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ADVANCED TRANSPORTATION SYSTEM STUDY

Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO

FINAL REPORT (DR-4)

Contract NAS8-39207

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Submitted by:

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Forward

This report is submitted in compliance with DR-4 of Contract NAS8-39207, Advanced Transportation System Studies for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center. The report describes the results of Rockwell International's work for the analysis of Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO during the Basic Contract and the Option 1 Contract period of performance (February 15, 1992 through December 15, 1993).

This report is organized in three volumes; an Executive Summary, a Final Report, and a Cost Estimates Report.

The Executive Summary is a condensation of the study's major findings and a summary of results for the several study activities. The findings and results are current with the study progress as of December 10, 1993.

The Final Report volume is an in-depth description of work performed during the study, with accompanying illustrations of briefing charts and other documents which were generated during the course of the study. This volume is organized by subject matter and includes an appendix with research reports of detailed analyses on selected special topics. Sufficient data is presented in this volume to reveal the depth of work performed and to provide data which supports the findings presented in the Executive Summary.

The Cost Estimates Report is a compilation of the Work Breakdown Structure and cost estimating techniques which were used to evaluate the several booster concepts during the course of the study. A summary of data used and generated during the evaluation of each booster type (or family of boosters) is provided. The data is organized by booster types which represent unique cost estimating conditions, such as the reusable Space Shuttle, the proposed low cost NLS family, existing expendable launch vehicles, and the Russian (C.I.S.) launch vehicles.

ABSTRACT

The purpose of the Advanced Transportation System Study (ATSS) Task Area 1 study effort is to examine manned launch vehicle booster concepts and two-way cargo transfer and return vehicle concepts to determine which of the many proposed concepts best meets NASA's needs for two-way transportation to low Earth orbit. The study identified specific configurations of the normally unmanned, expendable launch vehicles (such as the National Launch System family) necessary to fly manned payloads. These launch vehicle configurations were then analyzed to determine the integrated booster/spacecraft performance, operations, reliability, and cost characteristics for the payload delivery and return mission. Design impacts to the expendable launch vehicles which would be required to perform the manned payload delivery mission were also identified. These impacts included the implications of applying NASA's man-rating requirements, as well as any mission or payload unique impacts.

The booster concepts evaluated included the National Launch System (NLS) family of expendable vehicles and several variations of the NLS reference configurations to deliver larger manned payload concepts (such as the Crew Logistics Vehicle (CLV) proposed by NASA JSC). Advanced, clean sheet concepts such as an F-1A engine derived liquid rocket booster (LRB), the Single-Stage-to-Orbit rocket, and a NASP-derived aerospace plane were also included in the study effort. Existing expendable launch vehicles such as the Titan IV, Ariane V, Energia, and Proton were also examined.

Although several manned payload concepts were considered in the analyses, the reference manned payload was the NASA Langley Research Center's HL-20 version of the Personnel Launch System. A scaled up version of the PLS for combined crew/cargo delivery capability, the HL-42 configuration, was also included in the analyses of CTRV booster concepts.

In addition to strictly manned payloads, two-way cargo transportation systems (Cargo Transfer & Return Vehicles) were also examined. The study provided detailed design and analysis of the performance, reliability, and operations of these concepts. The study analyzed these concepts as unique systems and also analyzed several combined CTRV/booster configurations as integrated launch systems (such as for launch abort analyses). Included in the set of CTRV concepts analyzed were the Medium CTRV, the Integral CTRV (in both a pressurized and unpressurized configuration), the Winged CTRV, and an attached cargo carrier for the PLS system known as the PLS Caboose.

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1.0 Introduction

The ATSS Task Area 1 team has conducted studies for the design, performance, and evaluation of a number of expendable booster concepts currently being considered for the launch of reusable (manned or two-way cargo) spacecraft. These analyses were performed to determine which of the many proposed booster concepts best meets NASA's needs for two-way transportation to low Earth orbit. Detailed design and analysis of the two-way cargo spacecraft were also performed under the study. Analysis of the integrated configurations of these boosters and spacecraft revealed an improved understanding of the strengths and limitations of each element. Integrated booster/spacecraft analysis were performed for launch vehicle performance and controllability, reliability, cost, launch rate capability and facility utilization, and launch abort. Additional analyses of the launch vehicle ground processing activities were also performed to identify means of improving the operability of any launch vehicle concept, new or old. The product of these studies is a better understanding of the roles which both the booster and the spacecraft play in achieving improved access to space.

Manned Booster Studies

The study identified specific configurations of the normally unmanned, expendable launch vehicles (such as the National Launch System family) necessary to fly manned payloads. These launch vehicle configurations were then analyzed to determine their performance, operations, reliability, and cost characteristics for the manned payload delivery mission. Design impacts to the expendable launch vehicles which would be required to perform the manned payload delivery mission were also identified. These impacts included the implications of applying NASA's man-rating requirements, as well as any mission or payload unique impacts.

Booster concepts evaluated included the National Launch System (NLS) family of expendable vehicles, several variations of the NLS reference configurations, and the ESA Ariane V. Advanced, clean sheet concepts such as an F-1A engine-derived Liquid Rocket Bosoter (LRB), the Single-Stage-to-Orbit rocket, and a NASP-derived aerospace plane were also included in comparisons of the several candidate booster configurations. Existing expendable launch vehicles such as the Titan IV and the Russian Energia and Proton launch vehicles were also compared to the proposed new booster designs.

Although several manned payload concepts were considered in the analyses, the reference manned payload was the NASA Langley Research Center's HL-20 version of the Personnel Launch System (PLS). Other concepts such as the Crew Logistics Vehicle (CLV) proposed by NASA JSC and a scaled-up version of the PLS for combined crew/cargo delivery capability (the HL-42 configuration) were also included in the analyses. These concepts could be used in either the manned mode or an unmanned mode for delivery and return of cargo (such as for the Space Station cargo resupply mission).

Cargo Transfer & Return Vehicle (CTRV) Studies

A wide range of concepts for the delivery and return of cargo payloads to low Earth orbit (LEO) were designed and analyzed during the study. These concepts are generally referred to as Cargo Transfer and Return Vehicles (CTRV). These concepts include vehicles which deliver crew or cargo separately as well as those vehicles which deliver combined crew and cargo payloads. Combined crew/cargo delivery vehicles evaluated in the study included LaRC's scaled-up version of the Personnel Launch System (HL-42) and JSC's Crew Logistics Vehicle (CLV). Variations of all three vehicle concepts were created during the study to provide varying payload delivery and return capabilities. The CTRV, PLS, and CLV concepts, combined with appropriate launch vehicle(s), formed the architectural framework of NASA's Access to Space, Option 2 study. Identifying competitive design configurations of the CTRV to support the Access to Space study objectives was a principle activity of the study.

Launch Abort Studies

Trajectory analyses were performed to determine the ability of a variety of expendable boosters to provide a mission abort capability for the HL-42 and CLV-P crew/cargo vehicles during the ascent mission phase. The analysis was performed for several Access to Space study boosters (Boosters 2A', 2C, and 2D for the HL-42, and Booster 2B for the CLV-P). Specific design characteristics of the booster concepts were provided by NASA MSFC. The HL-42 and CLV-P design data were provided by NASA LaRC and JSC, respectively. The abort modes considered in the analysis included; Return To Launch Site (RTLS), Trans-Atlantic Abort (TAL), Engine Out (EO), Abort To Orbit (ATO), and Abort Once Around (AOA). A North America Landing (NAL) abort mode was added for the Booster 2D concept to compensate for this two-stage (series burn) booster design.

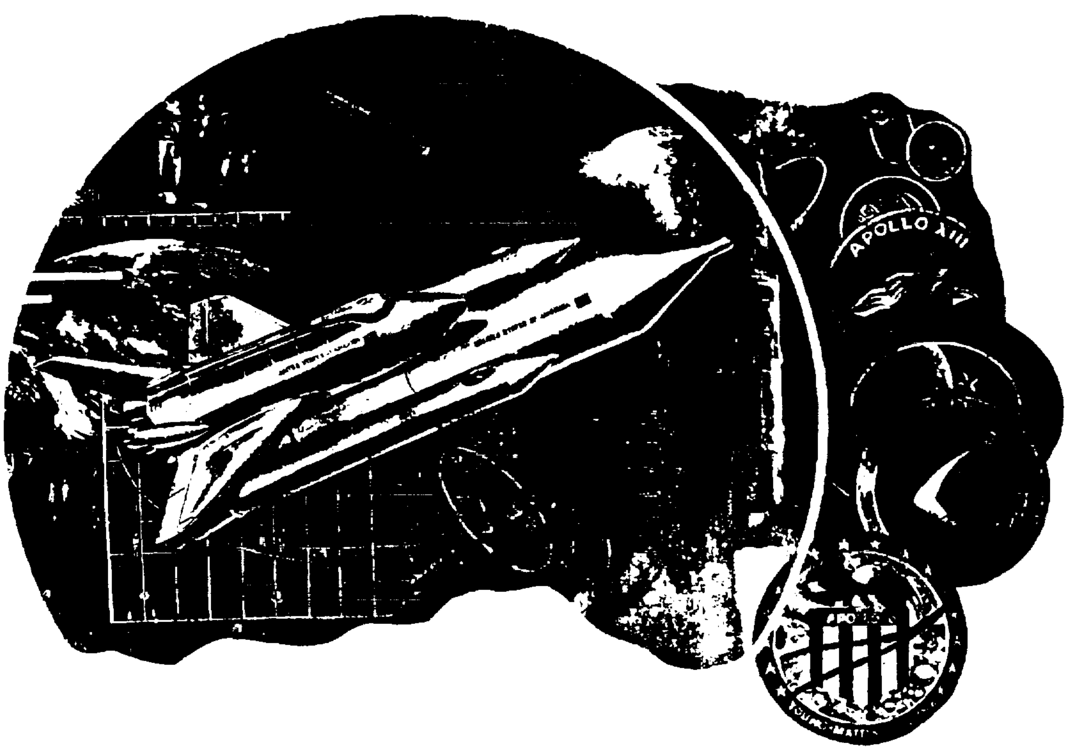
SSTO MPS Operability Studies

An investigation into means of achieving high operability in any new SSTO concept was accomplished by evaluating an SSTO main propulsion system's flight operations, ground operations, and design configuration characteristics. All of these aspects of a system design interact to produce a launch vehicle's net reliability and maintenance performance. Methods used for improving aircraft operability were adapted for the ATSS study to analyze the operability characteristics of SSTO concepts defined by NASA in their recently completed Access to Space study, Option 3. The method used simulation models to provide detailed assessment of the SSTO propulsion system components and checkout activities and also provide a system-level simulation of the SSTO launch rate capability, facility requirements, and resource utilization needs. The simulations included component level reliability and maintainability data as determined from actual Space Shuttle MPS processing history. The MPS serves as a useful benchmark for comparing the operability of the many competing SSTO concepts.

The investigation included the evaluation of optimum SSTO main engine operating techniques for maximizing both engine reliability and life, while also providing adequate abort coverage. Variations in engine throttle profiles and shutdown sequences were performed to find the minimum engine operating times, the minimum engine operating time at 100% throttle level, and the maximum engine out abort capability. Additionally, a design study for an SSTO MPS aft fuselage was used to rigorously apply the MPS design groundrules which were identified in the Operationally Efficient Propulsion System Study (OEPSS) by NASA KSC and Rocketdyne.

1.1 Significant Achievements

The ATSS Task Area 1 study has examined a wide range of launch systems for two-way space transportation payloads. Launch vehicles for future NASA spacecraft such as the PLS and CTRV concepts have been analyzed to determine which boosters best meet NASA objectives. Impacts to these boosters to perform manned payload missions have also been identified, including the implications of NASA's man-rating requirements. Design and analysis of several CTRV concepts which would compliment the PLS system were also provided to support the NASA Access to Space (Option 2) study. The ability to provide launch abort coverage for the return of reusable systems was also examined, for both expendable launch vehicles and for fully reusable SSTO concepts. The significant achievements and findings reached during the conduct of these studies should be of value to NASA and contractor engineers as plans for the next generation of manned launch vehicle concepts are developed.

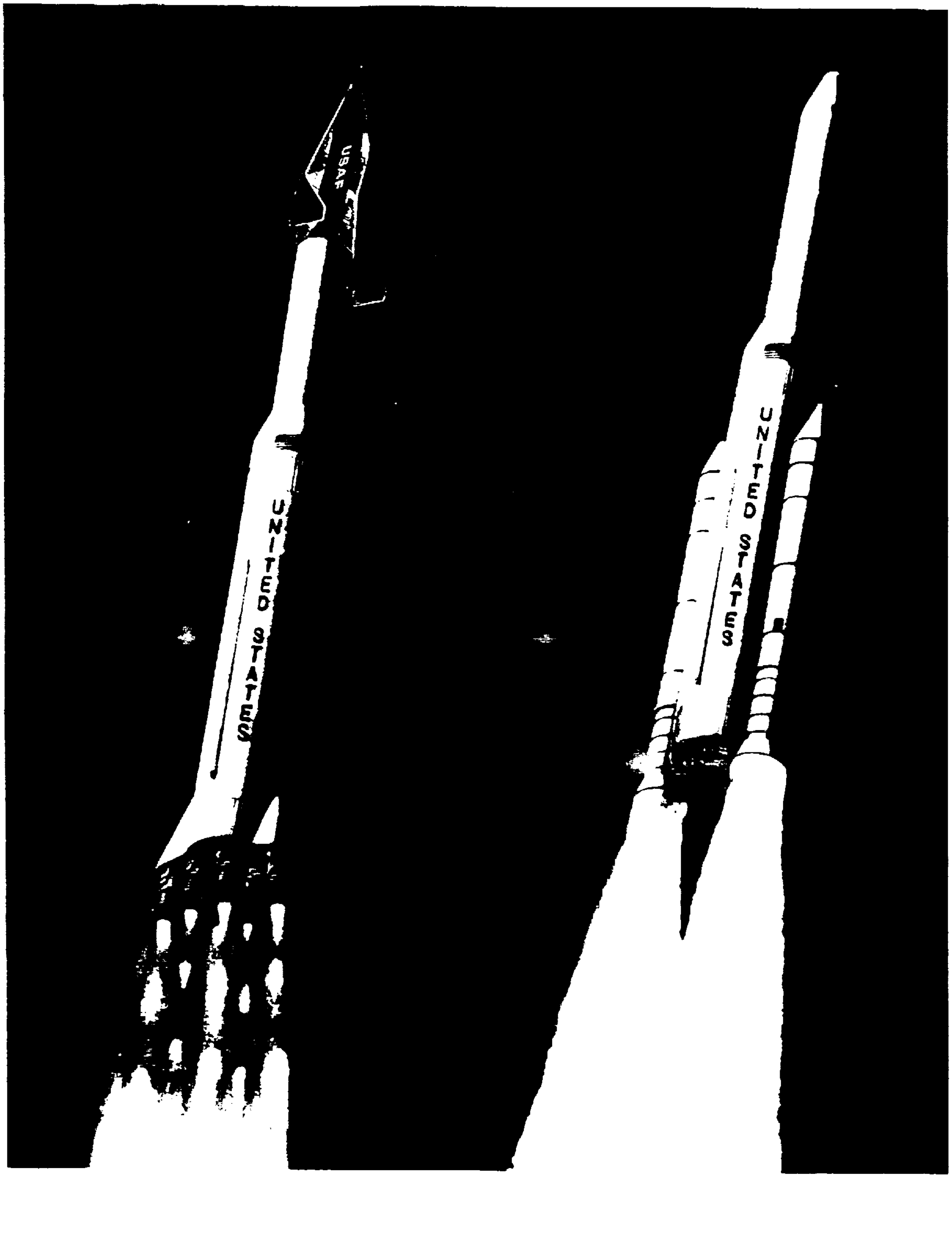


1.1.1 Manned Booster Studies

The ATSS contract has studied a number of booster concepts currently being considered for the launch of reusable (manned or two-way cargo) spacecraft. Analyses were performed to determine which of the many proposed booster concepts best meets NASA's needs for two-way transportation to low Earth orbit. The study identified specific configurations of the normally unmanned, expendable launch vehicles (such as the National Launch System family) necessary to fly manned payloads. Design impacts to the expendable launch vehicles which would be required to perform the manned payload delivery mission were also identified. These impacts included the implications of applying NASA's man-rating requirements, as well as any mission or payload unique impacts. The booster concepts evaluated included the National Launch System (NLS) family of expendable vehicles, the ESA Ariane V, the Titan IV, and the Russian Energia and Proton launch vehicles. Advanced, clean sheet concepts such as an F-1A engine-derived booster (LRB), the Single-Stage-to-Orbit rocket, and a NASP-derived aerospace plane were also included in comparisons of the several candidate booster configurations.

A review of current and past NASA man-rating requirements was performed and a compilation of these requirements documented in a single reference report. The requirements were summarized and translated into functional design requirements which would have to be added to an otherwise non-manrated launch vehicle. The study also identified design impacts to an expendable launch vehicle which are associated with the induced aerodynamic loads produced by PLS (winged) types of payloads. Structural (weight) impacts for these increased aerodynamic loads were calculated for the NLS-2 booster.

The study evaluated several manned booster concepts and found that the NLS-2 family of launch vehicles is an excellent booster for NASA's next manned spacecraft concepts (PLS or CLV systems). Although somewhat overpowered (excess lift capability) for these payloads, the NLS-2 provides a highly reliable and safe launch system and can be operated in a mixed fleet mode with the current Space Shuttle system during a manned transportation system transition period. The variable cost per launch estimates for all of these PLS/ELV systems was found to be greater than the current demonstrated cost per flight of the Space Shuttle system. Significant reductions in fixed costs over the current Space Shuttle (by nearly an order of magnitude) will be required by these new systems in order to be cost effective.


USAF
UNITED STATES
UNITED STATES

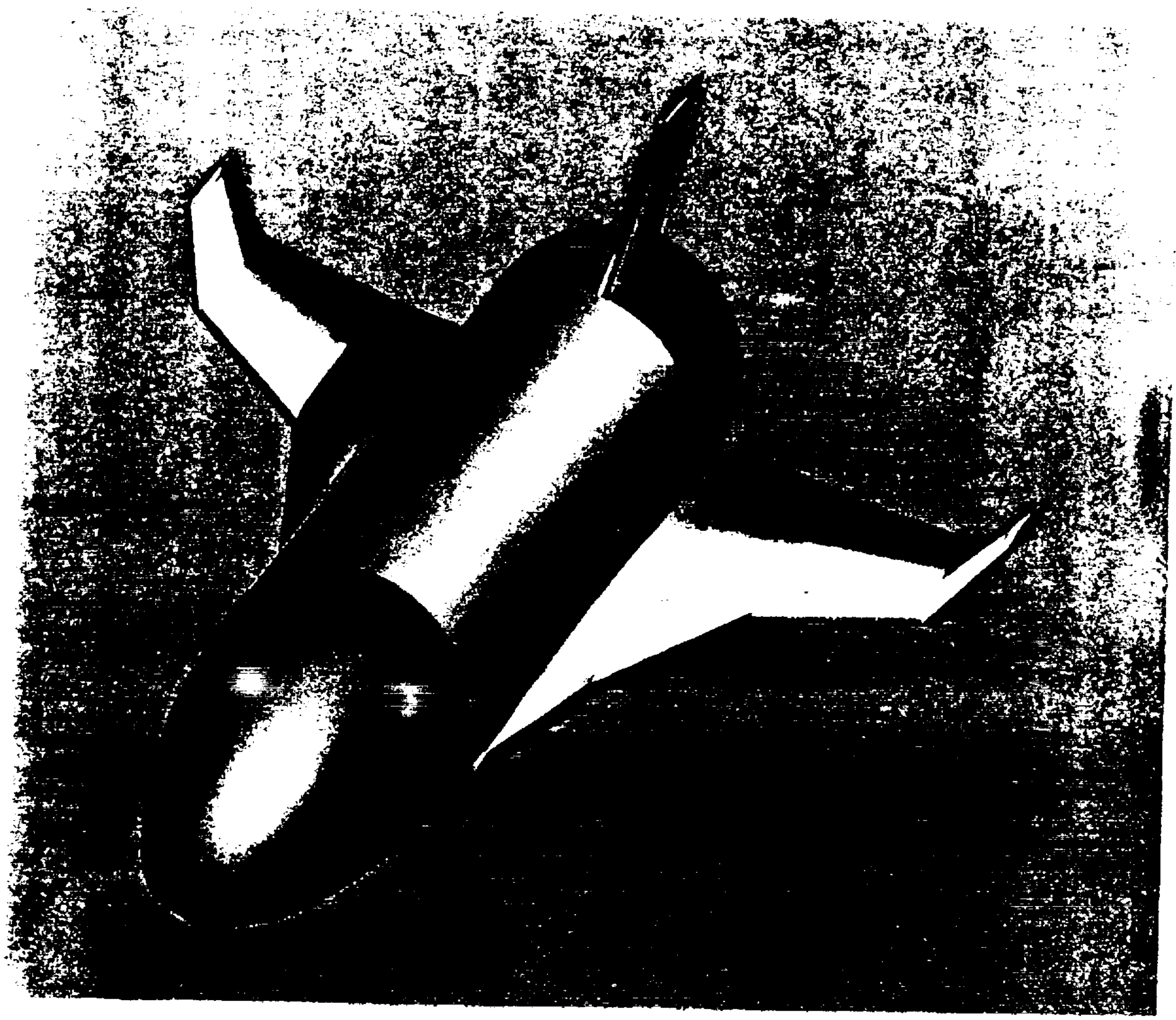
1.1.2 Cargo Transfer and Return Vehicle Studies

The study examined a number of Cargo Transfer and Return Vehicle (CTRV) concepts in support of the NASA Access to Space (Option 2) study. Concepts ranging from small ballistic re-entry systems to large payload capacity, winged systems were designed and analyzed. These CTRV designs were an integral part of NASA's Access to Space Option 2 evaluation of candidate launch system architectures. Although the design concepts were not selected in the final architecture definition, they provided highly competitive options to the HL-42 and CLV concepts.

The study provided improved design definitions for several CTRV concepts, including the Medium CTRV, the Integral CTRV, and Winged CTRV concepts. A detailed analysis of the Medium CTRV concept was performed, including design layouts and finite element stress analysis of the structural design. Aerodynamic, aeroheating, and trajectory analyses were performed for all CTRV designs, from the hypersonic re-entry to touchdown. Parachute landing system design and analysis was conducted for the non-winged CTRV concepts. Reliability, maintainability, and ground operations analyses were also provided for each of the CTRV concepts, including launch processing sequences, ground processing timelines, maintenance and spares requirements, and facility/resource utilization. Significant differences were found in ground operations requirements between the Medium and the Integral CTRV concepts.

CTRV Impacts to Space Station

The study found that a CTRV-based logistics system would cause both flight and ground operations changes to the Space Station program. The extent of the operations impacts will be determined by the CTRV concept selected - Medium, Winged, or Integral CTRV. All CTRV concepts will result in a greater amount of time spent by the Space Station crew in logistics operations (rendezvous and docking, transfer of payloads). The CTRV concepts also provide a reduced capability for the transfer (delivery or return) of large Space Station ORUs. A CTRV system would also cause changes to Space Station payload integration operations and facilities at KSC. Impacts to other KSC facilities related to launch and/or payload processing of Space Station elements and cargo would also be expected.



1.1.3 Launch Abort Studies

The study evaluated several launch vehicle concepts to determine the level of launch abort capabilities they could provide for two-way transportation systems (high value payloads such as reusable CTRV, CLV, and HL-42 cargo carriers, astronauts (when aboard), and even the reusable booster itself).

Abort Coverage Capability

The study provided a comparison of the launch abort capability of several manned booster concepts, including both expendable and fully reusable systems. Of all the expendable booster concepts studied, only one provided abort coverage during the entire launch trajectory. All other expendable booster concepts exposed the reusable spacecraft/crew to a water ditching option for large portions (40% to 60%) of the launch trajectory. Surprisingly, even some fully reusable (SSTO) concepts provided insufficient abort coverage. The tri-propellant propulsion SSTO design based on three Russian RD-701 engines provided only 90% abort coverage of its launch trajectory.

Launch Vehicle Design for Abort Coverage

The study found that for all of these expendable boosters, the second stage (or core stage) design was the key to providing abort coverage. Full coverage usually was available during all of the first stage flight, but rarely during second stage flight. This is principally due to the number of engines in the vehicle's second stage design. Two stage boosters (serial burn) have additional abort coverage issues related to their second stage engine ignition. For SSTO concepts, the number of engines and their thrust levels were the key abort coverage factor.


United States

1.1.4 SSTO MPS Operability Studies

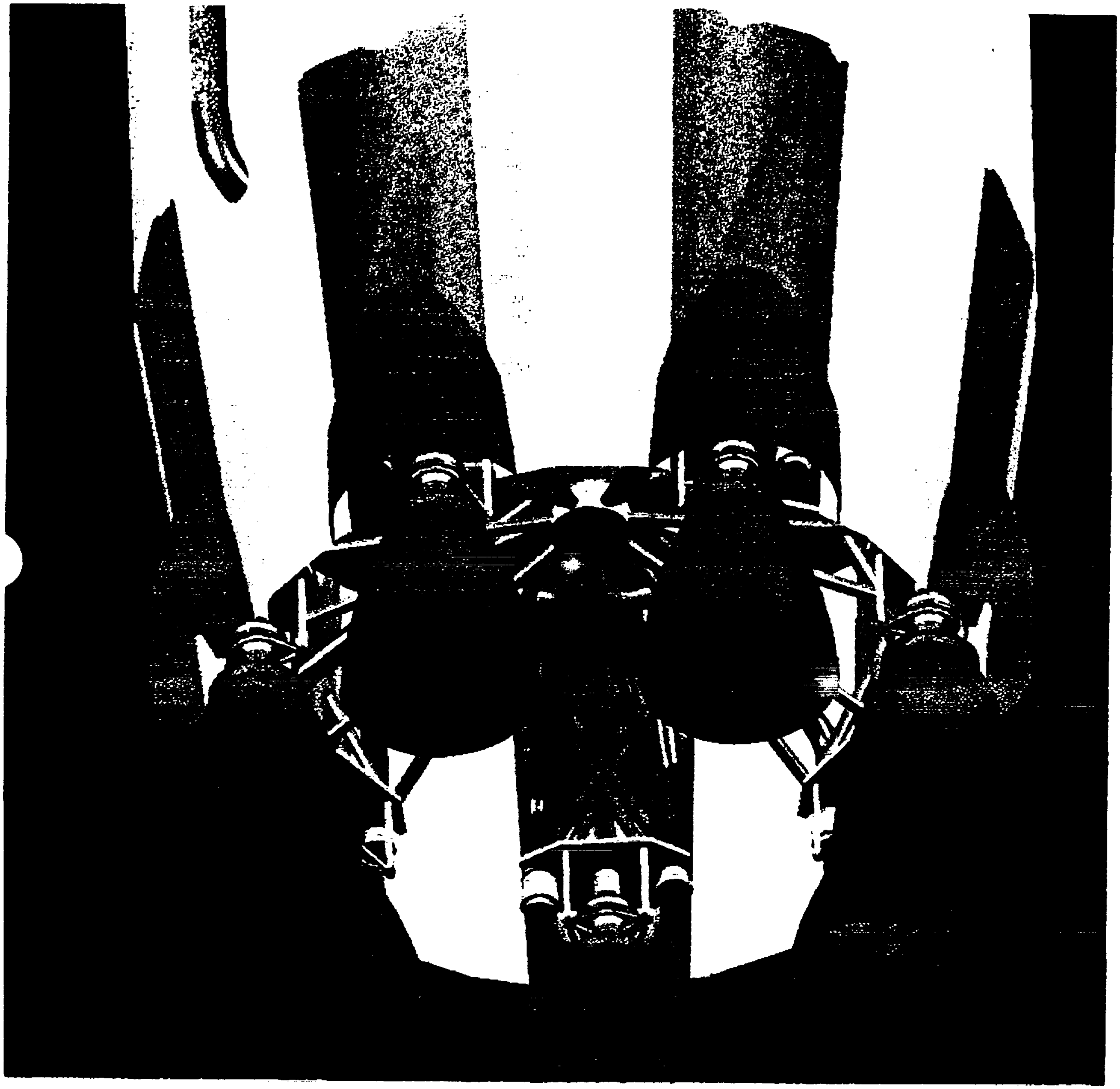
The study examined the operability of Single Stage To Orbit concepts to determine the relative impacts which new technologies would have on the current state-of-the-art for reusable launch vehicles. The Space Shuttle's current level of operability does not meet SSTO objectives, but it does provide the benchmark to measure how much improvement is needed. Detailed operations simulations at the Main Propulsion System (MPS) component level were performed to determine the integrated effects that flight hardware and test technology improvements would have on SSTO operability.

Achieving High Flight Rates

The study found that achieving high flight rates with SSTO concepts is readily achievable with existing hardware if the vehicle ground processing schedules can approach SSTO program goals (subsystem checkout times reduced to about 28 shifts (14 workdays) and the launch pad time kept to 2 shifts). The Space Shuttle MPS component current reliability and maintenance levels are sufficient to satisfy the SSTO schedule objectives (but not the cost objectives). The key to a highly operable SSTO from a flight rate standpoint is ground test technologies.

Achieving Low Operations Costs

The study found that significantly reduced flight hardware failure rates are required to reduce the overall maintenance costs of an SSTO concept. Shorter subsystem checkout times are the most significant factor for achieving reduced net hardware failure rates. Improved hardware reliability and maintainability design will also help reduce the SSTO maintenance demands. Significantly reduced component removal/retest times (from days to hours), and to a lesser extent increased mean time between failure (+ 50%), will greatly reduce the maintenance burden on a reusable SSTO. The key to a highly operable SSTO from a cost standpoint is lower maintenance levels. Highly accessible MPS components can have an appreciable effect on improving maintenance times. Design layouts with as many as seven SSME engines in an SSTO aft fuselage region were achieved without having to locate one engine in the middle of the compartment.



1.2 Summary of Results

1.2.1 Manned Booster Studies

The ATSS contract has studied the performance of a number of expendable booster concepts currently being considered for the launch of reusable (manned or two-way cargo) spacecraft. Analyses were performed to determine which of the many proposed booster concepts best meets NASA's needs for two-way transportation to low Earth orbit. The study identified specific configurations of the normally unmanned, expendable launch vehicles (such as the National Launch System family) necessary to fly manned payloads. These launch vehicle configurations were then analyzed to determine their performance, operations, reliability, and cost characteristics for the manned payload delivery mission. Design impacts to the expendable launch vehicles which would be required to perform the manned payload delivery mission were also identified. These impacts included the implications of applying NASA's man-rating requirements, as well as any mission or payload unique impacts.

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Although several manned payload concepts were considered in the analyses, the reference manned payload was the NASA Langley Research Center's HL-20 version of the Personnel Launch System (PLS). Other concepts such as the Crew Logistics Vehicle (CLV) proposed by NASA JSC and a small cargo carrier to be launched with the PLS for combined crew/cargo delivery capability (the PLS Caboose configuration) were also included in the comparisons of booster configurations. The reference mission used in the analyses was the Space Station crew/cargo resupply mission.

1.2.1.1 Effects of Manned Payloads on ELVs

The impacts of launching a manned payload on an otherwise unmanned launch vehicle were analyzed from both a man-rating requirements aspect and from a flight performance aspect. In addition to the expected high reliability parts, redundancy levels, and traceability requirements, the analysis identified additional functional design requirements which the booster will have to perform.

Man-rating Requirements

Analysis of NASA's man-rating requirements from current and previous man-rated launch vehicle programs were condensed down to a small set of functional requirements which any new booster would have to provide. Essentially, these functions indicated that the booster will have to provide a direct two-way communications link between the booster and the manned payload (crew). The booster must monitor and provide status of its critical systems to the crew and must also permit manual crew override of certain critical booster functions. The level of detail to be provided in the status indication is a subject of great debate, as is the level of active crew intervention in booster functions. An approach was suggested in which the launch vehicle is only required to achieve a hierarchy of primary and secondary (abort) mission MECO targets during a launch. The only booster to crew communication would be notification of which target (if any) was achievable, and if there was an immediate safety hazard.

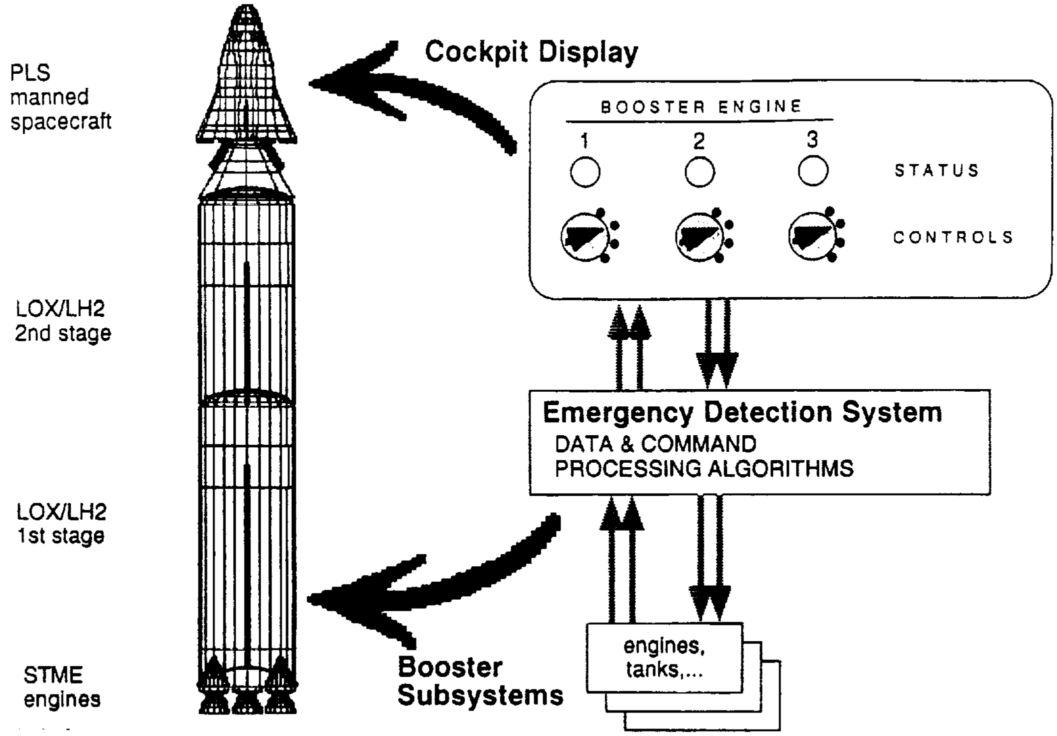


Figure 1.2-1 Booster to Spacecraft Interfaces

Emergency Detection System

A. **Monitor critical systems for out-of-limits conditions or failures**
B. **Make decisions for any corrective actions required and issue corrective action commands as required**
- **to protect crew safety**
- **to accomplish primary mission**

→ **C.** **If decision reached that primary mission cannot be met**
- **maintain vehicle within crew safetylimits**
- **select alternate mission and issue required commands**
- **notify crew of decision reached**

→ **D.** **If decision reached that alternate missions cannot be met**
- **maintain vehicle within crew safetylimits**
- **notify crew of decision reached**

→ **E.** **If decision reached that crew safety cannot be maintained**
- **notify crew of decision reached**
- **issue automatic crew escape commands**

→ = crew input capability (manual override)

The objective of the EDS is to detect malfunctions and provide commands to maintain the maximum mission completion capability while always maintaining the vehicle within crew safety limits

Figure 1.2-2 Booster Emergency Detection System Functions

Aerodynamic Launch Loads

Aerodynamic analysis of several manned spacecraft concepts found that such winged or lifting body payloads will cause significantly higher bending moments in the launch vehicle structure. A detailed stress analysis of the NLS-2 booster was performed to determine the impact of these higher loads for both on-pad and in-flight loads conditions. The analysis indicated that a structural beef-up will be required in certain sections, adding 4,000 lbs to the launch vehicle weight. Structural beef-up of the Titan IV and the Ariane V launch vehicles would also be expected to perform the PLS launch mission. The analysis also indicated that these winged payloads are not expected to exceed booster aerodynamic control capabilities during either liftoff or maximum dynamic pressure conditions.

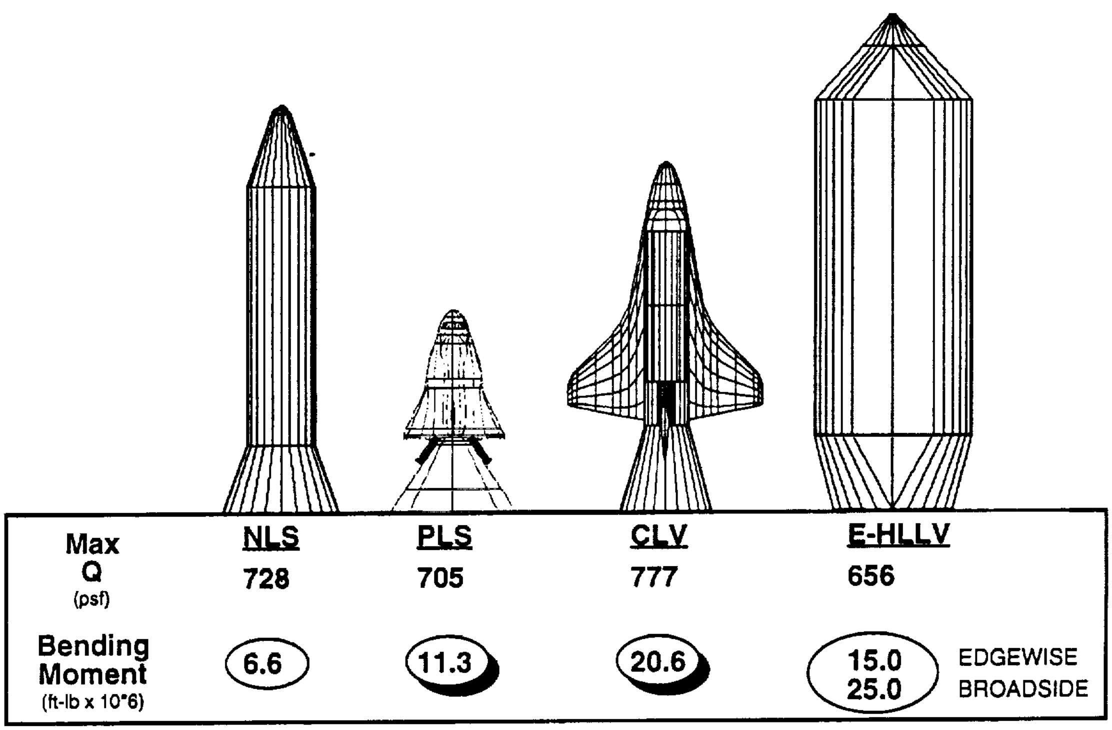


Figure 1.2-3 Aerodynamic Loads of some Manned Payloads

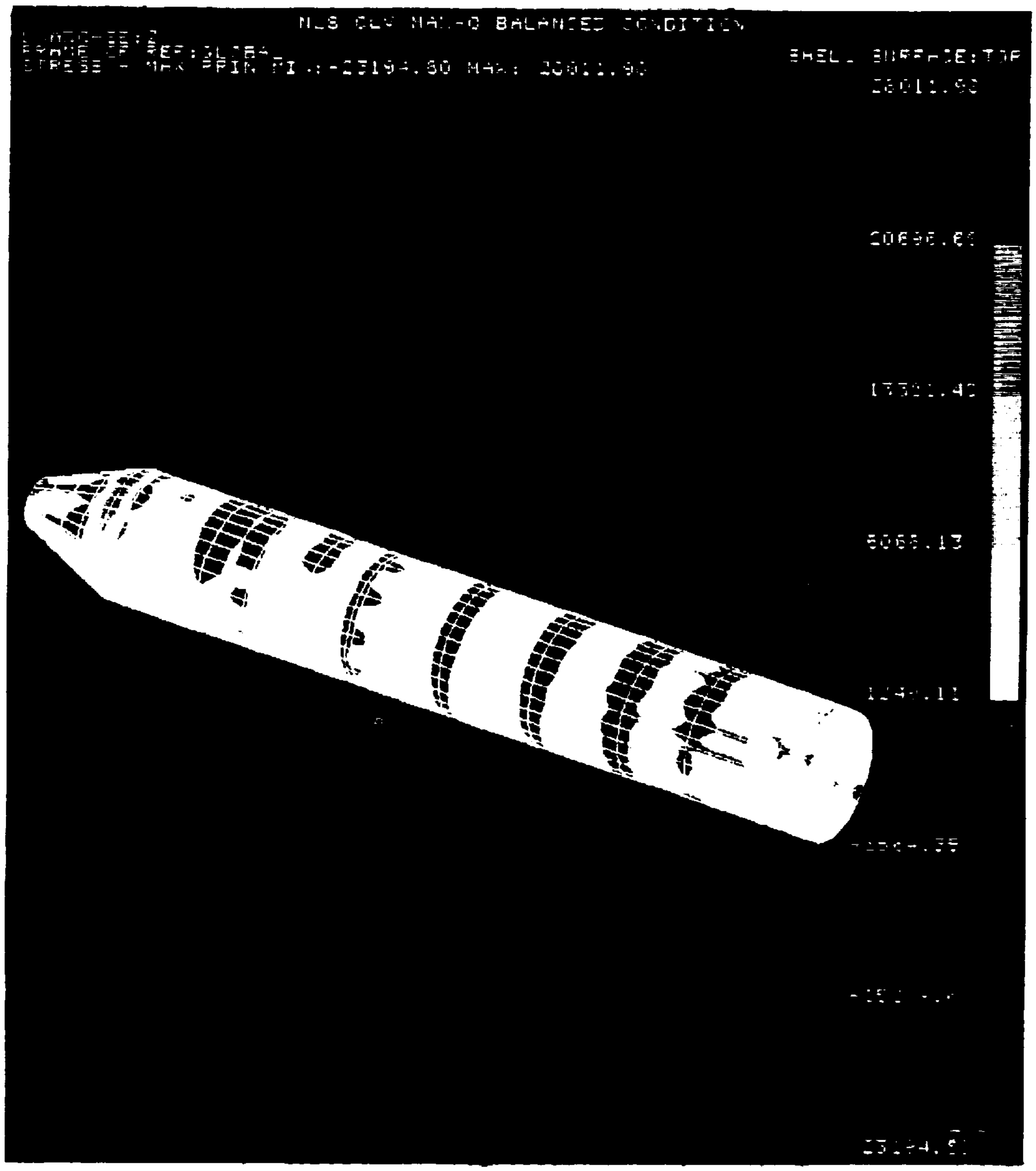


Figure 1.2-4 Booster Stresses at Max Q

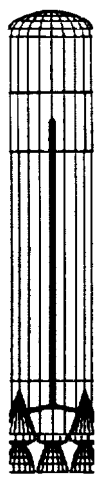
1.2.1.2 Comparison of Manned Booster Concepts

A number of existing and planned launch vehicles were analyzed to determine their suitability for launching either the PLS or the CLV manned payload concepts. The boosters were analyzed from a performance, cost, reliability, and launch processing perspective. The significant findings from these analyses are summarized below.

Performance Analyses

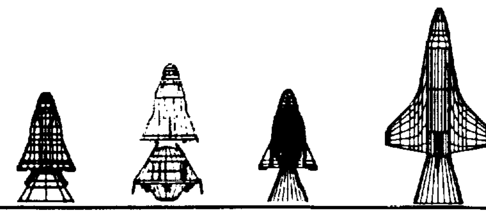
The NLS-2 1.5 Stage launch vehicle was found to be over-sized for launch of the PLS spacecraft and undersized for the CLV concept. The addition of approximately 15,000 to 20,000 pounds of mission equipment could be added to the PLS mission manifest to provide a better payload match for the NLS-2 booster. Use of the NLS booster to launch the CLV concept will require a two-stage version of the NLS-2 launch vehicle. Resizing of the ET-core based first stage and/or the STME thrust level is recommended to reduce high acceleration levels experienced in first stage flight of these configurations. A booster based on a Liquid Rocket Booster (LRB) concept utilizing a single F-1A engine in the first stage and a single J-2S engine for the second stage is a slightly better performance match for the PLS payload than the current NLS-2 booster configurations.

Launch of the PLS on existing expendable launch vehicles such as the Titan IV and the new Ariane V were found to generate high dynamic pressures. The effects of the PLS wings, as well as the implications of satisfying NASA's man-rating requirements, will have considerable design impacts on these boosters. Launch of the PLS on either of these boosters is not recommended.



NLS-2
1.5 STAGE (6/2)

Payload Options



Booster Description

	Booster	Core
Propellant Type	LOX/LH2	LOX/LH2
Engines	STME (4)	STME (2)
Thrust (lbf)	650 K	650 K
Isp (sec)	428.5	428.5
Engine out capab.	Yes	Yes
GLOW (lb)	78,312	1,877,641
Dry Weight (lb)	70,700	127,550
Length (ft)	34	179
Diameter (ft)	27.5	27.5

Figure 1.2-5 NLS-2 1.5 Stage (6/2) Description

Booster Maximum Performance (to 15 x 220 nmi.)

Maximum Dynamic Pressure (psf)	731.75
Maximum Acceleration (g's)	4.00
Booster Separation:	
Time (sec)	161.50
Altitude (ft)	245903
Relative Velocity (fps)	9031.30
MECO:	
Time (sec)	338.29
Weight (lbs)	219,055
Excess Propellant (lbs)	10,477
Payload to Transfer Orbit (lbs)	**62,696**

Annual Flight Rate = 10

Launch Cost = $100 M

Booster Payloads to SSF Orbit

- Maximum payload to SSF transfer orbit satisfies engine-out at liftoff requirement (*except the CLV payload)

	PLS + Caboose	CLV*	Scaled PLS	PLS
MECO Weight (lbs)	73,173	75,680	73,018	72,621
Gross Payload to Transfer Orbit (lbs)	**53,800**	**75,677**	**53,954**	**33,800**
Margin (lbs)	19,373	3	19,064	38,821

Figure 1.2-6 NLS-2 1.5 Stage (6/2) Performance

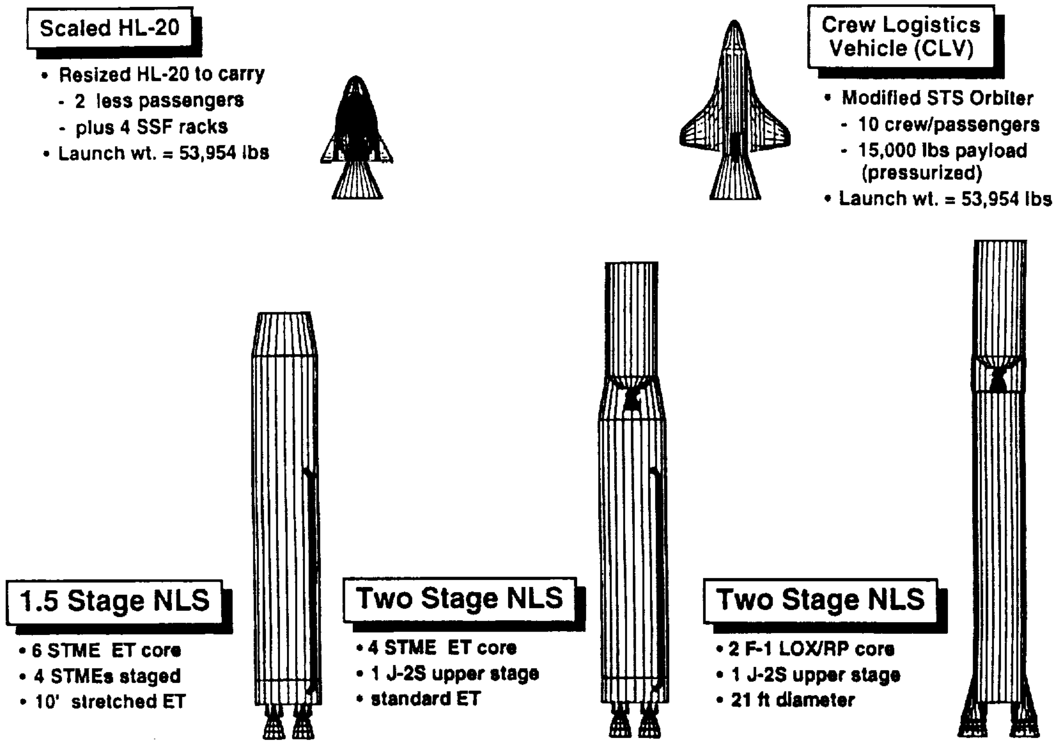
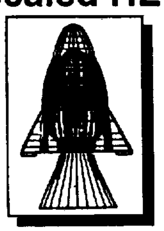


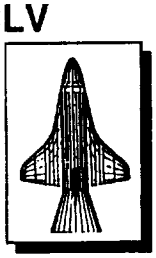
Figure 1.2-7 Booster Options for PLS & CLV

Scaled HL-20



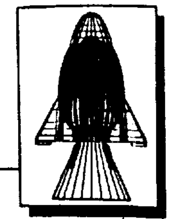
- Recommend 1.5 Stage NLS booster
  - Performance margin is good
  - Loads issues can be accomodated
  - Booster control authority is adequate
- 2 Stage NLS boosters have excessive performance capacity

CLV



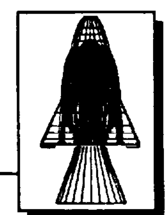
- Recommend 2 Stage NLS (STME) booster
  - Significant performance margin
  - Loads issues can be accomodated
  - Booster control authority is adequate
- F-1 based NLS booster is acceptable alternative
- 1.5 Stage NLS booster has no performance margin

Figure 1.2-8 Recommended Boosters for HL-42 & CLV



	1.5 Stage NLS	2 Stage NLS	
	6 STME	4 STME/ 1 J-2S	2 F-1/ 1 J-2S
Gross Liftoff Weight (lb)	2,009,907	1,776,716	2,933,622
1st stage propellant	1,788,657	1,310,119	2,387,695
2nd stage propellant	N/A	219,084	288,366
excess propellant	19,356	49,031	56,545
Payload to Transfer Orbit (lb)	**53,954**	**53,954**	**53,954**
Thrust-to-Weight (g)			
Pre-staging	4.00	3.90	2.78
Post-staging	1.86	0.88	0.71
Insertion	2.07	2.02	1.87
Maximum	**4.00**	**4.00**	**4.00**
Max Dynamic Pressure (psf)	795	554	677
Engine throttle - 1st stage	4 @ 70%	4 @ 70%	2 @ 75%, 1 off
- 2nd stage	2 @ 70%, 1 off	none	none

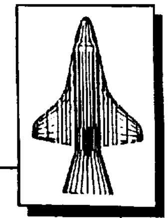
Figure 1.2-9 Booster Performances for HL-42



	1.5 Stage NLS	2 Stage NLS	
Conditions at:	6 STME	4 STME/ 1 J-2S	2 F-1/ 1 J-2S
Liftoff			
alpha (°)	90	90	90
gimbal reqd. (°)	-0.1	-0.2	0.04
CM (Kft-lb)	-161	-410	290
margin (°)	**3.9**	**3.8**	**3.9**
Max Q			
alpha (°)	6.6	5.4	4.7
gimbal reqd. (°)	3.0	0.6	1.0
CM (Kft-lb)	210	-7,459	18,450
margin (°)	**1.0**	**3.4**	**3.0**

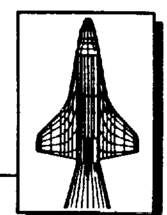
Notes
Available engine gimbal control assumed = 4°
CM = Control moment required at 6° alpha for computed C.G.
Gimbal requirements include effects of throttled engines

Figure 1.2-10 Booster Controllability with HL-42



	1.5 Stage NLS	2 Stage NLS	
	6 STME	4 STME/ 1 J-2S	2 F-1/ 1 J-2S
Gross Liftoff Weight (lb)	2,031,630	1,798,815	2,956,650
1st stage propellant	1,788,657	1,309,968	2,387,695
2nd stage propellant	N/A	219,589	288,544
excess propellant	0	30,334	36,479
Payload to Transfer Orbit (lb)	**75,677**	**75,677**	**75,677**
Thrust-to-Weight (g)			
Pre-staging	4.00	3.72	2.67
Post-staging	1.86	0.82	0.67
Insertion	2.05	1.97	1.83
Maximum	**4.00**	**4.00**	**4.00**
Max Dynamic Pressure (psf)	850	536	659
Engine throttle - 1st stage	4 @ 70%	4 @ 70%	2 @ 75%, 1 off
- 2nd stage	2 @ 70%, 1 off	none	none

Figure 1.2-11 Booster Performance for CLV



	1.5 Stage NLS	2 Stage NLS	
Conditions at:	6 STME	4 STME/ 1 J-2S	2 F-1/ 1 J-2S
Liftoff			
alpha (°)	90	90	90
gimbal reqd. (°)	0.4	0.1	0.3
CM (Kft-lb)	2,100	560	970
margin (°)	**3.6**	**3.9**	**3.7**
Max Q			
alpha (°)	5.9	5.5	4.8
gimbal reqd. (°)	2.5	0.6	0.9
CM (Kft-lb)	840	-8,006	16,880
margin (°)	**1.5**	**3.4**	**3.1**

Notes
Available engine gimbal control assumed = 4°
CM = Control moment required at 6° alpha for computed C.G.
Gimbal requirements include effects of throttled engines

Figure 1.2-12 Booster Controllability with CLV

Cost Analyses

Analysis of the recurring launch cost of a PLS/expendable launch vehicle booster system found that it will at best match the current Space Shuttle variable launch cost. The keys to achieving a cost effective PLS/ELV system will be the availability of a very low cost, man-rated booster, and the ability to eliminate the large fixed costs (standing army) currently absorbed by the Space Shuttle program. The Russian (C.I.S.) launch vehicles currently being offered for launch were analyzed and are believed to be offered at prices well below their true cost. As free market forces bring the C.I.S. economies into equilibrium with the rest of the world's launch businesses, the prices for launch vehicles such as the Zenit and Proton will approach the prices for similar boosters from American and European launchers. It is estimated, however, that the C.I.S. launch prices will stabilize at the 1-sigma lower band limit of Western launch prices as there will be insufficient market forces to drive them above that point. The Soyuz booster is projected to stabilize at a 2-sigma lower band limit of Western launch prices because of its significant rate and learning curve advantages over any other launch vehicle system.

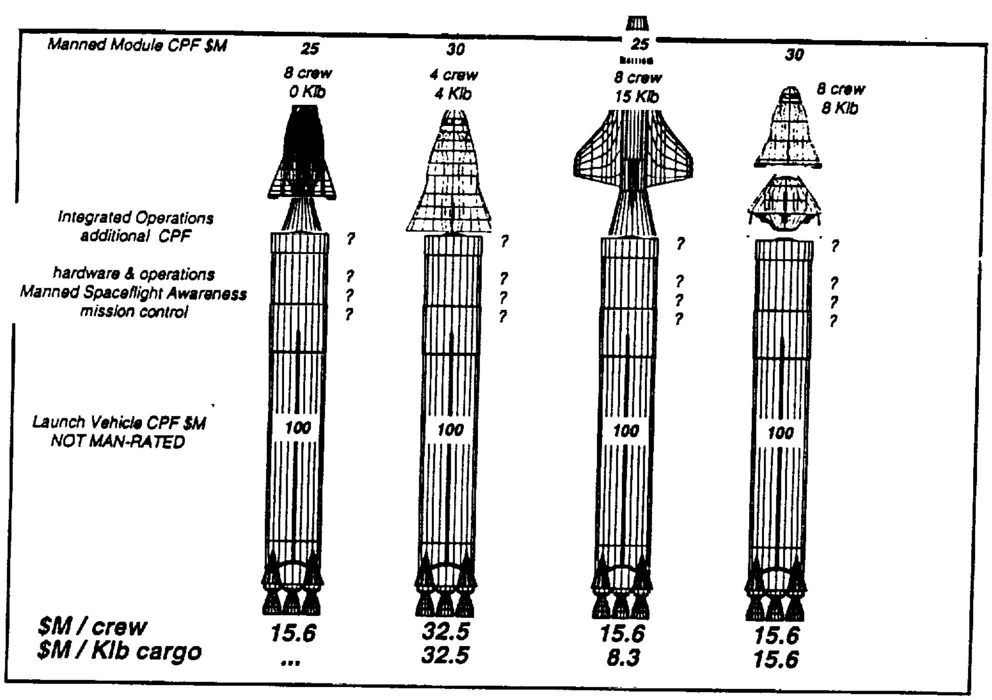


Figure 1.2-13 Launch Cost Comparisons for Selected Manned Payloads

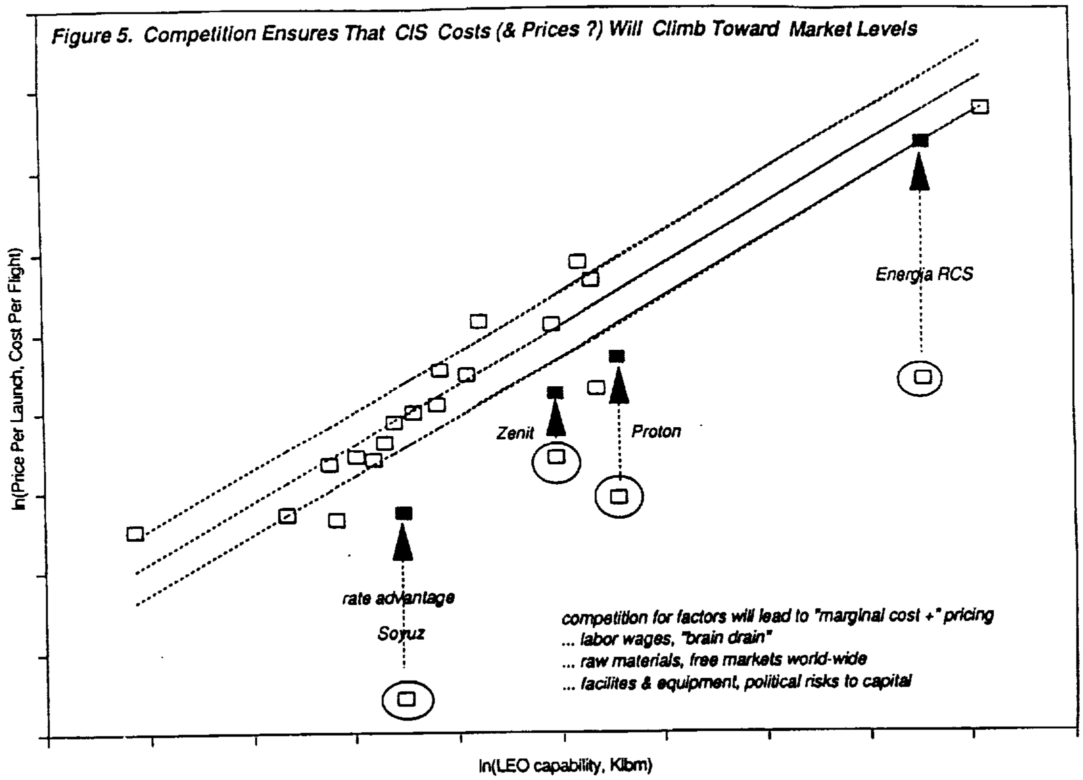


Figure 1.2-14 Estimation of True Russian Booster Launch Costs

Reliability Analyses

Reliability analysis of the PLS on a NLS-2 booster found that this particular combination should prove to be an exceptionally reliable and safe launch system. A highly reliable STME, a single engine out capability in the NLS, and the PLS-provided escape system all combined to provide high mission success rates and improved crew safety. A booster's engine out capability was found to be less a factor in comparing manned booster concepts reliability than is individual engine reliability, engine burn times, and number of stages employed by the boosters. Comparisons of crew safety levels on several manned booster concepts was most strongly governed by the crew escape system (escape motors), which is provided by the PLS system for the booster concepts studied.

- **Reliability assessment of the NLS 1.5 Stage vehicle's Propulsion system reveals potential for exceptional reliability**
- **Feasible levels of reliability based on "real" engine operational failure histories**

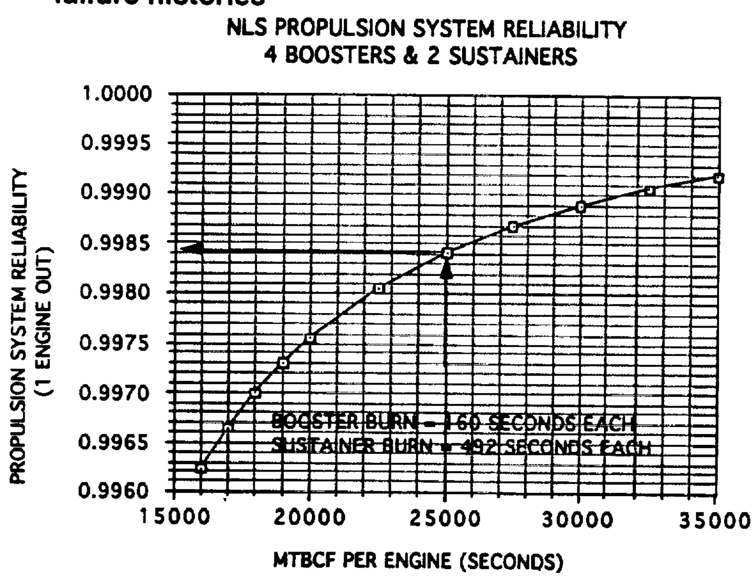


Figure 1.2-15 NLS-2 Propulsion System Reliability

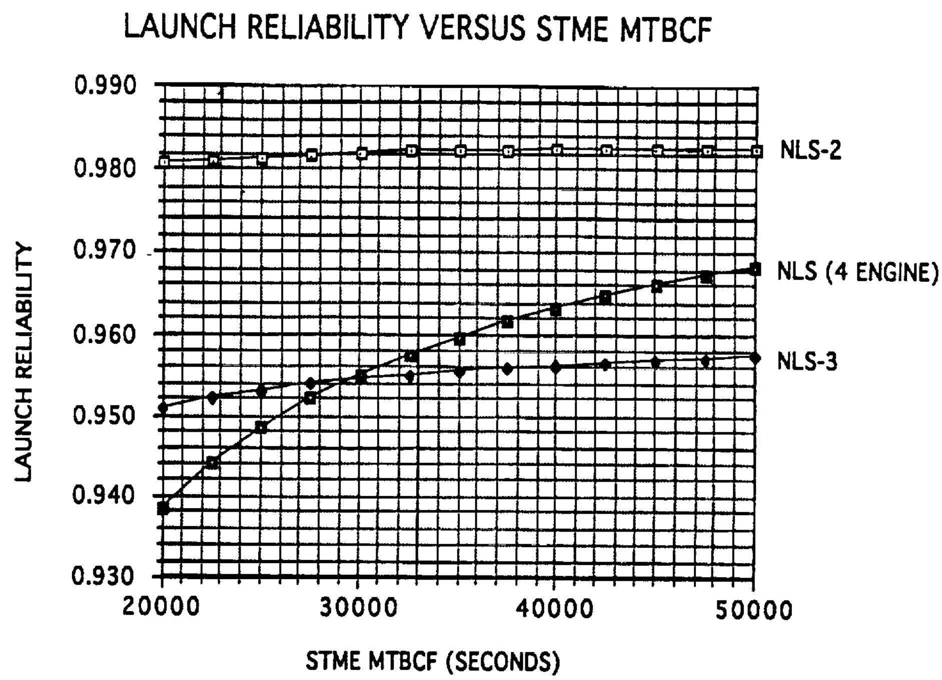


Figure 1.2-16 NLS-2 Launch Reliability vs. STME Reliability

Reliability was estimated on MTBCF basis

- **Key parameter is assumed MTBCF for STME (24,650 sec)**

System	Engine	Run Time	Failures	MTBCF
Atlas	boosters	60,758	7	8,679
	sustainer	55,594	8	6,949
Delta	sustainer	35,095	3	11,698
Shuttle	SSME	66,240	1	66,240

Calculation:

	Reliability
- booster engines (3 @ 195 sec.)	.9765
- sustainer engine (1 @ 455 sec.)	.9817
- avionics	.9999
- other (tanks, mech...)	.9886
Total System	.9478

Figure 1.2-17 NLS-2 and other Booster Engine Reliabilities

Launch Processing Analyses

Launch processing analyses of the combined PLS/NLS launch system were performed with software simulation programs. The simulations revealed that the combined effects of these two systems utilizing NASA KSC facilities can achieve the desired 10 flights per year launch rate with planned NLS and PLS program assets. Operation of mixed launch vehicle fleets (such as PLS/NLS and Space Shuttle) at KSC is considered achievable because of the low utilization rates of key facilities by concepts such as NLS. The mobile launch tower was consistently found to be the limiting resource of launch vehicle flight rate capability for all of the booster concepts analyzed. Analysis of the Space Shuttle system found that it has demonstrated a launch processing learning rate curve of 79% in the post-Challenger era. this learning curve effect should be included in ground processing analyses and flight rate planning of future launch systems.

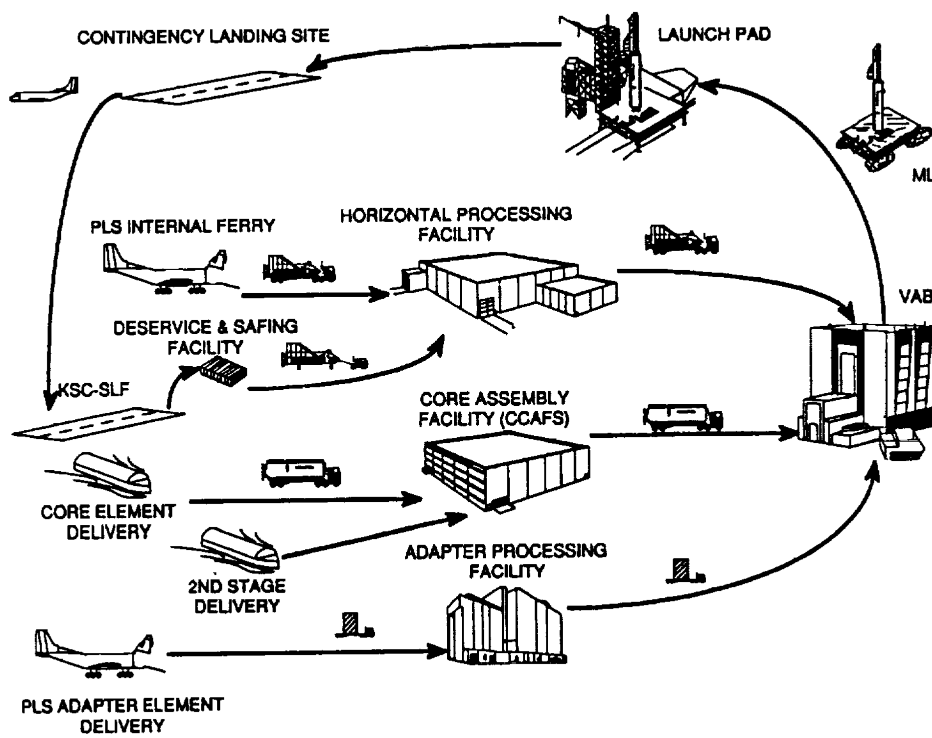


Figure 1.2-18 NLS-2 Launch Processing Diagram

Throughput Capacity:

Launch System Design Max	13.4 flights/year
Target (Derated to 80%)	10.7 flights/year
6-Month Surge Capability	7 flights

Facility/Resource Capability:

Name	HPF	VAB-4	CA/PF	MLT	Pad
Location	KSC	KSC	CCAFS	KSC	KSC
Status	New	Modified	New	New	Modified
Design Max Utilization	23%	37%	22%	100%	31%
Throughput Capacity:					
• Planning @ 80% (flows/yr)	46.8	28.6	48.4	10.7	34.3
• Max @ 100% (flows/year)	58.5	35.8	60.6	13.4	42.8

- MLT is the constraining resource
- Low utilization of KSC facilities

Figure 1.2-19 NLS-2 Launch Facility Utilization

1.2.2 Cargo Transfer & Return Vehicle Studies

The Cargo Transfer and Return Vehicle Concept, referred to as the CTRV, is a system for performing the mission of cargo delivery and return to low Earth orbit (LEO). As originally envisioned, this system would operate in conjunction with a crew delivery/return system (such as the PLS). The CTRV concept, together with a PLS concept and an appropriate launch vehicle(s), form the architectural framework of the Access to Space, Option 2 study. Identifying the specific configurations of the CTRV, PLS, and launch vehicles which best satisfied the Access to Space mission requirements was the principle objective of the study.

Design and analysis activities identified several CTRV configurations in response to key design issues for these systems. The PLS Caboose concept, and several larger CTRV concepts were developed. These concepts started with the Medium CTRV and Integral CTRV concepts (originally conceived of by MSFC and General Dynamics), and evolved into several versions of a Winged CTRV configuration. Payload volume and mass requirements associated with the Space Station logistics elements (which were designed for the Space Shuttle payload bay) placed the greatest constraint on the overall CTRV configurations. Aerodynamic stability and heating constraints during re-entry imposed design constraints which led to large aerodynamic control surfaces and limited the options for internal payload bay and subsystems layouts due to CG limits.

1.2.2.1 PLS Caboose Concept

The excess lift capability of the NLS-2 launch vehicle for the PLS payload led to an examination of synergistic systems which could utilize this additional lift capacity with the PLS system. The concept of a PLS Caboose (a mini-CTRV which operates attached to the PLS until SSF docking) evolved from this effort.

It was found that a cargo payload of 8,000 pounds (eight equivalent Space Station standard racks) could be delivered with each PLS mission to Space Station by utilizing the PLS caboose concept. Most of the mass of the PLS-to-booster adapter was converted to useful payload by integrating the adapter into the caboose structure. The PLS escape motors were also converted into useful payload mass by utilizing them for the orbital transfer maneuvers of the combined PLS/caboose mass. Use of this caboose concept with each PLS launch would reduce the number of annual launches required to deliver pressurized payloads to Space Station by one per year compared to separate CTRV and PLS launches.

A hot structure thermal protection system (TPS) approach was used on the caboose concept. This approach resulted in a TPS/structure combined weight which was 1.36 times the weight of an equivalent (conventional) tile/aluminum cold structure approach. The projected cost savings of this approach, however, was dramatic. The hot structure TPS fabrication and installation cost is estimated to be \$40 /ft^2, significantly lower than the \$16,000 /ft^2 for tile installation costs currently being experienced on the Space Shuttle. The combination of limited cross range and a low cost thermal protection system make this design approach a particularly good solution for water landing type systems.

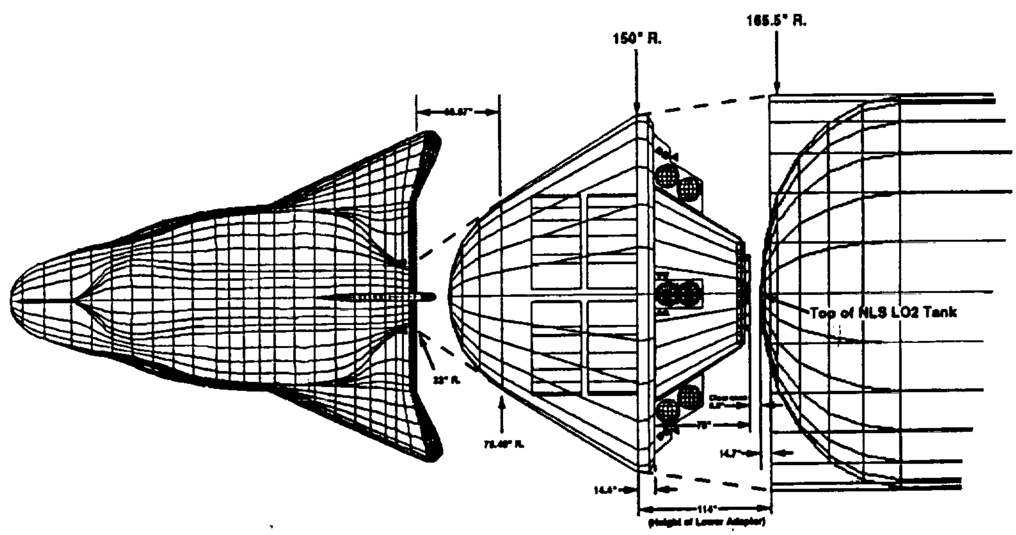


Figure 1.2-20 PLS/Caboose/NLS-2 Launch Configuration

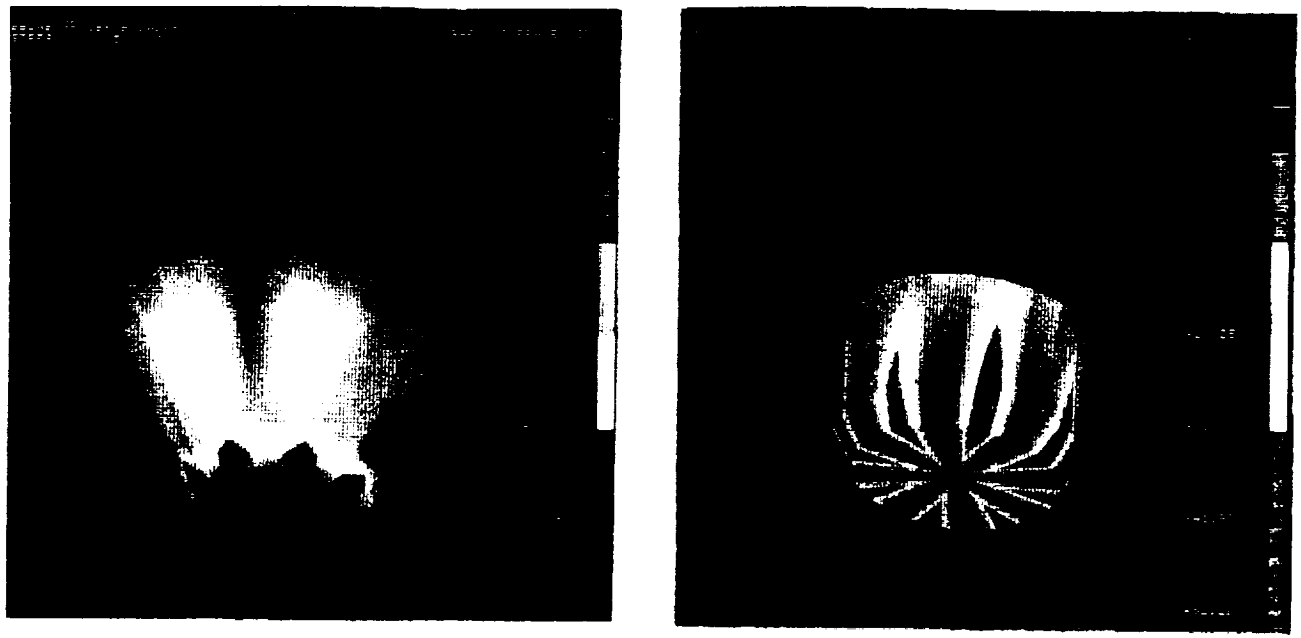


Figure 1.2-21 PLS Caboose Stresses During Re-entry

1.2.2.2 Medium CTRV Concept

A detailed design study of the Medium CTRV concept was performed to obtain a better understanding of the CTRV design requirements and to improve the design definition for this concept. The design effort included trajectory analysis, thermal analysis, detailed structural design layouts and stress analysis, and parametric analysis of the CTRV landing system options (parachutes). Key results and findings from these analyses are provided below.

Structural Design and Analysis

Detailed structural design of the Medium CTRV was performed to obtain better weight estimates of this concept and to also determine dynamic responses of this structure approach to flight loading conditions. Structural design layout drawings were first prepared of the Medium CTRV (C23LNF configuration) to better define the structural design. Analysis of load paths to carry flight loads through the structure resulted in some changes to the baseline structural concepts and in more detailed structural definition. Computer (CAD) drawings of the design and layout were utilized for the revised structural definition and stress analysis models were developed from the CAD drawings for use in the structural analysis.

A finite element model of the Medium CTRV structure design was completed which includes all structural elements and major subsystems. The model was used to determine stress levels for several flight load conditions: maximum acceleration (3.2 g's at MECO); maximum structural bending loads (aerodynamic loads at max Q); and landing loads (29 ft/sec vertical landing velocity). The analyses generally showed that the skin, frame, and longeron thicknesses can be reduced. The bulkhead structures at the forward and aft ends of the payload bay were found to require some stiffening for the high axial acceleration launch conditions (MECO & max Q).

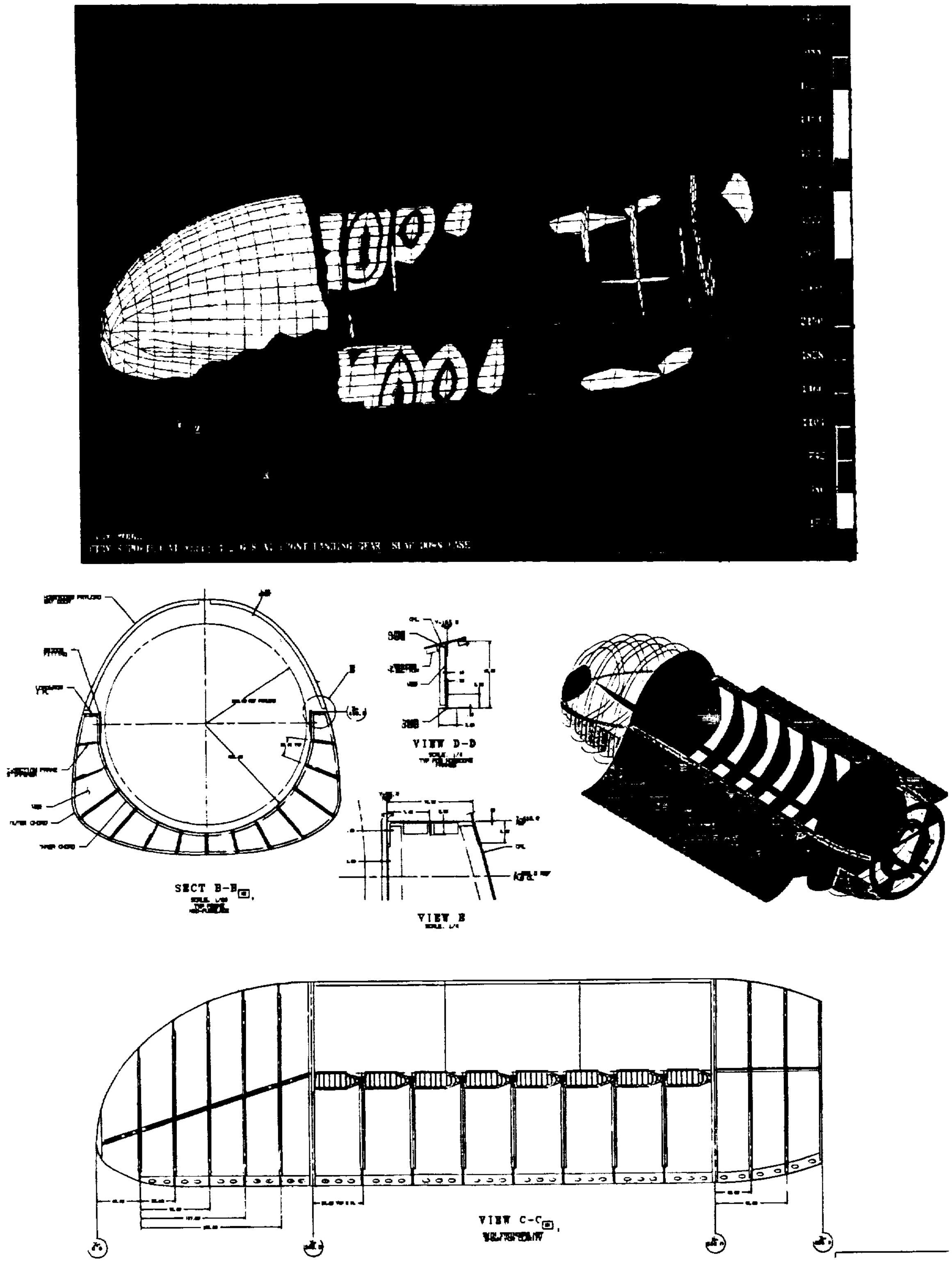


Figure 1.2-22 Medium CTRV Structure Design & Stress Analysis

Thermal Analysis and TPS Sizing

The TPS sizing analysis indicated that the required TPS thicknesses are actually slightly thinner than would be seen on a Shuttle for equivalent re-entry heat loads. The reason for the thinner tiles is the lower initial temperatures found on the CTRV structure and tiles. The Shuttle tiles were sized for a mission which required re-entry immediately after launch. The CTRV has no similar mission (or any equivalent abort missions) and temperatures are allowed to stabilize to orbital conditions prior to entry. Initial temperatures for the tiles and underlying structure were calculated for on-orbit solar heating conditions and allowed a 12 hour thermal conditioning period to lower the temperatures prior to entry.

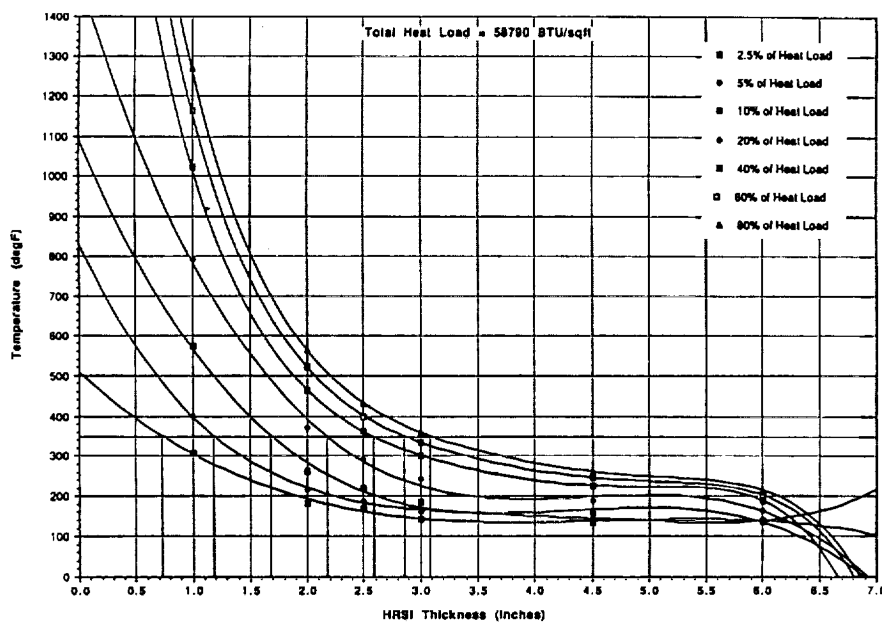


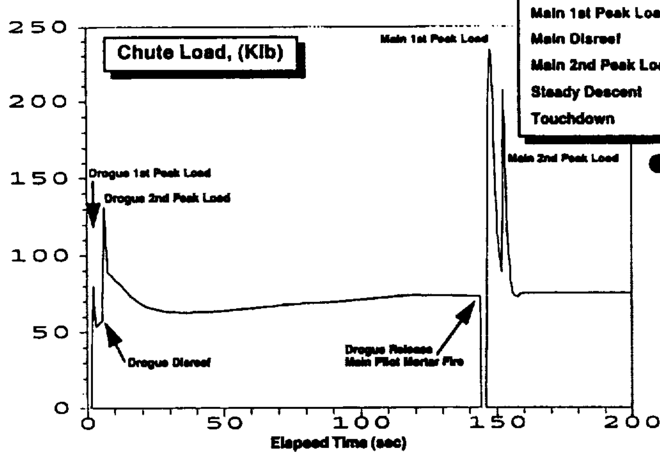
Figure 1.2-23 CTRV TPS Tile Sizing Analysis

Landing System Design

Landing system designs were based on a design approach which called for a supersonic deployment (Mach 1.4) of a drogue chute and parachute deployment loads under 4 g's. Trade studies of parachute sizing and deployment options, system cost and weight impacts, and parachute deployment loads were performed to determine an appropriate landing system design for the CTRV. The parachute system selected for the CTRV is an eight chute cluster of 137 foot parachutes. This main chute system results in a terminal descent rate of 28 ft/sec velocity. The total weight for the CTRV parachute system is estimated at 5,865 lbs, requiring a stowage volume of 140 cubic feet.

- **Estimated flight parameters for key events**

Event	Time (sec)	Alt (ft-MSL)	Vel (fps)	Mach No.	q (psf)	Chute Load (lb)	Accel (G)
Drogue Pilot Mortar Fire	0.0	87872	1376	1.4	55.1	0	1.0
Drogue 1st Peak Load	2.3	84714	1356	1.38	61.9	80291	2.18
Drogue Disreef	5.5	80530	1261	1.29	65.6	57367	1.94
Drogue 2nd Peak Load	6.3	79534	1223	1.25	64.8	131890	2.91
Drogue Release and Main Pilot Mortar Fire	144.0	16000	266	0.25	51.5	73010	1.0
Main 1st Peak Load	147.9	14831	254	0.24	48.4	234580	3.19
Main Disreef	152.0	14163	122	0.11	11.6	88585	1.2
Main 2nd Peak Load	152.6	14095	99	0.09	7.48	207505	2.77
Steady Descent	160.0	13800	33	0.03	0.83	74963	1.0
Touchdown	630.0	0	27	0.02	0.83	74945	1.0



- **Refined information required for subsequent iterations:**
  - C_d profile of CTRV (Super,Trans,Sub-sonic)
  - Angle of attack profile for CTRV under the influence of inflated drogue chute

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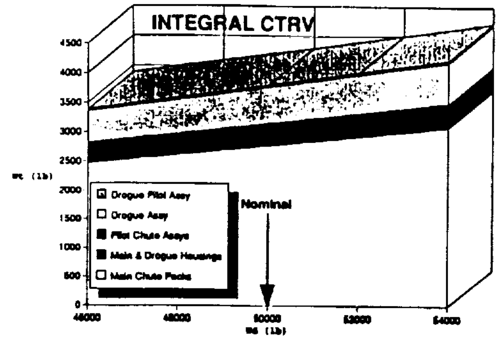
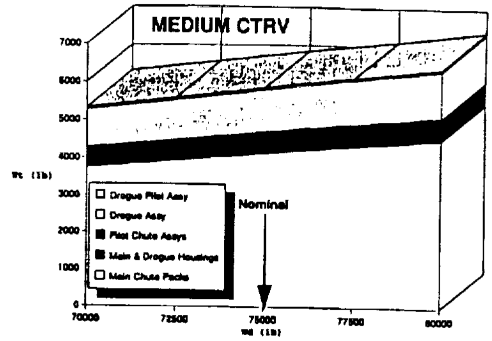
Figure 1.2-24 CTRV Parachute Landing System Performance





- **RECOVERY SYSTEM COMPONENT WEIGHTS**
  - Conventional Cluster recovery system designed for V_v=25 fps
  - System weight breakdowns vary with descent weight
- **MAIN CHUTE PACKS:** Includes main chutes with reefing provisions, deployment bags and risers
- **MAIN & DROGUE HOUSINGS:** Stowage for main and drogue parachute packs
- **PILOT CHUTE ASSYS:** Includes 2 pilot chutes with deployment bags, risers and mortars
- **DROGUE ASSY:** Drogue pack with reefing provisions, deployment bag and riser
- **DROGUE PILOT ASSY:** Drogue pilot chute pack with deployment bag, riser and mortar

COMPONENT WEIGHTS	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Main Chute Packs	2778	4137
Main & Drogue Housings	174	263
Pilot Chute Assys	206	299
Drogue Chute Pack	617	1114
Drogue Pilot Assy	30	51
TOTAL	3805	5864

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Figure 1.2-25 CTRV Parachute Weight Estimates

Launch Processing Analyses

Analysis of the CTRV ground (launch) operations was completed with a simulation of the combined Medium CTRV, PLS, and their launch vehicle (NLS-2 type) systems launched over a ten year period. The CTRV systems were modeled at the subsystem level to identify the payload integration differences between the Medium and the Integral CTRV concepts. The analysis demonstrated that the planned flight rates for PLS/CTRV to support Space Station Freedom logistics missions can be achieved when using the subsystems processing times as projected by the PLS program. The STARSIM analysis also demonstrated that at the anticipated mission reliability levels for this CTRV, a 90% Probability Of Sufficiency (POS) spares level will be satisfactory for this system. This translates into a mean ground processing delay period of just over five days.

1.2.2.3 Integral CTRV Concept

The Integral CTRV concept was developed to reduce the packaging overhead of Space Station logistics cargo payloads. Rather than load the logistics payloads into a pressurized module which is then loaded into the unpressurized payload volume of the CTRV, the Integral CTRV payload bay itself is pressurized and the payloads can be installed directly into the CTRV. The Integral CTRV thus directly replaces the Space Station Pressurized Logistics Module and remains at the Space Station for several months duration before returning to Earth. Space Station unpressurized logistics payloads are similarly delivered in an unpressurized version of the Integral CTRV. The Integral CTRV is a ballistic type re-entry vehicle (parachute landing) but the aft payload bay location requires large fins for aerodynamic stability during re-entry. The fins also served as CTRV orbital propulsion subsystem pods, leaving a clear opening for the end-opening payload bay door. Analysis of the Integral CTRV concept included trajectory analysis, aerodynamics and thermal analysis, preliminary structural design in the area of payload retention and deployment, and parametric analysis of the CTRV landing system options (parachutes).

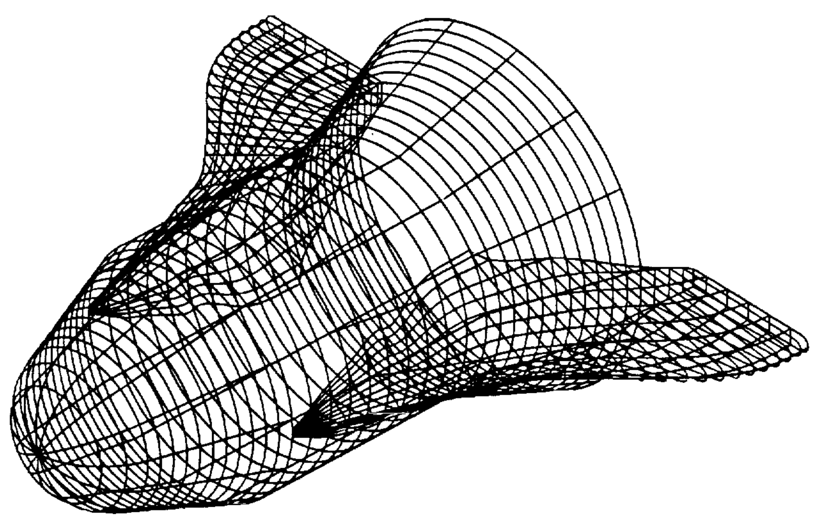


Figure 1.2-26 Integral CTRV Concept

Aerodynamic Analyses

Analysis indicated that an entirely new aerodynamic approach than that used for the Medium CTRV concept would be required for the Integral CTRV in order to arrive at acceptable structural weight fractions. The aerodynamic configuration for a low cross range version Integral CTRV was developed by performing a sensitivity study of lift (C_l) and drag (C_d) coefficients from a general ballistic re-entry CTRV configuration. POST computer simulations were used to optimize re-entry trajectory performance. The resulting preferred configuration was a basic cylindrical mid fuselage, a forward fuselage consisting of a 20° cone with 5 foot radius sphere nose, and a 20° flared skirt at the aft fuselage. A pair of large fins (not wings) were added to the cylinder/skirt to move the aerodynamic center of pressure aft for hypersonic flight stability and CG considerations.

Trajectory Analyses

Trajectory analyses with the selected Integral CTRV configuration showed that it could be flown to a maximum cross range of 70 Nmi without violating structural and thermal loads constraints. The trajectories were flown at a 7.5° to 15° angle of attack to minimize heat load to the TPS system. Because the vehicle is not flown at a high angle of attack (40° or higher like the Medium CTRV), there is not a large surface area which requires TPS tiles (a design goal). The majority of the outer surfaces could be covered by TPS blankets, rather than tiles.

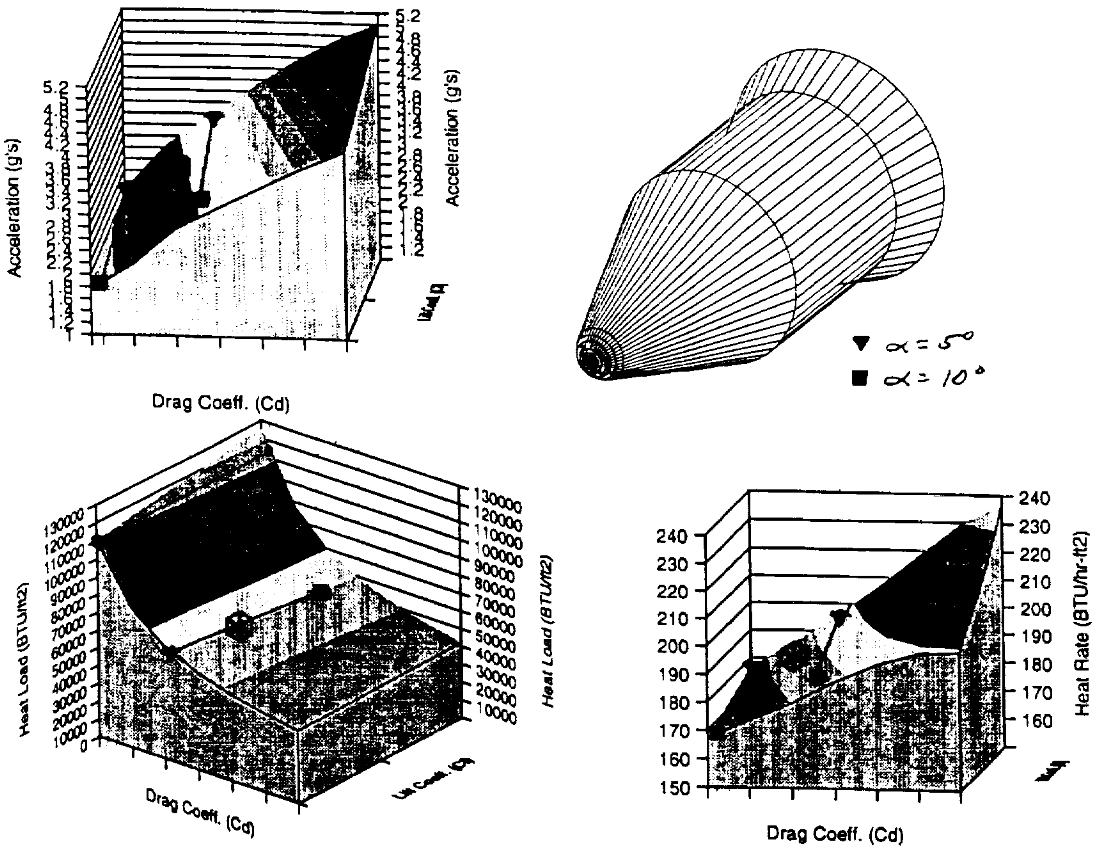


Figure 1.2-27 CTRV Aerodynamic Configuration Optimization

Thermal Analysis and TPS Sizing

The TPS sizing was based on flying a low angle of attack (7.5°) trajectory, as demonstrated in POST trajectory analyses of the selected configuration. These trajectories resulted in low heating rates to most of the surfaces and much lower total heat loads than as seen in the Medium CTRV trajectories. The total heat load for the Integral CTRV trajectory is less than 20,000 BTU/ft^2, compared to 50,384 BTU/ft^2 for the Medium CTRV (C23LNF). The thermal protection system (TPS) concept for the Integral CTRV was modified to avoid the problem of bonding TPS tiles to a pressure vessel (severe technical design issues related to tile gaps and on-orbit/re-entry structural temperature limits would be encountered). A debris shield will be used as an intermediate structural shell for attaching the tiles. Significant operational efficiencies may also be realized by this configuration as the tiles can be installed or maintained off the vehicle by removing the debris shield panels.

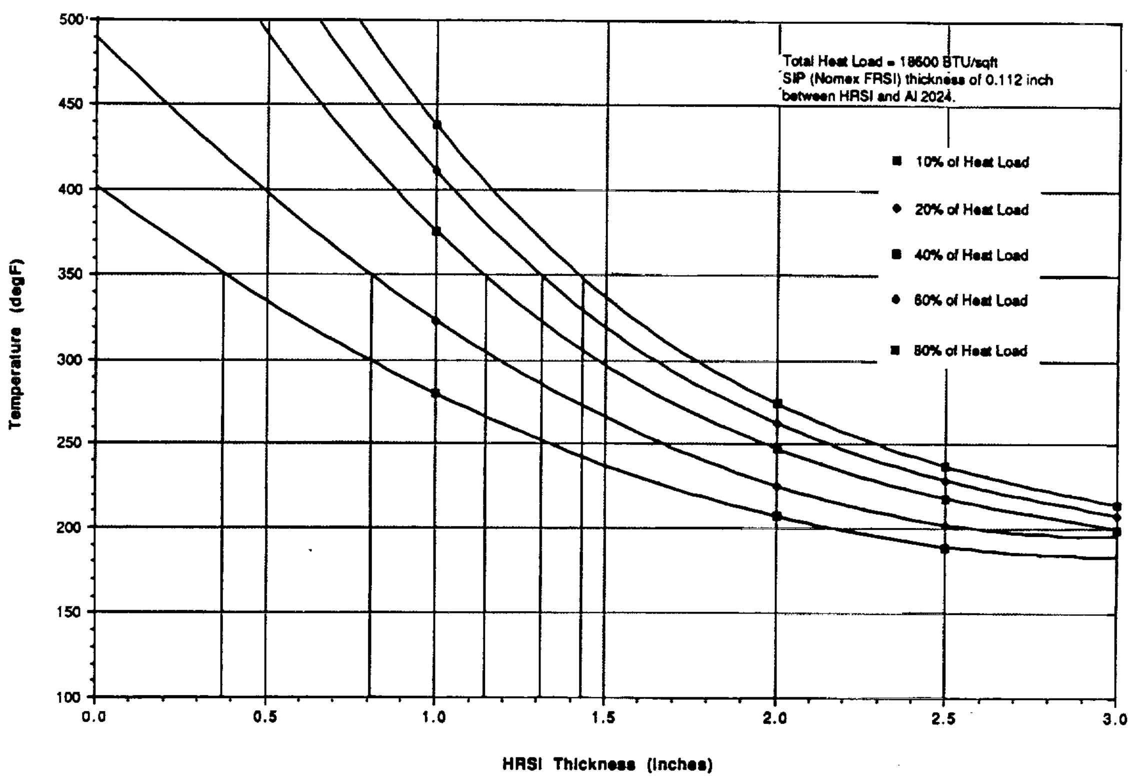


Figure 1.2-28 Integral CTRV TPS Tile Sizing Analysis

Reliability Analyses

A reliability analysis was performed using the MAtrix model for the Integral CTRV configuration to determine reliability levels and maintenance requirements down to the subsystem and major components level. With the Integral CTRV's long mission duration of 4320 hours (6 months), the analysis revealed a reliability of .969 and 25 unscheduled maintenance actions per mission. This low predicted reliability is due to the long operating period for the on-orbit needed systems during the 6-month mission. It is expected that better definition of actual component redundancies would improve the predicted reliability for the Integral CTRV concept.

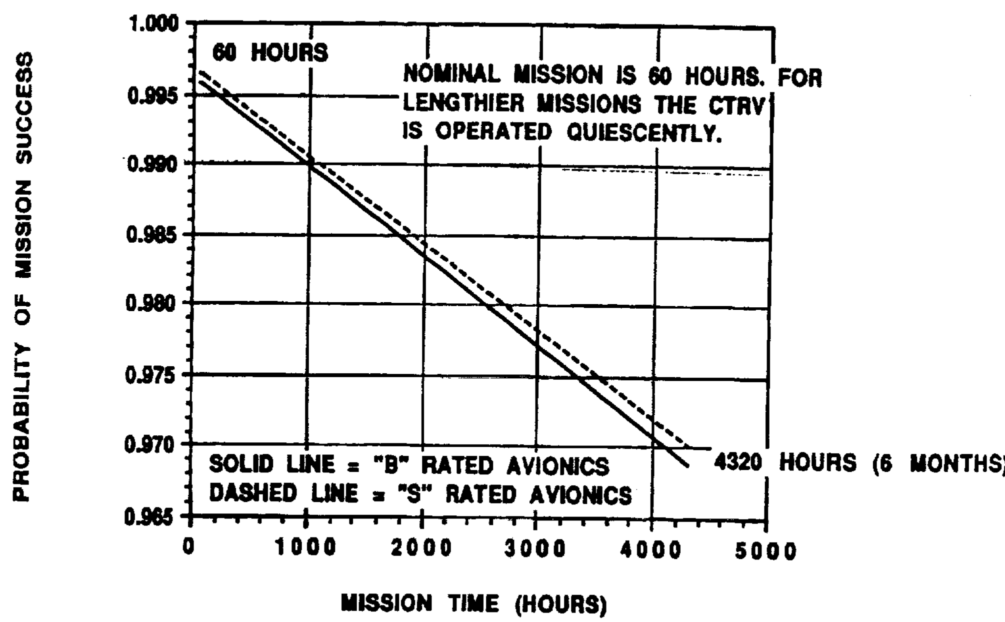


Figure 1.2-29 Integral CTRV Reliability vs. Mission Time

Launch Processing Analyses

Launch processing analysis of the Integral CTRV was performed similar to the Medium CTRV analysis. The CTRV was modeled at the subsystem level to permit expendable vs. reusable evaluations to be made at the total CTRV system level or at the individual subsystems level (such as propulsion, avionics, TPS/heat shield). Payload integration activities and propulsion systems were different from the Medium CTRV analysis. Integral CTRV propulsion system processing was performed off the vehicle and payload integration was performed on the vehicle, just the opposite from the Medium CTRV concept.

The Integral CTRV concept must fly up to 11 flights per year (versus 4 to 6 flights per year for the Med. CTRV). The launch processing simulation demonstrated that this higher flight rate can be achieved with the planned facilities and resources. A much higher manpower consumption is caused by the high flight rates, however. The Integral CTRV concept used almost twice as many hours of touch labor (direct "technician-hours") to accomplish the same SSF logistics supply mission as the Medium CTRV concept.

Of particular importance for the Integral CTRV processing analysis is the constraints imposed by the Space Station elements and payloads. The pressurized version of the Integral CTRV would be required to utilize the Space Station payloads processing facilities for integration of the payloads (Space Station racks) into the Integral CTRV payload compartment. This will require the propulsion systems for this CTRV to be either new (unflown) or removed and processed separately from the rest of the vehicle.

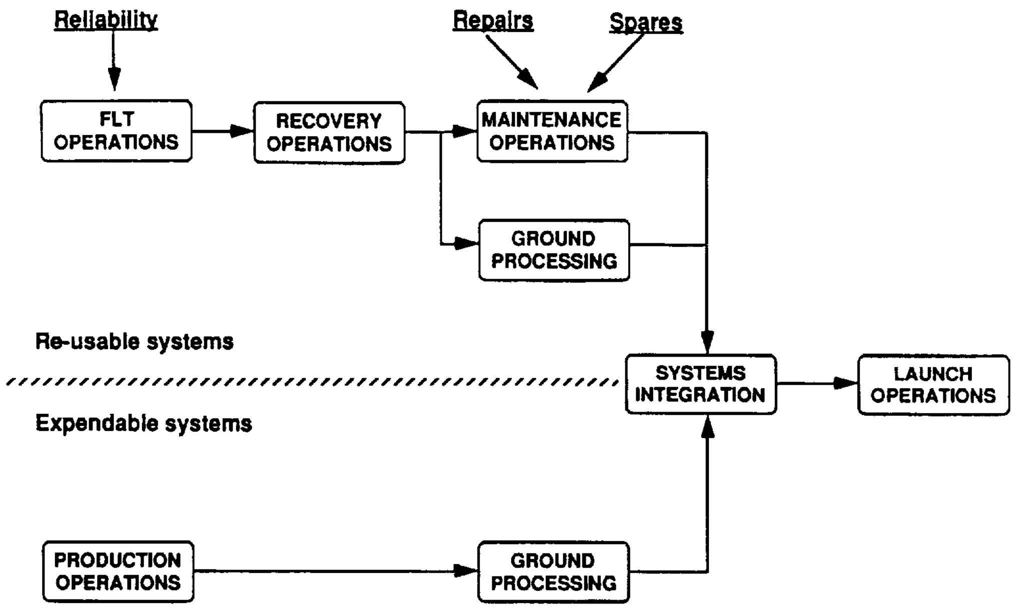


Figure 1.2-30 Integrated Launch Processing Approach

Throughput Capacity:

PLS Launch	4 flights/year
CTRV Launch Rate	6 - 11 flights/year
Avg. Time In System	49 days
Avg. Time Between Launches	26 days
Touch Labor Estimate	1,178 - 1,967 k-hours/year

Facility/Resource Capability:

Name	SSPF	HPF	VAB	Pad	MLT
Servers	2	2	2	2	3
Location	KSC	KSC	KSC	KSC	KSC
Status	New (Intgrtn. Cells)	Existing (SAEF-2 or PHSF)	Modified	Modified	Modified (MLP)
Utilization	2%	8%	19%	36%	86%

- MLT is the constraining resource

Figure 1.2-31 Integral CTRV Launch Facilities Utilization

CTRV Implications for Space Station

An assessment of the impacts that a CTRV system would have on the Space Station interfaces, and its flight or ground operations, was performed for both the Medium CTRV and the Integral CTRV concepts. The most significant differences noted between the current Space Shuttle based logistics system and a CTRV based system are listed below.

Space Station Flight Operations

The CTRV concept provides limited capability for delivery/return of large Space Station ORUs, severely restricting payload return mass and volume. The Space Station logistics mission will require a greater number of rendezvous and docking operations because of the smaller payload capabilities of some CTRV concepts, increasing SSF crew activity and training to support these functions (especially with no crew aboard the incoming element). The un-pressurized Integral CTRV configuration will also provide much reduced visibility and access to payloads in its cargo bay, limiting flexibility in payload deployment and transfer operations.

Space Station Ground Operations

Integration of Space Station logistics racks into the pressurized Integral CTRV payload bay will require payload installation to be performed in the Space Station Payload Processing Facility (as does the PLM). This will necessarily demand that some Integral CTRV post-flight turnaround and maintenance operations will also have to be performed in this facility. Integral CTRV concepts also provide no capability for late access (on the launch pad) to install special handling (refrigerated or live) payloads.

1.2.2.4 Winged CTRV Concept

As the analysis of Integral CTRV and Medium CTRV concepts progressed, it became apparent that the operating costs of these systems would not meet the goals of the NASA Access to Space study. A precision (runway) landing version of the CTRV concept was recognized as a key requirement for minimizing operations costs. This requirement led to the Winged CTRV concepts, which started with small payload capabilities (22,000 lbs), evolved to larger payload versions, and eventually to combined crew/cargo concepts. The Winged CTRV, the CLV, and the HL-42 concepts all became competitors for the crew/cargo element of a launch system architecture based on expendable launch vehicles. The definition of the Winged CTRV concept evolved as the NASA Access to Space study (Option 2) continued to refine the design requirements. The Winged CTRV concept evolved from the (original) small Winged CTRV, to a larger payload version (the Medium Winged CTRV), and finally to a combined crew/cargo version similar in function to the HL-42 but with a larger payload volume and weight capability (the Single Development Winged CTRV).

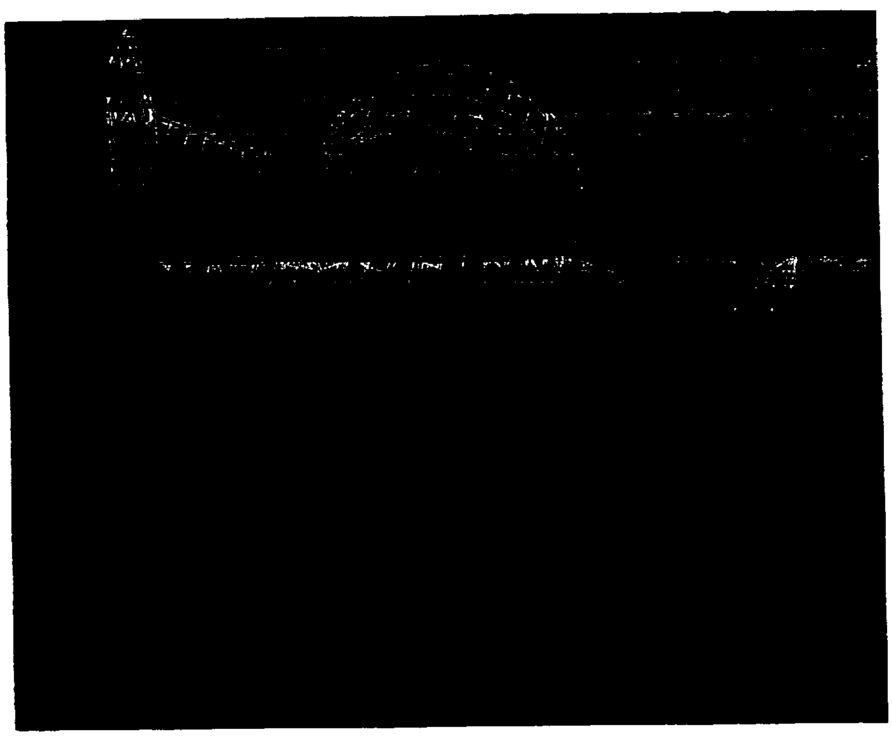


Figure 1.2-32 Winged CTRV Concept

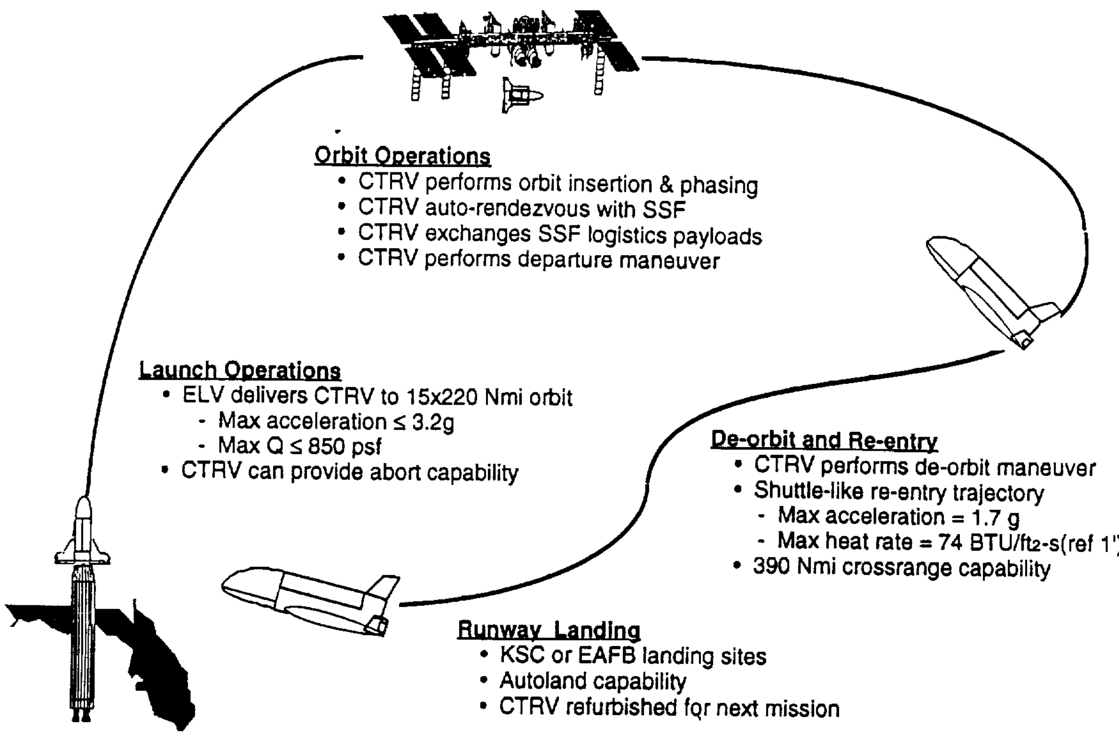


Figure 1.2-33 Winged CTRV Mission Profile

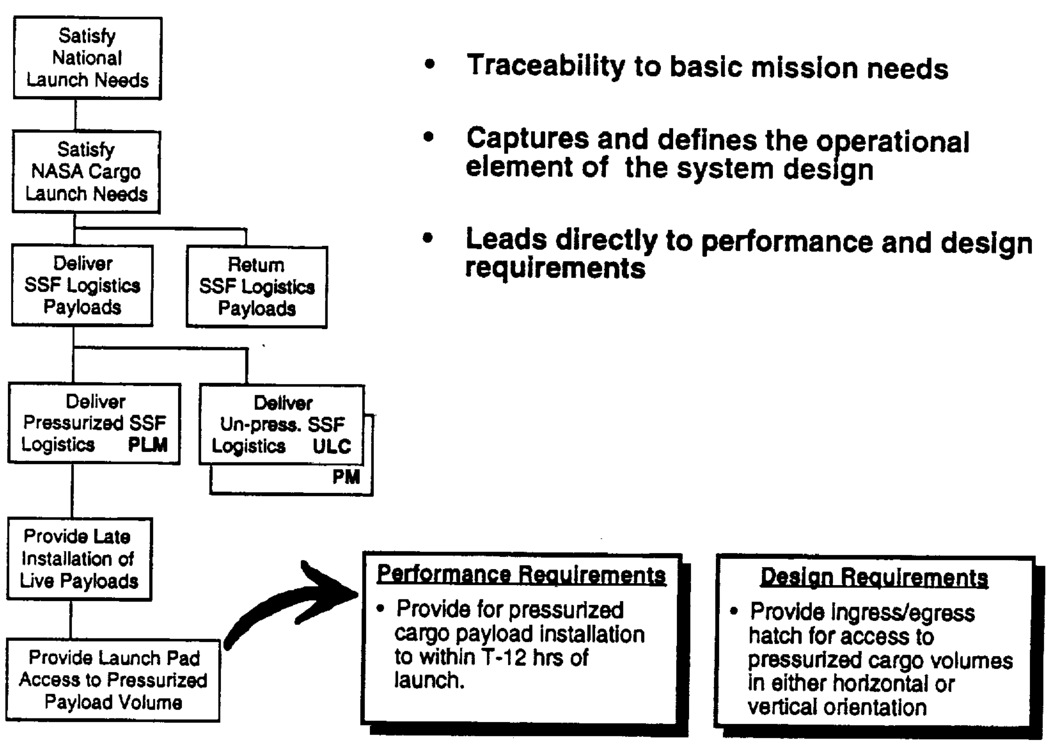


Figure 1.2-34 CTRV Functional Requirements Flowdown

Aerodynamics Analyses

Modifying the Integral CTRV into a runway landing system required sufficient low speed lift to counter the high drag caused by the wide and short fuselage. The Winged CTRV aerodynamic design was achieved by adding just enough wing area and stretching the fuselage just enough to reach the desired subsonic L/D ratio of 4.0. The aerodynamic characteristics of this configuration were calculated with the APAS analysis tool at both hypersonic and sub-sonic speeds. Trade studies of the configuration found the pitch stability to be very good, and the wing loading was found to be low compared to the current Space Shuttle design.

Analysis of the aerodynamic loads caused by the Winged CTRV (and by the other competing concepts) on the launch vehicle was also performed. The analysis evaluated the booster bending moments and static stability margins at maximum aerodynamic pressure (max Q) conditions (an NLS-2 booster was used for the launch trajectory conditions). The analysis showed that Winged CTRV concepts produced only low aerodynamic moments because of their relatively low-lift wings and a low normal force coefficient at the 5° angle of attack condition at max Q. The Winged CTRV wings were designed expressly to minimize the booster's max Q loads. The HL-42 concept produced only moderate launch vehicle bending moments at max Q, but the CLV concept imposed very high launch vehicle bending moments with associated large booster engine gimbal offsets.

Aerodynamic stability analysis of the Winged CTRV was performed to determine directional stability characteristics. A combination of various winglet sizes, nose cone shapes and vertical stabilizer sizes were evaluated. The results showed that retaining the initial winglet size and adding a vertical stabilizer of approximately 100 square foot area would provide a positive stability margin ($C_{\eta\beta}$= 0.008/deg at M=0.3).

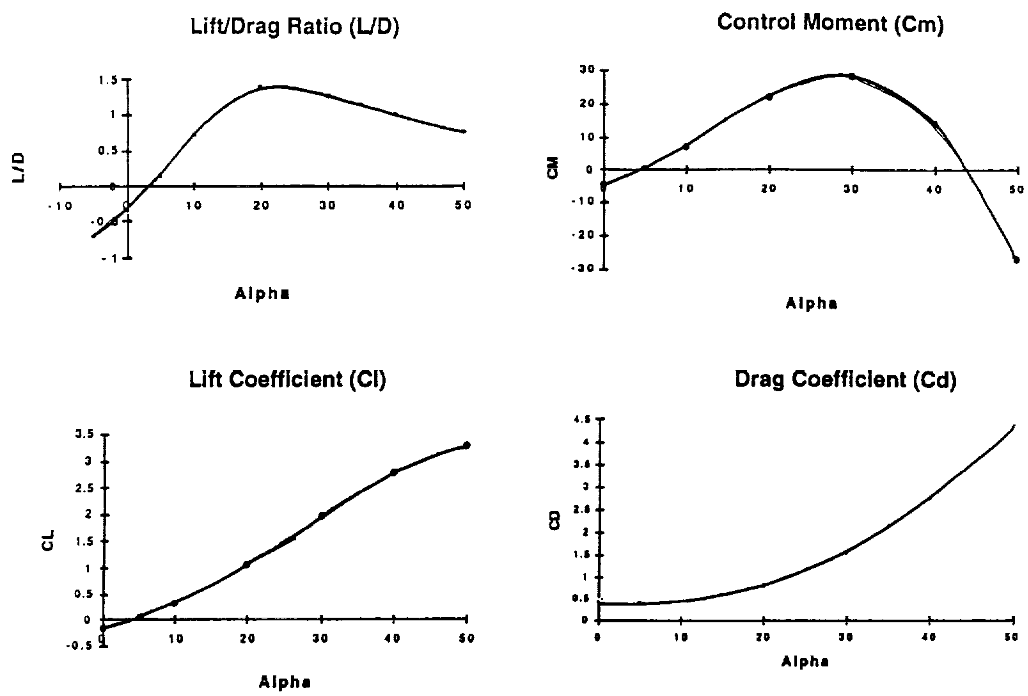
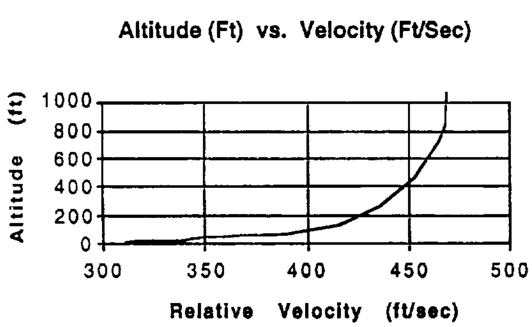


Figure 1.2-35 Winged CTRV Hypersonic Aerodynamic Characteristics



- Outer glideslope γ = -19°, inner γ = -1°
- Initial flare maneuver at 1.5 g's
- Landing flare maneuver at α = 6° to 14°
- Touchdown at α = 21°, V = 185 Kt

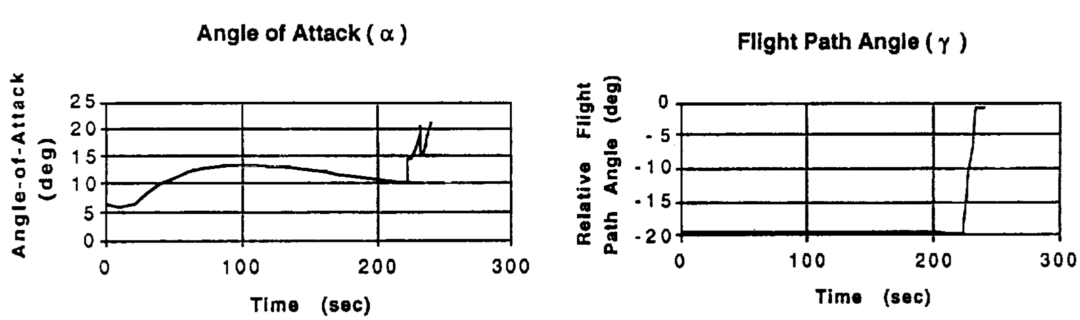


Figure 1.2-36 Winged CTRV Landing Characteristics

Trajectory Analyses

Trajectory analyses were performed for all versions of the Winged CTRV configurations and included the complete re-entry trajectory, from entry interface (400,000 ft altitude) down to the runway threshold. Improved re-entry trajectory simulation techniques from those used in the Medium CTRV analyses were required to more accurately predict the heating environment for CTRV's wings. A bank angle steering guidance mode was added to the CTRV's 3-DOF trajectory simulation. This permitted both angle of attack and bank angle profiles to be optimized by POST for a minimum heat rate trajectory. Maximum heating rates for the wing leading edges were reduced by a factor of almost two with these trajectories. The improved trajectory analysis eliminated the need for a re-design of the CTRV wing based on heating rates.

An analysis of the impact footprint for an uncontrolled CTRV re-entry was also performed to determine just how large an area of populated land mass might a re-entry vehicle such as the CTRV pose a danger to if system failures occurred during the entry phase of flight. This analysis showed that the potential debris impact footprint included most of the United States (from Hawaii to San Francisco, Chicago, Washington DC., and Florida) and the entire upper half of Mexico. This result indicates that a CTRV concept must have sufficient redundancy in flight critical systems to ensure that the vehicle can be guided to a controlled impact area in the event that primary or secondary landing sites cannot be reached due to system failures.

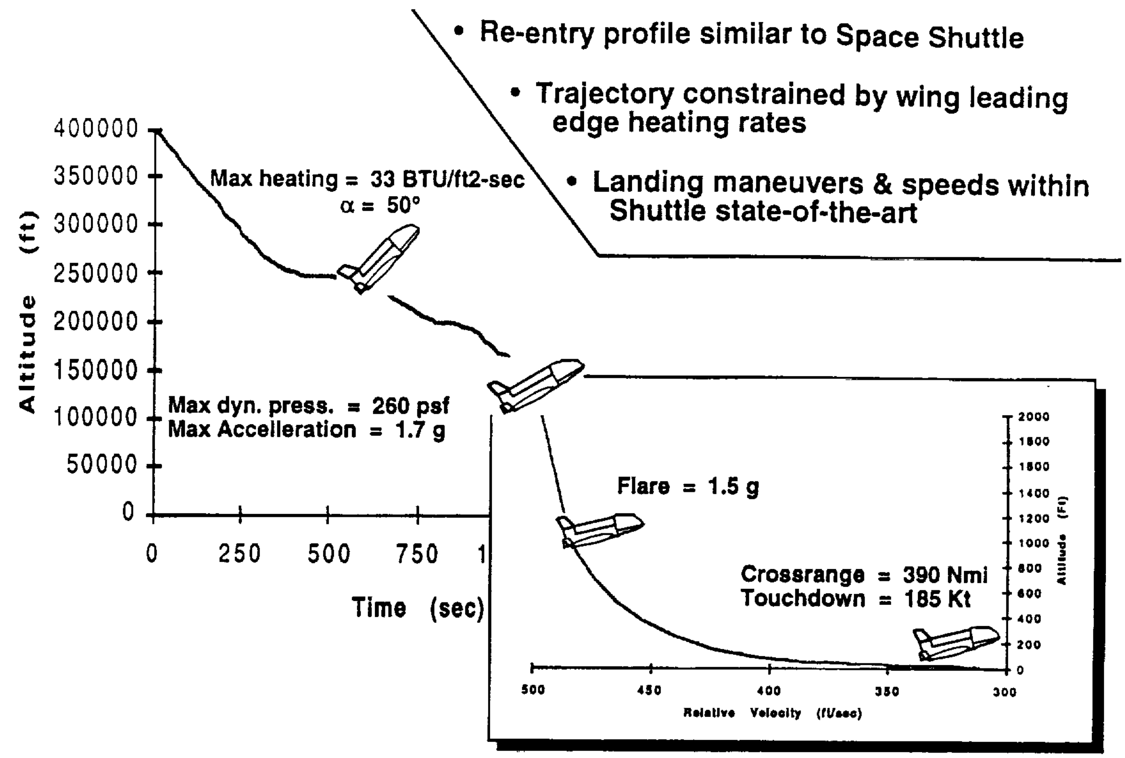


Figure 1.2-37 Winged CTRV Re-entry Performance

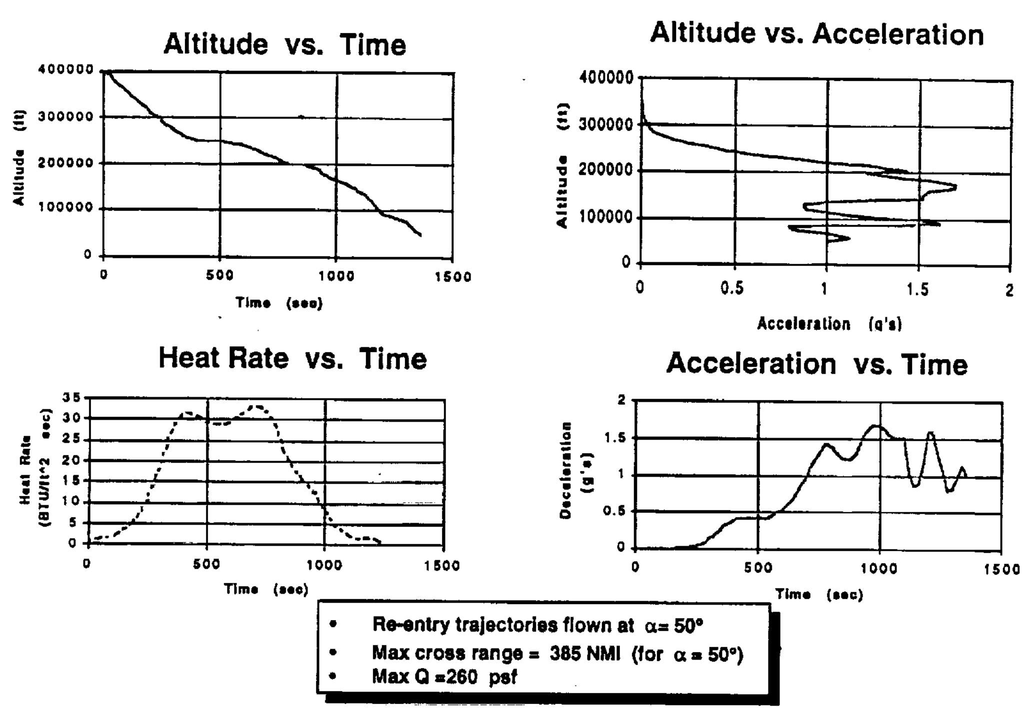


Figure 1.2-38 CTRV Re-entry Trajectory

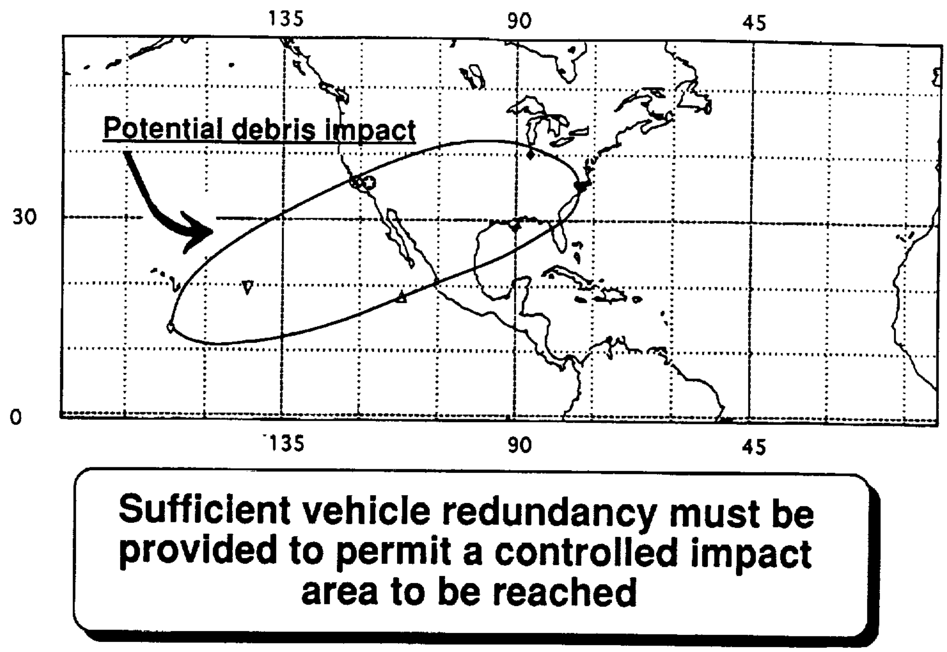


Figure 1.2-39 Potential Winged CTRV Impact Footprint

Aeroheating Analysis and TPS sizing

Sizing of the Winged CTRV thermal protection system (TPS) was calculated from heating rate distributions over the vehicle based on flying a moderate angle of attack (25°) trajectory, as demonstrated in POST trajectory analyses. The CTRV thermal protection system (TPS) weight was determined from TPS tile thicknesses as sized from the calculated heating rates.

A shock layout (position vs. mach number) for the Winged CTRV configuration showed that the bow shock wave would not reach the wing's vertical stabilizers (tip fins) until after maximum dynamic pressure (Mach 5.0). The shock position during maximum heating was well inboard of these surfaces.

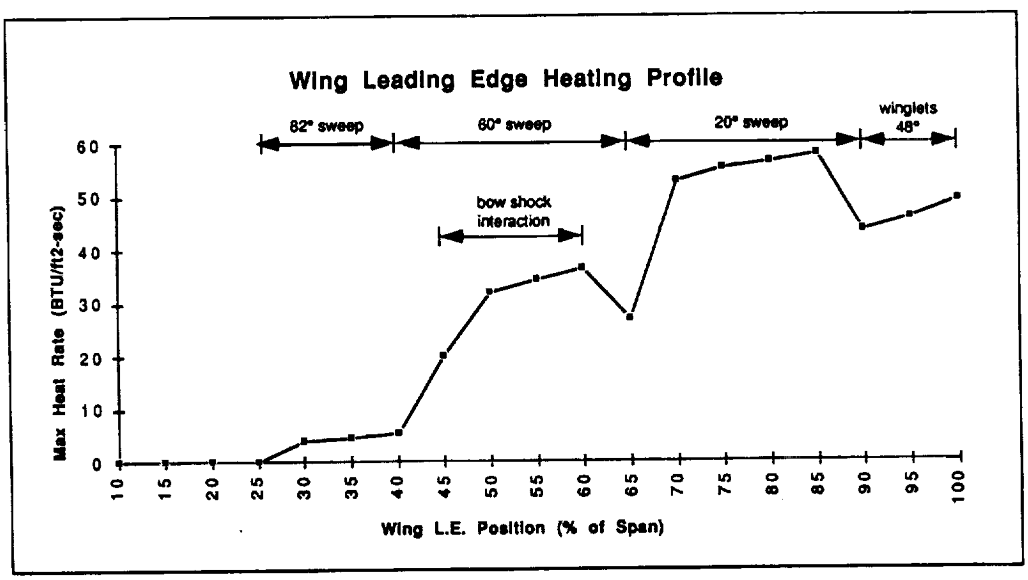


Figure 1.2-40 CTRV Wing Leading Edge Heating Rates

Subsystems Design

Achieving commonality of the Winged CTRV's Orbital Maneuvering System (OMS) and Attitude Control System (ACS) with the PLS represented a significant design trade study between the CTRV's hypergolic propellants and alternative propellants such as the hydrogen peroxide/RP-1 propellants of the PLS. An assessment of the launch processing impacts resulting from hypergolic propellant systems servicing based on actual Shuttle processing of OMS and RCS systems revealed that hypergolic systems processing typically resulted in six shifts of serial, hazardous operations (Actual servicing of the Shuttle hypergolic system components is performed off-line in a controlled facility removed from the main system processing).

Impacts to KSC operations of the proposed hydrogen peroxide system were identified based on the special characteristics of H2O2. It was found that there are currently no production facilities in the United States or Europe for propellant grade H2O2 and there are no storage facilities at KSC or CCAFS for any quantities of this propellant.

- **MONOMETHYLHYDRAZINE(MMH)**
  - CAUSTIC, LOCALLY DAMAGING TOXIC AGENT & HIGHLY FLAMMABLE
  - PROVEN WELL UNDERSTOOD SAFETY PROCEDURES IN PLACE
- **OPF*: MINOR SPILL (DROP, <1/2 CUP) DRIVES "CLEAR" AREA**
  - EVACUATE OPF BAY (100 - 200 PEOPLE)
  - UP TO 1/2 SHIFT CLEAN-UP, "SCAPE" CREW
- **OPF*: MAJOR SPILL (> 1/2 CUP)**
  - EVACUATE OPF BAY 1 & 2 (200 - 400 PEOPLE)
  - EVACUATE ANNEX OFFICES (~ 100 PEOPLE)
  - UP TO 2 SHIFT CLEAN-UP "SCAPE" CREW
- **N2O4 REQUIRES SAME PROCURES, DIFFERENT SPILL KIT**

- **HYDROGEN PEROXIDE (H2O2)**
  - UNSTABLE, SUSCEPTIBLE TO HEAT & CONTAMINATION
  - STRONG IRRITANT
  - NON FLAMMABLE, BUT ACTIVE OXIDIZER REACTING WITH FLAMMABLE MATERIALS
- **OPF*: MINOR SPILL (DROP, <1/2 CUP) DRIVES "CLEAR" AREA**
  - EVACUATE AFFECTED AREA SMALLER AREA? FEWER PEOPLE?
  - WATER DELUGE CLEAN-UP
  - SIMILAR CLEAN-UP, BREATHING APPARATUS
- **OPF*: SPILL PROCEDURES SPECIFIED BUT NOT IN PLACE**
- **NO EXISTING MANUFACTURING FACILITY (REFINERY) FOR 90 + % H202**
- **NO EXISTING STORAGE FACILITY FOR H2O2 AT KSC/CCAFS**

* OMS/RCS TANKS PURGED PRIOR TO ENTRY INTO ORBITER PROCESSING FACILITY (OPF)
SCAPE: SELF CONTAINED ATMOSPHERIC PROTECTIVE ENSEMBLE
REFERENCES: AFM 161-30, VOL, LIQUID PROPELLANTS ; GP 1098-F, KSC GROUND OPERATIONS SAFETY PLAN

Figure 1.2-41 Hypergolic vs. Hydrogen Peroxide Propellant Trade Study

Reliability and Maintainability Analyses

Reliability and maintainability analyses of the Winged CTRV concept were performed to reflect the subsystem design changes (including the addition of the CTRV's wings and aerodynamic control surfaces). The effect of these relatively more complex subsystems of the Winged CTRV (than the Medium CTRV concept) had a strong effect on the launch operations simulations. The increase in subsystems complexity resulted in an increase of predicted in-flight failures from 2.1 to 6.5. This produced a logistics delay factor of 21 times the Mean-Time-to-Repair (MTTR), a significant increase from the 8X factor associated with the earlier Medium CTRV reliability estimates. It was further found that a spares level of 95% would be required to meet the CTRV and PLS launch rates (rather than the 90% level required for the Medium CTRV).

Analysis of uncertainty in the predicted Winged CTRV and PLS subsystems turnaround processing times was also examined with the STARSIM model. The CTRV and the PLS subsystems were analyzed using both PLS-predicted fast processing timelines and using current Shuttle subsystems processing timelines. The desired flight rates were found to be still be achievable even with the Shuttle processing times. The effect of the longer subsystems processing timelines was not as significant as the effect of longer maintenance delays for spares! This demonstrated that the CTRV (and the PLS) reliability and maintainability parameters are at least as important as the launch processing times.

Comparison of manpower expenditures for both the fast (PLS-predicted) and Shuttle processing times showed only minor differences. A manpower consumption of 1.60 million man-hours per year required to process the 9.5 flights/year with the PLS timelines increased only to 1.69 million man-hours for the Shuttle timelines. This small difference is due to the low average utilization rates of several key facilities when the faster processing times are simulated. This demonstrates an important observation about launch processing costs: reducing launch processing timelines does not directly reduce launch processing costs. A better means of reducing direct (and even indirect) labor costs is to reduce the number of processing facilities required (e.g. high utilization of fewer facilities).

WINGED CTRV SPACECRAFT SOURCE: WCTRV WEIGHT STATEMENT OF 6/4/93	QTY	UNIT WT (lbs)	TOTAL WT (lbs)	LRU?	MTBR (Flt Hrs)	RECOMMENDED SPARES QTY (POS = 0.90)
ATTITUDE REACTION CONTROL & OMS	-	-	1598	NO	56	N/A
TANK (MMH + NTO)	2	502.50	1005	NO	3,027	N/A
THRUSTER (FORWARD)	18	6.10	110	YES	176	3
THRUSTER (AFT)	18	6.10	110	YES	176	3
PLUMBING	1	80.00	80	NO	968	N/A
THRUSTER (OMS)	1	35.00	35	YES	552	3
PLUMBING, VALVES, ETC.	1	258.00	258	NO	300	N/A
AVIONICS	-	-	695	NO	361	N/A
GUIDANCE, NAVIGATION AND CONTROL	-	-	435	NO	497	N/A
IMU, HEXAD	1	55.00	55	YES	6,231	1
GN&C COMPUTER	2	10.00	20	YES	17,134	1
GPS RECEIVER/PROCESSOR	2	9.00	18	YES	19,037	1
FLOODLIGHT, PTZ	2	6.00	12	YES	999	1
CAMERA, PTZ	1	78.00	78	YES	4,393	1
CONTROLLER, GIMBAL DRIVE	1	25.00	25	YES	61,965	1
CONTROLLER, RCS	1	33.00	33	YES	46,942	1
CONTROLLER, ACTUATOR	1	40.00	40	YES	38,723	1
ALTIMETER	2	5.00	10	YES	34,267	1
RF ASSEMBLY, MSBLS	2	7.00	14	YES	24,479	1
RECEIVER/DECODER, MSBLS	2	21.00	42	YES	8,159	1
TRANSDUCER, AIR DATA	2	19.00	38	YES	9,018	1
SENSOR, AIR DATA	2	25.00	50	YES	6,853	1
COMMUNICATIONS AND TRACKING	-	-	260	NO	1,318	N/A
POWER AMPLIFIER, RF	2	6.00	12	YES	28,557	1

Figure 1.2-42 Winged CTRV Spares Quantity Recommendations

Turnaround Times	Aircraft-like Operations	Shuttle-like Operations
CTRV Time in System (Work-Days)	44.4	86.7
PLS Time in System (Work-Days)	58.8	101.0
Launch Interval (Work-Days)	26.4	26.4
Manpower Requirements		
Annual Touch Labor (Million Hours)	1.512	1.690
Facility Utilization		
Landing Facility Utilization	0.042	0.042
HPF Utilization	0.326	0.857
VAB Utilization	0.648	0.935
Launch Pad Utilization	0.389	0.394
MLT Utilization	0.694	1.000
Crawler-Transporter Utilization	0.028	0.375

Figure 1.2-43 CTRV Launch Processing Manpower Utilization Analysis

1.2.3 Launch Abort Studies

An important element of manned launch systems is the ability to safely perform a launch abort in the event of a failure in a critical system component. This launch abort capability is vital to crew safety considerations and is also a major cost factor for un-manned but fully re-usable launch systems in which the reusable element represents a significant financial investment. The abort analyses performed in this study were conducted during the NASA Access to Space Option 2 study period and utilized launch vehicle and manned element concepts as defined in the NASA work. These launch abort studies directly contributed to the NASA study effort.

1.2.3.1 Abort Analysis Approach

Launch abort analyses were performed to determine the ability of four launch vehicles to perform a mission abort during the launch portion of the nominal mission (ascent trajectory phase). The analysis was performed for each Option 2 booster as defined by NASA (Boosters 2A', 2C, and 2D for the HL-42, and Booster 2B for the CLV-P). The abort trajectory analyses were performed with POST for both the ascent and re-entry/landing conditions of the reusable spacecraft. No predictions of booster stage impact points were attempted during these analyses. The launch abort modes considered in the analysis included; Return To Launch Site (RTLS), Trans-Atlantic Landing (TAL), Engine Out (EO), Abort To Orbit (ATO), and Abort Once Around (AOA). For those periods of the trajectory where above described intact abort modes are not available, the HL-42 would perform a water landing which would permit a safe recovery of the crew or cargo. The CLV-P would descend to a stable, low altitude/velocity condition for the crew bail out.

1.2.3.2 HL-42 on Booster 2A'

The abort analyses performed using the HL-42 on Booster 2A', a 1.5-stage booster for the Option 2 architecture, showed this configuration to be particularly effective for manned flights. The abort analysis revealed that 100% of the launch trajectory has an intact abort mode coverage for the HL-42 (<u>no water ditching required</u>). The EO and RTLS abort mode periods overlap, thereby providing full abort capability during the entire first stage flight.

During second stage flight, the abort modes available are the TAL, ATO, and EO aborts. These abort modes are available through the use of excess core stage propellant (since there are two engines in the core stage). At no time during the launch is the HL-42 exposed to a water landing contingency. An EO abort capability (that is, successful completion of the mission after suffering the loss of a single engine) exists for 31% of the trajectory and the alternate landing site exposure (TAL) is only 30% (48% and 26% for 28.5° trajectories).

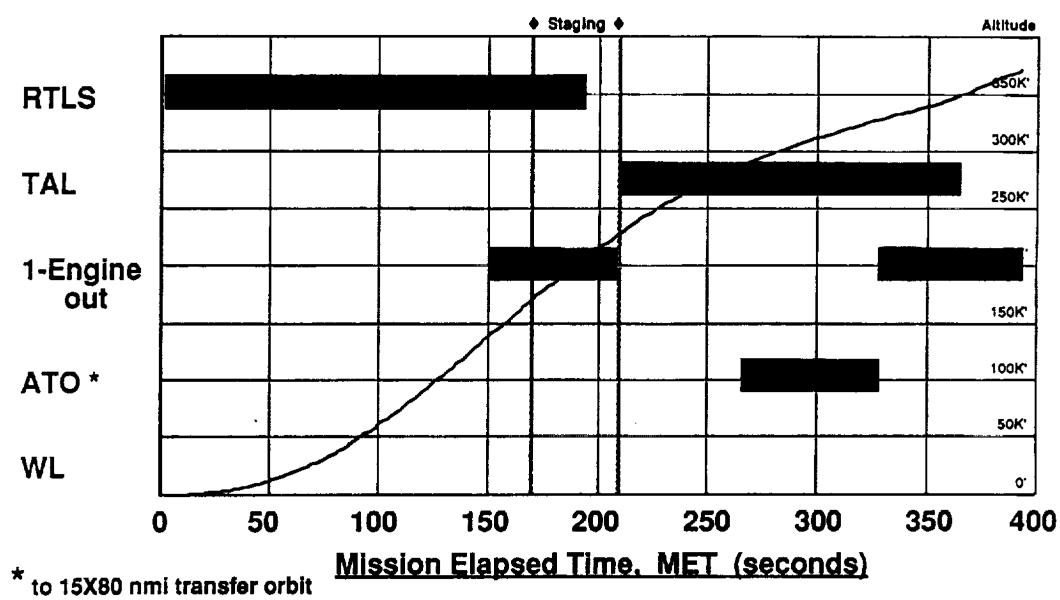


Figure 1.2-44 HL-42 Abort Coverage on Booster 2A'

1.2.3.3 CLV-P on Booster 2B

Abort analyses were next performed for the CLV-P on Booster 2B, a 2-stage, parallel burn booster for the Option 2 architecture. The abort analysis revealed that a large percentage (43%) of the launch trajectory has no abort mode coverage for the CLV-P (requiring a crew bailout over water and loss of the vehicle and payload) . During first stage flight, the RTLS abort mode is provided only after a minimum of 20 seconds after liftoff from the launch pad and until 200 seconds into the launch. Also during first stage flight, if a booster engine fails late in the burn duration (last 95 seconds), either an ATO or an EO abort option can be flown. The ATO, EO, and RTLS abort mode periods overlap, thereby providing some abort capability during all but the initial 20 seconds of first stage flight.

During second stage flight, the only abort mode available is the TAL abort, available only for the last 17 seconds of flight time. The booster second stage flight during this 198 second time period, an engine failure would force the CLV-P to perform a water ditching (and crew bailout if manned).

The Booster 2B configuration was found to provide only limited abort coverage for the CLV-P. During the launch, the CLV-P (and crew) is exposed to a water ditching/bailout contingency for 43% of the trajectory. An EO abort capability exists for 23% of the trajectory and the alternate landing site capability (TAL) is only 4% (24% and 10% for 28.5° inclination trajectories). This level of abort coverage is not considered acceptable for a new manned launch system.

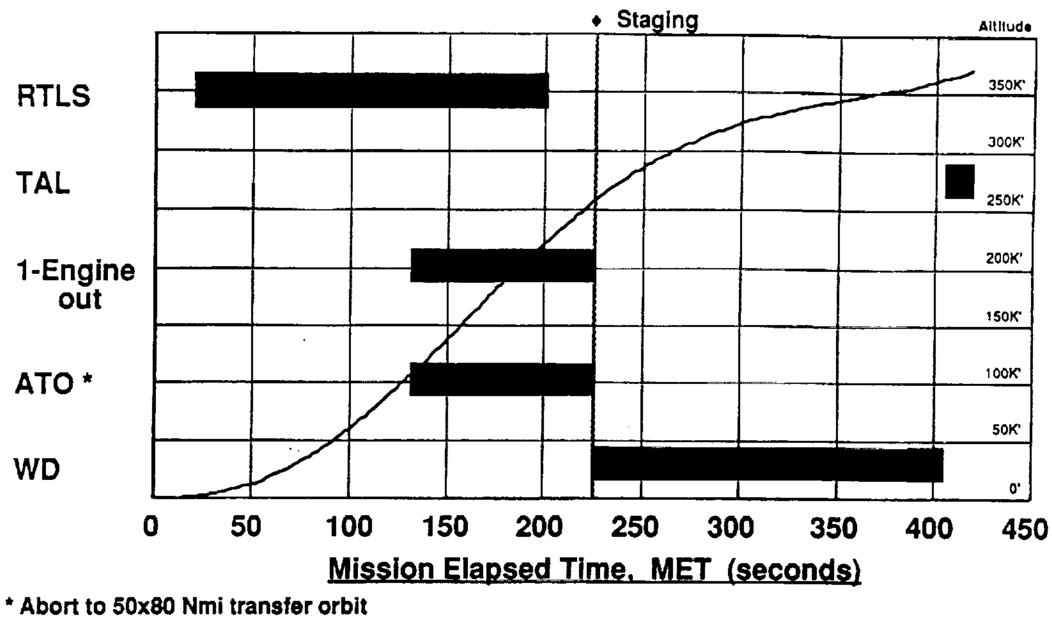


Figure 1.2-45 CLV-P Abort Coverage on Booster 2B

1.2.3.4 HL-42 on Booster 2C

Abort analyses were also performed for the HL-42 on Booster 2C, a 2-stage parallel burn, hybrid booster for the Option 2 architecture. Because the booster stages are large thrust, single engine stages (nearly 1.5 million lbs thrust each), loss of one booster stage will require shutdown of the other booster stage to maintain control of the launch vehicle. The abort analyses revealed that a large percentage (66%) of the launch trajectory has no abort mode coverage for the HL-42 (requiring a water landing/ditching). During first stage flight, the RTLS abort mode is provided from the launch pad until 130 seconds into the launch, at which time the vehicle is too far downrange for the HL-42 to return to KSC. No EO abort capability exists for first stage flight, and since the ATO capability is so limited, the RTLS abort is the only practical option available during first stage flight.

During second stage flight, the only abort mode available is the TAL abort, available only for the last 37 seconds of flight time. During this 365 second time period, an engine failure would force the HL-42 to perform a water ditching.

The Booster 2C configuration was found to provide only limited abort coverage for the HL-42. During the launch, the HL-42 (and crew) is exposed to a water landing contingency for 66% of the trajectory. No EO abort capability exists for the trajectory and the alternate landing site capability (TAL) is only 6% (7% and 11% for 28.5° inclination trajectories). This level of abort coverage is not considered acceptable for a new manned launch system.

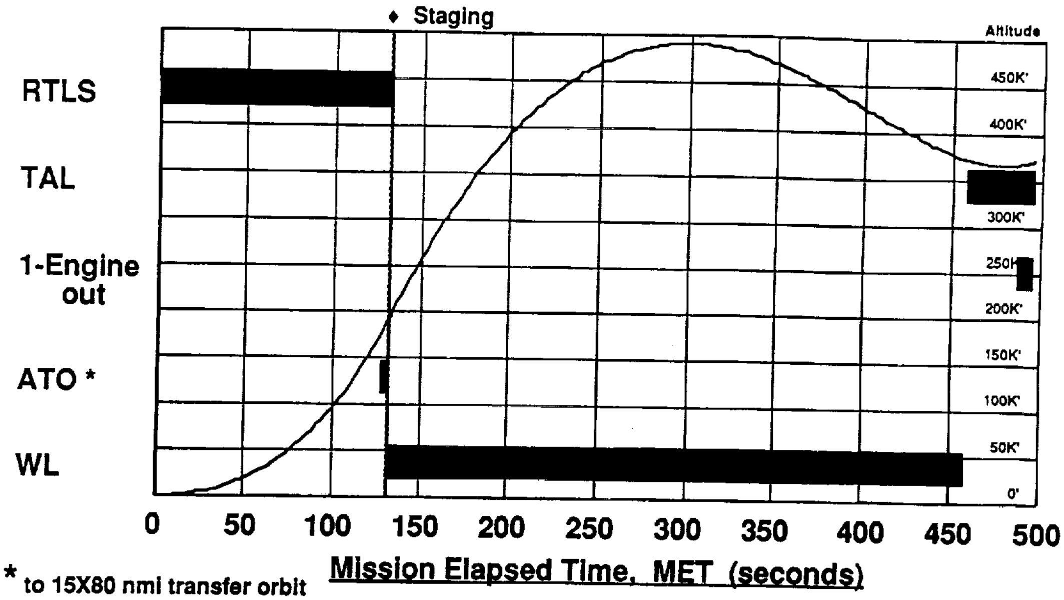


Figure 1.2-46 HL-42 Abort Coverage on Booster 2C

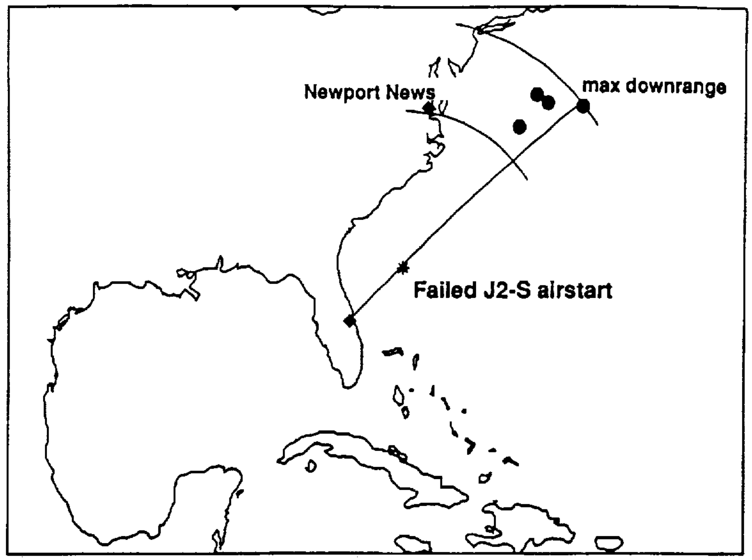
1.2.3.5 HL-42 on Booster 2D

The abort analysis performed for HL-42 using the Booster 2D, a 2-stage booster for the Option 2 architecture, was particularly interesting. This booster first stage uses three Russian RD-180 engines (LOX/RP 1st stage propellant), and the second stage uses a single J-2S engine. Because the HL-42 configuration weight is well below the maximum payload capability of this launch vehicle, there is considerable excess propellant in the booster second stage (assuming the tanks are filled to capacity for the launch). This excess propellant is valuable for abort capabilities when recovering from first stage propulsion failures.

The abort analysis revealed that a large percentage (53%) of the launch trajectory has no abort mode coverage (other than the water ditching) for the HL-42. During first stage flight, the RTLS abort mode is provided from the launch pad until 185 seconds into the launch. Also during first stage flight, if an engine fails late in the burn duration (last 58 seconds), either an ATO or an EO abort option can be flown. Both of these abort modes utilize propellant margins in the second stage to make up the velocity shortfall of the first stage failure. The ATO and RTLS abort mode periods overlap, thereby providing some abort capability during the entire first stage flight.

During second stage flight, the only abort modes available are the TAL and engine out (EO) aborts. The TAL abort mode is available for the last 65 seconds of flight time and the second stage EO abort mode (a short duration of just 8 seconds) is enabled only by use of the HL-42 abort motors. The booster second stage flight is thus found to have no abort coverage from its single engine start until the last 65 seconds.

Of particular concern for this launch vehicle was the impact of a failure to start the second stage engine (generally regarded as a high risk event). Because of the HL-42 vulnerability to this risk (a water landing), a determined effort was made to find a means of performing a runway recovery for the HL-42 for this condition. Specifically, a North America Landing (NAL) abort mode was devised to protect the system from failure of the Booster 2D second stage engine (a J-2S) to ignite. No other intact abort modes were available to the HL-42 for this failure event (too far downrange for RTLS, not enough downrange for TAL).



HL-42 abort motors & on-board propulsion plus another 1,600 fps required to achieve a runway landing

Figure 1.2-47 Second Stage Engine Ignition Failure Effects

To achieve the NAL abort, the Booster 2B must be flown with off-loaded second stage propellant in order to increase the staging velocity. A propellant off-load of 116,050 lbs in the second stage is possible, resulting in a faster and longer first stage trajectory. Operation of the booster in this fashion increased the staging velocity by 3,000 feet per second and thus created enough energy at staging that the HL-42 can reach landing sites in New England and Canada should the J-2S engine fail to start. The RTLS capability was reduced by 29 seconds and the EO and ATO capabilities were completely eliminated. The total exposure to water landing, however, was reduced significantly; from 297 seconds during second stage flight to only 55 seconds during first stage flight. The booster second stage propellant load was found to play the deciding role in the HL-42 abort capabilities on this launch vehicle. Excess propellant in the second stage was the key parameter for extending the coverage of the RTLS, ATO, and EO aborts. Removal of this excess propellant, on the other hand, was required to protect against failure of second stage engine ignition. The significantly reduced HL-42 exposure to water landing (from 53% to only 13% of the trajectory) was noted as a key factor favoring the off-loaded propellant approach.

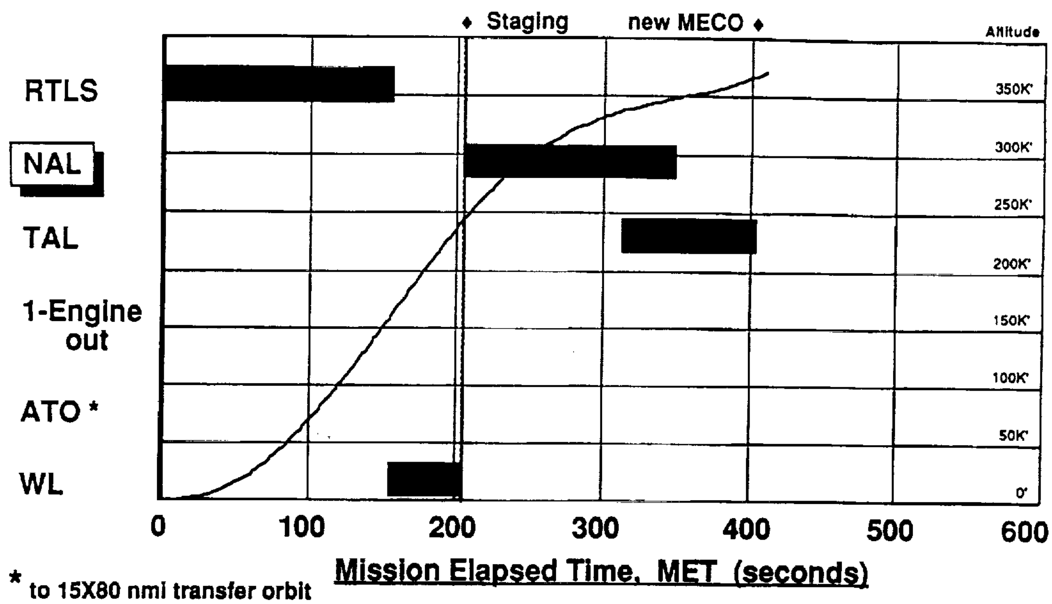


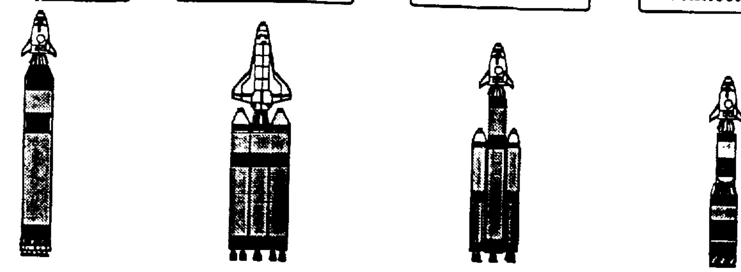
Figure 1.2-48 HL-42 Abort on Booster 2D, Off-loaded Propellant

1.2.3.6 Launch Abort Findings

Only the 2A' booster configuration has sufficient intact abort coverage to completely eliminate the spacecraft water landing exposure. For manned spacecraft flights, the Booster 2A' configuration is clearly superior to the other booster configurations analyzed.

For booster configurations 2B, 2C, and 2D, the analyses showed that a large percentage (generally 40 to 60%) of the launch trajectory has no intact abort mode coverage. This is caused by the single engine operation of these designs during second stage flight. During first stage flight, the RTLS abort mode is generally provided from the launch pad until approximately 200 seconds into the launch. The ATO and EO abort mode periods usually overlap with the RTLS abort mode, thereby providing some abort capability during the entire first stage flight. During second stage flight (post booster or engine staging), the only intact abort modes available are the TAL, ATO, and EO aborts. Where there is only a single engine in the booster's second (or core) stage, these abort modes are available only at the very end of the trajectory.

The booster second (or core) stage was clearly found to play a crucial role in the HL-42 and CLV-P abort capabilities on these launch vehicles. Utilization of this stage's engine and propellant plays a significant role in late RTLS capabilities. A second engine on these stages (available only on the Booster 2A' configuration) provides a much needed extended intact abort coverage.



	Architecture 2A'	Architecture 2B	Architecture 2C	Architecture 2D
Mission Success after Failure (%)				
28.5°	48	24	2	7
51.6°	31	23	2	0
Alternate Landing Site Exposure (%)				
28.5°	26	10	11	12
51.6°	30	4	6	49
Water Landing Exposure (sec)				
28.5°	0	156	304	297
51.6°	0	181	328	55

Figure 1.2-49 Summary of Launch Abort Analyses

1.2.4 SSTO MPS Operability Studies

The principal objective of all proposed SSTO launch vehicle concepts is a dramatic improvement in vehicle operability. Regardless of which SSTO design concept is eventually selected for development (VTOHL or VTOVL, LOX/LH2 or tri-propellant, etc.), all concepts must achieve significant reductions in ground processing timelines and the supporting workforce in order to meet the proposed cost benefits of an SSTO program. Under the ATSS study, a method which was first used in the design of the Lockheed L-1011 aircraft was adapted for analyzing the operability characteristics of the reference SSTO concept as defined by NASA in their recently completed Access to Space study, Option 3.

1.2.4.1 Operability Analysis Approach

The operability approach used is one in which the vehicle flight and ground operations are analyzed by computer simulations. The simulations include the flight operation and both scheduled & unscheduled maintenance operations for all vehicle components. The components' performance is determined from current (or projected) component reliability and maintainability histories. The SSTO Main Propulsion System (MPS) was selected for this operability analysis. The MPS is not only a critical subsystem of any SSTO concept, but it is, historically, one of the most difficult to process. Differences among the several SSTO concepts are clearly reflected in their MPS designs, so this subsystem also serves as a useful benchmark for comparing the operability of competing concepts.

The source of component reliability and maintenance data to support the simulation models was obtained from the Space Shuttle program. The Space Shuttle program's PRACA database was used to collect all problem reports (PRs) on the MPS components since the Challenger accident. Simulation models for analysis of the SSTO MPS ground processing were developed from Rockwell's SIMtrix and STARSIM computer codes. The SIMtrix model analyzes reliability, maintainability, and logistic parameters to determine the effects of unscheduled maintenance on the planned SSTO MPS ground processing estimates. The STARSIM model was used to analyze the SSTO launch rate capability and launch facility needs/utilization based upon data provided by the SIMtrix model. This model operates at the system level, but accepts data at the subsystem level (such as from SIMtrix) for calculating the total vehicle ground turnaround timelines. The SSTO ground processing timelines were based on the Access to Space Option 3 study's SSTO groundrules.

SIMtrix

Monte Carlo simulation of scheduled and unscheduled maintenance & repair activities for specified ground processing sequences and timelines

Includes:
- component MTBF, MTBM
- component MTTR
- spares POS and RTAT
- undetected failures

STARSIM

Probabilistic simulation of launch systems and facility/resource utilization for specified launch rates and launch processes

Includes:
- launch vehicle subsystems
- payload integration
- facility constraints
- manpower allocations

Figure 1.2-50 Operability Analysis Software Tools

Functional Failures: Component fails to perform to specified levels during ground processing (e.g. leaks, valve fails to open, ...)

Inspection Defects: Component has been improperly installed or damaged (e.g. scratched, dented, contaminated, misaligned, ...)

	Shuttle MPS Component PRs		
	Funct.	Defect	Total
Valves	81	91	172
Lines & manifolds	39	313	352
Helium tanks	41	54	95
Regulators	47	21	68
Disconnects	44	201	245
Filters (He system)	0	0	0
Sensors (temp, press)	46	24	70
Total	298	704	1002

Figure 1.2-51 Shuttle MPS Component Maintenance Record

1.2.4.2 MPS Operability Analysis Results

The SIMtrix model was first checked against the actual Shuttle experience as a benchmark test. The simulation predicted Space Shuttle MPS mean down time and a mean unscheduled maintenance manhour requirement which correlated reasonably close with the actual Shuttle experience. The SIMtrix model of Shuttle's MPS ground processing provides a good representation of the current state of the art (SOA) for reusable MPS hardware. When the scheduled ground test time for the SSTO's MPS was reduced from 700 to the specified 40 hours, the PRs dropped to 19 per flow. This resulted in a mean downtime of only 160 hr's, and only 1,323 unscheduled maintenance manhours, less than half of what was estimated for the Space Shuttle today. This significant improvement in required maintenance is an indicator of how much of the Shuttle MPS hardware life is being consumed by ground testing.

The SIMtrix analysis identified the time to perform the subsystem test and checkout as the most important factor for reducing turnaround times and costs. By drastically reducing the test time of flight hardware, equipment operating times are reduced and the number of failures (PRs) decrease accordingly. The next most significant factor was the reduction of time to remove and replace (or just to repair) a defective component. This factor directly reduces the maintenance time (MDT) and labor (UMMHR) to return the vehicle to an operational condition. These two factors both result in shortening the total time the SSTO is in the processing facility.

The total time the SSTO is in its processing facility was the most important factor in achieving high flight rates. The effect of maintenance down periods (OMDP) was found to not be a strong factor in achieving high flight rates, but it did affect the SSTO operations costs. Significant variations in both the frequency and time to perform OMDP maintenance can be tolerated without reducing the annual flight rates. The additional labor required to perform the maintenance, however, is directly related to the time and frequency of these events.

Comparison of a tri-propellant propulsion system concept with the reference (LH2/LOX propulsion) SSTO found that either concept can achieve the SSTO flight rate objectives, but higher maintenance costs should be expected with the tri-propellant design. Even with the 3-engine RD-701 concept, higher maintenance costs were found than the 7-engine SSME concept.

Extrapolation from Shuttle Technology to SSTO Technology

	PRs #	MDT (hrs)	UMMHR (hrs)	Total HPF Time (hrs)	Flight Rate	
Shuttle Reference with 7 SSMEs	60	499	4,493	539	7	Today's
IVHM to checkout subsystems in 40 hrs.	19.4	160	1,323	200	43	Testing
Improved designs to reduce remove/repair times (MTTR) to 6 hrs	19.6	55	600	95	51	Design
Improved component reliability (MTBF or MTBM) by 50%	16.5	51	535	90	51	Components

Figure 1.2-52 Technology Steps from Space Shuttle to SSTO

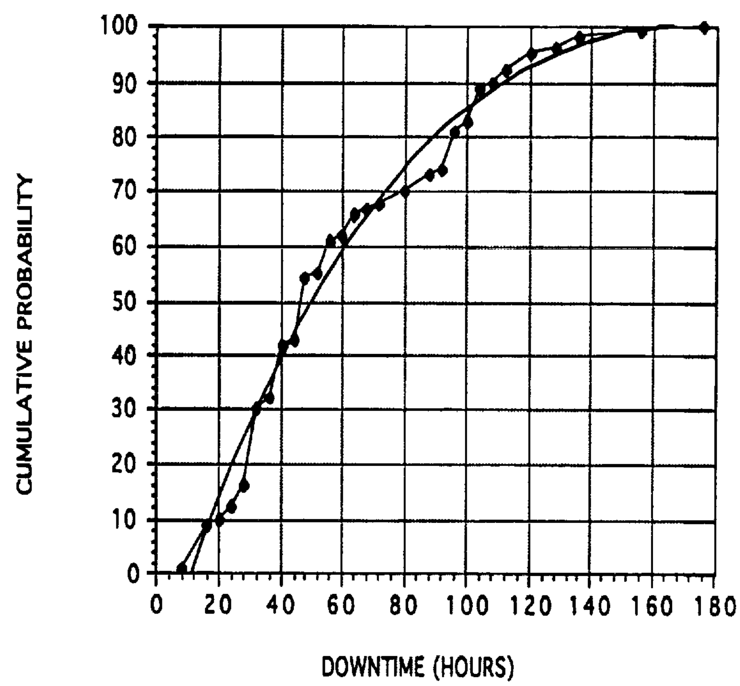


Figure 1.2-53 SSTO MPS Mean Maintenance Downtime

The STARSIM model was used to perform a basic sensitivity analysis of the annual SSTO flight rate capability. The model varied several operational parameters such as the interval and duration of the SSTO Operational Maintenance Down Periods (OMDP), fleet size (number of SSTO vehicles), and the mean Horizontal Processing Facility (HPF) turnaround time. The sensitivity analysis showed that the planned SSTO flight rate (43 flights/year) is easily achievable under the current groundrules. The effects of a wide range of OMDP variables was found to have a weak effect on the SSTO annual flight rate. The results indicate that considerable margin exists in this scenario for scheduled depot maintenance and/or modification for the SSTO vehicles. The resulting cost per flight (as measured in direct labor manhours per launch) of these OMDPs was more significant, however. The SSTO cost per flight varied by +45% to -20% over the range of the OMDP variables.

The effect of a longer than ground ruled HPF processing time had a direct and strong effect on the SSTO flight rate capability and on the manhours per flight. The flight rate correlation with HPF processing time was not linear, and the rate did not drop below 50 flts/yr until the HPF time increased to 120 hours. The planned 43 flts/yr was not met when HPF times increased to approximately 220 hours. The effect on per launch manpower (cost) was very strong, but also not linear. The processing times increased by a factor of 12 in the worst scenario, but the manpower per launch only increased by a factor of 2.5. This effect is caused in part by the reduced number of OMDPs performed each year (from 2.5 to 1) because of the reduced flight rate.

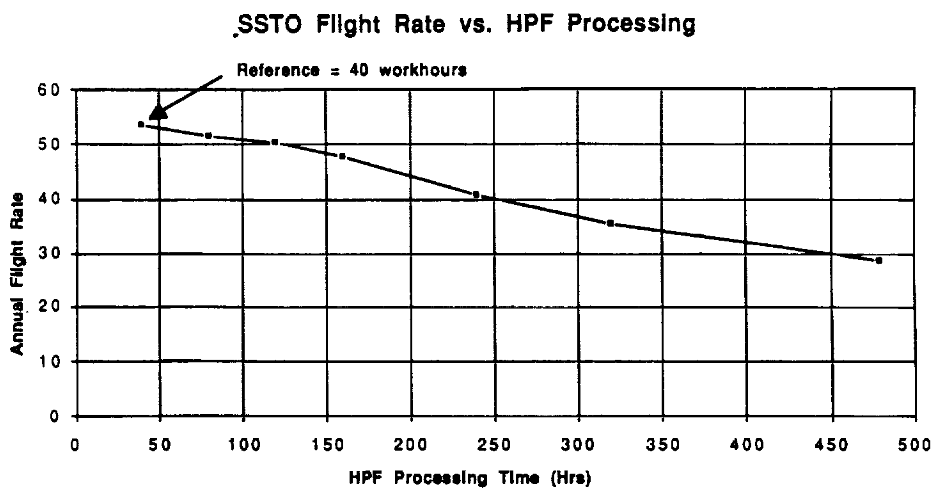


Figure 1.2-54 SSTO Flight Rate vs. HPF Processing Time

1.2.4.3 SSTO Engine Throttling Analyses

A key factor in any SSTO launch vehicle's maintenance characteristics is the amount of time the main propulsion system engines have to operate on each mission. An analysis was performed to determine what engine throttling and shutdown schemes might be devised to minimize total engine operating time and engine operating times at full throttle for a reference SSTO concept. Several engine operating schemes were investigated for the 7 SSME SSTO concept, each evaluated for total engine operating time, time at full and reduced power levels, payload impacts, and also for single or two engine out abort capability. The tri-propellant RD-701 propulsion system SSTO concept was also evaluated for comparison with the reference SSTO configuration.

Throttle profile variations produced little change in payload performance and all resulted in the eventual shutdown of five engines to meet the 3g acceleration limit. A range of 400 seconds from maximum to minimum total operating times was found over the nominal 2200 sec. total engines operating time. This is not a great variation for a single mission, but when applied to a planned 20-mission life between engine removals, this translates to an equivalent of two additional missions before planned removal of the engines, which is a significant maintenance improvement.

