- AL-LI/AL core honeycomb
- composite/AL core honeycomb
- composite/non-metallic core honeycomb

#### Separate vs. Nested Tank Trade

The baseline has 2 conventional tanks separated by an intertank. This needs to be traded against "nested" tanks with the elimination of the intertank.

Note that the proposed nested configuration is not a true common bulkhead. A common bulkhead has a single dome between the tanks. One of the nested tanks would have a concave dome which nests with the other tank. The intervening space would be insulated and contain thermally non-conductive spacers if needed. The tank with the concave dome would be determined by the higher pressure tank and if the LH2 feedline goes through or around the LOX tank.

# **Thrust Cone Angle**

The angle of the thrust cone and the length of the aft skirt need to be optimized to accommodate the engine and TVC mounting and loads, the APUs, and feedlines.

### **End Ring Manufacturing Trade**

In the baseline` end rings of components are assumed to be 4 segments of rolled extrusions. A manufacturing study is required to determine if single piece forgings may be preferable.

#### Non-Pressurized Joints

The baseline connection for non-pressurized joints is a rolled L bracket with shear fasteners to the component skin, and a bolted face joint to the next component. Alternate possibilities are:

- double lap shear joints which provide a more efficient load path but require more tooling setup.
- butt welded joints which would provide a highly efficient structural path, but makes destacking of components essentially impossible.

### Single vs. Multi Piece domes

Current stretching and spin forming tooling cannot support the 18' baseline domes. 18' spun form domes are not new technology, but the up front tooling is needed. This would be a manufacturing trade to determine the return on the tooling investment and if it could be ready in time. If domes are spun formed, then the end cap might be able to be formed directly and not be a separate welded piece.

# 4.2 SPACECRAFT ADAPTER

The Spacecraft Adapter is shown in Figure 4.3. Most of the Upper Stage General Design Assumptions, Typical Joints, and Trades apply as well to the Spacecraft Adapter. The Adapter is 105" long and maintains the nominal 216.5" structural outer diameter. The length of the adapter has to be worked in conjunction with the CEV nozzle length. The Spacecraft Adapter has a rigid bolted connection to the Upper Stage and a separation system interface to the Spacecraft. It also contains most of the avionics for control of the launch vehicle. On the pad it will require a purge and an electrical umbilical and includes an access door. The baseline assumption is that passive cooling of the avionics is adequate.

# 4.2.1 Spacecraft Separation System

The separation uses 6 discreet separation points. These are internal to the cylinder and align with the 6 internal structural panels of the Spacecraft Service Module. Each point includes a counterbored bracket which helps carry shear loads. Within each bracket is a 3 bolt pattern with explosive nuts. Springs are also mounted on the bracket to provide a push-off impulse to the spacecraft. The bracket with the nuts is part of the Adapter assembly, with a simple mechanical interface to the spacecraft.

# 4.2.2 Spacecraft Adapter Specific Trades

#### Active/Passive Avionics Cooling

The baseline and preferred approach is that passive cooling on the pad via the purge is adequate to keep avionics cool. The passive cooling assumption must be verified.

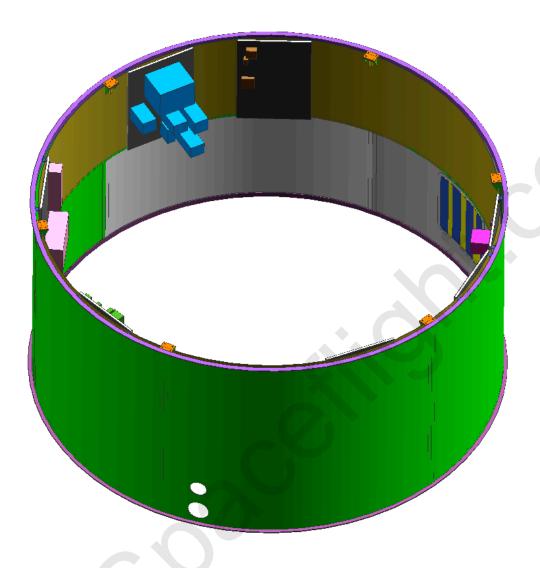


Figure 4.3 Spacecraft Adapter

### Separation System

Explosive nuts are a relatively common and clean method of payload separation. To minimize shock and pyrotechnic handling concerns, mechanical separation nuts such as the Starsys FASSN or the Hi-Shear Low-shock Separation Nut can be used. Both of these systems are flight-proven, commercially available alternatives to explosive nuts with similar actuation times and power requirements. Also, since the Spacecraft separation occurs during a coast phase, slower or soft release methods are also possible. The tradespace may include mechanical systems such as the Lightband or pyrotechnic systems such as the SuperZip.

# 4.3 INTERSTAGE

An expanded view of the Interstage is shown in Figure 4.4. Most of the Upper Stage General Design Assumptions, Typical Joints, and Trades apply as well to the Interstage. The Interstage consists of the Interstage Cylinder and the Frustum.

# 4.3.1 Interstage Cylinder

The forward section of the Interstage is the Interstage Cylinder, which is 240.8" long. It interfaces with the Upper Stage and contains four solid settling motors for Upper Stage engine startup, and two separation rings. The baseline structure for the Cylinder is an open truss to eliminate issues of engine bleed flow prior to Upper Stage engine start. An avionics conduit will pass cabling from the Upper Stage to the SRB.

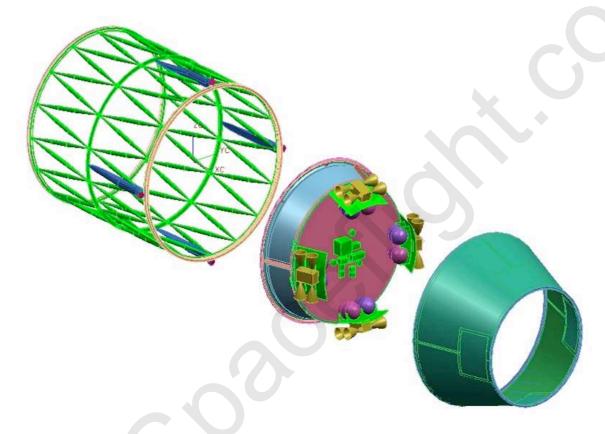


Figure 4.4 Interstage Expanded View

# 4.3.2 Interstage Cylinder Structure

The truss is made of tapered composite struts with a maximum diameter at the center of 5.5". The struts are attached to end fittings of titanium to minimize galvanic issues. The titanium fittings are mounted on AL-LI end rings. An intermediate set of titanium brackets and an AL-LI ring are used to reduce the slenderness ratio of the struts to eliminate buckling without making the struts too large in diameter.

# 4.3.3 Interstage Separation System

Separation rings are mounted fore and aft of the truss structure. These are C-section rings with a shape charge for cutting. This is essentially the same separation as in the forward skirt of the SRB.

The stage separation is shown in Figure 4.5. As SRB thrust is tailing off, the aft separation ring initiates and Booster Separation Motors on the SRB aft skirt fire to pull the booster away from the

Upper Stage. The SRB is yawed to an angle to prevent it from thrusting back into the Upper Stage. After the booster is



Figure 4.5 Stage Separation

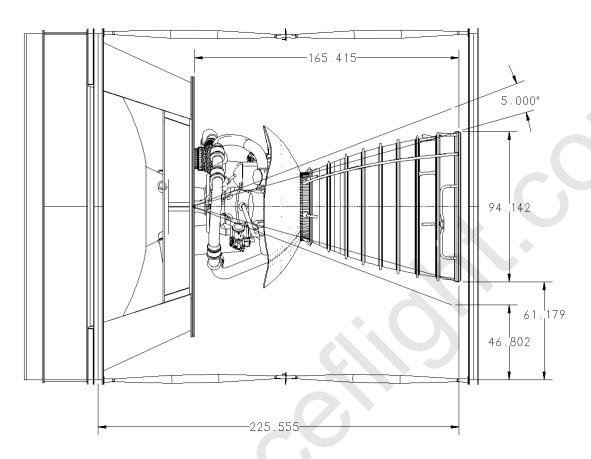


Figure 4.6 Interstage/Engine Clearance

separated, the settling motors fire to allow the Upper Stage engine to start. Once the engine reaches an SSME thrust level of 100% the forward ring separates and the Interstage Cylinder structure falls away. This is the same stage separation sequence used for the Saturn V. While simulations must verify the dynamic clearance of the interstage as it falls away, Figure 4.6 shows significant clearance even if the engine is gimbaled as far as 5 degrees. Should clearance prove to be inadequate then a thrusting separation system would be needed to split the Interstage Cylinder and push it aside, similar to conventional fairing separations.

# 4.3.4 Interstage Frustum

The Interstage Frustum is 120" long and transitions from the Upper Stage diameter to the SRB diameter. It contains the booster Roll Control system, the booster avionics, and the recovery system. It interfaces to a clevis ring on the booster, which is fastened to the forward segment. On the pad, the Frustum will need umbilicals for a purge of the enclosed volume and an electrical connection to the booster avionics.

# 4.3.5 Interstage Specific Trades

## Titanium Bracket Manufacturing

The best method of manufacturing the titanium brackets of the truss Cylinder needs to be determined. They could be welded, cast or machined.

Open vs. Closed Interstage Cylinder

The baseline Cylinder is an open truss. The alternative is a closed Cylinder. This trade would be driven primarily by the Upper Stage MPS needs and any complexities involved in getting the RS-25 engine bleed flow gases vented overboard. Any potential aerodynamic effects from the open Cylinder must also be assessed.

# Upper Stage Propellant Settling

The baseline configuration shows solid settling motors mounted on the interstage which are fired immediately prior to RS-25 start up. This is the same solution used by the Saturn V. The alternative is to oversize the RCS pods on the Upper Stage to provide enough aft thrust to settle the propellant.

## Interstage Cylinder Separation Clearance

The dynamics of the separation of the Cylinder around the RS-25 must be determined. If a direct fall away of the Cylinder cannot be relied upon, then the Cylinder must be separated into pieces and jettisoned sideways, away from the vehicle.

### 4.4 BOOSTER

An expanded view of the booster is shown in Figure 4.7. A clevis ring is mounted on the forward stub of the forward segment to provide an interface to the



Figure 4.7 Booster Expanded View

Interstage Frustum. The baseline configuration is essentially unchanged from the Shuttle 4 segment SRM and Aft Skirt. A second systems tunnel, 180 deg from the existing tunnel, is added to help balance aerodynamic roll disturbances and provide extra data capability for development flights or specialized communications between the Spacecraft Adapter avionics and the booster.

### 4.4.1 Booster Avionics

Most of the booster avionics are mounted on an equipment shelf just aft of the parachutes. Antennas are mounted on the skin in roughly the same locations as on the forward skirt of the Shuttle Solid Rocket Boosters. They are mounted on the aft side of the recovery system bulkhead, keeping them in a watertight compartment, similar to the existing SRB forward skirt.

# 4.4.2 Booster Recovery System

In the system models, the booster recovery system uses mass properties from the Shuttle industry team, and parachute volumes from the existing SRB. Because the booster separates higher, the reefing of the parachutes must be different to account for the supersonic deploy of the initial drogue chute.

Forward of the parachutes is the forward Frustum Bulkhead to protect the recovery system until it is deployed. It is a flat composite facesheet, aluminum honeycomb sandwich with external TPS. Prior to deployment of the parachutes the bulkhead is cut around the circumference by shape charge and pushed away by a series of springs around the edges.

#### 4.4.3 Booster Roll Control

Roll control is needed on the booster to counter potentially large roll disturbances generated by the motor itself. The roll control system consists of 4 sets of 4 900# thrusters. Each set uses a blow down bipropellant hydrazine system. The thrusters and tanks are self contained and mounted on a panel for easy integration.

## 4.5 CHANGES FROM ESAS 60 DAY STUDY

#### 4.5.1 GENERAL

- completed a primary structure loads and sizing cycle.
- refined structural joint details based on design practices and manufacturing preferences
- domes modeled in 8 gores with weld lands to accommodate manufacturing. 18.5' DIA outside current spin form tooling.
- dome end caps and interfaces added
- barrels and rings modeled in 4 pieces with end buildups to accommodate manufacturing
- corrected insulation density from 4.5 to 2.5 PCF per Thermal Team definition

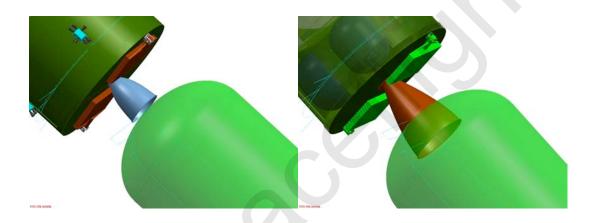
# 4.5.2 SPACECRAFT ADAPTER

- avionics refined per P-1 equipment list
- avionics mounting equipment shelves added
- separation bolt & bracket details added
- lengthened to accommodate CEV 6/22 delivered nozzle size

The length of the Spacecraft Adapter was changed from the ESAS 60 day study to accommodate the length of the CEV nozzle from a model received 6/22/05. Towards the end of the update a new CEV model was received on 8/17/05 with a much longer nozzle. The

Spacecraft Adapter was not updated to this new length because it would have invalidated much of the analysis data and significantly increased the weight. The CEV/CLV interface must be worked to attain the best system solution.

Assy Name:	ESAS_CM_SM_5HM	ESAS_	CM_SM_ASM_32HD-5HM
CM:	ESAS_LUNAR_CM_5HM_25DEG	ESAS_	LUNAR_CM_5HM_32HD
SM:	ESAS_SM_ASSEMBLY_5_5_MCS	ESAS_	SM_ASSEMBLY_5HM
at MSFC:	6/22/05	8/17/05	;
LES, in:	not included	216	(tip to bottom of cylinder)
CM, in:	152.4	111.8	(docking ring to SM)
SM, in:	138.6	138.6	
Nozzle, in:	62.5	110.5	



## 4.5.3 UPPER STAGE

#### LH2 tank

- changed from 4 to 3 barrels (holding 14' length to increase tool availability)

#### Forward Skirt

- shortened (in conjunction with Spacecraft Adapter lengthening)
- Routing/Harness mass placeholder refined with Avionics and MPS equipment lists

### Thrust Structure/Aft Skirt

- Aft skirt combined with thrust structure to avoid factory joint complications
- aft skirt cylinder lengthened to fit 12"ID RD-25 feedline.
- APU pods mass/volume placeholders added, each pod mass based on Advanced Concepts value
- small diameter of thrust cone reduced to minimum size for engine mounting and crossbeam resized
- honeycomb base closeout removed (negated by the smaller cone and thermal blanket)

#### <u>Intertank</u>

- access door added

## **MPS**

- thermal curtain added around powerhead.

#### RCS

- enclosure weight reevaluated

#### LOX Tank

- placeholder slosh baffles modified to ET cross section and spacing
- sump replaced with single outlet directed at engine inlet
- internal fill/drain piping added to accommodate single fill/drain umbilical for ground ops.

## 4.5.4 INTERSTAGE

#### Cylinder

- went from 2 ullage motors to 4 based on Saturn history
- sized ullage motors as TX-280-10, Saturn S-IV motor of approximately the right size

#### Frustum

- modified Roll control packs to LRU's for ground operations
- matched avionics to Avionics team equipment list

#### **4.5.5 BOOSTER**

- matched delivered SRM model masses to existing lightweight values provided by Advanced Concepts (some equipment not included, hence a desire to maintain 5% margin on this existing hardware)
- matched system tunnel masses to Advanced Concept values based on existing hardware.
- went to 4 from 2 BSM's from a VIPA Systems team decision.
- matched aft skirt top level weight to a rolled up aft skirt weight provided by Advanced Concepts

## 4.5 SRB ACOUSTIC MODE

The SRB has an axial acoustic thrust mode with an oscillating axial load of A sin f where A is as high as 80 Klb, i.e., 160 Klb peak to peak. For the 4 segment booster f is 15 Hz. This means that there is something like a 0.1-0.2 g oscillation in acceleration out of 2 g or so total acceleration, depending on the time of flight. The CLV first axial frequencies are well separated, so the structure should not tune with the 15 Hz mode. The second axial frequencies sweep through the 15 Hz mode. But this should not be a problem since it is a forced oscillation, not an instability. Also, there would be a lower response at the second mode frequencies. Payloads may have a requirement for component frequencies to be detuned from 15 Hz.

## 4.6 CLV UPPER STAGE MPS AND ENGINE OVERVIEW

This initial input for the Exploration System Architecture Study was a very brief and accelerated effort resulting in a rough draft concept and hopefully a useful set of issues and recommended studies or technical trades for consideration. This effort was organized by a very small number of

personnel with little time for detailed analysis. As yet, the full technical capability of NASA and the aerospace community have not brought to bear on this task due to the required brevity and initial confidentiality of the study.

The upper stage main propulsion and upper stage engine information was organized by engineers from the NASA-MSFC Propulsion Systems Department, Liquid Engines and Main Propulsion Systems Branch (ER21), they are: Matt Devine, Marc Neely, Steve Sexton, and R. H. Coates. A limited number of experts were consulted as time would permit for assistance with information on the SSME and Shuttle MPS and Orbiter as well as similar propulsion systems and sub-components.

#### **Assumptions**

The initial conceptual input for the Crew Launch Vehicle (CLV) second stage main propulsion system described herein was based on the following "going-in" assumptions:

- One Block II Space Shuttle Main Engine (aka "RS-24D") used in an upper stage (2nd stage) application with minimal changes to the engine and minimal changes to interface conditions
- 2. Single burn, approximately 350 seconds with no re-start
- 3. Start to 104.5 percent "power level" (percent of nominal engine main chamber pressure)
- 4. Shutdown from 104.5 percent power level
- 5. Second stage vehicle and propulsion system will not be re-used

#### **Ground Rules**

The basic ground rules for our technical input to this rough draft concept were:

- safety
- minimize calendar time for development

There was no attempt by this small team focused on the upper stage MPS and engine subsystem to consider hardware commonality to future applications, such as:

- MPS subsystem commonality between single-engine and multi-engine applications
- MPS or engine subsystem commonality between the current second stage concept and any Earth Departure Stage (EDS)

#### Resources

For this brief study the focus for the main propulsion system and upper stage engine effort was on comparable systems. Basic functional overviews, schematics, interface documents, and any applicable mass/energy balance data for the following systems were utilized to initialize inputs to the ESAS study.

- Shuttle Main Propulsion System (MPS), External Tank (ET), and Orbiter
- Space Shuttle Main Engine (SSME)
- Apollo Saturn V, S-IVB (3<sup>rd</sup> stage)
- Apollo Saturn V, S-II (2<sup>nd</sup> stage)

- Delta II upper stage and Delta IV boost and upper stage
- Atlas III/V upper stages

# 4.6.1 Upper Stage Main Propulsion System Overview

The Crew Launch Vehicle (CLV) Upper Stage Main Propulsion System (USMPS) provides the second stage ascent velocity increment from first stage separation to orbital insertion. The magnitude of the velocity increment supplied by the upper stage depends on payload mass, mission trajectory and system limitations.

The USMPS consists of the following major subsystems: one upper stage main engine, one main engine controller, one oxygen tank, one hydrogen tank, the propellant management subsystem and helium subsystem, two redundant auxiliary power units, two thrust vector control units, and two engine hydraulic thrust vector control (gimbal) servoactuators.

The Upper Stage Main Engine (USME) is the Block II Space Shuttle Main Engine, and will also be designated the RS-24D. The engine was originally designed as the reusable booster main engine subsystem for the Space Shuttle Program. The engine system will be used as an expendable engine system for the CLV. The engine is a high performance fuel-rich staged combustion rocket engine with variable thrust capability and two-axis gimbal capability. The propellant fuel is liquid hydrogen and the oxidizer is liquid oxygen. The propellant is carried in separate tanks in the second stage vehicle above the engine and supplied to the engine under pressure. The engine can be gimbaled plus or minus 10.5 degrees in the yaw axis and plus or minus 10.5 degrees in the pitch axis for thrust vector control by hydraulically powered thrust vector control actuators.

The upper stage main engine (USME) can be throttled over a range of 65 to 109 percent of the design rated power level (based on main chamber static pressure) in 1-percent increments, however, 104.5 percent power is selected as the desired thrust level for nominal second stage operation. The thrust level of 104.5 percent power is the current nominal booster ascent thrust level for the Shuttle program. A value of 100 percent power level corresponds to a thrust level of 469,449 pounds-force (lbf) in a vacuum (or 375,191 lbf at sea level). A value of 104.5 percent corresponds to 490,847 lbf vacuum (or 396,569 lbf at sea level). A value of 109 percent corresponds to 512,271 lbf in a vacuum (or 417,992 lbf at sea level).

The upper stage main engine is currently used as a booster main engine system for the Space Shuttle Program. The engine was originally intended for 7.5 hours of cumulative operation over a span of 55 starts for the Shuttle program. Throughout the throttling range, the ratio of the liquid oxygen-liquid hydrogen mixture is maintained at about 6-to-1 (currently 6.032 for Shuttle). The nozzle area ratio is approximately 69-to-1. The engine is approximately 167 inches long and 94 inches in diameter at the nozzle exit with a nominal dry mass of 7748 lbm (8212 lbm wet).

The main engine controller is a digital computer system and electronics package mounted on the engine. It operates in conjunction with engine sensors, valve actuators and spark igniters to provide a self-contained system for monitoring engine control, checkout and status. The controller is attached to the forward end of the engine.

Currently on the Shuttle system main engine data and status collected by the controller are transmitted to an engine interface unit (EIU), which is mounted in the vehicle. The EIU transmits commands from the vehicle computers to the main engine controller. When engine data and

status are received by the EIU, the data are held in a buffer until the EIU receives a request for data from the vehicle computers.

Two redundant hydraulic systems provide hydraulic pressure to position the main engine valves and servoactuators for thrust vector control during the engine's operation. Two redundant auxiliary power units (APUs) provide mechanical shaft power through a gear train to drive the hydraulic pumps that provide hydraulic pressure to their respective hydraulic systems. Both the Space Shuttle Orbiter and the Solid Rocket Booster (SRB) employ APU / hydraulic subsystems and both systems are similar in many respects. Initial evaluation of the Orbiter and SRB APU / hydraulic systems indicates that the SRB system may be better matched to the capability required by the main engine. There are no hydraulically actuated flight control surfaces on the CLV as there are on the Shuttle Orbiter. It will likely make more sense to adapt the SRB APU and hydraulic system to supply hydraulic power to the engine control valves and the thrust vector control actuators for the upper stage main engine.

The two thrust vector control units receive commands from the vehicle computers and send commands to the engine thrust vector control actuators. The units are electronics packages mounted in the vehicle. Hydraulic isolation commands are directed to engine gimbal actuators that indicate faulty servovalve position. In conjunction with this, a servovalve isolation signal is transmitted to the computers.

The main engine hydraulic servoactuators are used to gimbal the main engine and provide thrust vector control. There are two actuators per engine, one for pitch motion and one for yaw motion. They convert electrical commands received from the vehicle computers and position servovalves, which direct hydraulic pressure to a piston that converts the pressure into a mechanical force that is used to gimbal the engine. The hydraulic pressure status of each servovalve is transmitted to the thrust vector control units.

The upper stage MPS propellant management subsystem consists of the distribution ducts or lines, and valves by which the liquid propellants pass from ground systems to the vehicle propellant tanks or from the tanks to the main engine, and by which gaseous propellants pass from the engine back to the respective tanks for autogenous pressurization. The oxygen and hydrogen feed lines will be 12-inch diameter, the same as for the Shuttle, with the same size prevalves. The ground supply ducts and valves, umbilical / disconnects, and onboard upper stage fill / drain ducts and valves will be sized for safety and performance. The existing Shuttle fill/drain systems are sized for 8-inch diameters. It is proposed that 2-inch diameter components be evaluated because the 2-inch size ducts, valves, and disconnects are used for EELV upper stage fill/drain subsystems. All the valves in the propellant management subsystem are under direct control of either ground-based controls (pre-launch) or vehicle control computers (flight) and are either electrically or pneumatically actuated. Some of the same valves may be monitored and manually controlled by the flight crew as required by mission requirements.

The upper stage MPS helium subsystem consists of a series of helium supply tanks and regulators, check valves, distribution lines and control valves. The subsystem supplies the helium used within the engine to purge the high-pressure oxidizer turbopump intermediate seal and preburner oxidizer domes and to actuate valves during emergency pneumatic shutdown. The upper stage main engine will use approximately 25 lbm for the 350 second second stage burn (through engine cutoff purges), or roughly half the helium used on each Shuttle mission per SSME. This translates to roughly half the needed volumetric helium capacity. Including substantial margin, the current engine pneumatic helium supply could be reduced to using three (3) of the existing 4.75 cubic-foot 4500 psig helium tanks (composite overwrapped pressure

vessels). This will also allow for a redundant capability since two tanks could supply the helium requirement for a single engine (a single tank could be isolated should a problem occur). One or possibly two more 4.75 cubic-foot tanks could be used for the vehicle to actuate all the pneumatically operated valves within the propellant management subsystem on the vehicle, and to supply any required inerting of enclosed vehicle volumes to protect against fire and explosion risks. The exact requirement for the MPS valves and hazardous gas mitigation must be determined.

# 4.6.2 Preliminary Assessment Of Contractor Capability

The following is a very rough, initial status of the engineering and production capability for a subset of the major MPS subsystem suppliers (as used on the Space Shuttle system). This effort must be completed and advanced development activities pursued in detail.

Several components may be good candidates for advanced development or fabrication demonstration efforts, including but not necessarily limited to: large 12-inch pre-valves, 2-inch to 4-inch fill/drain valves, large umbilical disconnects, a new simplified H2 or O2 tank vent / relief valve, an oxygen recirculation pump, and adapting the current SRB APU / hydraulic system for use with a single main engine (Block II SSME).

Critical consideration must be given to the appropriate program support to human space flight qualification / re-qualification for new or updated designs for the CLV.

- Propellant feed lines and gimbal (flex) joints Arrowhead is the original supplier for the manifold, fill/drain, 12-inch and 17-inch feed lines on the Shuttle orbiter MPS. They supply Atlas V and Delta IV vehicles with similar components. United Space Alliance (USA) recently placed an order (then cancelled) for a complete set of Shuttle MPS feed lines. Arrowhead still has full capability to provide the main feed line systems for the SSME.
- 2) Pre-valves for LH2 and LOX 12-inch feed lines Fairchild was the original supplier for these components but was acquired by Vacco Aerospace Products. Vacco's Director of Aerospace Engineering and Programs indicated Vacco has the design authority for the pre-valves. They have the blueprints, tooling, and all specifications. However, there are some issues that must be addressed:
  - a) Vacco's drawings are not up to date. USA has made several changes over the years via cumulative engineering orders (EO's) and bypassed Vacco's data management process. Therefore their internal specifications are not up to date nor are the blue prints.
  - b) Some of the tooling was unique to the original equipment that is no longer available or out of date.
  - c) Many of the non-unique tools are worn or damaged beyond use. Production processes will have to be modernized for the old designs to be built on current equipment (specifically CNC machining).
- 3) Propellant fill/drain valves and quick disconnects The original Shuttle MPS design uses 8-inch diameter fill/drain valves made by Fairchild. The 8-inch valves and ducts may be too large for the proposed size of the upper stage concept and if used would constitute a weight penalty. Since the Delta IV unmanned Evolved Expendable Launch Vehicle (EELV) uses a 2-inch fill/drain valves and ducts for their upper stage, this may be functionally capable for the CLV upper stage. Vacco is the current manufacturer of just

such 2-inch valves, but a detailed analysis of the components versus our requirements (including human-rating) would have to be conducted to see if this component can be used "as-is".

- 4) Helium pressure regulators Eaton was the original supplier, but Vacco purchased that company as well. The CLV upper stage may require similar components compared to the Shuttle or EELV programs so this is probably a relatively obtainable component depending on requirements.
- 5) Umbilicals This hardware will require advanced development effort because it will be unique to the vehicle interfaces. Current EELV designs (Atlas/Delta) are similar in deliverable materials and subsystem components for umbilicals for hydrogen, oxygen, nitrogen, helium, and electrical. These should be evaluated; however, the CLV upper stage will still require a unique system depending on specific vehicle requirements and interface locations.
- 6) Tank vent/relief valve The original supplier for this component was Eaton. Although Vacco now owns Eaton and has the design authority, USA has changed the contract award to Ketema. The Ketema valve is similar to the original Eaton valve but is cheaper. For the CLV, this item needs to be an advanced development effort due to the issues and problems that have occurred with this component on the Shuttle MPS. It has also been proposed in discussions for this study that a new concept be considered. The new concept would split the vent and relief functions into two separate valves thus simplifying the functionality.
- 7) Hydrogen recirculation pump The original supplier for this component was Sundstrand, now part of Hamilton/Sundstrand and United Technologies/Pratt and Whitney. P&W has the design authority for this component. Since three pumps were used on Shuttle, one for each SSME, the assumption for the CLV upper stage is that one pump will work for the single engine but this assumption needs validated. We are currently trying to contact vendor for further status of component manufacturability. Also, the capability for producing a oxygen recirculation pump must be assessed.
- 8) Auxiliary power unit (APU) The current system used on the Shuttle orbiter/MPS and SRB have proven to be reliable and appears to be suitable for the CLV upper stage engine valve and thrust vector control actuator needs. The APU and hydraulic system on the SRB may be more closely aligned with the needs of the upper stage main engine. It is recommended that the redundant APU / hydraulic systems from the Shuttle SRB be evaluated for use on the CLV upper stage.

Other original contractor / suppliers from the Space Shuttle Program MPS are listed below. These should also be assessed for their current capability for engineering design, fabrication, and assembly.

- Aeroflex Laboratories, Plainview, NY (MPS vibration mounts);
- Airite Division, Sargent Industries, El Segundo, CA (MPS surge pressure receiver);

- Ametek Calmec, Pico Rivera, CA (1.5-inch and 2-inch liquid oxygen and liquid hydrogen shutoff valve, 4-inch liquid hydrogen disconnect and 2-inch gaseous hydrogen/gaseous oxygen disconnect);
- Ametek Straza, El Cajon, CA (8-inch liquid hydrogen/liquid oxygen fill and drain, 2- and 4-inch liquid hydrogen recirculation lines, high-point bleed line manifold and gimbal joint);
- Arrowhead Products, Division of Federal Mogul, Los Alamitos, CA (12 to 17-inch-diameter liquid oxygen and liquid hydrogen feed lines and flexible purge gas connector);
- Astech, Santa Ana, CA (MPS heat shield);
- The Boeing Company, Rocketdyne Propulsion and Power, Canoga Park, CA, is the prime contractor for the Block II SSME;
- Brunswick, Lincoln, NB (17.3- and 4.7-cubic-foot capacity helium tanks);
- Brunswick-Circle Seal, Anaheim, CA (helium check valves, gaseous oxygen and gaseous hydrogen 1-inch helium pressurization line, 0.375-inch liquid hydrogen relief valve and engine isolation check valves);
- Brunswick-Wintec, El Segundo, CA (helium filter);
- Coast Metal Craft, Compton, CA (metal flex hose);
- Conrac Corp., West Caldwell, NJ (engine interface unit);
- Consolidated Controls, El Segundo, CA (oxygen pressure primary flow control valve and hydraulic valve, hydrogen/oxygen pressurant flow control valves, 20-psi helium regulator, 850-psi helium relief valve and 750-psi helium regulator);
- Fairchild Stratos, Manhattan Beach, CA (12-inch pre-valves, 1.5-inch liquid oxygen disconnect, 8-inch liquid oxygen and liquid hydrogen fill and drain valves, and gaseous nitrogen and gaseous hydrogen disconnects);
- Gulton Industries, Costa Mesa, CA (pogo pressure transducer);
- Hamilton-Sundstrand / United Technologies (hydrogen recirculation pumps);
- K-West, Westminister, CA (liquid oxygen and liquid hydrogen external tank ullage pressure signal conditioner, MPS differential pressure transducer and electronics propellant head pressure);
- Megatek, Van Nuys, CA (MPS line flange cryo seals);
- Moog Inc., East Aurora, NY (main engine thrust vector control / gimbal actuators);
- Parker Hannifin Corp., Irvine, CA (1-inch relief isolation valves, pogo check valves, 17-inch liquid hydrogen and liquid oxygen disconnects, 8-inch liquid oxygen and liquid hydrogen relief valves);
- Simmonds Precision Instruments, Vergennes, VT (liquid oxygen and liquid hydrogen point sensors and electronics);
- Sterer Engineering, Los Angeles, CA (main engine hydraulic solenoid shutoff valve);
- Whittaker Corp., North Hollywood, CA (750-psi / 250-psi helium regulator);
- Wright Components Inc. Clifton Springs, NJ (two-way pneumatic solenoid valve, threeway helium solenoid valve and hydraulic latching solenoid valve).

#### 4.6.3 Recommended Technical Studies / Trades

The initial input for the ESAS effort will obviously require the initiation of more rigorous analyses and a formal development process.

Several areas were discussed in the course of gathering the data for this brief study that will require specific detailed analysis in order to move toward an optimum solution or at least quantify likely areas of technical or programmatic risk. More issues will obviously surface as details are developed further. The issues discussed as of the date of this report include:

1. Starting the Block II Space Shuttle Main Engine in an upper stage application. The effects and sensitivities of propellant inlet conditions on the thermal conditioning (chill),

start sequence, ignition, and detailed operational and functional design implications to the SSME versus tank size and mass must be fully investigated. The team must evaluate inlet conditions trending with time (boost and 2<sup>nd</sup> stage) and look at the effects of acceleration on engine thermal conditioning, tank size and pressure loads, assess autogenous tank pressurization requirements, vehicle mass and performance, and overall vehicle structural loads for the entire vehicle.

- Engine to vehicle (main propulsion system) interfaces. An analysis of the operation of the engine at mission specific conditions should be conducted (with dispersions) and a rigorous review of all interfaces should be completed to establish a more complete list of issues and technical trades.
- 3. Trade an open versus closed interstage. A detailed assessment of the interstage arrangement and impacts to performance (size, weight) and safety (haz gas detection and an engine system leak check, as well as fire / explosion prevention) must be completed. The evaluation should consider the issues associated with routing the oxygen, hydrogen, and hydraulic drain line fluids overboard during pre-hotfire thermal conditioning and during operation. The drain lines are located at the main engine at the nozzle exit plane. The study must evaluate pro's and con's for both a closed interstage and the baseline "open truss" interstage, including overall vehicle performance (aero, structural loads).
- 4. Maintaining hydrogen tank ullage conditions. Evaluate impacts of RSRM causing tank sloshing during first stage operation and possible ullage collapse. Investigate possible alternatives (helium pre-press capability) or mitigating steps.
- 5. Pogo. Evaluate the necessity for the single engine upper stage and the impact to zero-g shutdown (versus settling motors).
- 6. Evaluate need for and issues associate with hydrogen and oxygen recirculation pump redundancy. Initial input does not have a redundant capability.
- 7. Evaluate placement of liquid hydrogen 12-inch pre-valve and recirculation pump, bleed valve, and line with respect to safety (potential engine MFV or hydrogen system leaks) and for optimal operation for thermal conditioning of the hydrogen feed line and engine system.
- 8. Evaluate need for hydrogen fill / drain relief (between inboard and outboard valves) back to hydrogen system. The current baseline input is relief to an overboard dump (flare stack).
- 9. Engine system safety. Analyze engine/vehicle system performance/sizing, and safety margins for abort scenario and mission maximum design conditions to set mission and operational requirements to envelope needed margins and to set requirements for failure detection, isolation, and recovery (FDIR).
- Sizing and adaptation of SRB APU / hydraulic for use with the upper stage main engine. Perform detailed analysis and design effort to adapt the SRB system for use with the main engine (valves and TVC).

## 4.7 SECOND STAGE CHEMICAL PROPULSION RCS SYSTEM

As an initial point of departure, the attitude control requirements for Saturn IB S-IVB was used to size the Crew Launch Vehicle (CLV) second stage reaction control system (RCS). Table 4.7-1 shows the total impulse, and corresponding propellant mass, required by the S-IVB broken down by mission function.

Scaling linearly with respect to mass provides an initial estimate for the CLV second stage attitude control propellant requirements. Like the Saturn IB S-IVB stage the RCS propulsion system was separated into two independent, but identical modules, located externally on the aft end of the second stage and mounted 180 degrees apart. As with the first stage roll control system, the modular concepts allows off-line processing, propellant loading and vehicle integration as a self contained, sealed unit. As before this modularity will allow one-for-one easy replacement and block upgrade replacements when non-toxic propellant options mature.

Table 4.7-1Attitude Control Total-Impulse Requirement

Apollo S-IVB Mass 299000 lbm		om		Apollo	S-IVB		CLV (est	imate)		
CLV (estimat	te) 475000 lt	om		Ispv	280.0 sec	Ispv	280.0 sec	Ispv	280.0 sec	
	•	Impulse R	equired	Propellant	Required	Propellant F	Required	Propellant	Required	
	Operation					w/ Ullage & Sep		w/o Ullage & Sep		
		(lbf-sec/m	(lbf-sec/module)		(lbm/module)		(lbm/module)		(lbm/module)	
		Nominal	Variable	Nominal	Variable	Nominal	Variable	Nominal	Variable	
Powered Flig	ht									
	Roll Control		1540		5.5		8.7		8.7	
Earth Orbit:										
	Initial Convergence		440		1.6		2.5		2.5	
	Establish Orbital Pitch Rate	40		0.14		0.2		0.2		
	Attitude Stabilization									
	Pitch Limit Cycle	1040		3.7		5.9		5.9		
	Yaw Limit Cycle	7480		26.7		42.4		42.4		
	Vent Distribution									
	LOX		585		2.1		3.3		3.3	
	LH2		610		2.2		3.5		3.5	
Total		8560	1775	30.6	6.3	48.6	10.1	48.6	10.1	
Allowances fo	Allowances for off nominal performance		178	3.1	0.6	4.9	1.0	4.9	1.0	
Total (regired	)	11369		40.6		64.5		64.5		
Reserve		1600		5.7		10.4		10.4		
Available		13000		46.4		74.9 lbm		74.9 lbm		
Ullage Settling and Stage Separation (0.01g's for 10 sec)					75.0 lbm		0.0 lbm			
Total						149.9 lbm		74.9 lbm	-	

\* Total of two (2) DCC propulation modules not atom for 2 DOC

Although additional analysis is needed, the initial design point for the RCS module contains six 890 N (200 lbf) and three 4,450 N (1,000 lbf) pressure-fed, nitrogen tetroxide and monomethyl hydrazine thrusters. The propulsion system is pressure regulated providing a consistent minimum impulse bit and thrust throughout second stage flight. Figure 4.7-1 shows a top level propulsion schematic of the CLV second stage chemical reaction control system. The system is not only sized to provide attitude control, but propellant ullage settling (0.01-g for 10 sec) and separation maneuver during second stage / SRB staging events.

Table 4.7-2 shows the top-level mass breakdown of the roll control propulsion module. The table lists the masses of the major system components as well as, the propellant required and the principle geometry of the propellant tanks.

Small solid motors are ideal for providing high thrust for short duration and this was the approach taken for the ullage settling and separation impulse for the S-IVB stage. Although a more detailed examination is needed, (utilizing data derived for the Saturn IB S-IVB) a quick trade between a dedicated solid motors for ullage settling versus adding this capability to the seconds RCS shows the solid motors did not trade favorability from a mass point of view.

Pressure
Closed Pyro
Valve
Capped
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Color Processor
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Figure 4.7 – 1 Second Stage Propulsion Schematic

Table 4.7-2 Second Stage RCS Propulsion Mass Breakdown

Roll Control Propuls	ion System							07/05/2005
With Prop Settling								
Second Stage & Payloa	ıd	475,000 lbm						
Dry mass								
	Prop/Press Tank	<	10.7 kg			23.6 lb		
	Helium		0.33 kg			0.7 lb		
	System Compor	nents*	9.4 kg			20.7 lb		
	Thrusters (3 X 1	000#, 6 X 200#)	52.0 kg			114.7 lb		
	15% Misc H/W		10.8 kg			23.8 lb		
	10% Dry Mass	Contingency	8.3 kg			18.3 lb		
	Propellant (resid	ual)	2.0 kg			4.5 lb		
Inert Propulsion				93.6 kg			206.3 lb	
Propellant (usable)				68.1 kg			150.0 lb	
Propulsion System					161.7 kg			356.3 lb
Initial System Mass				-	161.7 kg			356.5 lb
*Does not includes seco	ondary support str	uctures						
	Oxidizer	Fuel	Helium					
Tank Volumes	31,677 cc	34,513 cc	6,920 cc					
	1,933 in^3	2,106 in/3	422 in^3					
Dia	12.0 in	12.0 in	9.3 in					
Overall Length	22.6 in	22.6 in						
Propmass Per Tank	96.2 lb	58.3 lb	0.7 lb					
	43.6 kg	26.4 kg	0.3 kg					

The primary list of guidelines and assumption for this phase of the study are out lined in Table 4.7-3.

#### Table 4.7-3 Guidelines and Assumption

- Propulsion system is considered expendable, two-fault-tolerant (minimum)
- Pressure vessel characteristics:
  - Factors of Safety: Propellant Tanks 2.0; Pressurant Tanks 2.0
  - Materials: Propellant Tanks Ti (6Al-4V); Pressurant Tanks COPV
- Anti-slosh/propellant management device ~30% of propellant tank shell mass
- Operating Pressures
  - o Propellant Tanks 2.4 MPa (350 psia)
  - o Pressurant Tanks 31 MPa (4,500 psia) down to 5.5 MPa (800 psia)
- Propellant tank ullage volume
  - Second stage RCS, pressure regulated 5%
- Propellant residual 5%
- Propulsion component (valves, regulators, filters, etc) derived from spacecraft and STS OMS propulsion system.
- Helium pressurant sized for adiabatic blowdown
- Propulsion hardware TRL 9, design margin on all dry mass 10%
- Storage life (with loaded propellant) > 90 days
- Active system life, < 60 minutes</li>

## 4.8 ABORT SENSORS AND ABORT INITIATION

This section examines what set of abort sensing capabilities might be required in order to provide for detection of all realistic abort scenarios that can be detected. It also looks at how to do the abort initiation. Both topics are approached from a historical perspective. This section also includes a discussion of the abort modes used on Apollo and finishes with a few other pertinent facts related to abort. The primary source of data for this section is "Ascent Aborts: A Historical Perspective" by T. Harold Robertson of NASA/JSC. The OSP-ELV Human Flight Safety Certification Study (March 2004) was also used.

The Mercury-Atlas system had the following abort sensor detection capabilities:

- LOX tank pressure
- Differential pressure across the intermediate bulkhead
- Attitude rates on all axes
- Engine injector manifold pressures
- Sustainer engine hydraulic pressure
- Primary electrical power
- Atlas/spacecraft electrical interface circuitry
- Dual sensors for each of these measurements

The Gemini-Titan system had the following abort sensor detection capabilities (it had different propellants as opposed to Atlas and Saturn, so some of the same sensors do not apply to Gemini-Titan):

- ♦ Engine chamber underpressure
- Fuel tank pressure
- Oxidizer tank pressure
- ♦ Stage II engine fuel injector pressure
- Stage I and II disconnects
- Attitude Rate sensing

The Apollo-Saturn system had the following abort sensor detection capabilities:

- Tank pressures
- Attitude rates
- ♦ Engine pressures (thrust)
- Breakwires to detect premature staging

The OSP-ELV Human Flight Safety Certification Study (March 2004) examined ELV failures and concluded that a notional IVHM system could have caught most failures. The sensor parameters in the notional system were:

- Attitude/attitude rates
- ♦ Chamber pressure, turbopump speed
- Helium bottle rapid decay
- Engine stop or nonstart
- Poor engine performance
- ♦ Current/voltage
- Separation failure
- Component overheating

The following failures would not have been caught by the notional system and must be designed out:

- ♦ Lubrication oil flow loss
- ♦ SRM burn through/explosion or other SRM structural failure
- ♦ Liquid combustion instability/explosion
- Structural failures and/or associated fuel leaks

Considering the Shuttle SRB, if a burn through causes only a slow leak and reduction in performance, abort logic can determine whether the system has the capability to reach orbit or not and can determine whether to abort as a result. So depending on the failure modes, SRM burn through might be a failure that the current vehicle can handle in terms of crew survivability.

Following is a combined list that seems to provide as complete a list as necessary as to what should be monitored:

- Attitudes and attitude rates
- ♦ Chamber pressure

- Fuel injector pressure
- Turbopump speed
- Tank pressures (all tanks)
- Engine stop or nonstart
- Engine performance (thrust)
- ♦ Current/voltage
- Stage separation status
- Component overheating
- Hydraulic pressures
- Computer model of the ability to reach orbit from the current condition

#### **Abort Initiation**

In both Mercury and Apollo, certain aborts were automatically initiated, when it was felt there was insufficient time for crew interaction, for example:

Attitude rates exceeding limits

Two or more engines shut down in first stage (Apollo)

Otherwise, the information was presented to the crew for decision. Titan propellants were considered to be less explosive, so all Gemini aborts were manual.

On the Saturn, the Emergency Detection System (EDS) was used to detect launch vehicle conditions that would lead to an abort. If breakup of the vehicle was imminent, it automatically initiated the abort sequence. For slower-developing emergencies, it provided information to the crew to evaluate and manually initiate the abort sequence. It physically resided in the IU ring on the S-IVB stage (not the Apollo spacecraft) and was part of the LV GN&C system. There were modes where the auto system was inhibited (through T+30 seconds and during nominal first stage engine shutdown). Manually- initiated abort was preferred for most of flight. The Launch Director or Flight Director could command an abort request (warning light) that cued the crew to manually initiate an abort. Aborts could also be triggered by the flight termination sequence that would be commanded by the Range Safety Officer. The Saturn included triple redundant digital modules and sensors that allowed two of three voting logic.

### **Apollo Abort Modes**

- Mode I (pad abort (T-45 minutes) through the end of atmospheric flight (LET jettison about 3:07 sec MET): The launch escape tower (LET) was used to rapidly separate the spacecraft (Command module only) from the launch vehicle. The spacecraft followed a ballistic path. Touchdown ranged from the pad area to about 400nm downrange.
- The three subclasses of Mode I aborts were determined by the dynamic pressure environment at the time of the abort, requiring different techniques to orient the CM.
  - Mode I Alpha (low altitude mode) The LES used a pitch motor to orient the assembly. This ensured the vehicle was directed downrange for water touchdown and to escape the "fireball" of the LV explosion.
  - Mode I Bravo Canards were extended from the LES to aerodynamically orient the CM (initiated a tumble in the pitch plane which results in the blunt end forward)

- Mode I Charlie The RCS on the CM was used to establish the proper orientation following manual jettison of the LES. If the attitude platform was bad, then tumble was introduced via the RCS. Aerodynamic forces result in a blunt end forward for attitude (like Mode 1B).
- Mode II (post LET jettison until SPS could be used to make a minimum orbit or footprint encroachment on Africa for lunar missions or Europe for Skylab). The combined Command and Service Modules were manually separated from the launch vehicle, followed by CM separation from the SM. The CM was oriented for entry, which was open loop, full lift, and had high g's (16g limit). Touchdown was also in the Atlantic (typically 350 to 3200 nm downrange).
- Mode III [touchdown encroachment on Africa (for lunar missions) through nominal orbital insertion]. Similar to the Mode II abort except the Service Module SPS engine performed either a posigrade or a retrograde burn prior to separation from the SM. This abort was used to target to a recovery ship which was 3350 nm downrange.
  - There is a subclass of this abort late in the region defined as a "fixed delta V abort" where the SPS retrograde burn results in touchdown in the Indian Ocean and would be used if the CM was No Go for orbit.
- Mode IV overlaps some of the Mode III region and begins when orbit insertion could be achieved through use of delta V from the SPS engine (up to 3000 ft/s) to make up the velocity shortfall of the launch vehicle. It extends through nominal SIVB stage cutoff.
- ◆ The burn attitude for Mode III and Mode IV aborts was a manual maneuver that placed the thrust vector either 31.7 degrees above or below the horizon using a scribe-line on the window.

A few final abort-related findings are of note. For Mercury/Atlas, more than 90% of the mission analysis work was related to aborts. Abort decision criteria will be very difficult to finalize, and if too tight may result in a higher than anticipated abort occurrence. Since a large percentage of failures are propulsion related, which lead to loss of vehicle control, the engine chamber pressure and turbopump speed parameters are critical for monitoring, making the reliability of the measurements critical. Testing of the abort system against vehicle loss of control failure will be necessary to establish confidence in its adequacy. Rate capability of the abort system is critical and will be dependent on time of flight at which the failure occurs.

## 5.0 INTEGRATED VEHICLE ANALYSES

## 5.1 AERODYNAMICS

The aerodynamic characteristics for the CLV-4 crew launch vehicle were obtained from experimental data sets for existing launch vehicles. The power-on base force is from the STS Aero Data Base.

#### M - Mach Number

CA - Axial force coefficient (forebody) @ alpha = zero degrees

CNA - Normal force coefficient slope per degree

CYB - Side force coefficient slope per degree

CMA - Pitching moment coefficient slope per degree

CNM - Yawing moment coefficient slope per degree

CPP - Center of pressure (pitch) ft fwd of base

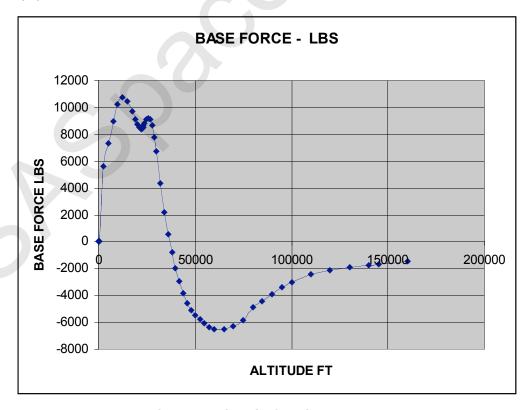
CPY - Center of pressure (yaw) ft fwd of base

CYB = -CNA

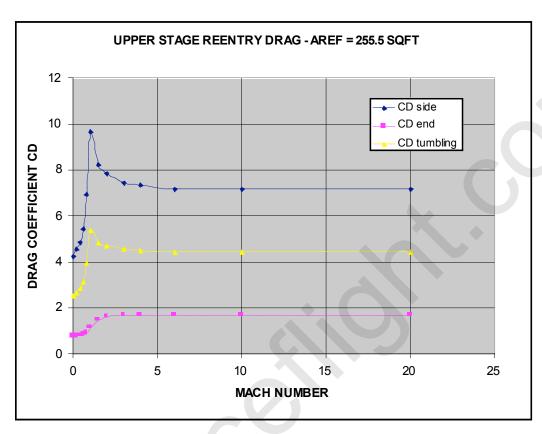
CPP = CPY

**AREF = 116.2 sqft** 

LREF = 12.16 ft



**CLV4 – BASE FORCE VS ALTITUDE** 



The CLV4 base force is presented as a function of altitude and can be modeled as a thrust table in trajectory analysis. Note: Positive base force is negative thrust.

The upper stage reentry aerodynamics are presented as a function of Mach number for three flight conditions. The end-first and side-first flight conditions should bound the debris foot print for the reentering upper stage. The average drag coefficient should yield a better representation of the actual debris splash down.

## 5.2 MASS PROPERTIES

The mass properties of an item include the item's weight, center of gravity, mass moments of inertia, and mass products of inertia. The mass properties for the CLV are reported in this section. CAD models were used to generate the mass properties for the Adapters, the Inter-stage, and the Second Stage. The First Stage mass properties (4 segment SRB derived) were scaled from the RSRM Block Model, ATK Thiokol document TPR07499, dated 3/15/1999. The CEV and LES mass properties were scaled based on Apollo data. Combining these data, the total mass properties for the vehicle stack was developed. These vehicle mass properties were provided for the performance analyses and stability and controls analyses.

The predicted mass properties of an item are the basic mass properties plus the mass growth allowance (MGA). Five percent MGA was applied to existing hardware and fifteen percent was applied new design. The reference coordinate system axes used for calculating the mass properties are a body fixed, rotating right-handed Cartesian coordinate system (CSYS). The CSYS origin (0,0,0) is located forward of the nose of the vehicle with the positive X-axis pointing aft. The mass properties for the CLV are computed about the centroid of the vehicle. A mass

properties report has been delivered under MSFC memorandum number EV12-05-019. The figures below are extracted from that report.

					CL	V4 WEIGHT S	UMMARY					
		-KIL	OGRA	MS-					-POU	NDS-		
	< BU	DGET>	<	CUE	RENT ST	ATUS>	- 1	< BUD	GET>	< CUF	RENT ST	ATUS>
SUBSYSTEM	CONTROL					ALLOWANCE		CONTROL		*WEIGHT		ALLOWANCE
CREW EXPLORATION VEH(CEV)										[ 47400		
LAUNCH ABORT SYSTEM(LAS)	1	1	]	4218	0	( 0.0%)]	1		1	[ 9300	0	( 0.0%)]
S/C ADAPTER	1	1	]	1510	197	(15.0%)]	1		1	[ 3328	434	(15.0%)
UPPER STAGE	[	1	]	183919	1959	( 1.1%)]	1 [		1	[ 405472	4318	( 1.1%)]
DRY LESS SSME				13732	1783	(14.9%)	ĺ			30274	3931	(14.9%)
SSME				3690	176	(5.0%)	ĺ			8135	387	(5.0%)
LOX PROPELLANT				142416	0	(0.0%)	i			313974	0	(0.0%)
LH2 PROPELLANT				23947	0	(0.0%)	i			52794	0	( 0.0%)
RCS PROPELLANT				134	0	(0.0%)	į			294	0	( 0.0%)
INTERSTAGE	ı	1	1	1615	197	(13.9%)]	1 1		1	[ 3560	434	(13.9%)]
DRY STAGE	-	-	-	1508	197	(15.0%)	i			3325	434	(15.0%)
ULLAGE MOTOR PROP				106	0	(0.0%)	į			235	0	(0.0%)
BOOSTER ADAPTER	1	1	1	7844	978	(14.2%)]	[		1	[ 17293	2156	(14.2%)]
DRY MASS				7499	978	(15.0%)	i 🖈			16533	2156	(15.0%)
RCS PROPELLANT				345	0	( 0.0%)				760	0	(0.0%)
SRB 4 SEGMENT	[	1	]	580722	776	( 0.1%))	1		1	[1280272	1711	( 0.1%)]
DRY MASS		_		78881	776	(1.0%)				173902	1711	(1.0%)
SRM/BSM PROPELLANT				501841	0	(0.0%)				1106370	0	(0.0%)
MANAGER RESERVE	1	1	[	3696	0	( 0.0%)]	1		1	[ 8148	0	( 0.0%)]
CLV4 @ IGN			-	805024	4107	(0.5%)		 >		1774773	9054	( 0.5%)

Figure 5.2-1: Mass Properties Summary

EVENT	TIME	WEIGHT	CENTER	OF GRA	VITY	1	MOMENTS OF IN	ERTIA	E	RODUCTS OF I	NERTIA
	(Seconds)	(Pounds)	(Statio	n - In	ches)		(Slug - Fee	t2)		(Slug - Fe	et2)
			Х	Y	Z	IX	IY	IZ	IXY	IXZ	IY
SRB IGN	0.000	1774773.0	2738.5	-0.2	0.0	1080571.	222676976.	222672656.	117634.	-19322.	18850
LO	0.230	1774121.0	2738.5	-0.2	0.0	1060644.	220069840.	220063712.	117359.	-19738.	18856
	20.000	1545525.2	2685.8	-0.2	0.0	890864.	204190480.	204183648.	107713.	-23788.	18960
	40.000	1314270.0	2612.1	-0.3	0.0	719108.	185070576.	185062640.	96054.	-28686.	19064
	60.000	1166292.5	2549.1	-0.3	0.1	624738.	172244672.	172237520.	97434.	-16633.	19037
	80.000	1018314.9	2466.5	-0.3	0.1	530366.	156476304.	156469616.	95951.	-8090.	19008
	100.000	870337.4	2354.4	-0.4	0.1	435989.	136262144.	136256128.	90117.	-4922.	18976
	120.000	722359.9	2194.5	-0.5	0.1	341600.	108868784.	108863208.	77225.	-10520.	18938
SRB SEP	128.042	662858.0	2109.6	-0.5	0.1	303640.	94743936.	94738640.	68976.	-16588.	18919
SSME START	133.741	477208.3	1705.6	-0.8	-0.3	110211.	20882604.	20871870.	-5244.	-71326.	19416
IS JET	138.043	475573.3	1705.6	-0.8	-0.3	110211.	20866292.	20855558.	-5246.	-71326.	19416
IS JET	138.043	472013.6	1702.5	-0.9	-0.3	101278.	20722318.	20711444.	-5519.	-71407.	19416
	140.000	469899.8	1702.4	-0.9	-0.3	101278.	20701222.	20690348.	-5524.	-71409.	19416
	160.000	448190.0	1701.3	-0.9	-0.3	101274.	20493938.	20483088.	-5617.	-71437.	19415
	180.000	426480.2	1699.3	-0.9	-0.3	101269.	20300830.	20289948.	-5792.	-71488.	19414
	200.000	404770.5	1696.3	-1.0	-0.3	101265.	20112890.	20102004.	-6058.	-71567.	19412
LAS JET	211.847	391910.7	1693.9	-1.0	-0.3	101262.	20002264.	19991376.	-6264.	-71629.	19411
LAS JET	211.847	382610.7	1719.4	-1.1	-0.3	101191.	16491458.	16480596.	-4031.	-62618.	16367
	220.000	373760.8	1718.0	-1.1	-0.3	101189.	16421458.	16410623.	-4148.	-62653.	16366
	240.000	352051.0	1713.7	-1.1	-0.3	101182.	16249730.	16238918.	-4518.	-62762.	16364
	260.000	330341.2	1707.9	-1.2	-0.4	101175.	16073162.	16062330.	-5022.	-62912.	16362
	280.000	308631.5	1700.2	-1.3	-0.4	101167.	15888107.	15877254.	-5692.	-63111.	16360
	300.000	286921.7	1690.2	-1.4	-0.4	101158.	15684845.	15673983.	-6563.	-63370.	16358
	320.000	265212.0	1677.2	-1.5	-0.5	101147.	15450752.	15439882.	-7686.	-63704.	16355
	340.000	243502.2	1660.6	-1.7	-0.5	101134.	15173518.	15162637.	-9127.	-64132.	16351
	360.000	221792.5	1639.3	-1.8	-0.5	101119.	14835347.	14824467.	-10980.	-64683.	16347
	380.000	200082.7	1611.6	-2.0	-0.6	101101.	14412291.	14401396.	-13379.	-65396.	16342
	400.000	178373.0	1575.4	-2.3	-0.7	101078.	13867267.	13856345.	-16525.	-66331.	16336
	420.000	156663.2	1527.0	-2.6	-0.8	101048.	13141954.	13131008.	-20728.	-67580.	16328
	440.000	134953.5	1460.3	-3.0	-0.9	101009.	12139527.	12128548.	-26513.	-69299.	16317
	460.000	113243.7	1364.6	-3.6	-1.1	100955.	10682826.	10671802.	-34822.	-71769.	16302
SSME CO	467.488	105115.6	1317.3	-3.8	-1.1	100929.	9954977.	9943930.	-38922.	-72988.	16295
MECO	472.807	104040.6	1310.5	-3.9	-1.1	100925.	9848475.	9837426.	-39518.	-73165.	16294
S/C SEP	503.065	47400.0	898.9	0.0	-0.1	32883.	4345350.	4332404.	309.	-68697.	14716

Figure 5.2-2: Sequenced Mass Properties

### 5.3. PERFORMANCE AND TRAJECTORIES

This section discusses the development of the reference trajectories and the various performance studies. The ground rules, assumptions, and the necessary input data are detailed.

## 5.3.1 GROUND RULES AND ASSUMPTIONS

The following is a summary of the ground rules and assumptions used to generate the Rev. 2 and Rev. 4 reference trajectories. Unless specified, the assumptions are valid for both revisions. A complete set of trajectory input data can be found in section 5.3.1. The POST input and output files can be obtained from Terri Schmitt (MSFC/EV42).

#### Crewed Launch Vehicle (CLV-4)

- ◆ Two Rev. 2 reference trajectories:
  - o Due East Mission: 30 x 160 nm orbit at 28.5° inclination
  - o ISS Mission: 30 x 160 nm orbit at 51.6° inclination
- One RSRM: lightweight, reusable, 60 deg PMBT, 0.368 ips burn rate (Thiokol TPR07499 3/15/99
- RSRM knockdown (Shuttle flight derived data) is incorporated (NSTS 08509, Volume I, Table 5-20).
- One Block II SSME at 100% and 104.5% nominal power level (Rocketdyne IL-2002-478-007 3/25/02). Start-up and shut-down transients are used (R.H. Coates/ER21, 7/5/05)
- Vehicle components: inline SRM, booster-to-interstage adapter, interstage, upper stage, upper stage-to-payload adapter, payload and launch escape system
- Rev. 2 mass properties data from Holly Chandler (MSFC/EV12 on 7/6/05) and documented in Section 5.2 of this report.
- Rev. 4 mass properties data from Holly Chandler and Roy Lutonsky (MSFC/EV12 on 9/8/05).
- ♦ 5% dry weight margin on SRB (non-propellant weight only)
- ◆ Maximum acceleration limit = 5.0 g's (No throttling requirement)
- ♦ No maximum Q-alpha limit
- No moment balance
- 3-DOF aerodynamic data and base force data provided by Joe Lowery (MSFC/EV33, 6/16/05)
- ♦ 1963 Patrick AFB atmosphere with KSC mean annual winds
- MECO altitude measured with respect to equatorial radius
- Flight Performance Reserve = 1% Ideal □V (Rev. 2)
- Flight Performance Reserve = 1.1% Ideal □V (Rev. 4)
- Trajectory event description:
  - Launch from KSC pad 39A
  - Lift-off at 100% power level
  - Start pitch maneuver after 350 ft vertical rise (roll maneuver not modeled)
  - o Ramp to zero alpha gravity turn after optimized pitch-over
  - Jettison SRB at 128.04 sec (thrust = 15,000 lb<sub>f</sub>)
  - Start SSME (start-up transient ends when 100% is reached)
  - Jettison interstage when SSME thrust = 100%
  - 100% thrust level reached 10 seconds after SRB separation.

- Ramp SSME to 104.5%
- Start optimized pitch profile 2 sec after engine reaches 104.5%
- Jettison LES at 300,000 ft
- Inject into 30 x 160 nm orbit at 28.5° inclination (due east mission) or 51.6° inclination (ISS mission).
- MECO occurs at 57 nm (target geocentric radius, inertial velocity, inertial flight path angle based on 30 x 160 nm orbit).
- Payload is responsible for transferring itself to the final orbit.

Since some of the trade studies presented in this document rely on the Rev. 1 trajectory, the major differences between the two ground rule sets are shown below.

#### Rev. 1 trajectory ground rules

- No RSRM knockdown
- No SSME start-up or shut-down transients
- Assume 100% instantaneous start on SSME
- Vehicle mass properties: 15% weight margins added to INTROS structures
- Vehicle components: separate adapters were not used.
- Difference in the trajectory event description:
  - Interstage jettisoned 0.1 seconds after 1<sup>st</sup> stage separation 10 second coast prior to 2<sup>nd</sup> stage ignition

  - Optimized pitch profile begins 2 seconds after 2<sup>nd</sup> stage ignition

# 5.3.2 TRAJECTORY INPUT DATA

This section contains the data necessary for the ascent trajectory simulation. Although some of this data may be contained in other sections of the main document, it is included here for completeness. The following subsections detail the following information.

Aerodynamic Data

SSME Block II Data

Solid Motor Data

**RSRM Performance Data** 

Five-Segment Booster Data

Seasonal RSRM Data

Wind Profiles

KSC Mean Annual Winds

Mean GRAM Wind Profiles

# 5.3.2.1 AERODYNAMIC DATA

The aerodynamics was determined for the Crewed Launch Vehicle -4 (CLV-4) and provided by Joe Lowery / MSFC/EV33. The 3-degree-of-freedom aerodynamic coefficients are shown in Table 5.3.2-1. Further details concerning the derivation of these coefficients may be found in Section 5.1 of this report. The configuration includes an inline SRM with the second stage employing a single Space Shuttle Main Engine (SSME). The upper stage diameter is 5.5 meters. Although a different vehicle configuration is shown, Figure 5.3.2-1 depicts the aerodynamic coordinate system definition used in this study.

For this vehicle, the reference area (A<sub>REF</sub>) is 116.2 ft<sup>2</sup> and the reference length (L<sub>REF</sub>) is 12.16 ft.

The coefficients in Table 5.3.2-1 use the following definitions.

<u>Variable</u>	<u>Definition</u>
М	Mach Number
CA	Axial force coefficient at zero degree alpha
CNA	Normal Force Coefficient per degree alpha
CMA	Pitching Moment Coefficient per degree alpha
CYB	Side Force Coefficient per degree beta
СРр	Center of Pressure in the pitch plane, measured from the base RSRB flare (ft)
СРу	Center of Pressure in the yaw plane, measured from the base RSRB flare (ft)

Table 5.3.2-1: Aerodynamic Coefficients for CLV-4 Vehicle

Aerodynamic Characteristics for the CLV-4 Vehicle							
Center of Pressure data measured from the base RSRB flare (ft)							
М	CA	CNA	СМА	СҮВ	СРр	СРу	
0.00	0.734	0.120	2.20	-0.120	223.0	223.0	

0.60	0.743	0.121	2.20	-0.121	221.8	221.8
0.80	0.899	0.128	2.25	-0.128	214.0	214.0
0.90	1.124	0.133	2.23	-0.133	204.0	204.0
0.95	1.280	0.135	2.28	-0.135	204.1	204.1
1.07	1.670	0.140	2.42	-0.140	208.8	208.8
1.23	1.930	0.138	2.42	-0.138	213.0	213.0
1.46	1.897	0.139	2.44	-0.139	217.7	217.7
1.96	1.600	0.140	2.69	-0.140	226.0	226.0
2.74	1.420	0.144	2.77	-0.144	234.0	234.0
3.48	1.255	0.148	2.81	-0.148	231.0	231.0
4.96	1.060	0.146	2.61	-0.146	217.0	217.0
30.00	1.060	0.146	2.61	-0.146	217.0	217.0

The recirculation of gases around the base area contributes to the aerodynamics. Table 5.3.2-2 provides the Base Force values as a function of the altitude.

Table 5.3.2-2: CLV-4 Base Force as a function of altitude

Altitude	Base Force	Altitude	Base Force
(ft)	(Ib <sub>f</sub> )	(ft)	(lb <sub>f</sub> )
0	0	38000	805
519	0	40000	1979
3200	-5598	42000	2937
5000	-7302	44000	3816
7500	-8985	46000	4583
10000	-10209	48000	5085
12500	-10718	50000	5467
15000	-10463	52500	5777
17500	-9689	53100	6095.2381
19000	-9085	56300	6371.42857
20000	-8705	60000	6533.33333
21000	-8472	65000	6506
22000	-8363	70000	6290
22500	-8424	75000	5858
23000	-8519	80000	4874
23500	-8634	85000	4425
24000	-8787	90000	3892
25000	-9078	95000	3394
26000	-9211	100000	3032

27000	-9119
28000	-8647
29000	-7784
30000	-6698
32000	-4327
34000	-2187
36000	-567

110000	2400
120000	2124
130000	1892
140000	1749
145000	1679
160000	1472
300000	0
<u>'</u>	

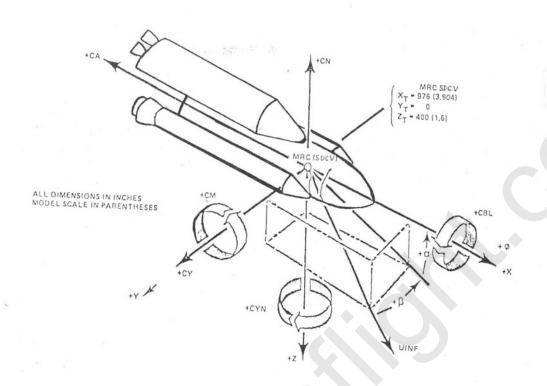


Figure 5.3.2-1: Aerodynamic Coordinate System definition (representative drawing)

# 5.3.2.2 2<sup>nd</sup> STAGE PROPELLANT INVENTORY DATA

The initial sizing was performed by Barney Holt / MSFC/NP11 using the sizing program INTROS. During this study, the second stage usable propellant value was assumed to be constant. The second stage jettison weight included the fuel bias and other unusable propellants. Table 5.3.2-3 shows the propellant breakdown. This study does not include propellants lost through mechanisms such as boil-off.

Table 5.3.2-3: CLV-4 2<sup>nd</sup> Stage Propellant Inventory

CLV-4 2 <sup>nd</sup> Stage Propellant Inventory							
	28.5 deg Inc.	51.6 deg Inc.					
Ascent Propellant (lb)	360,525	360,571					
Usable FPR (lb)	2,123	2,077					
Fuel Bias (lb)	534	534					
Residuals (unusable) (lb)	3,224	3,224					
Total (lb)	366,406	366,406					

## 5.3.2.3 SSME BLOCK II DATA

Two versions of the SSME will be considered: the Block II motor and an expendable motor. In this study, the Expendable SSME will use the same engine parameters as the current Block II motor.

The engine parameters shown in Table 5.3.2-4 were obtained from Rocketdyne's document IL-2002-478-007, entitled *SSME Block II Nominal Power Balance and Normal Operating Maximums and Minimums* and dated 3/25/2002. This paper was authored by J.A. Wendlandt / Rocketdyne, Canoga Park, CA, D938/478,055-AC58. In this document, the vacuum and sea level values were given. The exit area,  $A_E$ , was calculated by the equation  $A_E = (F_V - F_{SL}) / P_o$ . The atmospheric pressure at sea level,  $P_o$ , is 2124.07943 pounds per square foot (psf). This corresponds to the value used in POST for 1963 Patrick AFB atmosphere.

Table 5.3.2-4: SSME Block II Characteristics (*Mixture Ratio* = 6.032)

	SSME Block II Characteristics Mixture Ratio = 6.032							
Throttle	Vacuum Thrust	Vacuum Isp	Sea Level Thrust	Sea Level Isp	Exit Area			
Setting	(Ib <sub>f</sub> )	(sec)	(Ib <sub>f</sub> )	(sec)	(ft²)			
0.65	303,962	451.14	209,774	311.35	44.3430			
0.67	313,377	451.20	219,185	315.58	44.3449			
0.70	327,511	451.29	233,312	321.49	44.3482			
0.72	336,941	451.36	242,737	325.17	44.3505			
0.80	374,706	451.62	280,484	338.06	44.3590			
0.90	422,016	451.89	327,771	350.97	44.3698			
1.00	469,449	452.06	375,181	361.28	44.3806			
1.04	488,468	452.18	394,191	364.90	44.3849			
1.045	490,847	452.19	396,569	365.34	44.3853			

1.06	497,986	452.24	403,704	366.62	44.3872
1.09	512,271	452.33	417,982	369.07	44.3905
1.11	521,801	452.39	427,507	370.64	44.3929
1.13	531,335	452.45	437,037	372.15	44.3948
1.15	540,875	452.51	466,572	373.62	44.3971

Engines do not start nor shut down instantaneously. The transients, based upon the data within Shuttle Performance Assessment Databook (SPAD), were modeled in Revision 2 of the trajectory. The start-up and shut-down transients are shown in Tables 5.3.2-5 and 5.3.2-6, respectively.

Table 5.3.2-5: SSME Start-Up Transient

# SSME Start to 100% Block II Pc **Based on Shuttle Performance Assessment Databook** (SPAD)

NSTS 08209, Vol. I, Section 5 (Table 5.2.2)						
Time	Vacuum Thrust	Calculated Isp				
(sec)	(Ib <sub>f</sub> )	(sec)				
0.0000	0.0	0.00				
0.0625	28.2	5.31				
0.5000	42.3	5.26				
0.6250	75.1	7.66				
0.8125	70.4	6.47				
1.0000	1107.9	41.75				
1.2187	1765.1	26.88				
1.3750	3342.5	26.85				
1.4687	4300.2	31.04				
1.5937	5304.8	31.50				
1.7187	8403.1	47.16				
1.8437	37696.8	174.68				
1.9375	46146.8	193.80				
2.0937	62248.9	226.83				
2.2187	69901.0	236.96				
2.4062	73891.3	244.29				
2.6250	108489.7	272.72				
2.6875	122432.3	285.58				
2.9375	159659.6	322.53				
3.2500	259511.4	384.33				
3.3750	293077.0	394.50				
3.5937	358612.1	428.95				
3.6562	385042.1	437.95				
3.7187	411331.2	446.66				
3.8125	430062.2	451.21				
4.0000	448840.2	454.37				
4.2500	468838.7	457.85				
4.7500	474894.6	458.24				
5.0000	475457.9	457.96				
5.0625	475223.2	457.65				
5.0937	475129.3	457.67				
5.1250	475035.4	457.53				
5.1562	474894.6	457.33				
5.1875	474800.7	457.28				
5.2500	474612.9	457.01				
5.2812	474472.1	456.66				
6.3600	470857.3	453.27				

6.4800	470434.8	452.88
6.6000	470059.3	452.53
8.0000	469449.0	452.05

Table 5.3.2-6: SSME Shut-Down Transient

# SSME Shutdown from 104.5% Block II Pc **Based on Shuttle Performance Assessment Databook** (SPAD)

NSTS 08209,	NSTS 08209, Vol. I, Section 5 (Table 5.4.4, 104.5%)				
Time	Vacuum Thrust	Calculated Isp			
(sec)	(lb <sub>f</sub> )	(sec)			
0.0000	490847.0	452.19			
0.0800	486036.7	451.42			
0.2000	460463.6	451.38			
0.4800	325431.6	445.45			
0.6400	245767.1	437.42			
0.8400	179257.3	430.35			
1.0000	151671.7	426.26			
1.2400	124037.0	420.26			
1.3200	115152.7	417.81			
1.4000	105286.7	415.80			
1.4800	93653.6	411.77			
1.6400	62190.3	398. <i>4</i> 9			
1.7600	46041.4	388.89			
1.9200	32984.9	352.99			
1.9600	30088.9	327.64			
2.0800	25622.2	303.55			
2.1200	24885.9	293.44			
2.2000	21499.1	252.24			
2.4000	22039.0	259.66			
2.4800	23511.6	281.98			
2.5600	22922.6	294.81			
2.6400	22726.2	310.83			
2.8000	21891.8	339.41			
3.0000	21400.9	<i>364.15</i>			
3.2800	18799.4	377.91			
3.5200	15019.9	368.58			
4.4000	8540.7	391.64			
4.5200	8344.4	411.71			
4.7600	7313.6	430.53			
4.8800	6381.0	468.16			
5.0000	5497.5	486.98			

5.0800	5252.1	505.22
5.2800	4123.1	523.91
5.3200	3926.8	1169.58
5.4800	3485.0	0.00
5.5600	3583.2	0.00
6.0000	1914.3	0.00
7.2000	883.5	0.00
7.2400	0.0	0.00
8.0000	0.0	0.00

### 5.3.2.4 RSRM PERFORMANCE DATA

Table 5.3.2-7 represents the RSRM Block Model data that has been documented by ATK Thiokol in TPR07499, dated 3/15/1999. The file, TPR07499.368, represents the nominal RSRM performance at 60 deg F PMBT & 0.368 ips burn rate. Within this document, the SRB ignition weight is found to be 1,298,466.5 lb and the jettison weight is 186,859.5 lb.

For the purposes of this study, an additional margin of 5% was added to the structural weight of the booster. The SRB ignition weight becomes 1,305,307.7 lb. At separation, the expended weight is 1,111,915.7 lb and the jettison weight is 193,392. Since the SRB was not designed to be an inline component of the launch vehicle, a booster to interstage adapter is needed. This structural component also houses the booster's recovery system. The weight of this adapter is 18,689 lb.

From a telephone conference with Mark Tobias (ATK Thiokol) on 3/29/2004, the SRB separation occurs 4.42 sec after chamber pressure reaches 50 psi. The nominal SRB separation time occurs at 125.42 seconds. For this study, the separation time has been extended. Rather than the chamber pressure, the criterion for separation is the thrust level. Separation is assumed to occur when the thrust reaches  $15,000 \; \text{lb}_{\text{f}}$ . The jettison weight has been adjusted to reflect the additional expended weight.

The RSRM flight performance predictions have been modified to improve the Shuttle's ascent flight performance estimates. These modifications, known as knockdown, are based on trajectory reconstruction results from both analysis and flight data. To modify the SRB characteristics, two steps are described in the Shuttle Performance Assessment Databook (SPAD):

- 1. Reduce the SRB thrust versus time profile for all time points by 0.989797%.
- 2. Adjust the SRB thrust by the increments specified in Table 5.3.2-8 (SPAD Table 5.20).

Table 5.3.2-7: Lightweight SRB Characteristics

Lightweight SRB Characteristics Nominal Performance (3/15/1999) PMBT = 60 deg. F and Burn Rate = 0.368 ips							
Time	Time Head-End Vacuum Mass Flow Exit Are						
(sec)	(psia)	$(lb_f)$	(lb/sec)	(in²)			
0.000	14.690	258363.0	0.000	17587.68			
0.050	36.078	341724.6	595.361	17587.71			
0.100	53.622	361763.7	915.067	17587.74			
0.150	184.185	694385.8	2597.558	17587.77			
0.200	479.630	1512482.5	6991.335	17587.80			
0.231	563.500	1987890.5	7719.496	17587.81			
0.250	632.175	2202673.7	7980.343	17587.82			

0.300	738.003	2565673.4	9387.404	17587.85
0.350	809.874	2799192.5	10340.293	17587.88
0.400	852.711	2956528.8	10914.107	17587.91
0.450	879.049	3040796.8	11270.434	17587.94
0.500	895.073	3094196.2	11491.270	17587.97
0.550	903.179	3124451.5	11610.213	17588.00
0.600	906.383	3135198.7	11663.259	17588.03
0.612	906.832	3137421.4	11666.859	17588.04
0.650	906.035	3136828.4	11669.759	17588.06
0.700	905.773	3139106.3	11677.658	17588.09
0.750	904.787	3138465.3	11675.209	17588.12
0.800	904.156	3139373.4	11677.258	17588.15
0.850	903.025	3138592.4	11674.459	17588.18
0.900	902.768	3140299.5	11680.458	17588.21
0.950	902.178	3141548.8	11682.858	17588.24
1.000	901.612	3142301.7	11677.458	17588.27
1.500	896.127	3140190.9	11669.559	17588.56
2.000	891.346	3138625.0	11663.759	17588.86
2.500	887.648	3140423.9	11670.359	17589.15
3.000	885.060	3145087.1	11687.658	17589.45
3.500	883.408	3152524.7	11715.256	17589.74
4.000	882.679	3162998.7	11754.054	17599.04
4.500	882.929	3176613.7	11804.450	17590.33
5.000 5.000	883.983	3170013.7	11864.447	17590.33
5.500 5.500	885.443	3192623.3 3210157.4	11928.542	17590.63 17590.92
6.000	886.845	3226319.9	11988.539	17590.92
6.500	887.813	3238670.7	12034.536	17591.51
7.000	888.207	3247308.7	12034.536	17591.81
7.500	888.162	3254121.5	12000.934	17591.61
8.000		3260390.5		17592.10 17592.40
8.500	887.953 887.805		12116.131	
		3266698.6	12139.929	17592.69
9.000	887.740	3272961.0	12163.528	17592.99
9.500	887.585	3278636.8	12184.926	17593.28
10.000	887.128	3283089.9	12201.825	17593.58
10.500	886.272	3285962.6	12212.924	17593.87
11.000	885.116	3287473.6	12218.924	17594.17
11.500	883.851	3288289.3	12222.324	17594.46
12.000	882.620	3289069.4	12225.624	17594.76
12.500	881.498	3290168.9	12230.123	17595.05
13.000	880.507	3291665.1	12236.023	17595.35
13.500	879.632	3293466.6	12243.123	17595.64
14.000	878.846	3295425.0	12250.722	17595.94
14.500	878.124	3297443.9	12258.622	17596.23
15.000	877.460	3299545.6	12266.821	17596.53
15.500	876.870	3301835.2	12275.820	17596.82
16.000	876.371	3304423.7	12285.820	17597.12
16.500	875.975	3307351.4	12297.219	17597.41
17.000	875.668	3310524.1	12309.518	17597.71
17.500	875.404	3313727.6	12321.918	17598.00
18.000	875.115	3316704.8	12333.617	17598.30
18.500	874.737	3319238.4	12343.716	17598.59
19.000	874.235	3321234.9	12351.816	17598.89
19.500	873.628	3322806.7	12358.415	17599.18
20.000	872.991	3324275.7	12364.615	17599.48
20.500	872.357	3325743.5	12370.814	17599.78
21.000	871.526	3326310.9	12373.714	17600.07
21.500	869.868	3323293.7	12363.415	17600.37
22.000	866.118	3312237.8	12323.617	17600.66
22.500	859.100	3289339.7	12240.523	17600.96
23.000	850.502	3258824.2	12129.630	17601.25
23.500	841.560	3227905.6	12017.137	17601.55
24.000	833.077	3198191.0	11908.844	17601.84
24.500	825.111	3170185.4	11806.850	17602.14
25.000	817.613	3143999.8	11711.656	17602.43
25.500	810.542	3119489.7	11622.562	17602.73
26.000	803.860	3096414.0	11538.767	17603.02
26.500	797.522	3074541.7	11459.472	17603.32
27.000	791.479	3053672.7	11383.777	17603.61
27.500	785.680	3033636.0	11311.082	17603.91
28.000	780.081	3014286.6	11240.886	17604.20

28.500	774.646	2995504.4	11172.790	17604.50
29.000	769.344	2977190.9	11106.295	17604.79
29.500	764.154	2959267.2	11041.199	17605.09
30.000	759.059	2941672.3	10977.303	17605.38
30.500 31.000	754.047 749.112	2924359.9 2907296.3	10914.407 10852.411	17605.68 17605.97
31.500	744.249	2890458.9	10791.215	17606.27
32.000	739.456	2873832.9	10730.719	17606.56
32.500	734.733	2857409.5	10671.122	17606.86
33.000	730.080	2841183.4	10612.126	17607.15
33.500	725.496	2825151.2	10553.830	17607.45
34.000	720.981	2809309.7	10496.333	17607.74
34.500	716.531	2793655.4	10439.437	17608.04
35.000 35.500	712.141 707.804	2778184.1 2762889.8	10383.141 10327.544	17608.33 17608.63
36.000	707.504	2747765.0	10272.448	17608.92
36.500	699.266	2732798.7	10217.851	17609.22
37.000	695.053	2717977.0	10163.755	17609.52
37.500	690.873	2703281.8	10110.158	17609.81
38.000	686.724	2688691.4	10056.961	17610.11
38.500	682.608	2674187.3	10004.165	17610.40
39.000	678.532	2659765.2	9951.768	17610.70
39.500 40.000	674.506 670.545	2645447.0 2631292.8	9899.871 9848.575	17610.99 17611.29
40.500	666.671	2617406.1	9798.278	17611.58
41.000	662.907	2603916.2	9749.481	17611.88
41.500	659.277	2590947.7	9702.384	17612.17
42.000	655.798	2578589.2	9657.387	17612.47
42.500	652.483	2566869.1	9614.689	17612.76
43.000	649.328	2555759.9	9574.392	17613.06
43.500	646.319	2545197.3	9536.394	17613.35
44.000 44.500	643.429 640.641	2535100.1 2525390.9	9500.397 9466.099	17613.65 17613.94
44.500 45.000	637.961	2516015.8	9433.001	17613.94 17614.24
45.500	635.421	2506960.8	9400.903	17614.53
46.000	632.982	2498155.4	9369.105	17614.83
46.500	630.432	2489112.6	9336.007	17615.12
47.000	627.496	2478778.1	9298.709	17615.42
47.500	623.999	2466153.2	9253.912	17615.71
48.000	619.951	2451099.3	9200.216	17616.01
48.500	615.560	2434493.2	9140.220	17616.31
49.000 49.500	611.181 607.206	2417847.2 2402864.7	9079.623 9024.827	17616.60 17616.90
50.000	603.919	2390874.2	8981.130	17617.19
50.500	601.381	2382062.0	8950.032	17617.49
51.000	599.481	2375504.6	8928.033	17617.78
51.500	598.050	2370270.1	8910.334	17618.08
52.000	596.991	2366448.7	8896.935	17618.37
52.500	596.332	2364794.0	8891.235	17618.67
53.000	596.163	2365639.1	8895.135 8007.634	17618.96
53.500 54.000	596.544 597.430	2368741.7 2373535.1	8907.63 <i>4</i> 8926.233	17619.26 17619.55
54.500 54.500	598.670	2379390.5	8948.632	17619.85 17619.85
55.000	600.081	2385813.5	8972.930	17620.14
55.500	601.522	2392476.2	8998.129	17620.44
56.000	602.938	2399183.4	9023.427	17620.73
56.500	604.321	2405838.8	9048.525	17621.03
57.000	605.673	2412409.4	9073.224	17621.33
57.500	606.997 608.296	2418891.6 2425284.0	9097.722	17621.62 17621.02
58.000 58.500	608.296	2425284.0 2431581.7	9121.721 9145.419	17621.92 17622.21
59.000	610.838	2437776.6	9145.419 9168.618	17622.21 17622.51
59.500	612.084	2443859.6	9191.316	17622.80
60.000	613.312	2449821.5	9213.515	17623.10
60.500	614.517	2455655.0	9235.214	17623.39
61.000	615.692	2461355.9	9256.312	17623.69
61.500	616.833	2466924.5	9277.011	17623.98
62.000	617.933	2472367.2	9297.210	17624.28 17624.57
62.500 63.000	618.994 620.021	2477697.9 2482939.6	9316.908 9336.507	17624.57 17624.87
03.000	020.021	2 <del>4</del> 02939.0	9330.307	1/024.0/

64.000       622.023       2493295.8       9375.205       17         64.500       623.035       2498491.5       9394.703       17         65.000       624.077       2503746.0       9414.402       17         65.500       625.154       2509077.8       9434.501       17         66.000       626.260       2514482.4       9454.700       17         66.500       627.371       2519924.6       9474.898       17         67.000       628.454       2525338.3       9495.097       17         67.500       629.491       2530664.5       9515.096       17	7625.16 7625.46 7625.75 7626.05 7626.35 7626.64 7626.94
64.500     623.035     2498491.5     9394.703     17       65.000     624.077     2503746.0     9414.402     17       65.500     625.154     2509077.8     9434.501     17       66.000     626.260     2514482.4     9454.700     17       66.500     627.371     2519924.6     9474.898     17       67.000     628.454     2525338.3     9495.097     17       67.500     629.491     2530664.5     9515.096     17	7625.75 7626.05 7626.35 7626.64 7626.94
65.000     624.077     2503746.0     9414.402     17       65.500     625.154     2509077.8     9434.501     17       66.000     626.260     2514482.4     9454.700     17       66.500     627.371     2519924.6     9474.898     17       67.000     628.454     2525338.3     9495.097     17       67.500     629.491     2530664.5     9515.096     17	7626.05 7626.35 7626.64 7626.94
65.500     625.154     2509077.8     9434.501     17       66.000     626.260     2514482.4     9454.700     17       66.500     627.371     2519924.6     9474.898     17       67.000     628.454     2525338.3     9495.097     17       67.500     629.491     2530664.5     9515.096     17	7626.35 7626.64 7626.94
66.000     626.260     2514482.4     9454.700     17       66.500     627.371     2519924.6     9474.898     17       67.000     628.454     2525338.3     9495.097     17       67.500     629.491     2530664.5     9515.096     17	7626.64 7626.94
66.500     627.371     2519924.6     9474.898     17       67.000     628.454     2525338.3     9495.097     17       67.500     629.491     2530664.5     9515.096     17	7626.94
67.000 628.454 2525338.3 9495.097 17 67.500 629.491 2530664.5 9515.096 17	
67.500 629.491 2530664.5 9515.096 17	
	7627.23
	7627.53
	7627.82
	7628.12
	7628.41 7628.71
	7629.00
	7629.00 7629.30
	7629.50 7629.59
	7629.89
	7630.19
	7630.48
	7630.78
	7631.07
	7631.37
	7631.66
75.000 637.992 2580614.6 9704.284 17	7631.96
75.500 638.034 2581772.4 9708.483 17	7632.25
76.000 637.986 2582675.8 9711.383 17	7632.55
76.500 637.853 2583271.3 9713.383 17	7632.84
	7633.14
	7633.43
	7633.73
	7634.03
	7634.32
	7634.62
	7634.91
	7635.21
	7635.50 7635.80
	7635.60 7636.09
	7636.39
	7636.68
	7636.98
	7637.27
	7637.57
	7637.87
	7638.16
86.000 583.410 2376476.0 8951.732 17	7638.46
	7638.75
87.000 574.904 2343437.9 8827.339 17	7639.05
	7639.34
	7639.64
	7639.93
	7640.23
	7640.52
	7640.82
	7641.12
	7641.41
	7641.71 7642.00
	7642.00 7642.30
	7642.30 7642.59
	7642.59 7642.89
■ 30.000   001.000   21/1100.0   0100.000   1/	7642.69 7643.18
94.000 528.096 2159427.4 8144.183 17	/D4 < 4×
94.000     528.096     2159427.4     8144.183     17       94.500     524.733     2146392.2     8095.986     17	7643.48 7643.77
94.000     528.096     2159427.4     8144.183     17       94.500     524.733     2146392.2     8095.986     17       95.000     520.981     2131863.6     8042.089     17	7643.77
94.000     528.096     2159427.4     8144.183     17       94.500     524.733     2146392.2     8095.986     17       95.000     520.981     2131863.6     8042.089     17       95.500     516.971     2116030.9     7983.693     17	7643.77 7644.07
94.000     528.096     2159427.4     8144.183     17       94.500     524.733     2146392.2     8095.986     17       95.000     520.981     2131863.6     8042.089     17       95.500     516.971     2116030.9     7983.693     17       96.000     512.903     2099677.1     7923.597     17	7643.77 7644.07 7644.37
94.000     528.096     2159427.4     8144.183     17       94.500     524.733     2146392.2     8095.986     17       95.000     520.981     2131863.6     8042.089     17       95.500     516.971     2116030.9     7983.693     17       96.000     512.903     2099677.1     7923.597     17       96.500     508.976     2083838.0     7865.201     17	7643.77 7644.07 7644.37 7644.66
94.000     528.096     2159427.4     8144.183     17       94.500     524.733     2146392.2     8095.986     17       95.000     520.981     2131863.6     8042.089     17       95.500     516.971     2116030.9     7983.693     17       96.000     512.903     2099677.1     7923.597     17       96.500     508.976     2083838.0     7865.201     17       97.000     505.310     2069227.8     7811.104     17	7643.77 7644.07 7644.37

98.500	495.600	2031183.7	7670.813	17645.84
99.000	492.689	2019687.2	7627.716	17646.14
99.500	489.900	2008496.5	7586.918	17646.43
100.000	487.062	1997019.0	7546.021	17646.73
100.500	483.946	1984560.1	7500.524	17647.03
101.000	480.391	1970582.6	7448.127	17647.32
101.500	476.411	1954909.2	7389.531	17647.62
102.000	472.163	1937714.4	7326.135	17647.91
102.500	467.742	1919395.3	7249.540	17648.21
103.000 103.500	463.126 458.318	1900434.8 1881263.2	7162.745 7074.551	17648.50 17648.80
103.500	453.475	1862123.0	6985.856	17649.09
104.500	448.773	1842984.3	6896.862	17649.09 17649.39
105.000	444.118	1823659.3	6807.668	17649.69
105.500	439.286	1804030.3	6717.973	17649.98
106.000	434.357	1784354.0	6628.179	17650.28
106.500	429.753	1765709.7	6541.785	17650.57
107.000	425.923	1749973.0	6477.389	17650.87
107.500	423.076	1738440.6	6434.491	17651.16
108.000	421.070	1730509.0	6405.593	17651.46
108.500	419.393	1723840.2	6382.195	17651.75
109.000	417.226	1715108.1	6350.397	17652.05
109.500	413.524	1700247.5	6293.800	17652.34
110.000	407.154	1673941.5	6196.706	17652.64
110.500	397.746	1634586.6	6053.416	17652.94
111.000	386.197	1586122.2	5874.627	17653.23
111.500	372.403	1528509.9	5668.040	17653.53
112.000	356.013	1460295.1 1372781.8	5420.656	17653.82
112.500 113.000	334.891 309.188	1266421.3	5100.176 4707.801	17654.12 17654.41
113.500	279.367	1143337.8	4251.130	17654.71
114.000	249.284	1019896.6	3789.959	17655.01
114.500	222.897	912874.8	3387.085	17655.30
115.000	201.019	823344.8	3054.006	17655.60
115.200	193.896	793186.3	2941.013	17655.71
115.400	187.052	764927.5	2838.420	17655.83
115.600	180.494	738259.9	2739.926	17655.95
115.800	174.242	712879.1	2644.432	17656.07
116.000	168.257	688472.1	2552.638	17656.19
116.200	162.468	664755.0	2464.443	17656.31
116.400	156.808	641509.2	2378.549	17656.42
116.600	151.233	618605.9	2294.154	17656.54
116.800	145.722	595999.1	2210.760	17656.66
117.000	140.278	573705.1	2128.265 2047.270	17656.78
117.200	134.914	551783.1	1967.675	17656.90
117.400 117.600	129.655 124.524	530314.3 509381.6	1890.080	17657.02 17657.13
117.800	119.542	489049.4	1814.685	17657.15 17657.25
118.000	114.719	469343.4	1741.789	17657.37
118.200	110.048	450230.6	1671.294	17657.49
118.400	105.504	431609.7	1602.798	17657.61
118.600	101.047	413330.0	1535.502	17657.72
118.800	96.631	395219.2	1468.907	17657.84
119.000	92.211	377113.0	1402.011	17657.96
119.200	87.754	358884.9	1334.515	17658.08
119.400	83.245	340474.3	1266.220	17658.20
119.600	78.694	321915.9	1197.324	17658.32
119.800	74.142	303352.6	1128.428	17658.43
120.000	69.651	285006.5	1060.433	17658.55
120.200	65.286	267136.2	994.237	17658.67
120.400	61.108 57.156	249990.8	931.041	17658.79
120.600	57.156 52.451	233766.7	871.345 815.548	17658.91 17650.03
120.800 121.000	53.451 50.000	218575.8 204454.7	815.548 763.966	17659.03 17659.14
121.000	46.781	204454.7 191291.0	763.966	17659.14 17659.26
121.400	43.791	179063.8	670.957	17659.26 17659.38
121.600	41.010	167693.2	629.760	17659.50
121.800	38.417	157092.1	591.462	17659.62
122.000	35.993	147180.3	555.865	17659.73
122.200	33.721	137888.7	522.767	17659.85

122.400	31.586	129159.7	491.869	17659.97
122.600	29.578	120945.6	462.971	17660.09
122.800	27.685	113208.5	435.972	17660.21
123.000	25.903	105917.9	410.774	17660.33
123.200	24.223	99049.8	387.275	17660.44
123.400	22.642	92585.3	365.277	17660.56
123.472	22.100	90370.0	357.524	17660.60
123.600	21.156	86508.5	343.178	17660.62
123.800	19.761	80805.6	321.780	17660.62
124.000	18.455	75463.2	301.681	17660.62
124.200	17.233	70467.3	282.982	17660.62
124.400	16.092	65801.7	265.583	17660.62
124.600	15.027	61447.1	249.384	17660.62
124.800	14.032	57379.6	234.185	17660.62
125.000	13.101	53569.7	219.886	17660.62
125.200	12.223	49981.1	206.287	17660.62
125.400	11.389	46569.3	193.188	17660.62
125.600	10.586	43287.6	180.389	17660.62
125.800	9.809	40108.3	167.689	17660.62
126.000	9.061	37051.1	155.190	17660.62
126.080	8.776	35885.7	150.230	17660.62
126.200	8.367	34211.5	142.991	17660.62
126.400	7.773	31783.3	131.892	17660.62
126.600	7.321	29936.8	122.592	17660.62
126.800	6.927	28324.8	113.793	17660.62
127.000	6.300	25762.2	100.894	17660.62
127.200	5.824	23813.6	94.594	17660.62
127.400	4.940	20200.5	83.395	17660.62
127.600	4.434	18130.4	77.595	17660.62
127.800	3.732	15258.6	67.896	17660.62
128.000	3.465	14170.1	63.696	17660.62
128.200	3.078	12588.0	55.696	17660.62
128.400	2.766	11312.1	46.697	17660.62
128.600	2.548	10418.4	43.597	17660.62
128.800	2.388	9764.5	47.897	17660.62
129.000	2.260	9240.6	51.897	17660.62

Table 5.3.2-8: SRB Performance Knockdown

% Separation Time	% Action Time	Thrust Increment (lb <sub>f</sub> )
0	0	0
1	1.017349	10,000
24	24.416370	10,000
37	37.641900	-14,000
39	39.676600	-14,000
43	43.745990	2,000
48	48.832730	2,000
53	53.919480	-7,000
67	68.162360	2,500
69	70.197050	2,500
77	78.335840	-5,000
81	82.405240	-14,000
87	88.509330	-14,000
89	90.544030	-2,000
90	91.561370	-2,000
94	95.630770	2,000
95	96.648120	2,000
96	97.665470	0
100	101.734900	0

# 5.3.2.5 FIVE-SEGMENT BOOSTER (FSB) PERFORMANCE DATA

Table 5.3.2-9 represents the FSB model data that has been documented by ATK Thiokol in FSB\_HT\_266.2\_ADJMASS.txt and FSB\_HT\_3b.doc.These documents were obtained from Mark Tobias (ATK Thiokol).This data represents the nominal FSB performance at 60 deg F PMBT & 0.340 ips burn rate. Within this document, the FSB ignition weight is found to be 1,656,139.7 lb and the jettison weight is 221,233.6 lb. The average exit area is 126.926 square feet.

From a telephone conference with Mark Tobias (ATK Thiokol) on 3/29/2004, the FSB separation occurs 4.42 sec after chamber pressure reaches 50 psi. For the motor used in this study, the FSB separation time occurs at 132.52 seconds.

Table 5.3.2-9: FSB Characteristics

	Five Segment Booster (FSB) Characteristics PMBT = 60 deg. F and Burn Rate = 0.340 ips						
Time	Head-End Pressure	Vacuum Thrust	Propellant Mass Flow Rate	Inert Mass Flow Rate	Total Mass Flow Rate		
(sec)	(psia)	$(lb_f)$	(lb/sec)	(lb/sec)	(lb/sec)		
0.000	14.690	252462.8	0.000	0.000	0.000		
0.050	33.997	334281.9	0.000	272.944	272.944		
0.100	51.706	351212.0	245.638	226.731	472.369		
0.150	185.646	687541.7	1755.360	180.518	1935.878		
0.200	479.288	1482365.7	5283.605	134.305	5417.910		
0.250	632.311	2155050.2	7836.589	88.092	7924.681		
0.300	738.027	2508952.2	9262.688	72.922	9335.610		
0.350	819.583	2851272.1	10636.299	71.057	10707.356		
0.400	873.078	3132632.2	11676.434	69.192	11745.626		
0.450	910.472	3346521.6	12521.091	67.327	12588.418		
0.500	937.636	3531713.2	13236.585	65.461	13302.046		
0.550	956.834	3694427.0	13847.888	63.596	13911.484		
0.600	970.986	3835358.6	14386.759	61.731	14448.490		
0.650	970.573	3837261.5	14397.862	59.866	14457.728		
0.700	970.243	3840122.9	14410.736	58.000	14468.736		
0.750	969.200	3839243.1	14410.633	56.135	14466.768		
0.800	968.481	3840226.6	14416.030	54.270	14470.300		
0.850	967.366	3839510.6	14415.398	52.405	14467.803		
0.900	966.987	3841575.4	14425.770	50.540	14476.310		
0.950	966.288	3842589.9	14431.447	48.674	14480.121		
1.000	965.780	3844132.6	14439.885	46.809	14486.694		
1.134	963.310	3842335.9	14440.573	41.810	14482.383		
1.927	954.260	3840117.6	14476.235	12.228	14488.463		
2.895	944.440	3840192.0	14491.320	2.709	14494.029		
3.870	937.510	3844911.5	14507.769	4.241	14512.010		
4.849	939.800	3883158.2	14650.838	5.692	14656.530		
5.828	942.200	3918436.9	14782.875	7.028	14789.903		
6.809	943.360	3946335.5	14886.873	8.527	14895.400		
7.791	942.370	3962289.4	14944.677	10.770	14955.447		
8.774	941.590	3978584.0	15003.677	13.016	15016.693		
9.759	937.570	3980579.5	15010.001	14.429	15024.430		
10.746	939.190	4003218.2	15094.419	15.778	15110.197		
11.733	934.790	4000484.6	15082.635	17.333	15099.968		
12.724	929.580	3993952.7	15056.401	18.989	15075.390		
13.715	928.280	4003515.7	15090.967	20.646	15111.613		

14.708	926.740	4010214.6	15114.727	22.307	15137.034
15.702	918.710	3989094.0	15033.284	23.969	15057.253
16.698	913.620	3979086.2	14993.939	25.634	15019.573
17.697	905.290	3954810.7	14900.694	27.298	14927.992
18.700	890.150	3900914.0	14697.660	27.457	14725.117
19.707	877.480	3856270.2	14529.626	27.617	14557.243
		3805461.8			
20.718	863.470		14332.767	31.468	14364.235
21.733	846.740	3742220.1	14087.660	36.237	14123.897
22.753	830.780	3681825.3	13857.466	38.139	13895.605
23.778	818.150	3635566.1	13683.225	38.303	13721.528
24.807	802.380	3575126.2	13454.317	39.193	13493.510
26.027	787.430	3518322.6	13237.522	41.232	13278.754
27.440	773.880	3468690.2	13047.552	43.594	13091.146
28.859	754.080	3391057.7	12751.567	45.966	12797.533
30.287	738.810	3332658.7	12528.358	48.355	12576.713
31.722	724.960	3279676.2	12325.533	50.754	12376.287
33.165	709.340	3218248.7	12090.789	53.166	12143.955
34.618	693.830	3156979.7	11856.520	55.595	11912.115
36.079	677.520	3091423.5	11605.931	58.038	11663.969
37.548	663.390	3035247.8	11390.824	60.493	11451.317
39.023	653.560	2997522.1	11245.500	62.960	11308.460
40.503	642.460	2957522.1 2953983.1	11245.500 11078.162	65.436	11143.598
41.988	630.010	2903974.1	10886.213	67.920	10954.133
43.481	615.490	2843571.0	10654.946	70.415	10725.361
44.984	602.970	2791789.0	10456.256	72.927	10529.183
46.494	594.240	2757124.6	10322.248	75.451	10397.699
48.009	589.000	2738374.9	10248.426	77.984	10326.410
49.528	582.770	2714924.2	10158.924	79.143	10238.067
51.052	584.700	2728488.5	10210.927	79.277	10290.204
52.566	592.220	2767632.5	10359.731	79.410	10439.141
54.307	598.450	2801434.1	10488.506	79.563	10568.069
55.821	606.570	2843325.2	10647.811	79.696	10727.507
57.327	612.760	2876305.5	10773.416	79.829	10853.245
58.822	620.710	2917275.6	10929.223	79.960	11009.183
60.306	629.800	2963418.6	11104.731	80.091	11184.822
61.781	635.330	2993213.6	11218.240	80.221	11298.461
63.252	640.250	3020223.0	11321.350	80.350	11401.700
64.717	645.070	3046552.1	11421.761	80.479	11502.240
66.180	652.050	3083003.5	11560.572	80.608	11641.180
67.636	661.130	3129063.2	11735.883	80.736	11816.619
69.087	666.580	3158312.0	11847.396	80.864	11928.260
70.536	669.440	3175477.3	11913.209	80.991	11994.200
71.982	675.400	3207089.4	12033.822	81.118	12114.940
73.425	684.220	3251986.1	12204.735	81.245	12285.980
74.864	689.940	3282309.6	12320.449	81.372	12401.821
76.302	694.750	3308553.2	12420.663	81.498	12502.161
77.740	696.940	3322602.5	12474.777	81.625	12556.402
79.178	699.400	3337795.9	12533.191	81.751	12614.942
80.618	698.580	3337513.9	12532.905	81.878	12614.783
82.064	693.080	3315408.9	12449.918	82.005	12531.923
83.516	686.070	3286161.1	12339.831	82.133	12421.964
84.977	676.030	3242505.5	12175.042	82.262	12257.304
86.448	666.310	3200288.5	12015.752	82.391	12098.143
87.930	655.310	3151780.7	11832.560	82.521	11915.081
89.424	643.210	3098102.1	11629.767	82.653	11712.420
90.933	629.160	3035035.1	11390.674	83.193	11473.867
92.456	615.440	2973242.4	11153.431	85.741	11239.172
93.995	600.300	2973242.4 2904552.5	10890.051	88.313	10978.364
95.551	583.780	2829013.2	10599.417	91.671	10691.088
95.557 97.126	568.780	2760620.6	10399.417	96.938	10430.018
98.717	557.210	2708428.1	10133.247	98.730	10231.977
100.320	553.710	2694649.2	10080.857	99.403	10180.260
101.930	547.890	2669636.6	9986.757	99.403	10086.160
103.553	538.040	2625304.8	9819.357	99.403	9918.760
105.191	525.600	2568363.0	9604.157	99.403	9703.560
106.844	514.340	2516987.2	9409.957	99.403	9509.360
108.514	504.160	2470666.5	9234.857	99.403	9334.260
110.198	494.880	2428528.4	9075.759	99.402	9175.161
111.899	481.180	2364898.7	8834.958	99.403	8934.361
113.036	470.160	2313360.6	8639.757	99.403	8739.160
<u> </u>					

113.594	465.330	2290891.3	8554.657	99.403	8654.060
114.156	454.770	2240638.3	8364.057	99.403	8463.460
114.721	449.790	2217156.6	8275.157	99.403	8374.560
115.288	448.830	2213150.3	8260.157	99.403	8359.560
115.857	442.600	2183799.6	8148.957	99.403	8248.360
116.430	434.730	2146404.5	8007.157	99.403	8106.560
117.006	425.490	2102211.9	7839.557	99.403	7938.960
117.587	418.210	2067473.6	7707.757	99.403	7807.160
118.173	409.030	2023513.3	7541.057	99.403	7640.460
118.764	396.680	1964022.7	7315.257	99.403	7414.660
119.365	375.700	1862337.0	6929.257	99.403	7028.660
119.981	345.260	1714053.0	6366.056	99.404	6465.460
120.619	307.650	1530139.4	5669.531	98.060	5767.591
121.288	263.500	1313285.0	4849.606	95.422	4945.028
122.000	219.290	1095212.2	4026.615	91.741	4118.356
122.763	178.670	894048.9	3269.532	86.896	3356.428
123.586	143.970	721615.0	2622.110	81.671	2703.781
123.566	143.970	556317.5	2022.110	75.026	2078.899
124.461	100.000	500317.5 502341.5	2003.873 1802.999	75.026 72.144	2076.699 1875.143
124.647	89.872	451466.3	1602.999 1615.448	68.207	1675.143 1683.655
125.847	80.770	405743.5	1447.477	64.270	1511.747
126.347	72.590	364651.4	1297.102	60.334	1357.436
126.847	65.239	327720.9	1162.540	56.399	1218.939
127.347	58.632	294530.6	1042.192	52.464	1094.656
127.847	52.694	264701.7	934.617	48.529	983.146
128.347	47.357	237893.7	838.524	44.594	883.118
128.847	42.561	213800.8	752.751	40.657	793.408
129.347	38.250	192147.8	676.250	36.719	712.969
129.847	34.377	172687.9	608.083	32.782	640.865
130.347	30.895	155198.7	547.406	28.845	576.251
130.847	27.766	139480.8	493.459	24.908	518.367
131.347	24.954	125354.7	445.561	20.971	466.532
131.847	22.427	112659.2	403.099	17.034	420.133
132.347	20.156	101249.5	365.524	13.098	378.622
132.847	18.114	90995.4	340.690	3.478	344.168
133.347	16.280	81779.7	308.361	1.645	310.006
133.847	14.631	73497.4	275.492	2.595	278.087
134.347	13.149	66053.9	245.810	3.544	249.354
134.847	11.817	59364.2	218.993	4.494	223.487
135.347	10.621	53352.0	194.751	5.443	200.194
135.847	9.545	47948.7	172.822	6.392	179.214
136.347	8.578	43092.6	152.974	7.342	160.316
136.847	7.710	38728.4	134.994	8.291	143.285
137.347	6.929	34806.1	118.694	9.241	127.935
137.847	6.227	31281.1	103.903	10.190	114.093
138.347	5.596	28113.1	90.469	11.139	101.608
138.847	5.030	25265.9	78.254	12.089	90.343
139.347	4.520	22707.0	67.135	13.038	80.173
139.847	4.062	20407.4	57.001	13.988	70.989
140.347	3.651	18340.6	47.752	14.937	62.689
140.847	3.281	16483.1	39.299	15.887	55.186
141.347	2.949	14813.8	31.560	16.836	48.396
141.847	2.650	13313.5	24.464	17.785	42.249
142.347	2.382	11965.1	17.945	18.735	36.680
142.847	2.141	10753.3	11.945	19.684	31.629

## 5.3.2.6 SEASONAL RSRM DATA

The February and July RSRM and mean-GRAM wind profile data was used to develop the baseline dispersion trajectories. The February and July RSRM data listed in Table 5.3.2-10 were provided by Dr. John Hanson (MSFC/EV40). Dr. Hanson derived the February and July data using the reference RSRM data (60 deg PMBT) and the following equations from the SPAD:

 $P_{\text{scaled}} = P^* \exp[0.0011^*(PMBT-60)]$ 

 $F_{\text{scaled}} = F^* \exp[0.0011^*(PMBT-60)]$ 

 $W_{scaled} = W*exp[0.001063*(PMBT-60)]$ 

 $T_{scaled} = T*exp[-0.001063*(PMBT-60)]$ 

Where: P is pressure, F is thrust, W is flowrate, and T is time

PMBT = 61 deg for February

PMBT = 80 deg for July

Table 5.3.2-10: February and July RSRM Data

F	ebruary l	RSRM (61 c	leg PMBT)		July RSRM (80 deg PMBT)					
Time	Pressure	Fvac	Flowrate	Exit Area	Time	Pressure	Fvac	Flowrate	Exit Area	
(sec)	(psi)	(lbs)	(lb/sec)	(ft²)	(sec)	(psi)	(lbs)	(lb/sec)	(ft²)	
0.0000	14.706	258647.4	0.000	122.1367	0.0000	15.017	264110.0	0.000	122.1367	
0.0499	36.118	342100.7	595.994	122.1369	0.0489	36.881	349325.8	608.154	122.1369	
0.0999	53.681	362161.9	916.040	122.1371	0.0979	54.815	369810.7	934.730	122.1371	
0.1498	184.388	695150.0	2600.321	122.1373	0.1468	188.282	709831.6	2653.373	122.1373	
0.1998	480.158	1514147.1	6998.771	122.1375	0.1958	490.299	1546125.8	7141.562	122.1375	
0.2308	564.120	1990078.4	7727.706	122.1376	0.2261	576.034	2032108.7	7885.369	122.1376	
0.2497	632.871	2205098.0	7988.831	122.1376	0.2447	646.237	2251669.5	8151.821	122.1376	
0.2997	738.815	2568497.2	9397.388	122.1378	0.2937	754.419	2622743.7	9589.117	122.1378	
0.3496	810.765	2802273.3	10351.291	122.1381	0.3426	827.889	2861457.1	10562.481	122.1381	
0.3996	853.649	2959782.8	10925.715	122.1383	0.3916	871.679	3022293.2	11148.625	122.1383	
0.4495	880.016	3044143.5	11282.421	122.1385	0.4405	898.602	3108435.6	11512.609	122.1385	
0.4995	896.058	3097601.7	11503.492	122.1387	0.4895	914.983	3163022.8	11738.190	122.1387	
0.5494	904.173	3127890.3	11622.561	122.1389	0.5384	923.269	3193951.1	11859.689	122.1389	
0.5994	907.381	3138649.3	11675.664	122.1391	0.5874	926.544	3204937.4	11913.874	122.1391	
0.6113	907.830	3140874.5	11679.267	122.1392	0.5991	927.003	3207209.5	11917.552	122.1392	
0.6493	907.032	3140280.8	11682.171	122.1393	0.6363	926.189	3206603.3	11920.514	122.1393	
0.6993	906.770	3142561.2	11690.078	122.1395	0.6853	925.921	3208931.9	11928.583	122.1395	
0.7492	905.783	3141919.5	11687.626	122.1397	0.7342	924.913	3208276.6	11926.081	122.1397	
0.7992	905.151	3142828.6	11689.678	122.1399	0.7832	924.268	3209204.9	11928.174	122.1399	
0.8491 0.8990	904.019 903.762	3142046.8 3143755.7	11686.876 11692.881	122.1401 122.1403	0.8321 0.8811	923.112	3208406.6 3210151.6	11925.315 11931.443	122.1401 122.1403	
0.8990	903.762	3143755.7	11692.881	122.1403	0.8811	922.849 922.246	3210151.6	11931.443	122.1403	
0.9490	903.171	3145000.4	11689.878	122.1408	0.9300	922.246	3212198.4	11933.693	122.1408	
1.4984	897.113	3143647.0	11681.970	122.1400	1.4684	916.060	3210040.6	11920.379	122.1400	
1.9979	892.327	3142079.4	11676.164	122.1449	1.9579	911.173	3208439.9	11914.385	122.1449	
2.4973	888.625	3143880.3	11682.771	122.1469	2.4474	907.393	3210278.8	11921.127	122.1469	
2.9968	886.034	3148548.6	11700.089	122.1490	2.9369	904.747	3215045.7	11938.798	122.1490	
3.4963	884.380	3155994.4	11727.716	122.1510	3.4264	903.058	3222648.8	11966.989	122.1510	
3.9958	883.650	3166479.9	11766.555	122.1531	3.9159	902.313	3233355.8	12006.620	122.1531	
4.4952	883.901	3180109.9	11817.005	122.1551	4.4053	902.569	3247273.6	12058.099	122.1551	
4.9947	884.956	3196337.3	11877.066	122.1572	4.8948	903.646	3263843.8	12119.386	122.1572	
5.4942	886.418	3213690.5	11941.229	122.1592	5.3843	905.139	3281563.4	12184.858	122.1592	
5.9936	887.821	3229870.8	12001.290	122.1613	5.8738	906.572	3298085.5	12246.144	122.1613	
6.4931	888.790	3242235.2	12047.336	122.1633	6.3633	907.561	3310711.0	12293.129	122.1633	
6.9926	889.185	3250882.7	12079.768	122.1653	6.8528	907.964	3319541.1	12326.223	122.1653	
7.4920	889.140	3257703.0	12105.393	122.1674	7.3422	907.918	3326505.5	12352.372	122.1674	
7.9915	888.930	3263978.9	12129.017	122.1694	7.8317	907.704	3332913.9	12376.478	122.1694	
8.4910	888.782	3270293.9	12152.841	122.1715	8.3212	907.553	3339362.3	12400.787	122.1715	
8.9904	888.717	3276563.2	12176.465	122.1735	8.8107	907.487	3345764.0	12424.893	122.1735	
9.4899	888.562 888.104		12197.885	122.1756	9.3002	907.328 906.861	3351566.1	12446.751	122.1756	
9.9894	887.247	3286703.3	12214.802 12225.913	122.1776	9.7896	905.986	3356118.2	12464.013 12475.350	122.1776	
10.4888 10.9883	886.090	3289579.1 3291091.8	12231.920	122.1797 122.1817	10.2791 10.7686	905.966	3359054.8 3360599.5	12475.350	122.1797 122.1817	
11.4878	884.824	3291908.4	12231.920	122.1817	11.2581	904.604	3361433.3	12484.952	122.1817	
11.9873	883.591	3292689.4	12238.627	122.1858	11.7476	902.253	3362230.8	12488.323	122.1858	
12.4867	882.468	3293790.1	12243.131	122.1878	12.2371	901.106	3363354.7	12492.919	122.1878	
12.9862	881.476	3295287.9	12249.037	122.1899	12.7265	900.093	3364884.2	12498.946	122.1899	
13.4857	880.600	3297091.4	12256.144	122.1919	13.2160	899.198	3366725.8	12506.198	122.1919	
13.9851	879.813	3299052.0	12263.751	122.1940	13.7055	898.395	3368727.7	12513.961	122.1940	
14.4846	879.090	3301073.1	12271.660	122.1960	14.1950	897.657	3370791.5	12522.030	122.1960	
14.9841	878.426	3303177.1	12279.868	122.1981	14.6845	896.978	3372940.0	12530.406	122.1981	
15.4835	877.835	3305469.2	12288.876	122.2001	15.1739	896.375	3375280.5	12539.598	122.2001	
15.9830	877.336	3308060.6	12298.887	122.2022	15.6634	895.865	3377926.6	12549.813	122.2022	
16.4825	876.939	3310991.5	12310.298	122.2042	16.1529	895.460	3380919.4	12561.457	122.2042	
16.9819	876.632	3314167.7	12322.610	122.2063	16.6424	895.146	3384162.7	12574.020	122.2063	
17.4814	876.367	3317374.7	12335.023	122.2083	17.1319	894.876	3387437.4	12586.686	122.2083	
17.9809	876.078	3320355.2	12346.735	122.2104	17.6214	894.581	3390480.9	12598.637	122.2104	
18.4803	875.700	3322891.6	12356.844	122.2124	18.1108	894.194	3393070.8	12608.953	122.2124	
18.9798	875.197	3324890.3	12364.953	122.2145	18.6003	893.681	3395111.7	12617.227	122.2145	
19.4793 19.9788	874.590 873.952	3326463.8 3327934.4	12371.559 12377.766	122.2165 122.2186	19.0898 19.5793	893.061 892.410	3396718.5 3398220.2	12623.968 12630.301	122.2165	
20.4782	873.952	3329403.8	12377.766	122.2186	20.0688	892.410	3398220.2	12636.633	122.2186 122.2207	
20.9777	872.485	3329971.9	12386.874	122.2207	20.0686	890.912	3400300.6	12639.595	122.2207	
21.4772	870.825	3326951.3	12376.564	122.2227	21.0477	889.217	3397216.3	12629.075	122.2227	
21.9766	867.071	3315883.3	12336.724	122.2268	21.5372	885.384	3385914.5	12588.422	122.2268	

Table 5.3.2-10: February and July RSRM Data (Continued)

ľ	F	ebruary F	RSRM (61 c	lea PMBT)		July RSRM (80 deg PMBT)						
ŀ	Time	Pressure	Fvac	Flowrate	Exit Area	Time	Pressure	Fvac	Flowrate	Exit Area		
ı	(sec)	(psi)	(lbs)	(lb/sec)	(ft²)	(sec)	(psi)	(lbs)	(lb/sec)	(ft <sup>2</sup> )		
ŀ	22.4761	860.046	3292960.0	12253.542	122.2289	22.0267	878.210	3362507.1	12503.543	122.2289		
ı	22.9756	851.438	3262410.9	12142.531	122.2309	22.5162	869.420	3331312.8	12390.267	122.2309		
ı	23.4750	842.486	3231458.2	12029.918	122.2330	23.0057	860.279	3299706.4	12275.356	122.2330		
ı	23.9745	833.994	3201710.9	11921.510	122.2350	23.4951	851.608	3269330.9	12164.737	122.2350		
ı	24.4740	826.019	3173674.5	11819.407	122.2371	23.9846	843.465	3240702.3	12060.551	122.2371		
ı	24.9734	818.513	3147460.1	11724.112	122.2391	24.4741	835.800	3213934.3	11963.311	122.2391		
ı	25.4729	811.434	3122923.0	11634.923	122.2412	24.9636	828.572	3188879.0	11872.303	122.2412		
ı	25.9724	804.745	3099821.9	11551.039	122.2432	25.4531	821.741	3165290.0	11786.707	122.2432		
ı	26.4718	798.400	3077925.6	11471.660	122.2453	25.9426	815.262	3142931.1	11705.709	122.2453		
ı	26.9713	792.350	3057033.6	11395.884	122.2473	26.4320	809.084	3121597.9	11628.387	122.2473		
ı	27.4708	786.545	3036974.8	11323.112	122.2494	26.9215	803.156	3101115.5	11554.130	122.2494		
ı	27.9703	780.940	3017604.1	11252.841	122.2514	27.4110	797.433	3081335.7	11482.426	122.2514		
ı	28.4697	775.499	2998801.3	11184.673	122.2535	27.9005	791.877	3062135.8	11412.866	122.2535		
ı	28.9692	770.191	2980467.6	11118.107	122.2555	28.3900	786.457	3043414.9	11344.943	122.2555		
ı	29.4687	764.995	2962524.2	11052.942	122.2576	28.8794	781.152	3025092.5	11278.448	122.2576		
ı	29.9681	759.894	2944909.9	10988.978	122.2596	29.3689	775.943	3007106.2	11213.179	122.2596		
	30.4676	754.877	2927578.5	10926.015	122.2617	29.8584	770.820	2989408.7	11148.931	122.2617		
	30.9671	749.936	2910496.1	10863.953	122.2637	30.3479	765.775	2971965.6	11085.603	122.2637		
	31.4665	745.068	2893640.2	10802.692	122.2658	30.8374	760.804	2954753.6	11023.092	122.2658		
	31.9660	740.270	2876995.9	10742.132	122.2678	31.3269	755.904	2937757.8	10961.296	122.2678		
	32.4655	735.542	2860554.4	10682.471	122.2699	31.8163	751.076	2920969.1	10900.419	122.2699		
	32.9649	730.884	2844310.4	10623.413	122.2719	32.3058	746.320	2904382.1	10840.155	122.2719		
	33.4644	726.294	2828260.6	10565.055	122.2740	32.7953	741.634	2887993.3	10780.607	122.2740		
ı	33.9639	721.775	2812401.6	10507.497	122.2760	33.2848	737.018	2871799.4	10721.874	122.2760		
ı	34.4633	717.320	2796730.1	10450.540	122.2781	33.7743	732.469	2855796.9	10663.755	122.2781		
ı	34.9628	712.925	2781241.8	10394.184	122.2801	34.2638	727.982	2839981.4	10606.250	122.2801		
ı	35.4623	708.583	2765930.7	10338.528	122.2822	34.7532	723.548	2824346.9	10549.458	122.2822		
ı	35.9618	704.289	2750789.2	10283.373	122.2842	35.2427	719.164	2808885.7	10493.178	122.2842		
ı	36.4612	700.036 695.818	2735806.4	10228.718	122.2863	35.7322	714.820	2793586.5 2778435.1	10437.408	122.2863		
ı	36.9607 37.4602	691.633	2720968.4 2706257.0	10174.565 10120.911	122.2883 122.2903	36.2217 36.7112	710.514 706.241	2763413.0	10382.150 10327.401	122.2883 122.2903		
ı	37.9596	687.480	2691650.6	10067.657	122.2903	37.2006	700.241	2748498.1	10327.401	122.2903		
ı	38.4591	683.359	2677130.5	10007.007	122.2924	37.6901	697.792	2733671.3	10273.001	122.2944		
ı	38.9586	679.279	2662692.6	9962.352	122.2965	38.1796	693.625	2718928.4	10165.608	122.2965		
ı	39.4580	675.248	2648358.6	9910.400	122.2985	38.6691	689.510	2704291.8	10112.596	122.2985		
ı	39.9575	671.283	2634188.8	9859.050	122.3006	39.1586	685.460	2689822.7	10060.197	122.3006		
ı	40.4570	667.405	2620286.8	9808.699	122.3026	39.6481	681.500	2675627.1	10008.820	122.3026		
ı	40.9564	663.637	2606782.1	9759.850	122.3047	40.1375	677.653	2661837.2	9958.974	122.3047		
ı	41.4559	660.003	2593799.3	9712.703	122.3067	40.6270	673.942	2648580.2	9910.865	122.3067		
ı	41.9554	656.520	2581427.2	9667.658	122.3088	41.1165	670.385	2635946.8	9864.901	122.3088		
ı	42.4548	653.201	2569694.2	9624.915	122.3108	41.6060	666.997	2623966.0	9821.286	122.3108		
ı	42.9543	650.043	2558572.8	9584.575	122.3129	42.0955	663.772	2612609.7	9780.123	122.3129		
	43.4538	647.030	2547998.6	9546.537	122.3149	42.5850	660.696	2601812.1	9741.308	122.3149		
	43.9533	644.137	2537890.2	9510.501	122.3170	43.0744	657.741	2591490.3	9704.538	122.3170		
	44.4527	641.346	2528170.4	9476.167	122.3190	43.5639	654.891	2581565.2	9669.503	122.3190		
	44.9522	638.663	2518784.9	9443.034	122.3211	44.0534	652.152	2571981.5	9635.694	122.3211		
	45.4517	636.120	2509720.0	9410.901	122.3231	44.5429	649.555	2562725.1	9602.906	122.3231		
	45.9511	633.679	2500904.9	9379.070	122.3252	45.0324	647.062	2553723.8	9570.425	122.3252		
	46.4506	631.126	2491852.1	9345.936	122.3272	45.5218	644.455	2544479.9	9536.615	122.3272		
	46.9501	628.187	2481506.3	9308.599	122.3293	46.0113	641.454	2533915.5	9498.516	122.3293		
۱	47.4495		2468867.5	9263.754	122.3313	46.5008	637.879	2521009.8	9452.756	122.3313		
	47.9490	620.633	2453797.0	9210.001	122.3334	46.9903	633.741	2505621.0	9397.907	122.3334		
	48.4485	616.237	2437172.6	9149.941	122.3355	47.4798	629.252	2488645.5	9336.621	122.3355		
١	48.9479 49.4474	611.854 607.874	2420508.3 2405509.3	9089.280 9034.425	122.3375 122.3396	47.9693 48.4587	624.776 620.713	2471629.3 2456313.5	9274.722 9218.749	122.3375		
۱	49.4474	604.584	2393505.6	8990.682	122.3396	48.4587 48.9482	617.352	2444056.3	9218.749	122.3396 122.3416		
	50.4463	602.043	2384683.7	8959.551	122.3416	49.4377	614.758	2435048.1	9174.113	122.3410		
	50.4463	600.141	2378119.1	8937.529	122.3457	49.4377	612.816	2428344.8	9142.347	122.3457		
١	51.4453	598.708	2372878.8	8919.811	122.3437	50.4167	611.353	2422993.9	9101.796	122.3437		
۱	51.9448	597.648	2369053.2	8906.397	122.3478	50.9061	610.270	2419087.5	9088.109	122.3478		
	52.4442	596.988	2367396.7	8900.691	122.3519	51.3956	609.597	2417396.0	9082.286	122.3519		
	52.9437	596.819	2368242.7	8904.596	122.3539	51.8851	609.424	2418259.9	9086.270	122.3539		
	53.4432	597.201	2371348.7	8917.108	122.3560	52.3746	609.813	2421431.5	9099.038	122.3560		
	53.9426	598.088	2376147.4	8935.727	122.3580	52.8641	610.719	2426331.5	9118.036	122.3580		
	54.4421	599.329	2382009.3	8958.149	122.3601	53.3536	611.987	2432317.1	9140.917	122.3601		

Table 5.3.2-10: February and July RSRM Data (Continued)

F	ebruary F	RSRM (61 d	leg PMBT)		July RSRM (80 deg PMBT)						
Time	Pressure	Fvac	Flowrate	Exit Area	Time	Pressure	Fvac	Flowrate	Exit Area		
(sec)	(psi)	(lbs)	(lb/sec)	(ft²)	(sec)	(psi)	(lbs)	(lb/sec)	(ft²)		
54.9416	600.741	2388439.3	8982.473	122.3621	53.8430	613.429	2438883.0	9165.737	122.3621		
55.4410	602.184	2395109.4	9007.699	122.3642	54.3325	614.902	2445693.9	9191.477	122.3642		
55.9405	603.602	2401824.0	9033.024	122.3662	54.8220	616.350	2452550.3	9217.319	122.3662		
56.4400	604.986	2408486.7	9058.149	122.3683	55.3115	617.763	2459353.8	9242.956	122.3683		
56.9394	606.340	2415064.5	9082.874	122.3703	55.8010	619.145	2466070.5	9268.186	122.3703		
57.4389	607.665	2421553.8	9107.398	122.3724	56.2905	620.499	2472696.9	9293.210	122.3703		
57.9384	608.965	2427953.3	9131.423	122.3744	56.7799	621.827	2479231.5	9317.725	122.3744		
58.4378	610.247	2434257.9	9155.146	122.3744	57.2694	623.135	2485669.3	9341.932	122.3765		
58.9373	611.510	2440459.6	9178.369	122.3785	57.7589	624.425	2492002.0	9365.630	122.3785		
59.4368	612.758	2446549.3	9201.092	122.3703	58.2484	625.699	2498220.3	9388.815	122.3703		
59.9363	613.987	2452517.8	9223.314	122.3826	58.7379	626.954	2504314.8	9411.491	122.3826		
60.4357	615.193	2458357.7	9245.036	122.3847	59.2273	628.186	2510278.1	9433.657	122.3847		
60.9352	616.370	2464064.9	9266.157	122.3867	59.7168	629.387	2516105.8	9455.208	122.3867		
61.4347	617.512	2469639.6	9286.878		60.2063	630.554	2521798.2	9476.352			
				122.3888					122.3888		
61.9341	618.613	2475088.3	9307.098	122.3908	60.6958	631.678	2527362.0	9496.985	122.3908		
62.4336	619.675	2480424.9	9326.817	122.3928	61.1853	632.763	2532811.3	9517.106	122.3928		
62.9331	620.703	2485672.3	9346.437	122.3949	61.6748	633.813	2538169.6	9537.126	122.3949		
63.4325	621.708	2490863.9	9365.757	122.3969	62.1642	634.838	2543470.8	9556.840	122.3969		
63.9320	622.708	2496039.9	9385.176	122.3990	62.6537	635.859	2548756.1	9576.656	122.3990		
64.4315	623.721	2501241.4	9404.695	122.4010	63.1432	636.894	2554067.4	9596.573	122.4010		
64.9309	624.764	2506501.6	9424.415	122.4031	63.6327	637.959	2559438.8	9616.695	122.4031		
65.4304	625.842	2511839.3	9444.535	122.4052	64.1222	639.060	2564889.2	9637.226	122.4052		
65.9299	626.949	2517249.9	9464.756	122.4072	64.6117	640.190	2570414.0	9657.859	122.4072		
66.4293	628.061	2522698.0	9484.975	122.4093	65.1011	641.326	2575977.3	9678.491	122.4093		
66.9288	629.146	2528117.7	9505.196	122.4113	65.5906	642.433	2581511.4	9699.124	122.4113		
67.4283	630.184	2533449.8	9525.216	122.4134	66.0801	643.493	2586956.1	9719.553	122.4134		
67.9278	631.194	2538714.4	9545.536	122.4154	66.5696	644.525	2592331.8	9740.287	122.4154		
68.4272	632.258	2544071.6	9566.156	122.4175	67.0591	645.611	2597802.3	9761.328	122.4175		
68.9267	633.430	2549558.7	9586.477	122.4195	67.5485	646.808	2603405.2	9782.064	122.4195		
69.4262	634.434	2554556.6	9604.795	122.4216	68.0380	647.834	2608508.6	9800.756	122.4216		
69.9256	635.087	2558716.1	9620.611	122.4236	68.5275	648.500	2612756.1	9816.894	122.4236		
70.4251	635.683	2562270.7	9634.325	122.4257	69.0170	649.108	2616385.7	9830.888	122.4257		
70.9246	636.264	2565453.2	9646.637	122.4277	69.5065	649.701	2619635.5	9843.451	122.4277		
71.4240	636.787	2568402.4	9657.748	122.4298	69.9960	650.236	2622646.9	9854.788	122.4298		
71.9235	637.231	2571181.6	9668.059	122.4319	70.4854	650.689	2625484.8	9865.310	122.4319		
72.4230	637.598	2573800.2	9677.768	122.4339	70.9749	651.064	2628158.7	9875.217	122.4339		
72.9224	637.902	2576232.1	9686.778	122.4360	71.4644	651.375	2630642.0	9884.410	122.4360		
73.4219	638.162	2578437.0	9695.085	122.4380	71.9539	651.640	2632893.4	9892.888	122.4380		
73.9214	638.385	2580379.8	9702.593	122.4401	72.4434	651.867	2634877.2	9900.549	122.4401		
74.4208	638.569	2582048.3	9709.199	122.4421	72.9328	652.056	2636581.0	9907.290	122.4421		
74.9203	638.694	2583454.8	9714.605	122.4442	73.4223	652.183	2638017.2	9912.806	122.4442		
75.4198	638.736	2584613.9	9718.809	122.4462	73.9118	652.226	2639200.8	9917.095	122.4462		
75.9193	638.688	2585518.3	9721.712	122.4483	74.4013	652.177	2640124.3	9920.057	122.4483		
76.4187	638.555	2586114.5	9723.714	122.4503	74.8908	652.041	2640733.0	9922.100	122.4503		
76.9182	638.338	2586272.2	9724.615	122.4524	75.3803	651.819	2640894.1	9923.020	122.4524		
77.4177	637.988	2585725.5	9723.413	122.4544	75.8697	651.463	2640335.9	9921.794	122.4544		
77.9171	637.363	2583984.5	9717.708	122.4565	76.3592	650.824	2638558.1	9915.972	122.4565		
78.4166		2580211.8	9703.193	122.4585	76.8487	649.624	2634705.7	9901.161	122.4585		
78.9161	634.100	2573051.6	9675.065	122.4606	77.3382	647.492	2627394.3	9872.459	122.4606		
79.4155	630.924	2561064.8	9631.222	122.4626	77.8277	644.249	2615154.3	9827.721	122.4626		
79.9150	626.951	2545238.7	9575.266	122.4647	78.3172	640.192	2598994.0	9770.624	122.4647		
80.4145	622.849	2529471.6	9516.707	122.4667	78.8066	636.003	2582893.9	9710.870	122.4667		
80.9139	619.257	2516200.4	9465.456	122.4688	79.2961	632.335	2569342.4	9658.574	122.4688		
81.4134	616.356	2505002.4	9424.515	122.4708	79.7856	629.373	2557907.9	9616.797	122.4708		
81.9129	613.708	2494393.4	9388.278	122.4708	80.2751	626.669	2547074.9	9579.821	122.4708		
82.4123	610.737	2482828.0	9347.538	122.4720	80.7646	623.636	2535265.2	9538.250	122.4749		
82.9118	607.160	2469232.1	9296.287	122.4749	81.2540	619.983	2521382.1	9485.953	122.4749		
	603.037										
83.4113		2453503.9	9235.826	122.4790	81.7435	615.773	2505321.7	9424.259	122.4790		
83.9108	598.710	2436692.4	9173.164	122.4810	82.2330	611.355	2488155.2	9360.318	122.4810		
84.4102	594.595	2420363.6	9114.506	122.4831	82.7225	607.152	2471481.6	9300.463	122.4831		
84.9097	590.917	2405650.1	9061.753	122.4852	83.2120	603.397	2456457.2	9246.633	122.4852		
85.4092	587.534	2392385.3	9012.303	122.4872	83.7015	599.943	2442912.3	9196.175	122.4872		
85.9086	584.052	2379091.6	8961.253	122.4893	84.1909	596.387	2429337.8	9144.083	122.4893		
86.4081	580.111	2363982.5	8903.795	122.4913	84.6804	592.363 587.692	2413909.6	9085.453	122.4913		
86.9076	575.537	2346017.1	8836.727	122.4934	85.1699		2395564.8	9017.017	122.4934		

Table 5.3.2-10: February and July RSRM Data (Continued)

F	ebruary l	RSRM (61 c	leg PMBT)			July RS	RM (80 deg	PMBT)	
Time	Pressure	Fvac	Flowrate	Exit Area	Time	Pressure	Fvac	Flowrate	Exit Area
(sec)	(psi)	(lbs)	(lb/sec)	(ft²)	(sec)	(psi)	(lbs)	(lb/sec)	(ft²)
87.4070	570.346	2325254.4	8760.051	122.4954	85.6594	582.392	2374363.6	8938.777	122.4954
87.9065	564.718	2302629.8	8676.268	122.4975	86.1489	576.645	2351261.2	8853.284	122.4975
88.4060	558.966	2279666.7	8590.482	122.4995	86.6383	570.771	2327813.1	8765.748	122.4995
88.9054	553.508	2258169.3	8509.301	122.5016	87.1278	565.198	2305861.7	8682.910	122.5016
89.4049	548.797	2239803.6	8439.932	122.5036	87.6173	560.388	2287108.1	8612.126	122.5036
89.9044	545.167	2225598.8	8386.778	122.5057	88.1068	556.681	2272603.3	8557.888	122.5057
90.4038	542.670	2215627.1	8350.342	122.5078	88.5963	554.131	2262421.1	8520.708	122.5078
90.9033	541.002	2209009.2	8326.318	122.5098	89.0858	552.428	2255663.3	8496.195	122.5098
91.4028	539.659	2204098.1	8308.100	122.5119	89.5752	551.057	2250648.5	8477.604	122.5119
91.9023	538.188	2199019.1	8289.081	122.5139	90.0647	549.554	2245462.2	8458.197	122.5139
92.4017	536.368	2192397.2	8264.456	122.5160	90.5542	547.696	2238700.5	8433.071	122.5160
92.9012	534.176	2183808.5	8233.025	122.5180	91.0437	545.458	2229930.4	8400.998	122.5180
93.4007	531.622	2173496.1	8195.587	122.5201	91.5332	542.850	2219400.2	8362.796	122.5201
93.9001	528.677	2161804.1	8152.845	122.5221	92.0227	539.843	2207461.2	8319.182	122.5221
94.3996	525.311	2148754.5	8104.597	122.5242	92.5121	536.405	2194136.1	8269.949	122.5242
94.8991	521.554	2134209.9	8050.642	122.5262	93.0016	532.570	2179284.3	8214.894	122.5262
95.3985	517.540	2118359.8	7992.184	122.5283	93.4911	528.470	2163099.4	8155.243	122.5283
95.8980	513.468	2101988.0	7932.024	122.5303	93.9806	524.312	2146381.9	8093.856	122.5303
96.3975	509.536	2086131.5	7873.566	122.5324	94.4701	520.298	2130190.4	8034.205	122.5324
96.8969	505.866	2071505.2	7819.412	122.5344	94.9595	516.550	2115255.3	7978.946	122.5344
97.3964	502.452	2058101.9	7770.162	122.5365	95.4490	513.064	2101568.8	7928.692	122.5365
97.8959	499.224	2045486.4	7723.916	122.5385	95.9385	509.767	2088686.9	7881.502	122.5385
98.3954	496.145	2033419.2	7678.971	122.5406	96.4280	506.624	2076364.9	7835.640	122.5406
98.8948	493.231	2021910.1	7635.829	122.5426	96.9175	503.648	2064612.7	7791.617	122.5426
99.3943	490.439	2010707.1	7594.987	122.5447	97.4070	500.797	2053173.1	7749.943	122.5447
99.8938	487.598	1999216.9	7554.047	122.5467	97.8964	497.896	2041440.3	7708.167	122.5467
100.3932	484.479	1986744.3	7508.501	122.5488	98.3859	494.711	2028704.2	7661.692	122.5488
100.8927	480.920	1972751.4	7456.049	122.5508	98.8754	491.077	2014415.8	7608.169	122.5508
101.3922	476.935	1957060.8	7397.390	122.5529	99.3649	487.008	1998393.8	7548.314	122.5529
101.8916	472.683	1939847.1	7333.927	122.5549	99.8544	482.666	1980816.5	7483.556	122.5549
102.3911	468.257	1921507.8	7257.250	122.5570	100.3439	478.146	1962089.9	7405.315	122.5570
102.8906	463.636	1902526.4	7170.363	122.5590	100.8333	473.428	1942707.7	7316.655	122.5590
103.3900	458.822	1883333.7	7082.075	122.5611	101.3228	468.513	1923109.6	7226.566	122.5611
103.8895	453.974	1864172.5	6993.286	122.5631	101.8123	463.562	1903543.7	7135.965	122.5631
104.3890	449.267	1845012.7	6904.197	122.5652	102.3018	458.755	1883979.2	7045.059	122.5652
104.8884	444.607	1825666.4	6814.908	122.5673	102.7913	453.997	1864224.4	6953.948	122.5673
105.3879	439.769	1806015.8	6725.118	122.5693	103.2807	449.057	1844158.8	6862.326	122.5693
105.8874	434.835	1786317.9	6635.229	122.5714	103.7702	444.019	1824044.8	6770.603	122.5714
106.3869	430.226	1767653.0	6548.743	122.5734	104.2597	439.312	1804985.8	6682.352	122.5734
106.8863	426.392	1751899.0	6484.278	122.5755	104.7492	435.397	1788899.0	6616.573	122.5755
107.3858	423.542	1740353.9	6441.335	122.5775	105.2387	432.487	1777110.1	6572.753	122.5775
107.8853	421.533	1732413.6	6412.406	122.5796	105.7282	430.436	1769002.1	6543.234	122.5796
108.3847	419.855	1725737.5	6388.983	122.5816	106.2176	428.722	1762184.9	6519.333	122.5816
108.8842	417.685	1716995.8	6357.151	122.5837	106.7071	426.507	1753258.6	6486.852	122.5837
109.3837	413.979	1702118.8	6300.494	122.5857	107.1966	422.722	1738067.4	6429.039	122.5857
109.8831	407.602	1675783.8	6203.297	122.5878	107.6861	416.211	1711176.3	6329.858	122.5878
110.3826	398.184	1636385.6	6059.854	122.5899	108.1756	406.593	1670946.0	6183.489	122.5899
110.8821	386.622	1587867.9	5880.875	122.5919	108.6650	394.787	1621403.6	6000.859	122.5919
111.3815	372.813	1530192.2	5674.068	122.5940	109.1545	380.687	1562509.7	5789.833	122.5940
111.8810	356.405	1461902.3	5426.421	122.5960	109.6440	363.932	1492777.6	5537.133	122.5960
112.3805	335.260	1374292.7	5105.600	122.5981	110.1335	342.340	1403317.7	5209.767	122.5981
112.8799	309.528	1267815.1	4712.808	122.6001	110.6230	316.066	1294591.3	4808.960	122.6001
113.3794	279.674	1144596.2	4255.651	122.6022	111.1125	285.581	1168770.0	4342.477	122.6022
113.8789	249.558	1021019.1	3793.990	122.6042	111.6019	254.829	1042583.0	3871.396	122.6042
114.3784	223.142	913879.5	3390.687	122.6063	112.0914	227.855	933180.6	3459.865	122.6063
114.8778	201.240	824251.0	3057.254	122.6083	112.5809	205.490	841659.1	3119.629	122.6083
115.0776	194.109	794059.3	2944.141	122.6091	112.7767	198.209	810829.8	3004.208	122.6091
115.2774	187.258	765769.4	2841.439	122.6099	112.9725	191.213	781942.4	2899.411	122.6099
115.4772	180.693	739072.4	2742.840	122.6108	113.1683	184.509	754681.6	2798.800	122.6108
115.6770	174.434	713663.7	2647.245	122.6116	113.3641	178.118	728736.2 703786.3	2701.255	122.6116 122.6124
115.8768	168.442 162.647	689229.8 665486.6	2555.353	122.6124 122.6133	113.5599 113.7557	172.000 166.082		2607.488 2517.398	
116.0765 116.2763	162.647 156.981	665486.6 642215.2	2467.064 2381.079	122.6133 122.6140	113.7557 113.9515	166.082 160.296	679541.7 655778.8	2517.398 2429.658	122.6133 122.6140
116.2763	150.961	619286.7	2296.594	122.6140	114.1472	154.597	632366.0	2343.450	122.6140
110.4701	101.033	019200.7	2230.034	144.0148	117.1412	107.08/	002000.0	20 <del>4</del> 0.400	144.0143

Table 5.3.2-10: February and July RSRM Data (Continued)

F	ebruary F	RSRM (61 c	deg PMBT)		July RSRM (80 deg PMBT)						
Time	Pressure	Fvac	Flowrate	Exit Area	Time	Pressure	Fvac	Flowrate	Exit Area		
(sec)	(psi)	(lbs)	(lb/sec)	(ft²)	(sec)	(psi)	(lbs)	(lb/sec)	(ft²)		
116.6759	145.882	596655.1	2213.111	122.6157	114.3430	148.963	609256.4	2258.264	122.6157		
116.8757	140.432	574336.5	2130.529	122.6165	114.5388	143.398	586466.5	2173.996	122.6165		
117.0755	135.062	552390.4	2049.447	122.6174	114.7346	137.915	564056.8	2091.261	122.6174		
117.2753	129.798	530898.0	1969.768	122.6182	114.9304	132.539	542110.5	2009.956	122.6182		
117.4751	124.661	509942.2	1892.090	122.6190	115.1262	127.294	520712.2	1930.693	122.6190		
117.6748	119.674	489587.7	1816.615	122.6198	115.3220	122.201	499927.7	1853.678	122.6198		
117.8746	114.845	469860.0	1743.642	122.6206	115.5178	117.271	479783.4	1779.216	122.6206		
118.0744	110.169	450726.1	1673.072	122.6215	115.7136	112.496	460245.4	1707.206	122.6215		
118.2742	105.620	432084.7	1604.503	122.6223	115.9094	107.851	441210.3	1637.238	122.6223		
118.4740	101.158	413784.9	1537.135	122.6231	116.1052	103.295	422524.0	1568.496	122.6231		
118.6738	96.737	395654.2	1470.469	122.6239	116.3010	98.780	404010.4	1500.470	122.6239		
118.8736	92.312	377528.1	1403.502	122.6247	116.4968	94.262	385501.4	1432.137	122.6247		
119.0734	87.851	359279.9	1335.934	122.6256	116.6926	89.706	366867.9	1363.191	122.6256		
119.2731	83.337	340849.0	1267.567	122.6264	116.8883	85.097	348047.7	1293.428	122.6264		
119.4729	78.781	322270.2	1198.597	122.6272	117.0841	80.444	329076.5	1223.052	122.6272		
119.6727	74.224	303686.5	1129.628	122.6280	117.2799	75.791	310100.3	1152.675	122.6280		
119.8725	69.728	285320.2	1061.561	122.6288	117.4757	71.200	291346.1	1083.219	122.6288		
120.0723	65.358	267430.2	995.294	122.6297	117.6715	66.738	273078.3	1015.601	122.6297		
120.2721	61.175	250265.9	932.031	122.6305	117.8673	62.467	255551.5	951.047	122.6305		
120.4719	57.219	234024.0	872.272	122.6313	118.0631	58.427	238966.6	890.068	122.6313		
120.6717	53.510	218816.4	816.415	122.6322	118.2589	54.640	223437.8	833.072	122.6322		
120.8714	50.055	204679.7	764.779	122.6329	118.4547	51.112	209002.5	780.382	122.6329		
121.0712 121.2710	46.832 43.839	191501.5 179260.9	716.416 671.671	122.6338 122.6346	118.6505	47.822 44.765	195546.0 183046.9	731.033 685.374	122.6338 122.6346		
121.4708	43.039	167877.8	630.430	122.6346	118.8463 119.0421	41.922	171423.3	643.292	122.6354		
121.4708	38.459	157265.0	592.091	122.6363	119.0421	39.272	160586.4	604.171	122.6363		
121.8704	36.033	147342.3	556.456	122.6370	119.2379	36.794	150454.1	567.809	122.6370		
122.0702	33.758	138040.5	523.323	122.6378	119.6294	34.471	140955.9	534.000	122.6378		
122.2700	31.621	129301.9	492.392	122.6387	119.8252	32.289	132032.7	502.438	122.6387		
122.4697	29.611	121078.7	463.463	122.6395	120.0210	30.236	123635.9	472.919	122.6395		
122.6695	27.715	113333.1	436.436	122.6403	120.2168	28.301	115726.7	445.340	122.6403		
122.8693	25.932	106034.5	411.211	122.6412	120.4126	26.479	108273.9	419.601	122.6412		
123.0691	24.250	99158.8	387.687	122.6419	120.6084	24.762	101253.0	395.597	122.6419		
123.2689	22.667	92687.2	365.665	122.6428	120.8042	23.146	94644.7	373.126	122.6428		
123.3408	22.124	90469.5	357.904	122.6431	120.8747	22.592	92380.2	365.206	122.6431		
123.4687	21.179	86603.7	343.543	122.6432	121.0000	21.627	88432.8	350.552	122.6432		
123.6685	19.783	80894.5	322.122	122.6432	121.1958	20.201	82603.0	328.694	122.6432		
123.8683	18.475	75546.3	302.002	122.6432	121.3916	18.866	77141.8	308.163	122.6432		
124.0680	17.252	70544.9	283.283	122.6432	121.5874	17.616	72034.8	289.063	122.6432		
124.2678	16.110	65874.1	265.865	122.6432	121.7832	16.450	67265.4	271.290	122.6432		
124.4676	15.044	61514.7	249.649	122.6432	121.9790	15.361	62813.9	254.743	122.6432		
124.6674	14.047	57442.8	234.434	122.6432	122.1748	14.344	58655.9	239.217	122.6432		
124.8672	13.115	53628.7	220.120	122.6432	122.3706	13.392	54761.3	224.611	122.6432		
125.0670	12.236	50036.1	206.506	122.6432	122.5663	12.495	51092.9	210.720	122.6432		
125.2668	11.402	46620.6	193.393	122.6432	122.7621	11.642	47605.2	197.339	122.6432		
125.4666	10.598	43335.2	180.581	122.6432	122.9579	10.821	44250.5	184.265	122.6432		
125.6663	9.820	40152.4	167.867	122.6432	123.1537	10.027	41000.5	171.292	122.6432		
125.8661	9.071	37091.9	155.355 150.390	122.6432 122.6432	123.3495	9.263	37875.3	158.525	122.6432		
125.9460 126.0659	8.786 8.376	35925.2 34249.2	143.143	122.6432	123.4278 123.5453	8.971 8.553	36683.9 34972.5	153.458 146.064	122.6432 122.6432		
	7 700	040400					004000				
126.2657 126.4655	7.782	31818.3 29969.7	132.032 122.722	122.6432 122.6432	123.7411 123.9369	7.946 7.484	32490.3 30602.7	134.726 125.226	122.6432 122.6432		
126.4653	6.935	28356.0	113.914	122.6432	123.9309	7.464	28954.9	116.238	122.6432		
126.8651	6.307	25790.6	101.001	122.6432	124.1327	6.440	26335.2	103.062	122.6432		
127.0649	5.830	23839.8	94.695	122.6432	124.5243	5.954	24343.3	96.627	122.6432		
127.2646	4.945	20222.7	83.484	122.6432	124.7201	5.050	20649.8	85.187	122.6432		
127.4644	4.439	18150.4	77.678	122.6432	124.9159	4.533	18533.7	79.262	122.6432		
127.6642	3.736	15275.4	67.968	122.6432	125.1117	3.815	15598.0	69.355	122.6432		
127.8640	3.469	14185.7	63.764	122.6432	125.3074	3.542	14485.3	65.065	122.6432		
128.0638	3.081	12601.9	55.755	122.6432	125.5032	3.146	12868.0	56.893	122.6432		
128.2636	2.769	11324.6	46.747	122.6432	125.6990	2.828	11563.7	47.700	122.6432		
128.4634	2.551	10429.9	43.643	122.6432	125.8948	2.605	10650.1	44.534	122.6432		
128.6632	2.391	9775.2	47.948	122.6432	126.0906	2.441	9981.7	48.926	122.6432		
		9250.8	51.952	122.6432	126.2864	2.310	9446.1	53.012	122.6432		

## 5.3.2.7 KSC MEAN ANNUAL WINDS

The KSC mean annual winds data shown in Table 5.3.2-11 were provided by Lee Burns / Raytheon (MSFC Contractor) on 1/14/2004. The mean annual winds were derived from the Range Reference Atmosphere Dataset (Range Commanders' Council document 361-83, dated February 1983). The North and East components were derived by Dr. Greg Dukeman / MSFC/EV42 on 3/19/2004. These tables use the meteorological definitions for the directional components. For the north component, a southerly wind is shown as a positive value. The westerly winds are positive values for the east component of the wind speed vector.

Table 5.3.2-11: KSC Mean Annual Winds

				KSC	Mean A	nnual Wi	inds				
Altitud	North	East	Altitud	North	East	Altitud	North	East	Altitud	North	East
e (ft)	(fps)	(fps)	e (ft)	(fps)	(fps)	e (ft)	(fps)	(fps)	e (ft)	(fps)	(fps)
0	-0.6011	-0.6010	101706	5.8555	-0.4118	200131	22.111	38.225	301837	-9.0842	75.737 1
3281	3.8080	2.7377	104987	8.1010	2.1050	203412	19.580	40.416	305118	-8.9048	79.392 2
6562	3.5294	9.7828	108268	6.5814	6.1681	206693	17.272	42.245	308399	-8.7481	82.407 0
9843	3.2601	16.541 8	111549	5.1147	10.165 9	209974	14.920	44.136 1	311680	-8.5544	84.809 7
13123	3.0398	23.060 5	114829	2.8209	10.145 1	213255	10.646	44.144 4	314961	-8.3135	83.738 3
16404	2.7993	29.397 0	118110	0.6059	10.021 7	216535	6.4405	44.314 4	318241	-8.0052	82.090 6
19685	2.7210	35.365 5	121391	1.2064	8.8078	219816	3.4307	39.059 6	321522	-7.6830	79.941 7
22966	2.8034	41.445 3	124672	1.8323	7.5920	223097	0.5601	33.785 4	324803	-7.3272	77.373 8
26247	2.7513	47.720 8	127953	3.1491	6.7848	226378	0.0676	43.109 9	328084	-7.0211	74.278 9
29528	2.3720	54.118 0	131234	4.4018	6.0105	229659	3.8985	51.492 6	331365	-6.7021	68.603 4
32808	1.6918	60.576 4	134514	5.4732	4.3479	232940	3.7866	49.104 2	334646	-6.4043	62.834 5
36089	0.3258	66.699 2	137795	6.4665	2.8084	236220	3.6510	46.707 5	337927	-6.1568	57.048 7
39370	-1.0985	72.331 7	141076	9.1799	1.6098	239501	3.5698	44.276 3	341207	-6.0003	51.390 9
42651	-2.1938	73.917 5	144357	11.905 5	0.3256	242782	3.4549	41.847 6	344488	-5.9718	45.671 2
45932	-3.0890	69.091 0	147638	15.028 2	0.9788	246063	3.2970	39.432 4	347769	-5.9481	37.096 2
49213	-2.6084	59.983 3	150919	18.239 3	1.6074	249344	1.9853	38.528 9	351050	-6.0861	29.468 1
52493	-2.0352	48.157 0	154199	19.920 3	2.7557	252625	0.6172	37.624 9	354331	-6.4740	22.819 4
55774	-1.5456	34.705 6	157480	21.671 2	3.8470	255906	-0.6466	36.674 3	357612	-6.9937	17.062 3
59055	-1.2485	20.472 0	160761	22.804 0	6.6362	259186	-2.0687	35.770 2	360892	-7.6425	12.322 4
62336	-1.0340	7.1759	164042	24.047 1	9.4846	262467	-3.3246	34.912 1	364173	-8.5078	9.2532
65617	-0.6034	-2.5496	167323	24.213 9	11.345 2	265748	-3.7024	35.829 2	367454	-9.5518	6.7740
68898	0.1079	-8.7893	170604	24.412 0	13.279 3	269029	-4.2215	36.768 5	370735	- 10.865	4.8774
72178	0.1628	11.908	173885	24.412 0	15.542 8	272310	-4.6325	37.676 3	374016	5 - 12.404	3.5044
75459	0.1617	9 - 12.989 0	177165	24.476 4	17.877 4	275591	-5.0741	38.648 3	377297	5 - 14.116 2	2.5236
78740	0.0533	-	180446	24.254	21.012	278871	-5.4809	39.552	380577	-	2.1934

		13.679		0	2			0		16.020	
82021	-0.0562	9 - 13.059	183727	24.087 2	24.208 1	282152	-6.2451	45.008 8	383858	5 - 18.126	2.0319
85302	0.0528	9 - 11.939	187008	24.528 4	26.921 6	285433	-6.9591	50.543 2	387139	5 - 20.330	2.1139
88583	0.4628	9 - 10.459	190289	25.074 4	29.674 8	288714	-7.7163	55.970 6	390420	4 - 22.733	2.4962
91864	1.2319	8 -8.8749	193570	24.855 4	32.820 4	291995	-8.4288	61.535 4	393701	4 - 25.122	2.9895
95144	1.8605	-5.9143	196850	24.674 8	35.946 0	295276	-9.1572	67.027 4	100000 0	8 - 25.122	2.9895
98425	3.4939	-2.9767				298556	-9.1335	71.599 8		8	

# 5.3.2.8 MEAN GRAM WIND PROFILES

The mean February GRAM 99 and mean July GRAM 99 wind profiles were provided by Dr. Greg Dukeman / MSFC/EV42. These wind profiles assume a due-east launch from KSC and are listed in Table 5.3.2-12.