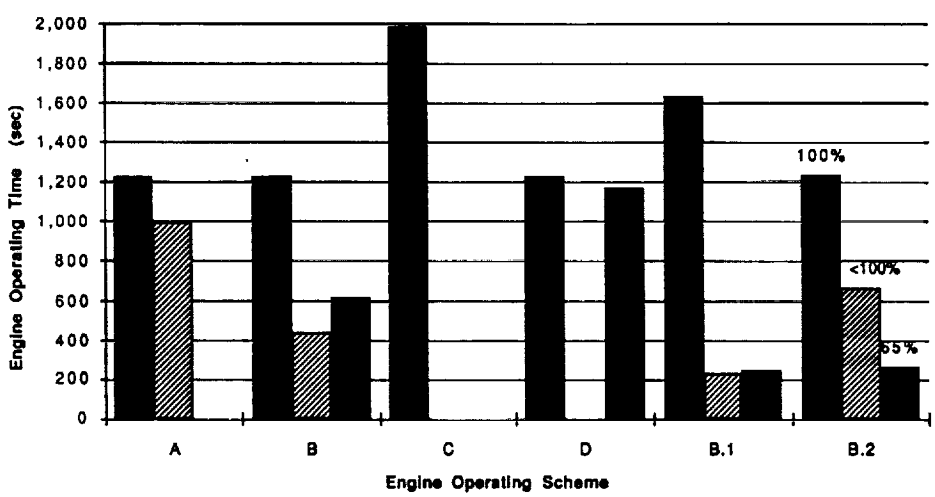


Figure 1.2-55 SSTO Engine Throttling Options

1.2.4.4 SSTO Engine Out & Abort Analyses

Engine out and abort analyses were also conducted for these engine throttling techniques to determine if they would improve (or reduce) the SSTO abort capabilities. The asymmetric throttle profile produced the best results for a single engine out (EO) abort. For 2 engine out conditions, the "no throttle" scenario proved best. These EO and RTLS abort analyses produced the similar results as was found in the Access to Space Option 3 study. The combined RTLS and engine out capability of this 7 SSME SSTO vehicle provides full abort coverage. That is, a runway landing option is available over the entire launch trajectory.

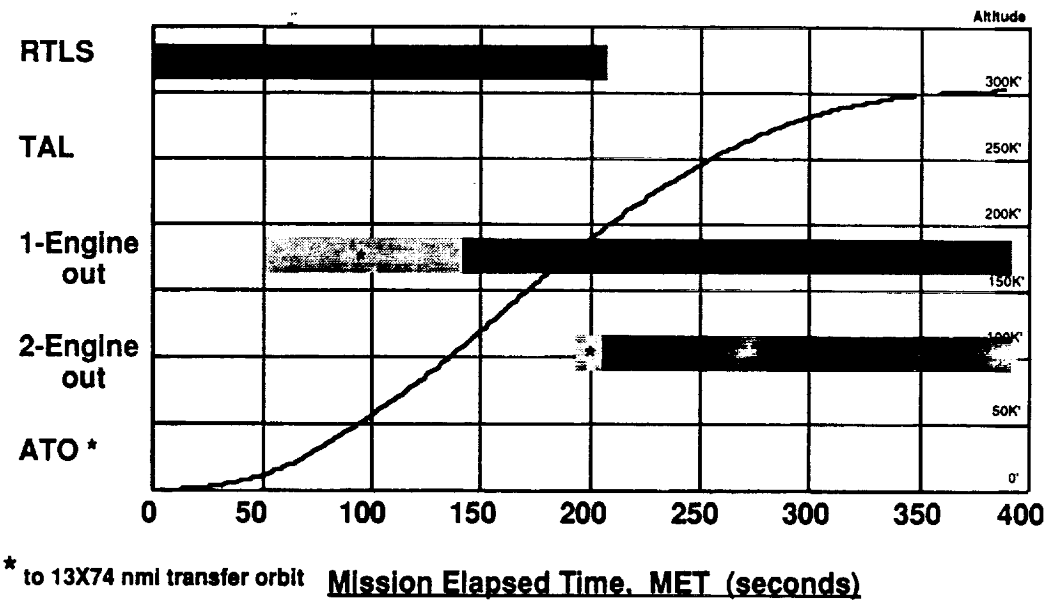


Figure 1.2-56 SSTO Launch Abort Coverage (7 SSME)

Trajectory and launch abort analyses of a tri-propellant SSTO propulsion system utilizing three Russian RD-701 engines were also performed. The analyses were similar to those performed for the seven SSME engine propulsion concept for SSTO. The fewer number of engines, coupled with their dual thrust level, was found to eliminate the need for engine throttling studies as was performed for the SSME concept. What was discovered, however, was that this concept has very limited engine out and RTLS abort capabilities. A single engine out capability was not achievable until 310 seconds into the trajectory (nominal MECO occurs at 373 sec.). Two engines out could not be tolerated at any time. The RTLS capability was greatly reduced by the significant thrust loss of an engine early in the trajectory. The RTLS could not be performed for an engine failure any earlier 37 seconds nor any later than 189 seconds after liftoff. An ATO abort is required to cover the gap (121 seconds) between RTLS and EO aborts . A single engine failure can not be tolerated early in the trajectory and a two engine failure cannot be tolerated at any time. The combined RTLS, EO, and ATO abort capabilities for this concept result in less than full abort coverage (90%). This level of abort coverage is only marginally acceptable for a fully reusable launch vehicle.

The same tri-propellant SSTO concept with seven, single-nozzle engines was also performed (the RD-704 engine concept). This version of the tri-propellant engine SSTO design produced a much improved abort performance. The RTLS capability was extended back to liftoff and out to 198 seconds. A single engine out capability was achievable at 262 seconds into the trajectory, and two engine out abort capability was achievable at 265 seconds (nominal MECO occurs at 390 seconds). An ATO abort mode was still required to span the gap (64 seconds) between RTLS and EO abort coverages. The EO abort performance of this concept is not as good as the 7 SSME propulsion system (33% vs. 64% of the trajectory), but at least this configuration achieved the full abort coverage which the RD-701 configuration could not.

These findings demonstrate the key engine parameters which determine the SSTO vehicle's abort coverage. Not only the number of engines, but also the thrust levels (especially for the dual thrust-level tri-propellant engines), determine the abort capability of the SSTO concept. The engine throttling/shutdown sequence and the vehicle aerodynamic characteristics (e.g. lift/drag ratio) have only secondary effects.

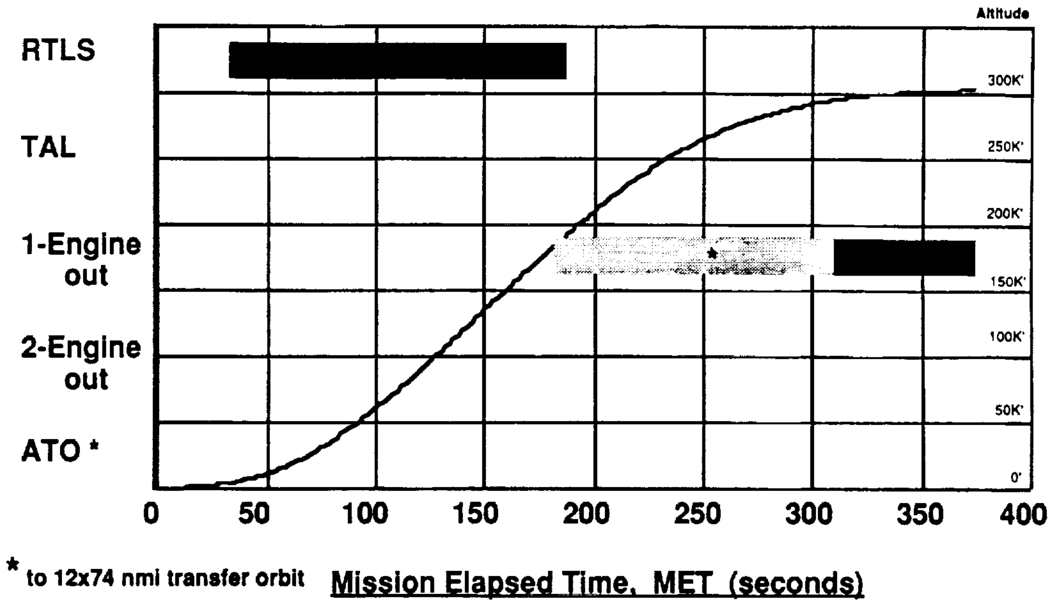


Figure 1.2-57 SSTO Launch Abort Coverage (3 RD-701)

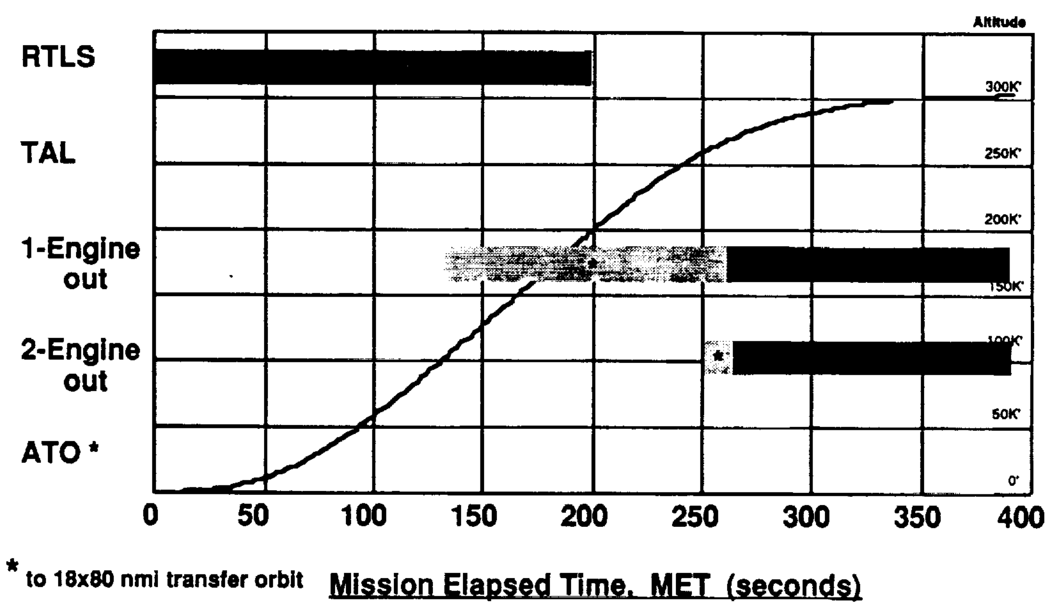


Figure 1.2-58 SSTO Launch Abort Coverage (7 RD-704)

1.2.4.5 SSTO MPS Design Layout

A key aspect of ground processing is access to MPS components to perform necessary maintenance and inspections. A design concept for an open boattail was initiated to support the SSTO MPS ground processing study. Many MPS design groundrules were identified in the Operationally Efficient Propulsion System Study (OEPSS) by NASA KSC and Rocketdyne were incorporated. The design concept is similar to that employed on the Saturn S-II and S-IVB stages in which the engine thrust structure is integrated with the tank lower bulkhead. The design also includes modular engine assemblies which integrate the engine with the TVC system and portions of the thrust structure. No closed compartments exist in the propulsion system region and considerable access is provided for engine and feedline maintenance. A three-point structural attachment was developed for the engine module to accommodate rapid engine replacement.

A rocket propulsion based SSTO(R) Access to Space Option 3

- **VTHL**
- **LOX/LH2 propellants**
- **7 evolved SSME engines**
- **Forward LOX tank with two 19" feedlines, toroidal manifold**
- **LH2 tank with spider manifold**
- **Electromechanical actuators**
- **Hot gas tank pressurization**

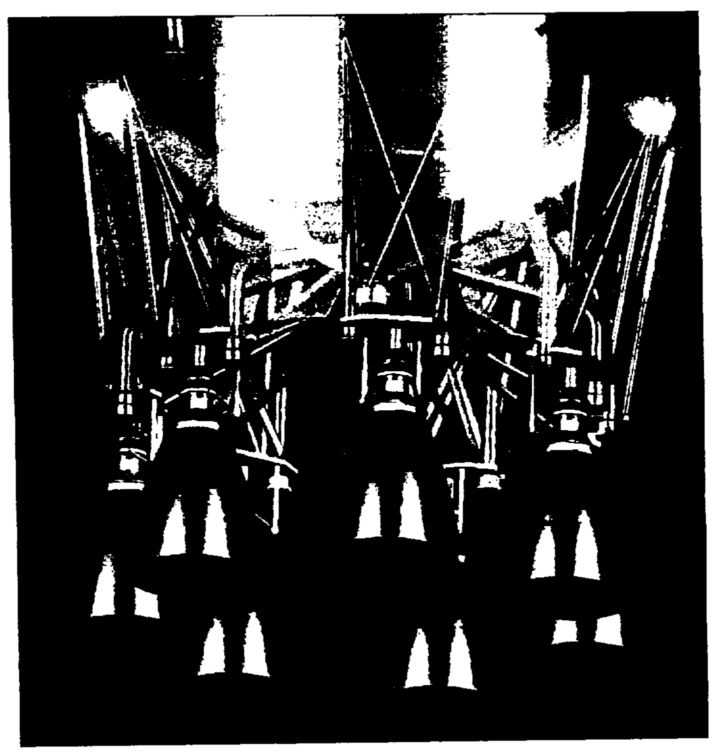


Figure 1.2-59 SSTO MPS Design Layout per OEPSS Guidelines

Additional Information

More in-depth discussions and more detailed information regarding the study findings which have been presented here may be found in Volume II and Volume III of this Final Report. Volume II contains additional information on the technical aspects of manned booster and cargo transfer vehicle concepts which were examined during the study. Volume III contains cost data and estimating techniques used for these concepts.

ADVANCED TRANSPORTATION SYSTEM STUDY

Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO

FINAL REPORT (DR-4)

Contract NAS8-39207

Submitted by:

J. B. Duffy
Rockwell International

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SSD93M0025-1

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Forward

This report is submitted in compliance with DR-4 of Contract NAS8-39207, Advanced Transportation System Studies for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center. The report describes the results of Rockwell International's work for the analysis of Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO during the Basic Contract and the Option 1 Contract period of performance (February 15, 1992 through December 15, 1993).

This report is organized in three volumes; an Executive Summary, a Final Report, and a Cost Estimates Report.

The Executive Summary is a condensation of the study's major findings and a summary of results for the several study activities. The findings and results are current with the study progress as of December 10, 1993.

The Final Report volume is an in-depth description of work performed during the study, with accompanying illustrations of briefing charts and other documents which were generated during the course of the study. This volume is organized by subject matter and includes an appendix with research reports of detailed analyses on selected special topics. Sufficient data is presented in this volume to reveal the depth of work performed and to provide data which supports the findings presented in the Executive Summary.

The Cost Estimates Report is a compilation of the Work Breakdown Structure and cost estimating techniques which were used to evaluate the several booster concepts during the course of the study. A summary of data used and generated during the evaluation of each booster type (or family of boosters) is provided. The data is organized by booster types which represent unique cost estimating conditions, such as the reusable Space Shuttle, the proposed low cost NLS family, existing expendable launch vehicles, and the Russian (C.I.S.) launch vehicles.

ABSTRACT

The purpose of the Advanced Transportation System Study (ATSS) Task Area 1 study effort is to examine manned launch vehicle booster concepts and two-way cargo transfer and return vehicle concepts to determine which of the many proposed concepts best meets NASA's needs for two-way transportation to low Earth orbit. The study identified specific configurations of the normally unmanned, expendable launch vehicles (such as the National Launch System family) necessary to fly manned payloads. These launch vehicle configurations were then analyzed to determine the integrated booster/spacecraft performance, operations, reliability, and cost characteristics for the payload delivery and return mission. Design impacts to the expendable launch vehicles which would be required to perform the manned payload delivery mission were also identified. These impacts included the implications of applying NASA's man-rating requirements, as well as any mission or payload unique impacts.

The booster concepts evaluated included the National Launch System (NLS) family of expendable vehicles and several variations of the NLS reference configurations to deliver larger manned payload concepts (such as the Crew Logistics Vehicle (CLV) proposed by NASA JSC). Advanced, clean sheet concepts such as an F-1A engine derived liquid rocket booster (LRB), the Single-Stage-to-Orbit rocket, and a NASP-derived aerospace plane were also included in the study effort. Existing expendable launch vehicles such as the Titan IV, Ariane V, Energia, and Proton were also examined.

Although several manned payload concepts were considered in the analyses, the reference manned payload was the NASA Langley Research Center's HL-20 version of the Personnel Launch System. A scaled up version of the PLS for combined crew/cargo delivery capability, the HL-42 configuration, was also included in the analyses of CTRV booster concepts.

In addition to strictly manned payloads, two-way cargo transportation systems (Cargo Transfer & Return Vehicles) were also examined. The study provided detailed design and analysis of the performance, reliability, and operations of these concepts. The study analyzed these concepts as unique systems and also analyzed several combined CTRV/booster configurations as integrated launch systems (such as for launch abort analyses). Included in the set of CTRV concepts analyzed were the Medium CTRV, the Integral CTRV (in both a pressurized and unpressurized configuration), the Winged CTRV, and an attached cargo carrier for the PLS system known as the PLS Caboose.

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Volume I Executive Summary

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Volume III Program Cost Estimates

2.1 Manned Booster Studies

The ATSS contract has studied the performance of a number of expendable booster concepts currently being considered for the launch of reusable (manned or two-way cargo) spacecraft. Analyses were performed to determine which of the many proposed booster concepts best meets NASA's needs for two-way transportation to low Earth orbit. The study identified specific configurations of the normally unmanned, expendable launch vehicles (such as the National Launch System family) necessary to fly manned payloads. These launch vehicle configurations were then analyzed to determine their performance, operations, reliability, and cost characteristics for the manned payload delivery mission. Design impacts to the expendable launch vehicles which would be required to perform the manned payload delivery mission were also identified. These impacts included the implications of applying NASA's man-rating requirements, as well as any mission or payload unique impacts.

The booster concepts evaluated included the National Launch System (NLS) family of expendable vehicles, the ESA Ariane V, and several variations of the NLS reference configurations to deliver a variety of manned payload concepts. Advanced, clean sheet concepts such as a F-1A engine-derived Liquid Rocket Booster (LRB), the Single-Stage-to-Orbit rocket, and a NASP-derived aerospace plane were also included in comparisons of the several candidate booster configurations. Existing expendable launch vehicles such as the Titan IV and the Russian Energia and Proton launch vehicles were also compared to the proposed new booster designs.

Although several manned payload concepts were considered in the analyses, the reference manned payload was the NASA Langley Research Center's HL-20 version of the Personnel Launch System (PLS). Other concepts such as the Crew Logistics Vehicle (CLV) proposed by NASA JSC and a small cargo carrier to be launched with the PLS for combined crew/cargo delivery capability (the PLS Caboose configuration) were also included in the comparisons of booster configurations. The reference mission used in the analyses was the Space Station crew/cargo resupply mission.

2.1.1 Man-rating Requirements Analysis

NASA's current man-rating definition is based on JSC-23211, "Guidelines for Man-rating Space Systems", prepared by NASA/JSC. The JSC-23211 man-rating guidelines are broken down into eight major topics, which constitute the major facets of a man-rated space program. These topics are consistent with a well-balanced and effective man-rated space program. Man-rating guidelines and functions were extracted from this and several other documents which address man-rating requirements for previous man-rated launch systems. Four of these documents are requirements documents, several of which are currently in effect for the Space Shuttle program. The other four documents provide useful perspectives on the subject of man-rating and how it has been applied on various programs. Guidelines and requirements are entered at the appropriate sub-topic level (where applicable). Man-rating philosophies which have been employed almost universally on all man-rated vehicles include the following:

- Keep designs as simple as possible.
- Follow proven, well-established design standards.
- Base design factors on industry standards.
- Provide redundancy for all single failure points. Where redundancy is impractical, provide a conservative safety factor to that point.
- Use proven technology. If advanced technology is required, a technology development program should precede any preliminary design phase.
- Effective unmanned testing of actual systems should be required prior to any manned tests. Testing should prove that such features perform as required.
- Safety decisions have precedence over other programmatic requirements.
- The assurance of man-rating guidelines/features should be constantly monitored at all management levels throughout the program.
- Risk assessments should be performed to determine the impact on the system from cost, manpower, and scheduling, and to identify areas susceptible to undue safety hazards.
- Hardware and processes with unresolved anomalies should not be used. All anomalies, failures, etc., should be reasonably understood and corrective measures verified prior to approval for use.
- Safety criteria and practices should be applied to all system elements and mission phases with equal rigor.

X-15	a rocket powered <u>airplane</u>
MERCURY - REDSTONE	escape system, emergency detection, manual abort
MERCURY - ATLAS	fully automatic flight & abort
GEMINI - TITAN II	redundancy & malfunction detection, manual abort
APOLLO / SATURN	multiple redundancy, high reliability, malfunction detection & correction, auto & manual aborts
SPACE SHUTTLE	multiple redundancy, high reliability, malfunction detection & correction, intact abort
• • •	??

Figure 2.1-1 Evolution of Man-rating Requirements

2.1.1.1 Requirements Analysis

To determine how these requirements would affect a new manned launch vehicle, the guidelines and requirements from the selected reference documents were separated into design guidelines and functional requirements. The functional requirements are of the greatest impact to the design of the launch system elements. Next, all of the identified functional requirements were converted into functional flow diagrams. This process permitted functional interactions among requirements to be identified. Allocation of these functions to selected system elements was then performed. This allows for a direct comparison of man-rating requirements (function flows) to the function flows of proposed man-rated launch vehicles. A software tool (System Architect) automated the process of generating, manipulating, organizing, and managing system functional requirements diagrams. This tool was used to store and analyze the man-rating requirements and to perform the process of allocating requirements among system elements.

This system engineering process resulted in the condensation of a great number of disparate requirements down to a short list of functional design requirements for an otherwise unmanned launch vehicle. In addition to the expected high reliability parts, redundancy levels, and traceability requirements, the analysis identified those functional design requirements which the booster will have to perform to launch manned payloads. The functional design requirements were ultimately able to be grouped into two basic functions which the booster must perform: provide two-way communications between the booster and the crew, and provide the capability to perform alternate missions in a single launch.

The first added design function is the requirement to establish a two-way communications link with the manned capsule. The booster's communication link must transmit the status of critical booster systems to the manned payload (crew) and at all times protect the crew from catastrophic failure of these critical booster systems. This protection of the crew from booster equipment malfunctions includes the issuance of automatic escape commands to the payload's escape system. The booster must also provide a means for the crew to directly communicate with the booster and issue commands to override certain booster functions. For example, shutting down the booster engines to permit the escape system to separate with sufficient velocity for a crew initiated abort. Together, these communication channels must provide the crew with the capability of safely separating from the booster under all flight conditions.

The second added design function is the requirement to perform alternate missions during the launch phase. The alternate missions can be expressed as alternate main engine cutoff (MECO) targets for the booster. These targets represent mission abort and crew escape scenarios which the booster must be capable of providing. The mission abort targets normally would represent flight conditions which permit the payload (crew capsule/system) an opportunity to perform one of several abort maneuvers (listed in order of priority):

1. Abort to orbit (at reduced velocity or altitude)
2. Trans-Atlantic abort (e.g. African landing site)
3. Return-to-launch-site abort
4. At-sea abort (water ditching)
5. Escape (immediate separation from the booster)

Execution of these alternate missions would require changes to some of the booster hardware and software systems. The booster must be capable of executing one or more of these missions at all times, including while on the launch pad. The goal would be to maintain the highest level mission capability for the maximum amount of time. The booster must have sufficient computational power to calculate the current mission MECO target capabilities and to choose the highest priority target.

The determination of just which data and commands are to be transmitted between the booster and the crew is a key design trade for which there are many solutions (as seen in previous man-rated programs) and even more opinions. A suggested list of data which should be monitored has been generated based on current definitions of the NLS booster design. These data would be monitored by the booster Emergency Detection System (EDS), but few if any would need to be actually transmitted to the crew.

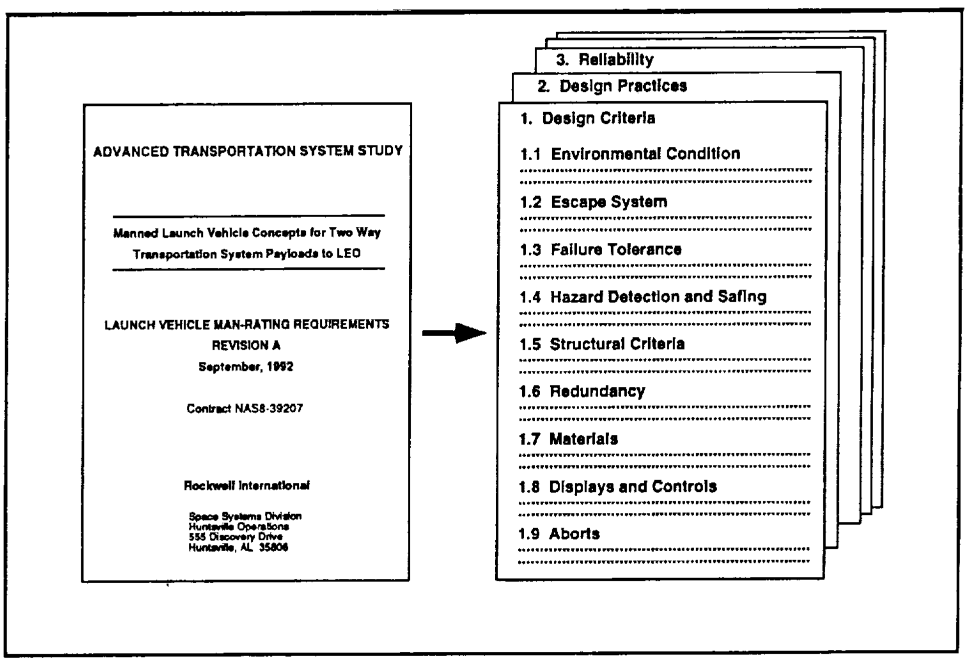


Figure 2.1-2 Requirements Based on JSC-23211

1.2 Escape System:

Guidelines:

Provisions should be made fora viable means of escape from the system in the event of an impending catastrophe. In situations where immediate and complete escape may not be feasible, an alternative approach can be considered if it can be proven reliable.(1)

The flight crew must be provided with a viable means of escape from the space vehicle in the event of an impending catastrophe. In situations where immediate escape from the space system may not be feasible, alternate approaches such as a safe haven may be considered if such can be provided to be viable.(4)

Functional Requirements:

Crew to be provided abort sensing and implementation data upon which to base an abort decision.(1)

Emergency considerations have become a distinguishing characteristic of man-rated systems; the resulting vehicle system changes and additions provide for emergency detection, control, and/or escape of the crew.(2)

The provision for a safe landing area for the spacecraft and the surrounding facilities is required.(2)

An escape system has the attendant requirement for providing the crew with abort sensing and implementation data upon which to base an abort decision.(4)

- Guidelines and their associated functional requirements are extracted from various man-rating documents.
- Guidelines and functional requirements are segregated in order to quickly identify requirements for FFBDs.
- Man-rating document sources are identified by the number in brackets.

Figure 2.1-3 Man-rating Guidelines and Functional Requirements

<u>Design Criteria</u>

- **Launch systems must be designed for alternate missions**
  (abort missions added to design mission)
- **Provide provisions for crew to escape from vehicle**
  e.g. - Emergency Detection System, crew commanded controls

<u>Redundancy</u>

- **Redundant flight control & electrical systems are required**
- **Redundant sensor outputs are required**
  to preclude sensor malfunctions from causing switchover to redundant systems
- **Systems shall not be less than Fail-Safe**
  except primary structure, pressure vessels, & TPS which are designed for no failures (design margins required)

Figure 2.1-4 Man-rating Guidelines for Boosters

Reliability

- **Reliability history required for all subsystems/components over their complete life**
    - from qual test, acceptance test, to ground & flight operations
    - all failures exhaustively investigated (from start of development)
- **FMEA required to support safety analyses**

Test & Verification

- **All redundant design features must be completely demonstrated during tests**
- **Astronauts must be trained in all aspects of the system**
  including man-in-the-loop simulation testing

Management Practices

- **Management assessment & review of man-rating criteria**
- **Verification of flight readiness required**
  disposition of any failed equipment
- **All documentation under strict management control**
  extensive procedural documentation for QC during fabrication and production (e.g. serial number traceability)
- **Formal test & verification program**

Figure 2.1-5 Man-rating Guidelines for Boosters

- Provide fault detection, isolation, and recovery systems to address problems in critical and non-critical systems over which the crew has control. Critical systems status shall be displayed as to prevent misinterpretation. Fire suppression capability should be provided, and may either be automatic or manual depending on the risk. (1)
- Provide a caution and warning system for the crew to identify equipment failures, fire, or other potential emergency situations. (4)

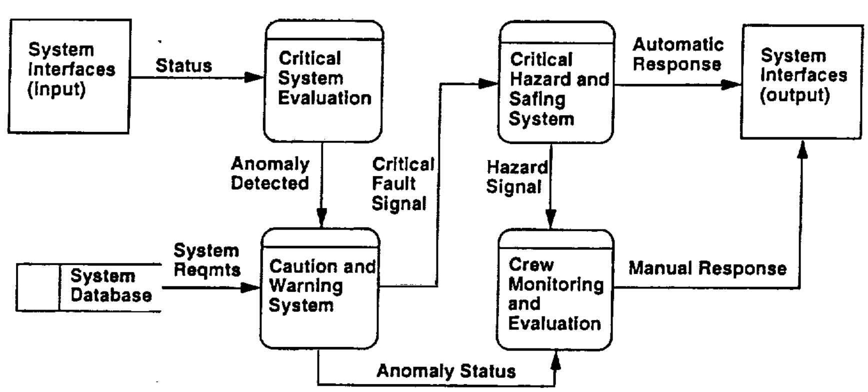


Figure 2.1-6 Man-rating Functional Diagrams

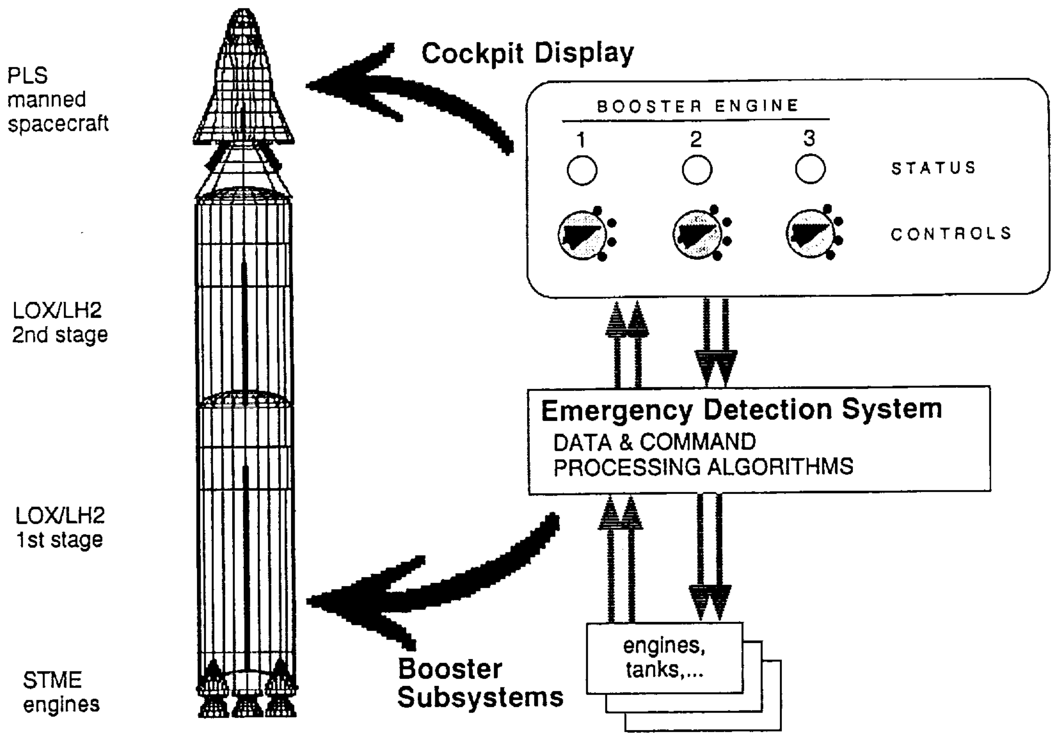


Figure 2.1-7 Booster-to-Crew Communications via EDS

- **Minimize reliance on crew for booster system corrective actions**
  - Allow only sufficient controls for implementing alternate missions selection and abort/escape functions.

  and manual override of critical functions ?

Crew Commands
- **Execute alternate mission command**
- **Execute abort mission command**
- **Execute escape command**

- **Minimize information flow to crew**
  - Data to crew presumes (invites) a crew response.
  - Avoid information overload to crew.

Booster Data Provided
- **Feedback response to commands issued by crew (see above)**
- **Notify crew of EDS decision to execute alternate missions**
- **Notify crew of EDS decision for crew to escape**

Figure 2.1-8 Booster-Crew Communications Rules

Booster Functions

A Get crew capsule to selected mission MECO target
& notify crew if unable to perform this mission

B_1 Get crew capsule to alternate mission MECO target
& notify crew if unable to perform this mission

B_2 Get crew capsule to abort mission MECO target
& notify crew if unable to perform this mission

C_1 Notify crew of inability to achieve any mission MECO target

C_2 Notify crew of immediate escape requirement

Crew Functions

1. Select an alternate mission MECO target
2. Select an abort mission MECO target
3. Issue escape command

Figure 2.1-9 Booster & Crew Functions

Emergency Detection System

A. Monitor critical systems for out-of-limits conditions or failures

B. Make decisions for any corrective actions required and issue corrective action commands as required
- to protect crew safety
- to accomplish primary mission

→ C. If decision reached that primary mission cannot be met
- maintain vehicle within crew safetylimits
- select alternate mission and issue required commands
- notify crew of decision reached

→ D. If decision reached that alternate missions cannot be met
- maintain vehicle within crew safetylimits
- notify crew of decision reached

→ E. If decision reached that crew safety cannot be maintained
- notify crew of decision reached
- issue automatic crew escape commands

→ = crew input capability (manual override)

The objective of the EDS is to detect malfunctions and provide commands to maintain the maximum mission completion capability while always maintaining the vehicle within crew safety limits

Figure 2.1-10 Booster EDS Functions

Emergency Detection System
Crew Commands to Booster

Engine
- manual engine cutoff
- engine shutdown override/inhibit
- manual engine throttle

Propulsion
- engine prevalves
- MPS fill/drain valves
- LH2 ullage flow control valves
- LH2 feedline relief isolation valve
- LOX feedline relief isolation valve

Hydraulics (if applicable)
- TVC enable (hydraulics isolation valves)

Helium
- engine/manifold pressurization (isolation valves)

Avionics
- engine controller power
- attitude rate override
- FCS (TVC) channel override
- TVC channel disable

Mechanical
- manual stage separation

Other
- manual abort initiation

Emergency Detection System
Booster Data to Crew Capsule

Engine
- HPOTP temperature
- HPOTP pressure
- combustion chamber pressure

Propulsion
- LOX manifold pressure
- LH2 manifold pressure
- LH2 tank ullage pressure
- LOX tank ullage pressure

Hydraulics (if applicable)
- engine hydraulic lockup

Helium
- helium (tank) pressure
- engine (regulated) helium pressure
- pneumatic valves (regulated) helium pressure

Avionics
- engine data channel loss
- engine command reject or channel loss
- electronic hold (no throttle)
- FCS bypass (to TVC)
- power supply voltage status
- vehicle roll rate (pitch, roll, yaw)

Mechanical
- stage separation

Other
- auto abort initiation

Figure 2.1-11 EDS Data & Command Monitor List

MAN-RATING REALLY MEANS:

Bring 'em back ALIVE!

Figure 2.1-12 The Man-rating Bottom Line

2.1.1.2 Booster Design Impacts

Manned payloads generally, but not always, have wings or lift generating body shapes. These manned payload design characteristics are another source of design impacts for a launch vehicle. Unmanned launch vehicles usually encase their payloads in a shroud which is axisymmetric and generates some nominal aerodynamic loads during launch. When a winged or lifting body shaped payload is installed on top of the launch vehicle instead of this shroud, significantly higher aerodynamic loads will be experienced by the booster. The magnitude of these loads were determined for a typical NLS-2 booster. Both the PLS HL-20 configuration and the much larger CLV configuration were analyzed to determine the effects of their large wings on the NLS-2 design. Detailed aerodynamic loads were calculated for both PLS/NLS-2 and the CLV/NLS-2 configurations. A finite-element model of the NLS-2 launch vehicle was then analyzed to determine the structural loads imposed by these payloads. It was found that the winged payloads induced significantly greater bending moments into the booster structure than normally encountered during ascent. A stress analysis with the NLS-2 structural model revealed that the increased bending loads would require a structural strengthening which would add approximately 4,000 pounds to the launch vehicle weight. Static control moment analysis of these manned payload configurations was also performed to determine if the booster could control the aerodynamic loads caused by the winged payloads. The analysis indicated that adequate control moment authority existed in the launch vehicle thrust vector control systems during both liftoff and maximum dynamic pressure conditions.

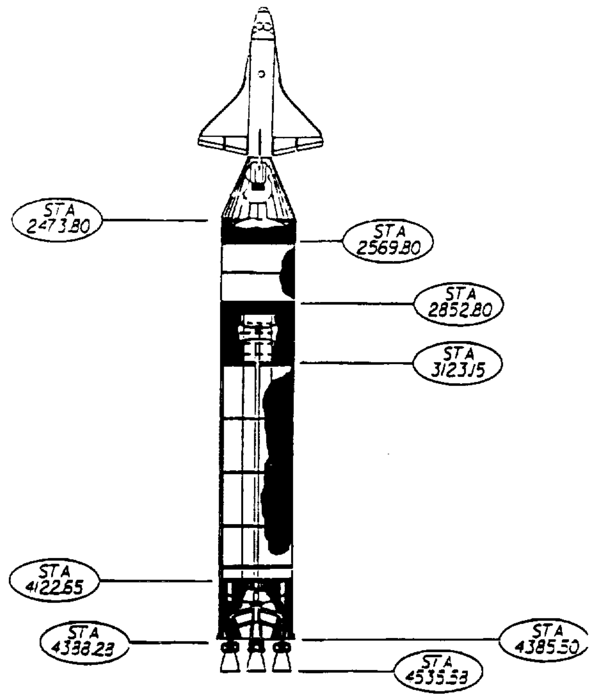


Figure 2.1-13 CLV Mounted on NLS Booster

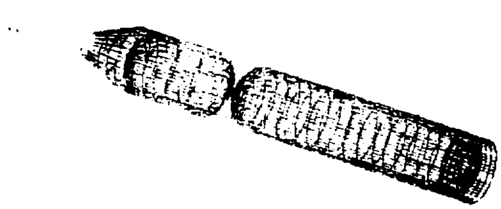


Figure 2.1-14 CLV & NLS Finite Element Model

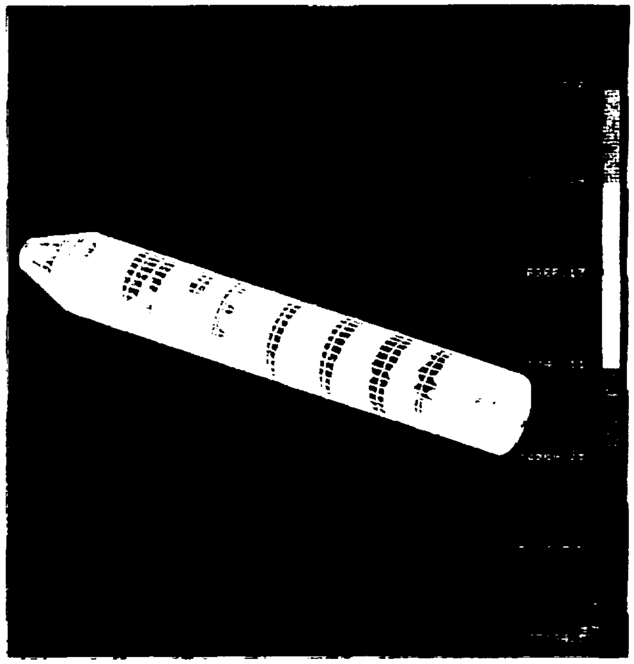


Figure 2.1-15 Booster Stresses at Max Q

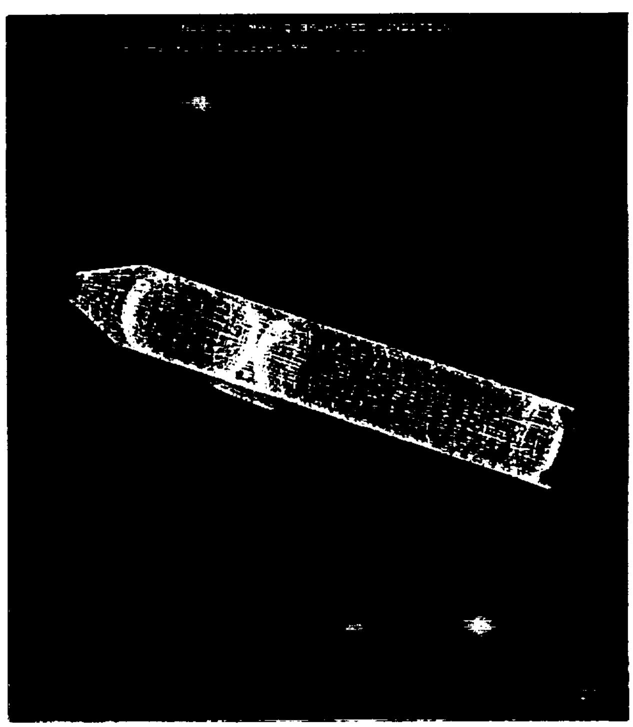


Figure 2.1-16 Booster Deflections at Max Q

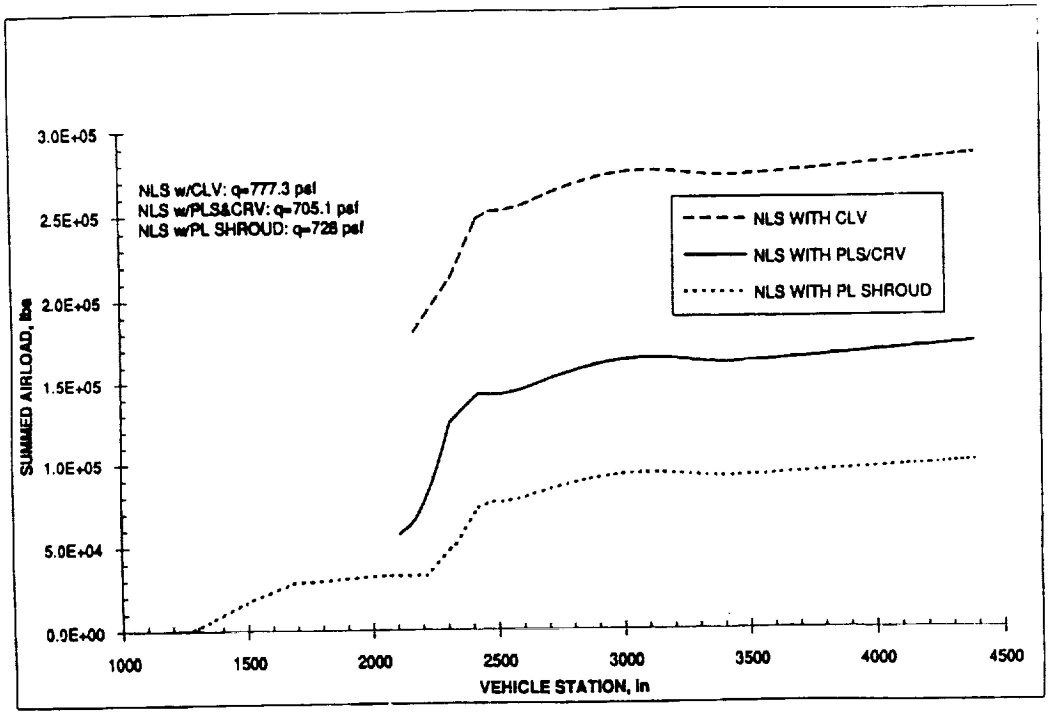


Figure 2.1-17 Comparison of Max Q Airloads

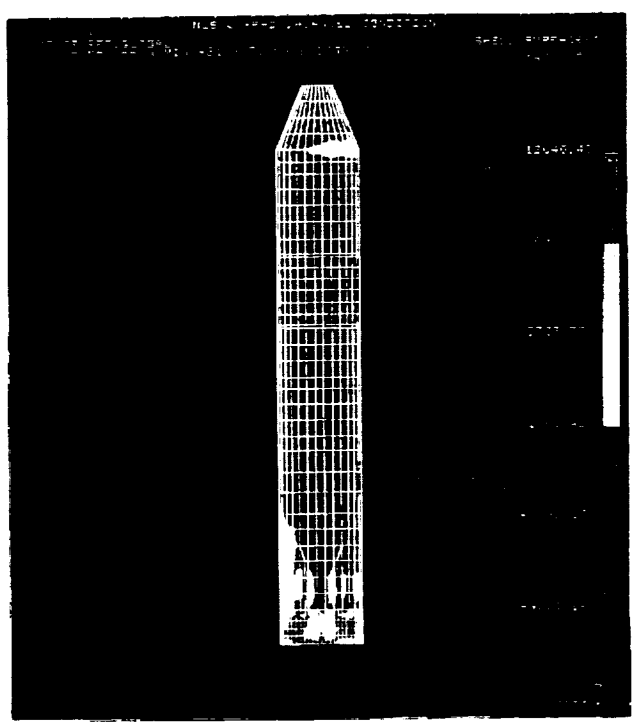


Figure 2.1-18 Booster Stresses, On-Pad Winds

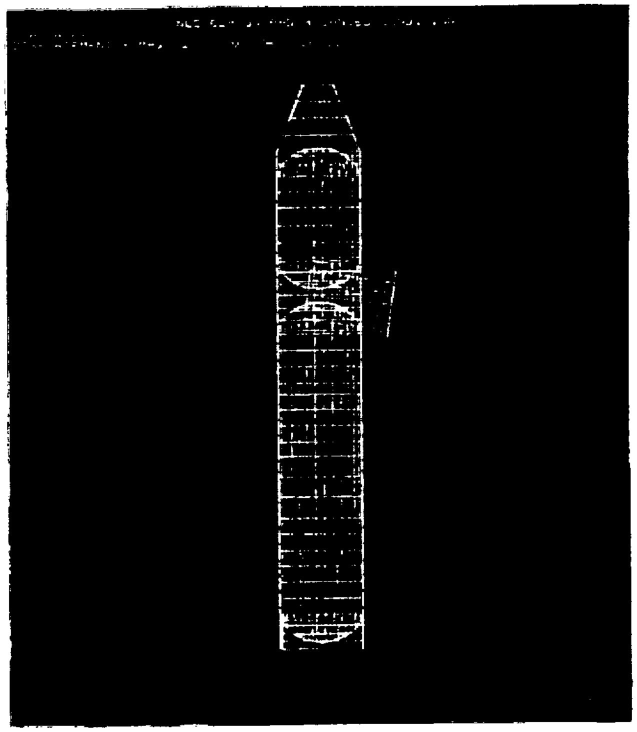


Figure 2.1-19 Booster Deflections, On-Pad Winds

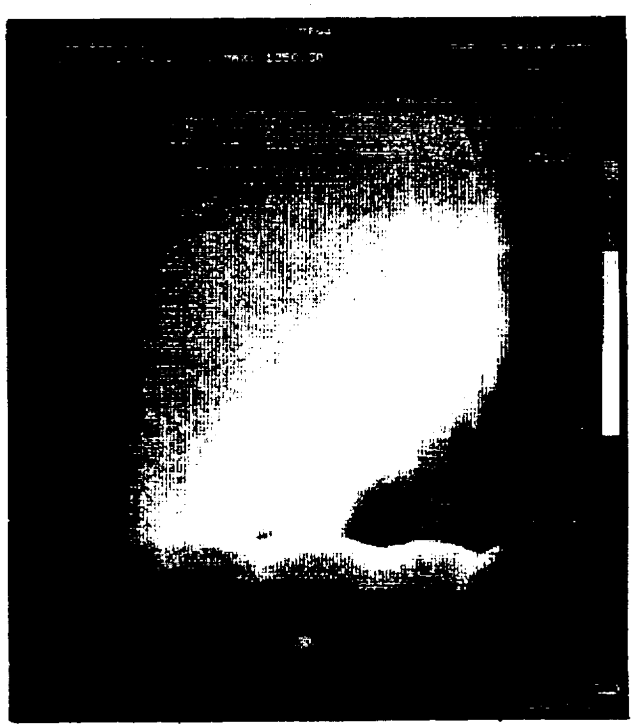
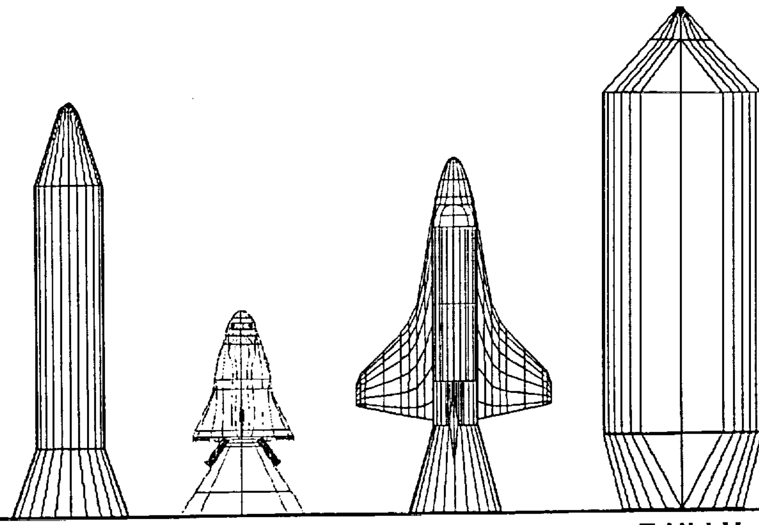


Figure 2.1-20 Booster LOX Tank Stresses (On-Pad)



	NLS	PLS	CLV	E-HLLV
Max Q (psf)	728	705	777	656
Bending Moment (ft-lb x 10^6)	6.6	11.3	20.6	15.0 EDGEWISE 25.0 BROADSIDE

Figure 2.1-21 Comparison of Payload Induced Loads

2.1.2 Performance Analysis

A number of candidate manned booster concepts were evaluated to determine their suitability for launching either the PLS or CLV manned payload concepts. In general, it was found that the payload performance of most of the current set of planned NLS launch vehicle configurations were not particularly well matched to either manned payload. The PLS and CLV system launch weights used for the analyses were 33,800 lbs and 75,700 lbs, respectively. The NLS-2 class of booster concepts analyzed generally optimized for payload masses in the range of approximately 60,000 to 65,000 lbs(1.5 stage versions of NLS-2), or in the range of 105,000 to 115,000 lbs (two stage versions of NLS-2). The manned payload masses fell considerably below the payload capabilities of either range. The boosters were thus somewhat overpowered for either the PLS or the CLV. This situation does, however, provide considerable room for weight growth in the booster to allow for manned mission trajectory options and additional systems necessary for satisfying man-rating requirements.

Existing expendable launch vehicles were also not well sized for these manned systems. Although their payload mass delivery capabilities (48,000 to 52,000 lbs to LEO) were closer to the PLS launch weight, some of these vehicles generated high dynamic pressures during launch which will cause significant structural modifications to these boosters. The Ariane V and the Titan IV generated max Q's of 835 psf and 925 psf during launch, respectively. While these pressures are within the normal operating limits for these vehicles, they will result in extremely high bending moments in the booster which are not within the normal booster limits (as we found in the case of NLS-2 boosters due to the high aerodynamic loads from the PLS payload).

A "clean sheet" concept utilizing the Liquid Rocket Booster (LRB) for launching the PLS payload was found to be a better performance match than the NLS-2 configurations. The LRB configuration included a single F-1A engine for the first stage and a single J-2S engine for the second stage. The LRB concept defined by MSFC as a Space Shuttle SRB replacement was used as a starting point for the first stage propellant sizing. A Saturn S-IVB stage was used for second stage sizing. This configuration provided lower payload mass capabilities (by 8,000 lbs) and equivalent dynamic pressures compared to the NLS-2 1.5 stage concepts. Compared to existing ELV boosters, this concept provided slightly more payload capability (2,000 lbs), and significantly lower dynamic pressures.

The ability to complete the PLS delivery mission with an engine out (from the launch pad) provides a booster with a slightly higher probability of mission success. For the NLS-2 1.5 stage booster, this increase amounted to approximately one percentage point (from 0.981 to 0.989). The individual engine reliability, and their burn duration and throttling requirements, play a stronger role in determining the various boosters reliability than the engine out capability. As will be seen later in this report (in the Reliability Analysis section), the number of engines, the number of stages, and the stage burn times span a wider range of variability among the several boosters analyzed than the effect of an engine out capability or not. These engine number and burn time characteristics were thus compared for each of the manned booster concepts. (This information will be of further utility during future analyses of abort mission capabilities of the various booster concepts.) A probability of mission success range of 0.917 to 0.989 was determined for the several booster concepts analyzed. This range is much greater than the effect of engine out capability as noted above.

The measure of crew safety for the manned booster comparisons is also influenced by the engine reliability and engine out capabilities (again, by the same relative strength levels). The overriding factor for crew safety, however, was the PLS provided escape capability. This capability was equally available for all the booster concepts. Crew safety levels were thus found to be only weakly influenced by the engine out capability (or lack thereof) for any of the manned boosters.

Performance of the several manned booster concepts at higher launch inclinations was determined to assess the effect that a Space Station orbit of 51.6° might have on the ability to launch PLS to the Space Station. The boosters' payload launch capacity at this orbit inclination was generally reduced by 6,000 to 8,000 lbs. This reduced payload capability did not, however, change any of the findings concerning which of the boosters was best suited to launching either the PLS or the CLV payloads. Because all of the boosters analyzed had significant performance margin for these payloads, the effect of a higher inclination orbit only reduced the excess lift capacity. All of the boosters recommended for the PLS or CLV systems still had excess lift capacity at this higher inclination orbit.

Comparison of the several boosters for manned payload (PLS) delivery missions led to a recommendation of either the NLS-2 1.5 stage booster (in the 4/1 STME configuration) or the F-1A based LRB for the PLS payload. Both of these boosters are somewhat overpowered for this payload, but the excess lift capacity may be used for mission flexibility (abort coverage), weight growth margins, or additional payload options. From among the concepts analyzed to date, the only recommended booster for the CLV payload is a two stage version of the NLS-2 (4 STME first stage, 1 J-2S second stage). This booster is also very overpowered (by > 30,000 lbs) for this manned payload, but the excess capacity may be useful for other considerations. A better matched launch vehicle than this NLS-2 two stage can be defined by modifying the engine and tank sizing to other than NLS constrained levels.

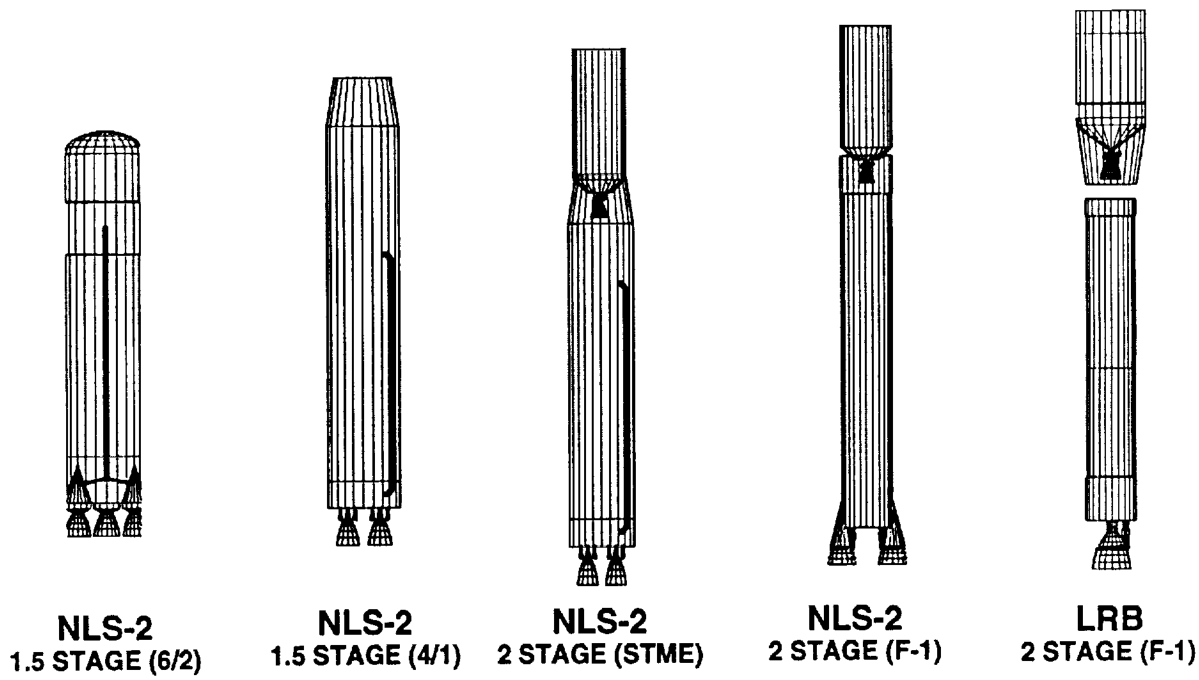


Figure 2.1-22 NLS Boosters Evaluated

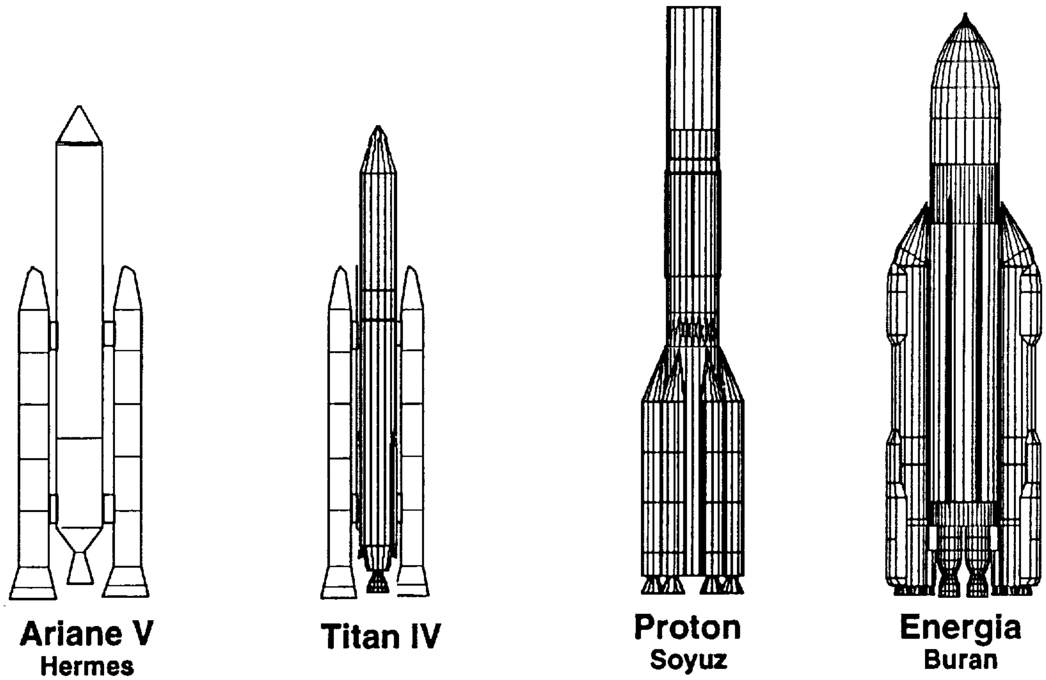


Figure 2.1-23 Existing ELVs Evaluated

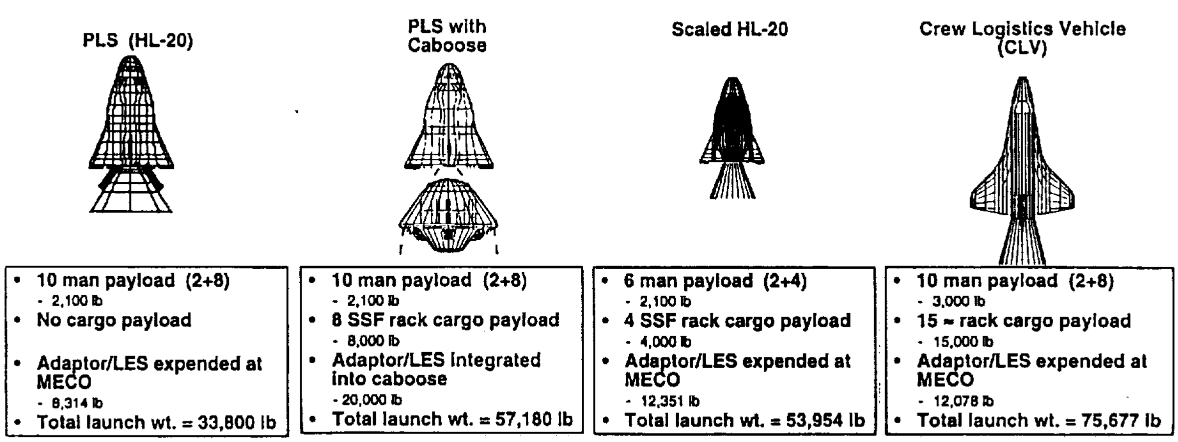
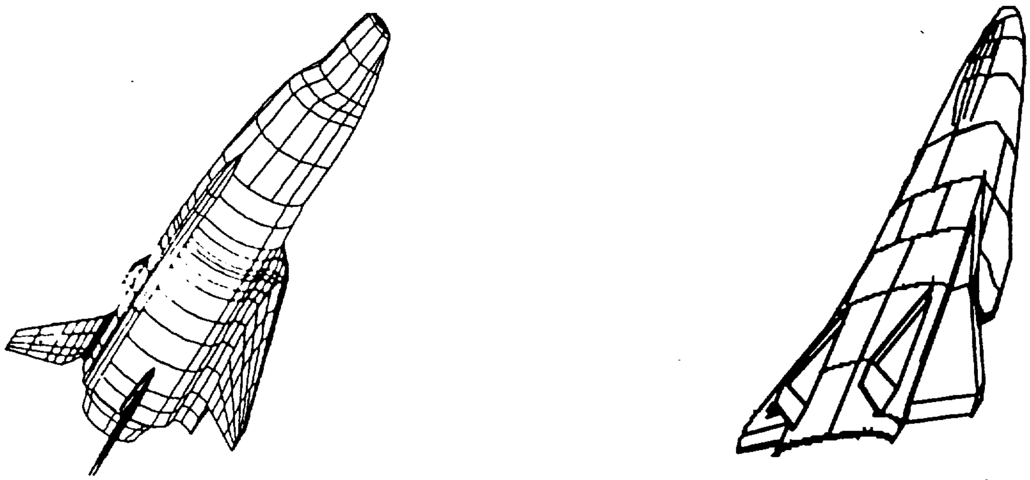


Figure 2.1-24 Booster Payload Options



SSTO
Rockwell Concept

NASP Derived
Single Stage to Orbit

Figure 2.1-25 Advanced Boosters Evaluated

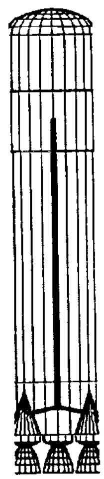
	Max Payload to LEO (lb)	Max Q (psf)	Max Accel (g)	Comments
NLS 1.5 Stage (6/2)	62,600	732	4.0	Engine out
NLS 1.5 Stage (4/1)	61,800	529	4.0	
NLS 2 Stage (STME)	107,100	524	4.0	
NLS 2 Stage (F-1)	113,800	649	4.0	
Ariane V	48,900	835	4.2	
Energia	227,300	800	4.4	Engine out
Proton	48,100	691	3.4	
LRB (F-1A)	54,423	725	4.0	
Titan IV	51,939	925	3.4	

Figure 2.1-26 Booster Maximum Performance (with PLS Payload)

Booster	PLS	PLS + Cab.	Scaled HL-20	CLV	Other
NLS-2 1.5 (6/2)	✔	★	★	✕	
NLS-2 1.5 (4/1)	★	✔	✔		
NLS 2stg (STME)	✕	✕	✕	★	
NLS 2stg (F-1)	✕	✕	✕	✕	
Ariane V	✔				
Proton	✔				
Energia					
LRB (F-1A)	★	★			
Titan IV	✔				
SSTO					★
NDV					★

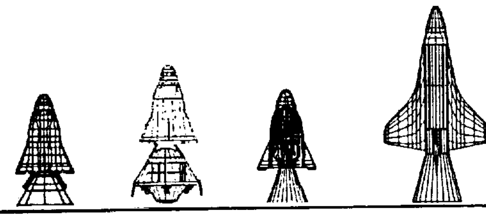
★ = Recommended
✔ = Acceptable
✕ = Not recommended

Figure 2.1-27 Recommended Boosters for Manned Payloads



NLS-2
1.5 STAGE (6/2)

Payload Options



Booster Description

	Booster	Core
Propellant Type	LOX/LH2	LOX/LH2
Engines	STME (4)	STME (2)
Thrust (lbf)	650 K	650 K
Isp (sec)	428.5	428.5
Engine out capab.	Yes	Yes
GLOW (lb)	78,312	1,877,641
Dry Weight (lb)	70,700	127,550
Length (ft)	34	179
Diameter (ft)	27.5	27.5

Figure 2.1-28 NLS-2 1.5 Stage (6/2) Configuration

Booster Maximum Performance (to 15 x 220 nmi.)

Maximum Dynamic Pressure (psf)	731.75
Maximum Acceleration (g's)	4.00
Booster Separation:	
Time (sec)	161.50
Altitude (ft)	245903
Relative Velocity (fps)	9031.30
MECO:	
Time (sec)	338.29
Weight (lbs)	219,055
Excess Propellant (lbs)	10,477
Payload to Transfer Orbit (lbs)	**62,696**

Annual Flight Rate = 10

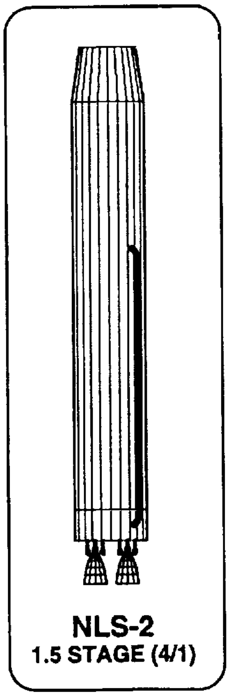
Launch Cost = $100 M

Booster Payloads to SSF Orbit

- Maximum payload to SSF transfer orbit satisfies engine-out at liftoff requirement (*except the CLV payload)

	PLS + Caboose	CLV*	Scaled PLS	PLS
MECO Weight (lbs)	73,173	75,680	73,018	72,621
Gross Payload to Transfer Orbit (lbs)	**53,800**	**75,677**	**53,954**	**33,800**
Margin (lbs)	19,373	3	19,064	38,821

Figure 2.1-29 NLS-2 1.5 Stage (6/2) Performance



Payload Options

Booster Description

	Booster	Core
Propellant Type	LOX/LH2	LOX/LH2
Engines	STME (3)	STME (1)
Thrust (lbf)	650 K	650 K
Isp (sec)	428.5	428.5
Engine out capab.	No	No
GLOW (lb)	62,878	1,714,809
Dry Weight (lb)	55,266	105,434
Length (ft)	34	172
Diameter (ft)	27.5	27.5

Figure 2.1-30 NLS-2 1.5 Stage (4/1) Description

Booster Maximum Performance (to 15 x 220 nmi.)

Maximum Dynamic Pressure (psf)	528.83
Maximum Acceleration (g's)	4.00
Booster Separation:	
Time (sec)	196.03
Altitude (ft)	238025
Relative Velocity (fps)	9291.59
MECO:	
Time (sec)	461.07
Weight (lbs)	185,105
Excess Propellant (lbs)	0
Payload to Transfer Orbit (lbs)	**61,796**

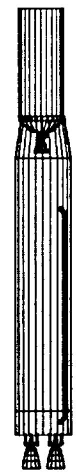
Annual Flight Rate = 10

Launch Cost = $90 M

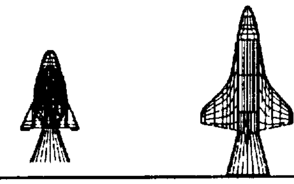
Booster Payloads to SSF Orbit

	PLS + Caboose	Scaled PLS	PLS
MECO Weight (lbs)	61,474	61,351	60,407
Gross Payload to Transfer Orbit (lbs)	**53,800**	**53,954**	**33,800**
Margin (lbs)	7,674	7,397	26,607

Figure 2.1-31 NLS-2 1.5 Stage (4/1) Performance



Payload Options



Booster Description

	Booster	Core
Propellant Type	LOX/LH2	LOX/LH2
Engines	STME (4)	J-2S (1)
Thrust (lbf)	650 K	265 K
Isp (sec)	428.5	436
Engine out capab.	No	No
GLOW (lb)	1,475,766	246,996
Dry Weight (lb)	165,647	27,912
Length (ft)	167.3	71.7
Diameter (ft)	27.5	21.0

Figure 2.1-32 NLS-2 2 Stage (STME) Description

Booster Maximum Performance (to 15 x 220 nmi.)

Maximum Dynamic Pressure (psf)	523.86
Maximum Acceleration (g's)	4.00
Booster Separation:	
Time (sec)	225.07
Altitude (ft)	296773
Relative Velocity (fps)	12087.89
MECO:	
Time (sec)	580.55
Weight (lbs)	138,079
Excess Propellant (lbs)	0
Payload to Transfer Orbit (lbs)	**107,143**

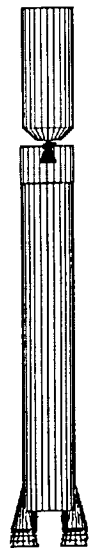
Annual Flight Rate = 10

Launch Cost = $120 M

Booster Payloads to SSF Orbit

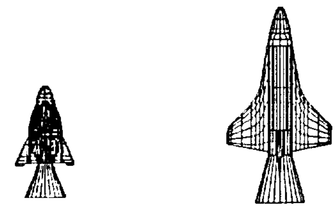
	Scaled PLS	CLV
MECO Weight (lbs)	100,642	103,474
Gross Payload to Transfer Orbit (lbs)	**53,954**	**75,677**
Margin (lbs)	46,688	27,797

Figure 2.1-33 NLS-2 2 Stage (STME) Performance



NLS-2
2 STAGE (F-1)

Payload Options



Booster Description

	Booster	Core
Propellant Type	LOX/RP	LOX/LH2
Engines	F-1A (2)	J-2S (1)
Thrust (lbf)	2,025 K	265 K
Isp (sec)	305	436
Engine out capab.	No	No
GLOW (lb)	2,559,969	320,059
Dry Weight (lb)	172,274	31,693
Length (ft)	151.6	71.7
Diameter (ft)	21.0	21.0

Figure 2.1-34 NLS-2 2 Stage (F-1) Description

Booster Maximum Performance (to 15 x 220 nmi.)

Maximum Dynamic Pressure (psf)	649.06
Maximum Acceleration (g's)	4.00
Booster Separation:	
Time (sec)	205.32
Altitude (ft)	311721
Relative Velocity (fps)	10859.59
MECO:	
Time (sec)	674.35
Weight (lbs)	148,853
Excess Propellant (lbs)	0
Payload to Transfer Orbit (lbs)	**113,870**

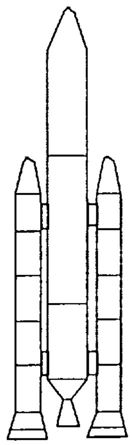
Annual Flight Rate = 10

Launch Cost = $140 M

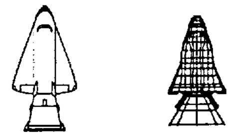
Booster Payloads to SSF Orbit

	Scaled PLS	CLV
MECO Weight (lbs)	107,535	110,055
Gross Payload to Transfer Orbit (lbs)	**53,954**	**75,677**
Margin (lbs)	53,851	34,378

Figure 2.1-35 NLS-2 2 Stage (F-1) Performance



Payload Options



Booster Description

	Booster	Core
Propellant Type	Solid	LOX/LH2
Engines	P230	HM60
Thrust (lbf)	1,430 K	225 K
Isp (sec)	273	430
Engine out capab.	No	No
GLOW (lb)	1,166,000	375,000
Dry Weight (lb)	154,000	33,000
Length (ft)	98	95
Diameter (ft)	9.9	17.7

Figure 2.1-36 Ariane V Description

Booster Maximum Performance (to 50 x 220 nmi.)

Maximum Dynamic Pressure (psf)	835.00
Maximum Acceleration (g's)	4.17
Booster Separation:	
Time (sec)	123.07
Altitude (ft)	201331
Relative Velocity (fps)	6928.92
MECO:	
Time (sec)	648.93
Weight (lbs)	83,945
Excess Propellant	0
Payload to Transfer Orbit (lbs)	**48,945**

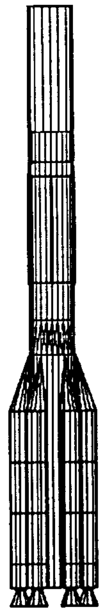
Annual Flight Rate = 10

Launch Cost = $90 M

Booster Payloads to SSF Orbit

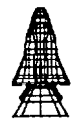
	Hermes	PLS
MECO Weight	48,945	?
Gross Payload to Transfer Orbit	**48,500**	**33,800**
Margin	445	?
Payload to SSF Orbit	**47,088**	**24,689**

Figure 2.1-37 Ariane V Performance



Proton

Payload Options



Booster Description

	1st Stage	2nd Stage	3rd Stage
Propellant Type	N2O4/UDMH	N2O4/UDMH	N2O4/UDMH
Engines	RD-253	RD-xx	RD-xx
Thrust (lbf)	368 K	135 K	142 K
Isp (sec)	316	322	442
Engine out capab.	?	?	No
GLOW (lb)	1,004,000	365,000	123,000
Dry Weight (lb)	100,000	35,000	13,000
Length (ft)	66.3	45	30
Diameter (ft)	24	13	13

Figure 2.1-38 Proton Description

Booster Maximum Performance (to 30 x 220 nmi.)

Maximum Dynamic Pressure (psf)	691.38
Maximum Acceleration (g's)	3.42
Booster Separation:	
Time (sec)	129.38
Altitude (ft)	131,266
Relative Velocity (fps)	4964.71
MECO:	
Time (sec)	661.07
Weight (lbs)	73,999
Excess Propellant	0
Payload to Transfer Orbit (lbs)	**48,149**

Performance:
- Launch Site - Baikonur

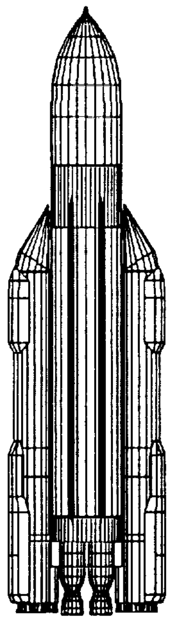
Annual Flight Rate = 10

Launch Cost = $140 M

Booster Payloads to SSF Orbit

	PLS
MECO Weight (lbs)	46,789
Gross Payload to Transfer Orbit (lbs)	**33,800**
Margin (lbs)	12,989

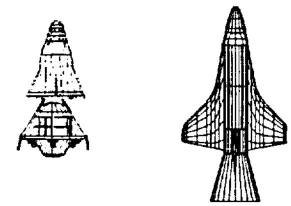
Figure 2.1-39 Proton Performance



Energia

Payload Options

Buran



Booster Description

	Booster	Core
Propellant Type	LOX/RP	LOX/LH2
Engines	RD-170	RD-xx
Thrust (lbf)	1,777,000	441,000
Isp (sec)	336	452.5
Engine out capab.	Yes	Yes
GLOW (lb)	3,031,932	1,982,901
Dry Weight (lb)	366,400	176,000
Length (ft)	131	197
Diameter (ft)	12.8	26

Figure 2.1-40 Energia Description

Booster Maximum Performance (to -38 x 220 nmi.)

Maximum Dynamic Pressure (psf)	800.00
Maximum Acceleration (g's)	4.37
Booster Separation:	
Time (sec)	136.67
Altitude (ft)	261273
Relative Velocity (fps)	6444.22
MECO:	
Time (sec)	463.51
Weight (lbs)	403,227
Excess Propellant (lbs)	0
Payload to Transfer Orbit (lbs)	$\boxed{227,300}$

Annual Flight Rate = 1

Launch Cost = $350M

Performance:
- Launch Site - Baikonur

Booster Payloads to SSF Orbit

	Buran	PLS + Caboose	CLV*
MECO Weight (lbs)	227,300	?	?
Gross Payload to Transfer Orbit (lbs)	**227,300**	**53,800**	**75,677**
Margin (lbs)	0	?	?

Figure 2.1-41 Energia Performance

Booster Maximum Performance (to 15 x 220 nmi.)

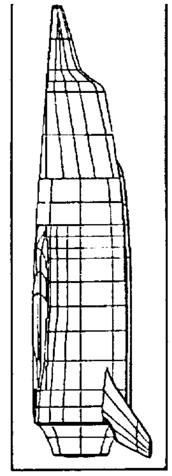
Maximum Dynamic Pressure (psf)
Maximum Acceleration (g's)

Booster Separation:
- Time (sec)
- Altitude (ft)
- Relative Velocity (fps)

MECO:
- Time (sec)
- Weight (lbs)
- Excess Propellant

Payload to Orbit (lbs) 23,780

Annual Flight Rate = 80
Launch Cost = $5.4M
($55.8M at min rate).



Booster Payloads to SSF Orbit

MECO Weight	
Gross Payload to Transfer Orbit	
Margin	
Payload to SSF Orbit	**21,100**

Figure 2.1-42 SSTO Performance

Booster Maximum Performance (to 15 x 220 nmi.)

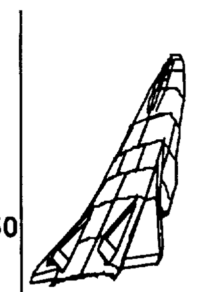
Maximum Dynamic Pressure (psf)
Maximum Acceleration (g's)

Booster Separation:
- Time (sec)
- Altitude (ft)
- Relative Velocity (fps)

MECO:
- Time (sec)
- Weight (lbs)
- Excess Propellant

Payload to Orbit (lbs) 26,500

Annual Flight Rate = 50
Launch Cost = $3M
(est. $50M at min rate)



Performance:
- Launch Site - Holloman AFB

Booster Payloads to SSF Orbit

MECO Weight	
Gross Payload to Transfer Orbit	
Margin	
Payload to SSF Orbit	**18,200**

Figure 2.1-43 NDV Performance

2.1.3 Cost Analysis

Cost analyses of all manned boosters were performed using an agreed upon Work Breakdown Structure which would be consistent with other space transportation studies being conducted under the ATSS contracts. Results of the recent Space Shuttle Zero-Base Operations Cost Study (completed by NASA in June, 1991) were updated to reflect the costing assumptions and WBS structure for this ATSS study. This Space Shuttle data was used as the current point of reference for comparison of all manned boosters. Both fixed and variable operating costs were identified to permit comparison among the many potential manned booster concepts, some of which are reusable like the Shuttle (AMLS, SSTO, NASP-derived, etc.), and some of which are expendable, such as NLS and ELV boosters for the PLS.

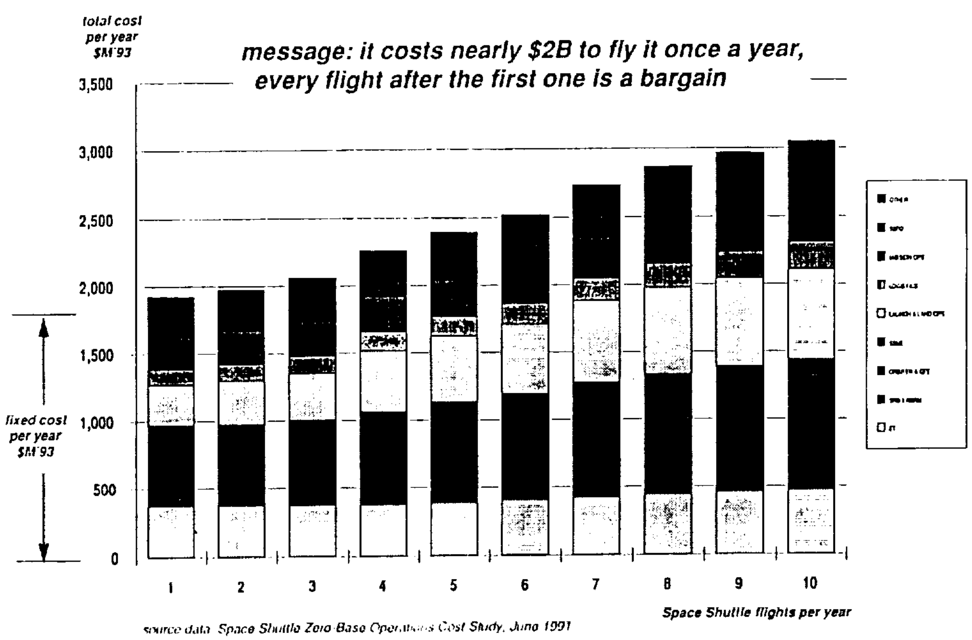
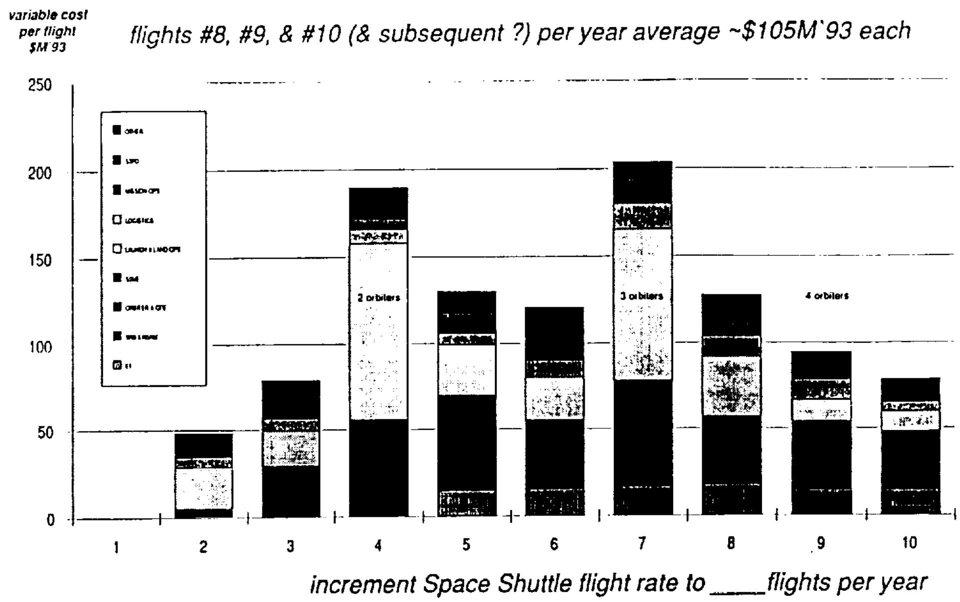


Figure 2.1-44 Space Shuttle Total Cost vs Flight Rate



source data: Space Shuttle Zero-Base Operations Cost Study, June 1991

Figure 2.1-45 Space Shuttle Variable Cost vs. Flight Rate

Cost estimates of the NLS family of boosters were extremely sensitive during the course of this study. To avoid causing interference in the active cost estimating activities on-going in the NLS program, the NLS-2 booster costs were pegged at a per launch cost of $100M (FY93). This number was consistent with the published NLS-2 booster cost per flight goal. The cost of any alternate configuration NLS-2 booster was calculated as a ratio of this reference cost. The cost ratio for alternate NLS-2 booster configurations was determined by identifying the relative hardware and operations complexity differences between the reference NLS-2 booster (1.5 stage, 6 STMEs) and the alternate. An analytical hierarchy process (AHP) analysis was used to determine relative weighting factors among the booster configurations. The AHP analysis compared relative differences in development, production, and operational complexities between the reference NLS-2 booster and the alternative configurations. The criteria used to gauge differences in the booster complexities included first and second stage technical characteristics, structural and propulsion systems test requirements, facility requirements or impacts, ground operations, logistics, and flight operations.

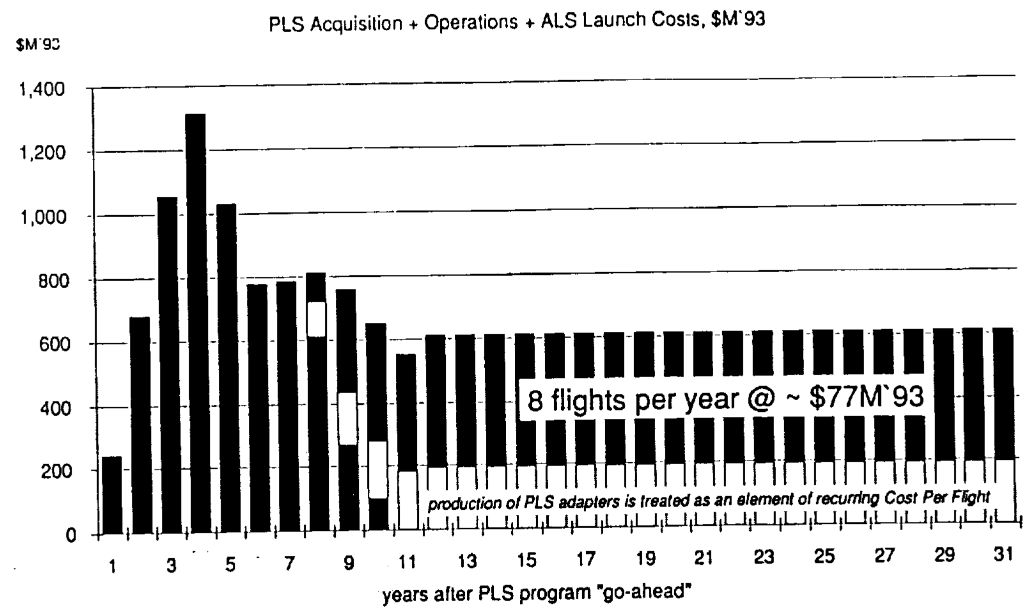


Figure 2.1-46 PLS Launch Costs with ALS Booster

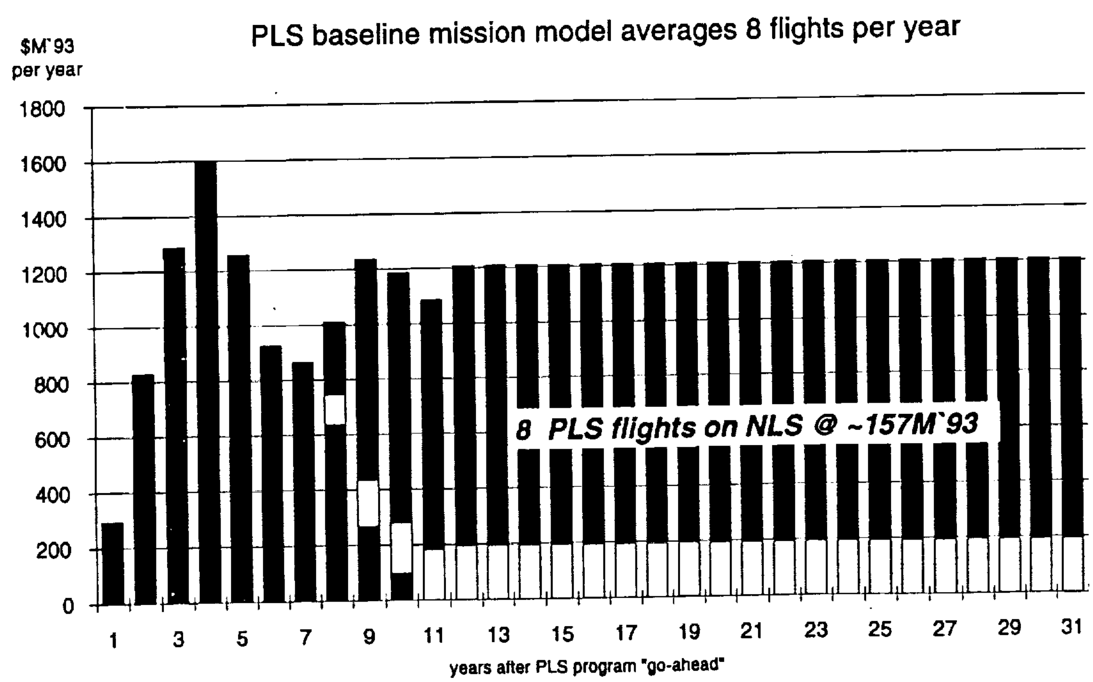


Figure 2.1-47 PLS Launch Costs with NLS

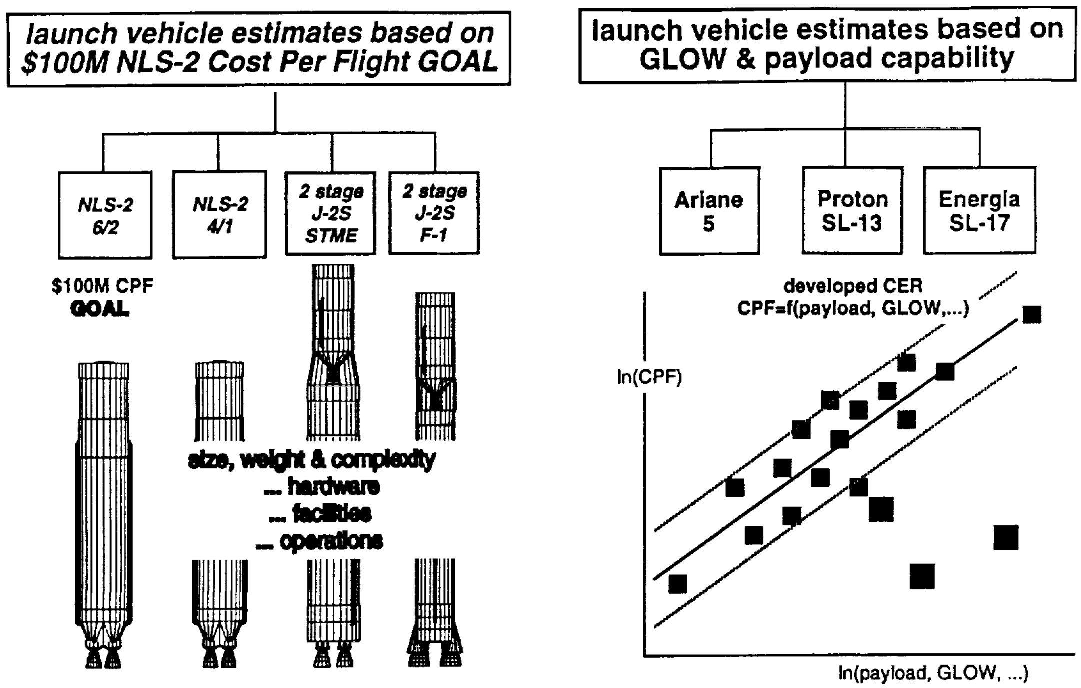


Figure 2.1-48 NLS costs are not directly comparable to other ELVs

	NLS-2 6/2	NLS-2 4/1	2 stage J-2S STME	2 stage J-2S F-1
hardware complexity	1.00	0.82	1.16	1.27
operations complexity	1.00	0.95	1.18	1.48
hardware %	71	58	82	90
operations %	19	18	22	28
reserves %	10	8	12	13
relative %	100	85	116	132

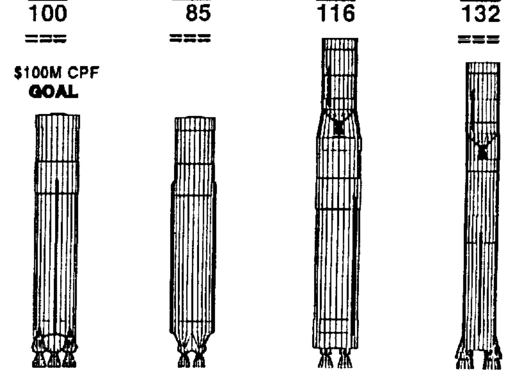


Figure 2.1-49 NLS booster costs were factored from a cost baseline

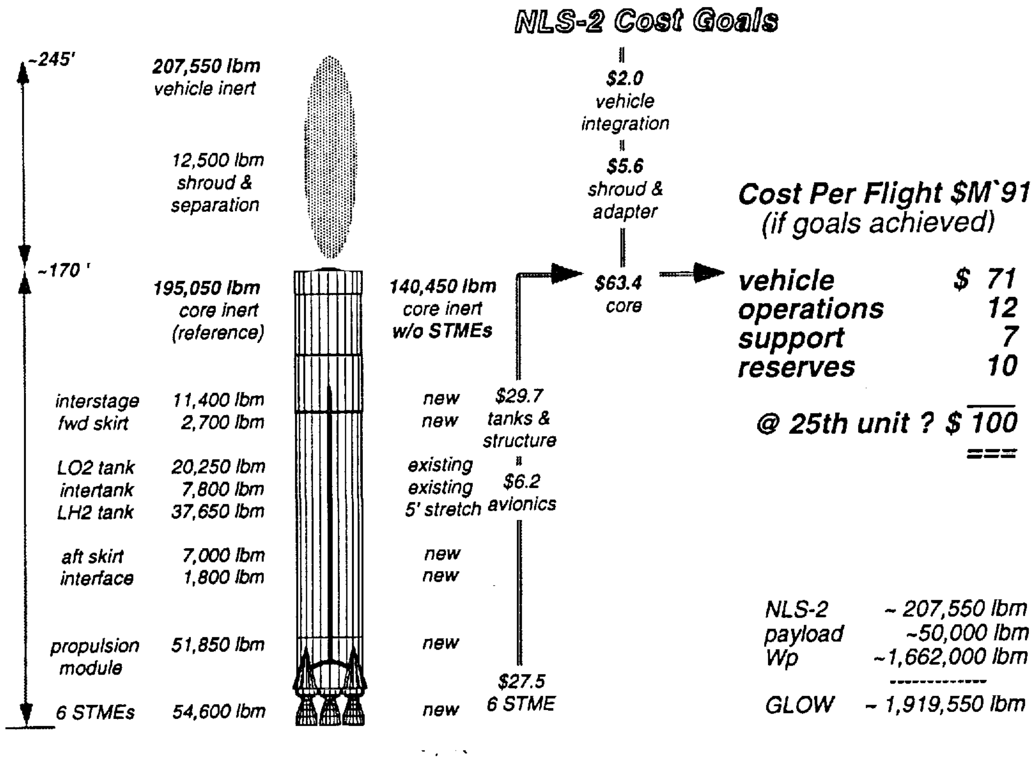


Figure 2.1-50 NLS-2 6/2 Cost Baseline Breakdown

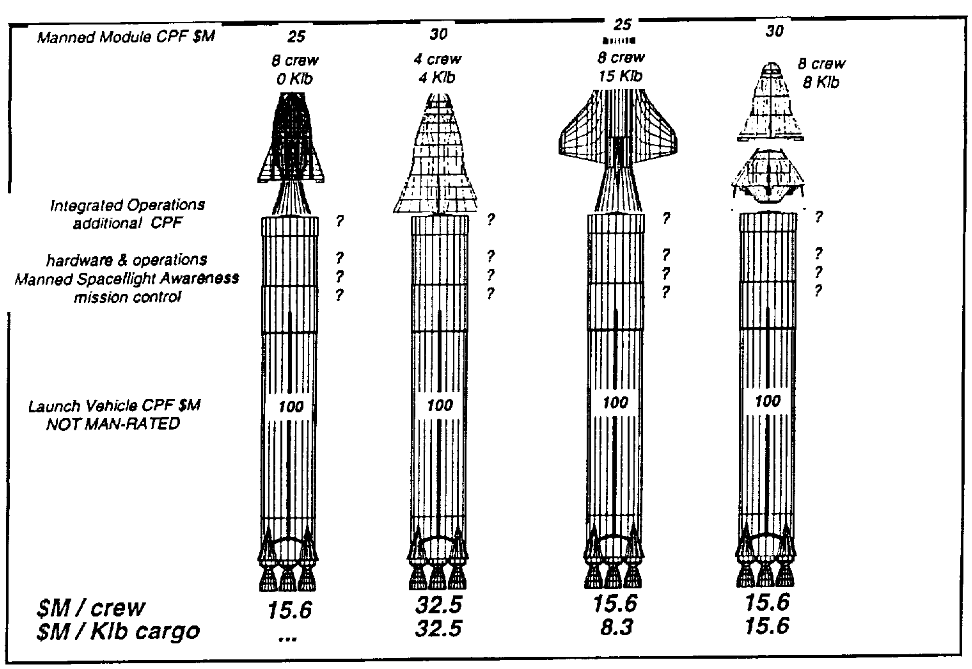


Figure 2.1-51 NLS-2 Baseline with Alternate Manned Payloads

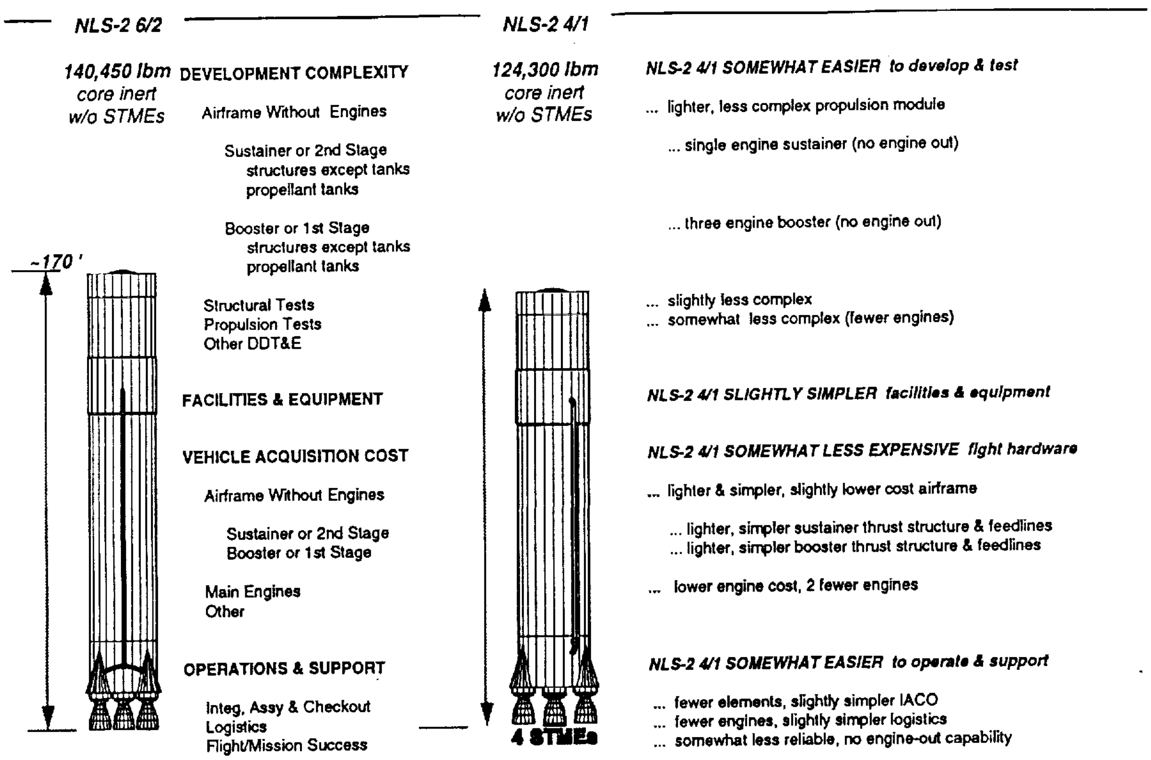


Figure 2.1-52 NLS-2 4/1 Cost

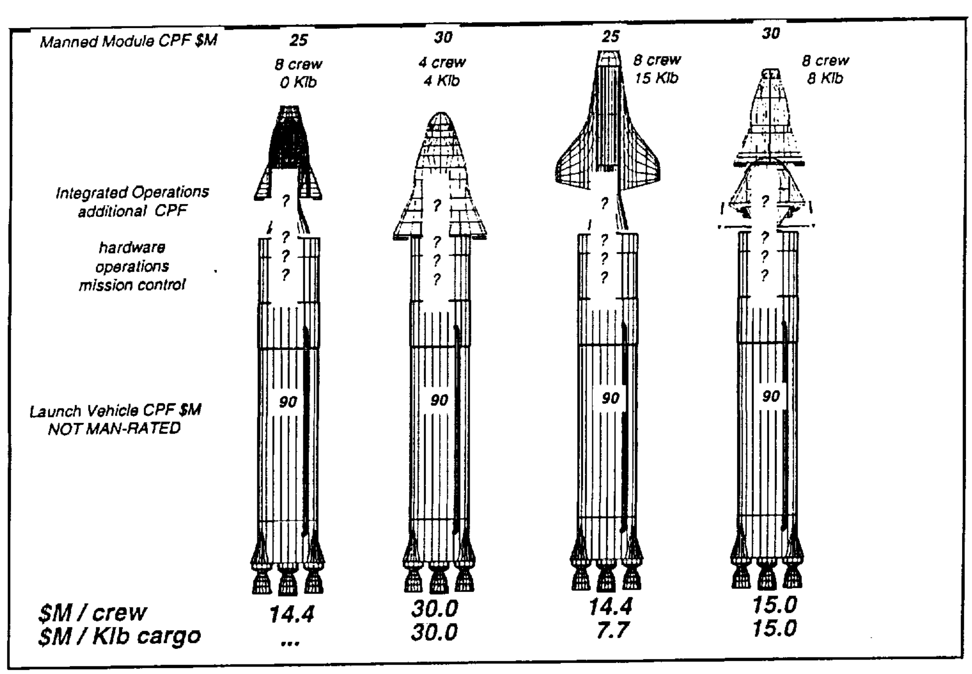


Figure 2.1-53 NLS-2 4/1 Cost with Alternate Manned Payloads

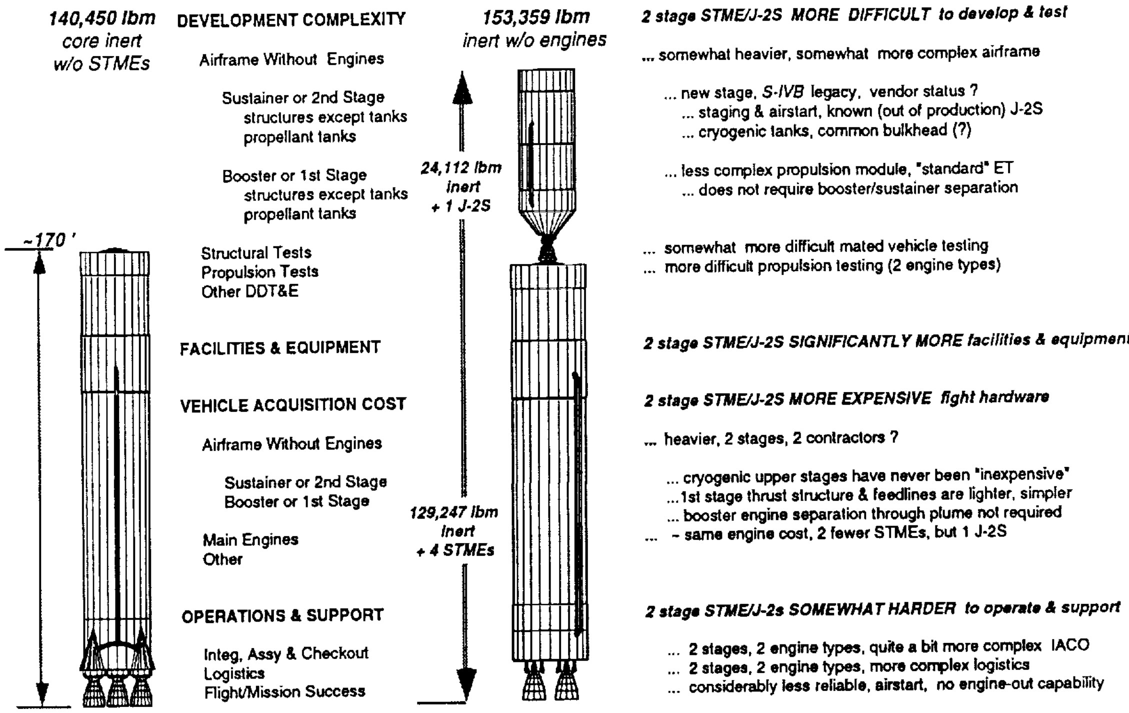


Figure 2.1-54 NLS 2 Stage (STME) Cost

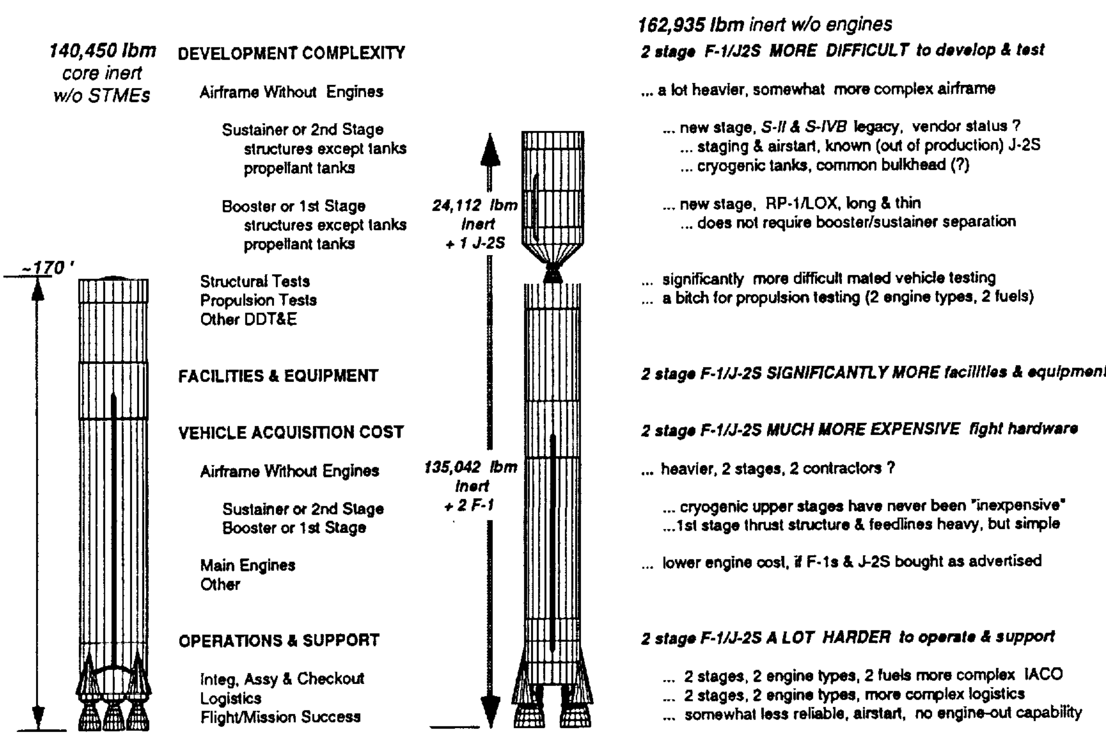


Figure 2.1-55 NLS 2 Stage (F-1) Cost

2 Stage F-1 & J-2S Is Most Costly Launcher Examined

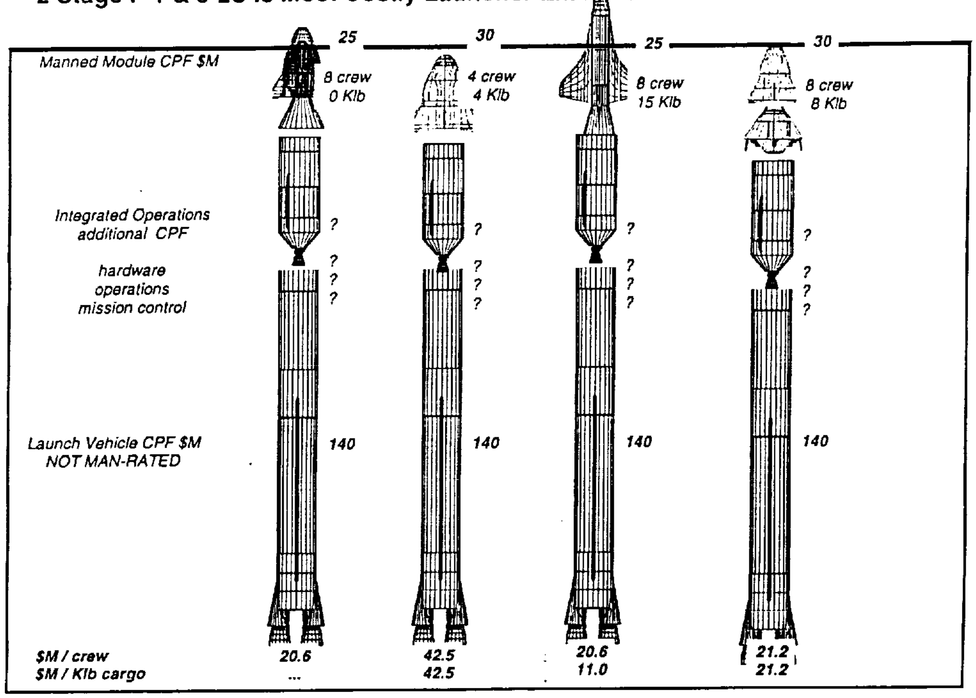


Figure 2.1-56 NLS 2 Stage with Alternate Manned Payloads

Results of the AHP process identified the NLS-2 4/1 configuration to be the cheapest NLS-derived booster capable of launching the PLS payload, at a cost ratio of 0.85 relative to the NLS-2 reference (or $85M per flight). The two stage versions of NLS-2, which were required to launch the CLV payload, were similarly estimated at $116M (STME first stage) to $132M (F-1A first stage) per flight. The higher two stage booster launch costs were highly influenced by the required operations costs associated with an additional stage and an additional engine to produce and process for each launch. These additional element's operations costs more than offset the fewer number of engines required per launch by the reference NLS-2.

Cost estimates for the C.I.S. launch vehicles is a highly judgmental exercise given the volatile economic and political conditions in that country. Nevertheless, a reasonable estimate was made of what these boosters might cost in the future, when economic equilibrium is reached between the C.I.S. and the Western markets. A fairly good cost per launch and booster payload capability correlation exists for all Western launch vehicles (often expressed as cost per pound to LEO). A 1-sigma error band on this cost correlation captures virtually all of the free world launch vehicles. If one assumes that the C.I.S. boosters have no technical advantage over the Western boosters, the economic forces should eventually drive the C.I.S. boosters into the 1-sigma error band. A check of C.I.S. booster technical capabilities showed no great advantage, but their boosters were in the upper range of performance comparisons with equivalent Western boosters. One cost advantage which the C.I.S. boosters do have is the benefit of a significant rate and learning curve advantage with the Soyuz launch vehicle. This factor has a direct influence on the ability to reduces launch costs, and the Soyuz is in a class by itself in this category. The Soyuz booster was therefore predicted to reach an equilibrium price below the current cost per pound error band. A price at the 2-sigma lower error band is predicted for the Proton launch vehicle, which equates to a per launch cost of $140M ('92$), assuming uninterrupted operations at the Baikonur (Tyuratam) launch site.

A cost estimate for the LRB concept was prepared based upon a combination of historical costs (for the F-1A and J-2S engines, as well as for the Saturn S-IVB second stage), and projected costs for a new LOX/RP booster first stage. The actual cost data which exists for three of the four elements of this booster concept was used for estimating production costs of these elements. Development costs for these elements was estimated from the percentage of new design, additional testing, and new or modified tooling which would be required to build and re-certify the systems for flight.

A point to remember concerning these manned booster cost estimate comparisons is that only the Titan IV and the Space Shuttle costs are real. Most of the booster concepts examined in this study are only paper systems. Their cost estimates are based on assumptions and often optimistic forecasts. Even the C.I.S. launch vehicle cost estimates are based on assumptions and favorable economic forecasts. The paper systems (and even the C.I.S. boosters) should therefore be compared only among themselves, as a class separate from the Shuttle and Titan IV launch systems.

Cost Estimates Do **NOT** Include

... extra costs to **MAN-RATE** launch vehicle
... extra costs to operate in *Manned Spaceflight Awareness* environment

Rough Order of Magnitude (ROM), parametric CER (SEE ~+- 20%)

... estimates in constant-year 1992 US$, commercial equivalent launch, circa 1998
... Ariane 5 development (DDT&E)
... Ariane 5, Proton & Energia cost per flight

Consistent With Level of Design Definition

... launch vehicle
... payload capability (maximum) at launch site latitude
... gross lift-off weight
... stage level data incomplete, inconsistent

Foreign Currency Exchange Rates

Primary Sources of Data

... *International Reference Guide To Space Launch Systems*, AIAA
... *Soviet Year In Space*, TRW
... *Aviation Week & Space Technology*
... anecdotal, US DoC

Estimates Are ***NOT COMPARABLE*** *To NLS-Based Estimates*

Figure 2.1-57 Cost Estimates for Foreign Launch Vehicles

* Methodology
* Data Base Limitations
* Exchange Rates
* Foreign Productivity (man-year equivalent)
* *ONLY SOYUZ Has Actually Launched Crew*

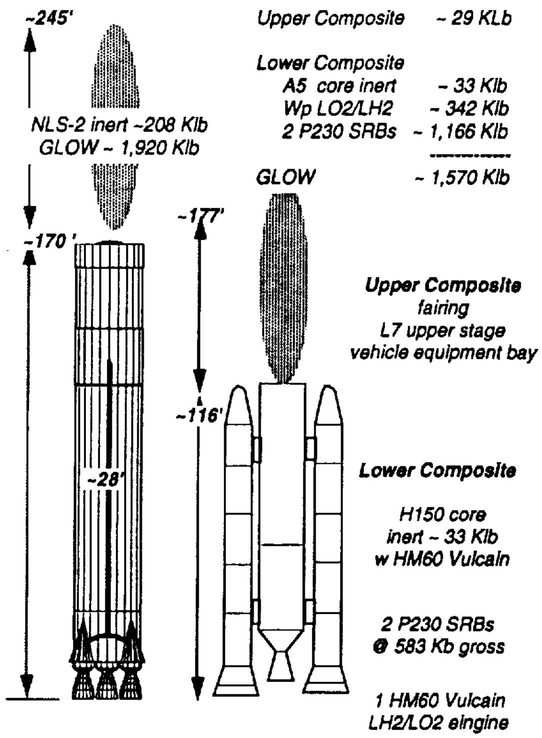
European Space Agency (ESA) & Ariane

* Commercial Operations
* Exchange Rates
* Hermes De-Scoped (unmanned X2000)
* Ariane 5 Man-Rating ?

Commonwealth Independent States (CIS) & Soyuz, Proton & Energia

* Political Stability
* Launch Rates
* Free-Market Economics (labor/factor mobility)
* Productivity
* Exchange Rates

Figure 2.1-58 Considerations for Unique Foreign Boosters



Specific Ariane 5 Groundrules

* commercial launch circa 1998
* '90-'91 average exchange rates
* 50-70 unit production lot buy
* ESA productivity ... capital/labor split
* annual fight rate ... 8 to 12
* profit (loss) Incentives

Development Cost $M'92	**$ 5,412**
DDT&E	4,326
N/R Production	347
Construction of Facilities	739
Cost Per Flight $M'92	**$ 90**
launch vehicle	66
operations	6
profit (support)	6
reserves & other direct costs	12

Cost Per Flight estimate based on
gross lift-off weight, payload capability & other data sources
NOT COMPARABLE TO NLS-BASED ESTIMATES

Figure 2.1-59 Ariane V Cost

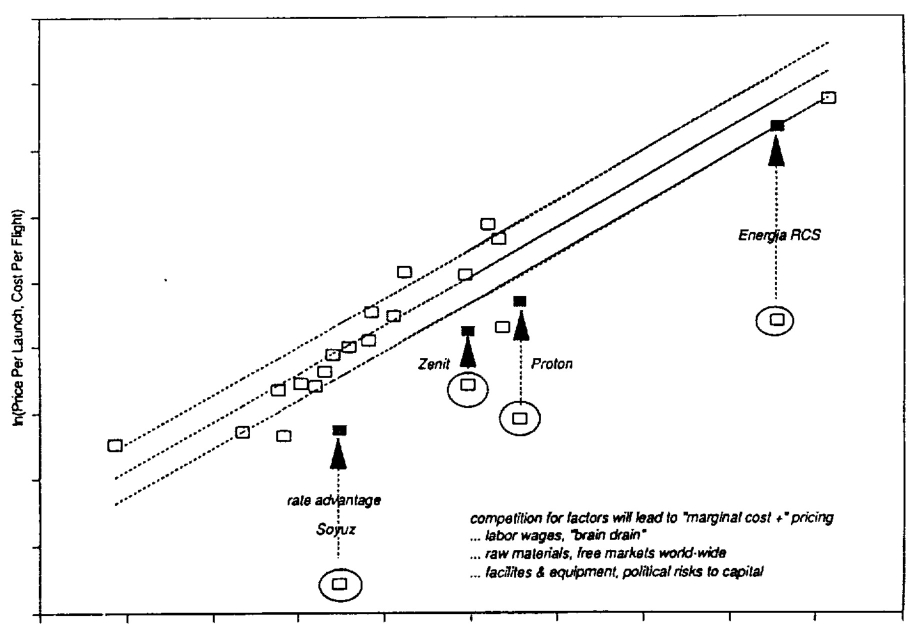


Figure 2.1-60 Estimating Russian Launch Vehicle True Costs

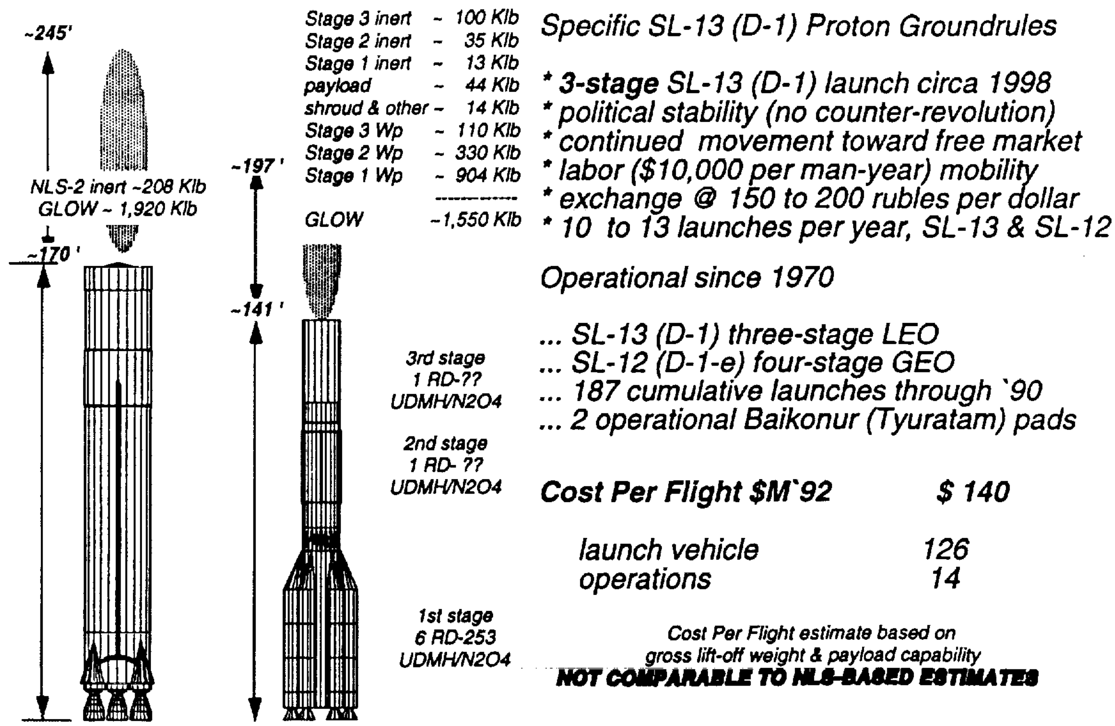


Figure 2.1-61 Proton Cost

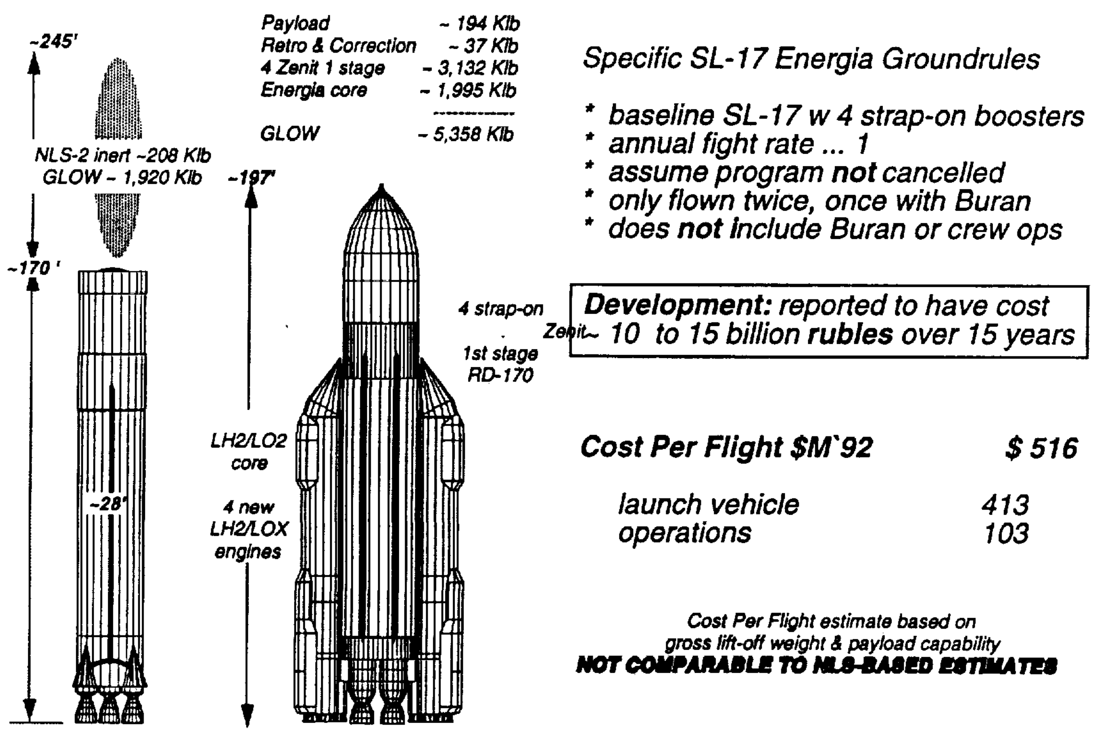


Figure 2.1-62 Energia Cost

Cost per pound delivered to LEO ($K/lb)

Booster	$$ (M)	PLS	PLS + Cab.	Scaled HL-20	CLV	Other
NLS-2 1.5 (6/2)	100	2.94	1.85	1.85	1.32	
NLS-2 1.5 (4/1)	85	2.50	1.57	1.57		
NLS 2stg (STME)	116			2.15	1.53	
NLS 2stg (F-1)	132			2.44	1.74	
Ariane V	**90**	**2.65**				
Proton	140	4.12				
Energia	516		9.56		6.79	
LRB (F-1A)	244	7.18	4.52			
Titan IV	**180**	**5.29**				
SSTO	5.4 (to 56)					.23 (2.33)
NDV	3 (to 50)					.11 (1.89)

Figure 2.1-63 Net Payload Delivery Cost Comparisons

2.1.4 Launch Processing Analysis

A launch operations analysis was performed on most of the manned booster concepts evaluated during this study. Insufficient data was available to fully analyze the C.I.S. Proton launch vehicle or the Ariane V launch vehicle. Launch processing flow diagrams and facility/resource scheduling networks were, however, prepared for these two vehicles. As data becomes available concerning these boosters, an analysis of their launch processing characteristics can be performed. All other manned booster concepts were fully analyzed to determine the facility and resource requirements needed to meet their design launch rate capabilities. The analyses were performed using the STARSIM model which simulates the booster's interactions with launch facilities and all required system elements. The model can be used to determine specific resources required to meet a given flight rate, or it can be used to determine the system's maximum flight rate given a specified number of resources. The model includes the scheduling constraints and algorithms needed to accurately predict launch processing capabilities at the NASA KSC facility.

Application of this simulation model to the Space Shuttle system revealed that the Shuttle currently has the capability to fly as many as 10 flights per year with existing assets and launch processing efficiencies (turnaround processing times). The design maximum flight rate for the Shuttle system is 12 flights per year, with a surge capability (6 month maximum rate) of 7 flights. The mobile launch platforms were the limiting resources for Space Shuttle flight rate capacity, with a utilization rate of 95%. Utilization of the OPF and ET checkout cells were also high (about 85%), but the launch pad utilization was less than 50%.

Similar analyses of the NLS-2 boosters revealed that these systems also were constrained by the mobile launch tower, of which the NLS program had planned only one. The NLS booster, operating with a parallel PLS launch processing system, was able to achieve a flight rate of 10 flights per year. The NLS-2 booster design maximum flight rate was determined to be 13 flights per year, with a surge capability of 7 flights in six months. Utilization of KSC facilities which would be shared with the current Shuttle systems (the VAB and launch pads) was very low at these maximum flight rates, 37% and 31% respectively. This implies that a mixed fleet of Shuttle and NLS-2 launch systems is feasible with the existing KSC facilities. An actual mixed fleet analysis has not been performed with the STARSIM model yet, but the low utilization of facilities indicates that this scenario can be implemented at projected NASA flight rate planning levels.

Analysis of the other NLS-2 configurations (2 stage versions) revealed that a flight rate of 10 per year was common to all the configurations and in all cases the mobile launch tower was the constraining resource.

An analysis of the effects of launch processing delays on achievable annual flight rates was also performed with the NLS-2 booster. This analysis was performed to determine what levels of system availability would be required to achieve a desired flight rate of 8 flights per year. The model simulated system stand-down periods of up to 100 days and at varying levels of probability of incurring the stand-down (0, 5, 10, and 20 %). This analysis revealed that the NLS-2 booster is highly tolerant of launch processing delays. The desired flight rate of 8 per year could be achieved with launch processing delays of up to 75 days with a 20% probability of occurrence. At a 10% probability of occurrence, delays of up to 100 days could be accommodated and still meet the 8 flight per year goal. As a point of comparison, the NLS-1 booster (an ET-based core with 2 Shuttle solid rocket boosters) was also modeled under similar conditions (a flight rate of 5 per year was the goal). This system was found to be very sensitive to launch processing delays. A 5% probability of just a 20 day delay was sufficient to reduce the flight rate by 1 per year. Any delays (> 5% probability) greater than 50 days would reduce the annual flight rate by two. This strong sensitivity to system availability is attributed to the NLS-1 solid booster's high utilization rate of the mobile launch platform.

An analysis of launch processing learning curve effects on the ability to meet flight rate goals was also performed. This analysis was based on data from the Space Shuttle program and attempted to determine if continued learning (as already demonstrated in the 50+ Shuttle flights to date) would permit higher flight rates to be achieved in the future. This analysis provides insight into whether a booster system is process limited or resource limited in its launch rate capability. The analysis identified that a 79 % learning curve has been established for launch processes on the Space Shuttle program in the post-51L era. Although many launch processes were lengthened following the Challenger accident, the launch process continues to experience learning at a respectable rate. Extrapolation of this learning rate into the future indicates that the Space Shuttle launch processing times should return to the pre-51L levels at approximately the eightieth Shuttle launch. Space Shuttle processing times of 50 days could be achieved by the 140th Shuttle launch. This analysis reveals that the Shuttle system flight rate is currently process time limited, not resource (facility or equipment) limited. A similar analysis of the NLS-2 booster system has not been performed.

- Computer-Aided System Engineering (CASE) analysis was used to develop concept processing flow diagrams
  - Same rule set as KSC Mission Planning Office
- STARSIM is programmed using object-oriented design for modular model reconfiguration
- STARSIM software objects are easily tailored to new concepts

Space Shuttle Launch Processing Flow Diagram

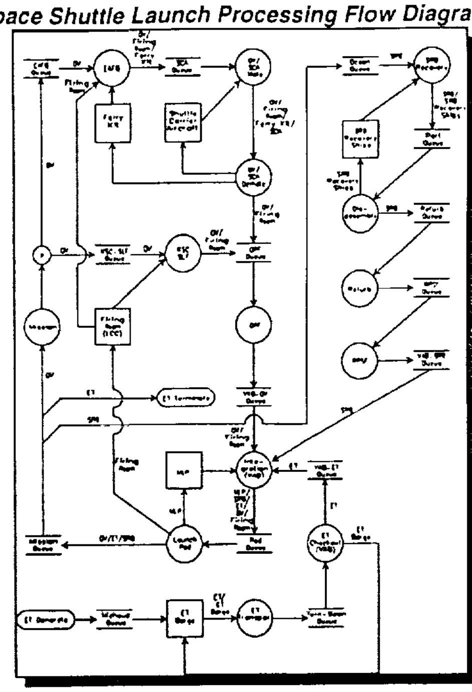


Figure 2.1-64 STARSIM Launch Simulation Software

• **Throughput Capacity:**	
• **Launch System Design Max**	**12.6 flights/year**
• **Target (Derated to 80%)**	**10.1 flights/year**
• **Post STS-51L Avg. Planned Days**	**9.5 flights/year**
• **6-Month Surge Capability**	**7 flights**
• **Launch Operability Index (LOI)**	**16**
• **Total Man-hours/Flow**	**407,000**

Additional launch rate capacity exists in today's Space Shuttle system

Figure 2.1-65 Space Shuttle Launch Simulations

Facility/Resource Capability:

Name	OPF	VAB-1&3	VAB-2&4	MLP	Pad
Location	KSC	KSC	KSC	KSC	KSC
Status	Existing	Existing	Existing	Existing	Existing
Design Max Utilization	82%	85%	49%	95%	48%
Throughput Capacity:					
• Planning @ 80% (flows/yr)	12.3	11.8	20.6	10.6	20.8
• Max @ 100% (flows/year)	15.4	14.8	25.7	13.2	26.1
Manpower (man-hours/flow)	213K	11K	15K	34K	78K

Figure 2.1-66 Space Shuttle Facility Utilization

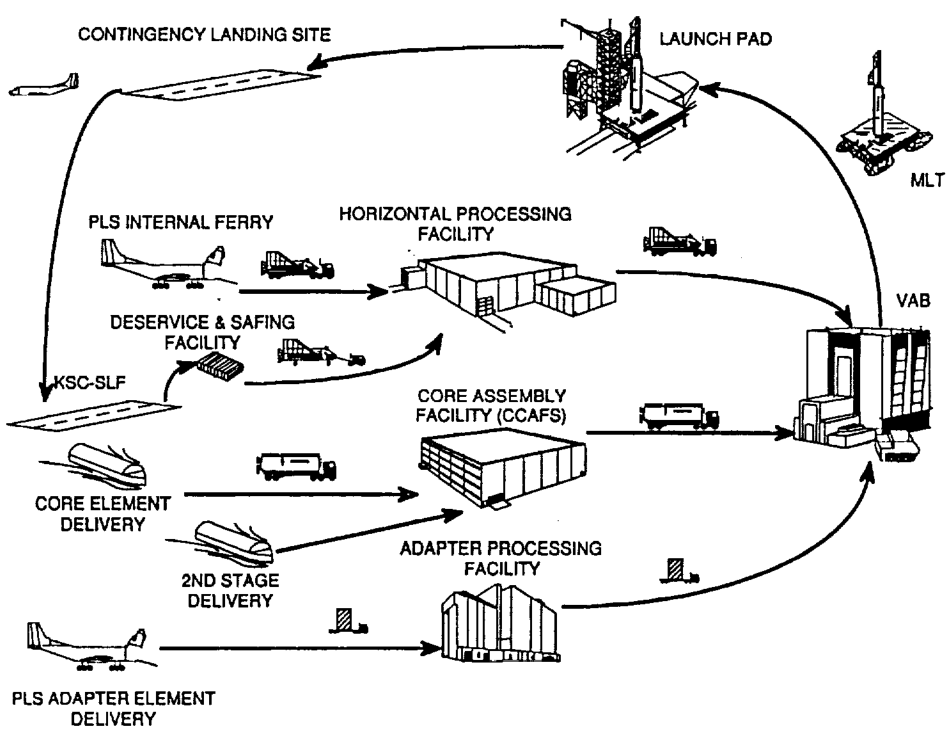
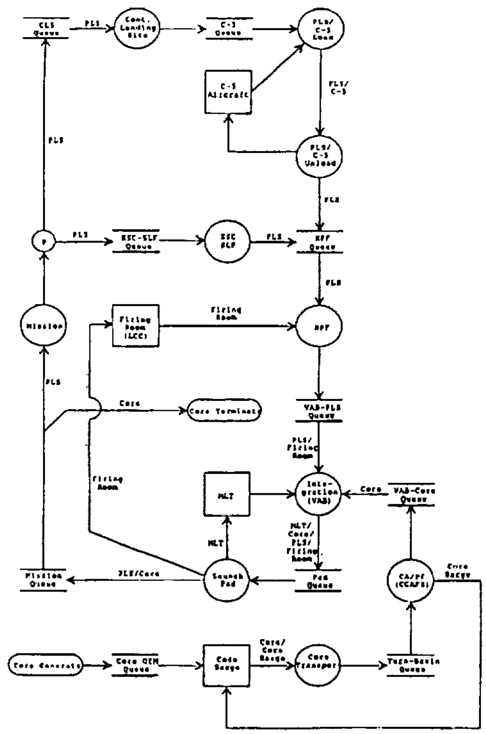


Figure 2.1-67 NLS-2 and PLS Launch Processing



PLS/NLS-2 Simulation Output

Output of STARSIM/PLS_NLS2: SERVER/RESOURCE Statistics

SERVER/ RESOURCE	TOTAL SERVED	AVG UTIL	AVG BLOCK	UNITS BUSY	UNITS FREE
CLS_CREW	11	0.015068	0.000000	0	1
CS	11	0.011781	0.000000	1	0
KSC_CREW	123	0.033699	0.000000	0	1
DSF	123	0.033699	0.000000	0	1
HPF_BAY	132	0.878904	2.204384	3	1
VAB_LBAY	130	0.394521	0.000000	1	0
LPAD	130	0.605479	0.000000	0	2
MLT	130	1.000000	0.000000	1	0
MIS_CTRL	130	0.213699	0.000000	1	0
FIR_ROOM	130	3.525205	0.000000	4	0
C_BARGE	133	0.366575	0.633425	1	0
APF	132	0.108493	0.891507	1	0

Output of STARSIM/PLS_NLS2: PLS QUEUE Statistics

QUEUE	AVERAGE LENGTH	AVERAGE WAITING	CURRENT LEN	MAX LEN	ENTITIES DEQUEUED
Q @ CLS_CREW	0.008219	3.000	0	1	10
Q @ CS	0.000000	0.000	0	1	11
Q @ KSC_CREW	0.000000	0.000	0	1	123
Q @ DSF	0.000000	0.000	0	1	123
Q @ HPF_BAY	0.000000	0.000	0	1	134
Q @ VAB_LBAY	2.203562	60.932	1	3	131
Q @ LPAD	0.000000	0.000	0	1	130
Q @ MIS_CTRL	0.000000	0.000	0	1	130

Output of STARSIM/PLS_NLS2: CORE/ADAPTER QUEUE Statistics

QUEUE	AVERAGE LENGTH	AVERAGE WAITING	CURRENT LEN	MAX LEN	ENTITIES DEQUEUED
CORE QUEUES:					
Q @ C_BARGE	0.000000	0.000	0	1	134
Q @ CA/PF	2.612603	71.699	2	3	131
Q @ VAB_LBAY	1.130356	31.477	1	2	131
ADAPTER QUEUES:					
Q @ APF	0.990904	27.621	1	1	131
Q @ VAB_LBAY	0.891233	24.644	1	1	131

Output of STARSIM/PLS_NLS2: MISSION Statistics

	MEAN	MAX	MIN	NO. OBSERVATION
Time on ground	104.8	162.0	24.0	130
Time bet launches	28.0	28.0	28.0	129

Figure 2.1-68 PLS/NLS-2 Simulation Model

Throughput Capacity:

Launch System Design Max	13.4 flights/year
Target (Derated to 80%)	10.7 flights/year
6-Month Surge Capability	7 flights

Facility/Resource Capability:

Name	HPF	VAB-4	CA/PF	MLT	Pad
Location	KSC	KSC	CCAFS	KSC	KSC
Status	New	Modified	New	New	Modified
Design Max Utilization	**23%**	**37%**	**22%**	**100%**	**31%**
Throughput Capacity:					
• Planning @ 80% (flows/yr)	46.8	28.6	48.4	10.7	34.3
• Max @ 100% (flows/year)	58.5	35.8	60.6	13.4	42.8

- MLT is the constraining resource
- Low utilization of KSC facilities

Figure 2.1-69 PLS/NLS-2 Launch Facilities Utilization

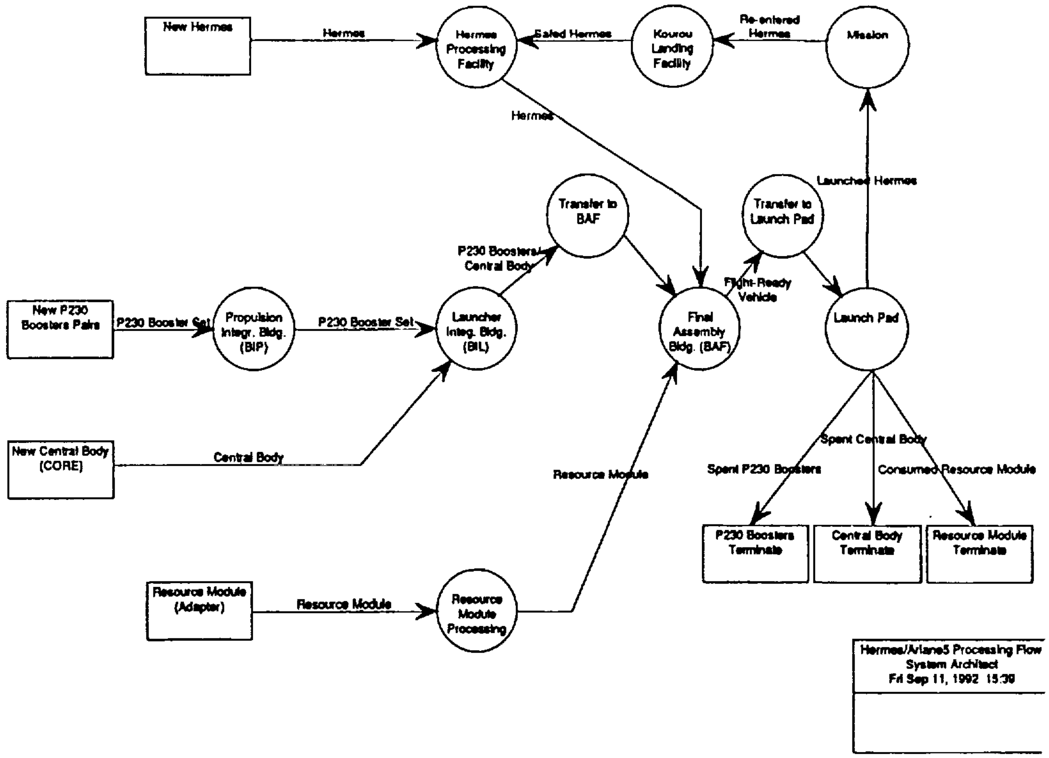


Figure 2.1-70 Ariane V Launch Processing Model

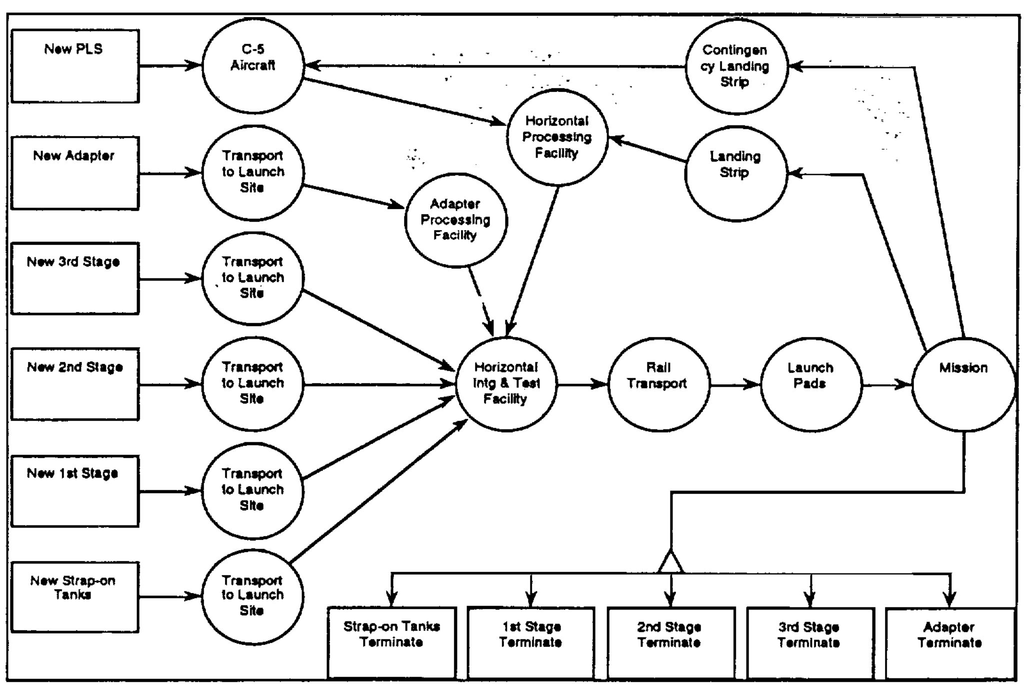
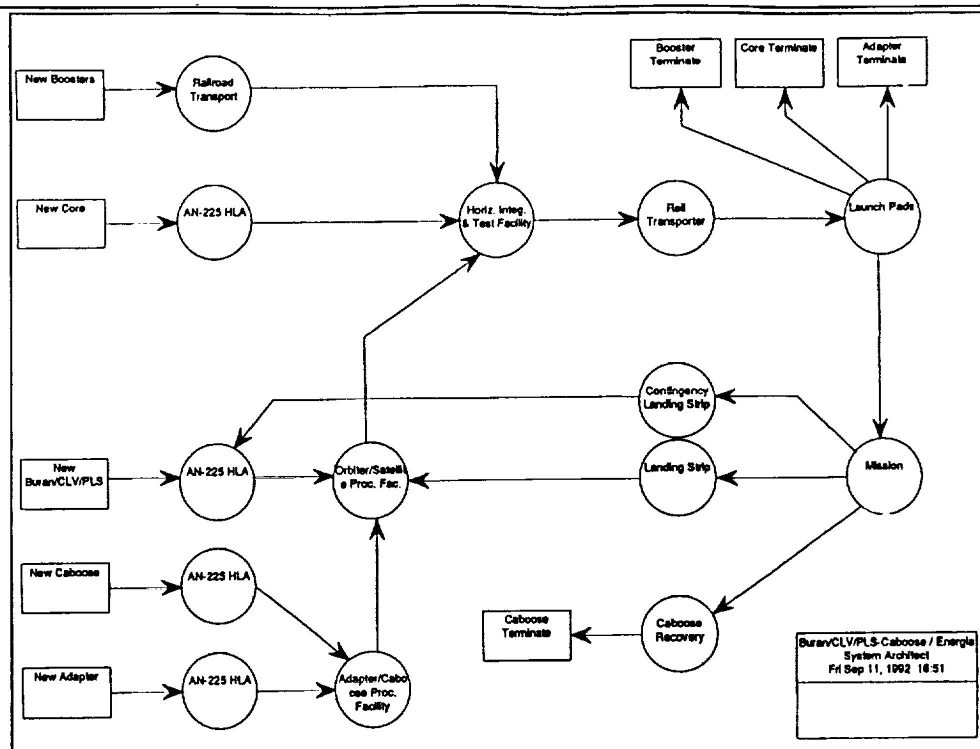


Figure 2.1-71 Proton Launch Processing Model

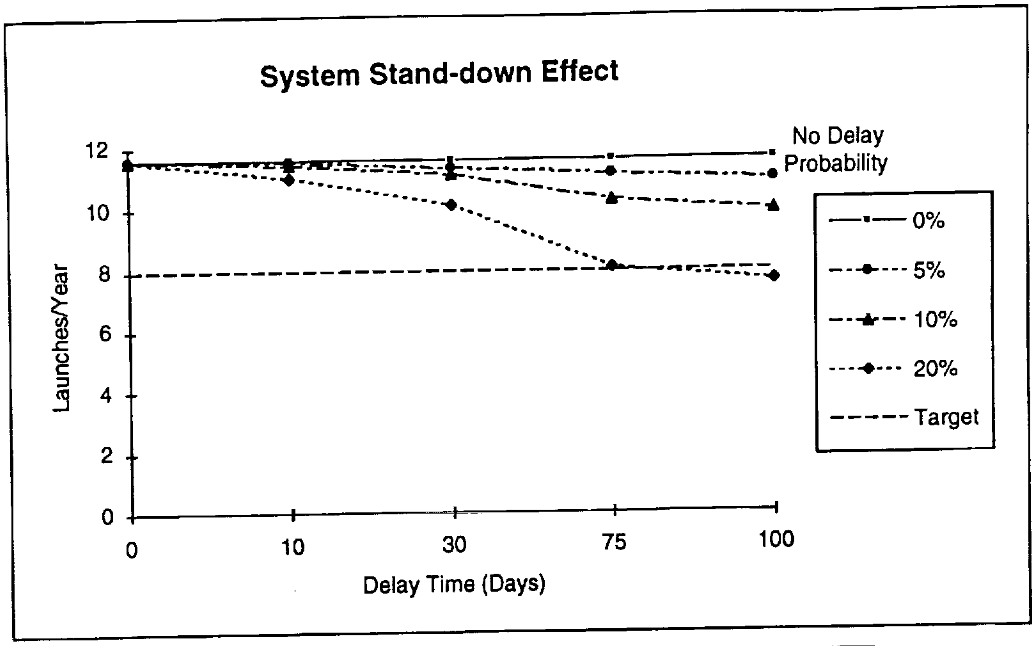


10/4/92

Figure 2.1-72 Energia Launch Processing Model

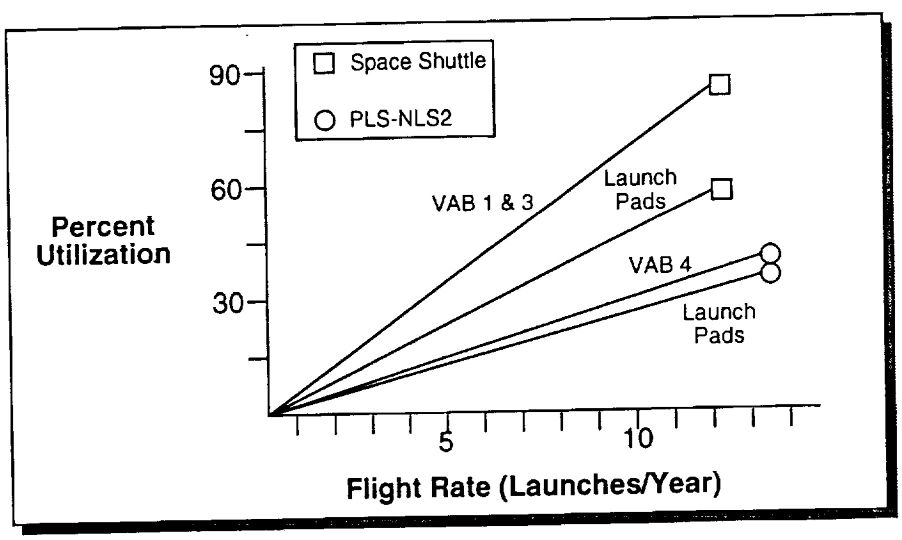
	Flight Rate (Flts/Yr)	Constraining Resource
NLS 1.5 Stage (6/2)	10	MLT
NLS 1.5 Stage (4/1)	10	MLT
NLS 2 Stage (STME)	10	MLT
NLS 2 Stage (F-1)	10	MLT
Ariane V	10	TBD
Energia	1	TBD
Proton	8	TBD

Figure 2.1-73 Comparison of Booster Launch Processing Simulations



- **High system tolerance for delays & short stand-down periods**

Figure 2.1-74 NLS-2 Availability Impact to Flight Rate



There should be sufficient capacity for Space Shuttle and PLS-NLS2 mixed fleet operations

Figure 2.1-75 NLS-2/PLS & Shuttle Mixed Fleet Operations

2.1.5 Reliability Analysis

As a prerequisite to conducting Reliability analyses of candidate manned launch vehicles, a thorough understanding of the reliability histories and operational problems of previous launch vehicles is required. The manned booster reliability analysis was performed by utilization of a computerized database, called RAM, which contains operational and design details for the following launch vehicles: Delta, Titan, Ariane, Space Shuttle, and Atlas. Each launch of each vehicle is accurately detailed with respect to:

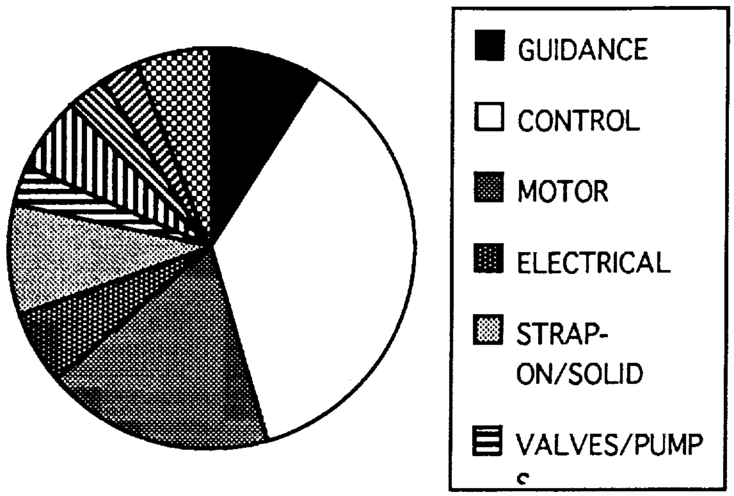
- When launched (date)
- Launch Site (ETR, WTR, etc.)
- Responsible agency (NASA, USAF, ESA, etc.)
- If mission was successful or not
- If unsuccessful, cause of failure and vehicle stage(s) responsible for failure
- Failure details (i.e., 2nd stage hydraulic system failure)

Moreover, the RAM database contains specific design and operational parameters for each vehicle and its associated stage(s). These parameters encompass: number of rocket motors, boosters and strap-ons, stages used, thrust levels by stage, burn times by stage, rocket motor designations (part numbers), fuels employed by stage, etc.

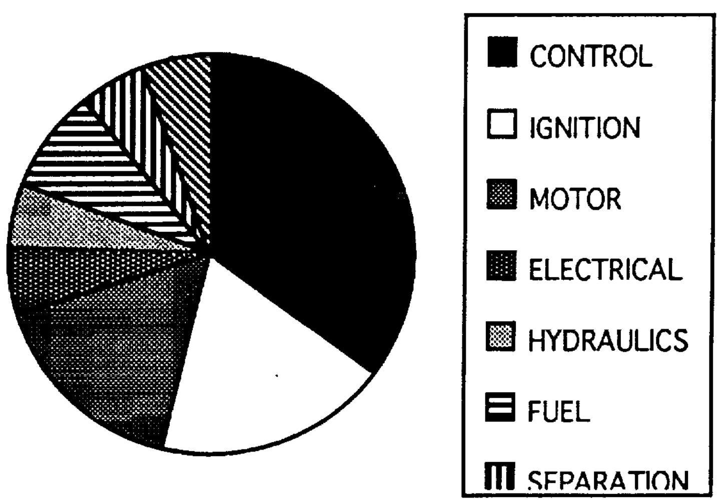
From the above information it was possible to generate statistics such as overall mission reliability by vehicle, mission reliability by stage, subsystem failure rates (MTBF's), subsystem percentage contributions to overall failures, etc. These statistics reveal design-related criteria that point the way to better designed and more reliable launch vehicles in the future. These parameters were also used as the means of comparing the several candidate manned booster concepts.

Pictorial displays from the RAM database show some of the more revealing statistical facts developed . Note the significant variation in failure causes by vehicle stage. To illustrate, the principal cause of 3rd stage failure appears to be ignition-related (40% of all 3rd stage failures), whereas 2nd stages suffer mostly from control problems. Frozen valves at high altitudes have contributed to a large fraction of all 2nd stage mission failures, and should be recognized as a design problem to be eliminated for future launch vehicles.

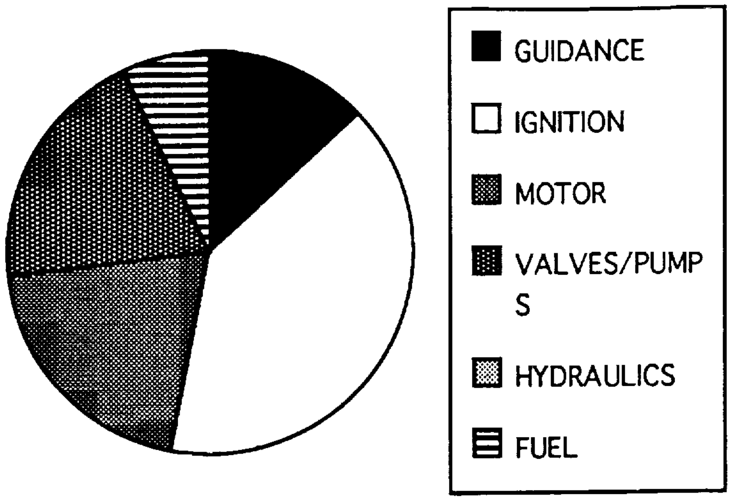
Pictorial Displays of Failure Causes



1st STAGE



2nd STAGE



3rd STAGE

When launch vehicle types are compared to one another it has been the usual practice to simply compare their mission reliability. If, for example, Vehicle "A" has an "observed" reliability of 0.915 and Vehicle "B" has an "observed" reliability of 0.892, the former is regarded as the more reliable of the two. Operationally, this is probably true, but this assessment fails to consider all pertinent data. To illustrate, from a time-dependent* MTBCF point-of-view the opposite might be true. If A's launch burn is 200 seconds and B's launch burn is 300 seconds it can be demonstrated that A's time-dependent MTBCF is less than B's time-dependent MTBCF. Therefore, on an equal exposure basis, B is more reliable than A. The table below shows time-dependent MTBCF and burn time related to reliability. Comparison of paper launch vehicle concepts, as was generally required in this study, must take these factors into account when estimating the booster's reliability.

* MTBCF has both a time-dependent component and a cycle dependent component. The RAM database contains a cycle-based or time-based "flag" for each reported failure.

VEHICLE	RELIABILITY	BURN TIME	MTBCF (time-dependent)
A	0.915	200 seconds	2251 seconds
B	0.892	300 seconds	2625 seconds

Reliability vs. Burn time vs. MTBCF

It follows that consideration must also be given to differences in the missions themselves. One vehicle may use two stages to reach LEO while another vehicle uses 3 stages and achieves a significantly higher orbit. A simple comparison of their numerical "observed" reliability may not fairly represent such vehicles when comparisons are attempted.

The degree of designed-in redundancy manifests itself in the levels of reliability actually achieved. It appears that numerous failure modes exist that could not be remedied by the use of redundancy. Valves freezing due to entrapped moisture would still freeze regardless of the number of redundant valves provided, whereas loss of guidance (i.e., IMU failure) might be virtually eliminated by adding an adequate number of redundant IMUs. However, the additional Life-Cycle Costs required to provide this redundancy must be weighed against the Life-Cycle Costs associated with vehicle loss and/or failure to meet mission objectives.

Application of the Above Methods to ATSS Studies.

When two or more candidate launch vehicles are compared the question sooner-or-later asked is: "Which booster is the most (or least) reliable, and why?" In fact, the question arose when the numerous NLS-2 booster configurations were under study. Details of each booster's design were basically limited to three-view drawings, but engine types and quantities were known, as was the booster performance and trajectory data.

For example, one of the proposed NLS-2 two stage vehicles was configured with 4 STMEs and 1 J-2S engines. Unfortunately, the reliability of the STME is not known since it has yet to be built and tested. A reasonable assumption, however, is that STME reliability will be as good as the current SSME (Space Shuttle Main Engine), an engine for which reliability is known with some precision and confidence. On 9/12/92, the SSME MTBCF was estimated to be 77,000 seconds. This 77,000 second MTBCF was utilized for all STME applications in candidate NLS booster configurations.

Similarly, MTBCFs for other engine types (i.e., the F-1, J-2S, etc.) were derived directly from operational experience data of these engines.

Reliability of all the candidate NLS manned booster configurations, as well as the currently operational vehicles such as the Proton (D-1-e), Energia, and Titan IV, were computed against the same "baseline" mission. That mission was the delivery of a manned payload (PLS or CLV) to a 15 X 220 Nmi transfer orbit to Space Station Freedom. For each vehicle, propellant masses and engine burn times (by stage) were provided from trajectory analyses of each booster as described in the Performance Analysis section of this report.

To illustrate how booster reliability estimates can be made in the absence of detailed design data, consider the following. Given that all we know is:

1. Vehicle "A" Has 3 SSME Engines (all must work).
2. Vehicle "B" Has 4 SSME Engines (all must work).
3. SSME burn times are identical for both vehicles.

The reliability of each vehicle can be calculated and comparisons made based strictly on the engine MTBCF. First, we find that Vehicle "A" is inherently more reliable than Vehicle "B" because the probability that 3 of 3 engines will work is higher than the probability that 4 of 4 engines will work. To illustrate, for an SSME MTBCF of 77,000 seconds and a burn time of 480 seconds, the following engine system reliability (R) will be achieved:

R_A (3 of 3) = 0.9815
R_B (4 of 4) = 0.9754

Conclusion = Vehicle "A" has higher reliability than Vehicle "B".

If, however, the 4-engine vehicle has single engine-out capability and the 3-engine vehicle does not, a substantially different answer results:

R_A (3 of 3) = 0.9815
R_B (3 of 4) = 0.9998

Conclusion = Vehicle "B" reliability is much better than Vehicle "A"!

This type of reliability analysis was performed for each of the manned booster concepts examined in this study. Reliability of each booster was estimated based on number of stages, number and type of engines in each stage, engine burn times for each engine, and engine-out capabilities. These reliability estimates were reported as the booster's Probability of Mission Success. Incorporating the PLS escape system (number and type of motors, motor burn times) as an additional stage to be employed if the earlier stage did fail, a crew safety estimate was also made for each booster (reported as Probability of Safe Return). It should be noted, however, that reliability <u>predictions</u> were made for the paper booster concepts. Reliability estimates for existing boosters (Space Shuttle, Proton, and Titan IV) are actual, demonstrated reliability values.

The reliability analyses showed that the NLS-2 boosters are predicted to achieve higher reliability than current expendable launch vehicles, and also higher than the soon to be introduced Ariane V. The NLS-2 boosters will also have higher reliability and higher crew safety than the current Space Shuttle. Results of the reliability analyses for each booster are presented below.

Table 3 Manned Booster Reliability Estimates

Vehicle	Engine Out ?	Reliability (Probability. of Mission Success)	Crew Safety (Probability. of Safe Return)
Non-NLS			
Space Shuttle	No	.9500	.9840
Ariane V	No	.9805	.9999028
Energia	No	.9771	.9998857
Proton	No	.9261	.9996305
Titan IV	No	.9169	.9995847
E-HLLV	No	.9677	.9998388
LRB (F-1A)	No	.9785	
NLS			
NLS-2 (6/2)	Yes	.9897	.9999488
" no engine out	No	.9811	.9999058
NLS 2 stage(STME)	No	.9765	.9998824
NLS 2 stage (F-1A)	No	.9845	.9999225
NLS-2 (4/1)	No	.9841	.9999205
Fully reusable			
AMLS	Yes	.9880	.9999947
" no engine out	No	.9500	.9999947
NASP derived	Yes	.9500	.9999950
SSTO (HTOHL)	Yes	.9500	.9999950
SSTO (VTOHL)	Yes	.9894	.9999946

The above results reveal that the variability in manned booster reliability (which has been estimated from the number of stages, engine type, and engine burn times) is much greater than the effects of engine out capability. Where a class of booster concepts uses the same engine (such as with the NLS-2 class of boosters), the engine out effect is strong and provides for a high probability of mission success. The single greatest reliability driver among these booster concepts is the engine reliability itself.

Also seen in the above results is the high crew safety estimates afforded by the PLS (or CLV) escape system. The capability for the crew (payload) to escape from the booster and return safely to Earth provides the greatest measure of crew safety. The effects of booster reliability or booster engine out capability on crew safety are minor compared to this escape capability.

- Reliability Analyses are based on "real world" historical data
  - Historical data for Atlas, Delta, Titan and Shuttle
  - Include all flights, through May 1992, for the above vehicles
  - Each flight is independently tracked
    - success, failure, what failed, when, why?

- Data includes:
  - vehicle model number
  - number of stages
  - burn time by stage
  - engines and fuels employed
  - avionics operating time

- Database Sort and Query capability

Figure 2.1-76 Reliability Estimates Based on Flight History

- **For manned launch vehicles, Failure ≠ Loss**
  - Most references to Shuttle reliability are in the context of vehicle and crew loss
  - This is understandable, because for ELVs
    - reliability is measured that way...
      mission and payload usually <u>are</u> lost

- **Unlike ELVS, manned systems can recover from critical failures and return without damage**
  - The measure of this capability is "Probability of Safe Return"
  - Currently estimated to be 0.983

- **Most Shuttle missions have been successful, but not 100%**
  - In the event of a mission terminating failure, the Orbiter returns safely to Earth
  - The measure of this capability is "Probability of Mission Success"
  - Currently estimated to be 0.948 for a 7-day mission
  - Varies as a function of mission "specifics"

Figure 2.1-77 Mission Success vs. Safe Return

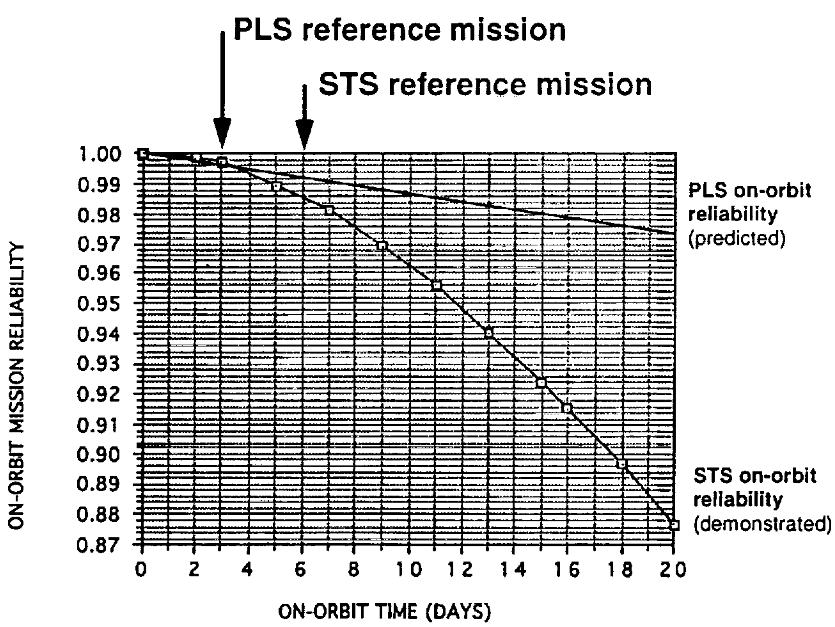


Figure 2.1-78 Mission Reliability vs. On-orbit Time

- SSMEs have proven to be extremely reliable
  - 46 flights flown to date
  - No in-flight engine failures
  - One commanded SSME shutdown
  - Total burn time is 66,240 seconds.