Table 5.3.2-12: Mean GRAM Wind Profiles

Alittude		Mean F	ebruary GF	RAM Wind F	Profile	Mea	n July GRA	M Wind Pro	file
Component Comp									
0.0 -2.119	A	ltitude	Component	Component	Component	Altitude	Component	Component	Component
3.8					\ I /		· · · · · ·		
23.2									
\$9.5									
113.1									
1848   1.758   3.631   0.0005   184.8   4.192   0.844   0.0009   3.85.8   1.366   4.224   0.0010   3.85.9   4.448   0.0532   0.0013   616.5   1.110   4.609   0.0013   516.7   4.615   0.329   0.0025   688.2   0.814   5.056   0.0011   668.4   4.808   0.0033   0.0024   4.448   0.0532   0.0015   688.2   0.476   5.566   0.0007   841.6   5.028   0.176   0.0023   1.255.0   0.329   6.779   0.0001   1.253.6   5.553   0.816   0.0020   1.253.0   0.329   6.779   0.0001   1.253.6   5.553   0.816   0.0020   1.253.0   0.329   6.779   0.0001   1.253.6   5.553   0.816   0.0020   1.402.3   0.797   7.485   0.0006   1.493.1   5.888   1.189   0.0019   1.754.2   1.309   8.256   0.0011   1.755.2   6.191   1.596   0.0017   2.038.8   1.866   9.095   0.0017   2.040.0   6.554   2.038   0.0015   2.346.3   3.133   10.976   0.0023   2.678.8   7.387   3.031   0.0013   3.007.6   3.305   3.205   0.0023   2.678.8   7.387   3.031   0.0013   3.007.6   4.332   13.307   0.0023   2.678.8   7.387   3.031   0.0011   3.300.6   3.303.4   4.401   15.052   0.0052   3.811.6   8.076   3.879   0.0007   3.807.9   4.401   15.052   0.0069   4.236.0   7.689   4.037   0.0002   4.678.4   4.675   18.846   0.0009   4.236.0   7.689   4.037   0.0002   4.678.4   4.675   18.846   0.0009   4.236.0   7.689   4.037   0.0002   4.678.4   4.675   18.846   0.0009   4.236.0   7.689   4.037   0.0002   4.578.2   4.582   4.007   0.0002   4.578.2   4.101   0.0004   4.555.3   5.577   5.548   0.0009   4.286.0   7.689   4.037   0.0002   4.578.4   4.555   5.577   5.548   0.0009   4.286.0   7.689   4.037   0.0002   4.584.5   5.568   4.102   0.0003   4.236.0   7.689   4.037   0.0002   4.584.5   5.568   4.102   0.0003   4.584.5   5.568   4.102   0.0003   4.584.5   5.568   4.102   0.0003   4.286.0   6.0001   6.165.2   6.782   4.172   0.0003   6.165.3   5.573   4.101   0.0004   6.165.2   6.782   4.172   0.0003   6.165.3   5.573   6.374   4.294   0.0007   5.164.8   7.285   6.314   4.294   0.0007   6.165.2   6.782   4.172   0.0003   6.165.3   6.314   4.294   0.00003   6.165.3   6.314   4.29									
275.4									
385.8									
516.5									
841.2									
1,036.0 - 0.095	(	668.2	-0.814	5.056	0.0011	668.4	4.808	-0.093	0.0024
1.253.0 0.329 6.779 -0.0001 1.253.6 5.553 0.816 0.0020 1.492.3 0.797 7.485 -0.0006 1.493.1 5.858 1.189 0.0019 1.754.2 1.309 8.256 -0.0011 1.755.2 6.191 1.596 0.0017 2.038.8 1.866 9.095 -0.0017 2.040.0 6.554 2.038 0.0015 2.346.3 2.467 10.002 -0.0023 2.347.9 6.946 2.517 0.0013 2.676.8 3.113 10.976 -0.0029 2.678.8 7.367 3.031 0.0011 3.030.6 3.805 12.019 -0.0036 3.033.0 7.818 3.582 0.0009 3.407.6 4.332 13.307 -0.0044 3.410.6 8.076 3.979 0.0007 3.807.9 4.440 15.052 -0.0052 3.811.6 7.888 4.007 0.0004 4.231.6 4.554 16.898 -0.0060 4.236.0 7.689 4.037 0.0002 4.678.4 4.675 18.846 -0.0069 4.683.8 7.478 4.068 -0.0001 5.148.2 4.801 20.894 -0.0075 5.154.8 7.257 4.101 -0.0004 5.640.6 4.934 23.040 -0.0073 5.648.7 7.025 4.135 -0.0004 5.640.6 4.934 23.040 -0.0073 6.682.2 5.212 27.577 -0.0069 6.704.1 6.552 6.762 4.172 -0.0003 6.892.2 5.212 27.577 -0.0069 6.704.1 6.552 6.762 4.172 -0.0003 7.251.1 5.335 29.806 -0.0066 7.2653 3.442 4.0003 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.848.9 6.081 4.366 -0.0002 7.832.0 5.463 32.124 -0.0064 7.2653 5.14 2.21 -0.0003 7.0000 7.0000 7.00000 7.00000 7.000000 7.000000 7.00000000	8	841.2	-0.476	5.566	0.0007	841.6	5.028	0.176	0.0023
1,492.3   0.797   7,485   -0.0006   1,493.1   5,858   1,189   0.0019   1,754.2   1,309   8,256   -0.0017   2,040.0   6,554   2,038   0.0015   2,346.3   2,467   0.002   0.0029   2,678.8   7,367   3,031   0.0011   3,030.6   3,805   12,019   -0.0029   2,678.8   7,367   3,031   0.0011   3,030.6   3,805   12,019   -0.0036   3,033.0   7,818   3,582   0.0009   3,407.6   4,332   13,307   -0.0044   3,410.6   8,076   3,979   0.0007   3,807.9   4,440   15,052   -0.0052   3,811.6   7,888   4,007   0.0004   4,231.6   4,554   16,898   -0.0060   4,236.0   7,689   4,037   0.0002   4,678.4   4,675   18,846   -0.0069   4,683.8   7,478   4,068   -0.0015   5,148.2   4,801   2,894   -0.0075   5,154.8   7,257   4,113   -0.0004   6,155.3   5,073   25,224   -0.0071   6,165.2   6,762   4,172   -0.0003   6,692.2   5,212   27,577   -0.0069   6,704.1   6,539   4,218   -0.0003   7,251.1   5,335   29,806   -0.0066   7,265.3   6,314   4,291   -0.0002   8,434.5   5,595   34,527   -0.0064   7,849.9   6,081   4,366   -0.0002   8,454.5   5,595   34,527   -0.0061   8,454.4   5,839   4,445   -0.0002   8,454.5   5,595   34,527   -0.0061   8,454.4   5,839   4,445   -0.0002   9,058.2   5,732   37,016   -0.0058   9,085.2   5,732   3,733   3,733   3,73									
1,754.2									
2,038,8 1,866 9,095 -0.0017 2,040,0 6,554 2,038 0,0015 2,346,3 2,467 10,002 -0.0023 2,678,8 7,367 3,031 0,0011 3,030,6 3,031 11,01976 -0.0029 2,678,8 7,367 3,031 0,0011 3,030,6 3,805 12,019 -0.0036 3,033.0 7,818 3,582 0,0009 3,407,6 4,332 13,307 -0.0044 3,410,6 8,076 3,979 0,0007 3,807,9 4,440 15,052 -0.0052 3,811,6 7,888 4,007 0,0004 4,231,6 4,554 16,898 -0.0060 4,236,0 7,889 4,037 0,0002 4,678,4 4,675 18,846 -0.0060 4,236,0 7,889 4,037 0,0002 4,678,4 4,675 18,846 -0.0069 4,883,8 7,478 4,068 -0.0001 5,148,2 4,801 20,894 -0.0075 5,154,8 7,257 4,101 -0.0004 6,155,3 5,073 25,284 -0.0071 6,165,2 6,782 4,172 -0.0003 6,692,2 5,212 27,577 -0.0069 6,704,1 6,599 4,218 -0.0003 7,832,0 5,483 32,124 -0.0064 7,848,9 6,081 4,366 -0.0002 7,832,0 5,483 32,124 -0.0064 7,848,9 6,081 4,366 -0.0002 7,832,0 5,483 32,124 -0.0064 7,848,9 6,081 4,366 -0.0002 9,058,2 5,732 37,016 -0.0058 9,081,5 5,588 4,527 -0.0001 10,367,7 6,187 4,205,7 -0.0055 9,728,8 5,329 4,611 -0.0002 9,058,2 5,732 37,016 -0.0056 9,728,8 5,329 4,611 -0.0001 11,367,7 6,187 4,205,7 -0.0055 9,728,8 5,329 4,611 -0.0001 11,367,7 6,187 4,2057 -0.0055 9,728,8 5,329 4,611 -0.0001 11,367,7 6,187 4,550 -0.0055 9,728,8 5,329 4,611 -0.0001 11,367,7 6,187 4,2057 -0.0055 9,728,8 5,329 4,611 -0.0001 11,367,7 6,187 4,550 -0.0055 9,728,8 5,329 4,611 -0.0001 11,367,7 6,187 4,550 -0.0055 9,728,8 5,329 4,611 -0.0001 11,367,7 6,187 4,2057 -0.0053 10,399,1 5,105 4,620 0,0000 11,561,8 9,284 4,554 4,00041 13,278,1 4,424 4,599 0,0000 11,561,8 9,284 4,555 4,400001 11,572,2 6,938 47,114 -0.0047 11,798,8 4,657 4,600 0,0000 11,728,4 10,331 67,893 -0.0055 14,849,4 14,844 4,599 0,0000 11,561,8 9,284 4,574 0,0041 13,278,1 4,424 4,599 0,0000 11,561,8 9,284 4,577 0,0061 13,600,8 1,4									
2.346.3									
2,676.8 3.113 10.976 -0.0029 2,678.8 7.367 3.031 0.0011 3.030.6 3.805 12.019 -0.0036 3.033.0 7.818 3.582 0.0009 3.407.6 4.332 13.307 -0.0044 3.410.6 8.076 3.979 0.0007 3.807.9 4.440 15.052 -0.0052 3.811.6 7.888 4.007 0.0004 4.231.6 4.554 16.898 -0.0060 4.236.0 7.689 4.0037 0.0002 4.678.4 4.675 18.846 -0.0069 4.868.3 7.478 4.088 -0.0001 5.148.2 4.801 20.894 -0.0075 5.154.8 7.257 4.101 -0.0004 5.640.6 4.934 23.040 -0.0073 5.648.7 7.255 4.101 -0.0004 6.165.3 5.073 25.284 -0.0071 6.165.2 6.762 4.172 -0.0003 6.692.2 5.212 27.577 -0.0069 6.704.1 6.539 4.218 -0.0003 7.251.1 5.335 29.806 -0.0066 7.265.3 6.314 4.291 -0.0002 8.434.5 5.595 34.527 -0.0064 7.265.3 6.314 4.291 -0.0002 9.058.2 5.732 37.016 -0.0058 9.081.5 5.588 4.527 -0.0001 10.367.7 6.187 42.055 -0.0055 9.728.8 5.329 4.611 -0.0001 11.052.6 6.557 44.550 -0.0055 11.0399.1 5.105 4.620 0.0000 11.052.6 6.557 44.550 -0.0050 11.052.6 6.557 44.550 -0.0050 11.052.6 6.557 44.550 -0.0050 11.052.6 6.557 44.550 -0.0050 11.052.6 6.557 44.550 -0.0051 11.052.6 6.557 44.550									
3,030.6 3,805 12,019 -0.0036 3,033.3 0 7,818 3,582 0,0009 3,407.6 4,332 13,307 -0.0044 3,410.6 8,076 3,979 0,0007 3,807.9 4,440 15,052 -0.0052 3,811.6 7,888 4,007 0,0004 4,231.6 4,554 16,898 -0.0060 4,236.0 7,689 4,037 0,0002 4,678.4 4,675 18,846 -0.0069 4,683.8 7,478 4,068 -0.001 5,148.2 4,801 20,894 -0.0075 5,154.8 7,257 4,101 -0.0004 5,640.6 4,934 23,040 -0.0073 5,648.7 7,025 4,135 -0.0004 6,155.3 5,073 25,284 -0.0071 6,165.2 6,762 4,172 -0.0003 6,692.2 5,212 27,577 -0.0069 6,704.1 6,539 4,218 -0.0003 7,251.1 5,335 29,806 -0.0066 7,265.3 6,314 4,291 -0.0002 7,832.0 5,463 32,124 -0.0064 7,848.9 6,081 4,366 -0.0002 8,434.5 5,595 34,527 -0.0061 8,454.4 5,839 4,445 -0.0002 9,058.2 5,732 37,016 -0.0058 9,081.5 5,588 4,527 -0.0001 10,367.7 6,187 4,057 -0.0055 9,729.8 5,329 4,611 -0.0001 11,052.6 6,557 44,550 -0.0055 11,088.8 4,884 4,614 0,0000 11,757.2 6,938 47,114 -0.0047 11,798.8 4,657 4,600 0,0000 11,757.2 6,938 47,114 -0.0047 11,798.8 4,864 4,614 0,0000 11,757.2 6,938 47,114 -0.0047 11,798.8 4,865 4,422 0,0001 13,323.8 7,741 52,474 -0.0041 13,278.1 4,179 4,578 0,0000 13,323.8 7,741 52,474 -0.0041 13,278.1 4,179 4,578 0,0000 11,561.8 9,284 61,521 -0.0032 15,640.7 3,353 4,342 0,0001 14,764.5 8,758 5,844 0,0032 15,640.7 3,353 4,342 0,0001 14,764.5 8,758 5,844 0,033 15,897 1 0,0064 12,481.0 7,328 49,748 -0.0044 12,528.7 4,424 4,599 0,0000 11,561.8 9,284 61,521 -0.0032 15,640.7 3,353 4,342 0,0001 15,561.8 9,284 61,521 -0.0032 15,640.7 3,353 4,342 0,0001 15,561.8 9,284 61,521 -0.0032 15,640.7 3,353 4,342 0,0001 18,057.1 10,848 71,177 -0.0046 19,941.3 2,239 2,827 0,0001 19,803.6 11,915 77,917 -0.0046 19,941.3 2,239 2,827 0,0001 19,803.6 11,915 77,917 -0.0046 19,941.3 2,239 2,827 0,0001 19,803.6 11,915 77,917 -0.0046 19,941.3 2,239 2,827 0,0001 19,803.6 11,915 77,917 -0.0046 19,941.3 2,239 2,827 0,0001 19,803.6 11,915 77,917 -0.0046 19,941.3 2,239 2,827 0,0001 19,803.6 11,915 77,917 -0.0046 19,941.3 2,239 2,2371 0,0002 2,542.5 13,613 8,607 -0.0052 2,546.5 10,000 2,7403.6 15,648.5 10,000 2,7403.6 15,648.5 10,000 2,7403									
3,407.6									
3,807.9									
4,231.6         4,554         16,888         -0,0060         4,236.0         7,689         4,037         0,0002           4,678.4         4,675         18,846         -0,0069         4,683.8         7,478         4,068         -0,0001           5,148.2         4,801         20,894         -0,0073         5,648.7         7,025         4,135         -0,0004           6,155.3         5,073         25,284         -0,0071         6,165.2         6,782         4,172         -0,0003           6,692.2         5,212         27,577         -0,0066         7,265.3         6,314         4,291         -0,0003           7,251.1         5,335         29,806         -0,0066         7,265.3         6,314         4,291         -0,0002           8,434.5         5,595         34,527         -0,0061         8,454.4         5,839         4,445         -0,0002           9,058.2         5,732         37,016         -0,0058         9,081.5         5,588         4,527         -0,001           10,367.7         6,187         42,057         -0,0053         10,399.1         5,105         4,620         0,0000           11,757.2         6,938         47,114         -0,0041         12,528.7						· ·			
5,148.2         4,801         20,894         -0,0075         5,154.8         7,257         4,101         -0,0004           5,640.6         4,934         23,040         -0,0071         6,165.2         6,782         4,172         -0,0004           6,692.2         5,212         27,577         -0,0069         6,704.1         6,539         4,218         -0,0002           7,251.1         5,335         29,806         -0,0064         7,265.3         6,314         4,291         -0,0002           7,832.0         5,463         32,124         -0,0064         7,848.9         6,081         4,366         -0,0002           8,434.5         5,595         34,527         -0,0061         8,454.4         5,839         4,445         -0,0002           9,085.2         5,732         37,016         -0,0055         9,081.5         5,588         4,527         -0,0001           10,367.7         6,187         42,057         -0,0053         10,399.1         5,105         4,620         0,0000           11,052.6         6,557         44,550         -0,0044         11,798.8         4,657         4,606         0,0000           12,481.0         7,328         49,748         -0,0044         12,528.7	4	,231.6	4.554	16.898	-0.0060	4,236.0	7.689	4.037	0.0002
5,640.6         4,934         23,040         -0.0073         5,648.7         7,025         4,135         -0.0004           6,655.3         5,073         25,284         -0.0071         6,165.2         6,782         4,172         -0.0003           7,251.1         5,335         29,806         -0.0066         7,265.3         6,314         4,291         -0.0002           7,832.0         5,463         32,124         -0.0064         7,848.9         6,081         4,366         -0.0002           8,434.5         5,595         34,527         -0.0061         8,454.4         5,839         4,445         -0.0002           9,088.2         5,732         37,016         -0.0058         9,081.5         5,588         4,527         -0.0001           10,367.7         6,187         42,057         -0.0055         9,729.8         5,329         4,611         -0.0001           11,052.6         6,557         44,550         -0.0050         11,088.8         4,884         4,614         0.0000           11,275.2         6,938         47,114         -0.0041         12,528.7         4,424         4,599         0.0000           12,481.0         7,328         49,748         -0.0044         12,528.7	4	,678.4	4.675	18.846	-0.0069	4,683.8	7.478	4.068	-0.0001
6,155.3         5,073         25,224         -0,0071         6,165.2         6,782         4,172         -0,0003           6,692.2         5,212         27,577         -0,0069         6,704.1         6,539         4,218         -0,0003           7,251.1         5,335         29,806         -0,0064         7,265.3         6,314         4,291         -0,0002           8,434.5         5,595         34,527         -0,0061         8,454.4         5,839         6,0002           9,088.2         5,732         37,016         -0,0058         9,081.5         5,588         4,527         -0,0001           9,702.8         5,874         39,587         -0,0055         9,729.8         5,329         4,611         -0,0001           11,052.6         6,557         44,550         -0,0050         11,088.8         4,884         4,614         0,0000           11,757.2         6,938         47,114         -0,0047         11,798.8         4,657         4,606         0,0000           12,481.0         7,328         49,748         -0,0041         13,278.1         4,179         4,578         0,0000           13,985.0         8,243         55,419         -0,0032         14,646.8         3,910									
6,692.2         5,212         27,577         -0,0069         6,704.1         6,539         4,218         -0,0003           7,251.1         5,335         29,806         -0,0066         7,265.3         6,314         4,291         -0,0002           8,434.5         5,595         34,527         -0,0061         8,454.4         5,839         4,445         -0,0002           9,702.8         5,874         39,587         -0,0058         9,081.5         5,588         4,527         -0,0001           10,367.7         6,187         42,057         -0,0053         10,399.1         5,105         4,620         0,0000           11,052.6         6,557         44,550         -0,0050         11,088.8         4,884         4,614         0,0000           11,757.2         6,938         47,114         -0,0047         11,798.8         4,657         4,606         0,0000           13,223.8         7,741         52,474         -0,0041         13,278.1         4,179         4,578         0,0000           14,764.5         8,758         56,436         -0,0035         14,834.4         3,635         4,422         0,0001           15,561.8         9,224         61,521         -0,0032         15,640.7 <th></th> <td></td> <td></td> <td></td> <td></td> <td></td> <td></td> <td></td> <td></td>									
7,251.1         5.335         29.806         -0.0066         7,265.3         6.314         4.291         -0.0002           7,832.0         5.463         32.124         -0.0064         7,848.9         6.081         4.366         -0.0002           8,434.5         5.595         34.527         -0.0058         9,081.5         5.588         4.445         -0.0001           9,702.8         5.874         39.587         -0.0055         9,729.8         5.329         4.611         -0.0001           10,367.7         6.187         42.057         -0.0053         10,399.1         5.105         4.620         0.0000           11,052.6         6.557         44.550         -0.0050         11,088.8         4.884         4.614         0.0000           12,481.0         7.328         49.748         -0.0047         11,798.8         4.657         4.606         0.0000           13,283.6         7.741         52.474         -0.0041         13,278.1         4.179         4.578         0.0000           14,764.5         8.758         58.436         -0.0032         15,640.7         3.353         4.342         0.0001           15,561.8         9.822         64.674         -0.0029         16,465.3 <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th>									
7,832.0         5.463         32.124         -0.0064         7,848.9         6.081         4.366         -0.0002           8,434.5         5.595         34.527         -0.0061         8,454.4         5.839         4.445         -0.0002           9,058.2         5.732         37.016         -0.0055         9,729.8         5.329         4.611         -0.0001           10,367.7         6.187         42.057         -0.0055         10,399.1         5.105         4.620         0.0000           11,052.6         6.557         44.550         -0.0050         11,088.8         4.884         4.614         0.0000           11,757.2         6.938         47.114         -0.0047         11,798.8         4.657         4.606         0.0000           12,481.0         7.328         49.748         -0.0044         12,528.7         4.424         4.599         0.0000           13,985.0         8.243         55.419         -0.0038         14,046.8         3.910         4.501         0.0001           14,764.5         8.758         58.436         -0.0032         15,640.7         3.353         4.322         0.0001           15,561.8         9.284         61.521         -0.0032         15,640.7 <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th>									
8,434.5         5.595         34.527         -0.0061         8,454.4         5.839         4.445         -0.0002           9,058.2         5.732         37.016         -0.0055         9,729.8         5.588         4.527         -0.0001           10,367.7         6.187         42.057         -0.0053         10,399.1         5.105         4.620         0.0000           11,052.6         6.557         44.550         -0.0050         11,088.8         4.884         4.614         0.0000           11,757.2         6.938         47.114         -0.0047         11,798.8         4.657         4.606         0.0000           12,481.0         7.328         49.748         -0.0044         12,528.7         4.424         4.599         0.0000           13,223.8         7.741         52.474         -0.0041         13,278.1         4.179         4.578         0.0000           14,764.5         8.758         58.436         -0.0035         14,834.4         3.635         4.422         0.0001           15,561.8         9.284         61.521         -0.0032         15,640.7         3.353         4.342         0.0001           16,376.5         9.822         64.674         -0.0029         16.455.3 </th <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th> <th></th>									
9,058.2 5.732 37.016 -0.0058 9,081.5 5.588 4.527 -0.0001 9,702.8 5.874 39.587 -0.0055 9,729.8 5.329 4.611 -0.0001 10,367.7 6.187 42.057 -0.0053 10,399.1 5.105 4.620 0.0000 11,052.6 6.557 44.550 -0.0050 11,088.8 4.884 4.614 0.0000 11,757.2 6.938 47.114 -0.0047 11,798.8 4.657 4.606 0.0000 12,481.0 7.328 49.748 -0.0044 12,528.7 4.424 4.599 0.0000 13,223.8 7.741 52.474 -0.0041 13,278.1 4.179 4.578 0.0000 13,395.0 8.243 55.419 -0.0038 14,046.8 3.910 4.501 0.0001 14,764.5 8.758 58.436 -0.0035 14,834.4 3.635 4.422 0.0001 16,376.5 9.822 64.674 -0.0029 16,465.3 3.071 4.242 0.0001 17,208.4 10.331 67.893 -0.0026 17,307.9 2.886 3.905 0.0001 18,922.2 11.376 74.525 -0.0019 19,046.1 2.504 3.210 0.0001 19,803.6 11.915 77.917 -0.0046 19,941.3 2.239 2.827 0.0001 20,700.9 12,471 81.246 -0.0077 20,853.4 1.792 2.371 0.0002 21,614.0 13.037 84.633 -0.0109 21,782.4 1.337 1.906 0.0002 22,542.5 13,613 88.077 -0.0141 22,728.1 0.873 1.434 0.0003 22,444.7 14.364 95.222 -0.0207 24,667.8 -0.145 0.362 0.0003 23,486.2 14.048 91.609 -0.0174 23,690.0 0.373 0.910 0.0003 24,444.7 14.364 95.222 -0.0207 24,667.8 -0.145 0.362 0.0003 25,417.3 14.685 98.889 -0.0214 25,660.9 -0.671 -0.194 0.0004 26,403.6 15.188 106.354 -0.0228 27,691.9 -1.805 -1.533 0.0008 29,443.9 15.535 114.004 -0.0223 37,691.9 -1.805 -1.533 0.0008 29,443.9 15.535 114.004 -0.0223 27,691.9 -1.202 -0.817 0.0006 27,403.6 15.188 106.354 -0.0228 28,729.4 -2.396 -2.259 0.0010 30,484.7 15.485 118.021 -0.0228 28,729.4 -2.396 -2.259 0.0010 31,695.1 14.489 15.296 68 -0.0169 33,033.4 -5.140 -5.412 0.0014 33,694.4 14.952 129.658 -0.0169 33,033.4 -5.140 -5.412 0.0014 33,694.4 14.952 129.658 -0.0169 33,033.4 -5.140 -5.412 0.0014 33,694.4 14.952 129.658 -0.0169 33,033.4 -5.140 -5.412 0.0014 33,694.4 14.952 129.658 -0.0169 33,033.4 -5.140 -5.412 0.0014 33,694.4 14.952 129.658 -0.0169 33,033.4 -5.140 -5.412 0.0014 33,694.4 14.952 129.658 -0.0169 33,033.4 -5.140 -5.412 0.0014 33,694.4 14.952 129.658 -0.0169 34,150.1 -5.922 -6.238 0.0013 34,795.1 14.489 32.938 -0.0132 35,284.0 -6.716 -7.077 0.0012									
9,702.8 5.874 39.587 -0.0055 9,729.8 5.329 4.611 -0.0001 10,367.7 6.187 42.057 -0.0053 10,399.1 5.105 4.620 0.0000 11,052.6 6.557 44.550 -0.0050 11,088.8 4.884 4.614 0.0000 11,757.2 6.938 47.114 -0.0047 11,798.8 4.657 4.606 0.0000 12,481.0 7.328 49,748 -0.0044 12,528.7 4.424 4.599 0.0000 13,223.8 7.741 52.474 -0.0041 13,278.1 4.179 4.578 0.0000 13,263.8 7.741 52.474 -0.0041 13,278.1 4.179 4.578 0.0000 14,764.5 8.758 58.436 -0.0035 14,834.4 3.635 4.422 0.0001 14,764.5 8.758 58.436 -0.0035 14,834.4 3.635 4.422 0.0001 15,561.8 9.284 61.521 -0.0032 15,640.7 3.353 4.342 0.0001 16,376.5 9.822 64.674 -0.0029 16,465.3 3.071 4.242 0.0001 17,208.4 10.331 67.893 -0.0026 17,307.9 2.886 3.905 0.0001 18,057.1 10.848 71.177 -0.0022 18,168.3 2.697 3.561 0.0001 18,057.1 10.848 71.177 -0.0022 18,168.3 2.697 3.561 0.0001 19,803.6 11.915 77.917 -0.0046 19,941.3 2.239 2.827 0.0001 19,803.6 11.915 77.917 -0.0046 19,941.3 2.239 2.827 0.0001 20,700.9 12.471 81.246 -0.0077 20,853.4 1.792 2.371 0.0002 21,614.0 13.037 84.633 -0.0109 21,782.4 1.337 1.906 0.0002 22,542.5 13.613 88.077 -0.0141 22,728.1 0.873 1.434 0.0003 24,444.7 14.364 95.222 -0.0207 24,667.8 -0.145 0.362 0.0003 24,444.7 14.364 95.222 -0.0207 24,667.8 -0.145 0.362 0.0003 24,444.7 14.364 95.222 -0.0207 24,667.8 -0.145 0.362 0.0003 24,444.7 14.364 95.222 -0.0207 24,667.8 -0.145 0.362 0.0003 25,417.3 14.685 98.889 -0.0214 25,660.9 -0.671 -0.194 0.0004 26,403.6 14.988 102.604 -0.0228 28,729.4 -2.396 -2.259 0.0010 29,443.9 15.535 114.004 -0.0228 28,729.4 -2.396 -2.259 0.0010 29,443.9 15.535 114.004 -0.0231 31,933.5 -4.415 -4.598 0.0015 33,699.4 14.952 129,658 -0.0165 34,150.1 -5.922 -6.238 0.0013 31,539.7 15.411 122.104 -0.0231 31,933.5 -4.415 -4.598 0.0015 33,699.4 14.952 129,658 -0.0165 34,150.1 -5.922 -6.238 0.0013 34,795.1 14.490 132,938 -0.0132 33,693.6 -7.614 -7.905 0.0010									
11,052.6         6.557         44.550         -0.0050         11,088.8         4.884         4.614         0.0000           11,757.2         6.938         47.114         -0.0047         11,798.8         4.657         4.606         0.0000           12,481.0         7.328         49.748         -0.0044         12,528.7         4.424         4.599         0.0000           13,223.8         7.741         52.474         -0.0041         13,278.1         4.179         4.578         0.0000           13,985.0         8.243         55.419         -0.0038         14,046.8         3.910         4.501         0.0001           14,764.5         8.758         58.436         -0.0032         15,640.7         3.353         4.342         0.0001           15,561.8         9.284         61.521         -0.0032         16,465.3         3.071         4.242         0.0001           17,208.4         10.331         67.893         -0.0026         17,307.9         2.886         3.905         0.0001           18,957.1         10.848         71.177         -0.0022         18,168.3         2.697         3.561         0.0001           18,9657.1         10.848         71.177         -0.0046         19,9									
11,757.2       6.938       47.114       -0.0047       11,798.8       4.657       4.606       0.0000         12,481.0       7.328       49.748       -0.0044       12,528.7       4.424       4.599       0.0000         13,223.8       7.741       52.474       -0.0041       13,278.1       4.179       4.578       0.0000         13,985.0       8.243       55.419       -0.0038       14,046.8       3.910       4.501       0.0001         14,764.5       8.758       58.436       -0.0035       14,834.4       3.635       4.422       0.0001         15,561.8       9.284       61.521       -0.0032       15,640.7       3.353       4.342       0.0001         16,376.5       9.822       64.674       -0.0029       16,465.3       3.071       4.242       0.0001         17,208.4       10.331       67.893       -0.0026       17,307.9       2.886       3.905       0.0001         18,957.1       10.848       71.177       -0.0022       18,168.3       2.697       3.561       0.0001         19,803.6       11.915       77.917       -0.0046       19,941.3       2.239       2.827       0.0001         20,601.6       12.471	10	),367.7	6.187	42.057	-0.0053	10,399.1	5.105	4.620	0.0000
12,481.0       7.328       49.748       -0.0044       12,528.7       4.424       4.599       0.0000         13,223.8       7.741       52,474       -0.0041       13,278.1       4.179       4.578       0.0000         13,985.0       8.243       55.419       -0.0038       14,046.8       3.910       4.501       0.0001         14,764.5       8.758       58.436       -0.0035       14,834.4       3.635       4.422       0.0001         15,561.8       9.284       61.521       -0.0032       15,640.7       3.353       4.342       0.0001         16,376.5       9.822       64.674       -0.0029       16,465.3       3.071       4.242       0.0001         17,208.4       10.331       67.893       -0.0026       17,307.9       2.886       3.905       0.0001         18,957.1       10.848       71.177       -0.0022       18,168.3       2.697       3.561       0.0001         19,803.6       11.915       77.917       -0.0046       19,941.3       2.239       2.827       0.0001         20,700.9       12.471       81.246       -0.0077       20,853.4       1.792       2.371       0.0002         21,614.0       13.037			6.557			11,088.8			0.0000
13,223.8         7.741         52.474         -0.0041         13,278.1         4.179         4.578         0.0000           13,985.0         8.243         55.419         -0.0038         14,046.8         3.910         4.501         0.0001           14,764.5         8.758         58.436         -0.0035         14,834.4         3.635         4.422         0.0001           15,561.8         9.284         61.521         -0.0032         15,640.7         3.353         4.342         0.0001           16,376.5         9.822         64.674         -0.0029         16,465.3         3.071         4.242         0.0001           17,208.4         10.331         67.893         -0.0026         17,307.9         2.886         3.905         0.0001           18,957.1         10.848         71.177         -0.0022         18,168.3         2.697         3.561         0.0001           18,922.2         11.376         74.525         -0.0019         19,046.1         2.504         3.210         0.0001           19,803.6         11,915         77.917         -0.0046         19,941.3         2.239         2.827         0.0001           20,700.9         12,471         81.246         -0.0077         20,									
13,985.0         8.243         55.419         -0.0038         14,046.8         3.910         4.501         0.0001           14,764.5         8.758         58.436         -0.0035         14,834.4         3.635         4.422         0.0001           15,561.8         9.284         61.521         -0.0032         15,640.7         3.353         4.342         0.0001           16,376.5         9.822         64.674         -0.0029         16,465.3         3.071         4.242         0.0001           17,208.4         10.331         67.893         -0.0026         17,307.9         2.886         3.905         0.0001           18,057.1         10.848         71.177         -0.0022         18,168.3         2.697         3.561         0.0001           19,803.6         11.915         77.917         -0.0046         19,941.3         2.239         2.827         0.0001           20,700.9         12.471         81.246         -0.0077         20,853.4         1.792         2.371         0.0002           21,614.0         13.037         84.633         -0.0109         21,782.4         1.337         1.906         0.0002           22,542.5         13.613         88.077         -0.0141         22									
14,764.5       8.758       58.436       -0.0035       14,834.4       3.635       4.422       0.0001         15,561.8       9.284       61.521       -0.0032       15,640.7       3.353       4.342       0.0001         16,376.5       9.822       64.674       -0.0029       16,465.3       3.071       4.242       0.0001         17,208.4       10.331       67.893       -0.0026       17,307.9       2.886       3.905       0.0001         18,057.1       10.848       71.177       -0.0022       18,168.3       2.697       3.561       0.0001         18,922.2       11.376       74.525       -0.0019       19,046.1       2.504       3.210       0.0001         19,803.6       11.915       77.917       -0.0046       19,941.3       2.239       2.827       0.0001         20,700.9       12.471       81.246       -0.0077       20,853.4       1.792       2.371       0.0002         21,614.0       13.037       84.633       -0.0109       21,782.4       1.337       1.906       0.0002         22,542.5       13.613       88.077       -0.0141       22,728.1       0.873       1.434       0.0003         23,486.2       14.048									
15,561.8         9.284         61.521         -0.0032         15,640.7         3.353         4.342         0.0001           16,376.5         9.822         64.674         -0.0029         16,465.3         3.071         4.242         0.0001           17,208.4         10.331         67.893         -0.0026         17,307.9         2.886         3.905         0.0001           18,057.1         10.848         71.177         -0.0022         18,168.3         2.697         3.561         0.0001           18,922.2         11.376         74.525         -0.0019         19,046.1         2.504         3.210         0.0001           19,803.6         11.915         77.917         -0.0046         19,941.3         2.239         2.827         0.0001           20,700.9         12.471         81.246         -0.0077         20,853.4         1.792         2.371         0.0002           21,614.0         13.037         84.633         -0.0109         21,782.4         1.337         1.906         0.0002           22,542.5         13.613         88.077         -0.0141         22,728.1         0.873         1.434         0.0003           23,486.2         14.048         91.609         -0.0174									
16,376.5         9.822         64.674         -0.0029         16,465.3         3.071         4.242         0.0001           17,208.4         10.331         67.893         -0.0026         17,307.9         2.886         3.905         0.0001           18,057.1         10.848         71.177         -0.0022         18,168.3         2.697         3.561         0.0001           18,922.2         11.376         74.525         -0.0019         19,046.1         2.504         3.210         0.0001           19,803.6         11.915         77.917         -0.0046         19,941.3         2.239         2.827         0.0001           20,700.9         12.471         81.246         -0.0077         20,853.4         1.792         2.371         0.0002           21,614.0         13.037         84.633         -0.0109         21,782.4         1.337         1.906         0.0002           22,542.5         13.613         88.077         -0.0141         22,728.1         0.873         1.434         0.0003           24,444.7         14.364         95.222         -0.0207         24,667.8         -0.145         0.362         0.0003           25,417.3         14.685         98.889         -0.0214 <t< th=""><th></th><td></td><td></td><td></td><td></td><td></td><td></td><td></td><td></td></t<>									
17,208.4         10.331         67.893         -0.0026         17,307.9         2.886         3.905         0.0001           18,057.1         10.848         71.177         -0.0022         18,168.3         2.697         3.561         0.0001           18,922.2         11.376         74.525         -0.0019         19,046.1         2.504         3.210         0.0001           19,803.6         11.915         77.917         -0.0046         19,941.3         2.239         2.827         0.0001           20,700.9         12.471         81.246         -0.0077         20,853.4         1.792         2.371         0.0002           21,614.0         13.037         84.633         -0.0109         21,782.4         1.337         1.906         0.0002           22,542.5         13.613         88.077         -0.0141         22,728.1         0.873         1.434         0.0003           23,486.2         14.048         91.609         -0.0174         23,690.0         0.373         0.910         0.0003           24,444.7         14.364         95.222         -0.0207         24,667.8         -0.145         0.362         0.0003           25,417.3         14.685         98.889         -0.0214         <									
18,057.1         10.848         71.177         -0.0022         18,168.3         2.697         3.561         0.0001           18,922.2         11.376         74.525         -0.0019         19,046.1         2.504         3.210         0.0001           19,803.6         11.915         77.917         -0.0046         19,941.3         2.239         2.827         0.0001           20,700.9         12.471         81.246         -0.0077         20,853.4         1.792         2.371         0.0002           21,614.0         13.037         84.633         -0.0109         21,782.4         1.337         1.906         0.0002           22,542.5         13.613         88.077         -0.0141         22,728.1         0.873         1.434         0.0003           23,486.2         14.048         91.609         -0.0174         23,690.0         0.373         0.910         0.0003           24,444.7         14.364         95.222         -0.0207         24,667.8         -0.145         0.362         0.0003           25,417.3         14.685         98.889         -0.0214         25,660.9         -0.671         -0.194         0.0004           26,403.6         14.988         102.604         -0.0219									
19,803.6         11.915         77.917         -0.0046         19,941.3         2.239         2.827         0.0001           20,700.9         12.471         81.246         -0.0077         20,853.4         1.792         2.371         0.0002           21,614.0         13.037         84.633         -0.0109         21,782.4         1.337         1.906         0.0002           22,542.5         13.613         88.077         -0.0141         22,728.1         0.873         1.434         0.0003           23,486.2         14.048         91.609         -0.0174         23,690.0         0.373         0.910         0.0003           24,444.7         14.364         95.222         -0.0207         24,667.8         -0.145         0.362         0.0003           25,417.3         14.685         98.889         -0.0214         25,660.9         -0.671         -0.194         0.0004           26,403.6         14.988         102.604         -0.0219         26,669.9         -1.222         -0.817         0.0006           27,403.6         15.168         106.354         -0.0223         27,691.9         -1.805         -1.533         0.0008           28,417.0         15.355         114.004         -0.0233				71.177					
20,700.9         12,471         81.246         -0.0077         20,853.4         1.792         2.371         0.0002           21,614.0         13.037         84.633         -0.0109         21,782.4         1.337         1.906         0.0002           22,542.5         13.613         88.077         -0.0141         22,728.1         0.873         1.434         0.0003           23,486.2         14.048         91.609         -0.0174         23,690.0         0.373         0.910         0.0003           24,444.7         14.364         95.222         -0.0207         24,667.8         -0.145         0.362         0.0003           25,417.3         14.685         98.889         -0.0214         25,660.9         -0.671         -0.194         0.0004           26,403.6         14.988         102.604         -0.0219         26,669.0         -1.222         -0.817         0.0006           27,403.6         15.168         106.354         -0.0223         27,691.9         -1.805         -1.533         0.0008           28,417.0         15.350         110.154         -0.0228         28,729.4         -2.396         -2.259         0.0010           30,484.7         15.485         118.021         -0.0233 <th></th> <td></td> <td></td> <td>_</td> <td></td> <td></td> <td></td> <td></td> <td></td>				_					
21,614.0       13.037       84.633       -0.0109       21,782.4       1.337       1.906       0.0002         22,542.5       13.613       88.077       -0.0141       22,728.1       0.873       1.434       0.0003         23,486.2       14.048       91.609       -0.0174       23,690.0       0.373       0.910       0.0003         24,444.7       14.364       95.222       -0.0207       24,667.8       -0.145       0.362       0.0003         25,417.3       14.685       98.889       -0.0214       25,660.9       -0.671       -0.194       0.0004         26,403.6       14.988       102.604       -0.0219       26,669.0       -1.222       -0.817       0.0006         27,403.6       15.168       106.354       -0.0223       27,691.9       -1.805       -1.533       0.0008         28,417.0       15.350       110.154       -0.0228       28,729.4       -2.396       -2.259       0.0010         29,443.9       15.535       114.004       -0.0233       29,781.9       -3.016       -3.006       0.0012         30,484.7       15.485       118.021       -0.0238       30,849.8       -3.710       -3.796       0.0013         31,539.7									
22,542.5       13.613       88.077       -0.0141       22,728.1       0.873       1.434       0.0003         23,486.2       14.048       91.609       -0.0174       23,690.0       0.373       0.910       0.0003         24,444.7       14.364       95.222       -0.0207       24,667.8       -0.145       0.362       0.0003         25,417.3       14.685       98.889       -0.0214       25,660.9       -0.671       -0.194       0.0004         26,403.6       14.988       102.604       -0.0219       26,669.0       -1.222       -0.817       0.0006         27,403.6       15.168       106.354       -0.0223       27,691.9       -1.805       -1.533       0.0008         28,417.0       15.350       110.154       -0.0228       28,729.4       -2.396       -2.259       0.0010         29,443.9       15.535       114.004       -0.0233       29,781.9       -3.016       -3.006       0.0012         30,484.7       15.485       118.021       -0.0238       30,849.8       -3.710       -3.796       0.0013         31,539.7       15.411       122.104       -0.0231       31,933.5       -4.415       -4.598       0.0015         32,609.5									
23,486.2       14.048       91.609       -0.0174       23,690.0       0.373       0.910       0.0003         24,444.7       14.364       95.222       -0.0207       24,667.8       -0.145       0.362       0.0003         25,417.3       14.685       98.889       -0.0214       25,660.9       -0.671       -0.194       0.0004         26,403.6       14.988       102.604       -0.0219       26,669.0       -1.222       -0.817       0.0006         27,403.6       15.168       106.354       -0.0223       27,691.9       -1.805       -1.533       0.0008         28,417.0       15.350       110.154       -0.0228       28,729.4       -2.396       -2.259       0.0010         29,443.9       15.535       114.004       -0.0233       29,781.9       -3.016       -3.006       0.0012         30,484.7       15.485       118.021       -0.0238       30,849.8       -3.710       -3.796       0.0013         31,539.7       15.411       122.104       -0.0231       31,933.5       -4.415       -4.598       0.0015         32,609.5       15.336       126.243       -0.0199       33,033.4       -5.140       -5.412       0.0014         33,694.4 <th></th> <td></td> <td></td> <td></td> <td></td> <td></td> <td></td> <td></td> <td></td>									
24,444.7       14.364       95.222       -0.0207       24,667.8       -0.145       0.362       0.0003         25,417.3       14.685       98.889       -0.0214       25,660.9       -0.671       -0.194       0.0004         26,403.6       14.988       102.604       -0.0219       26,669.0       -1.222       -0.817       0.0006         27,403.6       15.168       106.354       -0.0223       27,691.9       -1.805       -1.533       0.0008         28,417.0       15.350       110.154       -0.0228       28,729.4       -2.396       -2.259       0.0010         29,443.9       15.535       114.004       -0.0233       29,781.9       -3.016       -3.006       0.0012         30,484.7       15.485       118.021       -0.0238       30,849.8       -3.710       -3.796       0.0013         31,539.7       15.411       122.104       -0.0231       31,933.5       -4.415       -4.598       0.0015         32,609.5       15.336       126.243       -0.0199       33,033.4       -5.140       -5.412       0.0014         33,694.4       14.952       129.658       -0.0165       34,150.1       -5.922       -6.238       0.0013         34,795.1									
25,417.3         14.685         98.889         -0.0214         25,660.9         -0.671         -0.194         0.0004           26,403.6         14.988         102.604         -0.0219         26,669.0         -1.222         -0.817         0.0006           27,403.6         15.168         106.354         -0.0223         27,691.9         -1.805         -1.533         0.0008           28,417.0         15.350         110.154         -0.0228         28,729.4         -2.396         -2.259         0.0010           29,443.9         15.535         114.004         -0.0233         29,781.9         -3.016         -3.006         0.0012           30,484.7         15.485         118.021         -0.0238         30,849.8         -3.710         -3.796         0.0013           31,539.7         15.411         122.104         -0.0231         31,933.5         -4.415         -4.598         0.0015           32,609.5         15.336         126.243         -0.0199         33,033.4         -5.140         -5.412         0.0014           33,694.4         14.952         129.658         -0.0165         34,150.1         -5.922         -6.238         0.0013           34,795.1         14.489         132.938									
26,403.6         14.988         102.604         -0.0219         26,669.0         -1.222         -0.817         0.0006           27,403.6         15.168         106.354         -0.0223         27,691.9         -1.805         -1.533         0.0008           28,417.0         15.350         110.154         -0.0228         28,729.4         -2.396         -2.259         0.0010           29,443.9         15.535         114.004         -0.0233         29,781.9         -3.016         -3.006         0.0012           30,484.7         15.485         118.021         -0.0238         30,849.8         -3.710         -3.796         0.0013           31,539.7         15.411         122.104         -0.0231         31,933.5         -4.415         -4.598         0.0015           32,609.5         15.336         126.243         -0.0199         33,033.4         -5.140         -5.412         0.0014           33,694.4         14.952         129.658         -0.0165         34,150.1         -5.922         -6.238         0.0013           34,795.1         14.489         132.938         -0.0132         35,284.0         -6.716         -7.077         0.0012           35,912.0         14.020         136.266									
27,403.6     15.168     106.354     -0.0223     27,691.9     -1.805     -1.533     0.0008       28,417.0     15.350     110.154     -0.0228     28,729.4     -2.396     -2.259     0.0010       29,443.9     15.535     114.004     -0.0233     29,781.9     -3.016     -3.006     0.0012       30,484.7     15.485     118.021     -0.0238     30,849.8     -3.710     -3.796     0.0013       31,539.7     15.411     122.104     -0.0231     31,933.5     -4.415     -4.598     0.0015       32,609.5     15.336     126.243     -0.0199     33,033.4     -5.140     -5.412     0.0014       33,694.4     14.952     129.658     -0.0165     34,150.1     -5.922     -6.238     0.0013       34,795.1     14.489     132.938     -0.0132     35,284.0     -6.716     -7.077     0.0012       35,912.0     14.020     136.266     -0.0104     36,435.6     -7.614     -7.905     0.0010									
28,417.0     15.350     110.154     -0.0228     28,729.4     -2.396     -2.259     0.0010       29,443.9     15.535     114.004     -0.0233     29,781.9     -3.016     -3.006     0.0012       30,484.7     15.485     118.021     -0.0238     30,849.8     -3.710     -3.796     0.0013       31,539.7     15.411     122.104     -0.0231     31,933.5     -4.415     -4.598     0.0015       32,609.5     15.336     126.243     -0.0199     33,033.4     -5.140     -5.412     0.0014       33,694.4     14.952     129.658     -0.0165     34,150.1     -5.922     -6.238     0.0013       34,795.1     14.489     132.938     -0.0132     35,284.0     -6.716     -7.077     0.0012       35,912.0     14.020     136.266     -0.0104     36,435.6     -7.614     -7.905     0.0010									
29,443.9     15.535     114.004     -0.0233     29,781.9     -3.016     -3.006     0.0012       30,484.7     15.485     118.021     -0.0238     30,849.8     -3.710     -3.796     0.0013       31,539.7     15.411     122.104     -0.0231     31,933.5     -4.415     -4.598     0.0015       32,609.5     15.336     126.243     -0.0199     33,033.4     -5.140     -5.412     0.0014       33,694.4     14.952     129.668     -0.0165     34,150.1     -5.922     -6.238     0.0013       34,795.1     14.489     132.938     -0.0132     35,284.0     -6.716     -7.077     0.0012       35,912.0     14.020     136.266     -0.0104     36,435.6     -7.614     -7.905     0.0010									
31,539.7     15.411     122.104     -0.0231     31,933.5     -4.415     -4.598     0.0015       32,609.5     15.336     126.243     -0.0199     33,033.4     -5.140     -5.412     0.0014       33,694.4     14.952     129.658     -0.0165     34,150.1     -5.922     -6.238     0.0013       34,795.1     14.489     132.938     -0.0132     35,284.0     -6.716     -7.077     0.0012       35,912.0     14.020     136.266     -0.0104     36,435.6     -7.614     -7.905     0.0010									
32,609.5     15.336     126.243     -0.0199     33,033.4     -5.140     -5.412     0.0014       33,694.4     14.952     129.658     -0.0165     34,150.1     -5.922     -6.238     0.0013       34,795.1     14.489     132.938     -0.0132     35,284.0     -6.716     -7.077     0.0012       35,912.0     14.020     136.266     -0.0104     36,435.6     -7.614     -7.905     0.0010									
33,694.4     14.952     129.658     -0.0165     34,150.1     -5.922     -6.238     0.0013       34,795.1     14.489     132.938     -0.0132     35,284.0     -6.716     -7.077     0.0012       35,912.0     14.020     136.266     -0.0104     36,435.6     -7.614     -7.905     0.0010		,							
34,795.1     14.489     132.938     -0.0132     35,284.0     -6.716     -7.077     0.0012       35,912.0     14.020     136.266     -0.0104     36,435.6     -7.614     -7.905     0.0010									
35,912.0 14.020 136.266 -0.0104 36,435.6 -7.614 -7.905 0.0010									
01,000.2 14.140 105.454 1000.01 31,000.2 -0.140 -0.089 0.0000		,							
38,196.1 14.385 142.652 -0.0057 38,793.4 -9.901 -9.485 0.0002									

Table 5.3.2-12: Mean GRAM Wind Profiles (Continued)

	Mean F	ebruary GF	RAM Wind F	Profile	Mea	n July GRA	M Wind Pro	file
	Altitudo	North	East	Down		North	East	Down
	Altitude	Component	Component	Component	Altitude	Component	Component	Component
	(ft)	(fps)	(fps)	(fps)	(ft)	(fps)	(fps)	(fps)
	39,364.1	14.630	145.899	-0.0033	40,000.4	-11.078	-10.337	-0.0002
	40,550.0	14.633	145.603	-0.0016	41,226.6	-12.279	-11.245	-0.0007
	41,754.2	14.633	145.266	-0.0009	42,472.3	-13.500	-12.166	-0.0011
1	42,976.9	14.485	143.980	-0.0002	43,737.8	-14.200	-12.973	-0.0015
1	44,218.7	13.914	139.969	0.0005	45,023.3	-14.817	-13.770	-0.0019
1	45,479.9	13.334	135.896	0.0012	46,329.1	-15.010	-14.486	-0.0023
1	46,760.8	12.737	130.755	0.0014	47,655.4	-14.174	-14.990	-0.0026
1	48,061.8	12.125 11.516	124.966	0.0014	49,002.3	-13.326 -11.831	-15.502	-0.0027
1	49,383.2 50,725.1	10.979	118.970 112.047	0.0013 0.0012	50,370.2 51,759.0	-10.193	-16.067 -16.651	-0.0027 -0.0028
1	52,087.8	10.434	105.016	0.0012	53,168.9	-8.696	-17.610	-0.0020
1	53,471.5	9.570	97.352	0.0011	54,600.1	-7.365	-18.998	-0.0030
1	54,876.3	8.558	89.345	0.0011	56,052.6	-6.201	-20.684	-0.0028
1	56,302.4	7.527	81.087	0.0010	57,526.6	-5.848	-23.617	-0.0024
1	57,749.9	6.470	72.475	0.0010	59,022.1	-5.489	-26.593	-0.0021
1	59,219.0	5.449	63.705	0.0010	60,539.3	-4.829	-31.000	-0.0017
1	60,709.7	4.853	54.553	0.0009	62,078.1	-4.152	-35.508	-0.0013
1	62,221.9	4.248	45.268	0.0008	63,638.4	-2.728	-39.756	-0.0012
1	63,755.2	3.353	38.070	0.0008	65,219.9	-1.131	-43.994	-0.0011
1	65,309.5	2.420	30.968	0.0007	66,822.2	0.595	-47.270	-0.0009
1	66,884.4	1.311	25.963	0.0006	68,445.0	2.380	-50.239	-0.0008
1	68,479.7	0.147	21.448	0.0006	70,088.2	3.384	-51.839	-0.0007
1	70,095.0	-0.604	18.590	0.0006	71,751.2	4.082	-52.904	-0.0005
1	71,729.9	-1.209	16.317	0.0006	73,433.8	3.953	-53.968	-0.0004
1	73,384.3	-1.426	14.832	0.0005	75,135.6	3.528	-55.040	-0.0003
1	75,057.5	-1.493	13.644	0.0005	76,856.2	2.515	-56.207	-0.0003
1	76,749.0	-1.317	12.905	0.0005	78,595.1	1.350	-57.407	-0.0003
1	78,458.3 80,184.9	-1.061 -0.486	12.306 12.176	0.0004 0.0003	80,351.7 82,125.6	0.237 -0.859	-58.587 -59.773	-0.0003 -0.0005
1	81,928.7	0.159	12.170	0.0003	83,916.7	-1.594	-60.919	-0.0003
1	83,689.6	0.133	12.141	0.0003	85,724.9	-2.265	-62.138	-0.0004
1	85,467.5	1.497	13.930	0.0001	87,550.2	-2.703	-63.580	-0.0002
1	87,262.1	2.107	15.653	0.0001	89,392.2	-2.969	-65.331	0.0000
1	89,073.1	2.809	17.406	0.0000	91,250.6	-3.006	-67.487	0.0001
1	90,899.9	3.759	19.214	-0.0001	93,125.1	-3.031	-69.586	0.0003
1	92,742.4	4.752	22.003	-0.0001	95,015.2	-3.050	-71.665	0.0005
1	94,600.0	5.792	25.904	-0.0002	96,920.7	-2.150	-73.690	0.0006
1	96,472.6	6.920	30.034	-0.0003	98,841.3	-0.959	-75.966	0.0008
1	98,359.9	8.090	34.281	-0.0004	100,776.7	1.034	-79.159	0.0008
1	100,261.4	8.365	38.641	-0.0004	102,726.6	3.042	-82.376	0.0008
1	102,176.7	8.604	43.037	-0.0004	104,690.5	5.065	-85.617	0.0008
1	104,105.4	8.845	47.463	-0.0003	106,668.0	4.537	-86.757 97.513	0.0009
1	106,047.0 108,000.8	8.164 6.689	50.290 51.746	-0.0003 -0.0002	108,658.5 110,661.6	3.531 2.520	-87.513 -88.275	0.0009 0.0010
1	109,966.5	5.205	53.210	-0.0002	112,676.7	1.776	-89.834	0.0010
1	111,943.5	3.623	54.002	0.0000	114,703.4	1.776	-09.03 <del>4</del> -92.043	0.0010
1	113,931.7	1.665	51.964	0.0001	116,741.5	0.719	-94.265	0.0011
1	115,930.8	-0.304	49.915	0.0002	118,790.7	0.378	-96.830	0.0010
	117,940.2	-2.283	47.855	0.0001	120,850.5	0.429	-100.095	0.0009
	119,958.5	-1.762	45.428	0.0000	122,919.4	0.481	-103.374	0.0009
	121,983.7	-0.993	42.957	0.0000	124,995.5	0.503	-106.716	0.0008
	124,012.6	-0.222	40.482	-0.0001	127,075.6	0.357	-110.356	0.0007
	126,041.7	0.733	38.380	-0.0002	129,156.1	0.212	-113.997	0.0007
1	128,068.0	1.777	36.466	-0.0003	131,234.1	0.066	-117.634	0.0006
	130,089.4	2.818	34.555	-0.0004	133,307.6	-1.957	-122.181	0.0005
	132,104.1	4.453	33.112	-0.0006	135,374.7	-3.983	-126.718	0.0003
	134,110.6	6.881	32.289	-0.0006	137,433.9	-6.001	-131.238	0.0001
	136,107.3	9.297	31.470	-0.0007	139,483.7	-4.821	-137.500	0.0000
	138,092.8	11.773	31.004	-0.0008	141,522.7	-2.945	-144.117	-0.0002
1	140,066.1	14.664	32.583	-0.0008	143,549.9	-1.080	-150.695	-0.0004
1	142,026.0	17.535	34.151	-0.0009	145,564.2	1.821	-154.819	-0.0006
1	143,972.1 145,903.9	20.387 21.764	35.708 37.515	-0.0010 -0.0010	147,565.0 149,552.0	5.413 8.979	-157.280 -159.724	-0.0008 -0.0008
	140,903.9	∠1./04	31.313	-0.0010	149,552.0	0.979	- 109.724	-0.0008

Table 5.3.2-12: Mean GRAM Wind Profiles (Continued)

Mean F		RAM Wind F		Mea			Mean July GRAM Wind Profile				
Altitude	North	East	Down	Altitude	North	East	Down				
	•	Component				Component					
(ft)	(fps)	(fps)	(fps)	(ft)	(fps)	(fps)	(fps)				
146,768.5	21.764	37.515	-0.0010	150,441.6	8.979	-159.724	-0.0008				
147,820.8	22.761	39.374	-0.0014	151,524.5	11.766	-161.924	-0.0006				
149,721.5	23.749	41.218	-0.0015	153,480.7	12.803	-163.587	-0.0005				
151,607.2	24.631	43.322	-0.0004	155,421.9	13.832	-165.237	-0.0003				
153,479.5	25.333	45.897	-0.0006	157,349.8	14.854	-166.876	0.0034				
155,339.7	26.031	48.455	-0.0009	159,265.4	15.478	-167.874	0.0037				
157,188.5	26.724 27.214	50.997 53.884	-0.0011 -0.0014	161,169.2	16.068 16.655	-168.816 -169.753	0.0039				
159,025.9 160,851.8	27.214	56.831	-0.0014	163,061.0 164,940.5	17.215	-171.110	0.0042 0.0052				
162,666.3	28.100	59.769	-0.0010	166,807.7	17.213	-171.110	0.0032				
164,468.9	28.462	62.802	-0.0019	168,662.6	18.274	-174.736	0.0045				
166,259.7	28.578	66.157	-0.0027	170,505.3	18.795	-174.730	0.0043				
168,038.6	28.691	69.503	-0.0031	170,305.3	18.984	-176.526	0.0042				
169,805.8	28.799	72.842	-0.0039	172,333.7	19.148	-176.557	0.0039				
171,561.3	28.724	76.641	-0.0039	175,960.3	19.308	-176.516	0.0033				
173,305.1	28.494	80.825	-0.0042	177,754.4	19.300	-175.988	0.0029				
175,037.3	28.264	84.995	-0.0051	179,536.5	18.946	-174.459	0.0025				
176,757.9	28.032	89.152	-0.0055	181,306.6	18.592	-172.939	0.0028				
178,466.9	30.329	95.140	-0.0059	183,064.7	18.238	-171.425	0.0027				
180,164.4	33.405	101.683	-0.0063	184,810.9	17.265	-169.317	0.0027				
181,850.4	36.436	108.180	-0.0072	186,545.2	15.916	-166.852	0.0027				
183,524.9	39.424	114.631	-0.0073	188,267.6	14.577	-164.408	0.0026				
185,188.0	41.646	120.031	-0.0075	189,978.2	13.250	-161.984	0.0026				
186,839.8	43.727	125.252	-0.0077	191,677.1	13.325	-158.876	0.0026				
188,480.2	45.769	130.438	-0.0079	193,364.1	13.715	-155.638	0.0026				
190,109.3	47.771	135.590	-0.0081	195,039.5	14.095	-152.435	0.0026				
191,727.2	47.957	140.490	-0.0084	196,703.2	14.467	-149.267	0.0026				
193,333.8	47.894	145.328	-0.0087	198,355.3	14.213	-147.986	0.0025				
194,929.2	47.815	150.131	-0.0090	199,995.7	13.896	-146.896	0.0022				
196,513.5	47.721	154.898	-0.0094	201,624.6	13.578	-145.803	0.0019				
198,086.6	45.441	158.311	-0.0084	203,241.9	13.261	-144.707	0.0016				
199,648.7	42.580	161.313	-0.0079	204,847.7	15.055	-140.855	0.0013				
201,199.7	39.752	164.271	-0.0074	206,442.1	17.075	-136.702	0.0011				
202,739.7	36.960	167.181	-0.0069	208,025.0	19.055	-132.597	0.0008				
204,268.6	33.629	170.541	-0.0064	209,596.5	20.994	-128.538	0.0006				
205,786.7	29.894	174.233	-0.0060	211,156.7	21.138	-124.860	0.0004				
207,293.8	26.218	177.854	-0.0055	212,705.5	20.706	-121.331	0.0002				
208,790.0	22.605	181.399	-0.0051	214,243.0	20.275	-117.872	0.0001				
210,275.4	18.798	184.391	-0.0047	215,769.2	19.846	-114.478	0.0001				
211,750.0	14.035	185.416	-0.0043	217,284.2	21.286	-112.303	0.0001				
213,213.8	9.370	186.411	-0.0039	218,788.0	24.611	-111.357	0.0000				
214,666.9	4.828	187.330	-0.0036	220,280.7	27.851	-110.412	0.0000				
216,109.2	0.394	188.208	-0.0034	221,762.2	31.005	-109.466	0.0000				
217,540.9	-4.248	187.878	-0.0032	223,232.6	32.438	-108.062	0.0000				
218,961.9	-8.903	187.038	-0.0030	224,691.9	18.933	-103.369	0.0000				
220,372.3	-13.430	186.201	-0.0029	226,140.2	9.253	-101.350	0.0000				
221,772.1	-17.826	185.367	-0.0027	227,577.5	3.231	-101.877	-0.0001				
223,161.4	-21.751	184.428	-0.0026	229,003.9	0.697	-104.819	-0.0001				
224,540.2	-17.576	181.720	-0.0024	230,419.2	0.676	-105.905	-0.0001				
225,908.5	-12.883	180.287	-0.0023	231,823.7	0.707	-103.896	-0.0003				
227,266.4	-7.713	180.058	-0.0021	233,217.4	0.737	-101.903	-0.0004				
228,613.9	-2.104 2.603	180.964	-0.0020 0.0017	234,600.1	0.767	-99.926	-0.0005				
229,951.0	2.603	181.963	-0.0017	235,972.1	0.797	-97.965 06.010	-0.0006				
231,277.7	2.684	179.818	-0.0012 -0.0008	237,333.3	0.827	-96.019	-0.0007				
232,594.2	2.764	177.688 175.574		238,683.8	0.856	-94.089	-0.0008				
233,900.4	2.843	175.574	-0.0004	240,023.6	0.885	-92.175	-0.0009				
235,196.3	2.922	173.476	0.0000	241,352.7	0.914	-90.276	-0.0010				
236,482.0	3.000	171.394	0.0004	242,671.1	0.942	-88.393	-0.0011				
237,757.6	3.078	169.328	0.0008	243,979.0	0.971	-86.525	-0.0012				
239,023.0	3.155	167.277	0.0011	245,276.3	0.999	-84.672 91.073	-0.0013				
240,278.4	3.231	165.241 163.222	0.0015 0.0018	246,563.0	0.894	-81.073	-0.0012				
241,523.6	3.307	103.222	0.0018	247,839.2	0.578	-74.697	-0.0011				

Table 5.3.2-12: Mean GRAM Wind Profiles (Continued)

	Mean F	ebruary GF	RAM Wind F	Profile	Mea	n July GRA	M Wind Pro	file
	Altitude	North	East	Down	Altitude	North	East	Down
ı		•	Component				Component	'
	(ft)	(fps)	(fps)	(fps)	(ft)	(fps)	(fps)	(fps)
	243,984.1	3.456	159.229	0.0025	250,360.3	-0.043	-62.106	-0.0010
	245,199.3	3.530	157.256	0.0029	251,605.2	-0.350	-55.891	-0.0008
	246,404.7	3.407	155.311	0.0029	252,839.8	-0.653	-49.730	-0.0007
	247,600.1	2.781	153.415	0.0028	254,064.0	-0.953	-43.622	-0.0005
	248,785.6 249,961.4	2.159 1.541	151.532 149.661	0.0027 0.0026	255,277.9 256,481.5	-1.250 -1.544	-37.567 -31.566	-0.0003 -0.0001
	251,127.3	0.928	147.802	0.0025	257,674.9	-1.834	-25.617	0.0001
	252,283.5	0.319	145.956	0.0023	258,858.1	-2.122	-19.721	0.0003
	253,429.9	-0.285	144.121	0.0022	260,031.1	-2.407	-13.877	0.0006
	254,566.7	-0.885	142.299	0.0020	261,194.0	-2.689	-8.086	0.0009
	255,693.8	-1.481	140.489	0.0018	262,346.8	-2.968	-2.347	0.0012
	256,811.2	-2.073	138.692	0.0016	263,489.5	-2.701	1.462	0.0012
	257,919.1	-2.660	136.906	0.0014	264,622.2	-2.371	5.005	0.0011
	259,017.4	-3.242	135.132	0.0011	265,744.9	-2.045	8.516	0.0011
	260,106.2	-3.821	133.370	0.0009	266,857.6	-1.721	11.995	0.0011
1	261,185.5	-4.395	131.621	0.0006	267,960.3	-1.401	15.442	0.0011
1	262,255.4	-4.965	129.883	0.0003	269,053.2	-1.085	18.857	0.0011
1	263,315.8	-4.643	128.077	0.0013	270,136.2	-0.772	22.240	0.0010
1	264,366.9	-4.097 3.557	126.264	0.0024	271,209.4	-0.462	25.591	0.0010
1	265,408.6	-3.557 -3.022	124.465 122.680	0.0036 0.0046	272,272.8	-0.156 0.147	28.911 32.199	0.0010 0.0010
1	266,441.0 267,464.1	-3.022 -2.493	122.680	0.0046	273,326.4 274,370.3	0.147	35.455	0.0010
	268,478.0	-1.969	119.151	0.0067	275,404.5	0.742	38.680	0.0010
	269,482.6	-1.451	117.407	0.0076	276,429.1	1.035	41.874	0.0010
	270,478.1	-0.939	115.677	0.0086	277,444.0	1.324	45.037	0.0010
	271,464.5	-0.432	113.961	0.0094	278,449.3	1.611	48.169	0.0010
	272,441.7	0.070	112.257	0.0103	279,445.1	1.669	52.727	0.0011
	273,409.9	0.566	110.568	0.0111	280,431.4	1.559	58.323	0.0014
	274,369.0	1.057	108.892	0.0118	281,408.2	1.451	63.864	0.0017
	275,319.2	1.543	107.229	0.0126	282,375.5	1.344	69.350	0.0020
	276,260.4	2.023	105.579	0.0133	283,333.4	1.238	74.781	0.0023
	277,192.7	2.498	103.943	0.0139	284,282.0	1.133	80.158	0.0026
	278,116.0 279,030.6	2.968 3.316	102.320 100.837	0.0145 0.0165	285,221.2 286,151.1	1.030 0.928	85.481 90.750	0.0029 0.0032
	279,936.3	3.097	99.982	0.0162	287,071.7	0.827	95.965	0.0032
	280,833.2	2.879	99.132	0.0160	287,983.1	0.728	101.127	0.0039
	281,721.4	2.663	98.289	0.0158	288,885.3	0.630	106.235	0.0042
	282,600.8	2.448	97.452	0.0156	289,778.4	0.533	111.290	0.0045
	283,471.7	2.235	96.620	0.0153	290,662.3	0.437	116.292	0.0048
1	284,333.8	2.024	95.795	0.0151	291,537.1	0.342	121.241	0.0051
1	285,187.4	1.814	94.975	0.0149	292,402.9	0.249	126.138	0.0054
1	286,032.4	1.606	94.161	0.0147	293,259.6	0.157	130.983	0.0058
1	286,868.9	1.399	93.353	0.0146	294,107.4	0.066	135.775	0.0061
1	287,696.8	1.194	92.551	0.0144	294,946.2	-0.023	140.516	0.0064
1	288,516.4 289,327.5	0.990 0.788	91.754 90.963	0.0142 0.0141	295,776.1 296,597.1	-0.059 -0.058	143.974 146.538	0.0078 0.0081
1	290,130.2	0.788	90.963	0.0139	290,397.1	-0.058	149.026	0.0081
1	290,924.6	0.389	89.399	0.0138	298,212.7	-0.057	151.439	0.0085
L	291,710.7	0.192	88.625	0.0136	299,007.4	-0.057	153.781	0.0087
	292,488.5	-0.004	87.857	0.0135	299,793.3	-0.056	156.051	0.0089
	293,258.1	-0.198	87.095	0.0133	300,570.5	-0.056	158.253	0.0091
	294,019.5	-0.391	86.338	0.0132	301,339.1	-0.055	160.388	0.0093
	294,772.8	-0.582	85.586	0.0131	302,099.1	-0.054	162.458	0.0095
1	295,517.9	-0.715	85.066	0.0137	302,850.5	-0.054	164.464	0.0097
	296,254.9	-0.729	85.007	0.0137	303,593.3	-0.053	166.409	0.0098
	296,983.9	-0.743	84.923	0.0137	304,327.7	-0.052	168.293	0.0100
	297,704.9	-0.757 0.771	84.812	0.0137	305,053.6	-0.052	170.119	0.0102
1	298,418.0	-0.771	84.674	0.0137	305,771.1	-0.051	171.888	0.0104
1	299,123.1 299,820.3	-0.784 -0.798	84.509 84.317	0.0136 0.0136	306,480.2 307,180.9	-0.051 -0.050	173.602 175.261	0.0105
1	300,509.7	-0.798 -0.811	84.317 84.096	0.0136	307,180.9 307,873.4	-0.050 -0.050	175.261 176.869	0.0107 0.0108
1	301,191.2	-0.824	83.848	0.0136	308,557.6	-0.030	178.426	0.0108
1	301,865.0	-0.837	83.571	0.0136	309,233.5	-0.049	179.933	0.0111
	-0.,000.0	0.501	55.57 1	0.0.50	1 000,200.0	0.010	5.555	0.0.11

Table 5.3.2-12: Mean GRAM Wind Profiles (Continued)

Mean I		RAM Wind F		Mean July GRAM Wind Profile				
Altitude	North	East	Down	Altitude	North	East	Down	
		Component				Component		
(ft)	(fps)	(fps)	(fps)	(ft)	(fps)	(fps)	(fps)	
302,531.1	-0.849	83.267	0.0135	309,901.3	-0.049	181.393	0.0113	
303,189.4	-0.862	82.935	0.0135	310,560.9	-0.049	182.806	0.0114	
303,840.1	-0.874	82.574	0.0135	311,212.4	-0.049	184.174	0.0115	
304,483.2	-0.887	82.186	0.0134	311,855.8	-0.049	185.125	0.0080	
305,118.7	-0.899	81.771	0.0134	312,491.2	-0.049	185.045	0.0081	
305,746.7	-0.911	81.329	0.0133	313,118.6	-0.050	184.957	0.0081	
306,367.2	-0.922 -0.935	80.859 80.364	0.0133 0.0132	313,738.1	-0.050 -0.050	184.862	0.0082	
306,980.2 307,585.8	-0.935	79.843	0.0132	314,349.6 314,953.3	-0.050	184.760 184.654	0.0082 0.0082	
308,184.0	-0.958	79.296	0.0132	315,549.1	-0.052	184.545	0.0082	
308,774.9	-0.969	78.725	0.0131	316,137.1	-0.055	184.433	0.0083	
309,358.4	-0.980	78.129	0.0131	316,717.4	-0.057	184.319	0.0083	
309,934.7	-0.992	77.511	0.0129	317,290.0	-0.058	184.202	0.0084	
310,503.8	-1.003	76.869	0.0129	317,854.9	-0.061	184.087	0.0084	
311,065.8	-1.003	76.204	0.0128	318,412.2	-0.063	183.971	0.0084	
311,620.6	-1.024	75.519	0.0127	318,961.8	-0.067	183.857	0.0084	
312,168.2	-1.036	74.869	0.0093	319,504.0	-0.071	183.742	0.0085	
312,708.9	-1.046	74.204	0.0092	320,038.6	-0.074	183.634	0.0085	
313,242.5	-1.057	73.521	0.0092	320,565.8	-0.079	183.525	0.0085	
313,769.1	-1.068	72.815	0.0092	321,085.5	-0.082	183.417	0.0085	
314,288.9	-1.079	72.092	0.0092	321,597.9	-0.088	183.317	0.0085	
314,801.7	-1.090	71.349	0.0092	322,102.9	-0.094	183.217	0.0085	
315,307.7	-1.101	70.591	0.0091	322,600.6	-0.098	183.123	0.0085	
315,806.9	-1.113	69.813	0.0091	323,091.1	-0.103	183.032	0.0085	
316,299.3	-1.123	69.019	0.0091	323,574.4	-0.113	182.947	0.0085	
316,785.0	-1.135	68.213	0.0091	324,050.5	-0.121	182.870	0.0085	
317,264.0	-1.147	67.393	0.0090	324,519.5	-0.127	182.789	0.0085	
317,736.3	-1.156	66.555	0.0090	324,981.4	-0.133	182.720	0.0085	
318,202.1	-1.167	65.706	0.0090	325,436.2	-0.144	182.656	0.0085	
318,661.4	-1.180	64.848	0.0090	325,884.1	-0.154	182.595	0.0085	
319,114.1	-1.191	63.977	0.0089	326,325.0	-0.161	182.536	0.0085	
319,560.3	-1.203	63.096	0.0089	326,759.0	-0.171	182.487	0.0085	
320,000.2	-1.215	62.204	0.0089	327,186.1	-0.181	182.442	0.0084	
320,433.6	-1.227	61.304	0.0088	327,606.4	-0.192	182.399	0.0084	
320,860.7	-1.237	60.397	0.0088	328,020.0	-0.203	182.363	0.0084	
321,281.6	-1.252	59.483	0.0088	328,426.8	-0.215	181.950	0.0068	
321,696.2	-1.262	58.562	0.0087	328,826.9	-0.226	181.474	0.0068	
322,104.5	-1.274	57.635	0.0087	329,220.4	-0.239	181.022	0.0068	
322,506.8	-1.289	56.704	0.0087	329,607.3	-0.255	180.589	0.0068	
322,902.9	-1.300	55.769	0.0086	329,987.6	-0.264	180.167	0.0067	
323,292.9	-1.315	54.830	0.0086	330,361.4	-0.281	179.772	0.0067	
323,676.9	-1.326	53.885	0.0086	330,728.8	-0.294	179.389	0.0067	
324,054.9	-1.341	52.943	0.0085	331,089.8	-0.308	179.021	0.0066	
324,427.0	-1.355	51.995	0.0085	331,444.4	-0.325	178.675	0.0066	
324,793.2	-1.368	51.051	0.0085	331,792.6	-0.339	178.340	0.0066	
325,153.6	-1.382	50.102	0.0084	332,134.6	-0.353	178.020	0.0065	
325,508.1	-1.395	49.156	0.0084	332,470.4	-0.368	177.721	0.0065	
325,857.0	-1.409	48.208	0.0084	332,800.0	-0.391	177.435	0.0064	
326,200.1	-1.424	47.262	0.0083	333,123.5	-0.401	177.157	0.0064	
326,537.5	-1.440	46.324	0.0083	333,440.8	-0.416	176.891	0.0064	
326,869.3	-1.454	45.381	0.0083	333,752.2	-0.435	176.647	0.0063	
327,195.6	-1.469	44.445	0.0082	334,057.5	-0.455	176.411	0.0063	
327,516.3	-1.483	43.510	0.0082	334,356.9	-0.470	176.189	0.0063	
327,831.6	-1.498 1.515	42.582	0.0082	334,650.5	-0.490	175.982	0.0062	
328,141.4	-1.515 1.520	41.636	0.0068	334,938.1	-0.505	175.777	0.0062	
328,445.9	-1.530 1.540	40.603	0.0068	335,220.0	-0.525 0.544	175.586	0.0061	
328,745.0	-1.548	39.579	0.0068	335,496.2	-0.544	175.415	0.0061	
329,038.8	-1.560 -1.570	38.564	0.0067	335,766.6	-0.560	175.243	0.0061	
329,327.4	-1.578 1.507	37.557	0.0067	336,031.4	-0.580	175.076	0.0060	
329,610.8	-1.597 1.612	36.561 35.571	0.0067	336,290.6	-0.595 0.614	174.932	0.0060	
329,889.1	-1.612 1.630	35.571 34.594	0.0066	336,544.2 336,792.4	-0.614	174.794 174.666	0.0060	
330,162.3	-1.630 -1.642	34.594	0.0066 0.0066	330,792.4	-0.640 -0.654	174.000	0.0059	

Table 5.3.2-12: Mean GRAM Wind Profiles (Continued)

	Mean F	ebruary GF	RAM Wind P	Profile	Mea	n July GRA	M Wind Pro	file
	Altitude	North	East	Down	Altitude	North	East	Down
			Component			•	Component	
	(ft)	(fps)	(fps)	(fps)	(ft)	(fps)	(fps)	(fps)
	330,693.6 330,951.8	-1.665 -1.680	32.674 31.730	0.0065 0.0065	337,272.4 337,504.4	-0.675 -0.689	174.421 174.306	0.0058 0.0058
	331,205.1	-1.700	30.793	0.0065	337,731.0	-0.708	174.300	0.0058
	331,453.6	-1.715	29.866	0.0064	337,952.5	-0.723	174.204	0.0057
	331,697.3	-1.734	28.954	0.0064	338,168.7	-0.748	174.022	0.0057
	331,936.2	-1.750	28.050	0.0064	338,379.8	-0.767	173.941	0.0057
	332,170.4	-1.769	27.164	0.0064	338,585.9	-0.786	173.856	0.0056
	332,400.0	-1.785	26.283	0.0063	338,786.9	-0.800	173.782	0.0056
	332,625.0	-1.805	25.416	0.0063	338,982.9	-0.818	173.713	0.0055
	332,845.5	-1.824	24.566	0.0063	339,174.0	-0.837	173.654	0.0055
	333,061.5	-1.840	23.720	0.0062	339,360.2	-0.855	173.592	0.0055
	333,273.0	-1.860	22.896	0.0062	339,541.6	-0.873	173.536	0.0054
	333,480.2	-1.880	22.072	0.0062	339,718.2	-0.885	173.484	0.0054
	333,683.0	-1.896	21.267	0.0062	339,890.2	-0.908	173.439	0.0054
	333,881.5	-1.911	20.469	0.0061	340,057.4	-0.921	173.397	0.0053
1	334,075.8	-1.935	19.694	0.0061	340,220.1	-0.943	173.357	0.0053
1	334,266.0	-1.951	18.921	0.0061	340,378.2	-0.953	173.314	0.0053
1	334,452.0	-1.971	18.166	0.0060	340,531.9	-0.976	173.284	0.0053
	334,634.0	-1.995	17.423	0.0060	340,681.1	-0.987	173.249	0.0052
1	334,811.9	-2.011	16.690	0.0060	340,825.9	-1.001	173.215	0.0052
	334,985.9	-2.026	15.961	0.0060	340,966.4	-1.017	173.192	0.0052
	335,155.9	-2.046	15.254	0.0059	341,102.6	-1.033	173.160	0.0051
	335,322.2	-2.061	14.561	0.0059	341,234.6	-1.047	173.137	0.0051
	335,484.6 335,643.3	-2.081	13.878	0.0059	341,362.5	-1.061 1.075	173.116	0.0051
	335,798.3	-2.106 -2.120	13.207 12.541	0.0059 0.0058	341,486.3 341,606.0	-1.075 -1.089	173.093 173.075	0.0051 0.0050
	335,949.7	-2.120 -2.140	11.893	0.0058	341,721.8	-1.102	173.075	0.0050
	336,097.5	-2.160	11.259	0.0058	341,833.6	-1.115	173.041	0.0050
	336,241.7	-2.180	10.630	0.0058	341,941.5	-1.128	173.020	0.0050
	336,382.6	-2.200	10.019	0.0058	342,045.7	-1.140	173.003	0.0049
	336,520.0	-2.215	9.416	0.0057	342,146.1	-1.152	172.994	0.0049
	336,654.1	-2.234	8.822	0.0057	342,242.9	-1.163	172.972	0.0049
	336,784.8	-2.254	8.246	0.0057	342,336.0	-1.174	172.954	0.0049
	336,912.4	-2.273	7.679	0.0057	342,425.6	-1.192	172.951	0.0049
	337,036.8	-2.293	7.121	0.0056	342,511.6	-1.194	172.937	0.0048
	337,158.1	-2.313	6.576	0.0056	342,594.3	-1.211	172.923	0.0048
	337,276.3	-2.327	6.030	0.0056	342,673.5	-1.214	172.905	0.0048
	337,391.5	-2.347	5.507	0.0056	342,749.5	-1.224	172.896	0.0048
	337,503.8	-2.370	4.996	0.0056	342,822.2	-1.232	172.883	0.0048
1	337,613.2	-2.385	4.484	0.0055	342,891.7	-1.240	172.860	0.0047
1	337,719.9	-2.404	3.989	0.0055	342,958.1	-1.248 1.255	172.851	0.0047
1	337,823.7 337,924.9	-2.429 -2.443	3.507 3.028	0.0055 0.0055	343,021.5 343,081.9	-1.255 -1.262	172.834 172.817	0.0047 0.0047
	338,023.4	-2.443	2.561	0.0055	343,061.9	-1.262	172.808	0.0047
1	338,119.4	-2.481	2.102	0.0054	343,194.0	-1.209	172.796	0.0047
1	338,212.9	-2.500	1.655	0.0054	343,245.8	-1.282	172.730	0.0047
1	338,303.9	-2.519	1.209	0.0054	343,294.9	-1.288	172.758	0.0046
1	338,392.5	-2.533	0.781	0.0054	343,341.3	-1.301	172.739	0.0046
	338,478.8	-2.558	0.360	0.0054	343,385.2	-1.299	172.720	0.0046
	338,562.9	-2.570	-0.056	0.0053	343,426.5	-1.310	172.703	0.0046
	338,644.8	-2.595	-0.463	0.0053	343,465.4	-1.314	172.687	0.0046
	338,724.6	-2.614	-0.865	0.0053	343,502.0	-1.319	172.668	0.0046
	338,802.3	-2.627	-1.256	0.0053	343,536.2	-1.322	172.653	0.0046
1.	338,878.0	-2.645	-1.641	0.0053	343,568.2	-1.326	172.636	0.0046
	338,951.8	-2.670	-2.011	0.0052	343,598.0	-1.329	172.607	0.0046
	339,023.7	-2.683	-2.389	0.0052	343,625.7	-1.325	172.589	0.0046
	339,093.8	-2.707	-2.748	0.0052	343,651.4	-1.328	172.568	0.0046
	339,162.3	-2.727	-3.111	0.0052	343,675.1	-1.329	172.547	0.0045
1	339,229.0	-2.739	-3.460	0.0052	343,697.0	-1.338	172.521	0.0045
1	339,294.2	-2.757	-3.808	0.0051	343,717.0	-1.340	172.501	0.0045
	339,357.8	-2.782	-4.148	0.0051	343,735.4	-1.334	172.471	0.0045
<u> </u>	339,420.0	-2.800	-4.487	0.0051	343,752.0	-1.336	172.453	0.0045

Table 5.3.2-12: Mean GRAM Wind Profiles (Continued)

	Mean I	February GF	RAM Wind F	Profile	Mea	n July GRA	M Wind Pro	file
	Altitude	North	East	Down	Altitude	North	East	Down
			Component		Ailitude		Component	
	(ft)	(fps)	(fps)	(fps)	(ft)	(fps)	(fps)	(fps)
	339,480.8	-2.813	-4.825	0.0051	343,767.1	-1.337	172.423	0.0045
	339,540.3	-2.831	-5.147	0.0051	343,780.7	-1.339	172.407	0.0045
	339,598.6	-2.850	-5.465	0.0050	343,792.8	-1.344	172.383	0.0045
	339,655.7	-2.869	-5.785	0.0050	343,803.6	-1.339	172.350	0.0045
	339,711.6	-2.887	-6.097	0.0050	343,813.2	-1.339	172.322	0.0045
	339,766.6	-2.906	-6.412	0.0050	343,821.5	-1.346	172.295	0.0045
	339,820.6	-2.924	-6.714	0.0050	343,828.7	-1.339	172.268	0.0045
	339,873.7	-2.943 -2.962	-7.029 -7.325	0.0049 0.0049	343,834.9	-1.345 -1.345	172.248 172.221	0.0045
	339,926.1 339,977.6	-2.975	-7.625	0.0049	343,840.1 343,844.4	-1.343	172.221	0.0045 0.0045
	340,028.6	-2.998	-7.913	0.0049	343,847.9	-1.344	172.163	0.0045
	340,078.9	-3.011	-8.216	0.0049	343,850.8	-1.342	172.131	0.0045
	340,128.7	-3.036	-8.506	0.0048	343,852.9	-1.335	172.099	0.0045
	340,178.2	-3.055	-8.796	0.0048	343,854.6	-1.340	172.074	0.0045
	340,227.2	-3.073	-9.085	0.0048	343,855.7	-1.332	172.036	0.0045
	340,276.0	-3.092	-9.379	0.0048	343,856.5	-1.331	172.011	0.0045
1	340,324.5	-3.110	-9.668	0.0048	343,857.0	-1.337	171.979	0.0045
1	340,373.0	-3.129	-9.963	0.0047	343,857.3	-1.336	171.942	0.0045
1	340,421.4	-3.149	-10.252	0.0047	343,857.4	-1.333	171.910	0.0045
1	340,469.9	-3.168	-10.542	0.0047	343,857.5	-1.326	171.885	0.0045
	340,518.4	-3.186	-10.833	0.0047	343,857.7	-1.325	171.853	0.0045
	340,567.2	-3.206	-11.126	0.0047	343,858.0	-1.323	171.821	0.0045
	340,616.3	-3.218	-11.425	0.0046	343,858.5	-1.321	171.783	0.0045
	340,665.7	-3.245	-11.725	0.0046	343,859.4	-1.320	171.757	0.0045
	340,715.6	-3.257	-12.024	0.0046	343,860.7	-1.326	171.730	0.0045
	340,766.0	-3.282	-12.318	0.0046	343,862.4	-1.318	171.691	0.0045
	340,817.0	-3.296 -3.316	-12.632 -12.932	0.0046	343,864.8 343,867.8	-1.317 -1.323	171.658	0.0045
	340,868.7 340,921.2	-3.336	-12.932	0.0045 0.0045	343,871.7	-1.323	171.631 171.597	0.0045 0.0045
	340,974.6	-3.355	-13.565	0.0045	343,876.4	-1.314	171.563	0.0045
	341,029.0	-3.375	-13.882	0.0045	343,882.0	-1.314	171.535	0.0045
	341,084.4	-3.395	-14.209	0.0045	343,888.8	-1.313	171.500	0.0045
	341,140.9	-3.421	-14.532	0.0044	343,896.7	-1.313	171.471	0.0045
	341,198.7	-3.435	-14.869	0.0044	343,905.9	-1.319	171.447	0.0045
	341,257.8	-3.461	-15.209	0.0044	343,916.5	-1.313	171.417	0.0045
	341,318.3	-3.475	-15.549	0.0044	343,928.5	-1.313	171.385	0.0044
	341,380.3	-3.502	-15.908	0.0044	343,942.1	-1.313	171.354	0.0044
	341,444.0	-3.516	-16.276	0.0043	343,957.4	-1.314	171.322	0.0044
	341,509.3	-3.537	-16.635	0.0043	343,974.4	-1.315	171.290	0.0044
1	341,576.4	-3.558	-17.018	0.0043	343,993.3	-1.323	171.269	0.0044
1	341,645.4	-3.579	-17.390	0.0043	344,014.2	-1.317	171.235	0.0044
	341,716.3	-3.599	-17.783	0.0042	344,037.2	-1.326	171.213	0.0044
1	341,789.4	-3.620 -3.642	-18.181 -18.589	0.0042 0.0042	344,062.4 344,089.9	-1.322 -1.324	171.191 171.167	0.0044 0.0044
1	341,864.6 341,942.1	-3.664	-10.569	0.0042	344,119.9	-1.324	171.167 171.131	0.0044
	342,022.0	-3.693	-19.005	0.0042	344,152.3	-1.338	171.131	0.0044
1	342,104.4	-3.708	-19.877	0.0041	344,187.4	-1.334	171.088	0.0044
1	342,189.3	-3.729	-20.332	0.0041	344,225.2	-1.345	171.062	0.0044
	342,276.9	-3.758	-20.785	0.0041	344,265.9	-1.350	171.054	0.0044
	342,367.4	-3.773	-21.256	0.0041	344,309.6	-1.347	171.027	0.0043
	342,460.7	-3.796	-21.748	0.0040	344,356.4	-1.360	171.012	0.0043
	342,557.0	-3.819	-22.241	0.0040	344,406.4	-1.360	170.991	0.0043
	342,656.4	-3.849	-22.744	0.0040	344,459.7	-1.366	170.981	0.0043
1	342,759.1	-3.865	-23.271	0.0040	344,516.5	-1.380	170.915	0.0031
	342,865.1	-3.888	-23.806	0.0040	344,576.9	-1.380	170.797	0.0031
	342,974.5	-3.912	-24.358	0.0039	344,640.9	-1.396	170.680	0.0030
	343,087.5	-3.935	-24.919	0.0039	344,708.8	-1.398	170.550	0.0030
	343,204.2	-3.960	-25.505	0.0039	344,780.7	-1.414	170.418	0.0030
1	343,324.6	-3.983	-26.099	0.0039	344,856.6	-1.432	170.288	0.0030
1	343,448.9	-4.009	-26.720	0.0038	344,936.6	-1.437	170.144	0.0030
1	343,577.3	-4.033	-27.345	0.0038	345,021.1	-1.447	169.993	0.0029
	343,709.7	-4.059	-27.990	0.0038	345,110.0	-1.461	169.849	0.0029

Table 5.3.2-12: Mean GRAM Wind Profiles (Continued)

Mean F	ebruary GF			Mea	n July GRA		
Altitude	North	East	Down	Altitude	North	East	Down
(ft)	Component (fps)	Component (fps)	Component (fps)	(ft)	Component (fps)	Component (fps)	Component (fps)
343,846.5	-4.090	-28.653	0.0038	345,203.4	-1.480	169.704	0.0029
343,987.6	-4.116	-29.338	0.0038	345,301.7	-1.496	169.548	0.0028
344,133.1	-4.142	-30.039	0.0037	345,404.7	-1.509	169.392	0.0028
344,283.3	-4.169	-30.754	0.0037	345,512.7	-1.525	169.216	0.0028
344,437.9	-4.195	-31.497	0.0037	345,625.6	-1.543	169.055	0.0028
344,597.2	-4.222	-32.317	0.0030	345,743.5	-1.559	168.881	0.0027
344,761.0	-4.249	-33.192	0.0030	345,866.4	-1.579	168.724	0.0027
344,929.5	-4.284 -4.312	-34.081	0.0030	345,994.3	-1.599	168.543	0.0026
345,102.7 345,280.5	-4.312 -4.334	-34.996 -35.937	0.0029 0.0029	346,127.4 346,265.6	-1.621 -1.650	168.370 168.201	0.0026 0.0026
345,463.1	-4.368	-36.893	0.0029	346,408.9	-1.673	168.015	0.0025
345,650.4	-4.406	-37.875	0.0029	346,557.5	-1.698	167.841	0.0025
345,842.5	-4.428	-38.879	0.0029	346,711.3	-1.723	167.664	0.0024
346,039.4	-4.466	-39.902	0.0029	346,870.4	-1.758	167.492	0.0024
346,241.1	-4.504	-40.940	0.0028	347,034.8	-1.785	167.307	0.0023
346,447.7	-4.528	-42.013	0.0028	347,204.6	-1.822	167.142	0.0023
346,659.2	-4.568	-43.085	0.0028	347,379.8	-1.853	166.965	0.0022
346,875.6	-4.599	-44.200	0.0028	347,560.5	-1.885	166.795	0.0022
347,097.0	-4.632	-45.336	0.0028	347,746.7	-1.918	166.627	0.0021
347,323.4	-4.667	-46.487	0.0028	347,938.4	-1.960	166.461	0.0020
347,554.8	-4.710	-47.651	0.0027	348,135.7	-1.997	166.306	0.0020
347,791.3	-4.744	-48.845	0.0027	348,338.7	-2.043	166.155	0.0019
348,032.9	-4.787 -4.833	-50.069 51.300	0.0027	348,547.4	-2.082	166.010	0.0018
348,279.7 348,531.6	-4.869	-51.300 -52.575	0.0027 0.0027	348,761.7 348,981.9	-2.132 -2.175	165.869 165.730	0.0018 0.0017
348,788.8	-4.918	-53.842	0.0027	349,207.9	-2.173	165.610	0.0017
349,051.2	-4.956	-55.155	0.0027	349,439.8	-2.284	165.495	0.0015
349,318.9	-5.004	-56.475	0.0027	349,677.6	-2.322	165.387	0.0015
349,591.9	-5.046	-57.826	0.0027	349,921.4	-2.390	165.290	0.0014
349,870.3	-5.095	-59.186	0.0026	350,171.2	-2.442	165.196	0.0013
350,154.1	-5.148	-60.582	0.0026	350,427.1	-2.497	165.116	0.0012
350,443.3	-5.193	-61.994	0.0026	350,689.1	-2.561	165.065	0.0011
350,738.0	-5.247	-63.429	0.0026	350,957.3	-2.639	165.020	0.0010
351,038.2	-5.293	-64.888	0.0026	351,231.7	-2.700	164.984	0.0009
351,344.0	-5.361	-66.361	0.0026	351,512.4	-2.773	164.957	0.0008
351,655.4	-5.410	-67.859	0.0026	351,799.4	-2.838	164.956	0.0007
351,972.4	-5.479 5.532	-69.372	0.0026	352,092.8	-2.924	164.967	0.0006
352,295.1	-5.532 -5.595	-70.908 -72.467	0.0026 0.0026	352,392.5	-3.005 -3.089	165.016	0.0005
352,623.5 352,957.7	-5.595 -5.661	-74.051		352,698.8 353,011,5	-3.069 -3.177	165.061	0.0004
353,297.6	-5.738	-74.051	0.0026 0.0027	353,011.5 353,330.9	-3.177	165.129 165.223	0.0003 0.0002
353,643.3	-5.808	-73.031	0.0027	353,656.8	-3.356	165.342	0.0002
353,994.9	-5.877	-78.906	0.0027	353,989.4	-3.460	165.467	-0.0001
354,352.4	-5.962	-80.547	0.0027	354,328.7	-3.558	165.626	-0.0002
354,715.8	-6.037	-82.232	0.0027	354,674.7	-3.662	165.817	-0.0003
355,085.2	-6.114	-83.929	0.0027	355,027.5	-3.779	166.027	-0.0004
355,460.5	-6.196	-85.643	0.0027	355,387.1	-3.887	166.272	-0.0006
355,841.8	-6.287	-87.360	0.0028	355,753.6	-4.010	166.543	-0.0007
356,229.2	-6.384	-89.109	0.0028	356,126.9	-4.138	166.840	-0.0009
356,622.6	-6.471	-90.861	0.0028	356,507.2	-4.259	167.167	-0.0010
357,022.1	-6.575	-92.634	0.0028	356,894.5	-4.394	167.522	-0.0012
357,427.6	-6.677	-94.414	0.0029	357,288.7	-4.535 4.660	167.907	-0.0013
357,839.2	-6.786	-96.212	0.0029	357,689.9	-4.669 4.830	168.326	-0.0015 0.0016
358,256.9 358,680.7	-6.898 -6.999	-98.033 -99.864	0.0029 0.0030	358,098.1 358,513.2	-4.830 -4.998	168.778 169.274	-0.0016 -0.0018
358,680.7 359,110.5	-6.999 -7.130	-99.864 -101.704	0.0030	358,513.2 358,935.4	-4.998 -5.143	169.274 169.785	-0.0018 -0.0020
359,110.5	-7.130 -7.240	-101.70 <del>4</del> -103.531	0.0030	359,364.5	-5.143 -5.308	170.353	-0.0020 -0.0021
359,987.8	-7.240	-105.377	0.0031	359,800.4	-5.477	170.333	-0.0021
360,434.9	-7.507	-103.377	0.0031	360,243.0	-5.662	171.549	-0.0025
360,887.5	-7.639	-109.120	0.0032	360,692.1	-5.842	171.043	-0.0027
361,108.8	-7.639	-109.120	0.0032	361,117.7	-5.948	172.569	-0.0032
1,000,000	-7.639	-109.120	0.0032	361,117.7	-5.948	172.569	-0.0032

#### 5.3.2.9 FLIGHT MECHANICS REFERENCES

NSTS 08209, Volume I, Revision B, March 16, 1999, Shuttle Systems Design Criteria, Volume I, Shuttle Performance Assessment Databook (SPAD).

NSTS 08209, Volume III, Revision A, February 2, 1999, Shuttle Systems Design Criteria, Volume III, Systems and Environmental Dispersions.

NSTS 08209, Volume IV, Revision C, July 27, 1999, Shuttle Systems Design Criteria, Volume IV, Generic Ascent Flight Design Requirements.

Natural environments: NASA MSFC Environments Group/ED44, Dale Johnson (256) 544-1665, Jere Justus, Morgan, (256) 544-3260.

SRB reference data (including 5-segment booster); expendable versus reusable SRB: Mark Tobias/ATK Thiokol, (435) 863-2802. RSRM Block Model, ATK Thiokol document TPR07499, dated 3/15/1999.

SRB dispersions: Tim Olive, NASA MSFC Systems Analysis Group/TD53, (256) 544-1509.

SSME data and dispersions: Mark Hines, Boeing Ascent Performance (Houston), (281) 226-5525. SSME Block II Nominal Power Balance and Normal Operating Maximums and Minimums, document IL-2002-478-007, J.A. Wendlandt / Rocketdyne, dated 3/25/2002. SSME start up and shut down transients provided by R.H. Coates, NASA MSFC Liquid Engine & Main Propulsion Systems / ER21, (256) 544-0549.

Aerodynamic data: Thomas (Joe) Lowery, NASA MSFC Aerosciences Branch / EV33, (256) 544-0540.

Mass properties: Holly Chandler (256) 544-0496, NASA MSFC Systems Analysis and Performance Branch / EV12.

ET footprint size: STS 51-D 3 sigma footprint.

SRB Reentry data: email messages and excerpts from SE-019-053-2H provided by Bruce McWhorter / ATK Thiokol.

#### 5.3.3 REFERENCE TRAJECTORIES

Two reference trajectories were developed during this study: Rev. 1 and Rev. 2. All of the trade studies were conducted on Rev. 1. Once the updated information was incorporated into Rev. 2, most of the analyses were repeated. Within this document, the reference trajectory version will be specified. All trajectories were modeled using the Program to Optimize Simulated Trajectories (POST).

In addition to the ground rules, several assumptions were necessary to construct the reference trajectories. These ground rules and assumptions have been listed in Section 5.3.1. For completeness, the data used in the 3-DOF trajectories is detailed in Section 5.3.2.

For our reference trajectories, some of the key events needed to be specified. In future studies, these need to be analyzed to determine the optimal trajectory. These key events included the SRB separation condition, the time between the booster separation and full thrust of the second stage, and the LES jettison conditions.

For the typical STS mission, the SRBs separate from the external tank around 125 seconds. The thrust from each booster is around 100,000 lb<sub>f</sub>. Since they are pushed away from the ET, which still has the thrust of the 3 SSMEs, re-contact isn't a concern. For the CLV-4 vehicle, the separation conditions are greatly different. At separation, the upper stage has not been ignited. If this vehicle used the same criteria as the STS, the booster would still be thrusting toward the upper stage. By lengthening the burn time on the 1<sup>st</sup> stage, the thrust is rapidly decreasing. When the thrust had decreased to 15,000 lb<sub>f</sub>, the upper stage was able to separate cleanly.

Since the 2<sup>nd</sup> stage ignition is not an instantaneous event, a finite amount of time needed to be assumed. For purposes of this study, 10 seconds was used. After separation of the first stage, there is a short coast period. Then, the SSME is ignited and the start-up transient is incorporated into the trajectory. After the thrust reaches 100%, the engine ramps to the nominal power level of 104.5%. The time duration from 1<sup>st</sup> stage separation to a thrust level of 100% is the assumed 10 seconds.

The Exploration System Architecture Study (ESAS) levied a ground rule for the LES separation criteria. This stated the LES would be dropped 30 seconds after the 2<sup>nd</sup> stage was ignited. For the CLV-4 vehicle, this corresponded to an altitude around 190,000 to 195,000 feet.

Based on historical data from the Apollo missions, their LES was jettisoned at altitudes around 279,000 to 312,000 ft. In the case of Apollo 8, this corresponded to approximately 50 seconds after the S-IC separation.

Determination of the LES jettison altitude must include several factors. Some of the primary considerations are

LES location on the 2<sup>nd</sup> stage allows the crew to escape the still-burning SRB during a 1<sup>st</sup> stage abort.

The crew module must escape any explosion-driven compression wave when the atmospheric density is high enough to be problematic.

After the LES is jettisoned, the service module engine provides the abort separation burn. Sufficient time is needed before reentering the atmosphere for the SM burn to be completed, the SM to be jettisoned, and the CM to be correctly oriented for entry.

Separation altitudes around 190,000 to 195,000 ft appear to be low. For conservatism, this study adopted a separation altitude similar to those seen in the Apollo flights. Although there is an impact to the payload delivery capability, carrying the LES longer and higher aids in the abort modes.

A summary of performance and trajectory parameters for the two Rev. 2 reference trajectories are shown in Table 5.3.3-1. The Rev. 1 reference trajectories are also included for comparison.

Several trajectory parameters for the two Rev. 2 reference trajectories are plotted versus time in Figures 5.3.3-1 and 5.3.3-3.

The final main engine cut-off (MECO) conditions for the reference trajectories were identical. The conditions were the geocentric radius from the earth's center, the inertial velocity, and the inertial flight path angle. In Figure 5.3.3-1, a difference in the geodetic altitude may be seen. Since an oblate model was used for the earth, the altitude above the surface varies with latitude. So, equal distances in radius will produce different altitudes above the surface. The apparent discrepancy in the relative velocity is a similar situation. The inertial velocities for each trajectory are identical at MECO. Due to the slight differences in altitude, and thus, atmospheric density, there will be differences in the relative velocities.

Additionally, the mission profiles are depicted in Figures 5.3.3-4 and 5.3.3-5. Trajectory parameters for several key events are shown with a comparison to a typical STS mission.

Table 5.3.3-1: Reference Trajectory Comparison

CLV-4 Referen	ce Trajector	y Comparis	son (Rev. 1 &	<u>2)</u>
Trajectory description	REV 1	REV 1	REV 2	REV 2
Mission description	Due east crew	ISS Crew	Due east crew	ISS Crew
Gross mass at SRB ignition (lbm)	1,788,446	1,784,573	1,805,621	1,801,867
SRB usable propellant	1,111,917	1,111,917	1,111,916	1,111,916
SRB jettison mass	190,942	190,942	193,392	193,392
SRB-to-Interstage Adapter			18,940	18,940
Interstage	7,650	7,650	6,570	6,570
2nd Stage usable ascent LOX	309,200	309,240	309,249	309,288
2nd Stage usable ascent LH2	51,268	51,275	51,277	51,283
Launch Escape System (LES)	9,296	9,296	9,296	9,296
Injected mass (lbm)	108,172	104,252	104,982	101,181
2nd Stage usable FPR LOX	1,870	1,829	1,821	1,781
2nd Stage usable FPR LH2	310	303	302	295
2nd Stage usable fuel bias LH2	534	534	534	534
2nd Stage residual propellant	3,224	3,224	3,224	3,224
2nd Stage APU reactants & helium purge	120	120	120	120
2nd Stage pressurant gases	381	381	380	380
2nd Stage pressurant gases 2nd Stage dry weight	44,008	44,008	40,847	40,847
RCS propellant, reserves & residuals	301	301	317	317
Upper Stage-to-Spacecraft Adapter	501	301	3,223	3,223
Gross payload to delivery orbit (lbm)	57,424	53,551	54,214	50,459
Payload margin (10%)	5,742	5,355	5,421	5,046
Net payload to delivery orbit (lbm)	51,682	48,196	48,792	45,414
Net payload to delivery orbit (mt)	23.44	21.86	22.13	20.60
Net payload to delivery orbit (htt)	23,442	21,861	22,132	20,599
Net payload to derivery orbit (kg)	23,442	21,001	22,132	20,399
Total 2nd Stage usable propellant	363,182	363,182	363,182	363,182
Total 2nd Stage usable LOX	311,070	311,070	311,070	311,070
Total 2nd Stage usable LH2	52,113	52,113	52,113	52,113
TOWN A LIGHT CO.	1000	1.000	1.000	1 000
F/W at lift-off	1.000	1.000	1.000	1.000
F/W at 0.6 sec after SRB ignition	1.612	1.616	1.582	1.585
F/W prior to SRB staging	0.020	0.020	0.019	0.019
F/W after SRB staging (RPL=100%)	0.980	0.988	0.976	0.984
Launch azimuth (deg)	85.9	49.4	85.8	50.0
Total burn time (sec)	470.1	470.2	473.0	473.1
Total ascent ideal ÄV (fps)	29,618	30,067	29,708	30,156
Maximum dynamic pressure (psf)	823	832	776	783
Mach Number at Max. Q	1.32	1.34	1.26	1.28
Maximum acceleration (g's)	4.54	4.71	4.63	4.80
Maximum Q-alpha (psf-deg)	633	624	667	657
Maximum Q-beta (psf-deg)	63	14	65	15
	100.05	100.00	10004	10001
SRB jettison time (sec)	128.06	128.06	128.04	128.04
SRB jettison altitude (ft)	178,303	179,458	173,666	174,723
SRB jettison Mach Number	4.51	4.53	4.25	4.26
SRB relative fpa (deg)	25.17	25.28	25.87	25.97
SRB thrust prior to jettison (lbf)	15,000	15,000	15,000	15,000
SRB maximum altitude (ft)	249,142	251,474	240,626	242,704
SRB maximum allowable altitude (ft)	233,173	233,173	225,824	225,824
LES jettison time (sec)	213.3	210.8	220.0	217.5
EES jettison time (see)				
LES jettison time (see)	300,000	300,000	300,000	300,000

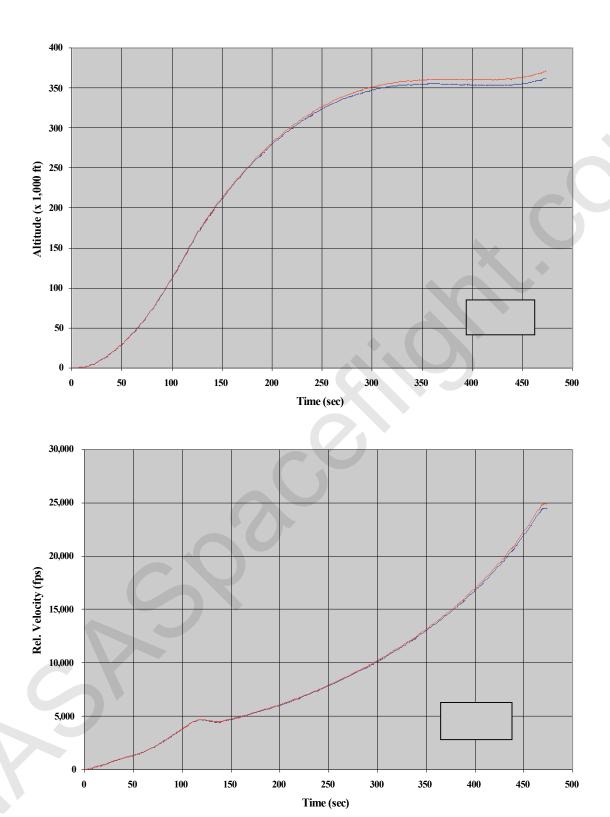
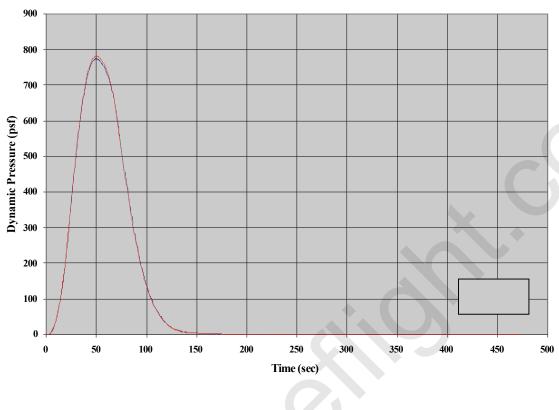


Figure 5.3.3-1: Rev. 2 Reference Trajectory Plots



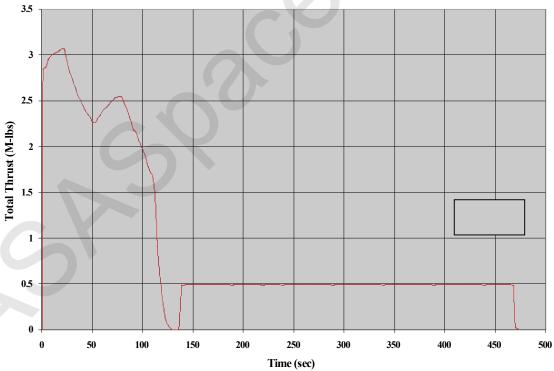


Figure 5.3.3-2: Rev. 2 Reference Trajectory Plots



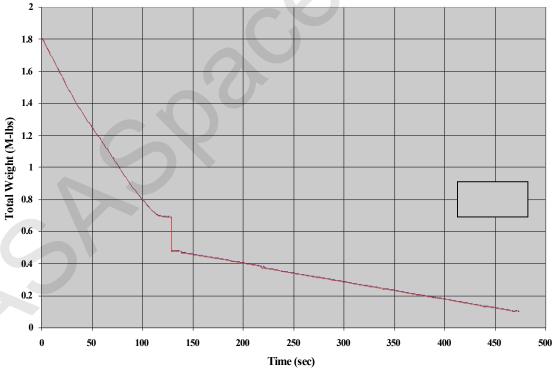


Figure 5.3.3-3: Rev. 2 Reference Trajectory Plots

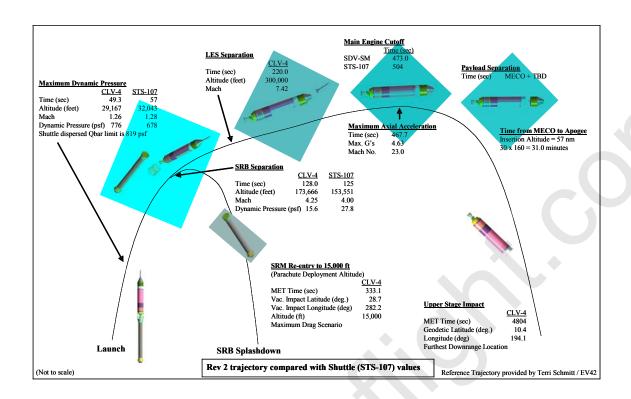


Figure 5.3.3-4: Due East Mission Profile for Rev. 2 Reference Trajectory

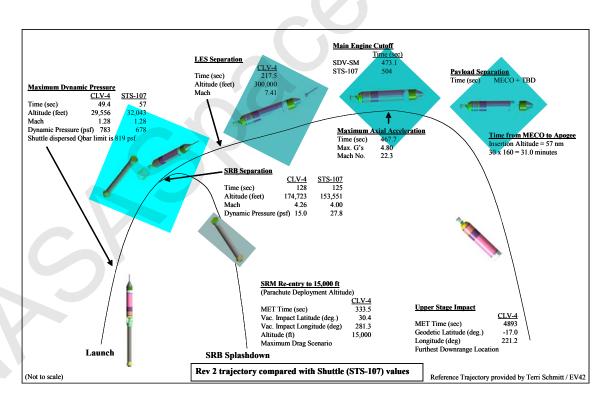


Figure 5.3.3-5: ISS Mission Profile for Rev. 2 Reference Trajectory

#### 5.3.4 TRAJECTORY TRADE STUDIES

Several studies were conducted during this analysis cycle. The parametric effects and the SSME power level studies were conducted with the Revision 2 trajectory. The SRB studies used the Rev. 1 reference trajectories.

## 5.3.4.1 PARAMETRIC EFFECTS ON PAYLOAD (Rev. 2 Trajectory)

The effects of different design parameters on the delivered payload were investigated. This analysis allows the user to see the impact of off-design conditions on the delivered payload. In this study, the Rev. 2 trajectory was used.

For clarity in the following figures, the parameters are grouped into three categories. Each parameter has been compared to the delivered payload for the reference trajectory. In each of these scenarios, only the payload capability was determined. Any effects on other flight phases, such as the abort situations, have not been analyzed.

Figure 5.3.4-1 shows the parameters where the component weights have been changed. These component changes are

- 1. A 1,000 lb increase in the booster's jettison weight.
- 2. A 1,000 lb increase in the Launch Escape System.
- 3. A 10,000 lb increase in the 2<sup>nd</sup> stage propellant, but no increase in the stage weight.
- 4. A 20,000 lb increase in the 2<sup>nd</sup> stage propellant, but no increase in the stage weight.
- 5. A 10,000 lb increase in the 2<sup>nd</sup> stage propellant and increase in the stage weight. A constant stage mass fraction (without the engine weight) was used to determine the new stage weight.
- 6. A 20,000 lb increase in the 2<sup>nd</sup> stage propellant and increase in the stage weight. A constant stage mass fraction (without the engine weight) was used to determine the new stage weight.

The increased propellant cases show an increased delivery capability. However, these situations do not account for any structural growth of the upper stage necessary to accommodate the increased propellant. Also shown on this chart is the net effect of increasing the propellant and corresponding increase to the structural weight of the upper stage. The larger propellant capability requires an increased structure that effectively negates any payload increase. Figure 5.3.4-2 shows the effect on the payload capability when structural growth is permitted. A constant stage mass fraction was used throughout.

In Figure 5.3.4-3, the effects of different trajectory shaping parameters are shown. The varied elements are

- 1. A 10% increase in the aerodynamic reference area.
- 2. A 10% increase in the base force.
- 3. The LES jettison altitude is increased by 2,000 feet, which is approximately a 2 second delay.
- 4. The  $1^{st}$  stage separation time is reduced by 2 seconds. The separation criterion becomes the time when the thrust =  $38,000 \text{ lb}_f$ .
- 5. Limit the maximum acceleration to 3.8 g's.
- 6. Limit the maximum acceleration to 3.2 g's.

Of these six elements, only the change in the booster separation time assists the vehicle's payload capability. In this case, the separation thrust of the first stage is greater than the reference trajectory. The effect of this higher thrust on the success of stage separation was not addressed.

When the engine characteristics are changed in either stage, there will be an effect on the payload capability. These sensitivities are shown in Figure 5.3.4-4. The parameters shown are

- A 1% increase in booster's thrust level with no changes to the flow rate. When the flow rate is left unchanged, the increase in thrust effectively increases the lsp.
- 2. A 2,000 lb increase in the booster's propellant. The lsp is assumed to be constant.
- A 4° increase in the propellant mean bulk temperature of the booster. The
  equations to modify the burn time, thrust, and mass flow rate tables may
  be found in NSTS 08509, Volume I.
- 4. Increase the nominal power level of the 2<sup>nd</sup> stage engine to 106%.
- 5. A 1 second increase in the specific impulse of the SSME.

Since only increases in the parameters are shown, the effect on the payload capability also increases. Reduction of the engine characteristics will result in a decrease in the payload.

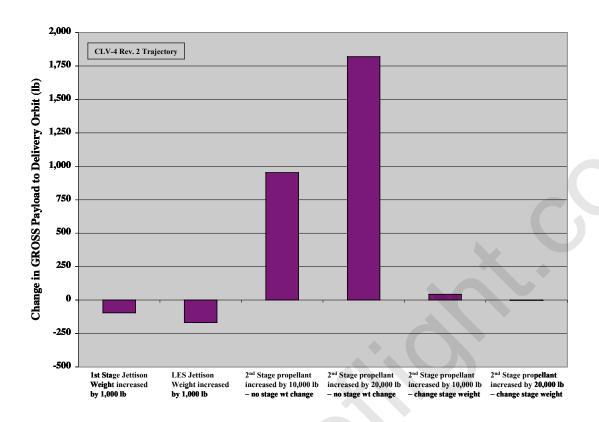
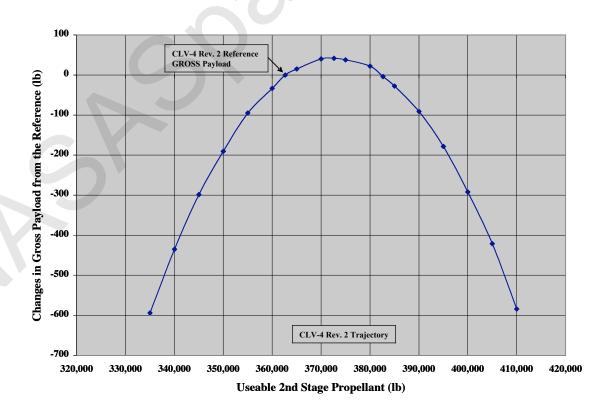


Figure 5.3.4-1: Parametric Effects on Delivered Payload (Rev. 2 Trajectory)





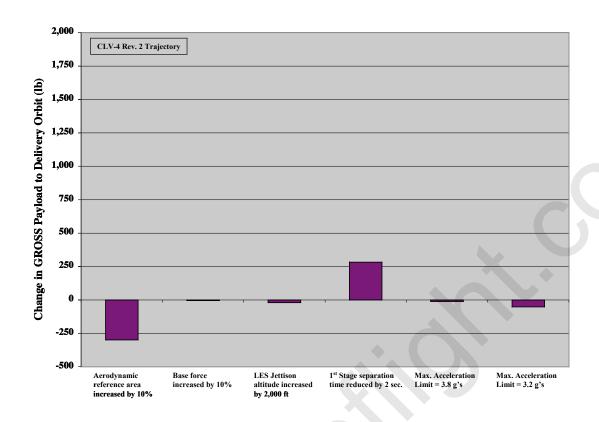
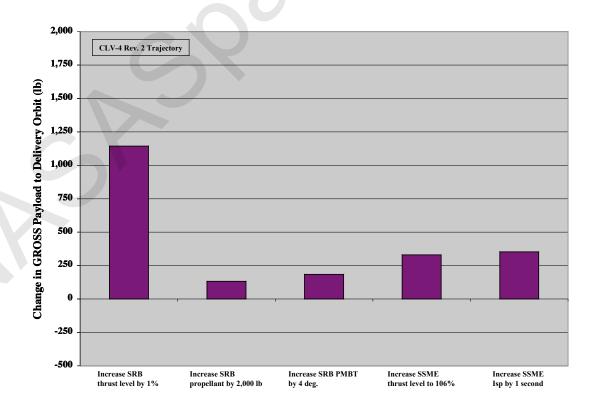


Figure 5.3.4-3: Parametric Effects on Delivered Payload (Rev. 2 Trajectory)





## 5.3.4.2 SRB TRADES (Rev. 1 Trajectory)

The reference design for the launch vehicle uses an inline solid rocket motor. This motor is a modification of the SRM currently flying with the Space Shuttle. The motor consists of 4 segments and it uses PBAN propellant. The reference data used in this analysis assumes a 60 deg. propellant mean bulk temperature (PMBT) and the KSC mean annual winds. For these trade studies, the Rev. 1 reference trajectory was used.

Several trade studies concerning the first stage have been conducted. These include the PMBT of the motor, the type of propellant and the number of segments, and the effects of knockdown. The knockdown phenomenon is described in Section 5.3.2.5.

Changing the temperature of the propellant impacts the payload capability of the launch vehicle as shown in Table 5.3.4-1. The reference motor is compared to motors typically seen in February (cold) and July (hot). For each of these trajectory runs, the corresponding KSC monthly winds were incorporated. These trajectory results were provided to the 6-DOF simulation.

As potential payloads for this vehicle become heavier, the delivery capability needs to be augmented. Two possible upgrades were investigated: (1) use HTPB as the propellant in the 4-segment booster, and (2) use a 5-segment booster. The 5-segment booster also uses the HTPB propellant. The payload increases with these upgrades can be seen in Table 5.3.4-2.

To more accurately model the SRB, the effect of knockdown was incorporated into the simulation. The modifications are based on trajectory reconstruction results from analysis and flight data. The effect on the CLV-4 Rev. 1 trajectory is shown in Table 5.3.4-3.

Table 5.3.4-1: SRB Temperature Trade Study Results (Rev. 1 Trajectory)

	CLV-4 SKI	o rempera	ture Trade	(Nev. I)		
Trajectory description Mission description	Reference Due east crew	Reference ISS crew	February Motor Due east crew	February Motor ISS crew	July Motor Due east crew	July Motor ISS crew
Gross mass at SRB ignition (lbm)	1,788,446	1,784,573	1,788,757	1,784,801	1,789,079	1,785,259
SRB usable propellant	1,111,917	1,111,917	1,111,917	1,111,917	1,111,919	1,111,919
SRB jettison mass	190,942	190,942	190,943	190,943	190,941	190,941
Interstage	7,650	7,650	7,650	7,650	7,650	7,650
2nd Stage usable ascent LOX	309,200	309,240	309,196	309,238	309,193	309,233
2nd Stage usable ascent LH2	51,268	51,275	51,268	51,275	51,267	51,274
Launch Escape System (LES)	9,296	9,296	9,296	9,296	9,296	9,296
Injected mass (lbm)	108,172	104,252	108,487	104,482	108,813	104,946
2nd Stage usable FPR LOX	1,870	1,829	1,873	1,832	1,877	1,837
2nd Stage usable FPR LH2	310	303	311	304	311	305
2nd Stage usable fuel bias LH2	534	534	534	534	534	534
2nd Stage residual propellant	3.224	3.224	3.224	3.224	3.224	3.224
2nd Stage APU reactants & helium purge	120	120	120	120	120	120
2nd Stage pressurant gases	381	381	381	381	381	381
2nd Stage dry weight	44,008	44.008	44,008	44,008	44,008	44,008
RCS propellant, reserves & residuals	301	301	301	301	301	301
Gross payload to delivery orbit (lbm)	57,424	53,551	57,735	53,779	58,057	54,237
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Payload margin (10%)	5,742	5,355	5,773	5,378	5,806	5,424
Net payload to delivery orbit (lbm)	51,682	48,196	51,961	48,401	52,251	48,813
Net payload to delivery orbit (mt)	23.44	21.86	23.57	21.95	23.70	22.14
Net payload to delivery orbit (kg)	23,442	21,861	23,569	21,954	23,701	22,141
Change from GROSS payload (lb)			310	227	633	686
Change from GROSS payload (mt)			0.14	0.10	0.29	0.31
Change from GROSS payload (kg)			141	103	287	311
Total 2nd Stage usable propellant	363,182	363,182	363,182	363,182	363,182	363,182
Total 2nd Stage usable LOX	311,070	311,070	311,070	311,070	311,070	311,070
Total 2nd Stage usable LH2	52,113	52,113	52,113	52,113	52,113	52,113
F/W at lift-off	1.000	1.000	1.000	1.000	1.000	1.000
F/W at 0.6 sec after SRB ignition	1.612	1.616	1.614	1.617	1.652	1.655
F/W prior to SRB staging	0.020	0.020	0.020	0.020	0.020	0.020
F/W after SRB staging	0.980	0.988	0.980	0.988	0.979	0.987
Launch azimuth (deg)	85.9	49.4	87.2	56.1	84.3	40.9
Total burn time (sec)	470.1	470.2	470.0	470.1	467.5	467.5
Total ascent ideal ÄV (fps)	29,618	30,067	29,583	30,040	29,555	29,994
Maximum dynamic pressure (psf)	823	832	813	822	882	888
Mach Number at Max. Q	1.32	1.34	1.29	1.32	1.40	1.41
Maximum acceleration (g's)	4.54	4.71	4.52	4.70	4.51	4.68
Maximum Q-alpha (psf-deg)	633	624	-768	-745	674	661
Maximum Q-beta (psf-deg)	63	14	58	-82	152	50
SRB jettison time (sec)	128.06	128.06	127.93	127.93	125.40	125.40
SRB jettison altitude (ft)	178,303	179.458	177,899	179.135	177,688	178,653
SRB jettison Mach Number	4.51	4.53	4.47	4.50	4.72	4.70
SRB relative fpa (deg)	25.17	25.28	25.04	25.18	25.20	25.29
SRB thrust prior to jettison (lbf)	15,000	15,000	15,000	15,000	15,000	15,000
SRB maximum altitude (ft)	249,142	251,474	248,873	251,268	250,174	252,465
SRB maximum allowable altitude (ft)	233,173	233,173	233,172	233,172	233,178	232,463
LES jettison time (sec)	213.3	210.8	213.4	210.8	209.6	207.2
LES jettison altitude (ft)	300,000 7.42	300,000 7.40	300,000 7.45	300,000 7.42	300,000 7.35	300,000 7.37
LES jettison Mach Number	1.42	7.40	7.45	1.42	1.33	1.31

Table 5.3.4-2: SRB Propellant Type and Segment Study Results (Rev. 1 Trajectory)

	Reference	Reference	HTPB SRB	HTPB SRB	5-Segment	5-Segment
rajectory description  Mission description	Due east crew	ISS crew	Due east crew	ISS crew	Due east crew	ISS crew
RB Segments	4	4	4	4	5	5
ropellant Type	PBAN	PBAN	НТРВ	нтрв	НТРВ	НТРВ
Topellain Type	12	15.11		2		
ross mass at SRB ignition (lbm)	1,788,446	1,784,573	1,820,226	1,816,254	2,159,732	2,155,313
SRB usable propellant	1,111,917	1.111.917	1,139,943	1,139,943	1,436,362	1,436,362
SRB jettison mass	190,942	190,942	192,349	192,349	225,968	225,968
Interstage	7,650	7,650	7,650	7,650	7,650	7,650
2nd Stage usable ascent LOX	309.200	309.240	309.163	309.205	309.004	309.050
2nd Stage usable ascent LH2	51,268	51,275	51,262	51,269	51,236	51,244
Launch Escape System (LES)	9,296	9,296	9,296	9,296	9,296	9,296
jected mass (lbm)	108,172	104,252	110,563	106,543	120,217	115,744
2nd Stage usable FPR LOX	1,870	1,829	1,907	1,865	2,066	2,019
2nd Stage usable FPR LH2	310	303	316	309	343	335
2nd Stage usable fuel bias LH2	534	534	534	534	534	534
2nd Stage residual propellant	3,224	3,224	3,224	3,224	3,224	3,224
2nd Stage APU reactants & helium purge	120	120	120	120	120	120
2nd Stage Pressurant gases	381	381	381	381	381	381
2nd Stage pressurant gases 2nd Stage dry weight	44,008	44,008	44,008	44,008	44,008	44,008
RCS propellant, reserves & residuals	301	301	301	301	301	301
ross payload to delivery orbit (lbm)	57,424	53,551	59,772	55,801	69,240	64,822
Payload margin (10%)	5,742	5,355	5,977	5,580	6,924	6,482
et payload to delivery orbit (lbm)	51,682	48,196	53,795	50,220	62,316	58,340
et payload to delivery orbit (mt)	23.44	21.86	24.40	22.78	28.27	26.46
et payload to delivery orbit (kg)	23,442	21,861	24,401	22,779	28,266	26,462
ifference in GROSS payload (lb) ifference in GROSS payload (mt) ifference in GROSS payload (kg)			2,347 1.06 1,065	2,249 1.02 1,020	11,816 5.36 5,360	11,270 5.11 5,112
Cotal 2nd Stage usable propellant	363,182	363,182	363,182	363,182	363,182	363,182
Total 2nd Stage usable LOX	311,070	311,070	311,070	311,070	311,070	311,070
Total 2nd Stage usable LH2	52,113	52,113	52,113	52,113	52,113	52,113
W at lift-off	1.000	1.000	1.000	1.000	1.000	1.000
W at 0.6 sec after SRB ignition	1.612	1.616	1.590	1.594	1.655	1.658
W prior to SRB staging	0.020	0.020	0.019	0.020	0.020	0.020
	0.980	0.988	0.975	0.984	0.958	0.967
	0.760					
W after SRB staging	85.9	49.4	85.9	49.3	85.8	49.6
W after SRB staging nunch azimuth (deg) otal burn time (sec)	85.9 470.1	470.2	470.0	470.1	483.6	49.6 483.7
W after SRB staging nunch azimuth (deg) otal burn time (sec)	85.9					
W after SRB staging nunch azimuth (deg) otal burn time (sec) otal ascent ideal ÅV (fps)	85.9 470.1	470.2	470.0	470.1	483.6	483.7
W after SRB staging  aunch azimuth (deg)  otal burn time (sec)  otal ascent ideal ÅV (fps)  (aximum dynamic pressure (psf)	85.9 470.1 29,618	470.2 30,067	470.0 29,547	470.1 29,996	483.6 29,442	483.7 29,895
W after SRB staging aunch azimuth (deg) otal burn time (sec) otal ascent ideal ÄV (fps) faximum dynamic pressure (psf) ach Number at Max. Q	85.9 470.1 29,618	470.2 30,067 832	470.0 29,547 833	470.1 29,996 842	483.6 29,442 746	483.7 29,895 753
W after SRB staging aunch azimuth (deg) otal burn time (sec) otal ascent ideal ÄV (fps)  (aximum dynamic pressure (psf) (ach Number at Max. Q (aximum acceleration (g's)	85.9 470.1 29,618 823 1.32	470.2 30,067 832 1.34	470.0 29,547 833 1.34	470.1 29,996 842 1.36	483.6 29,442 746 1.24	483.7 29,895 753 1.26
W after SRB staging nunch azimuth (deg) stal burn time (sec) stal ascent ideal ÅV (fps) aximum dynamic pressure (psf) ach Number at Max. Q aximum acceleration (g's) aximum Q-alpha (psf-deg)	85.9 470.1 29,618 823 1.32 4.54	470.2 30,067 832 1.34 4.71	470.0 29,547 833 1.34 4.44	470.1 29,996 842 1.36 4.61	483.6 29,442 746 1.24 4.08	483.7 29,895 753 1.26 4.24
W after SRB staging nunch azimuth (deg) total burn time (sec) total ascent ideal ÄV (fps) aximum dynamic pressure (psf) ach Number at Max. Q aximum acceleration (g's) aximum Q-alpha (psf-deg) aximum Q-beta (psf-deg)	85.9 470.1 29,618 823 1.32 4.54 633	470.2 30,067 832 1.34 4.71 624	470.0 29,547 833 1.34 4.44 642	470.1 29,996 842 1.36 4.61 629	483.6 29,442 746 1.24 4.08 606	483.7 29,895 753 1.26 4.24 593
W after SRB staging nunch azimuth (deg) stal burn time (sec) stal ascent ideal ÄV (fps) aximum dynamic pressure (psf) ach Number at Max. Q aximum acceleration (g's) aximum Q-alpha (psf-deg) aximum Q-beta (psf-deg) RB jettison time (sec)	85.9 470.1 29,618 823 1.32 4.54 633 63	470.2 30,067 832 1.34 4.71 624 14	470.0 29,547 833 1.34 4.44 642 63	470.1 29,996 842 1.36 4.61 629 14	483.6 29,442 746 1.24 4.08 606 60	483.7 29,895 753 1.26 4.24 593 14 141.73
W after SRB staging aunch azimuth (deg) otal burn time (sec) otal ascent ideal ÅV (fps) (aximum dynamic pressure (psf) (ach Number at Max. Q aximum acceleration (g's) (aximum Q-alpha (psf-deg) (aximum Q-beta (psf-deg) (aximum Q-beta (psf-deg) (aximum Q-beta (psf-deg) (aximum Q-beta (psf-deg) (aximum Q-beta (psf-deg) (ARB jettison time (sec) (ARB jettison altitude (ft)	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504	470.1 29,996 842 1.36 4.61 629 14 127.98 177,664	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402
W after SRB staging aunch azimuth (deg) otal burn time (sec) otal ascent ideal ĀV (fps) faximum dynamic pressure (psf) fach Number at Max. Q faximum acceleration (g's) faximum Q-alpha (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg)	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458 4.53	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504 4.69	470.1 29,996 842 1.36 4.61 629 14 127.98 177,664 4.71	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840 5.64	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402 5.69
W after SRB staging aunch azimuth (deg) otal burn time (sec) otal ascent ideal ÄV (fps) faximum dynamic pressure (psf) fach Number at Max. Q faximum acceleration (g's) faximum Q-alpha (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg)	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458 4.53 25.28	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504 4.69 24.69	470.1 29,996 842 1.36 4.61 629 14 127.98 177,664 4.71 24.80	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840 5.64 20.90	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402 5.69 21.04
W after SRB staging nunch azimuth (deg) btal burn time (sec) btal ascent ideal ÅV (fps) aximum dynamic pressure (psf) ach Number at Max. Q aximum acceleration (g's) aximum Q-alpha (psf-deg) aximum Q-beta (psf-deg) aximum Q-beta (psf-deg) RB jettison time (sec) RB jettison altitude (ft) RB jettison Mach Number RB relative fpa (deg) RB thrust prior to jettison (lbf)	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458 4.53 25.28 15,000	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504 4.69 24.69 15,000	470.1 29,996 842 1.36 4.61 629 14 127.98 177,664 4.71 24.80 15,000	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840 5.64 20.90 15,000	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402 5.69 21.04 15,000
W after SRB staging aunch azimuth (deg) botal burn time (sec) botal ascent ideal ÅV (fps) aximum dynamic pressure (psf) ach Number at Max. Q aximum acceleration (g's) aximum Q-alpha (psf-deg) aximum Q-beta (psf-deg) aximum Q-beta (psf-deg) RB jettison time (sec) RB jettison altitude (ft) RB jettison Mach Number RB relative fpa (deg) RB thrust prior to jettison (lbf) RB maximum altitude (ft)	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458 4.53 25.28	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504 4.69 24.69	470.1 29,996 842 1.36 4.61 629 14 127.98 177,664 4.71 24.80	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840 5.64 20.90	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402 5.69 21.04
W after SRB staging aunch azimuth (deg) total burn time (sec) total ascent ideal ÄV (fps) faximum dynamic pressure (psf) fach Number at Max. Q faximum acceleration (g's) faximum Q-alpha (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum G-beta (psf-deg) faximum G-beta (ft) faximum G-beta (ft) faximum G-beta (ft) faximum faximum altitude (ft) faximum altitude (ft) faximum altitude (ft) faximum altitude (ft)	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000 249,142 233,173	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458 4.53 25.28 15,000 251,474 233,173	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504 4.69 24.69 15,000 250,546 228,953	470.1 29,996 842 1.36 4.61 629 14 127,98 177,664 4.71 24.80 15,000 252,954 228,953	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840 5.64 20.90 15,000 278,724 263,000	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402 5.69 21.04 15,000 281,714 263,000
W after SRB staging aunch azimuth (deg) otal burn time (sec) otal ascent ideal ĀV (fps) faximum dynamic pressure (psf) faxh Number at Max. Q faximum acceleration (g's) faximum Q-alpha (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) faximum Q-beta (psf-deg) fax jettison time (sec) fRB jettison Mach Number fRB relative fpa (deg) fRB thrust prior to jettison (lbf) fRB maximum altitude (ft) fRB maximum altitude (ft) fRB maximum altitude (ft) fRB maximum altitude (ft) fRB maximum altitude (ft) fRB jettison time (sec)	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000 249,142 233,173 213.3	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458 4.53 25.28 15,000 251,474 233,173	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504 4.69 24.69 15,000 250,546 228,953	470.1 29,996 842 1.36 4.61 629 14 127.98 177,664 4.71 24.80 15,000 252,954 228,953	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840 5.64 20.90 15,000 278,724 263,000	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402 5.69 21.04 15,000 281,714 263,000
/W after SRB staging .aunch azimuth (deg) otal burn time (sec) otal ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Maximum dynamic pressure (psf) Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Q-beta (psf-deg) Maximum Getison (lbf) Maximum altitude (ft) Maximum altitude (ft) Maximum altitude (ft) Maximum (sec) Maximum (s	85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000 249,142 233,173	470.2 30,067 832 1.34 4.71 624 14 128.06 179,458 4.53 25.28 15,000 251,474 233,173	470.0 29,547 833 1.34 4.44 642 63 127.98 176,504 4.69 24.69 15,000 250,546 228,953	470.1 29,996 842 1.36 4.61 629 14 127,98 177,664 4.71 24.80 15,000 252,954 228,953	483.6 29,442 746 1.24 4.08 606 60 141.73 207,840 5.64 20.90 15,000 278,724 263,000	483.7 29,895 753 1.26 4.24 593 14 141.73 209,402 5.69 21.04 15,000 281,714 263,000

Table 5.3.4-3: SRB Knockdown Sensitivity (Rev. 1 Trajectory)

Trajectory description Mission description	Reference Due east crew	SRB Knockdown Due east crew
Wission description	Due cast crew	Due cast crew
Gross mass at SRB ignition (lbm)	1,788,446	1,787,249
SRB usable propellant	1,111,917	1,111,916
SRB jettison mass	190,942	190,944
Interstage	7,650	7,650
2nd Stage usable ascent LOX	309,200	309,218
2nd Stage usable ascent LH2	51,268 9,296	51,271 9,296
Launch Escape System (LES) Injected mass (lbm)	108,172	106,954
2nd Stage usable FPR LOX	1,870	1,852
2nd Stage usable FPR LH2	310	307
2nd Stage usable fuel bias LH2	534	534
2nd Stage residual propellant	3,224	3,224
2nd Stage APU reactants & helium purge	120	120
2nd Stage pressurant gases	381	381
2nd Stage dry weight	44,008	44,008
RCS propellant, reserves & residuals	301	301
Gross payload to delivery orbit (lbm)	57,424	56,227
Payload margin (10%)	5,742	5,623
Net payload to delivery orbit (lbm)	51,682	50,604
Net payload to delivery orbit (mt) Net payload to delivery orbit (kg)	23.44 23,442	22.95 22,954
Difference in GROSS payload (mt)		-1,197 -0.54 -543
Difference in GROSS payload (lb) Difference in GROSS payload (mt) Difference in GROSS payload (kg)		-0.54 -543
Difference in GROSS payload (mt) Difference in GROSS payload (kg) Total 2nd Stage usable propellant	363,182	-0.54 -543 363,182
Difference in GROSS payload (mt) Difference in GROSS payload (kg) Total 2nd Stage usable propellant Total 2nd Stage usable LOX	311,070	-0.54 -543 363,182 311,070
Difference in GROSS payload (mt) Difference in GROSS payload (kg) Total 2nd Stage usable propellant		-0.54 -543 363,182
Difference in GROSS payload (mt) Difference in GROSS payload (kg) Total 2nd Stage usable propellant Total 2nd Stage usable LOX	311,070	-0.54 -543 363,182 311,070
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2	311,070 52,113	-0.54 -543 363,182 311,070 52,113
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging	311,070 52,113 1.000	-0.54 -543 363,182 311,070 52,113 1.000
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging	311,070 52,113 1.000 1.612	-0.54 -543 363,182 311,070 52,113 1.000 1.598
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging	311,070 52,113 1.000 1.612 0.020	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging Launch azimuth (deg) Total burn time (sec)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging Launch azimuth (deg) Total burn time (sec)	311,070 52,113 1.000 1.612 0.020 0.980 85.9	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant Total 2nd Stage usable LOX Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÅV (fps)  Maximum dynamic pressure (psf)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison altitude (ft)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÅV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison altitude (ft) SRB jettison Mach Number	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÅV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison altitude (ft) SRB jettison Mach Number SRB relative fpa (deg)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64 128.04 176,758
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison Mach Number SRB relative fpa (deg) SRB thrust prior to jettison (lbf)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64 128.04 176,758 4.41 25.54 15,000
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison altitude (ft) SRB jettison Mach Number SRB relative fpa (deg) SRB thrust prior to jettison (lbf) SRB maximum altitude (ft)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000 249,142	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64 128.04 176,758 4.41 25.54 15,000 246,656
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison altitude (ft) SRB jettison Mach Number SRB relative fpa (deg) SRB thrust prior to jettison (lbf) SRB maximum altitude (ft)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64 128.04 176,758 4.41 25.54 15,000
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant     Total 2nd Stage usable LOX     Total 2nd Stage usable LOX     Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÅV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison Mach Number SRB relative fpa (deg) SRB thrust prior to jettison (lbf) SRB maximum allowable altitude (ft)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000 249,142 233,173	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64 128.04 176,758 4.41 25.54 15,000 246,656 233,168
Difference in GROSS payload (mt) Difference in GROSS payload (kg)  Total 2nd Stage usable propellant    Total 2nd Stage usable LOX    Total 2nd Stage usable LOX    Total 2nd Stage usable LH2  F/W at lift-off F/W at 0.6 sec after SRB ignition F/W prior to SRB staging F/W after SRB staging  Launch azimuth (deg) Total burn time (sec) Total ascent ideal ÄV (fps)  Maximum dynamic pressure (psf) Mach Number at Max. Q Maximum acceleration (g's) Maximum Q-alpha (psf-deg) Maximum Q-beta (psf-deg)  SRB jettison time (sec) SRB jettison altitude (ft) SRB jettison Mach Number SRB relative fpa (deg) SRB thrust prior to jettison (lbf) SRB maximum altitude (ft)	311,070 52,113 1.000 1.612 0.020 0.980 85.9 470.1 29,618 823 1.32 4.54 633 63 128.06 178,303 4.51 25.17 15,000 249,142	-0.54 -543 363,182 311,070 52,113 1.000 1.598 0.020 0.983 85.9 470.2 29,665 802 1.29 4.59 642 64 128.04 176,758 4.41 25.54 15,000 246,656

## 5.3.4.3 SSME POWER LEVEL TRADES (Rev. 2 Trajectory)

For the Rev. 2 reference trajectory, the upper stage follows a start transient, which is detailed in Section 5.3.2.3, until it reaches 100%. This is followed by a thrust ramp-up of 10% per second to the nominal power level (NPL). The reference NPL for these studies is 104.5%. Since the upper stage is an expendable component, increasing the operating thrust will help the payload capability of the vehicle. The effect of the increased thrust is shown in Table 5.3.4-4. As shown in Figure 5.3.4-5, the change in the payload capability is approximately linear between the power levels of 104.5% and 111%. Higher thrust levels may exceed the maximum acceleration limit of 5.0-g's and then throttling would be required.

The decision to increase the thrust level becomes a programmatic decision. Although the payload capability is increased, the reliability of the engine is believed to be slightly reduced.

Table 5.3.4-4: Upper Stage Nominal Power Level Study Results (Rev. 2 Trajectory)

Trajectory description	REFERENCE - REV. 2	Rev 2	Rev 2	Rev 2
Mission description	Due east crew	Due East	Due East	Due East
2nd Stage Power Level	NPL = 104.5%	NPL = 106%	NPL = 109%	NPL = 111%
Gross mass at SRB ignition (lbm)	1,805,621	1,805,951	1,806,574	1,806,957
SRB usable propellant	1,111,916	1,111,916	1,111,916	1,111,916
SRB jettison mass	193,392	193,392	193,392	193,392
SRB-to-Interstage Adapter	18,940	18,940	18,940	18,940
Interstage	6,570	6,570	6,570	6,570
2nd Stage usable ascent LOX	309,249	309,245	309,239	309,235
2nd Stage usable ascent LH2	51,277	51,276	51,275	51,274
Launch Escape System (LES)	9,296	9,296	9,296	9,296
njected mass (lbm)	104,982	105,316	105,947	106,335
2nd Stage usable FPR LOX	1,821	1,824	1,831	1,835
2nd Stage usable FPR LH2	302	302	304	304
2nd Stage usable fuel bias LH2	534	534	534	534
2nd Stage residual propellant	3,224	3,224	3,224	3,224
2nd Stage APU reactants & helium purge		120	120	120
2nd Stage pressurant gases	380	380	380	380
2nd Stage dry weight	40,847	40,847	40,847	40,847
RCS propellant, reserves & residuals	317	317	317	317
Upper Stage-to-Spacecraft Adapter	3,223	3,223	3,223	3,223
Gross payload to delivery orbit (lbm)	54,214	54,544	55,167	55,550
Payload margin (10%)	5,421	5,454	5,517	5,555
Net payload to delivery orbit (lbm)	48,792	49,090	49,650	49,995
Net payload to delivery orbit (mt)	22.13	22.27	22.52	22.68
Net payload to delivery orbit (kg)	22,132	22,267	22,521	22,677
Change in GROSS payload (lb) Change in GROSS payload (mt) Change in GROSS payload (kg)		330 0.15 150	953 0.43 432	1,337 0.61 606
Total 2nd Stage usable propellant	363,182	363,182	363,182	363,182
Total 2nd Stage usable LOX	311,070	311,070	311,070	311,070
Total 2nd Stage usable LH2	52,113	52,113	52,113	52,113
F/W at lift-off	1.000	1.000	1.000	1.000
F/W at 0.6 sec after SRB ignition	1.582	1.582	1.581	1.581
F/W prior to SRB staging	0.019	0.019	0.019	0.019
// W after SRB staging (RPL=100%)	0.976	0.976	0.974	0.974
Launch azimuth (deg)	85.8	85.8	85.9	85.9
Total burn time (sec)	473.0	468.3	459.3	453.6
Total ascent ideal ÄV (fps)	29,708	29,669	29,597	29,552
Maximum dynamic pressure (psf)	776	776	776	776
Mach Number at Max. Q	1.26	1.27	1.27	1.27
Altitude at Max. Q (ft)	29,167	29,183	29,214	29,233
Maximum acceleration (g's)	4.63	4.68	4.79	4.86
Maximum Q-alpha (psf-deg)	667	665	667	666
Maximum Q-beta (psf-deg)	65	65	66	66
st Stage Max. acceleration (g's)	2.53	2.53	2.53	2.53
nd Stage Max. acceleration (g's)	4.63	4.68	4.79	4.86
RB jettison time (sec)	128.04	128.04	128.04	128.04
SRB jettison altitude (ft)	173,666	173,204	172,321	171,778
SRB jettison Mach Number	4.25	4.25	4.24	4.24
SRB relative fpa (deg)	25.87	25.70	25.39	25.19
SRB thrust prior to jettison (lbf)	15,000	15,000	15,000	15,000
RB maximum altitude (ft)	240,626	239,447	237,229	235,864
RB maximum allowable altitude (ft)	225,824	225,824	225,824	225,824
LES jettison time (sec)	220.0	221.2	223.3	224.6
ES jettison altitude (ft)	300,000	300,000	302,000	300,000
LES jettison rel. velocity (fps)	6,711	6,808	7,000	7,127

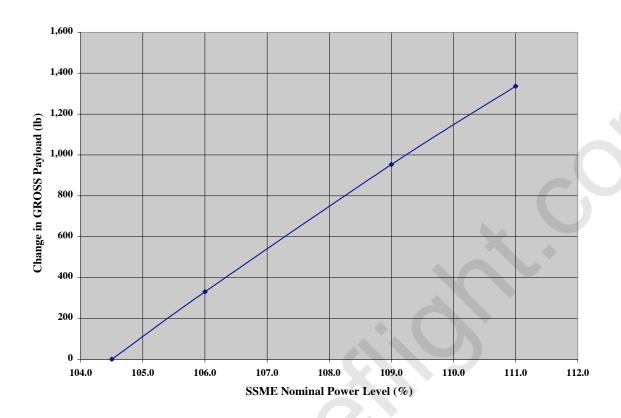


Figure 5.3.4-5: Influence of 2<sup>nd</sup> Stage Thrust on the Delivered Payload (Rev. 2 Trajectory)

#### 5.3.5 REENTRY STUDIES

The reentry of the booster and the upper stage were investigated. For the first stage, the trajectory began at the separation point in the reference trajectory. The booster stage was rotated for collision avoidance and it continued on a ballistic trajectory until impact. The upper stage reentry begins at the MECO location. The upper stage and the payload coast for 30 seconds prior to separation. After separation, the upper stage passively reenters the atmosphere and impacts into the Pacific Ocean.

For both reentry studies, the variation in impact location was incorporated. Two drag scenarios for the 1<sup>st</sup> stage were used and changes in the atmosphere were used for the 2<sup>nd</sup> stage flight.

## 5.3.5.1 1st STAGE REENTRY

The current ground rules for this study assume the SRM used in the first stage is recoverable. The reentry of the 1<sup>st</sup> stage was modeled from separation to 15,000 feet, which is the altitude where the parachutes are deployed. A pitch maneuver was incorporated to assure the booster would not re-contact the upper stage. This pitch event continued until the attitude reached the typical SRB reentry profile.

The reentry aerodynamics, as shown in Table 5.3.5-1, was provided by Bruce McWhorter / ATK Thiokol. Since zero bank angle was assumed for this study, only an excerpt of Universal Space Alliance's (USA) wind tunnel derived database for the solid rocket booster, Aero7N, is included in this document. The data contained in it matches the measured SRB reentry data (altitude, velocity, angle of attack, etc. versus time) very well.

The reentry profile of the first stage was based on the STS SRB reentry in document SE-019-053-2H and entry angle of attack ranges provided by Bruce McWhorter / ATK Thiokol. To model the behavior, an angle of attack profile was generated using this data and it is shown in Table 5.3.5-2. The two profiles shown approximate the high drag and low drag scenarios.

Figure 5.3.5-1 shows the reentry profiles of the two drag scenarios for each inclination. The downrange distance from the launch site ranges from 145 to 152 nm. The vacuum impact locations may be seen on the full flight profile in Figure 5.3.5-2.