- The MTBCF (Mean-Time-Before-Critical-Failure) is 66,240 seconds
  - Reliability of a single SSME is 0.99278
  - Probability of losing 2 or more SSMEs in the first 260 seconds of flight is 0.00004592.
  - SSME Safe Return reliability is 0.99995408

- SRBs are the limiting element in Shuttle reliability
  - One failure in 47 flights flown to date
  - Total burn time is 11,190 seconds
  - Reliability of a single SRB is 0.9925
  - SRB Safe Return reliability is 0.9925

Figure 2.1-79 Shuttle Propulsion System Reliability

- In-flight failures of a critical avionics system component usually results in loss of an ELV, its payload and its mission
  - Atlas has lost 4 of 204 missions due to avionics
  - Delta has lost 5 of 180 missions due to avionics
  - Titan has lost 2 of 170 missions due to avionics
  - Shuttle has lost 0 of 47 missions due to avionics

- Shuttle cumulative in-flight avionics operating times are 17 times the combined Atlas, Delta and Titan operating times
  - avionics redundancy is credited with this reliability achievement since a number of avionics failures have occurred in critical systems

- Avionics redundancy enables manned systems to remain on-orbit for extended periods

Figure 2.1-80 Shuttle Avionics System Reliability

- The Orbiter has in excess of 6,500 hours of flight avionics operating time

- Mission-terminating avionics failures occur for:
  - Atlas every 10.6 hours
  - Delta every 20.5 hours
  - Titan every 63.5 hours
  - Shuttle every 6,500 hours

Redundancy in avionics components yield startling improvements in reliability

Figure 2.1-81 Avionics Reliability Through Redundancy

PROBABILITY OF MISSION SUCCESS (Demonstrated)

ELEMENT	RELIABILITY	MISSION PHASE	RELIABILITY
SRB	.9850	Prelaunch	.8220
ET	.9990	Ascent	.9640
Orbiter	.9630	Orbit	.9841
		Entry	.9990
Total Mission Reliability			$\boxed{.948}$

PROBABILITY OF SAFE RETURN (Demonstrated)

ELEMENT	RELIABILITY	MISSION PHASE	RELIABILITY
SRB	.9850	Prelaunch	.9999
ET	.9990	Ascent	.9850
Orbiter	.9989	Orbit	.9990
		Entry	.9990
Total Vehicle Reliability			$\boxed{.983}$

PROBABILITY OF SAFE RETURN (Predicted)

ELEMENT	RELIABILITY	MISSION PHASE	RELIABILITY
SRB	.9900	Prelaunch	.9999
ET	.9990	Ascent	.9899
Orbiter	.9989	Orbit	.9990
		Entry	.9990
Total Vehicle Reliability			$\boxed{.988}$

Figure 2.1-82 Summary of Space Shuttle Reliability

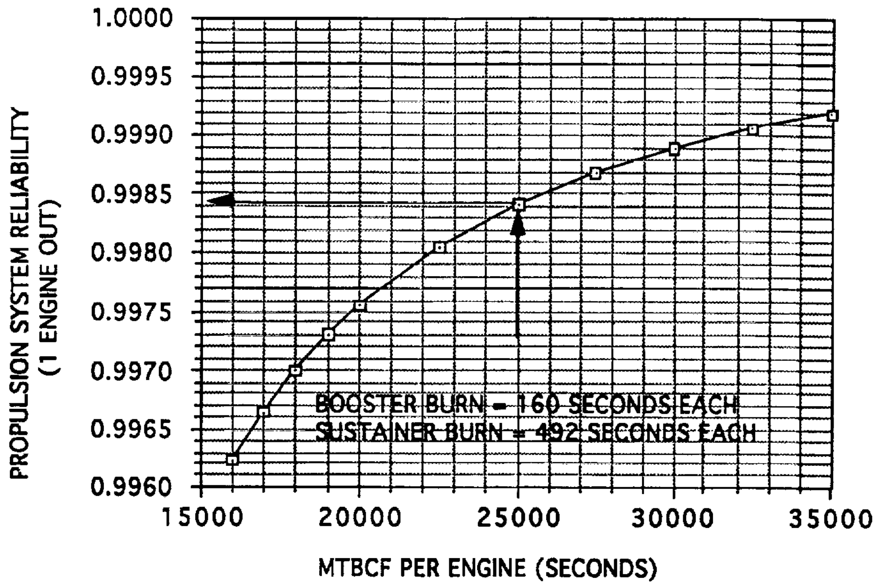


Figure 2.1-83 NLS-2 Propulsion System Reliability

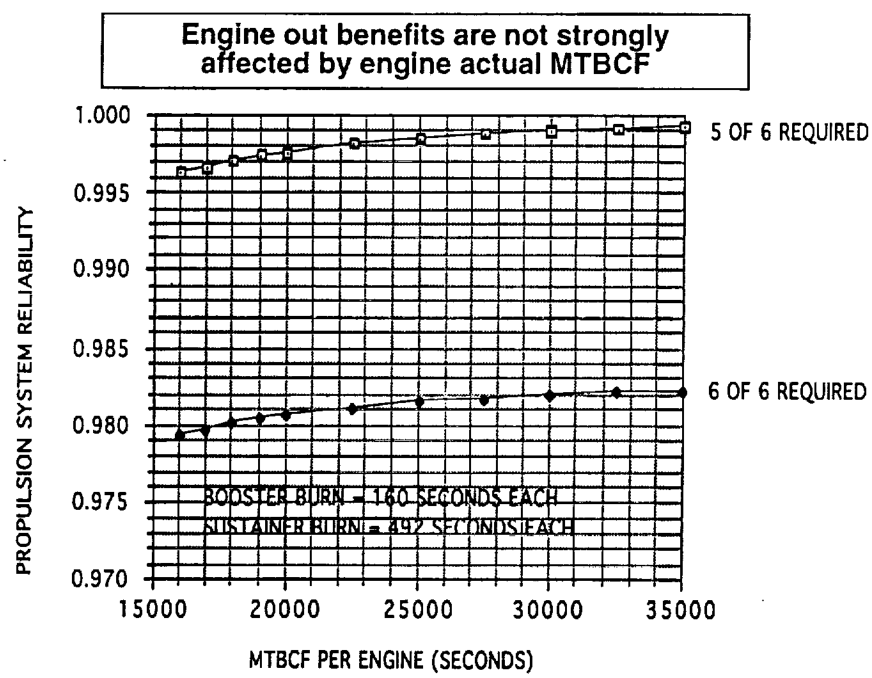


Figure 2.1-84 NLS-2 Engine Out Effects vs. MTBCF

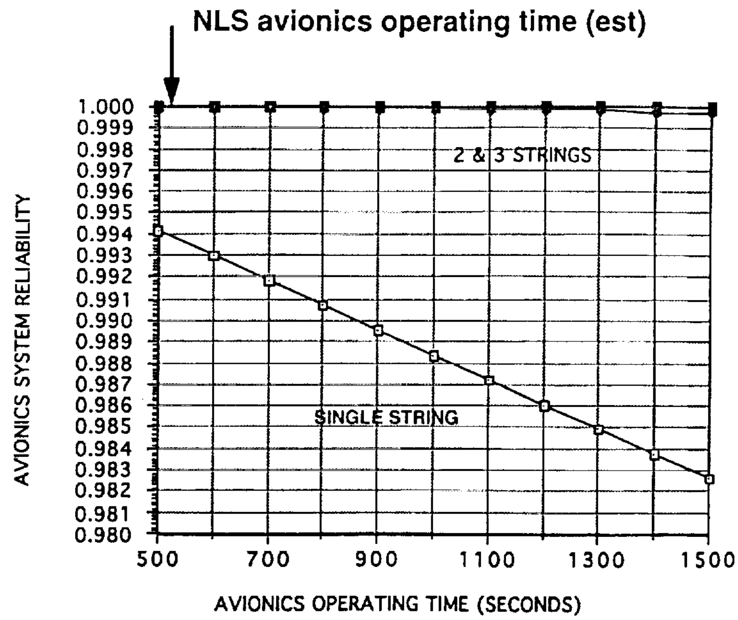


Figure 2.1-85 NLS-2 Avionics System Reliability

PROBABILITY OF MISSION SUCCESS (Predicted)

ELEMENT	RELIABILITY	MISSION PHASE	RELIABILITY
NLS	.981	Prelaunch	.846
PLS	.994	Ascent	.980
		Orbit	.996
		Entry	.999
	Total Mission Reliability		.975

PROBABILITY OF SAFE RETURN (Predicted)

ELEMENT	RELIABILITY	MISSION PHASE	RELIABILITY
NLS	.999	Prelaunch	.9999
PLS	.988	Ascent	.990
		Orbit	.999
		Entry	.999
	Total Vehicle Reliability		.987

Figure 2.1-86 NLS-2/PLS Reliability Summary

	NLS-2 1.5 Stage	AMLS
Prob. nothing goes wrong	.981125	.950054
Prob. something goes wrong - complete mission anyway	.989755	.988025
Prob. something goes wrong - cannot complete mission - safely return to ground	N/A	.998947
Prob. something goes wrong - cannot complete mission - cannot return to ground - SAFELY ESCAPE	.9999488	.9999947
Crew Loss Events (per 10,000 flights)	**.512**	**.053**

Figure 2.1-87 Expendable vs. Fully Re-usable Boosters

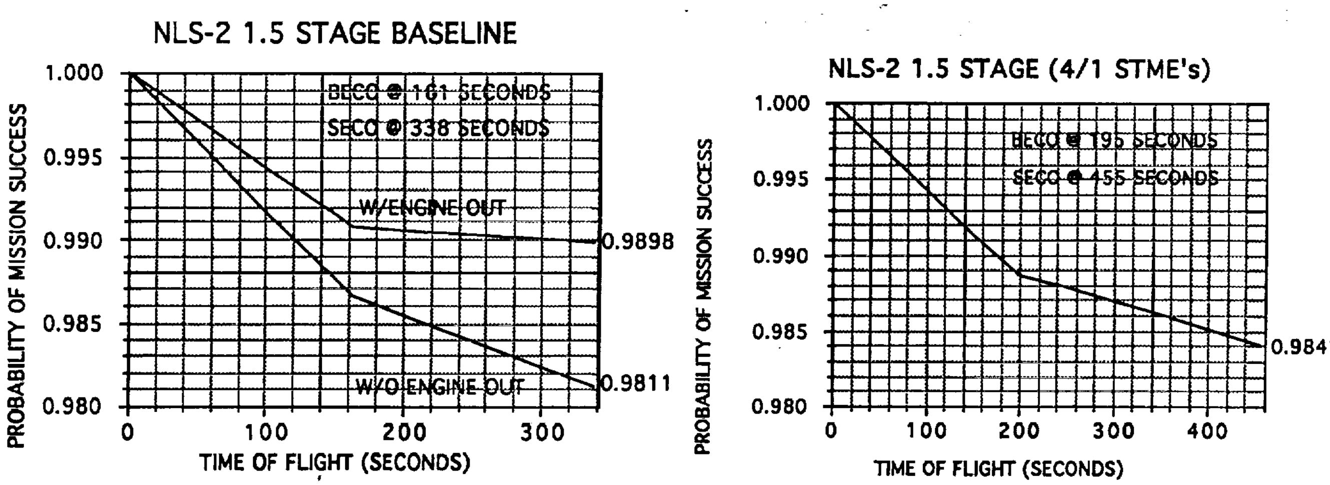


Figure 2.1-88 NLS-2 1.5 Stage Boosters Mission Success

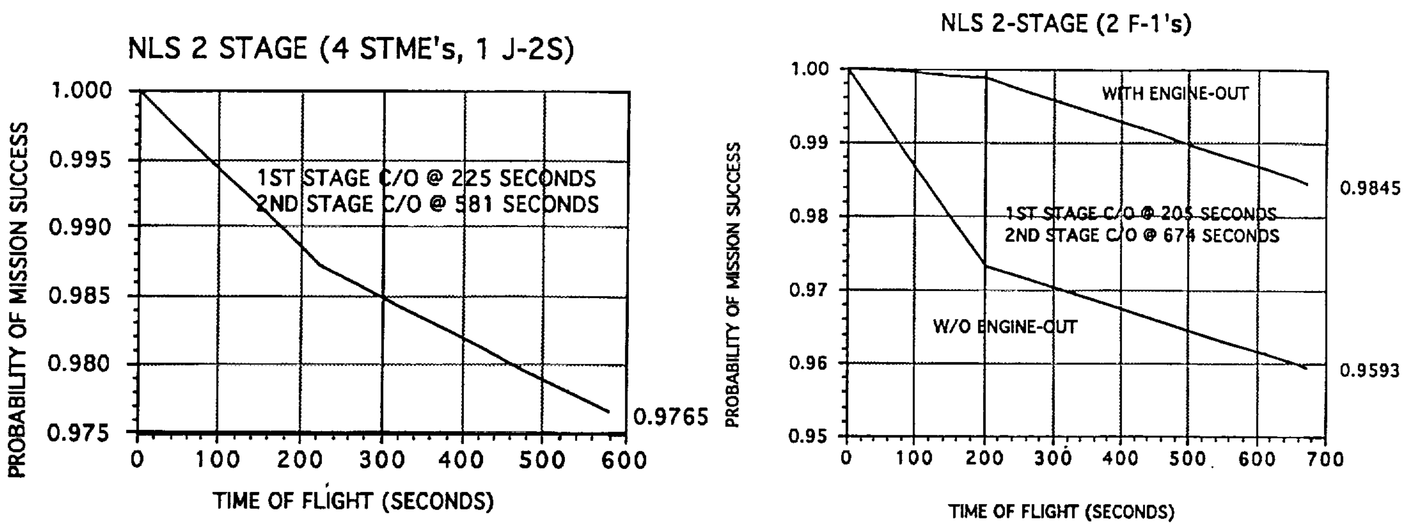


Figure 2.1-89 NLS-2 2 Stage Boosters Mission Success

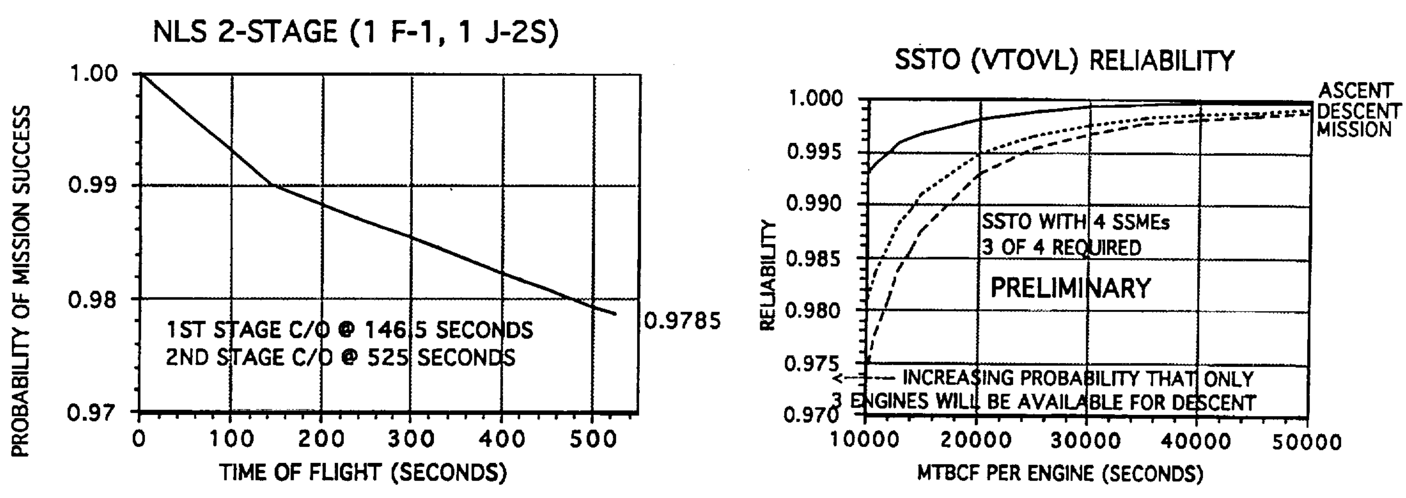


Figure 2.1-90 NLS-2 (F-1) and SSTO Boosters Mission Success

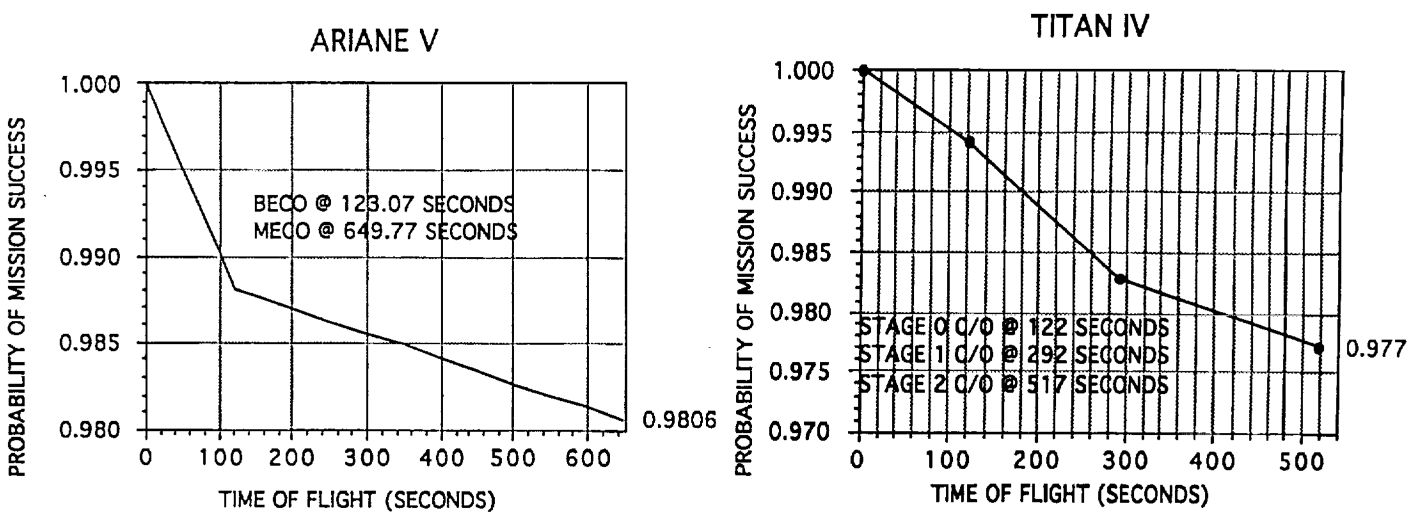


Figure 2.1-91 Ariane V and Titan V Boosters Mission Success

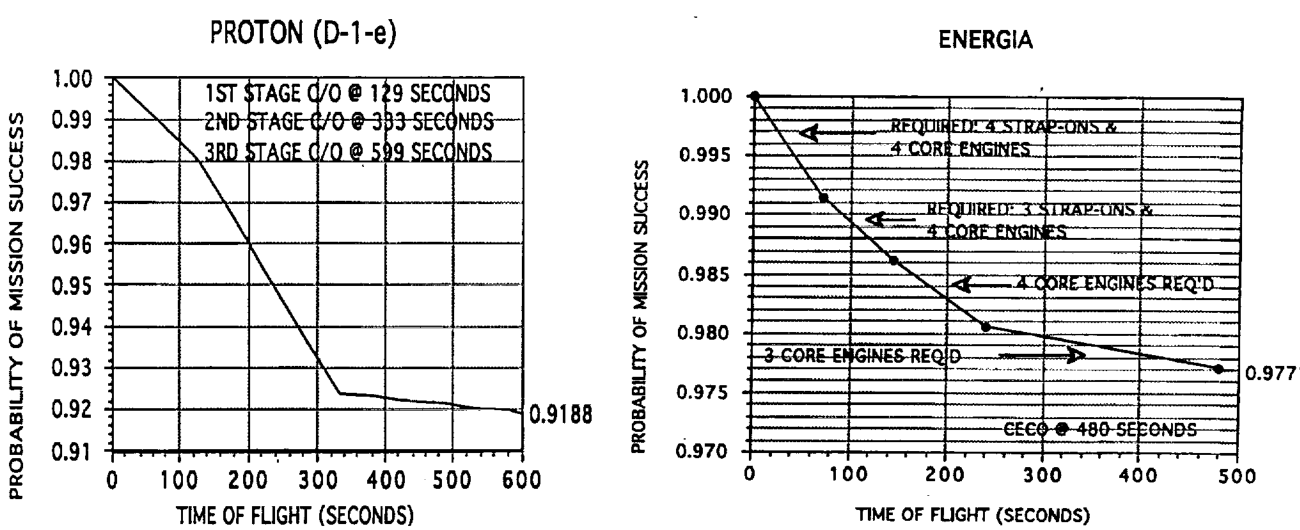


Figure 2.1-92 Russian Boosters Mission Success

2.2 Cargo Transfer & Return Vehicle Studies

The Cargo Transfer and Return Vehicle Concept, referred to as the CTRV, is a system for performing the mission of cargo delivery and return to low Earth orbit (LEO). As originally envisioned, this system would operate in conjunction with a crew delivery/return system (such as the PLS). The CTRV concept, together with a PLS concept and an appropriate launch vehicle, form the architectural framework of the Access to Space, Option 2 study. Identifying the specific configurations of the CTRV, PLS, and launch vehicles which best satisfied the Access to Space mission requirements was the principle objective of the study.

The key CTRV mission requirement is delivery and return of Space Station Freedom logistics elements. These logistics elements include both pressurized and unpressurized payloads which are normally carried in the Space Shuttle payload bay. The logistics elements are therefor standardized around a 15 foot diameter cylindrical volume. The payloads within these logistics elements are considerably smaller, but the key payload is a Space Station logistics rack. These racks are the largest pressurized payloads and constitute by far the greatest number of annual payload deliveries (and returns) for the Space Station. Annual delivery and return requirements of combinations of Pressurized Logistics Modules (PLMs), Unpressurized Logistics Carriers (ULCs), and Propulsion Modules (PMs) drove the CTRV payload volume and mass capabilities.

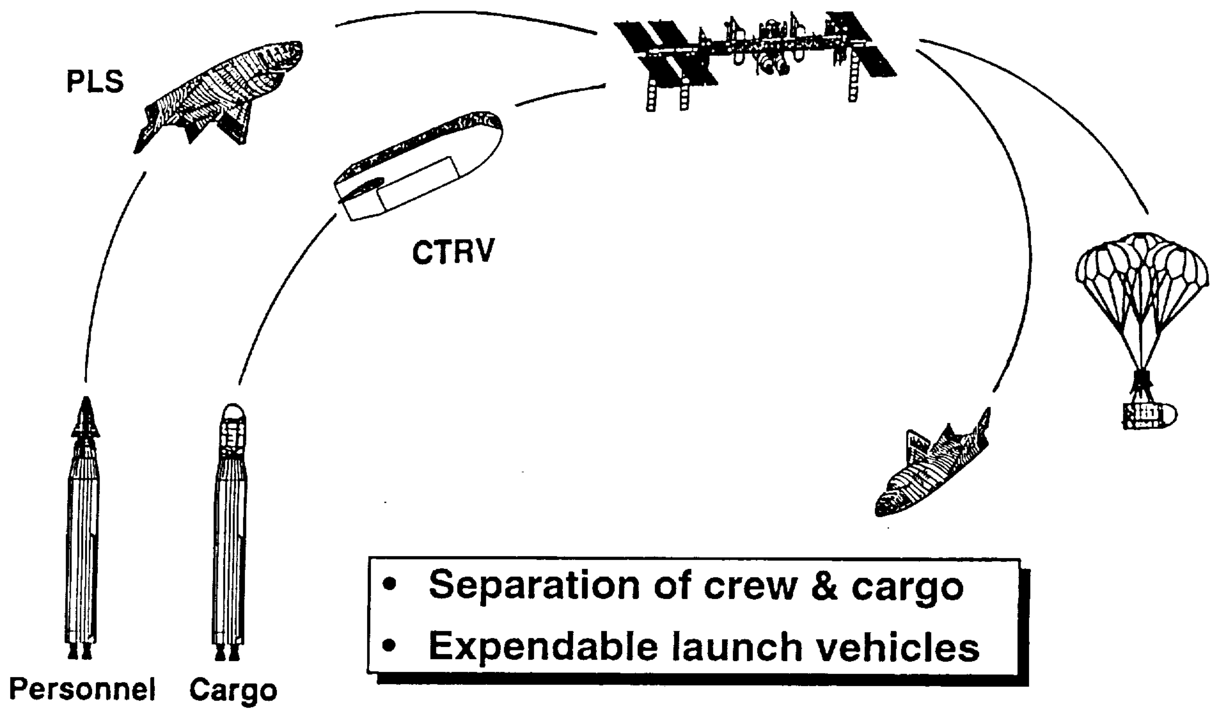


Figure 2.2-1 The CTRV Concept

CTRV concepts have been examined over the past several years by NASA MSFC and these concepts formed the starting point of current CTRV studies. The initial CTRV designs were based on ballistic or lifting body configurations with a parachute landing system. The reference configuration (or starting point) for the CTRV studies was the Medium CTRV concept. The Medium CTRV definition and initial configuration was provided by NASA MSFC and General Dynamics Corp., who had jointly developed the CTRV concept in earlier studies. Several versions of this basic approach were developed for varying payload sizes. The smallest of these was the PLS Caboose, which carried eight logistics racks and was launched concurrently with the PLS. A preliminary Integral CTRV concept from General Dynamics was also developed further. This concept attempted to reduce the payload packaging factor by replacing the Space Station logistics carriers and carrying only the logistics payloads themselves (racks, lockers, etc.). Later versions of this Integral CTRV concept included large fins which improved aerodynamic stability and which also provided subsystem installation volume. Precision (runway) landing of the CTRV concepts was recognized as a key requirement for minimizing operations costs. This requirement led to the Winged CTRV concepts, which started with small payload capabilities (22,000 lbs) and evolved to larger payloads and combined crew/cargo delivery capabilities.

Design and analysis activities for the various CTRV configurations identified several key design issues for these systems. Payload volume and mass requirements associated with the Space Station logistics elements (which were designed for the Space Shuttle payload bay) placed the greatest constraint on the overall CTRV configurations. Large payload bay doors are required to transfer these payloads once at the Space Station. Such large payload bays and doors limit aerodynamic and structural designs of the CTRV concepts. Aerodynamic stability and heating constraints during re-entry imposed design constraints which led to large aerodynamic control surfaces and limited internal layouts (payload and subsystems CG). The need to develop individual vehicles for the crew and the cargo missions (even if launched by the same booster) was a major cost factor which led to the combined crew/cargo CTRV configurations.

The selection process which determined the path taken from initial CTRV concepts to the final design configurations was derived from the CTRV design requirements and design issues. The mission requirements did not change during the course of the study, but the issues were uncovered only by the design process itself. The selection process thus followed a trail illuminated by the design issues. Initial CTRV concept selections were made on the basis of payload capabilities and launch weight. Later concept selections were based on aerodynamic characteristics and development/operations costs. The resulting final CTRV configuration selected was one which provided optimal payload and launch weights and best satisfied all design and cost issues. The final concept selected in the NASA Access to Space study was the HL-42, a scaled up version of the PLS concept with both crew and cargo delivery capability. The HL-42 performs the same cargo delivery and return functions as the CTRV concepts examined in this study, although with a reduced cargo volume and mass.

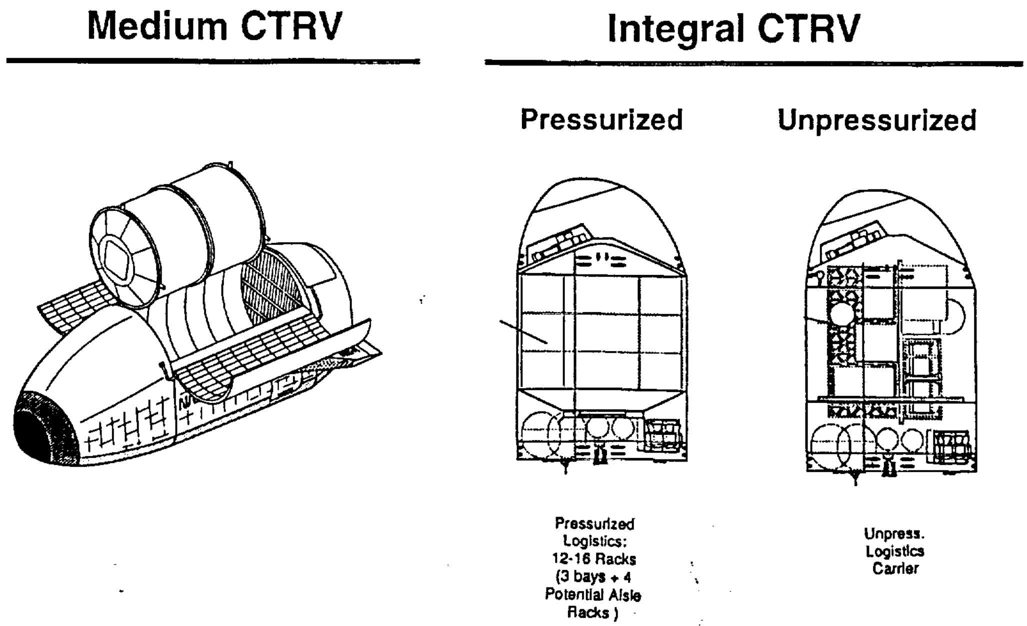
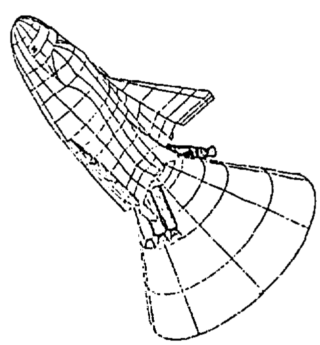


Figure 2.2-2 Initial CTRV Reference Concepts

2.2.1 PLS Caboose Concept

A concept which has not been featured in recent man/cargo launch system architectures is a cargo transfer and return vehicle which is an integral part of the PLS. This concept is similar to the Resource Module for the Hermes space plane, but has a cargo delivery mission of its own, independent of the manned capsule. The genesis of this concept, called the PLS Caboose, was the excess lift capability of the NLS boosters for the PLS delivery mission. Rather than re-size the booster downward from the NLS-2 reference, it was decided to determine how much of the excess lift capability could be converted into useful payload through a small cargo delivery and return vehicle. The goal for a cargo vehicle design was to utilize as much of the structure and functional capabilities which already existed in the adjacent PLS and the PLS adapter/escape system. The PLS adapter and escape system are normally carried all the way to the booster MECO conditions and then expended. At that point they are of no further use for the PLS but they can provide useful propulsion and structural elements for the cargo vehicle.

PLS Caboose


a Cargo Return Vehicle for PLS

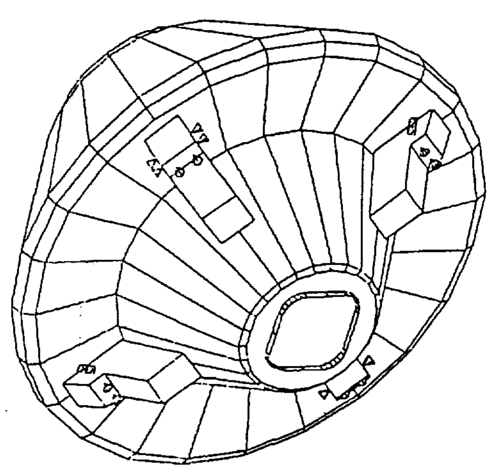


Figure 2.2-3 The PLS Caboose Concept

2.2.1.1 PLS Caboose Design

The PLS caboose was designed to deliver additional pressurized cargo on the PLS launch, which essentially extended the pressurized cargo capability of the PLS system it flies with. A pressurized volume and payload mass capability equivalent to eight Space Station standard racks was selected as the design point. The PLS caboose structural design was built around the PLS launch vehicle adapter. The adapter was utilized as an aeroshell for the caboose and an integrated load path for both PLS launch loads and the cargo payload was developed. Due to the adapter geometry limitations for the launch phase, a simple ballistic shape was selected for the PLS caboose. A Quality Functional Deployment (QFD) approach was used to guide the design process. The resulting design was highly modular and utilized simple designs, shapes and materials, especially in the high cost thermal protection system (TPS).

A hot structure approach to TPS was used for this relatively small sized re-entry system. Hot structure concepts are generally heavier than Shuttle-like tile systems, but are much cheaper. The caboose TPS concept represented an attempt to see just how much weight impact a non-tile system would cause for a small re-entry vehicle.

Propulsion systems (attitude control) were highly modular for the caboose to permit off-line ground processing of these systems (or even expendable units). The PLS escape motors were used for all orbit transfer propulsion maneuvers. The use of modular propulsion system (attitude control system) elements, coupled with the ability to utilize the caboose and PLS thruster firings for translation maneuvers, resulted in a reduction in the number of thrusters required to provide Fail Op/Fail Safe attitude control. A total of 24 thrusters was required on the caboose to perform all translation/rotation maneuvers with two failures anywhere in the system. An additional 14 thrusters would have been required to perform the required orbital maneuvers with the same level of redundancy without this PLS aided approach.

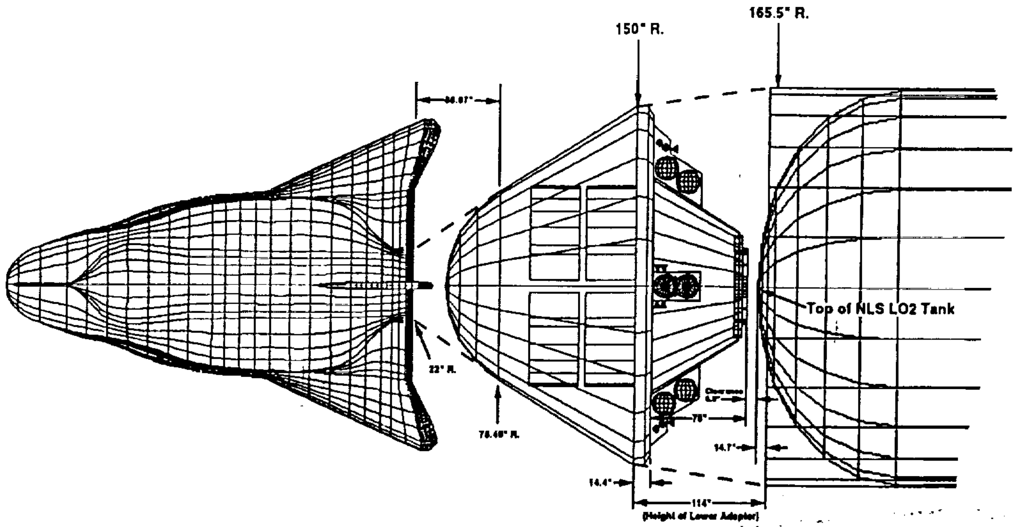


Figure 2.2-4 PLS/Caboose/NLS-2 Configuration

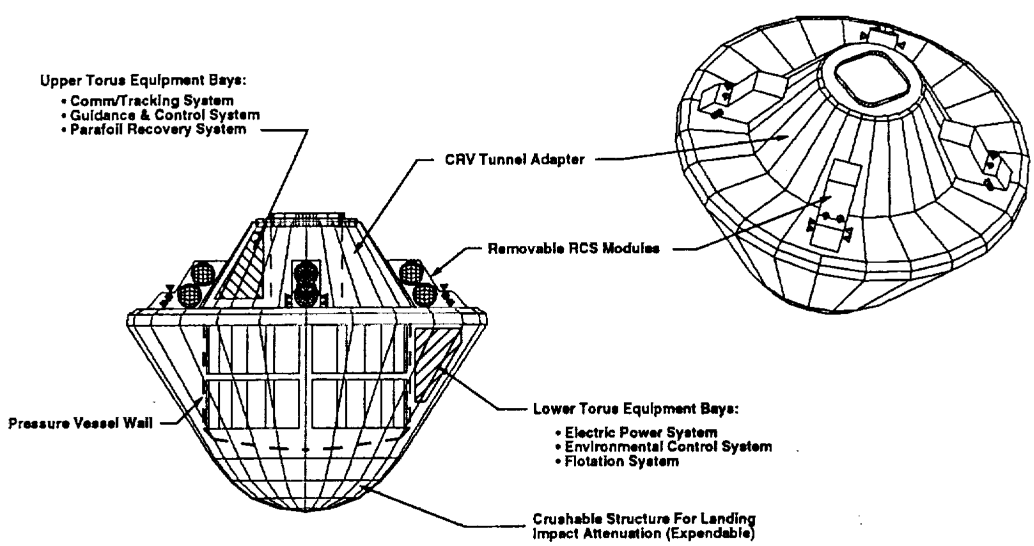


Figure 2.2-5 PLS Caboose Modular Subsystem

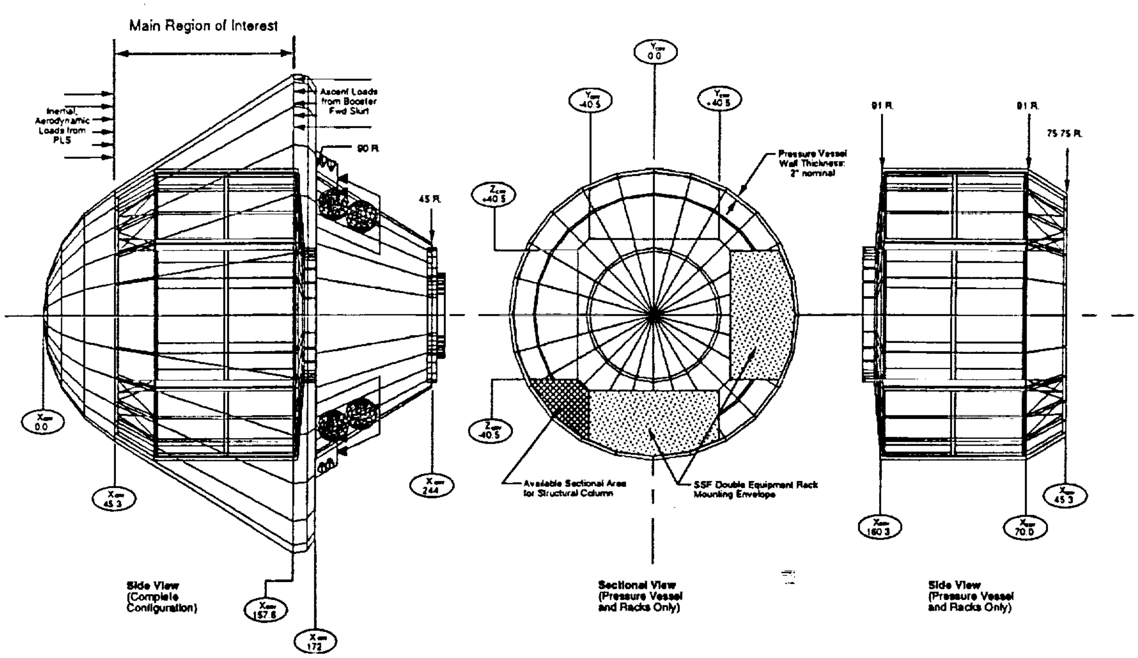


Figure 2.2-6 PLS Caboose Payload & Structural Approach

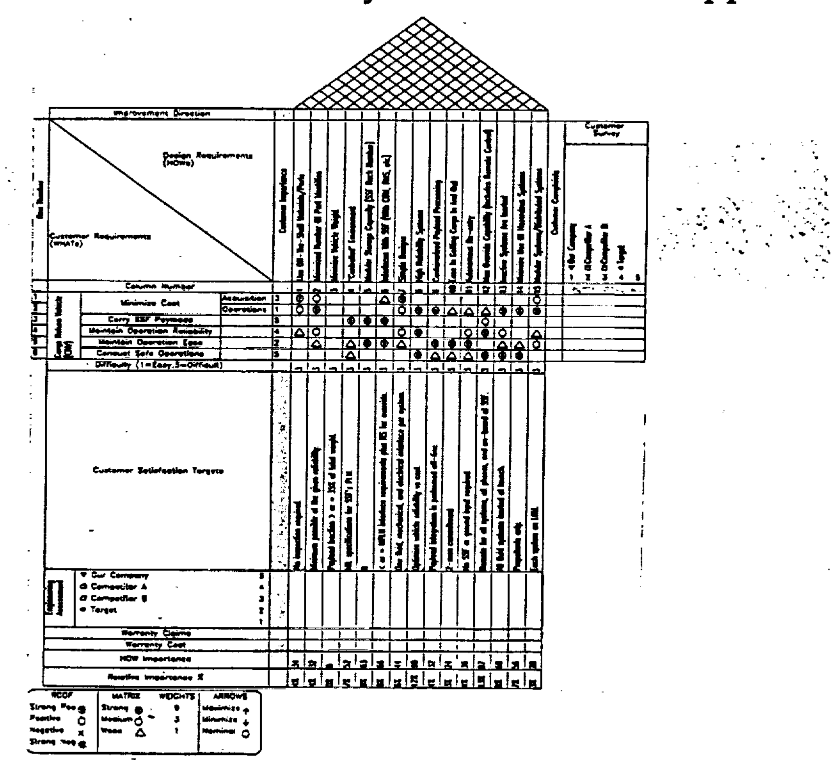


Figure 2.2-7 QFD Process for PLS Caboose

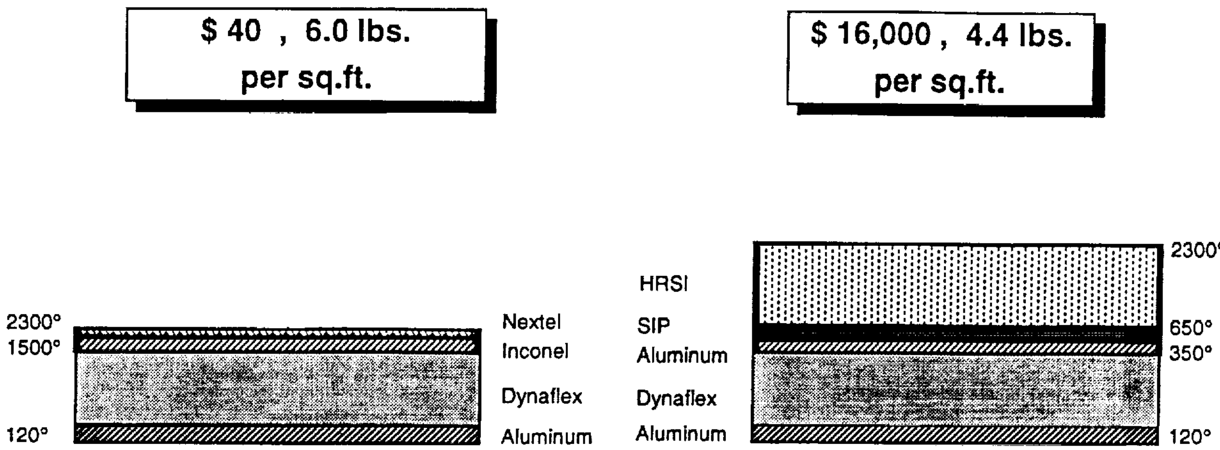


Figure 2.2-8 TPS Hot Structure vs. Thermal Tiles

2.2.1.2 PLS Caboose Analysis

Trajectory analysis of the caboose concept showed that the re-entry loads (3.5g deceleration) and the peak heating rates (40 BTU/ft^2-sec) are well within the desired range for the selected TPS and structures concepts. Cross range capabilities for this ballistic shape are very limited (less than 150 nmi.). The hot structure approach resulted in a TPS/structure combined weight of 6.0 lbs/square foot, 1.36 times the weight of an equivalent tile/aluminum structure approach. The projected cost savings of this approach, however, is dramatic. The hot structure TPS fabrication and installation cost is estimated to be \$40 $/ft^2$. This is significantly lower than the \$16,000 $/ft^2$ for tile installation costs currently being experienced on the Space Shuttle. Some amount of weight reduction could be achieved for an expendable structure by the use of ablator materials which would directly reduce the required structural thickness. The combination of limited cross range and an ablative type thermal protection system make this design approach a particularly good solution for water landing type systems.

A finite element structural analysis model of the caboose was prepared to analyze the structural loads in the outer aeroshell and in the pressure vessel. Both launch loads were calculated for the launch phase (integrated with a NLS-2 booster beneath and the PLS (HL-20) above the caboose) and re-entry phase. The analysis showed good isolation of the pressure vessel from the launch loads (aerodynamic bending loads at max Q) and good load distributions during the high deceleration re-entry. This model was used to calculate the caboose structural weight estimates.

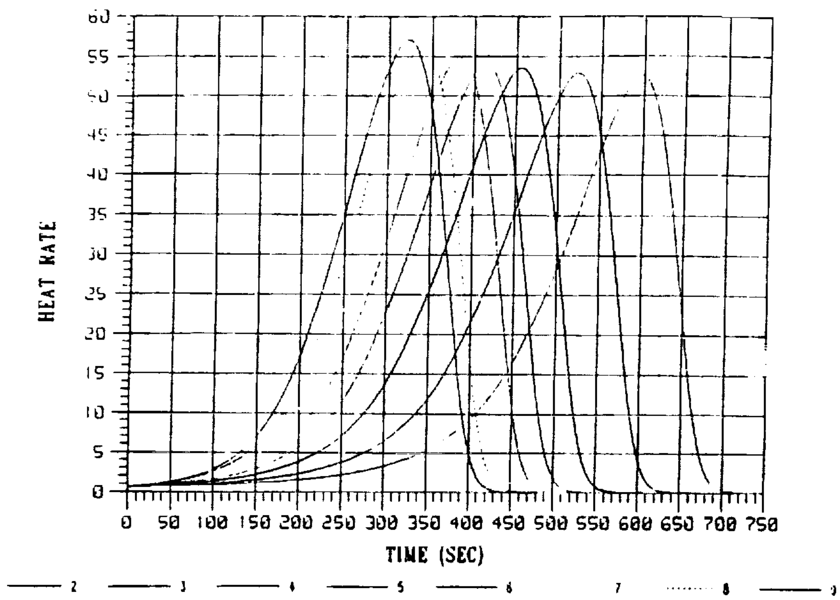


Figure 2.2-9 PLS Caboose Re-entry Heating Rates

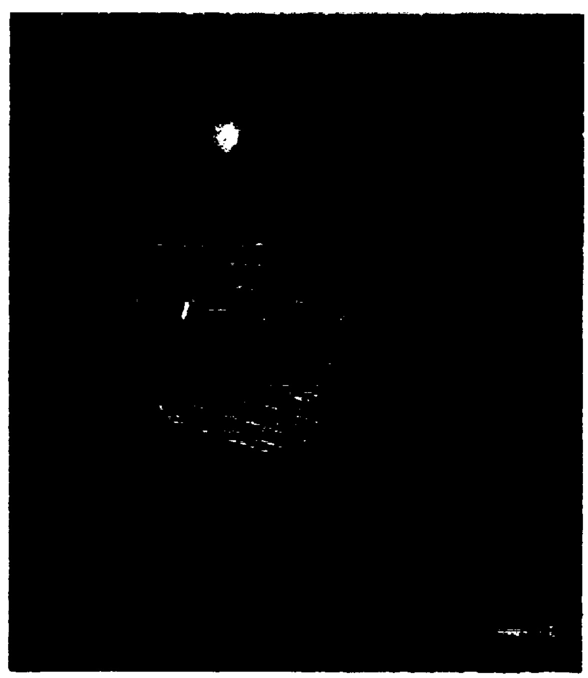


Figure 2.2-10 Caboose & NLS-2 Booster Finite-Element Model

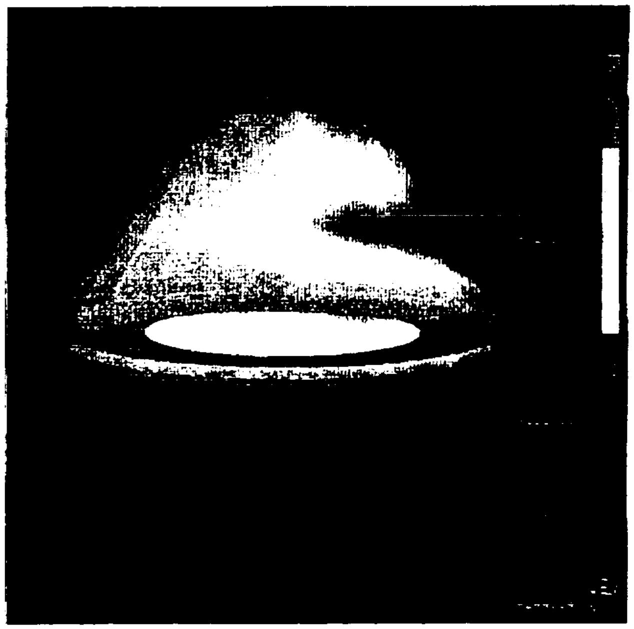


Figure 2.2-11 Aeroshell Stresses at Max Q

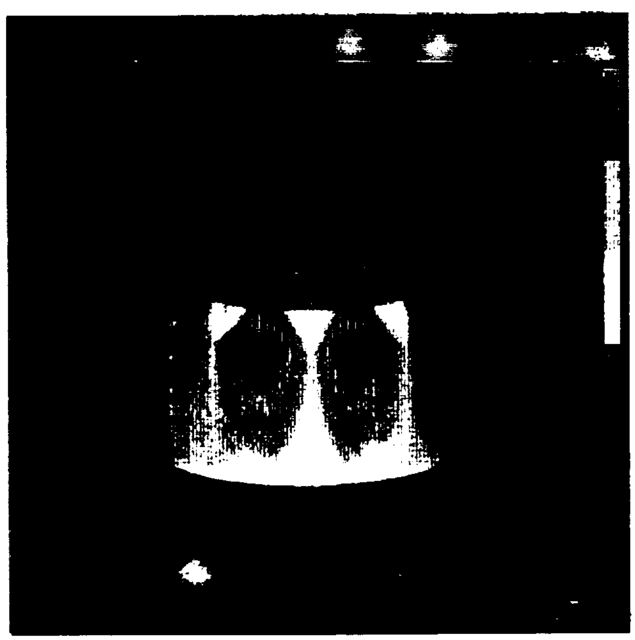


Figure 2.2-12 Pressure Vessel Stresses at Max Q

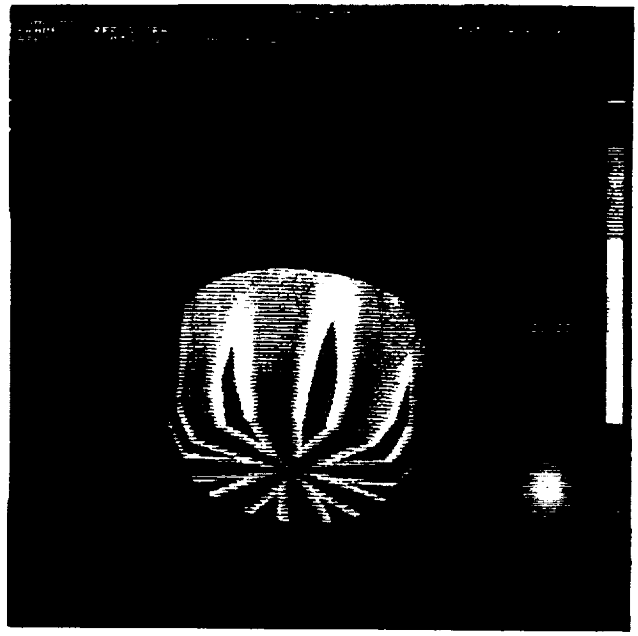


Figure 2.2-13 Pressure Vessel Stresses During Re-entry

Figure 2.2-14 Aeroshell Stresses During Re-entry

Structure	3730	calculated from structures analysis
TPS	1039	calculated from TPS analysis
TCS	400	.5 x Integ CTRV
OMS/RCS	1080	not including PLS abort motors (6200 lb)
EPG	1200	.66 X Med CTRV
EPD	1100	.66 X Med CTRV
Avionics	1065	same as Med CTRV
Env Control	150	.25 X Med CTRV
Landing	2800	.50 X Integ CTRV
	12,564	*Dry Weight*
Consumables		
OMS/RCS	724	calculated from delta-V analysis
FCP	850	.50 X Med CTRV
Payload	8000	
Total Launch Weight	$\boxed{22,138}$	

Figure 2.2-15 PLS Caboose Weight Estimate

2.2.2 Medium CTRV Concept

A detailed design study of the Medium CTRV concept was performed to obtain a better understanding of the CTRV design requirements and to improve the design definition for this concept. The design effort included trajectory analysis, thermal analysis, detailed structural design, and parametric analysis of the CTRV landing system options (parachutes) .

2.2.2.1 Requirements Analysis

A detailed set of design requirements was prepared for each CTRV subsystem and for the system level design. The requirements were compiled into a Requirements Definition Document (RDD) for each subsystem and one RDD for the complete CTRV system. These documents are structured such that the design requirements and the design solution (design definition and performance capabilities) can be documented in a single volume for future reference. Later definition of the system (and subsystem) verification plans can also be included in these same volumes. A total of 13 RDDs were prepared which included all subsystems except the Avionics subsystems. Included in each of these documents is the subsystem's definition, functional and design requirements, and subsystem interfaces with the Space Station, the launch vehicle, and with the CTRV payloads. The system level requirements documents are similar to the subsystem RDDs but include mission, payload, and ground operations requirements. These CTRV requirements documents were used to compile the list of impacts to Space Station Freedom for a CTRV-based logistics delivery system rather than the current Shuttle-based system.

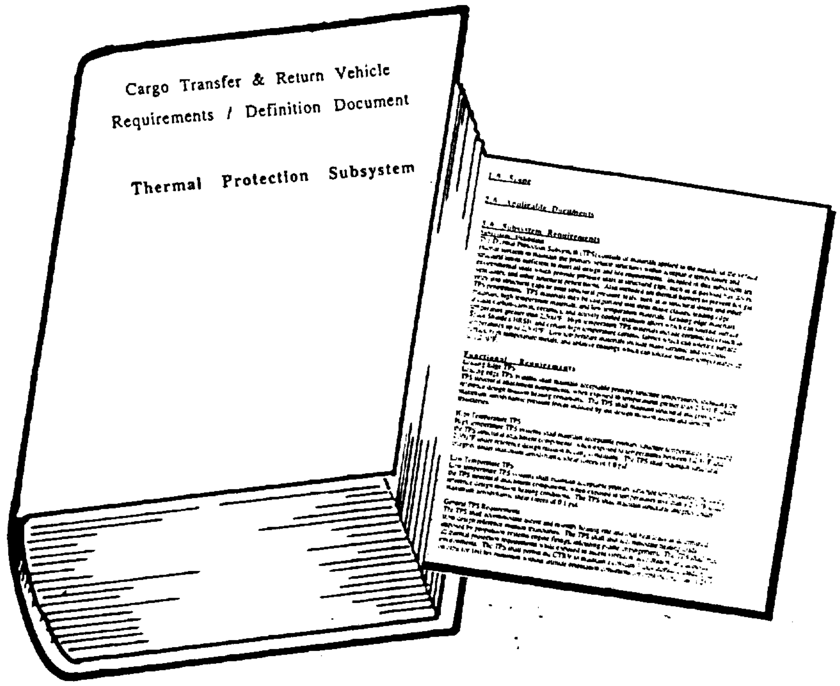


Figure 2.2-16 CTRV Requirements Document (TPS)

2.2.2.2 Structures Design

Detailed structural design of the Medium CTRV was performed to obtain better weight estimates of this concept and to also determine dynamic responses of this structure approach to flight loading conditions. Analysis of load paths to carry flight loads through the structure has resulted in some changes to the baseline structural concepts and in more detailed structural definition. Relocation of several structural beams was required in the forward and aft fuselage sections to redirect loads away from the upper portion of the payload bay bulkheads (which cannot transfer axial loads through the payload bay doors). Two keel beams were added to the lower structural areas to accommodate flight bending loads. Definition of structural beams and reinforcements in the aft fuselage section was required to accommodate launch loads, propulsion, and docking loads. CAD drawings of the design and layout were utilized for the current structural definition and stress analysis models were developed from the CAD system drawings for use in structural analysis.

Structural design layout drawings were prepared for the Medium CTRV (C23LNF configuration, which is the CTRV configuration baseline for our structural analysis). The layout drawings included dimensioned three view layouts of the exterior surfaces and of primary internal structural members (frames, longerons, etc.) Full color perspective drawings of the CTRV were also provided in several orientations, showing the concept with payload bay doors open and closed and with payloads installed.

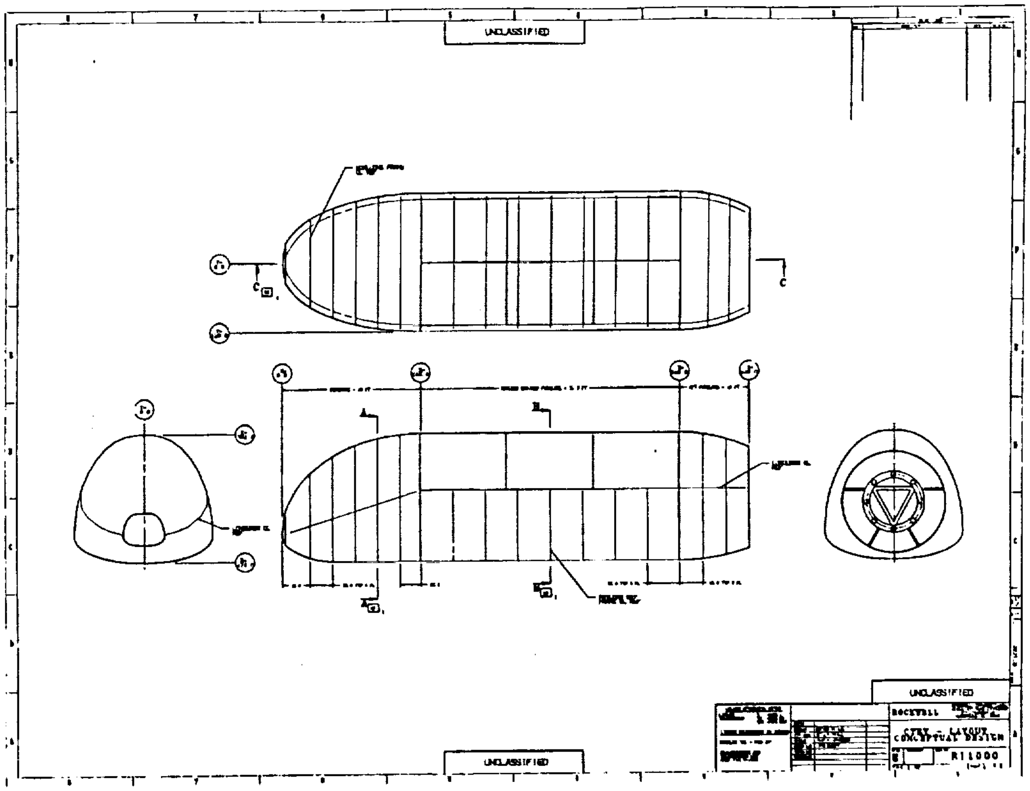


Figure 2.2-17 Medium CTRV Design Layout Drawing (1of2)

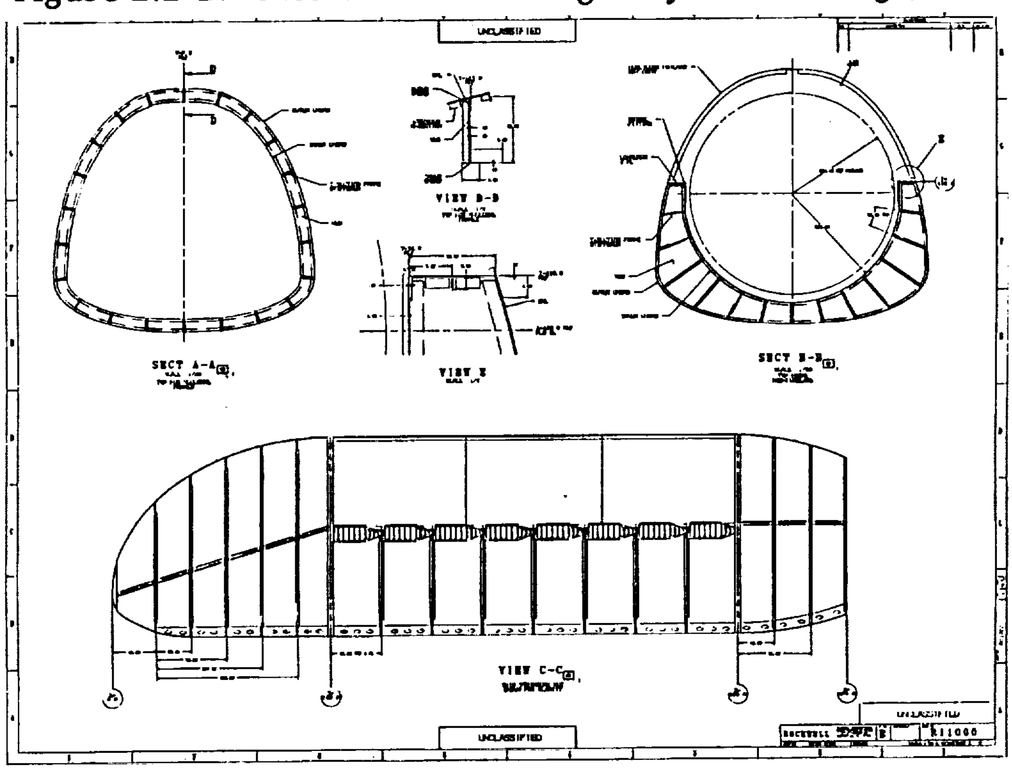


Figure 2.2-18 Medium CTRV Design Layout Drawing (2of2)

Structural Element	Quantity	Material	Surface Area	Thickness	Density	Weight (unit)	X Station	Weight (Total)	Wt. Factor	Weight Est.	Moment	
units	each	Type	sq. inches	inch	Lb/in3	Lb	inch	G x B (Lb)			H x I (in-lb)	
Forward Fuselage												
Skin	1	Al 2219	158185	0.05	0.102	807	140	807	1.17	943.89	132144.585	
Stringers		Al 2219			0.102	0		0		0.00	0	
Frames										0.00	0	
Xo 48	1	Al 2219	5916 +	0.125	0.102	196	48	196	1.17	229.32	11007.36	
Xo 87	1	Al 2219	7332 +	0.125	0.102	242	87	242	1.17	283.14	24633.18	
Xo 126	1	Al 2219	8104 +	0.125	0.102	269	126	269	1.17	314.73	39655.98	
Xo 165	1	Al 2219	8748 +	0.125	0.102	287	165	287	1.17	335.79	55405.35	
Xo 204	1	Al 2219	9036 +	0.125	0.102	297	204	297	1.17	347.49	70887.96	
Sill Beams	2	Al 2219	2880 +	0.125	0.102	93	140	186	1.17	217.62	30466.8	
Keel Beams	2	Al 2219	9820 +	0.125	0.102	126	140	252	1.17	294.84	41277.6	
Fwd. Bulkhead (Xo 240)	1	Al 2219	44129	0.2	0.102	900	240	900	1.38	1242.32	298156.706	
Nose Bulkhead (Xo 6)	1	Inco 718		0.2	0.297	168	10	168	1.17	196.56	1965.6	
										0.00	0	
Mid Fuselage										0.00		
Skin	1	Al 2219	203840	0.05	0.102	1019	464	1019	1.38	1406.22	652486.08	
Stringers		Al 2219			0.102	0		0		0.00	0	
Frames										0.00		
Xo 296	1	Al 2219	13454		0.102	313	296	313	1.17	366.21	108398.16	
Xo 352	1	Al 2219	13454		0.102	313	352	313	1.17	366.21	128905.92	
Xo 408	1	Al 2219	13454		0.102	313	408	313	1.17	366.21	149413.68	
Xo 464	1	Al 2219	13454		0.102	313	464	313	1.17	366.21	169921.44	
Xo 520	1	Al 2219	13454		0.102	313	520	313	1.17	366.21	190429.2	
Xo 576	1	Al 2219	13454		0.102	313	576	313	1.17	366.21	210936.96	
Xo 632	1	Al 2219	13454		0.102	313	632	313	1.17	366.21	231444.72	
Longerons	2	Al 2219	29230	0.1	0.102	298	464	596	1.17	697.66	323715.001	
Keel Beams	2	Al 2219	22390	0.1	0.102	245	464	490	1.17	573.30	266011.2	
										0.00		
Aft Fuselage										0.00		
Skin	1	Al 2219	85743	0.05	0.102	437	742	437	1.38	603.46	447766.752	
Stringers		Al 2219			0.102	0		0		0.00	0	
Frames										0.00		
Xo 728	1	Al 2219	8760 +	0.125	0.102	245	728	245	1.17	286.65	208681.2	
Xo 768	1	Al 2219	7992 +	0.125	0.102	403	768	403	1.17	471.51	362119.68	
Aft Bulkhead (Xo 688)	1	Al 2219	44129	0.2	0.102	900	688	900	1.38	1242.32	854715.89	
Sill Beams	2	Al 2219	1494 +	0.125	0.102	47	742	94	1.17	109.98	81605.16	
Keel Beams	2	Al 2219	4250 +	0.125	0.102	65	742	130	1.17	152.10	112858.2	
Thrust ring	1	Ti 6AL-4V	3619 +	.125 / .25	0.16	175	807	175	1.17	204.75	165233.25	
Bathtub fittings	4	Al 2219	735 +	0.5	0.102	48	806	192	1.17	224.64	181059.84	
Docking ring	1	Al 2219	33660	0.1	0.102	343	807	343	1.17	401.70	324170.641	
Base Bulkhead (Xo 808)	1	Al 2219	6780 +	0.05	0.102	160	808	160	1.38	220.80	178406.4	
										0.00		
Other Structure										0.00		
Payload bay doors	2	Gr/Ep h-comb	65633	4		1294	464	2588	1.07	2769.16	1284890.24	
Payload bay door hinges	18	Inco 718			0.297	20	464	360	1.07	385.20	178732.8	
Payload bay door latches	26	Al 2219	862	0.8	1	27	464	702	1.07	751.14	348528.96	
Landing system (gear) doors	4	Al 2219	8064	0.5	0.102	403	464	1612	1.17	1886.04	875122.56	X - C.G.
								16242		19356.80	8741155.86	451.603952

Figure 2.2-19 Medium CTRV Weight Estimate

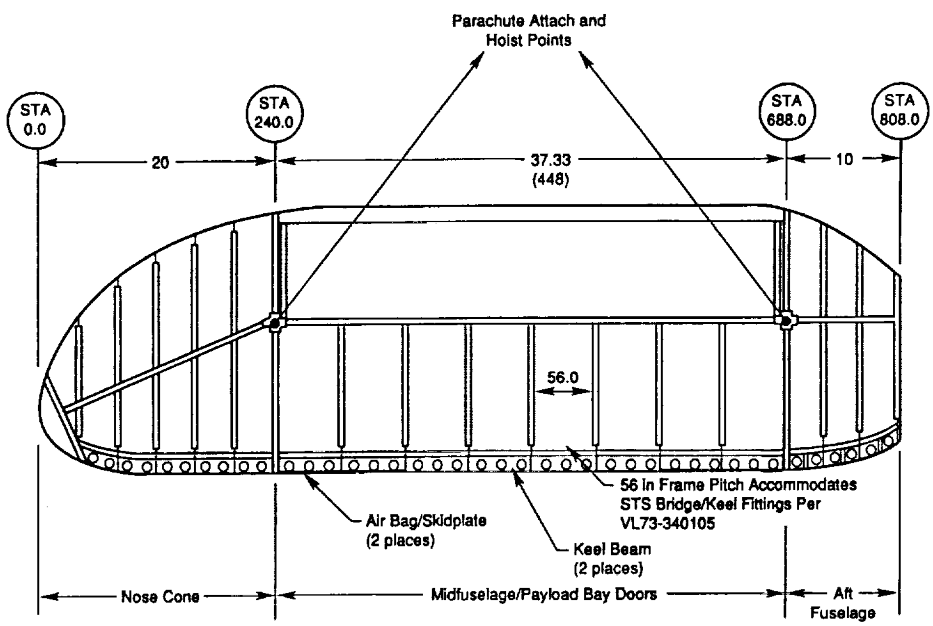


Figure 2.2-20 Medium CTRV Major Structural Elements

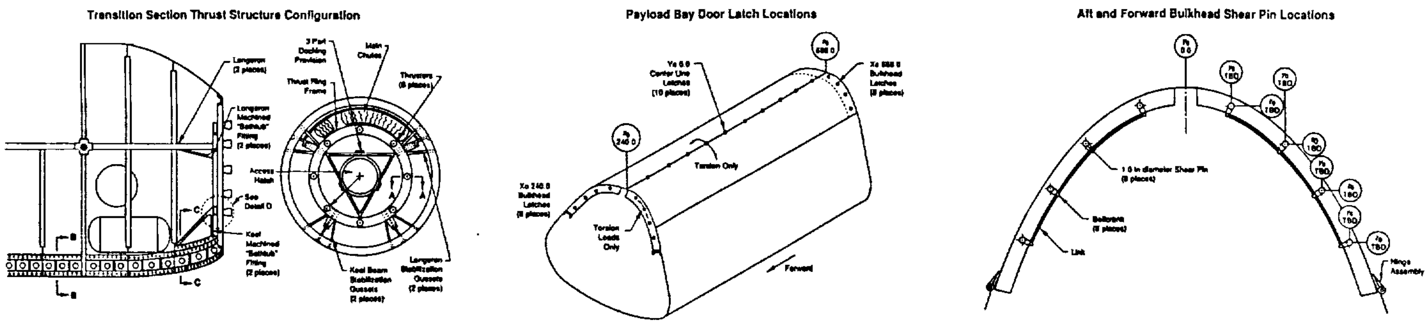


Figure 2.2-21 Medium CTRV Detail Design Layouts (1 of 3)

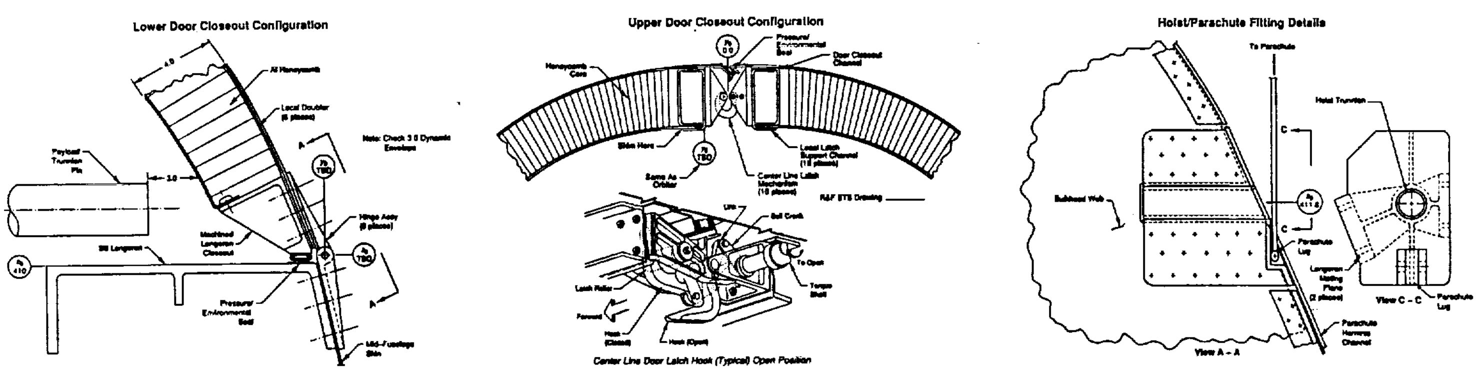


Figure 2.2-22 Medium CTRV Detail Design Layouts (2 of 3)

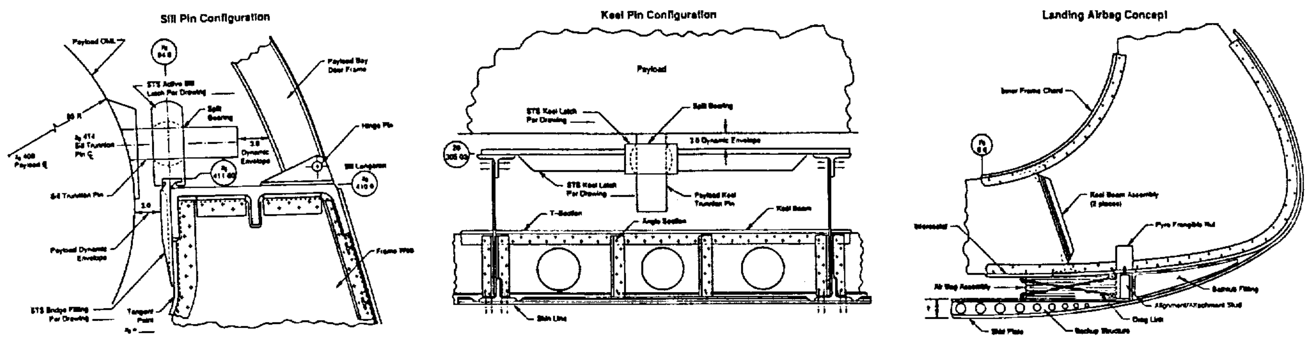


Figure 2.2-23 Medium CTRV Detail Design Layouts (3 of 3)

2.2.2.3 Structural Analysis

A finite element model of the Medium CTRV structure design was completed which includes all structural elements and major subsystems. The major subsystems were included as point masses in the model with the mass supported by the appropriate structural elements (beams, longerons, or frames). These subsystem masses were included in order to estimate the required loads for the primary structural members of the CTRV design. The finite element model was translated into a NASTRAN file for the structural sizing analyses under flight loads conditions. The NASTRAN file provided initial bending mode information for the structure as well as the vehicle moments of inertial about primary body axes. This file also provided a computer calculated weight estimate for the Medium CTRV structure.

The NASTRAN finite element model of the Medium CTRV structural design was used to determine stress levels for several flight load conditions: maximum acceleration (3.2 g's at MECO); maximum structural bending loads (aerodynamic loads at max Q); and landing loads (29 ft/sec vertical landing velocity). This last condition (landing) was run for a "perfect landing" (4-wheel initial contact) and for a nominal landing (2-wheel initial contact with 10 knot forward velocity). The analyses generally showed that the structure is oversized and skin, frame, and longeron thicknesses can be reduced. Most of the flight loads were being reacted by the skin sections rather than the keel beams and frames. The keel beams help transmit loads only during the landing conditions and thus can be greatly reduced or eliminated in most of the mid-fuselage (payload bay) region. The bulkhead structures at the forward and aft ends of the payload bay were found to require some stiffening for the high axial acceleration conditions (MECO and max Q). The payload bay frame structures were also too flexible and will require some bracing for axial loads. These frames did not carry high loads, however, so they could be replaced by a lighter truss structure instead of the solid frame design initially modeled. Color stress contour plots were prepared to illustrate the analysis results for each of the load conditions.

The stress analysis results described above generally indicate that the structural weight estimate for the CTRV can be reduced based on the expected loads. These findings are preliminary, however, as additional loads conditions such as skin panel loads and buckling due to aerodynamic pressure distributions were not examined (especially for the forward and mid fuselage sections). Total vehicle bending modes (buckling analysis) also must be determined before completing any redesign.

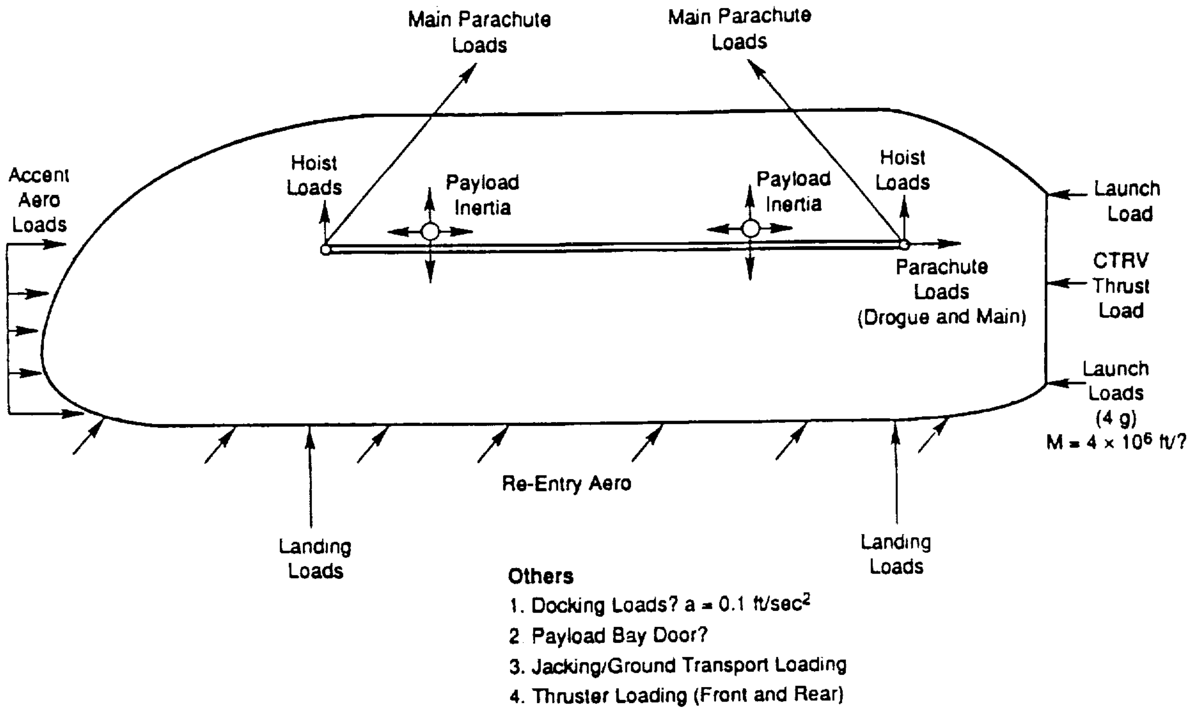


Figure 2.2-24 Medium CTRV Major Loads Conditions

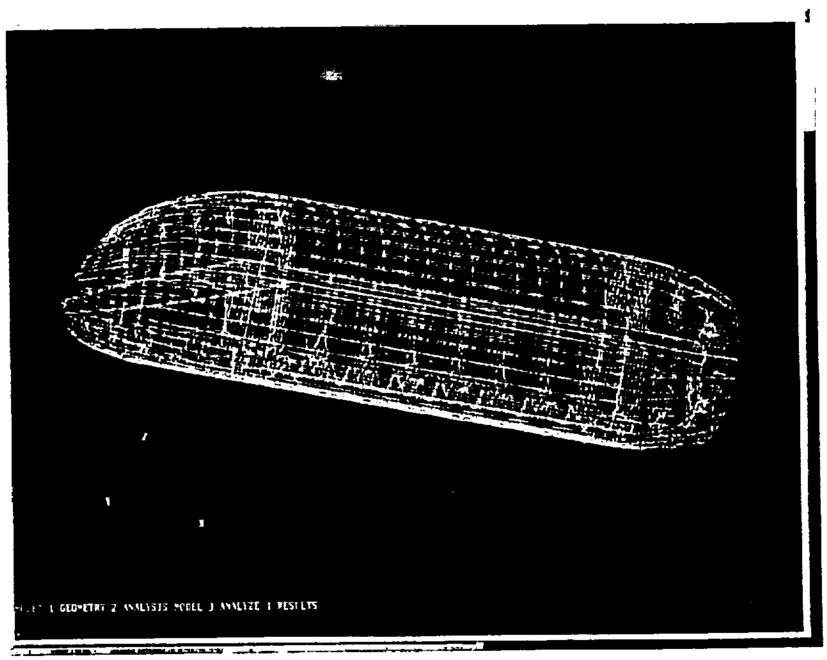


Figure 2.2-25 Medium CTRV Finite-Element Model

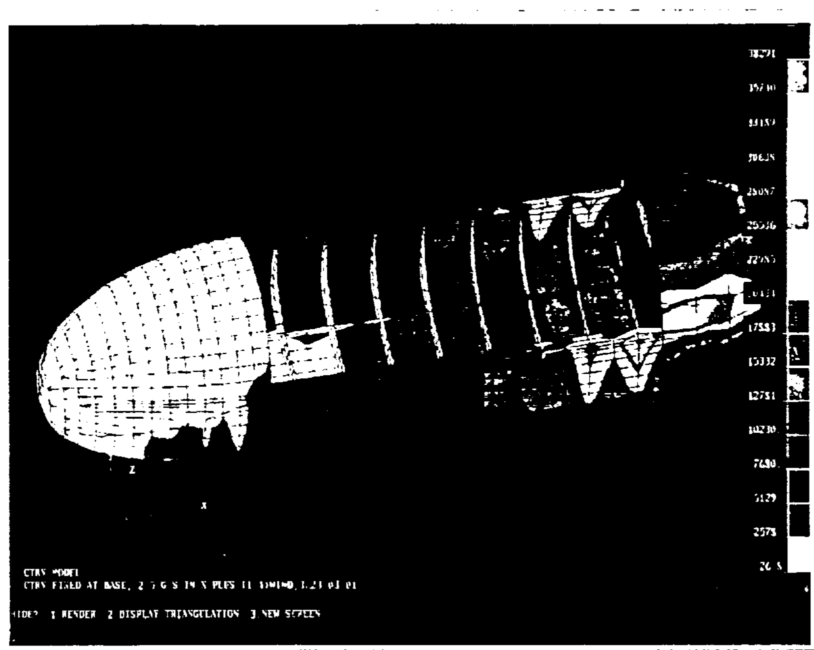


Figure 2.2-26 CTRV Stresses at Max Q

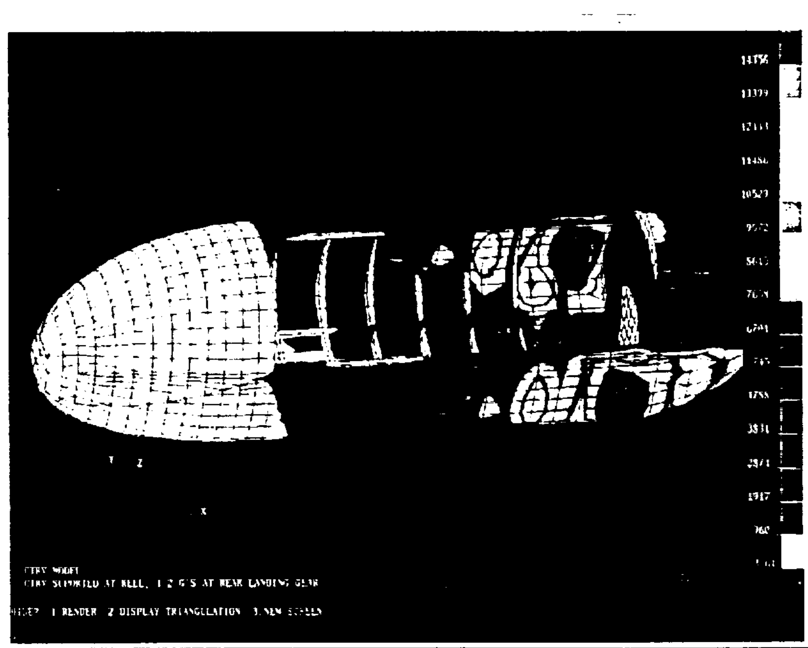


Figure 2.2-27 CTRV Stresses at Landing (Aft Gear Touchdown)

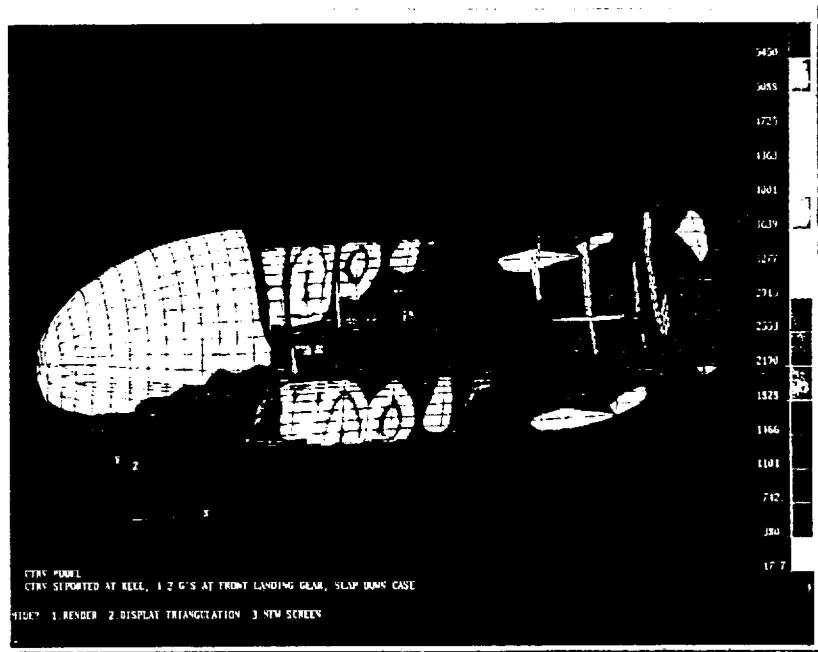


Figure 2.2-28 CTRV Stresses at Landing (Front Gear Slapdown)

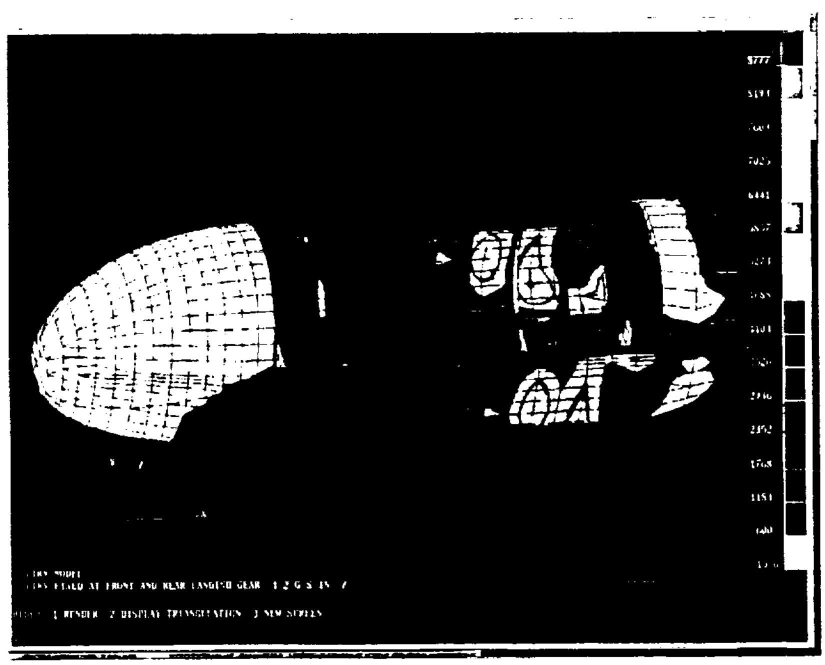


Figure 2.2-29 CTRV Stresses at Landing (4-point Touchdown)

2.2.2.4 Trajectory Analysis

Trajectory analyses of both the C23LNF and the more recent CV75 configurations of the Medium CTRV was performed with the POST trajectory analysis model to compare their cross range and TPS heating differences. The CV75 configuration was selected by NASA as the preferred Medium CTRV configuration in order to achieve sufficient cross range to reach an Edwards AFB landing site. The trajectory analyses demonstrated that the CV75 configuration does in fact provide sufficient cross range to reach Edwards AFB, but the total heat load on the vehicle increased by 70% (re-entry heat rates and accelerations were about the same for the two configurations). This significant increase in heat load will directly affect (increase) the TPS weight for this configuration. Re-entry environmental and performance parameters such as acceleration, heating rates, cross range, and dynamic pressure were obtained for both Medium CTRV shapes. The calculated re-entry trajectory environments for the reference CTRV shape (C23LNF configuration) were used for the subsequent detailed thermal analyses and landing systems design.

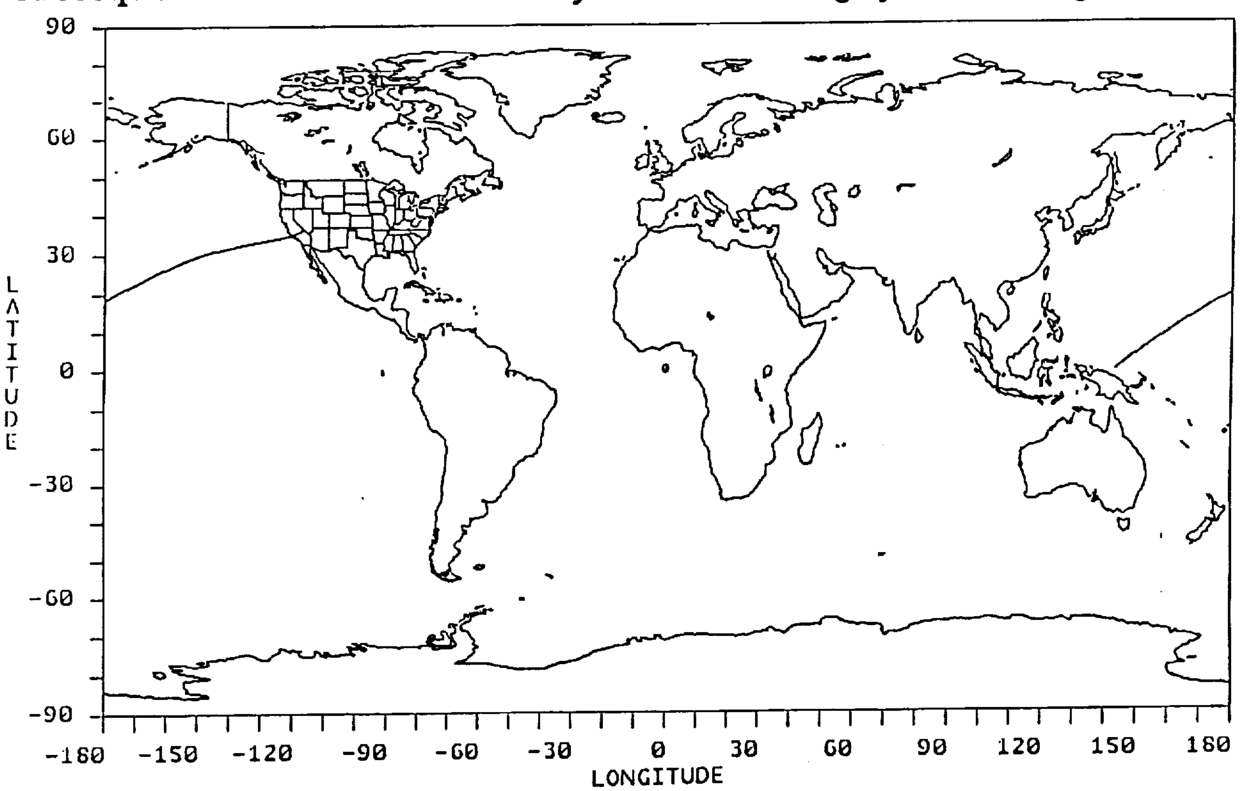


Figure 2.2-30 CTRV Re-entry Trajectory

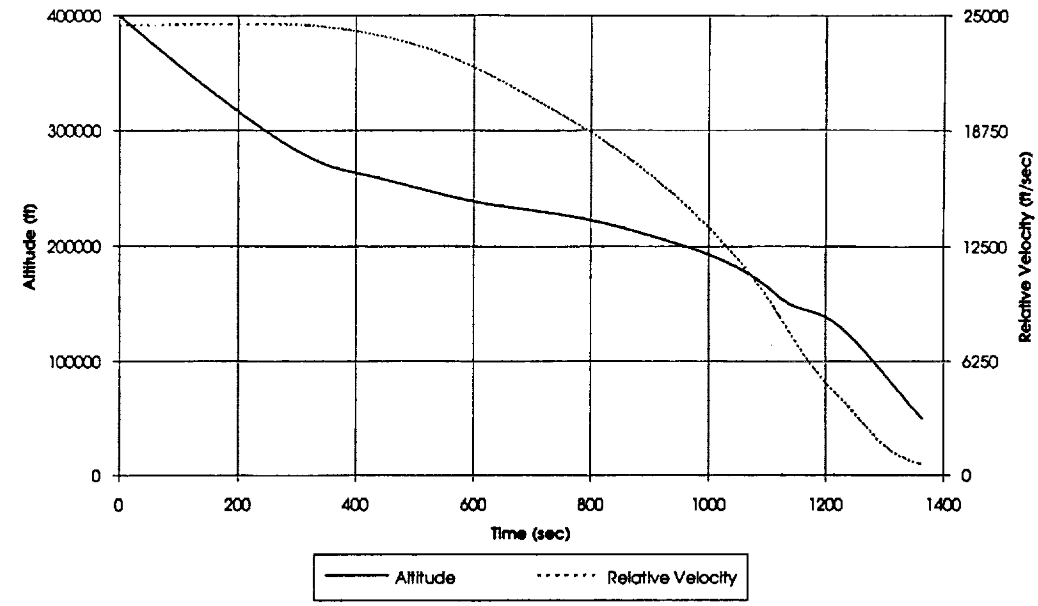


Figure 2.2-31 CTRV Re-entry Profile

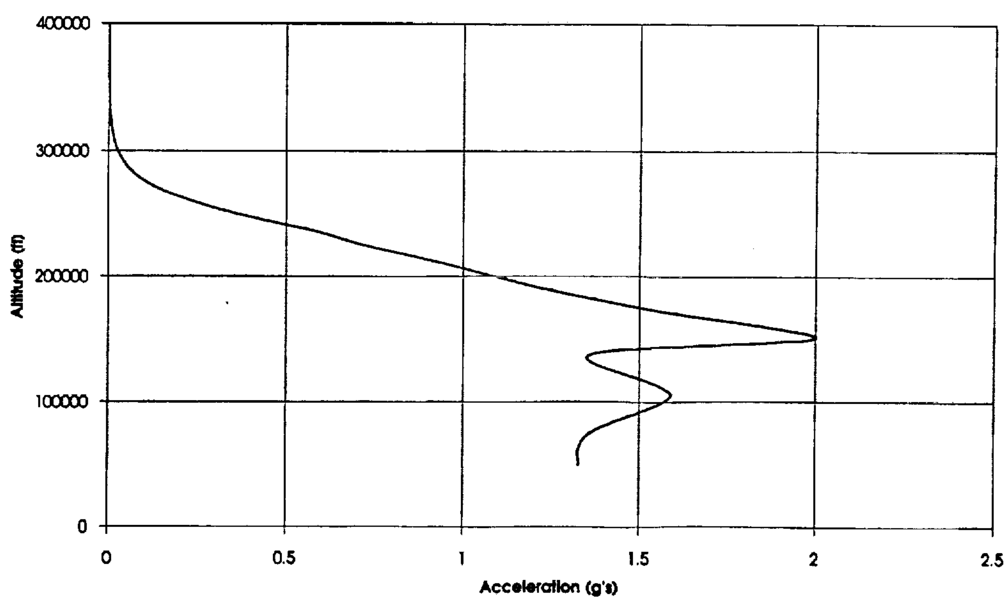


Figure 2.2-32 CTRV Re-entry Acceleration Profile

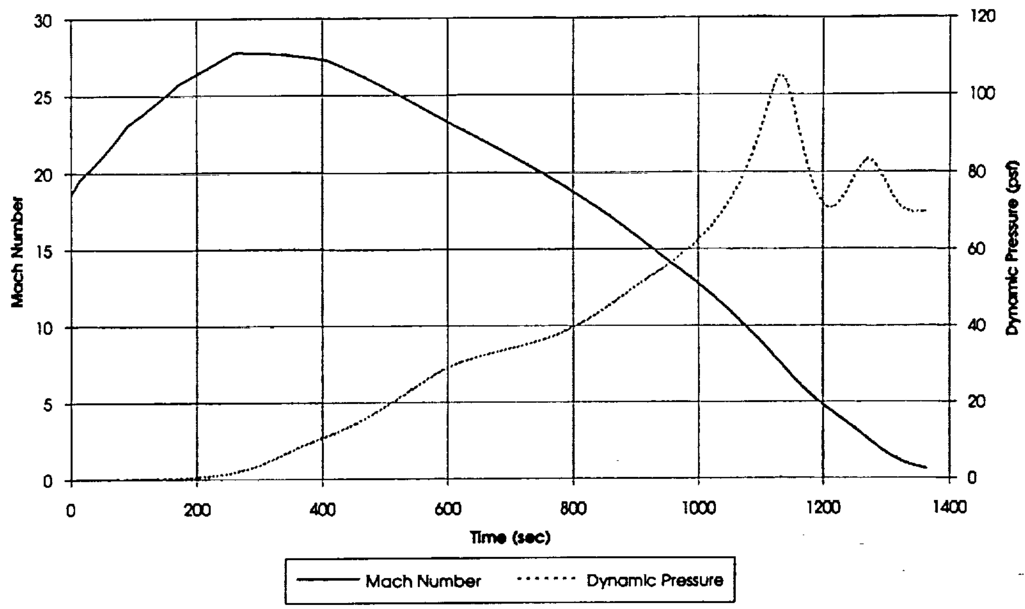


Figure 2.2-33 CTRV Re-entry Environments

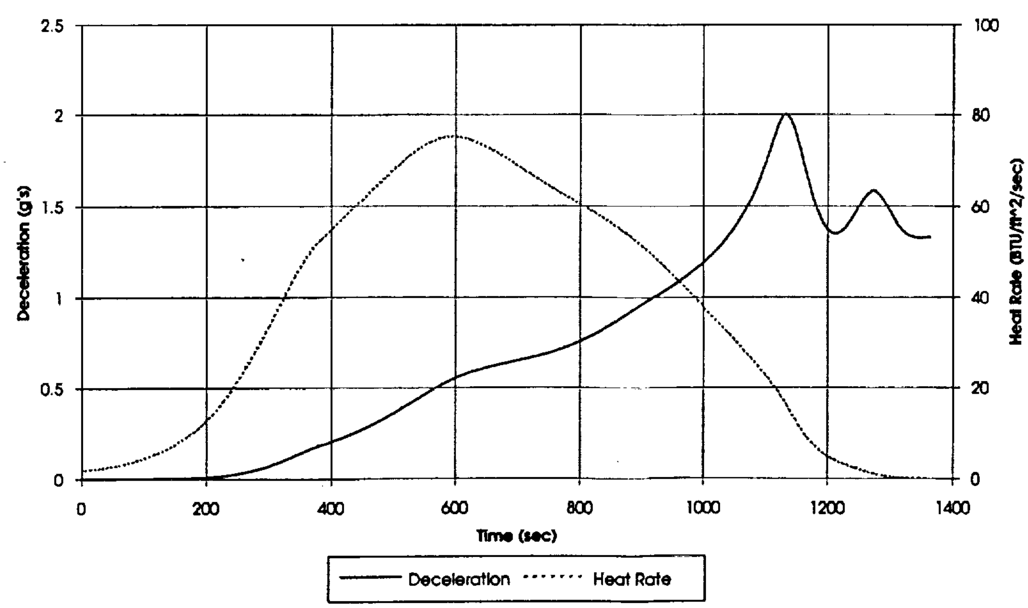


Figure 2.2-34 CTRV Re-entry Environments

2.2.2.5 Thermal Analysis & TPS Sizing

Thermal analysis was performed to determine the initial structure temperatures which would be encountered at the start of re-entry. The TPS/structure was subjected to thermal environments consistent with a long duration Space Station docked attitude (local vertical) and then allowed a 12-hour thermal conditioning period prior to the start of the re-entry mission phase. The analysis identified initial structure temperatures between 120°F and 70°F at the end of the thermal conditioning period. TPS sizing analysis was then performed using these initial structure and tile temperatures. Heating rates for several flow streamlines around the reference shape were computed and used to determine heat loads and TPS thicknesses at selected CTRV body points.

The TPS sizing analysis indicated a TPS weight of 6,415 lbs (excluding the RCC nose cap) for the CTRV. This translates to a net TPS tile weight per square foot of 1.60 lb/ft^2, which is more than the net TPS tile weight of 1.23 lb/ft^2 for the Space Shuttle. The greater average TPS tile thickness than Shuttle is due to the larger percentage of high temperature tiles for the CTRV (60%) vs. the Shuttle (50%). The required TPS thicknesses are actually slightly thinner than would be seen on a Shuttle for equivalent re-entry heat loads. The reason for the thinner tiles is the lower initial temperatures found on the CTRV structure and tiles. The Shuttle tiles were sized for a mission which required re-entry immediately after launch. The CTRV has no similar mission (or any equivalent abort missions) and temperatures are allowed to stabilize to orbital conditions prior to entry. The effect of a 12-hour pre-entry thermal conditioning period is seen in the tile thicknesses also. Structure and tile temperatures for the CTRV HRSI (black tiles) areas needed this thermal conditioning period to lower the temperatures they reached when exposed to direct solar radiation while in space.

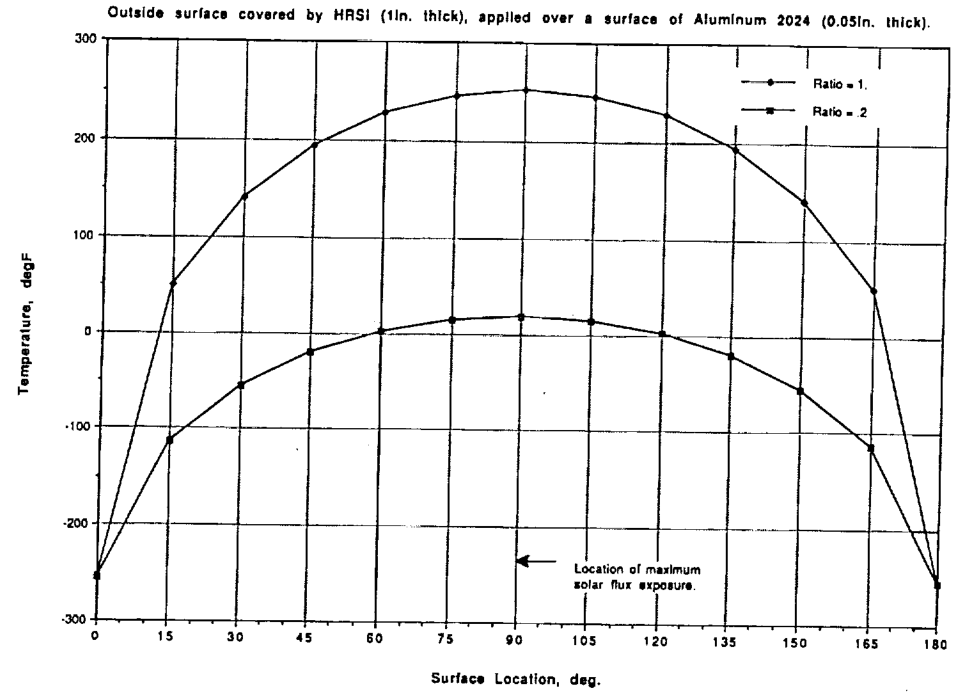


Figure 2.2-35 CTRV On-orbit Temperature Distribution

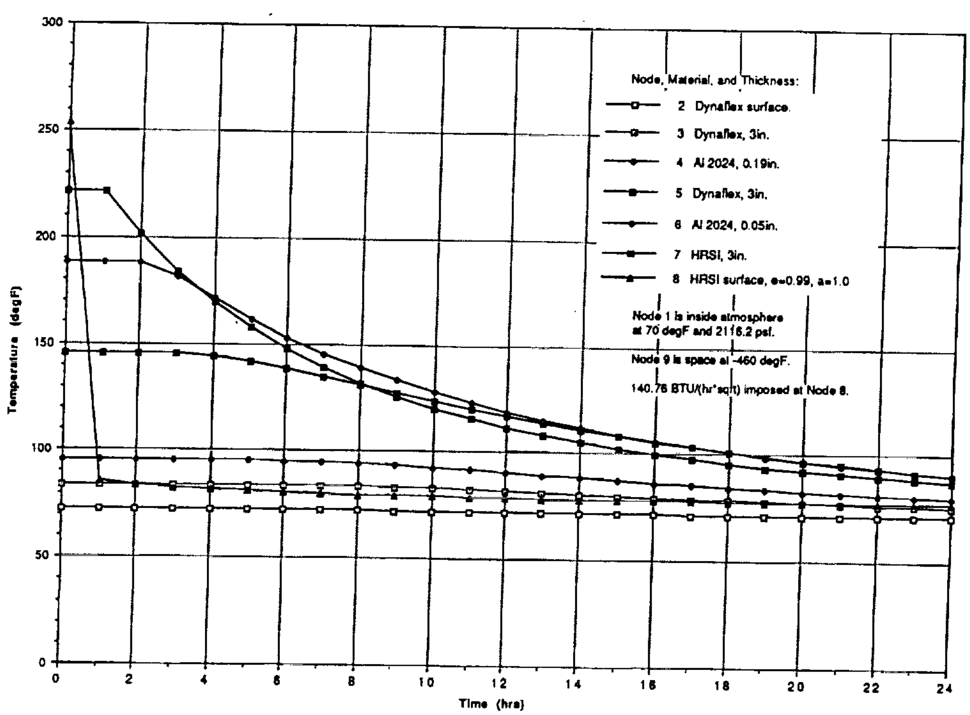


Figure 2.2-36 12-Hour Re-entry Thermal Conditioning

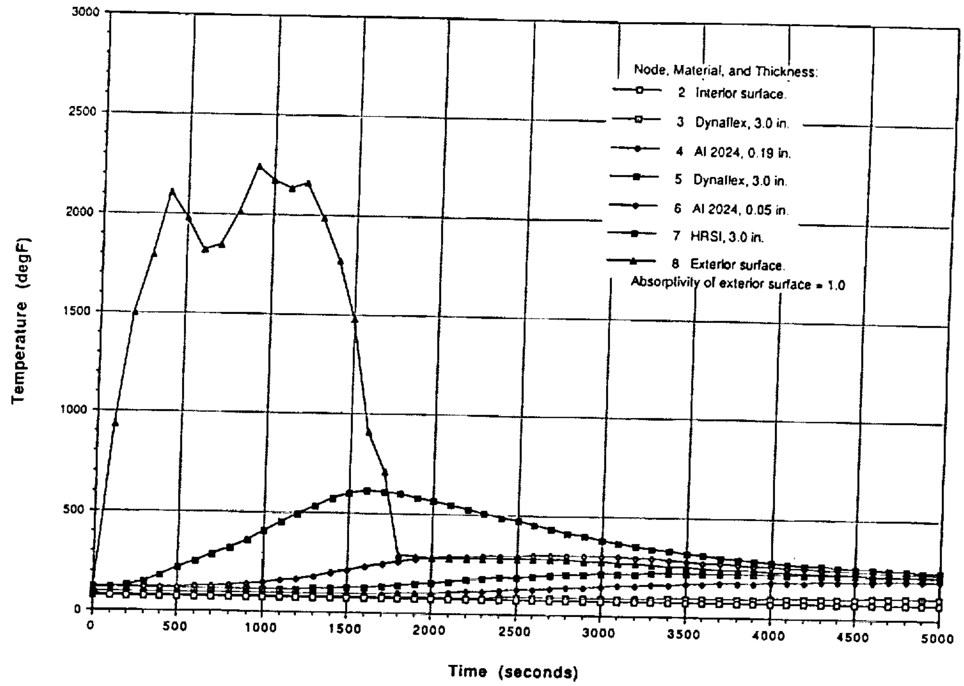


Figure 2.2-37 TPS/Structure Temperatures During Re-entry

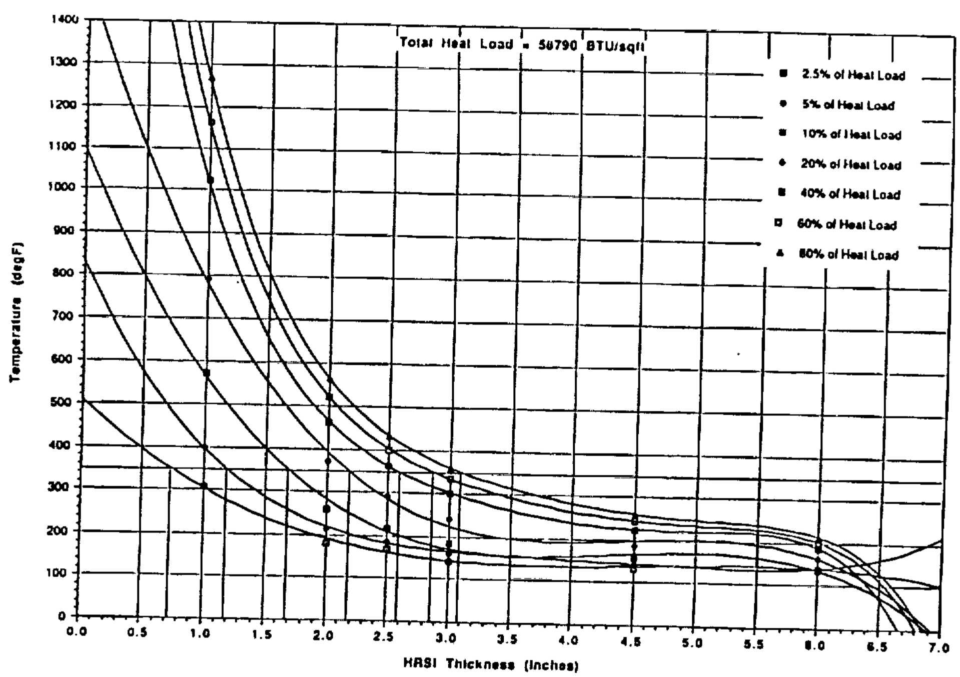


Figure 2.2-38 CTRV TPS Sizing Analyses

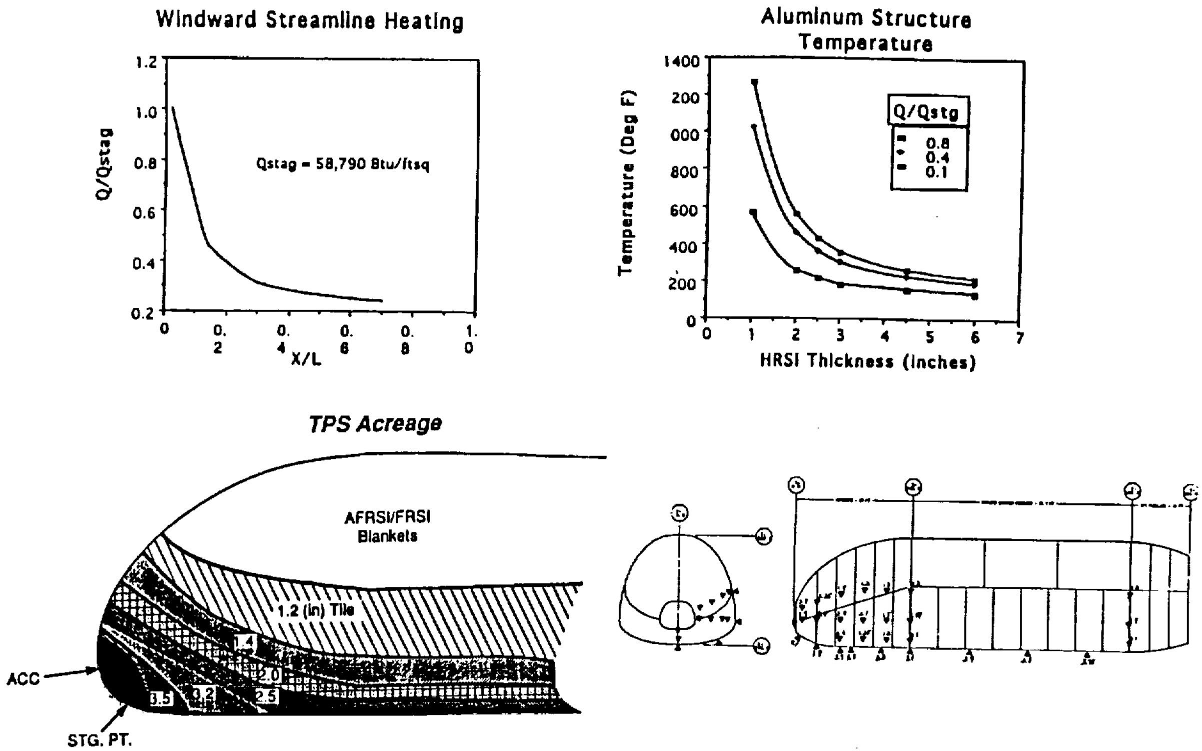


Figure 2.2-39 CTRV Aeroheating & TPS Thickness Distribution

- A passive subsystem consisting of materials applied externally to the primary structural shell of the CTRV in order to maintain exterior temperatures within acceptable limits.
- TPS materials and temperature limits are:
  1) Reuseable carbon-carbon (RCC). Used in areas where surface temperatures exceed 2300°F. RCC density is 98.5 lbm/cu.ft.
  2) High-temperature reusable surface insulation (HRSI). Acceptable for 1200°F≤T<2300°F. HRSI density is 9.0 lbm/cu.ft.
  3) Low-temperature reusable surface insulation (LRSI). Acceptable for 700°F≤T<1200°F. LRSI density is 9.0 lbm/cu.ft.
  4) Coated Nomex felt reusable surface insulation (FRSI). Acceptable for surface temperatures less than 700°F. FRSI density is 5.4 lbm/cu.ft.

TPS Comparison

Shuttle Orbiter

TPS Material	Area sq. ft.	Weight lb	Wt./Area lb/sq.ft.	% Total Weight
RCC	409	3,742	9.15	19.8
HRSI	5,164	9,728	1.89	51.5
LRSI	2,741	2,236	0.82	11.8
FRSI	3,581	1,173	0.33	6.2
Misc	—	2,025	—	10.7
TOTALS	11,895	18,904	1.59	100.0

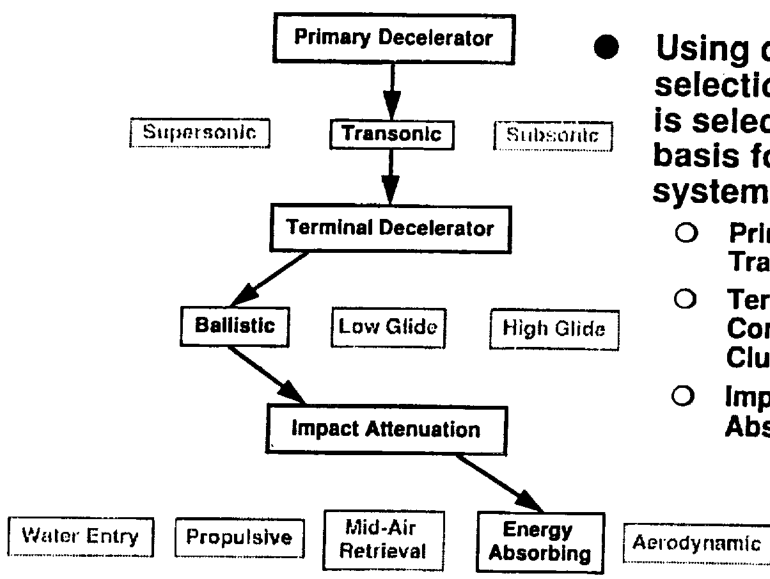
CTRV
Vertical tail included

TPS Material	Area sq. ft.	Weight lb	Wt./Area lb/sq.ft.	% Total Weight
RCC	143	1,307	9.14	14.2
HRSI	2,124	5,276	2.48	57.3
LRSI	1,611	1,760	1.09	19.1
FRSI	1,817	596	0.33	6.4
Misc	—	276	—	3.0
TOTALS	5,695	9,215	1.62	100.0

Figure 2.2-40 CTRV vs. Shuttle TPS Differences

2.2.2.6 Landing System Design

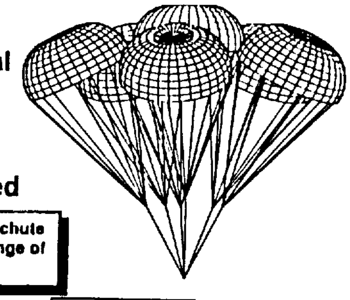
Landing system designs and performance were established for the CTRV concept based on a design approach which called for a supersonic deployment (Mach 1.4) of a drogue chute and parachute deployment loads under 4 g's. The current re-entry trajectory for the CTRV reaches the Mach 1.4 conditions at 80,000 ft altitude. Trade studies of parachute sizing and deployment options, system cost and weight impacts, and parachute deployment loads were performed to determine an appropriate landing system design for the CTRV. The parachute system selected for the CTRV is an eight chute cluster of 137 foot parachutes. This main chute system results in a terminal descent rate of 28 ft/sec velocity and weighs 4,100 lbs. In addition to the main parachutes, a drogue chute is required, as well as pilot chutes for both the main and drogue chutes. The CTRV parachute system weight is estimated at 5,865 lbs, requiring a volume of 140 cubic feet.



- **Using cost as the dominant selection driver, a scenario is selected to provide the basis for CTRV landing system preliminary design**
  - **Primary Decelerator - Transonic Drogue**
  - **Terminal Decelerator - Conventional Parachute Cluster**
  - **Impact Attenuation - Energy Absorbing Land landing**

Figure 2.2-41 Landing System Design Selection Path

Conventional Cluster: It is required that conventional parachutes be clustered for application to the CTRV weight range. A single chute would be impractically large.



The size/weight of the cluster is driven by the required terminal rate of descent:

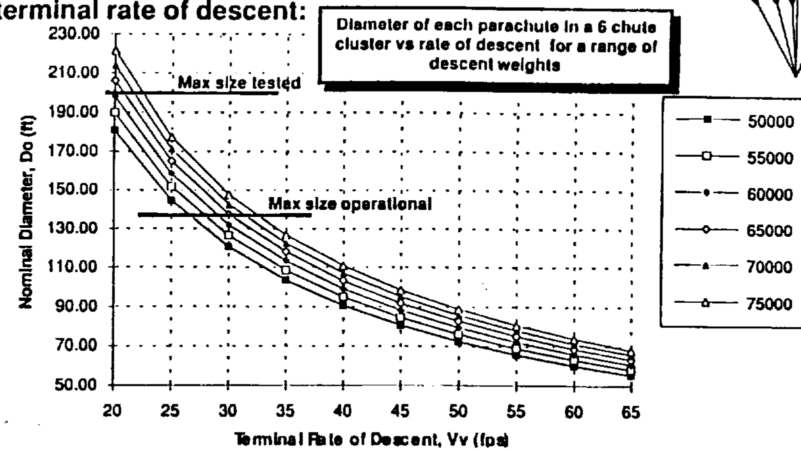


Figure 2.2-42 Parachute Sizing Trades

- Selection of Parachute Cluster Size depends heavily on required terminal rate of descent.
- Systems with retro rockets operate at a higher V_v than pure drag systems
- Weight curves for parachute/retro systems show a minimum system weight (W_s) for Steady State Rate of Descent (V_v)~50-60 fps.
- Pure drag systems are designed for the minimum V_v compatible with weight and volume constraints.
- For the CTRV, a vertical velocity range (A) of 20-60 fps provides a comprehensive basis for analysis.

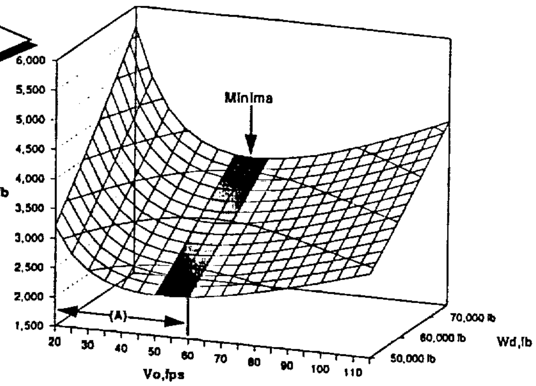


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Figure 2.2-43 Parachute Weight Sensitivity Analysis

- Vertical Velocities much lower than 30 fps result in excessively large canopies.
- Selecting an upper limit for D_o<150 ft. fits well with V_v<30 fps

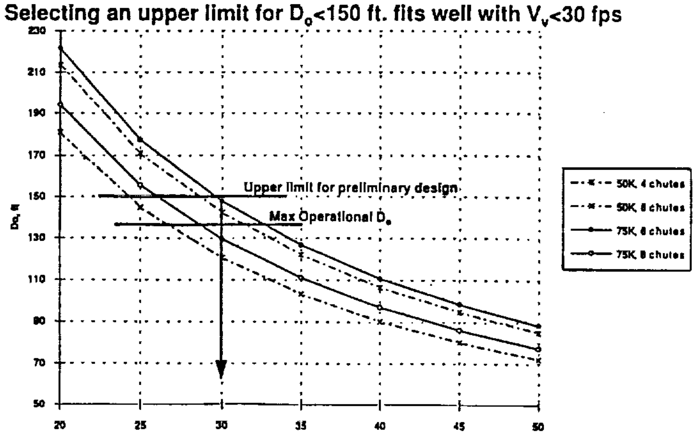

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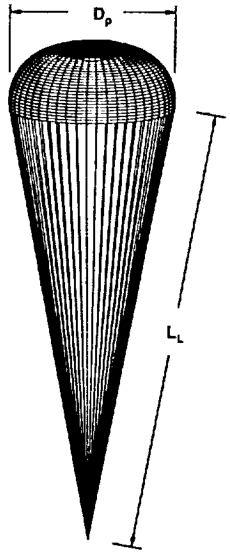
Figure 2.2-44 Landing Sink Rate Sensitivity Analysis



- **Drogue Parachute Type - Conical Ribbon**
- **Materials**
  - Canopy - Nylon
  - Suspension - Kevlar/Nylon
- **Reefing:**
  - Single Stage
  - S_r/S_o = 0.45 - 0.65
- **Physical Properties:**

	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Nominal Diameter, D_o (ft)	46.5	57.4
Projected Diameter, D_p (ft)	32.6	40.2
Number of Gores, N_g	56	68
Line Length, L_L (ft)	93	114.8
Drogue Pack Weight (lb) incl. bag	617	1114

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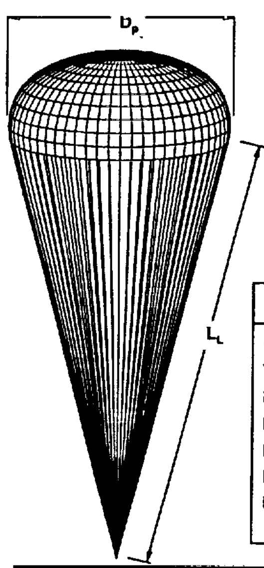
Figure 2.2-45 Medium CTRV Drogue Parachute Design



- **Main Parachute Type - Triconical**
- **Materials**
  - Canopy - Nylon
  - Suspension - Kevlar
- **Reefing:**
  - Single Stage
  - S_r/S_o = 0.85
- **Physical Properties:**

	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Cluster Size, N	5	8
Terminal Vertical Velocity, V_v (fps)	28	28
Nominal Diameter, D_o (ft)	139.1	137.1
Projected Diameter, D_p (ft)	96	95
Number of Gores, N_g	122	120
Line Length, L_L (ft)	201.7	198.8
Pack Weight (lb/chute) incl. bag	556	517

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Figure 2.2-46 Medium CTRV Main Parachute Design

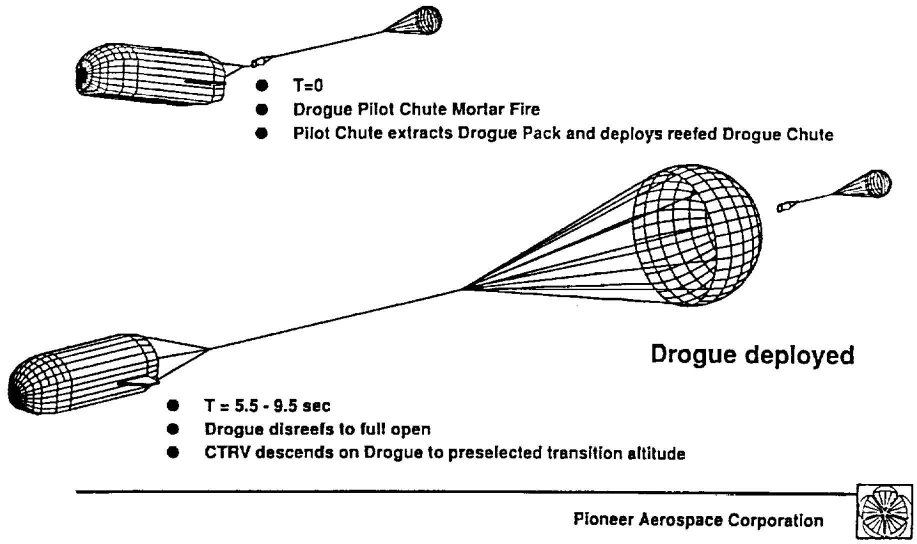


Figure 2.2-47 CTRV Drogue Parachute Deployment

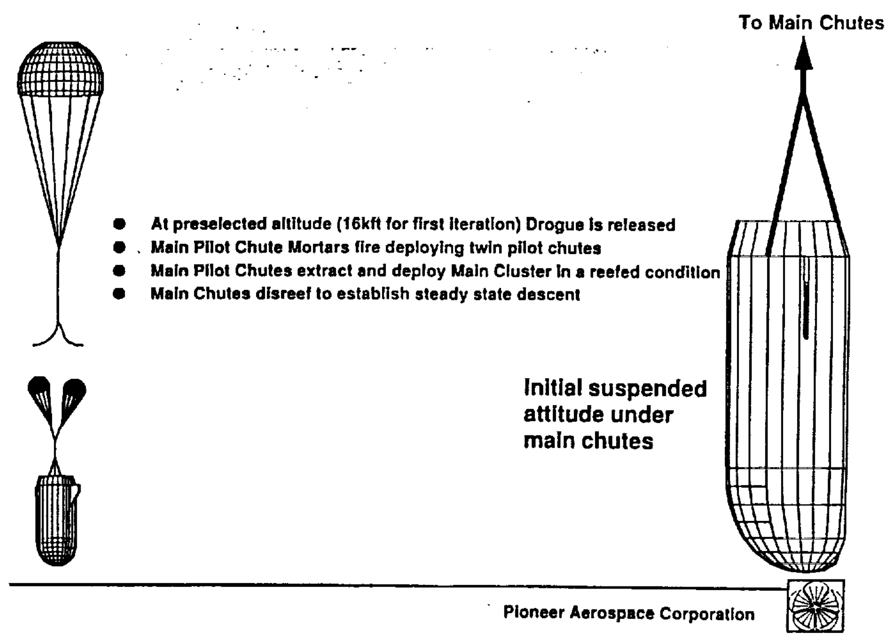


Figure 2.2-48 CTRV Main Parachute Deployment

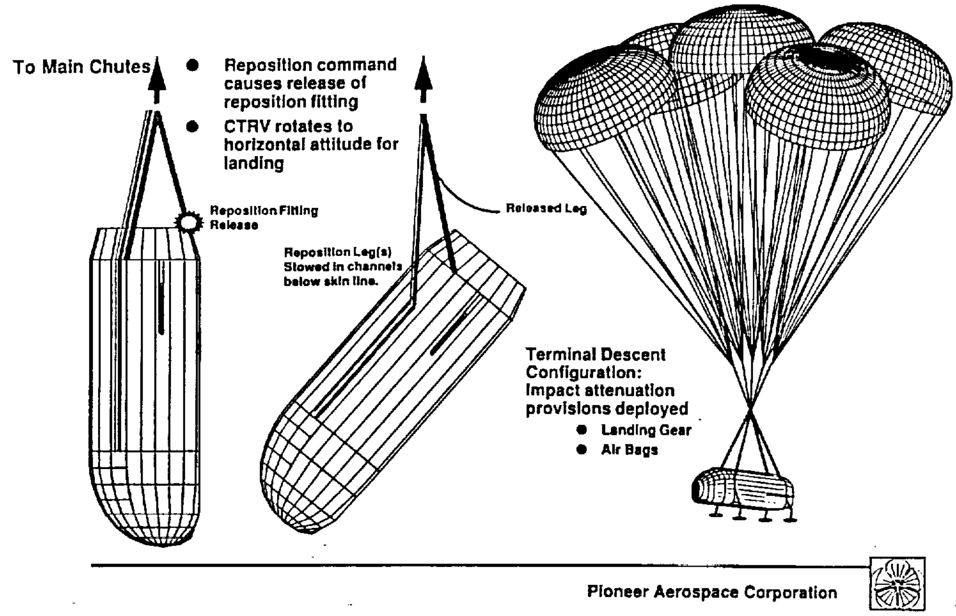
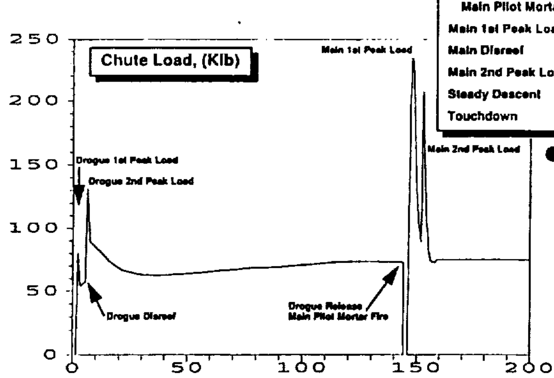


Figure 2.2-49 CTRV Parachute Reposition for Landing

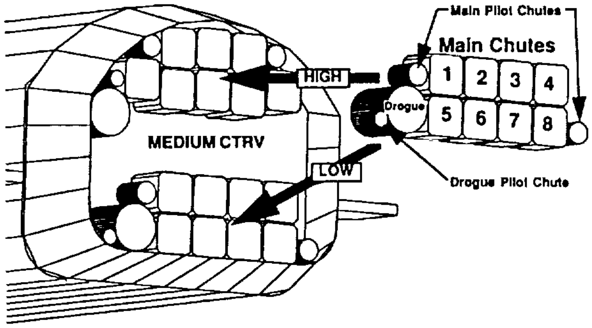
- **Estimated flight parameters for key events**

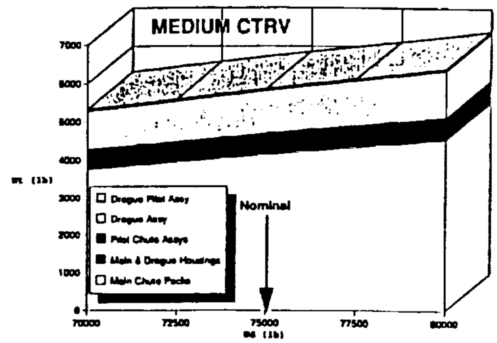
Event	Time (sec)	Alt (ft-MSL)	Vel (fps)	Mach No.	q (psf)	Chute Load (lb)	Accel (G)
Drogue Pilot Mortar Fire	0.0	87872	1376	1.4	55.1	0	1.0
Drogue 1st Peak Load	2.3	84714	1356	1.38	61.9	60291	2.18
Drogue Disreef	5.5	80530	1261	1.29	65.6	57367	1.94
Drogue 2nd Peak Load	6.3	79534	1223	1.25	64.8	131890	2.91
Drogue Release and Main Pilot Mortar Fire	144.0	16000	266	0.25	51.5	73010	1.0
Main 1st Peak Load	147.9	14031	254	0.24	40.4	234580	3.19
Main Disreef	152.0	14163	122	0.11	11.6	88585	1.2
Main 2nd Peak Load	152.6	14095	99	0.09	7.48	207505	2.77
Steady Descent	160.0	13800	33	0.03	0.83	74963	1.0
Touchdown	630.0	0	27	0.02	0.83	74945	1.0



- **Refined information required for subsequent iterations:**
  - C_d profile of CTRV (Super,Trans,Sub-sonic)
  - Angle of attack profile for CTRV under the influence of inflated drogue chute

Figure 2.2-50 CTRV Landing System Performance



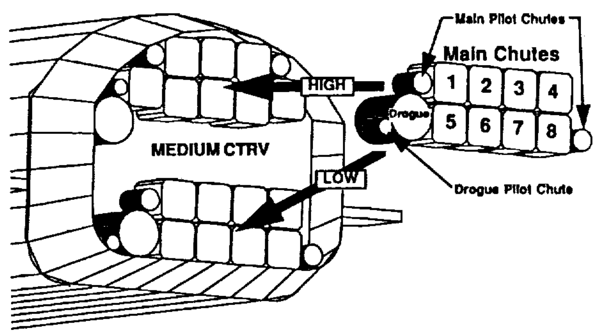


- RECOVERY SYSTEM COMPONENT WEIGHTS
  - Conventional Cluster recovery system designed for V_s=28 fps
  - System weight breakdowns vary with descent weight
- MAIN CHUTE PACKS: Includes main chutes with reefing provisions, deployment bags and risers
- MAIN & DROGUE HOUSINGS: Stowage for main and drogue parachute packs
- PILOT CHUTE ASSYS: Includes 2 pilot chutes with deployment bags, risers and mortars
- DROGUE ASSY: Drogue pack with reefing provisions, deployment bag and riser
- DROGUE PILOT ASSY: Drogue pilot chute pack with deployment bag, riser and mortar

COMPONENT WEIGHTS	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Main Chute Packs	2778	4137
Main & Drogue Housings	174	263
Pilot Chute Assys	206	299
Drogue Chute Pack	617	1114
Drogue Pilot Assy	30	51
TOTAL	3805	5864

Figure 2.2-51 CTRV Landing System Weight

- Since available volume is unknown, required volume is calculated based on maximum pack density
- Pack densities in excess of 40 lb/ft³ are expensive and difficult to achieve
- Component Volumes Include:
  - Main Chute Assy: Includes Main chute pack and container
  - Pilot Chute Assy: Includes Pilot chute pack and mortar
  - Drogue Chute Assy: Includes Drogue chute pack and container
  - Drogue Pilot Assy: Includes Drogue Pilot chute pack and mortar



COMPONENT VOLUMES	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Main Chute Assys:		
Each	14.1	13.1
Cluster	70.5	104.8
Pilot Chute Assys:		
Each	2.1	3.1
Pair	4.2	6.2
Drogue Chute Assy	15.6	28.1
Drogue Pilot Assy	0.5	1.0
TOTAL	90.8	140.1

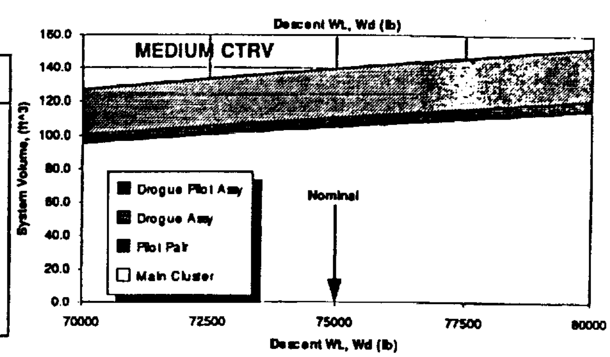


Figure 2.2-52 CTRV Landing System Volume

Q-3.

2.2.2.7 Reliability Analysis

A reliability analysis was completed for the reference (initial study definition) Medium CTRV configuration. Our MAtrix reliability model was used to determine reliability levels and maintenance requirements down to the subsystem and major components level. For the medium CTRV (mission duration of 60 hours), a reliability (Probability of Mission Success) of .995 was predicted. Maintenance requirements for this configuration are predicted to be an average of 4 unscheduled maintenance actions per mission.

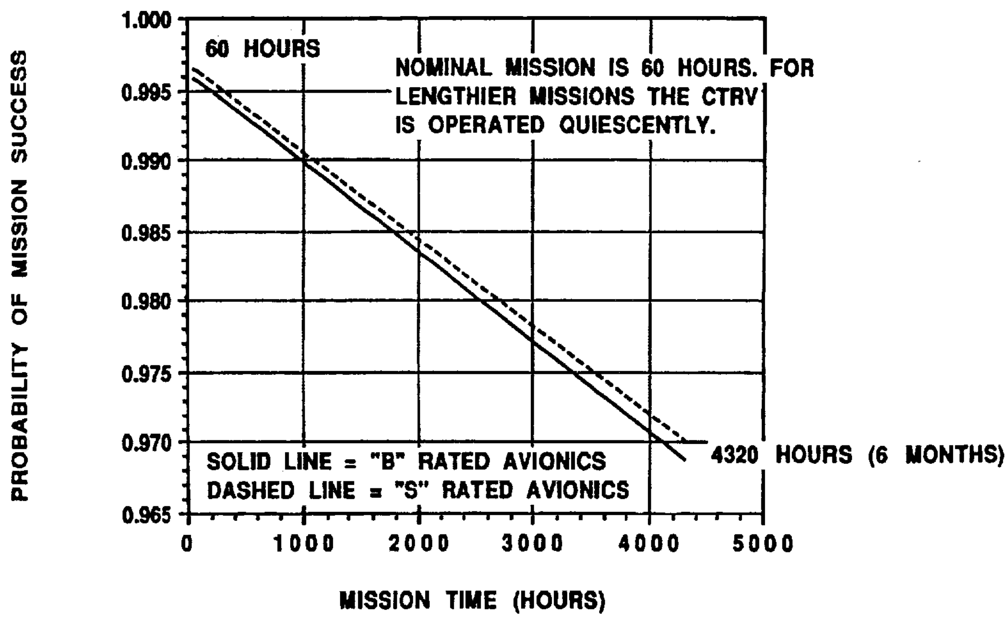


Figure 2.2-53 CTRV Reliability Prediction

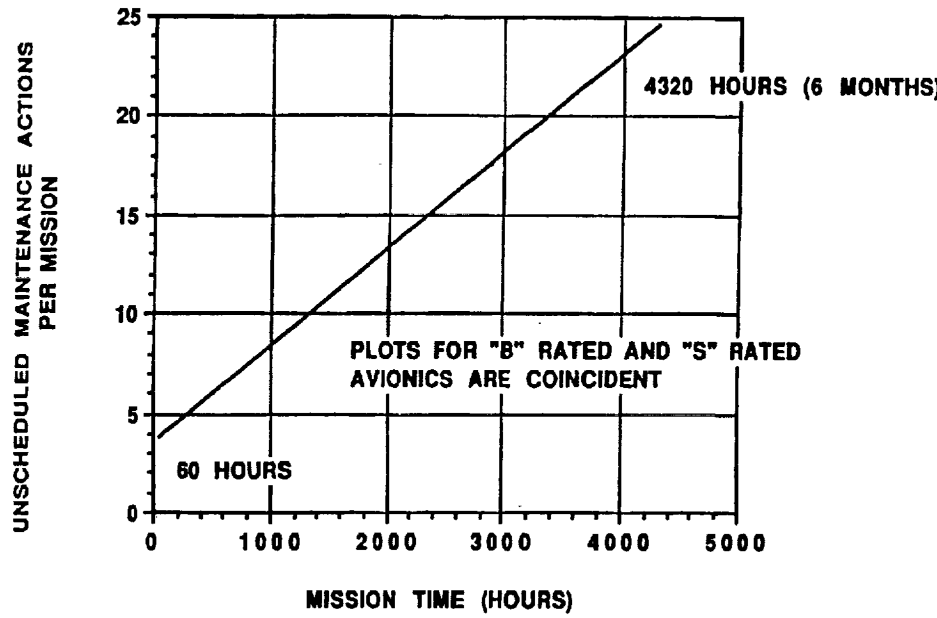


Figure 2.2-54 CTRV Unscheduled Maintenance Prediction

2.2.2.8 Launch Processing Analysis

Analysis of the CTRV ground (launch) operations was completed with a simulation of the combined Medium CTRV, PLS, and their launch vehicle (NLS-2 type) systems launched over a ten year period. The CTRV systems were modeled at the subsystem level to identify the payload integration differences between the Medium and the Integral CTRV concepts. The analysis demonstrated that the planned flight rates for PLS/CTRV to support Space Station Freedom logistics missions can be achieved when using the subsystems processing times as projected by the PLS program. The simulation included periodic maintenance down-periods for both PLS and CTRV, as well as unscheduled maintenance activities resulting from mission (flight) failures.

Analysis of the CTRV ground (launch) processing operations was also performed with an upgraded STARSIM model which included the effects of the predicted maintenance delays for the vehicle configuration. A maintainability/logistics simulation of the CTRV system was first performed to establish the launch processing delay factor as a function of the spares Probability Of Sufficiency POS. The simulation indicated that a logistics delay factor of eight times the system total mean time to repair (MTTR) would exist for this configuration. The STARSIM analysis then demonstrated that at the anticipated mission reliability levels for this CTRV, a 90% (POS) spares level will be satisfactory for this system. This translated into an average delay (per launch cycle) of just over five days. While this is not an insignificant factor, other launch systems analyzed have shown significantly higher delay times and/or required significantly higher spares inventory.

Integral CTRV

- Requires removable propulsion module
- Propulsion module deserviced in SAEF-2
- Vehicle transported from SAEF-2 to SSPF for payload integration, and then back to SAEF-2
- Payload/PLM's integrated into vehicle in SSPF
- Two vehicle types

Medium CTRV

- Propulsion system deserviced in OPF
- Payload/PLM's transported from SSPF to OPF
- Payload/PLM's integrated into vehicle in OPF
- One vehicle type

Modeling Assumptions

- Post 2007 "steady-state" scenario without a Space Shuttle program
- 350 work-days/year (245 work-days/year more realistic?)
- SSPF and VAB are non-hazardous processing facilities
- Hazardous fueling operations performed at launch pad
- 21-day minimum launch interval does not apply
- No simultaneous missions allowed (one vehicle going up or down at a time)
- Multiple vehicles may be docked to space station
- Maximum CTRV subsystem commonality with PLS (HL-20)

Figure 2.2-55 CTRV Launch Processing Assumptions

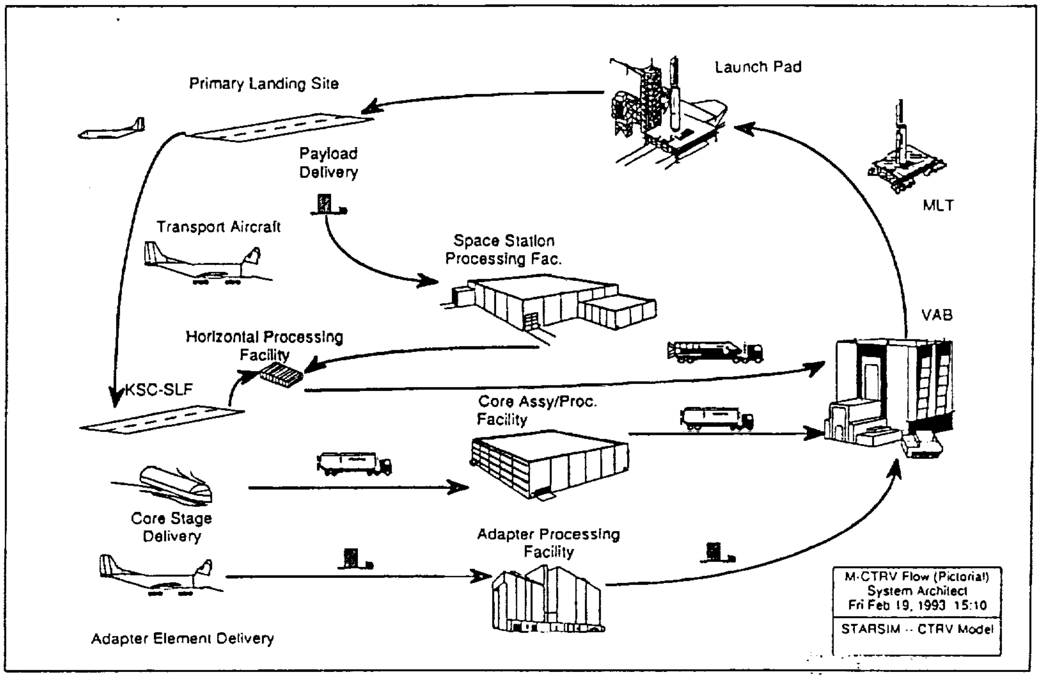


Figure 2.2-56 Medium CTRV Launch Processing Flow Diagram

<u>Throughput Capacity:</u>

PLS Launch Rate	4 flights/year
CTRV Launch Rate	4 - 6 flights/year
Avg. Time in System	46 days
Avg. Time Between Launches	39 days
Touch Labor Estimate	862 - 1,178 k-hours/year

<u>Facility/Resource Capability:</u>

Name	HPF	VAB	Pad	MLT
Servers	3	2	2	3
Location	KSC	KSC	KSC	KSC
Status	Modified (OPF)	Modified	Modified	Modified (MLP)
Utilization	17%	17%	25%	81%

• MLT is the constraining resource

Figure 2.2-57 Medium CTRV Launch Facilities Utilization

2.2.3 Integral CTRV Concept

The Integral CTRV concept was developed to reduce the packaging overhead of Space Station logistics cargo payloads. Rather than load the logistics payloads into a pressurized module which is then loaded into the unpressurized payload volume of the CTRV, the Integral CTRV payload bay itself is pressurized and the payloads can be installed directly into the CTRV. The Integral CTRV thus directly replaces the Space Station Pressurized Logistics Module and remains at the Space Station for several months duration before returning to Earth. Space Station unpressurized logistics payloads are similarly delivered in an unpressurized version of the Integral CTRV. The Integral CTRV is a ballistic type re-entry vehicle (parachute landing) but the aft payload bay location requires large fins for aerodynamic stability during re-entry. Analysis of the Integral CTRV concept was performed to determine the differences in design requirements for this alternative CTRV approach and to bring the design definition up to a level more consistent with the reference Medium CTRV. The study effort included trajectory analysis, aerodynamics and thermal analysis, preliminary structural design in the area of payload retention and deployment, and parametric analysis of the CTRV landing system options (parachutes).

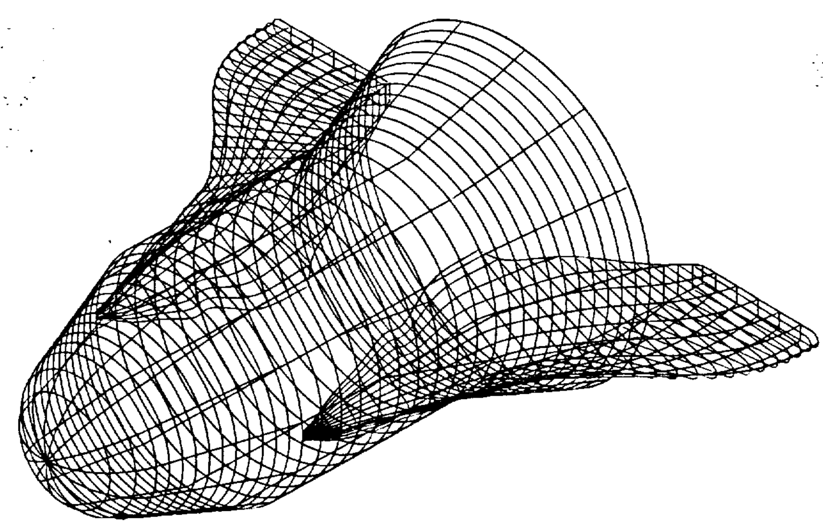


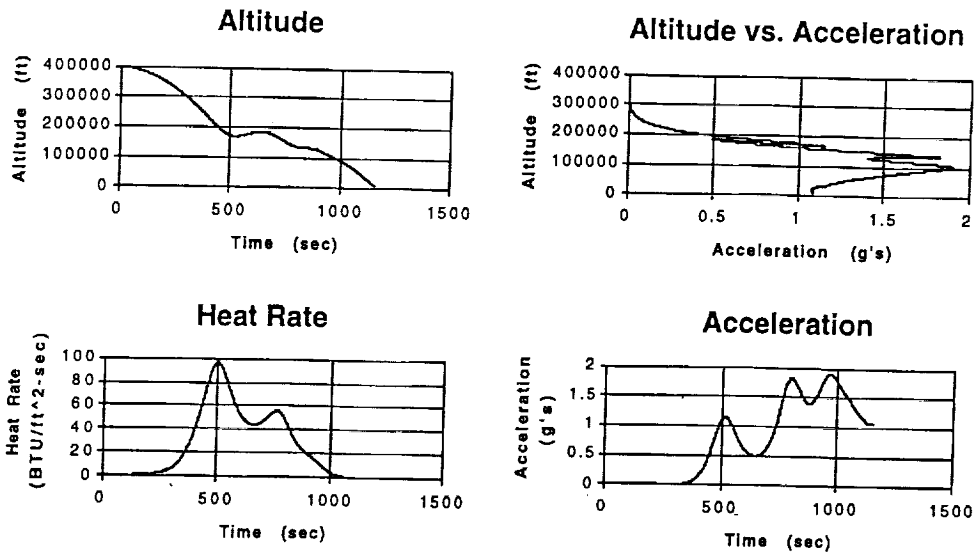
Figure 2.2.3-1 Integral CTRV Concept

2.2.3.1 Requirements Analysis

A new set of design requirements was prepared for those Integral CTRV subsystems which were unique to this configuration (pressurized volumes, hatches, and atmosphere circulation/conditioning). These requirements were added to the other subsystems requirements documents which had been prepared for the Medium CTRV concept. Modified system level requirements documents were also prepared for the Integral CTRV in both the pressurized and the unpressurized configurations.

2.2.3.2 Trajectory Analysis

Trajectory analyses with the selected Integral CTRV configuration were performed to determine the effects of flying various angle of attack trajectories and cross range. The analysis showed that this configuration could be flown to a maximum cross range of 70 NMi without violating structural and thermal loads constraints. The trajectories were flown at a 7.5° to 15° angle of attack to achieve these results. This low angle of attack also minimizes heat load to the TPS system, especially to the majority of the vehicle surfaces (along the fuselage sides). Since the vehicle is not flown at a high angle of attack (40° or higher like the Medium CTRV), there is not a large surface area which requires TPS tiles, a design goal for this CTRV configuration.



- Re-entry trajectories flown at $\alpha = 7.5°$
- Max cross range = 400 NMi (for $\alpha = 15°$)
- Max Q = 636 psf

Figure 2.2.3-2 Integral CTRV Design Trajectory

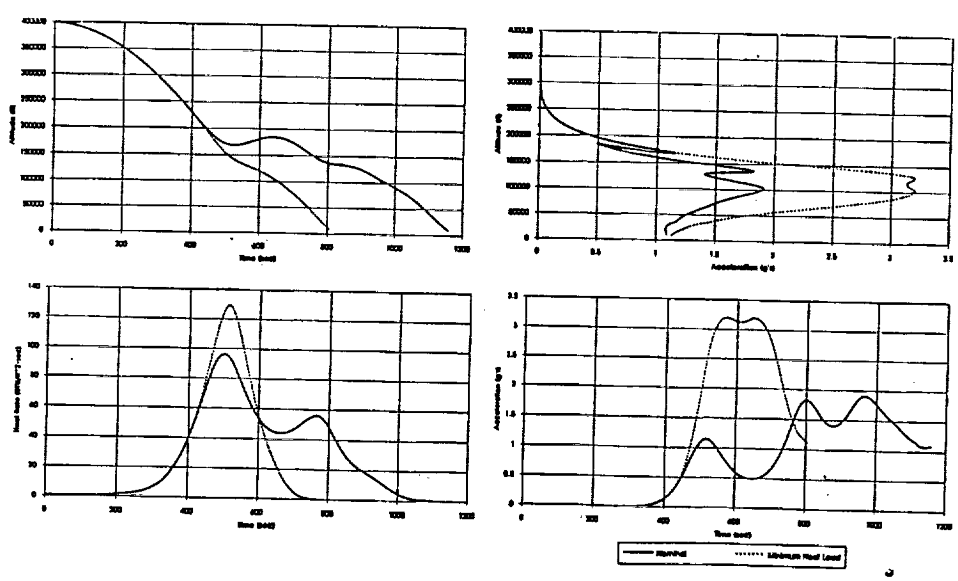


Figure 2.2.3-3 Integral CTRV Trajectory Options

2.2.3.3 Aerodynamics Analysis

The aerodynamic shape for an Integral CTRV is highly constrained by the required payload volume (diameter and length). The pressurized and unpressurized versions of an Integral CTRV are still required to carry Space Station logistics elements which result in a payload bay diameter of 15 feet. Given this diameter constraint, the aerodynamic shape which worked satisfactorily for the Medium CTRV will not scale down to the lower payload weight range of the Integral CTRV. The vehicle body length required to achieve both sufficient lift and required CG for this diameter payload bay remains at approximately seventy feet long, the same length as required for the Medium CTRV. Several lift and drag producing modifications were analyzed for the reference aerodynamic shape, including flared base regions and small to large wing sections. All of these configurations could be made to satisfy the re-entry heating, cross range, and acceleration limits for a CTRV, but they all ended up at a seventy foot fuselage length.

This analysis indicated that an entirely new aerodynamic approach will be required for the Integral CTRV in order to arrive at acceptable structural weight fractions. A significant reduction in the payload bay diameter would alleviate this requirement, but would force major redesigns of the Space Station logistics elements. Alternative aerodynamic concepts which were studied included the use of high drag devices (such as ballutes) and lift producing deployable (or even inflatable) structures. Deployable structures such as these can produce over twice as much lift and/or drag as the baseline configurations without long fuselage lengths and forward CG constraints.

The final aerodynamic configuration for a low cross range version Integral CTRV was developed by performing a sensitivity study of lift (C_l) and drag (C_d) coefficients from a general ballistic re-entry CTRV configuration. The variable C_l and C_d parameters were used to generate a map of optimized re-entry trajectory performance as calculated from several POST computer simulations. This process enabled a desirable combination of C_l and C_d to be determined which satisfied cross range, acceleration, heat rate, and heat load constraints for the Integral CTRV. The resulting configuration was a basic cylindrical mid fuselage with a forward fuselage consisting of a 20° cone with 5 foot radius sphere nose. A 20° flared skirt was added at the aft end of the cylindrical section to further increase drag. A pair of large fins (not wings) were added to the cylinder/skirt to move the aerodynamic center of pressure aft for hypersonic flight stability and CG considerations.

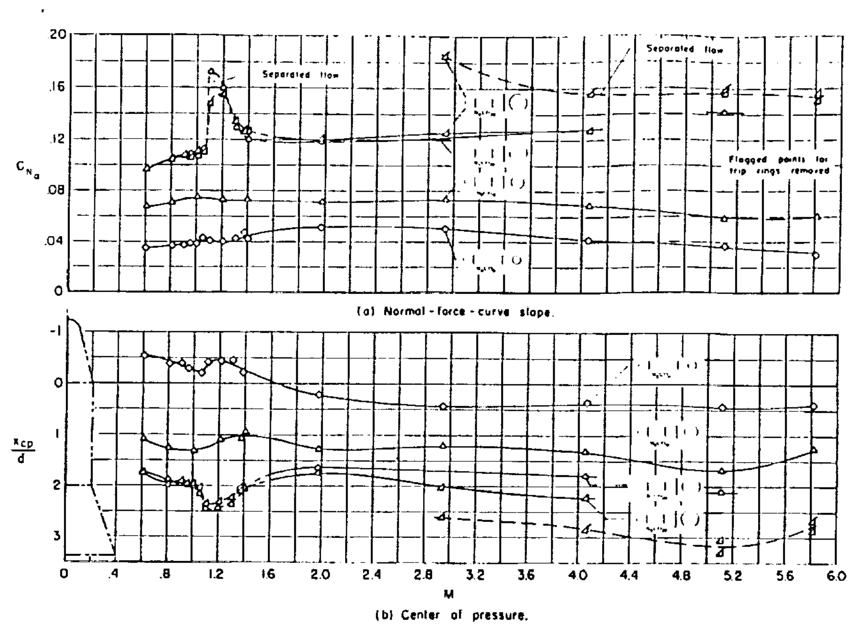


Figure 4.- Effect of flare angle on aerodynamic characteristics of flared bodies; $\alpha = 0^{\circ}$.

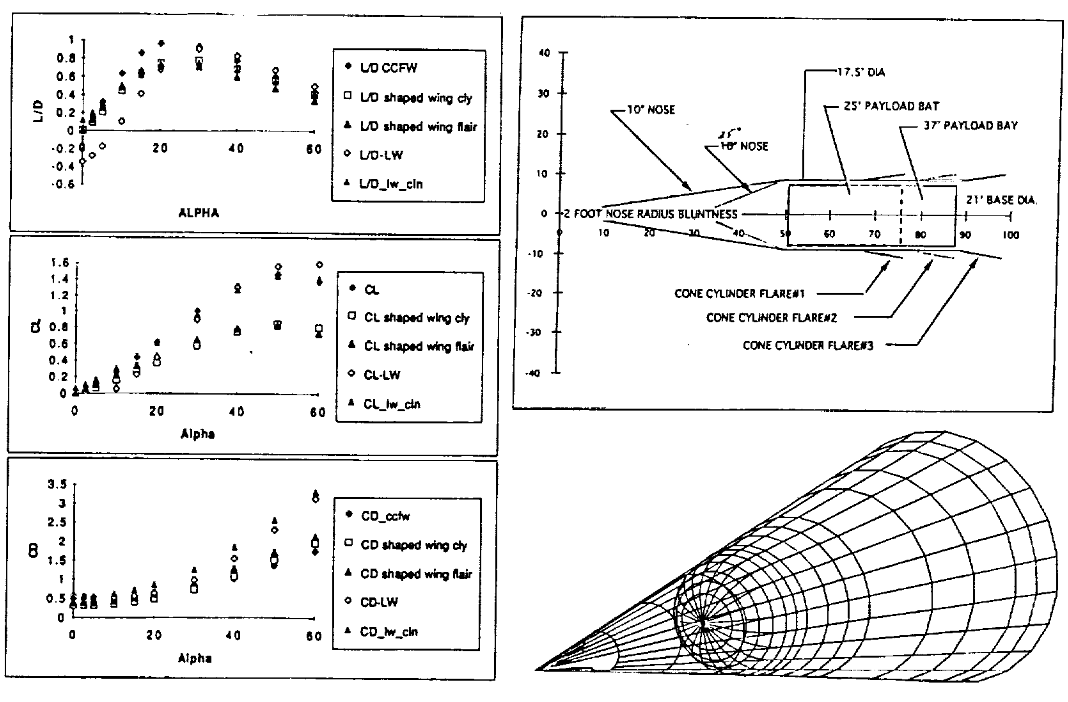


Figure 2.2.3-4 Aerodynamics Trade Study Options

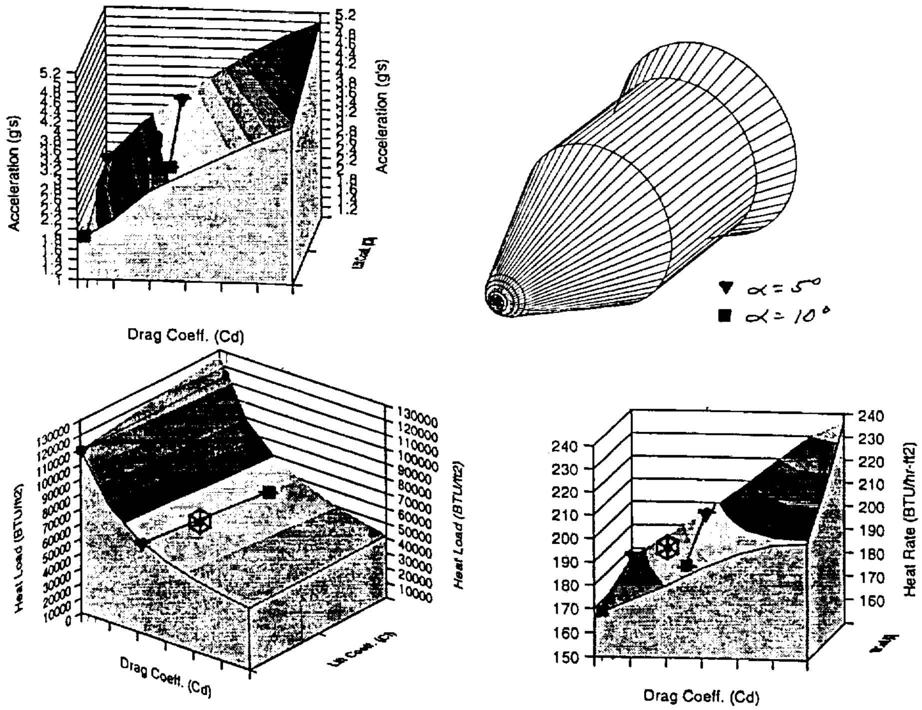


Figure 2.2.3-5 Aerodynamic Shape Optimization Map

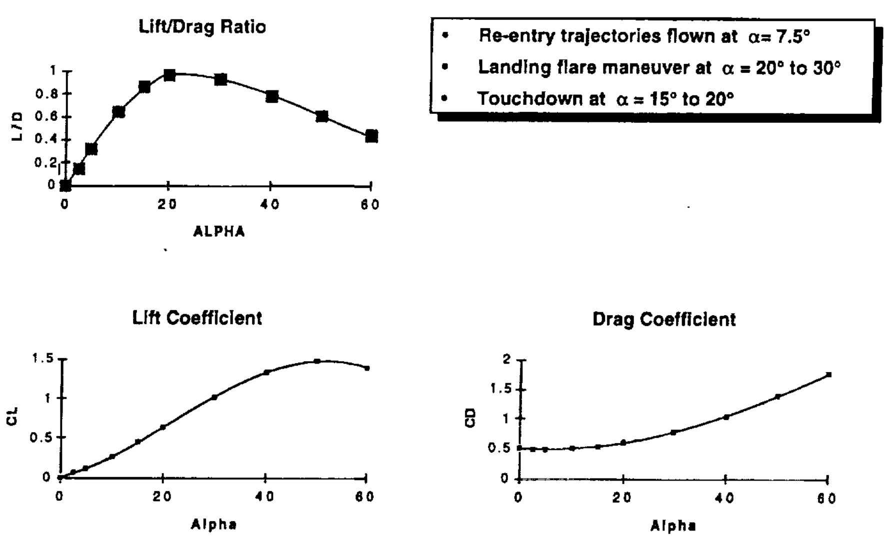


Figure 2.2.3-6 Aerodynamic Data - Hypersonic Regime

2.2.3.4 Structures Design and Analysis

Payload installation and deployment for the Integral CTRV concepts has generally been envisioned to be through end mounted doors. A large 15 foot diameter door for the unpressurized cargo version, and a smaller Space Station sized hatch for the pressurized cargo configuration. A payload bay configuration which is common to both pressurized and unpressurized cargo can be designed, but the unpressurized configuration presents some unique issues for on-orbit operations. In a Shuttle-like payload bay, the cargo is fully exposed for a 180° clear access and the payloads are removed radially. In an end opening payload bay as envisioned for the Integral CTRV, payloads must be removed axially. This presents limited access and deployment capabilities for the unpressurized configuration. (The pressurized configuration functions in the same manner as the Pressurized Logistics Module which it is replacing). The payloads must be installed and removed sequentially, moving only that cargo element which is closest to the door at any time. Payload retention and deployment concepts have been designed for the unpressurized configuration by use of a clocking arrangement, but still the installation and deployment sequence is restricted. Additionally, the clearance requirements and timing of opening or closing the end-mounted payload bay door (a 15 foot diameter structure) must be examined for Space Station impacts.

Structural design of the Integral CTRV configuration was completed to the point of generating a layout of major structural elements (such as beams, longerons, and frames) for the fuselage (payload bay) and fin surfaces. Three view dimensioned drawings , as well as perspective wire-frame and color surface views of the external surfaces, were also prepared. Estimates of the structure weight were prepared from the layout drawings. The fuselage design for this concept must be quite stiff since there are no wings to react flight bending loads (as in the Shuttle mid-fuselage design). Unlike the Medium CTRV concept with Shuttle-like payload bay/doors, this Integral CTRV design concept provides a circular fuselage which offers a more efficient structural shape.

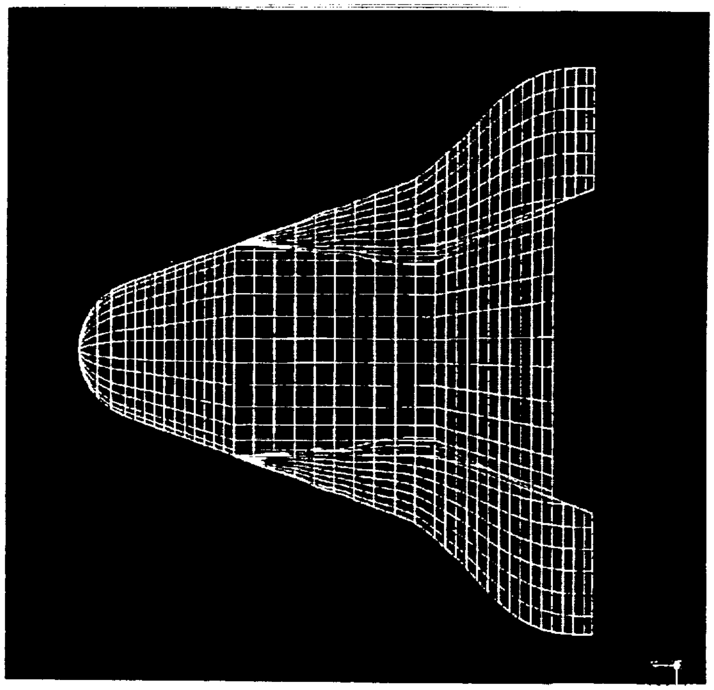


Figure 2.2.3-7 Integral CTRV - Plan View

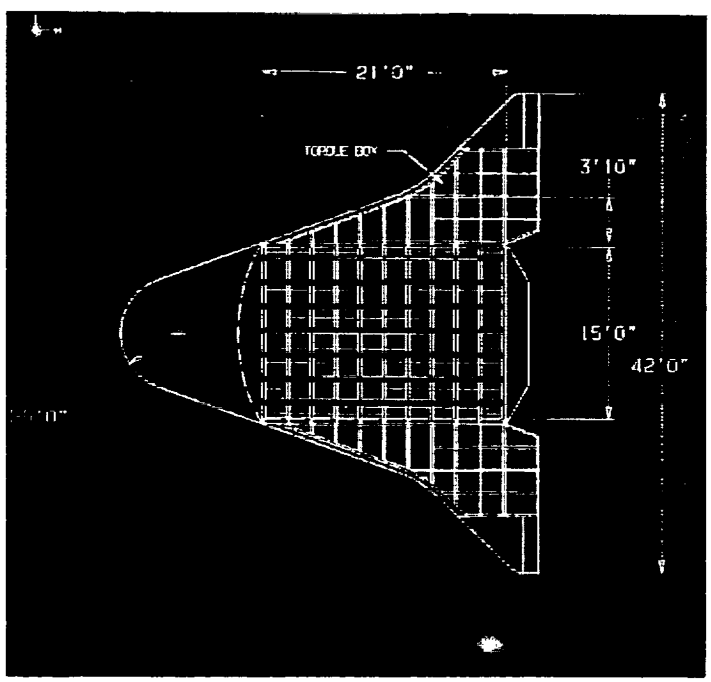


Figure 2.2.3-8 Integral CTRV - Structural Layout

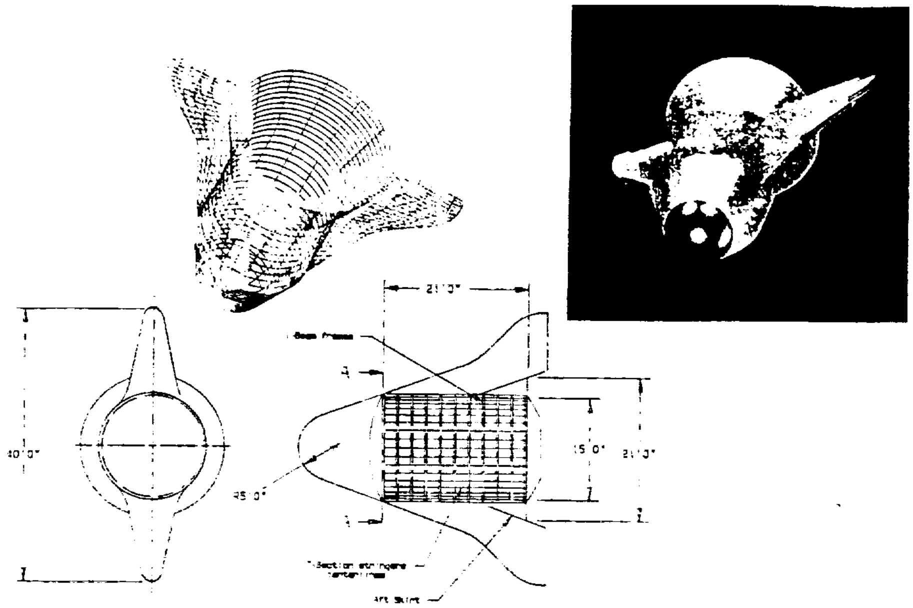


Figure 2.2.3-9 Integral CTRV Design Configuration

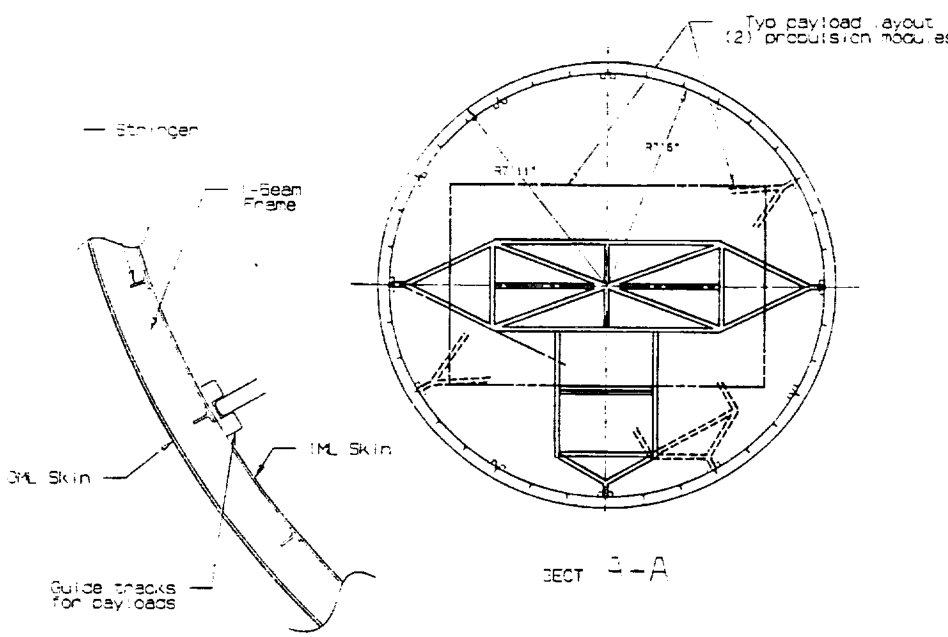
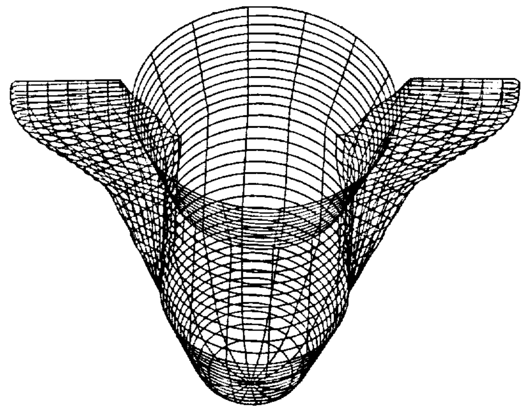


Figure 2.2.3-10 Integral CTRV Payload Installation

Structure	15,700
TPS	2,250
Active Thermal Control	800
OMS/RCS	1,500
Elec. Power Generation	1,780
Elec. Power Distribution	1,660
Avionics	1,065
Purge/Vent	380
Landing	5,600
	26,735 Dry Weight
Consumables	
OMS/RCS	4,000
Fuel cell reactant	1,700
Payload	22,000
Total Launch Weight	58,435
Re-entry Weight	53,300



Ballistic Coefficient = 77 psf

Payload Fraction = 38 %

Figure 2.2.3-11 Integral CTRV Weight Estimate

2.2.3.5 Thermal Analysis & TPS Sizing

Preliminary sizing of Integral CTRV thermal protection system (TPS) was estimated from an assumed heating rate distribution over the vehicle. The TPS sizing was based on flying a low angle of attack (7.5°) trajectory, as demonstrated in POST trajectory analyses of the selected configuration. These trajectories resulted in low heating rates to most of the surfaces and much lower total heat loads than as seen in the Medium CTRV trajectories. The total heat load for the Integral CTRV trajectory is less than 20,000 BTU/ft^2, compared to 50,384 BTU/ft^2 for the Medium CTRV (C23LNF). The maximum heat rate for the Integral CTRV is held within advanced carbon-carbon nose cap material limits by the large radius nose cone (5 ft).

The thermal protection system (TPS) concept for the Integral CTRV was modified to avoid the problem of bonding TPS tiles to a pressure vessel (severe technical design issues related to tile gaps and on-orbit/re-entry structural temperature limits would be encountered). A debris shield will be used as an intermediate structural shell for attaching the tiles. Current Space Station debris shield concepts (which have a several inch standoff from the primary pressure vessel structure) will permit underlying insulation blankets to protect the pressure vessel from the tile bondline temperatures (both on-orbit and during entry). The debris shield will protect the tiles from excessive gap changes caused by expansion of the pressure vessel in space. Use of either aluminum or higher temperature materials (titanium, Inconel, ...) for the debris shield will permit a wide range of TPS concepts to be explored for this configuration. Significant operational efficiencies may also be realized by this concept since the tiles can be installed or maintained off the vehicle by removing the debris shieldpanels.

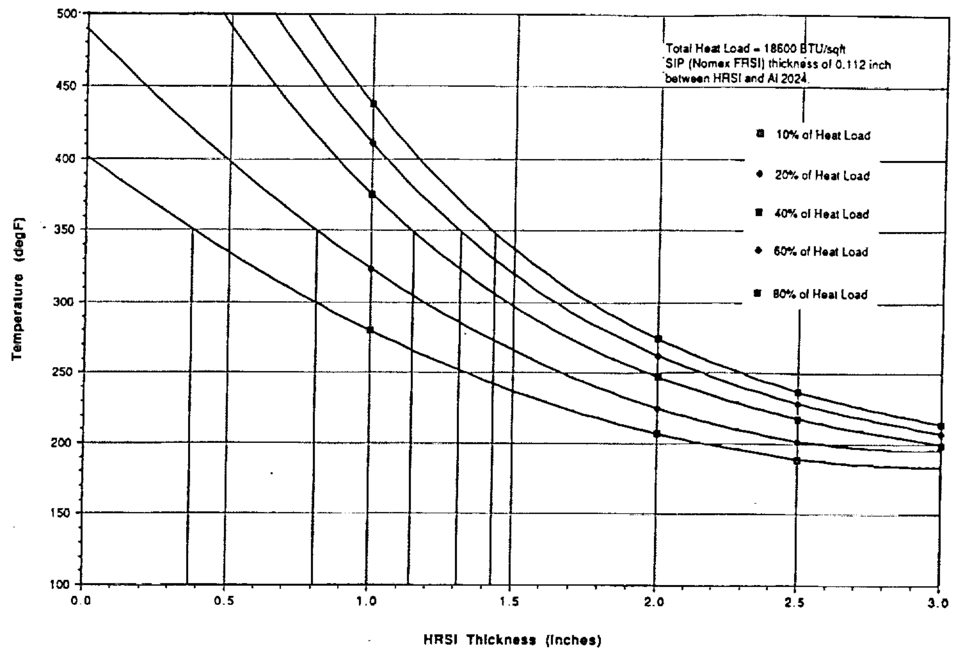


Figure 2.2.3-12 Integral CTRV Tile Sizing Chart

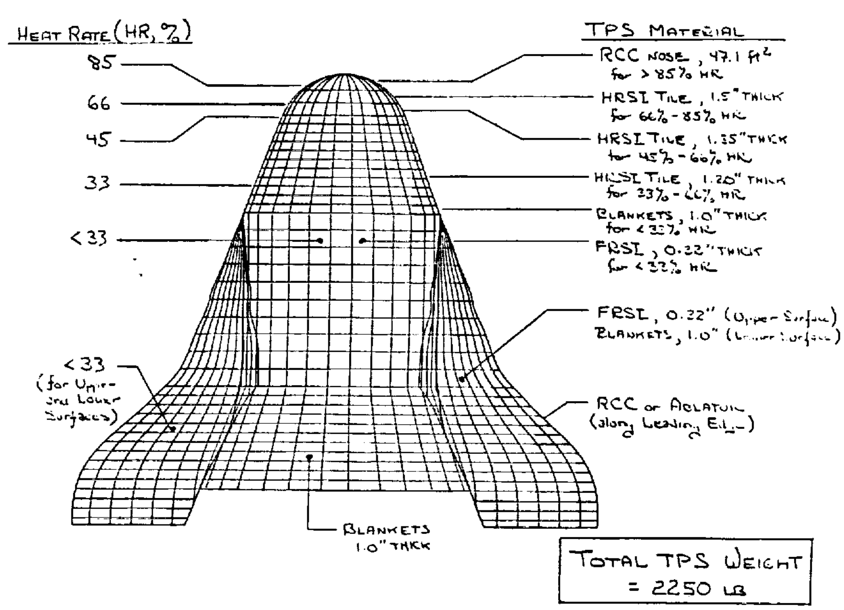


Figure 2.2.3-13 Integral CTRV TPS Distribution

2.2.3.6 Landing System Design

A parachute system design was also selected for the Integral CTRV configuration. The Integral CTRV system is similar to the Medium CTRV design and consists of a five chute cluster of 139 foot parachutes. This main chute system results in a terminal descent rate of 28 ft/sec velocity and weighs 2,800 lbs. In addition to the main parachutes, a drogue chute is required, as well as pilot chutes for both the main and drogue chutes. The total weight for the Integral CTRV parachute system is estimated at 3,800 lbs, requiring a stowage volume of 91 cubic feet.