Table 5.3.5-1: SRB Reentry Aerodynamics (3-DOF) for Bank Angle = 0°

# Solid Rocket Booster Reentry Aerodynamics (3-DOF)

# Bank Angle = 0°

Data Source: Aero7N Provided by Bruce McWhorter / ATK Thiokol

# $Aref = 116.2 ft^2$

				AIEI -	110.Z IL				
Mach	Alpha	CA	CN	CY	Mach	Alpha	CA	CN	CY
0.40	0	0.8605000	0.0713000	-0.0005000	0.50	0	0.8605000	0.0505000	-0.0 <del>009</del> 000
0.40				-0.0513000	0.50				
	5	0.9012000	0.4768000			5	0.9268000	0.4490000	-0.0551000
	10	0.9883000	0.9127000	-0.0717999		10	1.0185000	0.9696000	-0.0186000
	15	1.0352000	1.4309000	0.0121000		15	1.0530000	1.5072000	0.0121000
	20	1.0154000	2.0045000	0.1208000		20	1.0333000	2.0448000	0.0717000
	25	0.9692000	2.6503000	0.1845000		25	0.9980000	2.6732000	0.0289000
	30	0.8824000	3.4020000	-0.3690000		30	0.9319000	3.3178000	-0.5853000
	35	0.7665000	4.1666000	-1.8917000		35	0.8438000	4.3123000	-1.8869000
	40	0.6255000	5.1287000	-3.6734000		40	0.7051000	5.3289000	-3.1460000
	45	0.4804000	6.3395000	-5.3965000		45	0.5495000	6.5463000	-4.7467000
	50	0.3354000	7.4120000	-6.2266000		50	0.3772000	7.8978000	-5.2879500
	55	0.1576000	8.2280000	-6.4827000		55	0.1794000	8.7217000	-5.5700000
	60	-0.0002586	8.8469000	-5.8651000		60	-0.0502000	9.3803000	-5.8400000
	65	-0.1944907	9.2371667	-6.2385000		65	-0.2266667	9.6667667	-5.9722500
	70	-0.3538093	9.5513333	-7.9078700		70	-0.3056333	9.8434333	-6.1044900
	75	-0.4572020	9.8897000	-8.3015800		75	-0.3643040	10.2173000	-5.8536600
	80	-0.4226547	10.1247667	-7.0951800		80	-0.2533697	10.6046667	-5.5200000
	85	0.2588967	10.1972333	-6.8289100		85	0.4208757	10.8320333	-5.2226100
	90	0.1783920	10.2000000	-6.2883500		90	0.3417960	10.9501000	-4.9252200
	95	0.0744699	10.3450000	-5.6828700		95	0.2109850	11.0000000	-4.8534500
	100	-0.1432200	10.4500000	-5.0334400		100	-0.0672110	11.0100000	-4.7921700
	105	-0.4253600	10.4485000	-5.1510600		105	-0.3352350	10.9341000	-4.8855800
	110	-0.6847600	10.4080000	-5.9120000		110	-0.5990770	10.8225000	-5.0027900
	115	-1.0907000	10.2667000	-6.3485200		115	-1.0627000	10.5040000	-5.3149000
	120	-1.4033000	9.9964000	-6.1444200		120	-1.4138000	10.0768000	-5.2649000
	125	-1.6964500	9.6291000	-5.2538000		125	-1.7172000	9.3597000	-4.5557000
	130	-1.9693600	8.8000000	-4.2835400		130	-1.9992700	8.5022000	-4.0083300
	135	-2.1087800	7.5462000	-3.0062000		135	-2.1458400	7.4300000	-3.3980000
	140	-2.2930300	6.3660000	-1.9415000		140	-2.3205400	6.3736000	-2.8635100
	145	-2.4368300	5.4265000	-1.9865400		145	-2.4510300	5.1604000	-2.3025400
	150	-2.5062000	4.6043000	-1.2877100		150	-2.5344900	4.4244000	-1.6422000
	155	-2.3576300	3.7551000	-0.7988410		155	-2.3936300	3.6555000	-1.0445000
	160	-2.1620400	2.7408000	-0.0525480		160	-2.1927400	2.6316000	-0.0025241
	165	-1.9348500	1.6755000	0.0679587		165	-1.9428800	1.7566000	0.0215665
	170	-1.6790000	0.9199000	-0.0033460		170	-1.7032200	0.8126000	-0.0388926
	175	-1.3926600	0.3009000	0.0169999		175	-1.4180000	0.1317000	0.0043400
	180								
	100	-1.1785600	-0.1744000	0.0101000		180	-1.2048000	-0.2473000	0.0084000
0.55	0	0.8604500	0.0305500	-0.0222500	0.60	0	0.8604000	0.0106000	-0.0436000
0.55					0.00				
	5	0.9368500	0.4359500	-0.0593000		5	0.9469000	0.4229000	-0.0635000
	10	1.0295000	0.8956500	-0.0401500		10	1.0405000	0.8217000	-0.0617000
	15	1.0616500	1.4042000	0.0199000		15	1.0703000	1.3012000	0.0277000
	20	1.0421500	1.9435000	0.0623500		20	1.0510000	1.8422000	0.0530000
	25	1.0011500	2.5840500	-0.0484500		25	1.0043000	2.4949000	-0.1258000
	30	0.9149000	3.3305500	-0.4563000		30	0.8979000	3.3433000	-0.3273000
	35	0.8118000	4.3317833	-1.5927500		35	0.7798000	4.3512667	-1.2986000
	40	0.6555500	5.4242167	-2.7046500		40	0.6060000	5.5195333	-2.2633000
	45	0.5063000	6.7434000	-4.5203500		45	0.4631000	6.9405000	-4.2940000
	50	0.3331500	8.1590333	-5.3645200		50	0.2891000	8.4202667	-5.4411000
	55								-4.9997000
		0.1570500	9.4166167	-5.2848500		55 60	0.1347000	10.1115333	
	60	-0.0374000	10.5612000	-4.9364500		60	-0.0246000	11.7421000	-4.0329000
	65	-0.2131500	11.1774500	-4.9708300		65	-0.1996333	12.6880333	-3.9694100
	70	-0.2554030	11.4481000	-5.5250700		70	-0.2051727	13.0527667	-4.9456500
	75	-0.2950020	11.6841500	-5.6239100		75	-0.2257000	13.1511000	-5.3941600
	80	-0.1513638	11.8652000	-5.4139200		80	-0.0493576	13.1258333	-5.3078400
	85	0.3953550	11.9882500	-5.2825500		85	0.3698343	13.1443667	-5.3424800
	90	0.4172370	11.9970000	-5.0606100		90	0.4926770	13.0438000	-5.1959900
1	95	0.2572010	11.9916250	-4.9924300		95	0.3034170	12.9833500	-5.1314200
	100	-0.0156450	11.9790500	-4.6680900		100	0.0359203	12.9480000	-4.5440000
							-0.2371120		
	105	-0.2861730	11.8760750	-4.7400300		105		12.8179500	-4.5944900
	110	-0.5672380	11.5614000	-4.8238800		110	-0.5354000	12.3004000	-4.6449700
	115	-1.0546950	10.8225250	-5.1773200		115	-1.0466900	11.1411500	-5.0397400
	120	-1.4161300	10.2200000	-5.4320200		120	-1.4184700	10.3632000	-5.5991400
	125	-1.7224250	9.3834750	-4.9646200		125	-1.7276400	9.4072500	-5.3735400

	130	-1.9930700	8.3759000	-4.5737900		130	-1.9868800	8.2496000	-5.1392600
	135	-2.1634800	7.3710500	-3.9804200		135	-2.1811200	7.3121000	-4.5628300
I	140	-2.3439100	6.2245000	-3.4449000		140	-2.3672900	6.0754000	-4.0263000
				-2.5360200		145		5.1512000	
	145	-2.4793900	5.1558000				-2.5077600		-2.7695000
	150	-2.5532300	4.4482000	-1.7056700		150	-2.5719800	4.4720000	-1.7691500
	155	-2.4191950	3.6255250	-1.0256000		155	-2.4447600	3.5955500	-1.0067000
	160	-2.2192200	2.5774000	-0.3056120		160	-2.2457000	2.5232000	-0.6087000
	165	-1.9654950	1.6565250	-0.0924667		165	-1.9881100	1.5564500	-0.2065000
	170	-1.7235600	0.8691000	-0.0393037		170	-1.7439000	0.9256000	-0.0397148
	175	-1.4565000	0.3220000	-0.0009300		175	-1.4949900	0.5123000	-0.0062000
	180	-1.2405000	-0.0693500	0.0082000		180	-1.2762000	0.1086000	0.0080000
0.70	•	0.0700000	0.0400000	0.0400000	0.00	•	0.0040000	0.0040000	0.0540000
0.70	0	0.8729000	0.0106000	-0.0436000	0.80	0	0.9212000	0.0010000	-0.0546000
	5	0.9213000	0.4388000	-0.0635000		5	0.9747000	0.4631000	-0.0743000
	10	1.0165000	1.0656000	-0.0617000		10	1.0690000	0.9555000	-0.1683000
	15	1.0426000	1.9278000	-0.0093000		15	1.0631000	1.5127000	-0.3421000
	20	1.0237000	2.7648000	0.0913000		20	1.0418000	2.2210000	-0.7507000
	25	0.9863000	3.9381000	0.1784000		25	0.9928000	3.1062000	-1.3265000
	30	0.9128000	5.1251000	-0.0365000		30	0.9064000	4.6333000	-1.9433000
	35	0.7815000	6.4117333	-0.5621000		35	0.8015000	6.3667000	-2.5805000
	40	0.5915000	8.1820667	-1.2384000		40	0.7013000	8.1031000	-3.5315000
	45	0.3584000	10.4734000	-2.0696000		45	0.5741000	9.8309000	-3.8387000
	50	0.1725000	12.8037333	-3.2055000		50	0.4194000	11.3893000	-3.6555000
I	55	0.0701000	14.6109667	-3.8239000		55	0.2786000	12.5189000	-3.2693000
I	60	-0.0428807	15.8763000	-3.2797100		60	0.0810423	13.2213000	-2.8406900
I	65	-0.1327030	16.5539000	-3.2651900		65	-0.0260090	13.8598667	-2.6752100
I	70	-0.1327030 -0.1471420		-3.2031900 -3.7785200		70	-0.0200090	14.3073333	-2.4210000
I			16.8436000						
I	75	0.0078317	16.9246000	-4.1525800		75	0.0875000	14.6913000	-2.7569000
I	80	0.1115480	16.9168000	-4.2451600		80	0.3393333	14.9836667	-2.9420000
I	85	0.3465000	16.8339000	-4.1639000		85	0.5130687	15.2315333	-3.0147200
I	90	0.6573530	16.5544000	-4.0620100		90	0.7420400	15.4247000	-2.9324000
	95	0.4143640	16.1413000	-4.0108200		95	0.5921970	15.5299500	-2.9509500
	100	0.1344705	15.5838000	-3.7920000		100	0.2620400	15.4405000	-2.6333400
	105	-0.1498390	14.8403000	-3.6166600		105	-0.0780110	15.1978500	-2.4807000
	110	-0.5741780	13.9681000	-3.0663700		110	-0.5541000	14.7409000	-2.5552000
	115	-1.0392400	13.0219000	-3.1584600		115	-1.0266900	14.0920500	-2.8345600
	120	-1.4086600	11.9206000	-3.9005200		120	-1.4563300	13.2670000	-3.1070500
	125	-1.7416000	10.6915000	-4.3992600		125	-1.8266700	12.3323500	-3.3424500
	130	-2.0220200	9.2808000	-4.6274100		130	-2.1101400	11.1784000	-3.4693500
	135	-2.2391000	7.5953000	-4.2160000		135	-2.3266000	9.7406000	-3.5223200
	140	-2.4365000	5.9440000	-3.6220000		140	-2.5240000	8.0528000	-3.5060000
	145	-2.5990400	4.6557000	-3.2195000		145	-2.7075200	6.4065000	-3.2647300
	150	-2.6541100	3.7209000	-2.4730400		150	-2.7647000	5.0408000	-2.4335000
	155	-2.5248200	3.0564000	-1.1576000		155	-2.5938800	3.8849500	-0.9908000
	160	-2.3019400	2.3376000	0.0025957		160	-2.3861000	2.8445000	-0.5513330
	165	-2.0412500	1.7905000	-0.0593125		165	-2.1431700	1.7885500	-0.2586000
	170	-1.7887200	1.2225000	-0.0470425		170	-1.8772400	0.9140000	-0.0384708
	175	-1.5483000	0.7221000	-0.0024000		175	-1.6416000	0.4200000	0.0006590
	180	-1.3498100	0.2288000	0.0032000		180	-1.4372000	0.0668000	0.0068000
	100	-1.3430100	0.2200000	0.0032000		100	-1.4372000	0.0000000	0.0000000
0.00	•	1 1011000	0.000000	0.0540000	4.40	•	4 4470000	0.0070000	0.0044000
0.90	0	1.1041000	0.0008000	-0.0546000	1.10	0	1.4479000	0.0278000	-0.0011000
I	5	1.1182000	0.4631000	-0.0743000		5	1.5611000	0.5283000	-0.0747000
I	10	1.1690000	0.9248000	-0.1625000		10	1.6803000	1.2192000	-0.1899000
I	15	1.1519000	1.5127000	-0.3404000		15	1.6894000	2.3094000	-0.4733000
I	20	1.1195000	2.2314000	-0.6747000		20	1.6362000	3.8004000	-0.6978000
I	25	1.0703000	3.2944000	-1.1040000		25	1.5398000	5.5249000	-0.9237000
I	30	1.0172000	4.9683000	-1.6338000		30	1.4321000	7.2772000	-1.0964000
I	35	0.9481000	7.1492667	-2.3542000		35	1.3385000	9.1891667	-1.2712000
I									
	40	0.8582000	9.2283333	-2.7529000		40	1.2373000	11.3506333	-1.4463000
	45	0.7245000	11.1670000	-2.8039000		45	1.0870000	13.0949000	-1.6061000
	50	0.5583000	12.5263667	-2.7856000		50	0.9054000	14.7035667	-1.7584000
	55	0.3743000	13.5460333	-2.5557000		55	0.7829000	16.0754333	-1.9107000
	60	0.2067000	14.2446000	-2.4546700		60	0.6993570	17.3241000	-2.0044000
	65	0.0946667	14.9071333	-2.2989000		65	0.6403320	18.2737667	-2.0549700
.1	70	0.1417333	15.4352667	-2.2654900		70	0.5241940	18.9802333	-2.0316500
	75	0.2131840	15.9415000	-2.2960900		75	0.8268550	20.0152000	-1.9088300
	80	0.4684587	16.3043333	-2.2999100		80	0.9071000	21.3853667	-1.8363800
	85	0.6495503	16.5372667	-2.2950800		85	0.7164840	22.0979333	-1.7086000
	90	0.7491410	16.6660000	-2.2490000		90	0.3626340	22.1813000	-1.6293100
	95	0.6056000	16.6200000	-2.1299000		95	-0.0102370	21.9604000	-1.6444300
	100	0.2714000	16.2514000	-2.1019000		100	-0.3653340	21.2197000	-1.7327200
	105	-0.1063000	15.7626000	-2.0852700		105	-0.7184840	19.9664000	-1.7977500
1				-2.0800000			-1.1228700		
	110	-0.5501430	15.2007000			110		18.8500000	-1.7880000
	115	-1.0910200	14.3189000	-2.0800000		115	-1.5764700	17.9325000	-1.7757100
	120	-1.5792900	13.4561000	-2.0793300		120	-1.9547300	17.0631000	-1.7513700
	125	-1.9523800	12.3876000	-2.0725200		125	-2.2588400	16.0422000	-1.6524600
	130	-2.2245800	11.3040000	-2.1051900		130	-2.5237000	14.6800000	-1.5564000
	135	-2.4695800	9.9840000	-2.2860000		135	-2.7843500	13.0906000	-1.5203600
I	140	-2.6614000	8.5567000	-2.6330800		140	-3.0399900	10.8938000	-1.3789200
	145	-2.8433000	7.0575000	-2.5990000		145	-3.2635300	8.3426600	-1.2365000

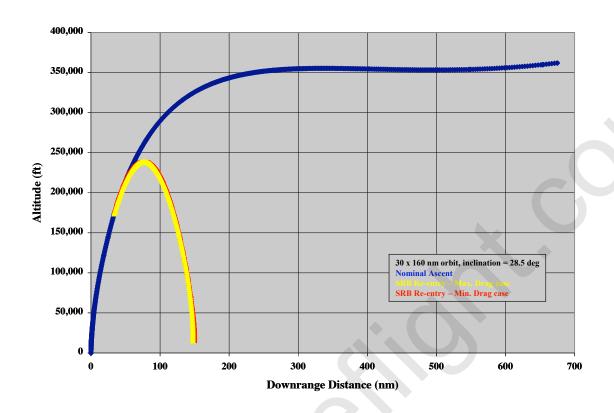
	150	-2.9491100	5.5893000	-1.6420900		150	-3.3999600	5.9155400	-1.1464100
	155	-2.7802200	4.2163000	-0.9175000		155	-3.2563500	4.0097900	-0.6993640
	160	-2.5611600	3.1141000	-0.4212090		160	-3.0765100	2.6693000	-0.2260000
	165	-2.2762000	1.9129000	-0.2599620		165	-2.8897500	1.7267200	0.9576320
	170	-2.0032600	1.1157000	-0.0416677		170	-2.5787700	0.8534820	0.2274330
	175	-1.7631000	0.5322000	-0.0084000		175	-2.3548200	0.3872780	-0.0573401
	175 180	-1.5338800	0.0847000	-0.0084000		175 180	-2.3546200 -2.1176000	0.3672760	-0.0573407
	100	-1.5556600	0.0647000	-0.0097200		100	-2.1176000	0.0193600	-0.0091600
1.20	0	1.4581000	0.0278000	-0.0011000	1.40	0	1.2708000	-0.0441000	-0.0011000
	5	1.5897000	0.5283000	-0.0420000		5	1.3324000	0.5112000	-0.0016000
	10	1.7273000	1.2192000	-0.1177000		10	1.4110000	1.2570000	-0.0402000
	15	1.7549000	2.3094000	-0.3616000		15	1.4404000	2.4020000	-0.2470000
	20	1.7201000	3.8004000	-0.5466000		20	1.4712000	3.9479000	-0.3948000
	25	1.6422000	5.5249000	-0.7330000		25	1.4905000	5.7273000	-0.5441000
	30	1.5529000	7.2772000	-0.8663000		30	1.4996000	7.5344000	-0.6220000
	35	1.4778000	9.1891667	-1.0016000		35	1.4983000	9.7392667	-0.6893000
	40	1.3950000	11.3506333	-1.1372000		40	1.4868000	11.9322333	-0.7525000
	45	1.2632000	13.0949000	-1.2576000		45	1.4690000	14.0027000	-0.8175000
	50	1.1000000	14.7035667	-1.3704000		50	1.4451000	15.7013667	-0.8777000
	55	0.9959000	16.0754333	-1.4833000		55	1.4157000	16.8859333	-0.9232000
	60	0.8548270	17.6754000	-1.5598700		60	1.3764200	18.1941000	-0.8940920
	65	0.7222847	18.5923333	-1.6171300		65	1.2895000	19.3666667	-0.7739000
	70	0.7910903	19.5698667	-1.6486300		70	1.2046600	20.5482333	-0.7494600
	75	1.0126700	20.6418000	-1.5960500		75	1.1058200	21.2897000	-0.6397640
	80	0.9118367	21.8205333	-1.3584000		80	0.9068900	21.7196667	-0.6032410
	85	0.6297783	22.5029667	-1.1233700		85		21.9233333	-0.5687400
							0.6340000		
	90	0.3082750	22.8044000	-1.0804000		90	0.2787980	22.0400000	-0.5387050
	95	-0.0413270	22.7596000	-1.0984300		95	-0.0782540	21.8168000	-0.5253980
Ī	100	-0.4231730	22.1871000	-1.2682000		100	-0.4732370	21.4971000	-0.5287480
	105	-0.8028050	21.1521000	-1.4652600	4	105	-0.9232860	20.8393000	-0.5639810
Ī	110	-1.2168100	20.1008000	-1.4727600		110	-1.4194400	20.0180000	-0.5899150
Ī	115	-1.6559300	18.9760000	-1.4887200		115	-1.8439800	18.9703000	-0.5985960
	120	-2.0274100	17.7897000	-1.5266200		120	-2.2041900	17.6562000	-0.6556580
	125	-2.3315500	16.6233000	-1.5304400		125	-2.5229900	16.1899000	-0.9186400
	130	-2.6437300	15.2870000	-1.4541600		130	-2.8016000	14.5405000	-1.0249500
	135	-2.8907600	13.5765000	-1.3599000		135	-3.0549000	12.6430000	-1.0400000
	140	-3.1046800	11.4844000	-1.2081000		140	-3.3331300	10.5223000	-1.0233600
Ī	145	-3.2995700	9.0842900	-1.0737400		145	-3.4891000	8.4834900	-0.9668000
	150	-3.4437900	6.7268300	-0.8771820		150	-3.6115900	6.6439200	-0.8671270
	155	-3.3047000	4.8270000	-0.5946000		155	-3.4341200	4.7466000	-0.7751930
		-3.0985000	3.2193200	-0.2260000			-3.4341200	3.2945400	-0.3733000
	160 165					160 165			
Ī	165	-2.9045500	1.9822200	0.7221210		165	-2.9292500	2.0166800	0.1192000
Ī	170	-2.6100100	1.1255800	0.5902490		170	-2.6472600	1.0657000	0.6055520
	175	-2.3983800	0.3582770	-0.0568876		175	-2.4503100	0.3444250	0.0048000
	180	-2.1991800	0.0365200	-0.0379800		180	-2.2976300	0.0236600	-0.0293200
1.96	0	1.0913000	0.0592000	-0.0011000	2.50	0	0.9692000	-0.1153000	0.0183000
1	5	1.1893000	0.6381000	-0.0011000	2.00	5	0.9611000	0.4905000	0.0226000
Ī	10	1.2744000	1.4045000	-0.0086000		10	0.9637000	1.2951000	-0.0337000
Ī	10 15	1.3278000	2.6389000	-0.0550000		10 15	0.9705000	2.6124000	-0.1469000
	20	1.3624000	4.2162000	-0.1019000		20	1.0169000	4.0459000	-0.2298000
Ī	25	1.3862000	6.0740000	-0.1618000		25 20	1.1013000	5.5519000	-0.2648000
Ī	30	1.4091000	7.9363000	-0.2189000		30	1.2209000	7.3094000	-0.2432000
	35	1.4295000	9.8119667	-0.2571000		35	1.3541000	8.9877667	-0.2373000
	40	1.4467000	11.8638333	-0.2960000		40	1.4543000	10.7882333	-0.2602000
	45	1.4614000	13.5436000	-0.3362000		45	1.4972000	12.4754000	-0.3197000
Ī	50	1.4640000	14.9628667	-0.3564000		50	1.5178000	13.9323667	-0.3741000
	55	1.4523000	16.2162333	-0.3639000		55	1.5089000	15.3380333	-0.4063000
	60	1.4220000	17.3728000	-0.3737560		60	1.4816000	16.5305000	-0.4191000
	65	1.3209000	18.4530667	-0.3855360		65	1.3436200	17.6189667	-0.4468240
	70	1.1909800	19.4786333	-0.4257040		70	1.1857300	18.7240333	-0.4696950
	75	1.0513700	20.2598000	-0.4531860		75	1.0296500	19.5410000	-0.4861240
	80	0.8733200	20.7894667	-0.4535460		80	0.8450300	20.0933667	-0.4881220
	85	0.6190240	21.0626333	-0.4139590		85	0.6358210	20.4133333	-0.4900820
1	90	0.3366290	21.1345000	-0.4052860		90	0.3861180	20.4523000	-0.4868290
	95	0.0527190	20.9040000	-0.4182180		95	0.1536970	20.2436000	-0.4798270
	100	-0.2926645	20.4237000	-0.4275750		100	-0.1429000	19.7431000	-0.4719850
	105	-0.6737290	19.7052000	-0.4128130		105	-0.4856860	18.9943000	-0.4618750
	110	-1.0948240	18.8241000	-0.3980510		110	-0.9123000	18.0800000	-0.4493000
	110 115	-1.5600200	17.5953000	-0.3969290		115	-1.4059000	16.9701000	-0.4337990
		-1.9957200		-0.3969290 -0.3810200					
	120 125		16.3351000			120 125	-1.8925100	15.4751000	-0.4103400
l '	125	-2.3663400	14.9074000	-0.3578960		125	-2.3509900	14.0498000	-0.3961970
1	130	-2.6823000	13.2907000	-0.3092040		130	-2.7065200	12.4945000	-0.3783820
I	135	-2.9179000	11.5909000	-0.2925560		135	-2.9649800	10.8336000	-0.3499760
I	140	-3.1095000	9.8039000	-0.2690000		140	-3.1465000	9.1415000	-0.2609000
I	145	-3.2345000	8.0360000	-0.3469000		145	-3.2849100	7.6162700	-0.2413900
I	150	-3.3508500	6.2729800	-0.5037620		150	-3.3476800	6.0253900	-0.2811800
I	155	-3.1865800	4.6939400	-0.5921600		155	-3.1750200	4.4131008	-0.3398450
I	160	-2.9599800	3.1648300	-0.2923000		160	-2.9549800	3.0497615	-0.2914730
I	165	-2.7326600	2.0153000	-0.0091000		165	-2.7162600	1.8832623	-0.0711183

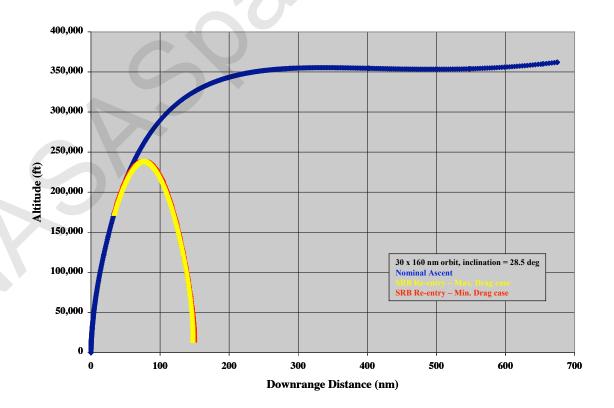
	170	-2.5108100	0.8486060	0.2076320		170	-2.5132500	0.8873611	0.0878881
	175	-2.3870500	0.2732110	0.0576000		175	-2.3872700	0.3275215	0.0368000
	180	-2.3129000	0.0000000	0.0171000		180	-2.3141100	0.0469000	0.0350000
3.00	0	0.9692000	-0.0831000	0.0183000	3.48	0	0.7879000	-0.0253000	0.0215000
0.00	5	0.9611000	0.4726000	0.0307000	0.40	5	0.8083000	0.5533000	-0.0069000
	10	0.9637000	1.2270000	-0.0268000		10	0.8526000	1.3110000	-0.0234000
	15	0.9705000	2.4942000	-0.1370000		15	0.9351000	2.3314000	-0.0929999
	20	1.0169000	3.8776000	-0.2168000		20	1.0360000	3.6501000	-0.0929999
	25 25	1.1013000	5.3335000	-0.2489000		25	1.1463000	4.9401000	-0.1744000
	30	1.2209000	7.0408000	-0.2243000		30	1.2648000	6.4746000	-0.1963000
	35	1.3541000	8.6690667	-0.2154000		35	1.4064000	8.2357667	-0.2171000
	40	1.4543000	10.4194333	-0.2353000		40	1.5198000	9.8609333	-0.2427000
		1.4543000	12.0564000	-0.2353000		40 45	1.5757000	11.5109000	-0.2427000
	<i>4</i> 5								
	50	1.5178000	13.4632667	-0.3432000		50	1.5776000	12.9848667	-0.2775000
	55	1.5089000	14.8188333	-0.3724000		55	1.5390000	14.3354333	-0.2921000
	60	1.4816000	15.9612000	-0.3822000		60	1.4577000	15.5765000	-0.3006000
	65	1.3297300	17.1070667	-0.3909500		65	1.3175300	16.7410667	-0.3102480
	70	1.1477400	18.1365333	-0.4038800		70	1.1571500	17.7175333	-0.3108660
	75	0.9923500	18.9278000	-0.4127120		75	1.0025200	18.4582000	-0.3094890
	80	0.8119800	19.5055667	-0.3982200		80	0.8234700	19.0532667	-0.3013880
	85	0.6174890	19.8533333	-0.4019000		85	0.6428440	19.3765333	-0.3057080
	90	0.4001350	19.9203000	-0.4019000		90	0.4400400	19.3753000	-0.3118650
	95	0.2085000	19.7156000	-0.4005400		95	0.2494060	19.1339000	-0.3155960
	100	-0.0585730	19.2958000	-0.3998560		100	-0.0036830	18.6678000	-0.3114500
	105	-0.3666832	18.6026000	-0.3978560		105	-0.3326214	18.0155000	-0.3187000
	110	-0.7807500	17.7067000	-0.4033060		110	-0.7833010	17.1868000	-0.3438540
	115	-1.2184060	16.5221000	-0.4112000		115	-1.2599550	15.9718000	-0.3499480
	120	-1.6889200	15.1744000	-0.3743800		120	-1.7577900	14.6304000	-0.3360400
	125	-2.1260300	13.7283000	-0.3364170		125	-2.2192800	13.2126000	-0.2990400
	130	-2.5392000	12.1370000	-0.3002320		130	-2.6337600	11.8206000	-0.2629610
	135	-2.8730000	10.5328000	-0.2840450		135	-2.9225800	9.9804600	-0.2374690
	140	-3.0628000	8.8447400	-0.2529040		140	-3.1189300	8.3721500	-0.2111450
	145	-3.2215400	7.2649000	-0.2129000		145	-3.2397200	6.6510800	-0.1513900
	150	-3.2940900	5.5989000	-0.1830000		150	-3.3034600	5.1070300	-0.1431440
	155	-3.1344200	4.0380300	-0.2850000		155	-3.0799200	3.7578000	-0.1394620
	160	-2.9599200	2.6892900	-0.1400000		160	-2.8774900	2.4408100	-0.0733121
	165	-2.7625600	1.6220800	0.0377268		165	-2.6755000	1.4424000	0.0041839
	170	-2.5520500	0.7572740	0.0327364		170	-2.4854700	0.7235310	-0.0022000
	175	-2.4695100	0.2732730	0.0368000		175	-2.4094200	0.2735200	-0.0096000
	180	-2.3832100	-0.0034112	0.0291000		180	-2.3296900	0.0173399	-0.0032000

Table 5.3.5-2: SRB Reentry - Angle of Attack Profiles

SRB Angle of Attack Profiles							
Altitude	Angle of Attack (degrees)						
(feet)	Minimum Drag	Maximum Drag					
20,000	113	97					
30,000	158	152					
50,000	167	158					
100,000	150	130					
150,000	119	106					

300,000 119 106







#### 5.3.5.2 UPPER STAGE REENTRY

The mission profile for Reference Trajectory 2 has been designed to deliver the upper stage, the upper stage—to—payload adapter, and the payload to the nominal MECO conditions. The launch vehicle places its payload into a 30 x 160 nm orbit, with the insertion altitude at 57 nm. For purposes of this study, it has been assumed the payload is separated from the upper stage 250 seconds after MECO. The optimal separation time will need to be determined by further analysis. As an integral unit, the upper stage and adapter will reenter the atmosphere.

The reentry aerodynamics, as shown in Table 5.3.5-3, was provided by Joe Lowery / EV33 on June 20, 2005. Since the orientation of the upper stage may vary, three sets of drag coefficients were provided. When it reenters on its side (angle of attack =  $90^{\circ}$ ), the drag causes upper stage to impact quicker. With an end entry (angle of attack =  $0^{\circ}$ ), the impact location is further downrange. A tumbling reentry gives an intermediate impact distance.

Since the atmospheric conditions are not constant, this study also varied the atmospheric density by 30%. The impact zones shown in Figure 5.3.5-2 and Figure 5.3.5-3 encompass the maximum and minimum impact locations. The shortest downrange distance represents a more dense atmosphere (+30% density multiplier) and a side-entry orientation on the upper stage. The furthest impact location is determined by a less dense atmosphere (-30% density multiplier) and an end-entry orientation.

Table 5.3.5-3: Upper Stage Reentry Aerodynamics

Upper Stage Reentry Drag									
Aref = 255.5 ft <sup>2</sup>									
Mach <u>Coefficient of Drag</u>									
Number	Side Entry	End Entry	Tumbling						
0.0	4.27	0.80	2.54						
0.2	4.56	0.80	2.68						
0.4	4.89	0.82	2.86						
0.6	5.46	0.85	3.16						
0.8	6.96	0.93	3.95						
1.0	9.68	1.16	5.42						
1.5	8.26	1.50	4.88						

2.0	7.87	1.64	4.75
3.0	7.47	1.72	4.60
4.0	7.36	1.72	4.54
6.0	7.19	1.72	4.46
10.0	7.19	1.72	4.46
20.0	7.19	1.72	4.46

Figure 5.3.5-2 shows the reentry ground track for each mission. The green ellipses indicate the nominal impact footprints. Each footprint has the same dimensions as the STS-51D nominal ET impact footprint: the footprint toe is located 439 nm downrange of the nominal impact point, the footprint heel is located 554 nm prior to the nominal impact point, and the maximum footprint width (located at the nominal impact point) is 36.6 nm.

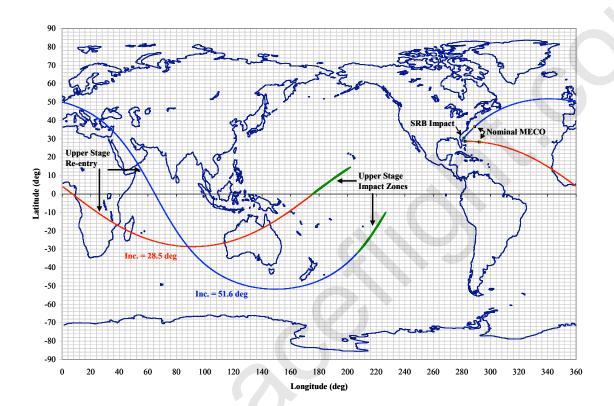


Figure 5.3.5-2: Upper Stage Reentry Ground Track (Rev. 2 Reference Trajectory)

There was concern that the upper stage would splash into the Pacific Ocean much further downrange than the nominal external tank impact point. As seen in Figure 5.3.5-3, it should not pose a threat to any land masses.

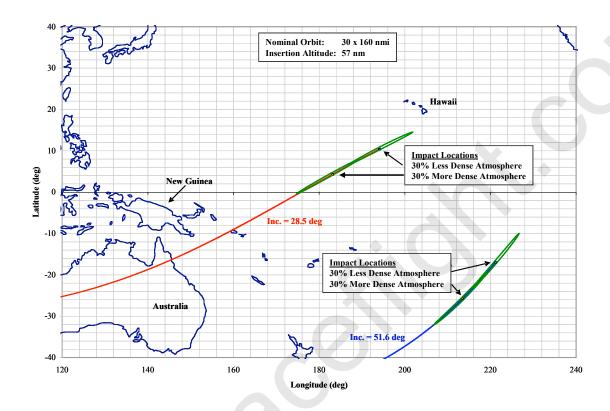
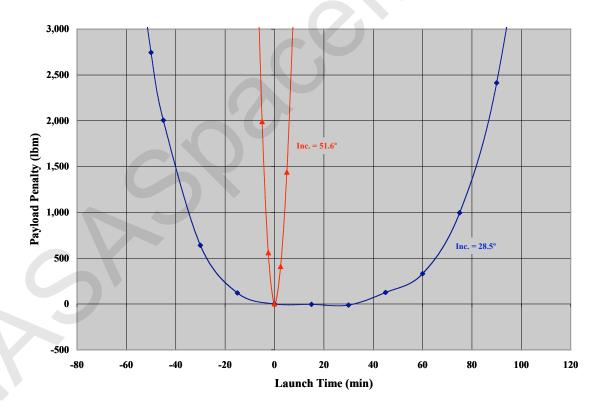


Figure 5.3.5-3: Enlargement of the Impact Location

## 5.3.6 LAUNCH WINDOW ANALYSIS

When launching for a rendezvous, additional constraints are placed on the mission. This has an impact on the available launch times. The first launch of a mission buildup will not be restricted to a specific orbit plane. The inclination will be predetermined, but the ascending node isn't fixed by the rendezvous requirements. Any subsequent launches must perform the rendezvous missions and they must be launched into the orbit plane of the first component.

The effect of the Earth's rotation and the need to launch into the required orbit plane as the launch site rotates past the target orbit is shown as a payload penalty in Figure 5.3.6-1. The curve for the 28.5° inclination orbit is not symmetrical about the zero time point. Since the bottom of the curve is very flat, the optimization program, POST, chose a zero point that was shifted to the left of a symmetrical location. The zero point could be moved to the right without changing the results. The following figure, Figure 5.3.6-2, shows the penalty for each total launch window duration. In this study, the subsequent launches are allowed to optimize the launch azimuth as well as perform yaw steering after the first stage is separated. The reference trajectory does not allow the yaw steering.





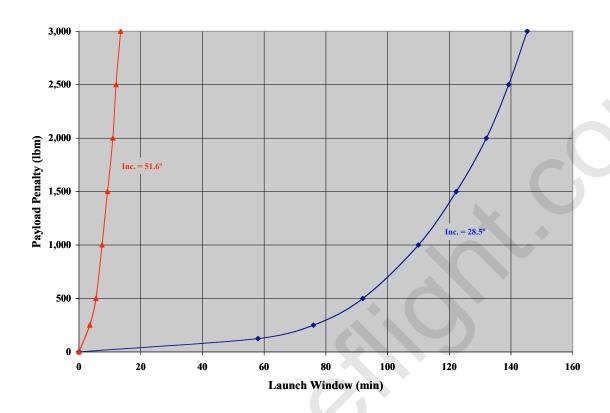


Figure 5.3.6-2: Launch Window Payload Penalties

# 5.3.7 POINT OF DEPARTURE CONFIGURATION (REV. 4)

At the conclusion of the "60-Day Study," one configuration had evolved to become the point-of-departure for future trade studies and system analyses. This configuration, as well as the trajectory philosophy, will be used by both the VIPA team and Advanced Concepts for the trade studies. The weight breakdown has been compiled by Roy Lutonsky and Holly Chandler / MSFC EV12 on September 8, 2005.

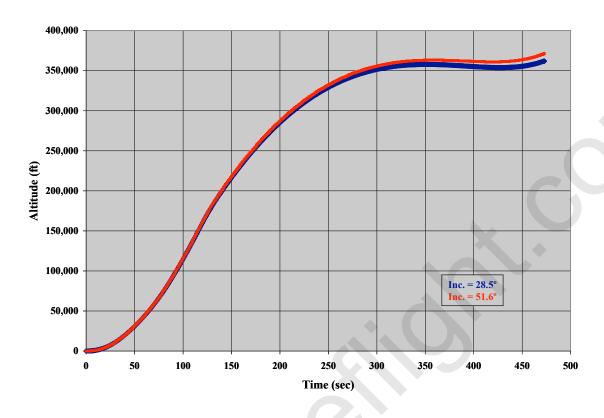
### 5.3.7.1 REV. 4 REFERENCE MISSIONS

The reference trajectory for this revision followed the philosophy detailed in Section 5.3.3 of this document. The only methodology change is the computation of the flight performance reserve (FPR) propellant. In the previous revisions, the FPR was equivalent to the propellant needed for 1% of the Ideal Delta Velocity. Based on Monte Carlo dispersion runs, there were some cases that required additional propellant. Based on information from Dr. Greg Dukeman / MSFC EV42, the FPR propellant has been increased to accommodate 1.1% of the Ideal Delta Velocity.

The following tables and graphs depict the ascent flight regimes of the CLV-4 Rev. 4 vehicle. Table 5.3.7-1 gives the summary of the two missions. Figures 5.3.7-1 to 5.3.7-3 show several of the trajectory parameters. The mission profiles are depicted in Figures 5.3.7-4 and 5.3.7-5.

Table 5.3.7-1: Rev. 4 Reference Mission Summaries

Performance Summary										
Trajectory description Mission description	REV 4 Due east crew		REV 4 Due east crew		REV 4 ISS Crew		REV 4 ISS Crew			
Units	SI		English		SI		English			
Gross mass at SRB ignition	806,770	kg	1,778,632	lb	805,020	kg	1,774,774	lb		
SRB usable propellant	504,354		1,111,916		504,354		1,111,916			
SRB jettison mass	76,710		169,117		76,710		169,117			
SRB-to-Interstage Adapter	7,499		16,533		7,499		16,533			
Interstage	1,615		3,560		1,615		3,560			
2nd Stage usable ascent LOX	140,167		309,018		140,188		309,062			
2nd Stage usable ascent LH2	23,241		51,238		23,244		51,246			
Launch Abort System (LAS)	4,218		9,300	_	4,218	_ ^	9,300	_		
Injected mass	48,966		107,952	_	47,192		104,041	_		
2nd Stage usable FPR LOX	930		2,050		910		2,006			
2nd Stage usable FPR LH2	154 242		340		151		333			
2nd Stage usable fuel bias LH2 2nd Stage residual propellant	1,462		534 3,224		242 1,462		534 3,224			
2nd Stage APU reactants & helium purge	1,462 54		120		54		120			
2nd Stage AFO feactants & nenum purge 2nd Stage pressurant gases	172		380		172		380			
2nd Stage pressurant gases 2nd Stage dry weight	17,360		38,274		17,360		38,274			
RCS propellant, reserves & residuals	134		295		134		295			
Upper Stage-to-Spacecraft Adapter	1,510		3,328		1,510		3,328			
Gross payload to delivery orbit	26,946	_ :	59,407		25,196		55,548	_		
Payload margin (15%)	4,042		8,911		3,779		8,332			
Net payload to delivery orbit	22,904	_ <sub>1ra</sub> .	50,496	- <sub>lb</sub>	21,417	— <sub>Ira</sub> –	47,216	— <sub>lb</sub>		
Net payload to delivery orbit (mt)	22,904	kg	22.90	mt	21,417	kg	21.42	mt		
ivet payload to delivery orbit (lift)	22.90	mt	22.90		21.42	mt	21.42			
Total 2nd Stage usable propellant	164,735	kg	363,180	lb	164,735	kg	363,180	lb		
Total 2nd Stage usable LOX	141,097		311,068		141,097		311,068			
Total 2nd Stage usable LH2	23,638		52,112		23,638		52,112			
F/W at 0.6 sec after SRB ignition	1.607		1.606		1.607		1.610			
F/W prior to SRB staging	0.020		0.020		0.020		0.020			
F/W after SRB staging (RPL=100%)	0.977		0.977		0.977		0.985			
Launch azimuth (deg)	85.9	deg	85.9	deg	49.4	deg	49.4	deg		
Total burn time (sec)	472.8	sec	472.8	sec	472.8	sec	472.8	sec		
Total ascent ideal ÄV	9,023	m/s	29,602	fps	9,159	m/s	30,051	fps		
Maximum dynamic pressure	39.078	N/m <sup>2</sup>	816	psf	39.471	$N/m^2$	824	psf		
Mach Number at Max. Q	1.31	7	1.31	L.v.	1.33	. 1/	1.33	Po.		
Maximum acceleration (g's)	4.50	g's	4.50	g's	4.67	g's	4.67	g's		
CDD inttinue time (cos)	120.04		120.04		120.04		120.04			
SRB jettison time (sec)	128.04 53,992	sec	128.04	sec ft	128.04 54,338	sec	128.04	sec ft		
SRB jettison altitude	53,992 4.50	m	177,138 4.50	It		m	178,274 4.51	π		
SRB jettison Mach Number SRB relative fpa (deg)	4.50 24.99	doc	4.50 24.99	daa	4.51 25.08	doc	4.51 25.08	doo		
SRB maximum altitude	75,214	deg m	24.99 246,766	deg ft	25.08 75,900	deg m	25.08 249,016	deg ft		
SRB maximum altitude	75,214 75,911	m m	249,051	ft	75,900 75,911	m m	249,016	ft		
Size internal anowable arrange	,5,711	***	2-7,031	11	15,711	111	2-7,031	11		
LAS jettison time (sec)	214.3	sec	214.3	sec	211.8	sec	211.8	sec		
LAS jettison altitude	91,440	m	300,000	ft	91,440	m	300,000	ft		
LAS jettison Mach Number	7.51		7.51		7.49		7.49			



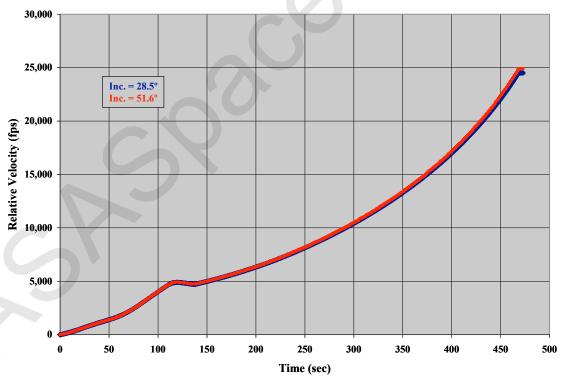
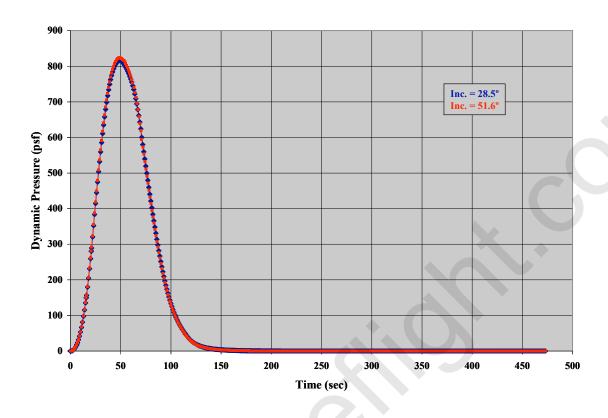


Figure 5.3.7-1: Rev. 4 Reference Trajectory Plots



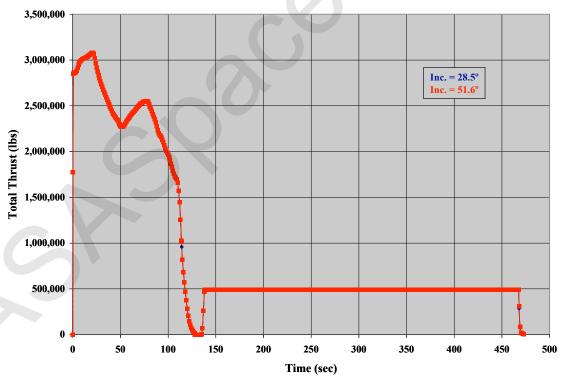
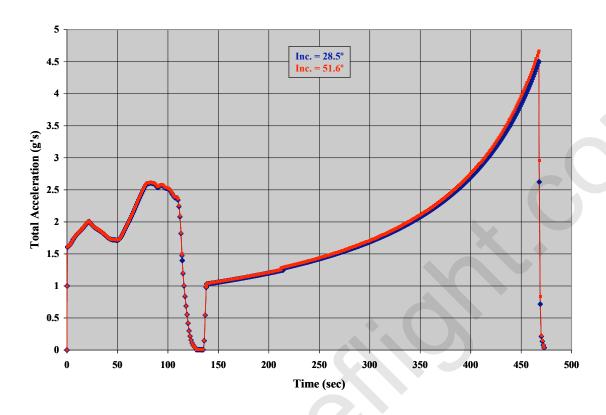


Figure 5.3.7-2: Rev. 4 Reference Trajectory Plots



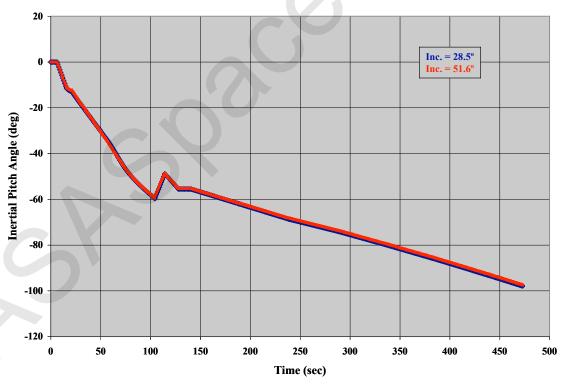


Figure 5.3.7-3: Rev. 4 Reference Trajectory Plots

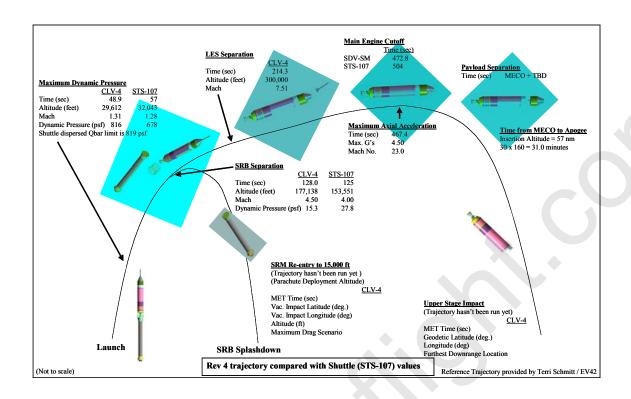


Figure 5.3.7-4: Due East Mission Profile for Rev. 4 Reference Trajectory

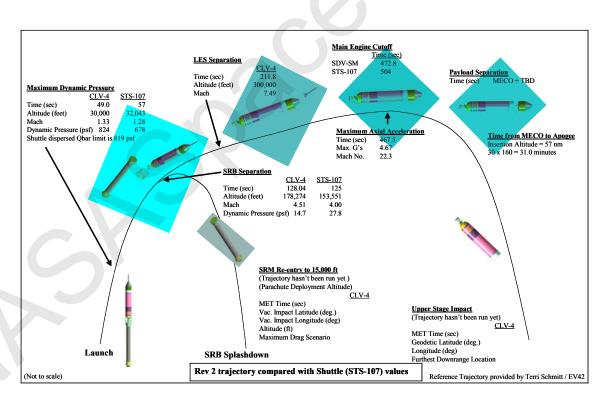


Figure 5.3.7-5: ISS Mission Profile for Rev. 4 Reference Trajectory

# 5.3.7.2 ISS CARGO MISSION (Rev. 4)

The CLV may be used as a cargo vehicle. The overall configuration is changed only by the removal of the Launch Assist System (LAS). Table 5.3.7-2 shows the comparison between the crewed and cargo missions to  $30 \times 160$  nm orbit at a  $51.6^{\circ}$  inclination.

Table 5.3.7-2: Crewed and Cargo Missions to ISS (Rev. 4)

	CL	V-4 IS	S Missio	ns (Rev. 4)	1			
	CL			`				
	DEVA	Perio	rmance Sun	ımary	DEV		REV 4	
Trajectory description Mission description	REV 4 REV 4 ISS Crew ISS Crew				ISS Cargo			
wission description	188 Clew		iss ciew		iss Cargo		ISS Cargo	
Jnits	SI		English		SI		English	
Gross mass at SRB ignition	805,020	kg	1,774,774	lb	801,510	kg	1,767,037	lb
SRB usable propellant	504,354	Č	1,111,916		504,354		1,111,916	
SRB jettison mass	76,710		169,117		76,710		169,117	
SRB-to-Interstage Adapter	7,499		16,533		7,499		16,533	
Interstage	1,615		3,560		1,615		3,560	
2nd Stage usable ascent LOX	140,188		309,062		140,175		309,035	
2nd Stage usable ascent LH2	23,244		51,246		23,242		51,241	
Launch Abort System (LAS)	4,218		9,300					
njected mass	47,192		104,041		47,916		105,636	
2nd Stage usable FPR LOX	910		2,006		922		2,033	
2nd Stage usable FPR LH2	151		333		153		337	
2nd Stage usable fuel bias LH2	242		534		242		534	
2nd Stage residual propellant	1,462		3,224		1,462		3,224	
2nd Stage APU reactants & helium purge	54		120		54		120	
2nd Stage pressurant gases	172		380		172		380	
2nd Stage dry weight	17,360		38,274		17,360		38,274	
RCS propellant, reserves & residuals	134		295		134		295	
Upper Stage-to-Spacecraft Adapter	1,510	L/ A.	3,328		1,510		3,328	_
Gross payload to delivery orbit	25,196		55,548		25,905		57,111	_
Payload margin (15%)	3,779		8,332		3,886		8,567	
Net payload to delivery orbit	21,417	kg -	47,216	lb	22,019	kg -	48,545	— <sub>lb</sub>
Net payload to delivery orbit (mt)	21.42	mt	21.42	mt	22.02	mt	22.02	mt
Total 2nd Stage usable propellant	164,735	kg	363,180	lb	164,735	kg	363,180	lb
Total 2nd Stage usable LOX	141,097	Kg	311,068	10	141,097	ĸg	311,068	10
Total 2nd Stage usable LOX  Total 2nd Stage usable LH2	23,638		52,112		23,638		52,112	
Total 2lid Stage usable L112	23,038		32,112		23,038		32,112	
/W at 0.6 sec after SRB ignition	1.607		1.610		1.607		1.617	
VW prior to SRB staging	0.020		0.020		0.020		0.020	
/W after SRB staging (RPL=100%)	0.977		0.985		0.977		1.001	
aunch azimuth (deg)	49.4	deg	49.4	deg	49.1	deg	49.1	deg
otal burn time (sec)	472.8	sec	472.8	sec	472.8	sec	472.8	sec
otal ascent ideal ÄV	9,159	m/s	30,051	fps	9,145	m/s	30,002	fps
forimum dynamic messyre	39.471	N/m <sup>2</sup>	824	mof	40.404	$N/m^2$	844	nof
Maximum dynamic pressure Mach Number at Max. O	1.33	IN/III	1.33	psf	40,404 1.35	IN/III	1.35	psf
Maximum acceleration (g's)	4.67	g's	4.67	g's	4.60	g's	4.60	g's
inaminam acceleration (g s)	7.07	g 3	7.07	53	7.00	80	т.00	80
RB jettison time (sec)	128.04	sec	128.04	sec	128.04	sec	128.04	sec
RB jettison altitude	54,338	m	178,274	ft	53,168	m	174,437	ft
RB jettison Mach Number	4.51		4.51		4.61		4.61	
RB relative fpa (deg)	25.08	deg	25.08	deg	23.13	deg	23.13	deg
RB maximum altitude	75,900	m	249,016	ft	72,740	m	238,650	ft
RB maximum allowable altitude	75,911	m	249,051	ft	75,911	m	249,051	ft
AS jettison time (sec)	211.8	sec	211.8	sec	211.8	sec	211.8	sec
AS jettison altitude	91,440	m	300,000	ft	91,440	m	300,000	ft
AS jettison Mach Number	7.49		7.49		7.49		7.49	

## 5.3.7.3 LAUNCH WINDOW ANALYSIS (Rev. 4)

When launching for a rendezvous, additional constraints are placed on the mission. This has an impact on the available launch times. The first launch of a mission buildup will not be restricted to a specific orbit plane. The inclination will be predetermined, but the ascending node isn't fixed by the rendezvous requirements. Any subsequent launches must perform the rendezvous missions and they must be launched into the orbit plane of the first component.

The effect of the Earth's rotation and the need to launch into the required orbit plane as the launch site rotates past the target orbit is shown as a payload penalty in Figure 5.3.7-6. The curve for the 28.5° inclination orbit is not symmetrical about the zero time point. Since the bottom of the curve is very flat, the optimization program, POST, chose a zero point that was shifted to the left of a symmetrical location. The zero point could be moved to the right without changing the results.

By launching into a slightly higher inclination, the launch window for a due east mission can be increased. When the vehicle is launched into the 29.0° inclination, two launch opportunities are present within a short period of time. These opportunities represent the ability to launch into either the ascending leg of the orbit or the descending leg. This produces the payload penalty oscillation seen in Figure 5.3.7-6.

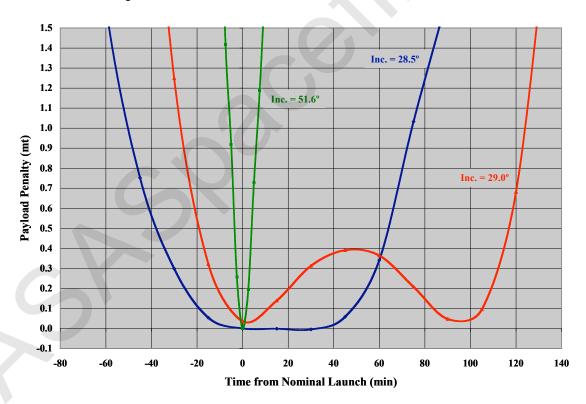


Figure 5.3.7-6: Launch Time Payload Penalties

The following figure, Figure 5.3.7-7, shows the penalty for each total launch window duration. In this study, the subsequent launches are allowed to optimize the launch azimuth as well as perform yaw steering after the first stage is separated. The reference trajectory does not allow the yaw steering. For this vehicle, the payload penalty initially is greater for the 29.0° inclination case. At approximately 87 seconds, similar payload penalties are seen for both the 28.5° and 29.0° inclination cases. After this time, the slightly greater inclination launches will produce a longer launch window.

The effect of the two launch opportunities appears to create a rapid change in the payload penalty. This transition occurs when the penalty exceeds the local maximum of the oscillation seen in Figure 5.3.7-6.

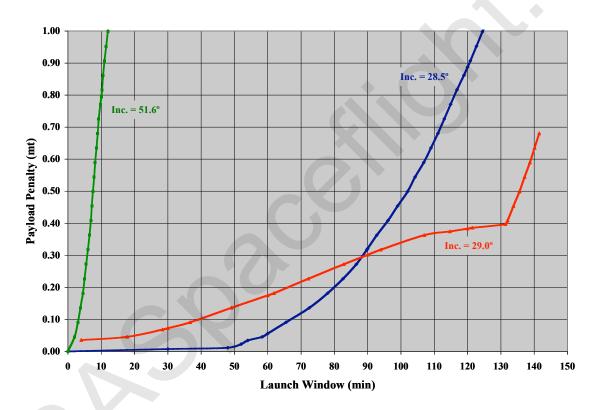


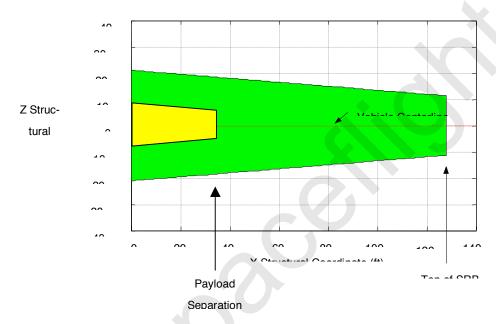
Figure 5.3.7-7: Launch Window Payload Penalties

## 5.4 FLIGHT CONTROL AND INTEGRATED SIMULATION

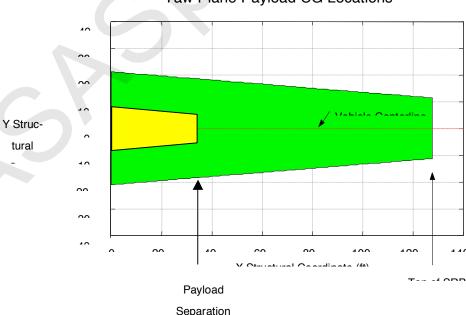
# **5.4.1 Center of Gravity Analysis**

To determine the approximate payload cg capability of the launch vehicle, we assumed a  $\pm$ -5 degree gimbal capability for the SSME and a  $\pm$ -6.36-degree gimbal range for the SRB. We allowed for a 2-deg margin for the SRB and a 1-deg margin for the upper stage for control purposes, and calculated the cg envelope which was trimmable by each engine. The results are in the following graphs.

## Pitch Plane Payload CG Locations



### Yaw Plane Payload CG Locations



485

The envelopes (in feet) show that the cg of the payload is quite large and is not a restrictive part of the vehicle design.

## 5.4.2 RCS Control Analysis and Sizing

## 5.4.2.1 First-Stage 4-Segment CLV RCS Design

#### Assumptions

- Sixteen 600 pound bipropellant thrusters
  - 8 positive roll, 8 negative roll in 4 pods spaced 90 degrees around
  - Pods mounted at top of SRB; aft skirt too crowded already
  - Isp = 293 seconds, Minimum impulse bit = 25 pounds
- Roll Moment Arm: 9 feet; pitch and yaw moment arm: 75 feet
- > Roll Moment of Inertia (T=0): 411271 slug-feet squared
- Pitch/Yaw Moment of Inertia (Burnout): 7717500 slug-feet squared
- > RCS also performs pitchover at burnout to protect parachute pack
- Placement on second stage ruled out due to weight considerations
- Redundancy not considered

#### Performance

- Max RCS Roll Torque: 9 x 600 x 8 = 43200 foot pounds
- ➤ Initial Roll Acceleration = (43200/411271) x 57.3 = 6.02 deg/sec/sec
- $\triangleright$  Initial min delta-omega = (25x6/411271) x 57.3 = .0201 deg/sec (single jet)
- Max RCS Pitch/Yaw Torque (burnout): 4 x 900 x 75 = 270000 ft-lbs
- Pitch/Yaw accel (b/o) = (270000/7717500) x 57.3 = 2.005 deg/sec/sec

#### Background

Worst case peak SRB roll disturbance torque of 36,000 foot pounds extrapolated from test data for small fixed-nozzle solid rocket motors (mostly DOD missiles) according to a methodology described in the paper "Roll torques produced by fixed nozzle solid rocket motors," by Roy Knauber, AIAA-1995-2874, plus a small (15 percent) safety factor. Largest known motor for which roll torque data exists is 50,000 pounds thrust, almost two orders of magnitude smaller than SRB.

No roll torque data exists for the SRB. The ATK/Thiokol test stand is not configured for roll measurement. No aerodynamic roll torque data is available either. Aero roll torque can be minimized by eliminating unnecessary external SRB appurtences.

ATK believes the roll torques caused by the SRB should not be significant.

#### Recommendations

Perform ground tests to measure peak SRB roll torque prior to flight. Design CLV GN&C and avionics so that it can fly-through a short roll disturbance torque which exceeds RCS capability. History includes two large booster failures (one Ariane and one Russian moon rocket) which occurred when roll control was lost, only because the GN&C and avionics assumed roll control would always be near-perfect. Peak roll disturbances typically last less than 10 seconds. Perform wind tunnel test to determine peak aerodynamic roll torque.

Measure roll torques on early CLV test flights

### 5.4.2.2 Evaluation of Single RCS Option (Upper Stage Only)

#### Assumptions

- Roll control torque requirement for US burns scaled up from Saturn J2
  - SSME thrust level 2.15 times that of similar J2 engine on S4-B
  - Saturn S4-B diameter 1.22 times that of CLV US
  - S4-B Roll control thruster size is 150 pounds
  - Scaling for CLV US specs yields 150 x 1.22 x 2.15 = 393.5 pounds
  - Throwing in about 100 more pounds as a safety factor (since we have no test data) gives us 500 pounds for the US RCS size.
  - This can be broken into multiple thrusters of smaller size (as was done with the RCS hardware analysis in section 4.3.2)
- Sixteen 500 pound bipropellant thrusters on upper stage
  - Number chosen to achieve required roll torque for first stage operation
  - No redundancy
  - 8 positive roll, 8 negative roll
  - mounted in four pods of four thrusters each
  - Pods spaced uniformly at 90 degree intervals
  - Pods mounted near the aft end of the stage
  - lsp = 293 seconds
  - Minimum impulse bit = 25 pounds
- Roll Moment Arm: 9 feet
- > Roll Moment of Inertia (T=0): 411271 slug-feet squared
- > Peak SRB Roll Disturbance not to exceed 36000 foot pounds

#### Performance

- Max RCS Roll Torque: 9 x 500 x 8 = 36000 foot pounds
- ➤ Initial Roll Acceleration = (36000/411271) x 57.3 = 5.02 deg/sec/sec
- Initial min delta-omega = (25x6/411271) x 57.3 = .0201 deg/sec (single jet)
- Estimated Roll Control Propellant (avg. 50% duty cycle): 648 pounds

#### Conclusions

Propellant savings achieved due to longer moment arm on upper stage. This reduction must be traded against higher RCS system weight carried to orbit relative to separate roll control systems, wherein the upper stage RCS hardware would be relatively light. A small solid motor will have to be added to the SRB to pitch it over at separation if all the Roll RCS is moved to the upper stage. There is a 10:1 ratio of payload penalty due to carrying weight on the upper stage versus carrying the weight on the SRB. Based on this, the baseline is to have separate RCS systems. Further trades should be conducted on this issue.

#### 5.4.3 Abort Simulation

This section includes a comparison between two 3-DOF abort simulations modeled using Maveric II and two 3-DOF trajectories modeled using POST. The vehicle configuration used for this comparison is a 5.0 meter diameter CEV mounted on top of a CLV-A launch vehicle. The latter includes a five-segment SRB and a LOX/LH2 second stage with four RL-85 engines. While this vehicle configuration differs from the CLV-4 configuration, the comparison between POST and Maveric II is still valid.

### 5.4.3.1 Ground Rules And Assumptions

The following ground rules and assumptions were used for this analysis:

- CLV-A ISS reference trajectory (Rev. 2B) provided by MSFC/EV42
- Block I CEV (5/5/05 mass properties)
- Two CEV abort trajectories:
  - Abort at 170 sec using Launch Escape System (LES)
  - Abort at 300 sec using Service Module (SM)
- POST abort trajectories provided by MSFC/EV42
- Maveric II abort guidance algorithm provided by MSFC/EV42
- Preliminary 3-DOF abort trajectories
- Apollo CM aerodynamics with capsule diameter increased to 5.0 m
- Fixed attitude during LES or SM escape burn
- Rotate to Apollo trim angle-of-attack profile after LES/SM burn
- Rotate to 45 degree bank angle after LES/SM burn
- 1963 Patrick reference atmosphere with no winds
- Abort trajectories stop at 20,000 ft altitude
- SM is not used to target or avoid specific landing sites
- No dispersions

## 5.4.3.2 Abort Simulation Comparison (MAVERIC vs POST)

Figures 5.4.3.2-1 - 5.4.3.2-5 on the following pages show the results of this comparison. While the POST & Maveric II trajectories have slight differences, the overall results compare very well. This close comparison provides validation for the Maveric II abort simulations found in the following two sections of this report.

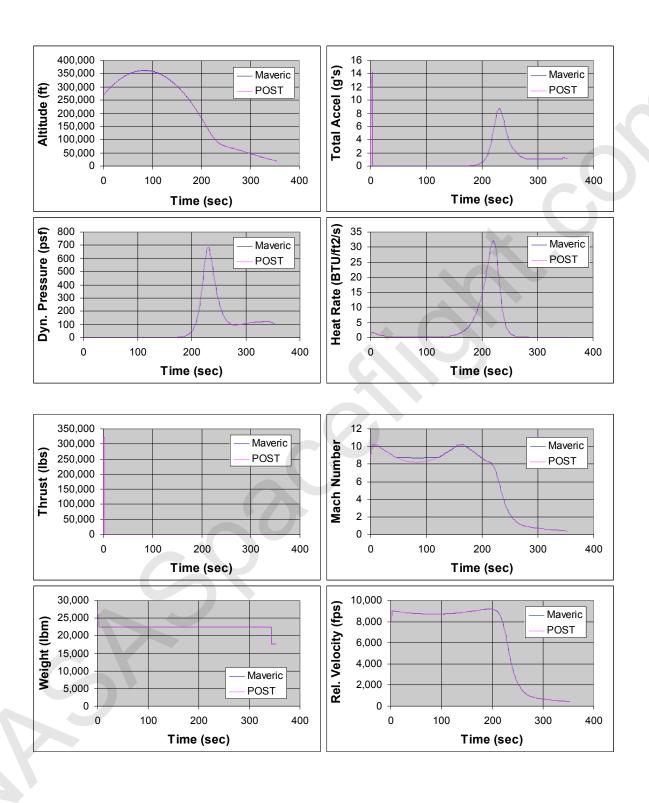


Figure 5.4.3.2-1 CEV Abort at 170 sec Using LES

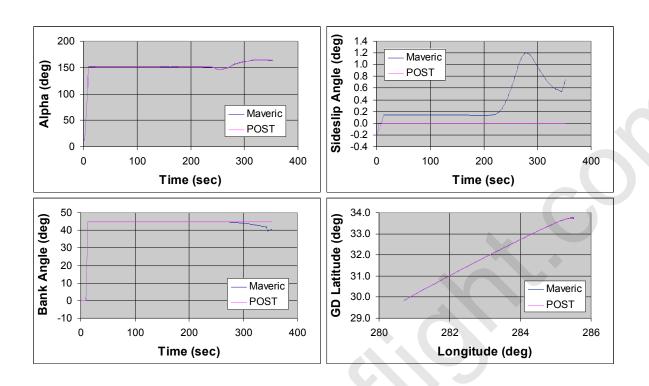


Figure 5.4.3.2-2 CEV Abort at 170 sec Using LES

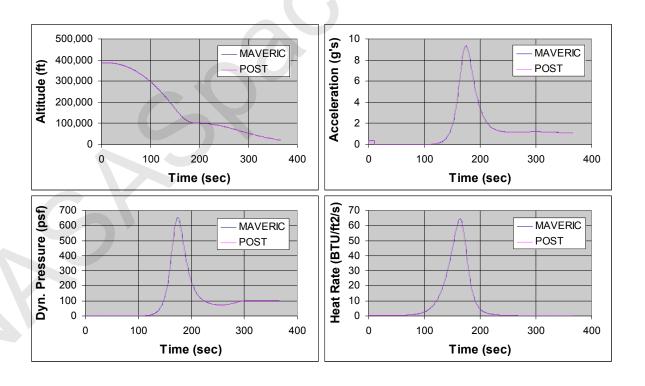


Figure 5.4.3.2-3 CEV Abort at 300 sec Using Service Module



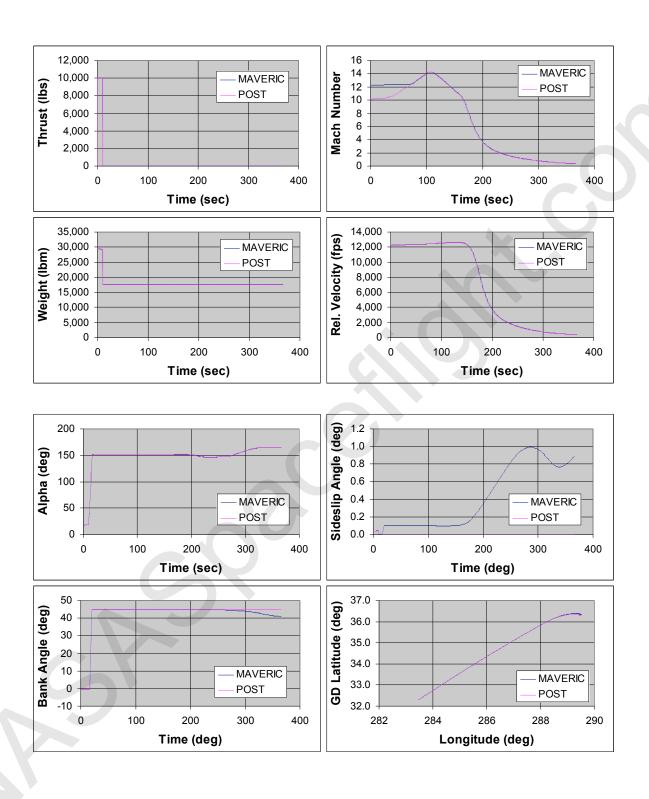


Figure 5.4.3.2-4 CEV Abort at 300 sec Using Service Module

	Abort at	170 sec	Abort at	300 sec
	POST	Maveric	veric POST Maveric	
Maximum Acceleration (g's)	14.33	14.28	9.34	9.37
Maximum Dynamic Pressure (psf)	675.6	686.8	642.7	653.3
Maximum Heat Rate (BTU/ft2/sec)	32.05	32.18	64.00	64.38
Geodetic Latitude at 20 k-ft (deg)	33.72	33.73	36.30	36.32
Longitude at 20 k-ft (deg)	285.49	285.51	289.46	289.50

Figure 5.4.3.2-5 POST / Maveric Comparison Summary

#### 5.4.3.3 Abort Simulation Results

Figures 5.4.3.3-1 – 5.4.3.3-8 on the following pages show the results of this analysis. Data is provided for several key trajectory parameters as a function of abort time for three ascent trajectories: ISS reference trajectory, ISS February trajectory, and ISS July trajectory. Abort landing ground tracks are provided for an ISS reference trajectory and a due east reference trajectory. A summary of all key trajectory parameters and worse case abort times is provided. Note that aborts during an ISS mission tend to have slightly higher maximum acceleration, dynamic pressure, and heat rate than aborts during a due east mission. Also note the high probability that an abort during a CLV-4 mission will end with landing in the mid-Atlantic.

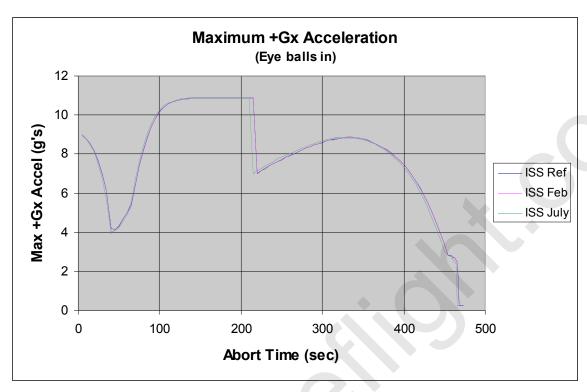
The peaks in eyeballs out and eyeballs down (Figures 5.4.3.3-1 and 5.4.3.3-2) accelerations occur because the aborts at high dynamic pressure involve a 3-second acceleration (eyeballs in) followed by free motion. The crew module (with spent launch escape system attached) is at high dynamic pressure and is relatively light. Immediately following LES burnout, the crew module is decelerated rapidly by the drag force (hence the large eyeballs out acceleration). The pitchover following LES burnout, made while dynamic pressure is still high, causes the large eyeballs down acceleration.

The eyeballs-down peak can be reduced by postponing the rotation to trim angle of attack. The eyeballs out acceleration is unavoidable since we have a relatively light vehicle traveling at high speed and at high dynamic pressure. Thus the crew will experience a high acceleration followed immediately by a high deceleration. This rapid change in acceleration may be an issue for crew health and may, for example, lead to a requirement that their heads be restrained.

This brings up a further issue with regard to abort near maximum dynamic pressure. The LES provides an acceleration of about 4g at its minimum just before 50 sec (Figure 5.4.3.3-1). This value is low due to the drag force induced by the relatively high speed at low altitude. The SRB-stack without the crew module at this time is accelerating at roughly 1.7 g towards the crew module (assuming the abort was caused by something other than SRB shutdown, vehicle disintegration, or loss of control). So the differential acceleration of the crew module relative to the stack is only 2.3 g, which is not very good for escaping a potential explosion compression wave.

After 3 seconds, the crew module acceleration is replaced by about 7.5 g deceleration. Allowing the SRB to continue burning towards the crew module is obviously not acceptable. Therefore we have a derived requirement to be able to shut the SRB down in case of abort. A trade is how to perform this shutdown: blowing the nozzle off, blowing the aft dome off, and separating at the aft segment are options. An alternative that would reduce the initial separation but aid in the later separation would be to command a SRB actuator hard-over to change the direction of thrust. This maneuver would likely result in vehicle breakup very shortly after abort, potentially when the CEV is still too close.

A larger LES is necessary in order to increase the escape capability. However, a larger LES to provide higher acceleration introduces problems. As an example, a LES providing 10 g for 3 seconds accelerates the crew module by 960 ft/sec, nearly doubling the relative velocity. The increase in velocity increases dynamic pressure by a factor of about 3.2 and increases the deceleration level felt after LES burnout by the same factor. A study is necessary for trading LES size versus deceleration levels/dynamic pressures versus successful abort escape. A simulation of the escape dynamics is needed to understand these effects clearly while examining solutions. The significant deceleration that is felt immediately after LES burnout could create a problem with respect to being too close to an explosive environment around the failed launch vehicle.



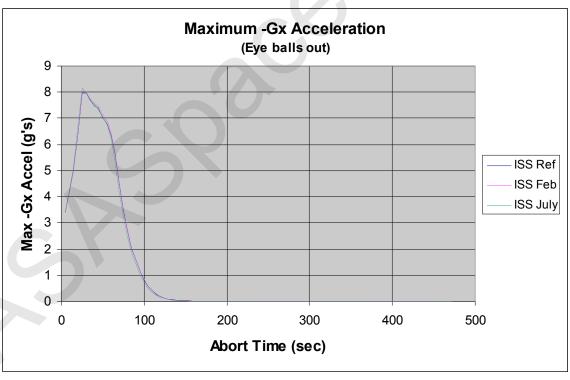
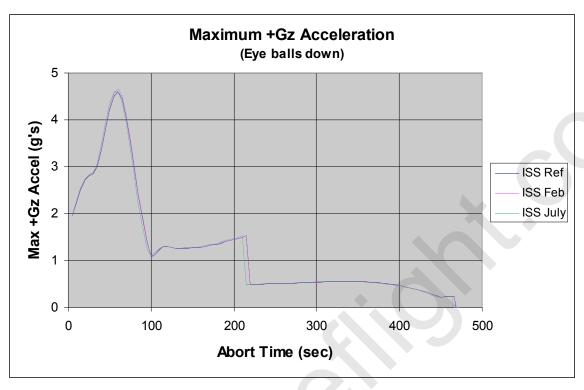


Figure 5.4.3.3-1 Maximum Axial Acceleration



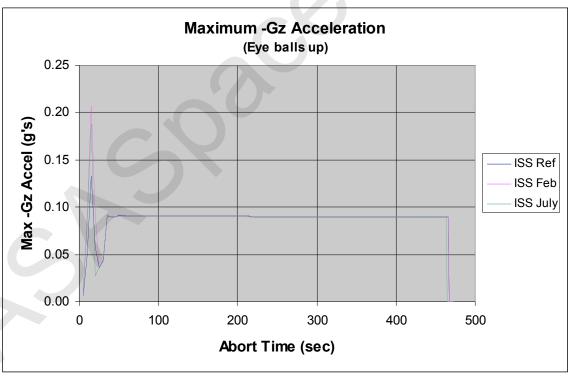


Figure 5.4.3.3-2 Maximum Normal Acceleration

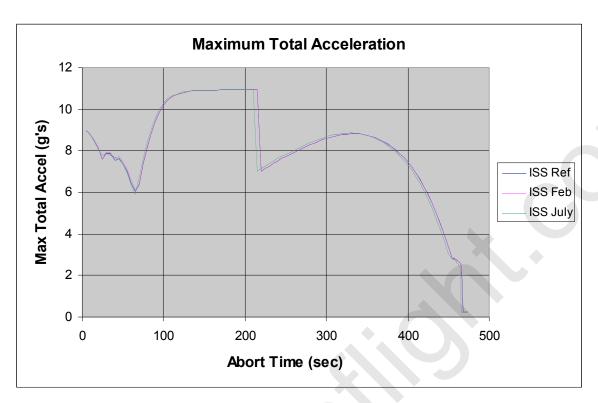


Figure 5.4.3.3-3 Maximum Total Acceleration

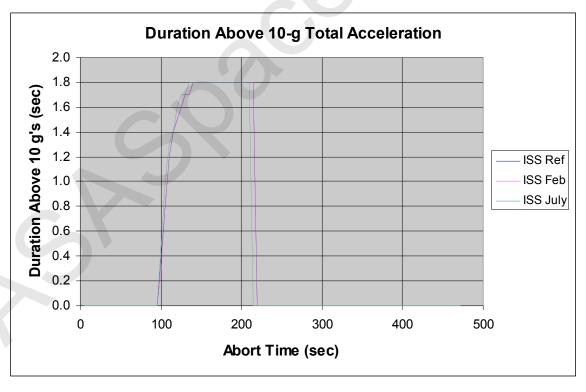
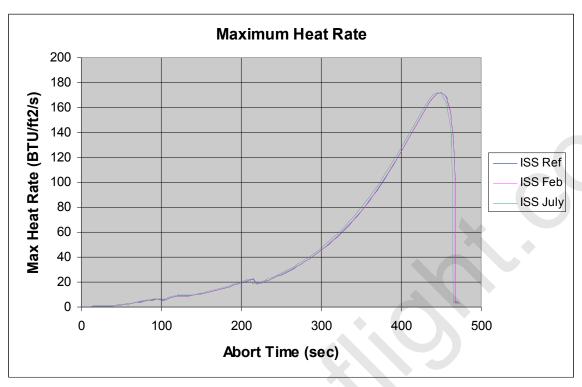


Figure 5.4.3.3-4 Duration Above 10-g Acceleration



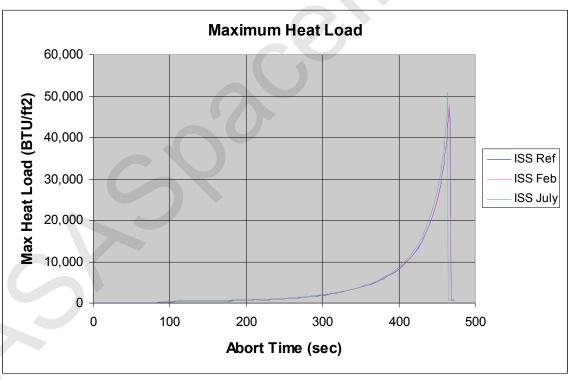


Figure 5.4.3.3-5 Maximum Heat Rate & Heat Load

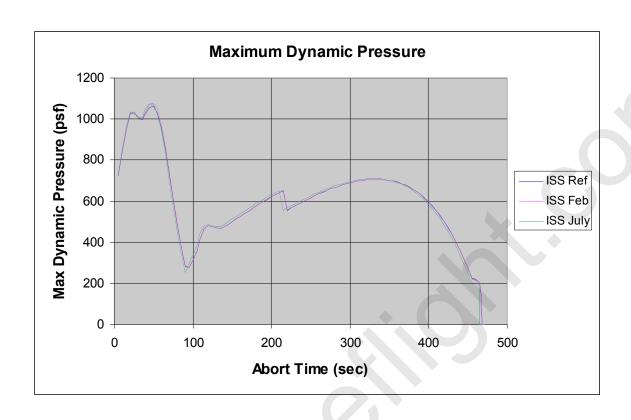


Figure 5.4.3.3-6 Maximum Dynamic Pressure

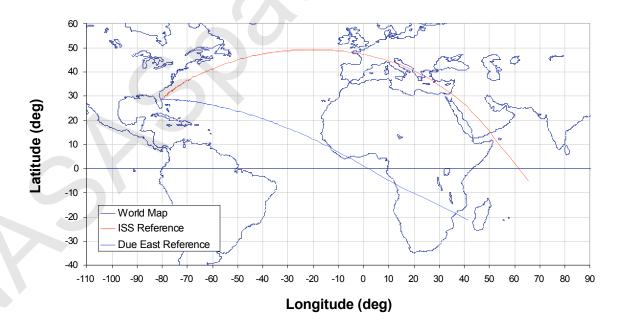


Figure 5.4.3.3-7 Abort Landing Ground Track

Mission Description CEV Configuration Ascent Trajectory Description	Due East	ISS	ISS	ISS
	Block 2	Block 1	Block 1	Block 1
	Reference	Reference	February	July
Number of Abort Simulations	109	109	108	106
Max +Gx Acceleration (eye balls in, g's) Abort Time (sec)	10.76	10.90	10.90	10.90
	220	215	215	210
Max -Gx Acceleration (eye balls out, g's) Abort Time (sec)	7.84	7.99	8.03	8.17
	30	25	25	25
Max +Gz Acceleration (eye balls down, g's) Abort Time (sec)	4.57	4.59	4.66	4.59
	60	60	60	60
Max -Gz Acceleration (eye balls up, g's) Abort Time (sec)	0.12	0.13	0.21	0.19
	15	15	15	15
Maximum Total Acceleration (g's) Abort Time (sec)	10.79	10.93	10.93	10.93
	220	215	215	210
Maximum Duration at 10 g's (sec) Abort Time (sec)	1.60	1.80	1.80	1.80
	220	215	215	210
Maximum Dynamic Pressure (psf) Abort Time (sec)	1,061	1,063	1,079	1,072
	50	50	50	50
Maximum Heat Rate (BTU/ft²/sec)	162.0	172.2	172.2	172.1
Abort Time (sec)	450	450	450	445
Maximum Heat Load (BTU/ft²)	51,199	47,486	48,047	50,903
Abort Time (sec)	467	466	466	464
Accumulative Abort Landing Times: Mid-Atlantic (sec)	462	435	435	430
North Atlantic (sec)	0	25	433 25	430 27
Europe & Middle-East (sec)	0	5	5	5
Africa (sec)	3	0	0	0
Indian Ocean (sec)	1	1	1	2
Orbital - No Landing (sec)	7	7	6	6

Figure 5.4.3.3-8 Abort Simulation Summary

# 5.4.3.4 Abort Monte Carlo Analysis

This section includes a 3-DOF abort Monte Carlo analysis for the CLV-4 launch vehicle. All simulations were modeled using Maveric II.

The undispersed abort analysis in the previous section used four Rev. 2 ascent trajectories. Due to lack of time and initial condition standard deviation data, this analysis uses the Rev. 1 ISS reference trajectory. In the future, this analysis should be updated using the Rev. 2 February and July ascent trajectories.

## 5.4.3.4.1 Ground Rules And Assumptions

All ground rules and assumptions from the previous section apply with the following exceptions:

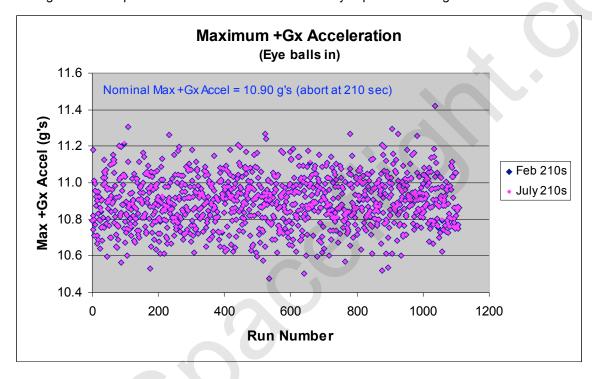
- CLV-4 ISS reference trajectory (Rev. 1) provided by MSFC/EV42
- February and July Monte Carlo simulations were run for aborts at specific times during the ascent trajectory. The selected abort times provide worse-case abort scenarios for key trajectory parameters, and were determined by an analysis similar to the one in the previous section (but not described in this report.)
   Following are the cases analyzed:
  - Abort at 25 sec (February and July)
  - Abort at 50 sec (February and July)
  - Abort at 60 sec (February and July)
  - Abort at 210 sec (February and July)
  - Abort at 330 sec (February and July)
  - Abort at 450 sec (February and July)
- Perturbed February and July GRAM atmosphere and winds
- 1108 3-sigma Monte Carlo runs for each case
- Following are the standard deviations used:
  - Lift & drag coefficients: 10%
    Trim angle-of-attack: 3 deg
    LES propellant mass: 2%
    SM fuel mass: 1%
  - SM oxidizer mass:
  - Total CEV dry mass: 246 lbm (1% of capsule mass)
- Initial condition standard deviations (provided by MSFC/EV42) are shown in the following table. Note that an additional 2 deg has been added to the angle-ofattack and sideslip standard deviations. This is to account for unknown attitude dispersions related to the launch vehicle failure which is requiring an abort.

Ca	se		l	nitial Cor	ndition St	andard [	Deviation	s	
Month	Abort Time (sec)	Altitude (ft)	Relative Velocity (fps)	Geodetic Latitude (deg)	Longitude (deg)	Relative FPA (deg)	Relative Heading (deg)	Angle-of- Attack (deg)	Sideslip (deg)
Feb.	25	231	19	0.0002	0.0003	0.540	1.925	3.654	3.688
	50	819	34	0.0011	0.0012	0.513	1.042	3.517	3.411
	60	1115	46	0.0017	0.0019	0.518	0.886	3.018	3.018
	210	4963	78	0.0202	0.0228	0.098	0.096	2.834	3.352
	330	3590	135	0.0414	0.0518	0.139	0.122	2.378	2.553
	450	1076	245	0.0753	0.1097	0.032	0.099	2.281	2.383
July	25	231	19	0.0002	0.0003	0.521	1.819	3.302	3.118
	50	821	33	0.0010	0.0011	0.425	0.824	2.875	2.864
	60	1119	46	0.0016	0.0017	0.424	0.685	2.968	2.944
	210	4961	77	0.0197	0.0220	0.098	0.095	2.892	3.403
	330	3590	134	0.0409	0.0512	0.139	0.113	2.377	2.707
	450	1074	245	0.0749	0.1091	0.031	0.096	2.281	2.427

Figure 5.4.3.4-1 Initial Condition Standard Deviations

#### 5.4.3.4.2 Abort Monte Carlo Results

Figures 5.4.3.4-1-5.4.3.4-11 on the following pages show the results of this analysis. Data is provided for several key dispersed trajectory parameters at worse-case abort times. Note the significant increase in several key parameters due to dispersions. The increase in maximum +Gz acceleration (eye balls up) is of particular interest because humans do not tolerate acceleration in this direction very well. Abort landing footprints are provided for the six abort times that were analyzed. Note that a late abort at 450 sec has a very large footprint in the North Atlantic. This large footprint could be significantly reduced in size using active guidance to target a specific landing site. Another option is to use the service module propulsive capability to target a specific landing site. A complete Monte Carlo statistical summary is provided in Figure 5.4.3.4-12.



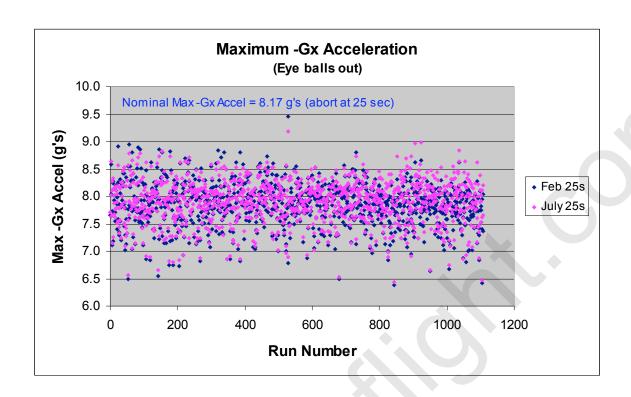
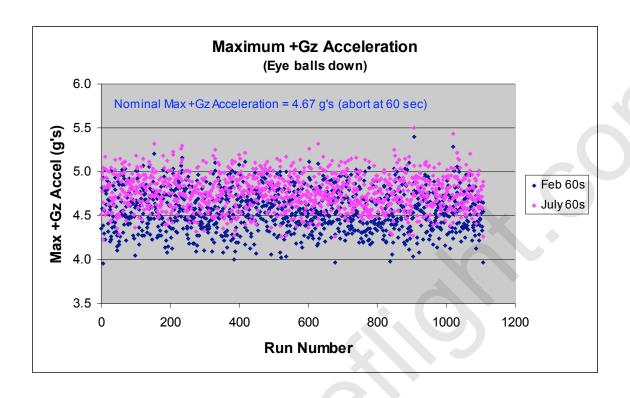


Figure 5.4.3.4-2 Dispersed Maximum Axial Acceleration



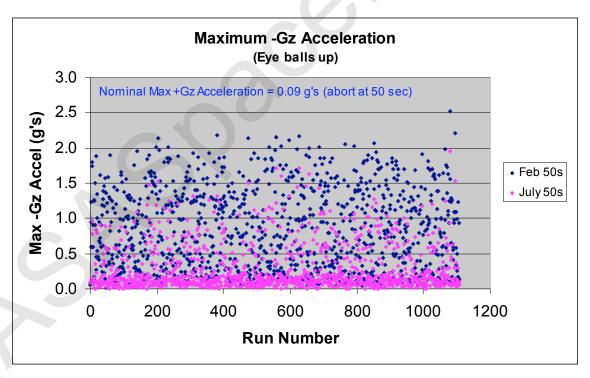


Figure 5.4.3.4-3 Dispersed Maximum Normal Acceleration

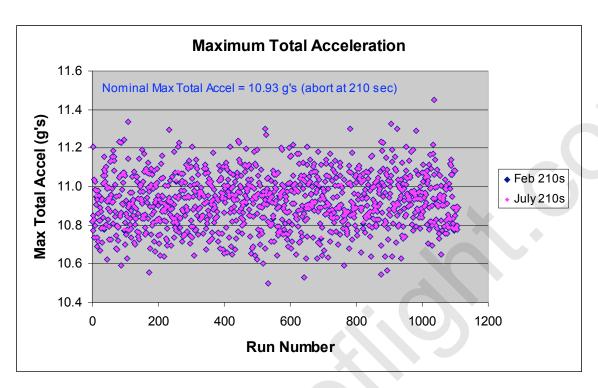


Figure 5.4.3.4-4 Dispersed Maximum Total Acceleration

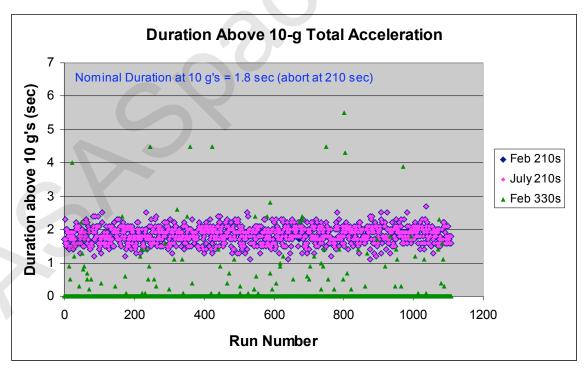


Figure 5.4.3.4-5 Dispersed Duration Above 10-g Acceleration

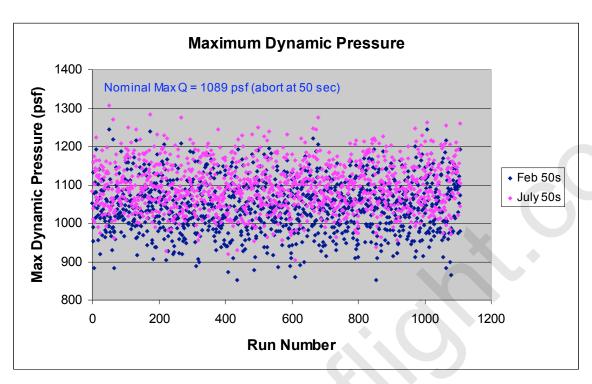


Figure 5.4.3.4-6 Dispersed Maximum Dynamic Pressure

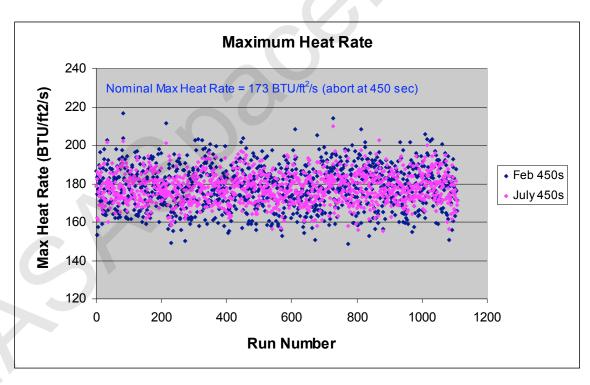
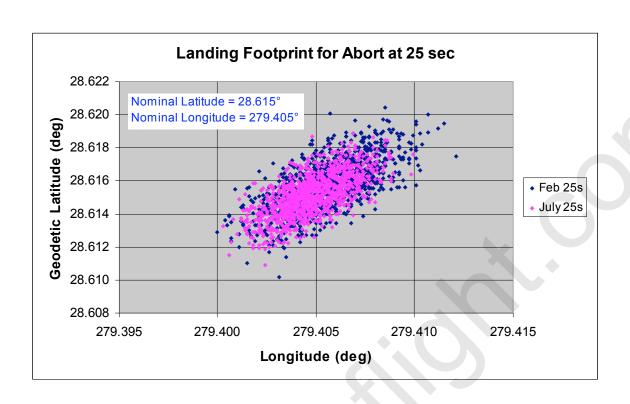


Figure 5.4.3.4-7 Dispersed Maximum Heat Rate



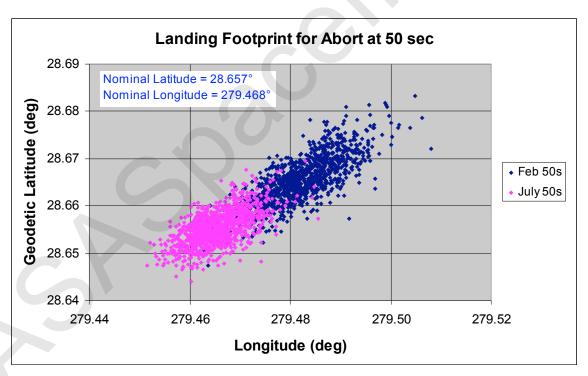
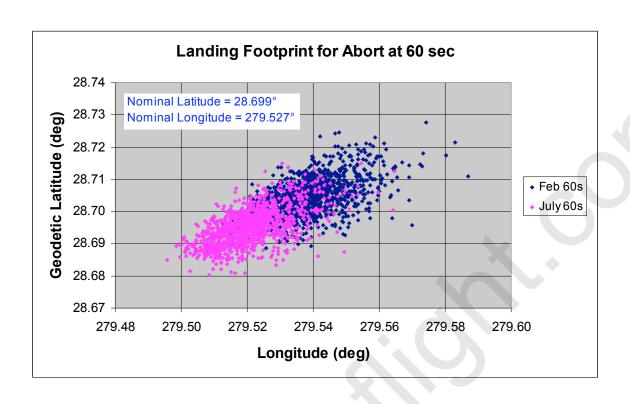


Figure 5.4.3.4-8 Landing Footprints for Aborts at 25 & 50 sec



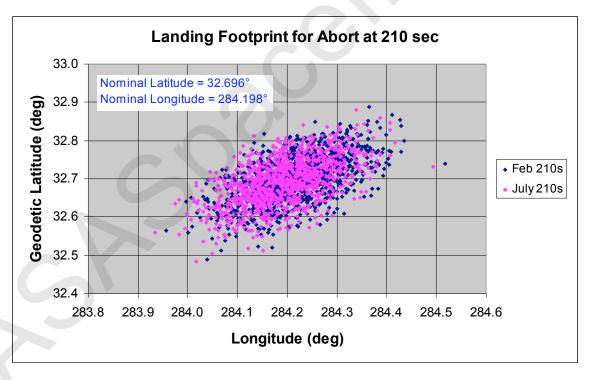
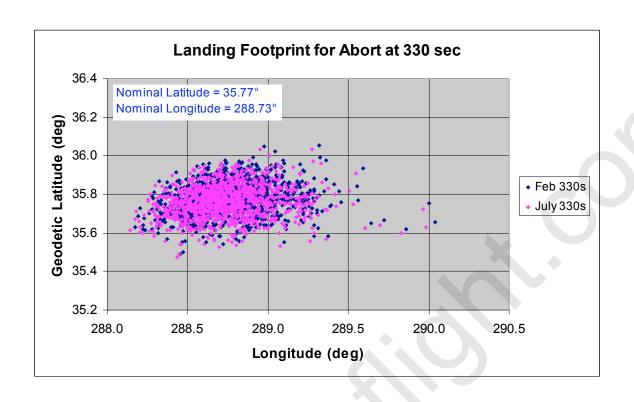


Figure 5.4.3.4-9 Landing Footprints for Aborts at 60 & 210 sec



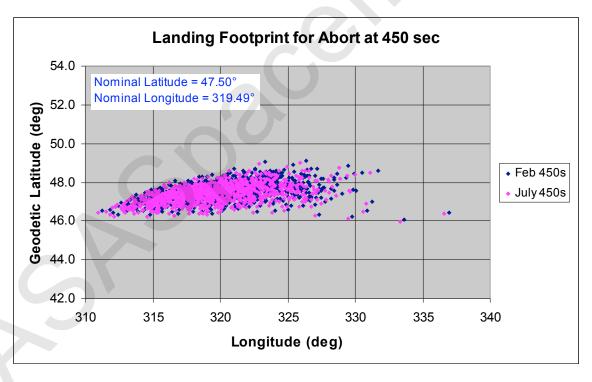


Figure 5.4.3.4-10 Landing Footprints for Aborts at 330 & 450 sec

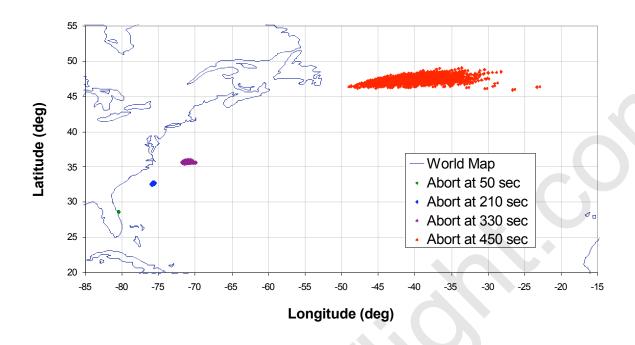


Figure 5.4.3.4-11 Dispersed Abort Landing Footprints

	Month	Abort Time (sec)	Minimum	Maximum	Average	Standard Deviation
Max +Gx Acceleration (eye balls in, g's)	Feb	210	10.4738	11.4209	10.8947	0.1369
	July	210	10.4738	11.4210	10.8948	0.1369
Max -Gx Acceleration (eye balls out, g's)	Feb	25	6.3810	9.4485	7.8390	0.3948
	July	25	6.4307	9.1841	7.9041	0.3885
Max +Gz Acceleration (eye balls down, g's)	Feb	60	3.9549	5.3922	4.5647	0.2345
	July	60	4.2261	5.5030	4.7682	0.1988
Max -Gz Acceleration (eye balls up, g's)	Feb	50	0.0199	2.5178	0.8071	0.5752
	July	50	0.0005	1.9584	0.2687	0.3159
Maximum Total Acceleration (g's)	Feb	210	10.4998	11.4519	10.9229	0.1376
	July	210	10.4998	11.4520	10.9230	0.1376
Duration Above 10-g Acceleration (sec)	Feb	210	1.10	2.70	1.83	0.25
	July	210	1.10	2.70	1.83	0.25
	Feb	330	0.00	5.50	0.11	0.50
Maximum Dynamic Pressure (psf)	Feb	50	853	1,244	1,045	68
	July	50	904	1,307	1,096	63
Maximum Heat Rate (BTU/ft <sup>2</sup> /sec)	Feb	450	148.5	216.4	178.0	11.1
	July	450	155.4	209.6	177.7	8.0
Landing Site Geodetic Latitude (deg)	Feb	25 50 60 210 330 450	28.610 28.647 28.689 32.489 35.488 46.067 28.611 28.644	28.620 28.683 28.728 32.888 36.056 49.098 28.619 28.669	28.616 28.666 28.704 32.698 35.777 47.502 28.615 28.655	0.002 0.005 0.006 0.060 0.084 0.467
	July	60 210 330 450	28.680 32.483 35.473 45.991	28.715 32.879 36.032 49.012	28.695 32.692 35.762 47.407	0.004 0.006 0.059 0.084 0.463
	Feb	25 50 60 210 330 450	279.400 279.463 279.512 283.956 288.172 311.240	279.412 279.508 279.587 284.517 290.038 336.974	279.406 279.482 279.539 284.215 288.765 320.005	0.002 0.007 0.010 0.079 0.243 3.525
Landing Site Longitude (deg)	July	25 50 60 210 330 450	279.400 279.452 279.496 283.935 288.144 310.993	279.410 279.486 279.564 284.494 289.986 336.598	279.405 279.465 279.521 284.193 288.739 319.674	0.002 0.005 0.010 0.078 0.242 3.495

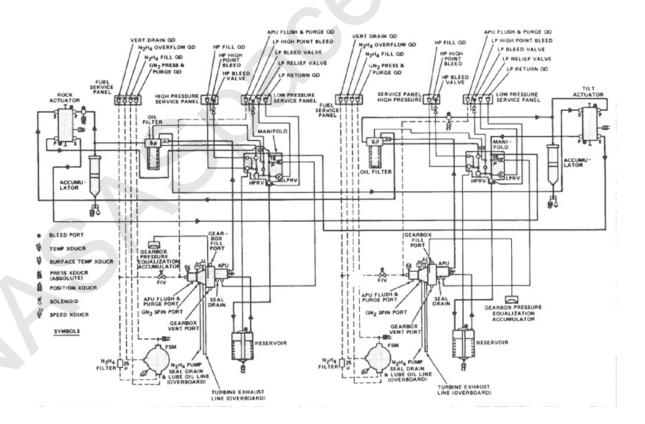
Figure 5.4.3.4-12 Monte Carlo Statistical Summary

## 5.4.4 CLV Launch Vehicle Thrust Vector Control System

#### 5.4.4.1 Description

The baseline Thrust Vector Control System (TVC) designs for first and second stage were chosen to be the same basic system as currently used on the RSRM in order to keep to the aggressive schedule for this vehicle and to keep cost down. Commonality between the booster and upper stage was seen as an advantage. Production for the current SRB Auxiliary Power Units (APU) is still active, while production for other TVC components could be reinstated within schedule to meet TVC system delivery. Furthermore, only a delta-qualification program would be required for the upper stage TVC, while no qualification program would likely be needed for the booster. No new technologies will be required, and the only modifications for upper stage TVC are likely to be limited to mounting structures, hydraulic lines and possibly the addition of cooling for the APU transmission oil.

The TVC system has redundancy in the APU's, and the actuator servovalves. The TVC system schematic is shown in figure below. Each APU is cross-fed to both actuators so that if one fails, hydraulic supply will still be fed to both actuators at slightly degraded velocity capability (3 deg/sec vs. 5 deg/sec).

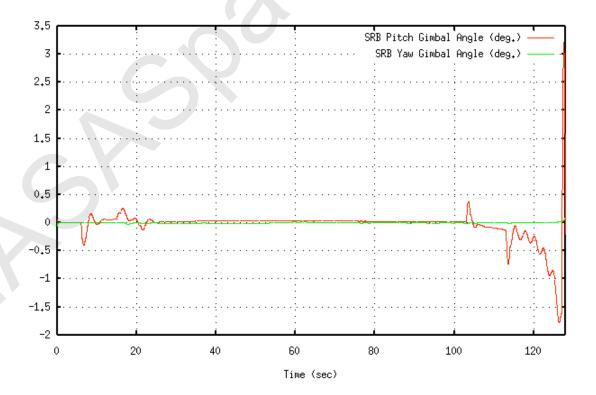


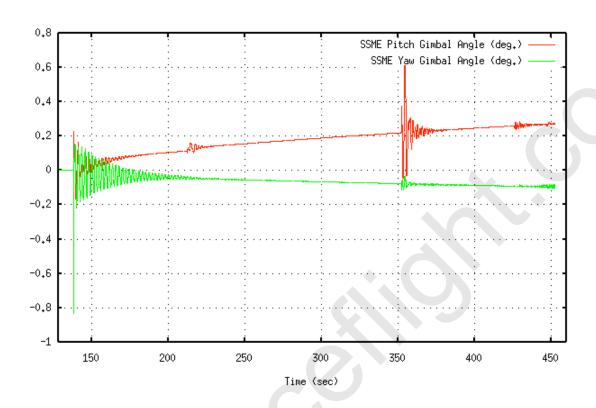
## 5.4.4.2 TVC Requirements

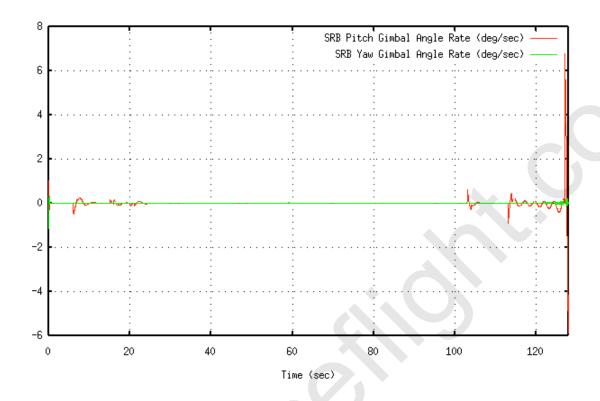
The current design SRB TVC requirements are

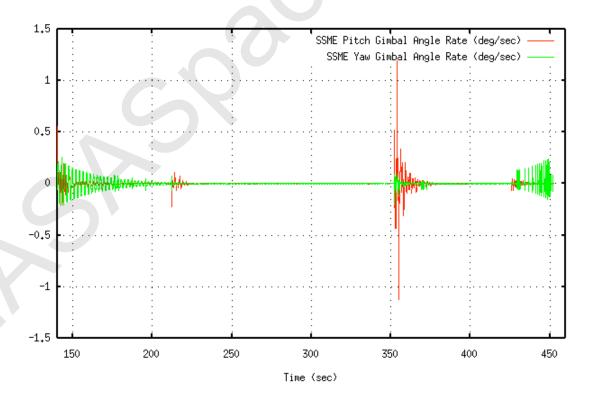
- ±5 degrees gimbal angle
- 6 deg/sec gimbal angle rate under rated load of 63,348 lbf
- 2 rad/sec^2 acceleration under rated load

Preliminary 6DOF simulations in Maveric are given in the following figures, for a nominal (non-dispersed vehicle or environment, no failures) flight with a reference wind. Among the models included in Maveric are a preliminary autopilot, a third order transfer function model of the TVC actuators and propellant slosh dynamics. These preliminary results show that gimbal angles are well within current design requirements. The SRB and SSME gimbal rates are also well within requirements except for a transient just before SRB burnout. This is not a realistic effect however, as SRB nozzle null commands prior to separation are not yet implemented in the simulation. The effects of slosh can be seen in the SSME gimbal and gimbal rate plots.









#### 5.4.5 CLV 3-DOF SIMULATION WITH COMPARISONS TO POST

This section includes a POST-to-MAVERIC comparison for the 28.5 deg inclination (AKA lunar mission) and the 51.6 deg inclination (AKA ISS mission), revision 1, 3-DOF reference trajectories using the CLV-4 launch vehicle model as described elsewhere in this report.

#### 5.4.5.1 GROUND RULES AND ASSUMPTIONS

The following ground rules and assumptions were used to generate these results:

- CLV-4 lunar and ISS reference trajectories (Rev. 1) (output from POST) provided by MSFC/EV42
- MAVERIC II version 2.0.0 with same general assumptions and modeling as those used in the POST simulations (described elsewhere in this report)
- Open-loop guidance during first-stage (using POST-derived attitude vs. altitude data) followed by closed-loop vacuum guidance during second-stage
- Closed-loop guidance uses the same target radius, inertial speed, inertial flight
  path angle, and orbital inclination as POST. In the 51.6 deg case, the ascent
  guidance in MAVERIC also targets for a specified nodal value to emulate the
  requirement for rendezvous in a particular orbital plane.
- Maximum command maneuver rates are set to 1 deg/sec each body axis applied to the guidance commands during second stage flight to stay within expected vehicle maneuverability limits.

## **5.4.5.2 COMPARISON RESULTS**

Figures 5.4.5-1-5.4.5-11 on the following pages show the 28.5 deg trajectory comparisons. While the POST & Maveric II trajectories have slight differences in some of the trajectory parameters, e.g., altitude, ground track, angles of attack and sideslip, the overall results compare very well. Most of these subtle differences are caused by differences in the steering and optimization formulations between POST and MAVERIC. For example, the closed-loop ascent guidance algorithm neglects the aerodynamic forces even though there is enough aerodynamic force at the beginning of second stage flight to perturb the trajectory. POST takes into account aerodynamic forces throughout the trajectory and also has perfect knowledge of the environment, whereas the ascent guidance in MAVERIC emulates an onboard software implementation and, hence, merely has estimates of the environment and vehicle states.

Figures 5.4.5-12 – 5.4.5-13 show the 51.6 deg trajectory comparisons of altitude and ground track. The other trajectory parameters have very minor differences comparable to the 28.5 deg mission comparisons and thus the corresponding plots are omitted. The injected mass-to-orbit difference between the POST and MAVERIC II simulations is typically less than 100 lbm.

### 5.4.5.3 Guidance Description and Results

The ascent guidance uses reference Euler angles vs. altitude tables for first stage flight and closed-loop vacuum guidance for second stage flight. The closed-loop guidance uses an optimal control solution with simplifying assumptions to achieve maximum payload to orbit (cf. Reference 1). The trajectory (as simulated within the guidance) is divided into the same flight phases as the Events Controller but with simplified start/end conditions. For example, the Launch Escape System is jettisoned at a particular altitude during the MAVERIC-simulated trajectory, but internal to the guidance calculations, the LES is jettisoned at some specified time after start of the second-stage burn, where the specified time is determined from the nominal trajectory. Guidance phases can also end at a particular vehicle mass and each phase has an end-of-phase jettison mass (possibly zero) associated with it. This simplified guidance formulation keeps the complexity of the onboard guidance software manageable while negligibly impacting guidance performance (even in the presence of dispersions.) Mass jettisons could be neglected in the guidance calculations, particularly when the jettison mass is small compared to the vehicle mass, but the resulting guidance attitude commands and predicted cutoff times have undesirable, albeit small, discontinuities in the region of the jettison event.

Figure 5.4.5-14 shows the predicted SSME cutoff time computed by guidance, that is, the predicted time at which the launch vehicle will attain the target orbit. The prediction is nearly constant over the duration of second-stage flight, changing by about one second over 250 sec of flight. Figure 5.4.5-15 shows the (implicitly) commanded angular velocity magnitude vs. time which verifies that the guidance is not commanding rates above 1 deg/sec, as specified via I-load. Figure 5.4.5-16 shows the number of iterations per guidance cycle. An iteration consists of a semi-analytical trajectory prediction, constraint error calculation, and independent variable correction. Most of the guidance cycles require one iteration with the worst-case guidance cycle requiring nine in the transition from open-loop to closed-loop flight.

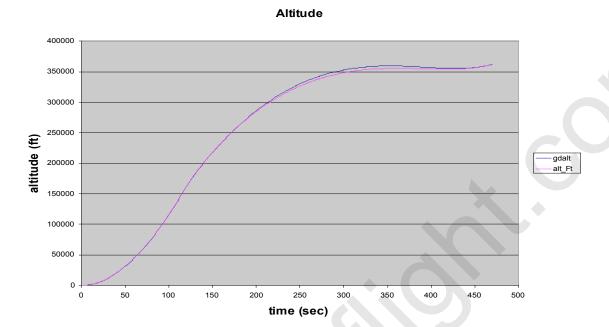


Figure 5.4.5-1: Altitude Comparison (28.5 deg mission)

#### **Ground Track**

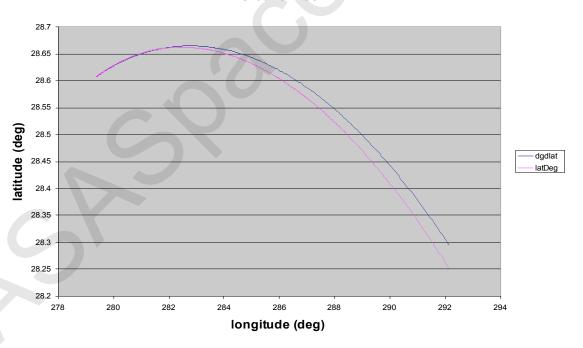


Figure 5.4.5-2: Ground Track Comparison (28.5 deg mission)

#### Earth-relative speed

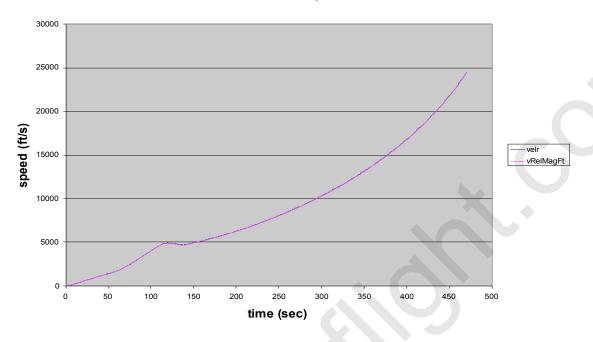


Figure 5.4.5-3 Earth-Relative Speed Comparison (28.5 Deg Mission)

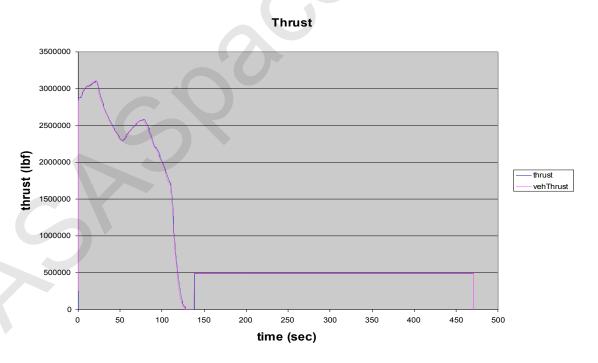


Figure 5.4.5-4 Thrust Comparison (28.5 Deg Mission)

## Vehicle Weight

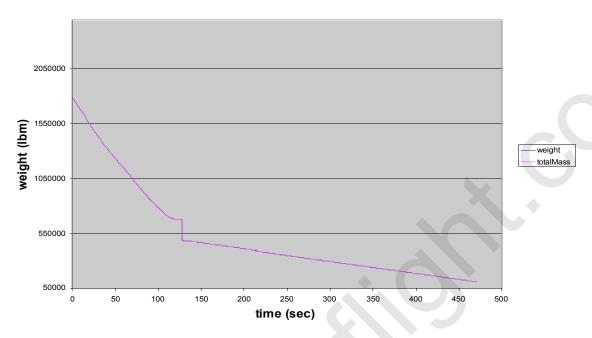


Figure 5.4.5-5 Earth-Relative Speed Comparison (28.5 Deg Mission)

# x-body Acceleration

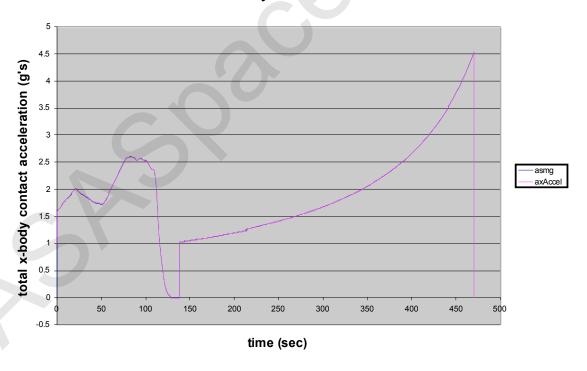


Figure 5.4.5-6 Axial Acceleration Comparison (28.5 Deg Mission)

## Angle of Attack

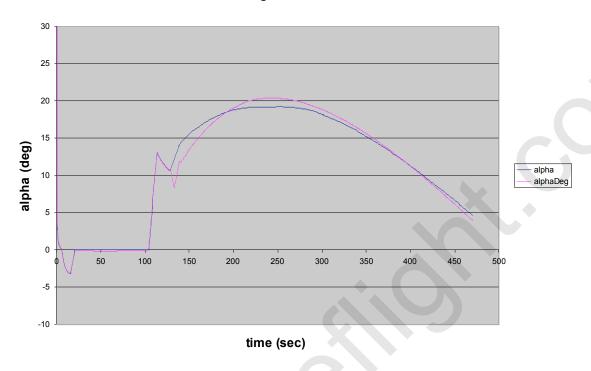


Figure 5.4.5-7 Angle Of Attack Comparison (28.5 Deg Mission)

# Angle of sideslip

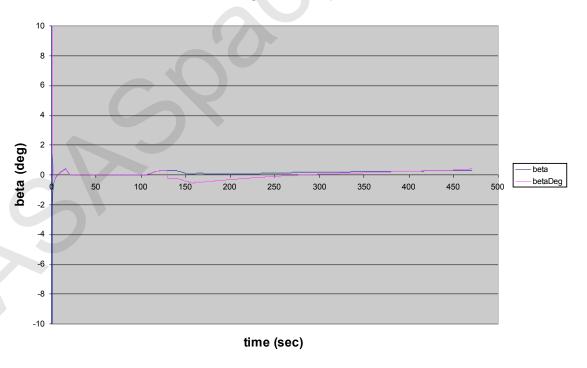


Figure 5.4.5-8 Angle Of Sideslip Comparison (28.5 Deg Mission)

#### product of dynamic pressure and angle of attack

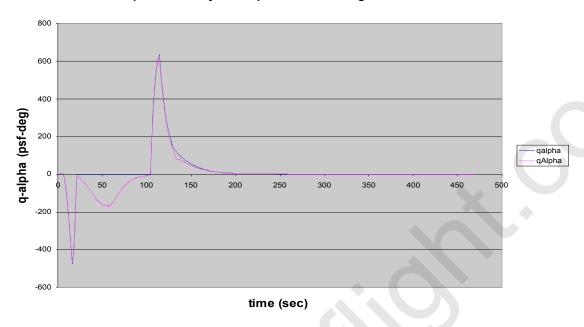


Figure 5.4.5-9 q-Alpha Comparison (28.5 Deg Mission)

## **Dynamic Pressure**

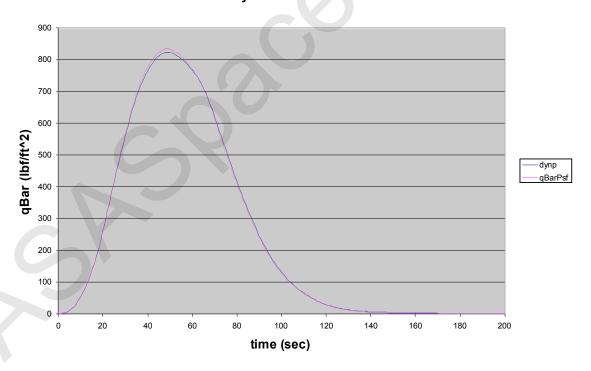


Figure 5.4.5-10 Dynamic Pressure Comparison (28.5 Deg Mission)

## **NED-relative Euler Angles**

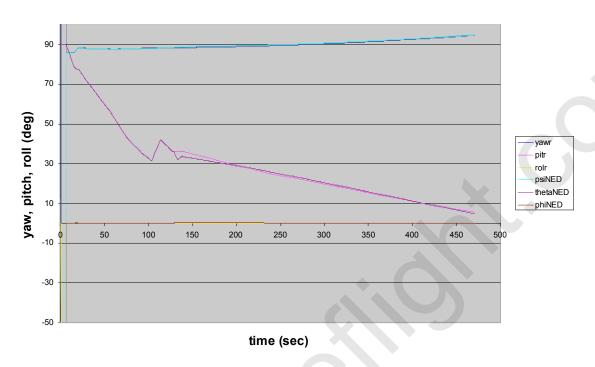


Figure 5.4.5-11: Euler Angles Comparison (28.5 Deg Mission)

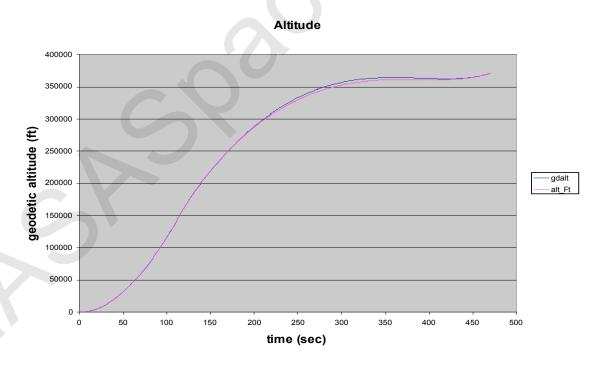


Figure 5.4.5-12 Altitude Comparison (51.6 Deg Mission)

#### **Ground Track**

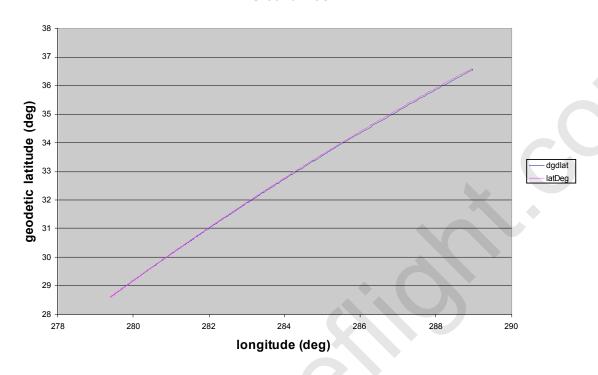


Figure 5.4.5-13 Ground Track Comparison (51.6 Deg Mission)

## Guidance Commanded Cutoff Time vs. Mission Time (s)

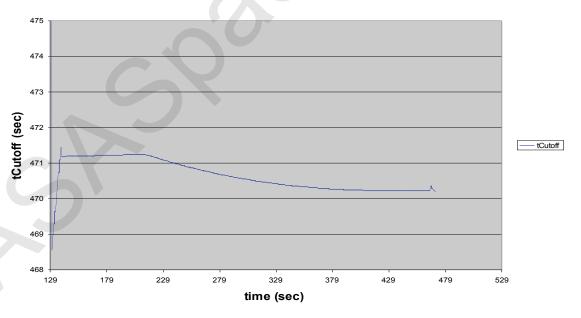


Figure 5.4.5-14: Guidance-Commanded Cutoff Time (28.5 Deg Mission)

#### Angular Velocity Magnitude (deg/s)

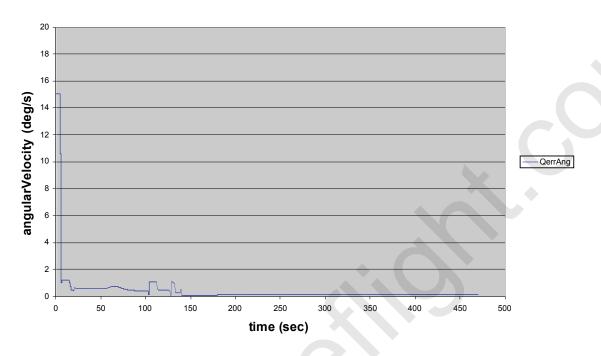


Figure 5.4.5-15: Guidance-Commanded Maneuver Rate (28.5 Deg Mission)

## Number of Closed-Loop Guidance Iterations Per Guidance Cycle

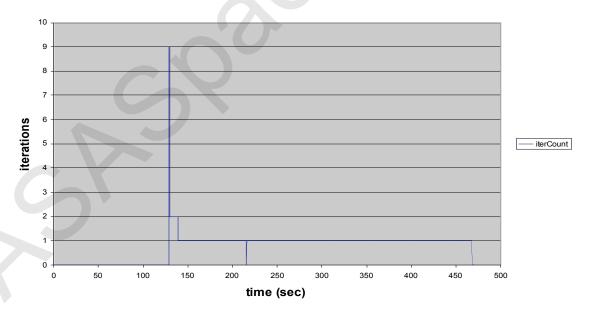


Figure 5.4.5-16 Ascent Guidance Iterations (28.5 Deg Mission)

#### 5.4.5.4 3-DOF DISPERSED TRAJECTORY RESULTS

This section includes results of a 3-DOF Monte Carlo analysis for the CLV-4 launch vehicle. All simulations were modeled using Maveric II. The Rev. 1 ISS trajectory was used as the nominal. The nominal trajectory was biased to a mean annual wind. In the future, this analysis will be updated using trajectories biased to mean February and July winds and corresponding SRB propellant mean bulk temperatures (PMBT). This will reduce the structural load indicators as compared to the values seen herein. The ISS mission is the worst case from a guidance and performance perspective because of the requirement to target a particular orbital plane.

#### 5.4.5.4.1 GROUND RULES AND ASSUMPTIONS

All ground rules and assumptions from section 5.4.5 apply with the following additions:

- 1) February and July Monte Carlo sets (1,350 trajectories per set)
- 2) Perturbed February and July GRAM atmosphere
- 3) 450 measured KSC wind profiles per season, Jan/Feb/Mar (for February set of runs), and Jun/Jul/Aug (for July set of runs)
- 4) nominal PMBT = 60 deg F for all trajectories (next analysis cycle the nominal PMBT will be appropriate to the season, i.e., 61 deg F Feb, 80 deg F July)

Table 5.4.5.4-1 lists the dispersion values used:

Table 5.4.5.4-1: Dispersion Models

	Dispersion Values (1-
	sigma)
SSME Dispersion Model (Bill Greene, April 2004)	
Mixture Ratio Dispersion	0.045
SSME thrust dispersion (lbf)	896.95
SSME Isp (s)	0.39
LOX loaded (nominal = 311070 lbm)	0.33%
LH2 Loaded (nominal = 52113 lbm)	0.33%
SSME pitch cant angle (deg)	0.1923
SSME yaw cant angle	0.1923
SRB Dispersion Model (Tim Olive, April 2004)	
PMBT (nominal = 60 deg F)	0.4667
burn rate (in/s)	0.0026
SRB loaded propellant (lbm)	774
SRB pitch cant angle (deg)	0.2477
SRB yaw cant angle (deg)	0.2477
Aero Coefficients (all, including base force)	3.33%
Navigation position-x (ft)	1123
у	1123
Z	1123
Navigation velocity-x (ft/s)	5.167
у	5.167
Z	5.167
KSC Day-of-launch winds (Frank Leahy/EV13), Nominal month, previous a	and following, 150 each
Atmospheric density, temp, pres - Perturbed GRAM for nominal month	

#### **5.4.5.4.2 MONTE CARLO RESULTS**

Figures 5.4.5.4-1 – 5.4.5.4-5 show the results of the February set of 1,350 dispersed trajectories. A complete Monte Carlo statistical summary for the February trajectories is provided in Table 5.4.5.4-2. It is seen from Figure 5.4.5.4-2 that seven trajectories out of 1,350 fail to make target insertion orbit due to running out of LOX or LH2. This means that both FPR and fuel bias need to be increased somewhat – this will reduce the payload by the same amount. The worst-case maximum dynamic pressure is less than 950 psf (nominal = 832 psf) The worst-case q- $\alpha$  (magnitude) is less than 5,800 psf-deg, the worst-case q- $\beta$  is less than 5,300 psf-deg. These load indicator results should improve somewhat by using a mean monthly wind for the trajectory design.

Figures 5.4.5.4-6 - 5.4.5.4-10 show the results of the July set of 1,350 dispersed trajectories. A complete statistical summary for the July trajectories is provided in Table 5.4.5.4-3. The results

are very similar to that of the February results. Future analyses will use seasonally-appropriate values of PMBT so that the February vs. July results will likely be more dissimilar. The worst-case maximum dynamic pressure is less than 970 psf. The worst-case  $q-\alpha$  (magnitude) is less than 4,200 psf-deg, the worst-case  $q-\beta$  is less than 4,500 psf-deg. The summer months apparently result in more benign values of the load indicators  $q-\alpha$  /  $q-\beta$ .

#### Injected Weight vs. Run Number

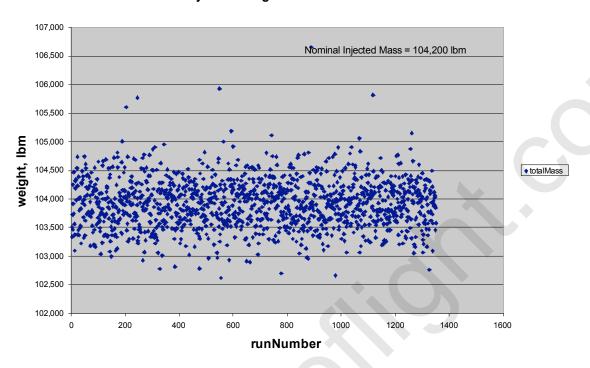


Figure 5.4.5.4-1: Injected Weights: 1,350 February Trajectories

#### Injection Apogee vs. Perigee Altitudes

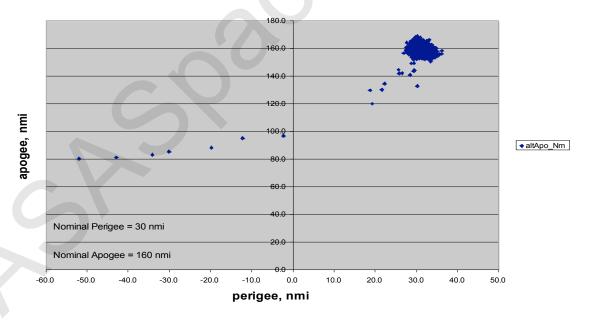


Figure 5.4.5.4-2: Injection Apogees, Perigees, 1,350 February Trajectories

# Flight Performance Reserve Remaining vs. Fuel Bias Remaining at MECO

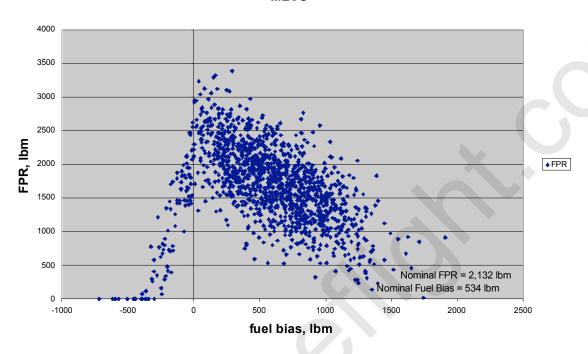


Figure 5.4.5.4-3: Flight Performance Reserve, Fuel Bias, 1,350 February Trajectories

#### Max Dynamic Pressure vs. Run Number

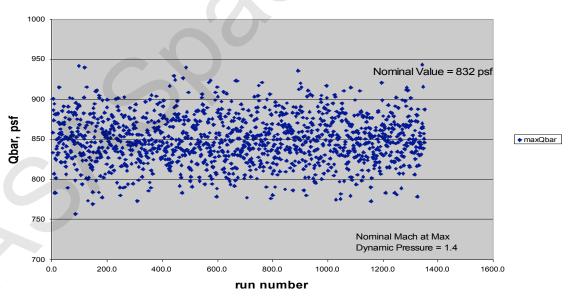


Figure 5.4.5.4-4: Maximum Dynamic Pressure, 1,350 February Trajectories

#### Maximum Abs Q-alpha / Max Abs Q-beta

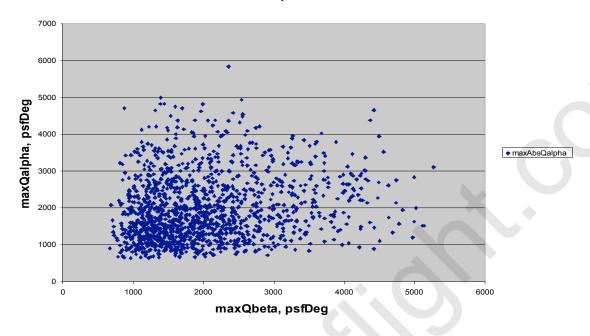


Figure 5.4.5.4-5: Max Q- $\alpha$  / Max Q- $\beta$ , 1,350 February Trajectories

## Injected Weight vs. Run Number

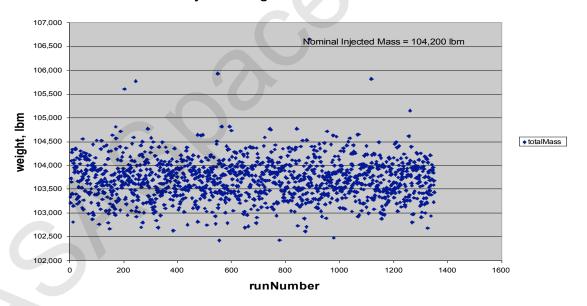


Figure 5.4.5.4-6: Injected Weights: 1,350 July Trajectories

#### Injection Apogee vs. Perigee Altitudes

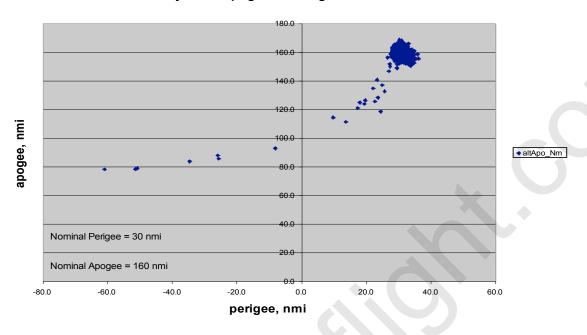


Figure 5.4.5.4-7: Injection Apogees, Perigees, 1,350 July Trajectories

# Flight Performance Reserve Remaining vs. Fuel Bias Remaining at MECO

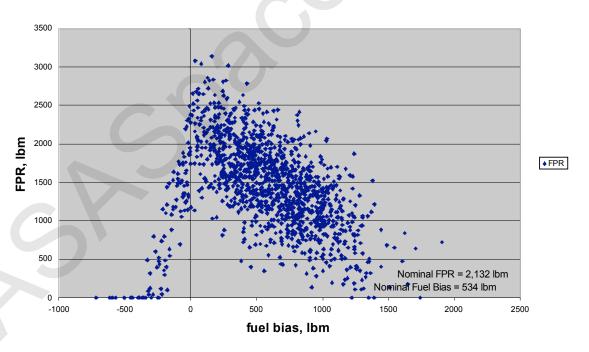


Figure 5.4.5.4-8: Flight Performance Reserve, Fuel Bias, 1,350 July Trajectories

#### Max Dynamic Pressure vs. Run Number

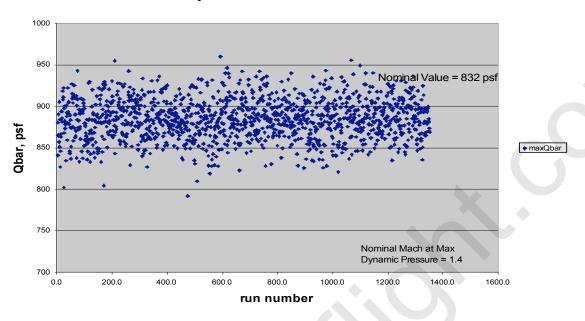


Figure 5.4.5.4-9: Maximum Dynamic Pressure, 1,350 July Trajectories

## Maximum Abs Q-alpha / Max Abs Q-beta

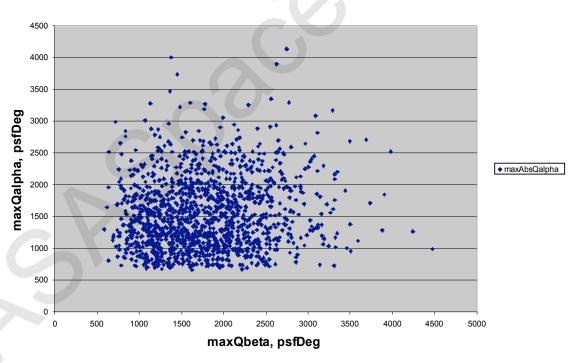


Figure 5.4.5.4-10: Max Q- $\alpha$  / Max Q- $\beta$ , 1,350 July Trajectories

Table 5.4.5.4-2 Monte Carlo Statistical Summary, 1,350 February Trajectories

				standard
PARAMETER	min	max	average	deviation
MECO injected mass, lbm	102,627	106,660	103,932	424
flight performance reserve remaining, lbm	0	3,387	1,677	590
fuel bias remaining, lbm	-722	1,906	571	388
LOX used, lbm	306,505			
LH2Used, lbm	50,375	52,250	51,323	
LOX remaining, lbm	15	5,074	1,555	
LH2 remaining, lbm	0	2,017	792	335
MECO altititude, ft	367,521	374,216	371,147	1,052
apogee altitude at MECO, nmi	80.5	169.0	159.4	6.3
perigee altitude at MECO, nmi	-52.0	36.1	30.4	4.6
MECO inclination, deg	51.53394		51.59959	
max dynamic pressure, psf	757	944	850	30
min q-alpha, psf-deg	-5853		-1858	979
max q-alpha, psf-deg	498		809	
min q-beta, psf-deg	-5263	-88	-1904	978
max q-beta, psf-deg	0	2801	785	
max axial accel, g's	4.59238		4.71251	0.01921
min normal accel, g's	-0.04776		-0.01833	
max normal accel, g's	0.00000		0.02514	
peak heat rate (Chapman), Btu/ft^2/s	3.78713		4.16210	
MECO inertial speed, fps	25,564.7			
MECO inertial Flight Path, deg	0.73092		0.84155	
SRB sep altitude, ft	176,148		179,868	
SRB sep Mach	4.16		4.48	
SRB sep time, s	123.85		128.05	
SRB sep dynamic pressure, psf	10.90		13.46	
Mach at Max Heat Rate	22.43			
MECO time, s	465.06			
LES jettison altitude, ft	300000			
LES jettison Mach	6.1	8.961624	7.37	0.51
LES jettison time, s	197.8	232.12	213.04	5.16

Table 5.4.5.4-3: Monte Carlo Statistical Summary, 1,350 July Trajectories

PARAMETER	Min	max	average	standard deviation
MECO injected mass, lbm	102,431	106,660	103,713	426
flight performance reserve remaining, lbm	0	3,142	1,456	577
fuel bias remaining, lbm	-722	1,903	566	388
LOX used, lbm	306,626	312,392	309,744	902
LH2Used, lbm	50,401	52,267	51,355	303
LOX remaining, lbm	0	5,074	1,367	551
LH2 remaining, lbm	0	1,991	760	333
MECO altitude, ft	367,498	374,192	371,131	1,053
apogee altitude at MECO, nmi	78.5	169.1	159.2	7.0
perigee altitude at MECO, nmi	-61.0	36.0	30.3	5.3
MECO inclination, deg	51.53374	51.65396	51.59957	0.01949
max dynamic pressure, psf	792	960	885	24
min q-alpha, psf-deg	-3474	0	-686	402
max q-alpha, psf-deg	621	4142	1461	600
min q-beta, psf-deg	-2550	0	-445	360
max q-beta, psf-deg	83	4470	1729	663
max axial accel, g's	4.59239	4.78148	4.72250	0.01937
min normal accel, g's	-0.05597	-0.00654	-0.02323	0.00833
max normal accel, g's	0.00000	0.04748	0.01195	0.00869
peak heat rate (Chapman), Btu/ft^2/s	4.06980	4.72254	4.41459	0.09630
MECO inertial speed, fps	25,543.9	25,877.8	25,861.1	21.3
MECO inertial Flight Path, deg	0.73058	0.90914	0.84147	0.02877

SRB sep altitude, ft	176,195	184,021	179,973	1,220
SRB sep Mach	4.39	4.97	4.65	0.10
SRB sep time, s	123.85	132.44	128.05	1.38
SRB sep dynamic pressure, psf	12.89	18.80	15.28	0.87
Mach at Max Heat Rate	22.14	26.40	24.79	0.31
MECO time, s	465.37	476.47	470.71	1.89
LES jettison altitude, ft	300000	300011	300,004.86	2.83
LES jettison Mach	6.0	8.79337	7.40	0.51
LES jettison time, s	197.8	231.99	212.90	5.15

#### References

1. Dukeman, G., "Atmospheric Ascent Guidance for Rocket-Powered Launch Vehicles", AIAA Paper 2002-4559, Proceedings of the AIAA Guidance, Navigation, and Control Conference, Monterey, CA, August 5-8, 2002...

## 5.4.6 CLV 6DOF Simulation: Modeling and Analyses

The ascent portion of the CLV flight has been modeled in MAVERIC II version 2.0. A preliminary SRB re-entry simulation has been modeled but not fully tested.

Trajectory: The trajectory is the rev 1 reference trajectory to the international space station as implemented in MAVERIC by EV42/Greg Dukeman documented in section 5.4.5.

Mass Properties: The mass properties are the rev 2 mass properties provided by EV12/Holly Chandler. The rev 2 mass properties were used with the rev 1 trajectory because 6dof mass properties were not generated for the rev 1 trajectory and the rev 2 trajectory was not yet available in MAVERIC.

RCS Model: Simple RCS model. No ramp up or down of thrust. No limiting of number of thruster firings or on time. The first stage has 16 900 lbf thrusters for roll control. The second stage has 18 thrusters. Six of them are 900 lbf thrusters for axial thrust and twelve of them are 100 lbf thrusters for 3 axis attitude control. Although the axial thrusters have been implemented, the control system is not commanding them yet.

Controller: A PID (proportional, integral, derivative) controller is used for ascent flight. For the first stage burn, nozzle gimballing is used for pitch and yaw control. Reaction control thrusters on the first stage are used for roll control. During the coast phase before SSME engine start reaction

control thrusters will be used for 3 axis attitude control. However, in the simulation, ideal torques are currently used during the coast phase. For the second stage burn, engine gimballing is used for pitch and yaw control. Reaction control thrusters on the second stage are used for roll control. The jet selection is based on an algorithm provided by EV41/Richard Dabney.

Slosh: Slosh models provided by ER41/José Perez-Batista are included for the upper stage lox and hydrogen tanks during first and second stage burn. The damping used is 7% and 2% for the lox and hydrogen tanks respectively. This assumes baffles in the lox tank will provide 7% damping and 2% damping for the fuel tank. These numbers are preliminary and it may be possible to reduce them. However, at this point these are the damping levels required.

Loads: A loads model provided by EV31/Dave McGhee is implemented and running but has not been fully tested yet.

Aerodynamics: The 6 DOF aerodynamics were provided by EV33/Joe Lowery for the stack. Aerodynamics for the second stage was not included at this point in the analysis.

Aero-thermal: An aero-thermal model provided by EV33/Dr. C. Mark Seaford is implemented and running for ascent but has not been fully tested yet.

Actuator Models: The SRB and SSME actuators are modeled using a 3<sup>rd</sup> order actuator model provided by EV41/Charles Hall. The SRB actuator coefficients are being used for both the SSME and the SRB since the SRB actuators are being base lined for both.

SRB Roll Torque Disturbance: Implementation of the Internal Solid Rocket Motor Roll Torque in MAVERIC was provided by EV41/Susan Elrod. It has been nominally tested but the controller gains will need to be adjusted to accommodate this disturbance. Therefore it was not included in the dispersion runs.

The internal roll torque that is generated by solid rocket motors is needed for vehicle design to assess the requirements for roll control. Roll torques generated by internal vortex flow and nozzle ablation are the two observed most prominent sources. There is no known method of predicting solid rocket motor generated roll torques, but Reference 1 (Roll Torques Produced by Fixed Nozzle Solid Rocket Motors by R. N. Knauber) indicates that torque values may be extrapolated from historical data. Mission-specific ground testing as well as flight testing will be required for the CLV to accurately determine the magnitude for the motor design. The following figure (provided by EV41/Richard Dabney) depicts historical rocket motor peak internal torques.

#### **Historical Rocket Motor Peak Internal Torques**

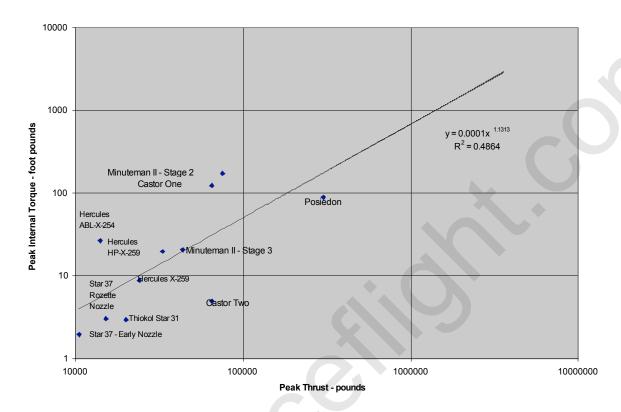


Figure 5.4.6-1 Historical Rocket Motor Peak Internal Torques

The Hercules X-259 was chosen as a basis for extrapolation of the SRB data because it was the closest in size to a Shuttle SRB (the X-259 provides 50,000 lb thrust whereas an SRB provides 3.3 M- lbs). The two also have the fin-propellant grain pattern. The following table depicts a spreadsheet that has been generated from the subsequent plot of the X-259:

time	time scaled	time scaled	x-259 thrust	SRB scaled
	for 5-segment	for 4-segment		thrust
	SRB	SRB		
C	0	0	0	0
1.25	11	7.9125	-0.3	-685.71
1.3	11.44	8.229	0	0
2.5	22	15.825	0.9	2057.13
3.1	27.28	19.623	3.1	7085.67
4	35.2	25.32	-2.5	-5714.25
4.6	40.48	29.118	-3	-6857.1
5	44	31.65	-0.65	-1485.705
6	52.8	37.98	8.75	19999.88
6.2	54.56	39.246	0	0
7.5	66	47.475	0.75	1714.275
8.5	74.8	53.805	0	0
9.3	81.84	58.869	-2.5	-5714.25
10	88	63.3	-0.5	-1142.85
10.2	89.76	64.566	0	0
10.3	90.64	65.199	-0.05	-114.285
11	96.8	69.63	0.1	228.57
12	105.6	75.96	-0.05	-114.285
13	114.4	82.29	0	0
14	123.2	88.62	0.1	228.57
14.3	125.84	90.519	-0.1	-228.57
14.5	127.6	91.785	0	0
15	132	94.95	0	0

Table 5.4.6-1 SRB Roll Disturbance Scaling

The third and fifth columns of the spreadsheet were input as a table into MAVERIC in the caseCLV4/aeroCoef.dat file. MAVERIC interpolates the necessary data.

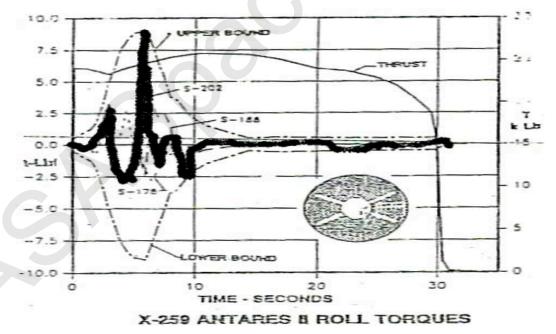


Figure 5.4.6-2 X-259 Antares Roll Torques

Teleconferences with SRB personnel indicated there were no roll control issues involving the SRB's during Shuttle operations. However, since two SRB's are used on the STS, ample roll

control is available by gimballing the SRB nozzles in opposite directions, making it possible to overcome any internal torques which may occur. On the proposed CLV, in contrast, the RCS must do this entire job. For the sake of conservatism and safety, we choose to include the roll torque as a disturbance, and sized RCS thrusters accordingly for roll control.

Forward Work: Future additions to the simulation include flex filters, jet selection instead of ideal torques during the coast phase, settling burn, thrust ramp up and down, bus delays, flex modeling, higher fidelity rcs models, control gains adjusted for stability margins, control gains adjusted to handle the SRB roll disturbance torque, and a thermal model for SRB re-entry. Full testing and check out of the loads and aero-thermal models is also planned. Additional dispersion parameters including mass properties and detailed RCS parameters.

6 DOF Simulation Results: Preliminary 6dof dispersions have been run for the July and February winds. The dispersion parameters are the same as those documented in the 3dof section with the addition of reaction control system thrust, isp, mixture ratio and thruster alignment. Some sample results can be seen in Figures 5.4.6-2 through 5.4.6-11. The pitch gimbal angle frequently hits the 5 deg limit during 1<sup>st</sup> stage burn in the dispersions as can be seen in Figure 5.4.6-6. More work is required to determine whether this can be improved. Updated gains from the stability analysis have not yet been folded into these results. It is expected that there will be control system performance degradation necessary to achieve stability margins. Statistics for pertinent variables for the February and July dispersions can be seen in Tables 5.4.6-2 and 5.4.6-3 respectively. There were 25 out of 1350 runs in the July dispersions that either lost control or failed to achieve the correct orbit. There were 41 out of 1350 runs in the February dispersions that either lost control or failed to achieve the correct orbit. The runs that lost control are not included in the statistics table. These results are very preliminary and should be viewed as a first cut at dispersions. Higher fidelity modeling and control system design work are still required. There is still significant work to be done before an accurate assessment of controllability and stability and can be made.

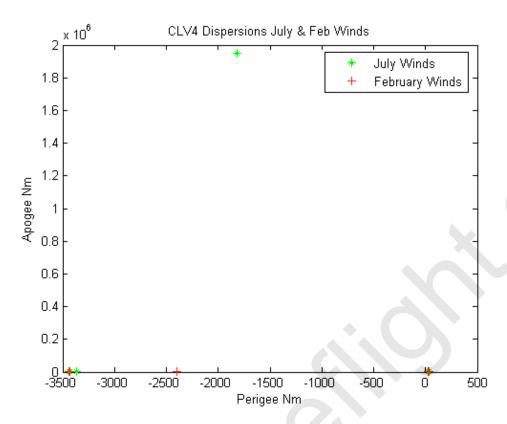


Figure 5.4.6-3 Apogee vs. Perigee at MECO

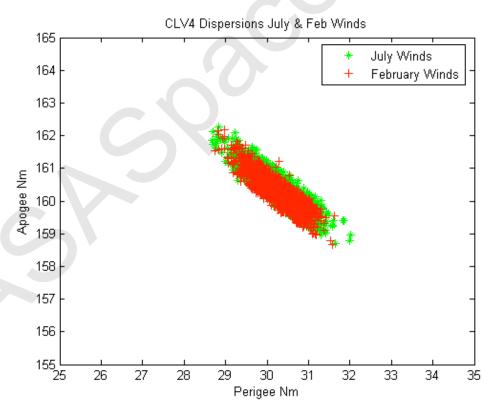


Figure 5.4.6-4 Apogee vs. Perigee at MECO Zoomed In

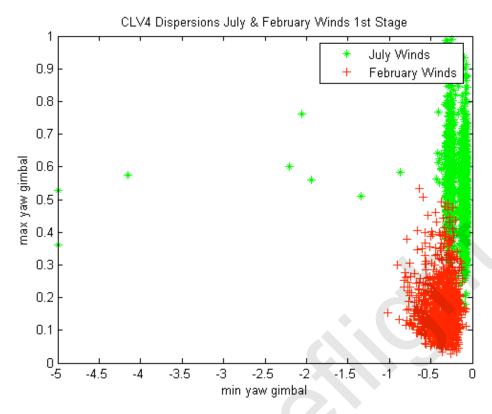


Figure 5.4.6-5 1<sup>st</sup> Stage Maximum vs. Minimum Yaw Gimbal Angle

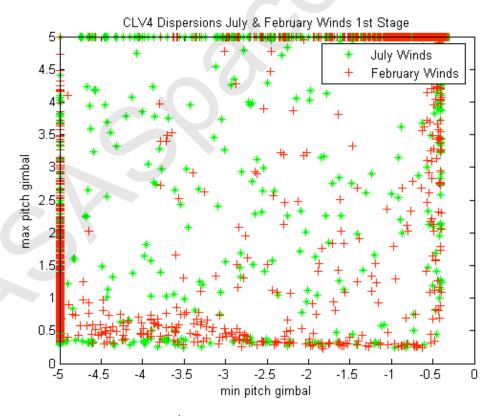


Figure 5.4.6-6 1<sup>st</sup> Stage Maximum vs. Minimum Pitch Gimbal Angle

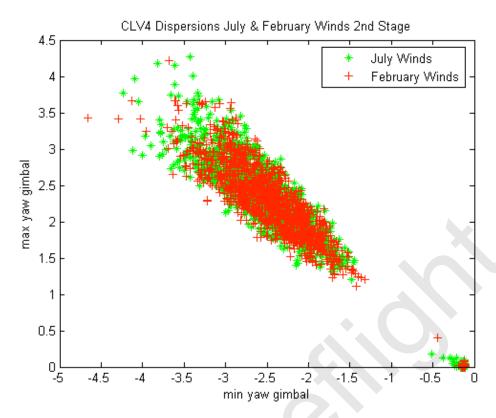


Figure 5.4.6-7 2<sup>nd</sup> Stage Maximum vs. Minimum Yaw Gimbal Angle

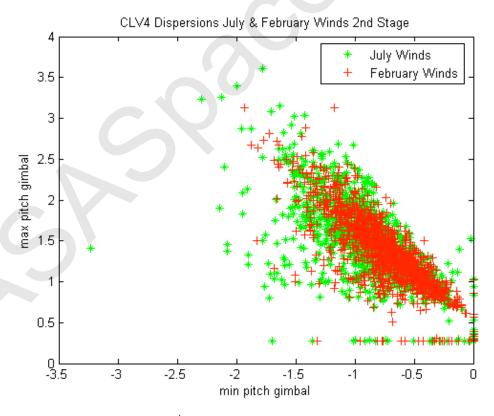


Figure 5.4.6-8 2<sup>nd</sup> Stage Maximum vs. Minimum Pitch Gimbal Angle

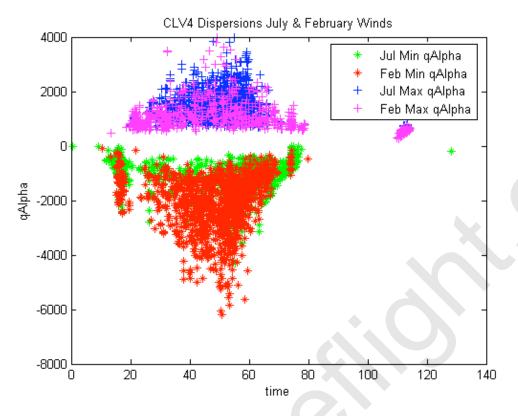


Figure 5.4.6-9 Min/Max qAlpha vs. Time

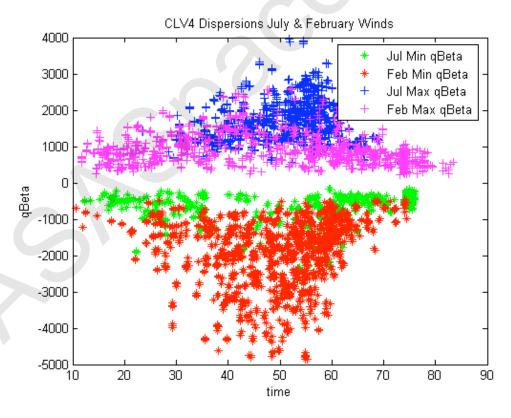


Figure 5.4.6-10 Min/Max qBeta vs. Time

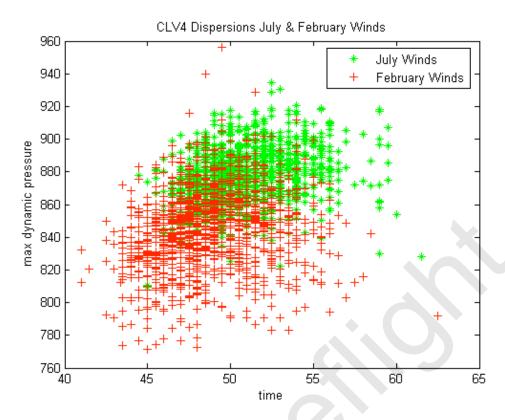


Figure 5.4.6-11 Maximum Dynamic Pressure vs. Time

	min alt_Ft	max alt_Ft	min latDeg	max latDeg	min lonDeg	max lonDeg	min phiNED	max phiNED
min	9.999946E-04	3.706477E+05	2.860841E+01	3.615845E+01	2.793959E+02	2.883665E+02	-1.799990E+02	9.000000E+01
max	9.999946E-04	3.733112E+05	2.860842E+01	3.628316E+01	2.793959E+02	2.885564E+02	-1.417145E+02	1.799999E+02
average	9.999946E-04	3.707462E+05	2.860842E+01	3.625318E+01	2.793959E+02	4.523762E+02	-1.573012E+02	1.067722E+02
standard deviation	1.195225E-01	8.547093E+01	2.742042E-07	1.167151E-02	2.843240E-12	1.781052E-02	1.308293E+01	3.491908E+01

Number of Monte Carlo runs:1330

	min thetaNED	max thetaNED	min psiNED	max psiNED	min alphaDeg	max alphaDeg	min betaDeg	max betaDeg
min	1.555082E+00	8.999992E+01	-1.799979E+02	1.236901E+02	-9.000000E+01	1.618303E+01	-28.14	4.40
max	6.387524E+00	8.999992E+01	-9.231488E+01	1.799969E+02	-9.000000E+01	2.641896E+01	-1.02	21.58
average	4.142941E+00	8.999992E+01	-1.159483E+02	1.323984E+02	-9.000000E+01	1.831900E+01	-5.27	5.57
standard deviation	8.468736E-01	1.265242E-12	2.883509E+01	1.911622E+01	0.000000E+00	4.898624E-01	3.91	2.71

	min phibkDeg	max phibkDeg	min gimPit.1	max gimPit.1	min gimYaw.1	max gimYaw.1	min gimPit.2	max gimPit.2
min	-1.799995E+02	1.791182E+02	-5.000000E+00	0.23	-1.016257E+00	2.82E-02	-1.93E+00	2.72E-01
max	-1.798858E+02	1.800000E+02	-3.078685E-01	5.00	-7.792233E-02	5.33E-01	0.00E+00	3.13E+00

average	-1.798885E+02	1.794388E+02	-2.794835E+00	2.72	-3.580286E-01	1.60E-01	-7.47E-01	1.37E+00
standard deviation	1.480934E-02	2.362531E-01	1.986186E+00	1.94	1.375611E-01	0.079728016	0.316830852	0.432844149

	min gimYaw.2	max gimYaw.2	min qAlpha	max qAlpha	min qBeta	max qBeta	min qBarPsf	max qBarPsf
min	-4.659403E+00	0.000000E+00	-6.18E+03	2.596275E+02	-4.85E+03	2.14E+02	2.48E-08	7.71E+02
max	-1.084705E-01	4.219659E+00	-7.86E+01	3.976134E+03	-4.71E+02	2.59E+03	2.67E-08	9.56E+02
average	-2.417464E+00	2.243103E+00	-2.05E+03	8.715388E+02	-1.92E+03	9.03E+02	2.57E-08	8.45E+02
standard deviation	5.822572E-01	5.877907E-01	996.3274529	4.668057E+02	922.2099832	415.0838217	4.04396E-10	24.04571426

	min altApo_Nm	max altApo_Nm	min altPer_Nm	max altPer_Nm
min	-2.630239E+00	1.586752E+02	-3.439328E+03	2.874248E+01
max	-2.630239E+00	1.621909E+02	-3.439318E+03	3.162995E+01
average	-2.630239E+00	1.602461E+02	-3.439320E+03	3.030327E+01
standard deviation	4.753542E-14	5.547929E-01	9.882399E-04	5.176679E-01

Table 5.4.6-2 February Dispersion Statistics

	min alt_Ft	max alt_Ft	min latDeg	max latDeg	min lonDeg	max lonDeg	min phiNED	max phiNED
min	9.999946E-04	3.706751E+05	2.860842E+01	3.620035E+01	2.793959E+02	2.884296E+02	-1.799998E+02	9.000000E+01
max	9.999946E-04	3.707796E+05	2.860842E+01	3.628353E+01	2.793959E+02	2.885547E+02	-1.431560E+02	1.799959E+02
average	9.999946E-04	3.707296E+05	2.860842E+01	3.624170E+01	2.793959E+02	4.525860E+02	-1.511131E+02	9.188484E+01
standard deviation	1.195224E-01	9.554702E+00	1.883643E-13	6.534145E-03	2.843234E-12	1.001862E-02	7.290201E+00	1.277473E+01

Number of Monte Carlo runs:1338

	min thetaNED	max thetaNED	min psiNED	max psiNED	min alphaDeg	max alphaDeg	min betaDeg	max betaDeg
min	1.411584E+00	8.999992E+01	-1.798124E+02	1.236901E+02	-9.000000E+01	1.716163E+01	-11.31	4.40
max	5.423582E+00	8.999992E+01	-9.375655E+01	1.796402E+02	-9.000000E+01	4.451332E+01	-0.91	15.92
average	4.041680E+00	8.999992E+01	-1.024980E+02	1.243790E+02	-9.000000E+01	1.858603E+01	-3.70	5.15
standard deviation	8.860986E-01	1.265239E-12	1.128356E+01	5.571776E+00	0.000000E+00	1.507782E+00	1.48	1.82

	min phibkDeg	max phibkDeg	min gimPit.1	max gimPit.1	min gimYaw.1	max gimYaw.1	min gimPit.2	max gimPit.2
min	-1.799998E+02	1.791325E+02	-5.000000E+00	0.24	-5.000000E+00	1.79E-01	-3.23E+00	2.72E-01
max	-1.798858E+02	1.799992E+02	-3.148120E-01	5.00	-6.134085E-02	9.90E-01	0.00E+00	3.61E+00

average	-1.798908E+02	1.794562E+02	-2.640334E+00	3.02	-1.873086E-01	5.40E-01	-8.45E-01	1.45E+00
standard deviation	2.079808E-02	2.467565E-01	2.054969E+00	2.04	2.545102E-01	0.137913821	0.364836552	0.470119803

	min gimYaw.2	max gimYaw.2	min qAlpha	max qAlpha	min qBeta	max qBeta	min qBarPsf	max qBarPsf
min	-4.228637E+00	0.000000E+00	-4.28E+03	5.266200E+02	-2.34E+03	5.43E+02	2.38E-08	8.09E+02
max	-1.117027E-01	4.270177E+00	-4.22E-04	3.979213E+03	-1.66E+02	3.96E+03	2.54E-08	9.34E+02
average	-2.482065E+00	2.307534E+00	-8.49E+02	1.428006E+03	-5.50E+02	1.70E+03	2.45E-08	8.78E+02
standard deviation	6.000733E-01	6.090271E-01	457.860775	5.529184E+02	292.3334807	541.1773919	3.35503E-10	15.85259464

	min altApo_Nm	max altApo_Nm	min altPer_Nm	max altPer_Nm
min	-2.630239E+00	1.586892E+02	-3.439324E+03	2.868371E+01
max	-2.630239E+00	1.622650E+02	-3.439318E+03	3.201225E+01
average	-2.630239E+00	1.603085E+02	-3.439320E+03	3.036934E+01
standard deviation	4.753533E-14	5.888155E-01	7.506620E-04	5.592547E-01

Table 5.4.6-3 July Dispersion Statistics

# 5.4.7 Stage Separation

This section includes three parts. The first is a description of the baseline separation plan used for this VIPA cycle. This section describes some of the rationale for choosing this separation plan. Next is a list of a few alternative ways to do separation that we did not use in this cycle. Finally, separation simulation results are displayed.