- High Glide devices such as the parafoil promise significant benefits for space payloads
  - Precise Landing at Selected Site
  - Wind Penetration
  - Soft Landing
- The State of the art is defined by the NASA (MSFC) Advanced Recovery Systems demonstration in which a 15,000 lb payload was recovered and guided to a soft landing in Nov '91.
- The US Army (Natick Labs) proposes to advance the ARS technology to demonstrate guided recovery of airdrop payloads of 42,000 lb by the end of FY'96.
  - The high glide option will become viable for the Integral CTRV at that payload level
  - Application to the Medium CTRV will require additional expansion of the payload envelope
- High glide recovery should be held as an evolution path for the CTRV (ICTRV first)

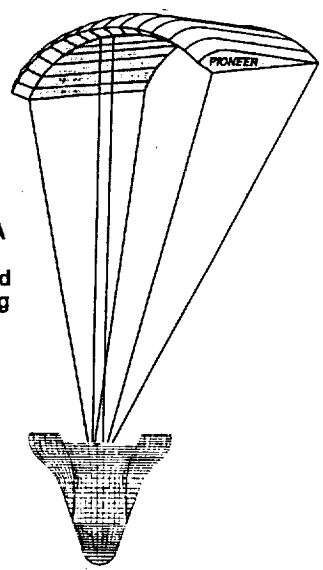


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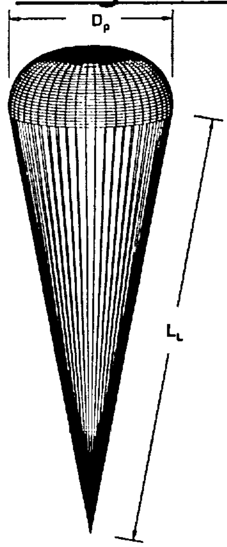
Figure 2.2.3-14 Potential Integral CTRV Landing System Option

Drogue Parachute

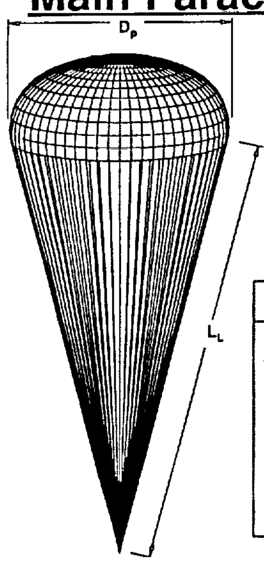


- Drogue Parachute Type - Conical Ribbon
- Materials
  - Canopy - Nylon
  - Suspension - Kevlar/Nylon
- Reefing:
  - Single Stage
  - S_r/S_o = 0.45 - 0.65
- Physical Properties:

	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Nominal Diameter, D_o (ft)	46.5	57.4
Projected Diameter, D_p (ft)	32.6	40.2
Number of Gores, N_g	56	68
Line Length, L_L (ft)	93	114.8
Drogue Pack Weight (lb) incl. bag	617	1114

Figure 2.2.3-15 Drogue Parachute Design

Main Parachute Cluster



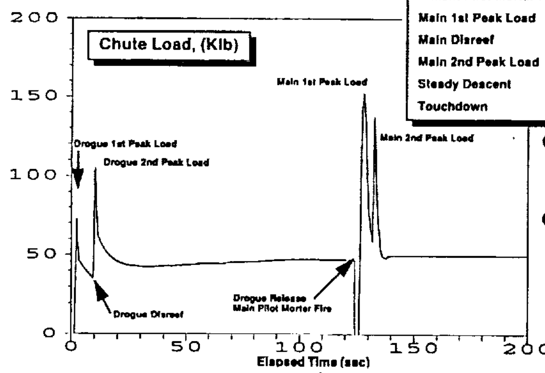
- Main Parachute Type - Triconical
- Materials
  - Canopy - Nylon
  - Suspension - Kevlar
- Reefing:
  - Single Stage
  - S_r/S_o = 0.85
- Physical Properties:

	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Cluster Size, N	5	8
Terminal Vertical Velocity, V_v (fps)	28	28
Nominal Diameter, D_o (ft)	139.1	137.1
Projected Diameter, D_p (ft)	96	95
Number of Gores, N_g	122	120
Line Length, L_L (ft)	201.7	198.8
Pack Weight (lb/chute) incl. bag	556	517

Figure 2.2.3-16 Main Parachute Design

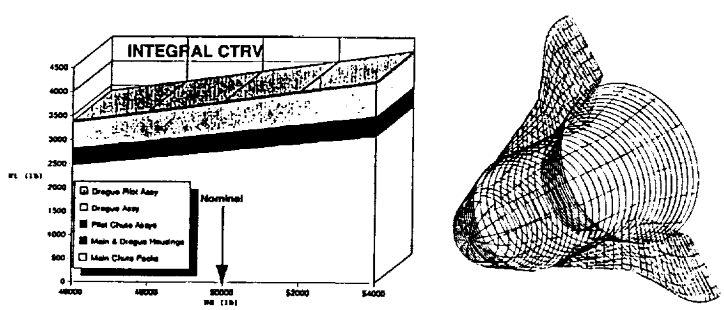
- **Estimated flight parameters for key events**

Event	Time (sec)	Alt (ft-MSL)	Vel (fps)	Mach No.	q (psf)	Chute Load (lb)	Accel (G)
Drogue Pilot Mortar Fire	0.0	71442	1361	1.4	119.5	0	1.0
Drogue 1st Peak Load	2.3	68415	1254	1.29	117.7	73143	3.56
Drogue Disreef	9.5	60886	877	0.91	83.5	34863	2.19
Drogue 2nd Peak Load	10.3	60201	828	0.86	76.7	105422	3.39
Drogue Release and Main Pilot Mortar Fire	124.0	16000	245	0.25	50.8	47105	1.0
Main 1st Peak Load	127.9	14880	250	0.24	47.2	152459	3.14
Main Disreef	132.0	14216	122	0.11	11.5	58550	1.19
Main 2nd Peak Load	132.6	14150	98	0.09	7.43	137494	2.76
Steady Descent	140.0	13860	33	0.03	0.83	49917	1.0
Touchdown	613.0	0	27	0.02	0.83	49937	1.0



- **The transonic drag area profile of the CTRV is essential to the parachute drag simulation**
- **Key assumptions include:**
  - CTRV Super sonic $C_d \sim 1.12$
  - CTRV Sub sonic $C_d \sim 0.42$
  - Angle of attack transition smooth from pre-recovery (high α) to steady state pre-reposition ($\alpha = 0$)

Figure 2.2.3-17 Landing System Performance



- **RECOVERY SYSTEM COMPONENT WEIGHTS**
  - Conventional Cluster recovery system designed for V_v=28 fps
  - System weight breakdowns vary with descent weight
- **MAIN CHUTE PACKS:** Includes main chutes with reefing provisions, deployment bags and risers
- **MAIN & DROGUE HOUSINGS:** Stowage for main and drogue parachute packs
- **PILOT CHUTE ASSYS:** Includes 2 pilot chutes with deployment bags, risers and mortars
- **DROGUE ASSY:** Drogue pack with reefing provisions, deployment bag and riser
- **DROGUE PILOT ASSY:** Drogue pilot chute pack with deployment bag, riser and mortar

COMPONENT WEIGHTS	Integral CTRV W_d = 50 Klb
Main Chute Packs	2778
Main & Drogue Housings	174
Pilot Chute Assys	206
Drogue Chute Pack	617
Drogue Pilot Assy	30
TOTAL	3805

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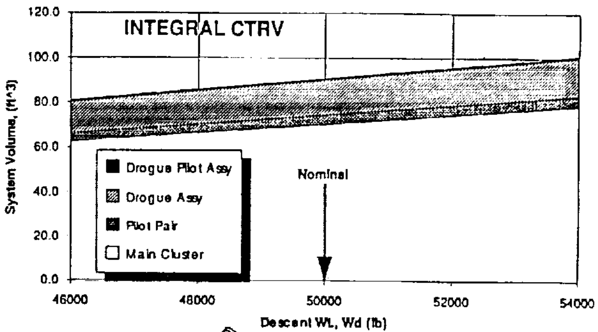
Figure 2.2.3-18 Landing System Weight

- Since available volume is unknown, required volume is calculated based on maximum pack density
- Pack densities in excess of 40 lb/ft^3 are expensive and difficult to achieve
- Component Volumes Include:
  - Main Chute Assy: Includes Main chute pack and container
  - Pilot Chute Assy: Includes Pilot chute pack and mortar
  - Drogue Chute Assy: Includes Drogue chute pack and container
  - Drogue Pilot Assy: Includes Drogue Pilot chute pack and mortar



COMPONENT VOLUMES	Integral CTRV W_d = 50 Klb	Medium CTRV W_d = 75 Klb
Main Chute Assys:		
Each	14.1	13.1
Cluster	70.5	104.8
Pilot Chute Assys:		
Each	2.1	3.1
Pair	4.2	6.2
Drogue Chute Assy	15.6	28.1
Drogue Pilot Assy	0.5	1.0
TOTAL	90.8	140.1

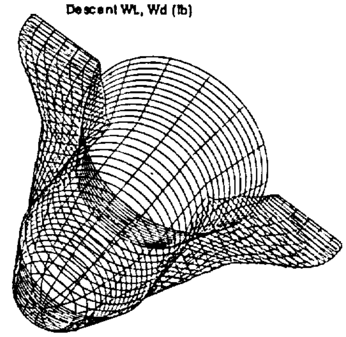


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Figure 2.2.3-19 Parachute Volume Requirements

2.2.3.7 Reliability Analysis

A reliability analysis was performed using the MAtrix model for the Integral CTRV configuration to determine reliability levels and maintenance requirements down to the subsystem and major components level. With the Integral CTRV's long mission duration of 4320 hours (6 months), the analysis revealed a reliability of .969 and 25 unscheduled maintenance actions per mission. Use of "S" rated ("S" for satellite, and very expensive) avionics parts for the Integral CTRV was assessed to try to improve the mission reliability over its long mission duration. This increased the reliability only to .970, and thus is not recommended. The minimal improvement in reliability is due to the long dormant period for most of the avionics systems during the 6-month mission. Because the level of redundancy was not specified in many of the reference CTRV avionics components (and no redundancy was modeled unless specified), it is expected that better definition of actual component redundancies will improve the avionics (and total Integral CTRV) reliability.

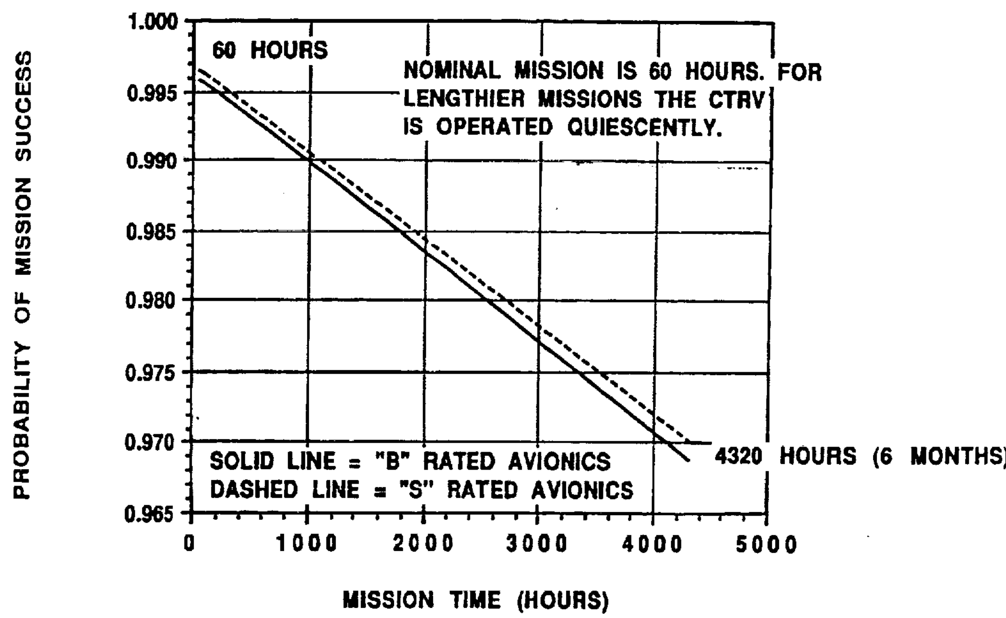


Figure 2.2.3-20 Integral CTRV Reliability

2.2.3.8 Launch Processing Analysis

Launch processing analysis of the Integral CTRV was performed similar to the Medium CTRV analysis. The CTRV was modeled at the subsystem level to permit expendable versus reusable evaluations to be made at the total CTRV system level or at the individual subsystems level (such as propulsion, avionics, TPS/heat shield). Subsystem processing times were identical to those used for the Medium CTRV. Only the payload integration activities and propulsion systems were different from the Medium CTRV analysis. Integral CTRV propulsion system processing was performed off the vehicle and payload integration was performed on the vehicle, just the opposite from the Medium CTRV concept. These differences did not significantly affect the results of the analysis, but the required system flight rate did affect the results. The Integral CTRV concept must fly up to 11 flights per year (versus 4 to 6 flights per year for the Med. CTRV). The launch processing simulation demonstrated that this higher flight rate can be achieved with the planned facilities and resources. A much higher manpower consumption is caused by the high flight rates, however. The Integral CTRV concept used almost twice as many hours of touch labor (direct "technician-hours") to accomplish the same SSF logistics supply mission as the Medium CTRV concept. This higher technician usage may or may not translate directly into higher launch processing costs, depending on the selected technician staffing levels and thus the resulting technician utilization factors (how much of the army is standing).

Of particular importance for the Integral CTRV processing analysis is the constraints imposed by the Space Station elements and payloads. The pressurized version of the Integral CTRV would be required to utilize the Space Station payloads processing facilities for integration of the payloads (Space Station racks) into the Integral CTRV payload compartment. This will require the propulsion systems for this CTRV to be either new (unflown) or removed and processed separately from the rest of the vehicle.

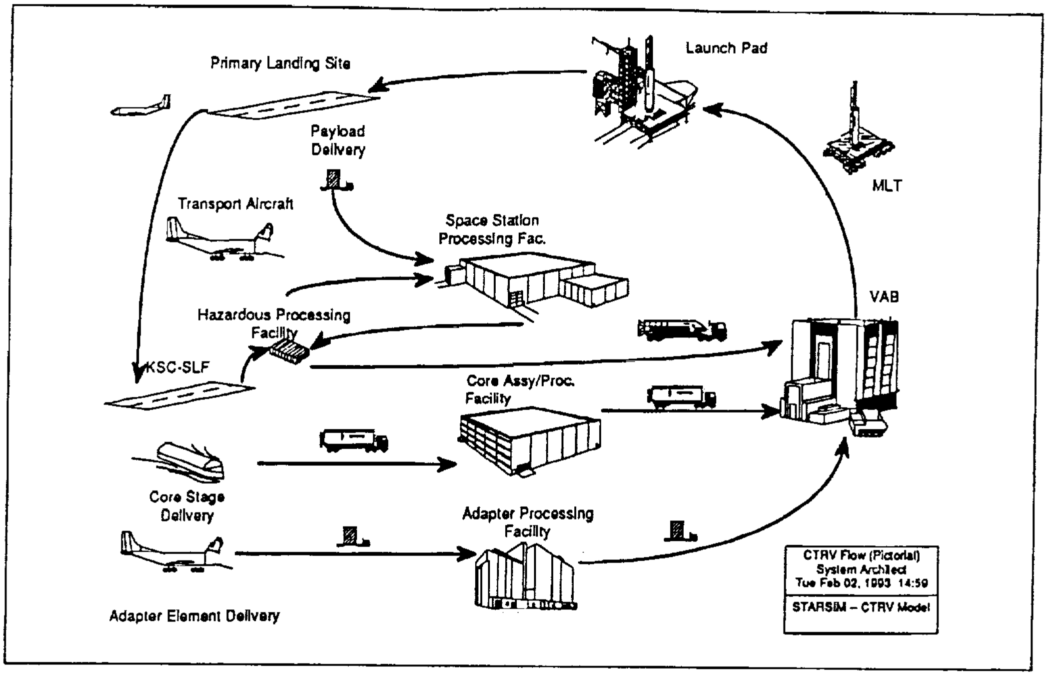


Figure 2.2.3-21 Launch Processing Flow Diagram

Throughput Capacity:

PLS Launch	4 flights/year
CTRV Launch Rate	6 - 11 flights/year
Avg. Time In System	49 days
Avg. Time Between Launches	26 days
Touch Labor Estimate	1,178 - 1,967 k-hours/year

Facility/Resource Capability:

Name	SSPF	HPF	VAB	Pad	MLT
Servers	2	2	2	2	3
Location	KSC	KSC	KSC	KSC	KSC
Status	New (Intgrtn. Cells)	Existing (SAEF-2 or PHSF)	Modified	Modified	Modified (MLP)
Utilization	2%	8%	19%	36%	86%

- MLT is the constraining resource

Figure 2.2.3-22 Facilities and Resources Utilization

2.2.4 Winged CTRV Concept

2.2.4.1 Introduction

As the analysis of Integral CTRV and Medium CTRV concepts progressed, it became apparent that the operating costs of these systems would not meet the goals of the NASA Access to Space study. A precision (runway) landing version of the CTRV concept was recognized as a key requirement for minimizing operations costs. This requirement led to the Winged CTRV concepts, which started with small payload capabilities (22,000 lbs), evolved to larger payload versions, and eventually to combined crew/cargo concepts such as the Crew Logistics Vehicle (CLV) and the scaled-up, cargo carrying version of the PLS (the HL-42). The Winged CTRV, the CLV, and the HL-42 concepts all became competitors for the crew/cargo element of a launch system architecture based on expendable launch vehicles. The development of each of these concepts evolved as the NASA Access to Space study (Option 2) continued to refine the design requirements. The CLV concept definition was continued by NASA's Johnson Space Center, the HL-42 definition by the Langley Research Center, and the Winged CTRV by the Marshall Space Flight Center with design support from this ATSS contract. The Winged CTRV concept evolved from the (original) small Winged CTRV, to a larger payload version (the Medium Winged CTRV), and finally to a combined crew/cargo version similar in function to the HL-42 but with a larger payload volume and weight capability (the Single Development Winged CTRV).

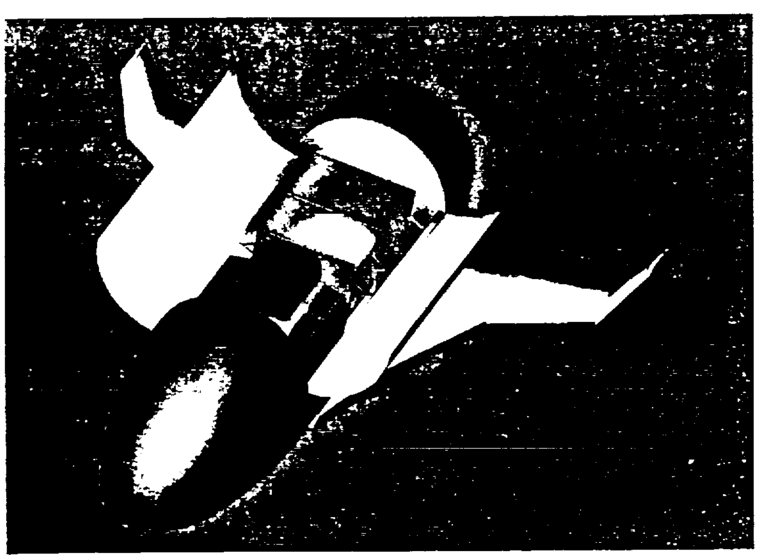


Figure 2.2.4-1 Winged CTRV Concept

2.2.4.2 Requirements Analysis

An analysis of the Access to Space study requirements was performed to identify traceability of CTRV and PLS requirements to NASA mission or other requirements. This task was performed with a Computer Aided System Engineering (CASE) software tool to track the flowdown of requirements and their allocation to system elements. The requirements provided were specifically for Option 2 of the Access to Space study, and only those requirements provided from the Access to Space study groundrules were used to populate the requirements database at this point. The launch systems database (of available or potential launch/space systems to which the requirements may be allocated) was limited to Option 2 systems. The analysis effort provided only top level requirements allocations, but the software model used is capable of being populated with requirements down to any level desired (e.g. to the subsystem level of selected launch or space systems). Several report formats were created to illustrate the requirements flowdown tree, the system elements tree, and the allocation of requirements and missions to each system element.

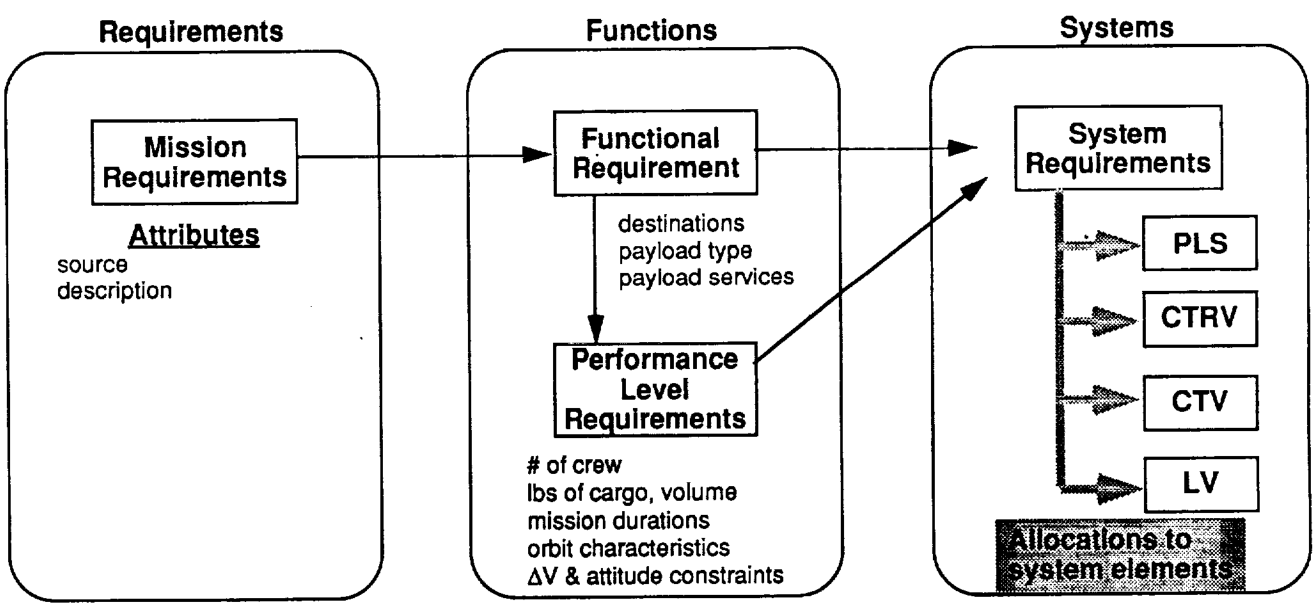


Figure 2.2.4-2 Access to Space - Requirements Flowdown

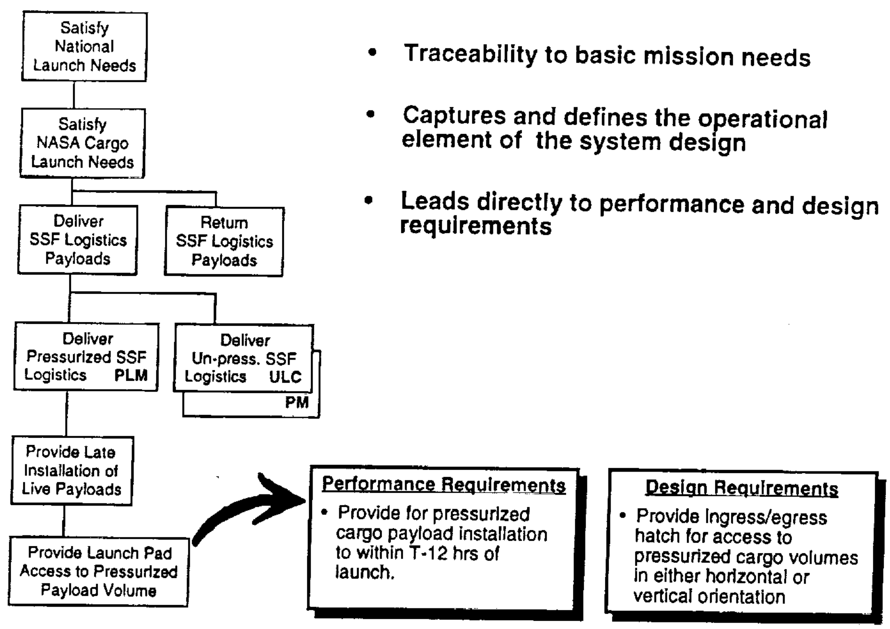


Figure 2.2.4-3 Functional Requirements Flowdown

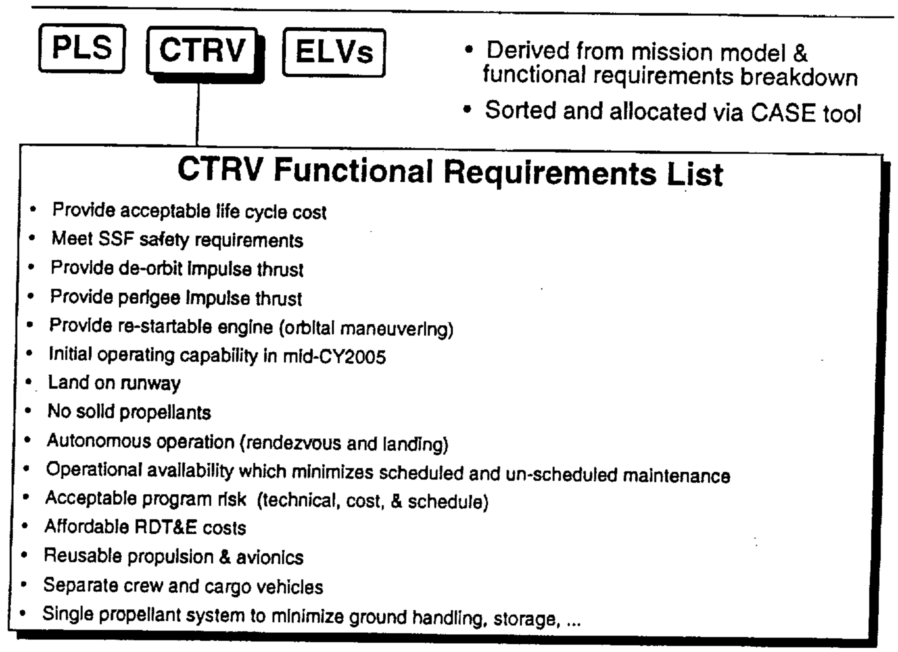


Figure 2.2.4-4 Element Requirements List

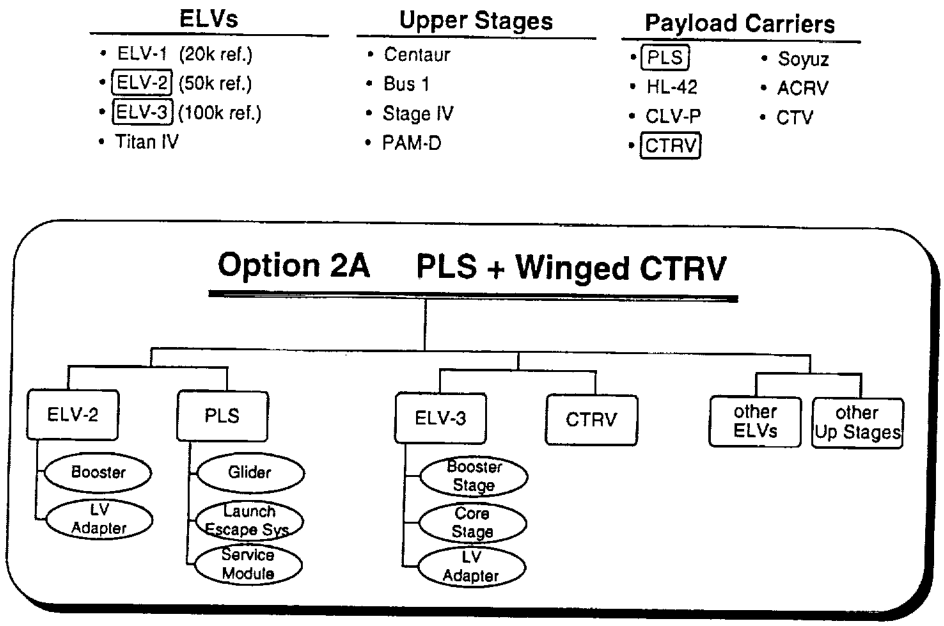


Figure 2.2.4-5 System Elements Tree

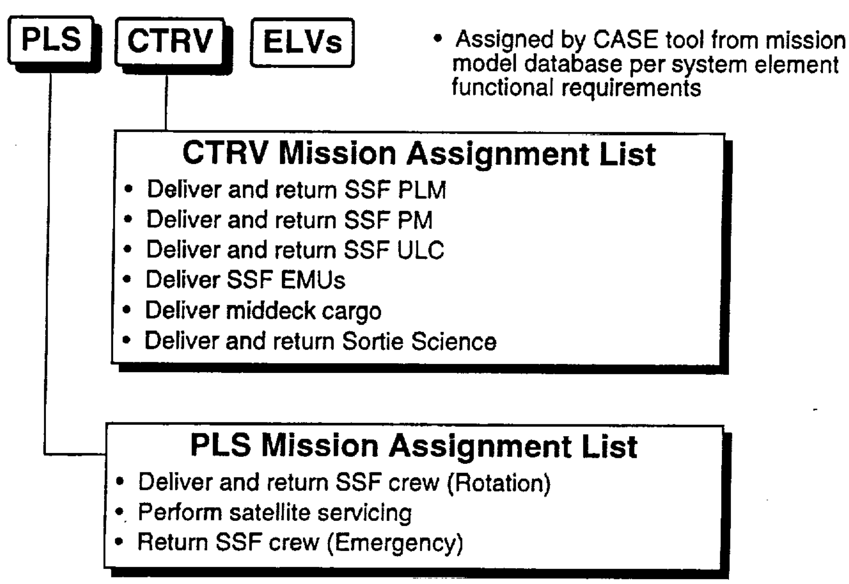


Figure 2.2.4-6 Element Mission Assignments

The evolution of the three Winged CTRV concepts was a direct result of evolving design requirements. There were three distinct design requirements which generated these three concepts:

Requirement	Design Solution
Reduce operations costs	Small Winged CTRV
Reduce annual launch rate	Medium Winged CTRV
Reduce development costs	Single Development Winged CTRV

The small Winged CTRV concept was a direct evolution of the Integral CTRV into a winged runway landing vehicle in order to reduce the high ground operations costs associated with parachute landing systems. The immediate design problem was to obtain a sufficient lift/drag ratio to perform a landing flare maneuver and actually fly the vehicle to a runway landing. The small Winged CTRV fuselage was sized to provide the same payload capability as the Integral CTRV concept (a 15' diameter by 25' long payload bay is provided, 22,000 lb payload mass). The Integral CTRV fins were converted to wings and moved to the bottom of the fuselage for landing. This design change made a Shuttle type payload bay more efficient and allowed the orbital propulsion systems to be located in an aft fuselage section. The Winged CTRV resembles a shortened Shuttle Orbiter but the wings have been modified (no camber and a higher aspect ratio) to reduce ascent aerodynamic loads on the booster and provide a much reduced entry cross range capability. A combined crew/cargo payload capability is possible with this winged CTRV concept by installing a small pressurized crew cabin in the forward fuselage section.

The Medium Winged CTRV concept is a larger version of the small Winged CTRV concept with increased payload capability (42,500 lb) in order to reduce the required number of launches. The payload bay was lengthened from 25' to 37.5' in length and wing span increased to handle the higher landing weight. A vertical tail (stabilizer) was added to this configuration to improve directional stability. This concept can be used in conjunction with a Spacehab module to provide a late access capability for refrigerated or biological payloads. This CTRV configuration represented the optimum cargo vehicle concept for pairing with the PLS HL-20 on the proposed booster concepts. The estimated CTRV launch weight of 99,000 lbs was just within the 100,000 lb limit study groundrule and thus represented the largest CTRV configuration possible for launch on the study's candidate expendable boosters.

The Single Development Winged CTRV concept was generated to evaluate how the Winged CTRV concept would perform in a combined crew/cargo payload mission, thereby eliminating the need for a separate vehicle for crew delivery and return. (This concept provided a trade study configuration for comparison with the HL-42 crew/cargo vehicle concept, but with a larger payload capability.) The forward fuselage of the Winged CTRV was replaced with the biconic PLS concept to allow crew to be carried on top (for launch abort) and the full 42,500 lb cargo payload to be carried in the payload bay. The biconic PLS concept included the full launch escape system as defined for individual PLS launches and a modified adapter for mating with the CTRV mid fuselage. This concept's 125,000 launch weight violated the 100,000 lb launch weight limit. The CTRV wing span was not increased in this configuration, choosing instead to let the entry heating rates and the landing angle of attack to increase. Leading edge TPS temperature limits were not exceeded and the angle of attack at landing increased from 21° to 26°.

A means for providing late access to pressurized payload volumes was also imposed as a requirement for the CTRV concept. This requirement is based on the current Space Shuttle mid-deck locker payload service which allows last minute loading of refrigerated (or even live) payloads to be installed on the launch pad. This type of payload has been identified as a specific payload design requirement for the CTRV. Since the winged CTRV does not contain an integral pressurized volume for such payloads, some other method of delivering these payloads is necessary. A downsized version of the Spacehab module was selected as the design solution for this requirement. The Spacehab module is designed especially for this type of payload, providing the Shuttle with the capability to deliver as much as 61 additional locker payloads. For the CTRV, a smaller version of the Spacehab would satisfy the mission needs (although the CTRV can deliver a Spacehab in its current configuration as well). Downsizing the Spacehab to a 12-foot diameter by 7-foot long module will permit installation of 42 lockers. Access to the new module can be provided through the CTRV forward fuselage while on the launch pad (or on the runway) without opening the payload bay doors. Access doors in the CTRV forward payload bay bulkhead and the sidewall will permit launch personnel to open the module door and install locker payloads. After launch and docking with the Space Station, the module would be attached to a pressurized port and utilized as a mini-"closet" module.

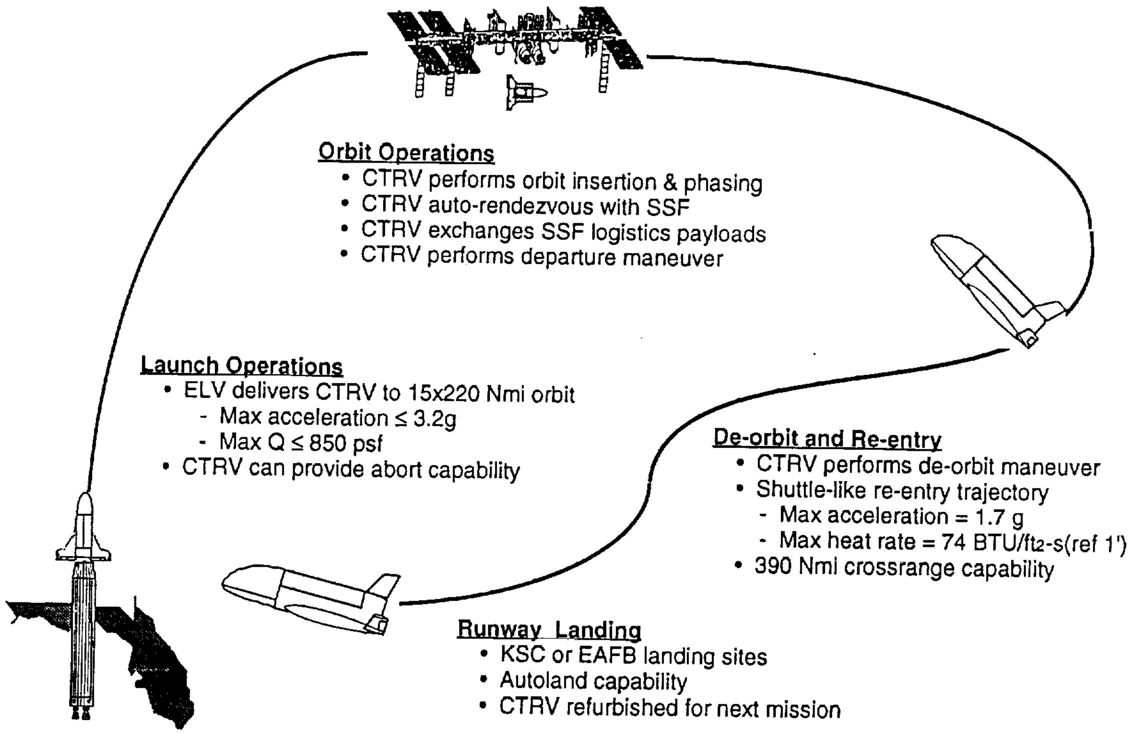


Figure 2.2.4-7 CTRV Mission Profile - SSF Logistics Resupply Mission

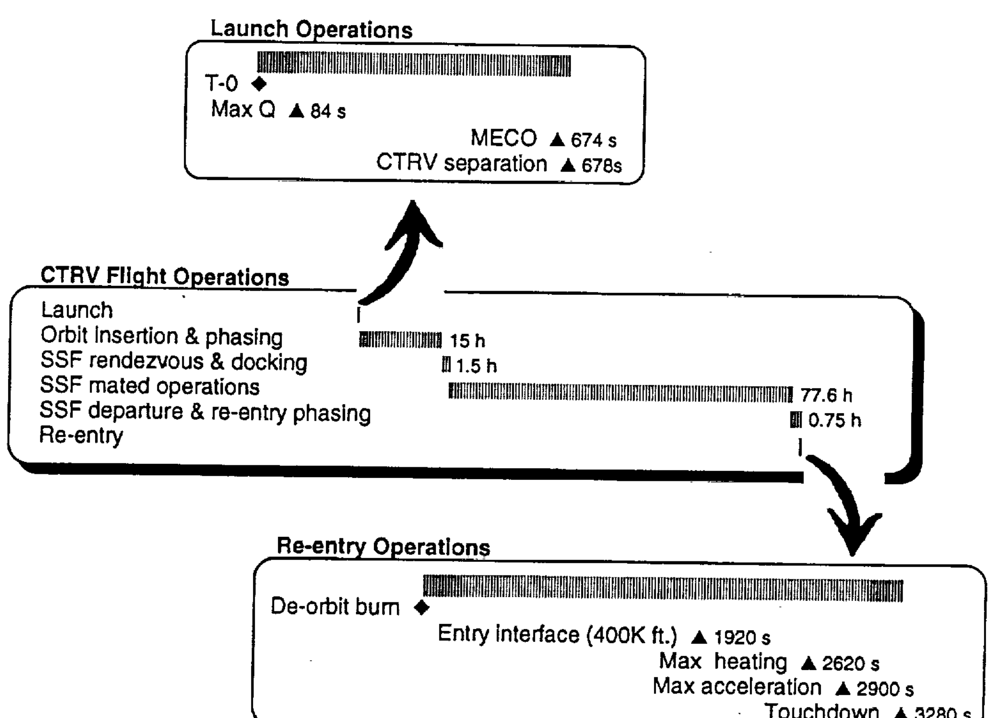
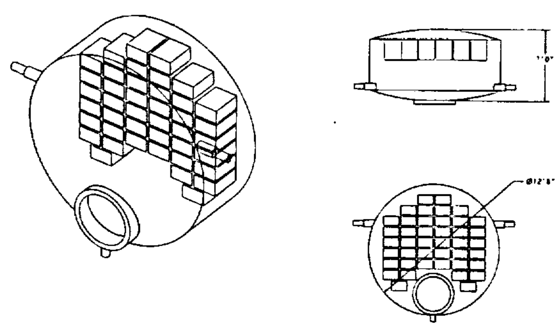


Figure 2.2.4-8 CTRV Mission Timeline-96 Hr SSF Logistics Resupply Mission

- Lockers provided via pressurized "Suitcase" module
  - Downsized Spacehab module
  - Sized for 24 "mid-deck" lockers
  - Self-contained electrical power (battery)
- CTRV provides access doors to Suitcase module
  - Vertical (launch pad)
  - Horizontal (runway)
- Module transferred to SSF after docking



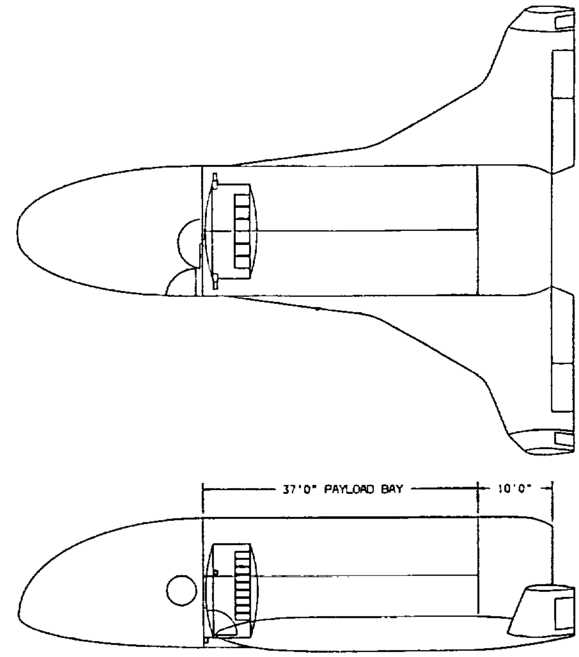


Figure 2.2.4-9 CTRV Late Access Capability - Locker Payloads Delivery

2.2.4.3 Structures Analysis & Weight Estimates

Weight estimates for the CTRV were based on the latest subsystem design definitions as provided by NASA MSFC engineers. Structural weight estimates were based on the stress analysis results from the Medium CTRV (the parachute landing version) and then scaled to the Winged CTRV dimensions. A weight reduction of 25% was applied to all fuselage and wing primary structure weight estimates based on the results of the NASTRAN analysis of Medium CTRV flight loads. This structural weight reduction was offset by some increased subsystem weights and the result was a launch weight for the Winged CTRV (including launch vehicle adapter) of just under 100,000 pounds. The CTRV thermal protection system (TPS) weight was determined from TPS tile thicknesses as sized from the calculated heating rates, in turn generated from re-entry trajectory analyses. Calculation of the CTRV center of gravity (CG) for both launch and landing conditions was also calculated.

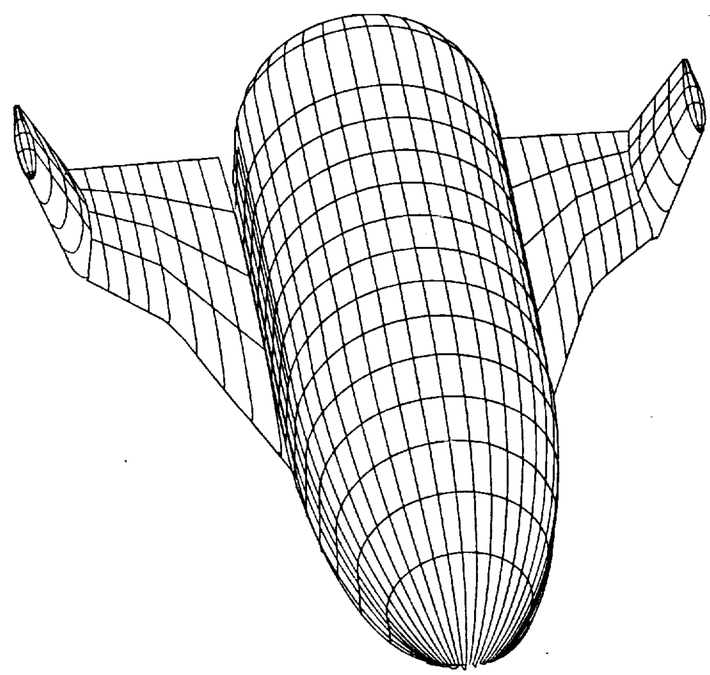
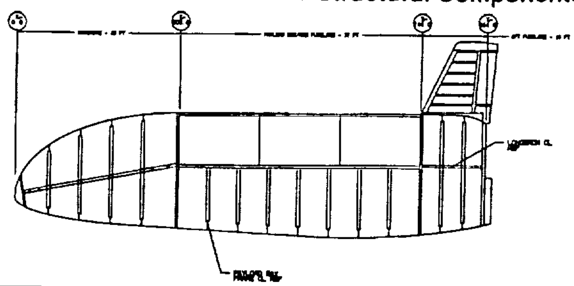


Figure 2.2.4-10 Winged CTRV Perspective View

CTRV Structural Components



Subsequent analyses will identify locations for purge ducts and vent doors to satisfy environmental control requirements.

Structure	Weight (lb)
Forward fuselage	6,000
Mid fuselage	7,190
Aft fuselage	4,250
Wings	5,518
Payload bay doors	2,770
Door hinges/latches	1,140
Vertical tail	875
TOTAL	27,743

Environmental Control	Weight (lb)
Purge ducts	106
Vent doors	598
TOTAL	704

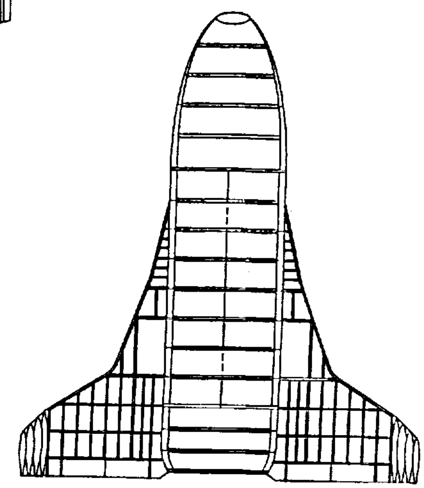


Figure 2.2.4-11 CTRV Structural Components

The CTRV is generally of conventional aluminum construction protected by reusable surface insulation.

- Forward fuselage: Composed of 2024 aluminum alloy skin/stringer panels, frames, and bulkheads.
- Mid fuselage: Includes the wing carry-through structure and the payload bay. Composed of 2124 aluminum alloy integral machined panels and honeycomb sandwich panels.
- Aft fuselage: Composed of 2124 aluminum alloy skin/stringer shell.
- Wings: Composed of 2024 aluminum alloy. Uses corrugated spar web, truss-type ribs, and riveted skin/stringer and honeycomb covers.
- Vertical tail: Composed of 2124 aluminum alloy construction consisting of a two-spar, multi-rib, integrally machined skin assembly.
- Payload bay doors: Graphite epoxy frames and honeycomb panel construction. Hinged along the side of the mid fuselage and split at the top centerline.

Figure 2.2.4-12 CTRV Structure

2.2.4.4 Aeroheating Analyses & TPS Sizing

Preliminary sizing of the Winged CTRV thermal protection system (TPS) was calculated from estimated heating rate distributions over the vehicle. The TPS sizing and weight estimates were based on flying a moderate angle of attack (25°) trajectory, as demonstrated in POST trajectory analyses of the CTRV configuration. These trajectories resulted in moderate heating rates to most of the lower surfaces. Heating rate distributions over the fuselage surfaces were estimated from analyses performed on the Medium CTRV configuration. A carbon-carbon TPS material was assumed required for the entire wing leading edge span.

A shock layout (position vs. mach number) was prepared for the Winged CTRV configuration to determine whether the bow shock would impinge on the wing tips during maximum heating and maximum dynamic pressure conditions of the trajectory. The layout showed that the shock wave would not reach the wing's vertical stabilizers (tip fins) until after maximum dynamic pressure (mach 5.0). The shock position during maximum heating was well inboard of these surfaces.

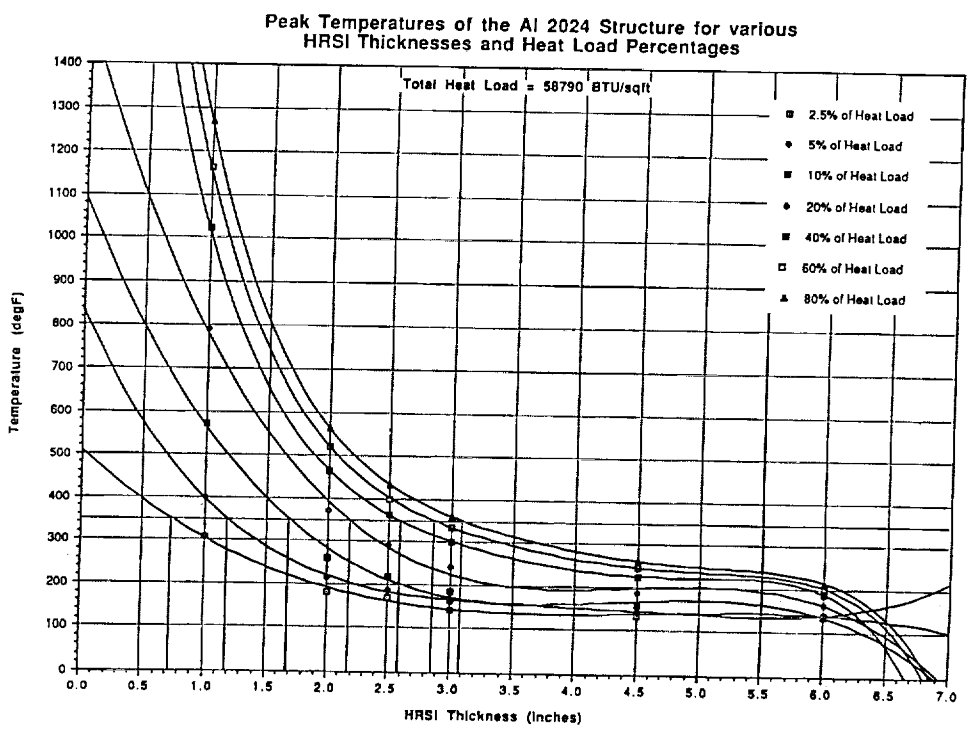
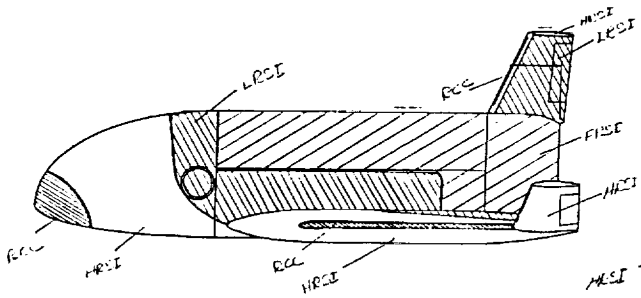


Figure 2.2.4-13 Peak Temperatures for the Al 2024 Structure



TPS Material	Area sq. ft.	Weight lb
RCC	143	1,307
HRSI	2,124	5,276
LRSI	1,611	1,760
FRSI	1,817	596
Misc	-----	276
TOTALS	5,695	9,215

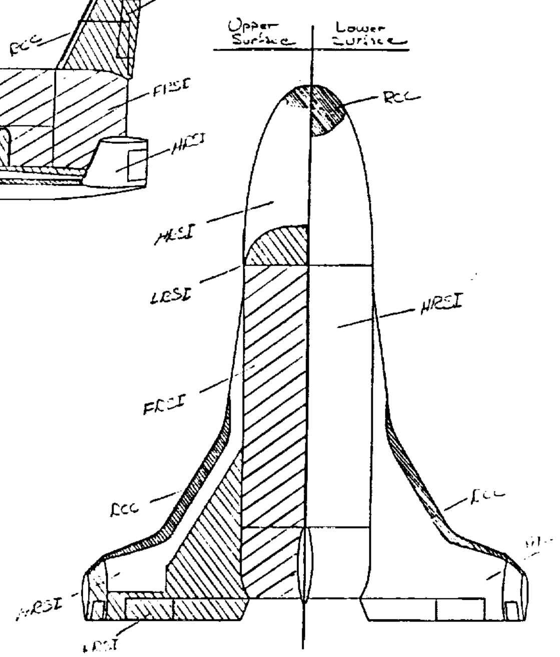


Figure 2.2.4-14 TPS Material Location

2.2.4.5 Trajectory Analyses

Trajectory analyses of the selected Winged CTRV configurations were performed with the POST trajectory simulation tool. The analyses were performed for all versions of the CTRV and included the complete re-entry trajectory, from entry interface (400,000 ft altitude) down to the runway threshold (50 ft altitude directly above the runway). The landing flare was performed with a 1.5 g pull-up from the -18° flight path angle (outer glide slope) to the final glide slope angle of -1°. This maneuver was successfully performed with a low speed L/D ratio as low as 3.5, but the landing angle of attack was too high (30°) for good controllability. The CTRV wing aspect ratio was increased until the L/D reached 4.0 to improve the landing characteristics.

Improved re-entry trajectory simulation techniques from those used in the Medium CTRV analyses were required to more accurately predict the heating environment for CTRV's wings. Lacking detailed aerodynamic coefficients for the CTRV (aerodynamic control surface trim data), previous trajectory designs were based on guidance schemes which optimized the angle of attack and kept fixed bank angles. By estimating control surface gains (from Shuttle 6-DOF trajectory simulations), a bank angle steering guidance mode was added to the CTRV's 3-DOF trajectory simulation. This permitted both angle of attack and bank angle profiles to be optimized by POST for a minimum heat rate trajectory. This improved trajectory capability permitted the Winged CTRV concept to fly constant heat rate and constant drag profiles during the re-entry (this is the basic Shuttle trajectory approach). Maximum heating rates for the wing leading edges were reduced by a factor of almost two in these trajectories. The improved trajectory analysis eliminated the need for a re-design of the CTRV wing based on heating rates.

An analysis of the impact footprint for an uncontrolled CTRV re-entry was also performed. The analysis objective was to determine just how large an area of populated land mass might a re-entry vehicle such as the CTRV pose a danger to if system failures occurred during the entry phase of flight. There are many failure modes which might cause a re-entry safety hazard (including both uncontrolled and controlled re-entry) which may or may not lead to structural breakup of the CTRV. System failures such as loss or degradation of GN&C systems, control surface(s) malfunction, TPS failures, and loss of electrical power can occur at any time in the re-entry trajectory. The effects of these failures range from a missed landing approach, to uncontrolled flight and structural breakup of the vehicle. Any prediction of the vehicle (or debris) footprint is highly subjective due to the high degree of variability in potential failure effects and times. Key trajectory parameters for the vehicle/debris are also highly variable and must be estimated for such criteria as debris mass & drag, velocity vectors, and time of vehicle breakup.

In order to obtain an estimate of the potential impact area, the CTRV vehicle (intact) trajectory was run to its geographic limits (cross range and downrange limits) without normally imposed heating and acceleration constraints. This technique assumes that the greatest lift and drag (or L/D) coefficients of any debris are less than that of the intact vehicle. The resulting trajectory footprints thus include maximum range dispersions of the intact vehicle under off-design heating and acceleration conditions.

The results of this analysis showed that the potential debris impact footprint included most of the United States (from Hawaii to San Francisco, Chicago, Washington DC., and Florida) and the entire upper half of Mexico. This result indicates that a CTRV concept should have sufficient redundancy in flight critical systems to ensure that the vehicle can be guided to a controlled impact area in the event the primary or secondary landing sites cannot be reached due to system failures.

Also considered in the failure scenario was the effect of system failures during the communications blackout period of the re-entry trajectory. This blackout period normally prevents vehicle communications with ground controllers because of highly ionized gasses surrounding the vehicle during portions of the re-entry. For a vehicle such as the CTRV, this period would be expected to last for approximately 400 seconds at an altitude of 250,000 to 200,000 feet (Mach 24 to 15), a region which is approximately 4,000 Nmi upgrange from the landing site. The CTRV is always aerodynamically stable during this blackout period, and would trim at an angle of attack of 55° without any control surface inputs required. This angle of attack produces a much less than maximum Lift/Drag ratio. Consequently, if a failure occurred at this point in the trajectory and no further control surfaces commands were issued to the vehicle (from on-board or ground sources), the vehicle would land well short of the targeted landing site. Specifically, it would land about 2,000 Nmi short of the landing site, which is a water impact off the coast of the US. or Mexico even for a KSC primary landing site. This condition thus results in a somewhat fail safe trajectory (that is, for several system failure scenarios) for the CTRV during this blackout period.

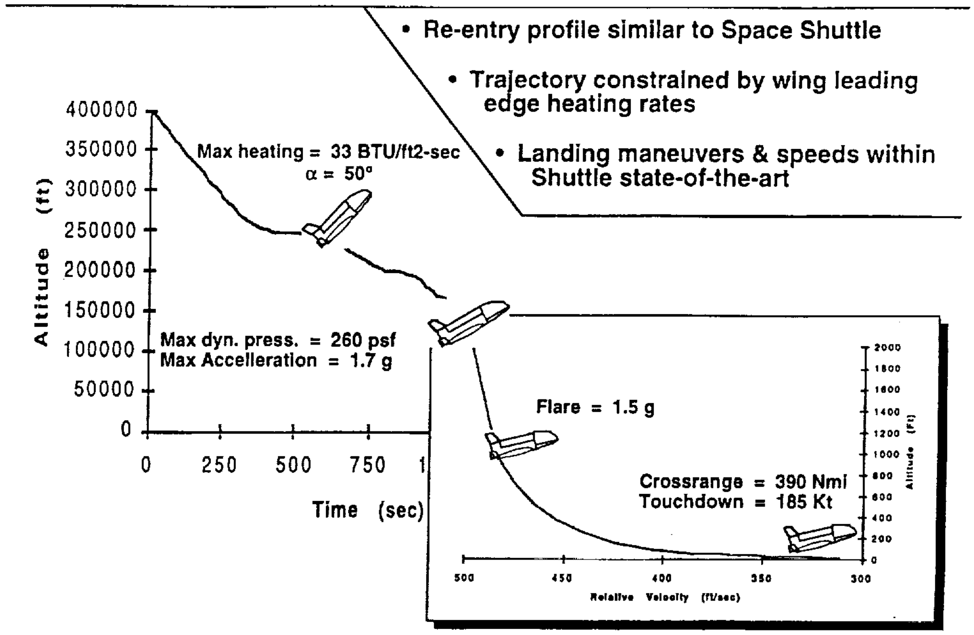


Figure 2.2.4-15 CTRV Re-entry Performance

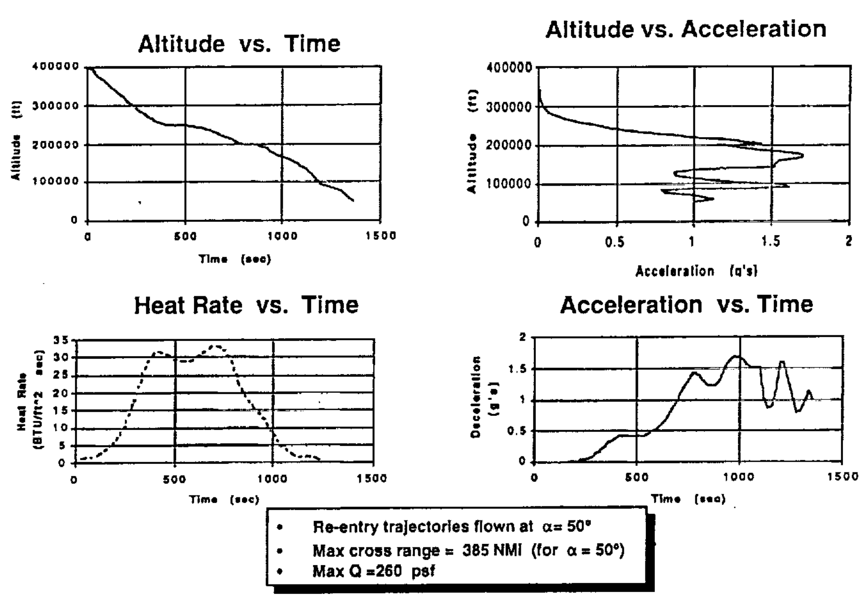
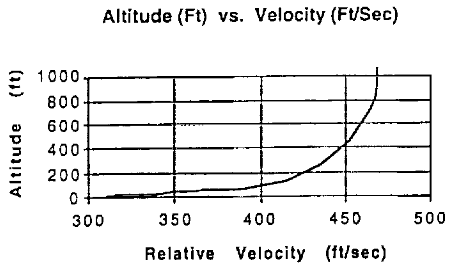
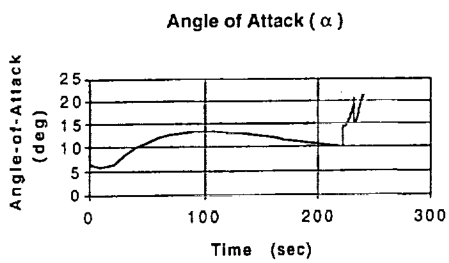


Figure 2.2.4-16 Trajectory Data - Hypersonic Regime



- Outer glideslope $\gamma = -19°$, inner $\gamma = -1°$
- Initial flare maneuver at 1.5 g's
- Landing flare maneuver at $\alpha = 6°$ to $14°$
- Touchdown at $\alpha = 21°$, V = 185 Kt



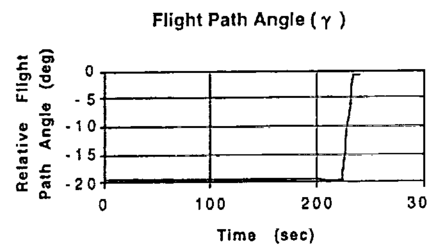


Figure 2.2.4-17 Trajectory Data- Landing Maneuver

Issue

What failure modes must be considered for re-entry of an um-manned spacecraft over populated land areas ?

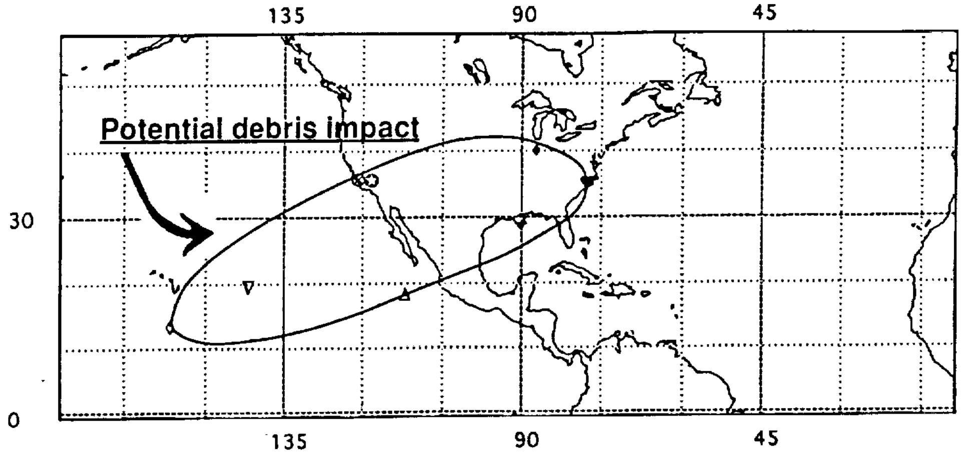
Failure Modes

- Loss or degradation of GN&C
- Control surface(s) malfunction
- TPS failure
- Structural failure
- Loss of electrical power

Failure Effects

- Structural breakup
- Missed landing
- Uncontrolled flight/trajectory
- Environmental contamination
- Civilian damage/casualties

Figure 2.2.4-18 CTRV Flight Safety - Re-entry Failure Modes



Sufficient vehicle redundancy must be provided to permit a controlled impact area to be reached

Figure 2.2.4-19 CTRV Impact Footprint - Failed Edwards AFB Landing

2.2.4.6 Aerodynamic Analyses

Modifying the Integral CTRV into a runway landing system required sufficient low speed lift to counter the high drag caused by the wide and short fuselage. The Winged CTRV aerodynamic design was achieved by adding just enough wing area and stretching the fuselage just enough to reach the desired subsonic L/D ratio of 4.0. This point was reached when the forward fuselage (nose section) was increased to 25 feet long and the wing span increased to 52 feet. The aerodynamic characteristics of this configuration were calculated with the APAS analysis tool at both hypersonic and sub-sonic speeds. Trade studies of the configuration options showed this design to be very sensitive to wing span, which became the key control parameter for adjusting the low speed L/D ratio. A subsonic L/D ratio of as much as 6.0 was generated with wing spans reaching 60 feet. Pitch stability of these higher (L/D > 4.0) lift configurations was found to be very good and the wing loading was found to be low compared to the current Space Shuttle design.

An analysis of the aerodynamic loads caused by several of the competing CTRV concepts on their launch vehicles was also performed. The analysis evaluated bending moments and static stability margins for typical boosters for this class of payload at maximum aerodynamic pressure (max Q) conditions (an NLS-2 booster was used for the launch trajectory conditions). The analysis showed that the HL-42 concepts produced moderate launch vehicle bending moments at max Q, but the CLV concepts imposed high launch vehicle bending moments and associated large booster engine gimbal offsets. Winged CTRV concepts produced only low aerodynamic moments because of their relatively low-lift wings and a low normal force coefficient at the 5° angle of attack condition at max Q. The Winged CTRV wings were designed expressly to minimize the booster's max Q loads, but had to make up for this design with higher aspect ratio wings for landing. The CLV wing designs were directly scaled from the Shuttle Orbiter (cambered) and thus were subject to the higher lift loading at the max Q condition.

Much of the analysis of the CTRV wings focused on the design's hypersonic heating issues. The initial CTRV configuration was selected on the basis of aerodynamic characteristics in the low speed (landing) regime. The re-entry and hypersonic aerodynamic characteristics of this configuration were analyzed to identify those configuration changes necessary to obtain satisfactory heating rates. The aerodynamic heating was initially found to be extremely high on the outboard sections of the wing leading edges (due to the low sweep angle and smaller leading edge radius). Heating rates on both the wing leading edges and the CTRV nose are limited by the surface temperature limit of the advanced carbon-carbon leading edge material. Alternative re-entry trajectories were attempted at various angle of attack profiles, but all were found to produce high heating rates. Analysis of the bow shock location on the wing during re-entry also indicated too high heating rates (bow and wing shock interaction effects).

Several wing re-design options were evaluated, including changes to the wing sweep angle, deployable wing sections, increased wing planform area (extended wing chine areas), and supercritical wing sections. Increasing the wing planform area produced higher drag during the high angle of attack re-entry (α=50°), which reduced the maximum heating rates but also added considerable structural weight. Use of supercritical wing sections would result in an increase in the wing thickness, and would increase the wing leading edge radius by a factor of about two greater than the current wing section. This would reduce heating (by a factor of $\sqrt{2}$) but reduce the low speed lift/drag ratio. The most promising options appeared to be a change to the outboard wing section sweep angle (from 20° to 30°) or deployable outer wing sections. The increased sweep angle produced some reduced heating effects for little area (weight) increase. A deployable wing section provided the best solution for both high speed and low speed performance but would only prove weight effective if the weight of the hinge and deployment mechanism could be offset by a reduced carbon-carbon leading edge area. That is, the deployed wing section would have to not require carbon-carbon leading edges while in the swept-back position. Aerodynamic coefficients were calculated for several increased wing area options and for the deployable wing configuration option. Re-entry trajectory analyses were then performed for each configuration to determine the resulting heating rate reductions. Eventually, the re-entry trajectory simulations were improved enough (see Section 2.2.4.5, Trajectory Analysis) to lower the predicted heating rates to acceptable conditions and no wing redesign was actually required.

Aerodynamic stability analysis of the winged CTRV was performed to determine directional stability characteristics. A combination of various winglet sizes, nose cone shapes and vertical stabilizer sizes were evaluated to determine which approach would provide positive stability margins at the lowest weight. The analysis results showed that retaining the current winglet size and adding a vertical stabilizer of approximately 100 square foot area would provide a positive stability margin ($C_{\eta\beta}$ = 0.008/deg at M=0.3). A single vertical stabilizer mounted at the top of the aft fuselage section of the vehicle was chosen as the design reference. Design configuration drawings and weight estimates for the CTRV were updated to reflect the stabilizer.

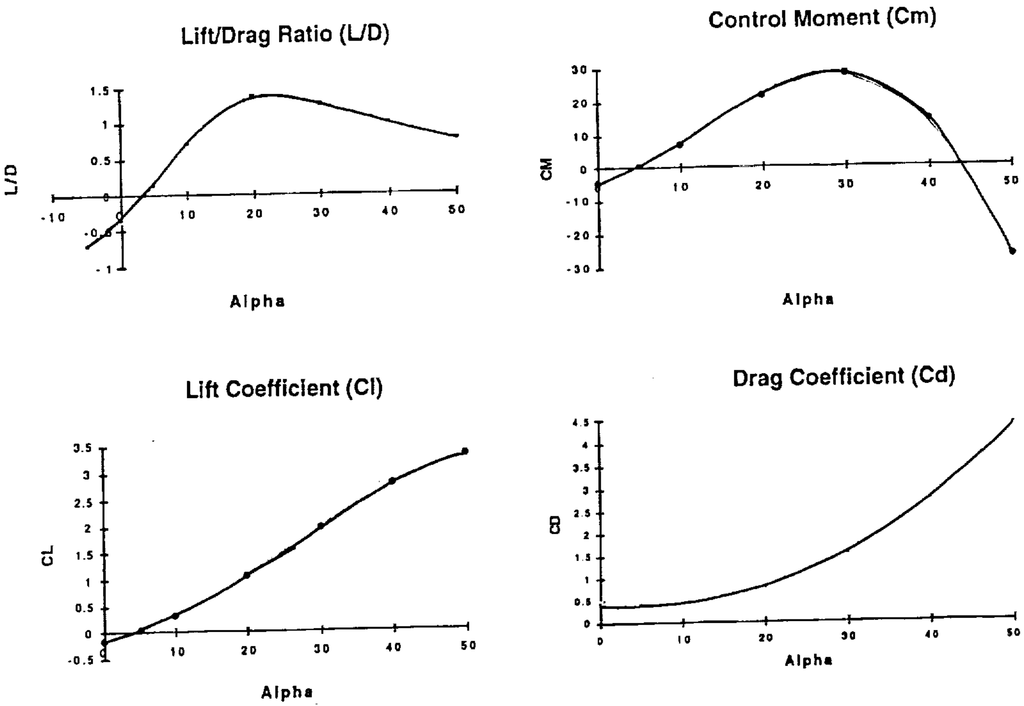
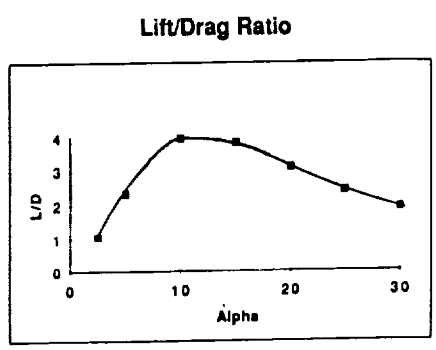


Figure 2.2.4-20 Aerodynamic Data - Hypersonic Regime



- Landing flare maneuver at $\alpha = 6°$ to $14°$
- Touchdown at $\alpha = 21°$, Vr = 185 Kt
- Wing loading = 67 lb./sq. ft.

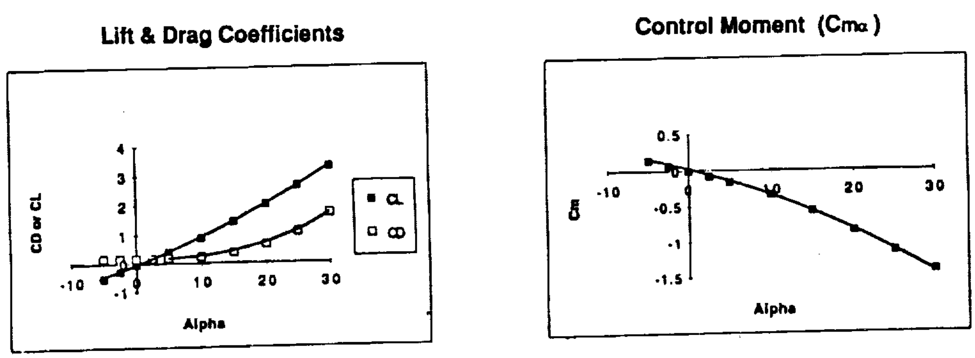
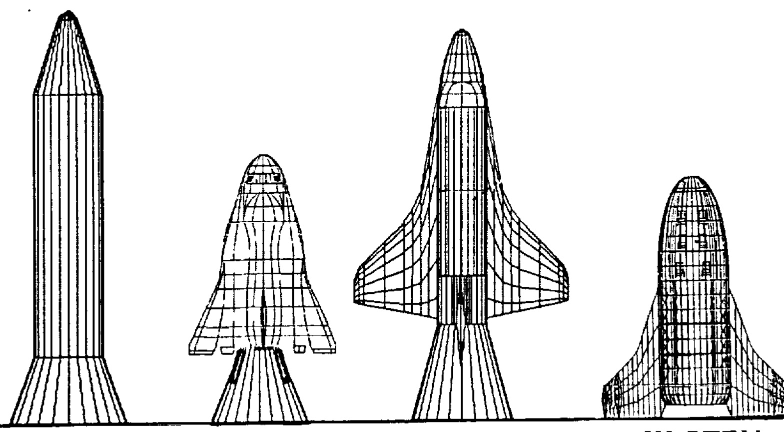


Figure 2.2.4-21 Aerodynamic Data - Low Speed Regime



	NLS	HL-40	CLV	W-CTRV
Max Q (psf)	728	777 est.	777	777 est.
Bending Moment (ft-lb x 10*6)	6.6	24.6 est.	20.6	12.4 est.

Figure 2.2.4-22 CTRV Induced Airloads - Comparison of Loads on NLS

2.2.4.7 Subsystems Definition

Subsystem definitions and weight estimates were updated from the Medium CTRV concept to reflect the final Winged CTRV design configuration and also to obtain maximum commonality with the PLS (HL-20) subsystems design. During coordination meetings with NASA JSC and LaRC personnel, it was decided that all of the CTRV concepts (winged CTRV, HL-42, and CLV) should be based on a common avionics system definition. Differences in the CTRV concepts for vehicle lengths and cockpit displays would have minor effects on most subsystems weights (wire harnesses, etc.). A review of all systems was conducted with NASA LaRC to identify where the CTRV subsystems could be common with the PLS (HL-20) subsystems definitions.

Achieving commonality of the Winged CTRV's Orbital Maneuvering System (OMS) and Attitude Control System (ACS) with the PLS represented a significant design trade study between the CTRV'S hypergolic propellants and alternative propellants such as the hydrogen peroxide/RP-1 propellants of the PLS. Studies performed by MSFC engineers showed only a minor weight and volumetric penalty for the CTRV between a hypergolic propellant system (MMH and N2O4) and the hydrogen peroxide (H2O2) system. An assessment of the launch processing impacts resulting from hypergolic propellant systems servicing was performed to identify how much of a savings might be realized if the H2O2 system was used. The assessment, based on actual Shuttle processing of OMS and RCS systems, revealed that hypergolic systems processing typically resulted in six shifts of serial, hazardous operations. This included removal and re-installation of a hypergolic propulsion module in the OPF, and fueling of the hypergolic systems at the launch pad. (Actual servicing of the Shuttle hypergolic system components is performed off-line in a controlled facility removed from the main system processing. This approach would be recommended for any propulsion system to allow high pressure testing of propulsion components.) Determination of the safety support requirements (e.g. SCAPE operations support equipment and personnel, fire trucks, etc.) were also identified for these hazardous operations.

Additional impacts to KSC operations of a hydrogen peroxide system were identified based on the special characteristics of H2O2. Although this propellant is not toxic, it is unstable and requires strictly controlled storage conditions. During trade studies of this propellant option under the PLS program, it was found that there are currently no production facilities in the United States or Europe for propellant grade H2O2 and there are no storage facilities at KSC or CCAFS for any quantities of this propellant.

SERIAL OPERATIONS IMPACTS
Hypergolic Propellant Systems

	Serial Impact (Shifts)
OPF Operations (1)	
• System removal (e.g. 1 each OMS pod)	1
• System installation (e.g. 1 each OMS pod)	1
LC-39 Operations (2)	
• Hypergolic propellant loading	4
Total Serial Impact= 6 shifts	

Notes

1. Typical OPF flow includes 6 facility clear operations (11 total shifts) and 39 local clear operations (85 total shifts). OMS pods and Fwd RCS module are not removed every flight, only as required for unscheduled maintenance actions.

2. Typical LC-39 hypergolic propellant loading is a scheduled 36 hour process with 31 hours of "pad clear" operations. During this 31 hour period, a total of 17 propellant tanks are filled in parallel operations (includes OMS, RCS, APU, and HPU systems propellant loading and pressurization)

Figure 2.2.4-23 Operations Impacts - Hypergolic Propellant Systems

OMS/RCS PROPELLANT TRADE
POTENTIAL DISCRIMINATORS

MMH/N2O4 HYPERGOLICS	JP-4/H202
• MONOMETHYLHYDRAZINE(MMH) - CAUSTIC, LOCALLY DAMAGING TOXIC AGENT & HIGHLY FLAMMABLE - PROVEN WELL UNDERSTOOD SAFETY PROCEDURES IN PLACE • OPF*: MINOR SPILL (DROP, <1/2 CUP) DRIVES "CLEAR" AREA - EVACUATE OPF BAY (100 - 200 PEOPLE) - UP TO 1/2 SHIFT CLEAN-UP, "SCAPE" CREW • OPF*: MAJOR SPILL (> 1/2 CUP) - EVACUATE OPF BAY 1 & 2 (200 - 400 PEOPLE) - EVACUATE ANNEX OFFICES (~ 100 PEOPLE) - UP TO 2 SHIFT CLEAN-UP "SCAPE" CREW • N2O4 REQUIRES SAME PROCURES, DIFFERENT SPILL KIT	• HYDROGEN PEROXIDE (H2O2) - UNSTABLE, SUSCEPTIBLE TO HEAT & CONTAMINATION - STRONG IRRITANT - NON FLAMMABLE, BUT ACTIVE OXIDIZER REACTING WITH FLAMMABLE MATERIALS • OPF*: MINOR SPILL (DROP, <1/2 CUP) DRIVES "CLEAR" AREA - EVACUATE AFFECTED AREA SMALLER AREA? FEWER PEOPLE? - WATER DELUGE CLEAN-UP - SIMILAR CLEAN-UP, BREATHING APPARATUS • OPF*: SPILL PROCEDURES SPECIFIED BUT NOT IN PLACE • NO EXISTING MANUFACTURING FACILITY (REFINERY) FOR 90 + % H2O2 • NO EXISTING STORAGE FACILITY FOR H2O2 AT KSC/CCAFS

* OMS/RCS TANKS PURGED PRIOR TO ENTRY INTO ORBITER PROCESSING FACILITY (OPF)
SCAPE: SELF CONTAINED ATMOSPHERIC PROTECTIVE ENSEMBLE
REFERENCES: AFM 161-30, VOL. LIQUID PROPELLANTS ; GP 1098-F, KSC GROUND OPERATIONS SAFETY PLAN

Figure 2.2.4-24 OMS/RCS Propellant Trade - Potential Discriminators

2.2.4.8 Reliability Analyses

Reliability and maintainability analyses of the Winged CTRV concept were performed to reflect the subsystem design changes (including the addition of the CTRV's wings and aerodynamic control surfaces). The effect of these relatively more complex subsystems of the Winged CTRV than the Medium CTRV concept had a dramatic effect on the CTRV's launch operations simulations. The reliability of all major components in each subsystem was calculated by our MAtrix program per the current subsystems weight and mission duration. These CTRV component reliability estimates were used to calculate maintenance requirements (MTBF, MTBR, ...) at the LRU level. The increase in subsystems complexity, and thus increase in maintenance actions (failures per flight increased from 2.1 to 6.5) was simulated in the SIMtrix maintainability model. This produced a logistics delay factor of 21 times the MTTR, a significant increase from the 8X factor associated with earlier (and simpler design) Medium CTRV reliability estimates. This data was in turn used to update the STARSIM simulation of integrated CTRV, PLS, and NELV launch processing. It was found that a spares level of 95% would be required to meet the CTRV and PLS launch rates (rather than the 90% level previously required).

The SIMtrix maintenance simulation tool was then used to establish a recommended spares quantity for each LRU based on the predicted reliability and maintenance data generated from MAtrix. The recommended spares level for most LRUs was only one each except for high quantity components such as thrusters (36 total per vehicle, 6 each spares recommended). The estimate of the required spares levels for the CTRV was used to support CTRV cost exercises.

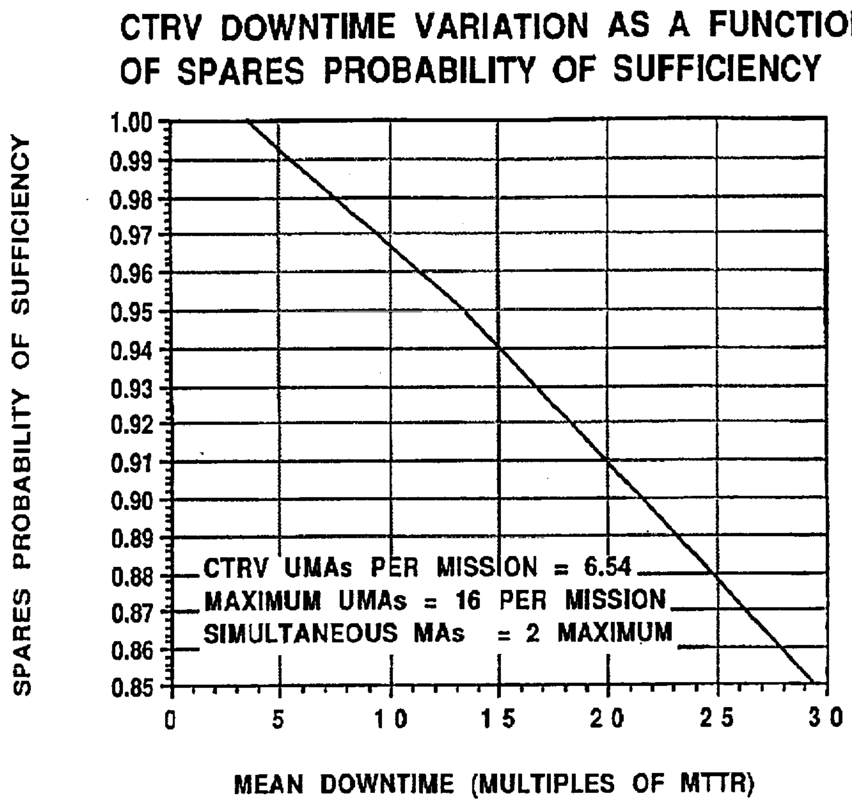


Figure 2.2.4-25 CTRV Downtime Variation

WINGED CTRV SPACECRAFT SOURCE: WCTRV WEIGHT STATEMENT OF 6/4/93	QTY	UNIT WT (lbs)	TOTAL WT (lbs)	LRU?	MTBR (Flt Hrs)	RECOMMENDED SPARES QTY (POS = 0.90)
ATTITUDE REACTION CONTROL & OMS	-	-	1598	NO	56	N/A
TANK (MMH + NTO)	2	502.50	1005	NO	3,027	N/A
THRUSTER (FORWARD)	18	6.10	110	YES	176	3
THRUSTER (AFT)	18	6.10	110	YES	176	3
PLUMBING	1	80.00	80	NO	968	N/A
THRUSTER (OMS)	1	35.00	35	YES	552	3
PLUMBING, VALVES, ETC.	1	258.00	258	NO	300	N/A
AVIONICS	-	-	695	NO	361	N/A
GUIDANCE, NAVIGATION AND CONTROL	-	-	435	NO	497	N/A
IMU, HEXAD	1	55.00	55	YES	6,231	1
GN&C COMPUTER	2	10.00	20	YES	17,134	1
GPS RECEIVER/PROCESSOR	2	9.00	18	YES	19,037	1
FLOODLIGHT, PTZ	2	6.00	12	YES	999	1
CAMERA, PTZ	1	78.00	78	YES	4,393	1
CONTROLLER, GIMBAL DRIVE	1	25.00	25	YES	61,965	1
CONTROLLER, RCS	1	33.00	33	YES	46,942	1
CONTROLLER, ACTUATOR	1	40.00	40	YES	38,723	1
ALTIMETER	2	5.00	10	YES	34,267	1
RF ASSEMBLY, MSBLS	2	7.00	14	YES	24,479	1
RECEIVER/DECODER, MSBLS	2	21.00	42	YES	8,159	1
TRANSDUCER, AIR DATA	2	19.00	38	YES	9,018	1
SENSOR, AIR DATA	2	25.00	50	YES	6,853	1
COMMUNICATIONS AND TRACKING	-	-	260	NO	1,318	N/A
POWER AMPLIFIER, RF	2	6.00	12	YES	28,557	1

Figure 2.2.4-26 MAtrix Model Spares Recommendations

2.2.4.9 Cost Analyses

The design complexity and percent new design cost factors for the CTRV concept were provided to MSFC for cost estimation activities by NASA. Definition of a scale for these cost factors was provided to JSC and to LaRC to permit a common costing approach among the competing CTRV concepts (Winged CTRV, PLS, HL-42, and the CLV). Using these definitions, cost factors for design complexity and for percent new design of the CTRV (as well as for the CLV, PLS, and HL-42) were established and coordinated with the NASA centers. This approach allowed a common cost basis for comparing the relative benefits of these concepts with respect to each other, without suffering from different programmatic or design groundrules which may be preferred for the individual concept programs (or governing NASA center).

The CTRV subsystems were evaluated to establish DDT&E cost factors in support of CTRV cost estimating efforts. The subsystems were compared to similar current (Shuttle or other space system) subsystem designs to establish relative new design and design complexity factors. As the Shuttle design experience provides the only re-usable space system with an available cost database, most of the factors are based on Shuttle comparisons.

SUBSYSTEM	D&D PERCENT NEW DESIGN	D&D DESIGN COMPLEXITY	FH DESIGN COMPLEXITY
STRUCTURE FWD BODY (NASCOM MANNEI	60%	1.00	1.00
STRUCTURE MIDBODY (NASCOM MANNED)	20%	1.00	1.00
STRUCTURE AFT BODY (NASCOM MANNED'	60%	1.00	1.00
STRUCTURE WING GROUP(NASCOM MANN	60%	1.00	1.00
PAYLOAD BAY DOOR (NASCOM MANNED)	20%	1.00	1.00
STRUCTUR CREW MODULE(NASCOM MANN	0%	0.00	1.00
RECOVERY (PARACHUTE) (STVCM)	0%	0.00	1.00
LANDING GEAR (NASCOM MANNED)	40%	1.00	1.00
TPS LEADING EDGE (NASCOM MANNED)	20%	1.00	1.00
TPS TILES (NASCOM MANNED)	30%	1.00	1.00
TPS (BLANKETS) (SPECIAL CER)	20%	1.00	1.00
MECHANISM (LVCM)	40%	1.00	1.00
ORBITAL MANUVERING SYS (OMS) (LVCM)	40%	1.00	1.00
REACTION CONTROL SYSTEM(RCS) (LVCM)	40%	1.00	1.00
TANKS (SSCM)	40%	1.00	1.00
GN & C (NASCOM UNMANNED)	50%	1.00	1.00
DATA MANAGEMENT SYS (NASCOM MANNE	30%	1.20	1.00
COMM. & TRACKING (NASCOM UNMANNED	40%	1.00	1.00
INSTRUMENTATION	50%	1.20	1.00
ELEC POWER GEN (BATTERY) (NASCOM UI	40%	1.00	1.00
ELEC POWER GEN (FUEL CELL)(SSCM)	20%	1.00	1.00
ELEC CONV & DISTRIBUTION (SSCM)	60%	1.20	1.00
THERMAL CONTROL (NASCOM UNMANNED	40%	1.00	1.00
AERO SURFACE CONTROL (EMA) (SSCM)	60%	1.00	1.00

NOTE:
D&D= DESIGN AND DEVELOPMENT
FH= FLIGHT HARDWARE (PRODUCTION)

Figure 2.2.4-27 Winged CTRV (WCTRV) (Unmanned Cargo Carrier) Cost Estimating Factors

2.2.4.10 Launch Processing Analyses

Analysis of uncertainty in the predicted Winged CTRV and PLS subsystems turnaround processing times was also examined with the STARSIM model. The CTRV and the PLS subsystems were analyzed using both the predicted fast processing timelines and using current Shuttle subsystems processing timelines. As expected, the PLS and CTRV vehicles occupied their processing facilities for a greater amount of time, but the desired flight rates could be still be achieved even with the Shuttle processing times. The effect of the longer subsystems processing timelines was not as significant as the effect of longer maintenance delays for spares! This demonstrated that the CTRV (and the PLS) reliability and maintainability parameters are at least as important as the launch processing times.

The effect of the subsystem processing timelines on manpower utilization was also demonstrated in these STARSIM simulations. Comparison of manpower expenditure for both the fast (predicted) and Shuttle processing times showed only minor differences. A manpower consumption of 1.60 million man-hours per year was required to process the required 9.5 flights with the fast timelines. The manpower increased only to 1.69 million man-hours for the Shuttle timelines. The small difference is due to the low average facility utilization rates of several key facilities when the faster processing times are simulated. This effect is consistent with staffing practices at KSC in which most direct labor staff are assigned to specific facilities and are not employed on a per flight basis. This demonstrates another important observation about launch processing costs: reducing launch processing timelines does not directly reduce launch processing costs. A better means of reducing direct (and even indirect) labor costs is to reduce the number of processing facilities required (e.g. high utilization of fewer facilities). Launch processing timelines thus need to be balanced among all facilities to produce the most efficient use of resources (just as a production line must balance the quantity and work content of all work stations in the manufacturing flow).

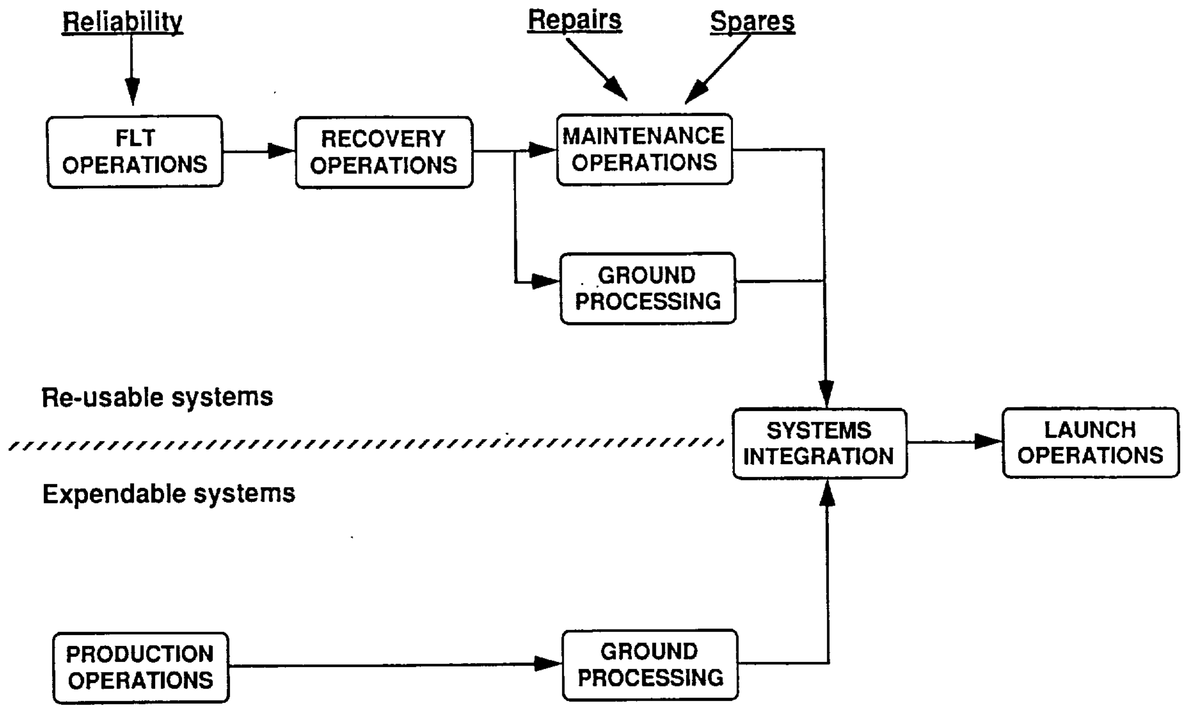


Figure 2.2.4-28 Launch Processing Operations

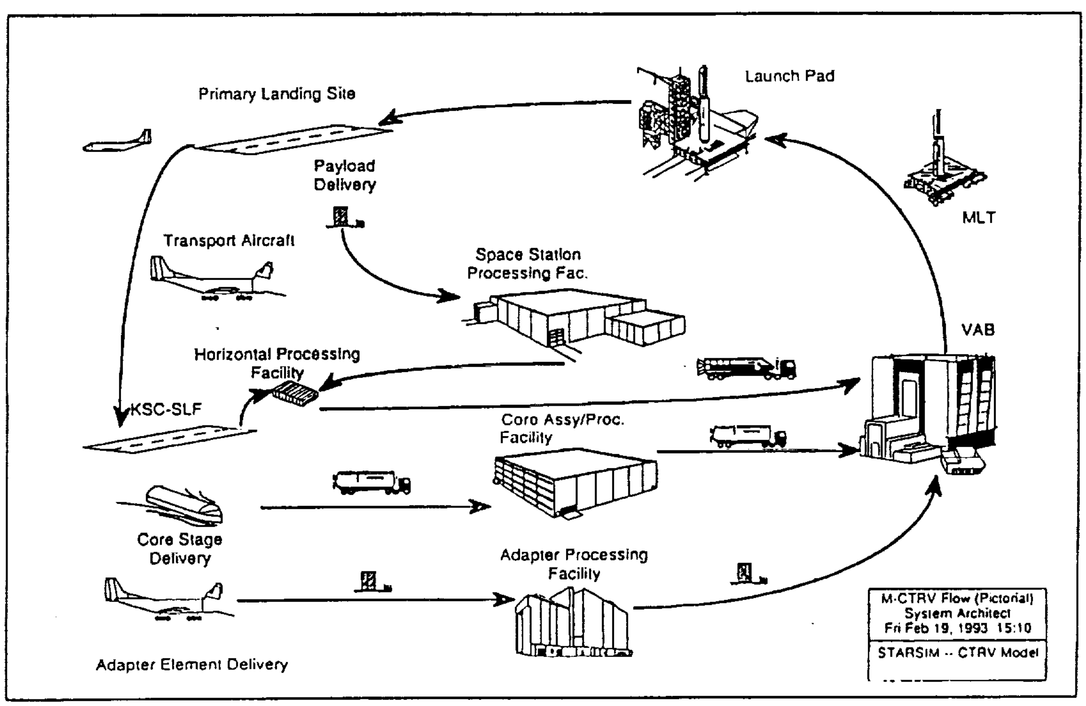


Figure 2.2.4-29 Medium CTRV - Launch Processing Flow

	Aircraft-like Operations	Shuttle-like Operations
Turnaround Times		
CTRV Time in System (Work-Days)	44.4	86.7
PLS Time in System (Work-Days)	58.8	101.0
Launch Interval (Work-Days)	26.4	26.4
Manpower Requirements		
Annual Touch Labor (Million Hours)	1.512	1.690
Facility Utilization		
Landing Facility Utilization	0.042	0.042
HPF Utilization	0.326	0.857
VAB Utilization	0.648	0.935
Launch Pad Utilization	0.389	0.394
MLT Utilization	0.694	1.000
Crawler-Transporter Utilization	0.028	0.375

Figure 2.2.4-30 CTRV Launch Processing Analysis - Simulation Results

• Assumed Staffing Levels	Facility Staff (EP)	Facility Quantity	Total Staff (EP)
OPF	85	3	255
VAB ①	84	2	168
Launch Pad	63	2	126
CA/PF	265	1	265
		Total	814

• Facility Utilization Levels	As Aircraft Processing	As Shuttle Processing
Landing Site	.04	.04
OPF	.33	.82
VAB	.65	.85
Launch Pad	.39	.48
CA/PF	?	?
Derived Manpower Utilization		
- Launch processing functions ②	70%	93%

① - VAB staff must perform launch pad functions also

② - A target utilization of 80% recommended, remainder for facility & GSE maintenance, training, ...

Figure 2.2.4-31 Direct Labor Utilization Estimate

2.2.4.11 Alternative Winged CTRV Concept

One alternative CTRV concept was briefly examined to evaluate potential changes to the CTRV requirements. This concept was the Single Development Winged CTRV. The concept was an attempt to reduce the development and operations costs of two separate airframes for crew and cargo delivery missions. The HL-42 already satisfied this objective, but at a very low payload capability. The Single Development Winged CTRV was used to determine the potential benefits of increasing this payload range.

2.2.4.11.1 Single Development Winged CTRV Concept

In order to reduce the Access to Space "Option 2" operating costs, a means of reducing the combined CTRV/PLS annual flight rate from 9 per year to 6 per year was investigated. The CTRV configuration which enables this cost savings is one which combines the Bi-conic PLS concept with the CTRV (PLS replaces the CTRV nose cone). This configuration was not previously considered because it results in a total launch weight greater than the 100,000 pound limit (study groundrule). The biconic PLS concept as reported by NASA JSC was used essentially intact as the CTRV forward fuselage. The "PLS" configuration includes the escape motors and the Space Station docking port as per a "stand-alone" PLS spacecraft. This approach permits the PLS to be a fully functional system when (if) separated from the CTRV at any time in the mission (launch, orbit, entry). The CTRV configuration was also left essentially intact, retaining the full 42,500 pound payload capability. CTRV subsystem weights were increased to allow for extra redundancy (e.g. FAIL OP/FAIL SAFE for critical functions) associated with the PLS. Life support and other crew related subsystem requirements were included in the PLS weight statement. The combined PLS/CTRV launch weight was 125,000 pounds. While greater than the study groundrules, this weight is within the performance levels of some candidate launch vehicles for the Option 2 architecture and thus could be viable.

Re-entry trajectory analyses were performed for this CTRV configuration to determine the impact of the increased weight (CTRV wing size was not changed). The analyses showed only small differences between the reference CTRV re-entry and the combined CTRV/PLS re-entry. The trajectory profile was essentially identical with only a small increase in the heating rates experienced by the vehicle. The maximum aerodynamic heating rate experienced on the CTRV's wing leading edge was raised from 58 BTU/sq ft-sec to 65 BTU/sq ft-sec. This increase merely used up the design margin which existed in the CTRV, raising the leading edge surface temperature up to its 3100° F temperature limit (same heating rate limit as Shuttle wing leading edges).

Analysis of the heavier CTRV landing maneuver showed the effects of not increasing the wing area. The landing flare maneuver was increased to a 2.0 g acceleration turn (was 1.5 g) and the final flare required a 29° angle of attack (was 21°) to maintain sufficient lift until touchdown. While the landing maneuver could be performed with the existing wing, a slightly larger wing would have brought the landing characteristics closer to conventional performance. Changes in the currently undefined body flap/elevon sizes and deployment schedules would also improve the landing performance of this concept.

- **Deliver & return all SSF logistics payloads, including crew**
- **Runway landing capability**
  - **EAFB crossrange capability**
- **Auto-rendezvous and autoland capability**

Payload volume = 15' dia X 37.5' L
Payload mass = 42,500 lbs. + 6 crew
Launch weight = 125,510 lbs.
Landing weight = 113,660 lbs.
Wing design (64000 series wing)
- 0° incidence angle
- symmetric (no camber, no twist)
- 82°, 60°, 20°, 48°(tips) sweep angles
- 60' span

Vertical stab. (64000 series wing)
- symmetric, supercritical
- 30° sweep

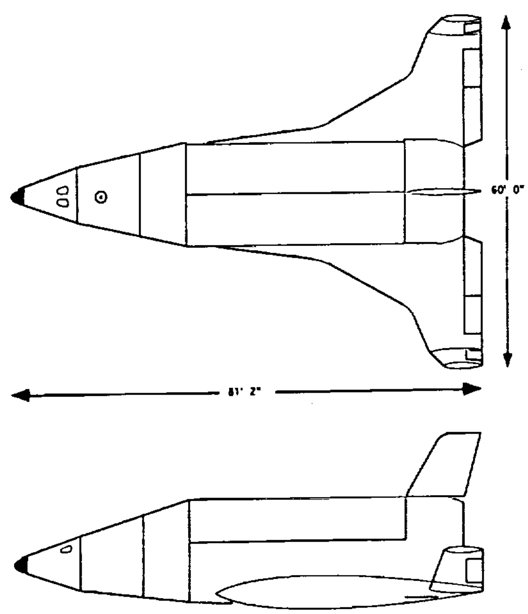
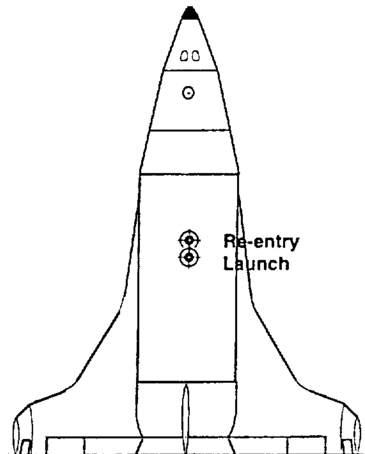


Figure 2.2.4-32 Winged CTRV with PLS Concept

Structure	34,550
TPS	10,600
Active Thermal Control	1,850
OMS/RCS	1,840
Elec. Power Generation	2,150
Elec. Power Distribution	1,900
Avionics	1,220
Purge/Vent	700
Landing Gear	3,590
PLS Abort motors	7,770
Other	1,210
	67,380 Dry Weight
Consumables	
OMS/RCS	9,410
Fuel cell reactant	2,350
Payload	42,500
Crew (2+4)	1,800
Launch veh. adapter	1,800
Total Launch Weight	125,510
Re-entry Weight	113,360



	X	Y	Z
Launch C.G. =	466	0	103
Re-entry C.G. =	445	0	105

Figure 2.2.4-33 Weight Estimate - Winged Medium CTRV with PLS

- Re-entry profile similar to Space Shuttle
- Trajectory constrained by wing leading edge heating rates
- Landing maneuvers & speeds within Shuttle state-of-the-art
- PLS capsule can escape at any point in trajectory

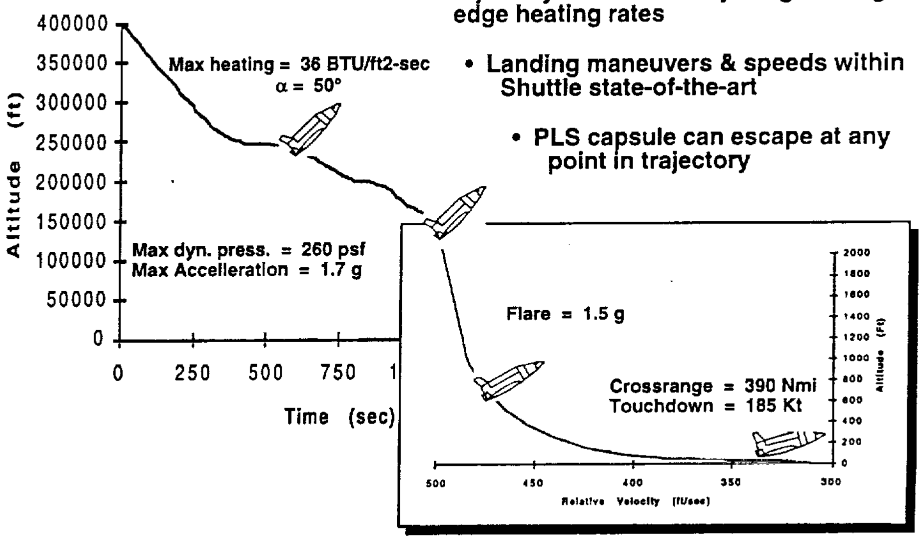


Figure 2.2.4-34 CTRV Re-entry Performance

2.3 Launch Abort Studies

An important element of manned launch systems is the ability to safely perform a launch abort in the event of a failure in a critical system component. This launch abort capability is vital to crew safety considerations and is also a major cost factor for un-manned but fully re-usable launch systems in which the reusable element represents a significant financial investment. The objective of the launch abort function is to return the crew (and/or reusable system element) safely to the ground so that they may be used again. The initial mission objectives of the launch are not retained. The capability to perform the launch abort function is typically provided by both the launch vehicle and the manned element, but neither element providing the full abort capability during the entire mission. Launch abort analysis of manned launch systems must therefor examine the capabilities of both elements over the entire launch mission phase. The abort analyses performed in this study were conducted during the NASA Access to Space study period and utilized launch vehicle and manned element concepts as defined in the NASA work. These launch abort studies directly contributed to the NASA study effort.

2.3.1 Abort Studies Approach

2.3.1.1 Launch System Elements

Launch abort analyses were performed to determine the ability of four launch vehicles to perform a mission abort during the launch portion of the nominal mission (ascent trajectory phase). The launch vehicles selected for analysis were those identified as the most promising from the NASA Access to Space, Option 2 study. These launch vehicles used either the HL-42 or the CLV-P crew/cargo concepts as the system's manned/reusable elements. The analysis was performed for each Option 2 booster as defined by NASA (Boosters 2A', 2C, and 2D for the HL-42, and Booster 2B for the CLV-P). The abort trajectories were analyzed with the POST trajectory software for nominal 15X220 Nmi insertion transfer orbits at both 28.5 and 51.6 degree inclinations. All analyses were based on launch from KSC using 3-DOF trajectories with mean annual KSC winds (peak wind velocity of 102 fps at 36,000 ft). The HL-42 vehicle includes a number of abort solid rocket motors which may be used to perform the abort maneuvers (for both rapid escape from the booster and post-escape velocity addition). These abort motors by themselves permit an abort from the launch pad and for the first 64 seconds of flight. The HL-42 and

CLV-P orbital maneuvering system propulsion systems may also be used for certain abort conditions (when the vehicle is at high altitudes). The HL-42 design also provides an option for emergency water landing by parachute. The CLV-P does not provide this option, but does provide for crew escape (bail out) at lower altitudes and velocities. The abort trajectory analyses were performed for both ascent and entry conditions with POST. During the re-entry/landing portions of the abort trajectories, design heating rate limits were maintained but acceleration limits were relaxed (to 8 g's) for the HL-42 per allowable abort requirements. Specific design characteristics of the booster were provided by NASA MSFC; HL-42 and CLV-P design data were provided by NASA LaRC and JSC, respectively. No predictions of booster stage impact points were attempted during these analyses.

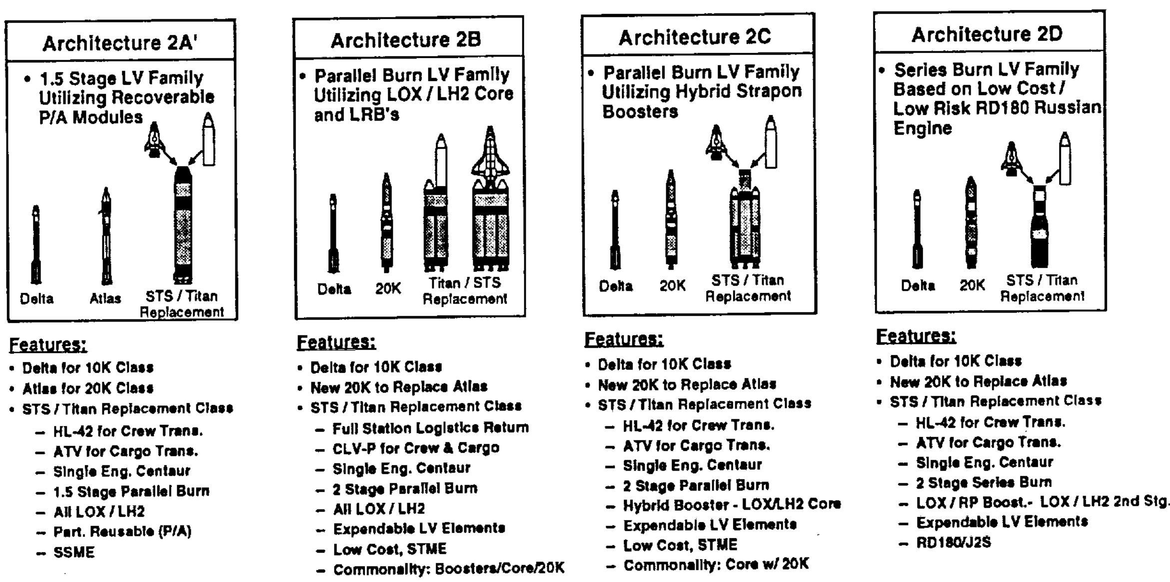


Figure 2.3-1 Option 2 - Architecture Overview

2.3.1.2 Launch Abort Modes

The launch abort modes considered in the analysis included; Return To Launch Site (RTLS), Trans-Atlantic Landing (TAL), Engine Out (EO), Abort To Orbit (ATO), and Abort Once Around (AOA). These five intact abort modes represent means of the spacecraft (HL-42 or CLV-P) to achieve a runway landing in the event of a malfunction during the launch phase.

The RTLS abort mode is used to return the spacecraft directly to the launch site. This means abandoning the launch trajectory, and reversing the ground track to gain sufficient velocity toward the launch site for a landing on the KSC runway. The TAL abort mode is similar to the RTLS in that the ascent trajectory is abandoned, but in this case the decision to abort is reached much later in the trajectory and the landing site is in Europe or Africa (depending on the launch inclination). The EO and ATO aborts permit the launch trajectory to continue to a MECO target (altitude and velocity) which places the spacecraft into orbit. The EO abort allows the nominal MECO target to be reached, and thus the mission continues as planned. In the case of the ATO abort, this orbit is a lower energy orbit than nominal (a 50X80 Nmi. ATO orbit was used for these analyses). The AOA abort is used when insufficient energy exists to reach orbit, but the MECO target can send the spacecraft once around the Earth for a landing back at the launch site. A North America Landing (NAL) was used for Booster 2D at high inclination orbits to compensate for this two-stage (series burn) booster design.

For those periods of the trajectory where above described intact abort modes are not available, the HL-42 would perform a water landing which would permit a safe recovery of the crew or cargo. The CLV-P would descend to a stable, low altitude/velocity condition for the crew bail out. All abort modes were initiated at an assumed single engine failure in either the booster or core/second stages (as appropriate) at various times of the launch trajectory. Multiple engine failures or system level failures were not analyzed. Failures of this type would generally incapacitate the entire booster and result in a water landing/bailout for the spacecraft.

The landing sites selected for the intact abort options included the Kennedy Space Center (for RTLS, EO, ATO and AOA), and either Banjul, Gambia or Brize-Norton, England (for TAL at 28.5° and 51.6° inclinations, respectively). Several additional landing sites were identified for the NAL abort mode for Booster 2D. This abort mode was identified for the HL-42 concept when the Booster 2D second stage engine fails to start during high inclination orbit launches (51.6°). Under these special circumstances, the HL-42 can land in Boston or other nearby cities which have a 10,000 foot runway. A similar abort mode was attempted for this launch configuration at low inclination orbits (28.5°), but no land masses (islands such as Puerto Rico, the Bahamas, Bermuda) were close enough to the trajectory path to enable a runway landing. The landing sites for the orbital abort modes (EO, ATO, AOA) was KSC.

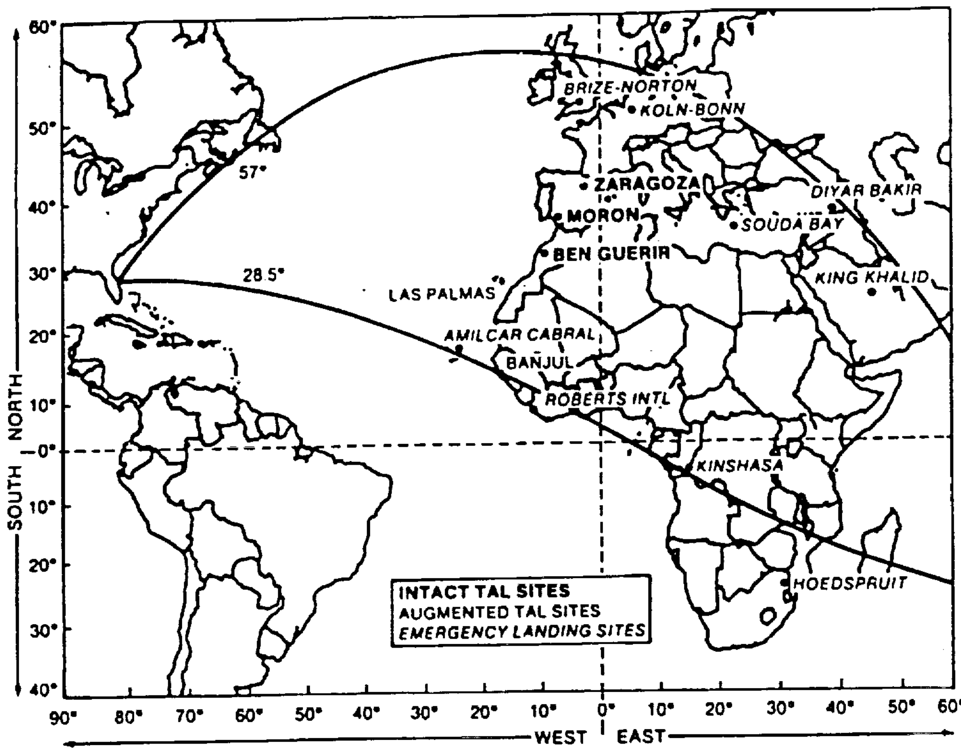


Figure 2.3-2 Abort Landing Sites

Discussions which follow of abort capabilities of the several launch vehicle/spacecraft configurations is based on the 51.6° inclination orbit trajectories to the Space Station. Analyses were performed to both 28.5° and 51.6° inclination orbits but the higher inclination orbit is of most interest because of the current Space Station redesign. Results at the 28.5° orbits are generally very similar to the 51.6° orbits and were not a major factor when comparing the booster configurations. Data from analysis of both inclination orbits is presented in enclosed tables and figures.

2.3.2 Abort Coverage of Selected Boosters

2.3.2.1 HL-42 Aborts on Booster 2A'

The abort analyses performed using the HL-42 on Booster 2A', a 1.5-stage booster for the Option 2 architecture, showed this configuration to be particularly effective for manned flights. This launch vehicle configuration consists of a core stage and several booster engines which are staged early in the flight. The booster engines are four SSME engines (staged in pairs), and the core stage uses two SSME sustainer engines (all LOX/LH2 propellant). Because the HL-42 configuration weight is well below the maximum payload capability of this launch vehicle, there is considerable excess propellant in the booster propellant tanks (assuming the tanks are filled to capacity for the launch). This excess propellant is valuable for abort capabilities when recovering from propulsion failures in either first or second stage flight.

The abort analysis revealed that 100% of the launch trajectory has an intact abort mode coverage for the HL-42 (no water ditching required). During first stage flight, the RTLS abort mode is provided from the launch pad until 192 seconds into the launch, at which time the vehicle is too far downrange for the HL-42 to return to KSC. During the first 64 seconds of flight, the HL-42 can return to KSC using just its own abort motors. The extended RTLS period (64 sec. to 192 sec.) is available by using the booster core stage thrust to perform a powered turnaround maneuver and to generate sufficient velocity back toward the launch site for the HL-42 range to reach the KSC landing site. Also during first stage flight, if a booster engine fails late in its burn duration (last 60 seconds), an engine out (EO) capability exists (nominal MECO target is achievable). Both of these abort modes utilize propellant margins in the core stage to make up the thrust loss of the booster stage engine failure. The EO and RTLS abort mode periods overlap, thereby providing full abort capability during the entire first stage flight.

During second stage flight, the abort modes available are the TAL, ATO, and EO aborts. These abort modes are available through the use of excess core stage propellant (since there are two engines in the core stage) and the HL-42's abort motors and on-board OMS systems (if necessary). The TAL abort mode is available for 158 seconds of flight time and returns the HL-42 to a landing site in northern Europe. The HL-42 abort and OMS propulsion systems, coupled with the long gliding range of the HL-42 aerodynamic shape, assist the remaining core stage engine in performing this abort mode. The ATO abort mode is available for 127 seconds during the second stage trajectory. The launch vehicle can adjust its trajectory to a lower energy MECO target if a core stage engine fails during this period. The HL-42 would remain in this orbit until a landing opportunity becomes available or it may perform some of the on-orbit mission objectives from this orbit. (Note: it is assumed that when an engine out capability becomes available, the ATO abort mode would no longer be needed and the EO abort mode would be the preferred option.) The second stage trajectory EO abort mode (a duration of 63 seconds) is enabled by use of the remaining core stage engine. Failure of a core stage engine during this period would not prevent the HL-42 from achieving the nominal MECO target. The booster second stage flight is thus found to provide full abort coverage. During this 182 second time period, an engine failure would not result in a HL-42 water landing.

The Booster 2A' configuration has thus been found to provide full abort coverage, providing at least one abort mode during the entire launch trajectory. At no time during the launch is the HL-42 exposed to a water landing contingency. An EO abort capability (that is, successful completion of the mission after suffering the loss of a single engine) exists for 31% of the trajectory and the alternate landing site exposure (TAL) is only 30% (48% and 26% for 28.5° trajectories).

HL-42 ABORT OPTIONS	Abort Window
• RETURN-TO-LAUNCH-SITE (RTLS) Any engine failure in core stage. Use remaining engines and HL-42 escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for HL-42 range to reach landing site.	0 sec ≤ MET ≤ 192 sec
• ENGINE OUT Single booster or sustainer engine failure. Utilize remaining engines to reach nominal MECO target.	150 sec ≤ MET ≤ 210 sec 329 sec ≤ MET ≤ 392 sec
• TRANS-ATLANTIC LANDING (TAL) Single booster or sustainer engine failure. Utilize HL-42 escape motors and OMS to establish sufficient velocity for HL-42 range to reach landing site in Europe or Africa.	210 sec ≤ MET ≤ 368 sec
• ABORT TO ORBIT (ATO) Single booster or sustainer engine failure. Utilize 2nd stage to reach lower energy MECO target for 15X80 Nmi. parking/transfer orbit. HL-42 circularizes to low altitude orbit, waits for first available landing opportunity.	265 sec ≤ MET ≤ 392 sec
• WATER LANDING (WL) Multiple engine or propulsion system failures, no runway landing available. HL-42 performs escape maneuver and glides to water landing.	Not Required

Figure 2.3-3 Booster 2A' Abort Capability

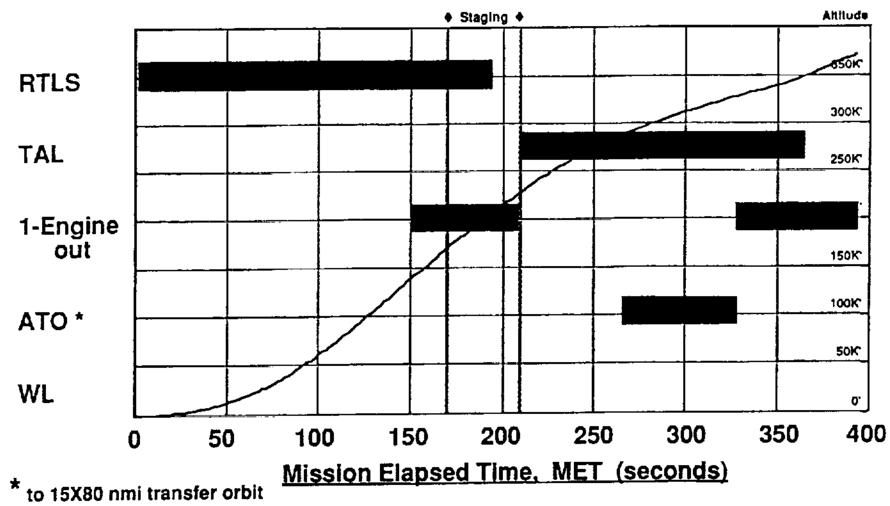


Figure 2.3-4 Booster 2A' Abort Coverage

HL-42 ABORT OPTIONS	Abort Window
RETURN-TO-LAUNCH-SITE (RTLS) Any engine failure in core stage. Use remaining engines and HL-42 escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for HL-42 range to reach landing site.	0 sec ≤ MET ≤ 187 sec
ENGINE OUT Single booster or sustainer engine failure. Utilize remaining engines to reach nominal MECO target.	103 sec ≤ MET ≤ 210 sec 311 sec ≤ MET ≤ 389 sec
TRANS-ATLANTIC LANDING (TAL) Single booster or sustainer engine failure. Utilize HL-42 escape motors and OMS to establish sufficient velocity for HL-42 range to reach landing site in Europe or Africa.	210 sec ≤ MET ≤ 368 sec
ABORT TO ORBIT (ATO) Single booster or sustainer engine failure. Utilize 2nd stage to reach lower energy MECO target for 15X80 Nmi. parking/transfer orbit. HL-42 circularizes to low altitude orbit, waits for first available landing opportunity.	214 sec ≤ MET ≤ 389 sec
WATER LANDING (WL) Multiple engine or propulsion system failures, no runway landing available. HL-42 performs escape maneuver and glides to water landing.	Not Required

Figure 2.3-5 Booster 2A' Abort Capability (28°)

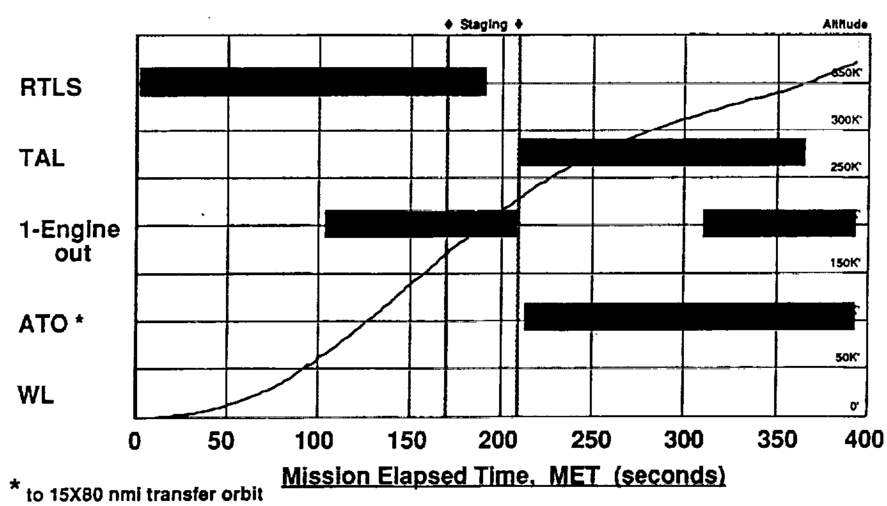


Figure 2.3-6 Booster 2A' Abort Coverage (28°)

2.3.2.2 CLV-P Aborts on Booster 2B

Abort analyses were performed for the CLV-P on Booster 2B, a 2-stage, parallel burn booster for the Option 2 architecture. The two booster stages each use two ATLCE engines (LOX/LH2 booster stage propellant), and the second (core) stage uses a single ATLCE engine (LOX/LH2 core stage propellant). Because the CLV-P configuration weight is the maximum payload capability of this launch vehicle, there is no excess propellant in the booster core stage available for extended abort capabilities when recovering from first stage propulsion failures.

The abort analysis revealed that a large percentage (43%) of the launch trajectory has no abort mode coverage for the CLV-P (requiring a crew bailout over water and loss of the vehicle and payload) . During first stage flight, the RTLS abort mode is provided only after a minimum of 20 seconds after liftoff from the launch pad and until 200 seconds into the launch, at which time the vehicle is too far downrange for the CLV-P to return to KSC. The RTLS period (180 seconds duration) is available only for an engine failure in one booster stage by using the remaining booster engines and the core stage thrust to perform a powered turnaround maneuver and to generate sufficient velocity back toward the launch site for the CLV-P range to reach the KSC landing site. Also during first stage flight, if a booster engine fails late in the burn duration (last 95 seconds), either an ATO or an EO abort option can be flown. Both of these abort modes utilize remaining engines and propellant in the booster and core stages to make up the thrust loss of the booster engine failure. The ATO, EO, and RTLS abort mode periods overlap, thereby providing some abort capability during all but the initial 20 seconds of first stage flight.

During second stage flight, the only abort mode available is the TAL abort. This abort mode is available only through the use of the CLV-P abort motors and on-board OMS systems since there is only a single engine in the booster core stage. The TAL abort mode is available only for the last 17 seconds of flight time and returns the CLV-P to a landing site in northern Europe. The CLV-P abort and OMS propulsion systems, coupled with the long gliding range of the CLV-P aerodynamic shape, enable this abort mode. The second stage EO abort mode (a short duration of only 8 seconds) is enabled only by use of the CLV-P abort motors. The booster second stage flight is thus found to have no abort coverage until the last 17 seconds. During this 198 second time period, an engine failure would force the CLV-P to perform a water ditching (and crew bailout if manned).

The Booster 2B configuration has thus been found to provide only limited abort coverage for the CLV-P. During the launch, the CLV-P (and crew) is exposed to a water ditching/bailout contingency for 43% of the trajectory. An EO abort capability exists for 23% of the trajectory and the alternate landing site capability (TAL) is only 4% (24% and 10% for 28.5° inclination trajectories). This level of abort coverage is not considered acceptable for a new manned launch system.

CLV-P ABORT OPTIONS	Abort Window
RETURN-TO-LAUNCH-SITE (RTLS) Single engine failure in booster stage. Utilize booster & core stages plus CLV escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for CLV to reach KSC.	20 sec ≤ MET ≤ 200 sec
ENGINE OUT Single engine failure in booster stage. Utilize remaining booster engines and core stage to reach nominal MECO target.	129 sec ≤ MET ≤ 224 sec
TRANS-ATLANTIC LANDING (TAL) Single engine failure in core stage. Utilize CLV escape motors/OMS to establish sufficient velocity for CLV range to reach landing site in Europe.	405 sec ≤ MET ≤ 422 sec
ABORT TO ORBIT (ATO) Single engine failure in booster stage. Utilize core stage to reach lower energy MECO target. CLV circularizes to low altitude orbit, waits for first available landing opportunity.	130 sec ≤ MET ≤ 224 sec
WATER DITCHING (WD) Single engine failure in core stage. CLV descends to low altitude for crew to safely bail out over water.	224 sec ≤ MET ≤ 405 sec

Figure 2.3-7 Booster 2B Abort Capability

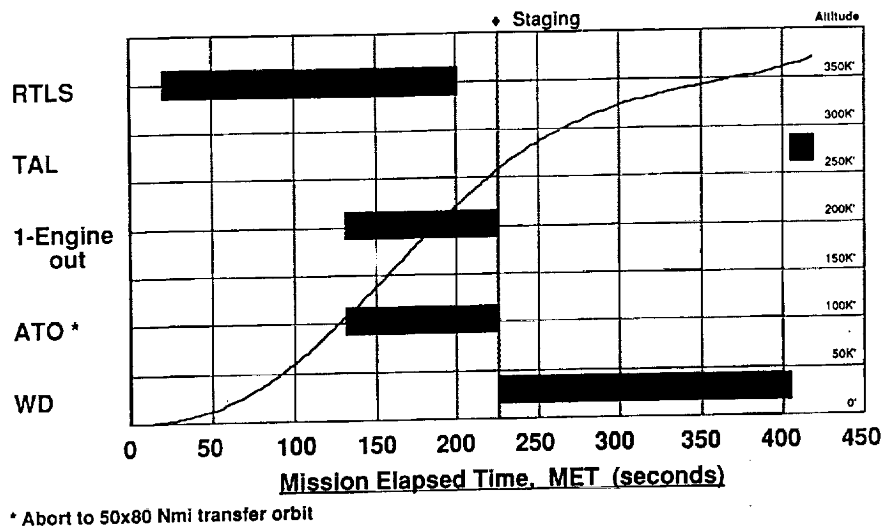
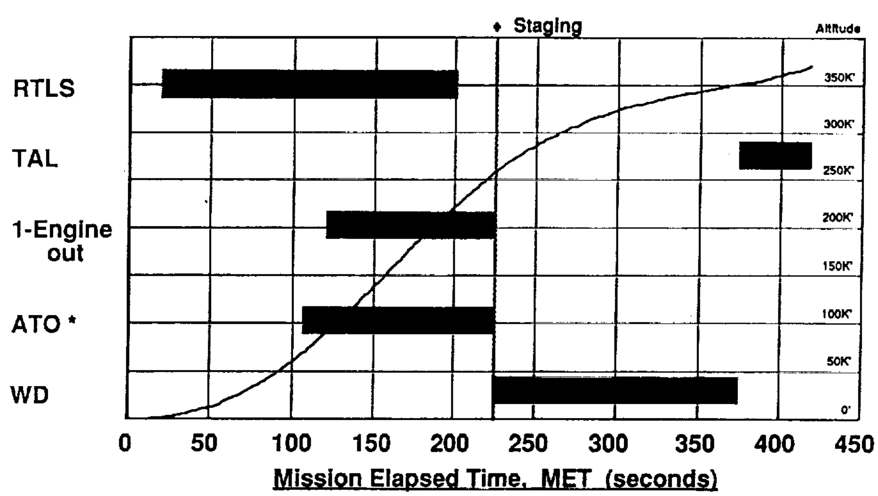


Figure 2.3-8 Booster 2B Abort Coverage

CLV-P ABORT OPTIONS

	Abort Window
• **RETURN-TO-LAUNCH-SITE (RTLS)** Single engine failure in booster stage. Utilize booster & core stages plus CLV escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for CLV to reach KSC.	20 sec ≤ MET ≤ 200 sec
• **ENGINE OUT** Single engine failure in booster stage. Utilize remaining booster engines and core stage to reach nominal MECO target.	120 sec ≤ MET ≤ 224 sec
• **TRANS-ATLANTIC LANDING (TAL)** Single engine failure in core stage. Utilize CLV escape motors/OMS to establish sufficient velocity for CLV range to reach landing site in Africa.	380 sec ≤ MET ≤ 422 sec
• **ABORT TO ORBIT (ATO)** Single engine failure in booster stage. Utilize core stage to reach lower energy MECO target. CLV circularizes to low altitude orbit, waits for first available landing opportunity.	107 sec ≤ MET ≤ 224 sec
• **WATER DITCHING (WD)** Single engine failure in core stage. CLV descends to low altitude for crew to safely bail out over water.	224 sec ≤ MET ≤ 380 sec

Figure 2.3-9 Booster 2B Abort Capability (28°)



* Abort to 50x80 Nmi transfer orbit

Figure 2.3-10 Booster 2B Abort Coverage (28°)

2.3.2.3 HL-42 Aborts on Booster 2C

Abort analyses were also performed for the HL-42 on Booster 2C, a 2-stage parallel burn, hybrid booster for the Option 2 architecture. The two booster stages each consist of a single engine hybrid motor (hybrid propellant booster stages), and the second (core) stage uses a single ATLCE engine (LOX/LH2 core stage propellant). Because the HL-42 configuration weight is near the maximum payload capability of this launch vehicle, there is no excess propellant in the booster core stage available for extended abort capabilities when recovering from first stage propulsion failures. Also, because the booster stages are large thrust, single engine stages (nearly 1.5 million lbs thrust each), loss of one booster stage will require shutdown of the other booster stage to maintain control of the launch vehicle.

The abort analyses revealed that a large percentage (66%) of the launch trajectory has no abort mode coverage for the HL-42 (requiring a water landing/ditching). During first stage flight, the RTLS abort mode is provided from the launch pad until 130 seconds into the launch, at which time the vehicle is too far downrange for the HL-42 to return to KSC. During the first 64 seconds of flight, the HL-42 can return to KSC using just its own abort motors. The extended RTLS period (64 sec. to 130 sec.) is available by using the booster core stage thrust to perform a powered turnaround maneuver and to generate sufficient velocity back toward the launch site for the HL-42 range to reach the KSC landing site. The RTLS abort mode is not available after 64 seconds for a core stage engine failure. Also during first stage flight, a limited ATO abort option can be flown if a booster engine fails late in the burn duration (last 2 seconds only). This abort mode utilizes the remaining core stage engine/propellant and selects a lower energy MECO target to make up the thrust loss of the booster stages. No EO abort capability exists for first stage flight, and since the ATO capability is so limited, the RTLS abort is the only practical option available during first stage flight.

During second stage flight, the only abort mode available is the TAL abort. This abort mode is available only through the use of the HL-42 abort motors and on-board OMS systems since there is only a single engine in the booster core stage. The TAL abort mode is available only for the last 37 seconds of flight time and returns the HL-42 to a landing site in northern Europe. The HL-42 abort and OMS propulsion systems, coupled with the long gliding range of the HL-42 aerodynamic shape, enable this abort mode. The second stage EO abort mode (a short duration of only 8 seconds) is enabled by use of the HL-42 abort motors. The booster second stage flight is thus found to have no abort coverage until the last 37 seconds. During this 365 second time period, an engine failure would force the HL-42 to perform a water ditching.

The Booster 2C configuration has thus been found to provide only limited abort coverage for the HL-42. During the launch, the HL-42 (and crew) is exposed to a water landing contingency for 66% of the trajectory. No EO abort capability exists for the trajectory and the alternate landing site capability (TAL) is only 6% (7% and 11% for 28.5° inclination trajectories). This level of abort coverage is not considered acceptable for a new manned launch system.

HL-42 ABORT OPTIONS	Abort Window
• **RETURN-TO-LAUNCH-SITE (RTLS)** Single engine failure in booster stage. Utilize core stage and HL-42 escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for HL-42 range to reach landing site.	0 sec ≤ MET ≤ 130 sec
• **ENGINE OUT** Single engine failure in core stage. Utilize HL-42 escape motors and OMS to reach nominal MECO target.	487 sec ≤ MET ≤ 495 sec
• **TRANS-ATLANTIC LANDING (TAL)** Single engine failure in core stage. Utilize HL-42 escape motors and OMS to establish sufficient velocity for HL-42 range to reach landing site in Europe.	458 sec ≤ MET ≤ 495 sec
• **ABORT TO ORBIT (ATO)** Single engine failure in booster stage. Utilize core stage to reach lower energy MECO target for 15X80 Nmi. parking/transfer orbit. HL-42 circularizes to low altitude orbit, waits for first landing opportunity.	128 sec ≤ MET ≤ 130 sec
• **WATER LANDING (WL)** Single engine failure in booster stage, single engine failure in core stage. HL-42 performs escape maneuver and glides to water landing.	130 sec ≤ MET ≤ 458 sec

Figure 2.3-11 Booster 2C Abort Capability

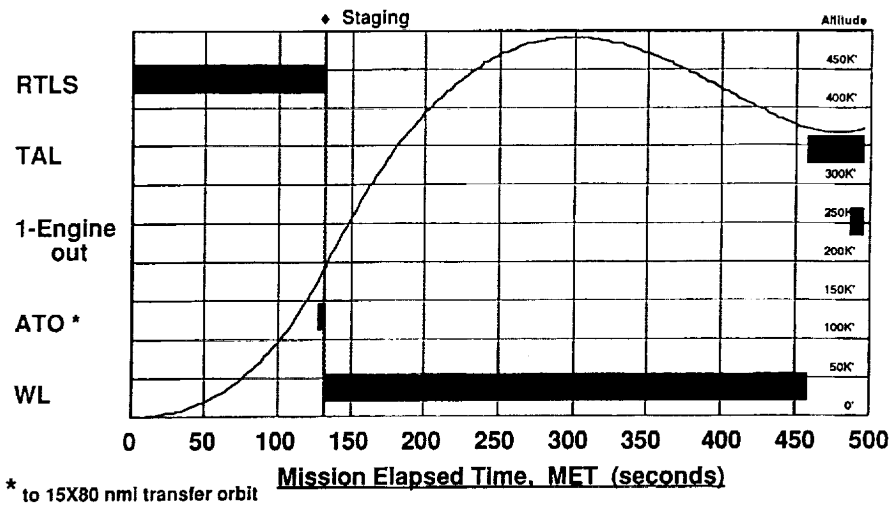


Figure 2.3-12 Booster 2C Abort Coverage

HL-42 ABORT OPTIONS	Abort Window
• **RETURN-TO-LAUNCH-SITE (RTLS)** Single engine failure in booster stage. Utilize core stage and HL-42 escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for HL-42 range to reach landing site.	0 sec ≤ MET ≤ 130 sec
• **ENGINE OUT** Single engine failure in core stage. Utilize HL-42 escape motors and OMS to reach nominal MECO target.	487 sec ≤ MET ≤ 495 sec
• **TRANS-ATLANTIC LANDING (TAL)** Single engine failure in core stage. Utilize HL-42 escape motors and OMS to establish sufficient velocity for HL-42 range to reach landing site in Africa.	434 sec ≤ MET ≤ 495 sec
• **ABORT TO ORBIT (ATO)** Single engine failure in booster stage. Utilize core stage to reach lower energy MECO target for 15X80 Nmi. parking/transfer orbit. HL-42 circularizes to low altitude orbit, waits for first landing opportunity.	128 sec ≤ MET ≤ 130 sec
• **WATER LANDING (WL)** Single engine failure in booster stage, single engine failure in core stage. HL-42 performs escape maneuver and glides to water landing.	130 sec ≤ MET ≤ 434 sec

Figure 2.3-13 Booster 2C Abort Capability (28°)

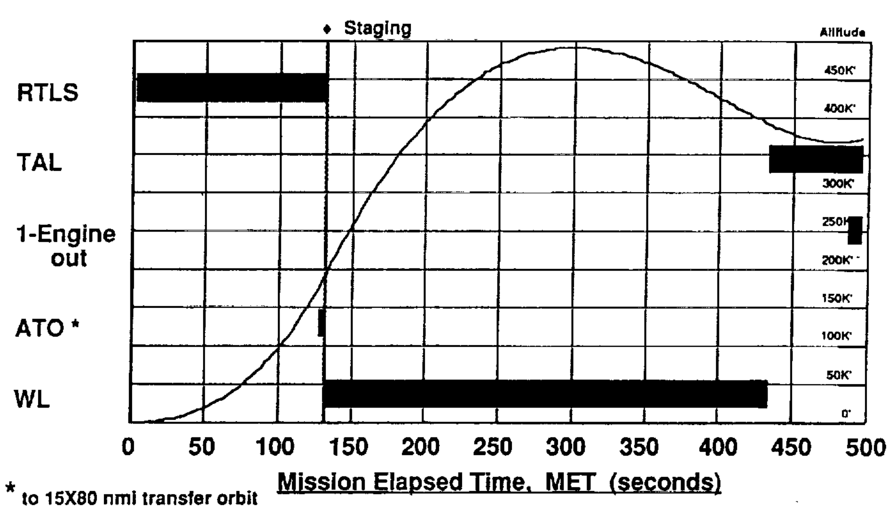


Figure 2.3-14 Booster 2C Abort Coverage (28°)

2.3.2.4 HL-42 Aborts on Booster 2D

The abort analysis performed using the Booster 2D, a 2-stage booster for the Option 2 architecture, was particularly interesting. This booster first stage uses three Russian RD-180 engines (LOX/RP 1st stage propellant), and the second stage uses a single J-2S engine (LOX/LH2 2nd stage propellant). Because the HL-42 configuration weight is well below the maximum payload capability of this launch vehicle, there is considerable excess propellant in the booster second stage (assuming the tanks are filled to capacity for the launch). This excess propellant is valuable for abort capabilities when recovering from first stage propulsion failures.

The abort analysis revealed that a large percentage (53%) of the launch trajectory has no abort mode coverage (other than the water ditching) for the HL-42. During first stage flight, the RTLS abort mode is provided from the launch pad until 185 seconds into the launch, at which time the vehicle is too far downrange for the HL-42 to return to KSC. During the first 64 seconds of flight, the HL-42 can return to KSC using just its own abort motors. The extended RTLS period (64 sec. to 185 sec.) is available by using the booster second stage thrust to perform a powered turnaround maneuver and to generate sufficient velocity back toward the launch site for the HL-42 range to reach the KSC landing site. Also during first stage flight, if an engine fails late in the burn duration (last 58 seconds), either an ATO or an EO abort option can be flown. Both of these abort modes utilize propellant margins in the second stage to make up the velocity shortfall of the first stage failure. The ATO and RTLS abort mode periods overlap, thereby providing some abort capability during the entire first stage flight.

During second stage flight, the only abort modes available are the TAL and engine out (EO) aborts. These abort modes are available only through the use of the HL-42 abort motors and on-board OMS systems since there is only a single engine in the booster second stage. The TAL abort mode is available for the last 65 seconds of flight time and returns the HL-42 to a landing site in northern Europe. The HL-42 abort and OMS propulsion systems, coupled with the long gliding range of the HL-42 aerodynamic shape, enable this abort mode. The second stage EO abort mode (a short duration of only 8 seconds) is enabled by use of the HL-42 abort motors. The booster second stage flight is thus found to have no abort coverage from its single engine start until the last 65 seconds. During this 297 second time period, an engine failure would force the HL-42 to perform a water landing. Of particular concern for this launch vehicle was the impact of a failure to start the second stage engine (generally regarded as a high risk event). Because of the HL-42 vulnerability to this risk (a water landing), a determined effort was made to find a means of performing a runway recovery for the HL-42 for this condition. Specifically, a North America Landing (NAL) abort mode was devised to protect the system from failure of the Booster 2D second stage engine (a J-2S) to ignite. No other intact abort modes were available to the HL-42 for this failure event (too far downrange for RTLS, not enough downrange for TAL).

To achieve the NAL abort, the Booster 2B must be flown with off-loaded second stage propellant in order to increase the staging velocity. A propellant off-load of 116,050 lbs in the second stage is possible for the booster to still insert the HL-42 into the nominal trajectory with no failures. This significant reduction in the total launch vehicle weight results in a faster and longer first stage trajectory and also reduces the nominal MECO time by 153 seconds. Operation of the booster in this fashion increased the staging velocity by 3,000 feet per second and thus created enough energy at staging that the HL-42 can reach landing sites in New England and Canada should the J-2S engine fail to start. Under these conditions, the HL-42 has sufficient energy to land at any 10,000 foot long runway along the North American East Coast (such as at Boston). Without the excess propellant in the second stage, however, the launch vehicle gives up significant capability to perform the first stage aborts (RTLS, ATO, and EO). The RTLS capability was reduced by 29 seconds and the EO and ATO capabilities were completely eliminated. The total exposure to water landing, however, was reduced significantly; from 297 seconds during second stage flight to only 55 seconds during first stage flight. An important design tradeoff exists between these two options (excess vs. off-loaded second stage propellant) to determine which of the potential engine failure modes (first stage or second stage) should be protected against. No attempt was made during the analysis to perform this design trade study, but the significantly reduced HL-42 exposure to water landing (from 53% to only 13% of the trajectory) was noted as a key factor favoring the off-loaded propellant approach. This trade study is an important design consideration for any manned two stage launch vehicle. Launch vehicles which employ parallel burn stages (all engines running before liftoff) do not experience this condition.

The booster second stage was clearly found to play a crucial role in the HL-42 abort capabilities on this launch vehicle. Ignition of the second stage engine is a requirement for most of the RTLS abort coverage and both of the ATO and EO abort modes. Excess propellant in the second stage was also a key parameter for extending performance of the RTLS, ATO, and EO aborts. Removal of this excess propellant, on the other hand, was required to protect against failure of second stage engine ignition. The other launch vehicle concepts (such as 1.5 stage boosters or parallel burn 2-stage boosters) provided significantly different abort coverage than was found with this two stage booster.

HL-42 ABORT OPTIONS	Abort Window
RETURN-TO-LAUNCH-SITE (RTLS) Single engine failure in 1st stage. Utilize 2nd stage and HL-42 escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for HL-42 range to reach landing site.	0 sec ≤ MET ≤ 185 sec
ENGINE OUT Single engine failure in 1st or 2nd stage. Utilize 2nd stage propellant margin/reserve or HL-42 escape motors to reach nominal MECO target.	168 sec ≤ MET ≤ 201 sec 555 sec ≤ MET ≤ 563 sec
TRANS-ATLANTIC LANDING (TAL) Single engine failure in 2nd stage. Utilize HL-42 escape motors and OMS to establish sufficient velocity for HL-42 range to reach landing site in Europe or Africa.	498 sec ≤ MET ≤ 563 sec
ABORT TO ORBIT (ATO) Single engine failure in 1st stage. Utilize 2nd stage to reach lower energy MECO target for 15X80 Nml. parking/transfer orbit. HL-42 circularizes to low altitude orbit, waits for first available landing opportunity.	143 sec ≤ MET ≤ 201 sec
WATER LANDING (WL) Multiple engine failures in 1st stage, single engine failure in 2nd stage. HL-42 performs escape maneuver and glides to water landing.	201 sec ≤ MET ≤ 498 sec

Figure 2.3-15 Booster 2D Abort Capability

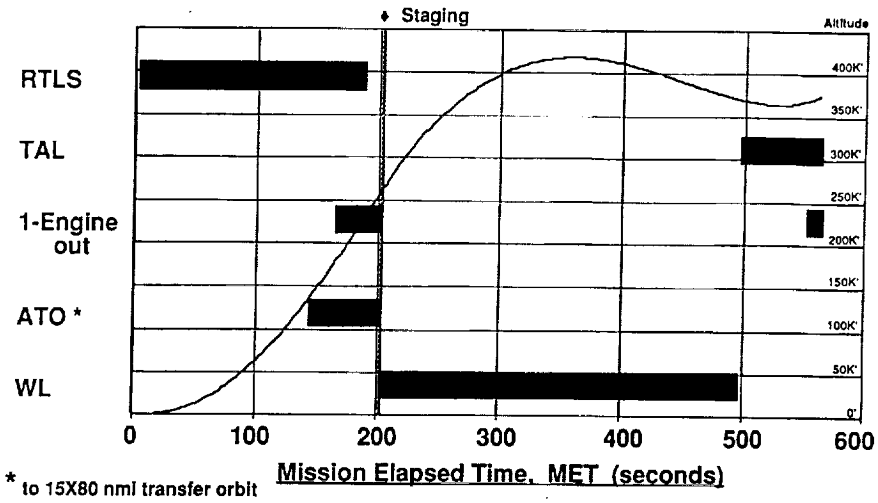
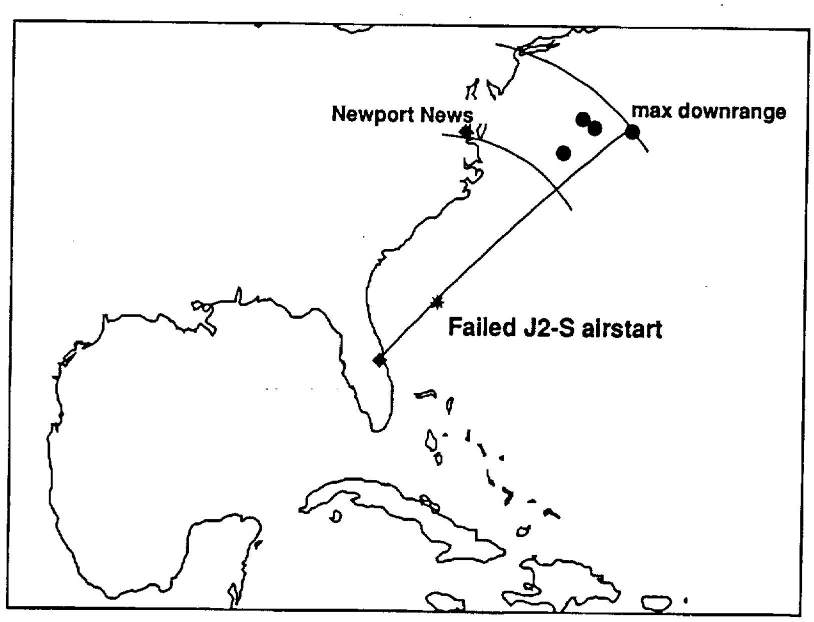


Figure 2.3-16 Booster 2D Abort Coverage



HL-42 abort motors & on-board propulsion <u>plus another 1,600 fps</u> required to achieve a runway landing

Figure 2.3-17 Failed Second Stage Ignition

HL-42 ABORT OPTIONS

HL-42 Abort Options	Abort Window
RETURN-TO-LAUNCH-SITE (RTLS) Single engine failure in 1st stage. Utilize 2nd stage and HL-42 escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for HL-42 range to reach landing site.	0 sec ≤ MET ≤ 156 sec
NORTH AMERICA LANDING (NAL) Second stage engine failure to start. HL-42 performs left bank and glides to landing site on east coast of U.S. or Canada. HL-42 uses abort motors and/or OMS as required.	211sec ≤ MET ≤ 350 sec
TRANS-ATLANTIC LANDING (TAL) Single engine failure in 2nd stage. Utilize HL-42 escape motors and OMS to establish sufficient velocity for HL-42 range to reach landing site in Europe or Africa.	316 sec ≤ MET ≤ 410 sec
WATER LANDING (WL) Multiple engine failures in 1st stage, single engine failure in 2nd stage. HL-42 performs escape maneuver and glides to water landing.	156 sec ≤ MET ≤ 211 sec
ENGINE OUT (EO)	Not Available
ABORT TO ORBIT (ATO)	Not Available

Figure 2.3-18 Booster 2D Abort Capability- Offloaded Propellant

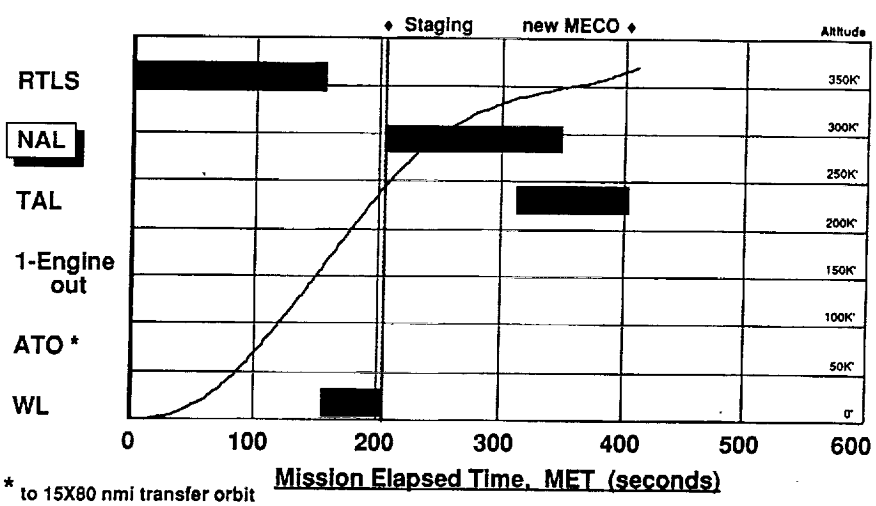


Figure 2.3-19 Booster 2D Abort Coverage- Offloaded Propellant

HL-42 ABORT OPTIONS	Abort Window
RETURN-TO-LAUNCH-SITE (RTLS) Single engine failure in 1st stage. Utilize 2nd stage and HL-42 escape motors to execute powered turnaround maneuver and establish sufficient velocity toward launch site for HL-42 range to reach landing site.	0 sec ≤ MET ≤ 185 sec
ENGINE OUT Single engine failure in 1st or 2nd stage. Utilize 2nd stage propellant margin/reserve or HL-42 escape motors to reach nominal MECO target.	170 sec ≤ MET ≤ 201 sec 555 sec ≤ MET ≤ 563 sec
TRANS-ATLANTIC LANDING (TAL) Single engine failure in 2nd stage. Utilize HL-42 escape motors and OMS to establish sufficient velocity for HL-42 range to reach landing site in Europe or Africa.	498 sec ≤ MET ≤ 563 sec
ABORT TO ORBIT (ATO) Single engine failure in 1st stage. Utilize 2nd stage to reach lower energy MECO target for 15X80 Nmi. parking/transfer orbit. HL-42 circularizes to low altitude orbit, waits for first available landing opportunity.	146 sec ≤ MET ≤ 201 sec
WATER LANDING (WL) Multiple engine failures in 1st stage, single engine failure in 2nd stage. HL-42 performs escape maneuver and glides to water landing.	201 sec ≤ MET ≤ 498 sec

Figure 2.3-20 Booster 2D Abort Capability (28°)

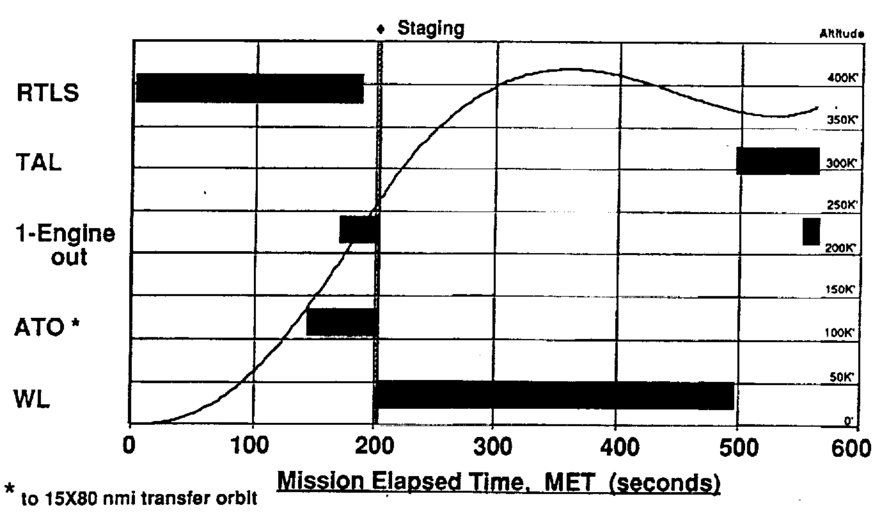
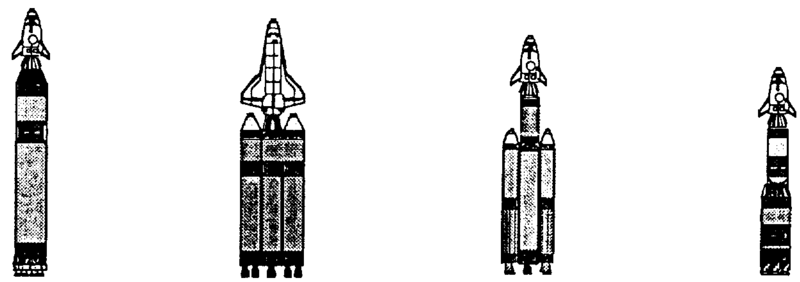


Figure 2.3-21 Booster 2D Abort Coverage (28°)

2.3.3 Abort Studies Findings

The abort analyses showed that a large percentage (generally 40 to 60%) of the launch trajectories have no intact abort mode coverage for booster configurations 2B, 2C, and 2D. This is caused by the single engine operation of these designs during second stage flight. During first stage flight, the RTLS abort mode is generally provided from the launch pad until approximately 200 seconds into the launch, at which time the vehicle is too far downrange for the spacecraft to return to KSC. The ATO and EO abort mode periods usually overlap with the RTLS abort mode, thereby providing some abort capability during the entire first stage flight. During second stage flight (post booster or engine staging), the only intact abort modes available are the TAL, ATO, and EO aborts. Where there is only a single engine in the booster's second (or core) stage, these abort modes are available only at the very end of the trajectory through the use of the spacecraft's abort motors and on-board OMS systems. The abort and OMS propulsion systems, coupled with the long gliding range of the HL-42 and CLV-P configurations, enable these abort modes.

The booster second (or core) stage was clearly found to play a crucial role in the HL-42 and CLV-P abort capabilities on these launch vehicles. Utilization of this stage's engine and propellant plays a significant role in late RTLS capabilities. A second engine on these stages (available only on the Booster 2A' configuration) provides extended intact abort coverage. Only the 2A' booster configuration has sufficient intact abort coverage to completely eliminate the spacecraft water landing exposure. For manned spacecraft flights, the Booster 2A' configuration is clearly superior to the other booster configurations analyzed.



	Architecture 2A'	Architecture 2B	Architecture 2C	Architecture 2D
Mission Success after Failure (%)				
28.5°	48	24	2	7
51.6°	31	23	2	0
Alternate Landing Site Exposure (%)				
28.5°	26	10	11	12
51.6°	30	4	6	49
Water Landing Exposure (sec)				
28.5°	0	156	304	297
51.6°	0	181	328	55

Figure 2.3-22 Summary of Abort Capabilities

2.4 SSTO MPS Operability Studies

The principal objective of all proposed SSTO launch vehicle concepts is a dramatic improvement in vehicle operability. Regardless of which SSTO design concept is eventually selected for development (VTOHL or VTOVL, LOX/LH2 or tri-propellant, etc.), all concepts must achieve significant reductions in ground processing timelines and the supporting workforce in order to meet the proposed cost benefits of an SSTO program. An often stated goal is that "the SSTO concept will be operated like an airplane." In practice, this is difficult to execute, but there are many facets of airplane operations which can be applied to the SSTO concept. In particular, the methods by which an airplane designer (or the airplane operator) identifies and incorporates operational requirements into the design process can be applied to a SSTO launch vehicle. Under the ATSS study, a method which was first used in the design of the Lockheed L-1011 aircraft (and subsequently other aircraft) was adapted for analyzing the operability characteristics of the reference SSTO concept as defined by NASA in their recently completed Access to Space study, Option 3. This method is one in which the vehicle flight and ground operations are analyzed by computer simulations. The simulations include the flight operation and both scheduled and unscheduled maintenance operations for all vehicle components. The components' performance is determined from current (or projected) component reliability and maintainability histories. Use of such a technique is viewed as a proven example of "operations driving the design." Use of this technique early in the design of the SSTO concept permits the vehicle operability to be designed in, not just allocated.

2.4.1 Operations Simulation Analysis

2.4.1.1 MPS Operability Study Approach

The SSTO Main Propulsion System (MPS) was selected for this operability analysis. There was insufficient time to model the entire SSTO concept, therefor this single subsystem was chosen. The MPS is not only a critical subsystem of any SSTO concept, but it is, historically, also one of the most difficult to process. Differences among the several SSTO concepts are clearly reflected in their MPS designs, so this subsystem also serves as a useful benchmark for comparing the operability of competing concepts.

The starting point of the MPS operability study is to determine the current Shuttle MPS component reliability (Mean Time Between Failure, MTBF) and maintainability (Mean Time To Repair, MTTR) characteristics and use these components as representative of the current state-of-the-art MPS capability (at least for a reusable MPS). This data is then used in a simulation of the planned ground processing of the SSTO to determine the probability that a component will fail (and therefore, its related ground process/test) and then will require unplanned/unscheduled maintenance actions beyond what might have otherwise been allocated. All of the MPS components are individually identified for each process or test performed during the SSTO turnaround and launch sequence. Changes in the component reliability, or changes in the MPS design which change component quantities, can thus be directly reflected in the resultant ground processing timelines. This permits new technology and hardware design improvements beyond the current state of the art (SOA) to be measured in terms of their impact on the SSTO operability. It also permits analysis of competing SSTO concepts (with their varying number and type of main propulsion engines, propellants, and components) to also be directly compared in terms of operability.

The goals of this initial operability analysis are to support trade studies of various SSTO concepts. The level of accuracy reflected in the operations simulations is thus not too exact. Only representative mean maintenance downtimes (MDT) and unscheduled maintenance manpower (UMMHR) estimates are needed to permit trade study comparisons among concepts. The simulation results provided in the study are considered to be good representations of projected SSTO operations, but should not be assigned high accuracy. Much more detail would be required in the simulation models before the analyses can be claimed accurate.

The operability analyses are performed with Rockwell's proprietary SIMtrix and STARSIM simulation models. These simulation models provide detailed assessment of the propulsion system components and checkout activities and also provide a complete system-level simulation of the SSTO launch capability, facility requirements, and resource utilization needs.

2.4.1.2 SSTO Main Propulsion System Definition

A detailed schematic of a typical Single Stage To Orbit (SSTO) vehicle main propulsion system (MPS) was created which includes a fluid flow diagram, and for each component in the system, a reference designator and a representative part number. The MPS schematic was based on a seven engine (SSME) propulsion system, but the number and type of engines may be changed to reflect alternate design concepts. Part numbers were selected from Space Shuttle MPS components in order that representative reliability and maintenance data for the SSTO components could be generated. The MPS system definition includes the complete fuel system, oxidizer system, and helium/nitrogen purge and pressurization systems. For the tri-propellant SSTO concepts, the schematic includes the complete RP-1 fuel system and any necessary changes to the helium pressurization system. The level of detail in these schematics includes items such as all isolation valves (such as engine pre-valves and tank fill/drain valves), check valves, He regulators and filters, propellant feedlines and manifolds(with vacuum jacketing), ground fluids disconnects, etc. The MPS helium system design assumed that pneumatic actuation would be required for the large feedline valves and for engine valves and purges (as per current Shuttle and SSME designs). Instrumentation for the tanks, feedlines, and engines were also included (Shuttle MPS instrumentation was used as the guideline for instrumentation requirements). No hydraulics were included in the SSTO MPS schematic as it was assumed the vehicle would utilize electro-mechanical actuators (EMAs). Individual wire harnesses (and connectors), engine insulation, test lines and test ports, and other ancillary equipment were not included.

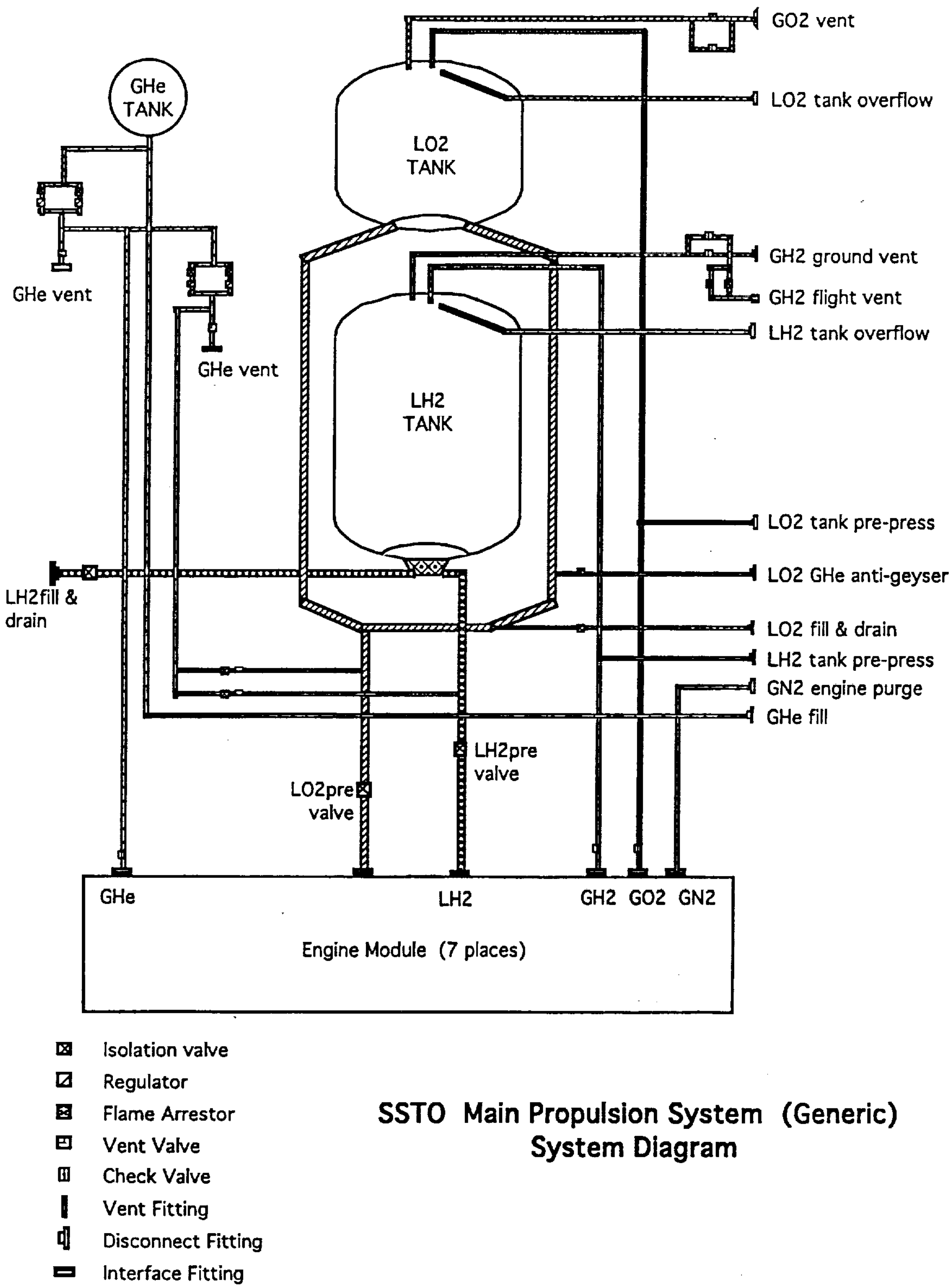


Figure 2.4-1 SSTO Main Propulsion System Schematic

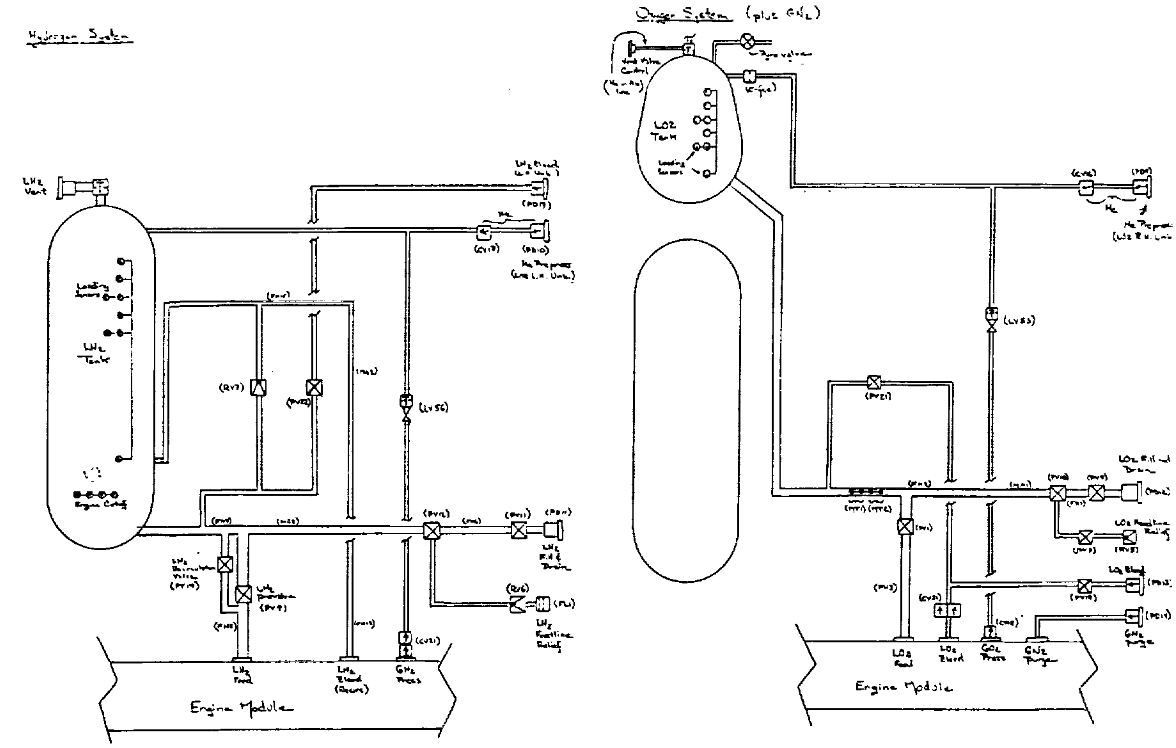


Figure 2.4-2 LH2 and LOX System Schematics

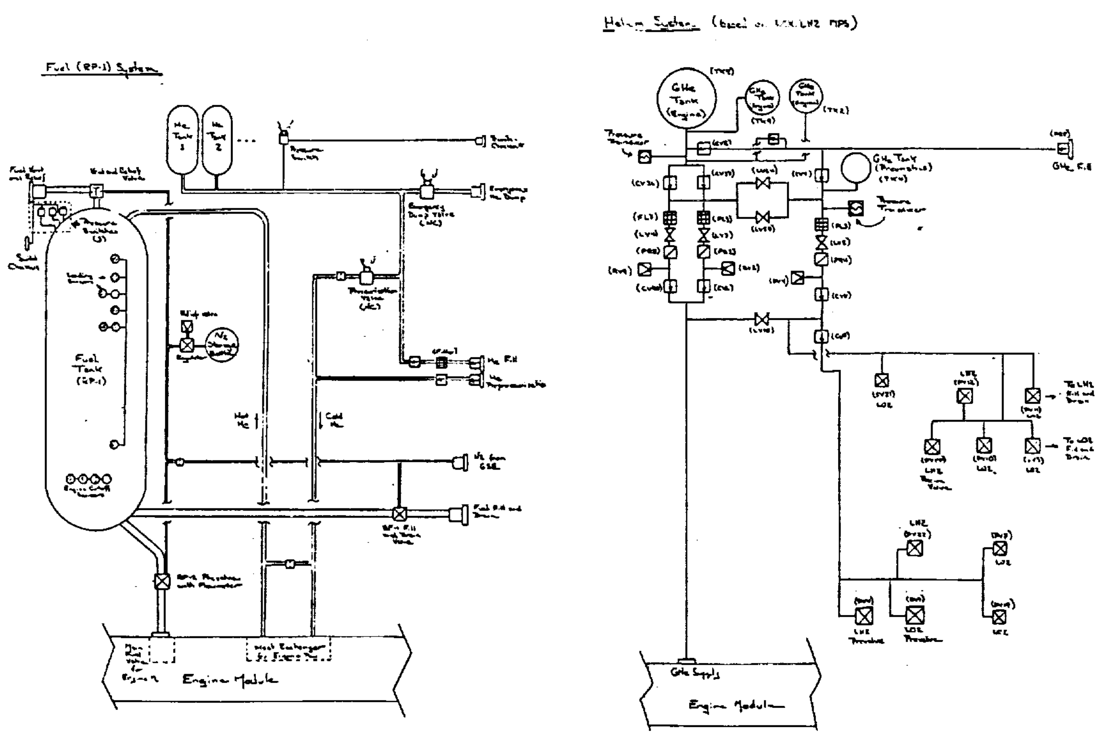


Figure 2.4-3 RP-1 & He/GN2 System Schematics

2.4.1.3 MPS Component Reliability/Maintenance Data

The source of component reliability and maintenance data to support the simulation models was obtained from the Space Shuttle program. The Shuttle is the only reusable propulsion system from which to obtain such data and an extensive failure database for all Shuttle components exists. The Space Shuttle program's Problem Reporting and Corrective Action (PRACA) database was used to collect all problem reports (PRs) on the MPS components which had been generated either during flight or during ground processing. This comprehensive NASA KSC database includes all functional failures of the components, as well as inspection defects. The database covers all launch processing activity since the Challenger accident, including all tanking tests, launch scrubs and aborts. Electronic sorting of the database was used to generate the failure list for the desired MPS components for the SSTO study. The PRs used in the simulation were segregated by functional and defect type failures to account for the different repair times associated with each of these type of PRs. A Shuttle logistics database (PEARL) was also used to determine the vendor repair activities of Shuttle MPS components in order to estimate logistics requirements for an SSTO MPS. This database provided a summary of all vendor repair actions for MPS components and the average repair turnaround time (time to return repaired parts to KSC from the vendor) for each component.