#### Stage Separation Plan

- a) When axial acceleration drops below a level corresponding to a SRB thrust of 15000 lbf TBR, command firing of two BSMs (immediately). We chose acceleration as the deciding factor because it directly ties to SRB thrust and we remove any chance that we might command separation while thrust is still too high. We can change to Pc later if that makes sense after more detailed analysis. The BSMs are on the aft skirt. Our separation condition holds the SRB longer than it is held for Shuttle, and longer than optimal performance would give, so that the thrust level required for separation is not too large. For example, at a typical separation time (10 sec earlier in our SRB model), 0.56 g would be required for 0.1g separation. This would require 6 BSMs for the first second, followed by more in the 2<sup>nd</sup> second and more in the 3<sup>rd</sup>. At separation, the SRB would be accelerating towards the stack at 0.46 g if we did not fire the BSMs. Delaying separation reduces this substantially.
- b) Timed for when BSM thrust has built up (milliseconds later): Command the shape charge ring at the base of the interstage to separate the SRB.
- c) BSMs fire for approximately 0.805 seconds to pull the SRBs back from the stack. They provide about 20000 lbf thrust each.
- d) Use RCS jets on second stage to provide 0.01 g acceleration for the duration of the stack coast, starting immediately after SRB separation, to provide for settling of propellants.
- e) Starting 0.5 seconds (TBR) after separation, fire the roll control RCS jets on the SRB (using a timer set when the command to the BSMs is sent) to rotate the SRB in the pitch plane. The purpose of this is two-fold: It removes the parachute area from the line of fire of the 2<sup>nd</sup> stage engine when it ignites, and it ensures that any residual thrust from the SRB will not move it towards the stack. An option here is to also command the SRB actuator to hard over in the direction that will assist the roll. We won't consider this right now since the magnitude of its help may have significant dispersions and is likely to be somewhat less than what we get from the RCS. Command the RCS jets to stop the rotation at about a pitch angle of 90 deg. Note that this stopping of the roll probably needs to be modified (in the next cycle), since tumbling is necessary to initiate the recovery process for the Shuttle SRBs.
- f) Start the 2<sup>nd</sup> stage ignition process timed so that significant thrust will not occur until the parachutes are sufficiently out of the line of fire. For now, we are assuming that the SSME is at full thrust 10 seconds after SRB separation.
- g) At 0.1 sec (TBR) after SSME at full thrust, separate the interstage up to the thrust cone with another shape charge. Use thrusting joints to separate the interstage in 2 halves (TBR). A trade is to separate the interstage prior to SSME startup.

Some alternative separation schemes that we are not using for the baseline:

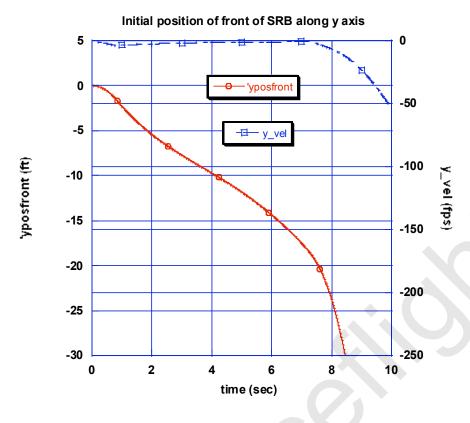
- a) Fire in the hole. This is simpler in certain respects, but raises issues of parachute survival and engine start shock wave reflection. Parachute survival becomes a non-issue if the SRB is not recovered.
- b) Using the LES to provide separation. We don't know whether or not we still need the LES for later aborts at this early design stage. Also, this makes successful LES solid burns part of the nominal mission success. It means we need to work the structural

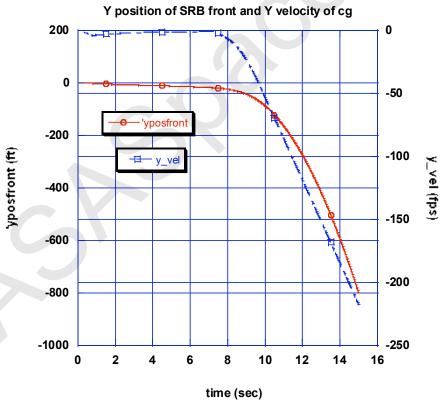
- impacts of tension throughout the second stage, service module, and crew module caused by the LES pulling the stack off the SRB. It would yield increased payload performance, other things being equal.
- c) Mechanism on the SRB to kill the thrust and remove the performance impact of holding the SRB longer than performance would dictate and to reduce/remove the need for the BSMs. We thought this would cause reliability/safety issues for the vehicle. However, we will probably need a method to kill the SRB thrust for aborts anyway.

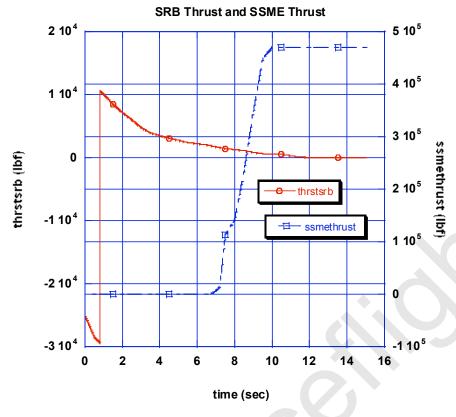
#### Separation Simulation Results

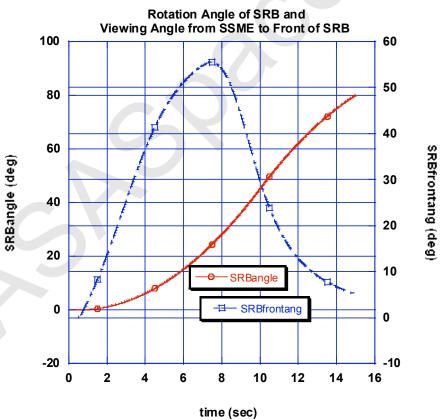
#### Assumptions and other information:

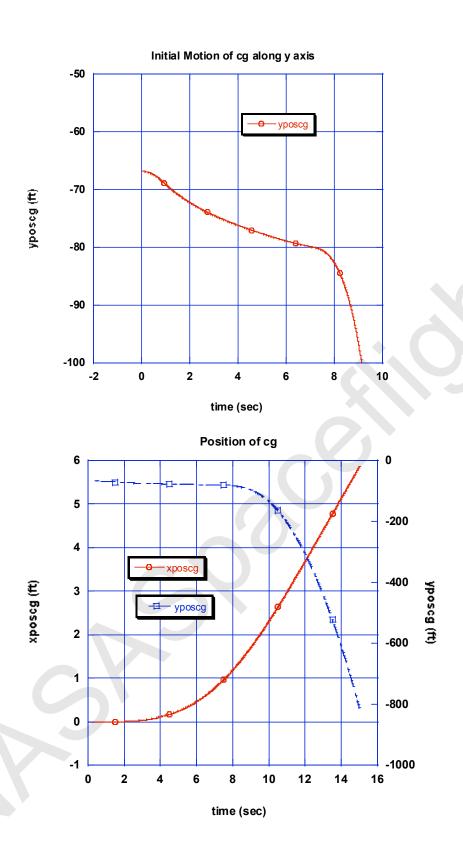
- a) Relative motion due to the various separation thrusting forces is numerically integrated. The effects of gravity are assumed to be equal on the two components.
- b) The origin is at the back end of the 2nd stage; measurements are relative to the back end of the 2nd stage
- c) 2nd stage is accelerating at 0.01 g forward
- d) 2 BSMs provide initial SRB separation; SRB is also still in the thrust tailoff phase
- e) Roll control thrusters used to provide pitch to rotate SRB away from 2nd stage nozzle
- f) Starting position of SRB cg is directly behind end of 2nd stage by 802 in (66.8 ft)
- g) y is longitudinal position behind the 2nd stage; x is position in the pitch plane
- h) SRB angle is the rotation angle of the SRB with respect to the y axis











The temperature and force impacts on the SRB, using this particular separation scheme, are in the thermal analysis section of the report.

# 5.4.8 CLV Linear Stability Analysis for First Stage Flight

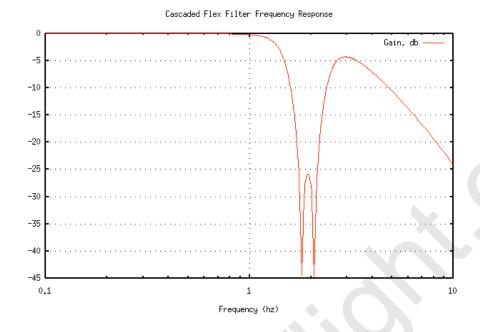
A preliminary linear stability analysis has been performed for the CLV Launch Vehicle for first stage flight. This analysis included upper stage propellant slosh dynamics and, for liftoff and SRB burnout conditions, flexible body dynamics. Rather than allocating phase and gain margins, reasonable estimates were made for effects that are not yet quantified in the vehicle design. The object of this analysis was to assess stability margins, provide modified control system gains and slosh damping needed to achieve stability margins.

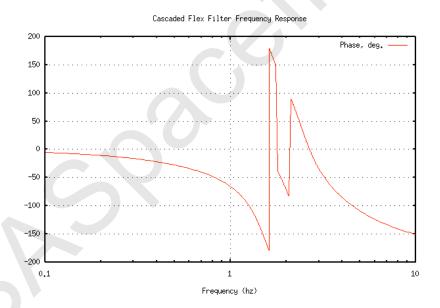
Not included in the stability analysis was the 15 Hz acoustic oscillation in the SRB. This mode can cause Y and Z body axis thrust oscillations for nonzero gimbal angles, but it is assumed that it is high enough in frequency that its effects will be filtered by the flex filters that reside in the ascent flight control system.

#### **Modeling and Data**

The following describes the models used in the stability analysis:

- Time delay model. A 30 ms delay between the controller outputs and the inputs to the
  control actuators was simulated. This is an estimate of delays due to computational time
  in the on-board computer and transport lags across data busses and other avionics.
- Engine TVC actuators. A third order transfer function model of the SRB actuators was used.
- GNC Sensors. GNC sensors were assumed to be "perfect" with no sensor dynamics on the variables used by the Ascent Flight Control System (AFCS).
- FCS model. A preliminary, Proportional-Integral-Derivative (PID) AFCS model, developed for the Maveric 6DOF simulation, was used.
- Flex filters. A digital notch filter, cascaded with a low pass filter to attenuate high frequency effects of flex body dynamics at the sensor was used. This filter was designed to filter flex modes above 1.5 Hz. Its frequency response is shown below.





- Flex model. First bending modes were included for two operating points; liftoff and SRB burnout.
- Propellant slosh. Slosh was modeled as a spring-mass-damper system acting in lateral directions only, that is along Y and Z body axes, not the X axis.

The following data used in the models were obtained from the Maveric 6DOF simulation:

- Mass properties
- Propulsion (thrust table and engine position)
- Aerodynamics
- Operating conditions such as relative velocity, aerodynamic pressure, Mach, etc.
- AFCS gains

The following were obtained from the Dynamics, Loads and Strength Branch, EV31:

- Flexible body modes (eigenvectors, natural frequencies), node locations for sensors and engine gimbal point
- Slosh data: slosh mass, locations, frequencies and smooth wall damping

## **Analysis Results**

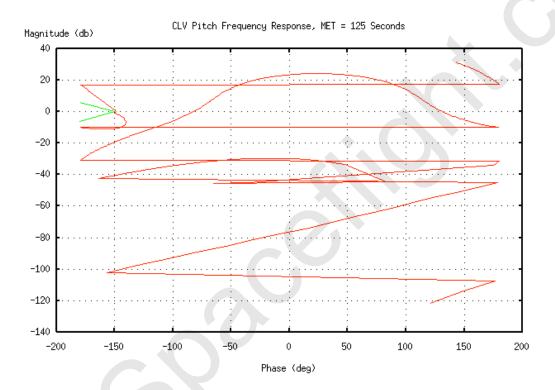
Stability margins from the analysis are given in the table below. Maximum aerodynamic pressure corresponds to 48.8 seconds Mission Elapsed Time (MET).

		High					
	Low Freq.	Freq.		Low Freq.	High		
	Pitch GM	Pitch GM		Yaw GM	Freq. Yaw		
MET	(dB)	(dB)	Pitch PM	(dB)	GM (dB)	Yaw PM	
1	21	11	31	15	5	33	Includes Flex
15.3	15	9	35	21	5.8	35	
38.1	9	10	35	12	5.6	30	
48.8	10	7	35	12	5.5	30	
105	၁၀	6	20	റ	10	ე∩	Indudos Elev

A preliminary requirement of 6 dB low and high frequency gain margin was assumed, and a 30 degree phase margin was also assumed. In the liftoff case, yaw high frequency gain margin was not met, although the requirement could probably be achieved with further gain or flex filter coefficient adjustments. The following table shows slosh damping values used in the analysis.

	LOXZ	LOXY	LH2 Z	LH2 Y	
	Axis	Axis	Axis	Axis	
,	Mode	Mode	Mode	Mode	
	Damping	Damping	Damping	Damping	
MET	(%)	(%)	(%)	(%)	
					Includes Flex
1	1	1	0.01	0.01	
15.3	1	1	0.01	0.01	
38.1	1	1	0.01	0.01	
48.8	1	1	0.01	0.01	
405	Λ	1	0.04	0.04	Indudes Flav

A damping of 4% was required at 125 seconds MET to stabilize slosh modes in the LOX tank so the required margins could be attained. The Nichols chart used to visualize the yaw axis frequency response and calculate margins is given below for this operating point. The slosh mode can be seen as the small lobe just touching the stability margin boundary represented by the half-triangle.



# 5.4.9 SRB Slag Impacts

Slag accumulates in the rear of the SRB during flight. It sloshes and sometimes sloshes overboard into the SRB plume. This sloshing causes slosh forces internally and also changes the thrust slightly when it comes out. There is a slosh frequency associated with this motion. The slag can also come out as the SRB burns out at the end of flight. A reference is "Shuttle STS-54 Pressure Perturbation Investigation" by Charles Martin/MSFC, AIAA 95-2882. Charles Martin is a good reference for additional information on the slag.

We did not analyze the slosh effects due to slag during this study cycle. Since the slosh mass is close to the gimbal point, this tends to mitigate the bad effects of the sloshing. We also did not analyze the effects of the thrust forces at the slosh frequency. These delta forces are fairly small, but should be analyzed. They may be small compared to the effects of the SRB acoustic mode thrust oscillation operating at a nonzero gimbal angle. We did not analyze these effects during this cycle because they seemed at first glance to be smaller than the other considerations we were examining for stability, such as the upper stage slosh modes and the vehicle vibration modes.

We do not know the precise effects of the slosh movement when the SRB is pushed backward at separation as opposed to earlier when it is accelerating. This needs to be studied to ensure there is no bad impact on the separation success.

# 5.4.10 CEV Separation

CEV separation was not simulated or analyzed during this early conceptual design study. Because the upper stage and CEV will be in orbit when CEV separation occurs, and because the SSME will not be thrusting at the time, any torques or forces on the stack should be very small. Thus a simple separation procedure with springs should be sufficient to result in a clean separation, without adding separation motors.

The stack should be controlled to a stable attitude for the separation. It may be possible to do this with the RCS engines that are already on the upper stage. Should more control capability be necessary, it makes sense to do this with the CEV control system rather than to add more RCS engines to the upper stage just for stability for separation.

#### 5.4.11 Liftoff Clearance

Liftoff clearance was not worked during this early conceptual design study. Analysis of liftoff clearance includes simulation of the clearance of the hold-down posts, as well as simulation of the clearance of the launch tower. This analysis must include worst case winds and dispersions to ensure that positive clearance always results. The design of the pad structures was not sufficiently mature to conduct the study at this time, and the team felt that liftoff clearance is not likely to be a problem since the vehicle has a high acceleration level immediately after liftoff.

#### 5.5 AEROTHERMODYNAMICS

CLV4 aerothermodynamic environments have been defined for two flow phenomena: aerodynamic heating and plume induced heating. All aerothermodynamic environments were generated using nominal trajectories. Trajectory dispersions caused by atmospheric, wind, systems, and other variations have not been taken into account at this time. Development of thermal design specific trajectories will take place at a later stage of vehicle development.

# 5.5.1 Aerodynamic Heating Environments

Aerodynamic heating is induced on external surfaces of the CLV4 launch vehicle during supersonic and hypersonic flight conditions during first and second stage ascent and re-entry.

Ascent aerodynamic heating environments were not defined at this time for flight after first stage separation on ascent, or for second stage re-entry. From a preliminary design perspective, the peak aerodynamic heat rate is encountered before first stage separation. Due to the relatively low overall magnitude of the first stage flight ascent aerodynamic heating, the second stage flight heat rate contribution is minimal to the overall thermal design. Re-entry aerodynamic heating is the dominant aerodynamic heating concern for the first stage Solid Rocket Booster. Second stage re-entry aerodynamic heating will be required to support future re-entry break up analysis. Thus, second stage re-entry aerodynamic heating is not required at this time. In addition, localized aerodynamic heating amplification due to protuberances should also be assessed. These environments will be generated at a later time as the vehicle configuration matures.

## 5.5.1.1 First Stage Ascent Aerodynamic Heating

Launch vehicle ascent aerodynamic heating peaks during the first stage of flight. For typical large launch vehicles, peak ascent aerodynamic heating will occur above Mach 2.0, and at altitudes below 150 kft. Virtually all launch vehicles begin lift-off with turbulent flow heating and then transition to laminar flow heating later in flight as the ambient flow density rapidly decreases. Turbulent flow is a result of the rather long overall surface running length, surface roughness, and assorted protuberances encountered on most large launch vehicles.

Preliminary design level first stage aerodynamic heating environments were defined for the CLV4 launch vehicle configuration at several axial stations down the vehicle length. The local heat transfer coefficient for each axial body point was computed using the local running length, boundary layer edge properties, and the Schultz-Grunow turbulent flat plate heating correlation. The appropriate Mangler transformation was applied to the flat plate turbulent heating correlation for each body point location. The local boundary layer edge pressure was approximated by modified Newtonian impact theory, or shock expansion theory as appropriate for the local geometry. The local flow properties were evaluated using Eckert's reference temperature concept. The enclosed environments were generated for the CLV4 ISS reference trajectory rev 1 dated June 6, 2005.

Solid Rocket Booster and second stage cold wall ( $T_{wall}$  =540 F) acreage heating rates were below 0.8 Btu/ft2-sec for all body points up to SRB separation at t=128 seconds. Ascent aerodynamic heating peaks around t=80 to 87 seconds at Mach numbers of approximately 2.85 – 3.25 depending on body point location. The highest levels of heating were encountered on the CEV nose and SRB aft skirt due to the higher local surface pressures for these areas. Figure 5.5.1.1.1 below defines the CLV4 body point locations and the peak aerodynamic heating encountered at

each location. Figures 5.5.1.1.2-3 define the aerodynamic heating as a function of time for each body point location.

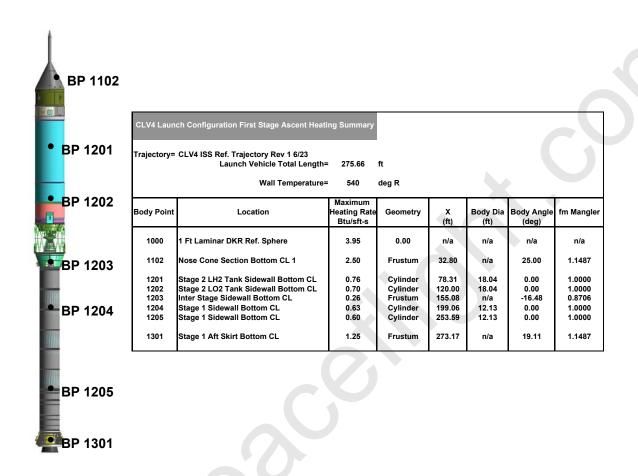


Figure 5.5.1.1.1 CLV4 Body Point Locations and Peak First Stage Aerodynamic Heat Rate vs. Time

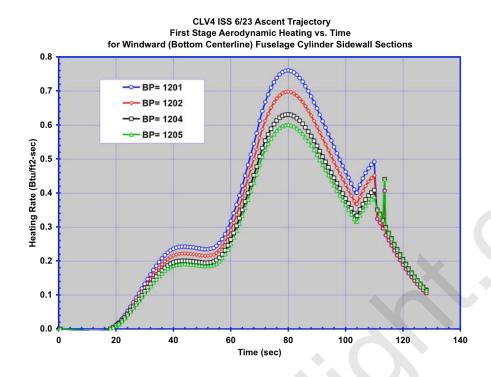


Figure 5.5.1.1.2 First Stage Aerodynamic Heat Rate vs. Time for CLV4 Cylindrical Sidewall Sections

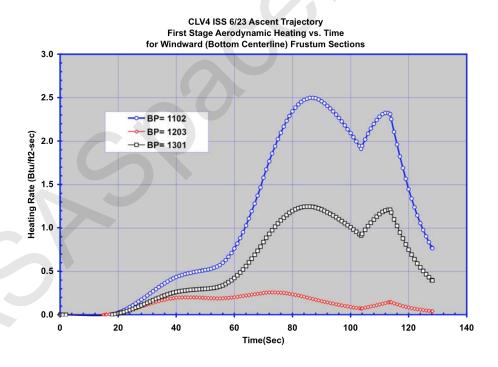


Figure 5.5.1.1.3 First Stage Aerodynamic Heat Rate vs. Time for CLV4 Frustum Sections

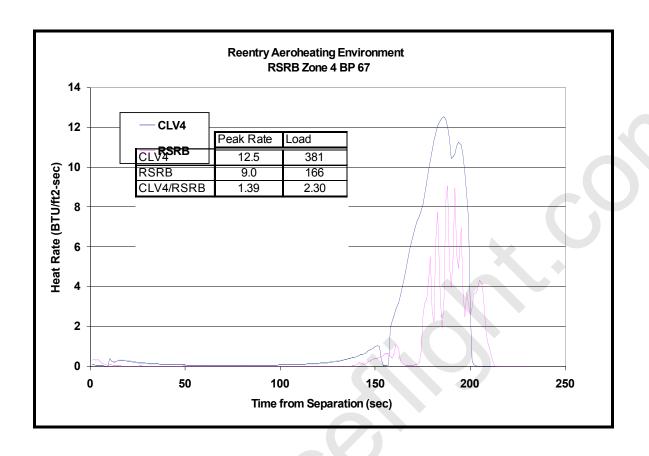
## 5.5.1.2 Re-entry Aerodynamic Heating Environments

Re-entry aeroheating environments for the CLV4 first stage vehicle were generated using the STATE computer code. The State program is a heating analysis code that establishes heat transfer coefficients, heating rates, and heating loads for the Space Transportation System Solid Rocket Boosters (SRB) during reentry from space. The STATE program is uniquely designed to model the tumbling and rolling trajectory of the RSRB that produces nearly every vehicle-flow field orientation possible. A complete analysis can be performed on some 200+ Monte Carlo trajectories to establish 0, 50, 95, and 100 percent maximum heat load trajectories; or a single reentry trajectory.

The preliminary CLV4 first stage re-entry aerodynamic heating environments were generated using a nominal 3DOF re-entry trajectory. This trajectory provides the altitude and velocity of the vehicle but does not accurately model the tumbling and rolling which results in a distribution of the peak heating (stagnation line) environments over the circumference of the vehicle. The result of this analysis is that the CLV4 re-entry environments generated by STATE are overly conservative. A comparison of the "Nominal" CLV4 environments with the "95% Design" RSRB environments for the RSRB Zone 4 BP 67 show a factor of 1.39 in the peak heating rate and a factor of 2.30 in the total heat load. A similar comparison in the total heat loads using the Swept Cylinder Heating Indicator Model at the peak heating circumferential location produces a factor of 2.0 in the total heat load. Preliminary Re-entry Aerodynamic Heating Environments have been generated for the CLV4 vehicle by applying the 2.0 factor to the existing RSRB environments for each body point of interest. This factor is conservative assuming that the RSRB aerodynamics accurately model the CLV4 re-entry vehicle.

The RSRB environments are generated using a Monte Carlo set of 200 trajectories which incorporate critical aerodynamic parameters which define the vehicle flight. The parameters differ greatly from the proposed CLV4 vehicle. For example, the RSRB has a significant roll which is produced by a variety of factors not existing in the CLV4 vehicle. The RSRM has forward and aft separation motors which produce roll. The RSRB separation motors and attach structure effect the CG and aerodynamic characteristics of the vehicle also effecting roll.

Because of differences in the aerodynamics and CG of the CLV4 and RSRB the trim angle of the CLV4 is likely to be different than the RSRB trim angle of 164 degrees. The internal aft skirt thermal design has small margins. A trim angle approaching 180 degrees will result in negative margins in the current TPS design. Aerodynamic characterization of the re-entry vehicle is a critical step in fully assessing the Re-entry vehicle TPS design.



## 5.5.2 Plume Induced Environments

During powered flight of all launch vehicles or missiles, the exhaust plumes and plume interaction flow fields combine to produce an environment affecting the aft end of the vehicle commonly referred to as the "base heating" environment. The base heating environment has three components: radiation from the hot gases to base region surfaces, convection resulting from air or recirculating plume gases flowing over the base surfaces, and direct impingement of plume gases on base or other structures.

Second stage CLV4 plume radiation and convection environments were not generated for this analytical cycle. Second stage radiation and convection will both be relatively low due to use of  $LO_2/LH_2$  propellants, which produce little radiation at high altitudes, and the low convective levels at these altitudes associated with a single engine configuration. These analyses will be conducted in the future.

# 5.5.2.1 First Stage Plume Radiation and Convection

CLV4 plume radiation and convection is induced by the single four-segment RSRM plume. The RSRM uses a composite propellant with 16% powdered aluminum. Aluminum oxide formed during combustion of the RSRM propellant is the dominant contributor to plume radiation.

CLV4 plume convection will be largely limited to the immediate base and nozzle area, and will be caused by plume / freestream flow field interaction. This interaction will cause a recirculation

zone to form on the aft portion of the nozzle exterior and to a lesser degree the thermal curtain and aft edge of the aft skirt.

The current analysis uses existing Space Shuttle SRB base environments at  $\theta_B$  = 225°. CLV4 and Space Shuttle ascent trajectories are similar enough to allow use of the shuttle heating time histories at this preliminary stage of analysis. CLV4 plume radiation and convection from the single RSRM plume will be similar to the Space Shuttle environments at the  $\theta_B$  = 225° location because this area does not experience heating from the three SSME or opposite RSRM plumes. These are the lowest environments on the current Shuttle SRB base. Future analyses will utilize actual CLV4 design trajectories, plumes, and base structures.

Since the existing SRB base area structure has been used as the basis for CLV base structure, and the CLV environments will be lower than existing Shuttle environments, base structure should easily accommodate the CLV plume induced radiative and convective heating environments.

## 5.5.2.2 First Stage Separation Plume Impingement

Once the CLV4 first stage has separated from the second stage, the single second stage SSME will start up and its plume will impinge on the first stage. Direct plume impingement environments were calculated using the Plume Impingement Program (PLIMP). Plume induced forces, moments, and heating rate environments were calculated on the second stage hardware. SSME plumes at 50% and 100% power levels were generated with a combination of the CEC, RAMP2 and SPF3 programs. Four separation conditions were analyzed using a separation simulation supplied by EV40.

A peak heating rate of approximately 52 BTU/ft²\*sec was found on the first stage forward skirt side at t=10 seconds. A centerline heating rate distribution for this case is shown in Figure 5.X. While the magnitude of these rates is high, they should be accommodated by the design due to the relatively short time they occur.

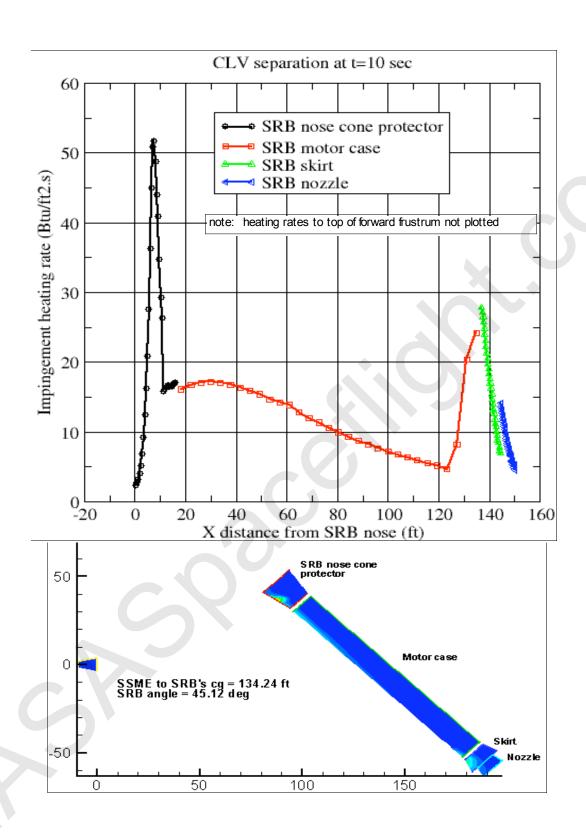


Figure 5.X First Stage Separation Centerline Heating Rate at T=10 seconds

# 5.6 CREW LAUNCH VEHICLE (CLV) THERMAL PROTECTION SYSTEM

This report describes a preliminary assessment of the CLV acreage TPS. The CLV design is a two-stage rocket utilizing a LOX-LH2 propellant combination and single Space Shuttle Main Engine (SSME) for Stage 2 and a 4-segment Solid Rocket Booster (SRB) for Stage 1. The vehicle layout is shown in Figure 5.6-1 below.



Figure 5.6-1. CLV Layout

Much of the specific TPS design challenge is related to the overall vehicle configuration, complexity, choice of propulsion systems, operational scenarios, mission, and the resulting effect of these variables on the aerothermal environments. Selection of the optimum TPS for a launch vehicle is a complex and challenging task that requires consideration of not only weight, but also operability, maintenance, durability, cost, and integration with the vehicle structure (including cryogenic propellant tanks).

TPS is used to protect the launch vehicle from the extreme temperatures experienced during launch and entry environments. TPS is not a single material, but is a system of materials working together in the most efficient manner. Factors that influence TPS design are: natural and induced environments, structural temperature limits, subsystem and component temperature requirements, and propulsion system requirements. The objective of the TPS is to minimize the vehicle's weight using materials whose thermal and mechanical integrity are maintained throughout the mission cycle. TPS is a key element in the vehicle design and critical to its ultimate success.

Based on these factors the current CLV TPS design will utilize, where possible, existing External Tank (ET) and SRB flight qualified TPS materials. In order to validate these TPS designs for this vehicle, a series of both generic and (sometimes) specific thermal models are/will be developed for locations of interest. These thermal models are utilized to perform architecture thermal assessments once the vehicle external and internal thermal environments are defined. The fidelity of these models will increase as the vehicle design progresses. The objective of these models is to verify the TPS and cryogenic tank insulation thermal designs and ensure vehicle thermal requirements are met.

Thermal protection materials currently used on the SRB [1] include:

- Marshall Convergent Coating (MCC-1)
  - This is certified for use on the forward assemblies, system tunnel covers and aft skirt acreage areas.

 Cork, silica-filled EPDM, phenolic glass, SLA-220, K5NA and BTA are used in protuberance areas (especially those exposed to high heating and high shear environments).

The ET thermal protection system [2] is composed of a Spray-on-Foam Insulator (SOFI) that serves to insulate the tank before and during launch. The ET also uses ablators on areas that are subjected to extreme heat, such as the ogive, or the top of the tank, and on the lines that feed the propellants to the Shuttle's Main Engines. SOFI is also considered as an ablator for heat flux environments less than 10 BTU/ft²-sec.

The following sections outline the heating environments used in this study and the acreage TPS sizing results.

#### **Heating Environments**

The CLV environments are a combination of ascent heating, plume impingement during separation and SRB re-entry heating. Ascent heating environments [3] were defined for 7 generic Body Points (BP) along the vehicle. The BP locations are defined in Figure 5.6-2.

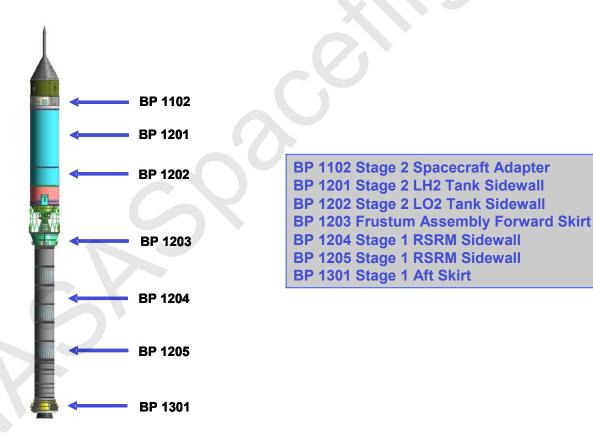


Figure 5.6-2: CLV Body Point Definition

Plume impingement environments were supplied for locations (as a function of time) along the entire length of the SRB [4]. All locations on the SRB are influenced by the SSME plume during separation as shown in Figure 5.6-3.

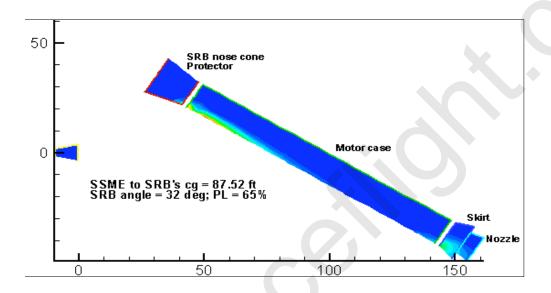


Figure 5.6-3: SRB Plume Impingement at 8.5 seconds ignition after SSME ignition

Since the ascent SRB environments were for generic locations only, maximum impingement heating corresponding to these locations (BP 1203, 1204, 1205 and 1301) was used in the analysis. For example, impingement heating for 31 axial locations was defined for the Frustum Assembly (BP 1203). The maximum axial location value at each time step was selected.

Re-entry Aerodynamic Heating Environments were generated for the CLV vehicle by applying a factor of 2.0 to the existing SRB environments for each body point of interest [5]. This factor is conservative, assuming that the current SRB aerodynamics accurately model the CLV re-entry vehicle.

Cold wall heating rates (used for comparative purposes only) for each BP are listed in the following section describing the thermal models and analysis results.

#### **TPS Analysis**

SINDA/ABL [6] was used to calculate the TPS design thickness. ABL, an in-house developed FORTRAN subroutine, calculates the in-depth thermal response of TPS materials. It provides a convenient and efficient way for the user to change values like TPS material and thickness.

The temperature dependent thermal properties such as thermal conductivity, specific heat and ablation recession logic were input as SINDA material arrays, which in turn, were utilized by ABL. The ablation rate has been determined in Hot Gas Facility tests. This is 95% recession data and as such is considered conservative. Material properties were obtained from References 1 and 2. All aeroheating environments were entered into SINDA/ABL in the form of time dependent heat transfer coefficients and recovery temperatures. An adiabatic boundary condition was assumed on the vehicle inner surface for maximum structural temperatures. Standard practice used to determine TPS thickness for large areas (acreage) is to define the TPS thickness required at the most severe heating location and apply that thickness of TPS over the entire component. The local BP environment determines TPS thickness on smaller components. In reality each structure or component has several environment BP locations identified with it.

## Stage 1

As stated previously, aerothermal environments consisting of ascent, plume impingement and reentry heating were supplied for 4 BP locations (BP 1203, 1204, 1205 and 1301). Figure 5.6-4 compares the cold wall heating for these locations. Based on the current environments plume impingement and re-entry heating are the most dominant and will drive the TPS design.

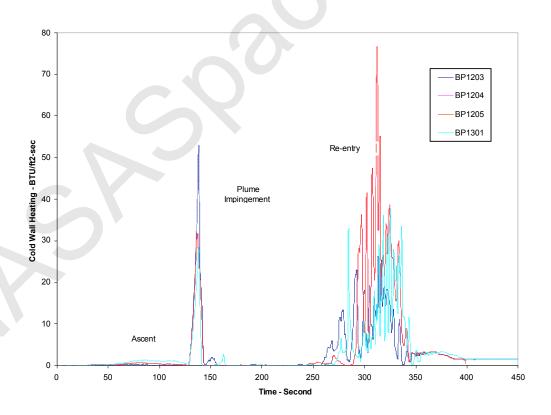


Figure 5.6-4. Stage 1 Cold Wall Heating Rates

#### BP 1203

BP 1203 is an acreage BP on the Frustum Assembly Forward Skirt. AL-2219 is the structural material and an effective thickness of 0.176 was used for analysis purposes. MCC-1 TPS was chosen for this location. Analysis results show that a thickness of 0.5 inches is required to maintain the substrate below its material temperature limit of 300°F. The ablation temperature of MCC-1 is 540°F. Analysis results are shown in Figure 5.6-5. It is clearly evident that it is during the plume impingement and re-entry phases that the MCC-1 ablates (surface temperature reaches the ablation temperature of 540°F). The rate of ablation is shown in Figure 5.6-6.

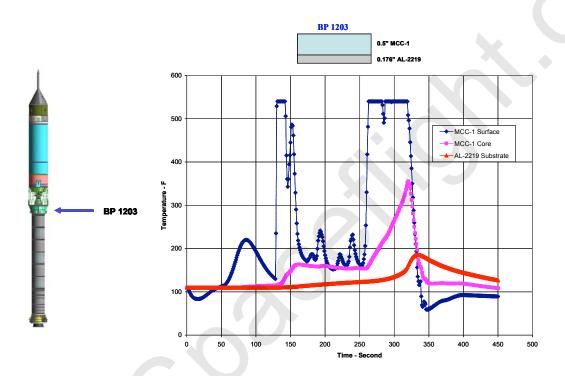


Figure 5.6-5. BP 1203 Temperature Profiles

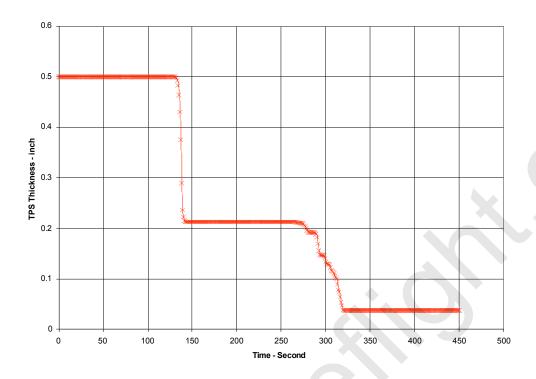


Figure 5.6-6. TPS Recession Rate

MCC-1 starts ablating during the separation and re-entry portions of the mission profile.

#### BP 1204 and BP 1205

BP 1204 and BP 1205 are located on the Reusable Solid Rocket Motor (RSRM) sidewall. D6AC steel is the structural material. Current RSRM case acreage steel thickness is 0.5 inches and requires no TPS. The CLV configuration analysis results for these locations are plotted in Figure 5.6-7. No TPS is required to maintain the D6AC steel below its structural temperature limit of 500°F.

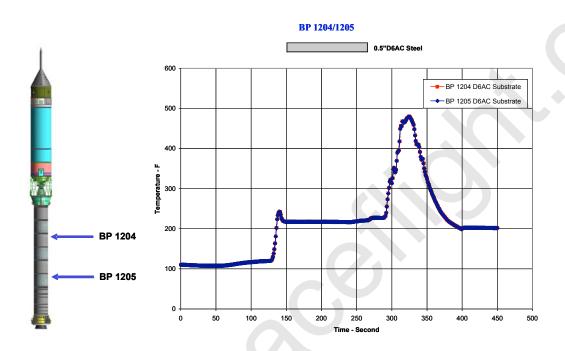


Figure 5.6-7. BP 1204 and BP 1205 Temperature Profiles

#### BP 1301

BP 1301 is an acreage BP on the Aft Skirt. AL-2219 is the structural material and a thickess of 0.5 inches was used in the analysis. A thickness of 0.375 inches of MCC-1 was required to maintain the substrate below 300°F. The resultant temperature profiles are shown in Figure 5.6-8.

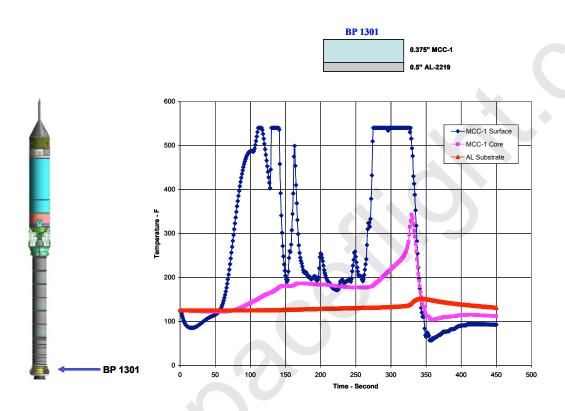


Figure 5.6-8. BP 1301 Temperature Profiles

The rate of ablation is shown in Figure 5.6-9. Again ablation occurs during separation and reentry.

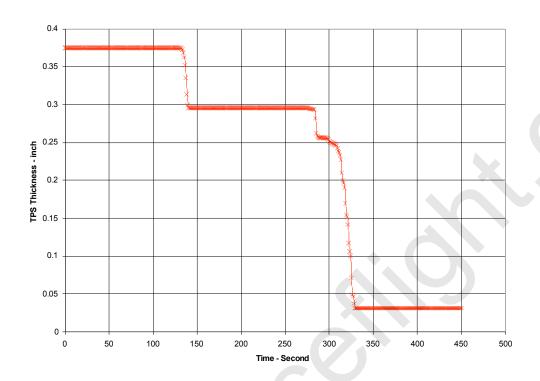


Figure 5.6-9. TPS Recession Rate

## Stage 1 Summary

Table 5.6-1 summarizes the 4 acreage locations analyzed.

Major Assembly	Component	TPS Material	Material Density (lbm/ft³)	TPS Thickness (inch)
Forward Skirt	Acreage	MCC-1	30.94	0.5
RSRM Case	Acreage	ı	-	-
Aft Skirt	Acreage	MCC-1	30.94	0.375

Table 5.6-1. CLV Stage 1 Acreage TPS Summary

SSME plume impingement heating after separation and re-entry heating are the dominant TPS drivers. Stage 1 ascent heating is minimal in comparison. The new Frustum Assembly design

will require further evaluation. The parachutes are housed in this section. They will need to be evaluated to ensure there is no temperature requirement violation. This analysis did not consider protuberances over the external surfaces. These will locally increase the heating to the exterior skin and could cause areas where thicker (than on the current SRB) TPS is needed. Protuberances could also disrupt flow near the aft end of the second stage (interstage region). If the external flow separates and reattaches on the open interstage, it could cause areas of localized heating that might compromise the structure. The Cable Systems Tunnel runs axially along the current SRB. MCC-1 and cork are currently used on the SRB to keep it within its temperature limit. This will also need to be assessed once environments are available.

The most significant TPS concern may be the need to protect the Thrust Vector Control (TVC) system components during re-entry (assuming the environments are more severe than current SRB re-entry environments). The aft skirt thermal curtain closes out the interior aft skirt against the heating environments experienced during flight. This flexible thermal curtain consists of several layers of blanket insulation that allows the nozzle system to gimbal while protecting the TVC system from plume radiation and recirculating hot gases and peak re-entry environments. The curtain is torn away by aerodynamic shear during re-entry exposing the TVC components. InstaFoam is used in the internal aft skirt, but is not sprayed near TVC components. Currently some of the stringers near the TVC violate the 300°F limit (for a short time). An increased reentry environment would cause longer and possibly more critical violations.

Base heating was not considered in this study. Two primary heating phenomena are associated with plumes: namely radiation heating where hot plume gases radiate at all altitudes and convection heating where hot plume gases are recirculated around the base of the launch vehicle. At this preliminary stage, based on data from the aerothermal community, the current SRB base region TPS design should be adequate. Once specific base region environments are available, this region will be evaluated. No Plume Induced Flow Separation (PIFS) environments were evaluated but it is not anticipated that such environments would cause the TPS design to change.

## Stage 2

Ascent aerothermal environments were provided for stage 2 [3]. Spray-on-Foam Insulation (SOFI) is used to protect this stage. NCFI 24-124, NCFI 24-57 and BX-265 are all SOFI's. Three (3) BP locations (BP 1102, 1201 and 1202) were assessed corresponding to acreage locations on the Spacecraft Adapter, LH2 tank and LOX tank respectively. Figure 5.6-10 compares the cold wall heating for these locations.

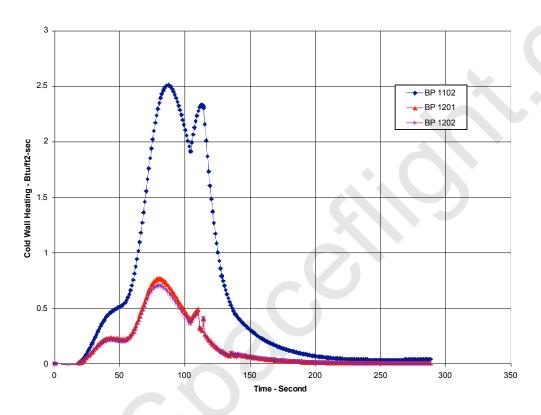


Figure 5.6-10. Stage 2 Cold Wall Heating Rates

## BP 1102

BP 1102 represents a location on the Spacecraft Adapter. The structural material is 0.08 inches of AL Lithium 2195. Analysis results show that 0.25 inches of NCFI 24-124 is adequate to maintain the substrate below its material temperature limit of 300°F. Recession of the foam occurs when the foam surface temperature reaches 650°F. The temperature profiles are shown in

Figure 5.6-11. The rate of material recession is plotted in Figure 5.6-12.

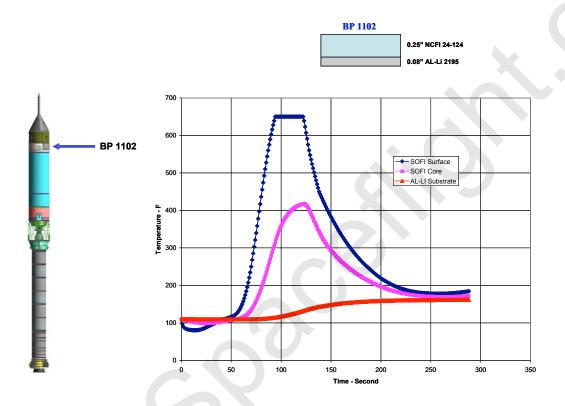


Figure 5.6-11. BP 1102 Temperature Profiles

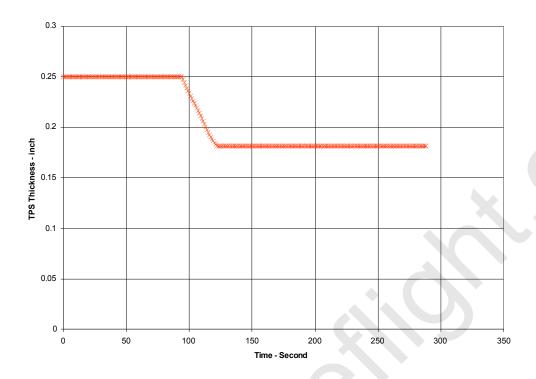


Figure 5.6-12. TPS Recession Rate

The Spacecraft Adapter houses the avionics and electronic instrumentation. The NCFI 24-124 was evaluated using an adiabatic boundary condition on the vehicle's interior surface. Any effect from the interior avionics and electronics boxes was not considered.

#### BP 1201

BP 1201 represents a generic location on the LH2 tank sidewall. Ascent heating is not the TPS driver in this area. The TPS thickness will depend on heat leak requirements, ice formation, stratification and air liquefaction. During ascent, induced environments result in LH2 heating. The amount of stratified propellant is directly proportional to the heat absorbed. The cryogenic insulation is designed to ensure that the weight of unusable propellants due to stratification is minimized and that the heat leak is within limits. Close-out areas or localized heating due to protuberances will have to be evaluated as the design progresses. One (1.0) inch of NCFI 24-124 was evaluated and as shown in Figure 5.6-13, the surface temperature is well below the recession temperature of 650°F. The tank structural material (AL Lithium 2195) is also below its temperature limit of 300°F.

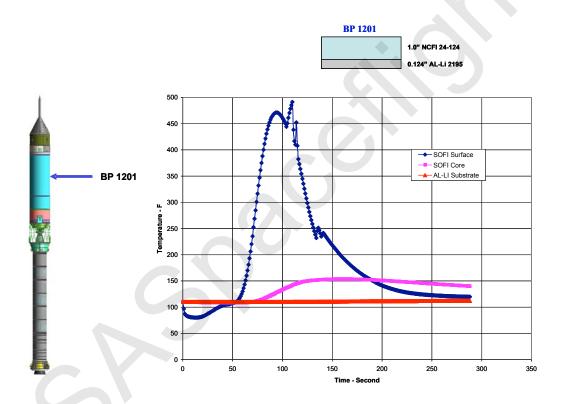


Figure 5.6-13. BP 1201 Temperature Profiles

## BP 1202

BP 1202 represents a location on the LOX tank sidewall. The TPS thickness will depend on heat leak requirements and ice formation. One-half (0.5) inch of NCFI 24-124 was evaluated and the temperature profiles are shown in Figure 5.6-14. The TPS surface temperature is well below the recession temperature of 650°F and the AL Lithium 2195 structural temperature does not exceed the material limit of 300°F.

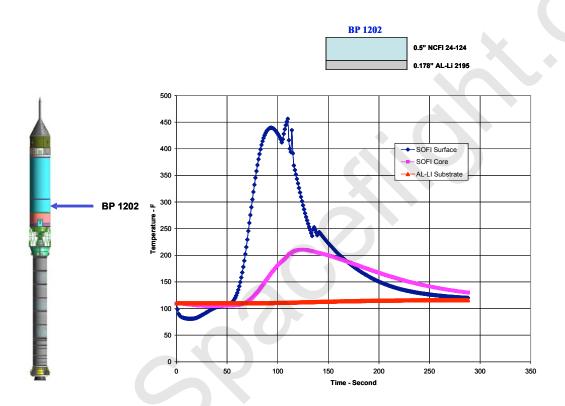


Figure 5.6-14. BP 1202 Temperature Profiles

#### Stage 2 Summary

TPS sizing details were based on ascent environment evaluation of structural and bondline temperature limits. Other considerations (not evaluated in this study) in the TPS sizing are icing, stratification, heat leak requirement and air liquefaction. No environments were available in the intertank area but 0.5 inches of SOFI would be adequate for weight estimation purposes. Also 0.5 inches of BX-265 would be sufficient for the LOX/LH2 domes. BX-265 has replaced SS-1171 for environmental reasons on ET domes. No attachment points or protuberances were evaluated. Table 5.6-2 summarizes the sizing results.

Major Assembly	Component	TPS Material	Material Density (lbm/ft <sup>3</sup> )	TPS Thickness (inch)
Spacecraft Adaptor	Acreage	NCFI 24-124	2.5	0.25
LH2 Tank	Acreage	NCFI 24-124	2.5	1
Intertank	Acreage	NCFI 24-124	2.5	0.5
LOX Tank	Acreage	NCFI 24-124	2.5	0.5
LOX/LH2 Domes	Acreage	BX-265	2.2	0.5

Table 5.6-2. CLV Stage 2 Acreage TPS Summary

#### References

- 1. Solid Rocket Booster Thermal Design Data Book, Rev. F, SE-019-068-2H
- 2. External Tank SLWT Thermal Data Book, Rev. E, 80900200102
- CLV Ascent Heating Environments, Electronic File, Victor Pritchett/EV33, September 9, 2005
- 4. CLV SSME Separation Impingement Environments, Electronic File, Mark D'Agostino/EV33, July 12, 2005
- 5. SRB Design Environments, Electronic File, Darrell Davis/EV34, July 13, 2005
- 6. SINDA/ABL Solution Routine Updates, NASA/MSFC Memo., ED66 (93-12), J. L. Clayton, Thermal Protection Systems Branch

### **5.6.1 Avionics Box Thermal Assessment**

A preliminary estimate of the box level base plate temperatures were calculated based on conservative assumptions (discussed below) about the thermal environment and box heat input. The preliminary analysis herein predicts the bulk temperature for each avionics box without considering radiation or conduction to the mounting surface or surrounding environment. A bulk temperature rise was predicted based on the launch timeline for both the core booster avionics boxes and the upper stage avionics boxes. This preliminary analysis was performed to determine any components of particular concern for development of a passive TCS.

#### **Modeling Approach**

Preliminary analysis was performed assuming an initial avionic box temperature of 21°C at the time of launch. The box input power and mass was obtained from the avionics equipment list [1]. It was assumed conservatively that the box power input was all dissipated within that box. For two components, the S-Band Power Amplifier and GPS Transmitter, estimated power dissipations

were used [2]. Any thermal contact of the avionics box with radiation or conduction to the mounting surface or surrounding environment was neglected. For the purpose of the analysis all boxes were assumed to be aluminum 2219. The analysis herein estimates bulk temperature rise for the whole avionic box which is considered comparable to the base plate temperatures of the avionics box. Determination of resulting internal component peak temperatures will require packaging analyses and evaluation of variance between base plate temperatures and these bulk avionic box temperature rise estimates for each box.

The duration over which the input power was applied varied depending on the mounting location. The upper stage avionics are mounted in the spacecraft adapter at the top of upper stage. Upper stage avionics are assumed operational from launch until upper stage separation. The duration from launch to Main Engine Cut Off (MECO) is 473 seconds [3]. It was assumed that separation occurs instantaneously after MECO for the purpose of this analysis. Any additional avionics operational time after MECO will be needed to be evaluated at a later date. The core booster avionics are mounted on the frustum at the top of the booster. The core booster avionics are assumed to be operational for 333 seconds from launch until 15,000ft [3] and an additional 59 seconds to splashdown [4] for a total duration of 392 seconds.

#### Results

A preliminary bulk temperature rise for each upper stage avionics box was estimated and the results are shown in Table 5.6.1-1. The temperatures shown are assumed equivalent to the avionic box base plate temperature; internal component temperatures will need to be determined by packaging analysis. For these components the estimated temperature rise varies from 0°C to 5°C for all but four components. The four components with higher preliminary temperature rise estimates are listed below:

- S-Band Transceiver estimated 9°C temperature rise,
- S-Band Power Amplifier estimated 12°C temperature rise,
- C-Band Transponder estimated 7°C temperature rise, and
- GPS Transmitter estimated 30°C temperature rise.

A preliminary bulk temperature rise for each core booster avionics box was estimated and the results are in Table 5.6.1-2. The temperatures shown are assumed equivalent to the avionic box base plate temperature; internal component temperatures will need to be determined by packaging analysis. For these components the estimated temperature rise varies from 0°C to 3°C for all but one component. The rate gyro has a preliminary temperature rise estimate or 10°C.

#### Conclusions

The objective of this preliminary study was to estimate the base plate temperature rise to provide data for preliminary evaluation of the thermal control system methodology needed. For the majority of components the temperature rise was 5°C or less. Although the bulk temperature rise was low for these components, the heat transfer path needs to be considered to adequately evaluate the design. For instance, if a component with a high heat dissipation is not in adequate thermal contact with the box exterior it may exceed its temperature requirement. For the remaining five components the temperature rise was estimated to be between 7°C to 30°C and they will require more detailed analysis to determine in a passive cooling method can be used for these components. This shows that more work will be needed to determine if a passive thermal

control system is feasible for the avionic system. Any updates to the timeline that would increase avionics operational time will have a significant impact on preliminary results herein and may impact the ability to provide a passive thermal control system. For example operation of the upper stage electronics beyond MECO will result in larger temperature rise predictions.

In addition to these preliminary results, the component temperature requirements and heat conduction path to the box base plate are needed. The factors relevant to TCS design for the CLV avionics include internal and external environments. The avionic box internal environment is determined by the following:

- Thermal heat dissipation and distribution for the box power input.
- Conduction path from internal components to the box base plate, and
- Flight operational timeline.

The avionic box external environment is determined by the following:

- Design of the base plate, mounting surface and interface material including possible use of phase change devices,
- Design of mounting locations to optimize heat dissipation,
- Ascent and re-entry environmental influences on avionics boxes, and
- Pre-launch purge conditions that define initial avionics temperatures.

Determination if a passive thermal control system is feasible for the avionics cannot be made based on the information provided herein. More detailed analysis of any areas of concern including those identified by this preliminary analysis is required to determine if a passively cooled TCS is feasible. Once environments are determined and temperature requirements for components are defined the TCS approach can be adequately evaluated. The final determination of TCS design is pending additional design definition and more detailed analyses that will be completed as the design matures. Any additional analysis will need to consider the factors listed above. We will facilitate modeling improvements as design and operational details emerge while the design matures.

Table 5.6.1-1. Preliminary Upper Stage Avionics Box Base Plate Temperature Estimates

Components	UnitWt (lb)	UnitPower (W)	UNIT SIZE (in)	Duration (s)	Initial Temp. (C)	UnitDelta Temp. (C)	Estimated Final Temp. (C)
UPPER STAGE AMONICS	1,42,	(1)	<b>\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\\</b>	473		10	
011211017102711100				Launch to ME	CO		
Communications	_	1	•	1			
S-band Transceiver	5.0	36.0	8x6.3x3.4	473	21	l 9	30
S Band Power Amp*	5.0	50.0	6x4x1.5	473	21	12	33
S Band Antennas	0.7	0.0	6x6x11.75	473	21	0	21
S-band Diplexer	2.0	0.0	2.5x3.5x2.5	473	21	0	21
S-band Bandreject Filter	0.9	0.0	1.2x2x5.5	473	21	0	21
S-band Bandbass Filter	0.9	0.0	1.2x2x5.5	473	21	0	21
S-band Transfer Switch	0.9	0.0	3x2x2.5	473	21	0	21
Coax	0.0	0.0					
*Estimated power dissipation from Bill Hopkins, 7.	/21/05						
Data System							
	150		0.7.10	473	21	5	26
Flight Control Computer CMD & TM Unit w/Encrypt	15.0 16.0	60.0 70.0	6x7x18 6x7x20	473	21	5	
1553 Data Bus Coupler	0.0	0.0	.9x.7x.5	473	21	0	20
Data Acquisition Unit	15.0	50.0	.9x.7x.5 7x7x12	473	21	4	
Upper Stage Data I/F Unit	60.0	100.0	7x7x12 7x7x16	473	21	2	
SSMEC/Interface Unit	100.0	180.0	14x18x15	473	21	2	
Data Bus Isolation Amplifier	5.4	22.3	8.5x4x7	473	21	5	
Data bus isolation Ampilier	5.4	22.5	0.3X4X7	473	21	3	20
GN&C			I				
GPS/INS	19.3	50.0	7x9x11	473	21	3	24
GPS Antenna	0.5	0.0	4.7x3.0x1.42	473	21	0	
GPS Hybrid Coupler	1.0	0.0	2.3x1.4x2.1	473	21	0	
Coax	0.0	0.0	-	110		J	21
Range Safety	_	1					
Command/Receiver/Decoder	6.0	3.9	3.0x7.0x7.5	473	21	1	
Hybrid Coupler, FTS	0.8	0.0	6.3x2.3x1.8	473	21	0	
Battery, FTS	12.0	0.0	6.5x8x5.5	473	21	0	
UHF RS Antenna	1.6	0.0	12.9x5.6x1.0	473	21	0	21
Coax	0.0	0.0	-				
C-Band Radar RSS Support	_					)	I
C-Band Transponder	2.7	16.8	4.7x5x2.5	473	21	7	29
C-band Antenna	0.3	0.0	6x6x1.75	473	21	0	21
Hybrid	0.8	0.0	3.33x2x2.5	473	21	0	21
Coax	0.0	0.0	-				
GPS/INS On-Vehicle RSS Support		L 50.0	1 764	470	0.1		1 04
GPS/INS (LN100)	19.3	50.0	7x9x11	473 473	21	30	24 51
Transmitter*	2.0	50.0	2.0x3.0x0.8				
Encoder CRS Astronomy	10.0	10.0	5x5x5	473	21 21	1 0	
GPS Antenna	0.5	0.0	4.7x3.0x1.42	473 473	21	0	
GPS Hybrid Coupler *Estimated power dissipation from Bill Hopkins, 7.	21.05	0.0	2.3x1.4x2.1	4/3	21	0	21
Electrical Power System							
Silver-Zinc Battery	40.0	7.4	25x6x6	473	21	0	
Power Distributor	80.0	65.0	22x15x9	473	21	1	
Cabling/Harness	500.0	0.0		473	21	0	21

Table 5.6.1-2. Preliminary Core Booster Avionics Box Base Plate Temperature Estimates

Components	Unit Wt. (lb)	Unit Power (W)	UNIT SIZE (in)	Duration (s)	Initial Temp. (C)	Unit Delta Temp. (C)	Estimated Final Temp. (C)
CORE BOOSTER AMONICS				392			
				Launch to re-e	ntry+splash	down estimat	e
Data System							
1553 Data Bus Coupler	0.0	0.0	.9x.7x.5	392	21	0	
Data Acquisition Unit	15.0	50.0	7x7x12	392	21	3	
Booster Stage Data I/F Unit	80.0	125.0	7x7x20	392	21	2	
TVC Controller	100.0	250.0	14x18x18	392	21	2	24
GN&C							
Rate Gyro	1.7	16.0	3.8x3.8 Dia	392	21	10	31
Range Safety							
Command/Receiver/Decoder	6.0	3.9	3.0x7.0x7.5	392	21	1	22
Hybrid Coupler, FTS	0.8	0.0	6.3x2.3x1.8	392	21	0	
Battery, FTS	12.0	0.0	6.5x8x5.5	392	21	0	
UHF RS Antenna	1.6	0.0	12.9x5.6x1.0	392	21	0	21
Coax	0.0	0.0					
Electrical Power System							
Silver -Zinc Battery	40.0	7.4	25x6x6	392		0	
Power Distributor	80.0	65.0	22x15x9	392	21	1	
Cabling/Hamess	500.0	0.0		392	21	0	21

#### References

- [1] Teamcenter Engineering Web /MSFC projects/VIPA/VAC08 CLV/HLV/Avionic /Manned LV P1 Avionics Equip List 06-21-05.xls
- [2] Bill Hopkins, MSFC, e-mail dated 7/21/05
- [3] Terri Schmitt, MSFC, CLV-4 Rev2 mission profile, telephone discussion 7/21/05
- [4] Teamcenter Engineering Web /MSFC projects/VIPA/VAC08 CLV/HLV/Reference Information/SRB Illustrated Systems Manual Final.pdf

# 5.6.2 Upper Stage Propellant Tank Ice Formation and Heat Leak Assessment

This report describes a preliminary assessment of the ice formation and heat leak into the Upper Stage propellant tanks while loaded and held on the ground and during ascent to first stage separation. The CLV design is a two-stage rocket utilizing a LOX-LH2 propellant combination and single Space Shuttle Main Engine (SSME) for the Upper Stage (Stage 2) and a 4-segment Solid Rocket Booster (SRB) for Stage 1. The vehicle layout is shown in Figure 5.6.2-1 below.



Figure 5.6.2-1. CLV Layout

The current CLV Thermal Protection System (TPS) design will utilize, where possible, existing External Tank (ET) and SRB flight qualified TPS materials. The ET TPS is composed of a Spray-on-Foam Insulator (SOFI) that serves to insulate the tank before and during launch. Table 5.6.2-1 summarizes the acreage TPS sizing results presented in reference 5.6.2-1. Reference 5.6.2-1 determined the LH2 and LOX tank acreage TPS thickness based on an ascent environment evaluation of structural and bondline temperature limits only.

Major Assembly	Component	TPS Material	Material Density (lbm/ft <sup>3</sup> )	TPS Thickness (inch)
Spacecraft Adaptor	Acreage	NCFI 24-124	2.5	0.25
LH2 Tank	Acreage	NCFI 24-124	2.5	1
Intertank	Acreage	NCFI 24-124	2.5	0.5
LOX Tank	Acreage	NCFI 24-124	2.5	0.5
LOX/LH2 Domes	Acreage	BX-265	2.2	0.5

Table 5.6.2-1. CLV Stage 2 Acreage TPS Summary

An expanded view of the Upper Stage is shown in Figure 5.6.2-2.

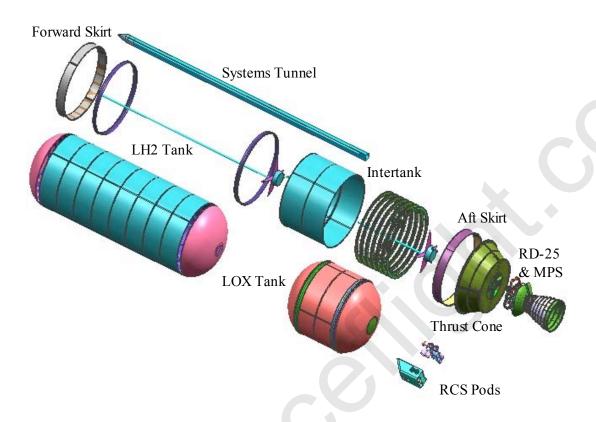


Figure 5.6.2-2. Upper Stage Expanded View

To optimally design cryogenic propellant vehicles, it is important that the heat transfer processes during the ground hold operations be analyzed. Analytical models must be capable of considering combinations of conduction, natural or forced convection, radiation, condensation and ice/frost formation. With this in mind, the methodology used to predict condensation or ice/frost formation in the SURFICEC [2] computer program was incorporated into the current sizing models.

#### **Thermal Model**

Figure 5.6.2-3 represents the LOX or LH2 tank TPS configuration. Also shown in this figure are the various terms considered in the calculation of the surface temperature for ground hold operations.

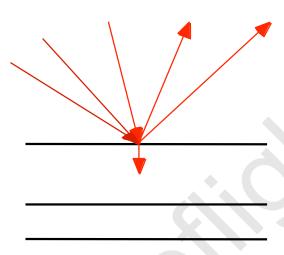


Figure 5.6.2-3. Typical LH2 & LOX Tank TPS Configuration

The surface energy balance is represented by equation 1.

$$q_{conduction} + q_{convection} + q_{latent} - q_{radiation(sky)} - q_{radiation(ground)} + q_{solar} = 0$$
 (1)

A steady-state one-dimensional SINDA/G [3] thermal model was created to calculate the thermal response of this TPS configuration during ground hold operations. All contributions from equation 1 were incorporated into this model to evaluate performance under various environmental conditions (ambient temperature, wind speed and relative humidity). The individual terms in equation 1 are described in detail in reference 5.6.2-2. This analysis does not consider the chilldown transients that occur as a cryogenic propellant tank is being loaded.

If the surface temperature is below the dew point temperature ( $T_{DP}$ ) and the freezing temperature, water vapor is assumed to be deposited on the surface in the form of ice/frost. The code does not distinguish between ice and frost. Condensation is formed if the surface temperature is greater than freezing but below  $T_{DP}$ . The tank is assumed to be fully loaded (inner wall of the tank is assumed to be at the propellant temperature). The model calculates a condensation or ice/frost rate of formation. This rate is an instantaneous rate and if applied to the time period for loading to determine a thickness will produce conservative results since the frost/ice layer is not modeled. In reality as a frost layer forms it has an insulating effect which causes the frosting (TPS) surface to rise in temperature. As the frost ages it increases in density and thermal conductivity which allows further growth. This continues until the frost becomes ice and a further

increase in conductivity is not possible. Since the conductivity is now constant the ice thickness increases until the outer surface reaches 32°F (or the dew point if lower than 32°F) and there is no further increase in thickness. Future work should include modeling the frost/ice layer and including this transient effect. Also not considered is the transient variation of the input conditions (for example the wind speed may vary with time). As mentioned previously the inner tank boundary condition is held constant at the propellant temperature, film boiling is not considered. Future work should include film boiling.

#### Results

#### Pre-launch

The following on-pad environmental conditions were analyzed:

Ambient Temperature: 55,75,90°F
Relative Humidity: 70,80,90%
Wind Speed: 0,5,10 knots
Solar Flux: 0,80 BTU/hr-ft-F

The heat balance assumes that there is no preexisting moisture on the outer TPS surface. Also the heat generated by the sun is set equal to zero (not a realistic assumption but does provide colder surface temperature predictions for ice formation) and

direct solar (80 BTU/hr-ft-F) for maximum heat leak calculations. It should be noted that results from this 1-D steady-state model are identical to Lockheed Martin Space System's ET HPSIM-solar [4] Window's based executable (when comparing current ET configuration).

For the LH2 tank condensation or ice/frost formation occurs at all analyzed conditions for zero solar input. The corresponding rate and heat leak per unit area into the tank are shown in Table 5.6.2-2. No condensation occurs when a direct solar input of 80 BTU/hr-ft-F is considered. The actual solar input will be somewhere in between these two extremes due to clouds, shading etc.

	Ī			ì			Direct solar	Ī		
TPS Thickness			Zero solar				input (80 BTU/hr			
1.0 inch			input				ft-F)			
Ambient			TPS Surface	Dew Point	Condensate	I	,	Dew Point	Condensate	
Temperature	Relative	Wind Speed	Temperature	Temperature	or Ice Rate	Heat Leak	TPS Surface	Temperature	or Ice Rate	Heat Leak
(F)	Humidity (%)	(knots)	(F)	(F)	(in/hr)	(BTU/hr-ft2)			(in/hr)	(BTU/hr-ft2)
55	70	0	19.41	45.30	0.00226	49.31	63.47	45.30	0	57.60
55	80	0	20.87	48.91	0.00263	49.54	63.72	48.91	0	57.60
55	90	0	22.30	52.11	0.00296	49.80	64.05	52.11	0	57.69
55	70	5	32.37	45.30	0.00263	51.67	60.17	45.30	0	56.94
55	80	5	34.20	48.91	0.00324	51.99	60.36	48.91	0	56.98
55	90	5	36.08	52.11	0.00388	52.33	60.53	52.11	0	56.99
55	70	10	39.51	45.30	0.00241	52.99	58.47	45.30	0	56.58
55	80	10	41.46	48.91	0.00345	53.35	58.65	48.91	0	56.65
55	90	10	43.34	52.11	0.00443	53.71	58.75	52.11	0	56.65
75	70	0	43.48	64.92	0.00320	53.74	81.25	64.92	0	61.10
75	80	0	45.66	68.67	0.00372	54.17	81.59	68.67	0	61.18
75	90	0	47.69	72.00	0.00424	54.55	81.89	72.00	0	61.22
75	70	5	55.16	64.92	0.00394	55.97	78.84	64.92	0	60.63
75	80	5	57.84	68.67	0.00488	56.48	79.06	68.67	0	60.64
75	90	5	60.37	72.00	0.00577	56.97	79.27	72.00	0	60.69
75	70	10	60.67	64.92	0.00348	57.05	77.65	64.92	0	60.39
75	80	10	63.31	68.67	0.00491	57.55	77.80	68.67	0	60.39
75	90	10	65.80	72.00	0.00626	58.05	77.97	72.00	0	60.44
90	70	0	62.25	79.64	0.00427	57.35	94.85	79.64	0	63.83
90	80	0	64.86	83.49	0.00494	57.86	95.26	83.49	0	63.92
90	90	0	67.22	86.92	0.00555	58.29	95.64	86.92	0	64.00
90	70	5	72.32	79.64	0.00494	59.29	93.04	79.64	0	63.49
90	80	5	75.46	83.49	0.00607	59.93	93.29	83.49	0	63.53
90	90	5	78.39	86.92	0.00714	60.55	93.53	86.92	0	63.56
90	70	10	76.50	79.64	0.00421	60.14	92.13	79.64	0	63.29
90	80	10	79.57	83.49	0.00589	60.75	92.31	83.49	0	63.30
90	90	10	82.42	86.92	0.00745	61.33	92.49	86.92	0	63.36

Table 5.6.2-2. LH2 Tank Acreage

Similar results are shown in Table 5.6.2-3 for the LOX tank. It is interesting to see that the heat leak into the tank is higher than those presented for the LH2 tank. This is due to the fact that 0.5 inches of NCFI 24-124 is used rather than 1.0 for the LH2 acreage areas. Also unlike the LH2 tank, condensation does occur for the direct solar case.

TPS Thickness										
							Direct solar			
			Zero solar				input (80 BTU/hr			
0.5 inch			input				ft-F)	5 5		
Ambient	<b>5</b>		TPS Surface	Dew Point	Condensate		TD0 0 1	Dew Point	Condensate	1
Temperature	Relative	Wind Speed	Temperature	Temperature	or Ice Rate	Heat Leak	TPS Surface	Temperature	or Ice Rate	Heat Leak
. ,	Humidity (%)	(knots)	(F)	(F)	(in/hr)	(BTU/hr-ft2)	Temperature (F)		(in/hr)	(BTU/hr-ft2)
55	70	0	4.35	45.30	0.0032	77.67	39.38	45.30	0.0005	90.42
55	80	0	5.97	48.91	0.0037	78.29	40.52	48.91	0.0007	90.82
55	90	0	7.51	52.11	0.0041	78.81	41.51	52.11	0.0009	91.20
55	70	5	21.72	45.30	0.0047	83.87	44.24	45.30	0.0002	92.22
55	80	5	23.72	48.91	0.0055	84.61	45.83	48.91	0.0008	92.84
55	90	5	25.68	52.11	0.0062	85.32	47.37	52.11	0.0014	93.45
55	70	10	33.00	45.30	0.0047	88.04	47.16	45.30	0.0000	93.35
55	80	10	35.02	48.91	0.0058	88.80	48.05	48.91	0.0005	93.67
55	90	10	37.01	52.11	0.0069	89.52	49.69	52.11	0.0013	94.31
75	70	0	29.60	64.92	0.0054	86.79	58.34	64.92	0.0010	97.61
75	80	0	31.90	68.67	0.0061	87.62	59.98	68.67	0.0014	98.25
75	90	0	34.09	72.00	0.0063	88.44	61.49	72.00	0.0018	98.84
75	70	5	45.78	64.92	0.0067	92.84	63.20	64.92	0.0008	99.52
75	80	5	48.77	68.67	0.0078	93.95	65.45	68.67	0.0016	100.42
75	90	5	51.57	72.00	0.0089	95.05	67.57	72.00	0.0025	101.23
75	70	10	55.44	64.92	0.0072	96.51	65.44	64.92	0.0000	100.42
75	80	10	58.28	68.67	0.0088	97.63	67.48	68.67	0.0012	101.16
75	90	10	60.96	72.00	0.0103	98.63	69.73	72.00	0.0024	102.04
90	70	0	49.76	79.64	0.0070	94.35	73.28	79.64	0.0016	103.47
90	80	0	52.70	83.49	0.0079	95.44	75.34	83.49	0.0021	104.31
90	90	0	55.44	86.92	0.0087	96.51	77.21	86.92	0.0026	105.03
90	70	5	64.78	79.64	0.0089	100.13	77.85	79.64	0.0013	105.28
90	80	5	68.30	83.49	0.0103	101.50	80.58	83.49	0.0024	106.36
90	90	5	71.53	86.92	0.0117	102.80	83.13	86.92	0.0034	107.38
90	70	10	72.40	79.64	0.0091	103.13	79.52	79.64	0.0002	105.95
90	80	10	75.69	83.49	0.0110	104.42	82.35	83.49	0.0018	107.11
90	90	10	78.76	86.92	0.0128	105.64	84.99	86.92	0.0033	108.21

# Table 5.6.2-3. LOX Tank Acreage (0.5 inch)

Table 5.6.2-4 shows the results if the LOX tank acreage TPS thickness is increased to 1.0 inch.