To calculate a component's MTBF, not only the number of failures, but also the operating time of the component is required. To determine the operating time on Shuttle components, KSC's scheduled OPF, VAB, and launch pad operations timelines were reviewed and equipment operating times estimated. The ground test sequences were then correlated to the specific MPS components being checked in the test in order to calculate the individual component ground processing times. These component ground operating times were added to the component's in-flight operating time (including on-orbit and re-entry timelines) to obtain an estimate of the total hardware operating time for all Shuttle flights since the Challenger accident (thus correlating the operating time period with the PR database time period). The Shuttle MPS components' operating times for all ground tests, launch countdowns, and flights were then combined with its PR history to calculate the MTBF. A separate MTBF estimate was calculated for functional and defect type failures. The calculated MTBF for each MPS component was used to represent the current level of reliability for SSTO MPS hardware. A detailed breakdown of the SSME component reliability as experienced in the Shuttle program (Mean Time Between Removal, MTBR) was provided by Rocketdyne to determine the mean engine removal rate for the SSTO.

The maintainability data for MPS components was similarly obtained from the Shuttle flight history. MPS engineers from Rockwell's Florida Operations who are directly involved in the Shuttle MPS processing provided repair time and removal time estimates for each component in the SSTO MPS schematic. Manpower requirements, including any special skills requirements (such as welding, brazing, x-ray, foaming skills) were also identified for each component maintenance action. This data provided the basis for estimating the unscheduled maintenance time and manpower which would be required to return the MPS system to a flight ready condition once a component failure has been experienced. The combined Space Shuttle MPS component MTBF and MTTR data comprised the required reliability and maintainability database for performing the SSTO MPS operations simulation. Additional data used in the simulation was the NASA KSC safety limitations on the maximum number of personnel which can be working in closed or open compartments at any one time.

MPS Ground Processing Maintenance Includes both Functional and Inspection (Defect) Failures

Functional Failures: Component fails to perform to specified levels during ground processing (e.g. leaks, valve fails to open, ...)

Inspection Defects: Component has been improperly installed or damaged (e.g. scratched, dented, contaminated, misaligned, ...)

	Shuttle MPS Component PRs		
	Funct.	Defect	Total
Valves	81	91	172
Lines & manifolds	39	313	352
Helium tanks	41	54	95
Regulators	47	21	68
Disconnects	44	201	245
Filters (He system)	0	0	0
Sensors (temp, press)	46	24	70
Total	298	704	1002

Figure 2.4-4 Shuttle MPS Component Failures

2.4.1.4 SSTO MPS Ground Processing

Definition of detailed SSTO MPS ground operations test and checkout processes were not found in any SSTO concept descriptions (NASA or contractor documentation). A default set of checkout processes was therefor generated from the combined Space Shuttle Orbiter and ET checkout processes. While these processes and tests may be greatly reduced or eliminated when advanced technologies are developed for the SSTO (such a integrated vehicle health management concepts), the basic functions these processes perform will still be required. The basic tasks of checking the MPS helium system, the LH2 system, and the LOX system will be performed. Engine checkout and verification of engines/flight control systems interfaces will also be performed. It is not known today how long these tasks will take and how many people may be required to execute them. Exactly how these task are to be automated is also not known.

For the simulation purposes, all basic Shuttle MPS checkout operations were included in a detailed ground processing timeline, but were allocated to a 40 hour period (2.5 days at two eight-hour shifts per day), consistent with ground ruled SSTO ground processing timelines. The basic sequencing and constraints which are currently used on the Shuttle MPS and SSME for these tasks were also retained. This allocation process can be changed for each process as more information of the envisioned SSTO ground processing is generated. For the purpose of conducting trade studies, the ground processes as described above served as the SSTO baseline turnaround processing.

These baseline SSTO checkout sequences provided the framework for evaluating the net effect of the reliability and maintainability characteristics of the SSTO MPS components. When a component failed (whether in flight or during ground checkout), the failure was correlated to a specific ground test and a specific time in the turnaround process. This permitted the simulation to "schedule" the SSTO MPS maintenance activities during each flow and more realistically model the ground processing. Included in the maintenance scheduling were constraints on the number of technicians which could be working on the MPS system at any one time. For example, only one engine was allowed to be removed or installed at a time. This allowed for known serial processing activities to be properly represented. The SSTO was assumed to have an "open boattail", which permitted a maximum of 16 people to be working on the MPS system at any one time.

The SSTO MPS component operating times per turnaround and flight were calculated from the ground ruled SSTO checkout timelines, just as was done for Shuttle MPS components and timelines. The Shuttle derived reliability and maintainability data for SSTO components, coupled with the detailed SSTO ground processes derived from ground ruled turnaround timelines, provided the simulation with the necessary information to calculate the SSTO component failure data.

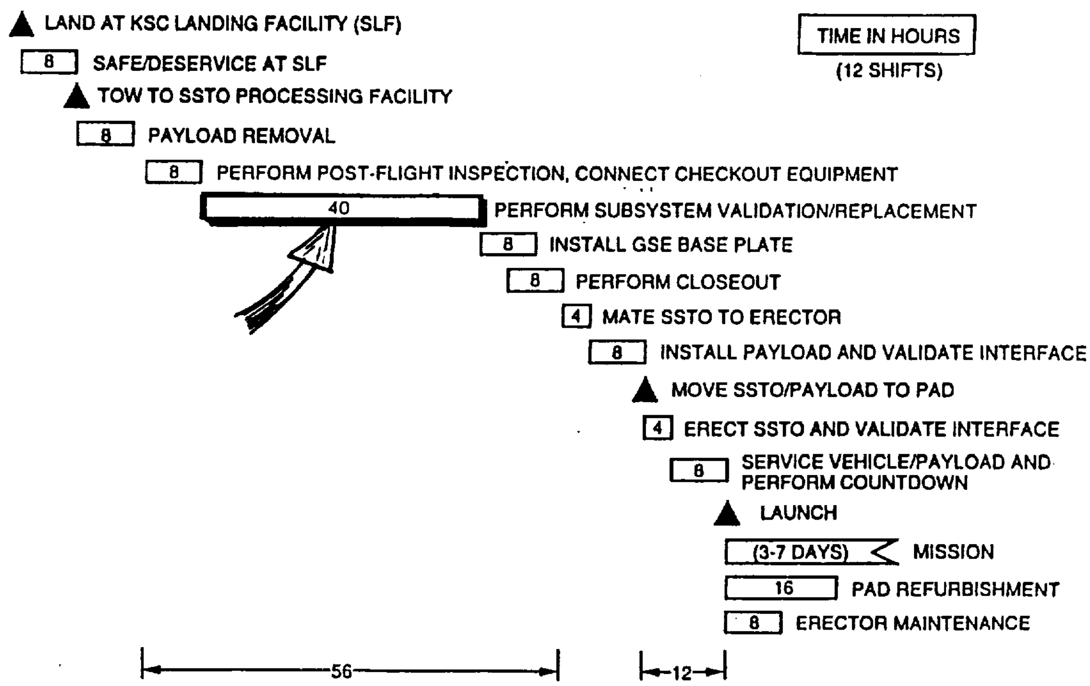


Figure 2.4-5 SSTO Baseline Ground Processing Timeline

Generic SSTO MPS & Tanks Processing Allocation

ID	Name	Duration
1	Engine Post Flight Low Pressure Pump Torques (V1011.03)	2.5h
2	Main Propulsion Sysem (MPS) VJ Line Checks (V9019)	2.5h
3	Engine Remove POSUs (V5058)	7h
4	Engine Post Flight High Pressure Pump Torques (V1011.03)	10h
5	MPS Electrical System Checkout (V1009.02)	10h
6	Engine Drying (V1011.01)	17.5h
7	Open Aft Swings (V3508)	1h
8	Aft Purge Air (V3555)	1.5h
9	Remove One Engine (V5058)	1.5h
10	Close Aft Swings (V3508)	1h
11	Offline Engine Operations	0h
12	MPS 17 Inch Pre Valve Screen (V1009.01)	4.5h
13	MPS He System Checkout (V1009.03)	14h
14	MPS LO2 System Checkout (V1009.04)	18.5h
15	MPS LH2 System Checkout (V1009.05)	9h
16	Open Swings (V3508)	1h
17	Aft Purge Air (V3555)	1.5h
18	Install One Engine (V5005)	1.5h
19	Close Swings (V3508)	1h
20	Engine Securing (V5005)	9.5h
21	MPS & Engine Checkout (V1009/V1011)	17.5h
22	Engine Pump Torques (V1011.03)	4.5h
23	Engine I/F Verification (V1001)	1.5h
24	MPS & Engine Integration Operations (V1009/V1011)	6h
25	MPS Flight Control TVC Checkout (V1063)	1.5h
26	MPS Valve Configuration for Rollout (V1171)	1h
27	Tank Preps for Checkout (T1102/T1103)	11h
28	LO2/LH2 Leak Tests (T1109)	4h
29	LH2 Fire Detection Installation (T5043)	3h
30	Leak Port Closeout (T5136)	5h
31	He Injector Box Checkout (T5238)	5h
32	All Tank Systems Test (T1150)	16h
33	Tank Inspection (T6248)	9.5h
34	LH2/LO2 Vent Disconnect (T1101)	1h
35	Pressure Sense Line Disconnect (T1102/T1103)	1.5h

S M T W T F S

Project: Generic SSTO MPS &
Date: 10/25/93

Critical
Noncritical
Progress
Milestone
Summary
Rolled Up

Page 1

Figure 2.4-6 MPS 40-Hour Ground Processing Detail

2.4.1.5 SSTO MPS System Modeling

Simulation models for analysis of the SSTO MPS ground processing were developed from Rockwell's SIMtrix and STARSIM computer codes. The SIMtrix model analyzes reliability, maintainability, and logistic parameters (as discussed above) to determine the effects of unscheduled maintenance on the planned SSTO MPS ground processing estimates. The calculated Shuttle MPS component level functional and defect failure rates, and all ground processing timelines, sequences, and constraints were used in this model. The model was used to simulate both the Shuttle MPS and the SSTO MPS ground processing. The Shuttle simulation results were checked against the actual Shuttle MPS experience as a benchmark test.

The STARSIM model was used to analyze the SSTO launch rate capability and launch facility needs/utilization based upon data provided by the SIMtrix model. This model operates at the system level, but accepts data at the subsystem level (such as from SIMtrix) for calculating the total vehicle ground turnaround timelines. The STARSIM reference turnaround scenario was based on the Access to Space Option 3 study's SSTO groundrules. The simulation included a five vehicle fleet of SSTO vehicles, mission times randomly distributed from one to seven days, total ground turnaround timelines of 100 hours, and scheduled maintenance down periods (OMDPs) every 20 flights.

A few additional processing groundrules had to be added to the STARSIM model for simulation completeness. The study manpower specifications were broken into a two-shift workforce and allocated a 246 workday year (consistent with current NASA KSC operating practices). To permit the effects of weather, pad aborts, and other launch pad delays to be measured, the current Space Shuttle history of launch delays and pad aborts was included in the model. (The SSTO design may or may not have similar sensitivities to these effects, but current design detail is far too vague to ignore them at this time. The Shuttle experience is typical of all launch vehicles and should serve as a good reference point for SSTO designers.) The model assigned a launch delay probability of 58%, with the cause of delay distributed among weather, MPS systems, avionics systems, ground systems, and other fluid systems as observed from the Shuttle launch history (post-Challenger). For MPS and fluid system delays, a percentage of these were assumed to require a rollback to the SSTO HPF for corrective action before launch. (The SSTO operational scenario is assumed to be geared to very minimal on-pad maintenance.)

SSTO Ground Operations Simulations are Performed with SIMtrix and STARSIM

SIMtrix

Monte Carlo simulation of scheduled and unscheduled maintenance & repair activities for specified ground processing sequences and timelines

Includes:
- component MTBF, MTBM
- component MTTR
- spares POS and RTAT
- undetected failures

STARSIM

Probabilistic simulation of launch systems and facility/resource utilization for specified launch rates and launch processes

Includes:
- launch vehicle subsystems
- payload integration
- facility constraints
- manpower allocations

Figure 2.4-7 Operability Analysis Software Tools

- Timeline follows Access to Space Option 3 (operational)
  - 100 hrs allocated for vehicle turnaround
  - 40 hrs allocated for subsystems processing
- Mission time uniformly distributed from 1 to 7 days
- Fleet of 5 SSTO vehicles
- 5 SSTO processing bays, 2 launch pads
- Single mission control center
- All missions return to KSC
- OMDP every 20 flights/vehicle, 90 workdays duration
  - 50 maintenance crew per shift for OMDPs
- 25 maintenance crew per vehicle per shift
  - 2 shift workdays, 246 days/yr
- Probability for on-pad launch delay (based on Shuttle)
  - 20% for MPS system scrub (60% require rollback)
  - 13% for weather scrub
  - 11% for fluid systems scrub (50% require rollback)
  - 7% for avionics system scrub
  - 7% for GSE scrub
- Reliability & logistics support
  - Component repair times (MTTR) per SIMtrix
  - Spares availability (POS) per SIMtrix

Figure 2.4-8 STARSIM Ground Processing Baseline

2.4.1.6 SSTO MPS Operability Trade Studies

2.4.1.6.1 SIMtrix Simulation Results

The SIMtrix model was first checked against the actual Shuttle experience as a benchmark test. The simulation returned a predicted Space Shuttle (MPS) mean down time of 370 hours and a mean unscheduled maintenance manhour requirement of 3,110 manhours. Combining this prediction with actual planned work schedules and planned manpower requirements resulted in a reasonably close correlation with the actual Shuttle experience (within 10% of actual Shuttle OPF processing times and MPS processing manpower). This means that the SIMtrix model of Shuttle MPS ground processing provides a good representation of the current state-of-the-art (SOA) for reusable MPS hardware. This SOA reference establishes the point of departure for all of the SSTO concept trade studies which followed.

The extrapolation from this SOA (Shuttle) reference to the envisioned SSTO MPS ground processing was performed in incremental steps in order to understand the individual effects of several parameters. The intermediate steps taken to get to the SSTO reference are:

1. Increase the number of SSME engines from 3 to 7.
2. Reduce the scheduled ground test time for MPS checkout to 40 hr's.
3. Reduce the MPS component repair times to 6 hours.
4. Increase the MPS component reliability for selected parts.

The effects of these incremental steps illustrated the very strong impact of extended (and intensive) ground testing on flight hardware. The addition of four extra engines (and the necessarily related additional MPS components) produced an additional 17 maintenance actions (PRs) per flow, resulting in a 135% increase in mean downtime and 4,493 unscheduled maintenance manhours. This result was not unexpected given the increase in system complexity that 7 SSMEs would create. When the scheduled ground test time for the MPS was reduced from 700 to 40 hours, the PRs dropped to 19 per flow. This resulted in a mean downtime of only 160 hr's, and only 1,323 unscheduled maintenance manhours, less than half of what was estimated for the Space Shuttle today. This significant improvement in required maintenance is an indicator of how much of the Shuttle MPS hardware life is being consumed by ground testing. (Because the Shuttle ground processing activities to date have been highly variable, the estimated component ground operating times can only be roughly estimated. The estimates made are believed to be reasonable, however, and thus the effects of ground testing believed to be representative of the actual Shuttle program.)

The next step to the SSTO reference simulation was to reduce component repair times. Many of the Shuttle MPS components are welded in place and covered with thermal protective foam insulation. This design creates for significant removal/repair times for these Shuttle components because of the need for the serial processes of foam removal, cutting, welding, weld x-ray, and finally foaming. Nevertheless, simulation of the SSTO required 6 hour component replacement groundrule was performed to show the benefits such a requirement might provide for an SSTO, given that it can be accomplished in the system design. The simulation revealed that the mean downtime was greatly reduced to only 55 hours and the resulting maintenance manhours further reduced to 600 hours (PRs were unchanged as the repair time does not influence the rate at which components fail).

The final step to the reference SSTO simulation was to increase the reliability (MTBF) of selected MPS components. Since the SSTO will not be placed into operational status for many years to come, there is every reason to expect that many of the components used will exhibit improved reliability from that seen today. This is even true for those components which currently demonstrate poor reliability, as they will encounter many opportunities for design improvements. The MTBF for several components was increased by a factor of 50% to allow for this component reliability growth. This final step also offers an insight into the sensitivity of the SSTO ground processing mean downtime to component reliabilities. The key components selected for this reliability improvement included: all electrically actuated isolation valves, all check valves, pneumatic operated fill and drain valves, helium supply regulators, relief valves, and vacuum jacketed lines. The simulation results revealed that the number of PRs per launch processing cycle fell to a mean of 16.5. The predicted mean down time was 50 hours, with an unscheduled maintenance manhours prediction of 534 hours. This final step of the simulated SSTO ground processing scenario indicates that the planned 40 hour maintenance period with 50 personnel would be expected to require 50 additional hours (6.25 shifts) and 534 additional manhours just for the MPS system alone.

NUMBER	OPF PROCESS (USES 7 SSMEs, 6 HOUR MTTR & IMPROVED R)		PRs/FLOW (MEAN)	PRs/FLOW S/D	FUNCT/ FLOW	DEFECTS/ FLOW	MDT (HOURS)	MDT S/D	MHRS/ FLOW	PERSONNEL (AVERAGE)	PERSONNEL (MAXIMUM)
V1011.03	ENGINE POST FLIGHT LOW PRESSURE PUMP TORQUES	1	-	-			-	-	-	-	-
V9019		2	1.17	1.04	0.20	0.97	4.73	3.56	43.85	9	16
V1011.03	ENGINE POST FLIGHT HIGH PRESSURE PUMP TORQUES	3	-	-			-	-	-	-	-
V1009.02	MPS ELECTRICAL SYSTEM CHECKOUT	4	0.01	0.10	0.01	0.00	0.16	1.60	0.32	1	2
	REMOVE ENGINES (IF REQUIRED)		0.40	0.64	0.40	0.00	0.60	0.64	7.20	12	14
V1009.01	MPS 17 INCH PRE-VALVE SCREEN	5	0.04	0.20	0.00	0.04	0.32	1.58	1.92	5	6
V1009.03	MPS He SYSTEM CHECKOUT	6	1.97	1.31	1.05	0.92	5.35	2.09	36.48	6	16
V1009.04	MPS LO2 SYSTEM CHECKOUT	7	2.49	1.51	0.50	1.99	8.14	4.58	94.15	11	16
V1009.05	MPS LH2 SYSTEM CHECKOUT	8	3.77	1.96	0.75	3.02	10.63	4.99	125.14	11	16
	INSTALL ENGINES PREVIOUSLY REMOVED		0.40	0.64	0.40	0.00	0.60	0.64	7.20	12	14
V1009/	MPS AND ENGINE CHECKOUT	9	5.38	2.39	0.78	4.60	15.17	6.20	201.36	13	16
V1171	MPS VALVE CONFIGURATION FOR ROLLOUT	10	0.05	0.22	0.01	0.04	0.30	1.31	1.32	3	6
T1109	LO2/LH2 LEAK TESTS	11	0.01	0.10			0.05	0.50	0.10	1	2
T1104/	ANCILLARY LEAK AND FLOW CHECKS	12	0.06	0.24			0.30	1.19	0.60	1	2
T1107		13	0.04	0.20			0.20	0.98	0.40	1	2
T5043	LH2 FIRE DETECTION INSTALLATION	14	0.01	0.10			0.16	1.60	0.64	3	4
T5136	LEAK PORT CLOSEOUT	15	0.01	0.10			0.05	0.50	0.10	1	2
T5238	He INJECTOR BOX CHECKOUT	16	-	-			-	-	-	-	-
T1160	ALL TANK SYSTEMS TEST	17	0.05	0.22			0.25	1.10	0.50	1	2
T6248	TANK INSPECTION	18	-	-			-	-	-	-	-
T1101	LH2/LO2 VENT DISCONNECT	19	-	-			-	-	-	-	-
V1011.03	ENGINE PUMP TORQUES	20	-	-			-	-	-	-	-
V1292	MPS TVC CHECKOUT	21	-	-			-	-	-	-	-
V1011		22	-	-			-	-	-	-	-
V1001	ENGINE I/F VERIFICATION	23	-	-			-	-	-	-	-
V1009/	MPS AND ENGINE INTEGRATION OPERATIONS	24	1.90	1.35	0.26	1.64	6.79	3.78	69.67	10	16
V1011		25	-	-			-	-	-	-	-
V1063	MPS FLIGHT CONTROL TVC CHECKOUT	26	-	-			-	-	-	-	-
	TOTALS =		17.76				53.80		590.95		

Figure 2.4-9 SIMtrix Data for SSTO Reference Conditions

NUMBER	OPF PROCESS RD-704 (7), & 6 HR MTTR		PRs/FLOW (MEAN)	PRs/FLOW S/D	FUNCT/ FLOW	DEFECTS/ FLOW	MDT (HOURS)	MDT S/D	MHRS/ FLOW	PERSONNEL (AVERAGE)	PERSONNEL (MAXIMUM)
V1011.03	ENGINE POST FLIGHT LOW PRESSURE PUMP TORQUES	1	-	-			-	-	-	-	-
V9019		2	1.64	1.30	0.12	1.52	6.29	4.19	68.52	10	16
V1011.03	ENGINE POST FLIGHT HIGH PRESSURE PUMP TORQUES	3	-	-			-	-	-	-	-
V1009.02	MPS ELECTRICAL SYSTEM CHECKOUT	4	0.01	0.10	0.01	0.00	0.16	1.60	0.32	1	2
	REMOVE ENGINES (IF REQUIRED)		0.40	0.64	0.40	0.00	0.60	0.64	7.20	12	14
V1009.01	MPS 17 INCH PRE-VALVE SCREEN	5	0.04	0.20	0.00	0.04	0.32	1.58	1.92	5	6
V1009.03	MPS He SYSTEM CHECKOUT	6	1.98	1.40	1.28	0.70	5.35	2.02	33.17	6	16
V1009.04	MPS LO2 SYSTEM CHECKOUT	7	4.30	1.92	0.91	3.39	12.46	5.26	156.11	12	16
V1009.05	MPS LH2 SYSTEM CHECKOUT	8	4.93	2.18	0.92	4.01	14.09	5.70	177.72	12	16
	INSTALL ENGINES PREVIOUSLY REMOVED		0.40	0.64	0.40	0.00	0.60	0.64	7.20	12	14
V1009/	MPS AND ENGINE CHECKOUT	9	8.17	2.84	0.84	7.33	23.42	7.81	332.11	12	16
V1171	MPS VALVE CONFIGURATION FOR ROLLOUT	10	0.14	0.38	0.06	0.08	4.06	14.37	16.70	4	6
T1109	LO2/LH2 LEAK TESTS	11	0.01	0.10			0.05	0.50	0.10	1	2
T1104/	ANCILLARY LEAK AND FLOW CHECKS	12	0.06	0.24			0.30	1.19	0.60	1	2
T1107		13	0.04	0.20			0.20	0.98	0.40	1	2
T5043	LH2 FIRE DETECTION INSTALLATION	14	0.01	0.10			0.16	1.60	0.64	3	4
T5136	LEAK PORT CLOSEOUT	15	0.01	0.10			0.05	0.50	0.10	1	2
T5238	He INJECTOR BOX CHECKOUT	16	-	-			-	-	-	-	-
T1160	ALL TANK SYSTEMS TEST	17	0.05	0.22			0.25	1.10	0.50	1	2
T6248	TANK INSPECTION	18	-	-			-	-	-	-	-
T1101	LH2/LO2 VENT DISCONNECT	19	-	-			-	-	-	-	-
V1011.03	ENGINE PUMP TORQUES	20	-	-			-	-	-	-	-
V1292	MPS TVC CHECKOUT	21	-	-			-	-	-	-	-
V1011		22	-	-			-	-	-	-	-
V1001	ENGINE I/F VERIFICATION	23	-	-			-	-	-	-	-
V1009/	MPS AND ENGINE INTEGRATION OPERATIONS	24	2.83	1.62	0.30	2.53	9.50	4.82	115.34	12	16
V1011		25	-	-			-	-	-	-	-
V1063	MPS FLIGHT CONTROL TVC CHECKOUT	26	-	-			-	-	-	-	-
	TOTALS =		25.02				77.86		920.65		
	PROCESSES 1, 2, 5, 6, 7 & 8 IN PARALLEL	1									
	PROCESSES 11-14, 16-18 IN PARALLEL	2									
	PROCESSES 9, 20, 22, 24 & 25 IN PARALLEL	3									

Figure 2.4-10 SIMtrix Data for Tri-Propellant SSTO

	PRs #	MDT (hrs)	UMMHR (hrs)	Total HPF Time (hrs)	Flight Rate	
Shuttle Reference with 7 SSMEs	60	499	4,493	539	7	Today's
IVHM to checkout subsystems in 40 hrs.	19.4	160	1,323	200	43	+ Testing
Improved designs to reduce remove/repair times (MTTR) to 6 hrs	19.6	55	600	95	51	+ Design
Improved component reliability (MTBF or MTBM) by 50%	16.5	51	535	90	51	+ Components

Figure 2.4-11 Technology Steps to Reach SSTO Operability

Mean Downtime Sensitivity Study to R&M Parameters

	PRs #	MDT (hrs)	UMMHR (hrs)	Total HPF Time (hrs)	Flight Rate
SSTO Reference with IVHM	19.9	164	1,380	204	43
Improved Reliability only (MTBF)	17.7	150	1,248	190	43
Improved Maintainability only (MTTR)	19.8	55	605	95	51
Improved Reliability & Maintainability	17.7	54	590	94	51

Figure 2.4-12 SSTO Operability Sensitivity Study

2.4.1.6.2 STARSIM Simulation Results

These SIMtrix simulations of MPS ground checkout operations were used to drive the STARSIM model of the SSTO, a complete system-level simulation of SSTO launch capability, facility requirements, and resource utilization. Prior to the SIMtrix results being available, the STARSIM model was used to perform a basic sensitivity analysis of the annual SSTO flight rate capability. The model varied several operational parameters such as the interval and duration of SSTO maintenance down periods (OMDP), fleet size (number of SSTO vehicles), and the mean HPF turnaround time.

The results of the STARSIM sensitivity analysis showed that the planned SSTO flight rate (43 flights/year) is easily achievable under the current groundrules. A mean (average) flight rate of 53 per year was achieved even with the launch pad delays included (a yearly average of 11 rollbacks were predicted at this flight rate). This level of flight activity resulted in 2.5 OMDPs per year and a net direct labor consumption of 10,000 manhours per launch. The two launch pads were utilized at a mean level of 75% , the HPFs (one per vehicle) at 54%, and the mission control center at 87%.

The effects of certain of the Access to Space Option 3 groundrules were modeled from this Reference SSTO simulation. The OMDP period was changed from once every 20 flights to one in ten, fifteen, and twenty. The duration of the OMDP period was also changed from 90 days to 60, 120, and 180 days. This rather wide range of OMDP variables was found to have a weak effect on the SSTO annual flight rate. Over the range of these two variables (each one taken independently), the minimum mean flight rate was 50 flts/yr and the maximum mean flight rate was 58 flts/yr. This results indicates that considerable margin exists in this scenario for scheduled depot maintenance and/or modification for the SSTO vehicles. The resulting cost per flight (as measured in direct labor manhours per launch) of these OMDPs was more significant, however. The annual direct manhours to both process the vehicles for launch and to perform the OMDP maintenance varied by +45% to -20% over the range of these OMDP variables. Thus, while the SSTO flight rate is not overly sensitive to the depot maintenance requirements, the SSTO cost per flight is.

The effect of a longer than ground ruled HPF processing time had a direct and strong effect on the SSTO flight rate capability and on the manhours per flight. The subsystem processing (checkout) time was varied from the ground ruled 40 hours to much longer times (up to a mean time of 480 hours, or 1200%). This resulted in the SSTO mean flight rate dropping to as low as 28 flights per year, with a resultant 250% increase in cost (manpower) per launch. The flight rate correlation with HPF processing time was not linear, and the rate did not drop below 50 flts/yr until the HPF time increased to 120 hours. The planned 43 flts/yr was not met when HPF times increased to approximately 220 hours. At the highest processing time simulated (480 hrs), the HPF mean utilization rate was 85%. This level indicates that the HPF processing itself, not the facility, was the constraint on flight rate. While the processing times had the strongest effect seen in the model on per launch manpower (cost), the effect was not direct. The processing times increased by a factor of 12, whereas the manpower per launch only increased by a factor of 2.5. This effect is caused in part by the reduced number of OMDPs performed each year (from 2.5 to 1) because of the reduced flight rate.

The HPF mean downtime predictions from the SIMtrix model all fell within the range of the STARSIM HPF processing sensitivity analysis (40 to 480 hours). The extrapolation of the SIMtrix maintenance analysis to the net SSTO flight rate capability was straightforward. The reference SSTO MPS maintenance estimate (Step 4 of the Simtrix sequence) translates into an annual SSTO mean flight rate of 51 flights per year. This is well above the SSTO baseline of 43 flights per year. In fact, this baseline flight rate can be met even at the Step 2 simulation. That is, even with the current Shuttle MPS component reliability levels, the current maintenance repair times, and the current launch delay/scrub rates, a five SSTO vehicle fleet could provide an average of 43 flights per year. This finding is based, however, on the SSTO ground processing and facilities being capable of checking out the systems within the 40 hour allocation and of performing a 12 hour launch pad countdown.

One interesting finding from the analysis was the observation that the SSTO launch rate was at some times being constrained by the number of vehicles which could be in space at any one time (one). This is a current Shuttle program constraint which may or may not apply to the SSTO program, but this result at least indicates increased facility and personnel requirements which would be required of NASA operations centers.

Case/Parameter	Launches	OMDPs	Rollbacks	HPF Utiliz.	Pad Utiliz.	MissionCntr	Direct labor	OMDP labor	Launch labor	Launch/OMPD	Labor/Launch
	# / yr	# / yr	# / yr	(%)	(%)	(%)	(K Mhrs)	(Manhrs)	(Manhrs)	labor ratio	(K Mhrs)
BASELINE	**53.6**	**2.5**	**11.6**	**54**	**75**	**87**	**536**	**270000**	**266000**	**0.99**	**10.00**
OMDP Interval											
every 10 flts	51.7	5.2	10.8	63	56	84	745	561600	183400	0.33	14.41
every 15 flts	54.8	3.7	11.8	57	68	89	619	399600	219400	0.55	11.30
Reference = every 20 flts	53.6	2.5	11.6	54	75	87	536	270000	266000	0.99	10.00
every 25 flts	57.4	2.2	12.3	52	79	93	495	237600	257400	1.08	8.62
OMDP Duration											
60 workdays	58	2.8	13.1	51	82	94	471	201600	269400	1.34	8.12
Reference = 90 workdays	53.6	2.5	11.6	54	75	87	536	270000	266000	0.99	10.00
120 workdays	53.2	2.6	11.6	57	69	87	607	374400	232600	0.62	11.41
180 workdays	50.5	2.5	10.7	63	57	81	730	540000	190000	0.35	14.46
Fleet Size											
OMDP every 10 Flts											
5 Vehicles	51.7	5.2	10.8	63	56	84	745	561600	183400	0.33	14.41
6 Vehicles	55.7	5.4	11.8	63	72	90	807	583200	223800	0.38	14.49
7 Vehicles	59.2	5.8	13.2	63	86	96	844	626400	217600	0.35	14.26
8 Vehicles	60	5.9	12.4	66	91	97	868	637200	230800	0.36	14.47
9 Vehicles	60.5	5.9	12.7	69	95	98	893	637200	255800	0.40	14.76
OMDP every 20 Flts											
3 Vehicles	43	2.3	9.4	59	32	69	428	248400	179600	0.72	9.95
4 Vehicles	52.1	2.6	11.2	54	55	84	511	280800	230200	0.82	9.81
Reference = 5 Vehicles	53.6	2.5	11.6	54	75	87	536	270000	266000	0.99	10.00
6 Vehicles	58.9	2.8	12.6	57	86	95	560	302400	257600	0.85	9.51
7 Vehicles	59.8	3	12.7	62	92	97	588	324000	264000	0.81	9.83
8 Vehicles	60.8	2.8	13	66	95	98	594	302400	291600	0.96	9.77
9 Vehicles	60.9	2.9	12.7	69	96	99	617	313200	303800	0.97	10.13
HPF Processing Time											
Reference = 40 hours	53.6	2.5	11.6	54	75	87	536	270000	266000	0.99	10.00
80 hours	51.5	2.6	10.9	58	67	84	596	280800	315200	1.12	11.57
120 hours	50.4	2.6	10.6	63	56	81	643	280800	362200	1.29	12.76
160 hours	47.9	2.4	9.9	68	46	77	669	259200	409800	1.58	13.97
240 hours	40.8	2.4	8.5	76	30	66	718	259200	458800	1.77	17.60
320 hours	35.5	1.9	7.8	80	24	58	720	205200	514800	2.51	20.28
480 hours	28.7	1.1	5.8	85	17	46	716	118800	597200	5.03	24.95
HPF Workforce											
12.5/shift	53.6	2.5	11.6	54	75	87	268	90000	178000	1.98	5.00
Reference = 25/shift	53.6	2.5	11.6	54	75	87	536	270000	266000	0.99	10.00
37.5/shift	53.6	2.5	11.6	54	75	87	804	270000	534000	1.98	15.00
50/shift	53.6	2.5	11.6	54	75	87	1072	360000	712000	1.98	20.00
SIMtrix Data Runs											
Shuttle Benchmark	7.8	0.72	*0.06*	96	40	13					
Step 1 (Shuttle w/7 SSME)	7.1	0.7	*0.09*	97	37	12					
Step 2 (SSTO timelines)	43.8	2.3	9.6	81	24	56	441	248400	192600	0.78	10.07
Step 3 (SSTO repair times)	50.9	2.6	11	60	64	82	537	374400	162600	0.43	10.55
Step 4 (SSTO@150% MTBF)	**51.1**	**2.6**	**11**	**60**	**65**	**82**	**545**	**374400**	**170600**	**0.46**	**10.67**

Figure 2.4-13 STARSIM Simulation Sensitivity Study

2.4.1.7 SSTO MPS Operability Analysis Findings

The data generated by the SIMtrix and STARSIM simulations of SSTO MPS operations has provided a comparison of competing SSTO concepts from an operability standpoint and has also identified the relative importance of several technology pursuits relative to the SSTO program. The analysis has identified the time to perform the subsystem test and checkout as the most important factor for reducing turnaround times and costs. By drastically reducing the test time of flight hardware, equipment operating times are reduced and the number of failures (PRs) decrease accordingly. The next most significant factor was the reduction of time to remove and replace (or just to repair) a defective component. This factor directly reduces the maintenance time (MDT) and labor (UMMHR) to return the vehicle to an operational condition. These two factors both result in shortening the total time the SSTO is in the processing facility. A direct link between processing time and the number of maintenance actions has been demonstrated on the current Shuttle program, and is also found in other programs (e.g. the X-15 program). The effect of improving flight hardware reliability was not found to be a strong factor in improving SSTO operability.

The total time the SSTO is in its processing facility was the most important factor in achieving high flight rates. The effect of maintenance down periods (OMDP) was found to not be a strong factor in achieving high flight rates, but did affect the SSTO operations costs. Significant variations in both the frequency and time to perform OMDP maintenance can be tolerated without reducing the annual flight rates. The additional labor required to perform the maintenance, however, is directly related to the time and frequency of these events.

Comparison of a tri-propellant propulsion system concept with the reference (LH2/LOX propulsion) SSTO found that either concept can achieve the SSTO flight rate objectives, but higher maintenance costs should be expected with the tri-propellant design. Even with the 3-engine RD-701 concept, higher maintenance costs were found than the 7-engine SSME concept. The components added for the additional fuel system generally required higher maintenance times than those added for a higher number of engines.

In summary, these simulations have demonstrated that low SSTO operations costs can best be achieved by reducing the time to test and checkout vehicle subsystems and by reducing component repair times. High SSTO flight rates can also best be achieved by reducing the per flight maintenance times. Improved component reliability and periodic maintenance effects are weaker.

Operability of Alternative SSTO Concepts

	PRs #	MDT (hrs)	UMMHR (hrs)	Total HPF Time (hrs)	Flight Rate
Reference SSTO with 7 SSME engines	20	55	535	95	51
Tri-propellant SSTO with 3 RD-701 engines	16	55	577	95	51
Tri-propellant SSTO with 7 RD-704 engines	25	78	921	108	50

Figure 2.4-14 SIMtrix Simulations Summary

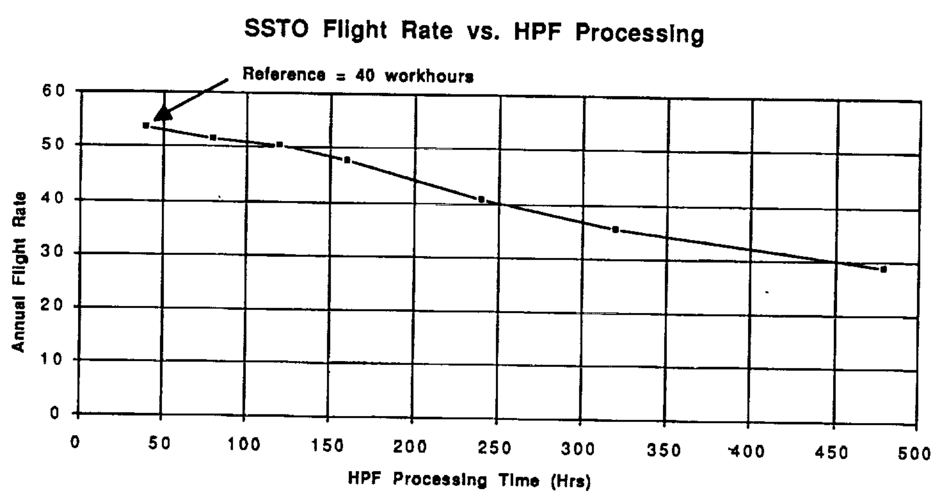


Figure 2.4-15 STARSIM Simulations Summary

2.4.2 SSTO Engine Throttling Studies

A key factor in any SSTO launch vehicle's maintenance characteristics is the amount of time the main propulsion system engines have to operate on each mission. Rocket engines generally have a short operating life between maintenance. Any SSTO design parameter which can minimize the engine operating time, or even the amount of time the engine is operated at full power levels, can be translated into reduced MPS maintenance costs. An analysis was performed to determine what engine throttling and shutdown schemes might be devised to minimize total engine operating time and engine operating times at full throttle for a reference SSTO concept. The Access to Space Option 3 reference SSTO rocket concept utilized a 7 SSME configuration main propulsion system. Concepts with a large number of engines such as this typically have to shutdown and/or throttle engines during the launch trajectory. Changes to the engines operating schemes may, however, detract from the vehicle's payload lift capability and its abort capabilities. Several engine operating schemes were investigated for the 7 SSME SSTO concept, each evaluated for total engine operating time, time at full and reduced power levels, payload impacts, and also for single or two engine out abort capability. The tri-propellant RD-701 propulsion system SSTO concept was also evaluated for comparison with the reference SSTO configuration.

2.4.2.1 SSTO Engine Throttling Analyses

Trajectory analyses of the reference SSTO launch vehicle (7-SSME propulsion system) were performed to determine optimum operating techniques for maximizing both engine reliability/life and maintaining adequate abort coverage. Variations in engine throttle profiles and shutdown sequences were performed to find the minimum engine operating times, the minimum engine operating time at 100% throttle level, and the maximum engine out abort capability. The analyses were performed with the POST trajectory analysis tool using 3-DOF trajectory simulation with a moment balance maintained in the vehicle pitch plane (consistent with other SSTO performance analyses performed in the Option 3 studies). Six different engine operating techniques were evaluated:

- A) Constant acceleration (maintain 3 g),
- B) Ramp throttle to 65%,
- C) No throttle,
- D) Step throttle to 65%,
- B.1) Ramp throttle 5 engines, leave two at 90%

B.2) Ramp throttle 5 engines, leave two at 100%

These throttle profile variations produced little change in the vehicle payload performance and all resulted in the eventual shutdown of five engines to meet the 3g acceleration limit. The time to final engine shutdown (MECO) varied from 372 seconds to 412 seconds among these launch profiles. The lowest performance was found with the no throttle (-1,000 lbs) and the step throttle (-1,950 lbs) profiles. The analysis of these throttle profile variations showed that total engine operating times (seven engines combined total burn time) could be reduced (or increased) by approximately 10%. A range of about 400 seconds from maximum to minimum total operating times was found over the nominal 2200 seconds total engines operating time. This is not a great variation for a single mission, but when applied to a planned 20-mission life between engine removals, this translates to an equivalent of two additional missions before planned removal of the seven engines, which is a significant maintenance improvement.

A much greater variation was found in total operating time at 100% for the engines. A range from nearly 2,000 seconds down to 1,230 seconds was found. Engine life and reliability may be significantly affected by these different operating techniques for the SSTO concept. The specific engine selection and its design characteristics will determine if this effect can be translated into maintenance savings.

The engine gimbal profile was examined for the Case B.1 engine throttle scenario to determine if the SSTO shutdown sequences would cause excessive gimbal ranges. The analysis was performed with the POST trajectory code and was based on the specific engine installation layout which is described in Section 2.4.3. (7 SSMEs in a circular arrangement near the fuselage outer diameter, no engine(s) installed in the center of the fuselage). The engines were required to gimbal to maintain the vehicle pitch balance and steer it to the MECO target while performing the engine throttle and shutdown sequences. The analysis showed that the maximum engine gimbal was 6° with a mean gimbal angle of 5° (±1°) for the high dynamic pressure periods of the trajectory (approximately 30 to 130 seconds after liftoff). During the later trajectory periods, the gimbal angle was a relatively constant 2.9°. This analysis indicates that the engine throttling and shutdown sequences will not create excessive gimbal ranges to be required, even for the asymmetric throttling scenario.

Another significant finding from this analysis was the observation that most of the throttle/shutdown techniques resulted in an engine shutdown within 10 to 20 seconds of MECO. At this time in the trajectory, very little propellant is left in the tanks (approximately 6%) and the engine shutdown results in an acceleration change from 3 g's to as low as 1.3 g's. This condition will cause significant propellant movement in the tank, potentially uncovering the tank outlet and shifting the CG of the vehicle. (Remember, at this point in the trajectory the vehicle is flying nearly horizontal, not vertical.) To prevent an engine shutdown late in the trajectory, two variations of the ramp throttle approach (Case B) were created. Two asymmetric throttle variations (Case B.1 and B.2) were devised in which five of the seven engines were allowed to be throttled but two engines were kept at 100% (or 90%) thrust until the end of the trajectory. This method allowed throttling of the last two engines to maintain smoother acceleration levels during the final minute of the trajectory and prevent large propellant shifts due to engine shutdowns. The Case B.1 was most successful, resulting in no engine shutdown for the last 60 seconds and producing constant 3 g's acceleration during the last 10 seconds.

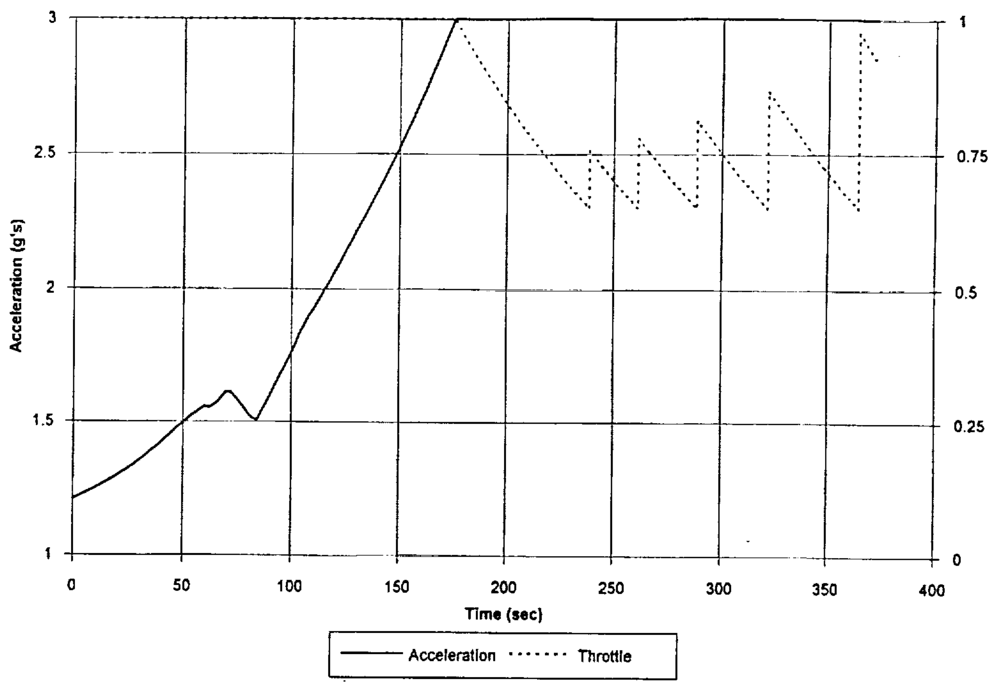


Figure 2.4-16 SSTO Engine Throttling - Case A

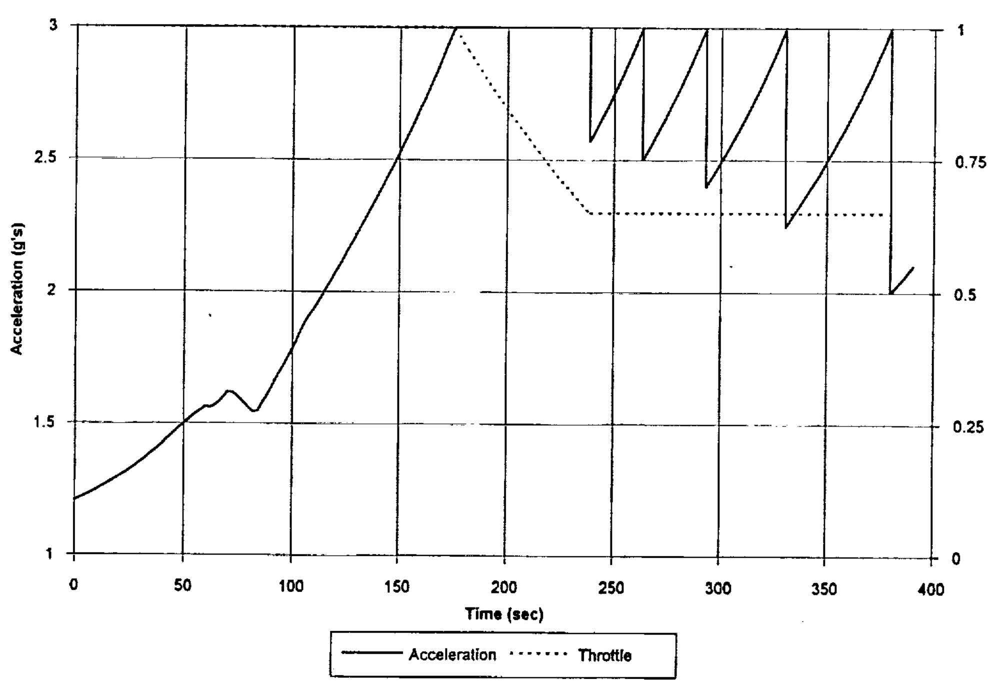


Figure 2.4-17 SSTO Engine Throttling - Case B

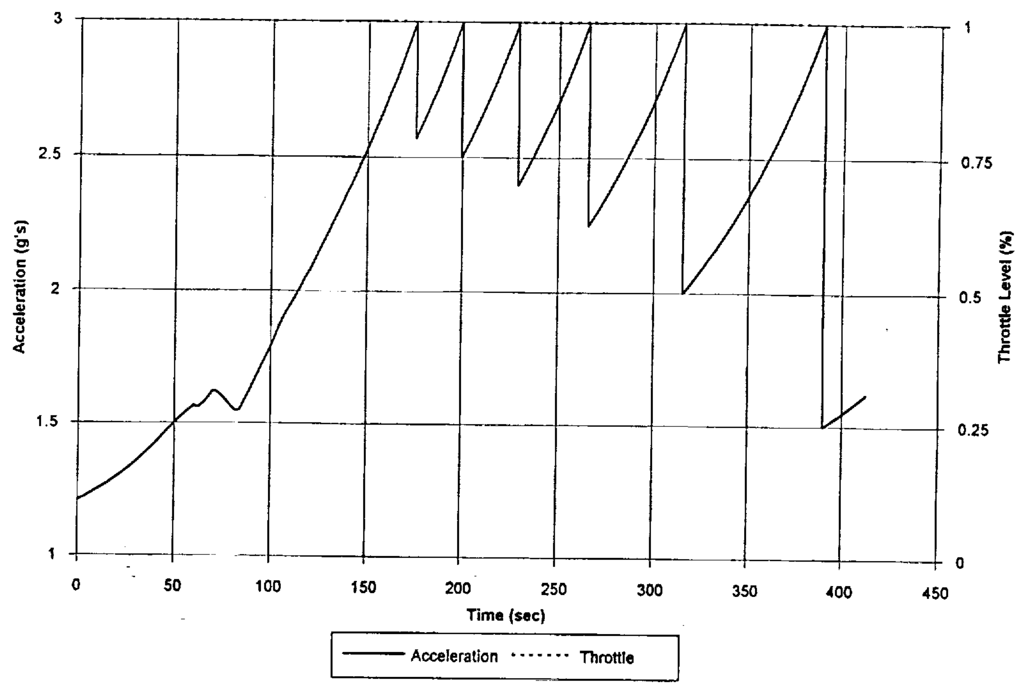


Figure 2.4-18 SSTO Engine Throttling - Case C

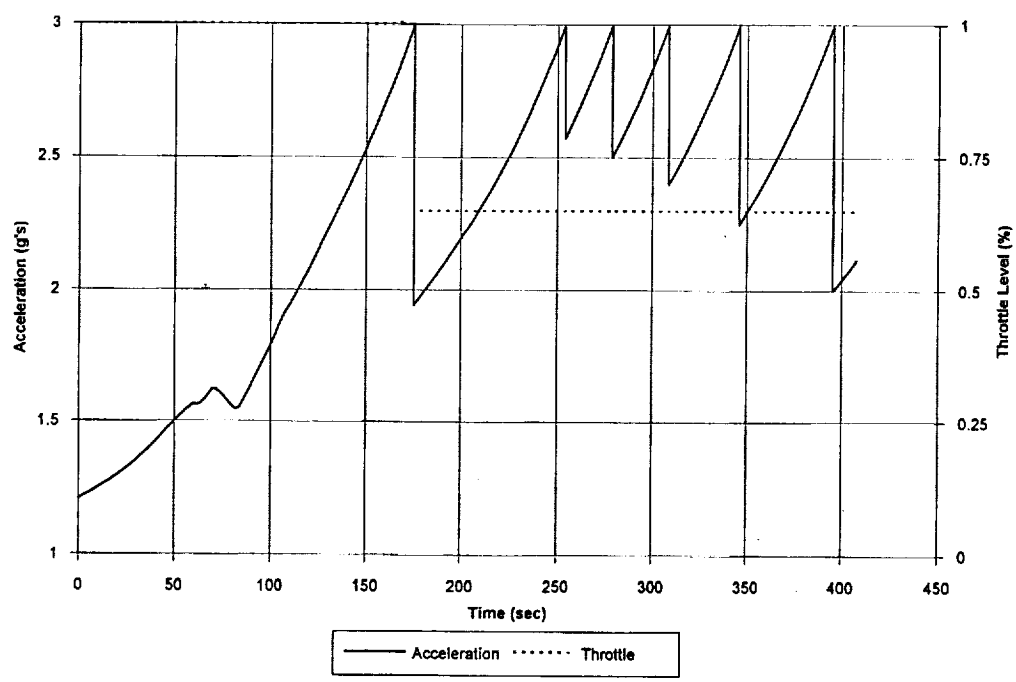


Figure 2.4-19 SSTO Engine Throttling - Case D

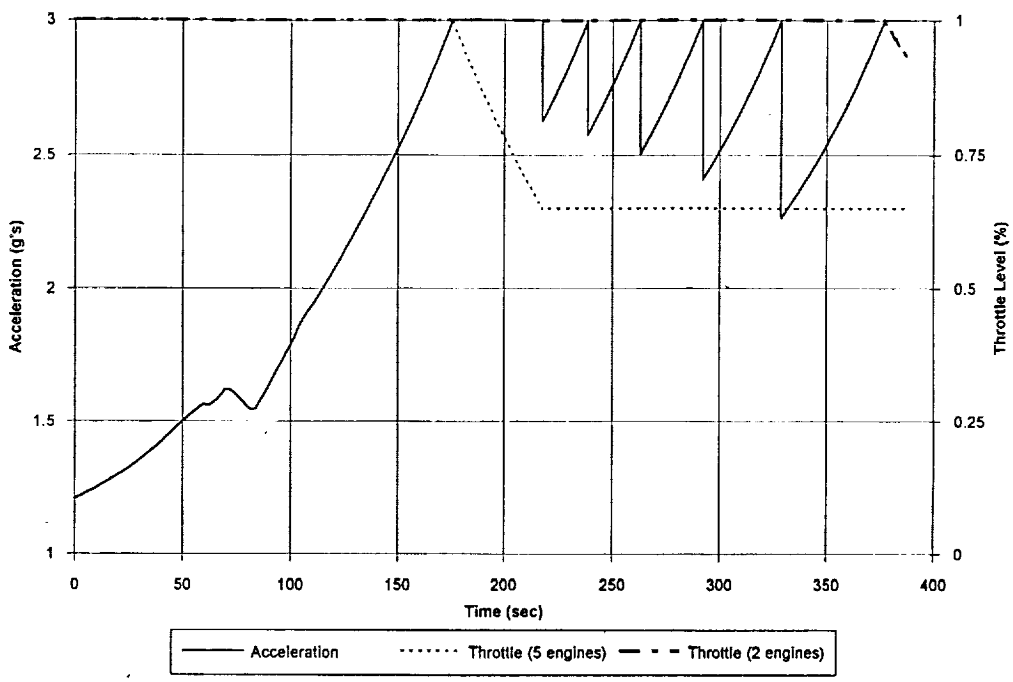


Figure 2.4-20 SSTO Engine Throttling - Case B.1

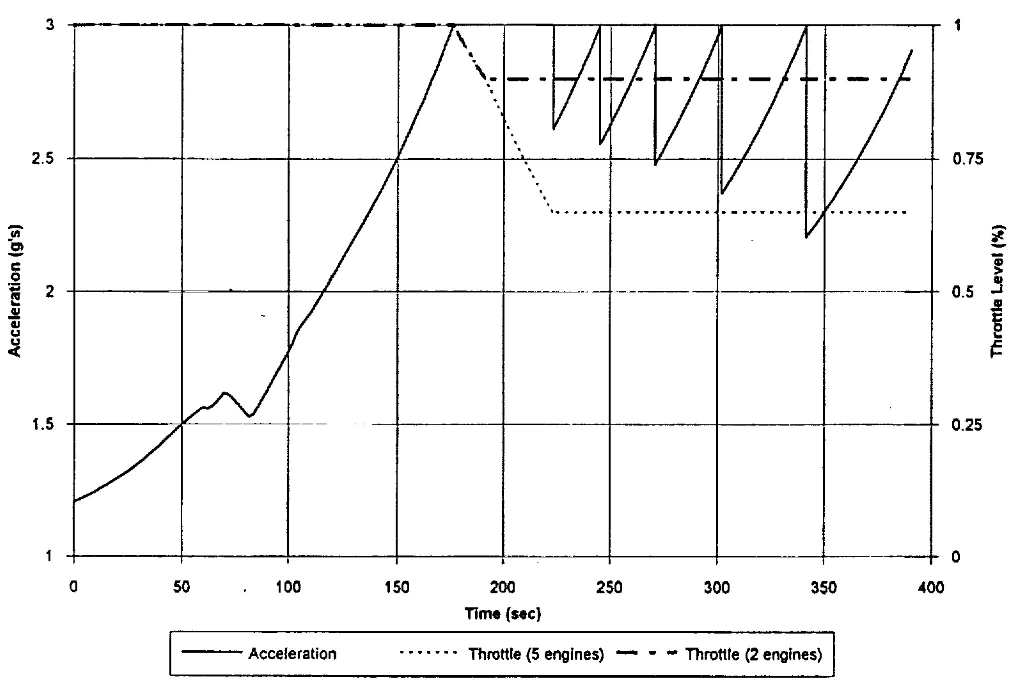


Figure 2.4-21 SSTO Engine Throttling - Case B.2

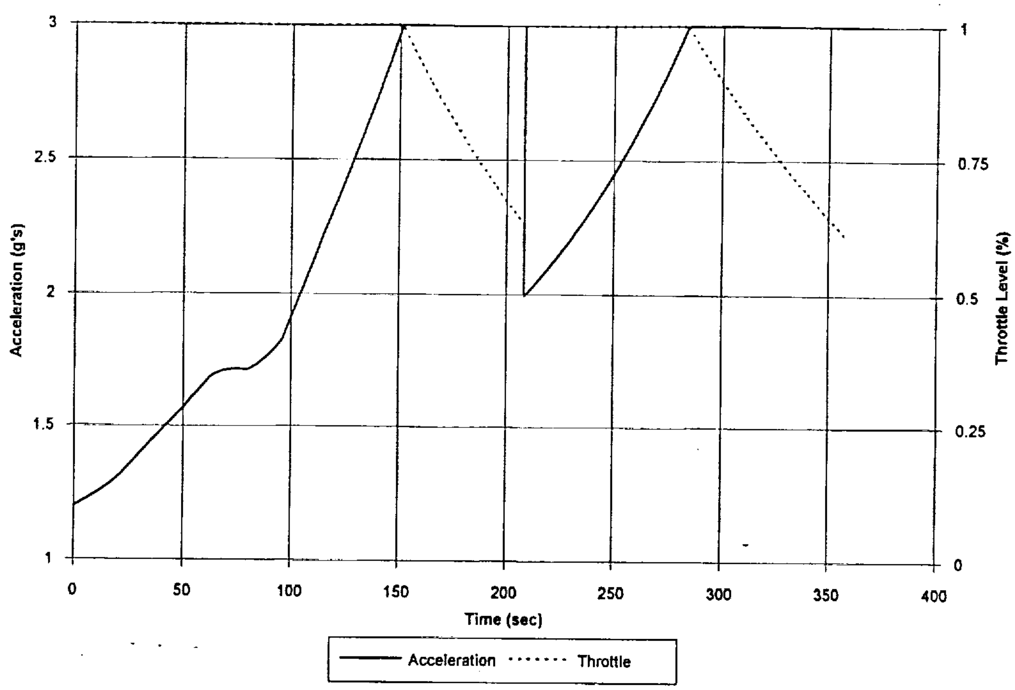


Figure 2.4-22 SSTO Engine Throttling - 3 RD-701 Engines

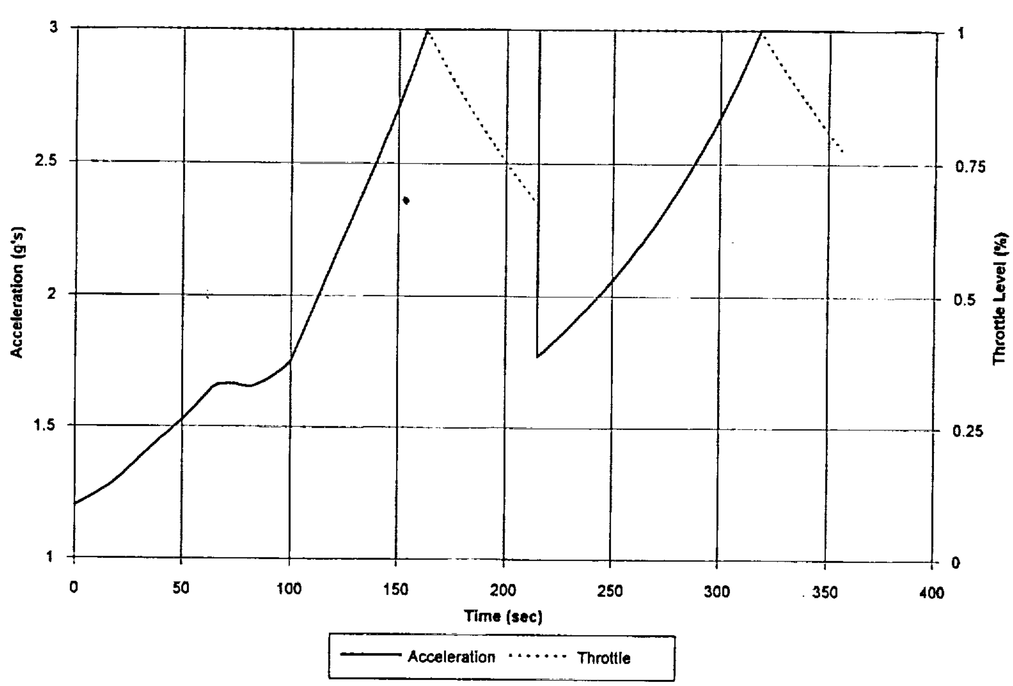


Figure 2.4-23 SSTO Engine Throttling - 7 RD-704 Engines

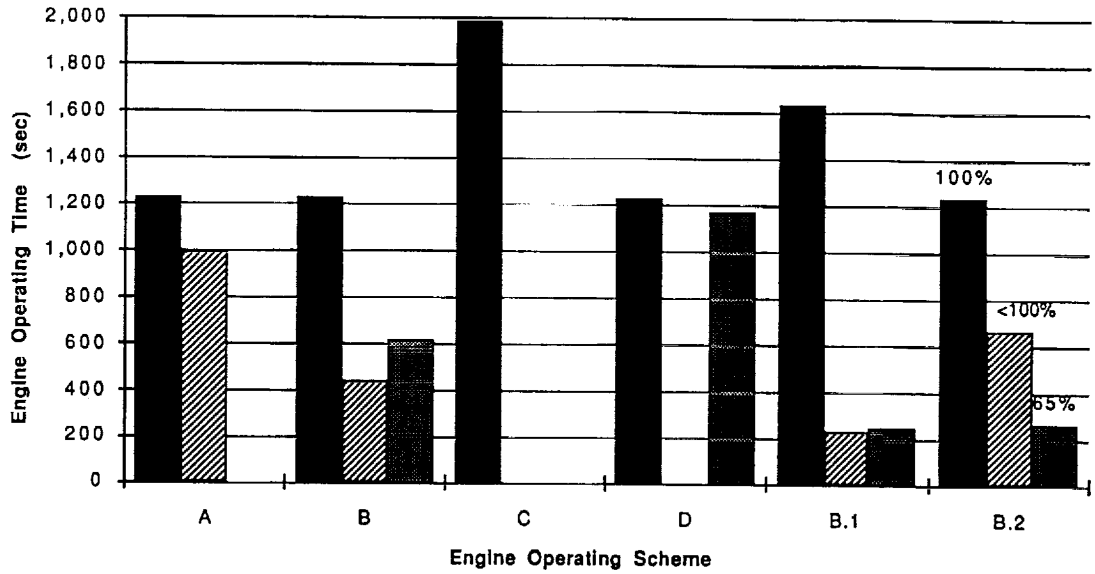


Figure 2.4-24 Engine Thrust & Operating Time Summary

2.4.2.2 SSTO Engine Out and Abort Analyses

Engine out and abort analyses were also conducted for these engine throttling techniques to determine if they would improve (or reduce) the SSTO abort capabilities. Both single engine out and two engine out aborts were analyzed. Engine throttling capability again improved the SSTO vehicle performance, but only under certain scenarios. A 30 second improvement was found with Case B.1 (asymmetric throttle, 100%) for the single engine out condition (22 seconds for 2 engines out). If the reserve propellant budgets are allowed to be consumed for the engine out conditions (a 1,000 lb propellant reserve just for aborts is a reasonable groundrule), all throttle profiles produced similar results for a single engine out condition. For 2 engine out conditions, the no throttle scenario proved best. Again, however, only the asymmetric throttle profiles produced smooth acceleration profiles near MECO.

The SSTO abort analysis produced similar results as was found in the Access to Space Option 3 study. The RTLS trajectory analyses included aerodynamic loads limits (maximum dynamic pressure and maximum normal force) as well as aeroheating limits on the SSTO vehicle (data as provided from the NASA Option 3 study). The capability exists to perform an RTLS from liftoff to as late as 207 seconds into the launch trajectory. Single engine out capability was available as early as 141 seconds into the trajectory, with two engine out capability at 205 seconds. An ATO abort capability is not required since the EO coverage overlaps with the RTLS coverage.

The combined RTLS and engine out capabilities of this SSTO vehicle provide full abort coverage, a runway landing option is available over the entire launch trajectory. The vehicle can achieve the nominal MECO target after a single engine failure during 64% of the trajectory.

SSTO Abort Capability

LOX/LH2 Propulsion (7 SSME)

51.6° Inclination Orbit

SSTO ABORT OPTIONS	Abort Window
• **RETURN-TO-LAUNCH-SITE (RTLS)** Any engine failure in booster. Use remaining engines to execute powered turnaround maneuver and establish sufficient velocity toward launch site for SSTO to land.	0 sec ≤ MET ≤ 207 sec
• **SINGLE ENGINE OUT** Single booster engine failure. Utilize remaining engines to reach nominal MECO target.	141 sec ≤ MET ≤ 388 sec
• **TWO ENGINE OUT** Two booster engine failures. Utilize remaining engines to reach nominal MECO target.	205 sec ≤ MET ≤ 388 sec
• **TRANS-ATLANTIC LANDING (TAL)** Multiple (>2) booster engine failure. Utilize remaining engines to establish sufficient velocity for SSTO range to reach landing site in Europe or Africa.	GROUND-RULED OUT
• **ABORT TO ORBIT (ATO)** Multiple booster engine failure. Utilize remaining engines to reach lower energy MECO target for 13X74 Nmi. parking/transfer orbit.	100 sec ≤ MET ≤ 205 sec

Figure 2.4-25 SSTO Abort Capability (7 SSME Engines)

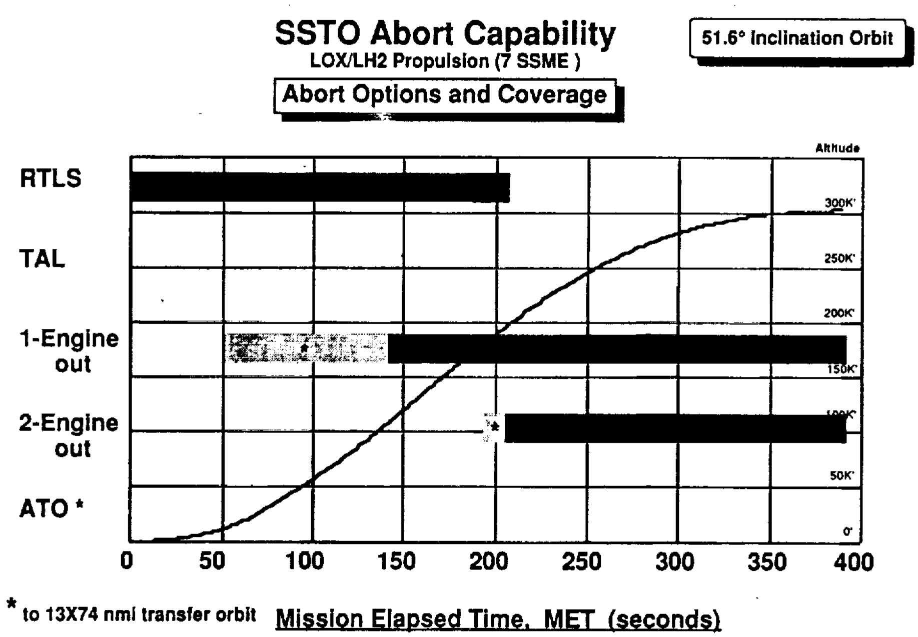


Figure 2.4-26 SSTO Abort Coverage (7 SSME Engines)

2.4.2.3 Tri-propellant SSTO Configuration Analysis

Trajectory and launch abort analyses of a tri-propellant SSTO propulsion system utilizing three Russian RD-701 engines were also completed. The analyses were similar to those performed for the seven SSME engine propulsion concept for SSTO. The fewer number of engines, coupled with their dual thrust level, was found to eliminate the need for engine throttling studies as was performed for the SSME concept. What was discovered, however, was that this concept has very limited engine out and RTLS abort capabilities. Because these engines have two thrust chambers per engine, they experience large thrust losses for an engine failure (the engine failures modeled assumed the two thrust chambers could not operate independently). A single engine out capability was not achievable until 336 seconds into the trajectory (nominal MECO occurs at 373 sec.). Two engines out could not be tolerated at any time. An allowance of 1,000 lbs reserve propellant just for the engine out condition extended the engine out capability back to 310 seconds. The RTLS capability was greatly reduced by the significant thrust loss of an engine early in the trajectory. The RTLS could not be performed for an engine failure any earlier 37 seconds nor any later than 189 seconds after liftoff. An ATO abort is required to cover the gap (121 seconds) between RTLS and EO aborts . A single engine failure can not be tolerated early in the trajectory and a two engine failure cannot be tolerated at any time. The combined RTLS, EO, and ATO abort capabilities for this concept result in less than full abort coverage (90%). This level of abort coverage is marginally acceptable for a fully reusable launch vehicle.

The same tri-propellant SSTO concept with seven, single-nozzle engines was also performed (the RD-704 engine concept). The Case B.1 engine throttling profile which seemed the best solution for the SSME concept was analyzed. This version of the tri-propellant engine SSTO design produced a much improved abort performance. The RTLS capability was extended back to liftoff and out to 198 seconds. A single engine out capability was achievable at 262 seconds into the trajectory, and two engine out abort capability was achievable at 265 seconds (nominal MECO occurs at 390 seconds). An ATO abort mode was still required to span the gap (64 seconds) between RTLS and EO abort coverages. The EO abort performance of this concept is not as good as the 7 SSME propulsion system (33% vs. 64% of the trajectory), but at least this configuration achieves the full abort coverage which the RD-701 configuration could not.

SSTO Abort Capability
Tri-propellant Propulsion (3 RD-701)

51.6° Inclination Orbit

SSTO ABORT OPTIONS

Option	Abort Window
• **RETURN-TO-LAUNCH-SITE (RTLS)** Any engine failure in booster. Use remaining engines to execute powered turnaround maneuver and establish sufficient velocity toward launch site for SSTO to land.	37 sec ≤ MET ≤ 189 sec
• **SINGLE ENGINE OUT** Single booster engine failure. Utilize remaining engines to reach nominal MECO target.	310 sec ≤ MET ≤ 373 sec
• **TWO ENGINE OUT** Two booster engine failures. Utilize remaining engines to reach nominal MECO target.	NOT POSSIBLE
• **TRANS-ATLANTIC LANDING (TAL)** Multiple (>2) booster engine failure. Utilize remaining engines to establish sufficient velocity for SSTO range to reach landing site in Europe or Africa.	GROUND-RULED OUT
• **ABORT TO ORBIT (ATO)** Single booster engine failure. Utilize remaining engines to reach lower energy MECO target for 12X74 Nml. parking/transfer orbit.	180 sec ≤ MET ≤ 310 sec

Figure 2.4-27 SSTO Abort Capability (3 RD-701 Engines)

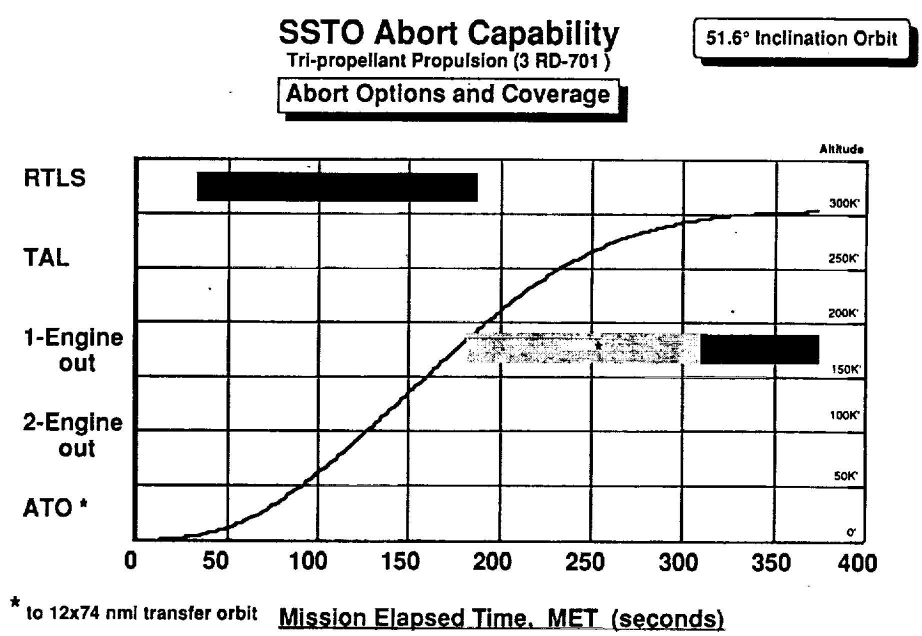


Figure 2.4-28 SSTO Abort Coverage (3 RD-701 Engines)

SSTO Abort Capability

Tri-propellant Propulsion (7 RD-704)

51.6° Inclination Orbit

SSTO ABORT OPTIONS

SSTO ABORT OPTIONS	Abort Window
RETURN-TO-LAUNCH-SITE (RTLS) Any engine failure in booster. Use remaining engines to execute powered turnaround maneuver and establish sufficient velocity toward launch site for SSTO to land.	0 sec ≤ MET ≤ 198 sec
SINGLE ENGINE OUT Single booster engine failure. Utilize remaining engines to reach nominal MECO target.	262 sec ≤ MET ≤ 390 sec
TWO ENGINE OUT Two booster engine failures. Utilize remaining engines to reach nominal MECO target.	265 sec ≤ MET ≤ 390 sec
TRANS-ATLANTIC LANDING (TAL) Multiple (>2) booster engine failure. Utilize remaining engines to establish sufficient velocity for SSTO range to reach landing site in Europe or Africa.	GROUND-RULED OUT
ABORT TO ORBIT (ATO) Multiple booster engine failure. Utilize remaining engines to reach lower energy MECO target for 15X80 Nmi. parking/transfer orbit.	135 sec ≤ MET ≤ 265 sec

Figure 2.4-29 SSTO Abort Capability (7 RD-704 Engines)

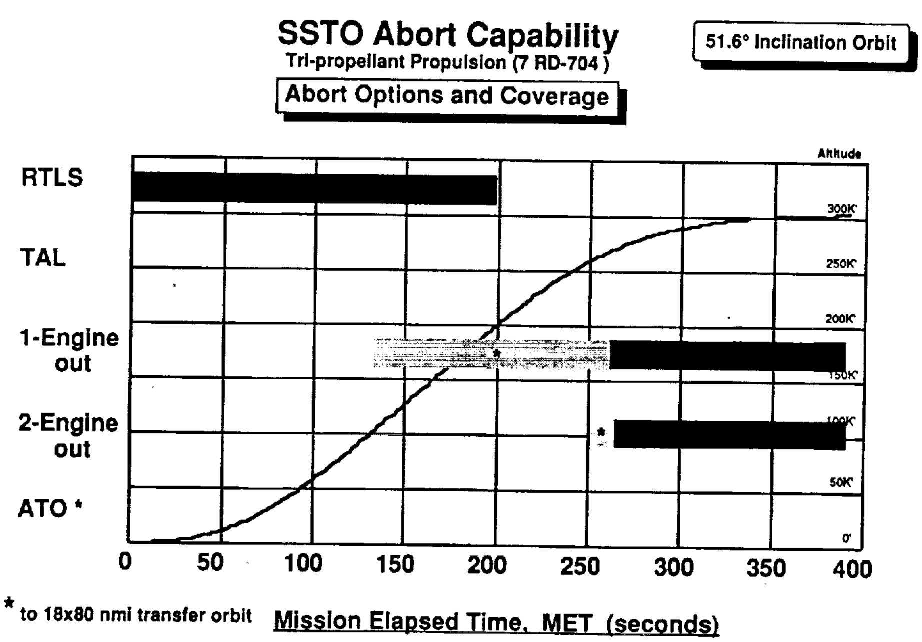


Figure 2.4-30 SSTO Abort Coverage (7 RD-704 Engines)

2.4.3 SSTO MPS Design Layout

A key aspect of ground processing is access to MPS components to perform necessary maintenance and inspections. A design concept for an open boattail was initiated to support the SSTO MPS ground processing study. Many MPS design groundrules were identified in the Operationally Efficient Propulsion System Study (OEPSS) by NASA KSC and Rocketdyne which would provide significant improvements in propulsion system processing costs. An open boattail was one of the key parameters identified in the study.

A design layout study was performed using 3-D CAD tools to vigorously apply the OEPSS groundrules. The design provided for a seven SSME propulsion system and included features such as no closed compartments, hardware integration, accessibility, no heat shields, lift-off umbilicals, and hardware commonality were incorporated. The design layouts currently include engine envelopes, LOX and LH2 feedlines, and an integrated tank/thrust structure arrangement. The design concept is similar to that employed on the Saturn S-II and S-IVB stages in which the engine thrust structure is integrated with the tank lower bulkhead. The design also includes modular engine assemblies which integrate the engine with the TVC system and portions of the thrust structure. No closed compartments exist in the propulsion system region and considerable access is provided for engine and feedline maintenance. A three-point structural attachment was developed for the engine module to accommodate rapid engine replacement.

A rocket propulsion based SSTO(R) as defined in NASA's Access to Space Option 3

- VTHL
- LOX/LH2 propellants
- 7 evolved SSME engines
- Forward LOX tank with two 19" feedlines, toroidal manifold
- LH2 tank with spider manifold
- Electromechanical actuators
- Hot gas tank pressurization

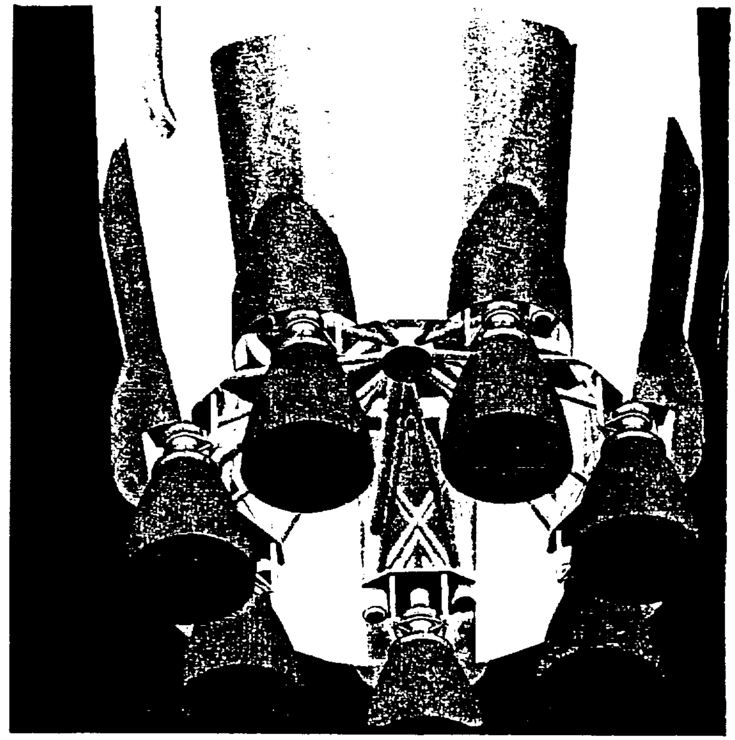


Figure 2.4-31 SSTO MPS Design Layout (7 SSMEs)

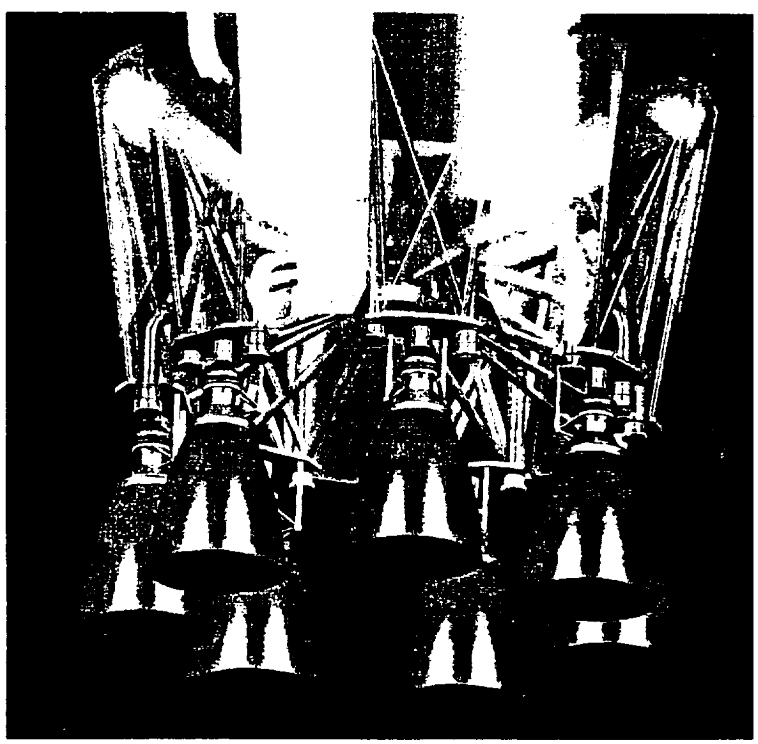


Figure 2.4-32 SSTO MPS Internal Components

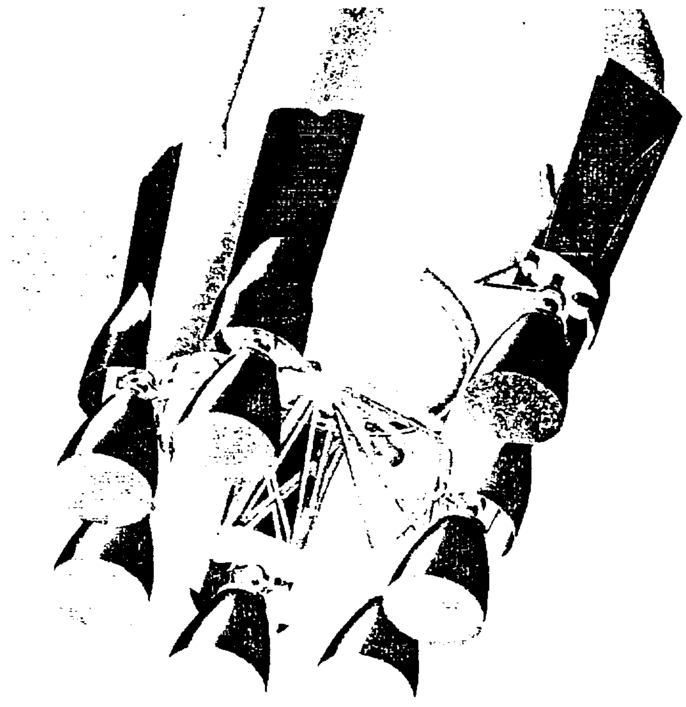


Figure 2.4-33 Modular Engine & Structure Integration

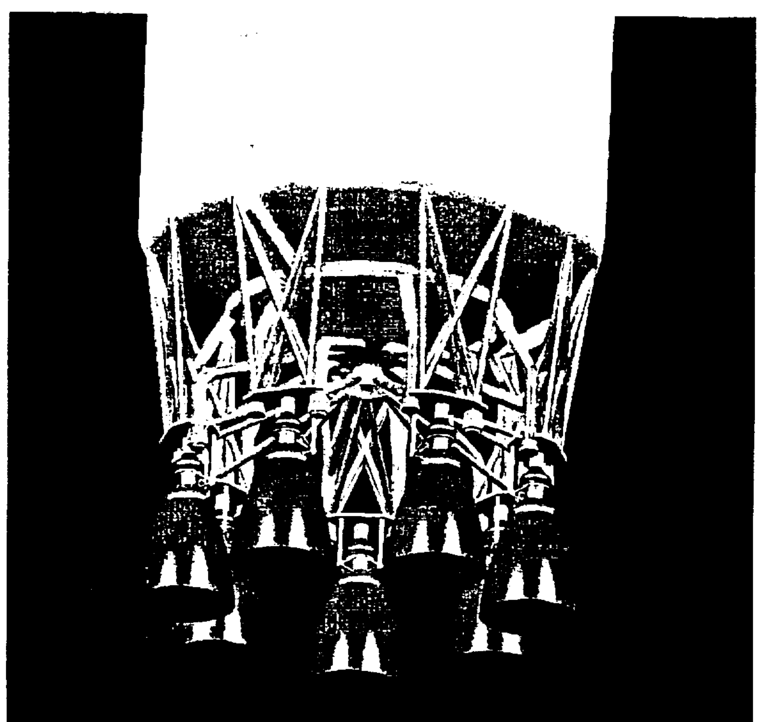


Figure 2.4-34 MPS Engines, Feedlines, & Thrust Structure

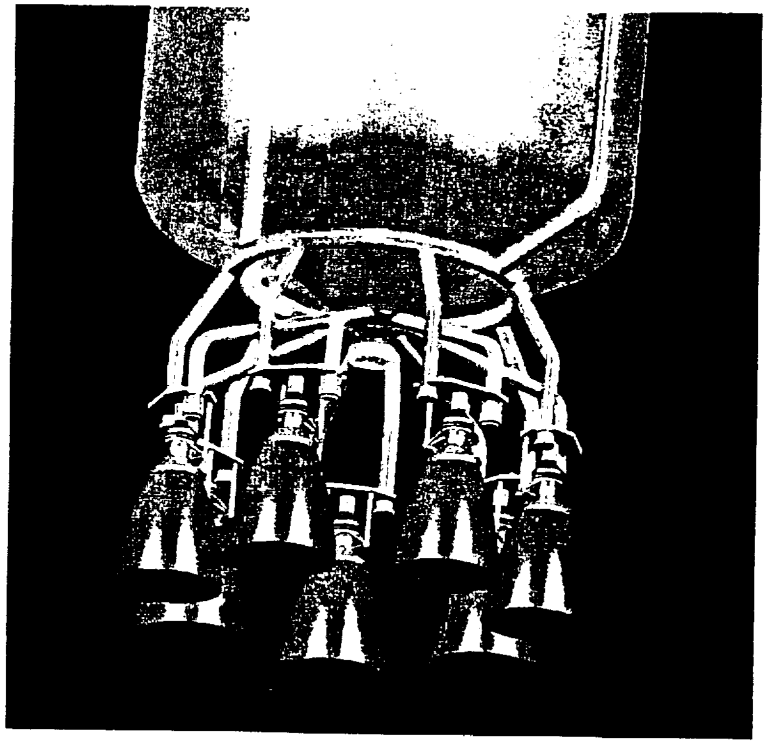


Figure 2.4-35 MPS Engines & Feedlines

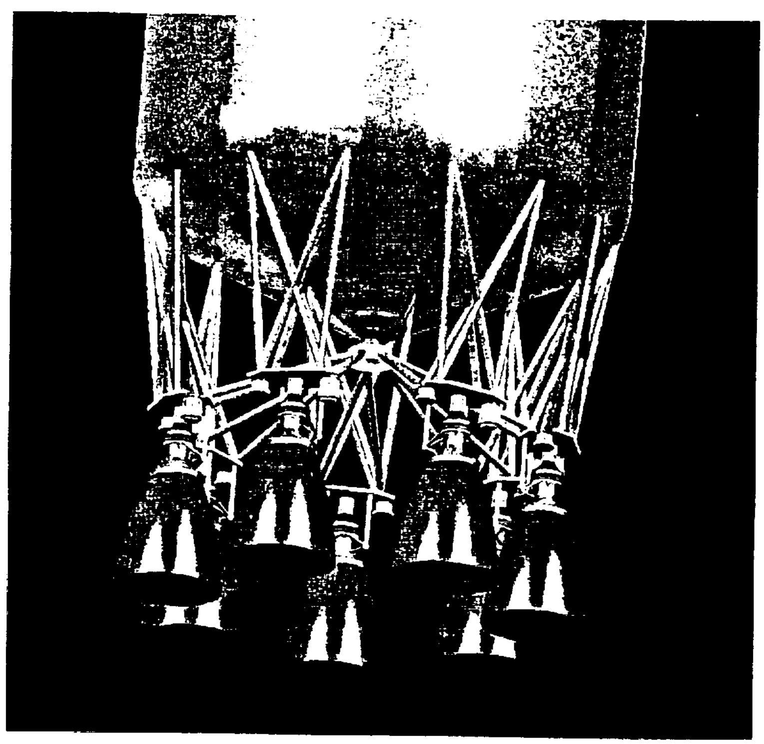


Figure 2.4-36 MPS Engines & Thrust Structure

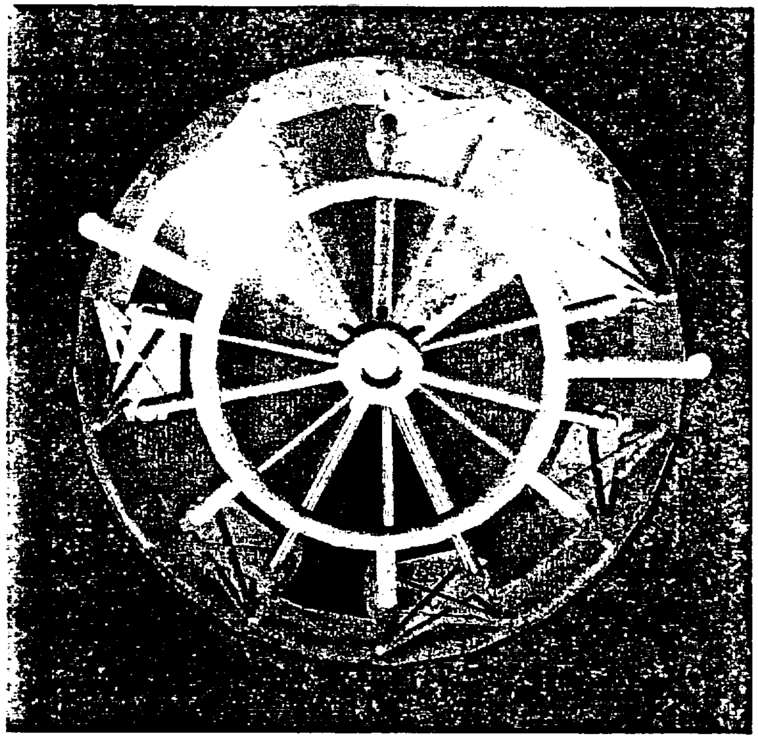


Figure 2.4-37 MPS Feedlines & Manifolds (Top View)

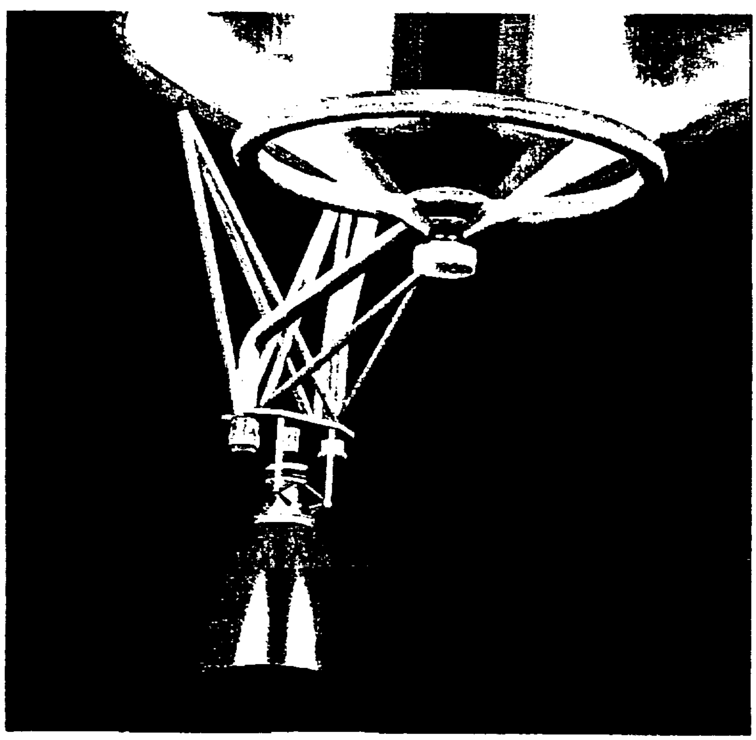


Figure 2.4-38 3-Point Engine Mounting (Pinned Joints)

2.5 Appendices

This section of the Interim Final Report contains several individual reports which were written over the course of the study period on selected key topics. These reports represent concentrated study efforts on key issues related to manned booster concepts. The first two reports summarize detailed analyses which were undertaken to fully understand the structural impacts to a booster when a winged payload (such as the CLV and the PLS concepts) is installed as the launch payload. The third report is the current release of a report which documents the NASA man-rating requirements and converts these requirements into functional design requirements for a manned booster. This detailed examination of man-rating requirements and consequent booster design impacts is a continuing study activity. This report will therefore be updated as the study progresses. The fourth report summarizes a detailed analysis of the NASA KSC launch process itself to determine if this process exhibits learning curve effects. The analysis was performed on the Space Shuttle system, but is applicable to other NASA booster and manned spacecraft launch processes.

<u>Enclosed Reports</u>

Appendix A	Structural Analysis of CLV on NLS-2
Appendix B	Structural Analysis of PLS with CRV on NLS-2
Appendix C	Man-Rating Requirements Report (Rev. A)
Appendix D	Learning Curve Analysis of Space Shuttle Processing

ADVANCED TRANSPORTATION SYSTEM STUDY

Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO

Structural Analysis of a Manned Payload on the NLS

Crew Logistics Vehicle (CLV)

September, 1992

Contract NAS8-39207

Prepared by: H. R. Grooms
Rockwell International

Rockwell International
Space Systems Division
Huntsville Operations

FORWARD

This report documents analyses conducted under Contract NAS8-39207, Advanced Transportation System Studies for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center. The report describes a preliminary analysis of aerodynamic loads, structural dynamics and stress, and weight estimation of the NLS-2 launch vehicle with a large manned payload (the Crew Logistics Vehicle concept as defined by the Johnson Spaceflight Center). This work was performed during the period of August and September, 1992 under the direction of Mr. Henry Grooms at Rockwell International, Space Systems Division, in Downey, CA. The primary technical analyses were performed by Vyto Baipsys (Aerodynamics), Keith Maeda (Weights), Al Yeung (Structural Dynamics), and Van Richardson (Stress), all of Rockwell International.

A PRELIMINARY ASSESSMENT OF A CREW LOGISTICS VEHICLE

INTRODUCTION

This report documents a preliminary analysis of a Crew Logistics Vehicle (CLV). This analysis took an initial design concept and performed a strength and weight assessment.

I. Ground Rules and Assumptions

The CLV is a scaled (58%) version of a Shuttle Orbiter that is mounted atop a National Launch System (NLS) booster (Figure 1). Existing information on the Orbiter and NLS was used directly/modified to expedite this study.

The main purpose of the study was to determine what impact carrying the CLV would have on the NLS booster. It was decided, because of this, to represent the mass of the CLV but not its stiffness. The aerodynamic forces on the CLV were included.

The analysis and assessment procedure is shown in Figure 2. Most of the work done for this report was done using the computer program "IDEAS" and other auxiliary programs that are compatible with it.

II. Aerodynamic Load Distributions

Design considerations were outlined as part of this effort in order to identify key issues to reduce the NLS/CLV technical development risk. Aerodynamic loads distributions have been defined for on-pad and high dynamic pressure (q) condition to support structural analysis for the NLS/CLV launch configuration depicted in Figure 3. The CLV is a 0.58-scale Orbiter geometry vehicle launched by a 10-foot tank stretch version of the 1.5 stage NLS booster. The CLV is attached to the NLS booster by an adapter and replaces the conventional NLS payload shroud. In order to show the relative difference between the NLS shroud and the CLV, Figure 3 shows the stack with the NLS shroud superimposed over the stack in dashed lines.

The key issues to be addressed for the NLS/CLV development are outlined below:

1. Evaluate ground wind loads to insure CLV wind load bearing capacity and structural design of NLS2 forward adapter.

<u>NLS Wind Criteria:</u>

(A) Normal Wind Operations (includes launch)
5% risk factor, windiest 1-hr exposure period
Wind speed of 34.4 knots at 60 ft altitude

(B) High Wind Operations
(Unfueled)
1% risk factor, windiest 180-day exposure period
Wind speed of 74.5 knots at 60 ft altitude

(Partly or fully fueled)
1% risk factor, windiest one-day exposure period
Wind speed of 47 knots, at 60 ft altitude

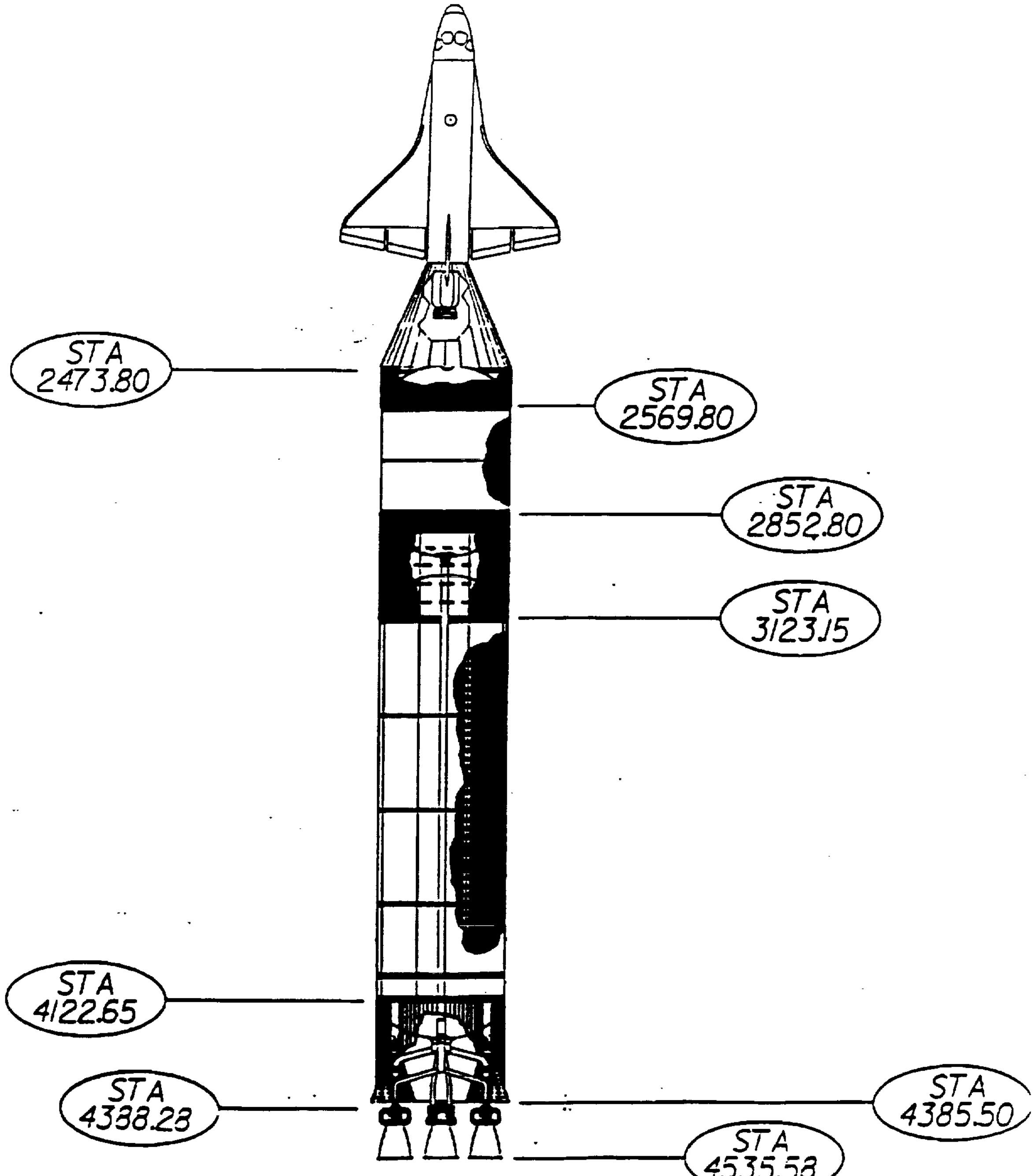


FIGURE. 1. CLV MOUNTED ON NLS BOOSTER

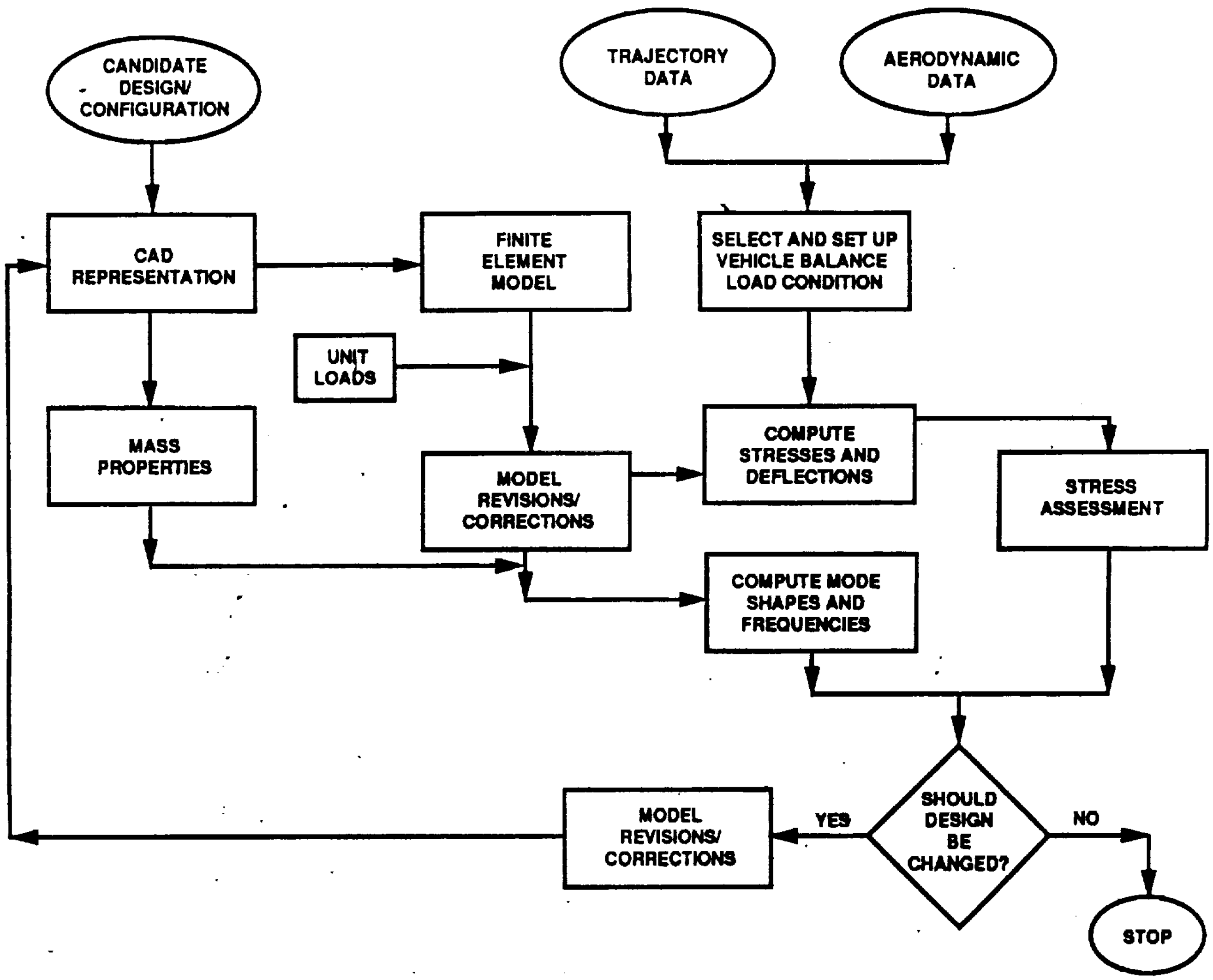


FIGURE 2. ANALYSIS PROCEDURE

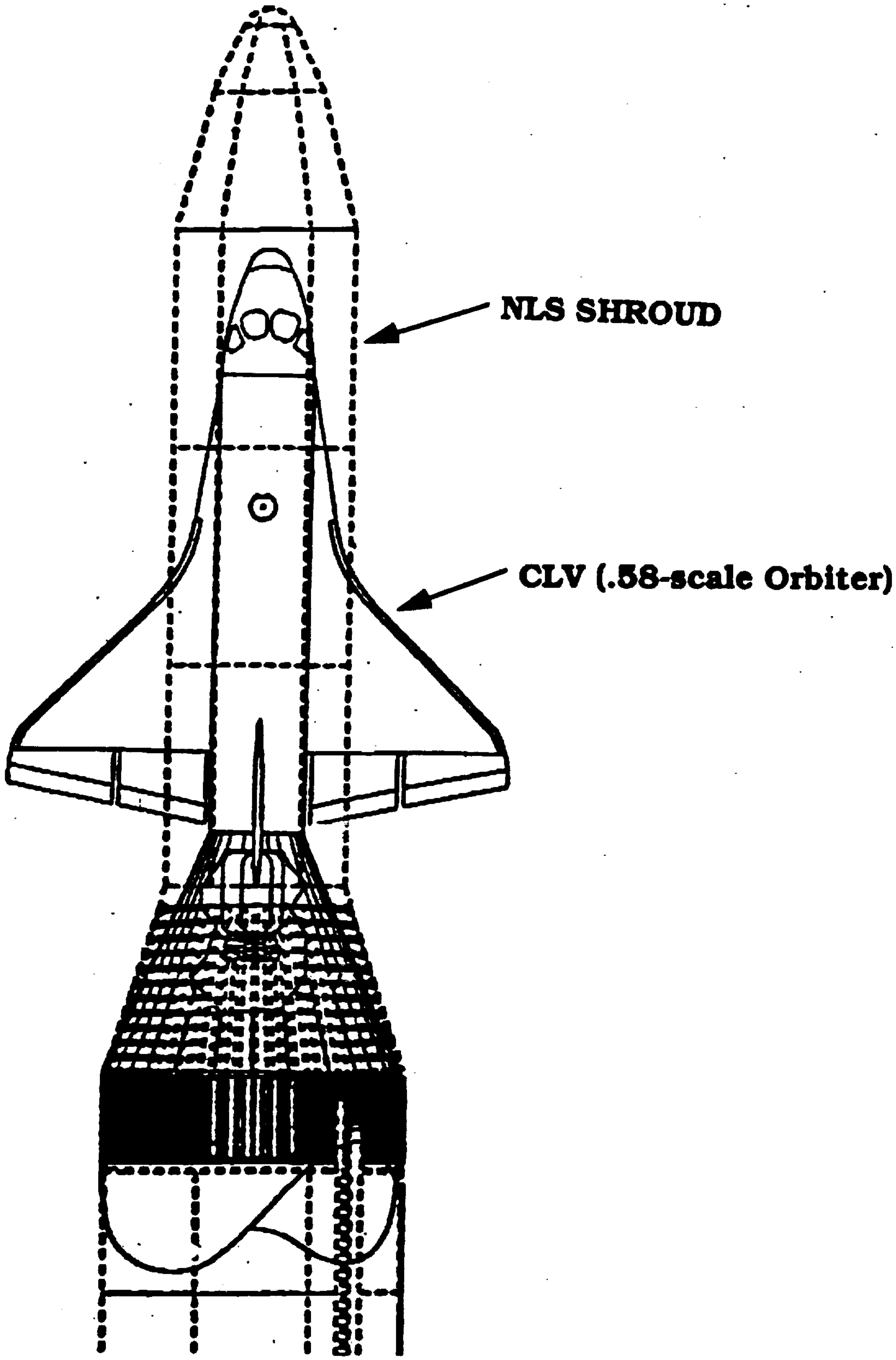


FIGURE 3. CLV ATTACHED TO THE NLS BOOSTER

2. Evaluate launch wind environment to insure NLS2 booster gimbal control authority.

   For wind criteria see (A) above.

3. Evaluate in-flight aerodynamic loads due to winds aloft-plus-gust to insure structural design of NLS2 forward adapter, booster gimbal control authority, and CLV wing load bearing capability.

   Use conservative winds alot with superimposed synthetic 10 m/s cosine-shaped gust at maximum $q\alpha$ condition.

   Insure that NLS2 gimbal authority is sufficient to handle de-stabilizing moment produced by CLV wing lift. If not, provide concepts for enhanced static stability and control.

   Evaluate CLV wing shear and bending moments to insure structural integrity. (Positive alpha can be experienced on the wings with in-line CLV exposed to winds. Orbiter ascent is typically fown with negative alphas, which limits wing loads.)

   Evaluate pressure and buffeting from CLV wing vortices on NLS 2 structure.

   - pressure spikes
   - panel flutter

4. Evaluate the effect of scaled wing and tail leading edges, and the smaller nose radius on thermal heating rates during entry.

   What is the impact on loading edte materials selection?

5. Explore and apply advanced TPS materials to reduce costs of production and operations relative to STS Orbiter.

   Advances made in studies for the STS Orbiter
   Advances made by the European Space Agency (ESA) for Hermes

6. Evaluate launch vehicle dynamic characteristics (mode shape and frequencies) associated with the CLV mass at the top of the NLS2 booster.

7. Conduct the CLV center of gravity (CG) range selections based on correct aerodynamic stability analysis. Text on page 60 and Figure 4.2-1 (April 1992 CLV Report) uses Xac and Xcp interchangeably. They are not, and therefore, erroneous calculations of static margin may result.

The distributed and summed ground wind loads for the on-pad wind condition are shown in the spreadsheet results in Table 1. The wind speed used was from the NLS wind criteria (References 1 and 2) which specifies the maximum wind speed at 74.5 knots at 60 foot height above ground level (the NLS base is 95 feet above ground). This is a 1% risk factor wind (99% probability of not exceeding it) for a 180-day exposure duration. The wind speed increases exponentially with height, as tabulated in the spreadsheet. The spreadsheet shows the distributed wind drag loading over the entire launch stack and the computation of the running drag load and bending moment summations along the NLS booster. These drag load and moment summations begin at the top of the NLS/CLV adapter (Station

TABLE 1. NLS GROUND WIND LOADING

1O' STRETCH NLV2 WITH INLINE CLV

VJB

11-Aug-92

ON-PAD AERO LOADS

SUMMED LOADS INCLUDE CLV

1% RISK FACTOR, 180 WINDIEST DAY EXPOSURE DURATION

Vwind=74.5 KTS @ 60 ft Ref. Altitude

NLS STRUCTURE BASE AT 95 ft ELEVATION

X STA. No in	Component	LOCAL DIAMETER ft	Height ft	Cd Drag Coeff (based on side area)	Vwind ft/sec	DRAG *1.5 (vortex shed) lbs/in	SUMMED LOAD VERSUS X lbs w/clv	SUMMED MOMENT ABOUT X ft-lbs w/clv	LOCAL PRESSURE DISTRIB. psia
2167.80		0.00	279.81	1.000	147.56	0.000	5.223E+04	1.112E+06	0.0000
2167.80	22.127 deg	10.17	279.81	1.000	147.56	32.617	5.223E+04	1.112E+06	0.2674
2301.00	Frustum	19.19	268.71	1.000	146.95	61.081	5.847E+04	1.723E+06	0.2652
2424.80	"	27.58	258.39	1.000	146.35	87.097	6.764E+04	2.371E+06	0.2631
2424.80	1st Stage	27.58	258.39	1.000	146.35	87.097	6.764E+04	2.371E+06	0.2631
2472.80	"	27.58	254.39	1.000	146.11	86.825	7.182E+04	2.650E+06	0.2623
2569.80	"	27.58	246.31	1.000	145.62	86.265	8.021E+04	3.264E+06	0.2606
2583.55	"	27.58	245.16	1.000	145.55	86.185	8.140E+04	3.357E+06	0.2604
2644.06	"	27.58	240.12	1.000	145.24	85.826	8.660E+04	3.780E+06	0.2593
2711.77	"	27.58	234.48	1.000	144.88	85.418	9.240E+04	4.285E+06	0.2581
2778.89	"	27.58	228.88	1.000	144.52	85.005	9.812E+04	4.818E+06	0.2568
2838.41	"	27.58	223.92	1.000	144.19	84.631	1.032E+05	5.317E+06	0.2557
2852.80	"	27.58	222.73	1.000	144.11	84.540	1.044E+05	5.442E+06	0.2554
2897.10	"	27.58	219.03	1.000	143.86	84.256	1.081E+05	5.834E+06	0.2545
2941.40	"	27.58	215.34	1.000	143.61	83.968	1.118E+05	6.240E+06	0.2537
2985.67	"	27.58	211.65	1.000	143.35	83.676	1.156E+05	6.660E+06	0.2528
3034.20	"	27.58	207.61	1.000	143.06	83.352	1.196E+05	7.135E+06	0.2518
3083.30	"	27.58	203.52	1.000	142.77	83.018	1.237E+05	7.633E+06	0.2508
3123.15	"	27.58	200.20	1.000	142.52	82.743	1.270E+05	8.049E+06	0.2500
3137.54	"	27.58	199.00	1.000	142.43	82.643	1.282E+05	8.202E+06	0.2497
3201.70	"	27.58	193.65	1.000	142.03	82.189	1.335E+05	8.902E+06	0.2483
3266.50	"	27.58	188.25	1.000	141.61	81.721	1.388E+05	9.637E+06	0.2469
3331.30	"	27.58	182.85	1.000	141.19	81.241	1.441E+05	1.040E+07	0.2454
3377.35	"	27.58	179.01	1.000	140.88	80.893	1.478E+05	1.096E+07	0.2444
3435.90	"	27.58	174.13	1.000	140.47	80.442	1.525E+05	1.169E+07	0.2430
3500.70	"	27.58	168.73	1.000	140.01	79.930	1.577E+05	1.253E+07	0.2415
3565.70	"	27.58	163.32	1.000	139.54	79.403	1.629E+05	1.340E+07	0.2399
3623.80	"	27.58	158.48	1.000	139.10	78.919	1.675E+05	1.420E+07	0.2384
3706.10	"	27.58	151.62	1.000	138.47	78.213	1.740E+05	1.537E+07	0.2363
3784.90	"	27.58	145.05	1.000	137.83	77.512	1.801E+05	1.653E+07	0.2342
3871.00	"	27.58	137.88	1.000	137.11	76.716	1.867E+05	1.785E+07	0.2318
3932.00	"	27.58	132.79	1.000	136.57	76.131	1.914E+05	1.881E+07	0.2300
3996.80	"	27.58	127.39	1.000	135.98	75.490	1.963E+05	1.986E+07	0.2281
4058.00	"	27.58	122.29	1.000	135.41	74.863	2.009E+05	2.087E+07	0.2262
4108.91	"	27.58	118.05	1.000	134.91	74.325	2.047E+05	2.173E+07	0.2245
4122.65	Thrust struct.	27.58	116.90	1.000	134.78	74.177	2.057E+05	2.197E+07	0.2241
4137.30	"	27.58	115.68	1.000	134.63	74.018	2.068E+05	2.222E+07	0.2236
4151.90	"	27.58	114.47	1.000	134.48	73.859	2.079E+05	2.247E+07	0.2231
4166.60	"	27.58	113.24	1.000	134.33	73.696	2.090E+05	2.272E+07	0.2226

TABLE 1. NLS GROUND WIND LOADING
(con't) 1O' STRETCH NLV2 WITH INLINE CLV

VJB

11-Aug-92

NLS STRUCTURE BASE AT 95 ft ELEVATION

						ON-PAD AERO LOADS SUMMED LOADS INCLUDE CLV — 1% RISK FACTOR, 180 WINDIEST DAY EXPOSURE DURATION Vwind=74.5 KTS @ 60 ft Ref. Altitude			
X STA. No in	Component	LOCAL DIAMETER ft	Height ft	Cd Drag Coeff (based on side area)	Vwind ft/sec	DRAG *1.5 (vortex shed) lbs/in	SUMMED LOAD VERSUS X lbs w/clv	SUMMED MOMENT ABOUT X ft-lbs w/clv	LOCAL PRESSURE DISTRIB. psia
4181.20	"	27.58	112.03	1.000	134.18	73.534	2.101E+05	2.298E+07	0.2222
4195.70	"	27.58	110.82	1.000	134.03	73.371	2.111E+05	2.323E+07	0.2217
4210.30	"	27.58	109.60	1.000	133.88	73.206	2.122E+05	2.349E+07	0.2212
4227.40	"	27.58	108.18	1.000	133.69	73.010	2.134E+05	2.379E+07	0.2206
4241.60	"	27.58	106.99	1.000	133.54	72.846	2.145E+05	2.405E+07	0.2201
4254.10	"	27.58	105.95	1.000	133.41	72.700	2.154E+05	2.427E+07	0.2196
4265.00	"	27.58	105.04	1.000	133.29	72.572	2.162E+05	2.447E+07	0.2193
4275.90	"	27.58	104.13	1.000	133.17	72.443	2.170E+05	2.466E+07	0.2189
4286.90	"	27.58	103.22	1.000	133.05	72.312	2.178E+05	2.486E+07	0.2185
4297.80	"	27.58	102.31	1.000	132.92	72.182	2.186E+05	2.506E+07	0.2181
4308.80	"	27.58	101.39	1.000	132.80	72.049	2.193E+05	2.526E+07	0.2177
4319.70	"	27.58	100.48	1.000	132.67	71.916	2.201E+05	2.546E+07	0.2173
4330.60	"	27.58	99.58	1.000	132.55	71.783	2.209E+05	2.566E+07	0.2169
4341.60	"	27.58	98.66	1.000	132.42	71.647	2.217E+05	2.587E+07	0.2165
4352.20	"	27.58	97.78	1.000	132.30	71.515	2.225E+05	2.606E+07	0.2161
4362.80	"	27.58	96.89	1.000	132.17	71.382	2.232E+05	2.626E+07	0.2157
4374.20	"	27.58	95.94	1.000	132.04	71.238	2.240E+05	2.647E+07	0.2152
4385.50	End of struct.	27.58	95.00	1.000	131.90	71.094	2.248E+05	2.668E+07	0.2148

2167.8) and continue to the NLS booster base at Station 4385.5. The effect of the CLV on the loads is represented as a concentrated drag force and moment acting at the top of the NLS/CLV adapter. The CLV drag value was obtained by multiplying its planform area of 1337 ft^2 by a drag coefficient of 1.5 and a dynamic pressure of 26 psf (the result of wind speed and air density at the hight corresponding to the centroid of the CLV area at Station 1912.27). The moment cuased by the CLV was the product of the drag load multiplied by the distance between the above area centroid and any X Station in question.

CLV drag = 1.5x1337x26.043 = 52,230 lbs
CLV moment = 52,230x(X Sta-1912.27)/12 ft-lbs
(moment varies with X Sta) Moment = 1.112E+06 ft-lbs at
X Sta = 2167.8, the top of the NLS/CLV adapter

Net pressure distributions along the booster were computed from the distributed drag loading and are shown in the table. In addition to the steady-state drag loads the total loads in the table include an amplification factor of 1.5 to account for transient loads in the dragwise and the transverse directions due to oscillatory vortex shedding from the booster sides.

Figures 4 and 5 depict the wind drag distribution (with the 1.5 amplification factor) and the resulting distribution of net pressures across the booster, respectively. The CLV concentrated drag load is depicted as a single force vector in Figure 2.

Plots of summed drag load and moment along the NLS booster due to ground wind are shown in Figures 6 and 7, respectively. The total drag load on the stack is 2.248E+05 lbs. The resulting moment about the NLS booster base is 2.67E+07 ft-lbs. These are the aerodynamic loads which the NLS structure has to withstand and which will have to be reacted against by the booster tie-down system.

Similar type of results were obtained for the in-flight condition of maximum product of dynamic pressure times alpha $(q\text{-alpha})_{max}$ to define the highest bending moment in flight. This condition occurred at an altitude of 31,116 feet and a Mach number of 1.32 at a corresponding dynamic pressure 777.34 psf. A conservative angle of attack (α) of 5.3 degrees was computed by superimposing an NLS wind of 96.44 ft/sec (Reference 3) and a 33 ft/sec gust on the vehicle speed of 1391 ft/sec.

$$\alpha = \mathrm{Tan}^{-1}\left(\frac{V_{wind} + V_{gust}}{V_{vehicle}}\right) = 5.3 \text{ deg}$$

The spreadsheet results defining the distributed airload distribution are shown in Table 2. The CLV normal force coefficient and center of pressure were derived from the STS Orbiter aerodynamic data (Reference 4). For this condition, the CLV normal force coefficient was found to be 0.39066 (based on the NLS reference area of 584 ft^2 and a center-of-pressure at X Sta=1917.67). Normal force coefficient distribution on the NLS/CLV adapter and the NLS booster was derived from the NLS distributions (Reference 5) with some modification to the distribution over the adapter section to reflect a longer NLS/CLV adapter.

Normal force distribution and the resulting distribution of net pressures across the booster are shown in Figures 8 and 9, respectively. For this condition, the distributed loads depend only on the steady-state aerodynamic normal force distributions and do not include

FIGURE 4. NLS ON-PAD DRAG LOAD DISTRIBUTION

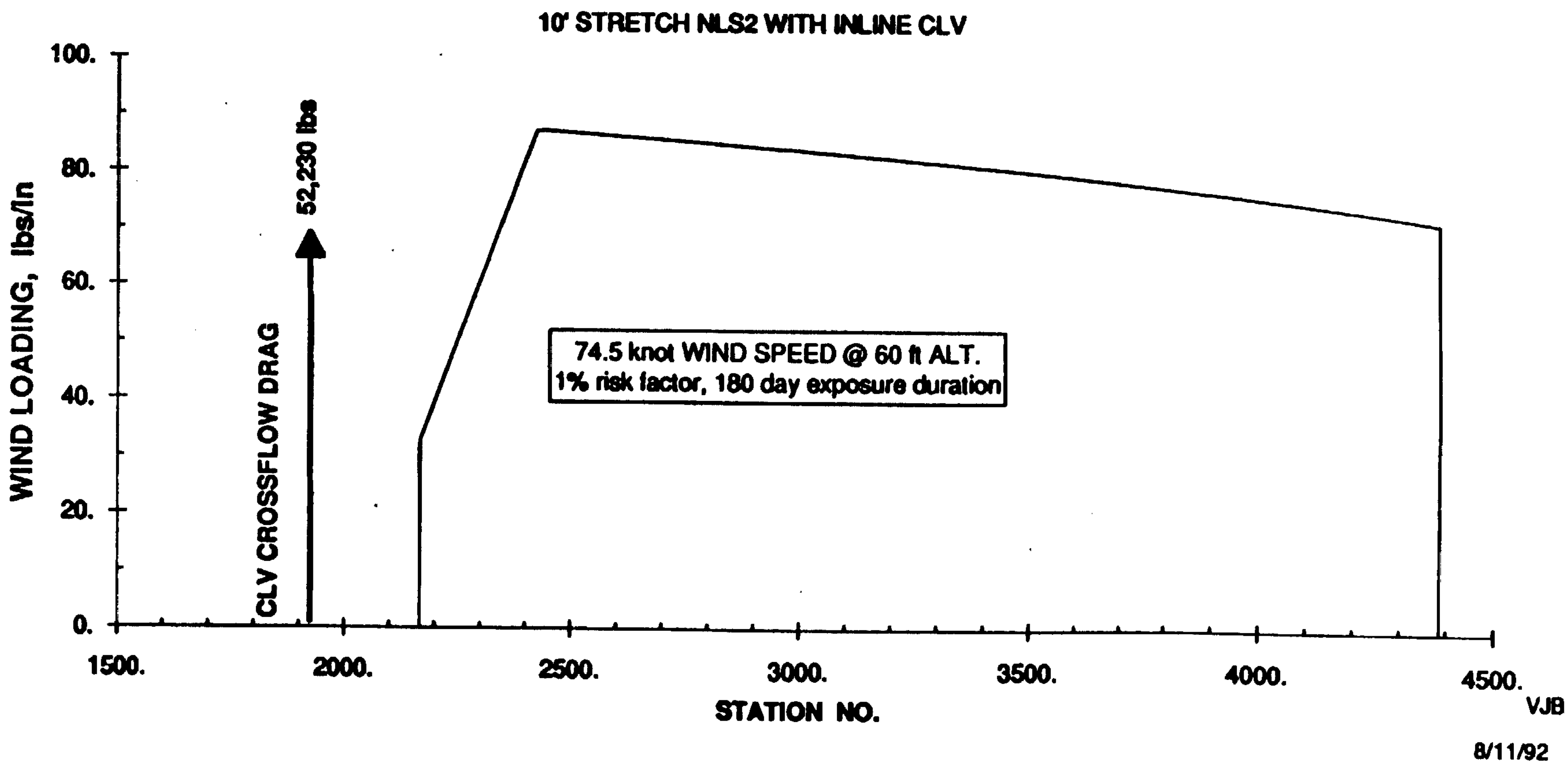


FIGURE 5. NLS ON-PAD SURFACE PRESSURE DUE TO WIND LOAD

10' STRETCH NLS2 WITH INLINE CLV

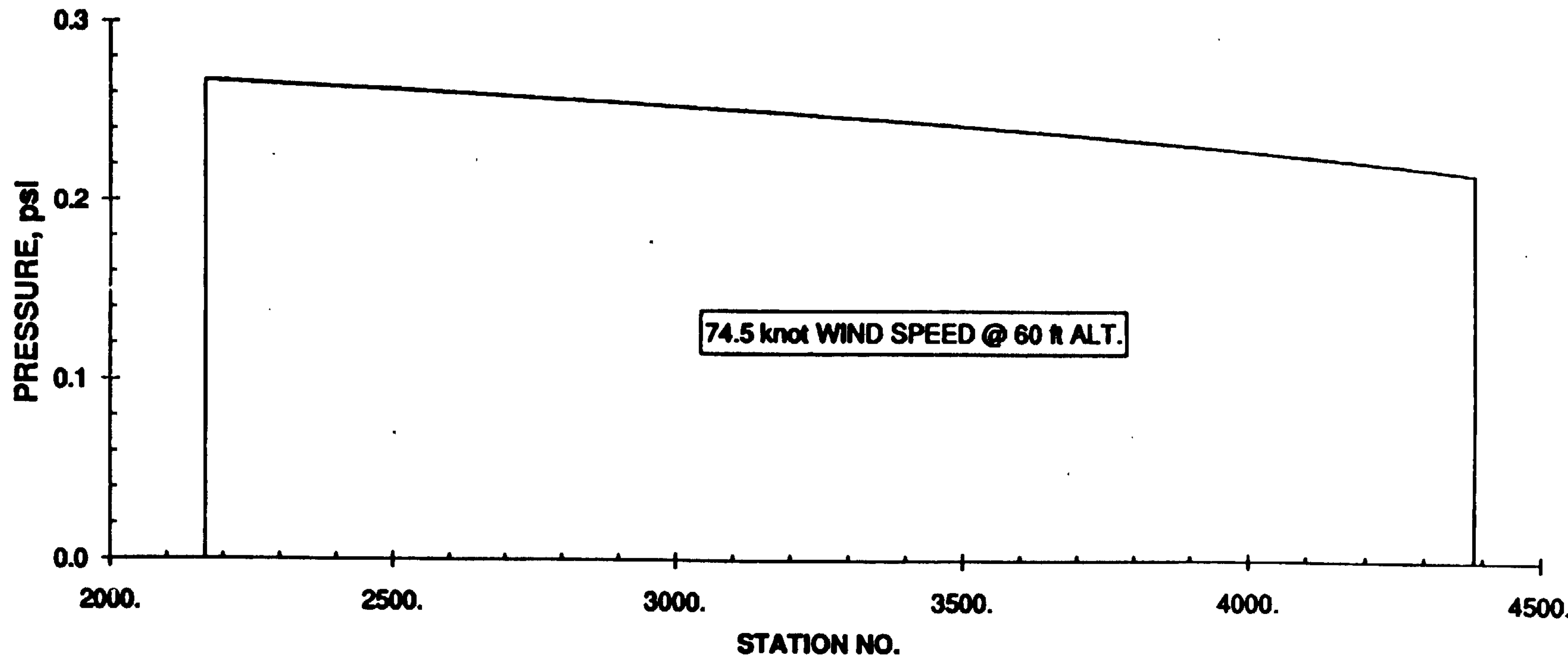


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FIGURE 6. ON-PAD WIND DRAG LOAD SUMMATION

10' STRETCH NLS2 WITH INLINE CLV

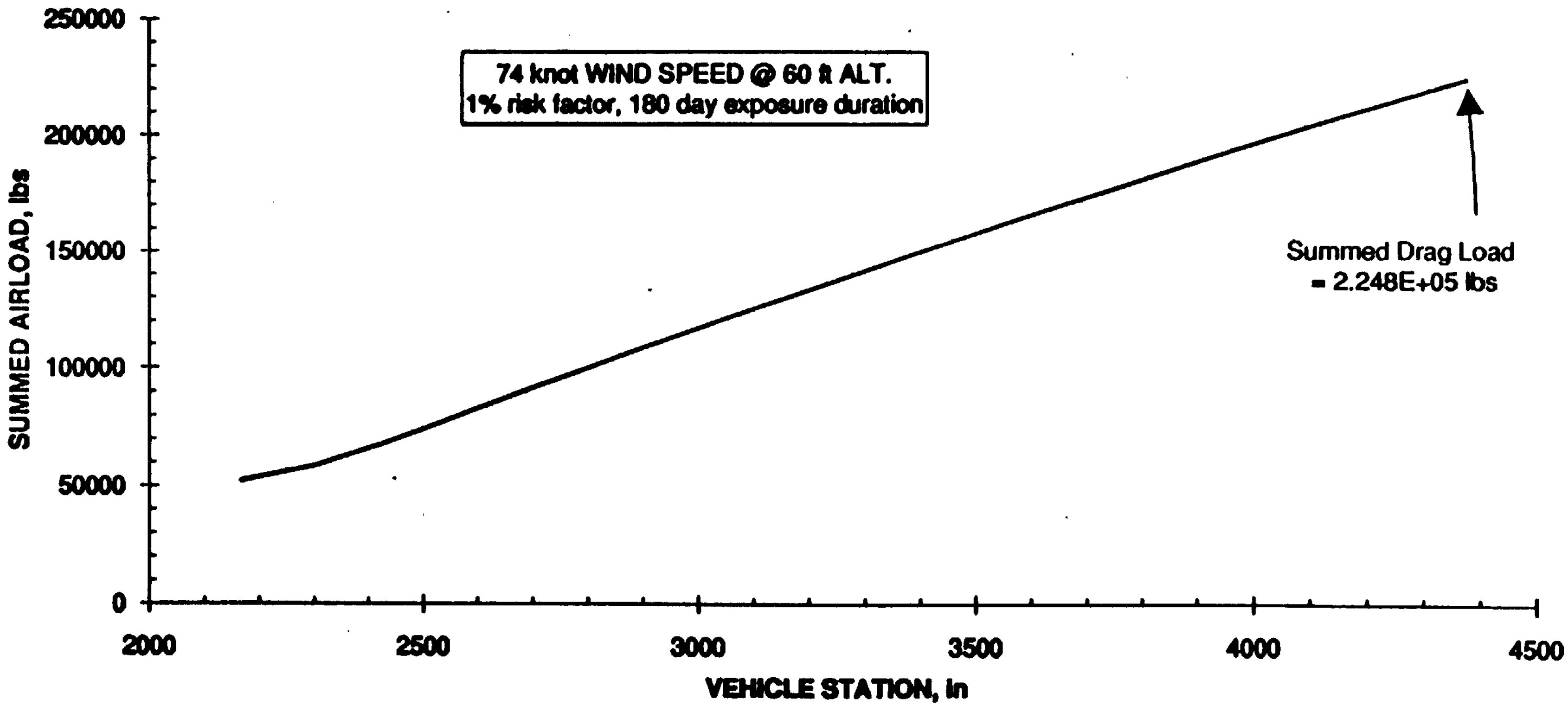

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FIGURE 7. NLS ON-PAD BENDING MOMENT

10' STRETCH NLS2 WITH INLINE CLV

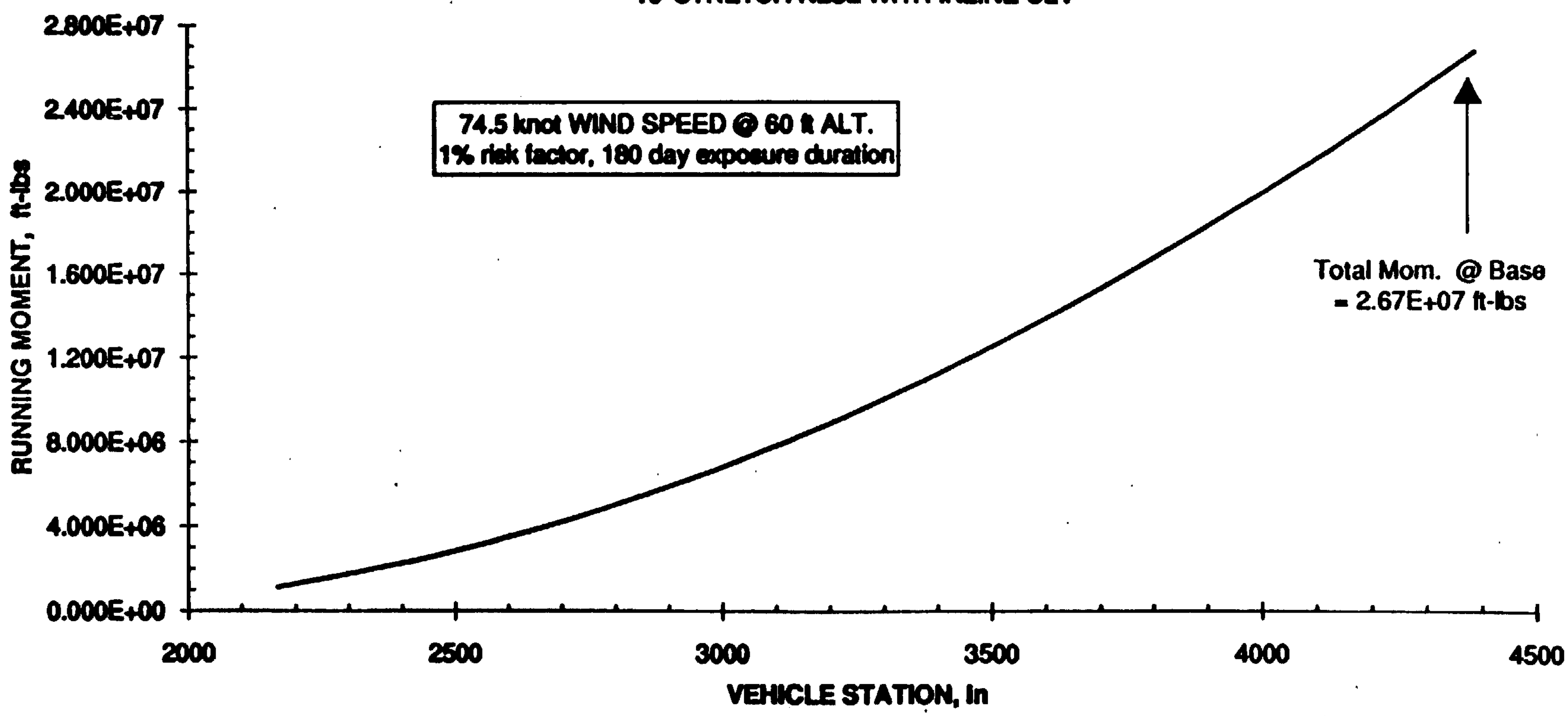

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TABLE 2. NLS/CLV MAX q AIRLOAD DISTRIB. ABOUT Xcg

10' STRETCH NLV2 WITH INLINE CLV

VJB

8/14/92

Sref = 593.96 ft^2 MACH = 1.3

Xcg(sta) = 3030 q (psf) = 777.34

α = 5.3 deg

CN clv = 0.39066

Xcp clv = 1917.67

MAX q AIRLOADS
SUMMED LOADS INCLUDE CLV

NOMINAL TRAJECTORY
MACH=1.3, α =5.3 deg

X STA. No in	Component	LOCAL DIAMETER ft	CNα per Radian per X/D	NORMAL AIR LOAD DISTRIBUTED lbs/in	SUMMED NORMAL F. FWD OF X lbs w/ clv	SUMMED MOMENT ABOUT Xcg ft-lbs w/ clv	LOCAL PRESSURE DISTRIB. psia
1917.67	<<< CLV Xcp	0.00	0.00000	0.00	1.804E+05	1.672E+07	0.000
2167.80	22.127 deg	10.17	0.91254	117.75	1.804E+05	1.672E+07	0.965
2308.80	Frustum	19.72	2.38900	308.25	2.104E+05	1.867E+07	1.302
2424.80	"	27.58	2.53390	326.95	2.472E+05	2.071E+07	0.988
2424.80	1st Stage	27.58	1.42650	184.06	2.472E+05	2.071E+07	0.556
2472.80	"	27.58	0.00000	0.00	2.517E+05	2.093E+07	0.000
2520.80	"	27.58	0.08000	10.32	2.519E+05	2.094E+07	0.031
2583.55	"	27.58	0.41000	52.90	2.539E+05	2.101E+07	0.160
2644.06	"	27.58	0.51000	65.81	2.575E+05	2.114E+07	0.199
2711.77	"	27.58	0.46000	59.35	2.617E+05	2.126E+07	0.179
2778.89	"	27.58	0.40000	51.61	2.654E+05	2.135E+07	0.156
2838.41	"	27.58	0.34000	43.87	2.683E+05	2.140E+07	0.133
2852.80	"	27.58	0.32000	41.29	2.689E+05	2.141E+07	0.125
2897.10	"	27.58	0.27000	34.84	2.706E+05	2.144E+07	0.105
2941.40	"	27.58	0.22000	28.39	2.720E+05	2.145E+07	0.086
2985.67	"	27.58	0.16000	20.64	2.731E+05	2.145E+07	0.062
3034.20	"	27.58	0.09000	11.61	2.739E+05	2.146E+07	0.035
3083.30	"	27.58	0.03000	3.87	2.742E+05	2.146E+07	0.012
3123.15	"	27.58	-0.03000	-3.87	2.742E+05	2.146E+07	-0.012
3137.54	"	27.58	-0.05000	-6.45	2.742E+05	2.146E+07	-0.019
3201.70	"	27.58	-0.12000	-15.48	2.735E+05	2.146E+07	-0.047
3266.50	"	27.58	-0.14000	-18.06	2.724E+05	2.148E+07	-0.055
3331.30	"	27.58	-0.07000	-9.03	2.715E+05	2.150E+07	-0.027
3377.35	"	27.58	-0.02000	-2.58	2.712E+05	2.151E+07	-0.008
3435.90	"	27.58	0.03000	3.87	2.713E+05	2.151E+07	0.012
3500.70	"	27.58	0.08000	10.32	2.717E+05	2.149E+07	0.031
3565.70	"	27.58	0.10169	13.12	2.725E+05	2.146E+07	0.040
3623.80	"	27.58	0.10169	13.12	2.732E+05	2.142E+07	0.040
3708.10	"	27.58	0.10169	13.12	2.743E+05	2.137E+07	0.040
3784.90	"	27.58	0.10169	13.12	2.754E+05	2.130E+07	0.040
3871.00	"	27.58	0.10169	13.12	2.765E+05	2.123E+07	0.040
3932.00	"	27.58	0.10169	13.12	2.773E+05	2.117E+07	0.040
3996.80	"	27.58	0.10169	13.12	2.781E+05	2.111E+07	0.040
4058.00	"	27.58	0.10169	13.12	2.789E+05	2.104E+07	0.040
4108.91	"	27.58	0.10169	13.12	2.796E+05	2.098E+07	0.040
4122.65	Thrust structure.	27.58	0.10169	13.12	2.798E+05	2.096E+07	0.040
4137.30	"	27.58	0.10169	13.12	2.800E+05	2.095E+07	0.040
4151.90	"	27.58	0.10169	13.12	2.802E+05	2.093E+07	0.040
4166.60	"	27.58	0.10169	13.12	2.804E+05	2.091E+07	0.040

TABLE 2. NLS/CLV MAX q AIRLOAD DISTRIB. ABOUT Xcg

(con't) 10' STRETCH NLV2 WITH INLINE CLV

VJB

8/14/92

Sref = 593.96 ft^2 MACH = 1.3

Xcg(sta) = 3030 q (psf) = 777.34

α = 5.3 deg

CN clv = 0.39066

Xcp clv = 1917.67

MAX q AIRLOADS
SUMMED LOADS INCLUDE CLV

NOMINAL TRAJECTORY
MACH=1.3, α =5.3 deg

X STA. No in	Component	LOCAL DIAMETER ft	CNα per Radian per X/D	NORMAL AIR LOAD DISTRIBUTED lbs/in	SUMMED NORMAL F. FWD OF X lbs w/ clv	SUMMED MOMENT ABOUT Xcg ft-lbs w/ clv	LOCAL PRESSURE DISTRIB. psia
4181.20	•	27.58	0.10169	13.12	2.806E+05	2.089E+07	0.040
4195.70	•	27.58	0.10169	13.12	2.808E+05	2.087E+07	0.040
4210.30	•	27.58	0.10169	13.12	2.809E+05	2.085E+07	0.040
4227.40	•	27.58	0.10169	13.12	2.812E+05	2.083E+07	0.040
4241.60	•	27.58	0.10169	13.12	2.814E+05	2.081E+07	0.040
4254.10	•	27.58	0.10169	13.12	2.815E+05	2.080E+07	0.040
4265.00	•	27.58	0.10169	13.12	2.817E+05	2.078E+07	0.040
4275.90	•	27.58	0.10169	13.12	2.818E+05	2.077E+07	0.040
4286.90	•	27.58	0.10169	13.12	2.819E+05	2.075E+07	0.040
4297.80	•	27.58	0.10169	13.12	2.821E+05	2.074E+07	0.040
4308.80	•	27.58	0.10169	13.12	2.822E+05	2.072E+07	0.040
4319.70	•	27.58	0.10169	13.12	2.824E+05	2.071E+07	0.040
4330.60	•	27.58	0.10169	13.12	2.825E+05	2.069E+07	0.040
4341.60	•	27.58	0.10169	13.12	2.827E+05	2.068E+07	0.040
4352.20	•	27.58	0.10169	13.12	2.828E+05	2.066E+07	0.040
4362.80	•	27.58	0.10169	13.12	2.829E+05	2.065E+07	0.040
4374.20	•	27.58	0.10169	13.12	2.831E+05	2.063E+07	0.040
4385.50	•	27.58	0.10169	13.12	2.832E+05	2.061E+07	0.040
4385.50	End of struct.	0.00	0.00000	0.00	2.832E+05	2.061E+07	0.000

THRUST, lbs=	2,941,220
Xgimbal=	4385.50
Xcp=	2156.75
CNa(per deg) =	0.11575

δ/α =	0.012	<<rad/deg			
δ =	3.555	<<deg gimbal req'd			
			Veh. Weight =	1,600,000	lbs
Axial Thrust=	2,935,561	lbs	Axial Accel=	1.8347	g's (Axial T)
Tangential T=	182,709	lbs	Tang. Accel. =	0.1770	g's (aero+T)

FIGURE 8. MAX q AIRLOAD DISTRIBUTION

10' STRETCH NLS2 WITH INLINE CLV

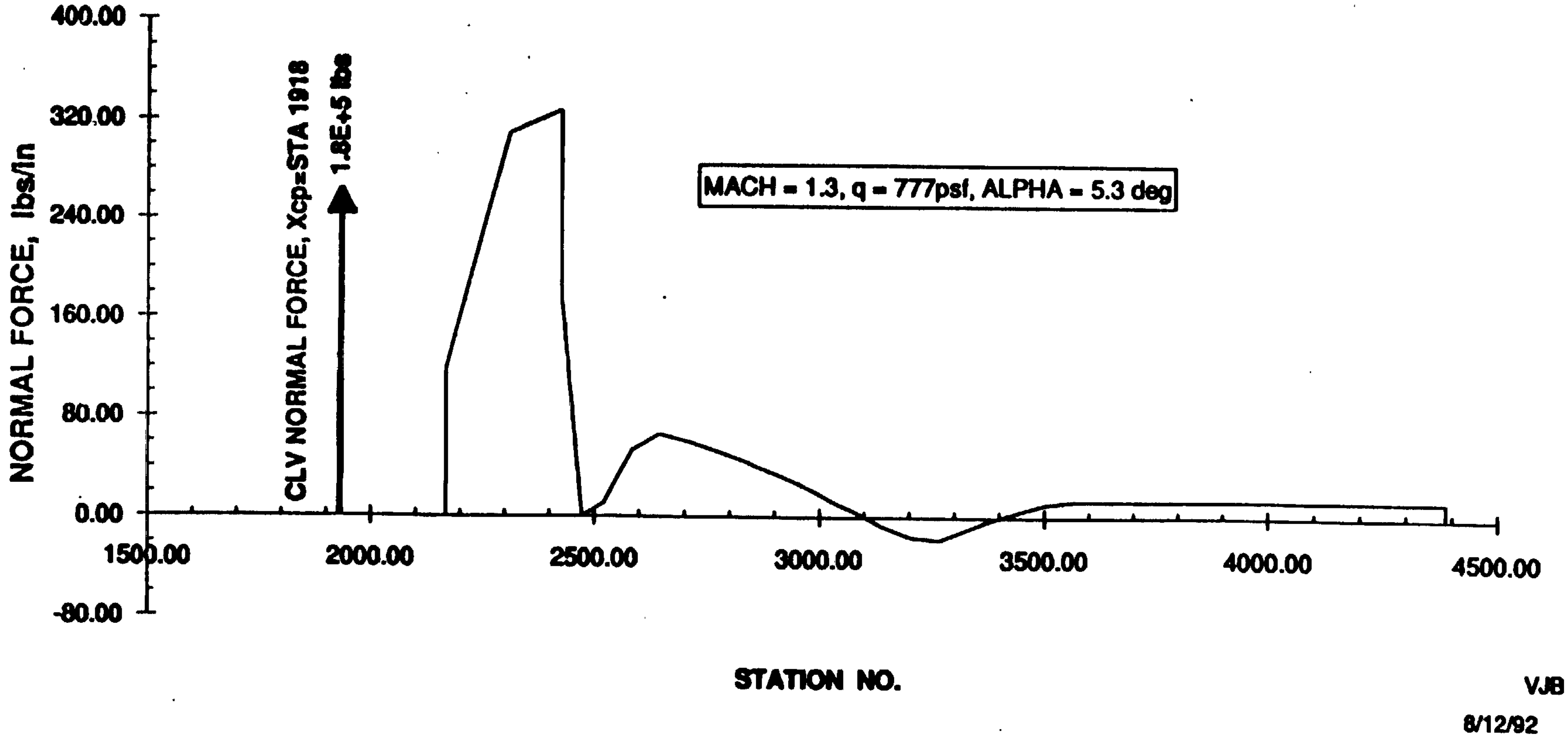
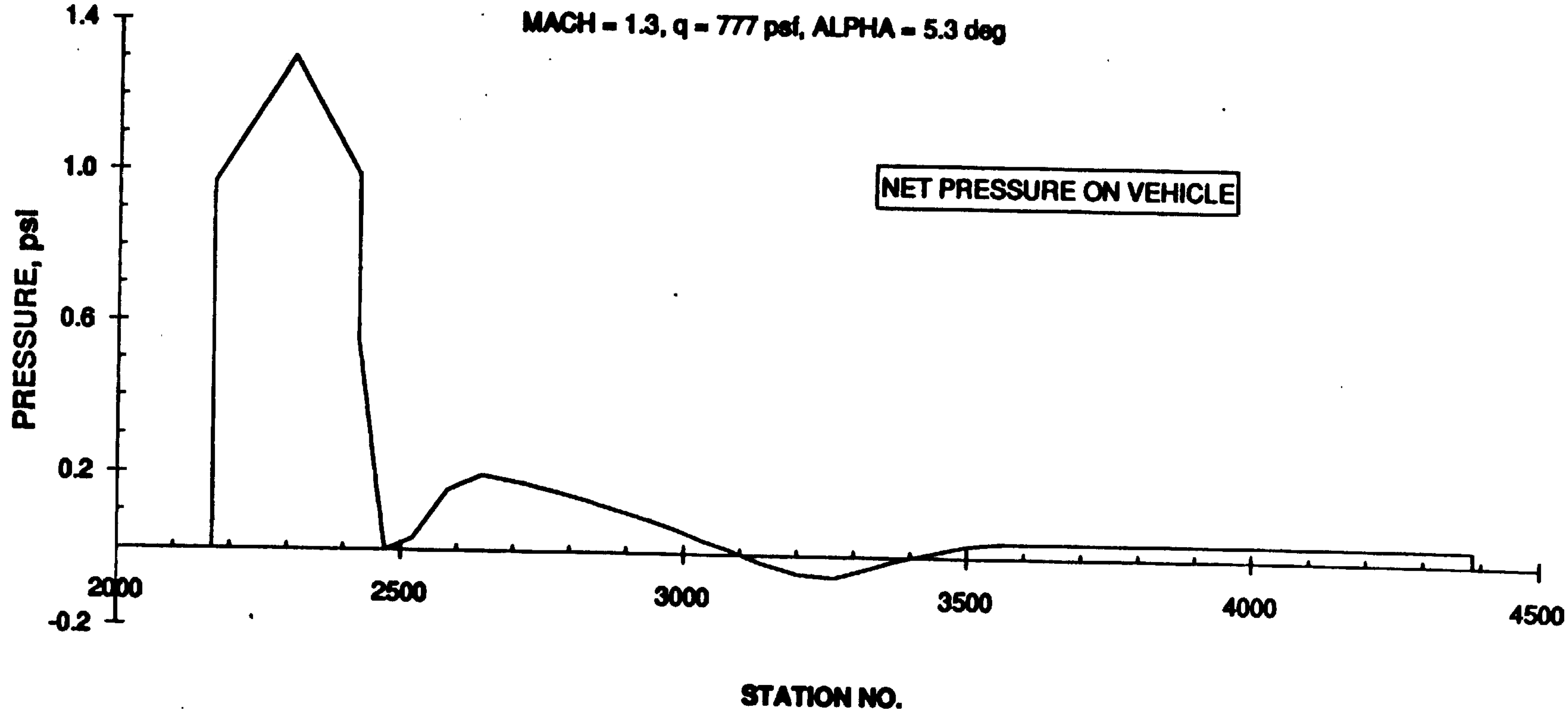


FIGURE 9. MAX q PRESSURE DISTRIBUTION

10' STRETCH NLS2 WITH INLINE CLV

MACH = 1.3, q = 777 psf, ALPHA = 5.3 deg



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any amplification factors which had been applied for on-pad loads due to vortex shedding. Figure 8 shows the CLV normal force of 1.8E+6 lbs concentrated at X Sta=1917.67.

The summed moments were computed about the vehicle Xcg at Station 3030 (adjusted for the presence of CLV from available NLS mass properties). The total loads and moments are shown in Figures 10 and 11 for the NLS/CLV configuration. These airloads show a summed drag of 2.248E+05 lbs and a moment about the Xcg of 2.06E+07 ft-lbs.

Static balance calculations were included in the spreadsheet of Table 2 to determine the amount of engine gimbal angle required in order to overcome the aerodynamic moment induced by the airload. This was computed from the moment balance require between the aerodynamic moment and the engines, as shown below.

$$T \sin(\delta)(X_{gimbal} - X_{cg}) = C_{N\alpha} q \alpha S_{ref}(X_{cg} - X_{cp})$$

Assuming small angles, $\sin(\delta)$ can be represented by δ, in radians

$$\text{then: } \frac{\delta}{\alpha} = \frac{C_{N_\alpha} q S_{ref} \; (X_{cg} - X_{cp})}{T(X_{gimbal} - X_{cg})} \quad \text{radians/degree}$$

where:

δ	=	engine gimbal angle, radians
α	=	angle of attack, degrees
$C_{N\alpha}$	=	normal force coefficient slope, per degree
q	=	dynamic pressure lbs/sq ft
S_{ref}	=	reference area, sq ft
T	=	engine thrust, lbs
X_{cg}	=	center of gravity station, in
X_{cp}	=	center of pressure station, in
X_{gimbal}	=	engine gimbal station, in

This relation is then solved for the appropriate α from the trajectory to solve for the gimbal angle, δ.

With the gimbal angle defined, the axial and tangential thrust values can be calculated. These are shown in the boxed area at the end of the spreadsheet.

These thrust components can then be used to compute the axial acceleration and the tangential acceleration

$$\text{Axial Acceleration} = \frac{\text{Axial Thrust}}{\text{Vehicle Weight}}$$

$$\text{Tangential Acceleration} = \frac{\text{Tangential Thrust} + \Sigma(\text{Airload})}{\text{Vehicle Weight}}$$

FIGURE 10. MAX q alpha AIRLOAD SUMMATION

10' STRETCH NLS2 WITH INLINE CLV

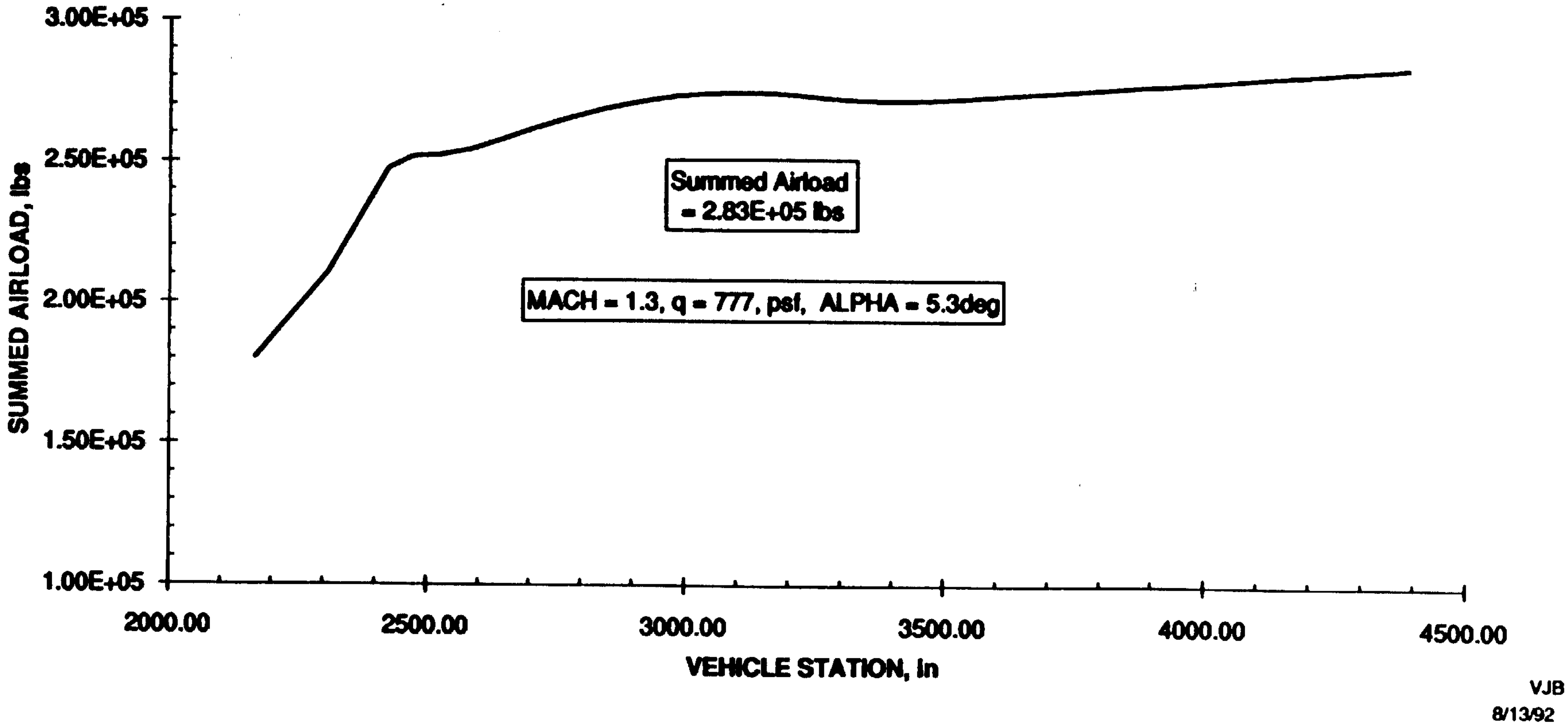
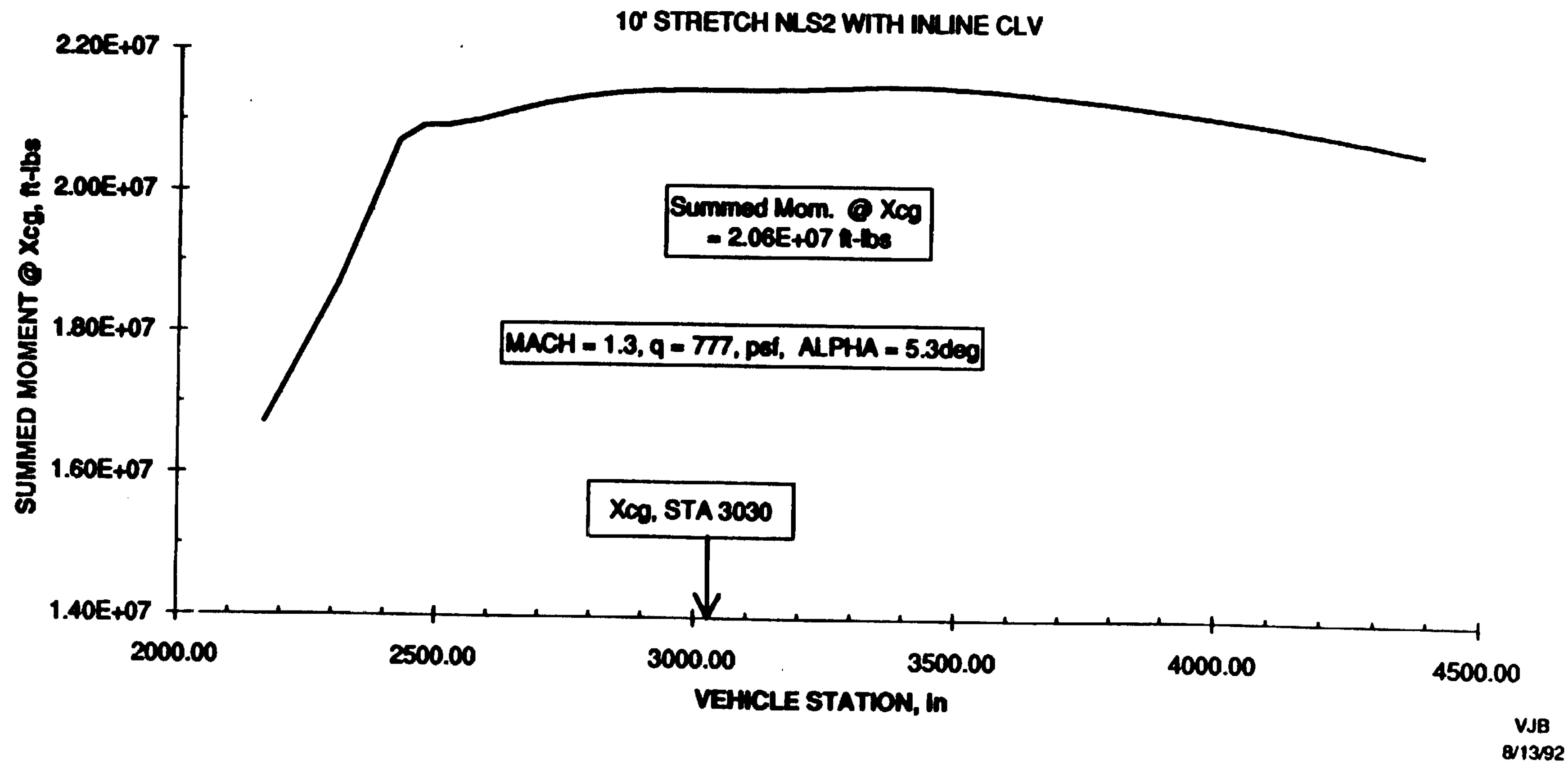


FIGURE 11. MAX q alpha AIRLOAD BENDING MOMENT @ Xcg



The results of this calculation are shown at the bottom of Table 2 where δ and the accelerations are shown in the boxed area at the bottom of Table 2. In addition, Figures 12 and 13 show the δ sensitivity with Xcg and the product of $q\alpha$.

This added gimbal angle can be used to evaluate potential launch configurations for gimbal control feasibility. It is also used to improve the finite element model (FEM) structural analyses by specifying the magnitudes of the inertial acceleration and correct thrust component inputs to the FEM.

III. Finite Element Model

A finite element representation was created to aid in assessing the impact on the tankage of carrying a CLV. The finite element model uses the structural sizing that was generated for the NLS baseline design. The model has the following characteristics:

1. Nodes ≈ 1800
2. Degrees of Freedom ≈ 11,000
3. Elements ≈ 4600

The model was run with unit load cases for checkout purposes and then with two balanced conditions (on-pad wind, high-Q). The stresses and deflections for the balanced conditions are presented in a later section of this report.

The model is shown in Figures 14 through 16.

IV. Mass Properties

A weight breakdown (Table 3) has been generated to support the stress, dynamic, and performance analyses. The mass properties are used as the starting point for creating inertial loads and a mass matrix (used in computing mode shapes and frequencies).

The vehicle was broken down into its basic systems, the mass properties found for each system, then the systems reassembled. Some systems were not included on each vehicle, but were included in the analysis if they were applicable. The general breakdown included: structure, payload, propulsion, TPS, avionics, and fuel. The total vehicle weight breakdown and distribution was then found by summing up each system. Spreadsheets were created to help in the determination of the weight breakdown for each vehicle.

Much of the rational used to find these weights were taken from both the Shuttle and Saturn programs. The Shuttle external tank and engines, and Saturn designed bulkheads, propulsion cones, and subsystems were used as guidelines for mass properties determination.

V. Vehicle Load Conditions

The vehicle was analyzed for two balanced conditions-- (1) on-pad winds, and (2) a high-Q flight condition. The first condition includes inertial (one-G) and aerodynamic effects. The second condition includes inertial, thrust, and aerodynamic forces.

FIGURE 12. SENSITIVITY OF GIMBAL REQM'T WITH Xcg

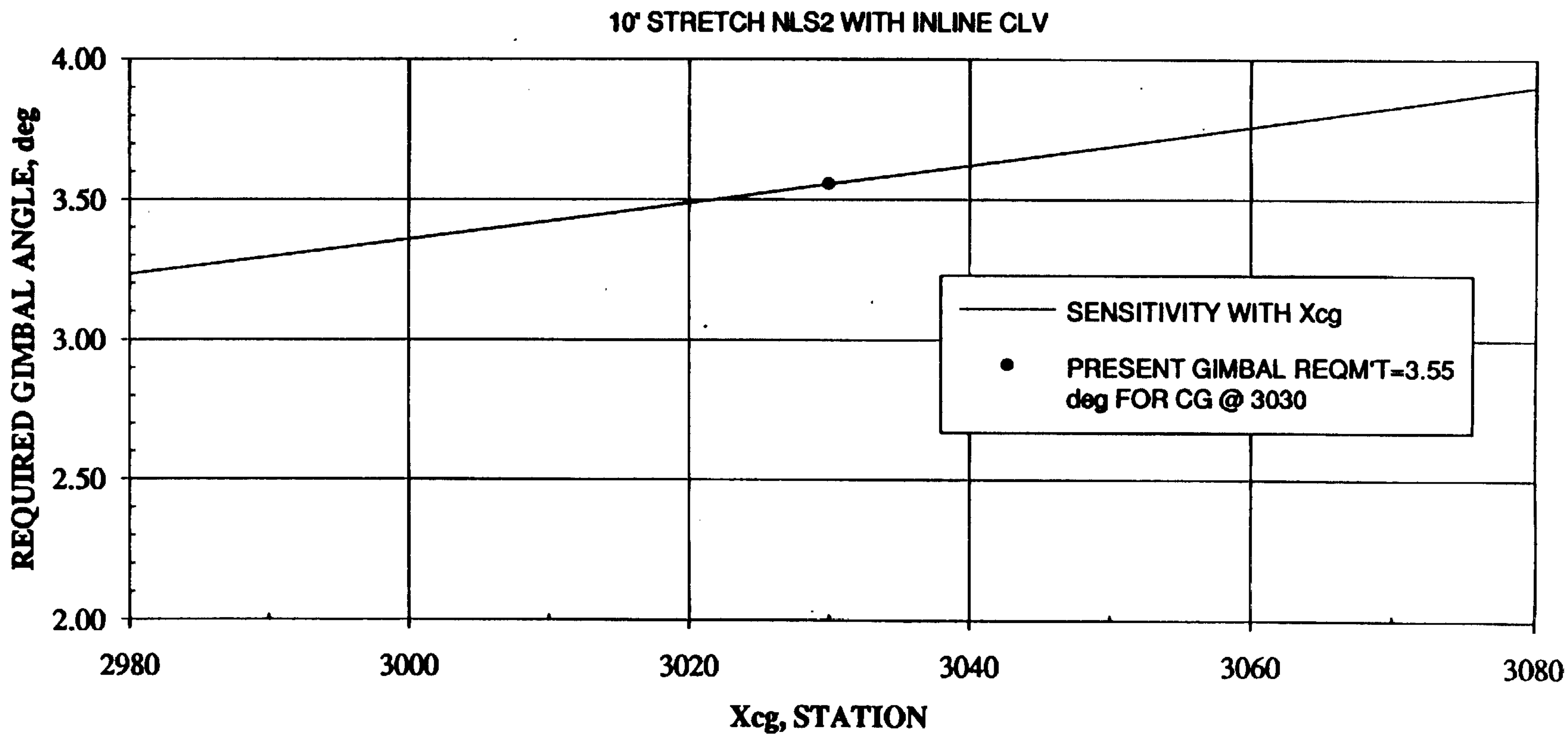
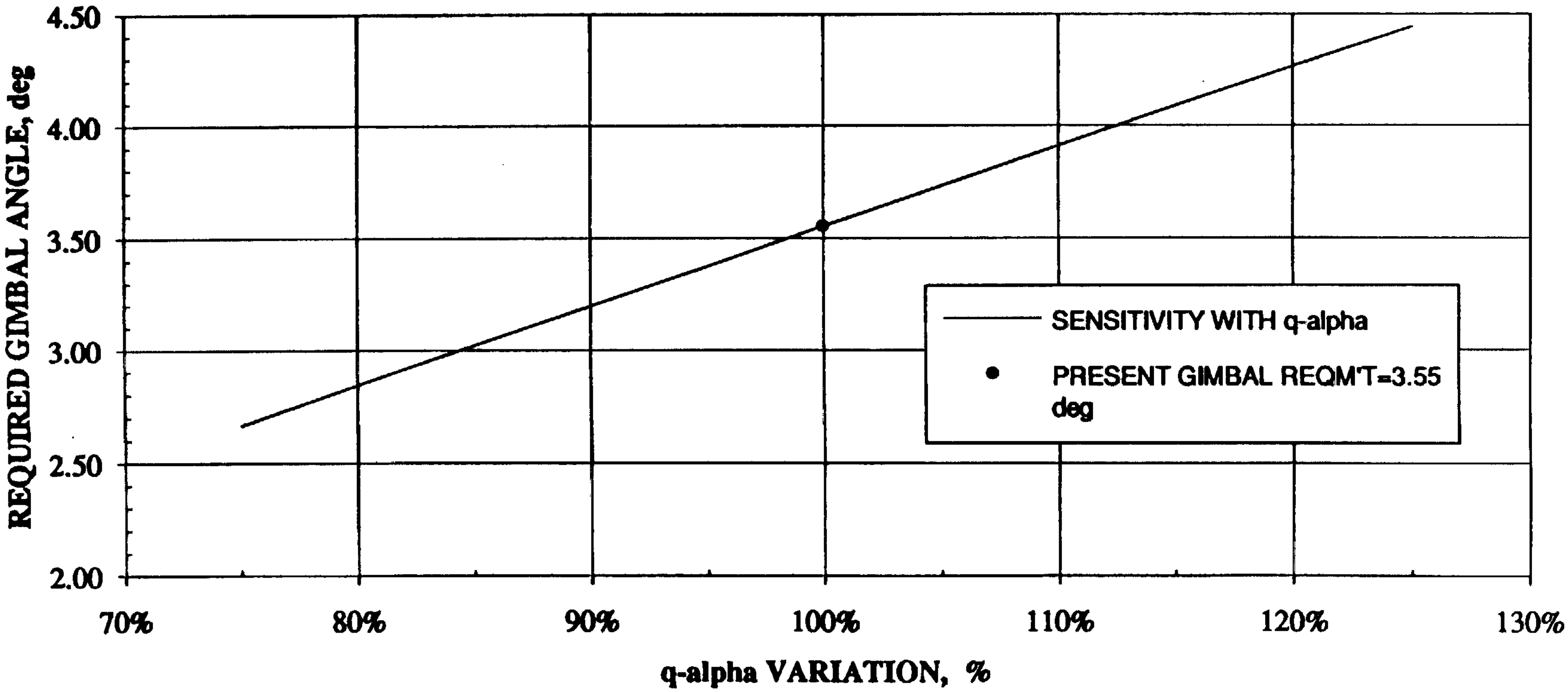


FIGURE 13. SENSITIVITY OF GIMBAL REQM'T WITH q-alpha

10' STRETCH NLS2 WITH INLINE CLV



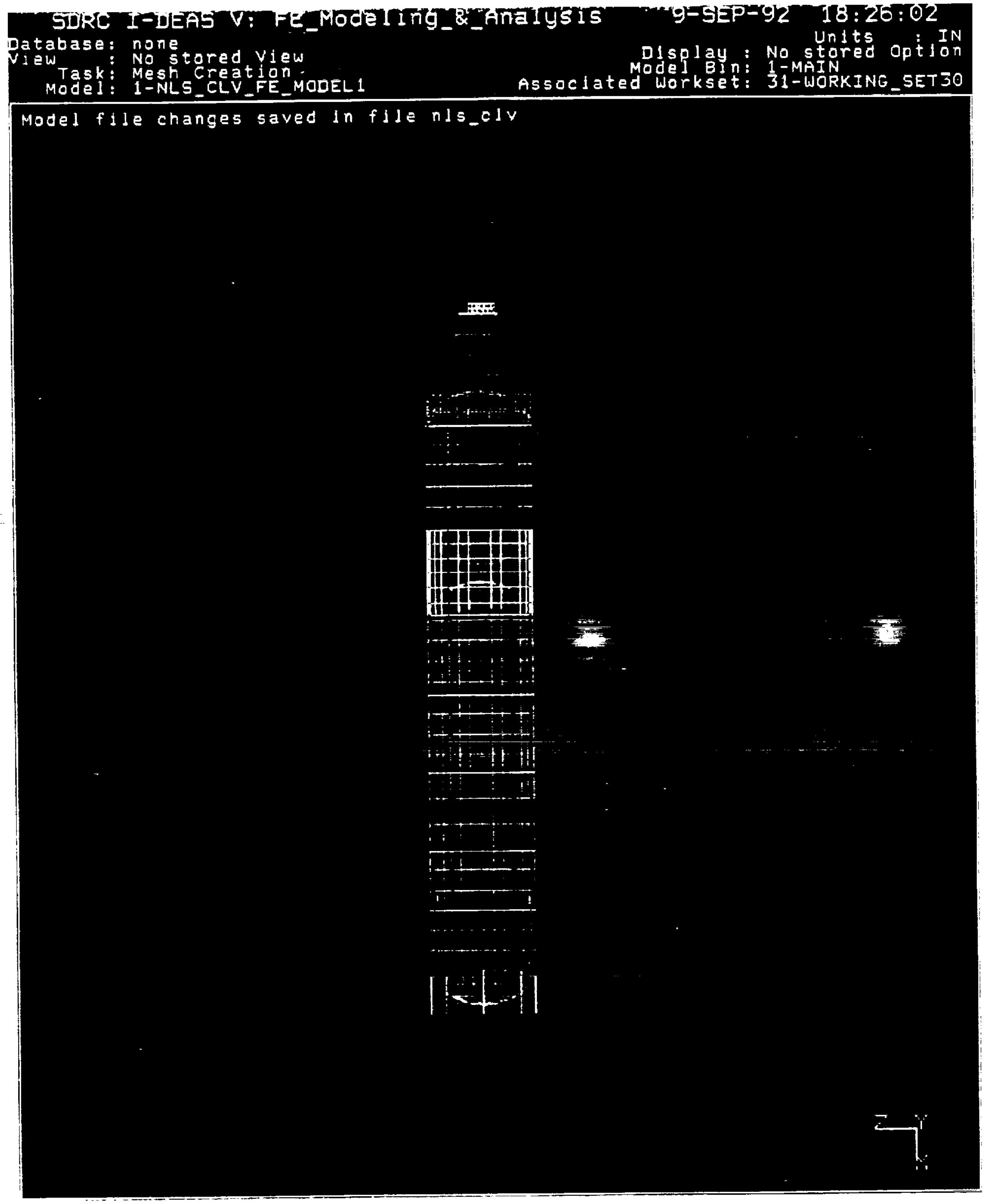


FIGURE 14. FINITE ELEMENT MODEL

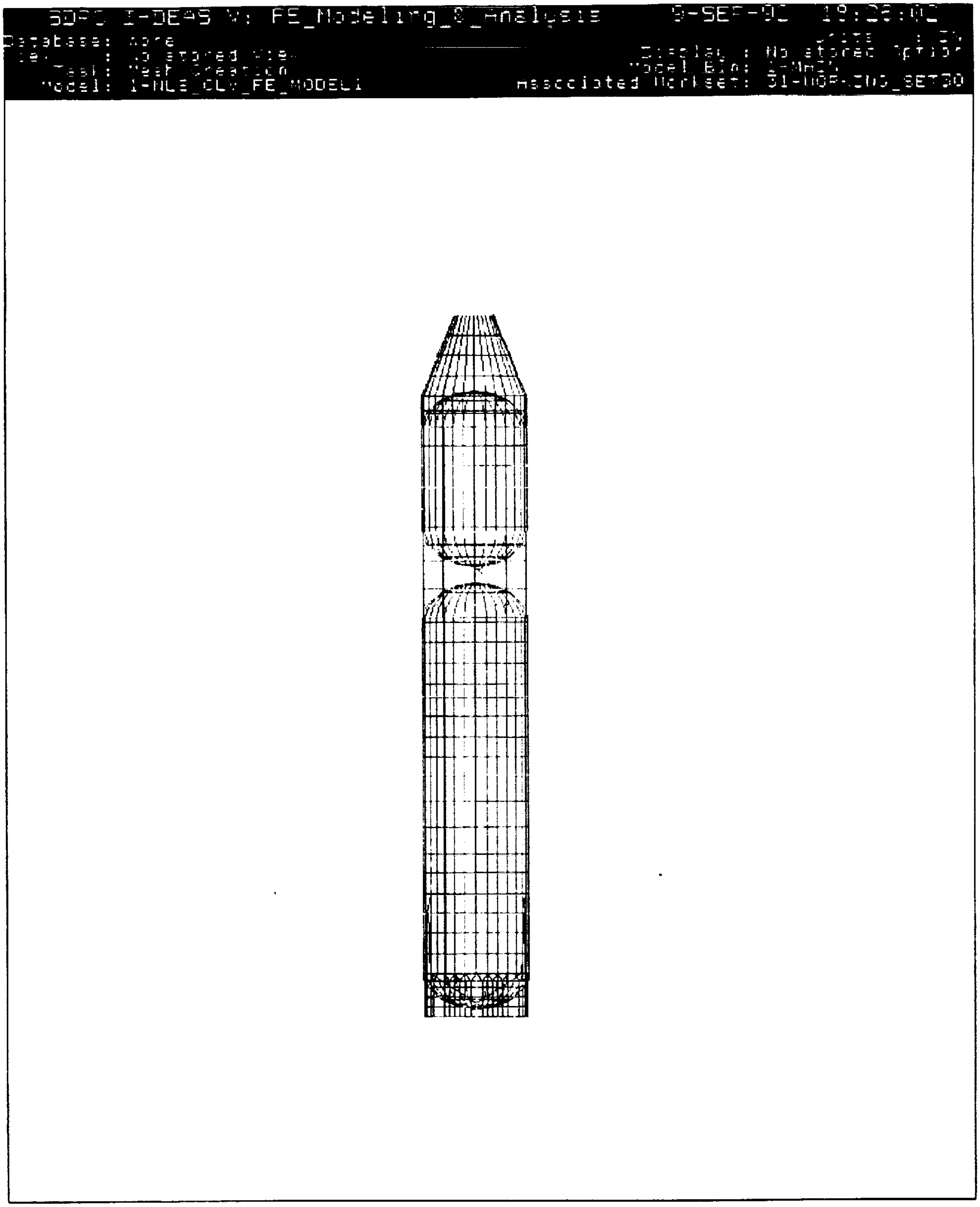


FIGURE 15. FINITE ELEMENT MODEL

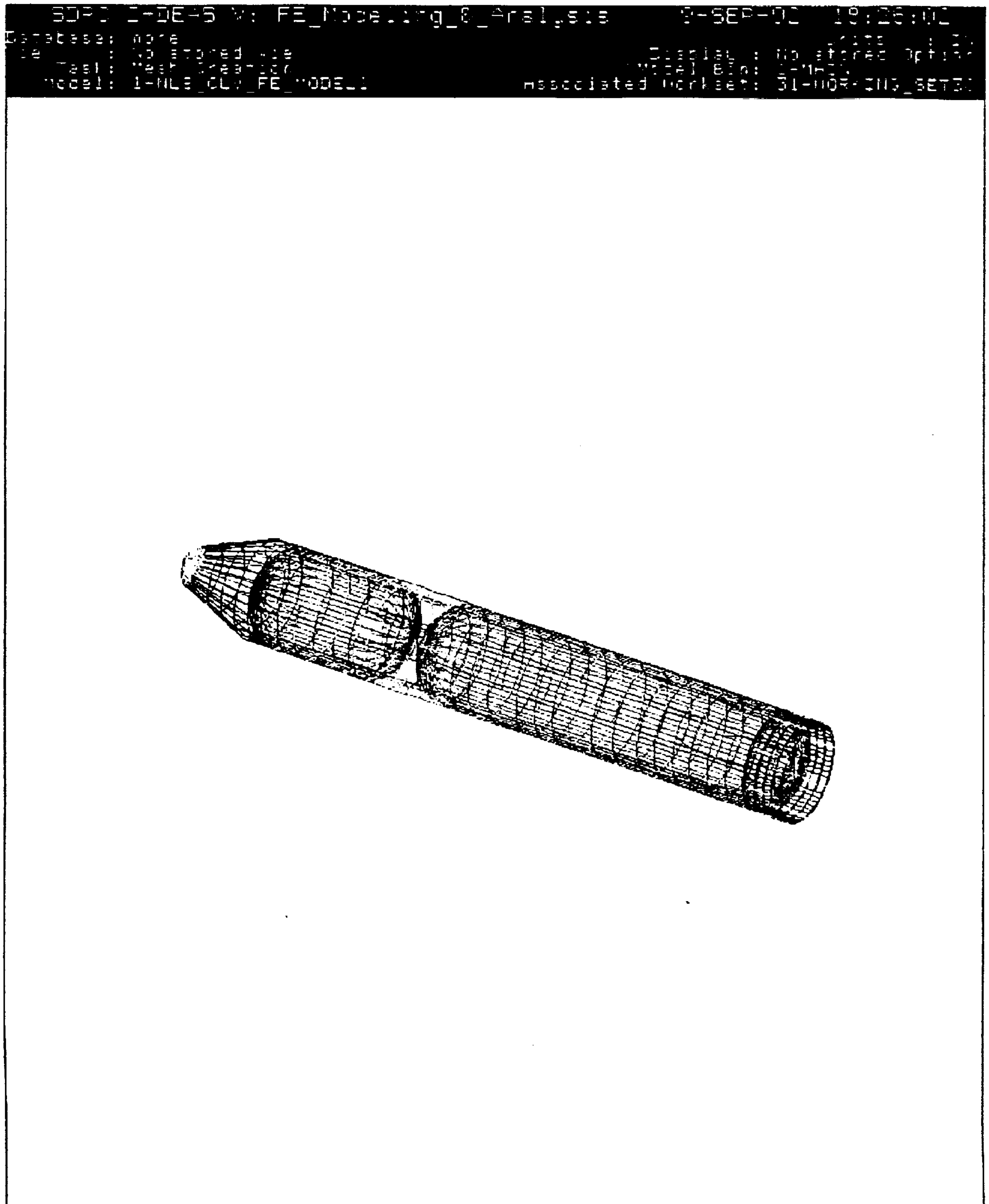


FIGURE 16. FINITE ELEMENT MODEL

TABLE 3. NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CLV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
CLV					(81444)
CG (Xclv)	456.5***	0	81444		
Σ	*** FROM TAIL OF CLV			81444	
CORE TANK					(88213)
FWD SKIRT	2480.4	1696	1781		
	2520.8	1084	1138		
Σ				2919	
LO2 TK-F DOME	2410.1	26	27		
(FIXED WT)	2417.6	209	219		
	2440.6	388	407		
	2480.4	545	572		
	2520.8	281	295		
Σ				1521	
CYLINDER	2520.8	816	857		
	2580.5	998	1048		
	2644	1724	1810		
	2711.7	1814	1905		
	2778.8	1724	1810		
	2838.4	1089	1143		
	2852.8	907	952		
Σ				9526	
LO2 TK-A DOME	2852.8	86	90		
(FIXED WT)	2892.5	690	725		
	2932.3	1285	1349		
	2955.3	1803	1893		
	2962.7	931	978		
Σ				5035	
INTERTANK	2852.8	1306	1371		
	2897.1	1979	2078		
	2941.4	1966	2064		
	2985.6	2067	2170		
	3034.2	2143	2250		
	3083.3	1979	2078		
	3123.1	1243	1305		
Σ				13317	
LH2 TK-F DOME	3013.1	430	452		
(FIXED WT)	3020.6	833	875		
	3043.6	594	624		
	3083.4	319	335		
	3123.1	40	42		
Σ				2327	
CYLINDER	3137.5	943	990		
	3201.7	1362	1430		
	3266.5	2200	2310		
	3331.3	2235	2347		
	3377.3	1886	1980		

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TABLE 3 (con't). NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CLV) - STRETCH "B" VERSION

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
	3435.9	1781	1870		
	3500.7	2096	2201		
	3565.5	2200	2310		
	3623.8	2096	2201		
	3706.1	2410	2531		
	3784.9	2759	2897		
	3871.0	2829	2970		
	3932.0	2515	2641		
	3996.8	2165	2273		
	4058	2165	2273		
	4122.6	1921	2017		
	4187.6	1118	1174		
	4252.2	244	256		
Σ				36671	
LH2 TK-A DOME	4252.2	55	58		
(FIXED WT)	4292.6	444	466		
	4332.4	826	867		
	4355.4	1159	1217		
	4362.8	598	628		
Σ				3236	
AFT SKIRT	4355.4	2211	2211		
(*NO STATION LOC)	4362.8	2210	2210		
Σ				4421	
EXT HARDWARE	2520.8	30	32		
	2580.5	35	37		
	2644	62	65		
	2711.7	65	68		
	2778.8	61	64		
	2838.4	38	40		
	2852.8	29	30		
	2897.1	43	45		
	2941.4	43	45		
	2985.6	44	46		
	3034.2	46	48		
	3083.3	43	45		
	3123.1	27	28		
	3137.5	38	40		
	3201.7	62	65		
	3266.5	62	65		
	3331.3	53	56		
	3377.3	50	53		
	3435.9	59	62		
	3500.7	62	65		
	3565.5	59	62		
	3623.8	68	71		
	3706.1	77	81		
	3784.9	79	83		

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TABLE 3 (con't). NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CLV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	3871.0	71	75		
	3932.0	61	64		
	3996.8	61	64		
	4058	55	58		
	4122.6	30	32		
	4187.6	6	6		
Σ				1595	
TPS-LO2 TANK	2520.8	15	16		
	2580.5	21	22		
	2644	36	38		
	2711.7	38	40		
	2778.8	36	38		
	2838.4	23	24		
	2852.8	23	24		
Σ				202	
TPS-LH2 TANK	3137.5	27	28		
	3201.7	38	40		
	3266.5	62	65		
	3331.3	63	66		
	3377.3	53	56		
	3435.9	50	53		
	3500.7	59	62		
	3565.5	62	65		
	3623.8	59	62		
	3706.1	68	71		
	3784.9	78	82		
	3871.0	80	84		
	3932.0	71	75		
	3996.8	61	64		
	4058	61	64		
	4122.6	54	57		
	4187.6	32	34		
	4252.2	8	8		
Σ				1035	
INSULATION	2852.8	94	99		
	2897.1	142	149		
	2941.4	141	148		
	2985.6	148	155		
	3034.2	154	162		
	3083.3	142	149		
	3123.1	89	93		
Σ				956	
LO2-FEED	2985.6	136	143		
	3034.2	139	146		
	3083.3	129	135		
	3123.1	81	85		
	3137.5	115	121		

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TABLE 3 (con't). NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CLV) - STRETCH "B" VERSION

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
	3201.7	186	195		
	3266.5	190	200		
	3331.3	163	171		
	3377.3	152	160		
	3435.9	180	189		
	3500.7	190	200		
	3565.5	180	189		
	3623.8	207	217		
	3706.1	234	246		
	3784.9	240	252		
	3871.0	213	224		
	3932.0	183	192		
	3996.8	183	192		
	4058	163	171		
	4122.6	95	100		
	4187.6	29	30		
Σ				3557	
LO2 - PRESSURE	2330.3	11	12		
	2424.8	9	9		
	2472.8	6	6		
	2580.5	4	4		
	2644	4	4		
	2711.7	8	8		
	2778.8	8	8		
	2838.4	8	8		
	2852.8	5	5		
	2897.1	4	4		
	2941.4	5	5		
	2985.6	5	5		
	3034.2	6	6		
	3083.3	6	6		
	3123.1	5	5		
	3137.5	3	3		
	3201.7	5	5		
	3266.5	8	8		
	3331.3	8	8		
	3377.3	7	7		
	3435.9	6	6		
	3500.7	7	7		
	3565.5	8	8		
	3623.8	7	7		
	3706.1	9	9		
	3784.9	10	11		
	3871	10	11		
	3932	9	9		
	3996.8	8	8		
	4058	8	8		

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TABLE 3. (con't) NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CLV) - STRETCH "B" VERSION

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
	4122.6	7	7		
	4187.6	4	4		
	4252.2	1	1		
Σ				230	
LO2 - VENT	2330.3	27	28		
	2424.8	21	22		
	2472.8	14	15		
	2580.5	9	9		
	2644	11	12		
	2711.7	19	20		
	2778.8	20	21		
	2838.4	18	19		
	2852.8	12	13		
	2897.1	9	9		
	2941.4	13	14		
	2985.6	13	14		
	3034.2	14	15		
	3083.3	14	15		
	3123.1	13	14		
	3137.5	8	8		
	3201.7	12	13		
	3266.5	19	20		
	3331.3	19	20		
	3377.3	16	17		
	3435.9	15	16		
	3500.7	18	19		
	3565.5	19	20		
	3623.8	18	19		
	3706.1	20	21		
	3784.9	23	24		
	3871	24	25		
	3932	21	22		
	3996.8	18	19		
	4058	18	19		
	4122.6	16	17		
	4187.6	9	9		
	4252.2	2	2		
Σ				548	
LH2 - FEED	4187.6	84	88		
	4252.2	84	88		
				176	
LH2 - PRESSURE	3034.2	11	12		
	3083.3	12	13		
	3123.1	11	12		
	3137.5	7	7		
	3201.7	10	11		
	3266.5	16	17		

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TABLE 3 (con't). NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CLV) - STRETCH "B" VERSION

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
	3331.3	16	17		
	3377.3	13	14		
	3435.9	13	14		
	3500.7	15	16		
	3565.5	16	17		
	3623.8	15	16		
	3706.1	17	18		
	3784.9	20	21		
	3871	20	21		
	3932	18	19		
	3996.8	15	16		
	4058	15	16		
	4122.6	14	15		
	4187.6	8	8		
	4252.2	2	2		
Σ				298	
LH2 - VENT	3034.2	14	15		
	3083.3	15	16		
	3123.1	14	15		
	3137.5	8	8		
	3201.7	12	13		
	3266.5	19	20		
	3331.3	20	21		
	3377.3	17	18		
	3435.9	16	17		
	3500.7	19	20		
	3565.5	20	21		
	3623.8	19	20		
	3706.1	21	22		
	3784.9	24	25		
	3871	25	26		
	3932	22	23		
	3996.8	19	20		
	4058	19	20		
	4122.6	17	18		
	4187.6	10	11		
	4252.2	2	2		
Σ				370	
RANGE SAFETY	2330.3	11	12		
	2424.8	7	7		
	2472.8	5	5		
	2580.5	6	6		
	2644	10	11		
	2711.7	10	11		
	2778.8	10	11		
	2838.4	6	6		
	2852.8	4	4		

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TABLE 3 (con't). NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CLV) - STRETCH "B" VERSION

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
	2897.1	7	7		
	2941.4	7	7		
	2985.6	7	7		
	3034.2	7	7		
	3083.3	7	7		
	3123.1	4	4		
	3137.5	6	6		
	3201.7	10	11		
	3266.5	10	11		
	3331.3	8	8		
	3377.3	8	8		
	3435.9	9	9		
	3500.7	10	11		
	3565.5	9	9		
	3623.8	11	12		
	3706.1	12	13		
	3784.9	13	14		
	3871	11	12		
	3932	10	11		
	3996.8	10	11		
	4058	9	9		
	4122.6	5	5		
	4187.6	1	1		
Σ				273	
TOTAL WEIGHT				169657	

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TABLE 3 (con't). DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION (CLV)

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
PODS					(36528)
STRUCT - BEAMS	4252.2	140	168		
	4272.8	701	842		
	4295.5	1402	1684		
	4325.5	1402	1684		
	4355.5	701	842		
	4385.5	327	393		
Σ				5613	
LONGERONS	4252.2	24	29		
	4272.8	119	143		
	4295.5	238	286		
	4325.5	238	286		
	4355.5	119	143		
	4385.5	56	67		
Σ				952	
COVER	4252.2	48	58		
	4272.8	241	290		
	4295.5	482	579		
	4325.5	482	579		
	4355.5	241	290		
	4385.5	113	135		
Σ				1931	
S/S INSTAL	4295.5	49	59		
	4325.5	244	293		
	4355.5	244	293		
	4385.5	73	88		
Σ				731	
TPS - COVER	4252.2	33	39		
	4272.8	33	39		
	4295.5	33	39		
	4325.5	33	39		
	4355.5	33	39		
	4385.5	33	39		
Σ				236	
CANNISTERS	4295.5	198	237		
	4325.5	395	475		
	4355.5	395	475		
	4385.5	198	237		
Σ				1424	
SEPARATION	4355.5	102	123		
	4385.5	102	123		
Σ				245	
FEED	4252.2	82	99		
	4272.8	412	495		

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TABLE 3 (con't). DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION (CLV)

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	4295.5	824	990		
	4325.5	824	990		
	4355.5	412	495		
	4385.5	192	231		
Σ				3299	
PNEUMATICS	4252.2	506	607		
	4272.8	506	607		
Σ				1214	
TVC	4295.5	95	115		
	4325.5	477	573		
	4355.5	477	573		
	4385.5	143	172		
Σ				1432	
DE-ORBIT	4355.5	770	925		
	4385.5	770	925		
Σ				1850	
ENGINES	4435	17600	17600		
Σ				17600	
JETTISON					(53907)
STRUCT	4252.2	317	248		
	4272.8	1587	1241		
	4295.5	3175	2483		
	4325.5	3175	2483		
	4355.5	1587	1241		
	4385.5	741	579		
Σ				8276	
S/S INSTAL	4252.2	35	27		
	4272.8	173	135		
	4295.5	346	271		
	4325.5	346	271		
	4355.5	173	135		
	4385.5	81	63		
Σ				902	
TPS - CYL	4252.2	5	4		
	4272.8	26	20		
	4295.5	52	41		
	4325.5	52	41		
	4355.5	26	20		
	4385.5	12	9		
Σ				135	
TPS - BHS	4355.5	295	231		
	4385.5	295	231		
Σ				461	
TPS - CANNISTERS	4435	1976	1545		
Σ				1545	

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TABLE 3 (con't). DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION (CLV)

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
SEPARATION	4252.2	102	80		
	4272.8	102	80		
Σ				160	
FEED	4187.6	204	160		
	4252.2	885	692		
	4272.8	1021	799		
	4295.5	1702	1331		
	4325.5	1702	1331		
	4355.5	1021	799		
	4385.5	272	213		
Σ				5324	
PNEUMATIC	4187.6	2	1		
	4252.2	7	5		
	4272.8	8	6		
	4295.5	13	10		
	4325.5	13	10		
	4355.5	8	6		
	4385.5	2	2		
Σ				39	
TVC	4406	2384	1864		
Σ				1864	
ENGINES	4463	35200	35200		
Σ				35200	
AVIONICS					(7334)
JETTISON - BATT	4252.2	231	231		
	4272.8	455	455		
	4295.5	446	446		
	4325.5	465	465		
	4355.5	481	481		
	4385.5	282	282		
Σ				2360	
BATT CNTRL	4325.5	52	52		
	4355.5	108	108		
	4385.5	53	53		
Σ				213	
EIU	4355.5	28	28		
	4385.5	29	29		
Σ				57	
RETAINED - BATT	4252.2	139	139		
	4272.8	243	243		
	4295.5	227	227		
	4325.5	227	227		
	4355.5	229	229		
	4385.5	115	115		
Σ				1180	
RETAINED - BATT	4325.5	27	27		

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TABLE 3 (con't). DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION (CLV)

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	4355.5	52	52		
	4385.5	27	27		
Σ				106	
EIU	4295.5	29	29		
Σ				29	
OTHER	4295.5	519	519		
Σ				519	
CONICAL ADAPTER	2330.3	242	242		
	2424.8	604	604		
	2472.8	362	362		
Σ				1208	
OTHER	2424.8	33	33		
	2472.8	45	45		
	2580.5	30	30		
	2644	35	35		
	2711.7	68	68		
	2778.8	68	68		
	2838.4	68	68		
	2852.8	40	40		
	2897.1	26	26		
	2941.4	40	40		
	2985.6	40	40		
	3034.2	43	43		
	3083.3	45	45		
	3123.1	41	41		
	3137.5	25	25		
	3201.7	36	36		
	3266.5	68	68		
	3331.3	68	68		
	3377.3	54	54		
	3435.9	51	51		
	3500.7	56	56		
	3565.5	59	59		
	3623.8	56	56		
	3706.1	73	73		
	3784.9	89	89		
	3871	76	76		
	3932	73	73		
	3996.8	100	100		
	4058	58	58		
	4122.6	56	56		
	4187.6	35	35		
	4252.2	7	7		
Σ				1662	
R&R					(17859)

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TABLE 3 (con't). DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION (CLV)

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
JETTISON					
LO2-FEED	4252.2	132	115		
	4272.8	231	201		
	4295.5	215	187		
	4325.5	216	188		
	4355.5	110	96		
Σ				786	
LO2-TORUS	4272.8	675	587		
	4295.5	1354	1178		
	4325.5	675	587		
Σ				2352	
LO2-LINES	4272.8	250	218		
	4295.5	376	327		
	4325.5	751	653		
	4355.5	751	653		
	4385.5	376	327		
Σ				2178	
LH2	4252.2	6	5		
	4272.8	30	26		
	4295.5	60	52		
	4325.5	60	52		
	4355.5	30	26		
	4385.5	14	12		
Σ				174	
RETAINED					
LO2-LINES	3435.9	477	466		
	3500.7	499	488		
	3565.5	477	466		
	3623.8	543	531		
	3706.1	621	607		
	3784.9	637	623		
	3871	571	558		
	3932	488	477		
	3996.8	488	477		
	4058	432	422		
	4122.6	249	243		
	4187.6	61	60		
	4252.2	143	140		
	4272.8	267	261		
	4295.5	375	367		
	4325.5	193	189		
Σ				6374	
LO2 - TANK					
	2424.8	660	645		
	2472.8	1270	1241		
	2580.5	910	890		
	2644	490	479		

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TABLE 3 (con't). DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION (CLV)

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	2711.7	60	59		
Σ				3314	
LH2 - LINE	4252.2	293	286		
	4272.8	579	566		
	4295.5	493	482		
	4325.5	143	140		
Σ				1474	
LH2 - TANK	2985.6	42	41		
	3034.2	43	42		
	3083.3	40	39		
	3123.1	25	24		
	3137.5	36	35		
	3201.7	58	57		
	3266.5	58	57		
	3331.3	49	48		
	3377.3	47	46		
	3435.9	56	55		
	3500.7	58	57		
	3565.5	56	55		
	3623.8	63	62		
	3706.1	73	71		
	3784.9	74	72		
	3871	67	65		
	3932	57	56		
	3996.8	57	56		
	4058	51	50		
	4122.6	30	29		
	4187.6	38	37		
	4252.2	36	35		
	4272.8	20	20		
	4295.5	69	67		
	4325.5	32	31		
Σ				1207	
TOTAL WEIGHT				115628	
PROPELLANT-USABLE					(1894498)
LO2 TK-F DOME	2472.8	10147	11353		
(FIXED WT)	2580.5	89909	100598		
	2644	147435	164964		
Σ				276915	
CYLINDER	2580.5	123355	138021		
	2644	212243	237477		
	2711.7	224941	251684		
	2778.8	211336	236462		
	2838.4	135145	151212		
Σ				1014856	
LO2 TK-A DOME	2852.8	5272	5899		
(FIXED WT)	2897.1	42176	47190		

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TABLE 3 (con't). DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION (CLV)

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
	2941.4	78494	87826		
	2985.6	110126	123219		
	3034.2	56821	63576		
Σ				327711	
LINE	2985.6	231	258		
	3034.2	371	415		
	3083.3	501	561		
	3123.1	463	518		
	3137.5	291	326		
	3201.7	414	463		
	3266.5	667	746		
	3331.3	678	759		
	3377.3	576	644		
	3435.9	544	609		
	3500.7	646	723		
Σ				6022	
LH2 TK-F DOME	3083.3	652	730		
(FIXED WT)	3123.1	5281	5909		
Σ				6638	
CYLINDER	3137.5	8701	9735		
	3201.7	14138	15819		
	3266.5	14138	15819		
	3331.3	12181	13629		
	3377.3	11528	12899		
	3435.9	13486	15089		
	3500.7	14138	15819		
	3565.5	13486	15089		
	3623.8	15443	17279		
	3706.1	17619	19714		
	3784.9	18054	20200		
	3871	16096	18010		
	3932	13703	15332		
	3996.8	13703	15332		
	4058	12398	13872		
	4122.6	7178	8031		
	4187.6	1523	1704		
Σ				243373	
LH2 TK-A DOME	4122.6	373	417		
(FIXED WT)	4187.6	3037	3398		
	4252.2	5650	6322		
	4272.8	7906	8846		
Σ				18983	
TOTAL WEIGHT				2125755	

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VI. Resultant Stresses and Deflections

Stresses and deflections for the on-pad and high-Q conditions are shown in Figures 17 through 23. Figures 17 and 19 show vehicle deflections while the remainder show stresses. The tank stresses shown in the contour plots reflect a zero internal tank pressure. The tank pressures have been accounted for in the assessment summary shown in Table 4. The detailed computations are shown on the pages that follow Table 4.

VII. Dynamic Characteristics

The first five mode shapes and frequencies were computed for the vehicle with 100% fuel. The original mass and stiffness matrices were not reduced from over 11,000 degrees of freedom (DOF). Simultaneous vector iteration was used to compute the first five mode shapes and frequencies. The results are summarized in Table 5.

VIII. Comparisons with NLS/Other Vehicles

The on-pad and in-flight aerodynamic load distributions are compared with such distributions for the NLS with the standard payload shroud. The replacement of the payload shroud with the CLV produced negligible change in on-pad aero loads. However, it contributed to a significant change for the high-Q in-flight condition.

Comparison plots of on-pad summed drag load and moment along the NLS booster are shown in Figures 29 and 30 between the NLS/CLV and the basic NLS with the standard payload shroud, respectively. Due to the similar size between the CLV and the NLS payload shroud, these loads and moments are nearly identical for the two configurations.

Comparisons are shown in Figures 31 and 32 for the high-Q summed loads and moments between the NLS/CLV configuration (for the trajectory q and alpha), and the NLS with payload shroud (for its trajectory q and alpha). The normal force and moment for the NLS/CLV are seen to be greater than for the NLS with payload shroud. As a result of such differences, these comparisons show that the CLV geometry has a profound effect on the magnitude of the aero loads.

Table 6 gives a comparison of dynamic characteristics between the CLV stack and the basic STS external tank.

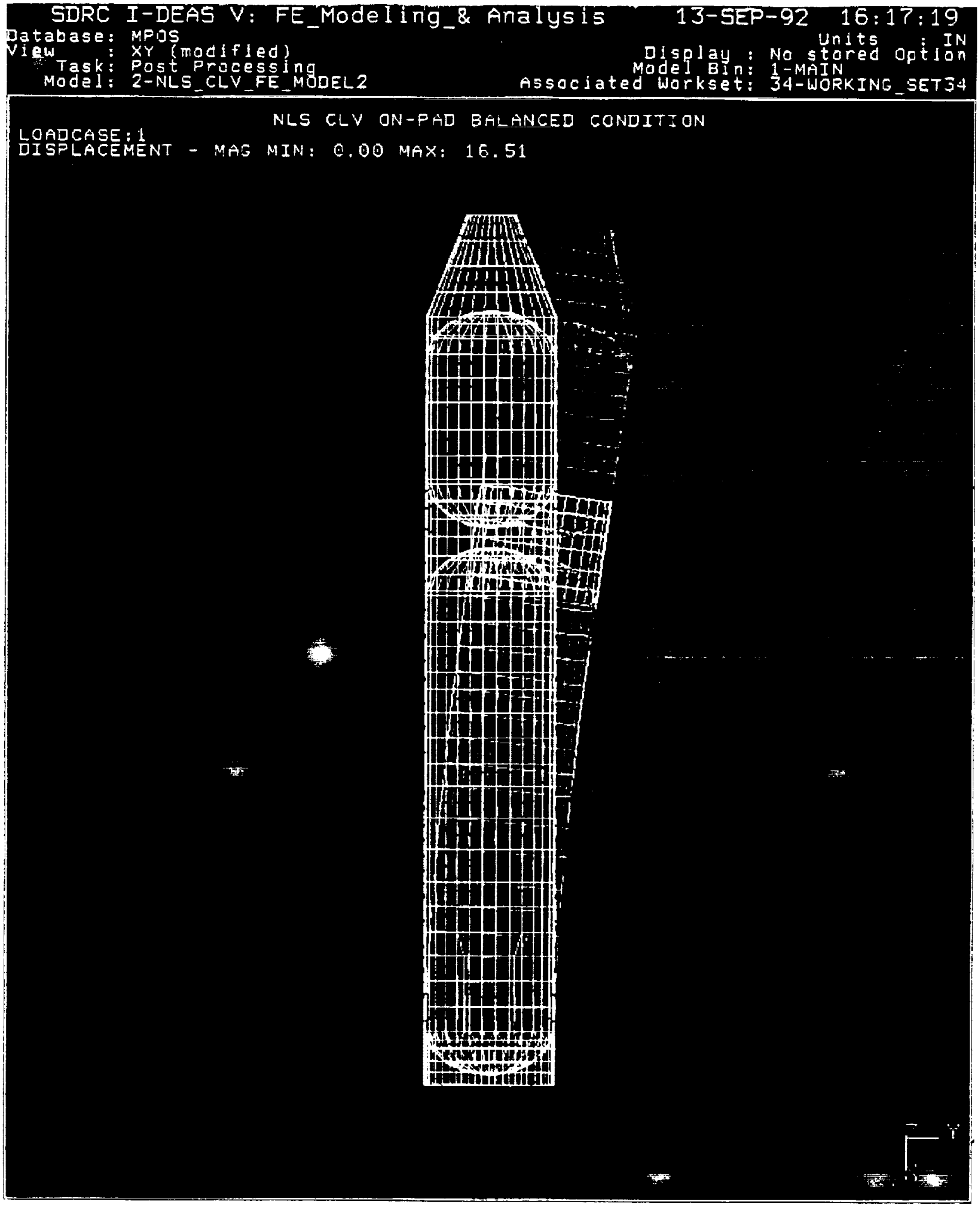

FIGURE 17. ON-PAD DEFLECTIONS

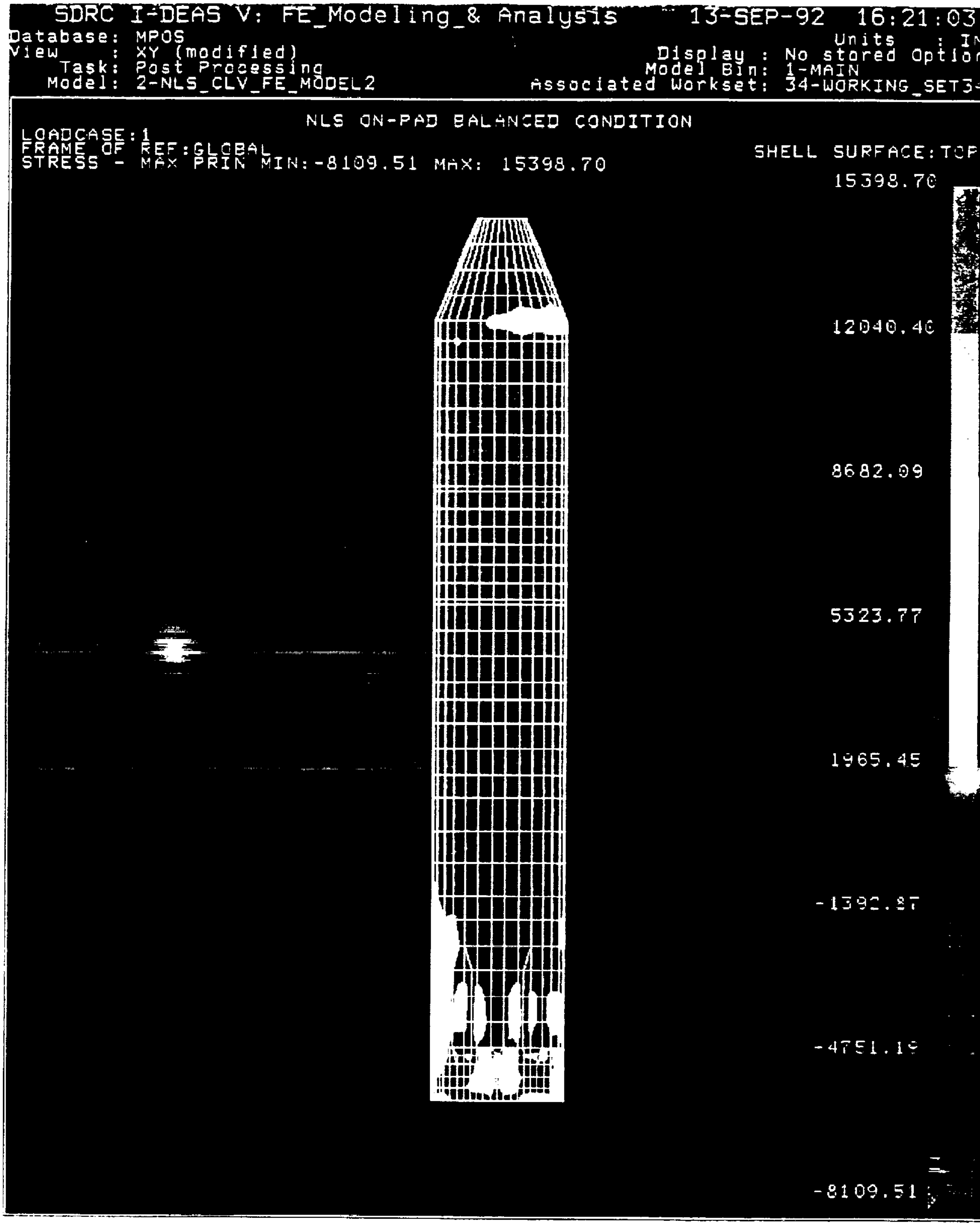

FIGURE 18. ON-PAD CONDITION, OUTER SHELL SURFACE STRESSES

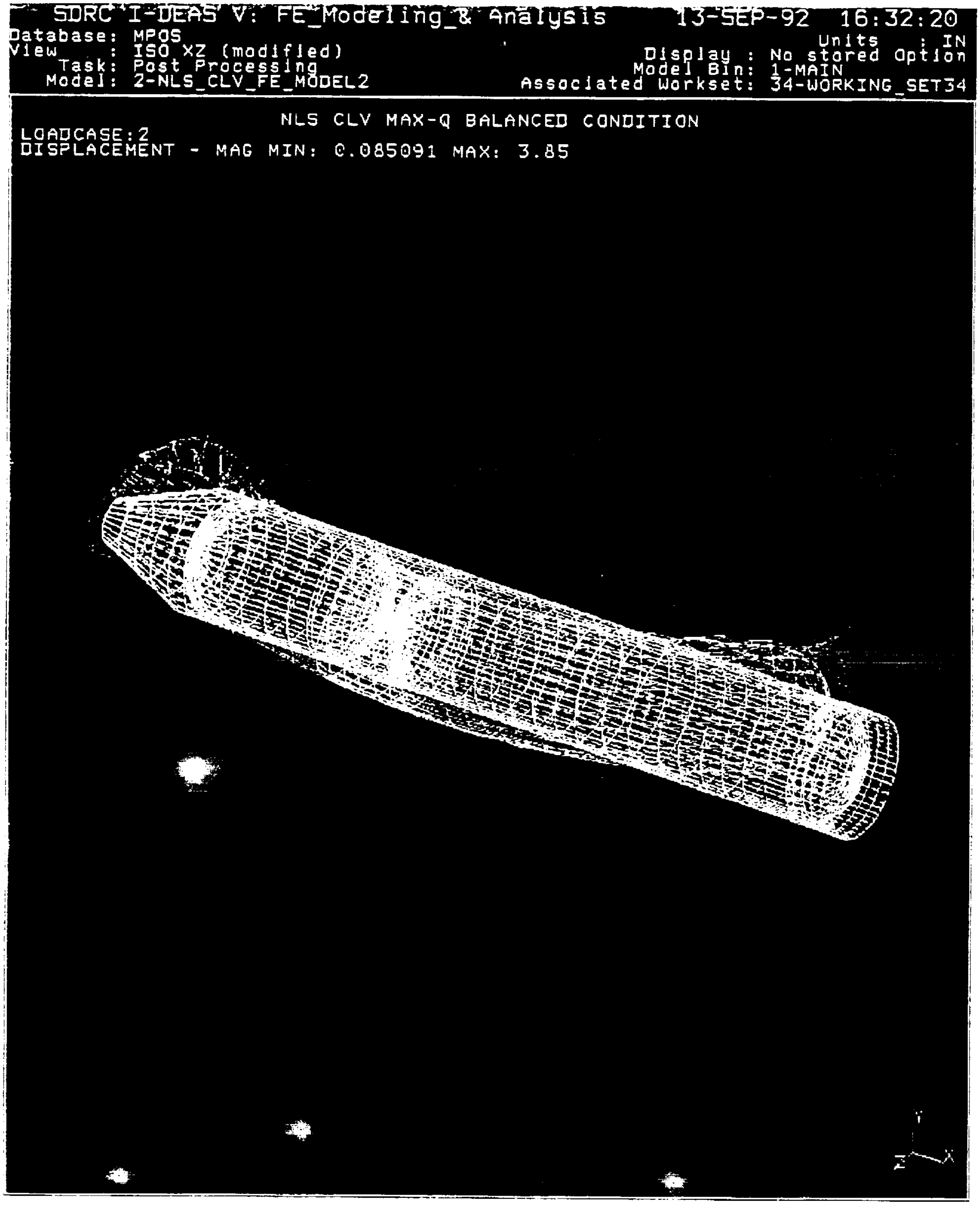

FIGURE 19. MAX Q DEFLECTIONS

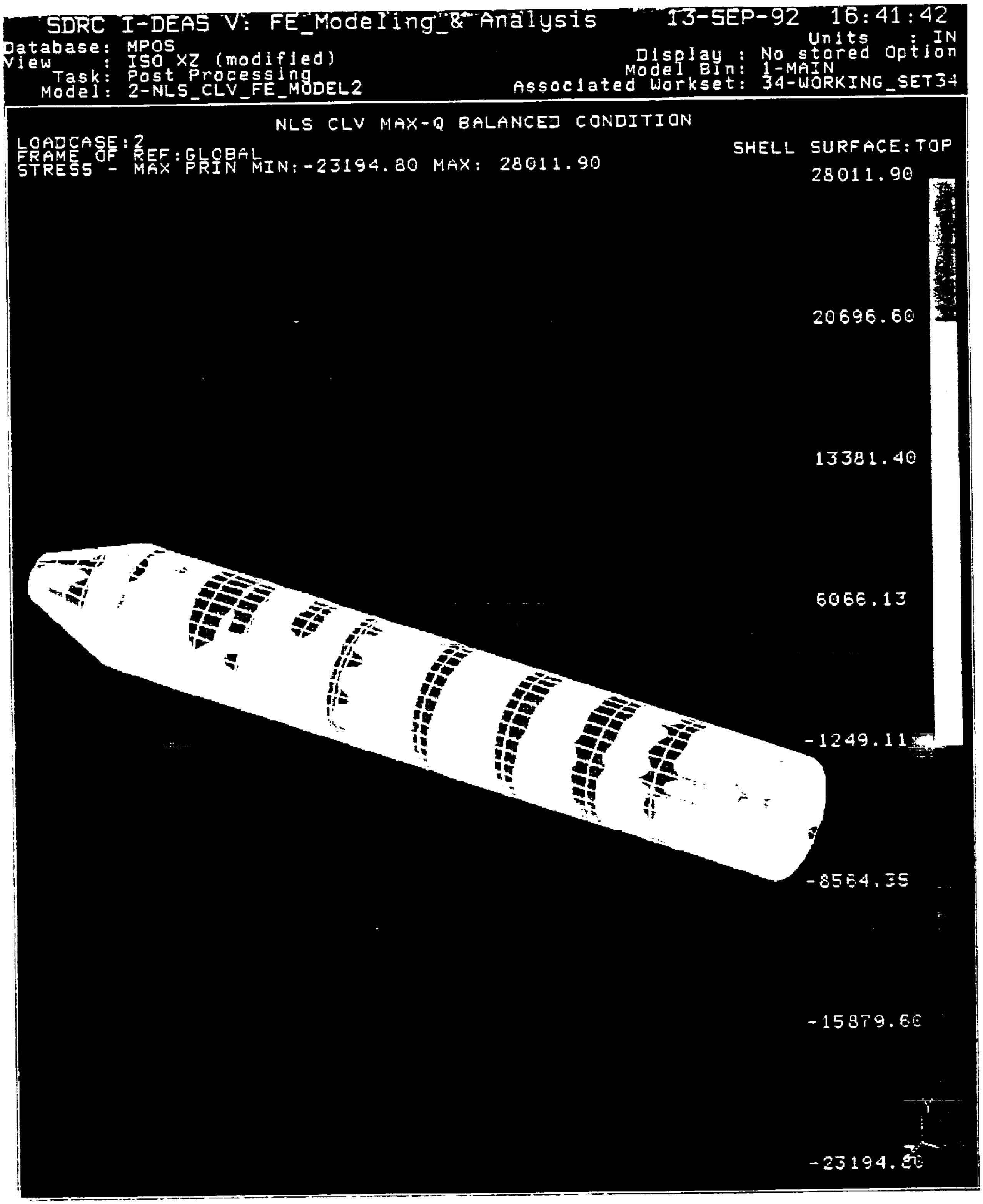

FIGURE 20. MAX Q STRESSES (SHELL OUTER SURFACE)

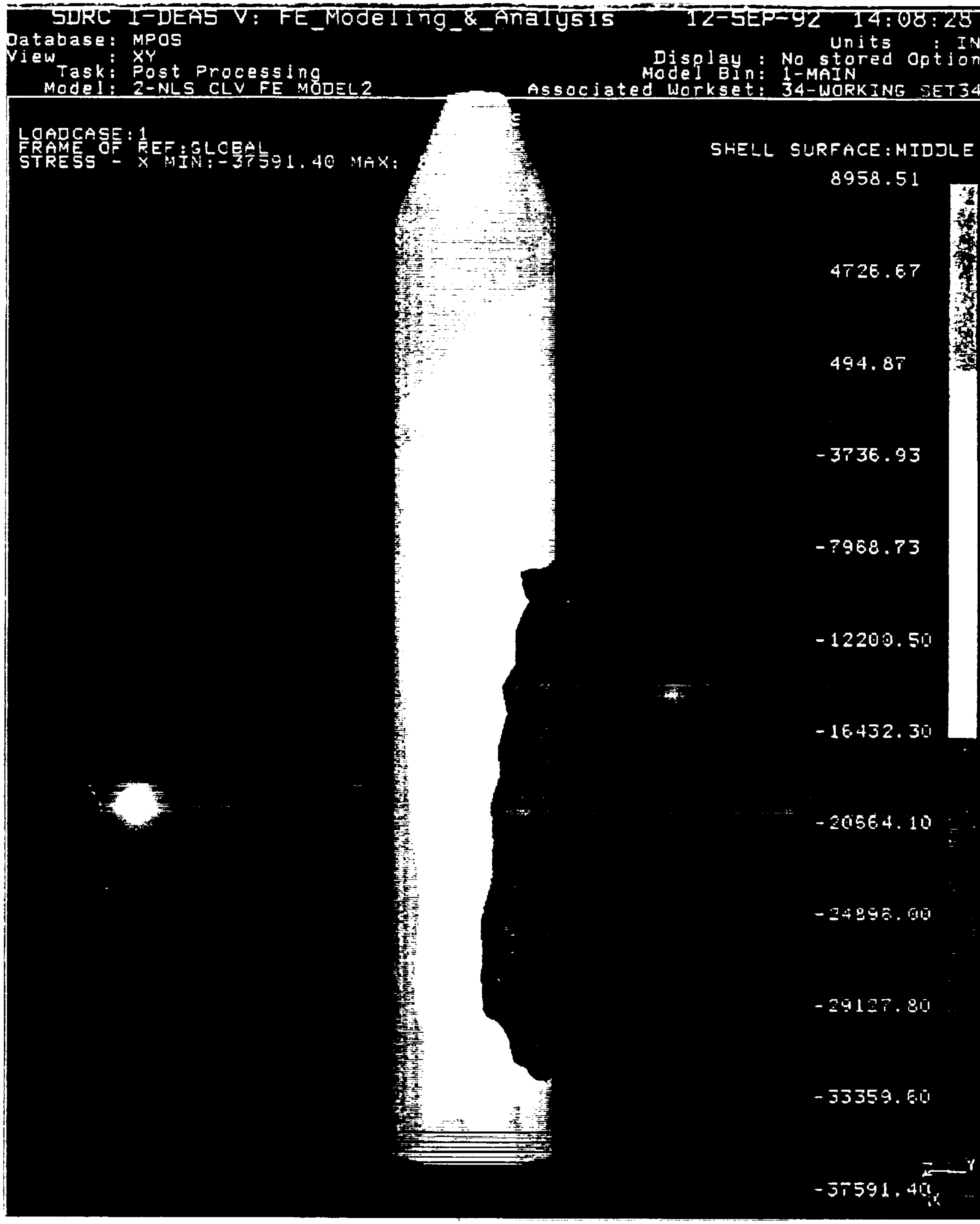

FIGURE 21. ON-PAD STRESSES (SHELL MIDDLE SURFACE, LONGITUDINAL)

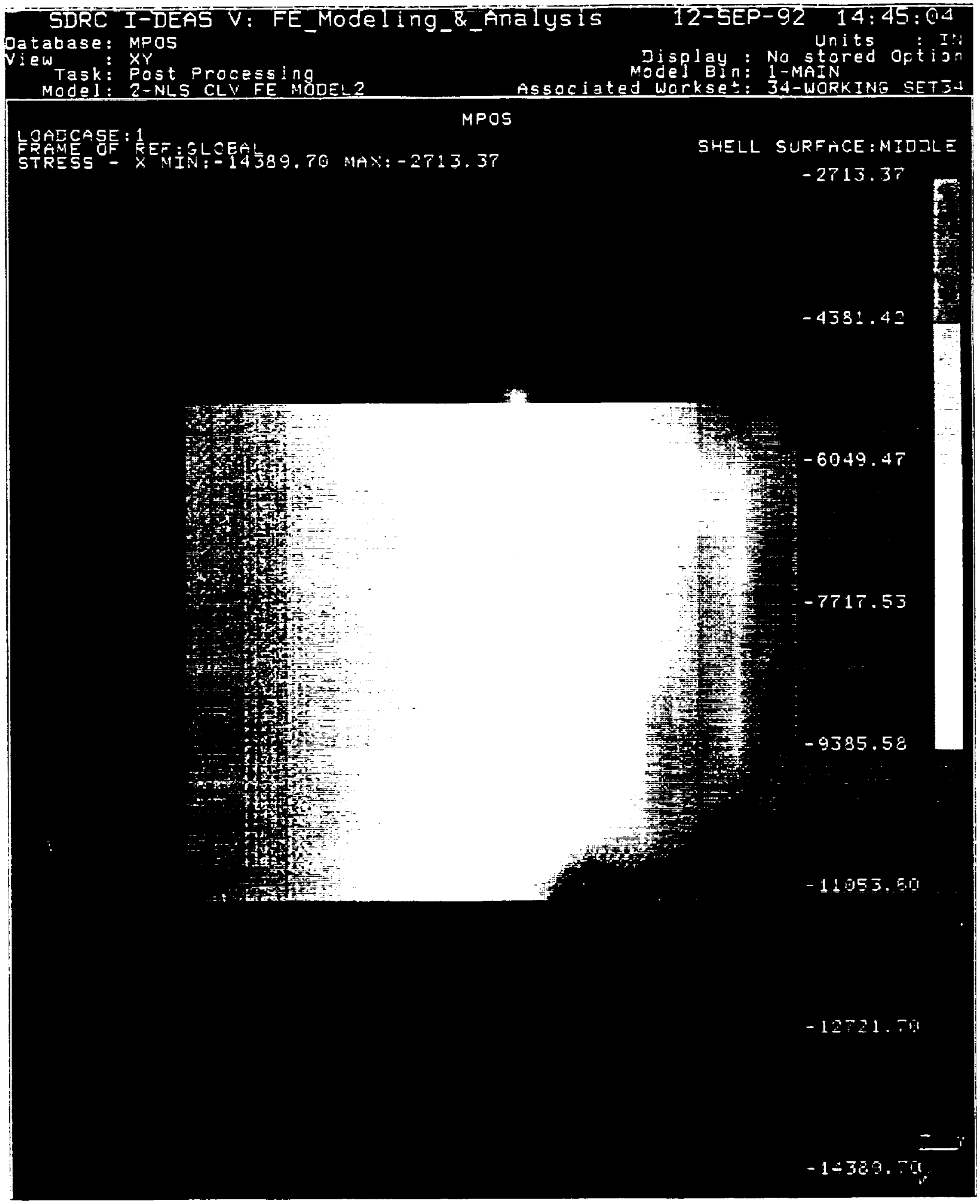

FIGURE 22. ON-PAD STRESSES, INTERTANK SECTION (SHELL MIDDLE SURFACE, LONGITUDINAL)

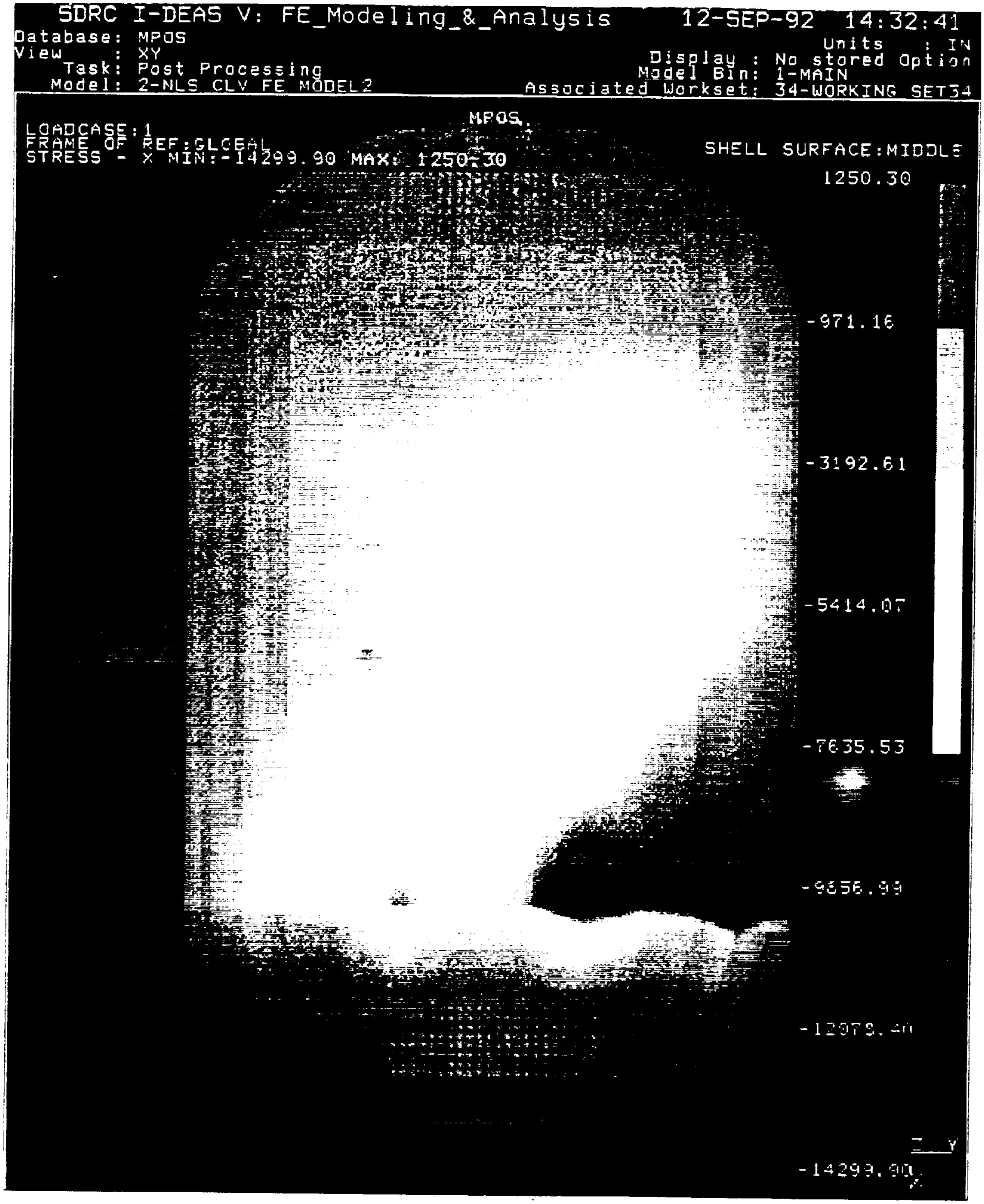

FIGURE 23. ON-PAD STRESSES IN LOX TANK
(SHELL MIDDLE SURFACE, LONGITUDINAL)

PREPARED BY: V. RICHARDSON	Rockwell International Space Systems Division	PAGE NO.
CHECKED BY:		REPORT NO.
DATE: SEPT.22, 1992	NLS / CLV	MODEL NO.
REF.		DWG. NO.

SEGMENT	FEM				HI Q		ON PAD		tank pres.	σ pres.	σ total max	σ total min	Nx ult max	Nx ult min
	skin	stringer		t bar	σ max	σ min	σ max	σ min						
	t	w	d											
	in.	in.	in.	in.	psi	psi	psi	psi	psi	psi	psi	psi	lbs/in	lbs/in
ADAPTER	0.007	2.70	1.02	0.034	9958.	-16350.	1350.	-4660.	0.	0.	9958.	-16350.	468.	-768.
FWD SKIRT	0.063	2.80	2.30	0.125	11891.	-18740.	2650.	-5500.	0.	0.	11891.	-18740.	2083.	-3283.
LOX TANK	0.100	2.20	1.80	0.138	4580.	-27832.	1250.	-14300.	29.	34624.	39204.	-14300.	7585.	-2767.
INTER TANK	0.150	2.30	2.80	0.212	-2129.	-22921.	-2713.	-14400.	0.	0.	-2129.	-22921.	-632.	-6807.
LH2 TANK	0.100	2.20	1.80	0.138	2284.	-30471.	8958.	-37600.	34.	40594.	49552.	-37600.	9587.	-7275.

SEGMENT	optimized t bar in	M.S. ult. tension	M.S. ult. comp.
ADAPTER			
FWD SKIRT			
LOX TANK	0.160	0.37	0.09
INTER TANK		n/a	
LH2 TANK	0.250	0.69	0.01

TABLE 4. STRESS ASSESSMENT

Mode	Freq (Hz)	Description
1	1.68	X-Y Bending
2	1.68	X-Z Bending
3	2.34	Shell Mode - LO_2 , LH
4	2.34	Local Mode - LO_2
5	2.39	Local Mode - LO_2

Table 5. Summary of CLV Modes

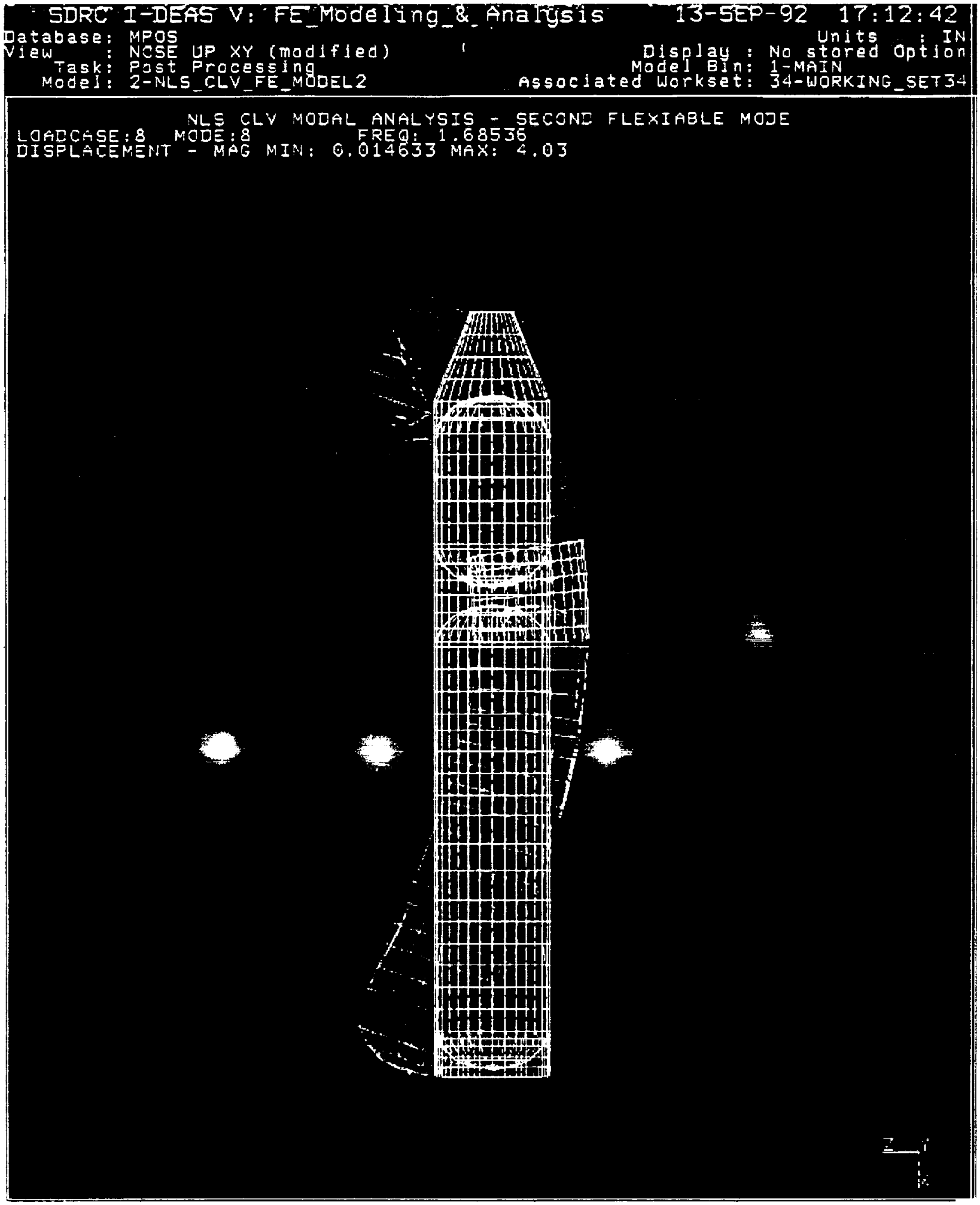

FIGURE 24. FIRST MODE (BODY BENDING)

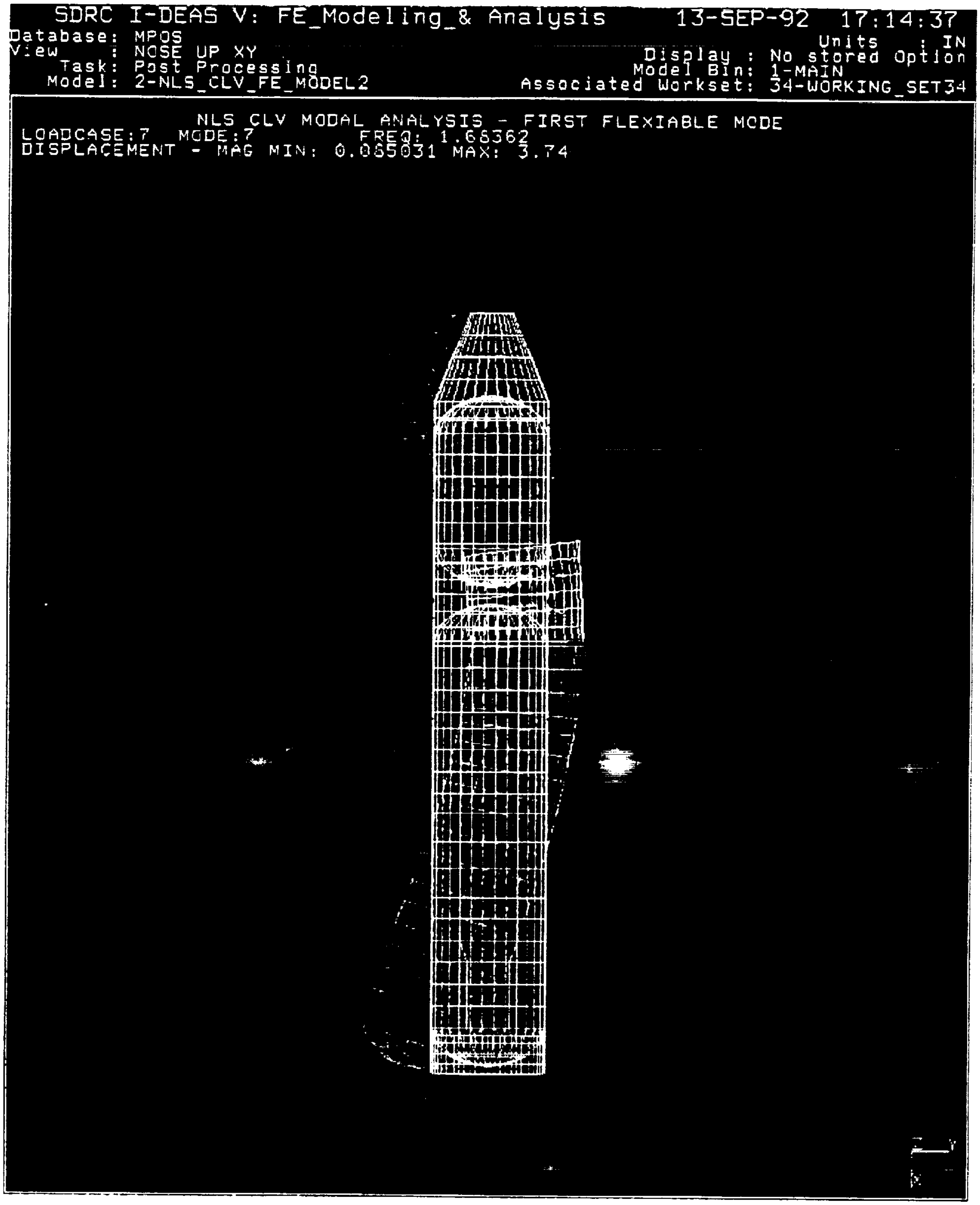

FIGURE 25. SECOND MODE (BODY BENDING)

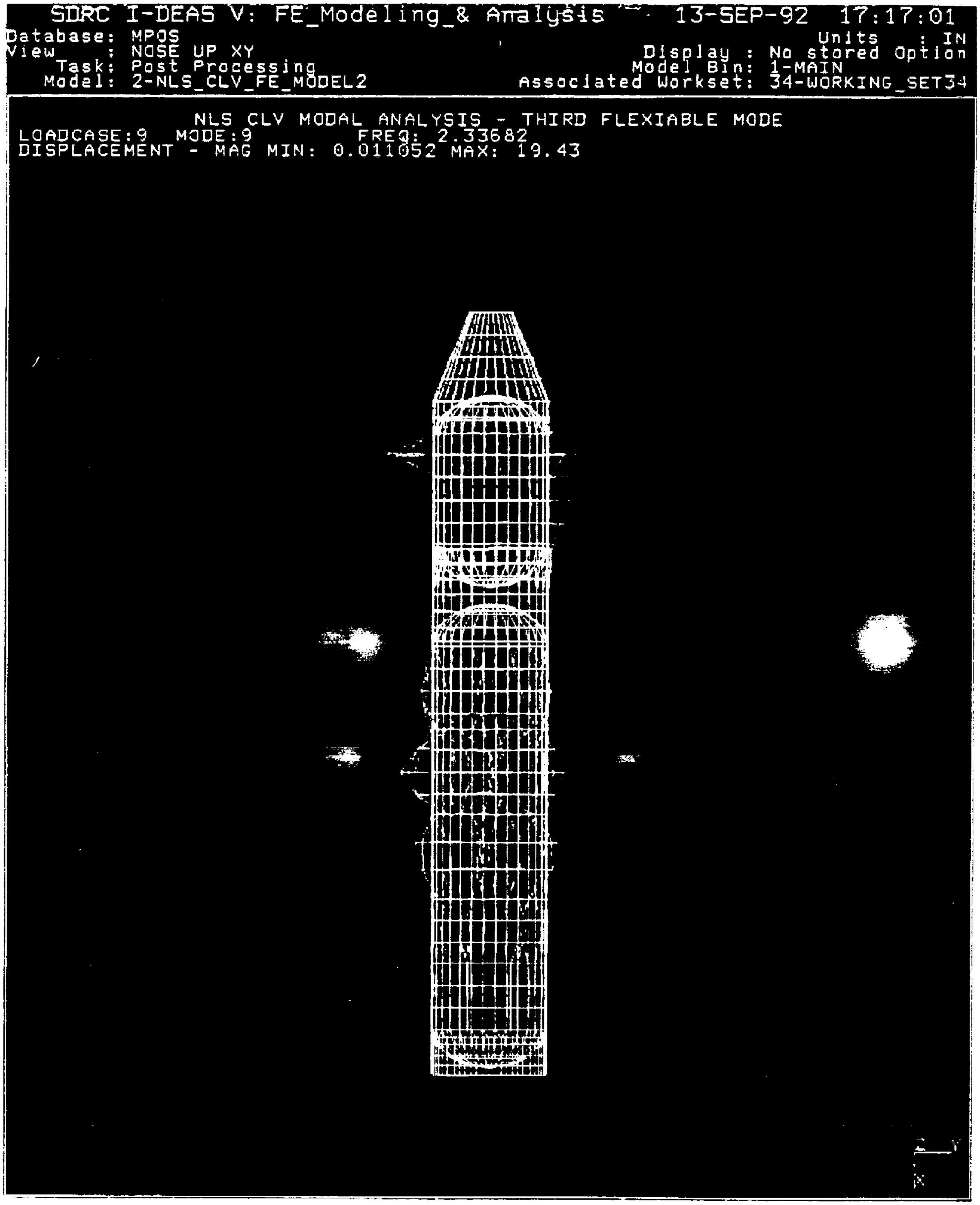

FIGURE 26. THIRD MODE (SHELL MODE, LO_2, LH_2)

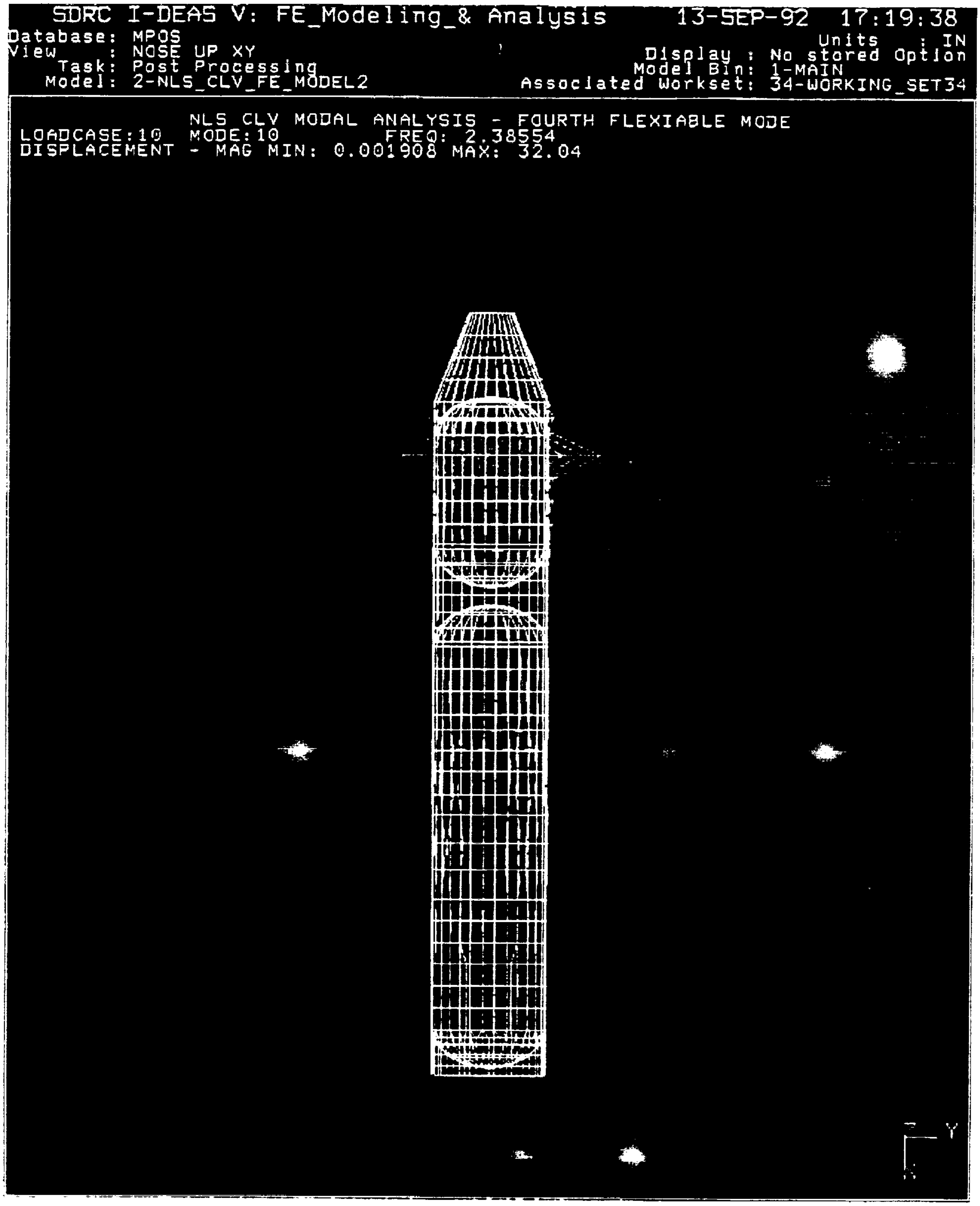

FIGURE 27. FOURTH MODE (LO$_2$ MODE)

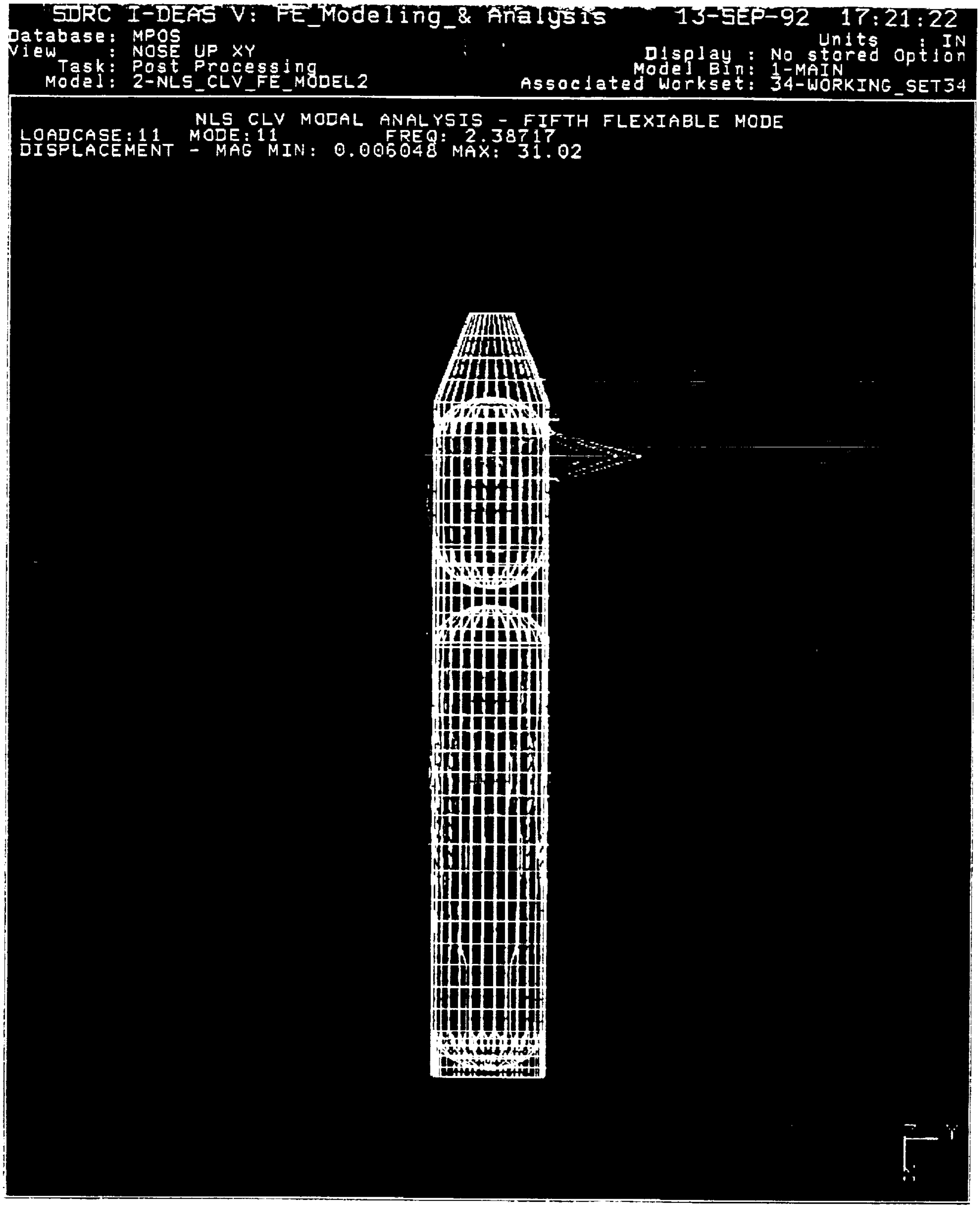

FIGURE 28. FIFTH MODE (LOCAL LO_2 MODE)

FIGURE 29. ON-PAD WIND DRAG LOAD SUMMATION COMPARISON

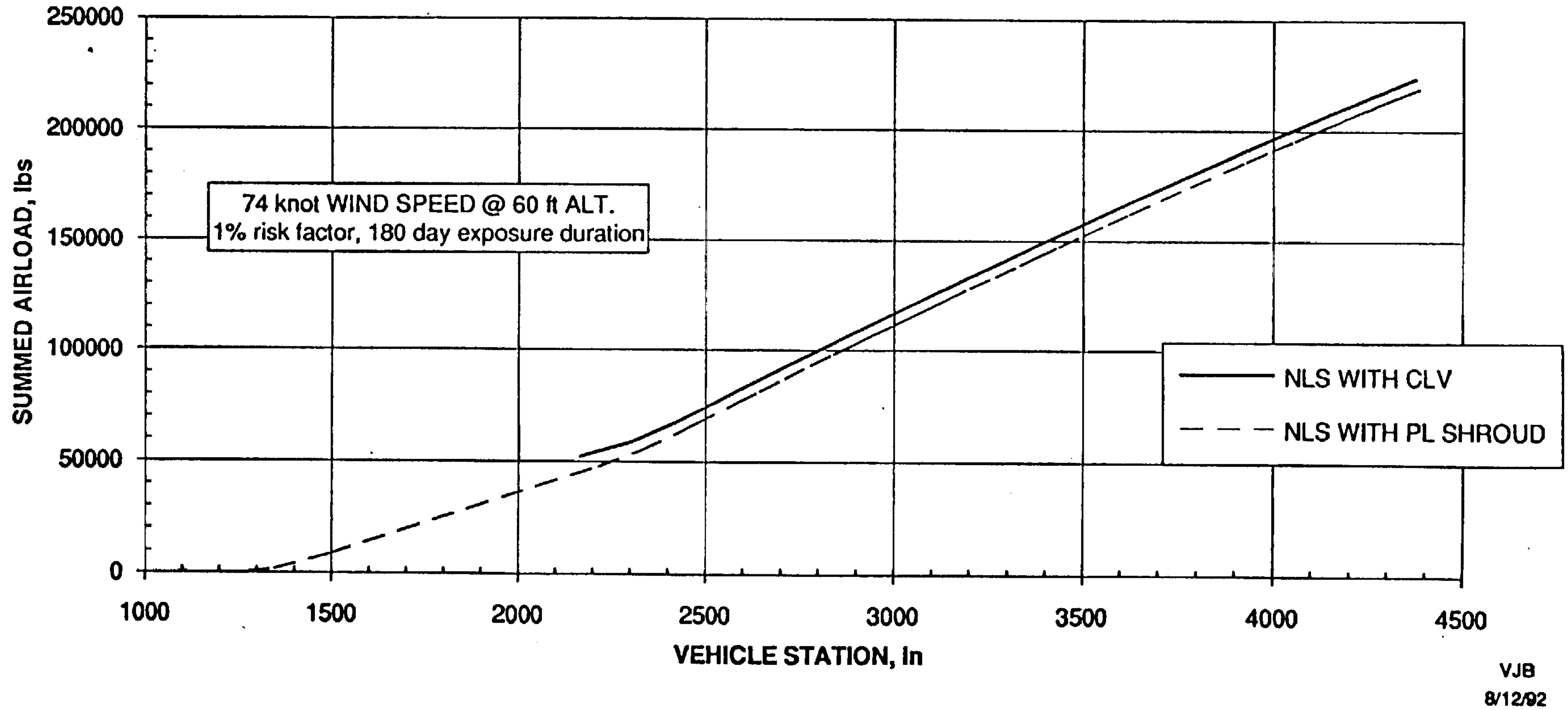

FIGURE 30. NLS ON-PAD BENDING MOMENT COMPARISON

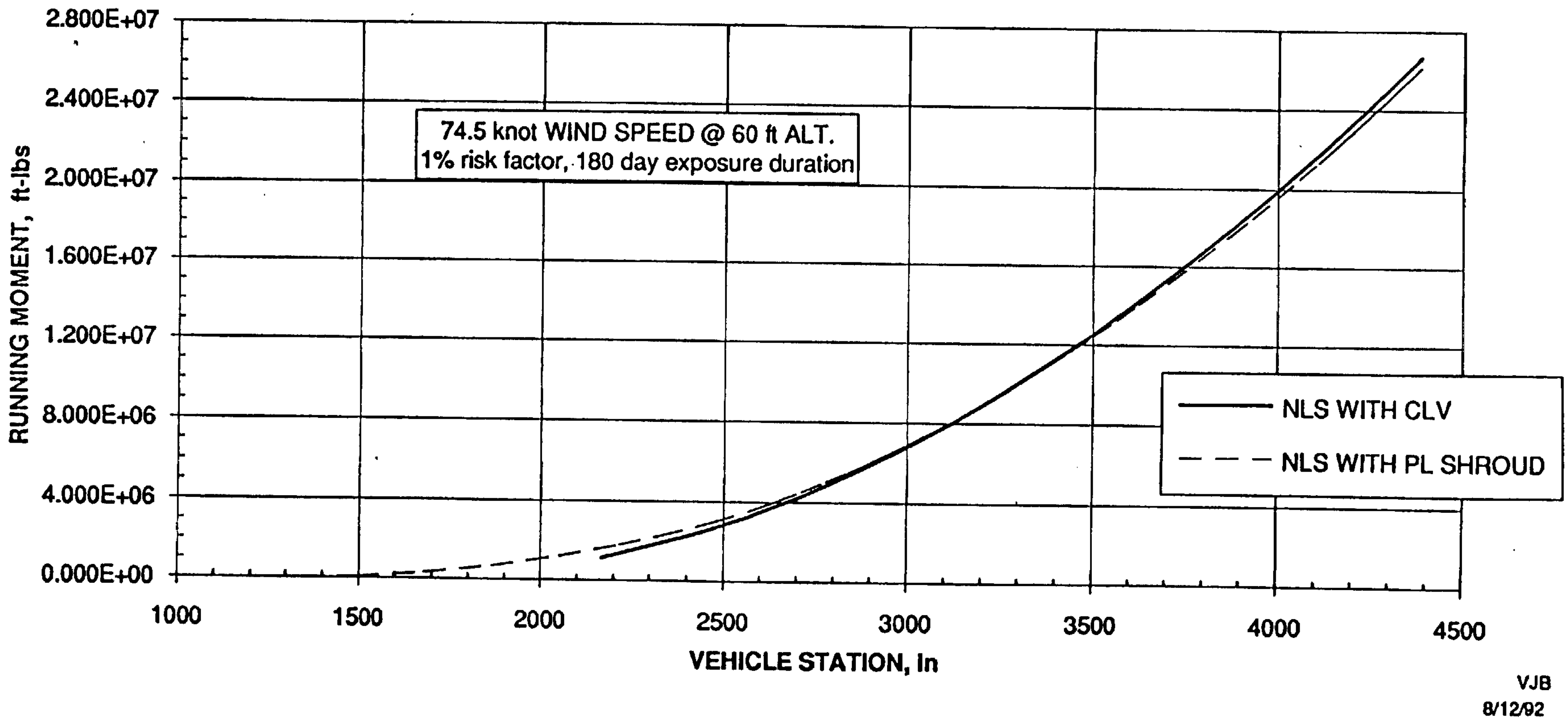

FIGURE 31. MAX q AIRLOAD SUMMATION COMPARISON

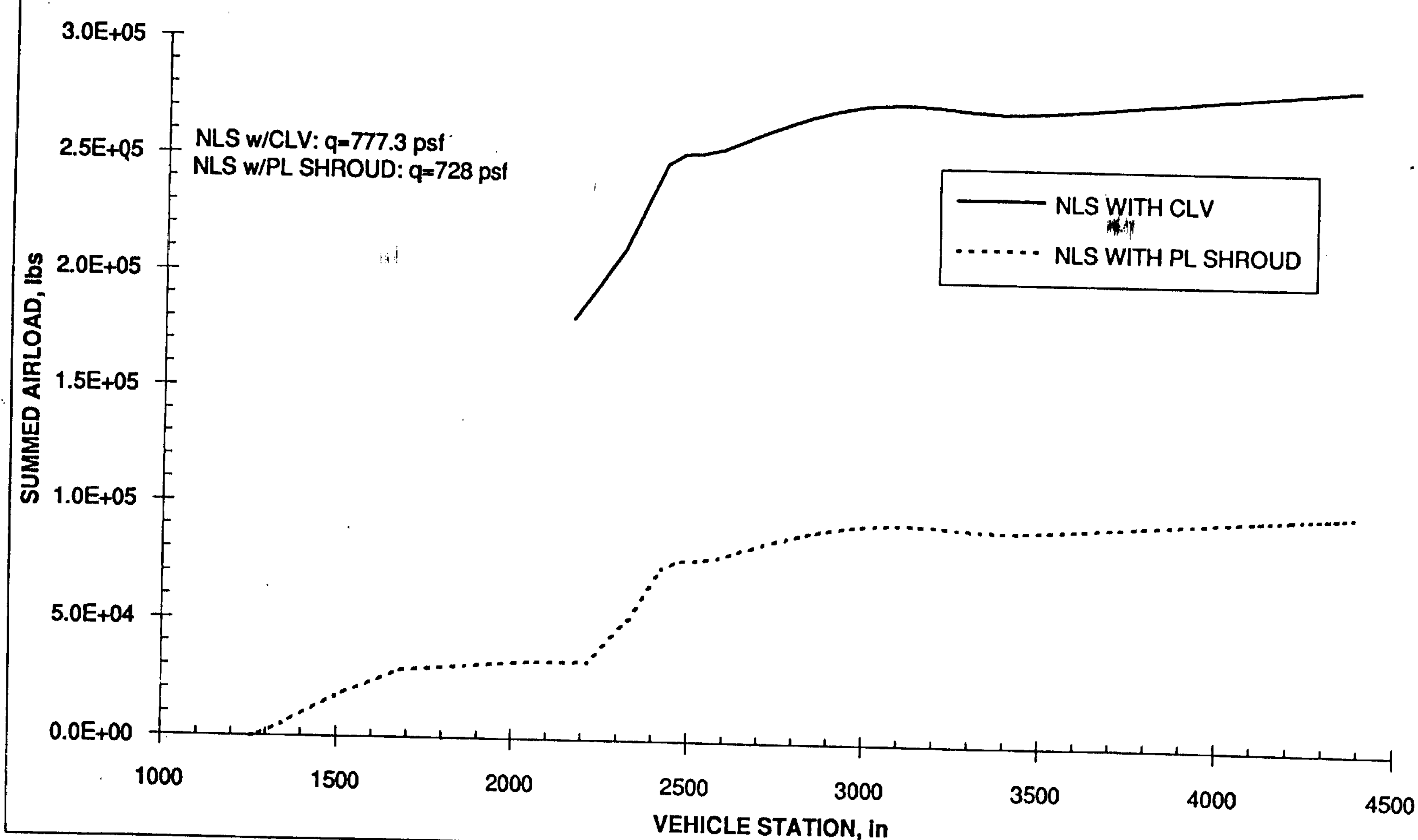

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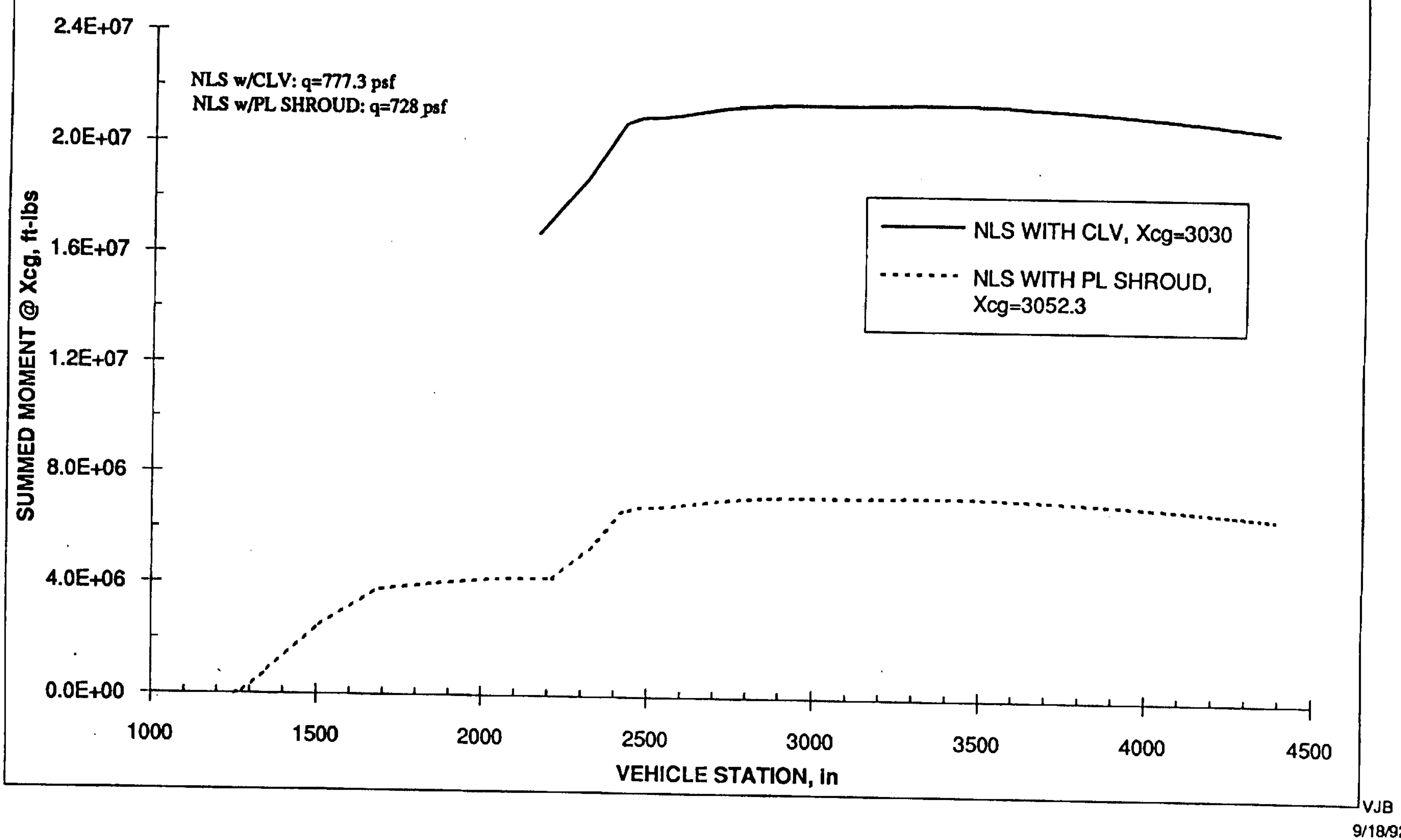
FIGURE 32. MAX q AIRLOAD MOMENT @ Xcg COMPARISON



Mode	CLV Freq (Hz)	ET Freq (Hz)	ET Mode Description
1	1.68	4.43	Body Bending
2	1.68	4.56	Body Bending
3	2.34	4.93	Shell Mode
4	2.34	5.02	Shell Mode
5	2.39	5.73	Shell Mode

Table 6. Dynamic Characteristics (ET vs CLV)

REFERENCES

1. MSFC Letter ED35-114-91, "NLS On-Pad Aerodynamic Data Base," September 12, 1991

2. Rockwell International IL SAS/NB/91-080, "Updated On-Pad Aerodynamic Analysis Tasks (3-FM-001) for the national Launch System (NLS) Configurations," September 5, 1991

3. MSFC Letter ES44-(89-90), "Statistical Parameters that Envelope the KSC Monthly Wind Ellipses for Shuttle-C," May 2, 1990

4. STS85-0118, "Operational Aerodynamic Data Book," Volume 3, Orbiter Vehicle Aerodynamic Data, Rockwell International, Space Division, Downey, California, September 1985

5. MSFC Letter ED35-88-91, "Distributed Loads on the National Launch System 1 1/2 Stage Launch Vehicle," September 18, 1991

ADVANCED TRANSPORTATION SYSTEM STUDY

Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO

Structural Analysis of a Manned Payload on the NLS

Personnel Launch System (PLS) with Cargo Return Vehicle (CRV)

October, 1992

Contract NAS8-39207

Prepared by: H. R. Grooms
Rockwell International

Rockwell International
Space Systems Division
Huntsville Operations

FORWARD

This report documents analyses conducted under Contract NAS8-39207, Advanced Transportation System Studies for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center. The report describes a preliminary analysis of aerodynamic loads, structural dynamics and stress, and weight estimation of the NLS-2 launch vehicle with a large manned payload (the Personnel Launch System concept as defined by the NASA Langley Research Center) combined with a Cargo Return Vehicle. This work was performed during the period of September and October, 1992 under the direction of Mr. Henry Grooms at Rockwell International, Space Systems Division, in Downey, CA. The primary technical analyses were performed by Vyto Baipsys (Aerodynamics), Keith Maeda (Weights), Susan Chen and Jack Barrett (Structural Dynamics), Van Richardson (Stress), and Ben Thompson and Bill Blanchard (Structural Design), all of Rockwell International.

A PRELIMINARY ASSESSMENT OF A CARGO RETURN VEHICLE (CRV)

INTRODUCTION

This report documents a preliminary design and analysis effort on a Cargo Return Vehicle (CRV). This work includes the computation of stresses, deflections, weights, aerodynamic distributions and dynamic characteristics for a preliminary CRV design.

I. Ground Rules and Assumptions

The CRV fits on top of a National Launch System (NLS) booster and below a Personnel Launch System (PLS) vehicle. This is shown in Figure 1. Figures 2 and 3 show some of the details of the CRV. Figure 4 shows the analytical flow for this work.

The main objective of this study was to produce a credible weight estimate for CRV. The structural evaluation was done to corroborate/revise the estimated structural sizing.

II. Aerodynamics

Results are presented in Figures 6 - 19 and Tables 1 - 5 to document the work performed in support of the CRV structural design. These results provided inputs to structural analysis for the launch stack elements depicted in Figure 5 and for the entering CRV capsule. The stack consists of a 10-foot tank stretch version of the 1.5 stage NLS vehicle (Config. "B") analyzed in Reference 1 atop of which ride the CRV and the PLS. The CRV is attached to the NLS booster and the PLS by adapter sections which transmit on-pad and ascent aero loads to the CRV. During entry the CRV is subjected to high dynamic pressure and deceleration (g's) loads.

Table 1 and Figures 6 - 9 show the spreadsheet results and graphs of the distributed ground wind loads for the on-pad wind condition. The wind speed used was from the NLS wind criteria (References 2 and 3) which specifies the maximum wind speed of 74.5 knots at a 60 foot height above ground level. The NLS base is 95 feet above ground. This is a 1 percent risk factor wind (99 percent probability of not exceeding it) for a 180 day exposure duration. The wind speed increases exponentially with height, as shown in the spreadsheet (Table 1). The airload distribution is depicted in Figure 6 by a normal force distribution beginning with PLS adapter just aft of the PLS and continuing over the NLS booster. The resulting net pressure distribution, which is used as input to the finite element structural analysis, is shown in Figure 7 over the same sections of the stack. The PLS-produced airload is treated as a point load in its affect on the adapters, CRV, and the NLS structure. Comparison plots are shown in Figure 8 between the NLS/CRV/PLS and the basic NLS with the standard payload shroud for the summed running air load and in Figure 9 for the moment along the NLS booster. Due to the lower projected area of the CRV/PLS relative to the payload shroud, the air load and moment are lower than for the basic NLS vehicle.

Table 2 and Figures 10 - 14 depict similar type of results for the condition of maximum product of dynamic pressure times alpha (q-alpha$_{max}$) to define the highest bending moment in flight. The dynamic pressure (q) variation with flight altitude, along with other trajectory parameters was obtained from the RI Huntsville CRV/NLS trajectory. A conservative angle of attack (a) was computed by superimposing an in-flight wind (Reference 4) and a 33 ft/sec gust at each altitude to find the maximum product of q-alpha.

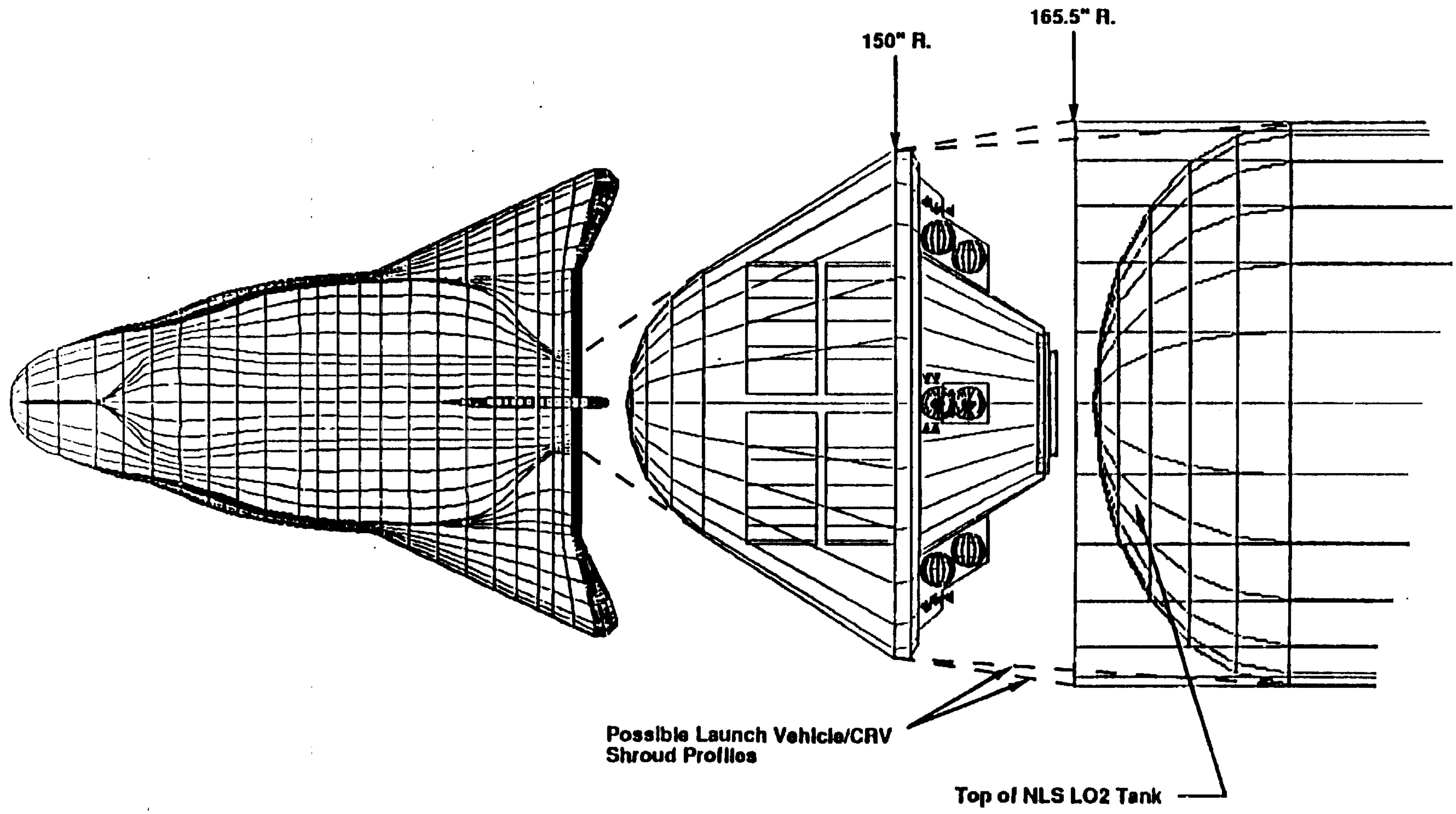


FIGURE 1. CARGO RETURN VEHICLE CONCEPT

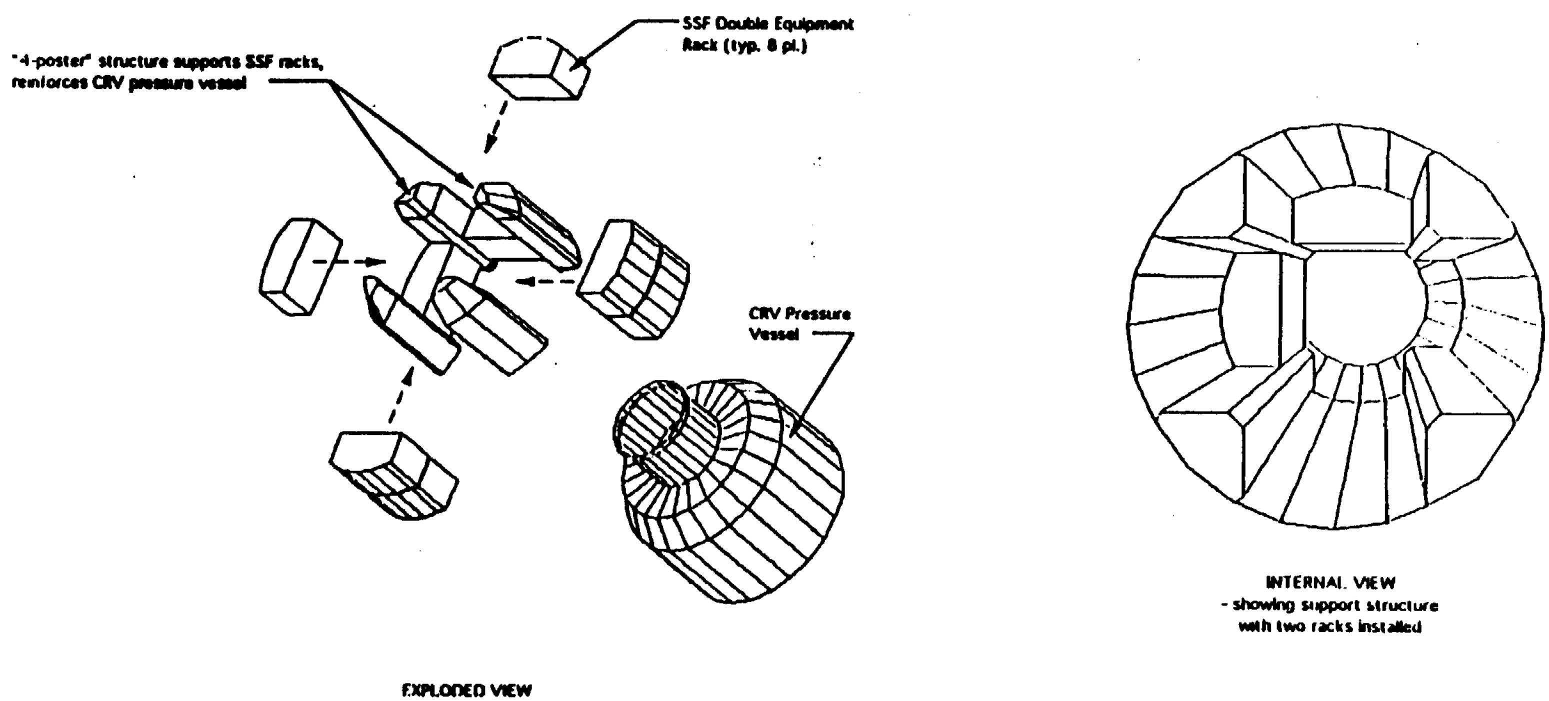


FIGURE 2. CRV INTERIOR ARRANGEMENT

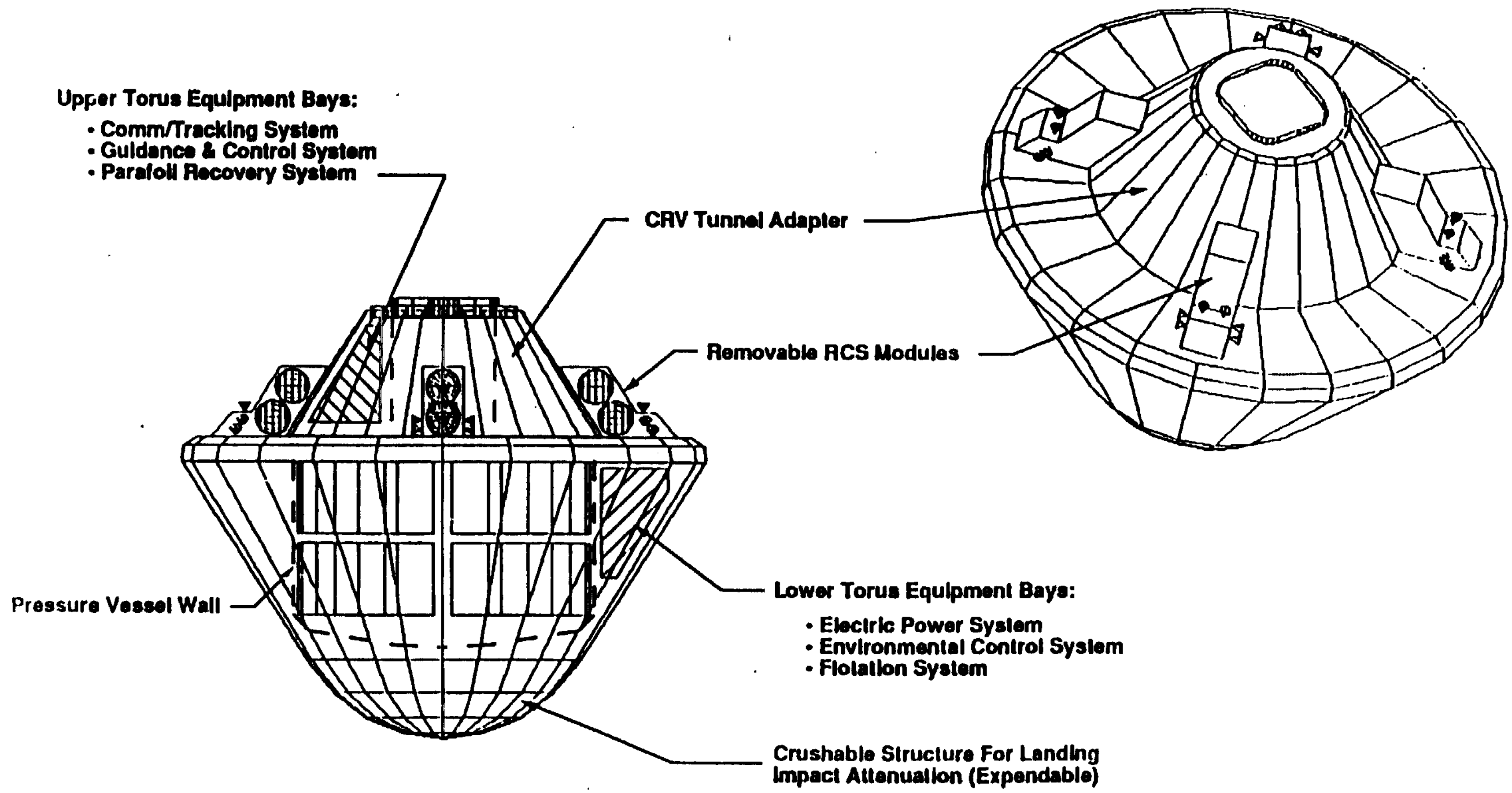


FIGURE 3. MAIN FEATURES OF THE CRV

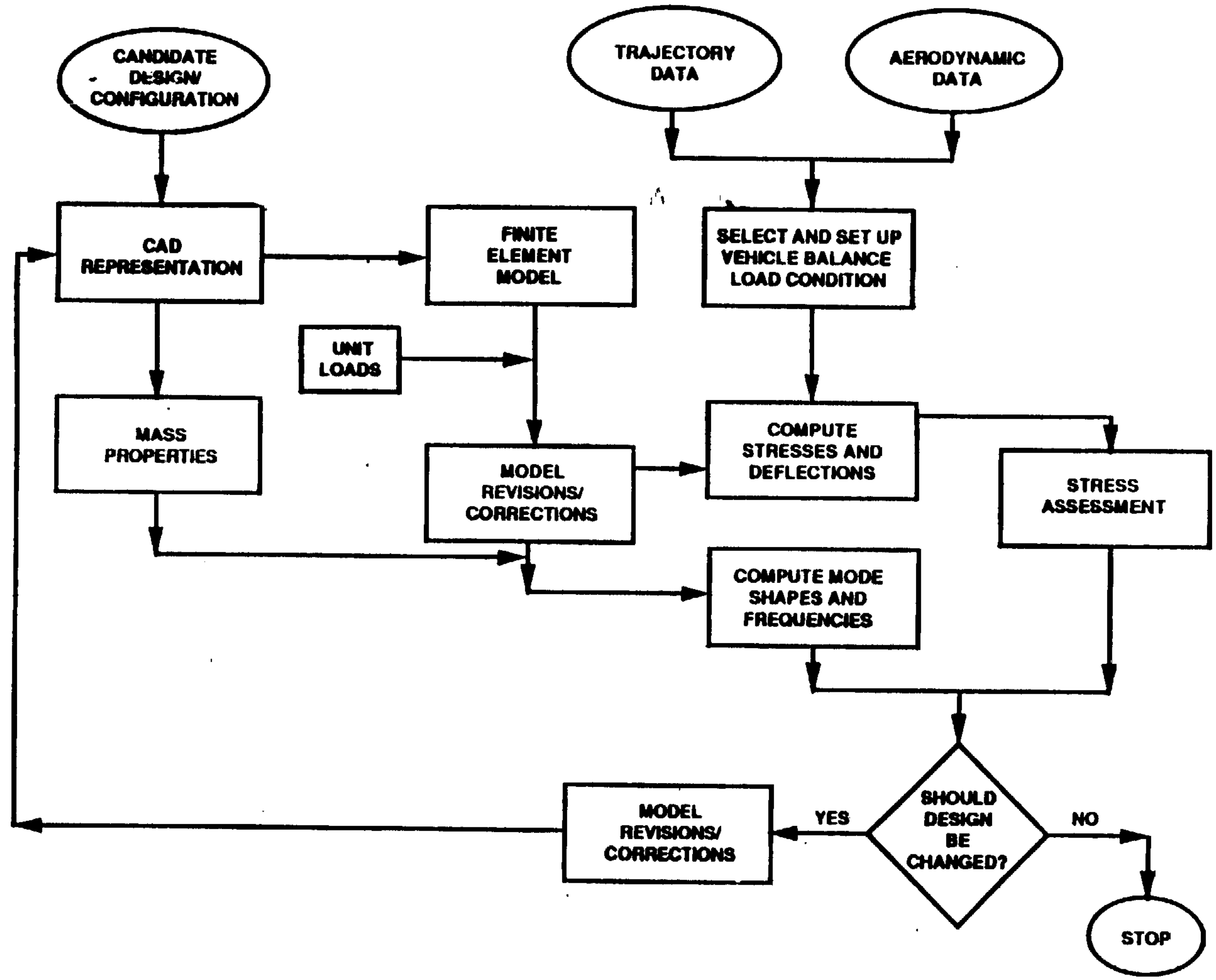


FIGURE 4. THE ANALYTICAL PROCESS

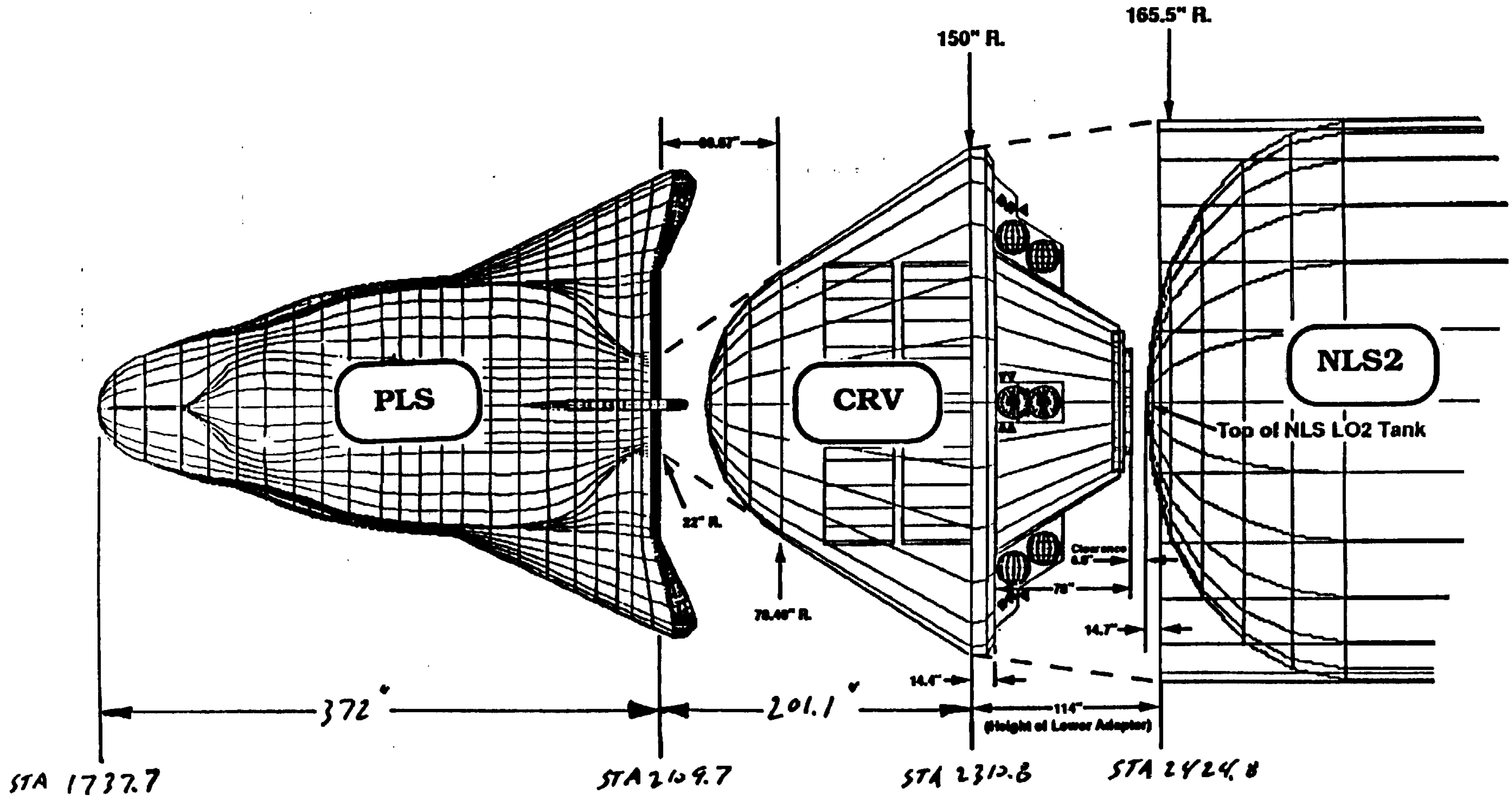


FIGURE 5. CLV ON NLS 2

TABLE 1. NLS/PLS/CRV GROUND WIND LOADING

10' STRETCH NLV2 WITH INLINE PLS/CRV

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4-Sep-92

NLS STRUCTURE BASE AT 95 ft ELEVATION

						ON-PAD AERO LOADS LOADS INCLUDE PLS/CRV 1% RISK FACTOR, 180 WINDIEST DAY EXPOSURE DURATION Vwind=74.5 KTS @ 60 ft Ref. Altitude			
X STA. No in	Component	LOCAL DIAMETER ft	Height ft	Cd Drag Coeff (based on side area)	Vwind ft/sec	DRAG *1.5 (vortex shed) lbs/in	SUMMED LOAD VERSUS X lbs w pls/crv	SUMMED MOMENT ABOUT X ft-lbs w pls/crv	LOCAL NET PRESSURE DISTRIB. psi normal to surface
2109.70		0.00	284.65	1.000	147.83	0.000	13398.3	1.603E+05	0.0000
2109.70	32.5 deg	3.67	284.65	1.000	147.83	11.804	13398.3	1.603E+05	0.3181
2153.70	fairing	8.33	280.98	1.000	147.63	26.760	14246.7	2.111E+05	0.3173
2175.70	"	10.67	279.15	1.000	147.53	34.209	14917.3	2.404E+05	0.3168
2198.37	<fwd bulkhead	13.07	277.28	1.000	147.42	41.864	15779.8	2.710E+05	0.3164
2219.70	"	15.34	275.48	1.000	147.33	49.047	16749.2	3.018E+05	0.3160
2241.70	"	17.67	273.65	1.000	147.22	56.436	17909.5	3.356E+05	0.3156
2263.70	"	20.00	271.82	1.000	147.12	63.804	19232.1	3.696E+05	0.3152
2285.70	"	22.34	269.98	1.000	147.02	71.152	20716.8	4.062E+05	0.3147
2310.80	<aft bulkhead	25.00	267.89	1.000	146.90	79.510	22607.8	4.515E+05	0.3143
2310.80	7.74 deg	25.00	267.89	1.000	146.90	79.511	22607.6	4.515E+05	0.2675
2340.80	fairing	25.68	265.39	1.000	146.76	81.521	25022.9	5.515E+05	0.2670
2370.80	"	26.36	262.89	1.000	146.61	83.521	27498.5	6.586E+05	0.2665
2424.80	"	27.58	258.39	1.000	146.35	87.097	32105.2	8.699E+05	0.2656
2424.80	1st Stage	27.58	258.39	1.000	146.35	87.097	32105.2	8.699E+05	0.2631
2472.80	"	27.58	254.39	1.000	146.11	86.825	36279.3	1.077E+06	0.2623
2569.80	"	27.58	246.31	1.000	145.62	86.265	44674.2	1.545E+06	0.2606
2583.55	"	27.58	245.16	1.000	145.55	86.185	45859.8	1.616E+06	0.2604
2644.06	"	27.58	240.12	1.000	145.24	85.826	51064.0	1.948E+06	0.2593
2711.77	"	27.58	234.48	1.000	144.88	85.418	56861.5	2.350E+06	0.2581
2778.89	"	27.58	228.88	1.000	144.52	85.005	62580.9	2.779E+06	0.2568
2838.41	"	27.58	223.92	1.000	144.19	84.631	67629.2	3.187E+06	0.2557
2852.80	"	27.58	222.73	1.000	144.11	84.540	68846.4	3.289E+06	0.2554
2897.10	"	27.58	219.03	1.000	143.86	84.256	72585.3	3.613E+06	0.2545
2941.40	"	27.58	215.34	1.000	143.61	83.968	76311.4	3.950E+06	0.2537
2985.67	"	27.58	211.65	1.000	143.35	83.676	80022.2	4.301E+06	0.2528
3034.20	"	27.58	207.61	1.000	143.06	83.352	84075.2	4.700E+06	0.2518
3083.30	"	27.58	203.52	1.000	142.77	83.018	88159.6	5.121E+06	0.2508
3123.15	"	27.58	200.20	1.000	142.52	82.743	91462.3	5.475E+06	0.2500
3137.54	"	27.58	199.00	1.000	142.43	82.643	92652.3	5.605E+06	0.2497
3201.70	"	27.58	193.65	1.000	142.03	82.189	97940.1	6.203E+06	0.2483
3266.50	"	27.58	188.25	1.000	141.61	81.721	103250.8	6.836E+06	0.2469
3331.30	"	27.58	182.85	1.000	141.19	81.241	108530.8	7.496E+06	0.2454
3377.35	"	27.58	179.01	1.000	140.88	80.893	112263.9	7.982E+06	0.2444
3435.90	"	27.58	174.13	1.000	140.47	80.442	116987.0	8.621E+06	0.2430
3500.70	"	27.58	168.73	1.000	140.01	79.930	122183.1	9.353E+06	0.2415
3565.70	"	27.58	163.32	1.000	139.54	79.403	127361.4	1.012E+07	0.2399
3623.80	"	27.58	158.48	1.000	139.10	78.919	131960.7	1.082E+07	0.2384
3706.10	"	27.58	151.62	1.000	138.47	78.213	138426.7	1.180E+07	0.2363

TABLE 1. NLS/PLS/CRV GROUND WIND LOADING (CONT.) 10' STRETCH NLV2 WITH INLINE PLS/CRV

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4-Sep-92

NLS STRUCTURE BASE AT 95 ft ELEVATION

						ON-PAD AERO LOADS LOADS INCLUDE PLS/CRV 1% RISK FACTOR, 180 WINDIEST DAY EXPOSURE DURATION Vwind=74.5 KTS @ 60 ft Ref. Altitude			
X STA. No in	Component	LOCAL DIAMETER ft	Height ft	Cd Drag Coeff (based on side area)	Vwind ft/sec	DRAG *1.5 (vortex shed) lbs/in	SUMMED LOAD VERSUS X lbs w pls/crv	SUMMED MOMENT ABOUT X ft-lbs w pls/crv	LOCAL NET PRESSURE DISTRIB. psi normal to surface
3784.90	•	27.58	145.05	1.000	137.83	77.512	144562.3	1.289E+07	0.2342
3871.00	•	27.58	137.88	1.000	137.11	76.716	151201.8	1.406E+07	0.2318
3932.00	•	27.58	132.79	1.000	136.57	76.131	155863.7	1.492E+07	0.2300
3996.80	•	27.58	127.39	1.000	135.98	75.490	160776.2	1.586E+07	0.2281
4058.00	•	27.58	122.29	1.000	135.41	74.863	165376.9	1.677E+07	0.2262
4108.91	•	27.58	118.05	1.000	134.91	74.325	169174.5	1.754E+07	0.2245
4122.65	Thrust struct.	27.58	116.90	1.000	134.78	74.177	170194.7	1.775E+07	0.2241
4137.30	•	27.58	115.68	1.000	134.63	74.018	171280.3	1.798E+07	0.2236
4151.90	•	27.58	114.47	1.000	134.48	73.859	172359.8	1.820E+07	0.2231
4166.60	•	27.58	113.24	1.000	134.33	73.696	173444.3	1.843E+07	0.2226
4181.20	•	27.58	112.03	1.000	134.18	73.534	174519.1	1.866E+07	0.2222
4195.70	•	27.58	110.82	1.000	134.03	73.371	175584.1	1.889E+07	0.2217
4210.30	•	27.58	109.60	1.000	133.88	73.206	176654.2	1.913E+07	0.2212
4227.40	•	27.58	108.18	1.000	133.69	73.010	177904.3	1.940E+07	0.2206
4241.60	•	27.58	106.99	1.000	133.54	72.846	178939.9	1.963E+07	0.2201
4254.10	•	27.58	105.95	1.000	133.41	72.700	179849.5	1.983E+07	0.2196
4265.00	•	27.58	105.04	1.000	133.29	72.572	180641.3	2.001E+07	0.2193
4275.90	•	27.58	104.13	1.000	133.17	72.443	181431.6	2.018E+07	0.2189
4286.90	•	27.58	103.22	1.000	133.05	72.312	182227.8	2.036E+07	0.2185
4297.80	•	27.58	102.31	1.000	132.92	72.182	183015.3	2.054E+07	0.2181
4308.80	•	27.58	101.39	1.000	132.80	72.049	183808.5	2.072E+07	0.2177
4319.70	•	27.58	100.48	1.000	132.67	71.916	184593.1	2.091E+07	0.2173
4330.60	•	27.58	99.58	1.000	132.55	71.783	185376.3	2.109E+07	0.2169
4341.60	•	27.58	98.66	1.000	132.42	71.647	186165.2	2.127E+07	0.2165
4352.20	•	27.58	97.78	1.000	132.30	71.515	186923.9	2.145E+07	0.2161
4362.80	•	27.58	96.89	1.000	132.17	71.382	187681.3	2.163E+07	0.2157
4374.20	•	27.58	95.94	1.000	132.04	71.238	188494.2	2.182E+07	0.2152
4385.50	End of struct.	27.58	95.00	1.000	131.90	71.094	189298.4	2.201E+07	0.2148

FIGURE 6.
NLS/PLS/CRV ON-PAD DRAG LOAD DISTRIBUTION

10' STRETCH NLS2 WITH INLINE PLS/CRV

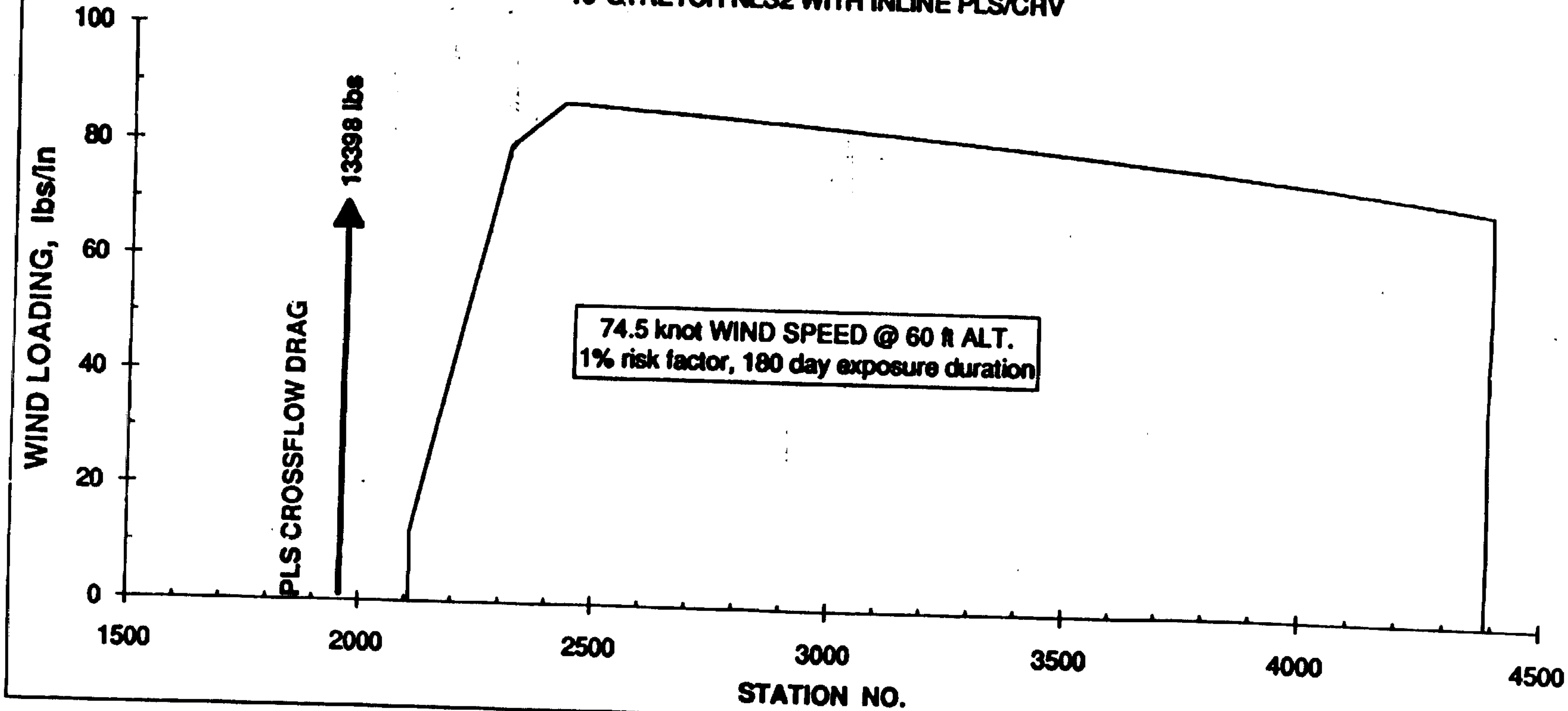

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FIGURE 7.
NLS/PLS/CRV ON-PAD SURFACE NET PRESSURE DUE TO WIND LOAD

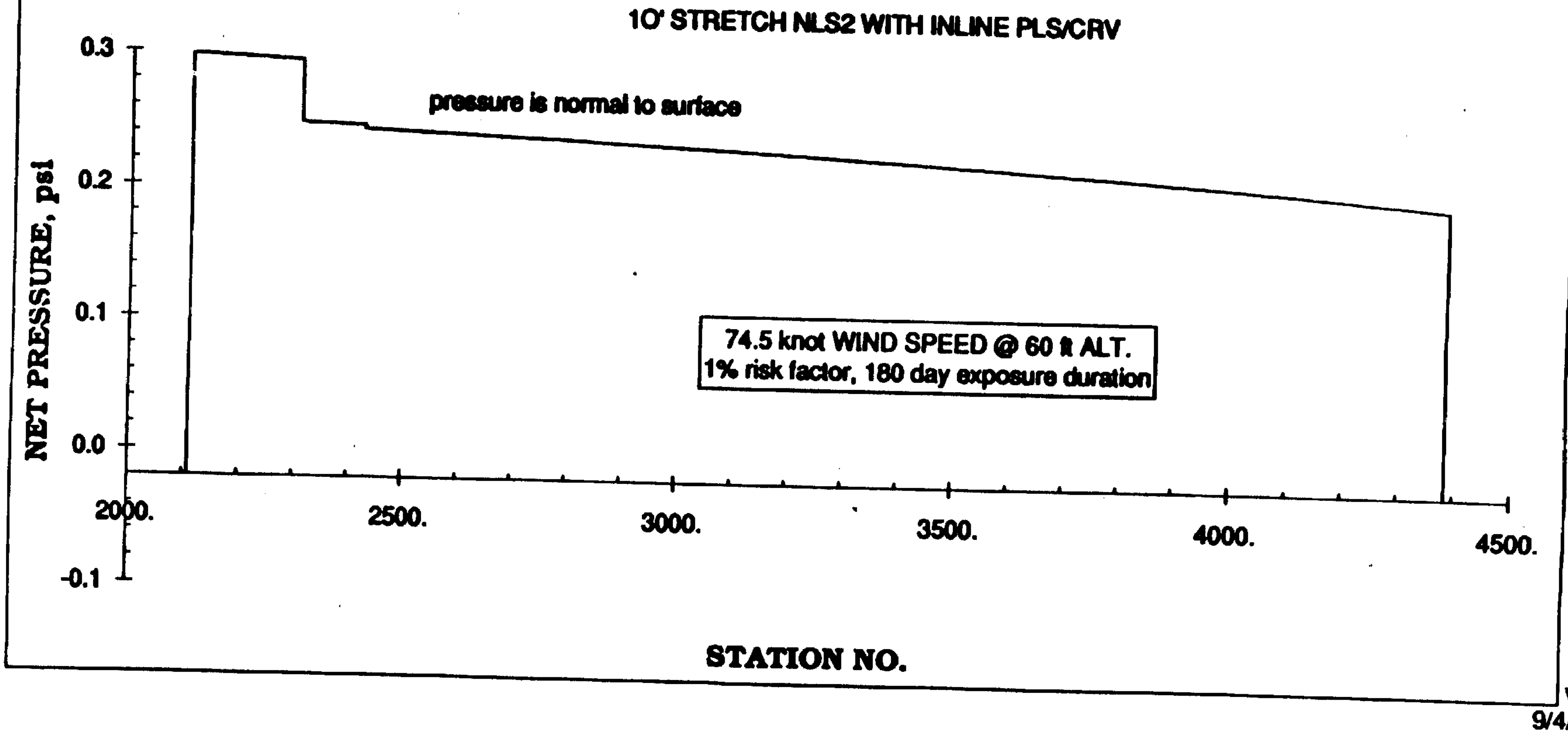

FIGURE 8-
NLS/PLS/CRV ON-PAD WIND DRAG LOADIS LOWER THAN FOR NLS WITH PL SHROUD

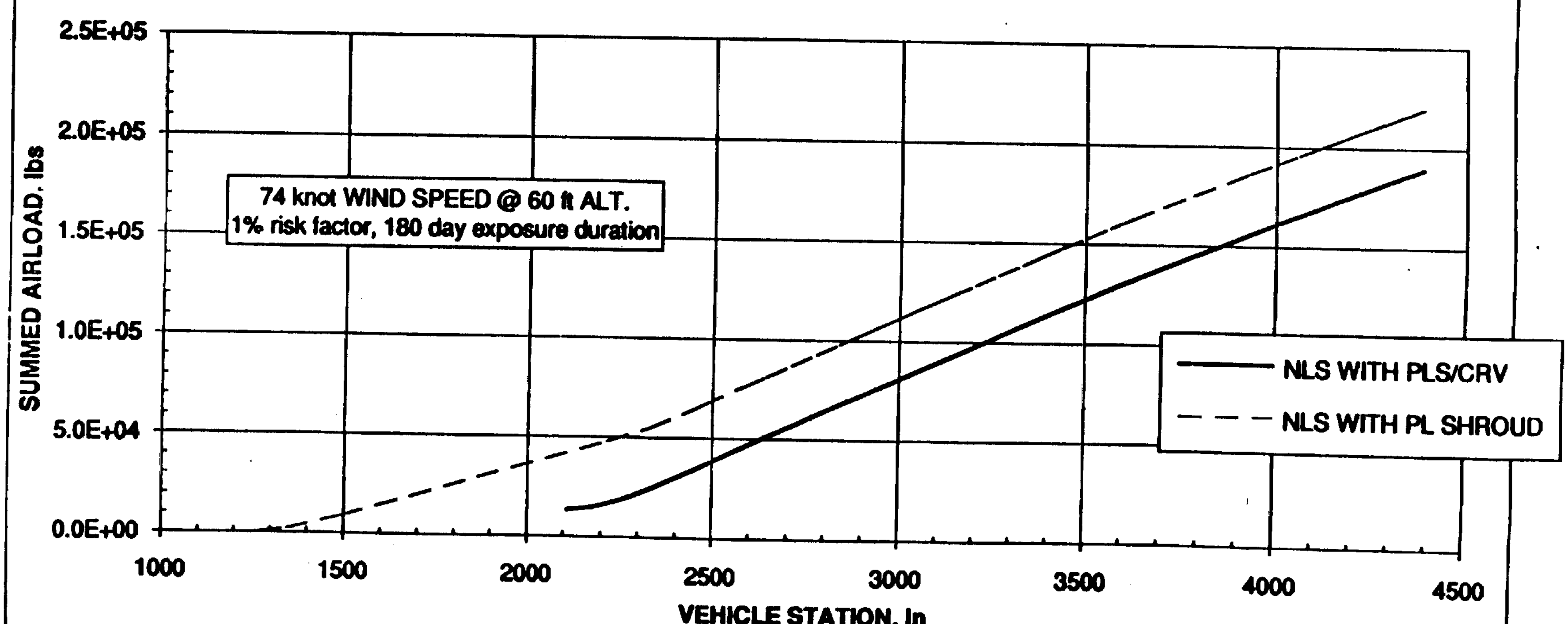

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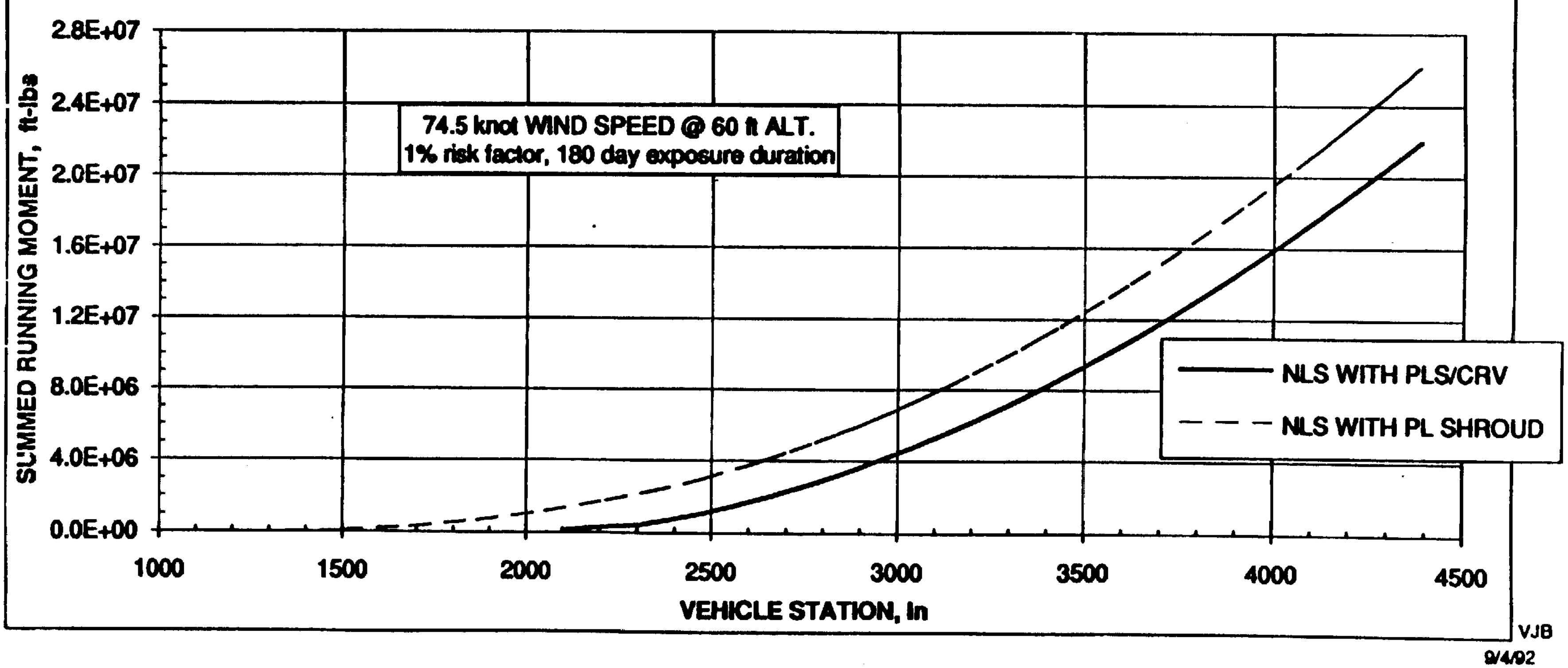
FIGURE 9.
NLS/PLS/CRV ON-PAD BENDING MOMENT IS LOWER THAN FOR NLS WITH PL SHROUD



The maximum product, q-alpha$_{max}$, of 3907 psf-deg. was calculated for a vehicle speed of 1353.46 ft/sec, q of 705.1 psf, and a corresponding wind speed of 98.3 ft/sec. At this condition a was computed as follows:

$$\alpha = \mathrm{Tan}^{-1}\left(\frac{V_{wind} + V_{gust}}{V_{vehicle}}\right) = \mathrm{Tan}^{-1}\left(\frac{98.3 + 33}{1353.46}\right) = 5.5 \text{ deg}$$

The airload distribution methods of Reference 5 were used to modify the normal force distribution over the PLS/CRV adapter sections from earlier-defined (Reference 6) NLS normal force distribution. The summed moments in this analysis were computed about the vehicle Xcg at Station 3051 (adjusted from earlier NLS mass properties). Tabulated and plotted normal force distribution, summed loads and moments, and net pressures are shown in the spreadsheet of Table 2 and Figures 10 - 13 with stations beginning at the PLS adapter (just aft of the PLS) and continuing over the NLS booster. Static balance calculations were included in the spreadsheet of Table 2 to determine the amount of engine gimbal angle required in order to overcome the aerodynamic moment induced by the airload. This was computed from the moment balance require between the aerodynamic moment and the engines, as shown below.

$$T\mathrm{Sin}(\delta)(X_{gimbal} - X_{cg}) = C_{N\alpha} q \alpha S_{ref}(X_{cg} - X_{cp})$$

Assuming small angles, Sin(δ) can be represented by δ, in radians

$$\text{then: } \frac{\delta}{\alpha} = \frac{C_{N_\alpha} q S_{ref}\ (X_{cg} - X_{cp})}{T(X_{gimbal} - X_{cg})} \quad \text{radians/degree}$$

where:

δ	=	engine gimbal angle, radians
α	=	angle of attack, degrees
$C_{N\alpha}$	=	normal force coefficient slope, per degree
q	=	dynamic pressure lbs/sq ft
S_{ref}	=	reference area, sq ft
T	=	engine thrust, lbs
X_{cg}	=	center of gravity station, in
X_{cp}	=	center of pressure station, in
X_{gimbal}	=	engine gimbal station, in

This equation was then solved for the appropriate α from the trajectory to solve for the gimbal angle, δ. A gimbal angle of 1.85 degrees was required to provide static balance against the aerodynamic bending moment. Figures 15 and 16 show the δ sensitivity with X_{cg} and the product of q-alpha variation. The sensitivity is seen to me greater with the product of q-alpha.

With the gimbal angle defined, the axial and tangential thrust values were calculated. These are shown in the boxed area at the end of the spreadsheet of Table 2. These thrust components were then used to compute the axial acceleration and the tangential acceleration.

$$\text{Axial Acceleration} = \frac{\text{Axial Thrust}}{\text{Vehicle Weight}}$$

$$\text{Tangential Acceleration} = \frac{\text{Tangential Thrust} + \Sigma(\text{Airload})}{\text{Vehicle Weight}}$$

The results of this calculation are shown at the bottom of Table 2 where δ and the accelerations are shown in the boxed area at the bottom of the table. This added gimbal angle is used to evaluate the configurations for gimbal control feasibility. It is also used to enhancee the finite element model (FEM) structural analyses by specifying the magnitudes of the inertia acceleration and correct thrust component inputs to the FEM.

Normal force distribution and resulting net pressure distribution along the vehicle are shown in Figures 10 and 11, respectively. For summed airloads and moments, the PLS-produced airload is treated as a point load in its affect on the adapters, CRV, and the NLS structure. Comparisons are shown for the total normal forces and moments (about Xcg) in Figures 12 and 13 between (1): the NLS/CRV/PLS configuration with current trajectory data (q and alpha) from RI Huntsville, (2): the NLS with payload shroud for the same q and alpha as in (1), and (3): the NLS with payload shroud for the q and alpha from the NLS design trajectory. These comparisons show that both the CRV/PLS geometry and the trajectory have a profound effect on the magnitude of the aero loads.

Plotted and tabulated summed moments are also shown (Figure 14 and Table 3) in the PLS/CRV/NLS adapter region as running moment versus X-station and taken about each X-station. This was done to allow better evaluation of local loads in the adapter and CRV area of the stack.

Figures 17 - 18 and Table 4 present the entry net pressure distribution over the CRV capsule forebody during a ballistic trajectory (assumes $\alpha = 0$ degrees). These pressures were computed using the Modified Newtonian pressure method and are shown for the entry conditions obtained from trajectory runs at RI Huntsville. Base region (all the aft-facing portions shielded from the airstream) pressure values are also specified. In addition, the entry drag coefficient variation with Mach was also estimated to assist in trajectory analysis and is included in Figures 19 and Table 5.

III. Finite Element Model

A finite element model of the CRV (capsule) has been created (Figure 20) and mounted on top of the NLS booster (Figures 21 and 22). The complete model has the following characteristics:

1. Nodes ≈ 2100
2. Degrees of Freedom ≈ 12, 600
3. Elements ≈ 5,100

The model does not include a stiffness representation of the PLS but its mass is incorporated.

TABLE 2.

NLS/PLS/CRV MAX q alpha AIRLOAD & MOMENT @ Xcg

10' STRETCH NLV2 WITH INLINE PLS/CRV

VJB

9/2/92

Sref = 593.96 ft^2 MACH = 1.3

Xcg(sta) = 3051 q (psf) = 705.1

α = 5.5

CN pls = 0.1378

Xcp pls = 1966

MAX q alpha AIRLOADS
SUMMED LOADS INCLUDE PLS/CRV

NLS/PLS/CRV TRAJECTORY
MACH=1.3, α =5.5 deg

X STA. No in	Component	LOCAL DIAMETER ft	CNα per Radian per X/D	NORMAL AIR LOAD DISTRIBUTED lbs/in	SUMMED NORMAL F. FWD OF X lbs w/ pls/crv	SUMMED MOMENT ABOUT Xcg ft-lbs w/ pls/crv	LOCAL NET PRESSURE DISTRIB. psi normal to surface
2109.70	32.5 deg	3.67	0.48575	59.00	5.771E+04	5.218E+06	1.5898
2153.70	fairing	8.33	1.47957	179.70	6.296E+04	5.618E+06	2.1305
2175.70	"	10.87	2.00025	242.94	6.761E+04	5.961E+06	2.2502
2198.37	<fwd bulkhead	13.07	2.53443	307.82	7.385E+04	6.411E+06	2.3268
2219.70	"	15.34	3.02903	367.89	8.106E+04	6.916E+06	2.3704
2241.70	"	17.87	3.53158	428.93	8.983E+04	7.515E+06	2.3986
2263.70	"	20.00	3.99804	485.59	9.989E+04	8.184E+06	2.3986
2285.70	"	22.34	4.46449	542.24	1.112E+05	8.915E+06	2.3986
2310.80	<aft bulkhead	25.00	4.99672	606.88	1.256E+05	9.820E+06	2.3986
2310.80	7.74 deg	25.00	1.32707	161.18	1.256E+05	9.820E+06	0.5422
2340.80	fairing	25.68	1.26833	154.05	1.303E+05	1.011E+07	0.5045
2370.80	"	26.36	1.26541	153.69	1.350E+05	1.037E+07	0.4903
2424.80	"	27.58	1.17137	142.27	1.429E+05	1.081E+07	0.4338
2424.80	1st Stage	27.58	-0.05000	-6.07	1.429E+05	1.081E+07	-0.0183
2472.80	"	27.58	0.00000	0.00	1.428E+05	1.080E+07	0.0000
2520.80	"	27.58	0.08000	9.72	1.430E+05	1.081E+07	0.0294
2583.55	"	27.58	0.41000	49.80	1.449E+05	1.089E+07	0.1504
2644.06	"	27.58	0.51000	61.94	1.483E+05	1.101E+07	0.1871
2711.77	"	27.58	0.46000	55.87	1.523E+05	1.113E+07	0.1688
2778.89	"	27.58	0.40000	48.58	1.558E+05	1.122E+07	0.1468
2838.41	"	27.58	0.34000	41.30	1.585E+05	1.128E+07	0.1248
2852.80	"	27.58	0.32000	38.87	1.590E+05	1.129E+07	0.1174
2897.10	"	27.58	0.27000	32.79	1.606E+05	1.131E+07	0.0991
2941.40	"	27.58	0.22000	26.72	1.619E+05	1.133E+07	0.0807
2985.67	"	27.58	0.16000	19.43	1.630E+05	1.133E+07	0.0587
3034.20	"	27.58	0.09000	10.93	1.637E+05	1.134E+07	0.0330
3083.30	"	27.58	0.03000	3.64	1.641E+05	1.134E+07	0.0110
3123.15	"	27.58	-0.03000	-3.64	1.641E+05	1.134E+07	-0.0110
3137.54	"	27.58	-0.05000	-6.07	1.640E+05	1.134E+07	-0.0183
3201.70	"	27.58	-0.12000	-14.57	1.633E+05	1.134E+07	-0.0440
3266.50	"	27.58	-0.14000	-17.00	1.623E+05	1.136E+07	-0.0514
3331.30	"	27.58	-0.07000	-8.50	1.615E+05	1.138E+07	-0.0257
3377.35	"	27.58	-0.02000	-2.43	1.612E+05	1.138E+07	-0.0073
3435.90	"	27.58	0.03000	3.64	1.613E+05	1.138E+07	0.0110
3500.70	"	27.58	0.08000	9.72	1.617E+05	1.137E+07	0.0294
3565.70	"	27.58	0.10169	12.35	1.624E+05	1.134E+07	0.0373
3623.80	"	27.58	0.10169	12.35	1.631E+05	1.130E+07	0.0373
3706.10	"	27.58	0.10169	12.35	1.641E+05	1.125E+07	0.0373

TABLE 2. (CON'T)
NLS/PLS/CRV MAX q alpha AIRLOAD & MOMENT @ Xcg
10' STRETCH NLV2 WITH INLINE PLS/CRV

VJB

9/2/92

Sref = 593.96 ft^2 MACH = 1.3

Xcg(sta) = 3051 q (psf) = 705.1

α = 5.5

CN pls = 0.1378

Xcp pls = 1966

MAX q alpha AIRLOADS
SUMMED LOADS INCLUDE PLS/CRV

NLS/PLS/CRV TRAJECTORY
MACH=1.3, α =5.5 deg

X STA. No in	Component	LOCAL DIAMETER ft	CNα per Radian per X/D	NORMAL AIR LOAD DISTRIBUTED lbs/in	SUMMED NORMAL F. FWD OF X lbs w/ pls/crv	SUMMED MOMENT ABOUT Xcg ft-lbs w/ pls/crv	LOCAL NET PRESSURE DISTRIB. psi normal to surface
3784.90	•	27.58	0.10169	12.35	1.651E+05	1.120E+07	0.0373
3871.00	•	27.58	0.10169	12.35	1.662E+05	1.113E+07	0.0373
3932.00	•	27.58	0.10169	12.35	1.669E+05	1.107E+07	0.0373
3996.80	•	27.58	0.10169	12.35	1.677E+05	1.101E+07	0.0373
4058.00	•	27.58	0.10169	12.35	1.685E+05	1.095E+07	0.0373
4108.91	•	27.58	0.10169	12.35	1.691E+05	1.090E+07	0.0373
4122.65	Thrust structure.	27.58	0.10169	12.35	1.693E+05	1.088E+07	0.0373
4137.30	•	27.58	0.10169	12.35	1.695E+05	1.087E+07	0.0373
4151.90	•	27.58	0.10169	12.35	1.696E+05	1.085E+07	0.0373
4166.60	•	27.58	0.10169	12.35	1.698E+05	1.083E+07	0.0373
4181.20	•	27.58	0.10169	12.35	1.700E+05	1.082E+07	0.0373
4195.70	•	27.58	0.10169	12.35	1.702E+05	1.080E+07	0.0373
4210.30	•	27.58	0.10169	12.35	1.704E+05	1.078E+07	0.0373
4227.40	•	27.58	0.10169	12.35	1.706E+05	1.076E+07	0.0373
4241.60	•	27.58	0.10169	12.35	1.708E+05	1.074E+07	0.0373
4254.10	•	27.58	0.10169	12.35	1.709E+05	1.073E+07	0.0373
4265.00	•	27.58	0.10169	12.35	1.710E+05	1.072E+07	0.0373
4275.90	•	27.58	0.10169	12.35	1.712E+05	1.070E+07	0.0373
4286.90	•	27.58	0.10169	12.35	1.713E+05	1.069E+07	0.0373
4297.80	•	27.58	0.10169	12.35	1.714E+05	1.067E+07	0.0373
4308.80	•	27.58	0.10169	12.35	1.716E+05	1.066E+07	0.0373
4319.70	•	27.58	0.10169	12.35	1.717E+05	1.065E+07	0.0373
4330.60	•	27.58	0.10169	12.35	1.719E+05	1.063E+07	0.0373
4341.60	•	27.58	0.10169	12.35	1.720E+05	1.062E+07	0.0373
4352.20	•	27.58	0.10169	12.35	1.721E+05	1.060E+07	0.0373
4362.80	•	27.58	0.10169	12.35	1.722E+05	1.059E+07	0.0373
4374.20	•	27.58	0.10169	12.35	1.724E+05	1.057E+07	0.0373
4385.50	•	27.58	0.10169	12.35	1.725E+05	1.056E+07	0.0373
4385.50	End of struct.	0.00	0.00000	0.00	1.725E+05	1.056E+07	0.0000

THRUST, lbs=	2,941,220
Xgimbal=	4385.50
Xcp=	2316.71
CNa(per deg) =	0.07490

δ/α =	0.006	<<rad/deg			
δ =	1.849	<<deg gimbal req'd			
			Veh. Weight =	1,545,323	lbs
Axial Thrust=	2,939,688	lbs	Axial Accel=	1.9023	g's (Axial T)
Tangential T=	94,965	lbs	Tang. Accel. =	0.1116	g's (aero+T)

FIGURE 10. MAX q alpha AIRLOAD DISTRIBUTION

10' STRETCH NLS2 WITH INLINE PLS/CRV

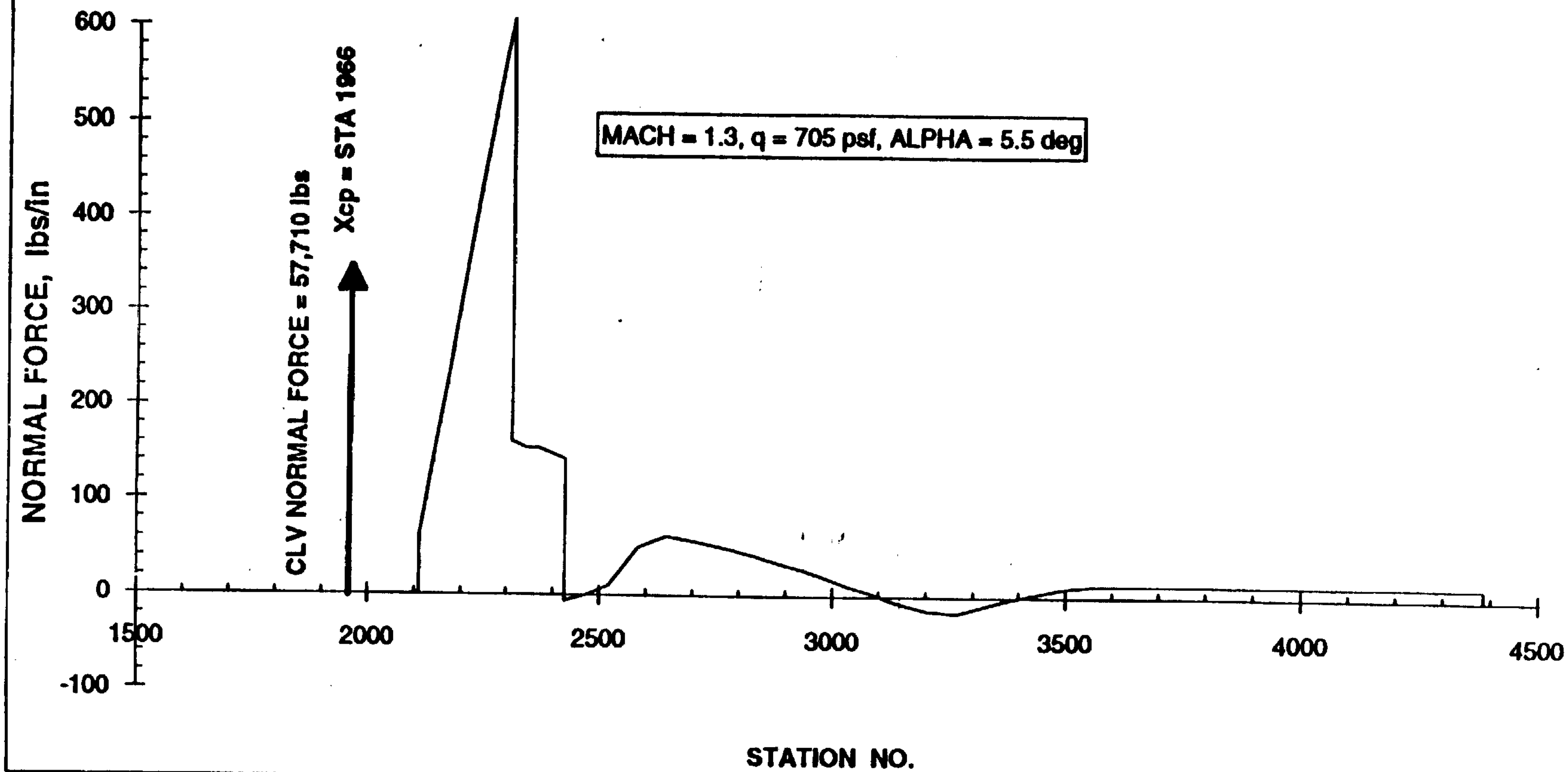


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FIGURE 11.
MAX q alpha NET PRESSURE DISTRIBUTION
10' STRETCH NLS2 WITH INLINE PLS/CRV

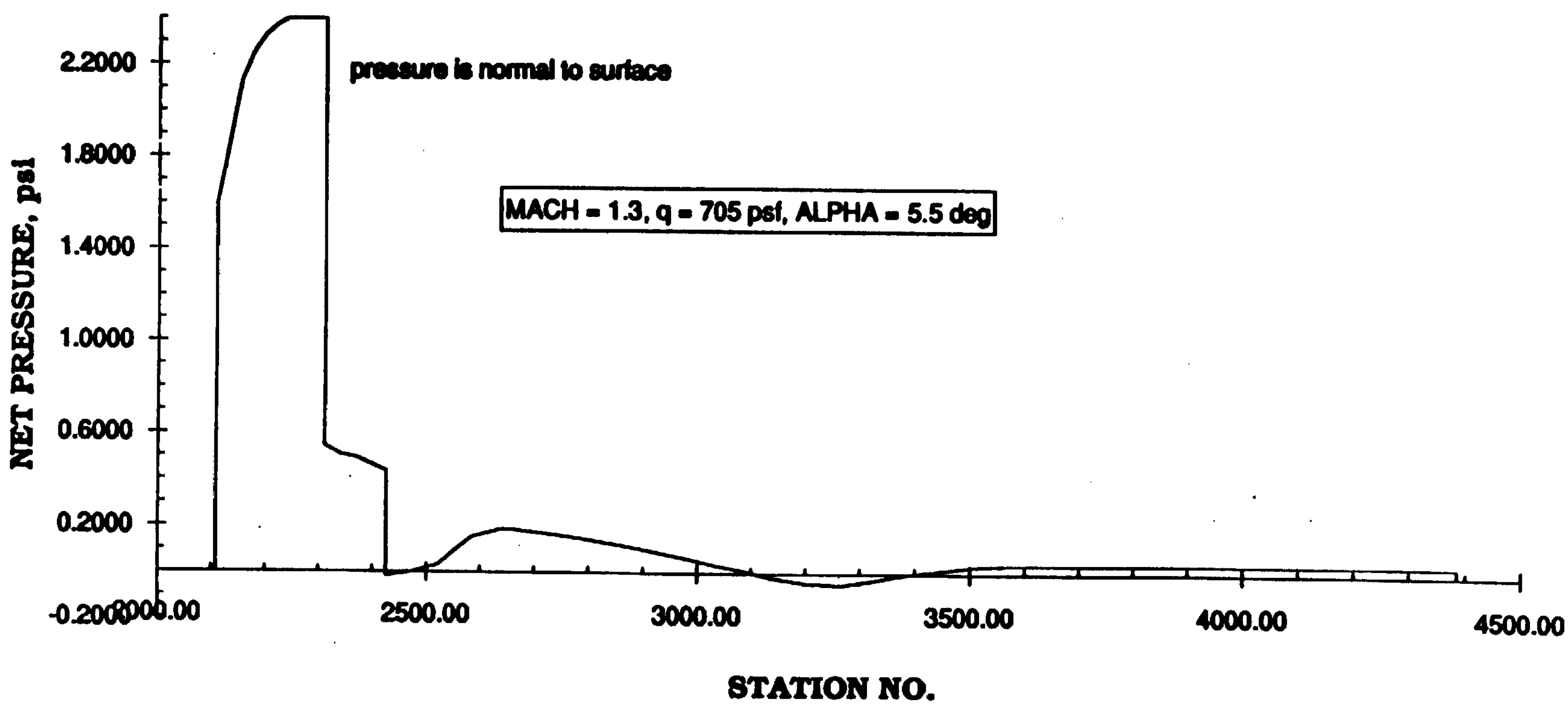

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FIGURE 12.
MAX q alpha AIRLOAD SUMMATION COMPARISON

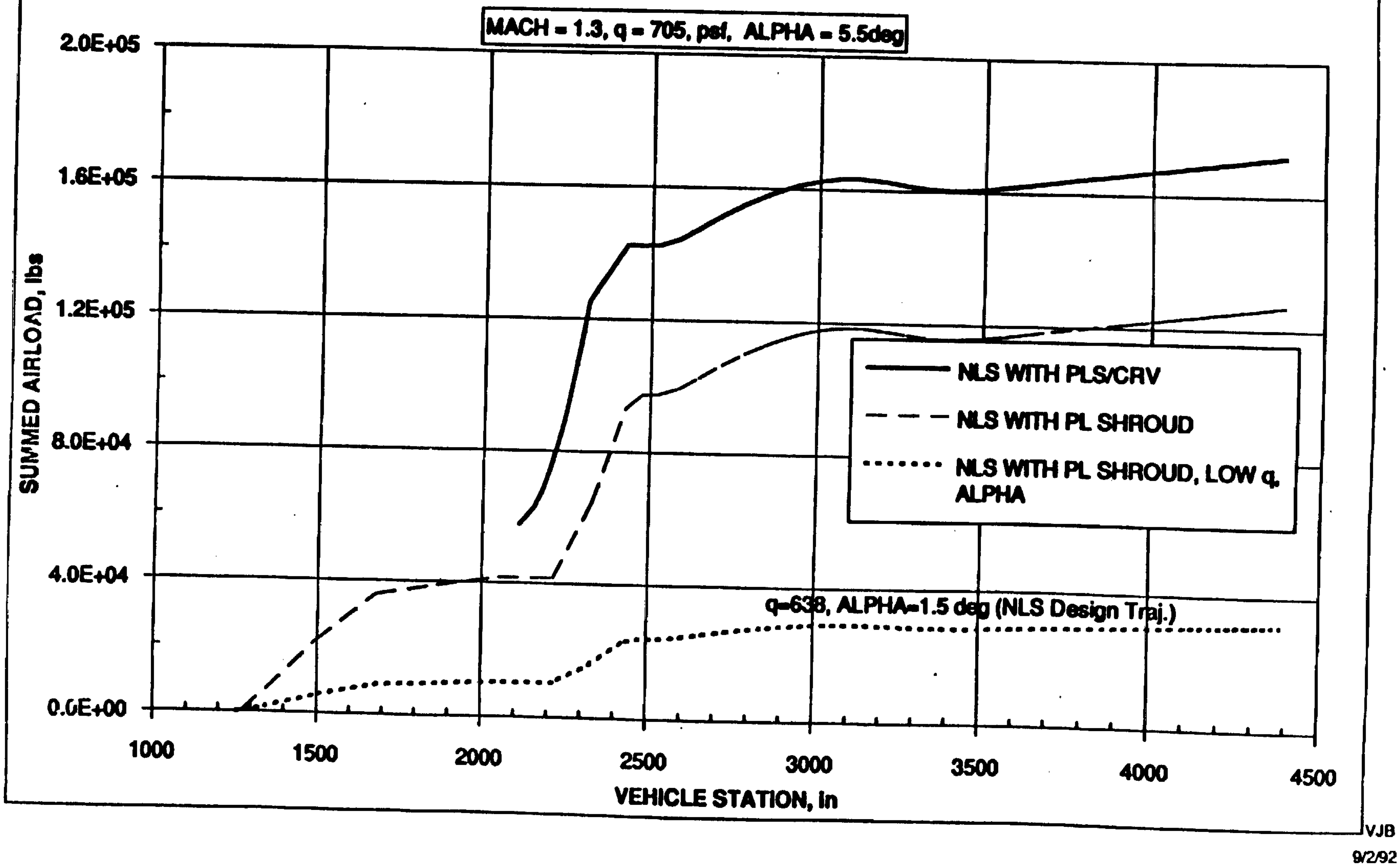

FIGURE 13.
MAX q alpha AIRLOAD MOMENT @ Xcg COMPARISON

MACH = 1.3, q = 705, psf, ALPHA = 5.5 deg

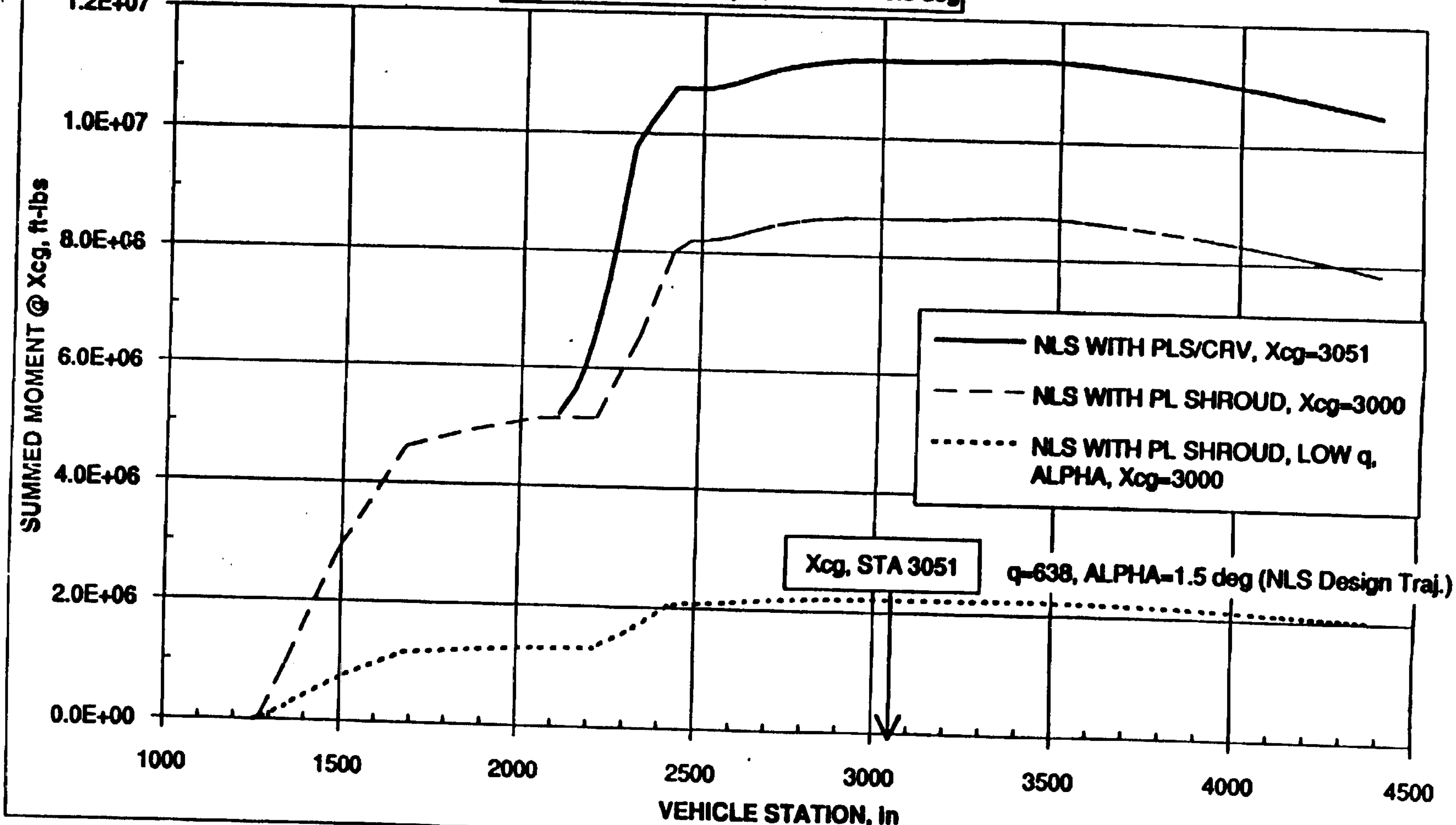

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FIGURE 14.
MAX q alpha SUMMED RUNNING MOMENT FROM LOADS FORWARD OF X-STA

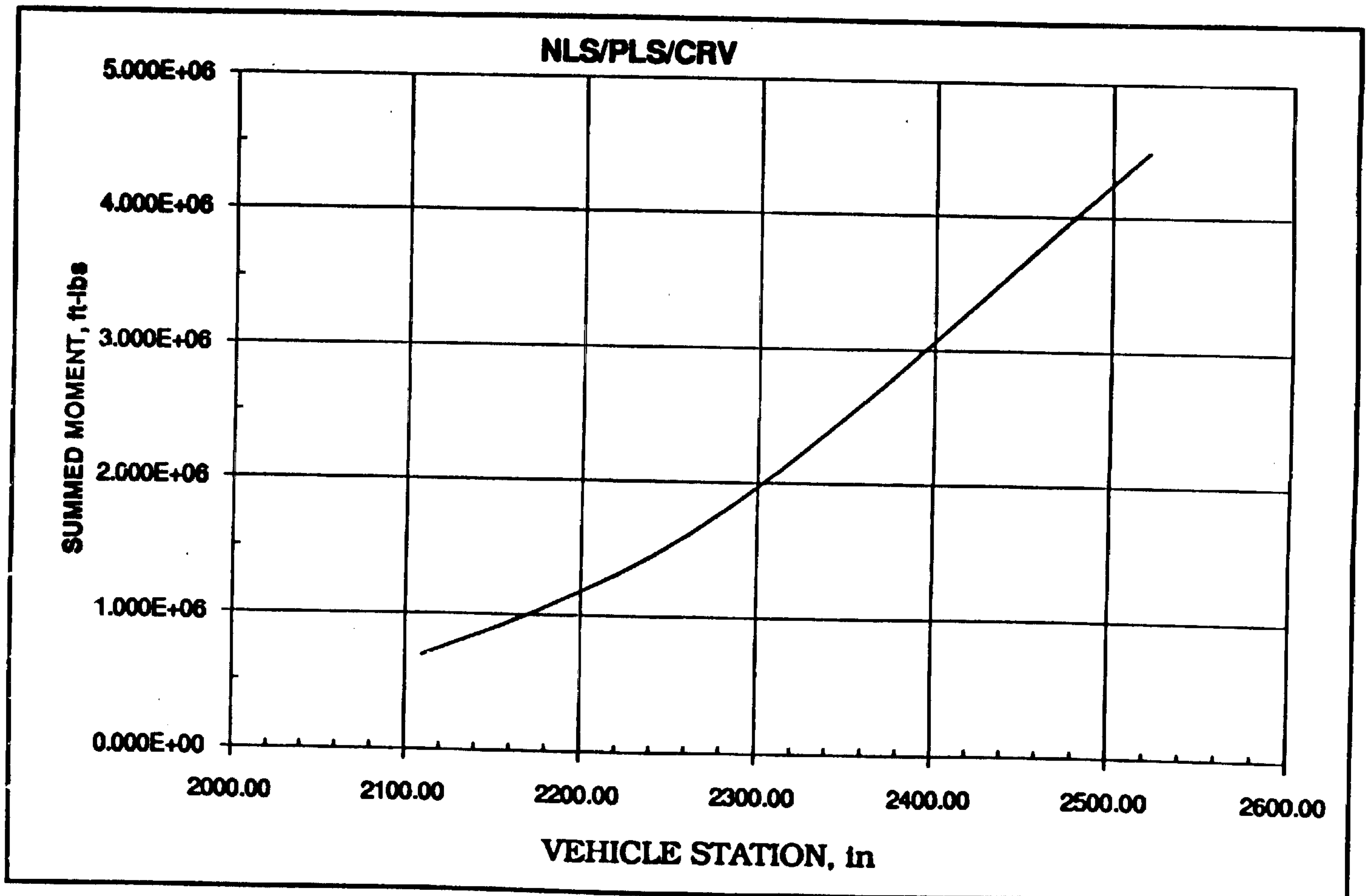

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TABLE 3.