	1						Direct solar			
TPS Thickness			Zero solar				input (80			
1.0 inch			input				BTU/hr-ft-F)			
			·				NCFI			
	1		NCFI Surface	Dew Point	Condensate		Surface	Dew Point	Condensate	
Ambient	Relative	Wind Speed	Temperature	Temperature	or Ice Rate	Heat Leak	Temperature	Temperature	or Ice Rate	Heat Leak
Temperature (F)	Humidity (%)	(knots)	(F)	(F)	(in/hr)	(BTU/hr-ft2)	(F)	(F)	(in/hr)	(BTU/hr-ft2)
55	70	0	23.04	45.30	0.0020	42.19	68.25	45.30	0	50.76
55	80	0	24.48	48.91	0.0023	42.44	68.53	48.91	0	50.80
55	90	0	25.86	52.11	0.0026	42.71	68.79	52.11	0	50.86
55	70	5	34.63	45.30	0.0022	44.31	63.46	45.30	0	49.82
55	80	5	36.52	48.91	0.0028	44.66	63.67	48.91	0	49.85
55	90	5	38.36	52.11	0.0034	45.01	63.84	52.11	0	49.88
55	70	10	40.84	45.30	0.0019	45.47	60.73	45.30	0	49.29
55	80	10	42.76	48.91	0.0029	45.85	60.88	48.91	0	49.31
55	90	10	44.61	52.11	0.0039	46.19	61.00	52.11	0	49.35
75	70	0	46.47	64.92	0.0028	46.54	85.88	64.92	0	54.24
75	80	0	48.57	68.67	0.0033	46.94	86.19	68.67	0	54.31
75	90	0	50.52	72.00	0.0038	47.32	86.53	72.00	0	54.38
75	70	5	56.88	64.92	0.0033	48.53	82.03	64.92	0	53.51
75	80	5	59.47	68.67	0.0042	49.01	82.28	68.67	0	53.53
75	90	5	61.95	72.00	0.0051	49.51	82.48	72.00	0	53.58
75	70	10	61.62	64.92	0.0027	49.44	79.86	64.92	0	53.06
75	80	10	64.21	68.67	0.0042	49.98	80.03	68.67	0	53.09
75	90	10	66.66	72.00	0.0055	50.43	80.18	72.00	0	53.11
90	70	0	64.63	79.64	0.0037	50.05	99.46	79.64	0	57.02
90	80	0	67.12	83.49	0.0043	50.50	99.83	83.49	0	57.10
90	90	0	69.44	86.92	0.0049	50.98	100.16	86.92	0	57.18
90	70	5	73.60	79.64	0.0042	51.80	96.18	79.64	0	56.37
90	80	5	76.66	83.49	0.0053	52.39	96.42	83.49	0	56.39
90	90	5	79.51	86.92	0.0063	52.99	96.67	86.92	0	56.45
90	70	10	77.19	79.64	0.0033	52.52	94.31	79.64	0	55.97
90	80	10	80.21	83.49	0.0050	53.12	94.51	83.49	0	56.01
90	90	10	83.02	86.92	0.0065	53.69	94.68	86.92	0	56.06

Table 5.6.2-4. LOX Tank Acreage (1.0 inch)

#### **Ascent**

Ascent aerothermal environments corresponding to acreage locations on the LH2 tank (BP 1201) and LOX tank (BP 1202) were provided for the Upper Stage [5]. Figure 5.6.2-4 compares the cold wall heating for these locations and shows the baseline tank structure and TPS configurations [1].

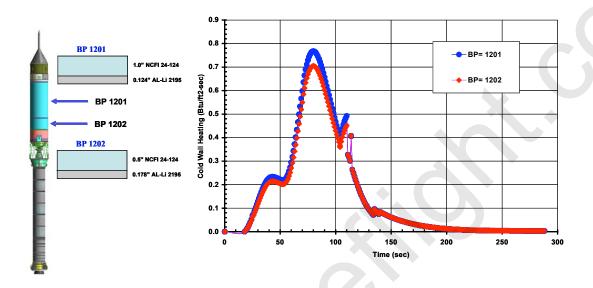


Figure 5.6.2-4. Upper Stage Cold Wall Heating Rates

Heat leak calculations are shown in Figures 5 and 6 for BP 1201 (LH2 acreage) and BP 1202 (LOX acreage) respectively. The calculations were terminated at separation prior to engine firing. Initial conditions for the transient model were determined using the steady-state code for an ambient temperature of 75°F, relative humidity of 70%, wind speed of 5 knots with direct solar input. Also shown in Figures 5.6.2-5 and 5.6.2-6 is the TPS surface temperature for each analyzed configuration. Note that this temperature is slightly cooler than those presented in reference 5.6.2-1. Reference 5.6.2-1 assumed an adiabatic condition for the tank surface (conservative for sizing purposes) while this analysis assumes the tank to be fully loaded (inner wall of the tank is assigned the propellant temperature). An adiabatic interior wall assumption eliminates the effect of the cold propellant absorbing heat from the tank wall. This adiabatic assumption (combined with the insulation effect of the foam) results in a higher foam surface temperature.

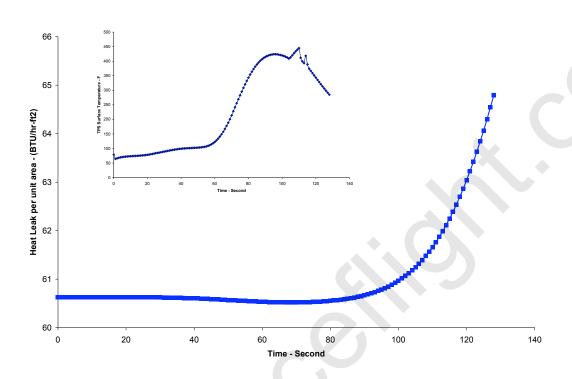


Figure 5.6.2-5. Ascent Heat Leak and TPS Surface Temperature for LH2 tank acreage location – BP 1201

Figure 5.6.2-6 compares the results using 0.5 and 1.0 inches of TPS on the LOX tank. The thicker insulation reduces the tank heat leak due to its insulating effect.

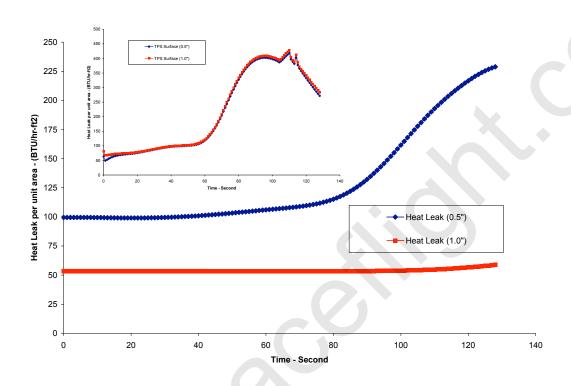


Figure 5.6.2-6. Ascent Heat Leak for LOX tank acreage location – BP 1202

#### Conclusions

A steady-state one-dimensional SINDA/G thermal model was created to calculate the thermal response of this TPS configuration during ground hold operations. This model predicts condensation or ice/frost rate for different environmental conditions. The tank heat leak is also calculated. This model compares favorably to existing Windows based executables for the ET model. This new model allows the analyst to change materials and thickness for CLV design options. A transient model was also developed to determine the heat leak into the tank for ascent conditions to separation/SSME firing. This model assumed any ice that formed during ground hold would fall off at lift-off. Film boiling was not considered, the tank inner wall was assumed to be fixed at the propellant temperature. This will produce conservative heat leak values.

## References

- 1. Crew Launch Vehicle (CLV) Thermal Protection System (TPS), report included in the Crew Launch Vehicle (CLV) VIPA Team Design Reference Manual
- 2. Documentation for Revision C of the Ice/Frost Computer Program (SURFICEC), M. Sporleder, Martin Marietta, Jan. 19, 1989, MMC 3522/T-89-002
- 3. SINDA/G, Systems Improved Numerical Differencing Analyzer, Network Analysis Inc., Version 2.3
- 4. HPSIM-solar, Lockheed Martin Space System, Michoud Operations ET Window's based executable
- 5. CLV Ascent Heating Environments, Electronic File, Victor Pritchett/EV33, September 9, 2005

#### 5.7 STRUCTURAL LOADS AND DYNAMICS

## 5.7.1 Study Objective

The objective of the VIPA Loads and Dynamics (L&D) team for this study was threefold. The first objective was to initiate the L&D assessment of a provided point-of-departure vehicle (PODV). These initial assessments are designed to drive out and assess the major load and dynamic contributors to the design. Given that the provided PODV relied on existing hardware, specifically the Shuttle SRB, the second objective was to compare the existing capabilities to the new configurations loads. The third objective was to provide insight to others regarding L&D considerations that will be present in the new design while understanding it's too early to quantitatively assess them. A detailed description of the data used, analyses conducted, and recommended design loads for the Point-of-Departure Vehicle is contained in the "Crew Launch Vehicle (CLV) System Structural Dynamics, Loads, And Models Databook," Revision 0.0, TBD. These recommended loads should be used for the next phase of CLV system design and analysis.

# 5.7.2 Point-of-Departure Vehicle and Previous Studies

VIPA L&D has participated in several studies pertaining to the definition and assessment of a crew capsule launched on a Shuttle SRB with a liquid upper stage. The first study was a "Human Rating" study lead by Gary Langford in the March to April 2005 timeframe. This concept consisted of a 4-segment SRB and a 5 meter upper stage with 4 expander cycle engines as described in Figure 5.7.2-1.

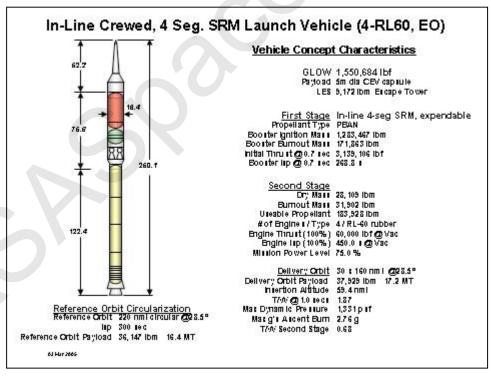


Figure 5.7.2-1 Human Rating Study Concept Description

Early in the 60-Day study a PODV was chosen that consisted of a 5-segment SRB and a 5 meter upper stage with either 4 expander cycle engines or 1 J-2S engine. This concept is described in Figure 5.7.2-2.

Finally the VIPA team was directed to work on what has emerged as the 60-Day study's PODV. This concept consisted of a 4 segment SRB and an upper stage with a single SSME/RS-25 or derivative. Figure 5.7.2-3 shows this concept. The pictured concept originally was a 5 meter upper stage but this was soon changed to 5.5 meters.

VIPA L&D was able to perform sufficient analysis on these three concepts to formulate assessments of different aspects of their loads and dynamics behavior. Taken all together they form a sound basis for an assessment of the general concept.

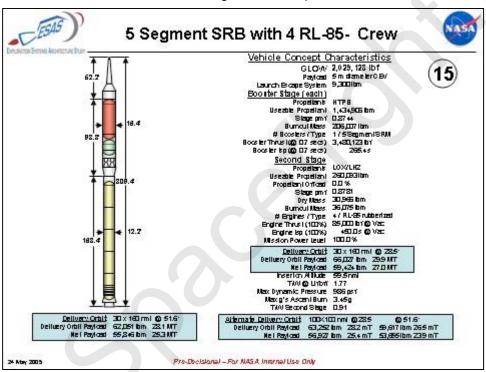


Figure 5.7.2-2 Early 60-Day Study Concept Description

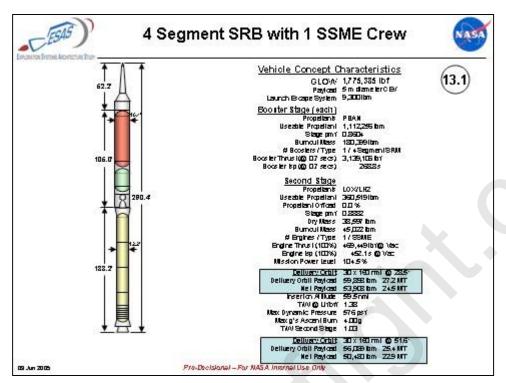


Figure 5.7.2-3 Emerged 60-Day Study Concept Description

# 5.7.3 Loads and Dynamics Assessments

A number of different assessments were made by the VIPA L&D team and are discussed here.

## 5.7.3.1 Structural Modeling

All the structural finite element models (FEM's) for the three concepts were very similar. The SRB models were the same models used to generate Shuttle Level II integration models and provided by United Space Alliance (USA). A 5<sup>th</sup> segment was copied from the forward center segment to generate the 5 segment SRB model. The upper stage, crew module and escape tower are simple beam element models generated for this task. Finally, the Mobile Launch Platform (MLP) model is the same model used for the Shuttle rollout testing analysis. Figure 5.7.3.1-1 shows a picture of one of these models along with a deflected shape plot (exaggerated).



Figure 5.7.3.1-1 Structural FEM Model and Deflected Shape (Exaggerated)

Structural frequencies for the different concepts are shown in Figure 5.7.3.1-2. These results indicate some low frequencies, particularly when constrained to the MLP. This indicates a supporting tower with a vehicle support will likely be required on the MLP. This is particularly true for the MLP rollout. Gross liftoff weight (GLOW) free-free or flight frequencies are around 1 Hz which is comparable to Saturn V.

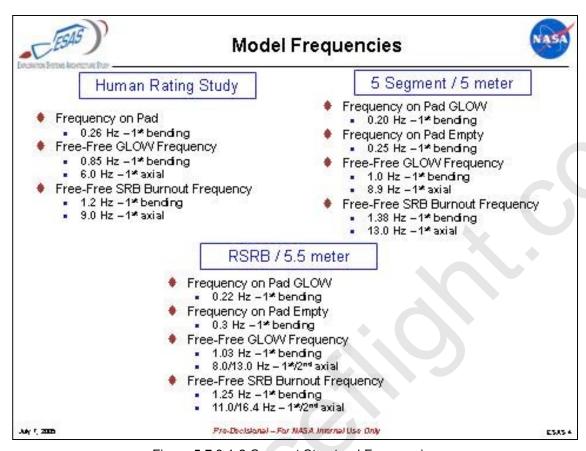


Figure 5.7.3.1-2 Concept Structural Frequencies

## 5.7.3.2 Sizing Cycle

At the start of the 60-Day study an initial sizing effort was conducted on the upper stage of the 5 segment concept. The design loads were taken from the Human Rating study pre-launch and 3ơ maximum dynamic pressure case. A factor of 1.5 was applied to these loads to account for unknowns. The VIPA Stress team used an Isogrid Stiffening Spreadsheet by Chiroux (12/2002) to size the upper stage structure. The resulting weight was found to be comparable with the Intros primary structure weight provided by Advanced Concepts. This result offers more confidence that the vehicle weight is reasonable.

#### 5.7.3.3 Pre-Launch Ground Winds

Pre-launch ground winds were looked at for all three concepts. Figures 5.7.3.3-1 through 5.7.3.3-3 illustrate the bending moments created within the vehicles. Comparison data for the 4 segment SRB from the SRB Loads Databook are also plotted.

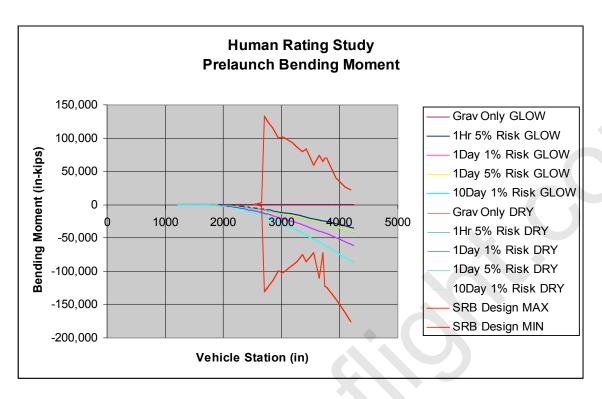


Figure 5.7.3.3-1 Human Rating Study Pre-Launch Bending Moments

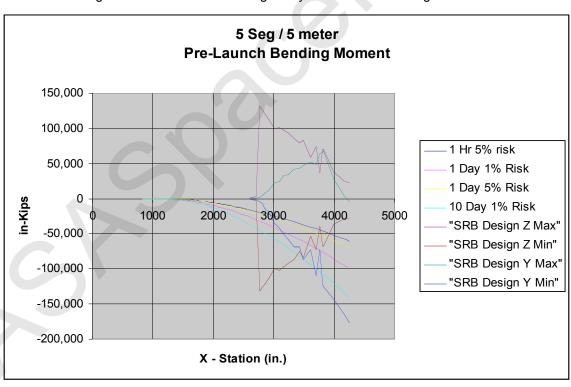


Figure 5.7.3.3-2 5 Seg / 5 meter Study Pre-Launch Bending Moments

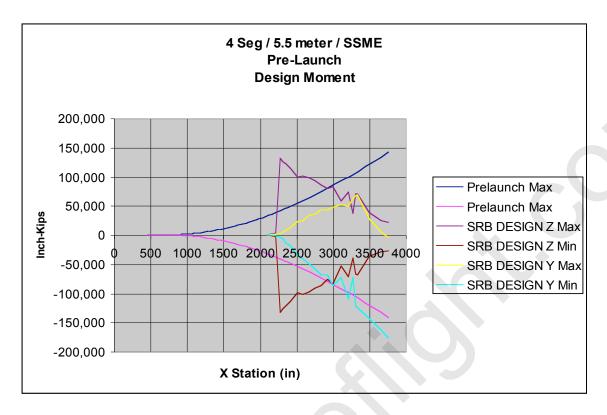


Figure 5.7.3.3-3 4 Seg / 5.5 meter /RS-25 Study Pre-Launch Bending Moments

Given the relatively simple steel case design of the SRB the 4 segment design loads should equally apply to the 4 aft most segments of the 5 segment SRB. Likewise, the directional bias shown in the SRB Design moments should only be an artifact of the Databook and not the structure itself. It can be seen that for the most part the Pre-launch loads fall near or below the SRB design moments. The longer vehicles are beginning to encroach on the design loads.

Table 5.7.3.3-1 shows the SRB hold down post loads for the three configurations as well as the capability indicators from the SRB Loads Databook (Reference 5.7.5-6). All configurations are well within the post limits.

Hold down Post Load Indicator										
<u>Load</u>	Indicator(KIPS)	5 Segment Concept	4 <u>Segment</u> <u>PODV</u> <u>Pre-</u> <u>Launch</u>							
Maximum F(X)	1591.49	354.8	876.9	830.5						
Minimum F(X)	-706.67	70.7	37.4	-37.6						

+ Post Compression - Post Tension

Table 5.7.3.3-1 SRB Hold down Post Loads

## 5.7.3.4 Ascent Flight

Ascent flight loads were looked at for all three concepts. A number of assumptions were made to facilitate these early assessments.

First, only quasi-static loads were considered. In this case, the vehicle FEM was considered to be in equilibrium between applied and inertia loading for each time point in the trajectory that was analyzed. The vehicle was considered to be in a trimmed state; i.e. the roll, pitch, and yaw rotational accelerations were trimmed to zero using engine control forces. This trimming can be affected by the inertial properties of the vehicle and hence how they are modeled in the FEM. The FEM masses were based primarily on Intros data provided by Advanced Concepts. These masses were distributed over the length of the vehicle as logically as possible during the modeling to provide a reasonable approximation to the rotational inertia characteristics of the vehicle. No dynamic or stiffness effects have yet been assessed. These would include static aeroelastics, gust, buffet, etc. The effects of these dynamics can be significant design drivers and can accounting for load increases on the order of 25-30%. The impact of these effects will be one of the first priorities as a concept is selected and a dedicated effort is begun.

Second, aerodynamic loading distributed down the length of the vehicle was not readily available during these studies. Net total normal force (lift), drag, and center-of-pressure data was available; primarily for performance work but also available to the VIPA L&D team. For this reason, L&D constructed some simplified engineering aerodynamic models of the vehicle using the Zonair code produced by ZONA Technologies. As documented in a previous VIPA VAC04 report (Reference 5.7.4-5), these models tend to predict normal force data significantly lower than that provided by the VIPA Aerosciences Team as well as having centers-of-pressure farther forward. The previous report also indicates that the Zonair code significantly under predicts the normal force coefficient on the SRB aft skirt. Therefore, to provide an alternate aerodynamic loading distribution, an additional normal force coefficient was added to the aft skirt of such a magnitude to correct for the displaced center-of-pressure. The resulting distribution was then scaled to match the provided net total normal force coefficient. Both sets of aerodynamic distributions were assessed to provide confidence that the loads were adequately enveloped. The altered distribution generally produced the worst loading. Additionally, these Zonair predictions are linearly increasing with AOA. This assumption does not necessarily hold for large AOA. During the later portions of flight as the dynamic pressure declines the vehicle tends to fly at increased angles of attack. The combination of these two effects could produce unrealistically high loads during this low dynamic pressure flight. For this reason, the aerodynamic loads were reduced by the following two rules. If the altitude was above 150,000 feet and the dynamic pressure was below 100 psf the aerodynamic loading was reduced by 50%. If the altitude was above 150,000 feet and the dynamic pressure was below 50 psf the aerodynamic loading was reduced by 75%. The provided performance drag loads were used for both sets of distributions. Obtaining realistic aerodynamic environments and assessing the impact of these assumptions will also be one of the first priorities as a concept is selected and a dedicated effort is begun.

Finally, these assessments generally do not have a 6-dof control simulation with dispersions available at the time of the loads analysis. This was true for the 5 segment RSB and the 4 segment / RS-25 configurations. For these cases the 3-dof performance trajectory is used to define the loading environments. An additional AOA and sideslip dispersion are added to the trajectory data to simulate possible dispersions that would result from a 6-dof simulation. In this

case a 5 degree dispersion was added to both AOA and sideslip. This results in a worst case dispersion of a little over 7 degrees. This dispersion would normally be considered severe however it was deemed prudent at this point given the other assumptions. The Human rating study had a defined  $3\sigma$  dispersed trajectory that was used for that configuration's load assessment without additional AOA and sideslip dispersions added by L&D. Maximum acceleration loads were also calculated in these assessments with the exception of the Human Rating study.

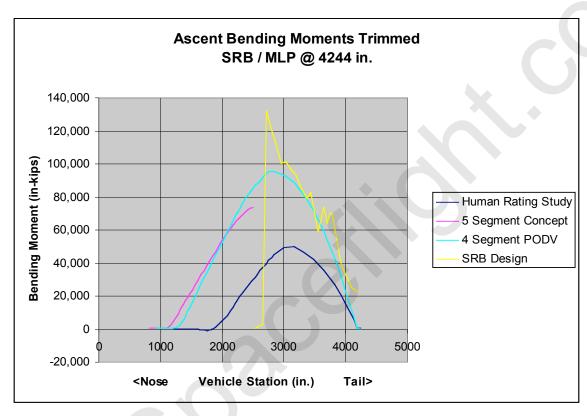


Figure 5.7.3.4-1 Ascent Bending Moment Comparisons

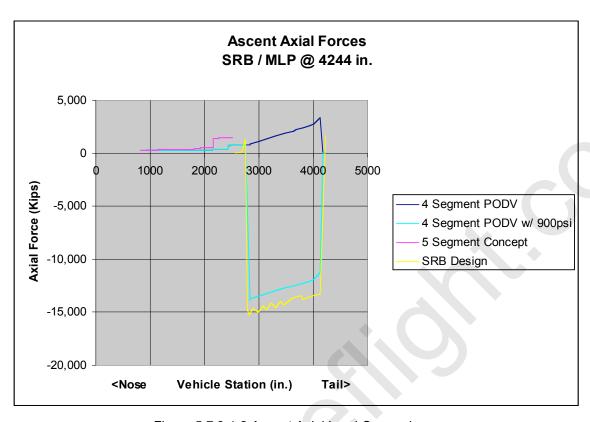


Figure 5.7.3.4-2 Ascent Axial Load Comparisons

Figures 5.7.3.4-1 and 5.7.3.4-2 show comparisons of the resulting ascent bending moments and axial loads for the various concepts. The axial load plot (Figure 5.7.3.4-2) shows the PODV with and without an SRB tension load due to a 900 psi internal pressure. This is for comparison purposes with the SRB design data which includes internal pressure.

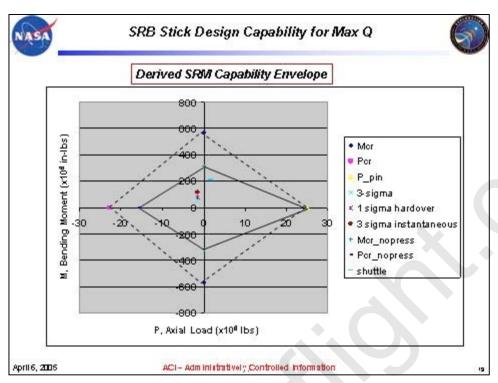


Figure 5.7.3.4-3 Derived SRM Capability Envelope

Additionally, the VIPA Stress team took resulting loads from the Human Rating study 3 $\sigma$  design case and compared them to the SRB casing capability. These results assume a simple steel cylinder of comparable thickness and no benefits from pressure within the SRB. From Figure 5.7.3.4-3 one can easily see the design loads are well within the derived SRM case capabilities.

These plots indicate the ascent loads are currently within the design of the existing SRB. The 5 segment concept analysis did not recover loads within the SRB, however the trends from the plots are apparent. The maximum dynamic pressure bending moments for the PODV are encroaching on these design limits. However, given the relatively simple steel case design of the SRB the 4 segment load capabilities should equally apply to the remaining aft segments of the SRB.

#### 5.7.3.5 SRB Actuator Hard Over Failure

One of the key aspects looked at during the Human Rating study was that of an SRB actuator failure. There were several assumptions made to define a credible failure. A complete actuator failure, such as a structural failure or complete loss of hydraulic pressure, was not deemed credible. Rather the failure was assumed to be from a software or command failure or a gradual hydraulic pressure release which would eliminate the ability to control the vehicle. This led to a 6-dof simulation where the SRB failure took a finite amount of time to develop. Secondly, the actuator failure was not assumed to be coincident with a  $3\sigma$  design case; a  $1\sigma$  case was used. Thirdly, the SRB failure was assumed to be initiated 5 seconds prior to a design wind gust hitting the vehicle. Finally, the actuator failure was assessed for 2 cases; failure "up" which would contribute to the pitch up of the vehicle due to angle-of-attack (AOA), and failure "down" which would pitch the vehicle down against the initial AOA and push it through zero into a growing

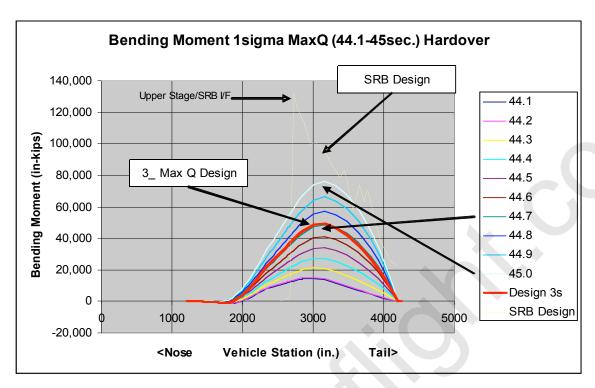


Figure 5.7.3.5-1 1σ Max Q Hardover (Gimbal Down)

negative AOA. In both cases it took about 0.6 seconds from the failure for the load in the SRB to increase to the CLV  $3\sigma$  design case. For the gimbal "up" failure the forward structures (CEV and Upper Stage) reached the  $3\sigma$  design case instantly.

Initial results of the Iteration 2 PRA indicate the probability of an actuator hard-over is 1.06E-05 per hour per actuator. These numbers have not been approved or seen by the project office yet, but they have HEI and USA approval. This number is for loss of control and does not include structural failure or loss of hydraulic pressure.

The following calculations have been made by the VIPA GN&C team. The current PODV concept will be flying and using the actuators for a maximum of 133 seconds, therefore the probability of an actuator failing is 1.06x10-5\*133/3600 = 3.91x10-7. Given 2 actuators, the probability of failure is 7.94x10-7.

This indicates that loss of control due to actuator hard-over failure is not a credible failure.

These cases will necessitate further study with consideration being given to limiting the actuator travel and designing the forward structure to survive for a given period of time during such a failure.

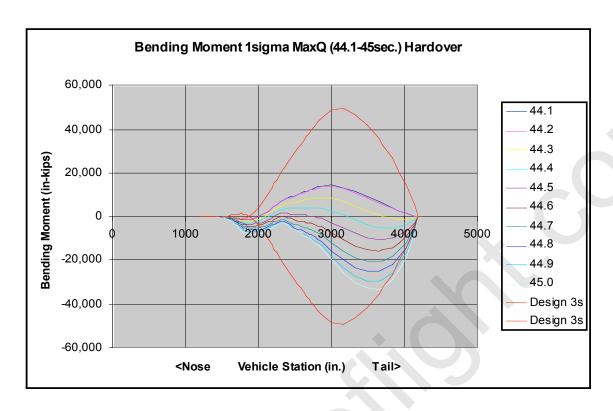


Figure 5.7.3.5-2 1σ Max Q Hardover (Gimbal Up)

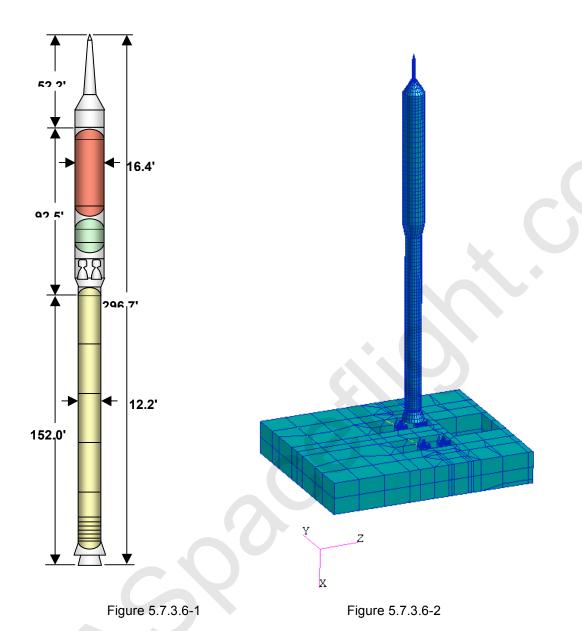
#### 5.7.3.6 Liftoff

A liftoff analysis for the 5-segment configuration of the CLV was performed for the 60-Day study. The analysis calculated vehicle responses (displacements and accelerations) due to vehicle/pad interface release while applying dynamic thrust build-up and static wind forces. Only one load case was considered, and dynamic overpressure and propellant slosh were not included in this analysis.

The analysis was performed using custom written FORTRAN programs designed and written by MSFC personnel. Documentation and verification for the method of this program can be found in reference 5.7.5-1 and 5.7.5-2. Reference 5.7.5-3 gives a relatively detailed look at the liftoff analyses performed using this method for the X33 vehicle.

## 5.7.3.6.1 Models

The 5-segment configuration of the CLV is depicted in figure 5.7.3.6-1. This vehicle is essentially a Space Shuttle 5 segment booster with a liquid fueled upper stage. The mathematical model used for the vehicle in the liftoff analysis was constructed in MSC-PATRAN, and preliminary analysis performed in NX-NASTRAN. Figure 5.7.3.6-2 shows the finite element model in MSC-PATRAN, along with the FE model of the mobile launch platform (MLP). For the liftoff



analysis, the model was extracted from NX-NASTRAN in the form of "Craig-Bampton" mass and stiffness matrices. These matrices contained 12 physical degrees of freedom and 54 generalized degrees of freedom. The 12 physical degrees of freedom corresponded to the attachments points of the launch pad and are constrained to the launch pad until the commanded release time. The remaining 54 generalized degrees of freedom contain the cantilevered component modes up to 35 Hertz. Propellants were modeled using RBE3's and concentrated masses.

The FEM of the MLP was also constructed in MSC-PATRAN, and a Craig-Bampton reduction was performed in NX-NASTRAN. The mass and stiffness matrices were extracted from NX-NASTRAN and used in the liftoff analysis. Along with the mass and stiffness matrices, load transformation matrices for the CLV were also extracted from NX-NASTRAN for data recovery. These Load Transformation Matrices (LTM's) were formulated using the modal acceleration technique in order to minimize modal truncation errors.

## 5.7.3.6.2 Analysis

Once the models and LTM's are extracted from NASTRAN, the liftoff analysis is performed in three main phases as listed below.

- Model transformation to un-coupled equations of motion and formatting of LTM's and transient and dynamic loads
- 2. Transient response analysis of un-coupled equations simulating various liftoff load cases (i.e. different wind loads, release times, etc...)
- 3. Data recovery of desired response quantities using aforementioned LTM's

One half percent modal damping was used for all modes in the response analysis, this is true for both the vehicle and pad models. This number is probably low for the SRB segment of the model, but should be conservative. For further details of each of these phases, see reference 5.7.5-3.

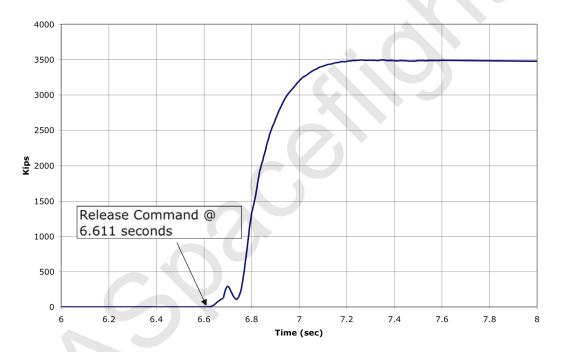


Figure 5.7.3.6-3 Approximated 5 Segment SRB Thrust Build-up

For the current analysis, only one load case was used. This load case consisted of an assumed nominal thrust build-up, a nominal release time, gravity load, and a static wind load corresponding to a 1 hour, 5% risk of exceedance.

The thrust build-up profile is shown in figure 5.7.3.6-3. This build-up profile was

developed by scaling the space shuttle SRB thrust loads, used for payload coupled loads analysis (see reference 5.7.5-4), to the 5 segment Engineering Test Motor 3 (ETM3) steady state thrust level. Figure 5.7.3.6-3 also indicates the commanded release time, chosen to correspond to the shuttle commanded release time.

#### 5.7.3.6.3 Results

Results from the liftoff analysis consist of element forces, nodal accelerations and nodal displacements. Figure 5.7.3.6-4 shows a comparison of the liftoff bending moment with the prelaunch wind and ascent loads generated for the 5 segment concept. The results for the SRB were not computed for this concept. Figure 5.7.3.6-4 does indicate that the ascent case is the primary load driver for the majority of the upper stage. It also indicates that liftoff dynamics may be the principle loads driver for the spacecraft and the extreme forward end of the upper stage.

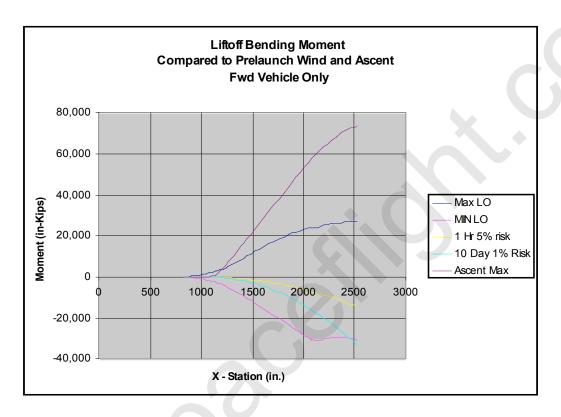


Figure 5.7.3.6-4 5 Segment Concept Load Comparison

Table 5.7.3.6-1 shows maximum and minimum acceleration data for various grid points in the model. Table 5.7.3.6-2 shows the vehicle stations corresponding to each grid point.

GRID	Vehicle Station	GRID	Vehicle Station
4002	829.990	6018	2109.480
4003	854.840	6019	2139.480
4004	879.690	6020	2167.440
4005	913.290	6021	2202.240
4006	946.890	6022	2237.040
4007	980.490	6023	2275.800
4008	1014.090	6024	2323.800
4009	1047.690	6025	2371.800
4010	1075.690	6026	2401.680
4011	1103.690	6027	2431.680
4012	1131.690	6028	2461.680
4013	1159.690	6029	2491.680
4014	1187.760	6030	2521.680
4015	1214.700	6102	1441.128
4016	1241.640	6103	1487.760
4017	1268.580	6113	1928.280
4018	1295.520	6114	1974.912
4019	1322.460	6116	2032.848
4020	1352.460	6117	2079.480
4021	1382.460	6120	2167.440
4022	1412.160	6121	2214.072
6001	1412.160	9999	3075.964
6002	1449.960	1000706	2529.830
6003	1487.760	1000707	2689.330
			·

600	4 1523	.760 10	000708	2849.330
600	5 1571	.760 10	000709	3009.330
600	6 1619	.760 10	000710	3169.330
600	7 1667	.760 10	000711	3329.330
600	8 1715	.760 10	000712	3489.330
600	9 1763	.760 10	000713	3649.330
601	0 1811	.760 10	000714	3703.310
601	1 1859	.760 10	000715	3828.850
601	2 1895	.760 10	000716	3931.350
601	3 1928	.280 10	000717	4015.390
601	4 1968	.280 10	000718	4141.700
601	5 2008	.280 10	000719	4191.370
601	6 2048	.280		
601	7 2079	.480		

Table 5.7.3.6-2 5 Segment Concept FEM Grid Stations

# TABLE 5.7.3.6-1 5 SEGMENT LIFTOFF ACCELERATIONS PAGE 1

#### COMPOSITE MAX-MIN SUMMARY

\*\*LIMIT LOADS\*\* LIFTOFF ACCELERATIONS (G'S) CEV 60 DAY STUDY \*\*LIMIT LOADS\*\*

## THE FOLLOWING RUN NUMBERS HAVE BEEN SEARCHED IN FORMING THIS TABLE-VAC-08

					MAXIMUM		ľ	MUMININ	
ROW		ROW							
NUMBER	1	DESCRIPTION		VALUE	TIME	RUNNO	VALUE	TIME	RUNNO
1	GRID	4002 ACCELERATION	X	1.033E+00	6.995	VAC-08	-2.178E+00	7.296	VAC-08
2	GRID	4002 ACCELERATION	Υ	3.498E-01	8.141	VAC-08	-4.906E-01	7.156	VAC-08
3	GRID	4002 ACCELERATION	Z	2.237E+00	7.106	VAC-08	-2.890E+00	7.195	VAC-08
7	GRID	4003 ACCELERATION	X	1.028E+00	6.995	VAC-08	-2.173E+00	7.296	VAC-08
8	GRID	4003 ACCELERATION	Y	2.875E-01	8.140	VAC-08	-4.071E-01	7.157	VAC-08
9	GRID	4003 ACCELERATION	Z	1.912E+00	7.105	VAC-08	-2.468E+00	7.196	VAC-08
13	GRID	4004 ACCELERATION	X	1.024E+00	6.995	VAC-08	-2.170E+00	7.296	VAC-08
14	GRID	4004 ACCELERATION	Y	2.281E-01	8.140	VAC-08	-3.274E-01	7.157	VAC-08
15	GRID	4004 ACCELERATION	Z	1.605E+00	7.103	VAC-08	-2.064E+00	7.196	VAC-08
19	GRID	4005 ACCELERATION	X	1.018E+00	6.995	VAC-08	-2.165E+00	7.296	VAC-08
20	GRID	4005 ACCELERATION	Y	1.509E-01	8.138	VAC-08	-2.236E-01	7.158	VAC-08
21	GRID	4005 ACCELERATION	Z	1.219E+00	7.098	VAC-08	-1.537E+00	7.198	VAC-08
25	GRID	4006 ACCELERATION	X	1.010E+00	6.995	VAC-08	-2.157E+00	7.296	VAC-08
26	GRID	4006 ACCELERATION	Y	1.139E-01	7.452	VAC-08	-1.295E-01	7.161	VAC-08
27	GRID	4006 ACCELERATION	Z	9.068E-01	7.088	VAC-08	-1.058E+00	7.202	VAC-08
31	GRID	4007 ACCELERATION	X	9.976E-01	6.995	VAC-08	-2.147E+00	7.296	VAC-08
32	GRID	4007 ACCELERATION	Υ	8.312E-02	7.451	VAC-08	-8.287E-02	8.029	VAC-08

33	GRID	4007 ACCELERATION	Z	7.303E-01	7.073	VAC-08	-8.172E-01	9.362	VAC-08
37	GRID	4008 ACCELERATION	Х	9.827E-01	6.995	VAC-08	-2.134E+00	7.296	VAC-08
38	GRID	4008 ACCELERATION	Y	7.254E-02	7.676	VAC-08	-8.519E-02	7.099	VAC-08
39	GRID	4008 ACCELERATION	Z	6.492E-01	7.064	VAC-08	-7.687E-01	9.365	VAC-08
43	GRID	4009 ACCELERATION	Х	9.648E-01	6.995	VAC-08	-2.119E+00	7.296	VAC-08
44	GRID	4009 ACCELERATION	Y	7.821E-02	7.677	VAC-08	-9.633E-02	7.096	VAC-08
45	GRID	4009 ACCELERATION	Z	5.956E-01	8.774	VAC-08	-7.165E-01	9.366	VAC-08
49	GRID	4010 ACCELERATION	Х	9.535E-01	6.995	VAC-08	-2.110E+00	7.296	VAC-08
50	GRID	4010 ACCELERATION	Y	7.298E-02	7.678	VAC-08	-9.174E-02	7.095	VAC-08
51	GRID	4010 ACCELERATION	Z	5.683E-01	8.773	VAC-08	-6.715E-01	9.367	VAC-08
55	GRID	4011 ACCELERATION	Х	9.453E-01	6.995	VAC-08	-2.103E+00	7.296	VAC-08
56	GRID	4011 ACCELERATION	Y	6.634E-02	7.678	VAC-08	-8.504E-02	7.095	VAC-08
57	GRID	4011 ACCELERATION	Z	5.359E-01	8.773	VAC-08	-6.267E-01	9.367	VAC-08
61	GRID	4012 ACCELERATION	Х	9.379E-01	6.995	VAC-08	-2.096E+00	7.296	VAC-08
62	GRID	4012 ACCELERATION	Y	5.936E-02	7.679	VAC-08	-7.778E-02	7.094	VAC-08
63	GRID	4012 ACCELERATION	Z	5.020E-01	8.773	VAC-08	-5.823E-01	9.368	VAC-08

# TABLE 5.7.3.6-1 5 SEGMENT LIFTOFF ACCELERATIONS (CONTINUED) PAGE 2

## COMPOSITE MAX-MIN SUMMARY

\*\*LIMIT LOADS\*\* LIFTOFF ACCELERATIONS (G'S) CEV 60 DAY STUDY \*\*LIMIT LOADS\*\*

		MAXIMUM	MINIMUM
ROW	ROW		

NUMBER		DESCRIPTION		VALUE	TIME	RUNNO	VALUE	TIME	RUNNO
67	GRID	4013 ACCELERATION	Х	9.307E-01	6.995	VAC-08	-2.090E+00	7.296	VAC-08
68	GRID	4013 ACCELERATION	Y	5.795E-02	6.858	VAC-08	-7.036E-02	7.094	VAC-08
69	GRID	4013 ACCELERATION	Z	4.676E-01	8.772	VAC-08	-5.505E-01	7.260	VAC-08
73	GRID	4014 ACCELERATION	Х	9.237E-01	6.995	VAC-08	-2.084E+00	7.296	VAC-08
74	GRID	4014 ACCELERATION	Y	5.688E-02	6.858	VAC-08	-6.281E-02	7.093	VAC-08
75	GRID	4014 ACCELERATION	Z	4.328E-01	8.772	VAC-08	-5.257E-01	7.264	VAC-08
79	GRID	4015 ACCELERATION	Х	9.165E-01	6.995	VAC-08	-2.078E+00	7.296	VAC-08
80	GRID	4015 ACCELERATION	Y	5.583E-02	6.858	VAC-08	-5.558E-02	7.092	VAC-08
81	GRID	4015 ACCELERATION	Z	3.993E-01	8.771	VAC-08	-5.059E-01	7.269	VAC-08
85	GRID	4016 ACCELERATION	Х	9.083E-01	6.995	VAC-08	-2.072E+00	7.296	VAC-08
86	GRID	4016 ACCELERATION	Y	5.478E-02	6.858	VAC-08	-4.845E-02	7.090	VAC-08
87	GRID	4016 ACCELERATION	Z	3.658E-01	8.771	VAC-08	-4.916E-01	7.275	VAC-08
91	GRID	4017 ACCELERATION	Х	8.991E-01	6.995	VAC-08	-2.064E+00	7.296	VAC-08
92	GRID	4017 ACCELERATION	Y	5.370E-02	6.858	VAC-08	-4.155E-02	7.088	VAC-08
93	GRID	4017 ACCELERATION	Z	3.322E-01	8.770	VAC-08	-4.841E-01	7.283	VAC-08
97	GRID	4018 ACCELERATION	Х	8.888E-01	6.995	VAC-08	-2.055E+00	7.296	VAC-08
98	GRID	4018 ACCELERATION	Y	5.267E-02	6.858	VAC-08	-3.521E-02	7.083	VAC-08
99	GRID	4018 ACCELERATION	Z	2.986E-01	8.769	VAC-08	-4.828E-01	7.288	VAC-08
103	GRID	4019 ACCELERATION	X	8.775E-01	6.995	VAC-08	-2.046E+00	7.296	VAC-08
104	GRID	4019 ACCELERATION	Y	5.159E-02	6.858	VAC-08	-3.421E-02	7.249	VAC-08
105	GRID	4019 ACCELERATION	Z	2.790E-01	7.882	VAC-08	-4.845E-01	7.292	VAC-08
109	GRID	4020 ACCELERATION	X	8.586E-01	6.995	VAC-08	-2.030E+00	7.296	VAC-08
110	GRID	4020 ACCELERATION	Y	5.042E-02	6.858	VAC-08	-3.918E-02	7.255	VAC-08
111	GRID	4020 ACCELERATION	Z	2.644E-01	7.880	VAC-08	-4.882E-01	7.294	VAC-08
115	GRID	4021 ACCELERATION	Х	8.314E-01	6.996	VAC-08	-2.007E+00	7.296	VAC-08
116	GRID	4021 ACCELERATION	Y	4.923E-02	6.858	VAC-08	-4.488E-02	7.257	VAC-08
117	GRID	4021 ACCELERATION	Z	2.500E-01	7.879	VAC-08	-4.923E-01	7.296	VAC-08
121	GRID	4022 ACCELERATION	Х	8.044E-01	6.996	VAC-08	-1.985E+00	7.295	VAC-08

122	GRID	4022 ACCELERATION	Y	4.802E-02	6.858	VAC-08	-5.065E-02	7.259	VAC-08
123	GRID	4022 ACCELERATION	Z	2.358E-01	7.877	VAC-08	-4.959E-01	7.297	VAC-08
127	GRID	6002 ACCELERATION	Χ	7.180E-01	6.996	VAC-08	-1.914E+00	7.295	VAC-08
128	GRID	6002 ACCELERATION	Y	4.651E-02	6.858	VAC-08	-5.884E-02	7.261	VAC-08
129	GRID	6002 ACCELERATION	Z	2.170E-01	7.876	VAC-08	-5.010E-01	7.298	VAC-08
133	GRID	6003 ACCELERATION	Χ	6.303E-01	6.996	VAC-08	-1.844E+00	7.294	VAC-08
134	GRID	6003 ACCELERATION	Y	4.741E-02	8.261	VAC-08	-6.570E-02	7.262	VAC-08

TABLE 5.7.3.6-1 5 SEGMENT LIFTOFF ACCELERATIONS (CONTINUED)

PAGE 3

#### COMPOSITE MAX-MIN SUMMARY

\*\*LIMIT LOADS\*\* LIFTOFF ACCELERATIONS (G'S) CEV 60 DAY STUDY \*\*LIMIT LOADS\*\*

			И	MUMIXA		1	MUMINIM	
ROW		ROW				<del></del>		
NUMBER		DESCRIPTION	VALUE	TIME	RUNNO	VALUE	TIME	RUNNO
135	GRID	6003 ACCELERATION Z	2.458E-01	8.943	VAC-08	-5.007E-01	7.298	VAC-08
139	GRID	6004 ACCELERATION X	5.691E-01	6.997	VAC-08	-1.797E+00	7.294	VAC-08
140	GRID	6004 ACCELERATION Y	5.242E-02	8.262	VAC-08	-7.005E-02	7.263	VAC-08
141	GRID	6004 ACCELERATION Z	2.696E-01	8.944	VAC-08	-4.930E-01	7.299	VAC-08
145	GRID	6005 ACCELERATION X	5.094E-01	6.997	VAC-08	-1.751E+00	7.293	VAC-08
146	GRID	6005 ACCELERATION Y	6.147E-02	7.613	VAC-08	-7.402E-02	7.263	VAC-08
147	GRID	6005 ACCELERATION Z	2.951E-01	8.944	VAC-08	-4.761E-01	7.300	VAC-08
151	GRID	6006 ACCELERATION X	4.494E-01	6.997	VAC-08	-1.707E+00	7.292	VAC-08

15	2 GRID	6006 ACCELERATION	Y	6.880E-02	7.613	VAC-08	-7.626E-02	7.264	VAC-08
15	3 GRID	6006 ACCELERATION	Z	3.150E-01	8.945	VAC-08	-4.939E-01	8.362	VAC-08
15	7 GRID	6007 ACCELERATION	Х	4.189E-01	6.782	VAC-08	-1.666E+00	7.291	VAC-08
15	8 GRID	6007 ACCELERATION	Y	7.420E-02	7.613	VAC-08	-7.820E-02	7.036	VAC-08
15	9 GRID	6007 ACCELERATION	Z	3.288E-01	8.945	VAC-08	-5.054E-01	8.363	VAC-08
16	3 GRID	6008 ACCELERATION	Х	3.897E-01	6.781	VAC-08	-1.635E+00	7.169	VAC-08
16	4 GRID	6008 ACCELERATION	Y	7.761E-02	7.613	VAC-08	-8.016E-02	7.036	VAC-08
16	5 GRID	6008 ACCELERATION	Z	3.367E-01	8.946	VAC-08	-5.090E-01	8.363	VAC-08
16	9 GRID	6009 ACCELERATION	Х	3.617E-01	6.780	VAC-08	-1.603E+00	7.169	VAC-08
17	0 GRID	6009 ACCELERATION	Y	7.896E-02	7.612	VAC-08	-8.041E-02	7.033	VAC-08
17	1 GRID	6009 ACCELERATION	Z	3.436E-01	7.646	VAC-08	-5.046E-01	7.070	VAC-08
17	5 GRID	6010 ACCELERATION	Х	3.357E-01	6.779	VAC-08	-1.572E+00	7.169	VAC-08
17	6 GRID	6010 ACCELERATION	Y	7.823E-02	7.612	VAC-08	-7.917E-02	7.032	VAC-08
17	7 GRID	6010 ACCELERATION	Z	3.516E-01	7.646	VAC-08	-5.164E-01	7.069	VAC-08
18	1 GRID	6011 ACCELERATION	Х	3.123E-01	6.777	VAC-08	-1.541E+00	7.286	VAC-08
18	2 GRID	6011 ACCELERATION	Y	7.553E-02	7.612	VAC-08	-7.622E-02	7.031	VAC-08
18	3 GRID	6011 ACCELERATION	Z	3.512E-01	7.646	VAC-08	-5.190E-01	7.068	VAC-08
18	7 GRID	6012 ACCELERATION	Х	3.010E-01	6.773	VAC-08	-1.526E+00	7.286	VAC-08
18	8 GRID	6012 ACCELERATION	Y	7.222E-02	7.612	VAC-08	-7.288E-02	7.030	VAC-08
18	9 GRID	6012 ACCELERATION	Z	3.454E-01	7.646	VAC-08	-5.151E-01	7.068	VAC-08
19	3 GRID	6013 ACCELERATION	Х	2.960E-01	6.771	VAC-08	-1.514E+00	7.285	VAC-08
19	4 GRID	6013 ACCELERATION	Y	6.836E-02	7.612	VAC-08	-6.911E-02	7.030	VAC-08
19	5 GRID	6013 ACCELERATION	Z	3.362E-01	7.646	VAC-08	-5.074E-01	7.067	VAC-08
19	9 GRID	6014 ACCELERATION	X	2.924E-01	6.768	VAC-08	-1.484E+00	7.284	VAC-08
20	0 GRID	6014 ACCELERATION	Y	6.204E-02	7.611	VAC-08	-6.298E-02	7.029	VAC-08
20	1 GRID	6014 ACCELERATION	Z	3.185E-01	7.646	VAC-08	-4.911E-01	7.065	VAC-08
20	5 GRID	6015 ACCELERATION	Х	2.921E-01	6.767	VAC-08	-1.452E+00	7.282	VAC-08

# TABLE 5.7.3.6-1 5 SEGMENT LIFTOFF ACCELERATIONS (CONTINUED) PAGE 4

#### COMPOSITE MAX-MIN SUMMARY

### \*\*LIMIT LOADS\*\* LIFTOFF ACCELERATIONS (G'S) CEV 60 DAY STUDY \*\*LIMIT LOADS\*\*

				1	MAXIMUM		I.	MUMININ	
ROW		ROW					<u> </u>		
NUMBER	I	DESCRIPTION		VALUE	TIME	RUNNO	VALUE	TIME	RUNNO
206	GRID	6015 ACCELERATION	Y	5.413E-02	7.611	VAC-08	-5.561E-02	7.027	VAC-08
207	GRID	6015 ACCELERATION	Z	2.934E-01	7.646	VAC-08	-4.672E-01	7.064	VAC-08
211	GRID	6016 ACCELERATION	X	2.941E-01	6.766	VAC-08	-1.444E+00	7.055	VAC-08
212	GRID	6016 ACCELERATION	Y	4.519E-02	7.609	VAC-08	-4.750E-02	7.025	VAC-08
213	GRID	6016 ACCELERATION	Z	2.619E-01	7.646	VAC-08	-4.364E-01	7.062	VAC-08
217	GRID	6017 ACCELERATION	Х	2.961E-01	6.765	VAC-08	-1.449E+00	7.055	VAC-08
218	GRID	6017 ACCELERATION	Y	3.764E-02	7.607	VAC-08	-4.052E-02	7.019	VAC-08
219	GRID	6017 ACCELERATION	Z	2.382E-01	9.373	VAC-08	-4.075E-01	7.060	VAC-08
223	GRID	6018 ACCELERATION	X	2.974E-01	6.765	VAC-08	-1.450E+00	7.055	VAC-08
224	GRID	6018 ACCELERATION	Y	2.993E-02	7.602	VAC-08	-3.320E-02	7.018	VAC-08
225	GRID	6018 ACCELERATION	Z	2.193E-01	9.377	VAC-08	-3.715E-01	7.058	VAC-08
229	GRID	6019 ACCELERATION	X	2.981E-01	6.765	VAC-08	-1.451E+00	7.055	VAC-08
230	GRID	6019 ACCELERATION	Y	2.493E-02	8.006	VAC-08	-2.706E-02	7.011	VAC-08
231	GRID	6019 ACCELERATION	Z	2.029E-01	9.383	VAC-08	-3.354E-01	7.057	VAC-08
235	GRID	6020 ACCELERATION	X	2.988E-01	6.765	VAC-08	-1.452E+00	7.055	VAC-08
236	GRID	6020 ACCELERATION	Y	2.250E-02	7.578	VAC-08	-2.298E-02	6.999	VAC-08
237	GRID	6020 ACCELERATION	Z	1.906E-01	9.390	VAC-08	-2.998E-01	7.055	VAC-08
241	GRID	6021 ACCELERATION	X	2.711E-01	6.765	VAC-08	-1.390E+00	7.057	VAC-08

242	GRID	6021 ACCELERATION	Y	2.362E-02	7.145	VAC-08	-2.166E-02	6.990	VAC-08
243	GRID	6021 ACCELERATION	Z	1.795E-01	9.399	VAC-08	-2.623E-01	7.916	VAC-08
247	GRID	6022 ACCELERATION	X	2.409E-01	6.765	VAC-08	-1.339E+00	7.059	VAC-08
248	GRID	6022 ACCELERATION	Y	2.806E-02	7.152	VAC-08	-2.398E-02	6.982	VAC-08
249	GRID	6022 ACCELERATION	Z	1.938E-01	7.295	VAC-08	-2.556E-01	7.908	VAC-08
253	GRID	6023 ACCELERATION	X	2.071E-01	6.765	VAC-08	-1.304E+00	7.061	VAC-08
254	GRID	6023 ACCELERATION	Y	3.272E-02	7.152	VAC-08	-2.904E-02	6.974	VAC-08
255	GRID	6023 ACCELERATION	Z	2.263E-01	7.296	VAC-08	-2.512E-01	7.904	VAC-08
259	GRID	6024 ACCELERATION	X	1.598E-01	6.765	VAC-08	-1.285E+00	7.064	VAC-08
260	GRID	6024 ACCELERATION	Y	3.753E-02	7.153	VAC-08	-3.619E-02	8.264	VAC-08
261	GRID	6024 ACCELERATION	Z	2.626E-01	7.299	VAC-08	-2.458E-01	7.899	VAC-08
265	GRID	6025 ACCELERATION	X	1.158E-01	6.743	VAC-08	-1.279E+00	7.065	VAC-08
266	GRID	6025 ACCELERATION	Y	4.646E-02	7.034	VAC-08	-4.256E-02	8.264	VAC-08
267	GRID	6025 ACCELERATION	Z	2.937E-01	7.301	VAC-08	-2.643E-01	8.712	VAC-08
271	GRID	6026 ACCELERATION	X	1.467E-01	6.742	VAC-08	-1.277E+00	7.065	VAC-08
272	GRID	6026 ACCELERATION	Y	5.284E-02	7.034	VAC-08	-4.576E-02	8.264	VAC-08
273	GRID	6026 ACCELERATION	Z	3.092E-01	7.301	VAC-08	-2.822E-01	8.711	VAC-08

# TABLE 5.7.3.6-1 5 SEGMENT LIFTOFF ACCELERATIONS (CONTINUED) PAGE 5

## COMPOSITE MAX-MIN SUMMARY

\*\*LIMIT LOADS\*\* LIFTOFF ACCELERATIONS (G'S) CEV 60 DAY STUDY \*\*LIMIT LOADS\*\*

		MAXIMUM	MINIMUM
ROW	ROW		

NUMBER	Ι	DESCRIPTI(	ON		VALUE	TIME	RUNNO	VALUE	TIME	RUNNO
277	GRID	6027	ACCELERATION	Х	1.786E-01	6.741	VAC-08	-1.275E+00	7.065	VAC-08
278	GRID		ACCELERATION	A Y	5.825E-02	7.034	VAC-08	-4.827E-02	8.264	VAC-08
279	GRID		ACCELERATION	Z	3.211E-01	7.302	VAC-08	-2.973E-01	8.711	VAC-08
283			ACCELERATION	X	2.114E-01	6.741	VAC-08	-1.272E+00	7.065	VAC-08
284	GRID GRID		ACCELERATION	A Y	6.256E-02	7.034	VAC-08	-5.027E-02	8.671	VAC-08
285	GRID		ACCELERATION	Z	3.290E-01	7.302	VAC-08	-3.027E-02	8.710	VAC-08
289	GRID		ACCELERATION	Z X	2.471E-01	6.740	VAC-08	-1.271E+00	7.066	VAC-08
290	GRID		ACCELERATION	Y	6.564E-02	7.034	VAC-08	-5.253E-02		VAC-08
290				Z	3.324E-01	7.303	VAC-08		7.094	
291	GRID GRID		ACCELERATION ACCELERATION	Z X	2.869E-01	6.740	VAC-08	-3.175E-01 -1.271E+00	8.710 7.066	VAC-08
296	GRID		ACCELERATION	Y	6.706E-02	7.033	VAC-08	-5.368E-02	7.614	VAC-08
297	GRID		ACCELERATION	Z 	3.295E-01	7.304	VAC-08	-3.205E-01	8.709	VAC-08
301	GRID		ACCELERATION	Х	2.964E-01	6.740	VAC-08	-1.292E+00	7.066	VAC-08
302	GRID		ACCELERATION	Y	7.258E-02	7.034	VAC-08	-5.739E-02	7.614	VAC-08
303	GRID		ACCELERATION	Z	3.386E-01	7.303	VAC-08	-3.278E-01	8.709	VAC-08
307	GRID		ACCELERATION	Χ	3.314E-01	6.740	VAC-08	-1.273E+00	7.066	VAC-08
308	GRID		ACCELERATION	Y	7.520E-02	7.028	VAC-08	-6.108E-02	7.612	VAC-08
309	GRID		ACCELERATION	Z	2.954E-01	7.305	VAC-08	-3.129E-01	8.706	VAC-08
313	GRID		ACCELERATION	X	2.741E-01	6.741	VAC-08	-1.068E+00	7.134	VAC-08
314	GRID	1000708	ACCELERATION	Y	3.037E-02	6.998	VAC-08	-3.392E-02	7.571	VAC-08
315	GRID		ACCELERATION	Z	1.947E-01	8.394	VAC-08	-2.421E-01	6.958	VAC-08
319	GRID	1000709	ACCELERATION	X	1.994E-01	6.746	VAC-08	-1.091E+00	7.128	VAC-08
320	GRID	1000709	ACCELERATION	Y	6.945E-02	7.093	VAC-08	-7.850E-02	7.151	VAC-08
321	GRID	1000709	ACCELERATION	Z	2.370E-01	7.529	VAC-08	-2.638E-01	7.185	VAC-08
325	GRID	1000710	ACCELERATION	Χ	2.041E-01	6.825	VAC-08	-1.174E+00	7.183	VAC-08
326	GRID	1000710	ACCELERATION	Y	8.349E-02	6.959	VAC-08	-6.850E-02	7.026	VAC-08
327	GRID	1000710	ACCELERATION	Z	2.675E-01	8.300	VAC-08	-3.486E-01	7.714	VAC-08
331	GRID	1000711	ACCELERATION	Χ	2.523E-01	6.722	VAC-08	-1.173E+00	7.117	VAC-08

332	GRID	1000711 ACCELERATION	Y	1.027E-01	6.943	VAC-08	-8.074E-02	6.987	VAC-08
333	GRID	1000711 ACCELERATION	Z	3.097E-01	6.978	VAC-08	-3.245E-01	7.756	VAC-08
337	GRID	1000712 ACCELERATION	Х	2.979E-01	6.820	VAC-08	-1.251E+00	7.221	VAC-08
338	GRID	1000712 ACCELERATION	Y	1.116E-01	6.926	VAC-08	-9.656E-02	7.234	VAC-08
339	GRID	1000712 ACCELERATION	Z	4.023E-01	6.954	VAC-08	-3.688E-01	7.272	VAC-08
343	GRID	1000713 ACCELERATION	Χ	2.455E-01	6.820	VAC-08	-1.305E+00	7.221	VAC-08
344	GRID	1000713 ACCELERATION	Y	1.507E-01	7.298	VAC-08	-1.112E-01	6.939	VAC-08

TABLE 5.7.3.6-1 5 SEGMENT LIFTOFF ACCELERATIONS (CONTINUED)

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#### COMPOSITE MAX-MIN SUMMARY

\*\*LIMIT LOADS\*\* LIFTOFF ACCELERATIONS (G'S) CEV 60 DAY STUDY \*\*LIMIT LOADS\*\*

			M	AXIMUM		MINIMUM			
ROW		ROW					- <del></del>		
NUMBER	Ι	DESCRIPTION		VALUE	TIME	RUNNO	VALUE	TIME	RUNNO
345	GRID	1000713 ACCELERATION	Z	5.085E-01	7.857	VAC-08	-6.662E-01	7.289	VAC-08
349	GRID	1000714 ACCELERATION	X	2.045E-01	6.846	VAC-08	-1.293E+00	7.221	VAC-08
350	GRID	1000714 ACCELERATION	Y	1.347E-01	7.298	VAC-08	-1.244E-01	7.482	VAC-08
351	GRID	1000714 ACCELERATION	Z	5.873E-01	7.019	VAC-08	-6.732E-01	6.984	VAC-08
355	GRID	1000715 ACCELERATION	X	1.760E-01	6.713	VAC-08	-1.207E+00	7.222	VAC-08
356	GRID	1000715 ACCELERATION	Y	4.663E-01	6.883	VAC-08	-4.352E-01	6.942	VAC-08
357	GRID	1000715 ACCELERATION	Z	9.593E-01	6.886	VAC-08	-9.672E-01	6.872	VAC-08
361	GRID	1000716 ACCELERATION	X	1.881E-01	6.711	VAC-08	-1.228E+00	7.232	VAC-08

362	GRID	1000716 ACCELERATION	Y	2.480E-01	7.105	VAC-08	-2.641E-01	7.069	VAC-08
363	GRID	1000716 ACCELERATION	Z	1.753E+00	7.062	VAC-08	-1.620E+00	7.027	VAC-08
367	GRID	1000717 ACCELERATION	Χ	2.013E-01	6.709	VAC-08	-1.382E+00	7.104	VAC-08
368	GRID	1000717 ACCELERATION	Y	2.847E-01	6.939	VAC-08	-3.087E-01	6.903	VAC-08
369	GRID	1000717 ACCELERATION	Z	1.679E+00	6.899	VAC-08	-1.738E+00	6.931	VAC-08
373	GRID	1000718 ACCELERATION	Х	5.742E-01	6.838	VAC-08	-1.702E+00	7.002	VAC-08
374	GRID	1000718 ACCELERATION	Y	2.777E+00	7.287	VAC-08	-2.665E+00	7.274	VAC-08
375	GRID	1000718 ACCELERATION	Z	6.957E+00	6.873	VAC-08	-5.017E+00	6.843	VAC-08
379	GRID	1000719 ACCELERATION	Χ	6.244E-01	6.838	VAC-08	-1.777E+00	7.002	VAC-08
380	GRID	1000719 ACCELERATION	Y	3.659E+00	6.858	VAC-08	-3.705E+00	6.915	VAC-08
381	GRID	1000719 ACCELERATION	Z	8.287E+00	6.863	VAC-08	-5.178E+00	6.897	VAC-08

(END OF TABLE.)

As a check, the results were evaluated for consistency with basic force balances. The vehicle/MPL interface forces did match the vehicle weight and wind moment, and the rigid body vehicle acceleration after release did match the expected acceleration due to steady state thrust forces.

In general, the liftoff events for the load case analyzed seem relatively benign. This is due primarily to the commanded release time being so early in the thrust build-up profile, which prevents any significant elastic energy from being stored, and then released. Addition of overpressure loads and hydrodynamic propellant models would increase the accuracy of the analysis, may increase responses of some components, but such an increase is not expected to be drastic.

## 5.7.3.7 SSME/RS-25 Loads, Dynamics, Life Comments

The SSME/RS-25 today is a human-rated development engine. Its complexity and capability to both operate successfully and survive structurally between extremes demand carefully maintaining a balance between inspections and diagnostics. Care should be taken during evolution of this engine to a revised mission. For this reason VIPA L&D asked MSFC/ER41 Propulsion L&D to provide comments on the proposed SSME/RS-25 usage and development plan based on their extensive experience.

# 5.7.3.7.1 Fleet Leader Logic

Fleet leader logic is the methodology used to certify that the SSME/RS-25 in the present Space Shuttle system is safe for flight. It states that if a component or system can be tested to a certain amount of time with no anomalies, then identical hardware can be flown safely to 50% of that time without incident and without need of any additional inspections. A second fleet leader criteria is used when component hardware has been found cracked due to low cycle fatigue, high cycle fatigue or overload during flight. This documented wavier criteria, is based upon a combination of testing and analysis, that allows the hardware to continue to fly safely while the root cause of the anomaly is investigated. The wavier criteria will usually consider the amount of time on the component, how many engine starts the component has experienced, and the type of crack growth that was determined through inspection. As inspections continue throughout the fleet to determine the severity of the issue, the waiver constrains the component's life to 50% and requires an inspection at 25% of the component's fleet leader life.

Therefore, to maintain human rating of transitioned assets, it would be necessary to carry along or accommodate the associated fleet leader experience for appropriate engine flight hardware. It is important to maintain component hardware pedigree data because all internal engine interface design loads are based upon analyses and environments that evolved with design changes at differing timelines for individual components. Design loads assumptions are unique to each piece part. Fleet leader databases provide a way to normalize the variability of individual hardware experience by careful accounting of hot fire operational ground and flight experiences.

## 5.7.3.7.2 Certification

Certification for the SSME/RS-25 has been accomplished, as with all engine programs, by deciding a requirement for certification that includes a certain number of starts for a defined number of engines. In the case of the SSME/RS-25, the definition of certification was defined most recently for the ATD pumps to be 2 engines with 22 starts. This allowed the program to come out of the certification program with a fleet-leader status that would theoretically qualify the pumps for 1 Green Run and 10 flights, holding to the 50% fleet-leader rule.

The SSME/RS-25 is reusable in the Space Shuttle program. Through inspections of the returned flight hardware this program has had the ability to identify issues that were missed by the ground testing. The inspections have served as valuable tools; identifying the LPOTP nozzle vane cracking issues and HPOTP knife edge seal cracks, which happen at very high frequencies and are of low statistical samplings. As NASA considers moving into the next evolutionary stage of making the SSME/RS-25 expendable and changing the requirements to run possibly at new uncertified power levels or reducing the complexity that comes with reusable engines, the certification process must be re-examined.

An evaluation needs to be considered whether hardware issues can be properly identified by using the current certification series knowing that the hardware will not have the advantage of post-flight inspection as seen during the Space Shuttle Program. It is therefore recommended that developmental testing and inspection continue during the life of the program to enable the expensive long-life version of the SSME/RS-25 to evolve to lower cost units that are more quickly manufactured (as with a channel wall nozzle rather than the current tube-walled nozzles) without sacrificing safety.

# 5.7.3.7.3 System Testing

Engine system and MPS certification testing will be required. Similar to the main propulsion test article (MPTA) testing of the 1970's and 1980's, a hot fire test series of an engine integrated with the new flight vehicle feedlines and thrust structure would provide the most significant risk mitigation data for the largest design uncertainty, which is system loads, by enabling the characterization of a pre-selected set of mission based, self-induced engine operational environments. In addition, this testing will identify engine/MPS fluid structure interaction affected by turbopump backflow coupled with feedline acoustics and components, and it will provide operational statistical variation to validate key design assumptions. Finally, a limited number of test flights of the entire system would be helpful to capture a small number of low frequency data using special low frequency accelerometers to confirm flight design loads envelopes during rollout, ferry, ignition and steady state.

For a rapid preliminary evaluation of the certification status, coupled system frequencies and modes should be validated through modal tests. These tests are comparatively inexpensive, particularly while the hardware is integrated for the purposes of MPTA testing. Modal testing is desirable for several reasons. First, several design and operational requirements are affected by the coupled engine/vehicle dynamics. Guidance and control (GNC) indicates that low frequency modes of the engine system below 35 Hz can affect vehicle flight dynamic stability and controls. In addition, there may be active flight POGO modes in this regime. Second, integration hardware between the SSME/RS-25 and vehicle will likely become candidates for re-certification or redesign from a certification standpoint. This integration hardware provides the structural load path between the vehicle and engine and is especially critical for transient loads and other low frequency dynamic events. This includes structural members such as the thrust vector control gimbals and actuators, thrust structure, and supporting attachment hardware, possibly the engine quadropods. Finally, integrated modal testing would be a cheap way of evaluating the availability and sustainability of remaining resources and evolutionary hardware. Critical transfer functions, design spring rates and damping data to provide design operational loads insight can be verified with an integrated engine/aft vehicle thrust structure modal test.

## 5.7.3.7.4 Component Vibroacoustic Environments

There are a number of areas for which engine self-induced vibroacoustic environment changes may affect component and system loads. Components which are dominated by low frequency behavior may be more affected by the mission change, that is, the addition of new transient events. Additionally, changes in the power level operation and duration may affect some fatigue

design life calculations of a few high time component units (for example, not much operational engine system experience exists for 109-111% power level). Associated vibroacoustics environment updates may require some statistical evaluation to extend fleet leader databases to account for low sample populations and operational variability. In addition to a possible revision of operating environments including fluctuating pressures, it would be a good idea to revisit range safety vibroacoustic environment requirements for potential impact due to operational changes. Block II engine systems level change accepted strain gage histories as a primary method of design loads validation. Some consideration from the higher level Block II turbopumps environments is the potential benefit (credit) to an overall system level recertification. During this evolution work of the engine to a revised mission, original design requirements may need to be revisited to determine potential benefits which, if positive cost trades result, may substantially improve operational reliability in areas such as production automation, materials, in-process production inspections, mechanisms, and EEE parts.

Components currently in use are sensitive to changes in their operating environments. It would be productive to review turbomachinery component analyses for possible changes in resonance conditions. In addition to these assessments, components that are driven by the engine transients such as the nozzle would require assessment of the strain gage history. Vibroacoustic and unsteady pressure environments should be updated for the expected new mission profiles. Valves, actuators, sensors and other electronics may require new qualification specifications.

## 5.7.3.8 SRB Acoustic Mode Thrust Oscillations

SRB acoustic modes, similar to organ pipes, create thrust oscillations due to changes in the pressure at the nozzle. For the Shuttle 4 segment RSRM this is around 15 Hz. The 5 segment SRM mode is around 11 Hz as demonstrated by the Engineering Test Motor (ETM3) test. This phenomenon is stationary and does not couple like POGO, however it can excite coincident structural modes. Avoiding these frequencies is the best approach during design of structure. The first axial modes from the Human Rating study and the current PODV (6-9 Hz and 8-11 Hz) will not resonate with the 15 Hz RSRM mode. There may be some issues with the PODV 2<sup>nd</sup> axial mode passing through 15 Hz from 13-16.4 Hz. Similarly the 5 segment axial mode may couple while passing through 11 Hz from 9-13Hz.

These possible issues are not deemed significant to the overall system for several reasons. First, the structure should only pass through these oscillations briefly and not cause sustained resonance. As mentioned, this phenomenon doesn't "lock-in" like a POGO mode. Second, some local design effort such as crew seat or critical component isolation can alleviate problems without impacting system weight significantly. Finally, ATK believes that the thrust oscillations can be substantially reduced by joint chamfering and inhibitor changes. The "odd" segment number of the 5 segment booster also alleviates this effect because the acoustic anti-node does not coincide with a case joint. It is the inhibitors at these joints that trigger this effect much like blowing across a reed.

# 5.7.4 Significant Findings

Prior to and during this 60-Day study, VIPA L&D performed various analyses and assessments on three similar concepts consisting of a Shuttle SRB and a liquid propulsion upper stage to carry a spacecraft to orbit. These analyses, taken all together form a sound basis for generating confidence in the general concept.

Structural frequencies for the different concepts were estimated. The results indicate some low frequencies, particularly when constrained to the MLP indicating the need for a vehicle support

incorporated on the MLP; particularly rollout. The lowest free-free or flight frequencies are around 1 Hz which is comparable to previous launch vehicles.

SRB acoustic modes, which generate thrust oscillations, do not appear to be significant design drivers. There may be instances of passing through structural frequencies that resonate with the oscillations. However these resonances should be short lived and can be treated as design problems rather than concept drivers.

An initial sizing cycle was performed on the upper stage using loads from an early cycle. The results indicated that the preliminary weights for primary structure from Advanced Concepts are reasonable.

Pre-launch loads fall near or below the SRB design moments. The longer vehicles are beginning to encroach on the design loads. All configurations indicate SRB hold down post loads are within capability indicators.

Ascent loads for the three configurations are currently within or just encroaching on the design of the existing SRB. An initial reconstruction of the SRB casing capability based on its steel construction indicates the actual SRB capability is higher than the current design loads.

A study of an SRB actuator failure was conducted. It indicated a time period of roughly 0.6 seconds before the vehicle  $3\sigma$  design loads were realized for the SRB structure. The Spacecraft and upper stage loads reached design levels much sooner. There is some debate regarding the credibility of this failure. These cases will necessitate further study as crew safety and abort scenarios are considered.

A preliminary liftoff analysis of the 5 segment configuration indicates that the ascent case is the primary load driver for the majority of the upper stage. It also indicates that liftoff dynamics may be the principle loads driver for the spacecraft and the extreme forward end of the upper stage. The liftoff events for the load case analyzed seem relatively benign compared to Shuttle.

MSFC/ER41 Propulsion L&D has reviewed the proposed SSME/RS-25 usage and development plan based on their extensive experience in structural dynamics of propulsion systems. Integrating this engine system into a new vehicle with new flight requirements will require recertification for changes to both operational environment (new power levels and altitude starts) and due to new feedline and structural system changes. Further certification will be needed as the SSME/RS-25 evolves into a more easily manufactured, cost effective, expendable engine and the changes associated with this path of improvement. Additional flight instrumentation will likely be required to offset the loss of post-flight hardware inspections. Also, due to the lack of flight hardware inspections, a periodic testing program needs to be developed to address statistically anomalies that are not captured during the initial certification testing.

### 5.7.5 References

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- J. Peck, J. Brunty, "X33 Transient Liftoff Analysis", 41<sup>st</sup> AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, April 3-6 2000, Atlanta, GA, AIAA-2000-1585
- 4) Boeing Tech. Memo CINS-TM-04-002," Generic Space Shuttle Vehicle Structural Dynamic Math Models and Lift-off Forcing Functions (CM1.0A55)," dated 3/4/04
- 5) MSFC VIPA Report, "Clean Sheet Heavy Lift Vehicle Concepts TB & TC-SRB Engine Out Stages 1 & 2 Integrated Vehicle Baseline Characterization," August 31, 2004.
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