MAX q alpha SUMMED RUNNING MOMENT FROM LOADS FWD OF X-STA VJB 9/2/92

NLS2/CRV/PLS

MACH =1.3, q=705.1 psf

alpha = 5.5 deg

X STA. No in	Component description	SUMMED MOMENT ABOUT X-STA ft-lbs
2109.70	32.5 deg	6.906E+05
2153.70	fairing	9.102E+05
2175.70	"	1.030E+06
2198.37	<fwd bulkhead	1.163E+06
2219.70	"	1.301E+06
2241.70	"	1.457E+06
2263.70	"	1.631E+06
2285.70	"	1.824E+06
2310.80	<aft bulkhead	2.071E+06
2310.80	10.4 deg	2.071E+06
2340.80	fairing	2.391E+06
2370.80	"	2.723E+06
2424.80	"	3.349E+06
2424.80	1st Stage	3.349E+06
2472.80	"	3.920E+06
2520.80	"	4.491E+06
2583.55	"	5.243E+06

FIGURE 15.
SENSITIVITY OF GIMBAL REQM'T WITH Xcg

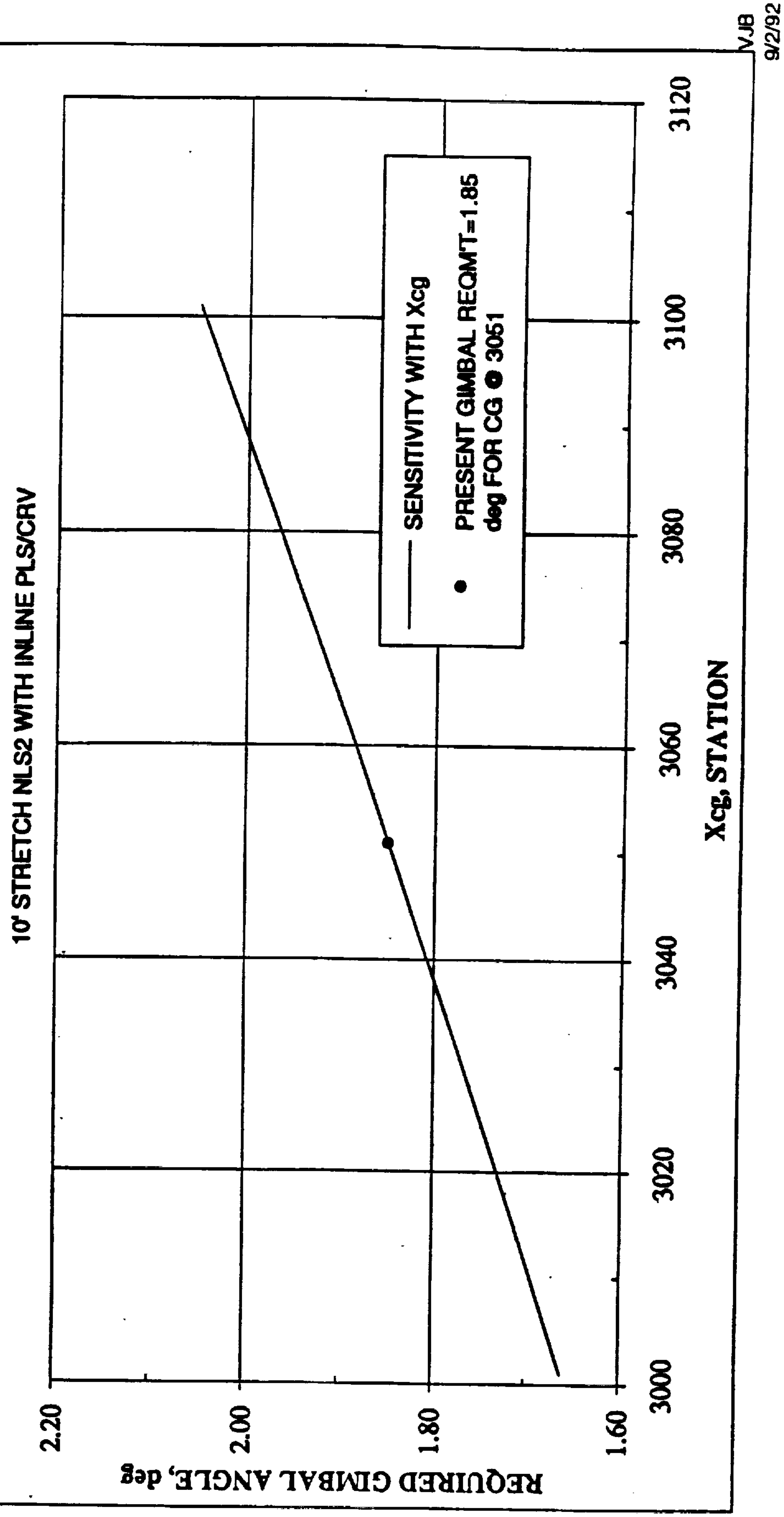


FIGURE 16.
SENSITIVITY OF GIMBAL REQM'T WITH q-alpha

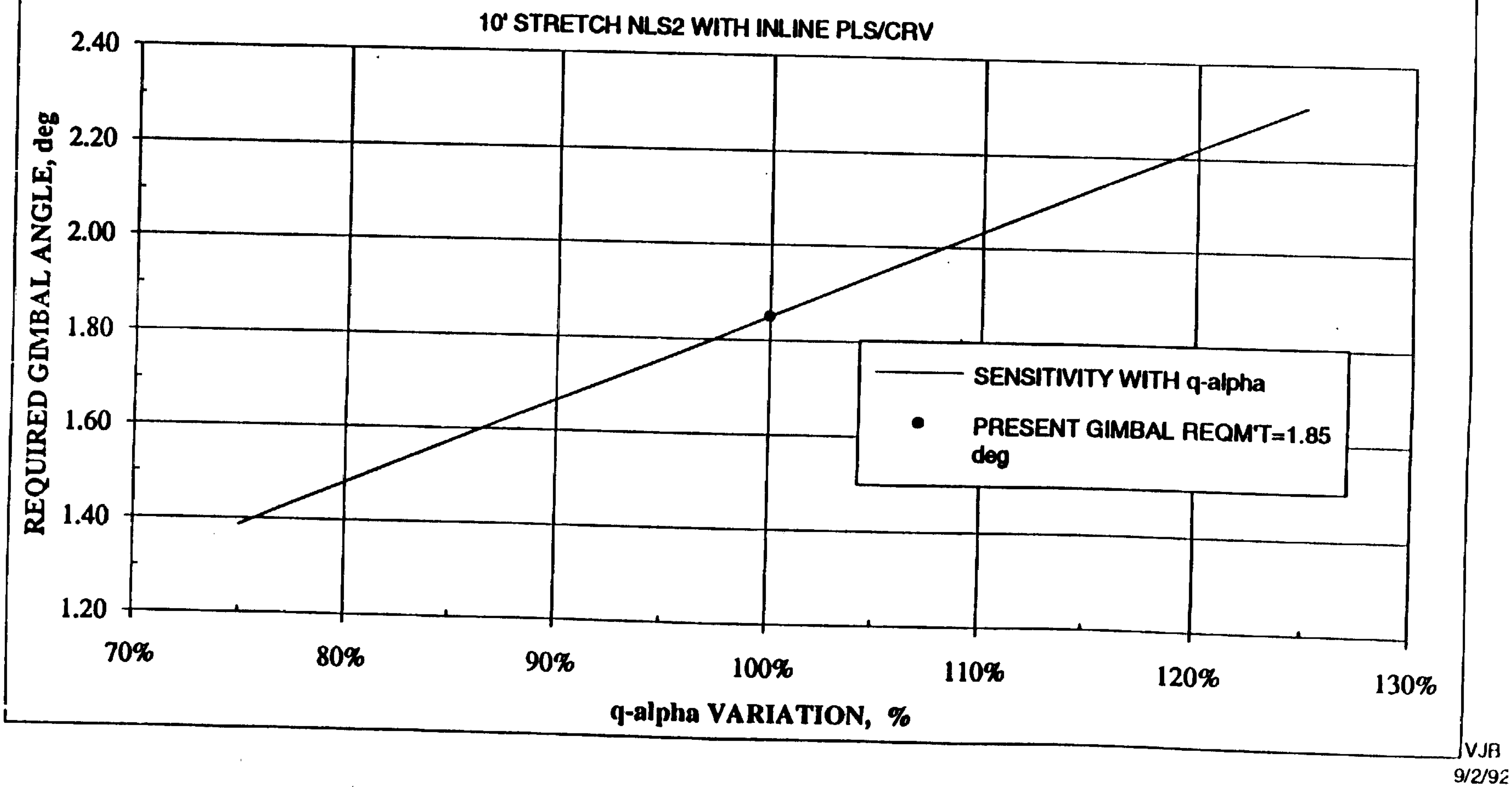
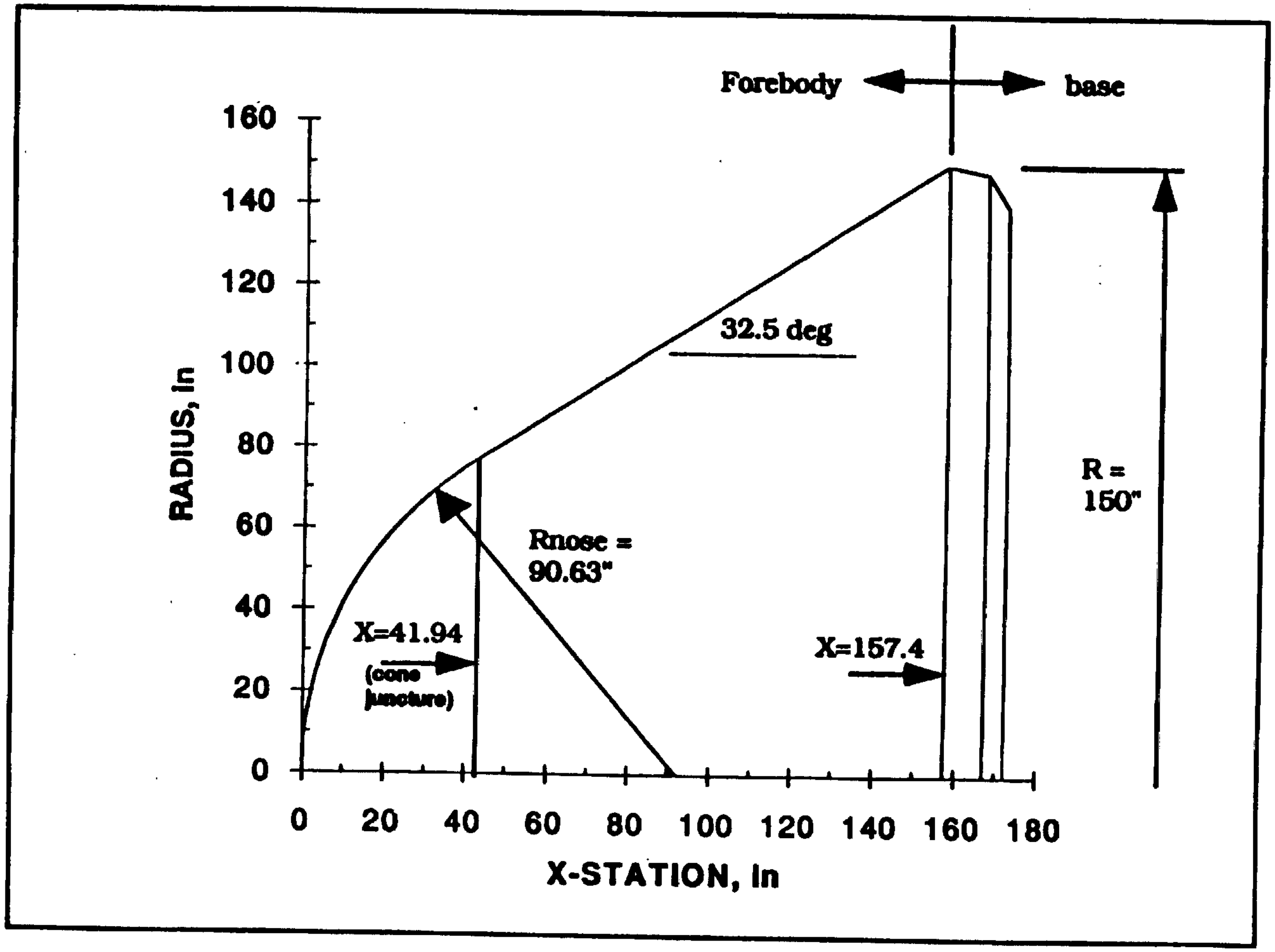


FIGURE 17.
CRV AERODYNAMIC
GEOMETRY



vjb
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FIGURE 18. CRV ENTRY NET PRESSURE DISTRIBUTION

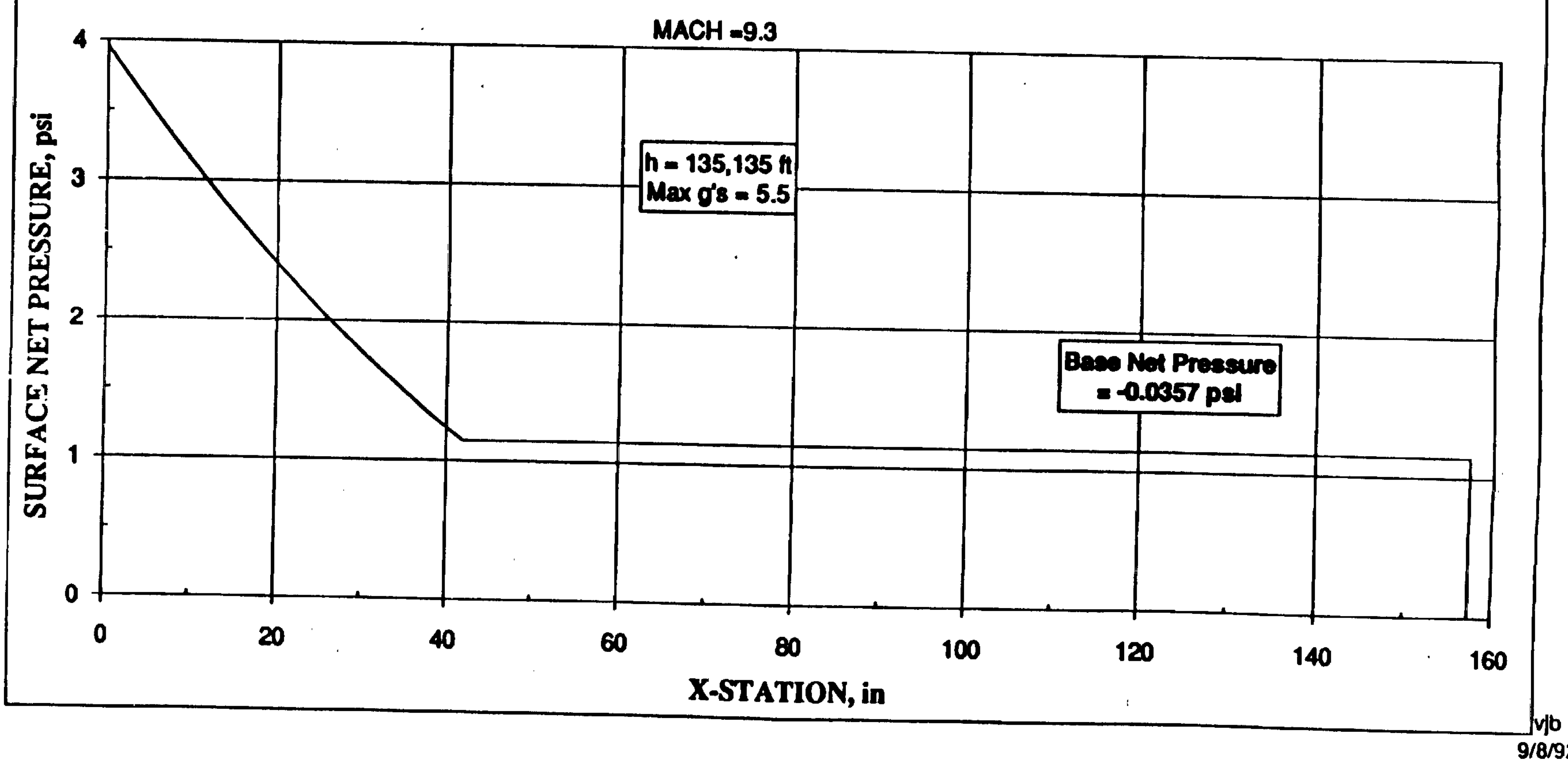


TABLE 4. CRV ENTRY PRESSURE AND DRAG LOADING

h=135,135 ft
Max g = 5.5
Max q = 312 psf
Alpha = 0.0

MACH = 9.3

Sref, ft^2 = 490.87

base net pressure (psi)= -0.0357

X in	radius in	COMPONENT	slope deg	Cp local coefficient	forebody surface net pressure psi	projected delta Area ft^2	forebody delta drag lbs
0.00	0.00	90.63° sphere	90.00	1.830	3.960	0.000	0.0
0.30	7.37	"	85.34	1.818	3.933	1.184	670.9
0.70	11.24	"	82.87	1.802	3.899	1.573	883.2
1.20	14.70	"	80.67	1.782	3.856	1.957	1086.3
2.00	18.94	"	77.94	1.751	3.787	3.108	1694.7
3.00	23.13	"	75.22	1.711	3.702	3.846	2050.0
4.00	26.63	"	72.91	1.672	3.618	3.802	1980.7
6.00	32.43	"	69.04	1.596	3.453	7.473	3715.6
8.00	37.23	"	65.75	1.522	3.292	7.298	3459.3
10.00	41.38	"	62.83	1.449	3.134	7.124	3215.1
12.00	45.07	"	60.18	1.378	2.981	6.949	2982.7
14.00	48.39	"	57.73	1.309	2.831	6.775	2761.7
16.00	51.42	"	55.43	1.241	2.685	6.600	2552.0
18.00	54.21	"	53.26	1.176	2.543	6.426	2353.1
20.00	56.79	"	51.20	1.112	2.405	6.251	2164.9
22.00	59.19	"	49.22	1.050	2.271	6.077	1987.0
24.00	61.44	"	47.32	0.989	2.140	5.902	1819.1
26.00	63.54	"	45.49	0.931	2.014	5.728	1660.9
28.00	65.51	"	43.71	0.874	1.891	5.553	1512.2
30.00	67.37	"	41.99	0.819	1.772	5.379	1372.6
32.00	69.11	"	40.31	0.766	1.657	5.204	1241.9
34.00	70.76	"	38.67	0.715	1.546	5.030	1119.8
36.00	72.32	"	37.07	0.665	1.439	4.855	1005.9
38.00	73.78	"	35.50	0.617	1.335	4.680	900.0
40.00	75.17	"	33.96	0.571	1.236	4.506	801.9
41.94	76.44	"	32.50	0.528	1.143	4.197	690.8
41.94	76.44	32.5 deg cone	32.50	0.528	1.143	0.000	0.0
48.00	80.30	"	32.50	0.528	1.143	13.209	2174.4
52.00	82.85	"	32.50	0.528	1.143	9.071	1493.1
56.00	85.40	"	32.50	0.528	1.143	9.354	1539.7
60.00	87.95	"	32.50	0.528	1.143	9.637	1586.4
64.00	90.50	"	32.50	0.528	1.143	9.921	1633.0
68.00	93.04	"	32.50	0.528	1.143	10.204	1679.7
72.00	95.59	"	32.50	0.528	1.143	10.487	1726.3
76.00	98.14	"	32.50	0.528	1.143	10.771	1772.9
80.00	100.69	"	32.50	0.528	1.143	11.054	1819.6
84.00	103.24	"	32.50	0.528	1.143	11.337	1866.2
88.00	105.79	"	32.50	0.528	1.143	11.621	1912.9
92.00	108.33	"	32.50	0.528	1.143	11.904	1959.5
96.00	110.88	"	32.50	0.528	1.143	12.187	2006.1
100.00	113.43	"	32.50	0.528	1.143	12.471	2052.8
104.00	115.98	"	32.50	0.528	1.143	12.754	2099.4
108.00	118.53	"	32.50	0.528	1.143	13.037	2146.1
112.00	121.08	"	32.50	0.528	1.143	13.321	2192.7
116.00	123.32	"	32.50	0.528	1.143	13.604	2239.3

TABLE 4 (CONT.)

CRV ENTRY PRESSURE AND DRAG LOADING

VJB
9/8/92

h=135,135 ft
Max g = 5.5
Max q = 312 psf
Alpha = 0.0

MACH = 9.3

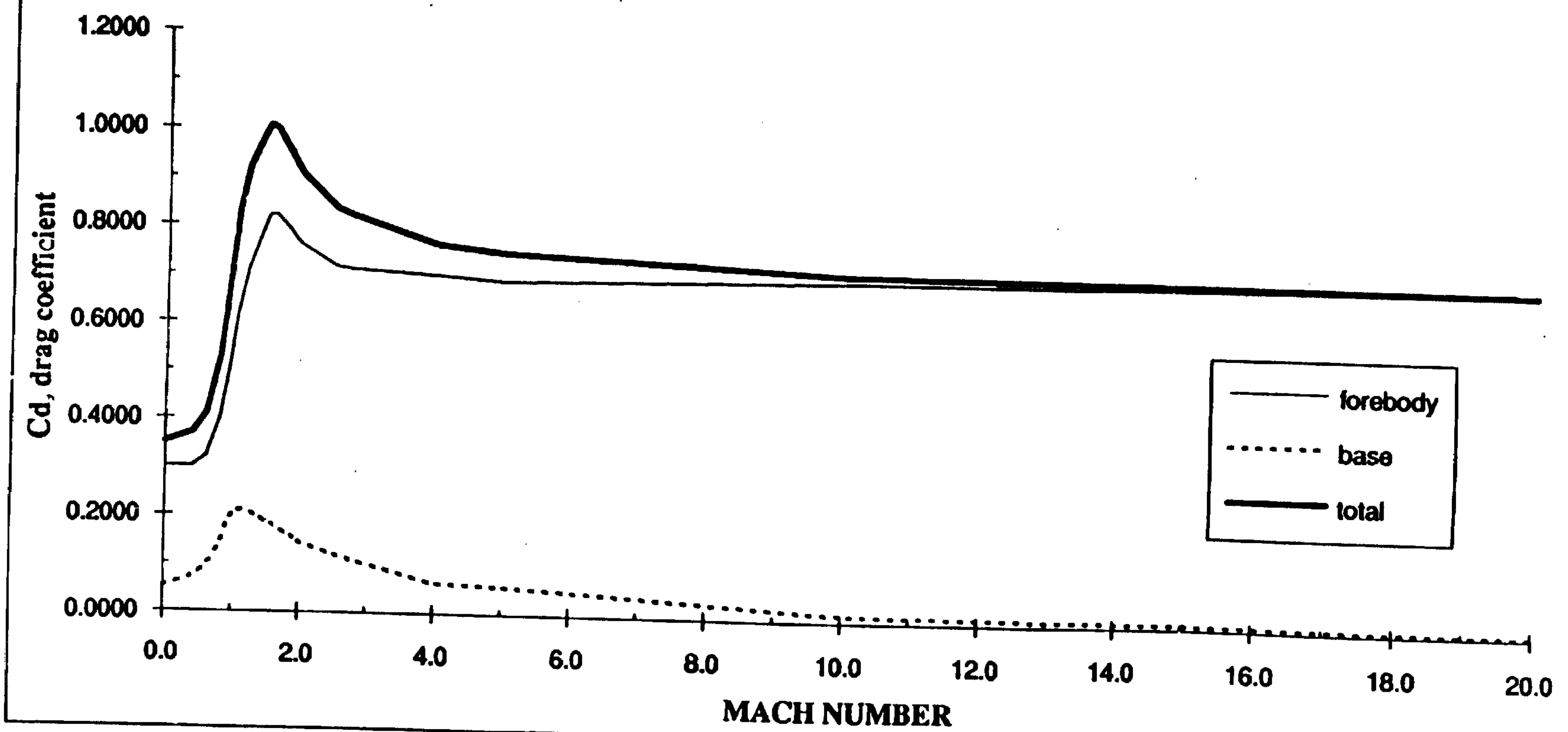
Sref, ft^2 = 490.87

base net pressure (psi)= -0.0357

X in	radius in	COMPONENT	slope deg	Cp local coefficient	forebody surface net pressure psi	projected delta Area ft^2	forebody delta drag lbs
120.00	126.17	"	32.50	0.528	1.143	13.887	2286.0
124.00	128.72	"	32.50	0.528	1.143	14.171	2332.6
128.00	131.27	"	32.50	0.528	1.143	14.454	2379.3
132.00	133.82	"	32.50	0.528	1.143	14.737	2425.9
136.00	136.37	"	32.50	0.528	1.143	15.021	2472.6
140.00	138.91	"	32.50	0.528	1.143	15.304	2519.2
144.00	141.46	"	32.50	0.528	1.143	15.587	2565.8
150.00	145.28	"	32.50	0.528	1.143	23.912	3936.2
157.40	150.00	max diameter	32.50	0.528	1.143	30.379	5000.7
157.40	0.00		0	0.000	0.000	0.000	0.0
157.40	150.00						
167.00	148.00						
167.00	0.00						
167.00	148.00						

drag =		105501	lbs
Cd =		0.6900	w/o Cd base
Cd =		0.7065	w/ Cd base

FIGURE 19. CRV DRAG COEFFICIENT
Sref = 491 ft^2



VJB
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TABLE 5.
CRV DRAG COEFFICIENT
Sref= 491 ft^2

MACH	Cd forebody	Cd base	Cd total
0.00	0.3000	0.0500	0.3500
0.40	0.3000	0.0700	0.3700
0.60	0.3200	0.0900	0.4100
0.80	0.4000	0.1300	0.5300
0.94	0.5100	0.1900	0.7000
1.05	0.6200	0.2100	0.8300
1.20	0.7100	0.2100	0.9200
1.40	0.7870	0.1950	0.9820
1.50	0.8200	0.1860	1.0060
1.60	0.8200	0.1790	0.9990
1.80	0.7900	0.1620	0.9520
1.90	0.7700	0.1550	0.9250
1.97	0.7600	0.1480	0.9080
2.00	0.7580	0.1450	0.9030
2.50	0.7200	0.1200	0.8400
2.75	0.7100	0.1100	0.8200
4.00	0.6900	0.0625	0.7525
5.00	0.6818	0.0571	0.7389
10.00	0.6904	0.0143	0.7047
20.00	0.6926	0.0036	0.6962

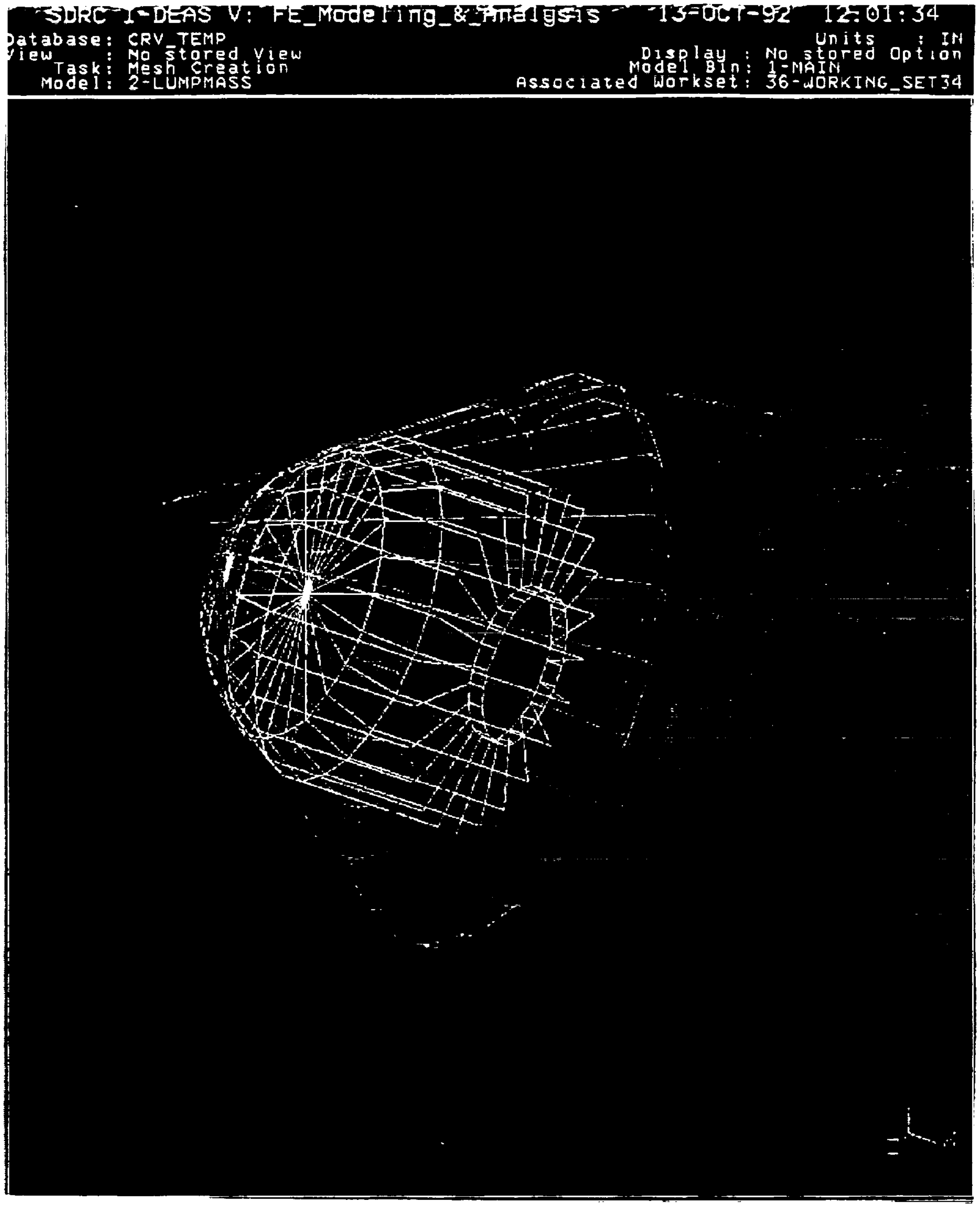

FIGURE 20. CRV CAPSULE

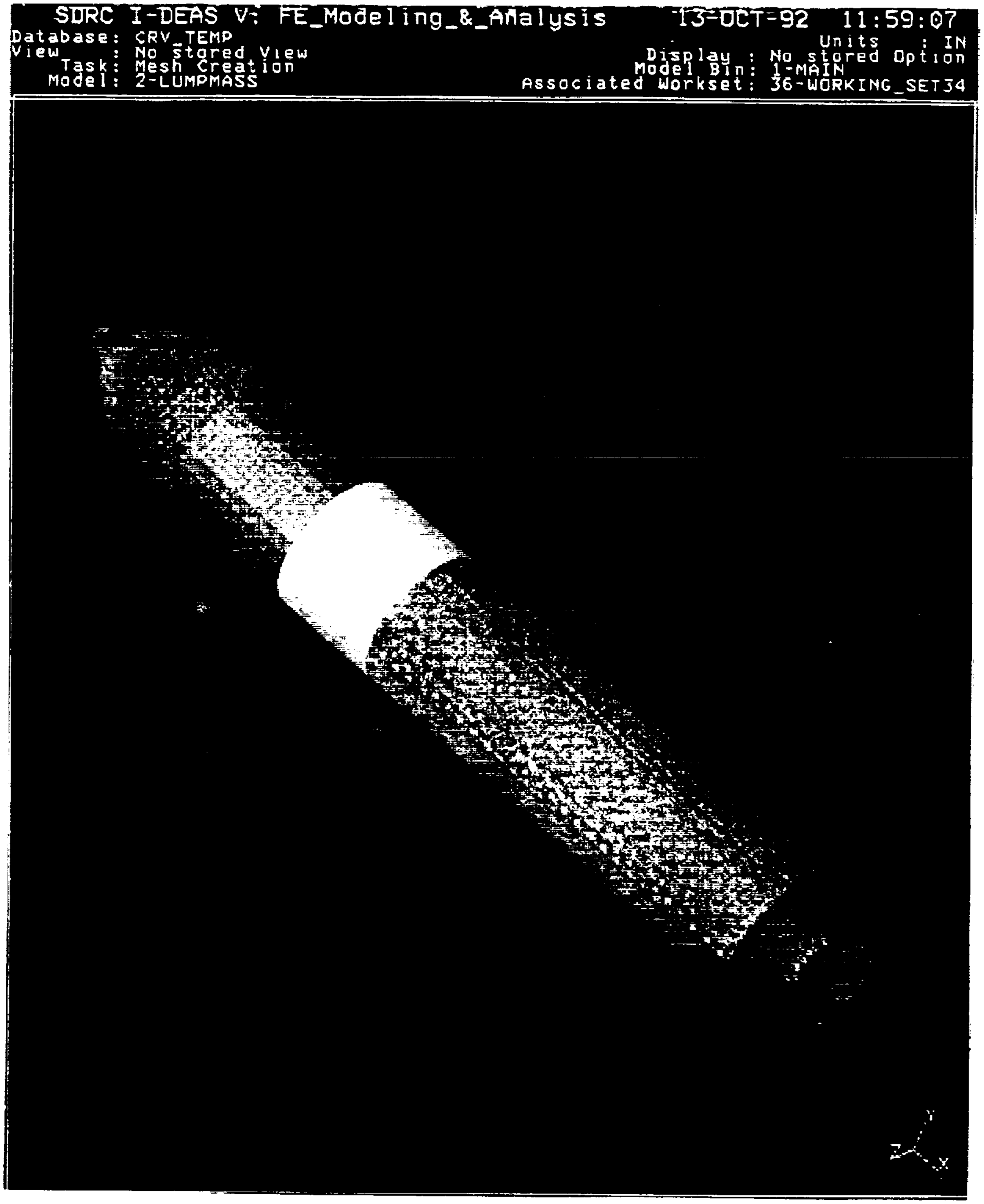

FIGURE 21. CRV ON TOP OF BOOSTER

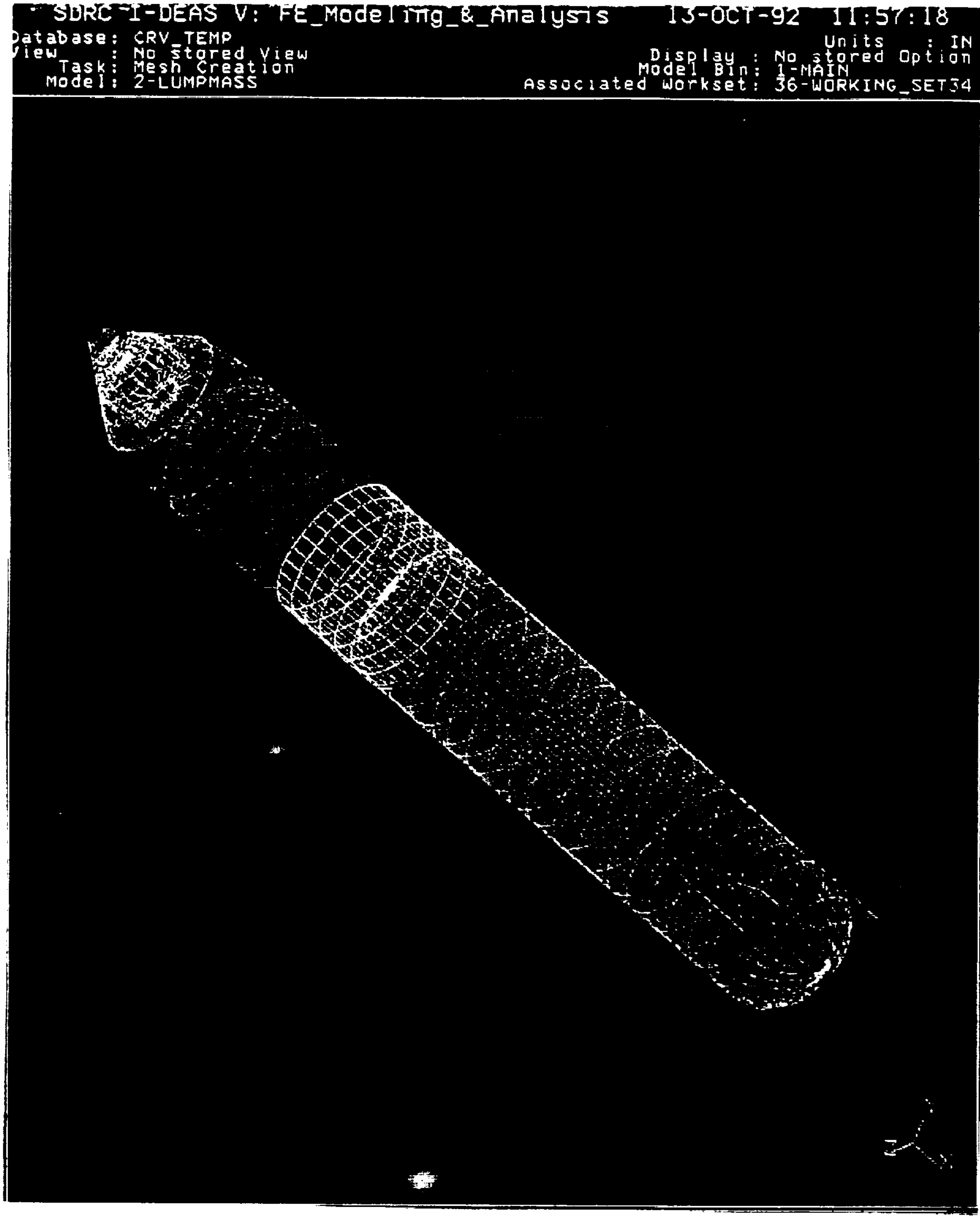

FIGURE 22. CRV ON BOOSTER

IV. Mass Properties

A weight breakdown (Table 6) has been generated to support the stress, dynamic, and performance analyses. The mass properties are used as the starting point for creating inertial loads and a mass matrix (used in computing mode shapes and frequencies).

The vehicle was broken down into its basic systems, the mass properties found for each system, then the systems reassembled. Some systems were not included on each vehicle, but were included in the analysis if they were applicable. The general breakdown included: structure, payload, propulsion, TPS, avionics, and fuel. The total vehicle weight breakdown and distribution was then found by summing up each system. Spreadsheets were created to help in the determination of the weight breakdown for each vehicle.

Much of the rational used to find these weights were taken from both the Shuttle and Saturn programs. The Shuttle external tank and engines, and Saturn designed bulkheads, propulsion cones, and subsystems were used as guidelines for mass properties determination.

V. Vehicle Load Conditions

The vehicle was analyzed for two balanced conditions: (1) on-pad winds and (2) a high-Q flight condition. The first condition includes inertial (one-G) and aerodynamic effects. The second condition includes inertial, thrust, and aerodynamic forces.

VI. Resultant Stresses and Deflections

Stresses and deflections for the on-pad and high-Q conditions are shown in Figures 23 - 27. The tank stresses shown in the contour plots reflect a zero internal tank pressure. The tank pressures have been accounted for in the assessment summary shown in Table 7. The detailed computations are shown on the pages that follow Table 7. Re-entry stresses are shown in Figures 28 and 29.

VII. Dynamic Characteristics

The first five mode shapes (Figures 30 - 34) and frequencies were computed for the vehicle with 100% fuel. The original mass and stiffness matrices were not reduced from over 11,000 degrees of freedom (DOF). Simultaneous vector iteration was used to compute the first five mode shapes and frequencies. The results are summarized in Table 8. A comparison with ET modes is given in Table 9.

VIII. Conclusions

The preliminary assessment of a CRV capsule has been performed. The concept appears to be structurally feasible.

TABLE 6.

NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CRV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
PLS + STRUCT ADPT					(32000)
CG (Xpls)	245***	0	32000		
Σ	*** FROM TOP OF CRV			32000	
CRV					(14000)
CG (Xnls)	2273.1	0	14000		
Σ				14000	
AFT SKIRT					(4000)
CG (Xnls)	2186	0	4000		
Σ				4000	
CORE TANK					(88213)
FWD SKIRT	2480.4	1696	1781		
	2520.8	1084	1138		
Σ				2919	
LO2 TK-F DOME	2410.1	26	27		
(FIXED WT)	2417.6	209	219		
	2440.6	388	407		
	2480.4	545	572		
	2520.8	281	295		
Σ				1521	
CYLINDER	2520.8	816	857		
	2580.5	998	1048		
	2644	1724	1810		
	2711.7	1814	1905		
	2778.8	1724	1810		
	2838.4	1089	1143		
	2852.8	907	952		
Σ				9526	
LO2 TK-A DOME	2852.8	86	90		
(FIXED WT)	2892.5	690	725		
	2932.3	1285	1349		
	2955.3	1803	1893		
	2962.7	931	978		
Σ				5035	
INTERTANK	2852.8	1306	1371		
	2897.1	1979	2078		
	2941.4	1966	2064		
	2985.6	2067	2170		
	3034.2	2143	2250		
	3083.3	1979	2078		
	3123.1	1243	1305		
Σ				13317	
LH2 TK-F DOME	3013.1	430	452		
(FIXED WT)	3020.6	833	875		
	3043.6	594	624		
	3083.4	319	335		
	3123.1	40	42		

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TABLE 6 (CONT.)

NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CRV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
Σ				2327	
CYLINDER	3137.5	943	990		
	3201.7	1362	1430		
	3266.5	2200	2310		
	3331.3	2235	2347		
	3377.3	1886	1980		
	3435.9	1781	1870		
	3500.7	2096	2201		
	3565.5	2200	2310		
	3623.8	2096	2201		
	3706.1	2410	2531		
	3784.9	2759	2897		
	3871.0	2829	2970		
	3932.0	2515	2641		
	3996.8	2165	2273		
	4058	2165	2273		
	4122.6	1921	2017		
	4187.6	1118	1174		
	4252.2	244	256		
Σ				36671	
LH2 TK-A DOME	4252.2	55	58		
(FIXED WT)	4292.6	444	466		
	4332.4	826	867		
	4355.4	1159	1217		
	4362.8	598	628		
Σ				3236	
AFT SKIRT	4355.4	2211	2211		
(*NO STATION LOC)	4362.8	2210	2210		
Σ				4421	
EXT HARDWARE	2520.8	30	32		
	2580.5	35	37		
	2644	62	65		
	2711.7	65	68		
	2778.8	61	64		
	2838.4	38	40		
	2852.8	29	30		
	2897.1	43	45		
	2941.4	43	45		
	2985.6	44	46		
	3034.2	46	48		
	3083.3	43	45		
	3123.1	27	28		
	3137.5	38	40		
	3201.7	62	65		
	3266.5	62	65		
	3331.3	53	56		
	3377.3	50	53		

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TABLE 6 (CONT.)

NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CRV) - STRETCH "B" VERSION

ITEM	STA	RI REF	"B" VERSION	TOTAL WT	
		WT (LBS)	WT (LBS)		
	3435.9	59	62		
	3500.7	62	65		
	3565.5	59	62		
	3623.8	68	71		
	3706.1	77	81		
	3784.9	79	83		
	3871.0	71	75		
	3932.0	61	64		
	3996.8	61	64		
	4058	55	58		
	4122.6	30	32		
	4187.6	6	6		
Σ				1595	
TPS-LO2 TANK	2520.8	15	16		
	2580.5	21	22		
	2644	36	38		
	2711.7	38	40		
	2778.8	36	38		
	2838.4	23	24		
	2852.8	23	24		
Σ				202	
TPS-LH2 TANK	3137.5	27	28		
	3201.7	38	40		
	3266.5	62	65		
	3331.3	63	66		
	3377.3	53	56		
	3435.9	50	53		
	3500.7	59	62		
	3565.5	62	65		
	3623.8	59	62		
	3706.1	68	71		
	3784.9	78	82		
	3871.0	80	84		
	3932.0	71	75		
	3996.8	61	64		
	4058	61	64		
	4122.6	54	57		
	4187.6	32	34		
	4252.2	8	8		
Σ				1035	
INSULATION	2852.8	94	99		
	2897.1	142	149		
	2941.4	141	148		
	2985.6	148	155		
	3034.2	154	162		
	3083.3	142	149		
	3123.1	89	93		

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TABLE 6 (CONT.)

NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CRV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
Σ				956	
LO2-FEED	2985.6	136	143		
	3034.2	139	146		
	3083.3	129	135		
	3123.1	81	85		
	3137.5	115	121		
	3201.7	186	195		
	3266.5	190	200		
	3331.3	163	171		
	3377.3	152	160		
	3435.9	180	189		
	3500.7	190	200		
	3565.5	180	189		
	3623.8	207	217		
	3706.1	234	246		
	3784.9	240	252		
	3871.0	213	224		
	3932.0	183	192		
	3996.8	183	192		
	4058	163	171		
	4122.6	95	100		
	4187.6	29	30		
Σ				3557	
LO2 - PRESSURE	2330.3	11	12		
	2424.8	9	9		
	2472.8	6	6		
	2580.5	4	4		
	2644	4	4		
	2711.7	8	8		
	2778.8	8	8		
	2838.4	8	8		
	2852.8	5	5		
	2897.1	4	4		
	2941.4	5	5		
	2985.6	5	5		
	3034.2	6	6		
	3083.3	6	6		
	3123.1	5	5		
	3137.5	3	3		
	3201.7	5	5		
	3266.5	8	8		
	3331.3	8	8		
	3377.3	7	7		
	3435.9	6	6		
	3500.7	7	7		
	3565.5	8	8		
	3623.8	7	7		

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TABLE 6 (CONT.)

NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CRV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	3706.1	9	9		
	3784.9	10	11		
	3871	10	11		
	3932	9	9		
	3996.8	8	8		
	4058	8	8		
	4122.6	7	7		
	4187.6	4	4		
	4252.2	1	1		
Σ				230	
LO2 - VENT	2330.3	27	28		
	2424.8	21	22		
	2472.8	14	15		
	2580.5	9	9		
	2644	11	12		
	2711.7	19	20		
	2778.8	20	21		
	2838.4	18	19		
	2852.8	12	13		
	2897.1	9	9		
	2941.4	13	14		
	2985.6	13	14		
	3034.2	14	15		
	3083.3	14	15		
	3123.1	13	14		
	3137.5	8	8		
	3201.7	12	13		
	3266.5	19	20		
	3331.3	19	20		
	3377.3	16	17		
	3435.9	15	16		
	3500.7	18	19		
	3565.5	19	20		
	3623.8	18	19		
	3706.1	20	21		
	3784.9	23	24		
	3871	24	25		
	3932	21	22		
	3996.8	18	19		
	4058	18	19		
	4122.6	16	17		
	4187.6	9	9		
	4252.2	2	2		
Σ				548	
LH2 - FEED	4187.6	84	88		
	4252.2	84	88		
				176	

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TABLE 6 (CONT.)

NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CRV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
LH2 - PRESSURE	3034.2	11	12		
	3083.3	12	13		
	3123.1	11	12		
	3137.5	7	7		
	3201.7	10	11		
	3266.5	16	17		
	3331.3	16	17		
	3377.3	13	14		
	3435.9	13	14		
	3500.7	15	16		
	3565.5	16	17		
	3623.8	15	16		
	3706.1	17	18		
	3784.9	20	21		
	3871	20	21		
	3932	18	19		
	3996.8	15	16		
	4058	15	16		
	4122.6	14	15		
	4187.6	8	8		
	4252.2	2	2		
Σ				298	
LH2 - VENT	3034.2	14	15		
	3083.3	15	16		
	3123.1	14	15		
	3137.5	8	8		
	3201.7	12	13		
	3266.5	19	20		
	3331.3	20	21		
	3377.3	17	18		
	3435.9	16	17		
	3500.7	19	20		
	3565.5	20	21		
	3623.8	19	20		
	3706.1	21	22		
	3784.9	24	25		
	3871	25	26		
	3932	22	23		
	3996.8	19	20		
	4058	19	20		
	4122.6	17	18		
	4187.6	10	11		
	4252.2	2	2		
Σ				370	
RANGE SAFETY	2330.3	11	12		
	2424.8	7	7		
	2472.8	5	5		

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TABLE 6 (CONT.)

NLS DISTRIBUTED WEIGHTS W/ PAYLOAD(CRV) - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	2580.5	6	6		
	2644	10	11		
	2711.7	10	11		
	2778.8	10	11		
	2838.4	6	6		
	2852.8	4	4		
	2897.1	7	7		
	2941.4	7	7		
	2985.6	7	7		
	3034.2	7	7		
	3083.3	7	7		
	3123.1	4	4		
	3137.5	6	6		
	3201.7	10	11		
	3266.5	10	11		
	3331.3	8	8		
	3377.3	8	8		
	3435.9	9	9		
	3500.7	10	11		
	3565.5	9	9		
	3623.8	11	12		
	3706.1	12	13		
	3784.9	13	14		
	3871	11	12		
	3932	10	11		
	3996.8	10	11		
	4058	9	9		
	4122.6	5	5		
	4187.6	1	1		
Σ				273	
TOTAL WEIGHT				138213	

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TABLE 6 (CONT.)
DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
PODS					(36528)
STRUCT - BEAMS	4252.2	140	168		
	4272.8	701	842		
	4295.5	1402	1684		
	4325.5	1402	1684		
	4355.5	701	842		
	4385.5	327	393		
Σ				5613	
LONGERONS	4252.2	24	29		
	4272.8	119	143		
	4295.5	238	286		
	4325.5	238	286		
	4355.5	119	143		
	4385.5	56	67		
Σ				952	
COVER	4252.2	48	58		
	4272.8	241	290		
	4295.5	482	579		
	4325.5	482	579		
	4355.5	241	290		
	4385.5	113	135		
Σ				1931	
S/S INSTAL	4295.5	49	59		
	4325.5	244	293		
	4355.5	244	293		
	4385.5	73	88		
Σ				731	
TPS - COVER	4252.2	33	39		
	4272.8	33	39		
	4295.5	33	39		
	4325.5	33	39		
	4355.5	33	39		
	4385.5	33	39		
Σ				236	
CANNISTERS	4295.5	198	237		
	4325.5	395	475		
	4355.5	395	475		
	4385.5	198	237		
Σ				1424	
SEPARATION	4355.5	102	123		
	4385.5	102	123		
Σ				245	
FEED	4252.2	82	99		
	4272.8	412	495		

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TABLE 6 (CONT.)

DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	4295.5	824	990		
	4325.5	824	990		
	4355.5	412	495		
	4385.5	192	231		
Σ				3299	
PNEUMATICS	4252.2	506	607		
	4272.8	506	607		
Σ				1214	
TVC	4295.5	95	115		
	4325.5	477	573		
	4355.5	477	573		
	4385.5	143	172		
Σ				1432	
DE-ORBIT	4355.5	770	925		
	4385.5	770	925		
Σ				1850	
ENGINES	4435	17600	17600		
Σ				17600	
JETTISON					(53907)
STRUCT	4252.2	317	248		
	4272.8	1587	1241		
	4295.5	3175	2483		
	4325.5	3175	2483		
	4355.5	1587	1241		
	4385.5	741	579		
Σ				8276	
S/S INSTAL	4252.2	35	27		
	4272.8	173	135		
	4295.5	346	271		
	4325.5	346	271		
	4355.5	173	135		
	4385.5	81	63		
Σ				902	
TPS - CYL	4252.2	5	4		
	4272.8	26	20		
	4295.5	52	41		
	4325.5	52	41		
	4355.5	26	20		
	4385.5	12	9		
Σ				135	
TPS - BHS	4355.5	295	231		
	4385.5	295	231		
Σ				461	
TPS - CANNISTERS	4435	1976	1545		
Σ				1545	

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DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
SEPARATION	4252.2	102	80		
	4272.8	102	80		
Σ				160	
FEED	4187.6	204	160		
	4252.2	885	692		
	4272.8	1021	799		
	4295.5	1702	1331		
	4325.5	1702	1331		
	4355.5	1021	799		
	4385.5	272	213		
Σ				5324	
PNEUMATIC	4187.6	2	1		
	4252.2	7	5		
	4272.8	8	6		
	4295.5	13	10		
	4325.5	13	10		
	4355.5	8	6		
	4385.5	2	2		
Σ				39	
TVC	4406	2384	1864		
Σ				1864	
ENGINES	4463	35200	35200		
Σ				35200	
AVIONICS					(7334)
JETTISON - BATT	4252.2	231	231		
	4272.8	455	455		
	4295.5	446	446		
	4325.5	465	465		
	4355.5	481	481		
	4385.5	282	282		
Σ				2360	
BATT CNTRL	4325.5	52	52		
	4355.5	108	108		
	4385.5	53	53		
Σ				213	
EIU	4355.5	28	28		
	4385.5	29	29		
Σ				57	
RETAINED - BATT	4252.2	139	139		
	4272.8	243	243		
	4295.5	227	227		
	4325.5	227	227		
	4355.5	229	229		
	4385.5	115	115		
Σ				1180	
RETAINED - BATT	4325.5	27	27		

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DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	4355.5	52	52		
	4385.5	27	27		
Σ				106	
EIU	4295.5	29	29		
Σ				29	
OTHER	4295.5	519	519		
Σ				519	
CONICAL ADAPTER	2330.3	242	242		
	2424.8	604	604		
	2472.8	362	362		
Σ				1208	
OTHER	2424.8	33	33		
	2472.8	45	45		
	2580.5	30	30		
	2644	35	35		
	2711.7	68	68		
	2778.8	68	68		
	2838.4	68	68		
	2852.8	40	40		
	2897.1	26	26		
	2941.4	40	40		
	2985.6	40	40		
	3034.2	43	43		
	3083.3	45	45		
	3123.1	41	41		
	3137.5	25	25		
	3201.7	36	36		
	3266.5	68	68		
	3331.3	68	68		
	3377.3	54	54		
	3435.9	51	51		
	3500.7	56	56		
	3565.5	59	59		
	3623.8	56	56		
	3706.1	73	73		
	3784.9	89	89		
	3871	76	76		
	3932	73	73		
	3996.8	100	100		
	4058	58	58		
	4122.6	56	56		
	4187.6	35	35		
	4252.2	7	7		
Σ				1662	
R&R					(17859)

BPROP1.XLS 9/22/92

TABLE 6 (CONT.)

DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
JETTISON					
LO2-FEED	4252.2	132	115		
	4272.8	231	201		
	4295.5	215	187		
	4325.5	216	188		
	4355.5	110	96		
Σ				786	
LO2-TORUS	4272.8	675	587		
	4295.5	1354	1178		
	4325.5	675	587		
Σ				2352	
LO2-LINES	4272.8	250	218		
	4295.5	376	327		
	4325.5	751	653		
	4355.5	751	653		
	4385.5	376	327		
Σ				2178	
LH2	4252.2	6	5		
	4272.8	30	26		
	4295.5	60	52		
	4325.5	60	52		
	4355.5	30	26		
	4385.5	14	12		
Σ				174	
RETAINED					
LO2-LINES	3435.9	477	466		
	3500.7	499	488		
	3565.5	477	466		
	3623.8	543	531		
	3706.1	621	607		
	3784.9	637	623		
	3871	571	558		
	3932	488	477		
	3996.8	488	477		
	4058	432	422		
	4122.6	249	243		
	4187.6	61	60		
	4252.2	143	140		
	4272.8	267	261		
	4295.5	375	367		
	4325.5	193	189		
Σ				6374	
LO2 - TANK					
	2424.8	660	645		
	2472.8	1270	1241		
	2580.5	910	890		
	2644	490	479		

BPROP1.XLS 9/22/92

TABLE 6 (CONT.)

DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	2711.7	60	59		
Σ				3314	
LH2 - LINE	4252.2	293	286		
	4272.8	579	566		
	4295.5	493	482		
	4325.5	143	140		
Σ				1474	
LH2 - TANK	2985.6	42	41		
	3034.2	43	42		
	3083.3	40	39		
	3123.1	25	24		
	3137.5	36	35		
	3201.7	58	57		
	3266.5	58	57		
	3331.3	49	48		
	3377.3	47	46		
	3435.9	56	55		
	3500.7	58	57		
	3565.5	56	55		
	3623.8	63	62		
	3706.1	73	71		
	3784.9	74	72		
	3871	67	65		
	3932	57	56		
	3996.8	57	56		
	4058	51	50		
	4122.6	30	29		
	4187.6	38	37		
	4252.2	36	35		
	4272.8	20	20		
	4295.5	69	67		
	4325.5	32	31		
Σ				1207	
TOTAL WEIGHT				115628	
PROPELLANT-USABLE					(1894498)
LO2 TK-F DOME	2472.8	10147	11353		
(FIXED WT)	2580.5	89909	100598		
	2644	147435	164964		
Σ				276915	
CYLINDER	2580.5	123355	138021		
	2644	212243	237477		
	2711.7	224941	251684		
	2778.8	211336	236462		
	2838.4	135145	151212		
Σ				1014856	
LO2 TK-A DOME	2852.8	5272	5899		
(FIXED WT)	2897.1	42176	47190		

BPROP1.XLS 9/22/92

TABLE 6 (CONT.)

DISTRIBUTED PROPULSION WEIGHTS - STRETCH "B" VERSION

ITEM	STA	RI REF WT (LBS)	"B" VERSION WT (LBS)	TOTAL WT	
	2941.4	78494	87826		
	2985.6	110126	123219		
	3034.2	56821	63576		
Σ				327711	
LINE	2985.6	231	258		
	3034.2	371	415		
	3083.3	501	561		
	3123.1	463	518		
	3137.5	291	326		
	3201.7	414	463		
	3266.5	667	746		
	3331.3	678	759		
	3377.3	576	644		
	3435.9	544	609		
	3500.7	646	723		
Σ				6022	
LH2 TK-F DOME	3083.3	652	730		
(FIXED WT)	3123.1	5281	5909		
Σ				6638	
CYLINDER	3137.5	8701	9735		
	3201.7	14138	15819		
	3266.5	14138	15819		
	3331.3	12181	13629		
	3377.3	11528	12899		
	3435.9	13486	15089		
	3500.7	14138	15819		
	3565.5	13486	15089		
	3623.8	15443	17279		
	3706.1	17619	19714		
	3784.9	18054	20200		
	3871	16096	18010		
	3932	13703	15332		
	3996.8	13703	15332		
	4058	12398	13872		
	4122.6	7178	8031		
	4187.6	1523	1704		
Σ				243373	
LH2 TK-A DOME	4122.6	373	417		
(FIXED WT)	4187.6	3037	3398		
	4252.2	5650	6322		
	4272.8	7906	8846		
Σ				18983	
TOTAL WEIGHT				2125755	

BPROP1.XLS 9/22/92

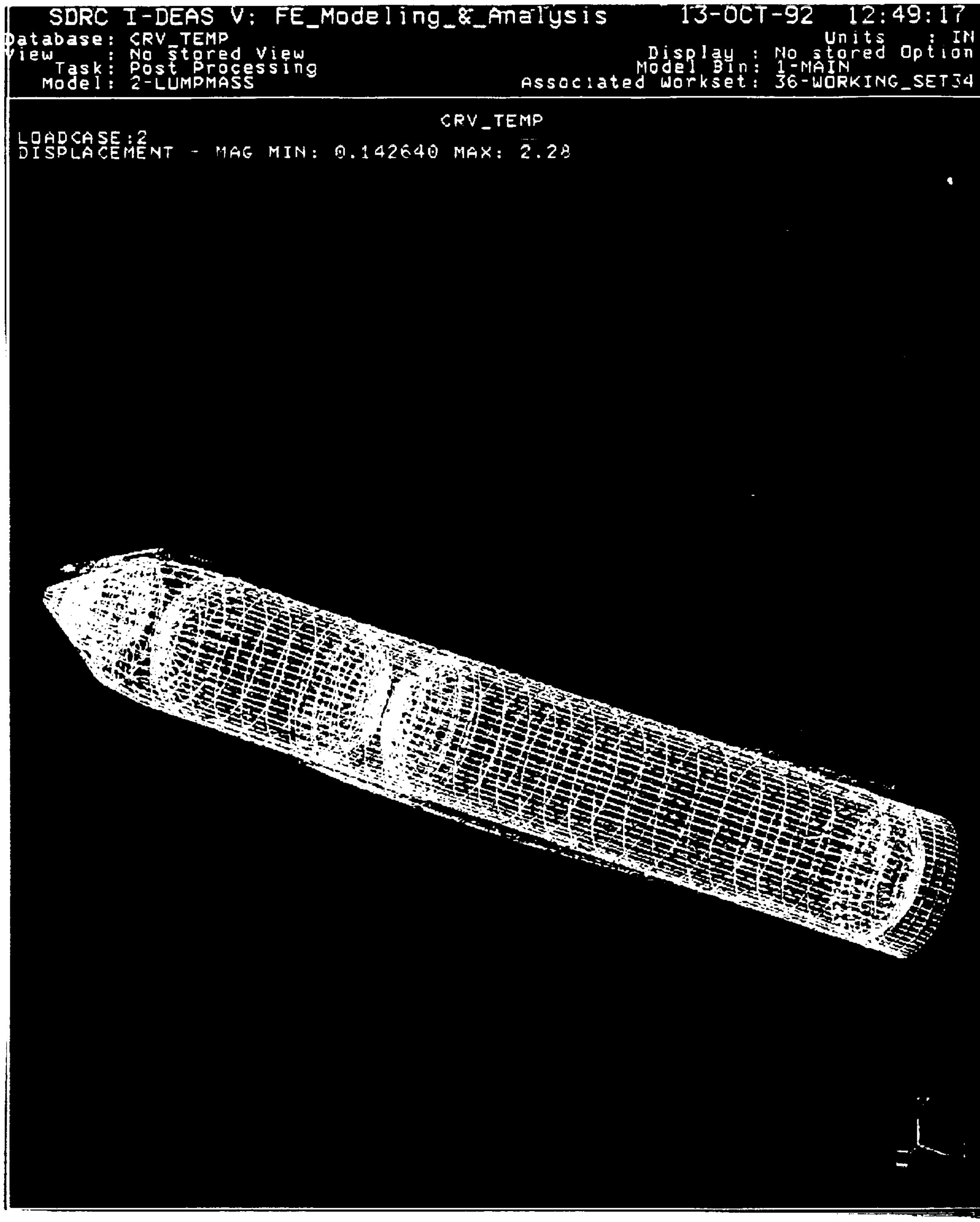


FIGURE 23. DEFLECTIONS (HI-Q)

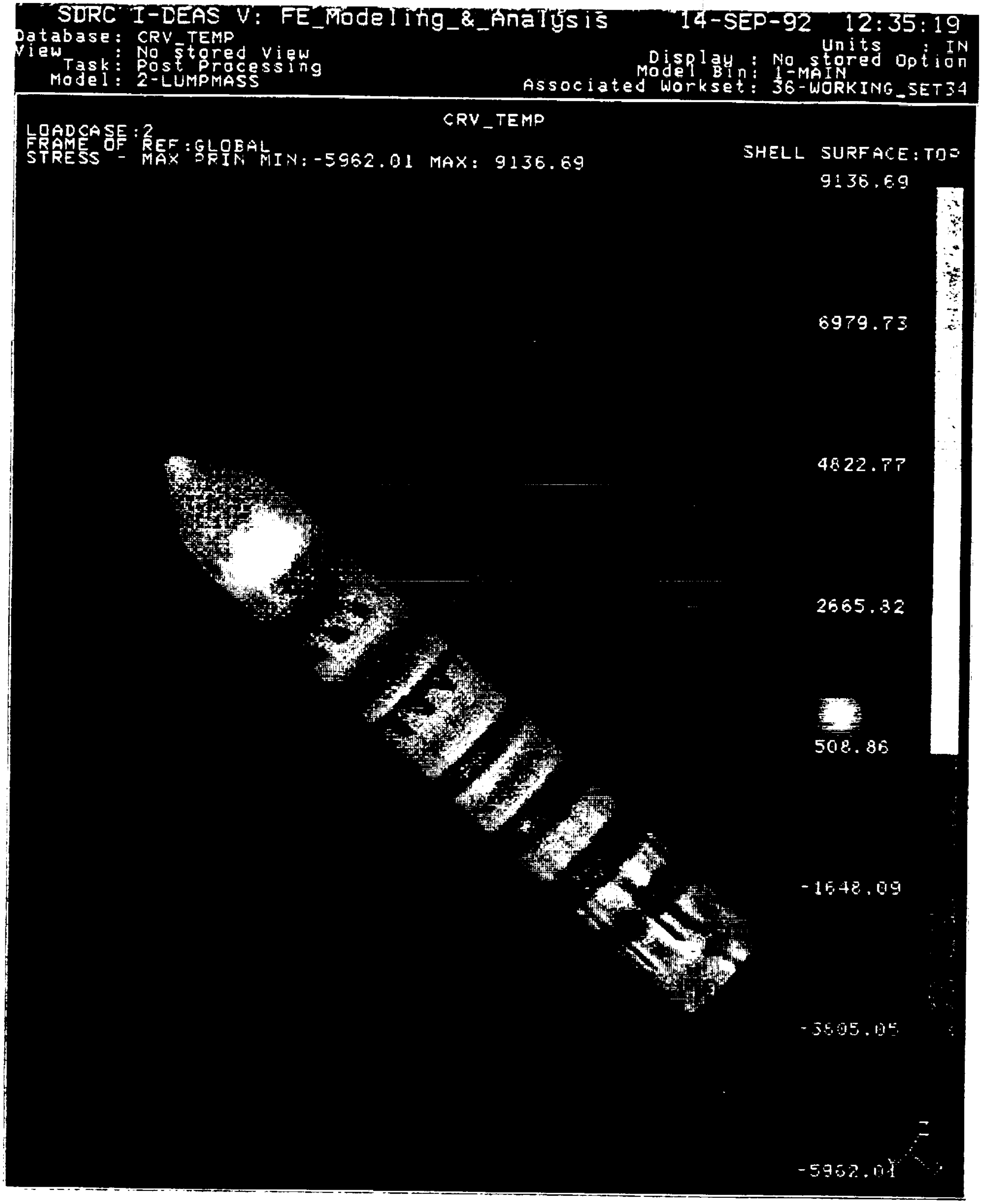


FIGURE 24. OUTER SURFACE STRESSES (HI-Q)

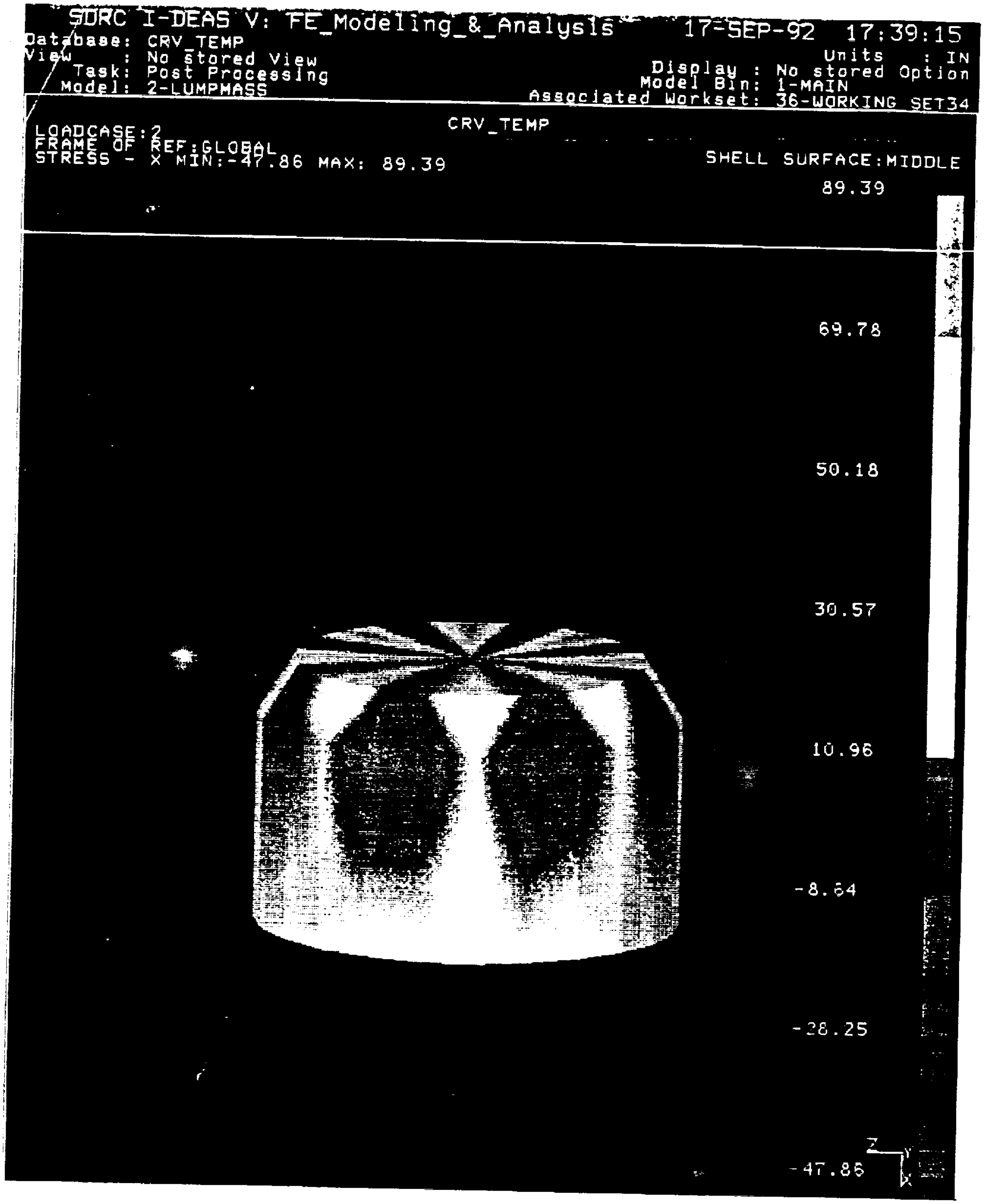


FIGURE 25. LONGITUDINAL SRESSES IN CRV PRESSURE VESSEL (HI-Q, NO INT. PRESS.) MIDDLE SURFACE

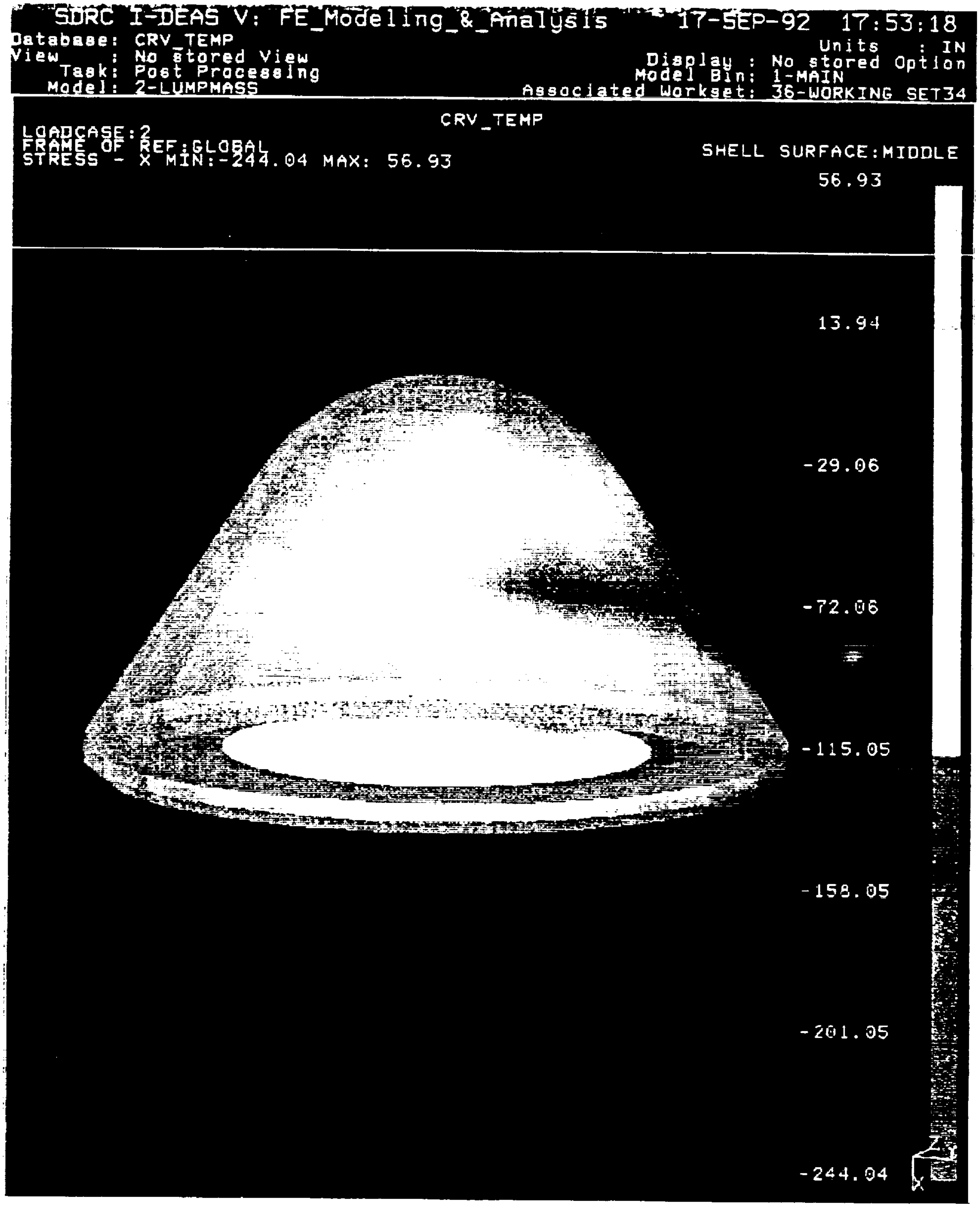


FIGURE 26. LONGITUDINAL STRESSES IN CRV SHELL MIDDLE SURFACE (HIQ)

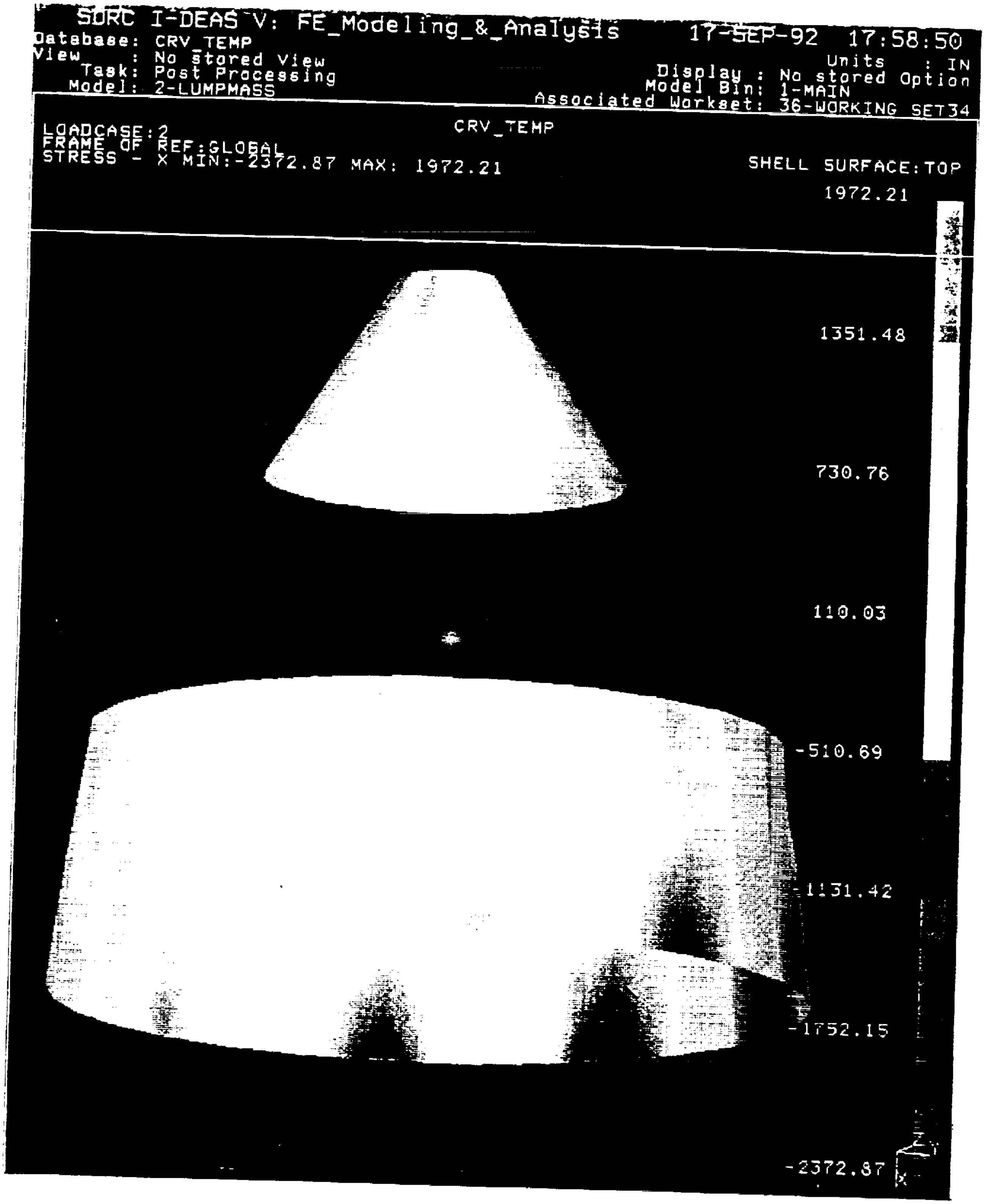


FIGURE 27. LONGITUDINAL STRESSES IN CRV UPPER AND LOWER ADAPTERS (HI-Q)

PREPARED BY: V. RICHARDSON	Rockwell International	PAGE NO.
CHECKED BY:	Space Systems Division	REPORT NO.
DATE: SEPT. 22 1992	NLS / CRV	MODEL NO.
REF.		DWG. NO.

SEGMENT	FEM skin t	FEM stringer w	FEM stringer d	t bar	HI Q σ max	HI Q σ min	ON PAD σ max	ON PAD σ min	tank pres.	σ pres.	σ total max	σ total min	Nx ult max	Nx ult min
	in.	in.	in.	in.	psi	psi	psi	psi	psi	psi	psi	psi	lbs/in	lbs/in
UPPER ADAPTER	2.000	0.00	0.00	2.000	1552.	-1910.	15.	-209.	0.	0.	1552.	-1910.	4346.	-5348.
CRV SHELL	3.000			3.000	65.	-273.	-11.	-82.	0.	0.	65.	-273.	273.	-1147.
LOWER ADAPTER	2.000	0.00	0.00	2.000	51.	-815.	18.	-264.	0.	0.	51.	-815.	143.	-2282.
FWD SKIRT	0.063	2.80	2.30	0.125	1050.	-10903.	-495.	-3691.	0.	0.	1050.	-10903.	184.	-1910.
LOX TANK	0.100	2.20	1.80	0.138	1600.	-22027.	1290.	-13289.	29.	34624.	36224.	-13289.	7009.	-2571.
INTER TANK	0.150	2.30	2.80	0.212	-5900.	-18515.	-4956.	-13176.	0.	0.	-4956.	-18515.	-1472.	-5498.
LH2 TANK	0.100	2.20	1.80	0.138	1000.	-25640.	2781.	-31803.	34.	40594.	43375.	-31803.	8392.	-6153.

SEGMENT	optimized t bar in	M.S. ult. tension	M.S. ult. comp.
UPPER ADAPTER	0.258	2.86	0.07
CRV SHELL	0.107	24.48	0.07
LOWER ADAPTER	0.150	67.28	0.02
FWD SKIRT	0.151	52.36	0.01
LOX TANK	0.151	0.40	0.02
INTER TANK	0.259	n/a	0.01
LH2 TANK	0.230	0.78	0.01

TABLE 7. STRESS ASSESSMENT

V. Richardson PREPARED BY:	Rockwell International	PAGE NO.
CHECKED BY:		REPORT NO.
DATE: Sept. 24, 1992	NLS	MODEL NO.
		DRAWING NO.

NLS CLV/CRV STRESS ANALYSIS

Assumptions: LH2 tank pressure = 34 psi
Lox tank pressure = 29 psi

The objective of the stress analysis is to develop the lowest weight skin/stringer configurations for the different segments using the FEM results. Since the FEM is a rough model with approximate sizing, the stresses may be unreasonable. Therefore, the following procedure was used. Both the 'On Pad' and 'Hi Q' conditions were reviewed. The max and min longitudinal skin stresses were combined with the effective skin thickness (t bar) of the FEM to determine the internal loading of each segment, Nx. Stresses due to tankage internal pressure were added when appropriate (no internal pressure for 'On Pad' condition). The loading and geometry was then used in a skin/stringer optimization program called BSOP. BSOP develops the lowest weight configuration that will sustain the loading based on the following analysis; strength, Euler stability, short column buckling, blade crippling, skin buckling and overall panel stability. Once skin/stringer configuring is complete the frames can be sized.

LH2 CRV

SECTION NUMBER: 26

WEIGHT	SKIN THICK.	HEIGHT	SPACING	BLADE THICK.
3.279	.105	2.05	3.70	.225

ANALYSIS RESULTS FROM LOAD CASE NUMBER 1

	SX SKIN	SY SKIN	SXY SKIN	SX BLADE
STRESSES DUE TO AXIAL LOADS:	-27557.7	.0	.0	-27557.7
STRESSES DUE TO PRESSURE:	.0			.0
COMBINED STRESSES:	-27557.7			-27557.7

CHECK	MARGINS OF SAFETY
FCYX SKIN:	.161
FCYX BLADE:	.161
FCYY SKIN:	.000
FSU SKIN:	.000
EULER STABILITY:	.509
SHORT COLUMN:	.010
BLADE STABILITY:	.337
~~SKIN STABILITY~~:	.133
PANEL STABILITY:	1.264
PANEL DEFLECTION:	.000

$\bar{t} = .230$

LO2 CRV

SECTION NUMBER: 47

WEIGHT	SKIN THICK.	HEIGHT	SPACING	BLADE THICK.
2.176	.069	1.80	3.24	.148

ANALYSIS RESULTS FROM LOAD CASE NUMBER 1

	SX SKIN	SY SKIN	SXY SKIN	SX BLADE
STRESSES DUE TO AXIAL LOADS:	-17356.2	.0	.0	-17356.2
STRESSES DUE TO PRESSURE:	.0			.0
COMBINED STRESSES:	-17356.2			-17356.2

CHECK	MARGINS OF SAFETY
FCYX SKIN:	.844
FCYX BLADE:	.844
FCYY SKIN:	.000
FSU SKIN:	.000
EULER STABILITY:	.095
SHORT COLUMN:	.024
BLADE STABILITY:	.159
SKIN STABILITY:	.015
PANEL STABILITY:	.643
PANEL DEFLECTION:	.000

$\bar{t} = .151$

INTERTANK CRV

SECTION NUMBER: 10

WEIGHT	SKIN THICK.	HEIGHT	SPACING	BLADE THICK.
3.728	.078	2.50	3.24	.235

ANALYSIS RESULTS FROM LOAD CASE NUMBER 1

	SX SKIN	SY SKIN	SXY SKIN	SX BLADE
STRESSES DUE TO AXIAL LOADS:	-21662.3	.0	.0	-21662.3
STRESSES DUE TO PRESSURE:	.0			.0
COMBINED STRESSES:	-21662.3			-21662.3

CHECK	MARGINS OF SAFETY
FCYX SKIN:	.477
FCYX BLADE:	.477
FCYY SKIN:	.000
FSU SKIN:	.000
EULER STABILITY:	.057
SHORT COLUMN:	.006
BLADE STABILITY:	.196
SKIN STABILITY:	.039
PANEL STABILITY:	.586
PANEL DEFLECTION:	.000

$\bar{t} = .259$

FWD. SKIRT

SECTION NUMBER: 33

WEIGHT	SKIN THICK.	HEIGHT	SPACING	BLADE THICK.
2.177	.060	2.10	3.24	.140

ANALYSIS RESULTS FROM LOAD CASE NUMBER 1

	SX SKIN	SY SKIN	SXY SKIN	SX BLADE
STRESSES DUE TO AXIAL LOADS:	-12886.6	.0	.0	-12886.6
STRESSES DUE TO PRESSURE:	.0			.0
COMBINED STRESSES:	-12886.6			-12886.6

CHECK	MARGINS OF SAFETY
FCYX SKIN:	1.483
FCYX BLADE:	1.483
FCYY SKIN:	.000
FSU SKIN:	.000
EULER STABILITY:	.099
SHORT COLUMN:	.090
BLADE STABILITY:	.006
SKIN STABILITY:	.033
PANEL STABILITY:	.648
PANEL DEFLECTION:	.000

$\bar{t} = .151$

FORM 994-B-1 REV 12-84

LOWER ADAPTER

SECTION NUMBER: 36

WEIGHT	SKIN THICK.	HEIGHT	SPACING	BLADE THICK.
2.158	.078	1.90	3.70	.140

ANALYSIS RESULTS FROM LOAD CASE NUMBER 1

	SX SKIN	SY SKIN	SXY SKIN	SX BLADE
STRESSES DUE TO AXIAL LOADS:	-15530.1	.0	.0	-15530.1
STRESSES DUE TO PRESSURE:	.0			.0
COMBINED STRESSES:	-15530.1			-15530.1

CHECK	MARGINS OF SAFETY
FCYX SKIN:	1.061
FCYX BLADE:	1.061
FCYY SKIN:	.000
FSU SKIN:	.000
EULER STABILITY:	.093
SHORT COLUMN:	.023
BLADE STABILITY:	.046
SKIN STABILITY:	.109
PANEL STABILITY:	.640
PANEL DEFLECTION:	.000

$\bar{t} = .150$

CRV SHELL

SECTION NUMBER: 30

WEIGHT	SKIN THICK.	HEIGHT	SPACING	BLADE THICK.
1.552	.060	1.60	3.37	.100

ANALYSIS RESULTS FROM LOAD CASE NUMBER 1

	SX SKIN	SY SKIN	SXY SKIN	SX BLADE
STRESSES DUE TO AXIAL LOADS:	-10853.7	.0	.0	-10853.7
STRESSES DUE TO PRESSURE:	.0			.0
COMBINED STRESSES:	-10853.7			-10853.7

CHECK	MARGINS OF SAFETY
FCYX SKIN:	1.948
FCYX BLADE:	1.948
FCYY SKIN:	.000
FSU SKIN:	.000
EULER STABILITY:	.104
SHORT COLUMN:	.093
BLADE STABILITY:	.069
SKIN STABILITY:	.131
PANEL STABILITY:	.657
PANEL DEFLECTION:	.000

$\bar{t} = .107$

FORM 904-B-1 REV 12-84

PREPARED BY:

CHECKED BY:

DATE:

REF.

Rockwell International

NLS / CRV

PAGE NO.

REPORT NO.

MODEL NO.

DWG. NO.

UPPER ADAPTER

SECTION NUMBER: 34

WEIGHT	SKIN THICK.	HEIGHT	SPACING	BLADE THICK.
3.704	.100	2.70	4.10	.240

ANALYSIS RESULTS FROM LOAD CASE NUMBER 1

	SX SKIN	SY SKIN	SXY SKIN	SX BLADE
STRESSES DUE TO AXIAL LOADS:	-21205.8	.0	.0	-21205.8
STRESSES DUE TO PRESSURE:	.0			.0
COMBINED STRESSES:	-21205.8			-21205.8

CHECK	MARGINS OF SAFETY
FCYX SKIN:	.509
FCYX BLADE:	.509
FCYY SKIN:	.000
FSU SKIN:	.000
EULER STABILITY:	.283
SHORT COLUMN:	.073
BLADE STABILITY:	.106
SKIN STABILITY:	.087
PANEL STABILITY:	.926
PANEL DEFLECTION:	.000

$\bar{t} = .258$

FORM 994-B-1 REV 12-84

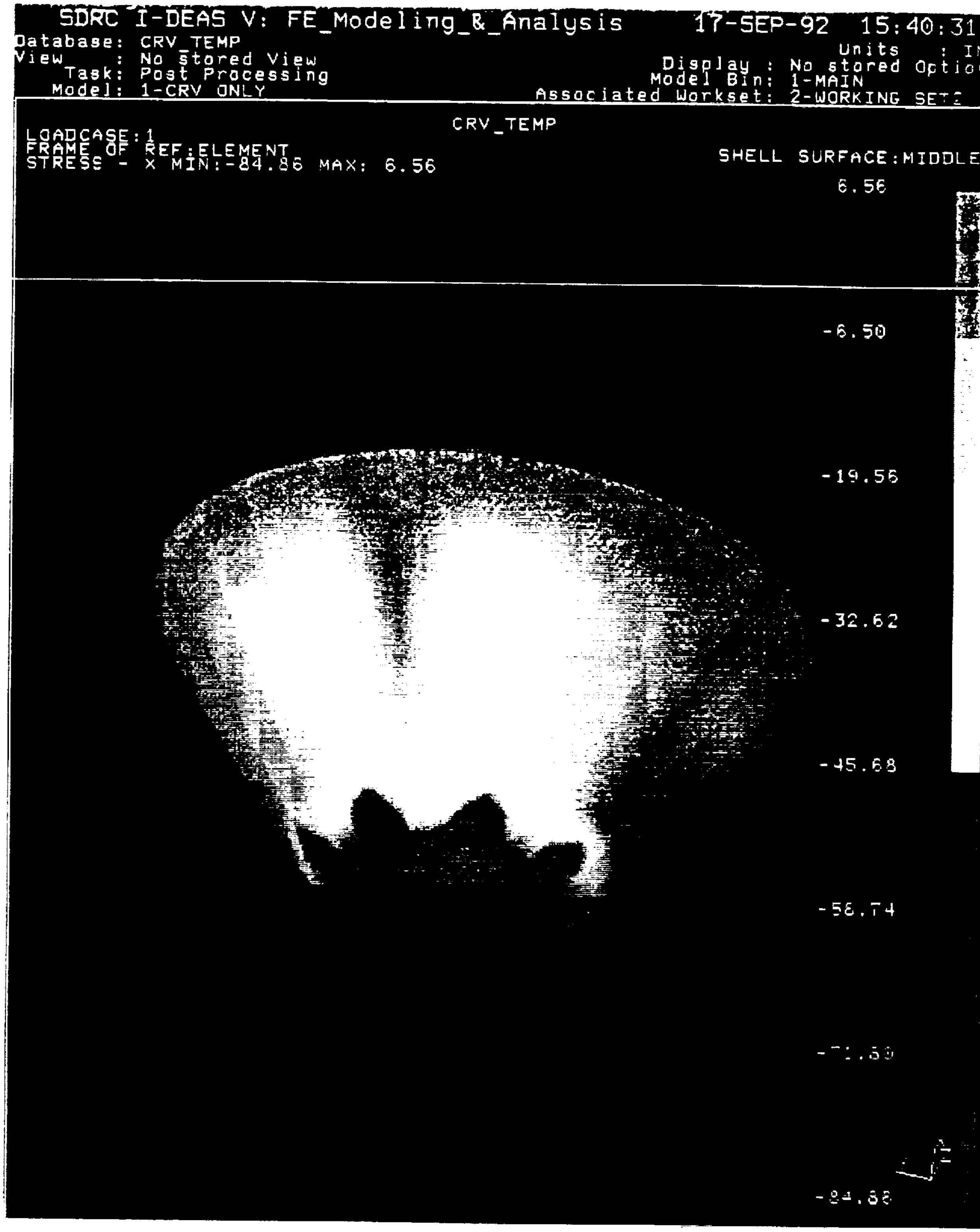


FIGURE 28. LONGITUDINAL STRESS IN CRV SHELL (RE-ENTRY)

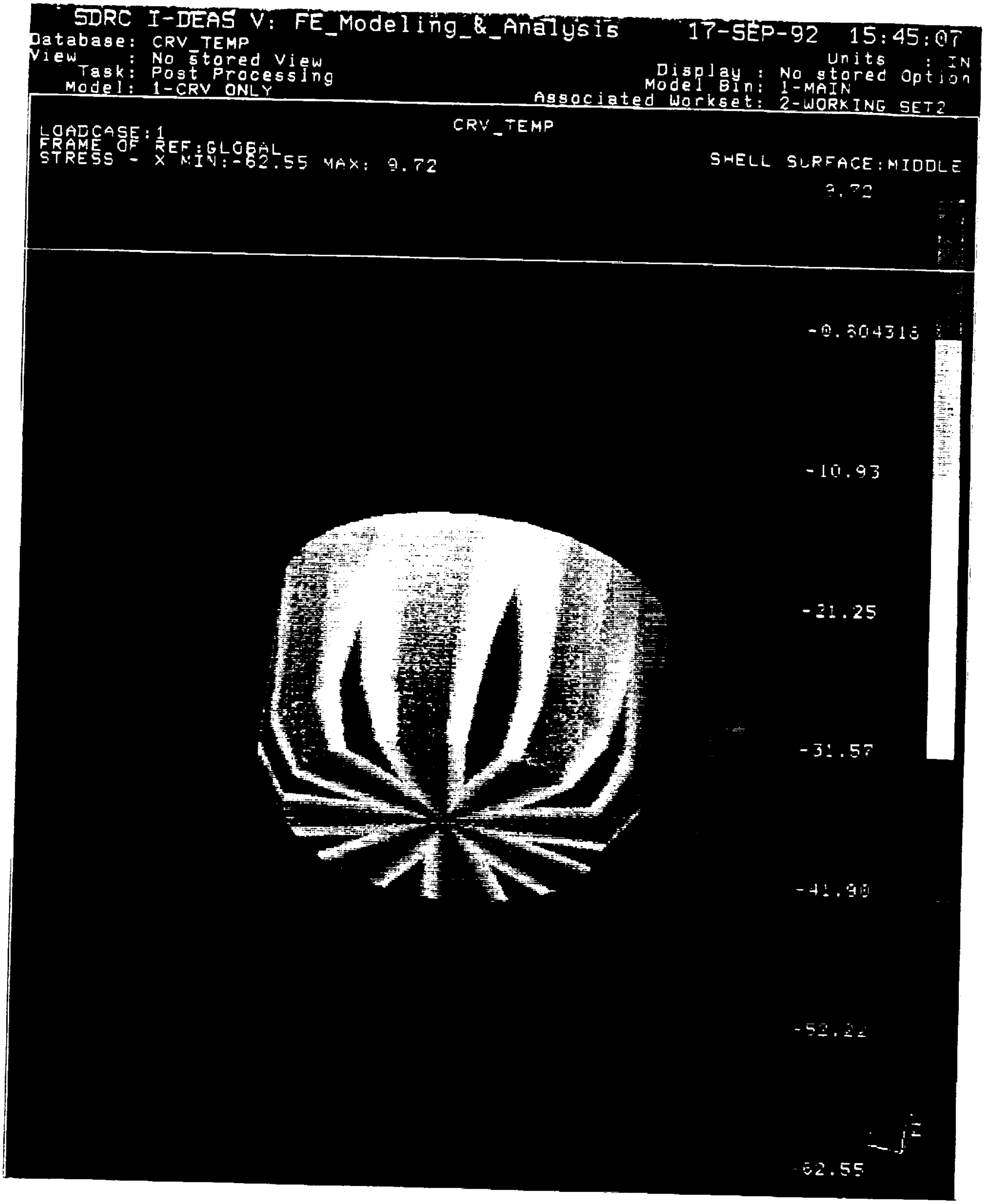


FIGURE 29. LONGITUDINAL STRESSES IN CRV PRESSURE VESSEL (RE-ENTRY)

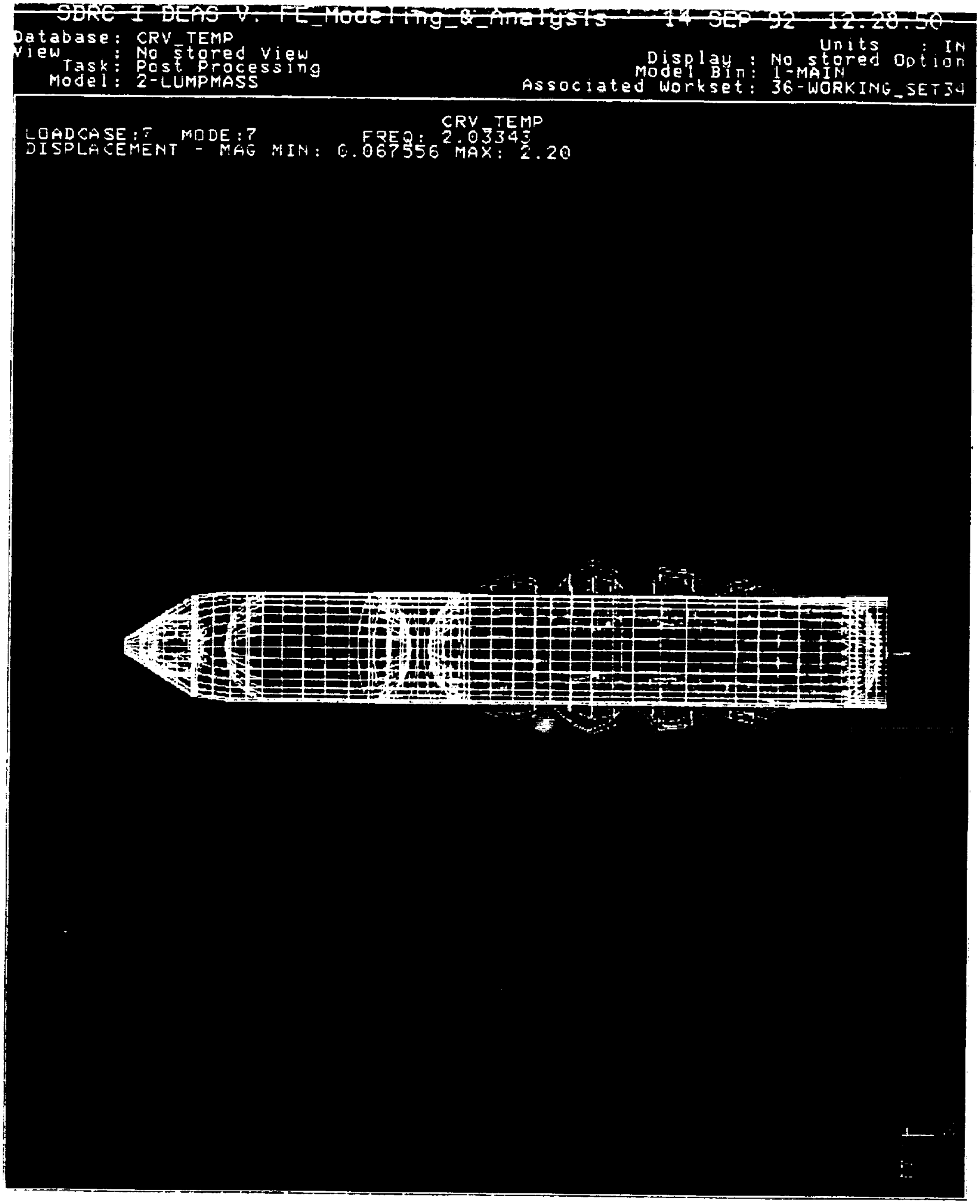


FIGURE 30. FIRST MODE (H_2 TANK SHELL MODE)

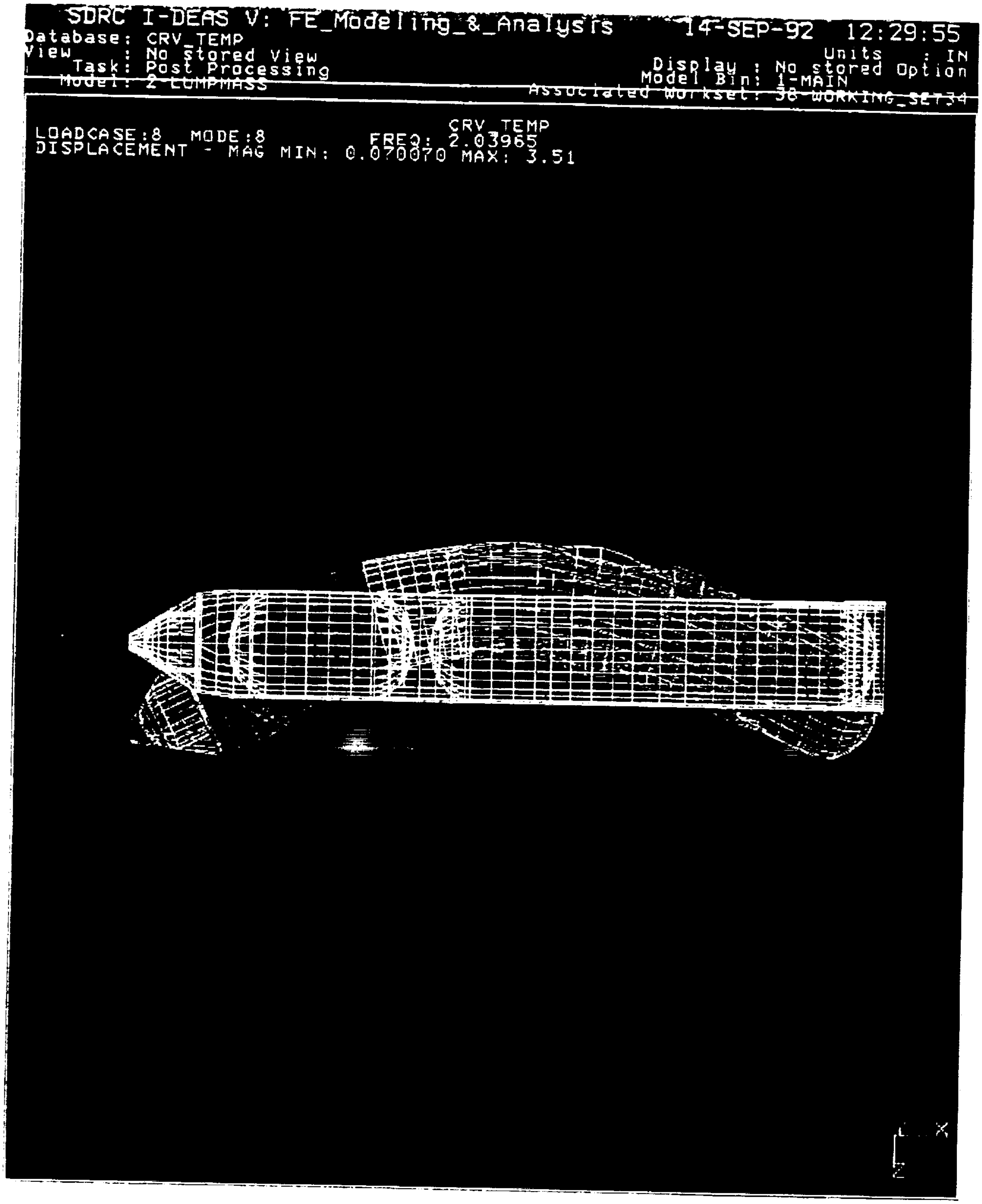


FIGURE 31. SECOND MODE (VEHICLE BENDING)

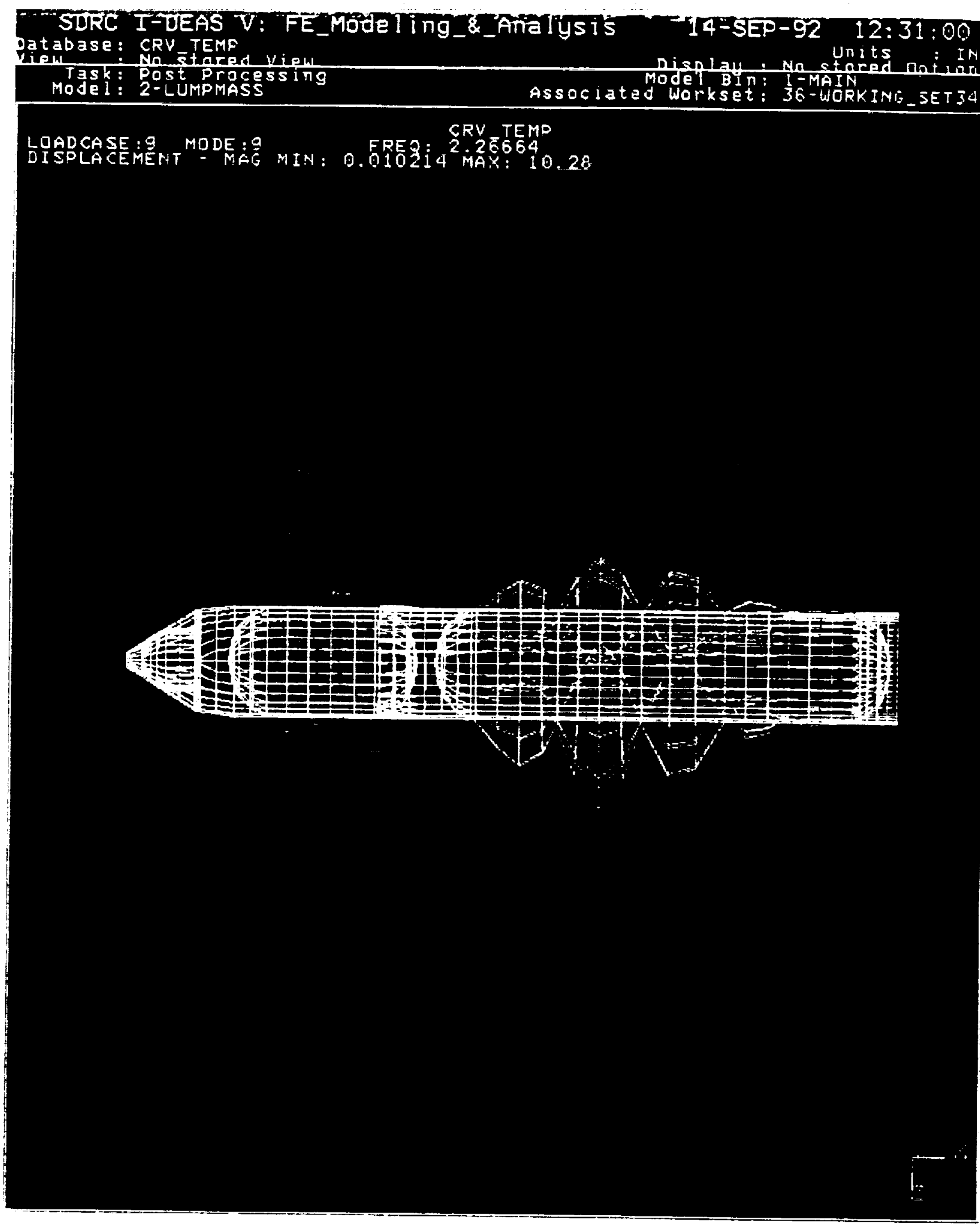


FIGURE 32. THIRD MODE (LOX AND H_2 TANK SHELL MODES)

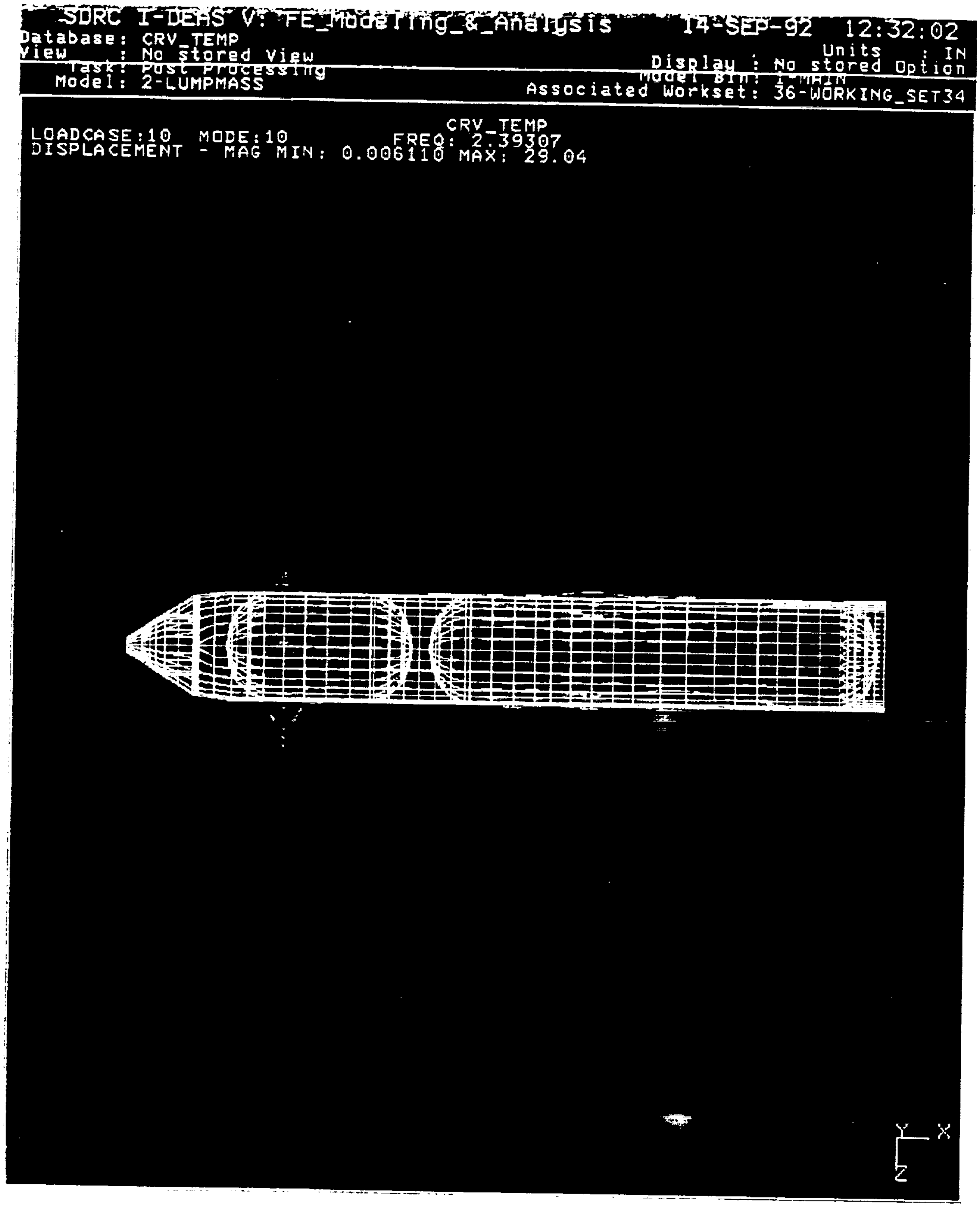


FIGURE 33. FOURTH MODE (LOX TANK SHELL MODE

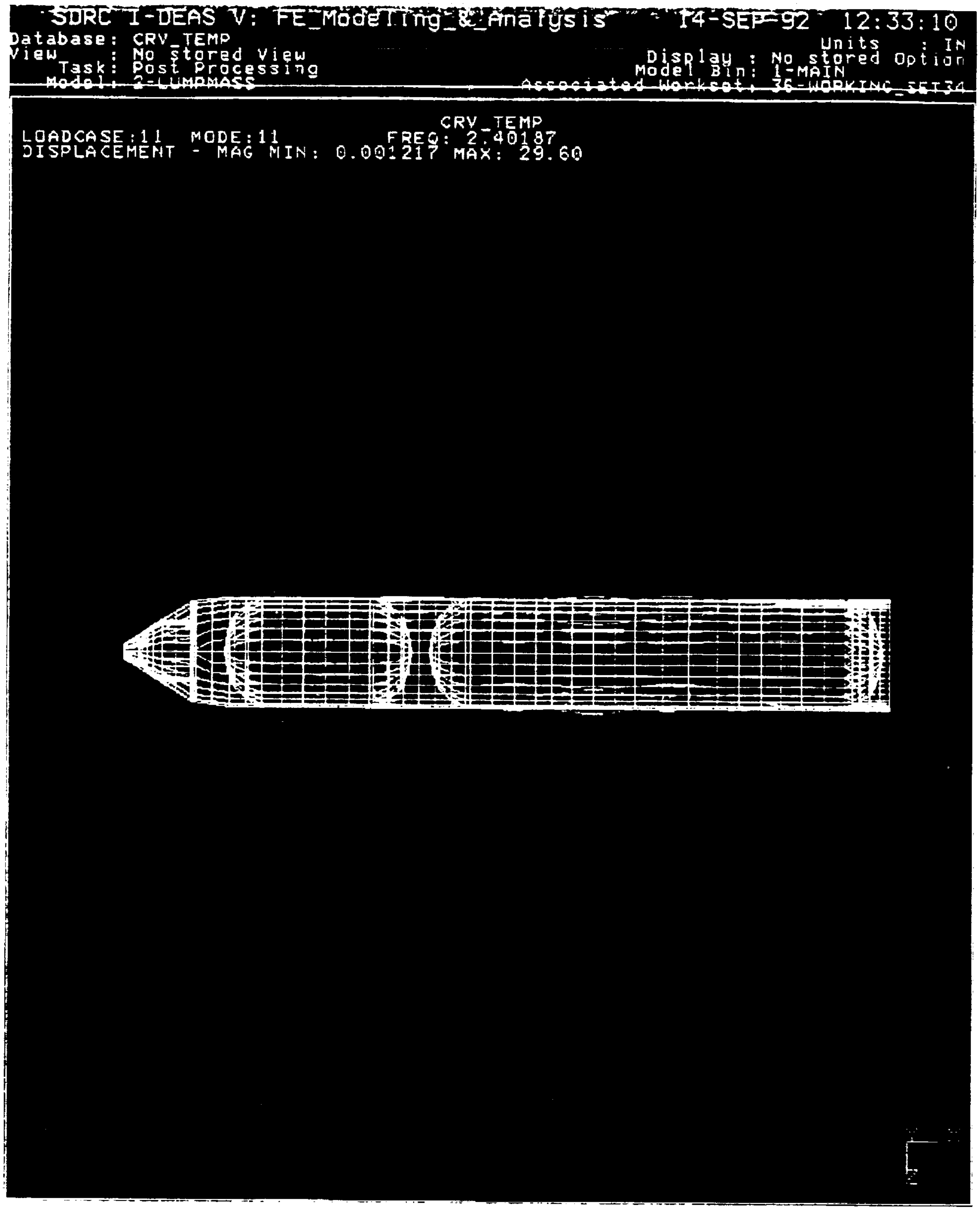


FIGURE 34. FIFTH MODE (LOX AND H_2 TANK SHELL MODES)

Table 8. Summary of CRV Modes

Mode	Freq (Hz)	Description
1	2.03	H_2 Tank Shell Mode
2	2.04	X-Y Bending
3	2.27	Shell Mode - LO_2, LH_2
4	2.39	Shell Mode - LO_2
5	2.40	Shell Mode - LO_2, LH_2

Table 9. Dynamic Characteristics (ET vs CRV)

Mode	CRV Freq (Hz)	ET Freq (Hz)	ET Mode Description
1	2.03	4.43	Body Bending
2	2.04	4.56	Body Bending
3	2.27	4.93	Shell Mode
4	2.39	5.02	Shell Mode
5	2.40	5.73	Shell Mode

REFERENCES

1. IL SAS/CFD/92-003, "Normal Force Distribution and Integrated Moments on the National Launch System (NLS) Stage 1.5 Configuration Due to Ground Winds and In-Flight maximum Dynamic Pressure Condition - NLS Tank Stretch IR&D Study", February 26, 1992

2. MSFC Letter ED35-114-91, "NLS On-Pad Aerodynamic Data Base", September 12, 1991

3. Rockwell International IL SAS/NB/91-080, "Updated On-Pad Aerodynamic Analysis Tasks (3-FM-001) for the National Launch System (NLS) Configurations", September 5, 1991

4. NASA TN D-3283, "An Empirical Method for Determining Static Distributed Aerodynamic Loads On Axisymmetric Multistage Launch Vehicles", March 1966

5. MSFC Letter ED35-88-91, "Distributed Loads on The National Launch System 1 1/2 Stage Launch vehicle", September 18, 1991

6. MSFC Letter ES44-(89-90), " Statistical Parameters That Envelope The KSC Monthly Wind Ellipses For Shuttle-C", May 2, 1990

ADVANCED TRANSPORTATION SYSTEM STUDY

Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO

LAUNCH VEHICLE MAN-RATING REQUIREMENTS

Revision A

September, 1992

Contract NAS8-39207

Rockwell International

Space Systems Division
Huntsville Operations
555 Discovery Drive
Huntsville, AL 35806

FORWARD

This document was prepared under Contract NAS8-39207, Advanced Transportation System Studies for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center by Rockwell International. The document identifies the requirements for man-rating of launch vehicles as defined in several referenced documents. The document will be updated periodically during the contract to incorporate additional references and further analysis of man-rating requirements. The document will be included in the Final Report (DR-4) at the completion of the contract.

Revision A

Revision A of this document adds Appendix A, which contains the functional flow block diagrams for the man-rating functional requirements.

TABLE OF CONTENTS

INTRODUCTION

American Astronauts have piloted a number of vehicles into space. The first of these vehicles were essentially rocket-propelled airplanes and converted ICBM missiles. The development of the Apollo/Saturn launch represented the first launch vehicle designed from inception to a set of man-rating requirements. The definition of man-rating evolved with the succession of manned launch vehicles, but was generally constrained by the specific requirements and vehicle configurations of each program. The Space Shuttle program now provides an extensive source of man-rating requirements. Application of these Space Shuttle requirements to other programs, however, would not be straight-forward because of the likely different missions and subsystems which would be found. The Space Shuttle requirements also would not provide a sufficient grasp of the basic objectives of man-rating, which have evolved over several programs and are now expressed in great detail only in the context of the Shuttle program. Designers of new manned launch vehicles have new missions and a freedom of design that stretches beyond the requirements found in the Shuttle program.

The objective of this document is to identify a set of man-rating requirements which represent both the spirit and the letter of man-rating requirements from past and present programs. It is further desired to isolate those requirements which result in launch system hardware or software functional requirements. By isolating this type of requirement from programmatic or design guidelines, it is possible to identify subsystem changes or additions which may be required for new manned launch systems.

The approach used in this document is to treat the most recent man-rating requirements document from the NASA Johnson Space Center (Reference 1, JSC 23211, November 1991) as the baseline and foundation of the effort. Requirements from other man-rating documents will be interleaved with the requirements of this reference, maintaining the outline of the reference document. The result is a grouping of man-rating requirements from a variety of sources/programs in the topical outline of the most current man-rating requirements document. This format permits direct comparison of man-rating requirements from current and past programs. Where current man-rating requirements may be stated in general terms, the comparison with past programs should serve as an interpretation or specific application of the requirement.

The functional flow block diagrams for each of the man-rating functional requirements are provided in Appendix A. Each diagram identifies the specific requirement from which it was derived. External interfaces, including data requirements and external functions, are shown for the man-rating functional flow diagrams.

Future additions to this document will organize all of the identified functional requirements by generic subsystems requirements. These addition will permit direct comparison of subsystem functions from an existing or planned launch system with the requirements for man-rating.

<u>Use of this Document</u>

The man-rating requirements are organized by the categories used in Reference (1) and appear in the same order. Requirements are segregated as either *Guidelines* or *Functional Requirements* within each topic. Within any specific category, requirements from all sources are provided and identified by the reference document from which they were obtained. Each requirement within a topic appears in ascending order per the reference document number (see References section of this document). Note that some of the requirements are stated as suggestions or recommendations in many of the reference documents. These are considered as requirements for the purposes of this document.

The functional flow block diagrams are keyed to the source requirements by a box in the top of each page. The diagram number is shown first, e.g., (MR_1.4_1). The source requirements are then listed, with the requirement paragraph number and the requirement statement printed.

Man-Rating Requirements

BASELINE DOCUMENT

1. DESIGN CRITERIA

Guidelines:

The responsible design element shall establish and maintain an effective strength analysis, structural test and structural assessment program to assess and verify the structural integrity of space vehicle structural and propulsion systems.(6)

Design criteria shall be furnished by the procuring activity. Criteria originated by the responsible design element shall be approved by MSFC prior to use.(6)

Functional Requirements:

None

1.1 Environmental Conditions

Guidelines:

Operational environmental conditions (e.g. temperature, atmosphere, radiation, g-forces), under normal and emergency modes, should be well within known human limits.(1)

...consider alternate missions ...instead of destroying the entire launch vehicle and aborting the crew.(2)

An alternate-mission plan must be planned in case problems occur during launch. ...The alternate mission that will satisfy the greatest number of flight objectives within the capability of the flight constraints must be chosen.(2)

Flight path constraints are placed on the mission because of the data-collection and voice communications requirements... This requires that telemetry contact with the vehicle be maintained.(2)

Since some malfunction detection systems (MDS) require continuous ground telemetry data or commands, use of such systems could also result in flight path constraints...(2)

Abort data link transmissions may modify the direction of the launch. Some systems may have only a narrow corridor where safe abort is possible ... as a result of forces on the crew during return.(2)

Surface temperatures within habitable volumes shall be designed to be less than 45°C (113°F) and greater than 4°C(41.2°F).(3)

The sound pressure levels, shall be measured at, or translated to, the external auditory meatus of the operating personnel. Exposures shall not exceed the allowable levels, periods, or repetition rates.(3)

Internally Generated Radiation: Sources emitting electromagnetic wavelengths between x-ray and visible light, sufficiently strong to pose a hazard, shall be positioned to preclude crew exposure or touch. Hazard distance shall be marked.(3)

Acoustic Noise Criteria: Acoustic noise sound pressure levels establish the maximum acceptable flight crew environmental limits which would result from operation of the total spacecraft system and from the expected environment during all phases of a mission.(3)

Functional Requirements:

None

1.2 Escape System

Guidelines:

Provisions should be made for a viable means of escape from the system in the event of an impending catastrophe. In situations where immediate and complete escape may not be feasible, an alternative approach can be considered if it can be proven reliable.(1)

The flight crew must be provided with a viable means of escape from the space vehicle in the event of an impending catastrophe. In situations where immediate escape from the space system may not be feasible, alternate approaches such as a safe haven may be considered if such can be provided to be viable.(4)

Functional Requirements:

Crew to be provided abort sensing and implementation data upon which to base an abort decision.(1) Emergency considerations have become a distinguishing characteristic of man-rated systems; the resulting vehicle system changes and additions provide for emergency detection, control, and/or escape of the crew.(2) The provision for a safe landing area for the spacecraft and the surrounding facilities is required.(2) An escape system has the attendant requirement for providing the crew with abort sensing and implementation data upon which to base an abort decision.(4)

1.3 Failure Tolerance

Guidelines:

Primary structure, pressure vessels and thermal protection systems should be designed for minimum tolerance for any failure or malfunction that would jeopardize the safety of the crew. When redundancy is unfeasible, designs should be based on conservative criteria. Positive margins should be demonstrated by means of appropriate test programs.(1)

Failure tolerance is the basic safety requirement that shall be used to control most payload hazards. A hazard which is critical can result in damage to launch system equipment, a nondisabling personnel injury or the use of unscheduled safing procedures that affect operations of the launch system or another payload. A hazard which is catastrophic can result in the potential for a disabling or fatal personnel injury, loss of the launch vehicle, ground facilities or launch system equipment.(5)

Fracture Control and Fracture Mechanics Analysis: All flight structures shall be examined to determine their fracture criticality and associated fracture control requirements.(6)

Fracture Mechanics Analysis and Proof Testing: Fracture mechanics analysis shall be performed on all fracture critical parts to demonstrate that the maximum size flaw or crack-like defect that could exist after proof testing and nondestructive evaluation (NDE) inspection will not grow to critical size and cause premature failure during the required service life. Proof testing, supplemented by NDE, shall be the preferred method for establishing the maximum size flaw or crack-like defect to be used in the service life analysis. All load sources and environments shall be considered in determining the appropriate loading spectrums for life analyses and proper application of flaw growth data. (6)

Fatigue Analysis: A detailed design life cycle history shall be developed in sufficient detail that a cumulative damage assessment can be analytically verified for all applicable components. The applicable loads and associated histories shall be determined and the indicated factors applied for the creep life analysis.(6)

Pressure vessels and rotating machinery shall be considered fracture critical and therefore subject to fracture control. Other flight structures with failures modes that could cause loss of vehicle or crew shall be considered fracture critical candidates and undergo a fracture mechanics evaluation. All fracture critical parts shall have a fracture control plan establishing responsibilities, criteria, and procedures for the prevention of structural failures associated with the initiation and propagation of flaws or crack-like defects during fabrication, testing, handling and transportation, and operational life.(6)

Current state-of-the-art NDE inspection techniques shall be utilized. The best current state-of-the-art fracture mechanics analytical techniques shall be utilized. The fracture mechanics analyses shall demonstrate a calculated life of 4.0 times the required service life. Stress concentration factors shall be included in the mean and cyclic stresses. The proof test factor shall be the larger of the values determined by fracture mechanics analysis/proof test requirements. Proof testing shall be performed in the actual expected environment (temperature and media) when feasible. In no case shall the adjusted proof test factor be less than 1.05 without MSFC approval.(6)

All structural elements shall be designed and analyzed to demonstrate the follows factors: 1) The limit stress/strain shall be multiplied by a minimum factor of 1.15 prior to entering the S-N curve to determine the low cycle/high cycle life, 2) the low cycle/high cycle fatigue analysis shall demonstrate a minimum calculated life of 4.0 times the required service life, and 3) the alternating and mean stress/strain shall include the effects of stress concentration factors when applicable.(6)

Functional Requirements:

All critical crew/system functions shall be designed to be two-failure tolerant as a minimum. Functions essential for critical mission support shall be designed to be single failure tolerant as a minimum. Noncritical functions shall be designed to fail in safe mode. Total functional failure of a subsystem should not be allowed to propagate to other systems.(1)

No single failure shall result in a critical hazard. No two failure shall result in a catastrophic hazard.(4)

The payload must tolerate a minimum number of credible failures and/or operator errors determined by the hazard level.

a) Critical hazards shall be controlled such that no single failure or operator error can result in damage to launch system equipment, a nondisabling personnel injury or the use of unscheduled safing procedures that affect operations of the launch system or another payload.
b) Catastrophic hazards shall be controlled such that no combination of two failures or operator errors can result in the potential for a disabling or fatal personnel injury, loss of the launch vehicle, ground facilities or launch system equipment.(5)

In case of failure of an essential function for the crew, the system must still be in a safe condition, i.e., the system must be at least fail-safe. For safety essential functions, redundancies or back-up features have to be incorporated into the design without considering any failure probabilities.(7)

1.4 Hazard Detection and Safing

Guidelines:

The fast response of the automatic system is utilized by setting discrete sensors to respond only to ultimate indicators of an imminent catastrophe.(2)

Manual systems are believed to prevent some false aborts; however, the system must be designed to preclude failures which would not allow for pilot reaction time.(2)

Abort parameter selection and display is based on the design failure philosophy and type of abort initiation (i.e., manual or automatic).(2)

The trajectory information with plots of velocity, acceleration, altitude, and dynamic pressure versus time are not all inclusive of various manned systems.(2)

All manned booster emergency detection systems have included electrical voltage and attitude rate sensors, and, generally, the display of all abort parameters has been by panel light.(2)

In review, the following parameters and signals are common to all systems:
a. Pitch, yaw, and roll rates.
b. electrical voltage.
c. Abort and destruct signals.
d. Staging command and separation signals (in staged vehicles).(2)

Protection of Electrical/Mechanical Systems from Debris: Malfunction of inadvertent operation of spacecraft electrical, electronic, or mechanical equipment caused by exposure to conducting or nonconducting debris or foreign material floating in a gravity-free state shall be prevented.(3)

Shatterable Material: Material that can shatter shall not be used in the habitable compartment unless positive protection is provided to prevent fragments from entering the cabin environment.(3)

Lightning Protection: Lightning protection shall be designed into spacecraft which will operate in earth's atmosphere such that, in the event of a lightning strike, flight hardware will not be damaged or affected to the extent that mission success or crew safety is compromised.(3)

The need for hazard detection and safing by the crew to control time-critical hazards will be minimized and implemented only when an alternate means of reduction or control of hazardous conditions is not available.(5)

A hazard status list shall be prepared and continuously updated.(7)

A malfunction detection system shall incorporate an airborne sensing system and both ground and crew monitoring.(8)

A manual switchover from a primary to a redundant system shall not occur until a malfunction has been detected by one data source and verified by a second independent data source.(8)

Functional Requirements:

A caution and warning system should be provided which will identify equipment failures, fire, or other potential emergency situations. This does not rule out an automatic hazard safing systems for instances of slow crew response time.(1)

The space system should provide a fault detection, isolation and recovery system addressing problems in critical and non-critical systems over which the crew has controls. The status of critical systems shall be displayed in a manner that prevents misinterpretation. Fire suppression capability should be provided in local and general areas, and may be either automatic or manual depending on the type of risk.(1)

It is the function of the emergency detection system to monitor predetermined launch vehicle parameters and supply a signal to selected crew warning displays and the abort initiation system when parametric limits are exceeded.(2)

Malfunction detection to survey, alert, and provide the signal for emergency action.(2)

The pilot or controller should have sufficient displays ... to separate actual malfunctions from false malfunctions.(2)

In all cases... the destruct command signal is displayed to the crew as a light. (2)

Abort, staging command, and stage separation signals are common to all EDS's.(2)

After selection of the sensor range and switching events, adequate provisions for sensor failure must be made.(2)

Provision for safe control and/or shutdown of critical elements by automatic or manual systems.(2)

Fire Control: The capability shall be provided to detect any fire. Means shall be provided for fire-resistant storage of all items that are not self-extinguishing when they are not in use.(3)

Critical mechanical items shall be provided with debris-proof covers or containers. Critical electrical items shall be provided with suitable containers, potting, or conformal coating. Filters, strainers, traps, or other devices shall be provided in all moving-fluid systems to eliminate such systems as a threat to critical mechanical or electrical components. All such filters, strainers, traps, or other devices shall be capable of inflight cleaning and/or replacement.(3)

Interior walls and the secondary structure shall be self-extinguishing. All combustible shall be self-extinguishing in the most severe oxidizing environment to which they will be exposed. The material used to extinguish fires must be nontoxic and capable of being easily cleaned up after use.(3)

The capability to withstand the strike shall be verified by test using, as a minimum a lightning waveform of 200,000-A peak, having average currents of at least 200 A for a minimum of 200 ms. The 200,000-A peak should be reached in 10 µs after strike start. The current shall be applied to the vehicle skin at each major protuberance and the vehicle shall be grounded at the furthermost extremity.(3)

The crew shall be provided with a caution and warning system which will identify equipment failures, fire or other potential emergency situations. Further, the vehicle shall be provided with a Fault Detection, Isolation and Recovery System which will address problems in critical and non-critical systems.(4)

Appropriate functions, when implemented, shall be capable of being tested for proper operations during both ground and flight phases.(5)

1.5 Structural Criteria

Guidelines:

a) Primary structures, pressurized lines, and fittings should be designed in accordance with proven aerospace standards. Structural margins should be demonstrated by formal tests. Vibration and fatigue analyses should be performed.
b) Primary structures should be designed in accordance with safe-life design concepts. Safe-life should be determined by analysis/test considered to be the total number of mission cycles expected during the entire service life of the structure.
c) Primary structures should be assessed for stress corrosion susceptibility.
d) Pressure vessels should be designed by fracture mechanics to leak-before-burst criteria.
e) Structures should be designed to accommodate critical design loads for the entire flight envelope, derived from applicable design principles such as crash load requirements and crash-worthy principles.(1)

Thermal Design and Analysis: Nominal and worst-case analyses shall be done for all temperature-sensitive components and structures.(3)

Structural Analysis: Structural analysis shall be performed on all spacecraft structures, including pressure vessels, to show that all elements of the design, such as the strength, stiffness, structural stability, and life, meet all specified criteria for the anticipated loads and environments.(3)

Worst hot and cold cases shall be analyzed single values of thermal parameters which, in combination, produce the worst-case conditions.(3)

The analyses shall include stress analyses, fatigue or fracture analyses, loads and environmental data, the appropriate reference, and, in general, shall meet the same requirements as analyses generated for the purpose of certifying the flightworthiness of structures.(3)

Primary structure must be assessed for stress corrosion susceptibility. Pressure vessels shall be designed by fracture mechanics to leak before burst criteria. Structures shall be designed to accommodate design loads derived from applicable crash-worthy design principles.(4)

Primary structure shall be designed to an ultimate factor of safety of 1.4 or greater. Structural margins are to be demonstrated by test. Pressurized lines and fittings shall be designed to an ultimate factor of safety of 4.0 or greater.(4)

The payload structural design shall provide acceptable factors of safety for all mission phases. Design compliance will be verified in accordance with proven standards. When structural failure can result in a catastrophic event, the design shall be based on fracture control procedures.(5)

The structural design shall comply with the ultimate design load factors for emergency landing loads that are specified.(5)

Materials used in the design of payload structures, support bracketry, and mounting hardware shall be rated for resistance to stress corrosion cracking.(5)

Pressure vessels shall be assessed for adequate stress rupture life.(5)

Pressure containers shall meet leak-before-burst (LBB) criteria as determined by fracture mechanics analysis. The fracture mechanics analysis must employ a safe-life approach for containers of hazardous fluids and non-LBB designs.(5)

In circumstances where pressure loads have a relieving or stabilizing effect on structural load capability, the minimum expected value of such loads shall be used and shall not be multiplied by the factor of safety in calculating the design yield or ultimate load. Stress calculations of structural members, critical for stability and compressive strength, may be performed using the mean drawing thickness as the maximum thickness. The thickness used in the stress calculations for pressure vessels and for tension-critical and shear-critical members shall be the minimum thickness shown on the drawing.(6)

Hardware shall be designed to minimize weight and yet resist all loads and combination of loads that may reasonably be expected to occur during all phases of fabrication, testing, transportation, erection, checkout, launch flight, and recovery.(6)

Factor of Safety: For components, or systems subjected to several missions, static strength safety factor requirements shall apply to all missions. Consideration shall be given to transient loads and pressure, such as surge phenomena, when required. Elongation criteria rather than the yield safety factors may be applied, if the structural integrity of the component affected is demonstrated by adequate analysis and test.(6)

Materials and mechanical parts used in load carrying structural elements, the integrity of which is a safety issue, together with related manufacturing processes are fully assessed, for fracture mechanics, fatigue and stress corrosion resistance.(7)

To assure structural integrity, special attention should be given to aerodynamic loads for the launch vehicle configuration, loading due to manual and automatic switchover from primary to redundant systems, and structural break-up for safe abort.(8)

Functional Requirements:

None

1.6 Redundancy

Guidelines:

All critical systems, except for primary structures, should be designed with the appropriate degree of functional redundancy as determined by supporting mission and design analyses. Redundant components should be chosen from different manufacturing lots. To prevent generic failure, the last level of redundancy should use dissimilar components or practices. Special attention to redundancy requirements should be given to power supplies, breathing air supplies, pyrotechnics and communications.(1)

...analysis has established:

- The effect of man-rating subsystems to increase crew safety.
- The gain in mission reliability (launch completion) by man-rating several subsystems.(2)

...redundant systems are switched in automatically using malfunction detection signals.
...manual controls are used to correct four types of failures when the automatic systems malfunction.(2)

Slow failures of the drift type may be corrected by overriding controls...(2)

Failures which occur as step functions without redundant backup are only correctable by automatic action. If the failure is of a critical nature then abort should be automatic.(2)

Design compatibility should be emphasized in the area of GSE.(2)

Critical Subsystems: The redundancy requirements for critical flight vehicle subsystems (except structure, thermal protective system, and pressure vessels) shall be established on an individual subsystem basis but shall not be less than fail-safe. Flight hardware and payloads will be designed, as a minimum, to sustain a failure of a single item of hardware in any subsystem without loss of life or vehicle. Where the above criteria are not met, a list shall be provided of critical nonredundant items, the failure of which could cause loss of the crew or require abort of missions.(3)

All critical systems, except for primary structures, pressure vessels and thermal protection systems, shall be designed with an appropriate degree of redundancy as determined by supporting mission and design analyses.(4)

Safety-critical redundant subsystems shall be separated by the maximum practical distance, or otherwise protected, to ensure that an unexpected event that damages one is not likely to prevent the others from performing their functions.(5)

All redundant functions that are required to prevent a catastrophic hazard must not be routed through a single connector.(5)

Redundant flight control and electrical systems are necessary. Switchover from primary to redundant systems can be manual for malfunctions that result in relatively slow divergence, and automatic for malfunctions that result in rapid divergence. All sensor outputs should be redundant to preclude any possibility of sensor malfunction causing switchover.(8)

Functional Requirements:

Systems should be designed so that the interruption of gas flow, fluid, or electrical current should not, by itself, cause a critical condition. Single point failures and credible single failure modes shall be guarded against through separation of redundant paths, failure propagation control, and redundancy management. Purely redundant systems or components should be prohibited from performing their function unless 1) a primary system or component has failed, or 2) the redundant systems or components are being intentionally tested. This "standby redundancy" uses redundant hardware items which are non-operative until they are switched into the subsystem upon failure of the primary element.(1)

Automatic detection and switching must be used for the violent control malfunction type of failure ... (2)

Man-rating of the AGE [GSE] and prelaunch operations is equally as important to a program success as the present studies of the vehicle and its inflight operations.(2)

Establish a list of all possible flight failure modes and analytically determine the probability of failures.(8)

1.7 Materials

Guidelines:

Designers should only use well-understood and characterized materials. The use of new or highly improved materials, for which there exists little detailed information on their properties, capabilities, long-term effects, and other related characteristics, should be avoided. The environment the materials will experience should be known as well as their reaction to that environment and each other.(1)

Design considerations may include such important aspects as ... limitation of propulsion-created effects such as toxicity... (2)

Selection and Review: Materials to be used in the fabrication of spacecraft structure, mission-essential flight equipment, and ground support equipment, where operational failure can adversely affect the integrity of flight hardware, shall be selected by consideration of both functional and compatibility requirements.(3)

Wiring Material Flammability: This standard applies to electrical wiring within habitable compartments of spacecraft.(3)

Mercury Contamination: Prior to initial manned operations, the breathing systems of spacecraft, environmental chambers, and auxiliary life support systems shall be tested for mercury contamination. Any level of mercury contamination in breathing systems shall be avoided. Continuous exposure to mercury at greater than 0.005 mg/m^3 is considered toxic to humans.(3)

Mercury Use: The use of equipment containing mercury in liquid or vapor form shall be avoided where the mercury could come in contact with the spacecraft of spaceflight equipment at any time during manufacturing, assembly, testing, checkout, or flight.(3)

Titanium Use: Titanium or its alloys shall be used where exposed to liquid oxygen.(3)

Beryllium Use: Unalloyed beryllium shall not be used within the crew compartment(s) of spacecraft unless suitably protected to prevent erosion, or formation of salts or oxidizes.(3)

Cadmium Use: Use cadmium and cadmium plating should be avoided under the following conditions:

1) Where cadmium in contact with breathing gas could reach temperatures that would generate toxic fumes.
2) In equipment containers where electrical and electronic equipment could be degraded to an unacceptable level by vaporization and deposition of cadmium on the equipment surfaces.
3) In applications where the combination of temperature and proximity of the cadmium or cadmium plating could adversely affect critical surfaces by cadmium deposition.
4) In applications where temperature of the cadmium or cadmium plating could exceed 232.2°C(450°F).(3)

Explosive Device Packaging: Explosive devices, such as pyrotechnics and electroexplosive components, shall be packaged in a conductive materials which provide protection from static electric charges.(3)

Tests shall be conducted using calibrated analytical instruments capable of detecting concentrations of mercury less than 0.005 mg/m^3. The minimum temperature of a system while undergoing testing shall be 20°C(68°F).(3)

Where the use of equipment containing mercury cannot be avoided, the following information shall be documented:

1) A list of equipment containing mercury to be used during manufacturing, assembly, testing, and checkout, along with justification for each use.
2) The amount of mercury contained in the equipment.
3) The protection provided to prevent the release of mercury.
4) A plan for decontamination in the event the mercury is release. The plan must note that, a) an environment containing mercury vapor in concentration of 0.005 mg/m^3 (or greater) is not acceptable, and that b) mercury must not be removed from metal surfaces with any abrasive cleaning method. The removal of oxide films on the metal will cause immediate mercury penetration.(3)

Titanium or its alloys shall not be used where exposed to gaseous oxygen at any pressure or with air at oxygen partial pressures above 34.5 kPa (5 lb psia).(3)

Beryllium is an extremely toxic material whose threshold limit is 0.002 mg/m^3. Alloys containing 4 percent or less of beryllium are an exception to this standard.(3)

Packaging materials shall have a maximum surface resistivity of 3×10^4 ohms at all levels of humidity when tested.(3)

Use only well understood and characterized materials. (Designers should definitely avoid the use of new or highly improved materials for which there is little detailed knowledge and understanding of materials characteristics, compatibilities, long term effects, etc.).(4)

Selected materials should be well understood and characterized. Test data should be available. Hazardous materials which are toxic or would threaten hardware if released, should be avoided. Particular attention should be given to materials used in systems containing hazardous fluids.(5)

The selection, application, qualification and procurement of parts and materials is a major safety consideration in any space project, since the end product is no better than the basic material used.(7)

All non-metallic materials shall be rated for their characteristics of flammability under elevated oxygen levels, toxic offgassing and odor, in addition to their "normal" evaluation for space application like outgassing and UV stability.(7)

Functional Requirements:

Functional requirements including, but not limited to, load distribution and magnitude, temperature, life, and use of exposure environments shall be met by consideration of such material properties as mechanical strength, fatigue, thermal stability, fracture toughness, and flaw propagation rates. Material compatibility requirements shall be met by consideration of possible degradative mechanisms including, but not limited to, stress corrosion, galvanic or dissimilar metal corrosion, hydrogen embrittlement, creep, cycle and thermal fatigue, oxidization, vacuum stability, and radiation exposure.(3)

Electrical wire insulation, wiring accessories, and materials in contact with electrical circuitry shall not be capable of sustaining combustion in the most severe oxidizing environment to be encountered during operations:
1) After removal of the source ignition.
2) Following melting of the electrical conductor by high currents, such as those resulting from short circuits or equipment malfunction.(3)

Toxicity: No materials that, when exposed to a short circuit, will generate toxic fumes in a concentration sufficient to impair crew safety shall be used for wire insulation, ties, identification marks, and protective covering on wiring. Nonmetallic materials used within crew compartments shall not provide a toxic atmosphere.(3)

a) For hazardous chemicals that must be contained, each level of containment will be free of leaks under maximum use conditions. Documentation of all chemicals used and their method of containment will be maintained.
b) Payloads shall not constitute an uncontrolled fire hazard, and a flammability assessment shall be documented.
c) Material used in the crew cabin and other habitable areas must be tested under worst-case cabin environment conditions. Offgassing tests shall be conducted.(5)

1.8 Displays and Controls

Guidelines:

...launch vehicles have used the crew of the vehicle to perform...
a. Detection of malfunctions...
b. Corrective action...to preclude aborting the mission.
c. Actuation of abort...
d. Checkout of the vehicle prior to launch and in an earth...parking orbit.
e. Flight control of the vehicle as a backup to an automatic system.(2)

... a wide variance was found in the degree to which the crew participated in the operations of the vehicle.
[Primary control by crew]
[Primary control by crew but with automatic override]
[Automatic control with manual override]
[Monitor only] (2)

The pilot or controller should have sufficient display ...to separate actual malfunctions from false malfunctions.(2)

The information presented to the pilot must include information required for him to make decisions and information required in an emergency to evaluate the situation and make a fast, accurate decision.(2)

The pilot or controller should have sufficient displays...to separate actual from false malfunctions.(2)

Operating Limits on Temperature-Controlled Equipment: For spacecraft equipment where the operating temperature is normally controlled by heating or cooling equipment and the temperature is monitored in ground test and flight, the test program and/or appropriate analyses shall define:
1) The maximum and minimum temperatures expected in normal operations.
2) The maximum and minimum temperatures at which equipment may be expected to:
   a. Fail to function until temperature is restored to normal range.
   b. Be permanently rendered inoperative.(3)

Monitoring circuits should be designed such that the information obtained is as directly related to the status of the monitored device as possible.(5)

Functional Requirements:

Displays and controls should be provided to the crew for monitoring system status and failure alerts.(1)

Must be available, accessible, and readable in emergency situations. Controls for critical functions must not be able to be activated or deactivated inadvertently.(1)

...a combination of automatic and pilot control is favored. Crew override should be provided where signals may be erroneous.(2)

Communications and data transmission are an integral part of the design aspect of "man-rating". ...Man-rating of a launch vehicle cannot be considered complete without a full analysis of the limitations imposed and the potentials offered by communications systems. The reliability of the data transmission is essential.(2)

Where abort is by astronaut command, a clear communications channel must exist for advice and recovery.(2)

...the computer should have all possible alternate missions preprogrammed so that the optimum alternate mission can be selected. Then, new commands can be supplied to the flight control system, and illuminated display lights on the crew's console tells them of the alternate mission and the new parameters that require resetting, if applicable.(2)

Indication of Failure: Those measurements systems which display critical flight information to the crew on panel indicators shall be designed so that when such a system fails, it should provide an indication of its failure.(3)

Attitude Control Authority: Spacecraft automatic attitude control circuitry shall be designed so that the crew can assume manual attitude control at all times.(3)

Separation Sensing System: Separation sensing systems used to detect separation of stages or modules of the space vehicle shall be designed so that actuation of separation sensors will not result from structural deformation or vibrations less severe than those associated with structural failure of the vehicle.(3)

Gyroscope Performance Verification: Guidance and navigation subsystems, stabilization subsystems, control subsystems, and any similar subsystems using gyroscopes for guidance or stabilization of spacecraft during propulsion subsystem operation shall provide continuous outputs for verification and proper gyroscope rotational speed or drift rate.(3)

Fluid Temperature/Pressure Monitoring: All spacecraft systems and ground support servicing equipment requiring storage of reactive fluids (i.e., oxidizers, monopropellants, etc.) shall be designed to include devices for monitoring temperature and pressure to permit accurate determination of the rates of active oxygen loss of the oxidizer contained in their respective systems.(3)

Atmospheric Pressure and Composition Control: Provisions shall be made to monitor and control oxygen, carbon monoxide, atmospheric pressure, and trace contaminants. Trace contaminants may be organic, inorganic, or biological.(3)

Explosive Devices: In the design and operation of spacecraft systems, provisions shall be made for arming explosive devices as near to the time of expected use as is feasible without compromising reliability or safety. Provisions shall be made to promptly disarm explosive devices when no longer needed.(3)

For critical components, automatic safing with manual override only shall be used.(3)

Where the separation sensing system is used to initiate automatically subsequent steps in a sequence of events, the sensing system shall be configured so that actuation or failure of a single sensor will not initiate the sequence of events.(3)

Immediately prior to engine ignition for launch, including inflight launches, the rotational speed or drift rate of all gyroscopes, normally required to operate at launch, shall be verified to be within required safe operating limits.(3)

These monitoring devices will provide time for corrective action in the event that abnormal decomposition of the oxidizer is initiated.(3)

"Arm" and "fire" shall be separate functions and separately displayed. Arm and fire switches shall be guarded switches.(3)

a) Monitor circuits shall be current limited or otherwise designed to prevent operation of the hazardous functions with credible failures. Loss of input or failure of the monitor should cause a change in state of the indicator. Notification of changes in the status of safety monitoring shall be given to the flight crew in either near-real-time or real-time.
b) When timers are used on deployable payloads to control inhibits to hazardous functions, complete separation of the payload from the launch vehicle must be achieved prior to timer initiation. If credible failure modes exist that could allow the timer to start prior to separation, a safing capability must be provided.
c) A function whose inadvertent operation could result in a critical hazard must be controlled by two independent inhibits, whenever the hazard potential exist.
d) A function whose inadvertent operation could result in a catastrophic hazard must be controlled by a minimum of three independent inhibits, whenever the hazard potential exist.(5)

1.9 Aborts

Guidelines:

The system should be designed to accommodate sufficient abort scenarios to allow the crew to return home safely or attain safe haven from any credible failure for all phases of the mission.(1)

Payloads must be safe for aborts and contingency returns.(5)

To meet fail-safe requirements, rescue provisions have to be foreseen in cases of launch or mission interruption.(7)

In instances where an immediate abort from a unstable launch vehicle is not feasible and the crew must "ride it out" for some finite time, incorporate hardware changes to the vehicle that assures that any vehicle break-up would not cause a fire-ball type failure, but rather permit eventual safe abort.(8)

In the event that a malfunction is not cured by a switchover from primary to redundant systems, an abort is required.(8)

Functional Requirements:

Credible failures for which abort procedures should be developed shall include, at a minimum, one engine out and loss of cabin pressure. All abort scenarios shall be tested and the crew fully trained in their execution.(1)

Hazard controls may include deployment, jettison or design provisions to change the payload configurations.(5)

A parachute system for the habitable compartment of the launch vehicle or emergency landing for a winged system.(7)

2. DESIGN PRACTICES

Guidelines:

Designers should use existing manned spacecraft design practices which have been developed and proven by NASA, the military, and aerospace industry.(1)

a) Overpressurization should be avoided.
b) All critical service lines should be routed and protected to preclude damage from any cause other than a catastrophic occurrence. Redundant systems should be adequately separated from one another to avoid common-cause damage.
c) Lines providing non-compatible materials or services should be run in separate protective shields.
d) All sharp corners and other protrusions should be eliminated to preclude damage to crew, space suits, or related equipment.
e) Toxic or explosive gases should be avoided.

f) All equipment and material flammabilities should be designed to be compatible with the expected pressure environments (normal and emergency) of the habitable areas.
g) Special attention should be given to the integrity of quick-disconnects (gas and fluid) that may be operated by the crew.
h) Oxygen partial pressure build-up or decrease should be avoided.(1)

Designers shall use existing manned spacecraft design practices which have been developed and documented.(4)

a) The maximum design pressure (MDP) for a pressurized system shall be the highest pressure defined by maximum relief pressure, maximum regulator pressure or maximum temperature.
b) All pressure vessels shall be designed to satisfy proof testing requirements.
c) The structural integrity of the payload design must be demonstrated for external load environments.
d) Flexible hoses and bellows shall be designed to exclude flow induced vibrations.
e) Payload sealed compartments within a habitable volume of the launch vehicle, including containers which present a safety hazard if rupture occurs, shall be capable of withstanding the maximum pressure differential associated with emergency depressurization of the habitable volume.
f) A proof test of each flight pressure vessel to a minimum of 1.5xMDP and a fatigue analysis showing a minimum of 10 design lifetimes may be used in lieu of testing a certification vessel to qualify a vessel design.
g) The proof test factor for each flight pressure container shall be a minimum of 1.1xMDP.(5)

Priorities in design shall be given in order to avoid safety critical situations. Designs are to be based on "leak-before-burst" principles.(7)

Special attention in the design is needed to maintain cleanliness, to avoid excessive generation of potentially damaging particulates.(7)

The order of precedence to avoid safety critical situations shall be 1. design for minimum hazard, 2. provide safety devices to limit hazards to an acceptable level, 3. incorporate warning devices for timely detection of hazardous conditions, and 4. develop procedures to encounter hazardous conditions.(7)

Functional Requirements:

a) Pressure relief valves or other safeguards should be provided to prevent overpressurization.
b) Cryogenic systems, with sections where cryogenic liquid may be trapped, should be designed to prevent line rupture if relief valves freeze. System should be provided with relief valves paralleled by burst discs.
c) Systems or materials, which are potentially hazardous if allowed to physically meet, shall be redundantly separated or shielded from one another, or adequately spaced apart.
d) TBD
e) A detection system and an appropriate exhaust or neutralizing system should be provided where toxic or explosive gases may be expected.
f) TBD

g) TBD
h) Oxygen flow limiters and/or monitoring devices should be required to insure against oxygen partial pressure build-up or decrease.(1)

Outer shells (i.e., vacuum jackets) shall have pressure relief capability to preclude rupture.(5)

Where pressure regulators, relief devices, and/or a thermal control system are used to control pressure, collectively they must be two-fault tolerant from causing pressure to exceed MDP.(5)

Relief devices must be redundant and sized to permit full flow at MDP.(5)

2.1 General Issues

Guidelines:

Design considerations may include such important aspects as engine-out capability and holddown.(2)

Limitation of propulsion-created effects, such as toxicity, noise, and vibration also should be considered.(2)

Separation of Redundant Equipment: Redundant systems, redundant subsystems, and redundant major elements of subsystems shall be separated to the maximum extent practicable or otherwise protected to ensure that an unexpected event which damages one is not likely to prevent the orbiter from performing the functions.(3)

Interior Design for Cleanliness: The greatest practicable precautions shall be taken to ensure freedom from debris within the spacecraft compartment and within individual systems or components.(3)

Equipment Protection from System Liquids: Location of sensitive equipment below plumbing, cold plates, or other equipment capable of leaking or generating condensate during ground operations shall be avoided.(3)

Equipment Protection from Moisture: Equipment within a pressurized compartment shall be designed so that performance of the equipment will not be degraded by humidity or moisture droplets in the spacecraft atmosphere or by condensation.(3)

Ingress of Undesirable Elements: In the design of pressurization, repressurization, and ventilation systems for habitable areas, provisions shall be made to minimize ingress of undesirable elements.(3)

Functional Requirement:

Electrical wiring of redundant systems, redundant subsystems, or redundant major elements of subsystems shall not be routed in the same wire bundle or through the same connector without wiring of the other redundant systems, subsystem, or subsystem element. Redundant systems and redundant components should be designed so as not to preclude concurrent operation.(3)

Protective covers shall be provided. The use of particulate-generating materials and surfaces is prohibited. If such material must be used they must either be coated, encased, or taped. The ventilation system shall include debris-collection screens on air inlets. Removal and/or control of biologically active components shall be considered when specifying the use of debris screens or filters.(3)

Design plumbing to be insensitive to the liquid leakage. Design plumbing or equipment containing the liquid to locate couplings, vents, service points, and other items where leakage from them could not reach the sensitive equipment from leakage during ground operation. Provide insulation to prevent condensate from falling on the equipment.(3)

Applies to manned spacecraft which have habitable areas pressurized at less than atmospheric pressure during normal mission.(3)

2.2 Electrical Issues

Guidelines:

Mating Provisions for Electrical Connectors: Electrical connectors, plugs, and receptacles which otherwise could be incorrectly mated shall be designed to prevent incorrect connection with other accessible connectors.(3)

Protection of Severed Electrical Circuits: Electrical circuits which are to be severed in the normal course of mission events shall be protected against short circuiting or compromising of other circuits during the mission.(3)

Moisture Protection for Electrical Connectors: Electronic and electrical equipment, both external to and within the crew compartment, which is not hermetically sealed or otherwise positively protected against moisture shall not be cooled below the dew point of the surrounding atmosphere.(3)

Corona Suppression: Electrical and electronic systems and components shall be designed such that corona discharge will not occur under any operating conditions.(3)

Protective Covering for Electrical Receptacles and Plugs: Electrical plugs and receptacles of flight equipment and ground equipment that connects with flight equipment shall be protected at all times.(3)

Control of Electrostatic Discharge (ESD): All ESD-susceptible parts and assemblies which make up the Line Replaceable Unit (LRU) shall be ESD-classified by test, while the LRU and higher level items my be ESD-classified by analysis.(3)

Bioinstrumentation Systems: Such systems shall be designed with sufficient resistance in series with each body electrode, and must limit to safe levels any electric shock current that could flow through an instrumented member as a result of contact with available voltage sources.(3)

Pressure Garment Wiring: Current entering a crew member's pressure garment through wiring shall not ignite or damage materials used within the garment under the worst condition.(3)

Extravehicular Activity (EVA) Electronic Connectors: Consideration will be given to improve the capability of the gloved astronaut to effect the mating and demating of electrical connectors on boxes designed for changeout in orbit.(3)

Ionizing Radiation Effects: Spacecraft electronics shall be designed to accommodate the orbital ambient ionizing radiation environment.(3)

Use of constraints built into a cable or harness. Selection of different sizes for connectors. Permanent identification of mating connectors.(3)

All parts and assemblies at the LRU level must have a certified ESD sensitivity level equal to or greater than 15000 V.(3)

The maximum safe shock current levels for DC and AC currents up to 2000 Hz are defined as 1.0μA applied internally and 50 μA applied externally to the body.(3)

Functional Requirement:

Electrical Circuits - De-energizing: Spacecraft electrical systems shall be designed so that all necessary mating and demating of connectors can be accomplished without producing electrical arcs that will damage connector pins or ignite surrounding materiels or vapors.(3)

Electrical Power Overload Protection: Maximum operating temperatures for electrical power distribution circuit elements of the circuit.(3)

Protective Devices for Solid-State Circuits: Protective devices used in critical electronic circuits to protect solid-state circuit elements shall be verified as ready to function.(3)

Engine Shutdown Circuitry: Design of circuitry for automatic shutdown of launch vehicle engine(s) shall include protection against possible engine shutdown coincidental with, or immediately after, launch vehicle release.(3)

Electronic and electrical equipment, electrical connectors and wiring junctions to connectors shall be protected from moisture by methods which are demonstrated by test or analysis to provide adequate protection to prevent open and short circuits.(3)

Unless connectors are specifically designed and approved for mating or demating in the existing environment under the loads being carried, they shall not be mated or demated until voltages have been removed from the powered side(s) of the connector.(3)

Device hardening, circuit fault tolerance, shielding.(3)

2.3 Fluid Issues

Guidelines:

Fluid System Service Point Protection: Service points for spacecraft fluid systems shall be designed with positive protection by location, connector size, or type to prevent connection to incorrect fluid service lines.(3)

Ground Service Points: Ground service points for fluid systems, including those for filling, draining, purging, or bleeding, shall be accessibly located external to the vehicle. Gas purge or bleed fittings shall exhaust external to the vehicle.(3)

Fluid Line Separation Provisions: Fluid lines that are required to be disconnected or severed on separation of space-vehicle modules shall be designed such that any breakage resulting from failure of the disconnecting device to function will occur on the side of the disconnect that is 1) the least hazardous and 2) the most easily repairable.(3)

Fluid Line Installation: Routing and installation of all fluid lines, including pressure-sensor lines, shall be specified in detail. Special precautions shall be taken to prevent the installation of such lines where they would be exposed to extreme temperature conditions. An adequate design analysis shall be made for each such line installation to show that the temperature extremes to which the line will be subjected are within limits acceptable for the fluid involved.(3)

Fluid System Component Protection: All ends of tubing, fittings, and components used in fluid systems shall be protected against damage and entry of contaminants in each step of the spacecraft manufacturing process and subsystem buildup.(3)

Fluid System Cleanliness: After manufacturing and after any subsequent exposure to the probable entry of contaminants, all spacecraft fluid systems and their servicing equipment shall be cleaned by flushing to remove all contaminants which could be detrimental to the system.(3)

Fluid System Flushing and Draining: Spacecraft fluid systems and related servicing equipment shall be designed to permit complete flushing and draining during ground and in-orbit servicing operations.(3)

Design drawings and/or process specifications shall designate the method of complying with this requirement. The degree of protection provided shall be compatible with the cleanliness requirement of the manufacturing specification.(3)

The flushing fluid shall be compatible with the system materials and the working fluid to be used in the system. Cleanliness levels of the flushing fluid and the maximum allowable contamination shall be specified.(3)

Functional Requirement:

Flow Restrictions - Pressurized Sources: Where pressurized gas lines could fail in such a way that the total gas supply dumped directly into a compartment would be greater than the relief valve or venting could handle without overpressurization of the compartment, necessary flow restrictions shall be incorporated at the pressure source to restrict the mass flow to a level that can be handled by the relief valve and/or venting.(3) The flow restriction must not interfered with the normal operation of the system.(3) The check valve for shutoff valve, used on the retained side of the disconnect for preventing loss of fluid after disconnection, shall be a type that will function; i.e., that will close in spite of such a failure.(3) The systems shall be free of dead-ended piping or passages through which flushing fluid cannot be made to flow. Drain and bleed ports shall be provided for attitudes anticipated during ground servicing of the systems.(3)

2.4 Mechanical/Structural Issues

Guidelines:

Each type of pressure vessel must be qualified prior to acceptance of the design for program use by tests designed to demonstrate that the selected design factors were achieved. Each pressure vessel accepted for program use shall pass acceptance tests designed to demonstrate its freedom from latent manufacturing defects or handling damage.(3)

Qualification tests shall include provisions to demonstrate the ability of this equipment to operate without significant wear or damage for a minimum of double the maximum number of cycles expected to occur during fabrication, testing, and flight. Qualification tests for parts, such as seals, that are intended to be replaced prior to launch shall demonstrate the capability of the part to operate without significant wear or damage for a minimum of double the maximum number of cycles expected during prelaunch checkout and flight.(3)

Threaded devices shall be applied in a manner to preclude the release of particles or foreign material where interference with proper operation of system components could occur. Devices such as self-tapping screws and bolts shall not be used.(3)

A fracture mechanics analysis shall be performed for each configuration of glass structure. A proof acceptance test consistent with the type of loading shall be conducted to screen flaws in each glass structural flight item based on the results of the fracture mechanics analysis. All proof testing will be performed in a suitable environment to limit flaw growth during the proof testing.(3)

1) Doors shall be used only if no other practicable methods exists to perform the desired function.(3)

3) Spacecraft systems requiring thermal protection doors shall be designed so that if a door fails, the spacecraft can make a safe ascent and reentry.(3)

Pressure Vessel Design: Each pressure vessel must be designed using a factor of safety selected to accommodate the most severe combination of environmental and pressure conditions expected in use.(3)

Hatches: Spacecraft hatches and associated hardware (hinges, latches, seals, actuators, etc.) shall be designed for repeated operation and inspection and shall be repairable or replaceable in orbit.(3)

Threaded Fittings: Threaded fittings and fasteners, such as nuts, nut plates, and bolts, used in manned spacecraft or spacecraft equipment shall be designed to minimize the generation of metallic particles or foreign material.(3)

Exposed Sharp Surfaces and Protrusions: Exposed sharp surfaces or protrusions which could injure crew members of damage equipment shall be eliminated or guarded so as to avoid accidental contact.(3)

Windows and Glass Structures: The design of all spacecraft windows and other glass structures shall include evaluation of flaw growth under the design stress and environment.(3)

Penetration of Inhabited Compartments: Inhabited spacecraft compartments shall be so designed that all penetrations shall take advantage of normal pressure-induced forces to aid in maintaining vessel structure and cabin pressure integrity.(3)

Functional Doors: Doors in the structure of the spacecraft or heat shield that open and close during flight must be designed such that a single-point failure will not cause the loss of crew or vehicle.(3)

Functional Requirement:

Pressure Vessel Relief: Pressure relief capability shall be provided for vessels where the contents, system design, or operation may cause an increase in internal pressure above the maximum designed operation pressure. Portions of fluid systems that trap fluids (become locked-up) should be considered pressure vessels and should be evaluated for the need of relief capabilities.(3)

All flight vessels shall be protected during servicing, either on the ground or in flight, by relief valves in the servicing equipment. The relief valves shall be sized for sufficient mass flow to protect the vessel in the event of servicing pressure regulator failure. Such as failure shall not cause the vessel to exceed the maximum design operating pressure.(3)

The primary flight crew ingress/egress hatch used during ground operations shall be designed to be outward opening from the pressurized spacecraft compartment. For designs where it is impractical to have an outward opening hatch, provisions will be made to rapidly equalize the pressure across the match.(3)

Reliable, redundant safety devices shall be provided to prevent inadvertent opening or rapid depressurization on orbit.(3)

All mechanisms shall have manual overrides. All mechanisms used on docking, berthing, and positioning systems shall have position indicators. External (outside actuator) limit switches shall not be used in areas susceptible to contamination.(3)

2) Doors shall be as small as practicable and shall be designed to cover only the devices they protect; specifically, they will not be enlarged for use as maintenance access doors. Designs requiring active vents will include relief features to prevent catastrophic failures should the door fail to operate. Vehicle aerodynamics and structural integrity will not be catastrophically degraded by a failed door.(3)

4) For door operation that is not time critical with respect to survival of crew and/or vehicle, an alternate manual means shall be provided if items 2 and 3 are not practicable.(3)

3. RELIABILITY

Guidelines:

All critical systems should be designed with known, highly reliable parts and components. Overall reliability should be demonstrated and verified by appropriate tests of components, subsystems, and systems. Risk analysis should be conducted to identify ares of weakness or concern. Probabilistic risk analyses may be used to assist in identifying problem areas or to rank risk levels in trade studies, but should not be used in place of testing as a verification of overall design reliability.(1)

Safety analyses including logic trees, Fault Tree analyses, and Failure Modes and Effects Analyses (FMEA) shall be employed to assess specific risks and determine how to minimize them.(1)

...malfunction analysis, including failure-mode and failure-effect analyses to minimize potential hazards to crew survival.(2)

...failure effects analysis (FEA) should include prelaunch operations within its scope.(2)

The reliability of the data transmission is essential.(2)

...attention should be given to the following reliability factors:

a. Failure effects analysis.
b. Reliability improvement methods, including redundancy.
c. Design for long life...
d. Reliability design reviews.
e. consideration of the use of the pilot, where advantageous, in checkout, ground support, operations, surveillance, and maintenance.
f. Quality assurance programs, training, etc.(2)

methods of promoting reliability are:

a. Reduction of the number of basic components or subsystems.
b. Simplification of components and systems.
c. Redundant design - duplication of functions through elements either identical or different in form.
f. Application of fail-safe features - modes of failure will not render the engine unsafe or induce a noncorrectable condition.(2)

The history of reliability for components and subsystems over their complete life, from first qualification through acceptance and system tests on to flight tests,...(2)

All critical systems shall be designed with known high reliability parts and components. Overall reliability shall be demonstrated and verified by appropriate tests on components, subsystems and systems. Numerical reliability analyses may be used to identify problem areas, however such analyses shall not be used to verify design capability.(4)

For a basic launcher, identified for manned mission, which is also used for unmanned missions, reliability of this vehicle is proven by previous unmanned flights.(7)

Functional Requirements:

None

4. MANUFACTURING AND PROCUREMENT

Guidelines:

Only proven manufacturing processes and techniques should be employed. Designers should avoid processes and approaches which are not well-understood for the manufacture and fabrication of hardware. Designers should be included in the approval loop for any changes and improvements to the manufacture/fabrication process. Prime contractors should be required to investigate manufacturing and fabrication processes of their vendors for any unusual or new approaches which may affect the man-rated status of components and materials being used.(1)

Inspection and tests shall be performed during the process and in the final product to assure strict adherence to specified engineering or operational requirements.(1)

Use only proven manufacturing processes and techniques. (Designers should avoid new processes and approaches for the manufacturing and fabrication of hardware. They should be well acquainted with the processes to be employed and should be included in the approval loop for any changes or improvements to such processes.)(4)

Functional Requirements:

None

4.1 Parts/Materials Identification

Guidelines:

Parts and materials procured or designated specifically for use in manned spacecraft shall be identifiable by an appropriate method and stored in controlled access ares.(3)

Identification necessary to control parts until they are installed in an assembly is required. The detailed methods for numbering and marking parts shall be provided in documentation.(3)

Functional Requirements:

None

4.2 Pressure Vessel Documentation

Guidelines:

Manufacturing, processing, and pressurization histories shall be maintained on each spaceflight pressure vessel considered to be critical to the safety and/or success of the mission.(3)

Data required is:
1) Material certification and composition.
2) Actual fabrication and processing sequence.
3) Fluid exposure and temperature during fabrication and testing.
4) Actual chronological tests and checkout history including all proof, leak, and cycling data along with the magnitude of pressure, type of pressurant, and number of pressure cycles to which the tank was subjected.
5) Discrepancy history.(3)

Functional Requirements:

None

5. QUALITY ASSURANCE

Guidelines:

A parts traceability program shall be established.(1)

The program/project should put a heavy emphasis on attaining a high level of quality for all critical hardware and software. A quality awareness program should be a primary feature of the quality plan for all critical hardware and software. The plane should definitize the specific inspection and quality requirements for all such hardware, including safeguards for handling and protecting the hardware during its manufacture, assembly, testing, and shipment.(1)

Quality assurance...including the following mechanisms is necessary:
- Measurement of total vehicle effectiveness.
- Controlling manufacture and assembly.(2)

The program should put heavy emphasis on attaining a high level of quality for hardware and software. (A quality awareness program should be a primary feature of the quality plan).(4)

The responsible design element shall submit strength analysis and qualification test reports which will verify the capability of hardware to meet design requirements with factors of safety as specified herein.(6)

a. Margin of Safety - The responsible design element shall show by analyses and/or tests that the hardware meets program design requirements with sufficient margin of safety to assure adequate strength, service, life, rigidity, and safety of personnel at all times.(6)

Hazard analyses shall be performed on system, subsystem, equipment and functional levels with iterations between the prime contractor and all tier subcontractors. Results shall be presented and assessed during all phased design reviews on all levels.(7)

Quality assurance is maintained through appropriate personnel education and training, appropriate component selection and testing, and through extensive qualification testing.(8)

Functional Requirements:

None

5.1 Shipping/Handling Protection of Hardware

Guidelines:

Spaceflight hardware shall be suitably packaged or supported to provide protection of the hardware from damage during handling and shipping.(3)

Contractor or National Aeronautics and Space Administration (NASA) personnel will determine when shock-indicating devices are required. The use of such devices must be approved by the appropriate program office or project office prior to installation.(3)

Functional Requirement:

None

5.2 Reuse of Flight Hardware

Guidelines:

Flight hardware that has been previously used in flight may be reused in manned flight if appropriate refurbishment, inspection, and testing have been accomplished between flights.(3)

Any elements not replaced will be within all shelf-life and operational-life time limits at the end of the mission. There shall be no evidence that the unit has been stressed beyond specification limits during previous use.(3)

Functional Requirements:

None

6. TEST AND VERIFICATION

Guidelines:

A formal verification program should be conducted which demonstrates all functional and performance design requirements and repair and maintenance capability associated with man-rating. All redundant design features should be completely demonstrated during such tests. Limits should be established and safety margins determined by off-limits testing. The system should be fully demonstrated unmanned prior to committing to manned operations. If this is not feasible, an alternative plan may be considered which will completely exercise and demonstrate the manned-safety aspects of the design.(1)

...final design must be based on full pilot-simulator tests with actual pilot-display equipment.(2)

Man-controlled launch vehicles must eventually be simulated using man in the loop of the simulation...(2)

Extra testing will be required in that the EDS and shutdown provisions must be tested.(2)

...the astronaut must be trained in all aspects of the systems operation,...(2)

...all test procedures that are accomplished must be correct and formally revised before the procedure is accomplished.(2)

A formal test and verification program shall be conducted which demonstrates and verifies before manned launch all functional and performance design requirements and maintenance capability associated with man-rating requirements. All redundant design features shall be demonstrated and exercised.(4)

Test, analysis, and inspection are common techniques for verification of design features used to control potential hazards. The successful completion of the safety process will require positive feedback of completion results for all verification items associated with a given hazard.(5)

Test loads shall duplicate or envelope all flight loads and include pressure and temperature effects. When a separate qualification unit is used, the tests shall be accomplished at the yield and ultimate levels specified by the factors of safety.

a. Static Tests: In general, strength qualification testing shall be static.
b. Flight Article Simulators: If the component to be tested is statically determinant, it may be tested as a stand-alone unit. If the component to be tested is not statically determinant, the interfacing structure through which the loads and reactions are applied to the qualification unit must be simulated in the test.(6)

The interfacing structure used in the test must simulate the stiffness and boundary conditions of the corresponding flight hardware.(6)

c. Protoflight Tests: Protoflight testing and associated test factors may be accepted in lieu of static qualification testing with MSFC approval.(6)

The test factors will be limited to values which will not subject the protoflight structure to detrimental deformations beyond the elastic limit.(6)

Functional Requirements:

The system should be fully demonstrated unmanned prior to committing to manned operations.(1)
Prior to committing man to flight, the system shall be fully demonstrated in flight, unmanned.(4)
Payload hazards being controlled by launch vehicle provided services will require post-mate interface test verification for both controls and monitors.(5)

6.1 Redundant Path Verification

Guidelines:

None

Functional Requirements:

The design of spacecraft systems and subsystems incorporating redundancies shall include a means of verifying satisfactory operation of each redundant path at any time the system and/or subsystem is determined to require testing prior to launch and during the mission.(3)

6.2 Adequate External Visibility Verification

Guidelines:

Simulations shall include mockups.(3)

Functional Requirements:

Visibility verification for manned spacecraft shall include tests, simulations, or analyses to verify that the crew will have adequate visibility during all anticipated phases and environmental conditions of the planned mission.(3)

6.3 Verification Tests for Electrical and Electronic Supplies and Loads

Guidelines:

None

Functional Requirements:

Ground support equipment, facilities, and other equipment to be connected to a spacecraft system for operation, testing, checkout, or maintenance shall be designed to that routine verification tests can be conducted before each connection is made, to ensure that each electrical and electronic input to the spacecraft is compatible with the spacecraft system.(3)

6.4 Fluid Supply Verification

Guidelines:

Procedures to accomplish the verification tests shall be provided with the equipment. Calibration fluids shall be furnished with analysis reports.(3)

Functional Requirements:

Ground Support equipment, facilities, fluid containers, and other equipment to be connected to a spacecraft system for operation, testing, checkout, or maintenance shall be designed so that routine verification tests can be conducted before each connection is made to ensure that each fluid input to the spacecraft will be compatible with the spacecraft system.(3)

6.5 Pressure Vessel Qualification

Guidelines:

The compatibility of pressure vessel materials with processing, inspecting, testing, and flight fluids will be verified by data obtained under conditions simulating the intended fluid-use environment.(3)

The scope of the program shall be sufficient to show compatibility of the fluid and material for the anticipated temperature range, pressure range, pressure cycle history of the vessel, and fluid composition range. The qualification program will include tests on pressure vessels containing the flight fluids at the maximum pressure allowed by the relief devices or maximum pressure expected for vessels without relief devices. In addition, the conditions of the tests will include exposure to the most deleterious operation temperature expected in use. The vessels, while containing the flight fluid, shall be subjected to the above pressure and temperature conditions for twice the expected pressurized life or for 1 year, whichever is less. The flight duration shall include the time from initial pressurization, with the flight fluid, and the anticipated launch pad hold times under pressure. For testing of pressure vessels with decaying pressure, flight duration can be taken as the time from initial pressurization with the flight fluid including the anticipated launch pad hold times under pressure to the time at which the internal pressure of the vessel reaches one-half the maximum pressure allowed by the relief device or one-half maximum operating pressure, whichever pressure is less.(3)

Functional Requirements:

None

7. MANAGEMENT REVIEW AND CONTROL

Guidelines:

Reporting of results by procedure/report number and date is required.(5)

Strength analysis reports shall be submitted to MSFC in support of the following four design reviews: PDR, CDR, DCR, and FRR. These reports shall be current with respect to loads and the design at the time of the review.

a. Strength Analysis: The responsible design element shall perform strength analysis and document them so that it is clearly demonstrated that strength requirements have been fulfilled.(

b. Documentation Content: The strength analysis reports shall be prepared in accordance with standard aerospace industry practices for flight hardware.(6)

The analysis shall clearly identify such items as geometric description of each component, identification of all applied loads, type of material and applicable strength allowables, environments and effects, proper identification of references for all input into the analyses, and a summary of all calculated margins of safety.(6)

Quality control is maintained through extensive physical inspection, extensive failure analysis, and through fully implemented corrective actions and the reporting and documentation of all such actions. (8)

Functional Requirements:

None

7.1 Hardware/Software Failure and Corrective Action Reporting

Guidelines:

All failures are to be exhaustively investigated to assure that the cause is reasonably understood and that all corrective actions have been implemented and verified. This discipline should be enforced from the start of development and continued throughout the operating life of the system.(1)

System requirements reviews, preliminary design reviews, critical design reviews, design certification reviews, readiness reviews, and special safety/risk assessment reviews.(1)

...malfunction analysis, including failure-mode and failure-effect analyses to minimize potential hazards to crew survival.(2)

Translating the needs for higher safety into systems and components has been and is being accomplished by starting with a review of the launch vehicle and a failure effects analysis (FEA). ...Consequently, reliability and abort studies were combined with test results and FEA's to make up the data necessary for defining the vehicle modifications and abort systems.(2)

...attention should be given to the following reliability factors:
d. Reliability design reviews.(2)

Hardware/Software Failure and Corrective Action Reporting: Verification of flight readiness - a) Where flight or flight-like equipment has failed, launch-to-orbit of like equipment, either as an initial assembly or as an on-orbit replacement, shall not be permitted unless a) an analysis of the failure has established that the basic deficiency which caused the failure is not present in the replacement equipment, and b) the basic deficiency has been counteracted by changes in operational procedures to a degree that eliminates it as a significant threat to the success of the mission or the safety or the crew.(3)

The basic deficiency is determined to represent no significant threat to the success of the mission and safety of the crew.(3)

Hardware/Software failure and Corrective Action Reporting: All failures shall be exhaustively investigated to assure that the cause is completely understood and that corrective action is implemented and verified. This discipline shall be enforced from the start of development and continued to mature operations.(4)

Management assessment and review of man-rating criteria and requirements shall be conducted in conjunction with the various design reviews, special risk assessment exercises, flight readiness reviews, etc.(4)

Hardware/Software failure and Corrective Action Reporting: All anomalies during the previous payload missions must be assessed for safety impact.(5)

Those anomalies affecting safety critical systems must be reported and corrected.(5)

The results of safety analyses and hazard close-outs are subject to each major project review. This includes the review of safety assurance approaches, the review of caution, warning, and fusing concepts, the review of the hazard close-out status, and a safety "walk-through" of the engineering models. The approval for parts and material processing should be strictly followed, involving all contract tiers up to the prime contractor and the sponsoring government agency. Any non-compliance has to be sanctioned by waiver.(7)

A technical acceptance for each launch vehicle shall be conducted, with the scope of this review in sufficient detail to evaluate all test data, discrepancies and corrective actions, and all problem encountered during fabrication, assembly and testing. Any disagreement with the contractor's resolution shall result in non-acceptance until further investigations and/or corrective actions have been performed.(8)

Functional Requirements:

None

7.2 Documentation

Guidelines:

All documentation concerning formal test and verification, manufacturing, configuration control, and related matters should be maintained under strict management control. This shall include analyses related to critical issues involving man-rating.(1)

Maintained as part of the official files of the program.(1)

Extensive procedural documentation is required for...quality control during fabrication stages or production stages.(2)

...a quality control program covers the operation from inception, through design, to utilization.(2)

...a program including the following mechanisms is necessary:
- Critical evaluation of all deviations.
- Timely feedback of deviation information to responsible areas.(2)

Special Processes Identified: Manufacturing, assembly, or installation drawings for spacecraft, spaceflight equipment, experiments, and mission essential ground support equipment shall identify on the appropriate drawings all special processes required to manufacture, assemble, and install the equipment.(3)

Explosive Device Identification: Explosive devices shall be identified as to source, purpose, and operating characteristics.(3)

Identification of Flight and Nonflight Equipment: The status of flight and nonflight equipment shall be identified and classified.(3)

Process specifications shall be referenced, or the processes shall be specified in detail on the respective drawings.(3)

The following information shall be furnished with each device:
1) Part name
2) Contractor
3) Manufacturer
4) Part, lot, and serial number
5) Date of manufacturer
6) Type of device and weight of explosive.(3)

Three types of classification:
Class I for equipment acceptable for flight use, Class II (Controlled Equipment) for equipment acceptable for use in ground tests or training in a hazardous environment, Class III (Not For Flight) for equipment acceptable for nonhazardous training and display purposes.(3)

All documentation having to do with formal test and verification, manufacturing, configuration control, etc., shall be maintained under strict management control.(4)

Critical procedure/process steps must be identified in the appropriate hazard report.(5)

a) A payload safety verification tracking log is required.
b) The payload organization must provide a summary of the hazards being controlled by launch system services in a safety assessment report.
c) The payload organization must document in individual hazard reports those launch vehicle interfaces used to control and/or monitor the hazards.(5)

Functional Requirements:

None

7.3 Certification for Use

Guidelines:

The system should be certified for usage only after a review board is satisfied that all verification requirements have been properly completed and that there are no open issues or anomalies that have not been resolved.(1)

The review board will be charged with ensuring the system is man-rated.(1)

The vehicle shall be certified for flight only after the Safety/Certification Board is satisfied that all requirements for verification have been satisfactorily completed and that there are no open issues or anomalies that have not been resolved.(4)

A payload shall be certified safe in the applicable worst case natural and induced environments, as defined in the payload integration plan (PIP).(5)

a) Series payloads and reflown hardware must be recertified as safe.
b) The acceptance rationale for all deviations from previous flight procedures and operations must be validated by the payload organization.(5)

Flight safety reviews shall include a complete vehicle history for flight readiness, and a final technical evaluation.(8)

Functional Requirements:

None

8. OPERATIONS AND MAINTENANCE

Guidelines:

None

Functional Requirements:

None

8.1 Safe Operations

Guidelines:

Space system hardware should not be operated beyond its design limits. Subsystems should also not be operated in such a way as to compromise the safety of other hardware of the system. Limits of safety for crew activities will be determined and tested during mission simulations prior to the conduct of the mission.(1)

Crewmembers shall provide input to all phases of the design, development, testing, and evaluation of the system.(1)

Rescue operations after the initial emergency are as extensively planned as the vehicle systems.(2)

On-the-pad emergency egress procedures are mandatory in manned space vehicle operations.(2)

Range safety requirements have also been modified for manned launches. ...with man as a payload, destruction cannot be instantaneous: instead, a three-second margin...allowed for abort and escape.(2)

...consider alternate missions...instead of destroying the entire launch vehicle and aborting the crew.(2)

Test and Operating Procedures: Procedures developed for testing and operating spacecraft or ground support equipment shall clearly indicate any step which, if not correctly followed, would result in injury to personnel, damage to a system or equipment, or an environmental impact.(3)

Pressure Venting System: Crew cabin module pressure venting systems shall be designed such that the relief valves do not vent the crew cabin module atmosphere into space through compartments or outlets that are used to vent other fluids.(3)

Pressure Garments: Systems which supply pressure to the crew's pressure garments shall be designed so that a major failure of one crew member's garment or garment pressure supply will not cause loss of life of other crew members.(3)

The cover sheet of the procedure should identify it as a safety-critical operation.(3)

Payloads shall be designed to maintain fault tolerances or safety margins consistent with the hazard potential without ground or flight services.(5)

a) During emergency conditions, power will be provided temporarily to payloads for payload safing and verification if necessary.
b) All hazardous commands that can be sent to the payload shall be identified.
c) Monitoring shall be available to the launch site when necessary to assure safe ground operations.(5)

a. Handling and transportation Factors for Flight Structures: As a goal, flight structure design shall be based on flight loads and conditions rather than on transportation and handling loads. Transportation equipment design shall ensure that flight structures are not subjected to loads more severe than flight design conditions.(6)

		Yield	Ultimate (6)
1.	<u>General Safety Factors for Metallica Flight Structures</u>		
	Verified by Analysis Only	1.25	2.00
	Verified by Analysis and Static Test	1.10	1.40
2.	<u>General Safety Factors for Non-Metallica Flight Structures</u>		
	Verified by Analysis and Static Test		
	Non-Discontinuity Areas		1.4
	Discontinuity Areas and Joints		* 2.0
	*Note: Structural Test Factor = 1.4		
3.	<u>General Safety Factor for Solid Propellants</u>		
	Solid Propellant, Insulation, Liner, and Inhibitor		2.0

4. <u>Safety Factors for Pressures</u>

a. Propellant Tanks:
- Proof Pressure = 1.05 x limit pressure
- Yield Pressure = 1.10 x limit pressure
- Ultimate Pressure = 1.40 x limit pressure

b. Solid Motor Castings:
- **Proof Pressure = 1.05 x limit pressure
- Yield Pressure = 1.20 x limit pressure
- Ultimate Pressure = 1.40 x limit pressure

c. Windows, Doors, Hatches, etc.
Internal Pressure Only:
- Proof Pressure = 2.00 x limit pressure
- Ultimate Pressure = 3.00 x limit pressure

d. Engine Structures and Components:
- **Proof Pressure = 1.20 x limit pressure
- Yield Pressure = 1.50 x limit pressure

e. Hydraulic and Pneumatic Systems, including reservoirs:

1. Lines and Fittings, less than 1.5 inches (38 mm) diameter:
   - Proof Pressure = 2.0 x limit pressure
   - Ultimate Pressure = 4.0 x limit pressure
2. Lines and Fittings, less than 1.5 inches (38 mm) diameter:
   - Proof Pressure = 1.2 x limit pressure
   - Ultimate Pressure = 1.5 x limit pressure
3. Reservoirs:
   - Proof Pressure = 1.5 x limit pressure
   - Ultimate Pressure = 2.0 x limit pressure
4. Actuating Cylinders, Valves, Filters, Switches:
   - Proof Pressure = 1.5 x limit pressure
   - Ultimate Pressure = 2.0 x limit pressure

f. Personnel Compartments, Internal Pressure Only:
- Proof Pressure = 1.50 x limit pressure
- Yield Pressure = 1.65 x limit pressure
- Ultimate Pressure = 1.65 x limit pressure

**Note: Proof factor determined from fracture mechanics service life analysis must be used in if greater than those shown.(6)

All flight constraints and procedures for nominal and abort operations shall be maintained. A thorough knowledge of the vehicle divergence characteristics is necessary, in particular, of vehicle break-up or of exceeding crew physiological limits.(8)

Functional Requirements:

...special testing to assure compatibility of the spacecraft with the launch vehicle.(2)
...proper operation of emergency detection...systems(2)
...proper operation of...abort systems(2)
...all test procedures that are accomplished must be correct and formally revised before the procedure is accomplished.(2)
Generally, on a manned launch, pad access is strictly controlled...(2)
GSE and ASE Protective Devices: Ground support equipment (GSE), airborne support equipment (ASE), facility equipment, or test equipment used in ground or flight operations shall be equipped with protective devices to preserve safe operating margins of the spacecraft subsystems.(3)
Cabin Ventilation: Crew cabin module ventilation fans shall be protected by screens or other devices to prevent entrance of debris that could damage or jam the fan blades during zero-gravity conditions. Such screens or other devices shall be serviceable and/or replaceable.(3)

8.2 Periodic Check-Out and Maintenance

Guidelines:

The operating life of each critical subsystem should be conservatively determined. A schedule for the periodic check-out of such subsystems should be maintained. Periodic maintenance should be performed as required, to sustain the safety margins of critical subsystems. Ease of access and maintainability should be designed in.(1)

Control of Limited-Life Components: Appropriate documentation shall accompany all time-critical and limited-life items and shall include the date of manufacture of the item and of its most time-critical component. Realistic life limits shall be assigned and documented for each item and shall be suitably altered as new data and new evidence are obtained.(3)

Servicing/Testing Port Capping: Servicing and test ports shall be designed such that they can be capped immediately after servicing or test in order to preclude leakage in flight.(3)

Venting-Induced Forces: Sources of venting that could occur during the mission shall be identified and an analysis made to ensure that the total vent condition is designed to be compatible with vehicle and/or mission control capabilities. Impingement of vent plumes on spacecraft elements shall be analyzed.(3)

Each time-critical or limited-life assembly, subassembly, component, and spare shall be clearly and indelibly marked with a serial number.
Status records shall be maintained on all such items after installation in the spacecraft.(3)

Special storage requirements shall be carefully defined and strictly observed.(3)

a) Limited Life Items: All safety critical age sensitive equipment must be refurbished or replaced.
b) Refurbishment: Safety impact of any changes, maintenance or refurbishment made to the hardware or operating procedures must be assessed and reported in safety assessment reviews.(5)

Functional Requirements:

A history of system performance shall be maintained and monitored to detect possible degradations over time. Fail-operational/fail-safe subsystems shall allow maintenance to upgrade the subsystem without being degraded below fail-safe during maintenance actions.(1)

Prelaunch Nozzle/Vent Protection: All nozzles and vents used in manned spacecraft systems, such as those of the reaction control system and environmental systems, shall be protected from entrance of rain, debris, or other contaminants prior to launch.(3)

Electrical and fluid-handling subsystems shall include checkout test points which will permit checkout tests to be made without disconnecting tubing or electrical connectors normally connected in flight.(3)

These ports will not utilize permanent closure methods.(3)

Nonpropulsive vent concepts, opposed venting, operational procedures, or similar methods shall be used to eliminate the undesirable effects of perturbing forces resulting from such vents.(3)

Protective covers for nozzles or vents located within the payload by shall be designed to be readily removable during the countdown prior to launch or prior to final closure of the Orbiter payload bay doors.(3)

8.3 Repairability

Guidelines:

Components of critical subsystems should have sufficient accessibility for replacement or repair during extended-duration missions. Spare requirements should be determined from detailed analysis of subsystem performance and life test data. The mission duration of a quiescent system should include the time it is dormant.(1)

Spare critical components should be available for the replacement during the missions.(1)

Equipment Accessibility: Systems, Subsystems, equipment, and components shall be designed with features that contribute to the ease and rapidity of maintenance by both humans and robots.(3)

Equipment expected to require servicing, replacement, or maintenance shall be accessible without the removal of other equipment, wire bundles, and fluid lines. This should include accessibility during ground operations as well as on orbit.equipment expected to require servicing, replacement, or maintenance shall be accessible without the removal of other equipment, wire bundles, and fluid lines. This should include accessibility during ground operations as well as on orbit.(3)

Electrical connectors and cable installations shall permit disconnection and reconnection without damage to wiring or connectors.(3)

Payloads shall be designed such that any required access to hardware during flight or ground operations (e.g., maintenance, repair) can be accomplished with minimum risk to personnel.(5)

Functional Requirements:

None

MAN-RATING REFERENCES

(1) NASA JSC-23211, "Guidelines for Man-rating Space Systems", M.P. Cerimele, November 1991.

(2) NASW 410-24-13-1, NAS-CR-118614, "Launch Vehicle Man-rating" (all volumes), F. E. Miller, R. E. Ehrlich and J. M. Sullivan, December 1968.

(3) NASA-JSC, JSCM 8080, "Manned Spacecraft Criteria and Standards", February,1980.

(4) Eagle Engineering/LEMSCO Report No. 88-193, Contract No. NAS17900, "A Review of Man-rating in Past and Current Manned Space Flight Programs", A. C. Bond, May 1988.

(5) NSTS-1700-7B, "Safety Policy and Requirements for Payloads Using the Space Transportation System", January 1989.

(6) MSFC-HDBK-505A, "Structural Strength Program Requirements", April 1989.

(7) "Man-rating of a Launch Vehicle", D. Soffker, ERNO Raumfahrttechnik GmbH, January 1982.

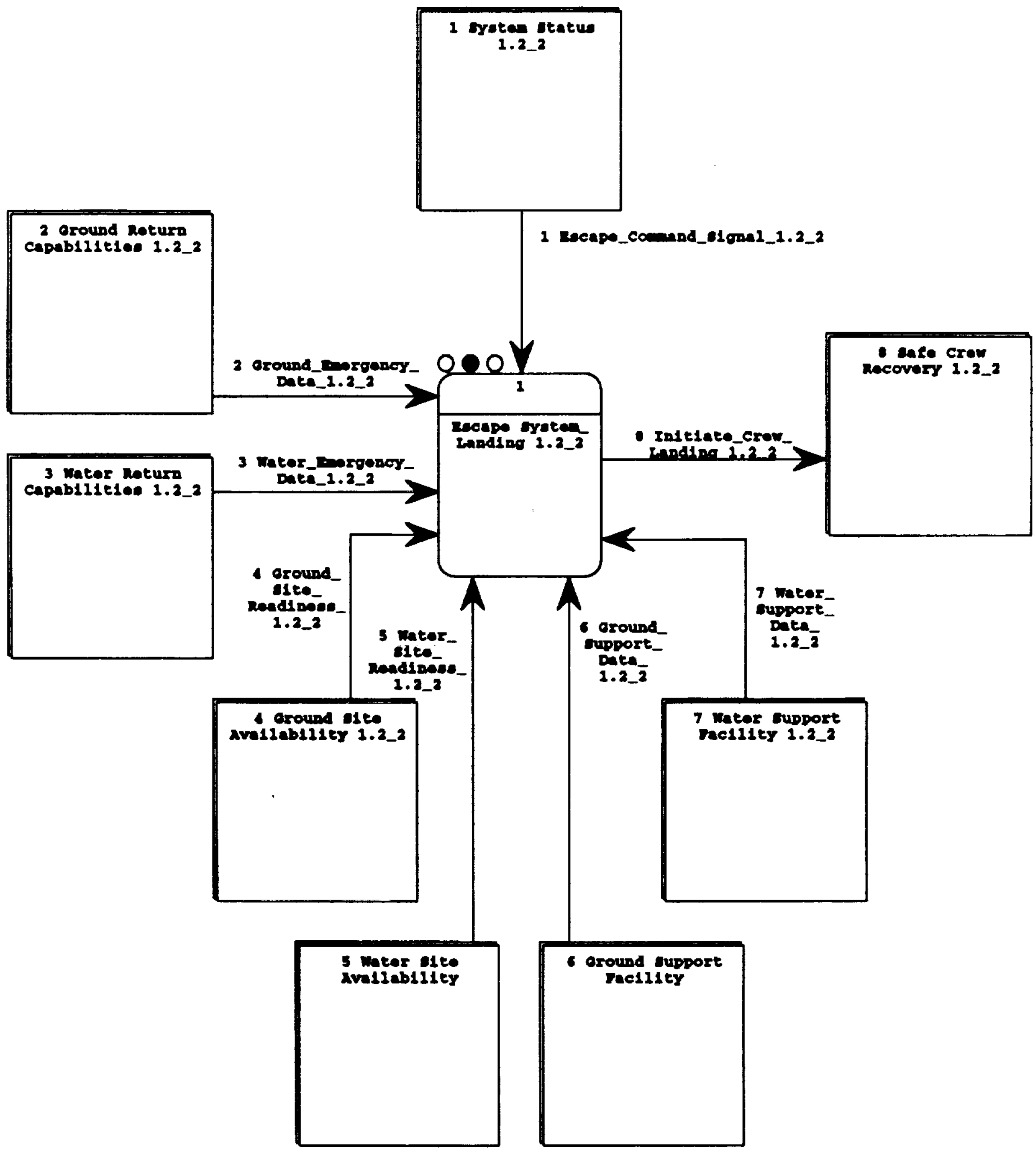
(8) USAF Report No. SSD-TR-66-202, "Structural Considerations in Man-rating the Titan Missile for the Gemini Program", B. A. Hohmann and T. Shiokari, September 1966.

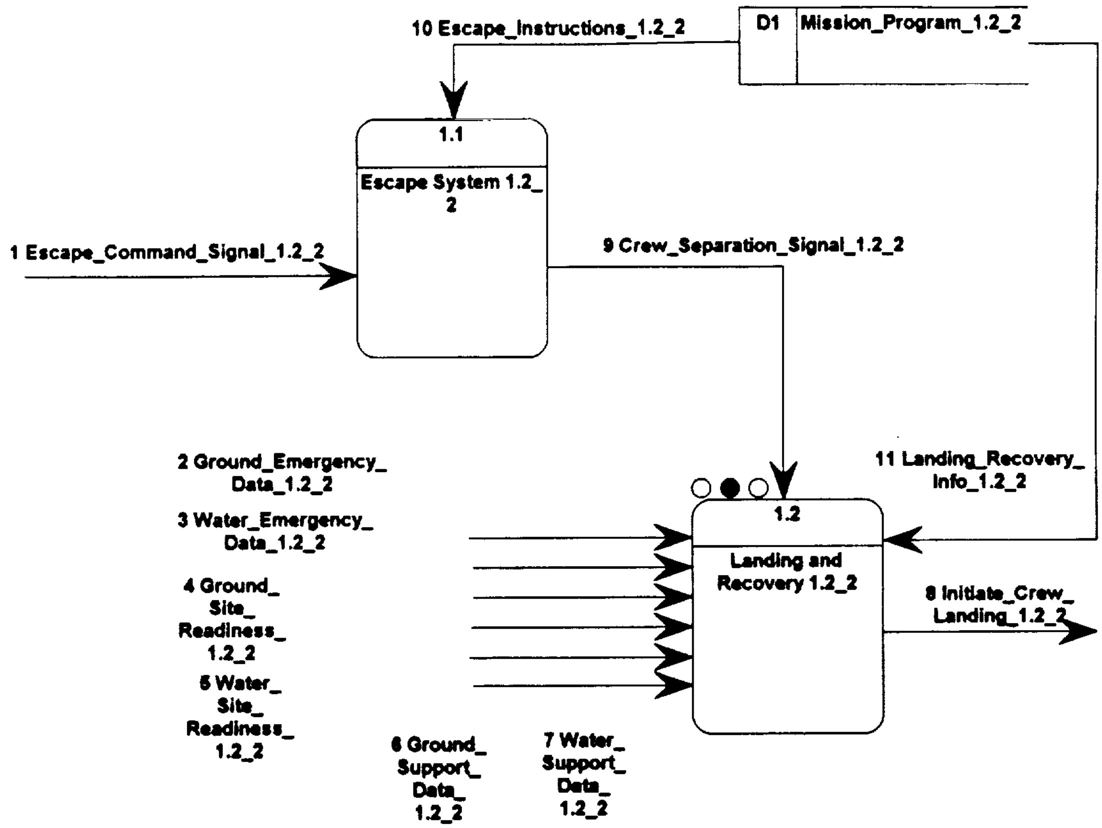
APPENDIX A

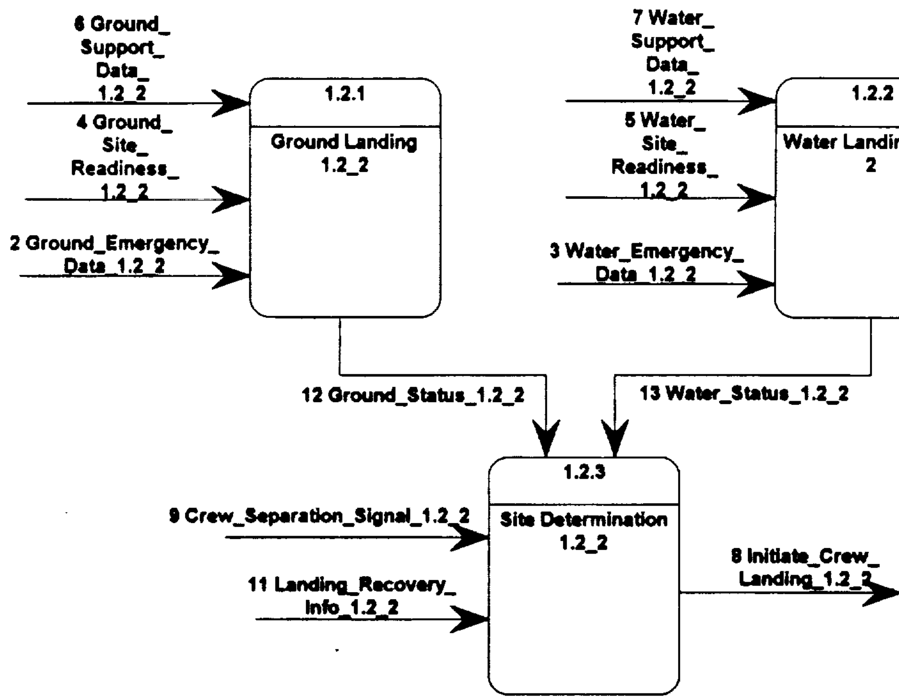
Functional Flow Block Diagrams

MR_1.2_2
System Architect
Mon Aug 31, 1992 08:41

1.2 Escape System:
The provision for a safe landing area for the spacecraft and the surrounding facilities is required. (2)



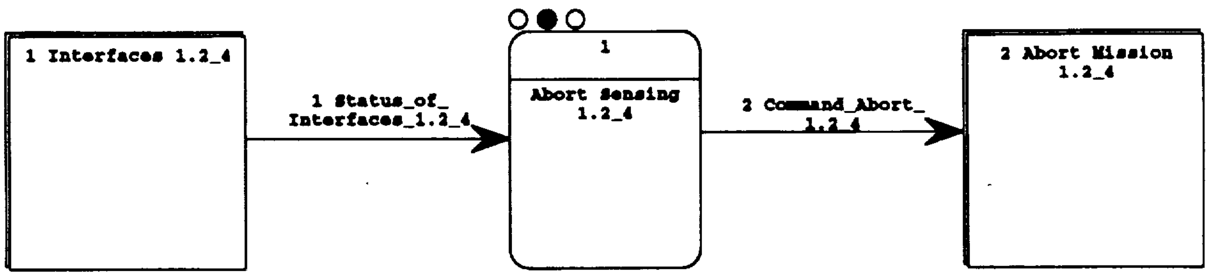

10 Escape_Instructions_1.2_2
D1
Mission_Program_1.2_2
1.1
Escape System 1.2_2
1 Escape_Command_Signal_1.2_2
9 Crew_Separation_Signal_1.2_2
2 Ground_Emergency_Data_1.2_2
3 Water_Emergency_Data_1.2_2
4 Ground_Site_Readiness_1.2_2
5 Water_Site_Readiness_1.2_2
6 Ground_Support_Data_1.2_2
7 Water_Support_Data_1.2_2
11 Landing_Recovery_Info_1.2_2
1.2
Landing and Recovery 1.2_2
8 Initiate_Crew_Landing_1.2_2

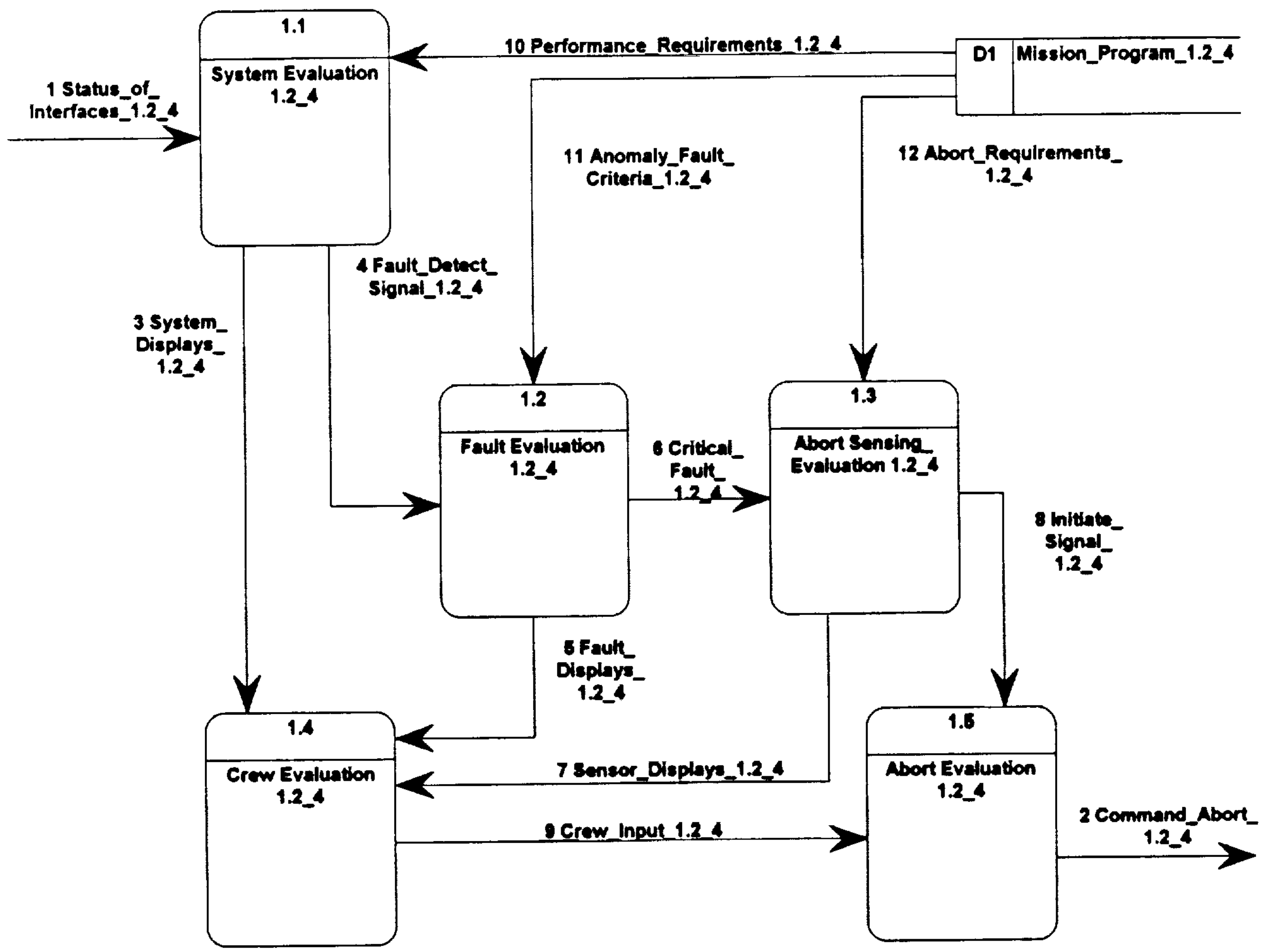


6 Ground_ Support_ Data_ 1.2_2
4 Ground_ Site_ Readiness_ 1.2_2
2 Ground_Emergency_ Data_1.2_2
1.2.1
Ground Landing 1.2_2
7 Water_ Support_ Data_ 1.2_2
5 Water_ Site_ Readiness_ 1.2_2
3 Water_Emergency_ Data_1.2_2
1.2.2
Water Landing 1.2_ 2
12 Ground_Status_1.2_2
13 Water_Status_1.2_2
9 Crew_Separation_Signal_1.2_2
11 Landing_Recovery_ Info_1.2_2
1.2.3
Site Determination 1.2_2
8 Initiate_Crew_ Landing_1.2_2

MR_1.2_4
System Architect
Mon Aug 31, 1992 08:55

1.2 **Escape System:**
An escape system has the attendant requirement for providing the crew with abort sensing and implementation data upon which to base an abort decision. (4)




1.1
System Evaluation 1.2_4
1.2
Fault Evaluation 1.2_4
1.3
Abort Sensing_ Evaluation 1.2_4
1.4
Crew Evaluation 1.2_4
1.5
Abort Evaluation 1.2_4
D1
Mission_Program_1.2_4
1 Status_of_ Interfaces_1.2_4
2 Command_Abort_ 1.2_4
3 System_ Displays_ 1.2_4
4 Fault_Detect_ Signal_1.2_4
5 Fault_ Displays_ 1.2_4
6 Critical_ Fault_ 1.2_4
7 Sensor_Displays_1.2_4
8 Initiate_ Signal_ 1.2_4
9 Crew_Input_1.2_4
10 Performance_Requirements_1.2_4
11 Anomaly_Fault_ Criteria_1.2_4
12 Abort_Requirements_ 1.2_4

MR_1.3_1
System Architect
Mon Aug 31, 1992 09:05

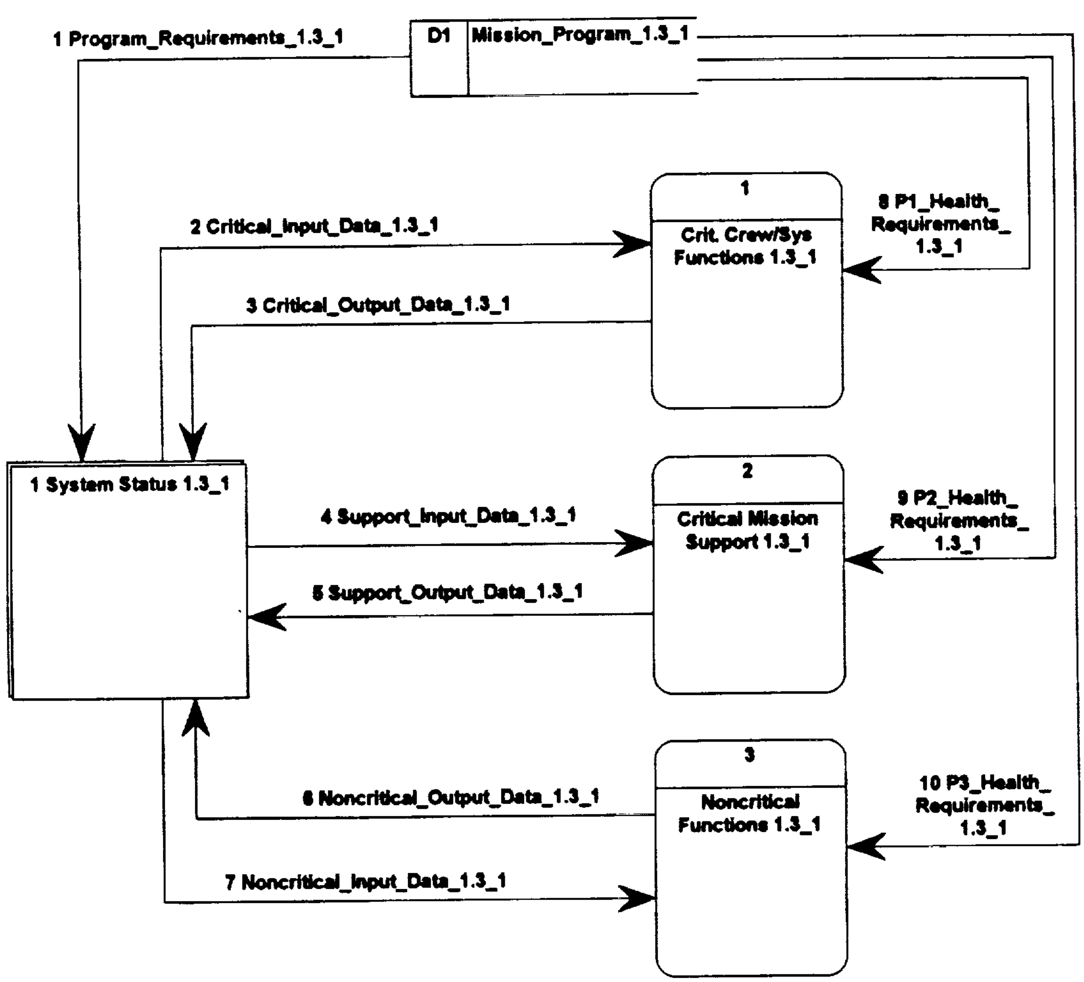
1.3 Failure Tolerance (1,4,5,7)

All critical crew/system functions shall be designed to be two-failure tolerant as a minimum. Functions essential for critical mission support shall be designed to be single failure tolerant as a minimum. Noncritical functions shall be designed to fail in safe mode. Total functional failure of a subsystem should not be allowed to propagate to other systems. (1)

No single failure shall result in a critical hazard. No two failures shall result in a catastrophic hazard. (4)

The payload must tolerate a minimum of credible failures and/or operator errors determined by the hazard level.
a) Critical hazards shall be controlled such that no single failure or operator error can result in damage to launch system equipment, a nondisabling personnel injury or the use of unscheduled safing procedures that effect operations of the launch system or another payload.
b) Catastrophic hazards shall be controlled such that no combination of two failures or operator errors can result in the potential for a disabling or fatal personnel injury, loss of the launch vehicle, ground facilities or launch system equipment. (5)

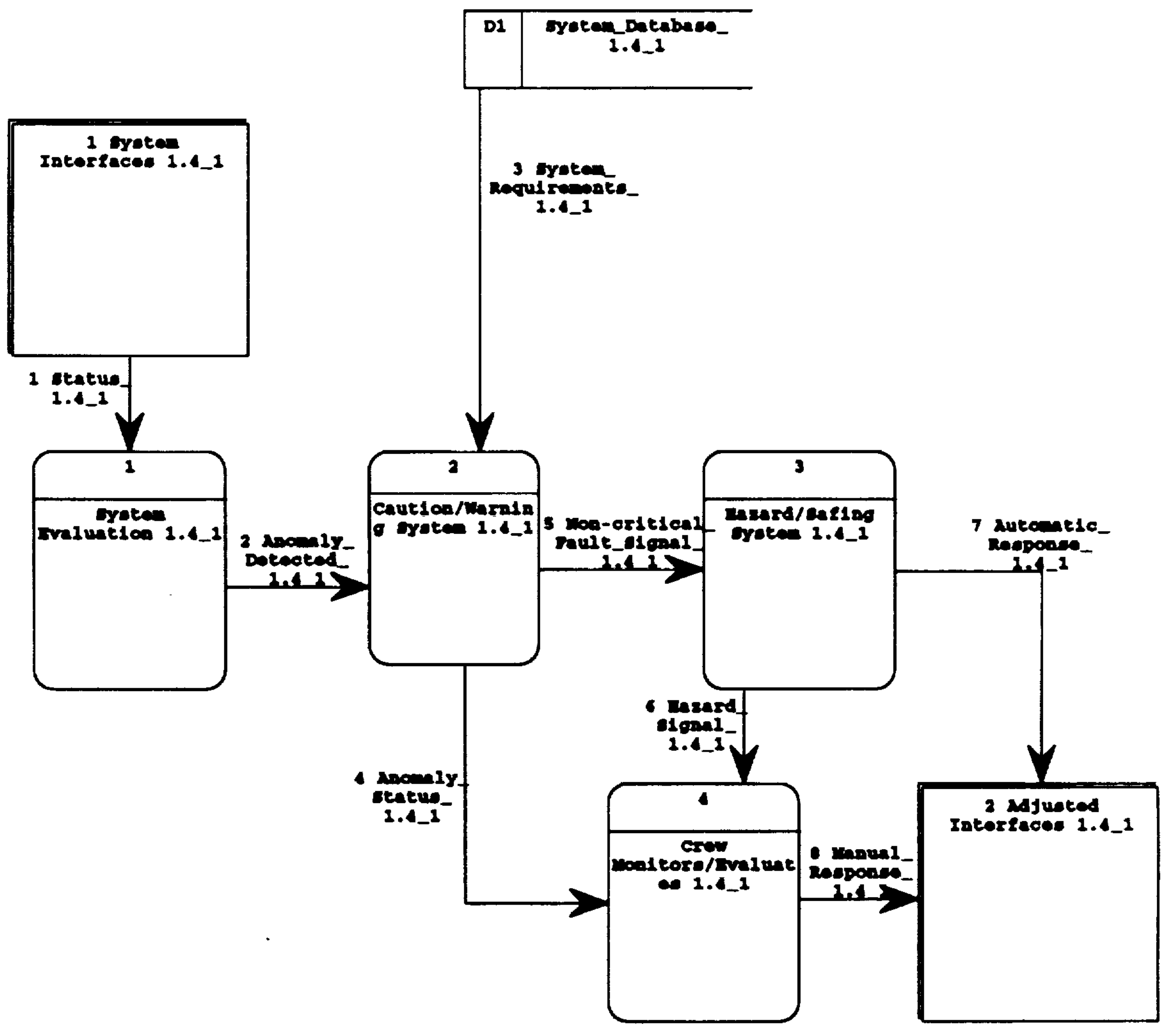
In case of failure of an essential function for the crew, the system must still be in a safe condition, i.e. the system must be at least fail-safe. For safety essential functions, redundancies or back-up features have to be incorporated in to the design without considering any failure probabilities. (7)



MR_1.4_1
System Architect
Mon Aug 31, 1992 10:24

1.4 Hazard Detection and Safing (1a)

A caution and warning system should be provided which will identify equipment failures, fire, or other potential emergency situations. This does not rule out an automatic hazard safing system for instances of slow crew response time. (1)

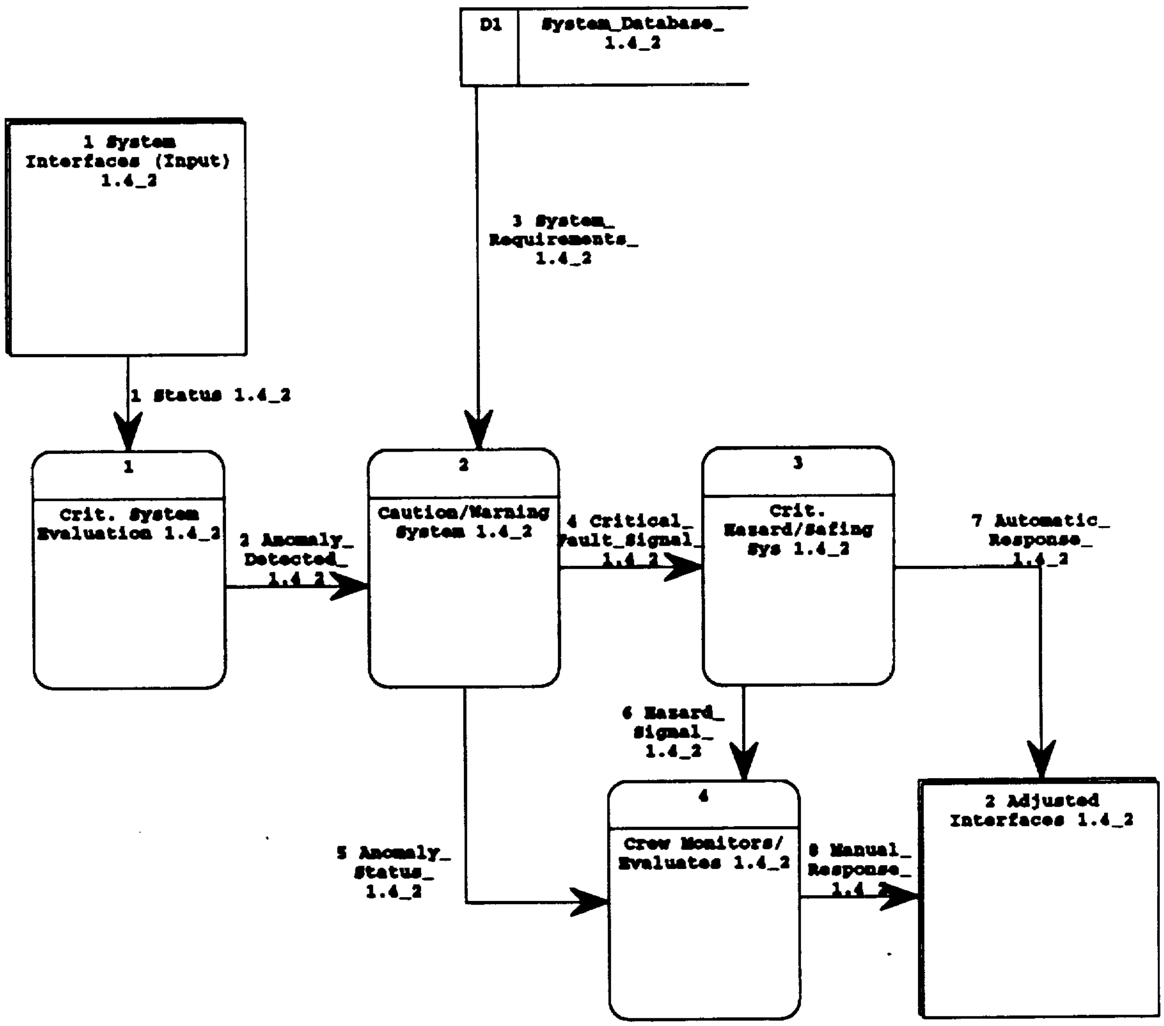


MR_1.4_2
System Architect
Mon Aug 31, 1992 16:59

1.4 Hazard Detection and Safing (1b)

The space system should provide a fault detection, isolation and recovery systems addressing problems in critical and non-critical systems over which the crew has control. The status of critical systems shall be displayed in a manner that prevents misinterpretation. Fire suppression capability should be provided in local and generadl areas, and may be either automatic or manual depending on the type of risk. (1)

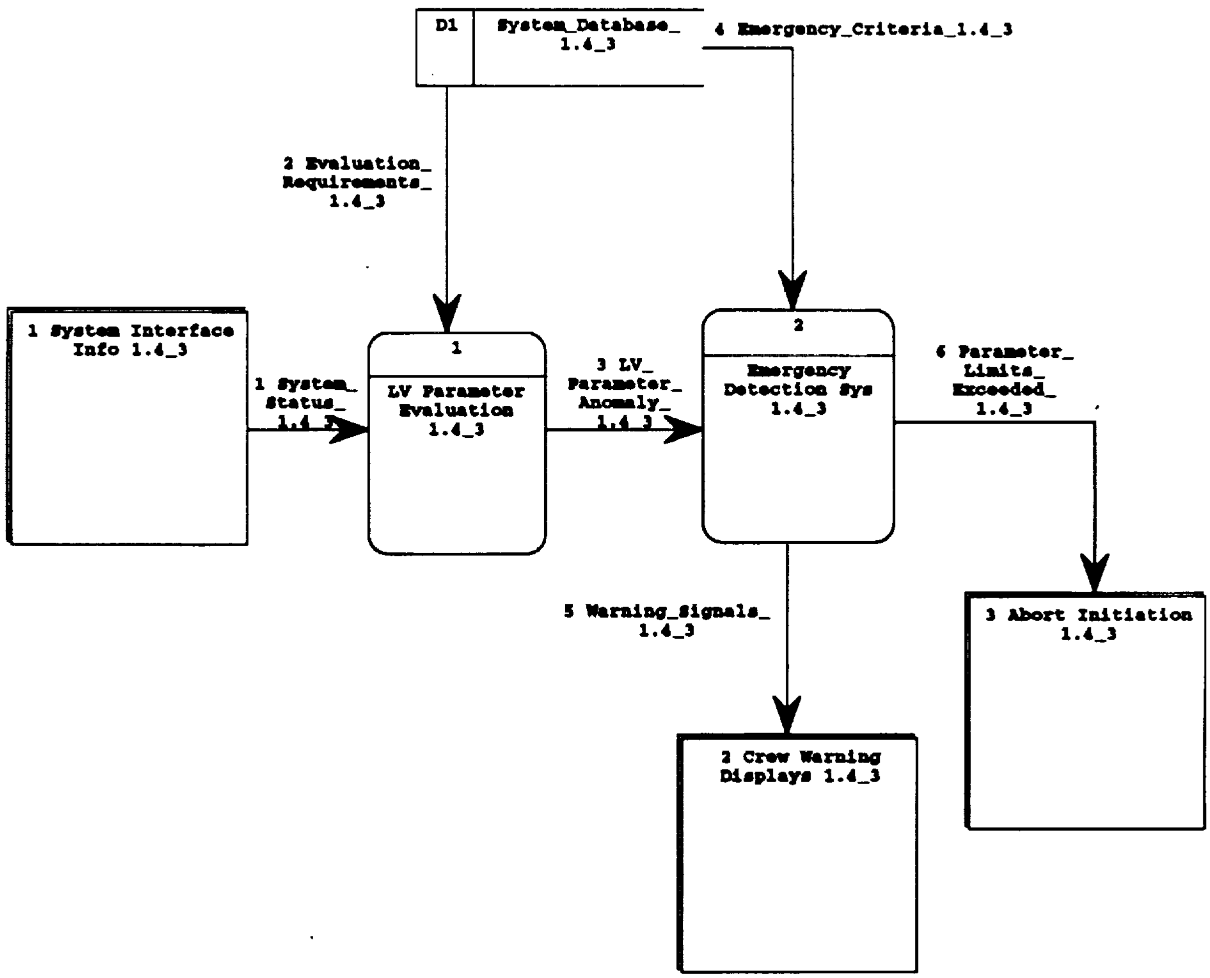
The crew shall be provided with a caution and warning system which will identify equipment failures, fire, or other potential emergency situations. Further, the vehicle shall be provided with a fault detection, isolation, and recovery system which will address problems in critical and non-critical systems. (4)



MR_1.4_3
System Architect
Mon Aug 31, 1992 12:03

1.4 Hazard Detection and Safing (2a)

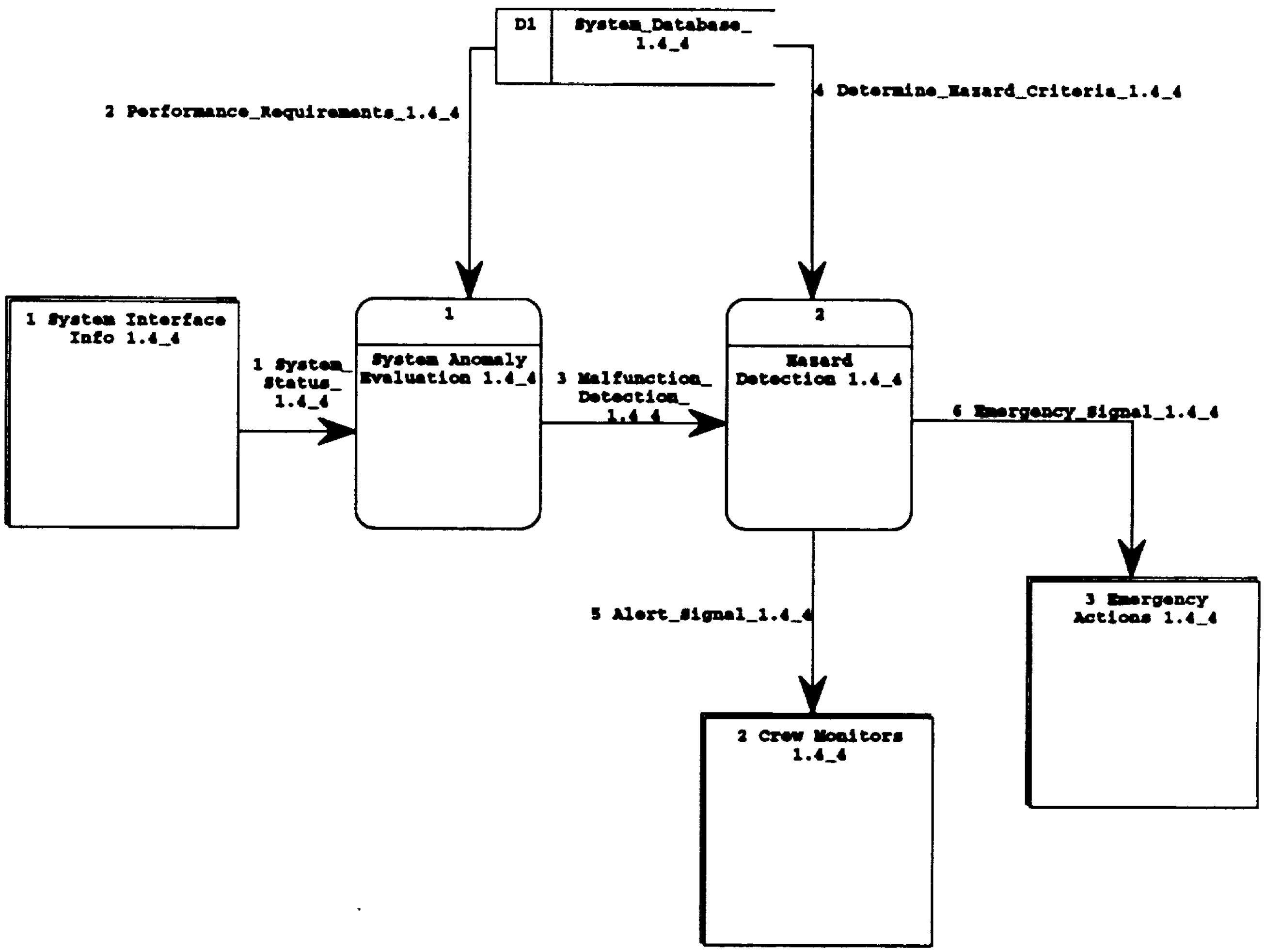
It is the function of the emergency detection system to monitor predetermined launch vehicle parameters, and supply a signal to selected crew warning displays and the abort initiation system when parametric limits are exceeded. (2)



MR_1.4_4
System Architect
Mon Aug 31, 1992 12:23

1.4 Hazard Detection and Safing (2b)

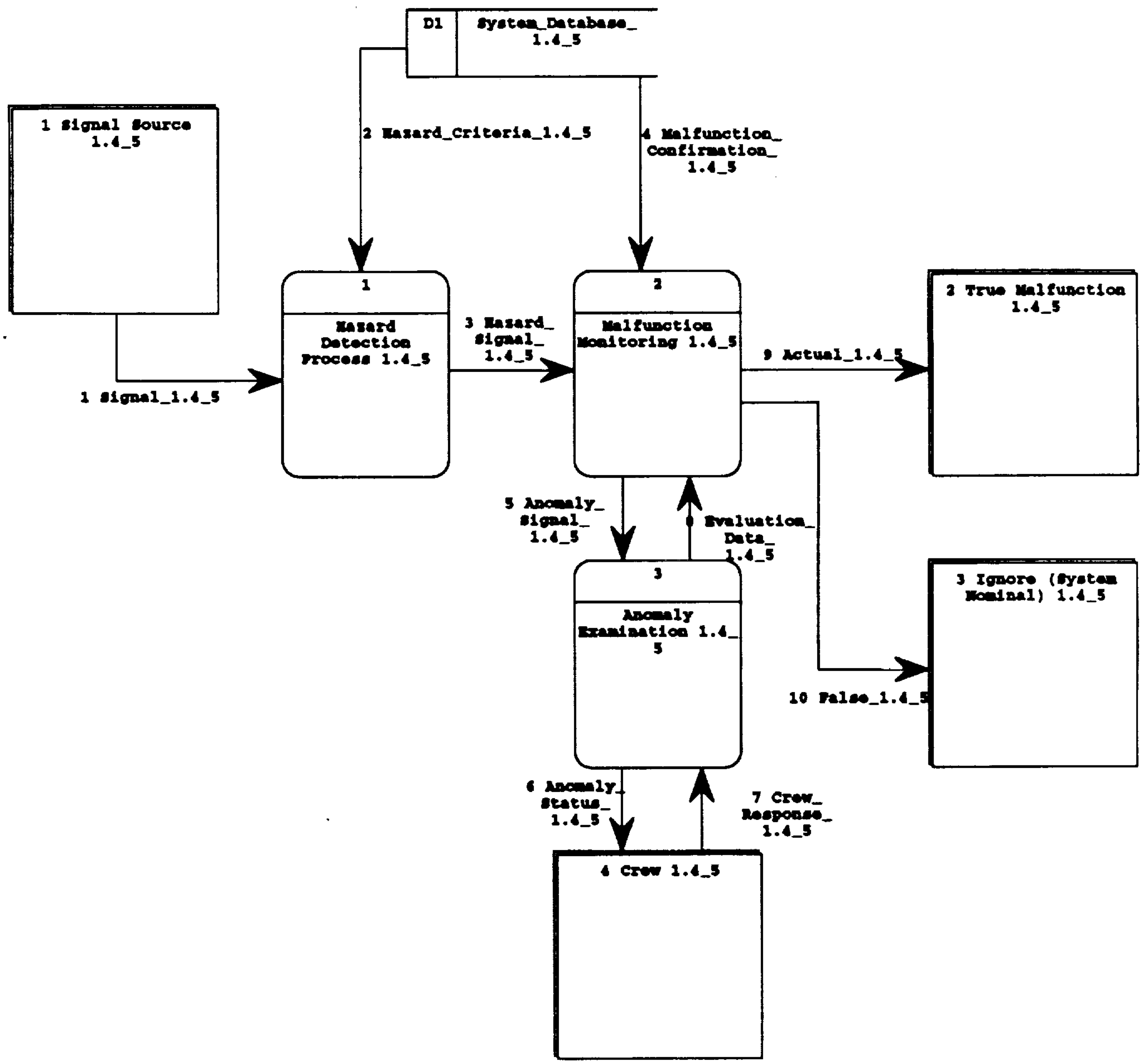
Malfunction detection to survey, alert, and provide the signal for emergency action. (2)



MR_1.4_5
System Architect
Mon Aug 31, 1992 12:50

1.4 Hazard Detection and Safing (2c)

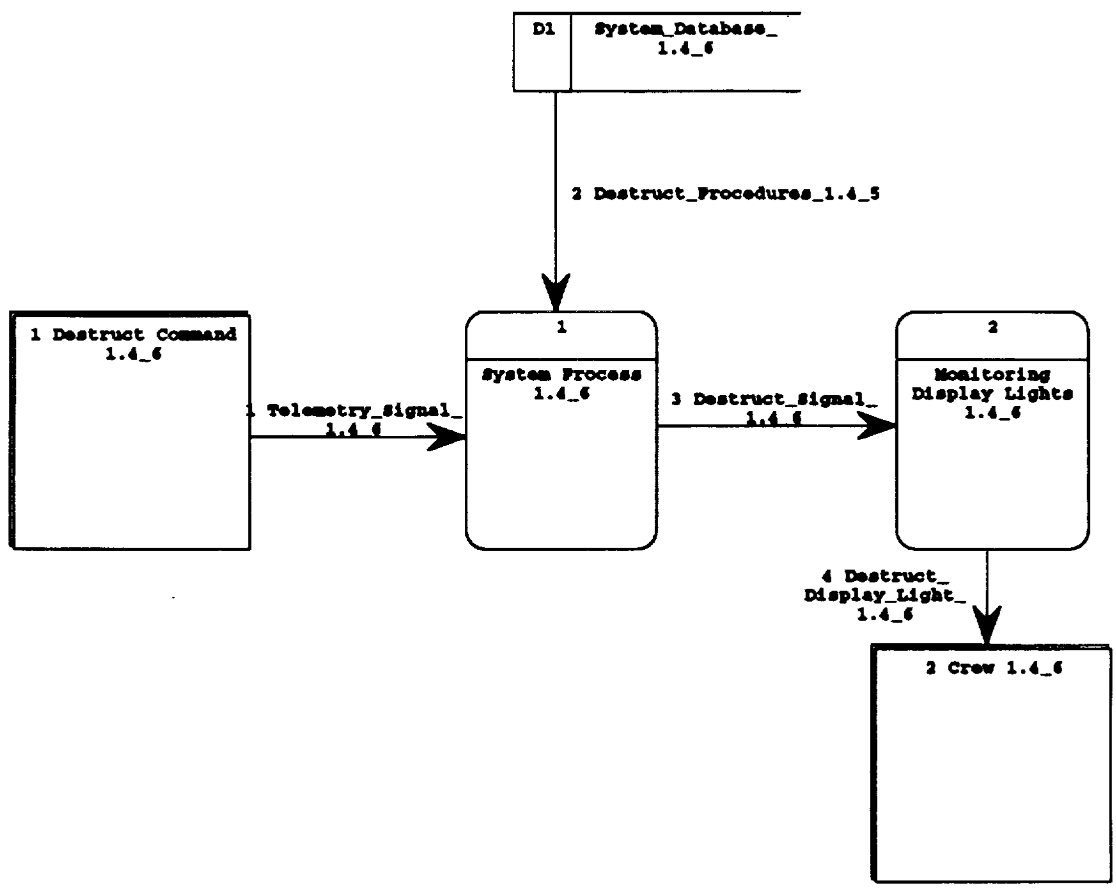
The pilot or controller should have sufficient displays...to separate actual malfunctions from false malfunctions. (2)



MR_1.4_6
System Architect
Mon Aug 31, 1992 12:58

1.4 Hazard Detection and Safing (2d)

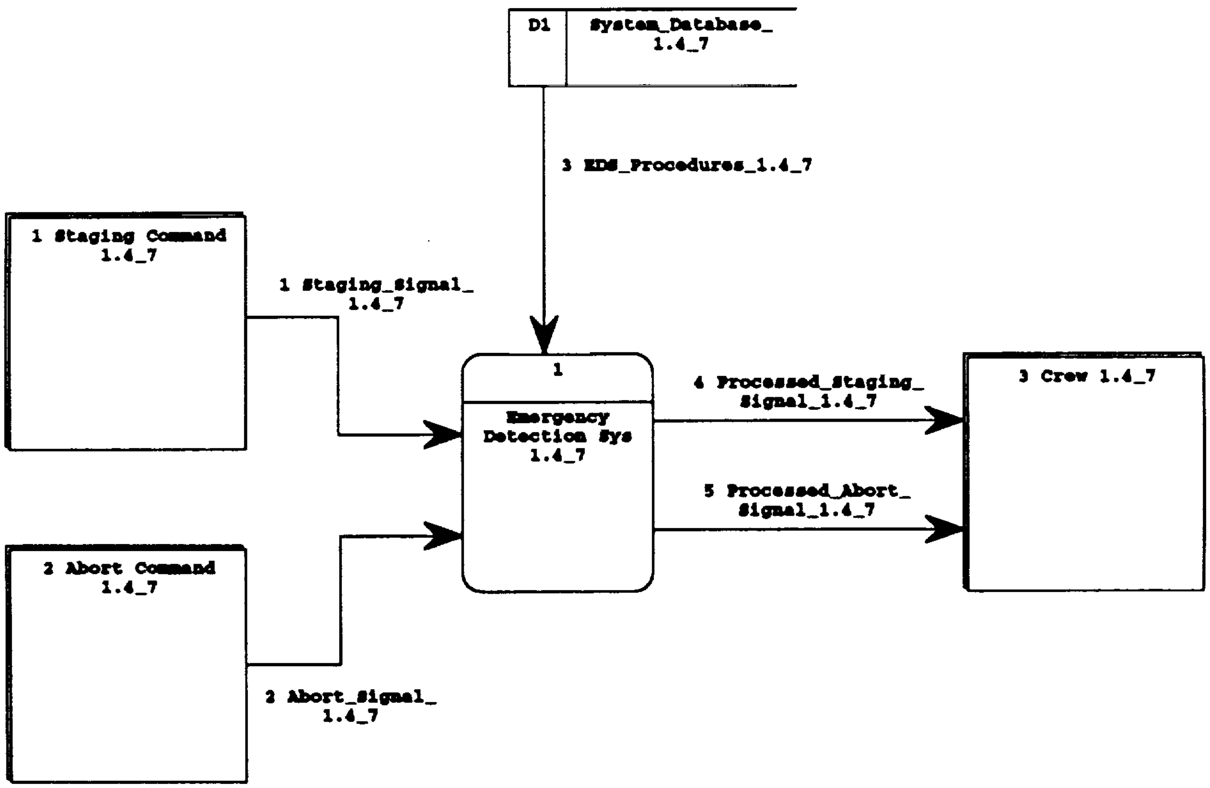
In all cases...the destruct command signal is displayed to the crew as a light. (2)



MR_1.4_7
System Architect
Mon Aug 31, 1992 14:36

1.4 Hazard Detection and Safing (2e)

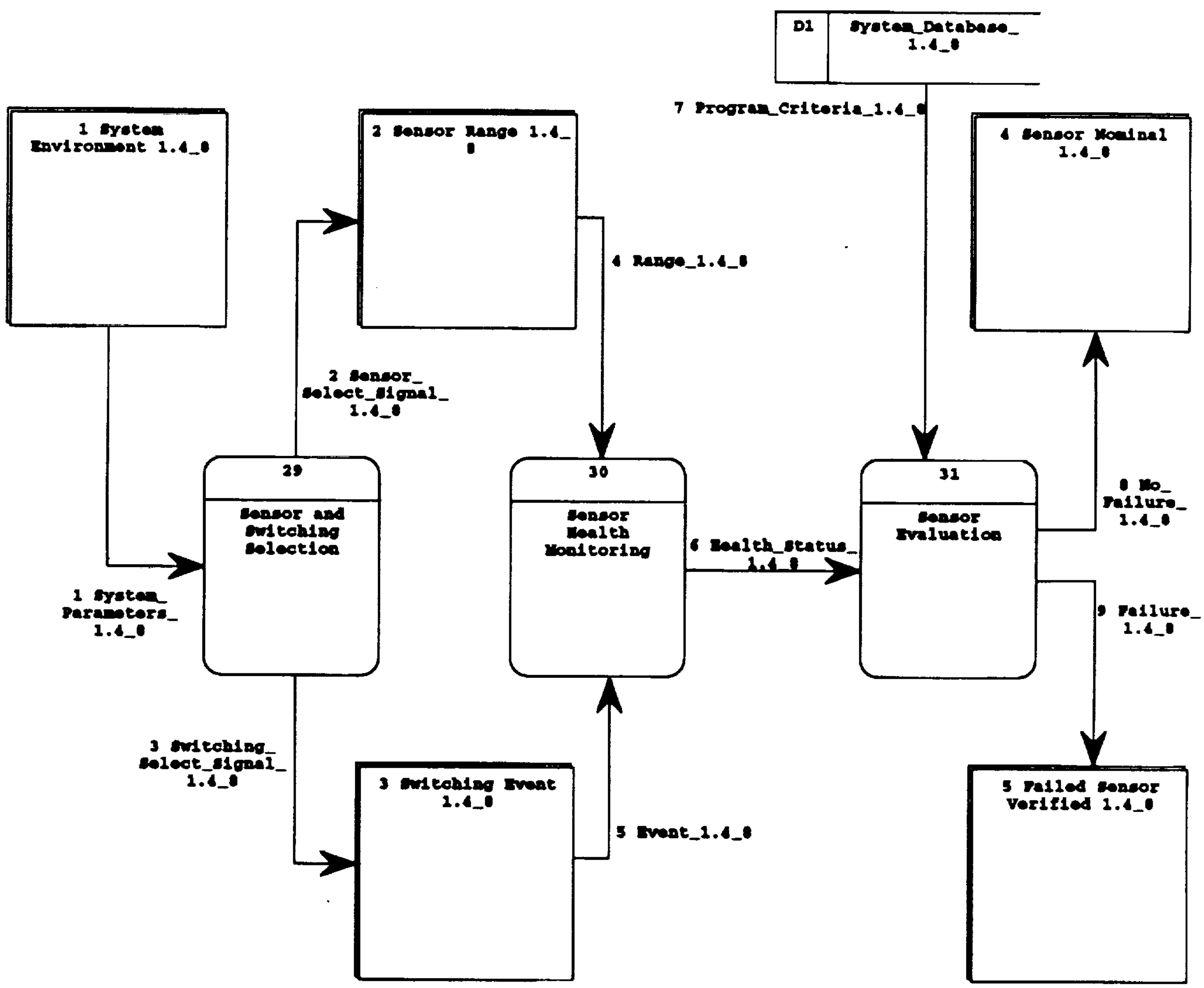
Abort, staging command, stage separation signals are common to all EDSs. (2)



MR_1.4_8
System Architect
Mon Aug 31, 1992 14:56

1.4 Hazard Detection and Safing (2f)

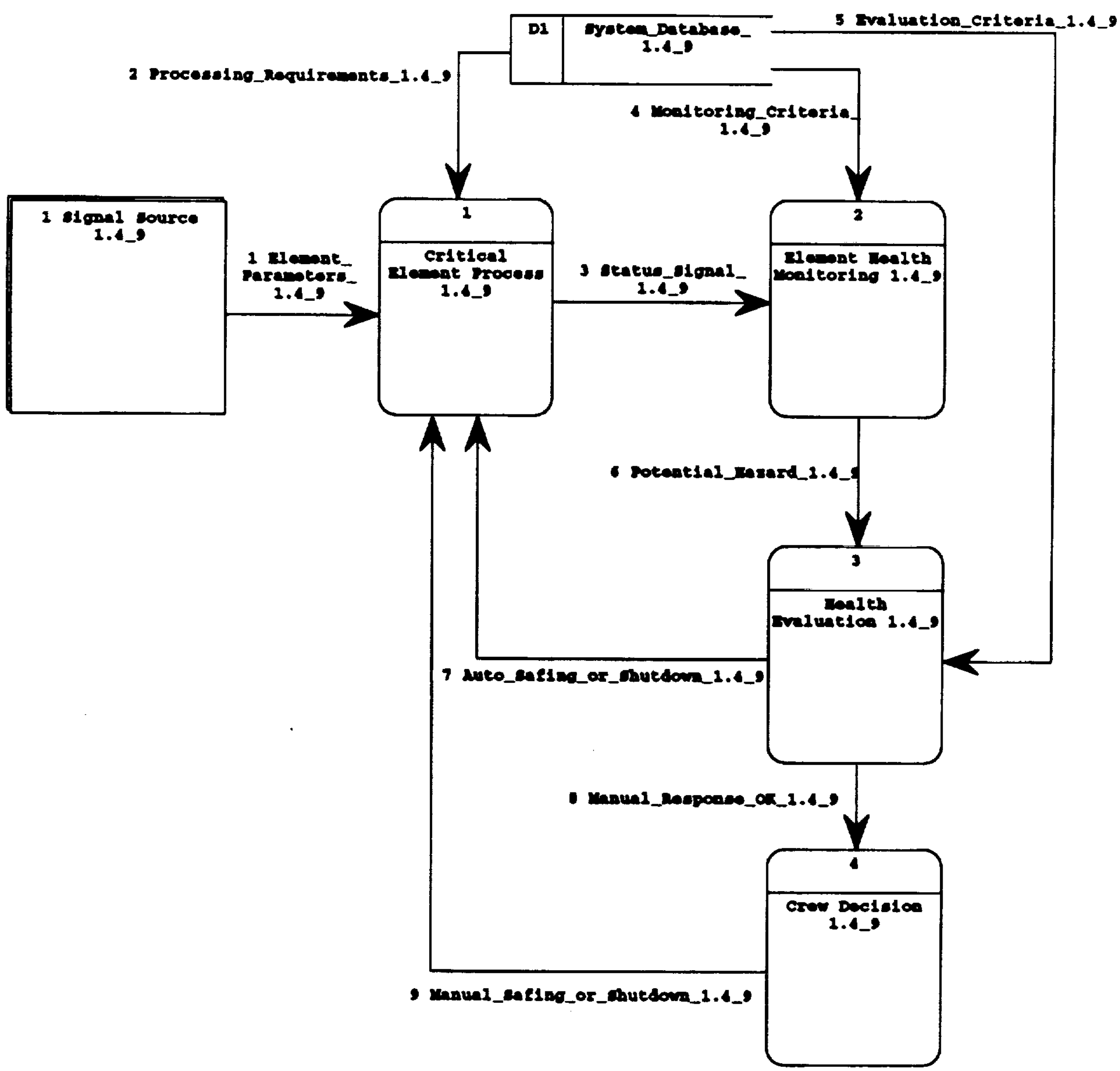
After selection of the sensor range and switching events, adequate provisions for sensor failure must be made. (2)



MR_1.4_9
System Architect
Mon Aug 31, 1992 15:16

1.4 Hazard Detection and Safing (2g)

Provision for safe control and/or shutdown of critical elements by automatic or manual systems. (2)

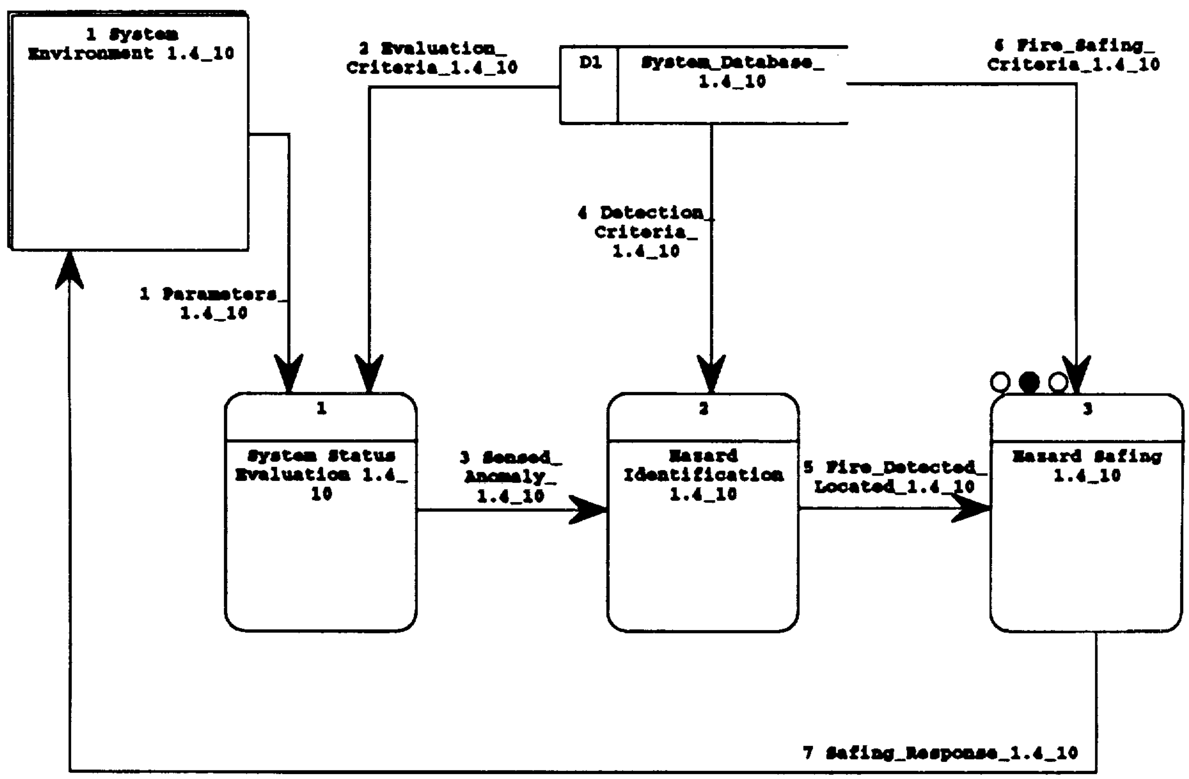


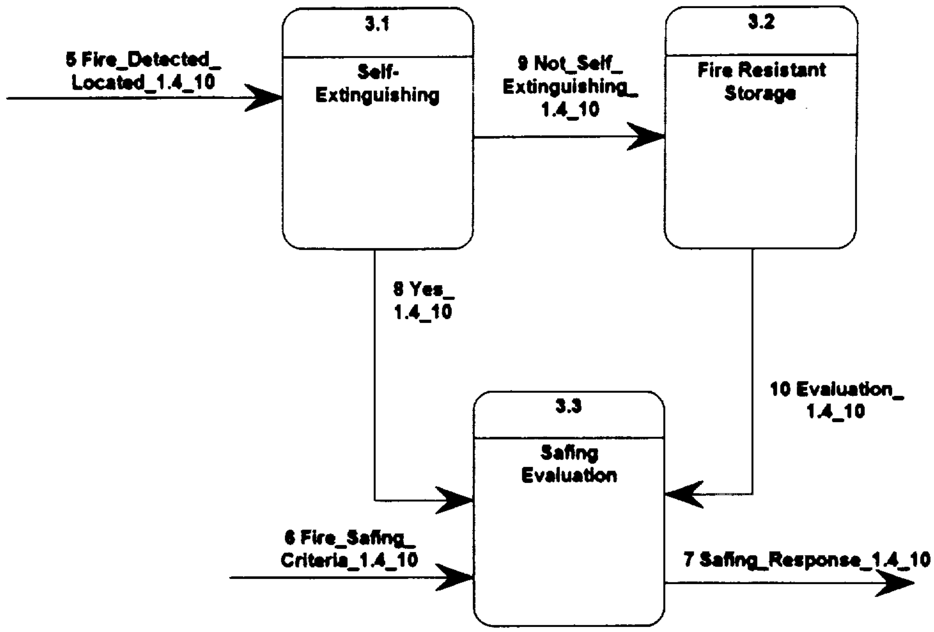
MR_1.4_10
System Architect
Mon Aug 31, 1992 16:47

1.4 Hazard Detection and Safing (3a)

Fire Control: The capability shall be provided to detect any fire. Means shall be provided for fire-resistant storage of all items that are not self-extinguishing when they are not in use. (3)

Interior walls and secondary structures shall be self-extinguishing. All combustibles shall be self-extinguishing in the most severe oxidizing environment to which they will be exposed. The material used to extinguish fires must be nontoxic and capable of being easily cleaned up after use. (3)

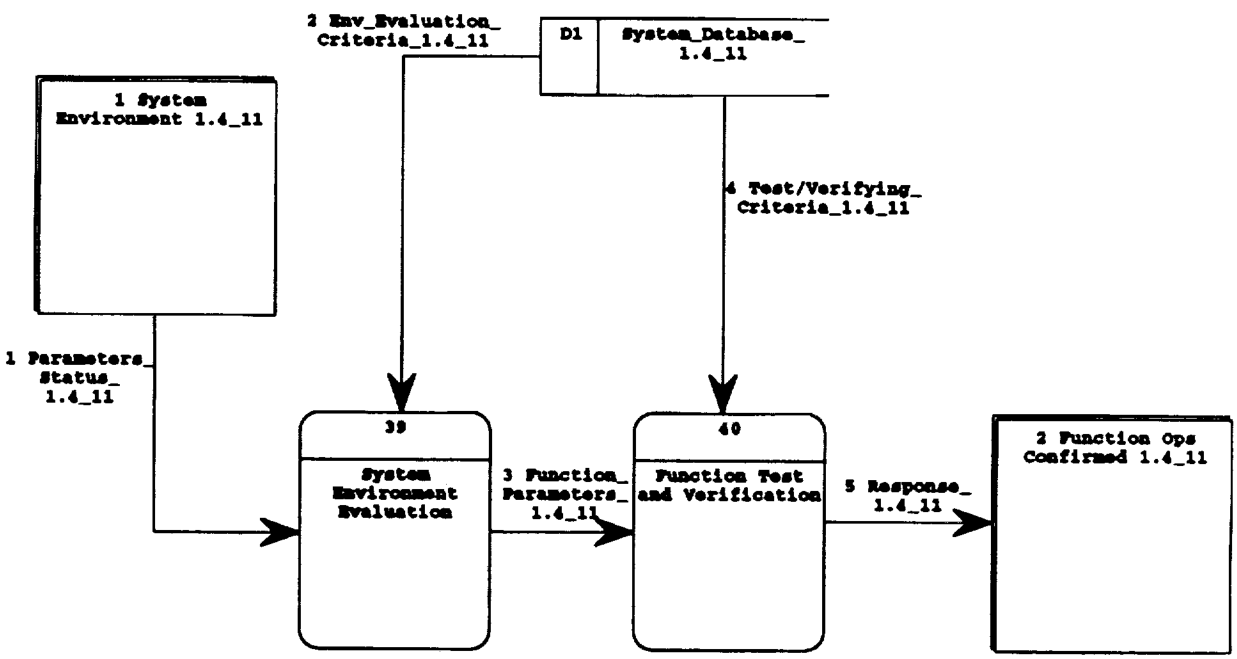



3.1
Self-
Extinguishing
3.2
Fire Resistant
Storage
3.3
Safing
Evaluation
5 Fire_Detected_
Located_1.4_10
9 Not_Self_
Extinguishing_
1.4_10
8 Yes_
1.4_10
10 Evaluation_
1.4_10
6 Fire_Safing_
Criteria_1.4_10
7 Safing_Response_1.4_10

MR_1.4_11
System Architect
Mon Aug 31, 1992 16:38

1.4 Hazard Detection and Safing (5a)

Appropriate functions, when implemented, shall be capable of being tested for proper operations during both ground and flight phases. (5)



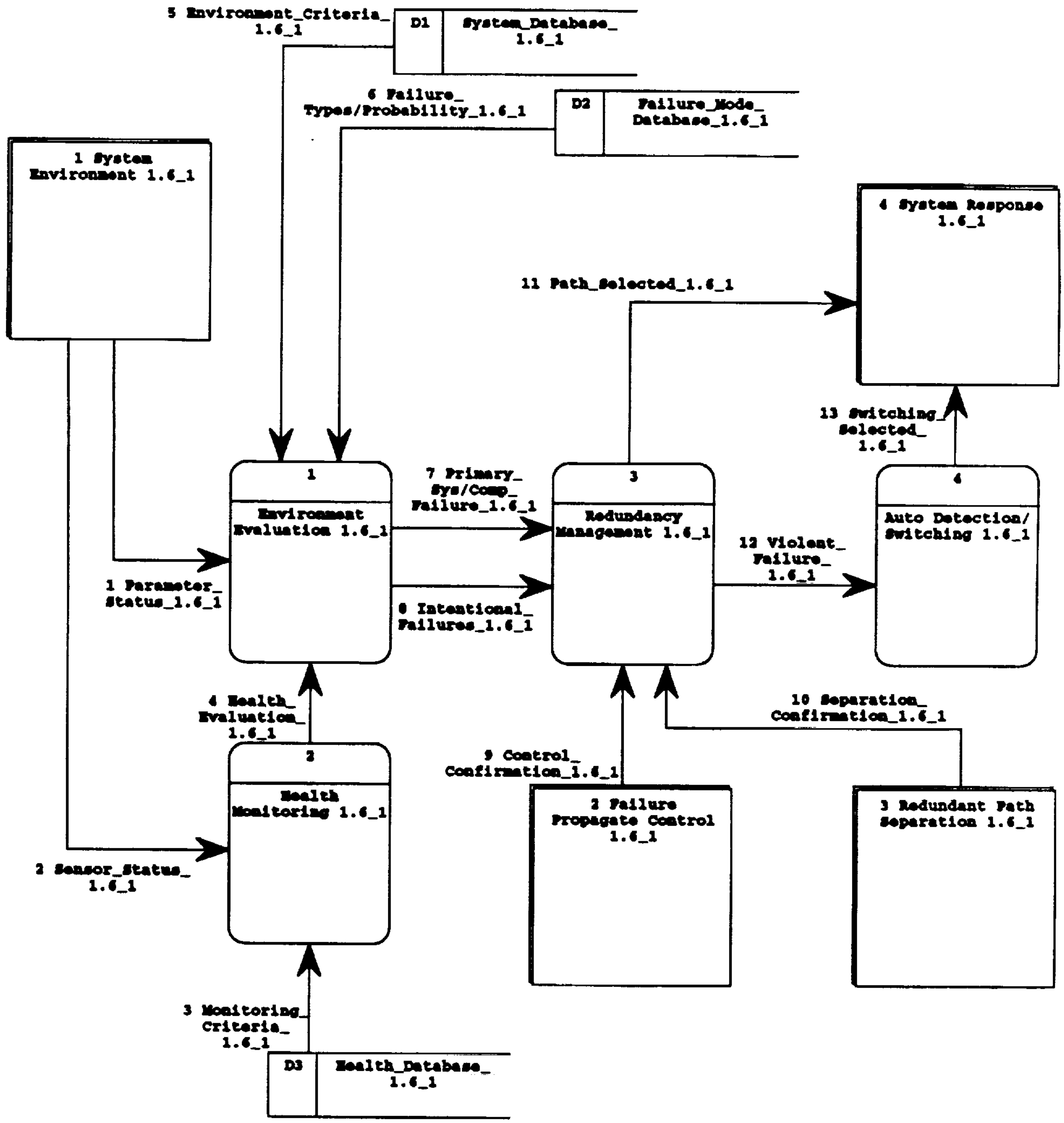
MR_1.6_1
System Architect
Tue Sep 01, 1992 10:42

1.6 Redundancy (1,2a)

Systems should be designed so that the interruption of gas flow, fluid, or electrical current should not, by itself, cause a critical condition. Single point failures and credible single failure modes shall be guarded against through separation of redundant paths, failure propagation control, and redundancy management.
Purely redundant systems or components should be prohibited from performing their function unless
1) a primary system or component has failed, or
2) the redundant systems or components are being intentionally tested.
This "standby redundancy" uses redundant hardware items which are non-operative until they are switched into the subsystem upon failure of the primary element. (1)

Automatic detection and switching must be used for the violent control malfunction type of failure...(2)

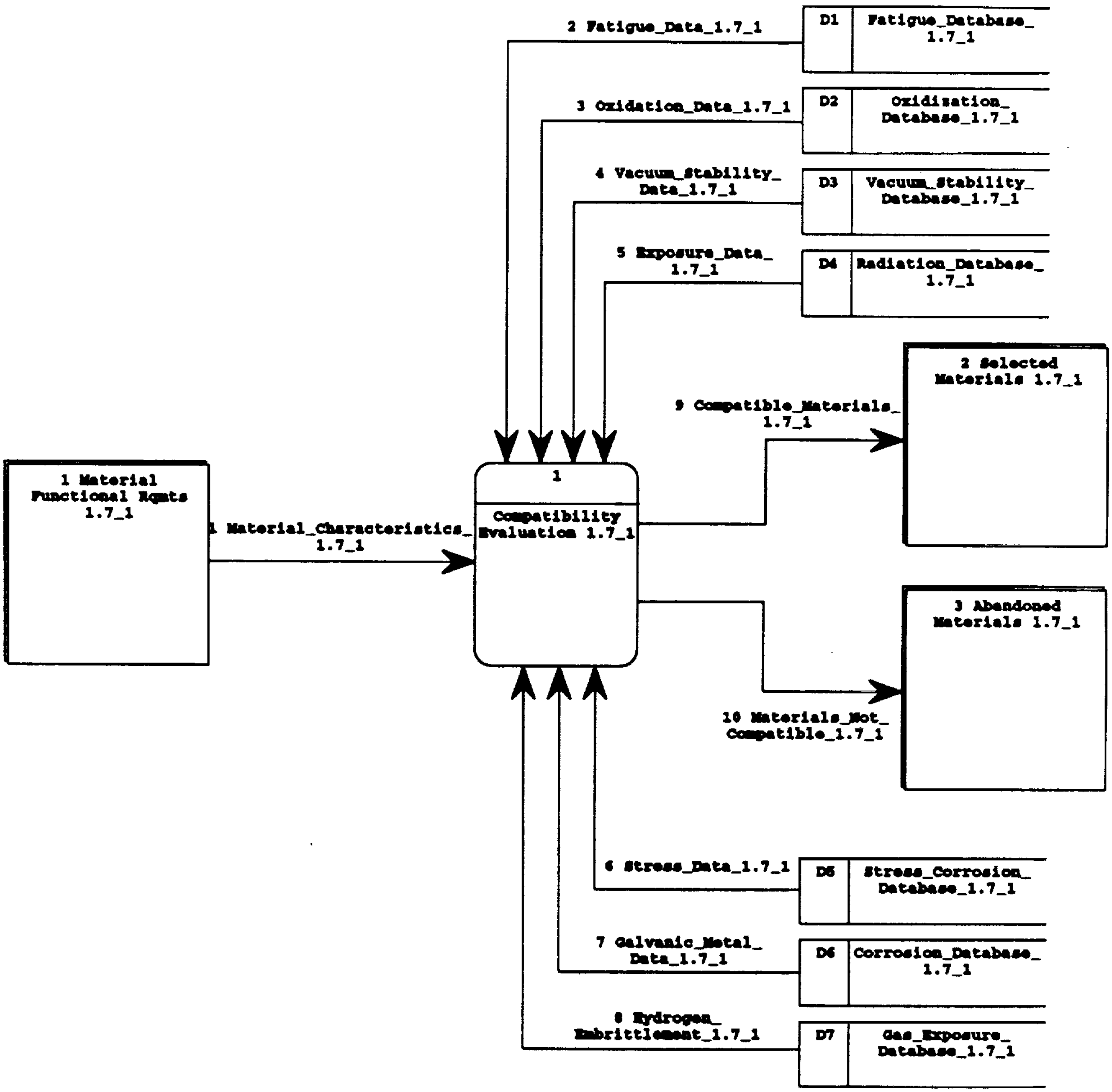
Establish a list of possible flight failure modes and analytically determine the probability of failures. (8)



MR_1.7_1
System Architect
Tue Sep 01, 1992 11:37

1.7 Materials (3a)

Functional requirements including, but not limited to, load distribution and magnitude, temperature, life, and use of exposure environments shall be met by consideration of such material properties as mechanical strength, fatigue, thermal stability, fracture toughness, and flaw propagation rates.
Material compatibility requirements shall be met by consideration of possible degradative mechanisms including, but not limited to, stress corrosion, galvanic or dissimilar metal corrosion, hydrogen embrittlement, creep, cycle and thermal fatigue, oxidation, vacuum stability, and radiation exposure. (3)

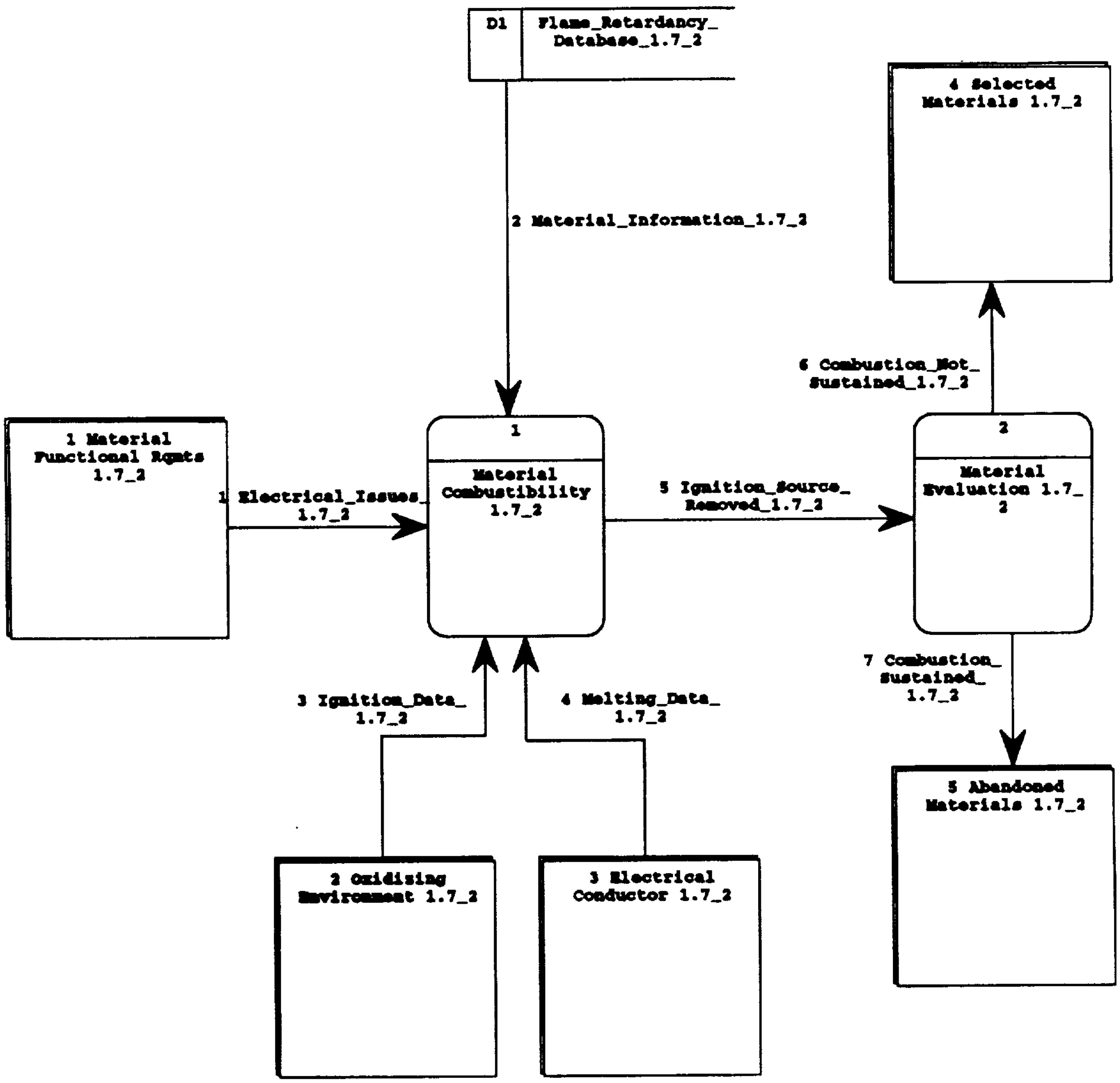


MR_1.7_2
System Architect
Tue Sep 01, 1992 15:00

1.7 Materials (3b)

Electrical wire insulation, wiring accessories, and materials in contact with electrical circuitry shall not be capable of sustaining combustion in the most severe oxidizing environment to be encountered during operations:

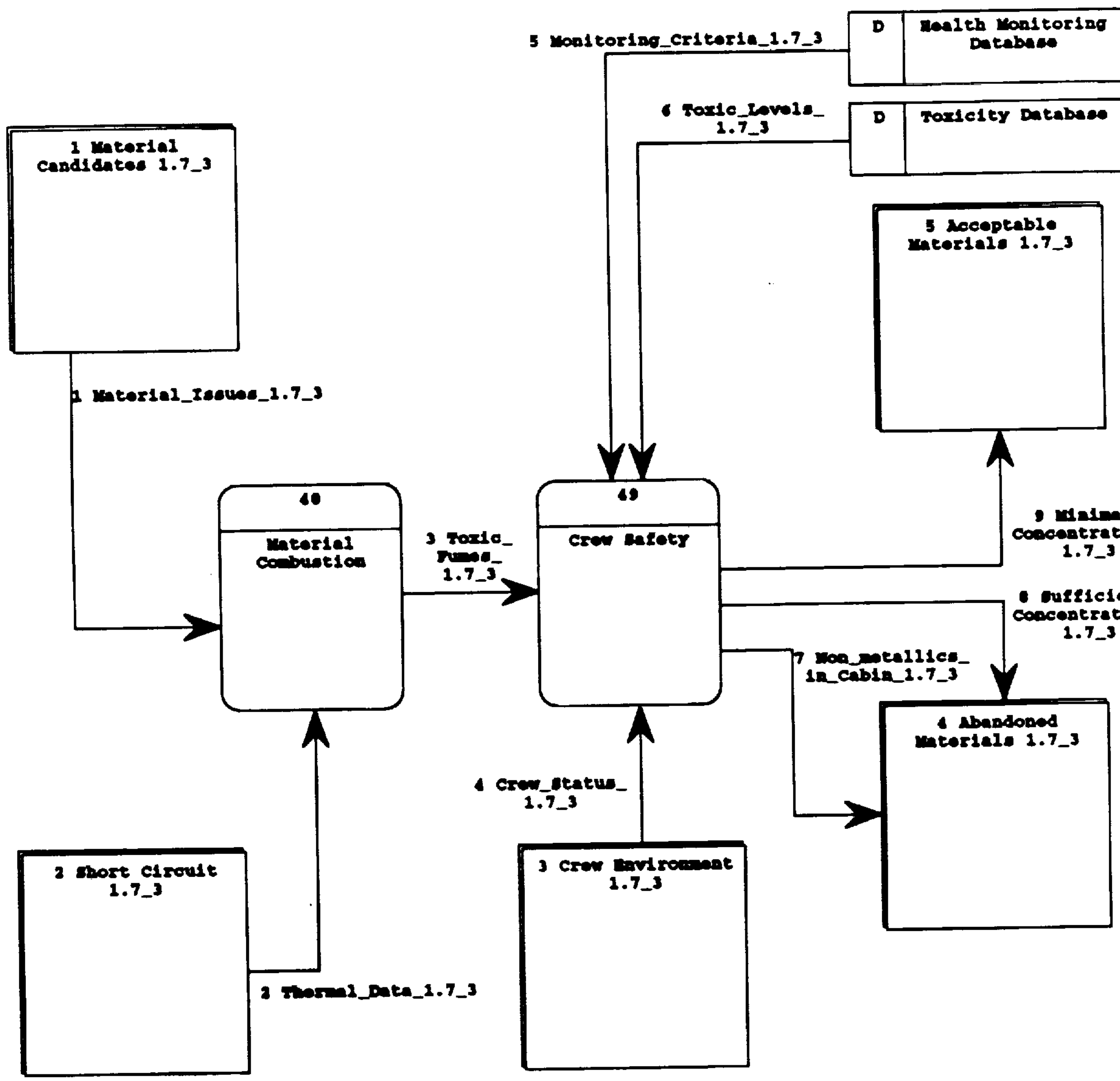
1) After removal of the ignition source.
2) Following melting of the electrical conductor by high currents, such as those resulting from short circuits or equipment malfunctions. (3)



MR_1.7_3
System Architect
Tue Sep 01, 1992 15:22

1.7 Materials (3c)

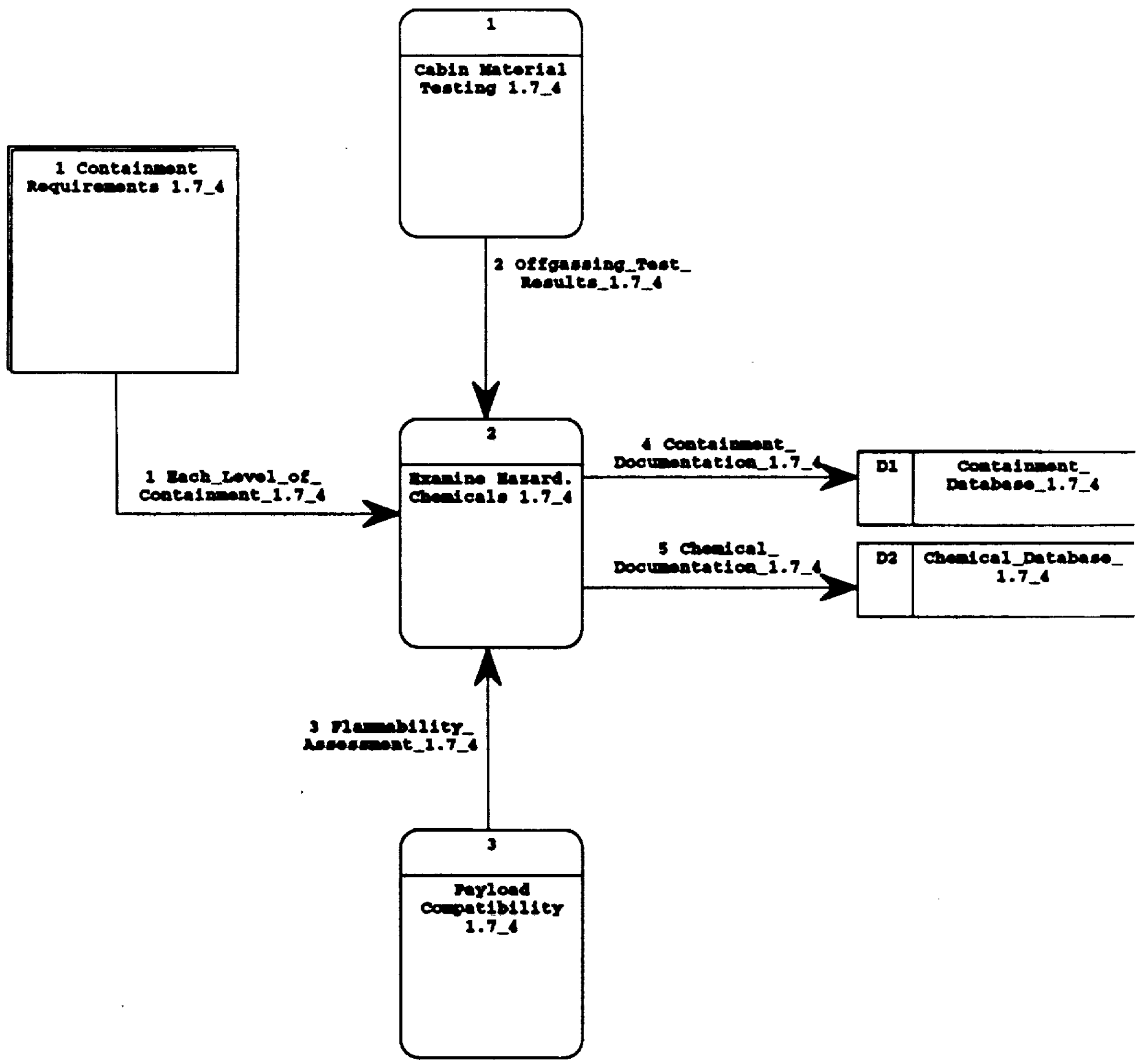
Toxicity: No material, when exposed to a short circuit, will generate toxic fumes in a concentration sufficient to impair crew safety shall be used for wire insulation, ties, identification marks, and protective covering on wiring. Non-metallic materials used within crew compartments shall not provide a toxic atmosphere. (3)



MR_1.7_4
System Architect
Fri Sep 04, 1992 10:23

1.7 Materials (5)

a) For hazardous materials that must be contained, each level of containment will be free of leaks under maximum use conditions. Documentation of all chemicals used and their method of containment will be maintained.
b) Payloads shall not constitute an uncontrolled fire hazard, and a flammability assessment shall be documented.
c) Materials used in the crew cabin and other habitable areas must be tested under worst-case cabin environment conditions. Offgassing tests shall be conducted. (5)



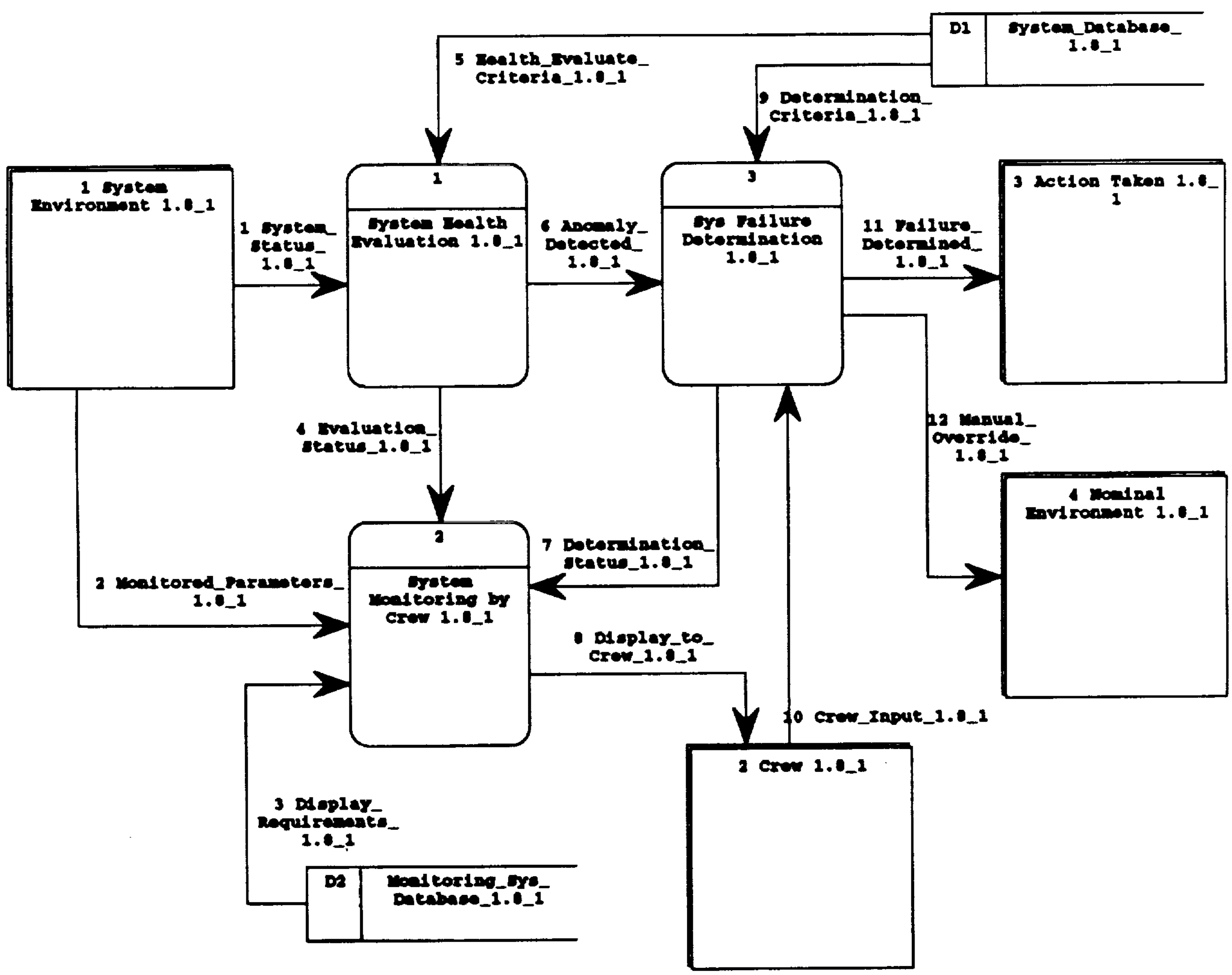
MR_1.8_1
System Architect
Tue Sep 01, 1992 16:08

1.8 Displays and Controls (1a, 1b, 2a)

Displays and controls should be provided to the crew for monitoring system status and failure alerts. (1)

Must be available, accessible, and readable in emergency situations. Controls for critical functions must not be able to be activated or deactivated inadvertently. (1)

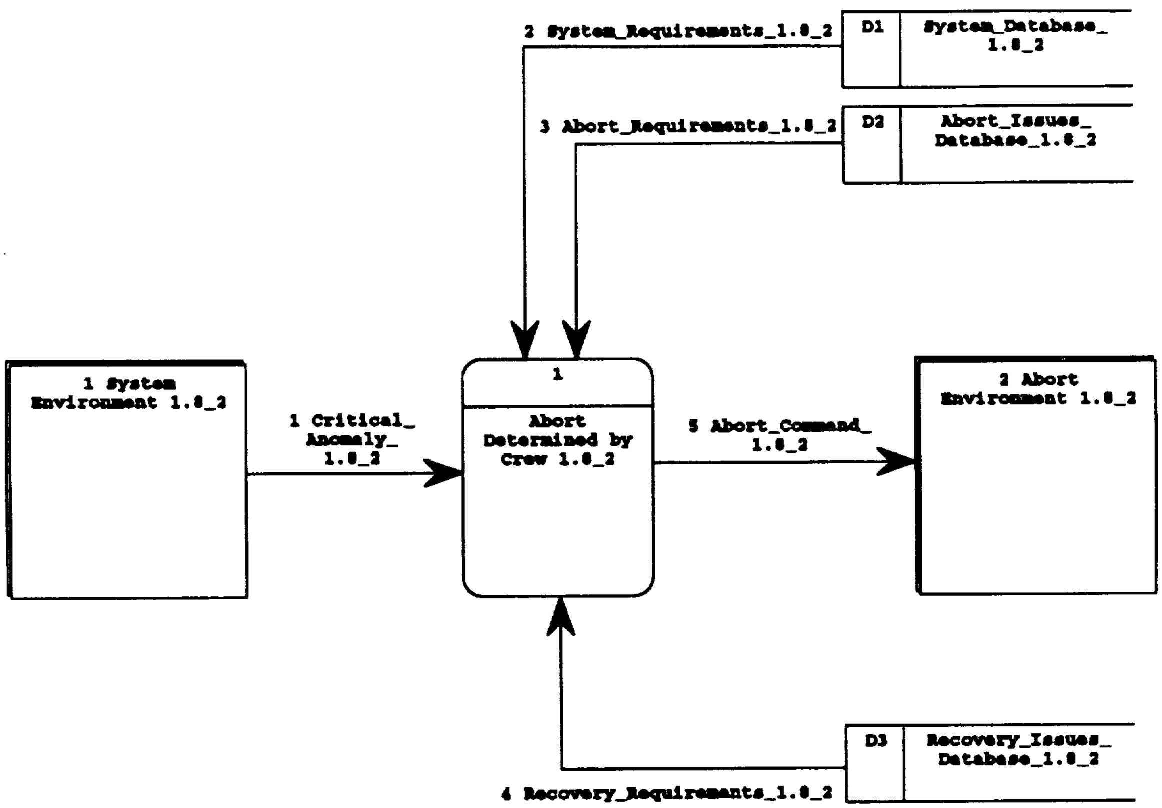
...a combination of automatic and pilot control is favored. Crew override should be provided where signals may be erroneous. (2)



MR_1.8_2
System Architect
Tue Sep 01, 1992 16:27

1.8 Displays and Controls (2c)

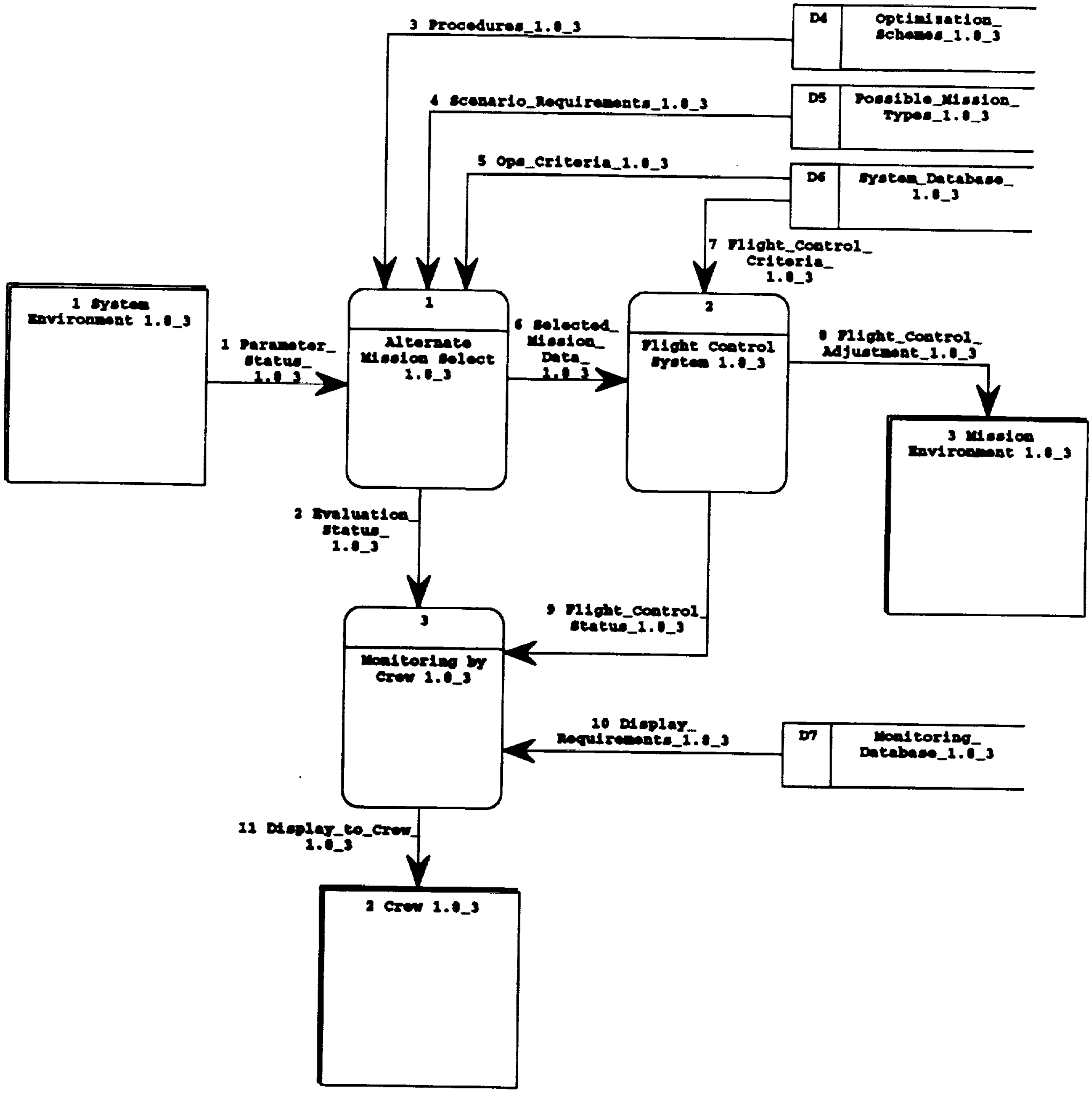
Where abort is by astronaut command, a clear communication channel must exist for advice and recovery. (2)



MR_1.8_3
System Architect
Tue Sep 08, 1992 13:21

1.8 Displays and Controls (2d)

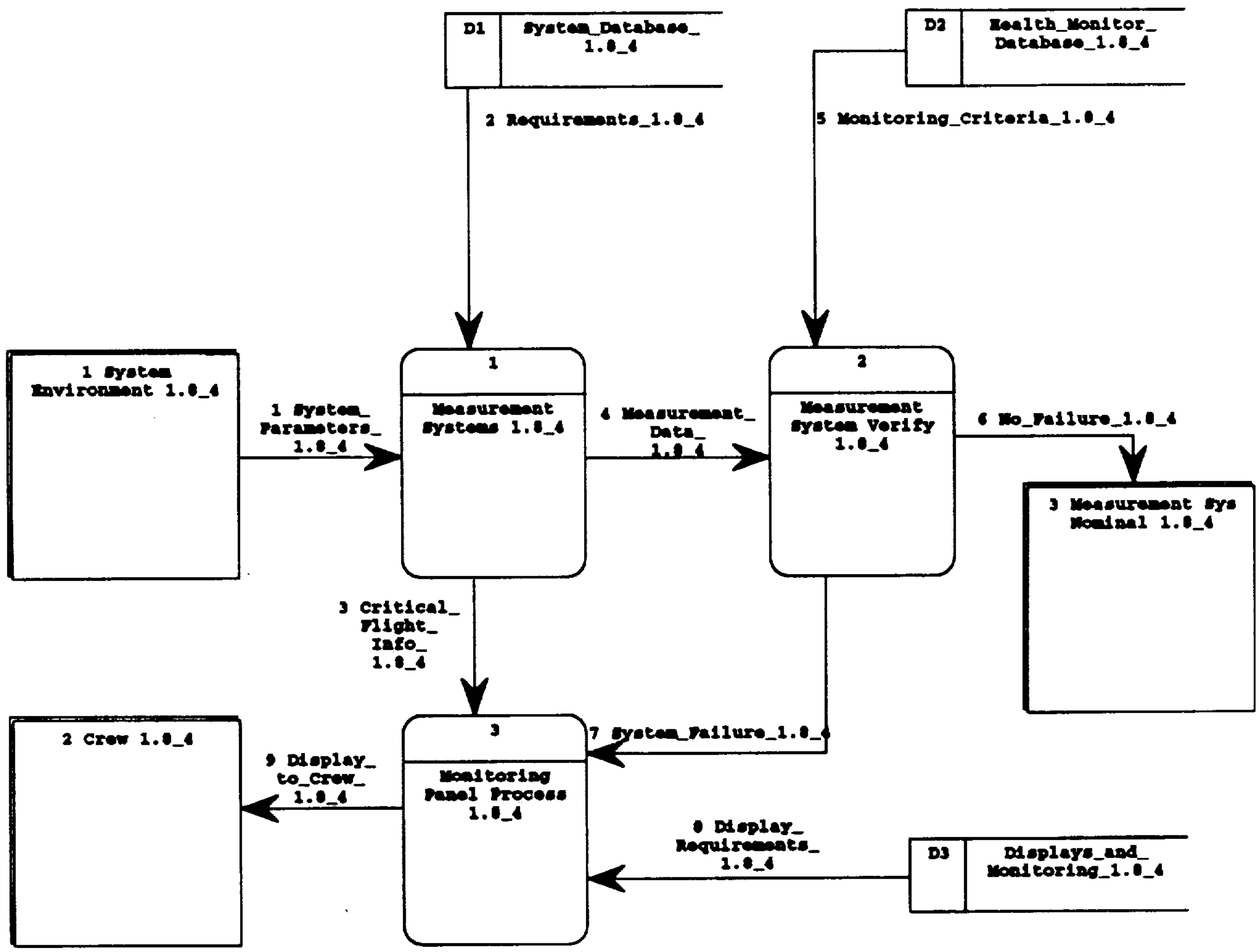
...the computer should have all possible alternate missions preprogrammed so that the optimum alternate mission can be selected. Then, new commands can be supplied to the flight control system, and illuminated display lights on the crew's console tells them of the alternate mission and the new parameters that require resetting, if applicable.(2)



MR_1.8_4
System Architect
Tue Sep 08, 1992 13:25

1.8 Displays and Controls (3a)
(Indication of Failure):

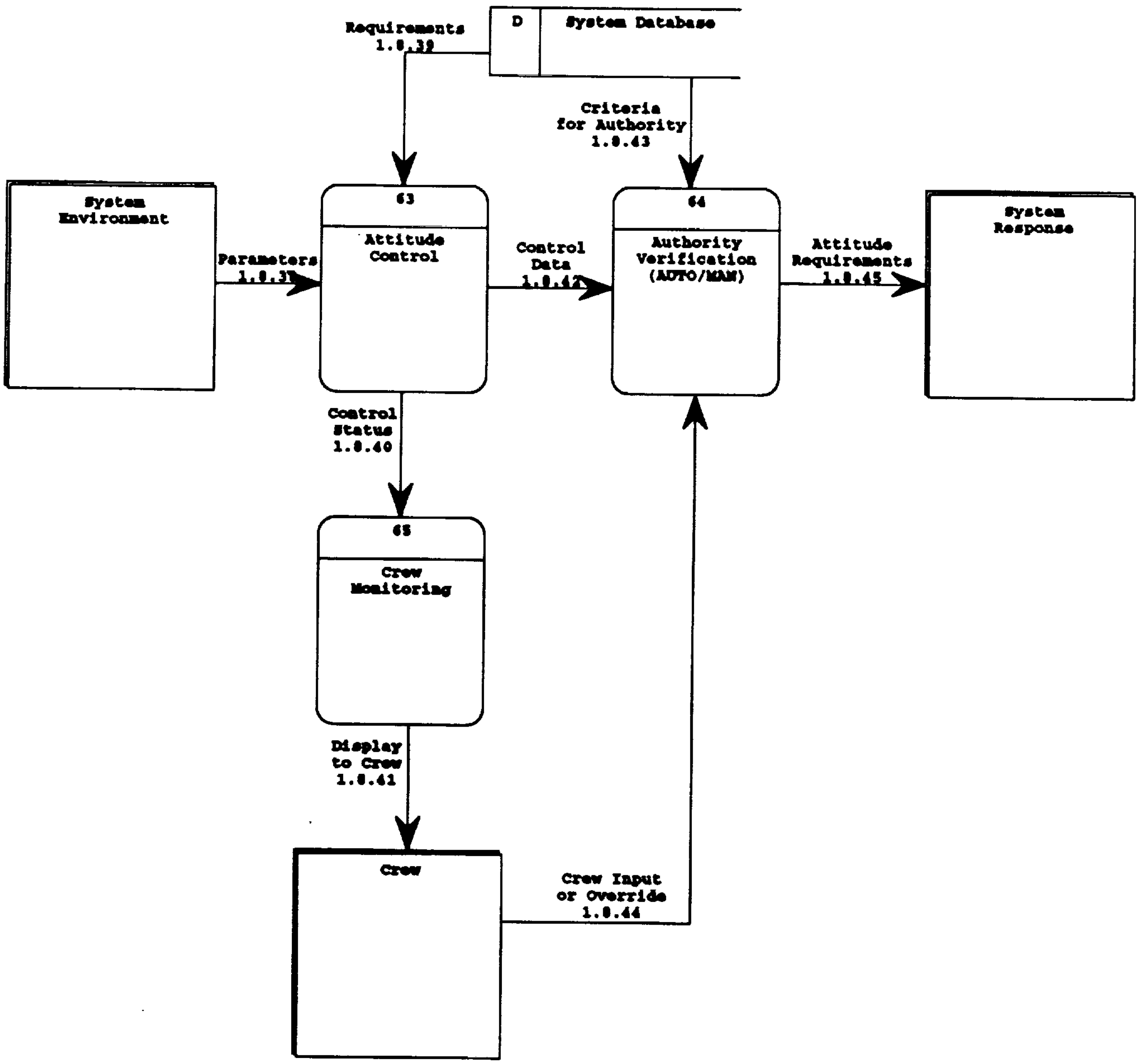
Those measurements systems which display critical flight information to the crew on panel indicators shall be designed so that when such a system fails, it should provide an indication of its failure.(3)



MR_1.8_5
System Architect
Tue Sep 08, 1992 13:28

1.8 Displays and Controls (3b)
(Attitude Control Authority)

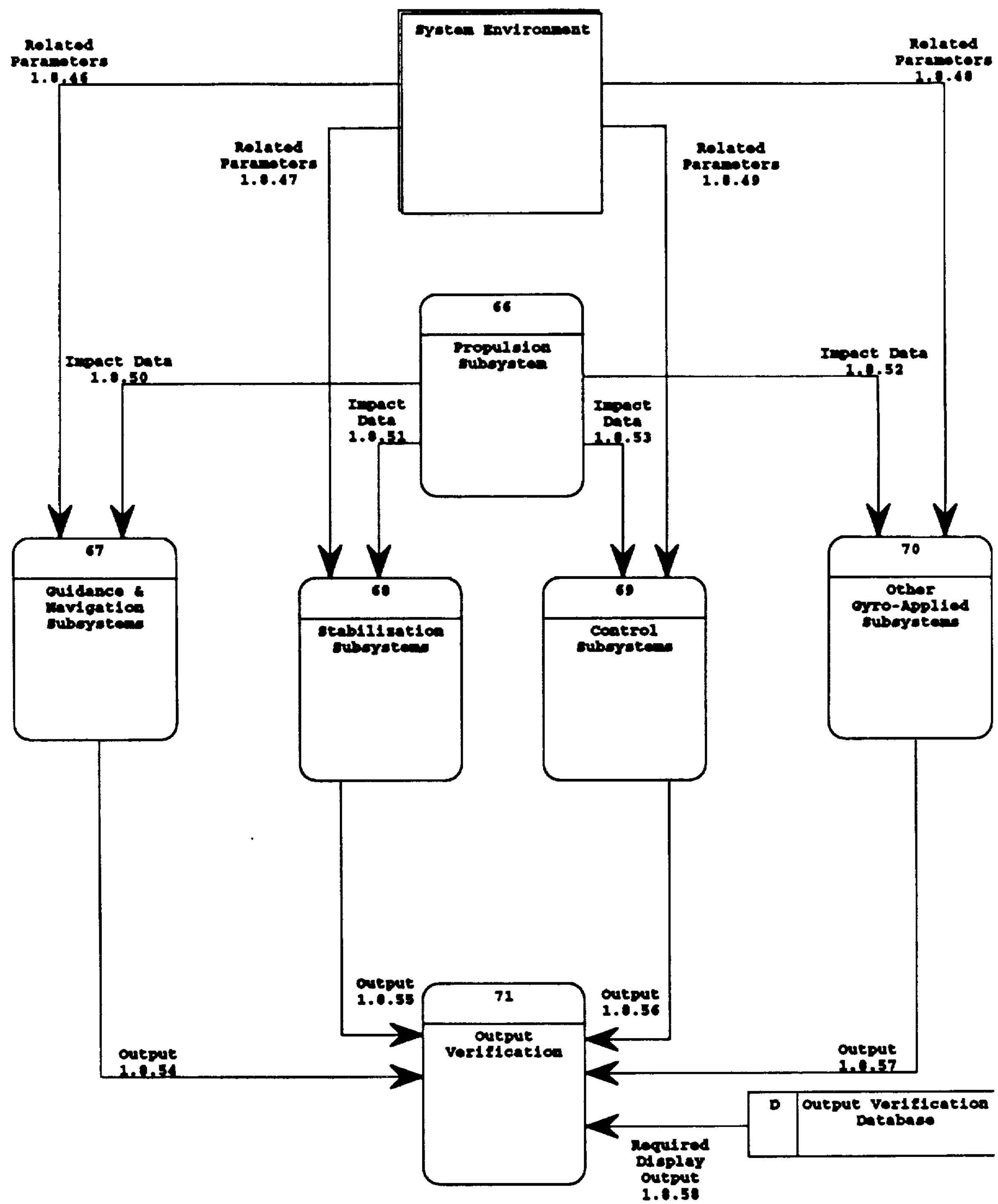
Spacecraft automatic attitude control circuitry shall be designed so that the crew can assume manual attitude control at all times.(3)



MR_1.8_6
System Architect
Fri Sep 04, 1992 11:52

1.8 Displays and Controls (3d)
(Gyroscopic Performance Verification

Guidance and naviagation subsystems, stabilization subsystems, control subsystems, and any similar subsystems using gyroscopes for guidance or stabilization of spacecraft during propulsion subsystem operation shall provide continuous outputs for verification and proper gyroscope rotational speed or drift rate.(3)



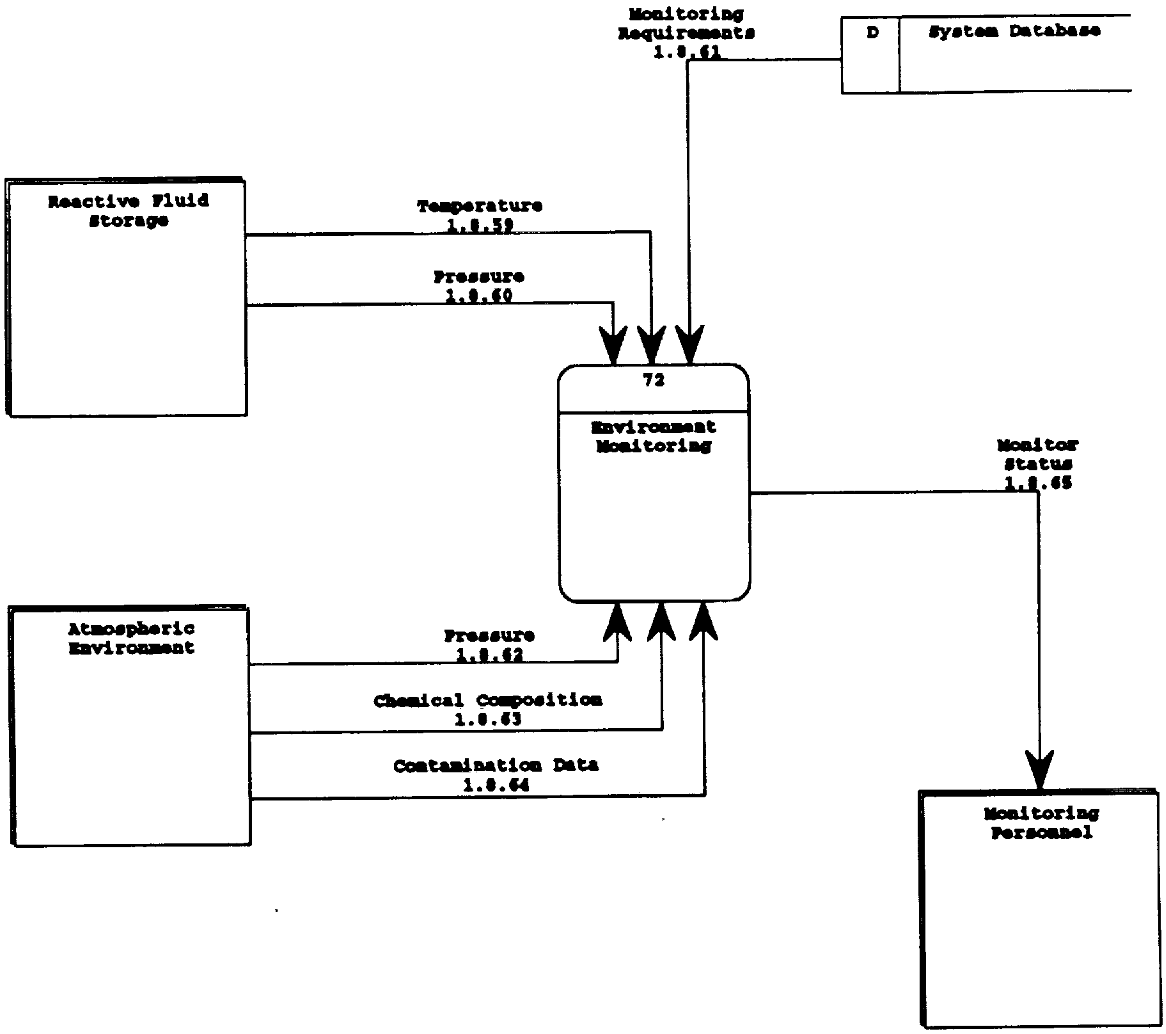
MR_1.8_7
System Architect
Fri Sep 04, 1992 16:53

1.8 Displays and Controls (3e,f)
(Environment Monitoring)

All spacecraft systems and ground support servicing equipment requiring storage of reactive fluids (i.e., oxidizer, monopropellants, etc.) shall be designed to include devices for monitoring temperature and pressure to permit accurate determination of the rates of active oxygen loss of the oxidizer contained in their respective systems.(3)

(Atmospheric Pressure and Composition Control)

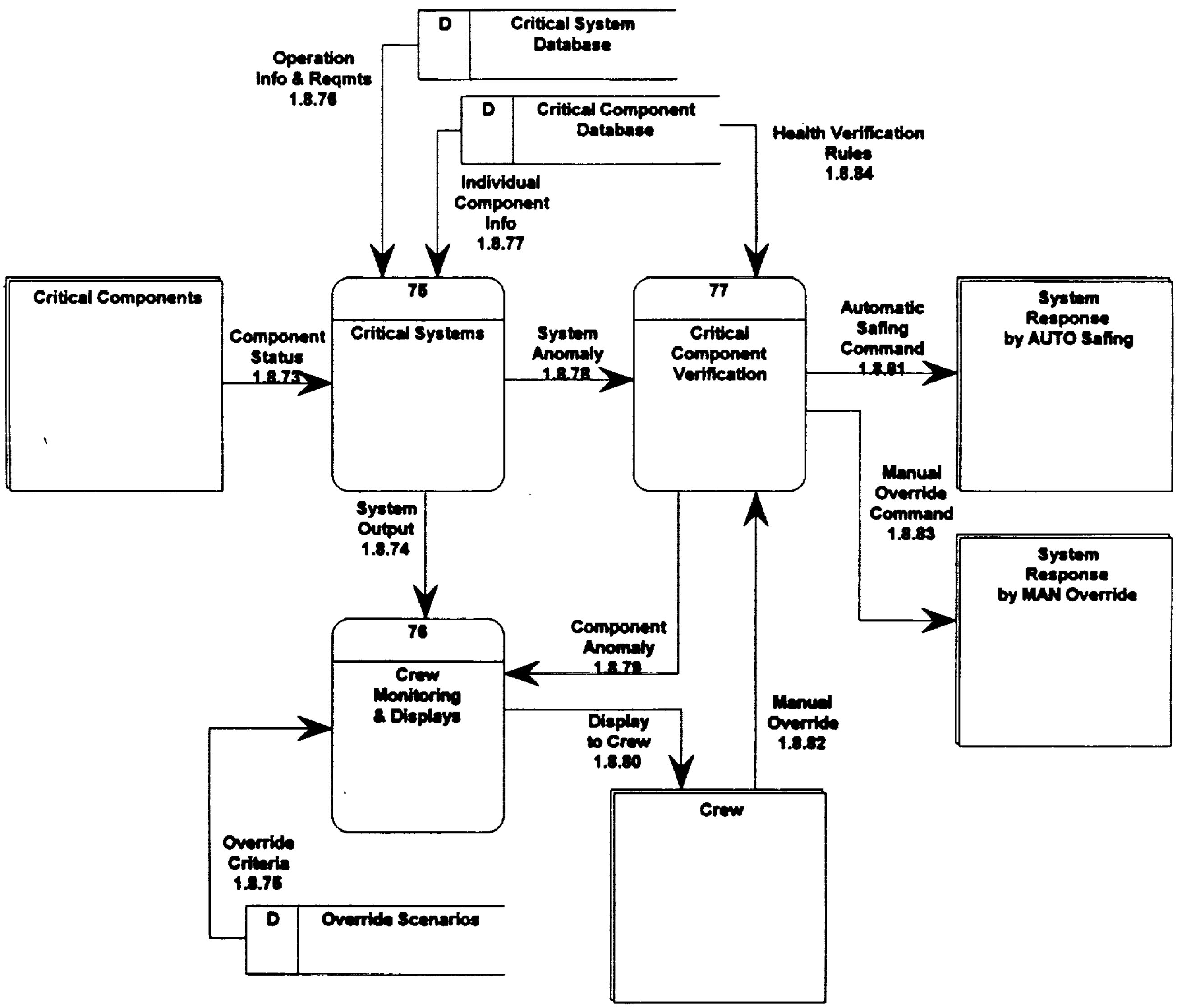
Provisions shall be made to monitor and control oxygen, carbon monoxide, atmospheric pressure, and trace contaminants. Trace contaminants may be organic, inorganic, or biological.(3)



MR_1.8_8
System Architect
Thu Sep 10, 1992 18:15

1.8 Displays and Controls (3h)
(Critical Components)

For critical components, automatic safing with manual override only shall be used.(3)

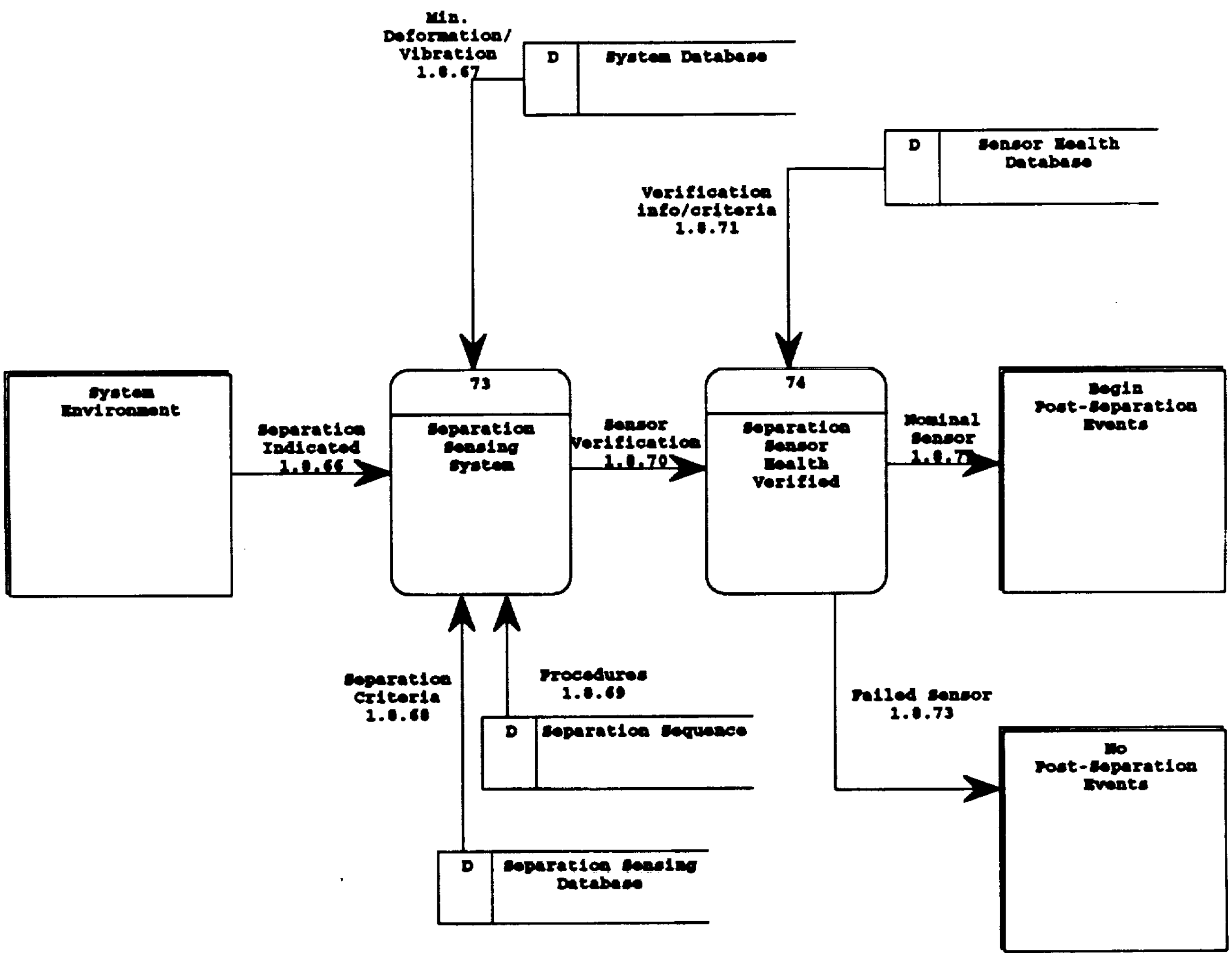


MR_1.8_9
System Architect
Fri Sep 04, 1992 13:32

1.8 Displays and Controls (3c, 31)
(Separation Sensing System)

Separation sensing systems used to detect separation of stages or modules of the space vehicle shall be designed so that actuation of separation sensors will not result from structural deformation or vibrations less severe than those associated with structural failure of the vehicle.(3)

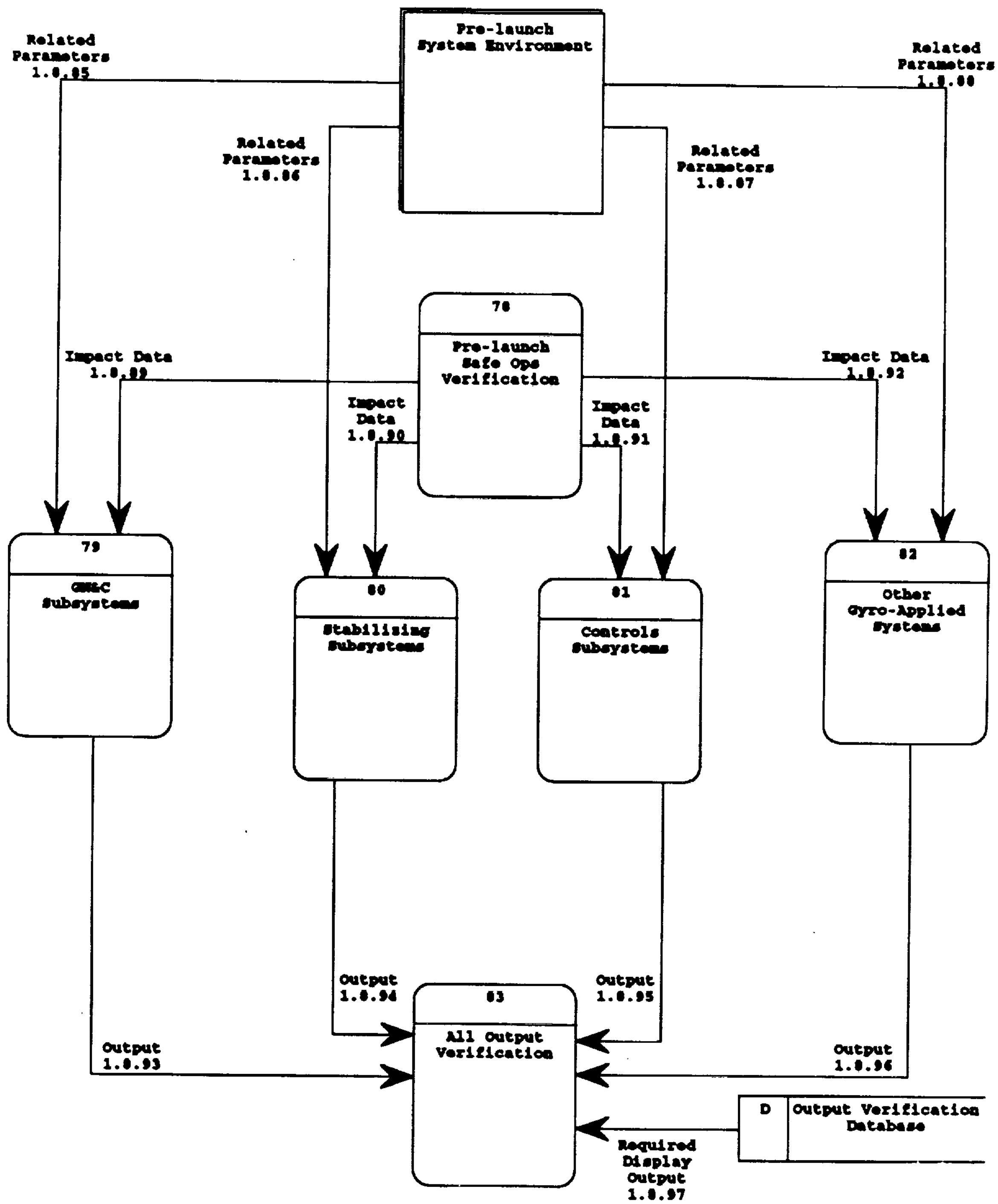
Where the separation sensing system is used to initiate automatically subsequent steps in a sequence of events, the sensing system shall be configured so that actuation or failure of a single sensor will not initiate the sequence of events.(3)



MR_1.8_10
System Architect
Tue Sep 08, 1992 13:41

1.8 Displays and Controls (3j)
(Gyroscopic Safe-ops Verification)

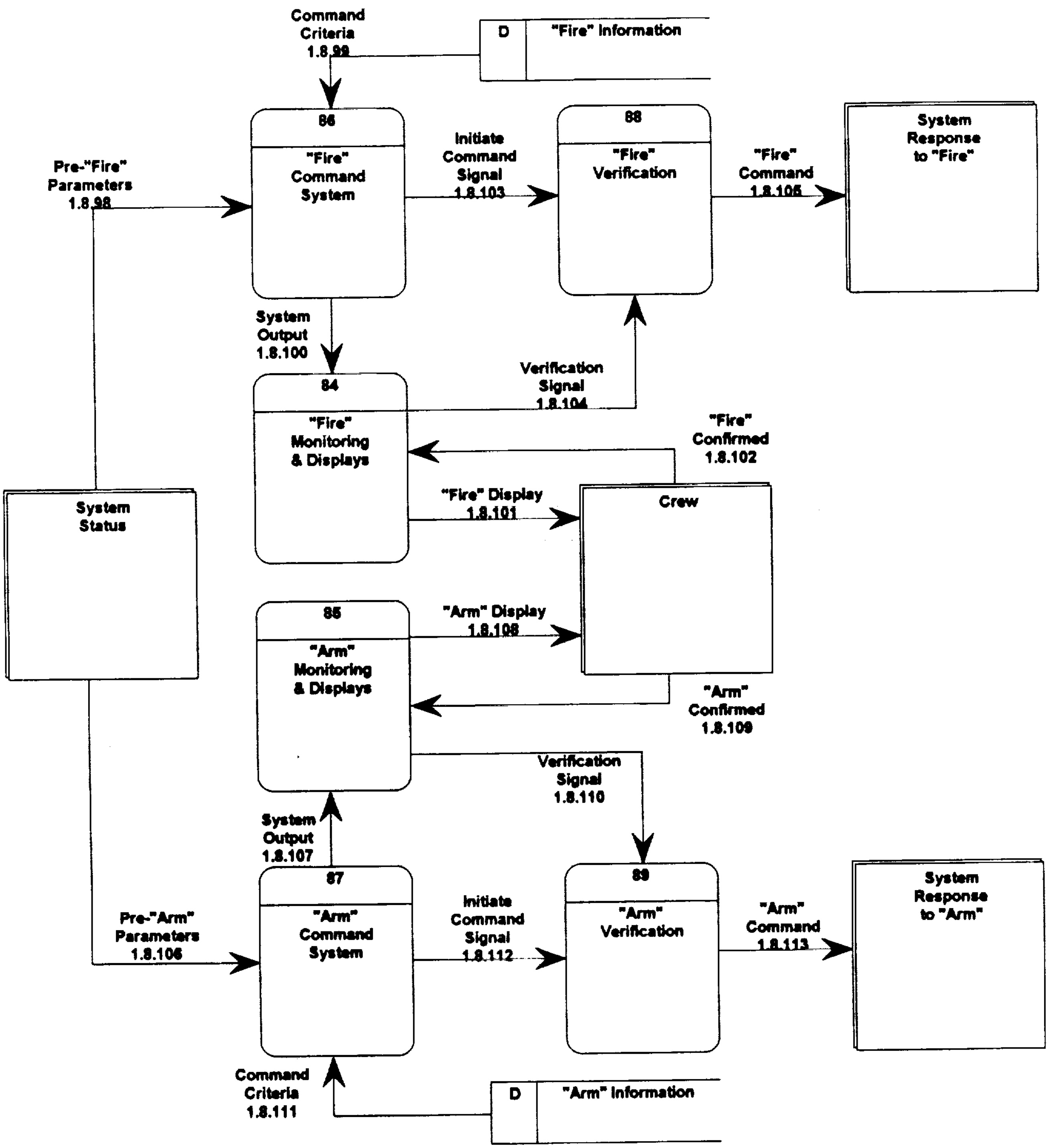
Immediately prior to engine ignition for launch, including inflight launches, the rotational speed or drift rate of all gyroscopes, normally required to operate a launch, shall be verified to be within required safe operating limits.(3)



MR_1.8_11
System Architect
Thu Sep 10, 1992 18:17

1.8 Displays and Controls (3L)

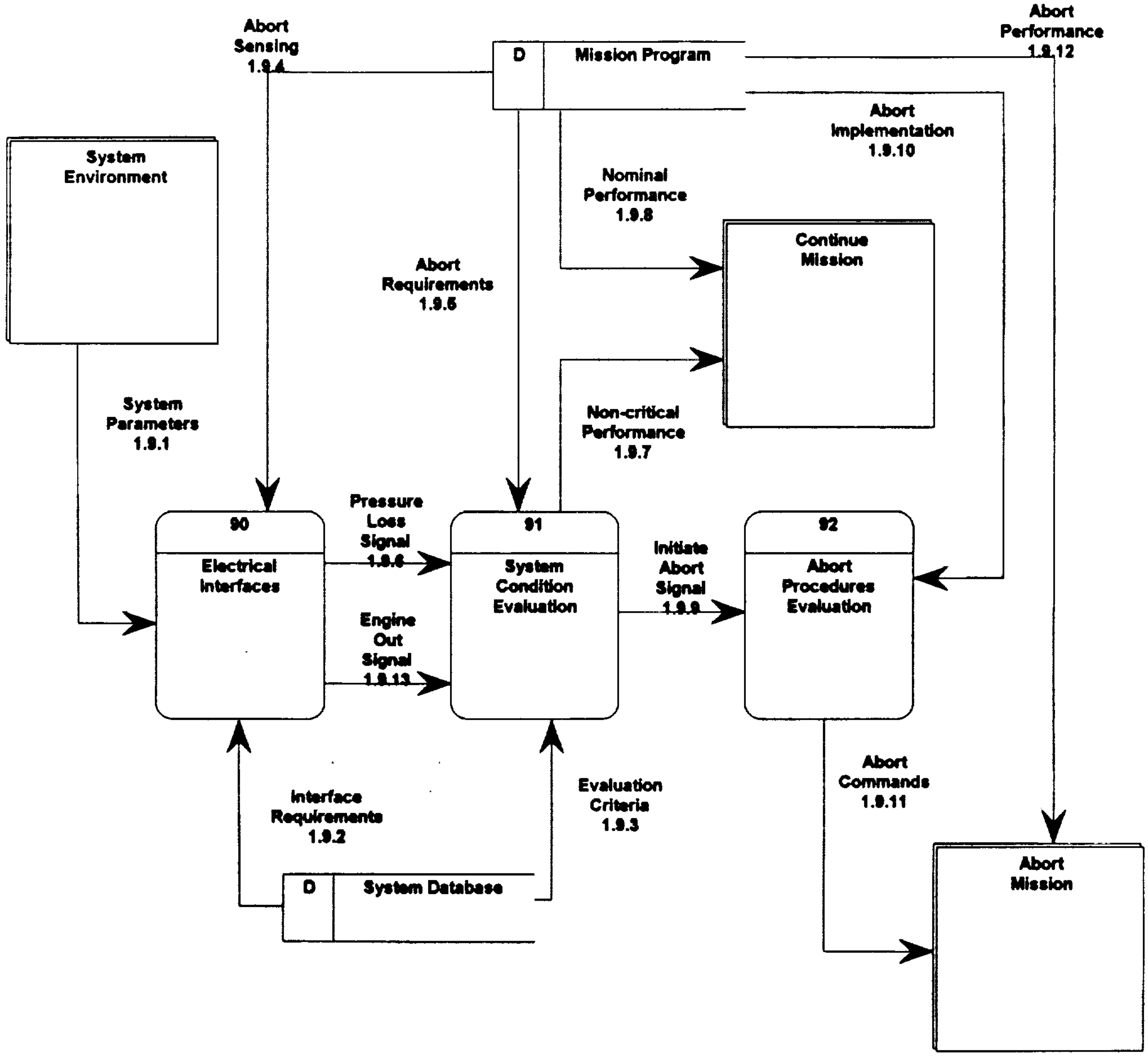
"Arm" and "fire" shall be separate functions and separately displayed. Arm and fire switches shall be guarded switches.(3)



MR_1.9_1
System Architect
Thu Sep 10, 1992 18:21

1.9 Aborts (1)

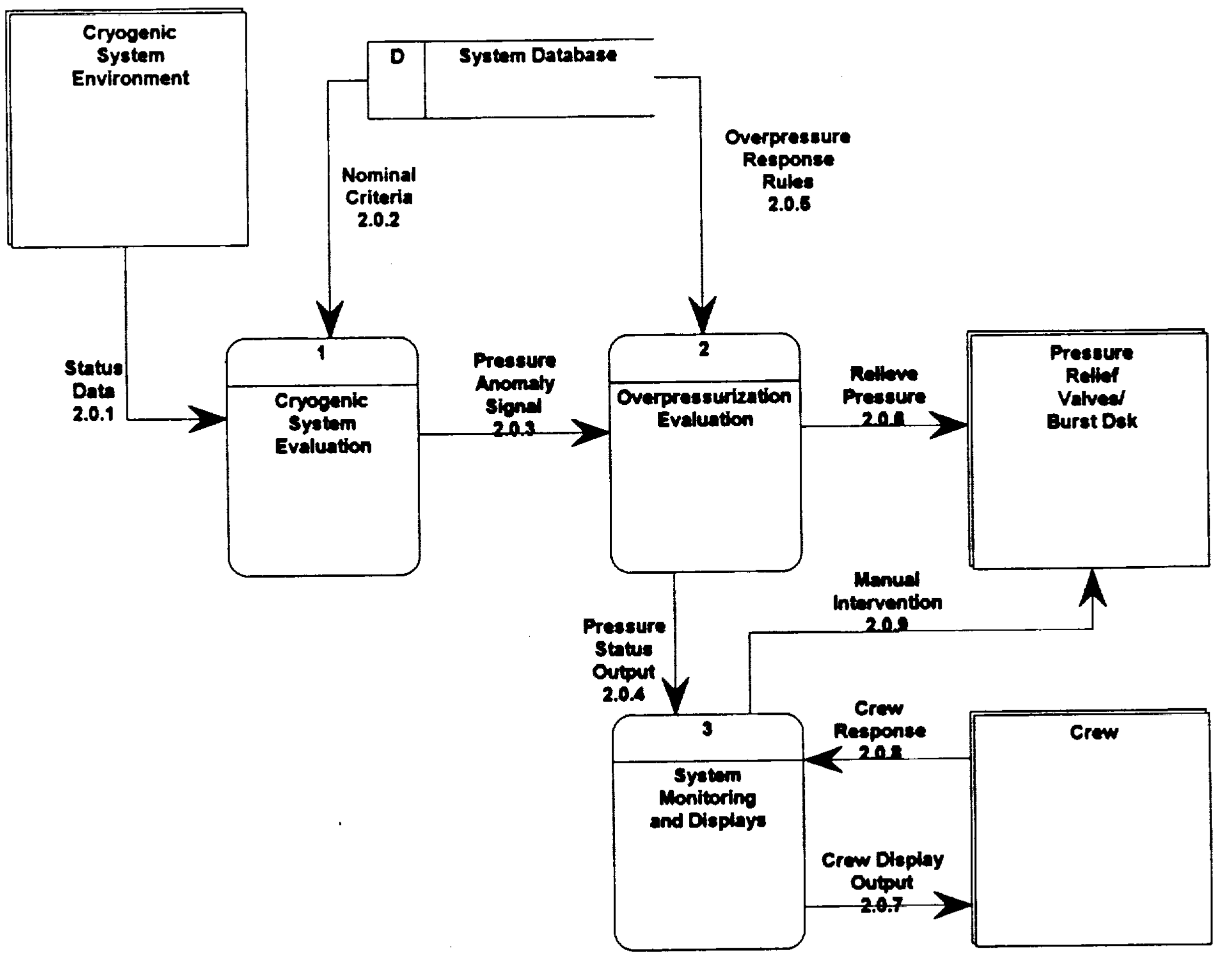
Credible failures for which abort procedures should be developed shall include, at a minimum, one engine out and loss of cabin pressure. All abort scenarios shall be tested and the crew fully trained in their execution.(1)



MR_2.0_1
System Architect
Thu Sep 10, 1992 18:24

2. Design Practices (1)

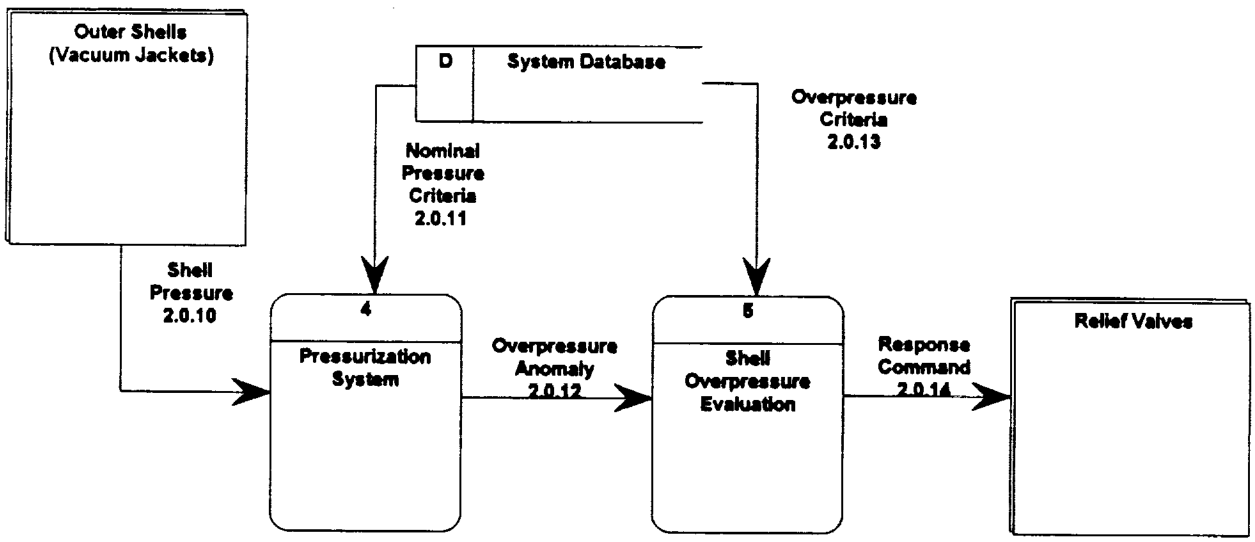
a) Cryogenic systems, with sections where cryogenic liquid may be trapped, should be designed to prevent line rupture
b) Cryogenic systems, with sections where cryogenic liquid may be trapped, should be designed to prevent line rupture if relief valves freeze. System should be provided with relief valves paralleled by burst discs.
c) Systems or materials, which are potentially hazardous if allowed to physically meet, shall be redundantly separated or shielded from one another, or adequately spaced apart.
d) A detection system and an appropriate exhaust or neutralizing system should be provided where toxic or explosive gases may be expected.
e) Oxygen flow limiters and/or monitoring devices should be required to insure against oxygen partial pressure build-up or decrese.(1)



MR_2.0_2
System Architect
Thu Sep 10, 1992 18:27

2. Design Practices (5a)

Outer shells (i.e., vacuum jackets) shall have pressure relief capability to preclude rupture.(5)

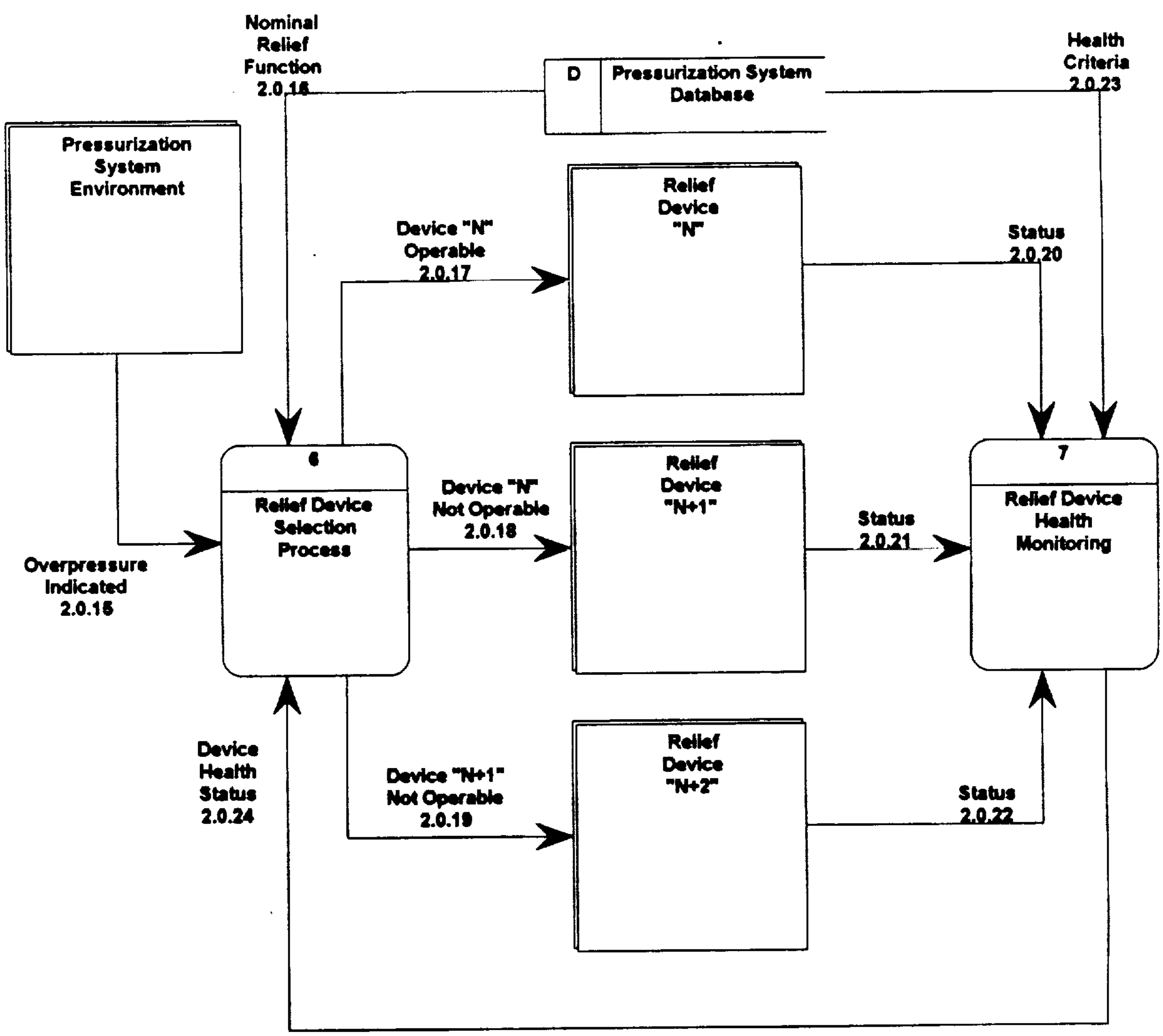


MR_2.0_3
System Architect
Thu Sep 10, 1992 18:29

2. Design Practices (5c)

Where pressure regulators, relief devices and/or a thermal control system are used to control pressure, collectively they must be two-fault tolerant from causing pressure to exceed MPD.(5)

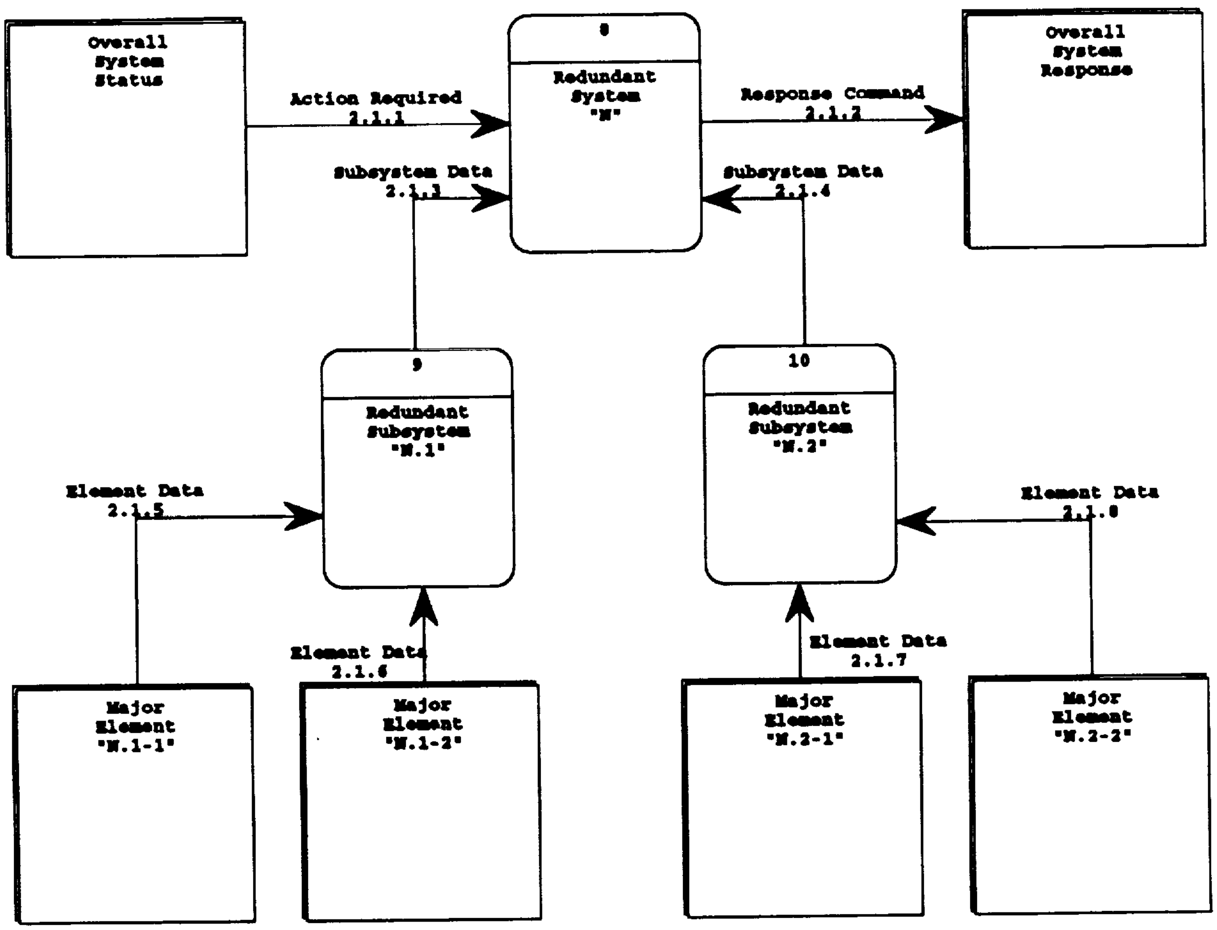
Relief devices must be redundant and sized to permit full flow at MDP.(5)



MR_2.1_1
System Architect
Fri Sep 04, 1992 13:57

2.1 General Issues (3a)

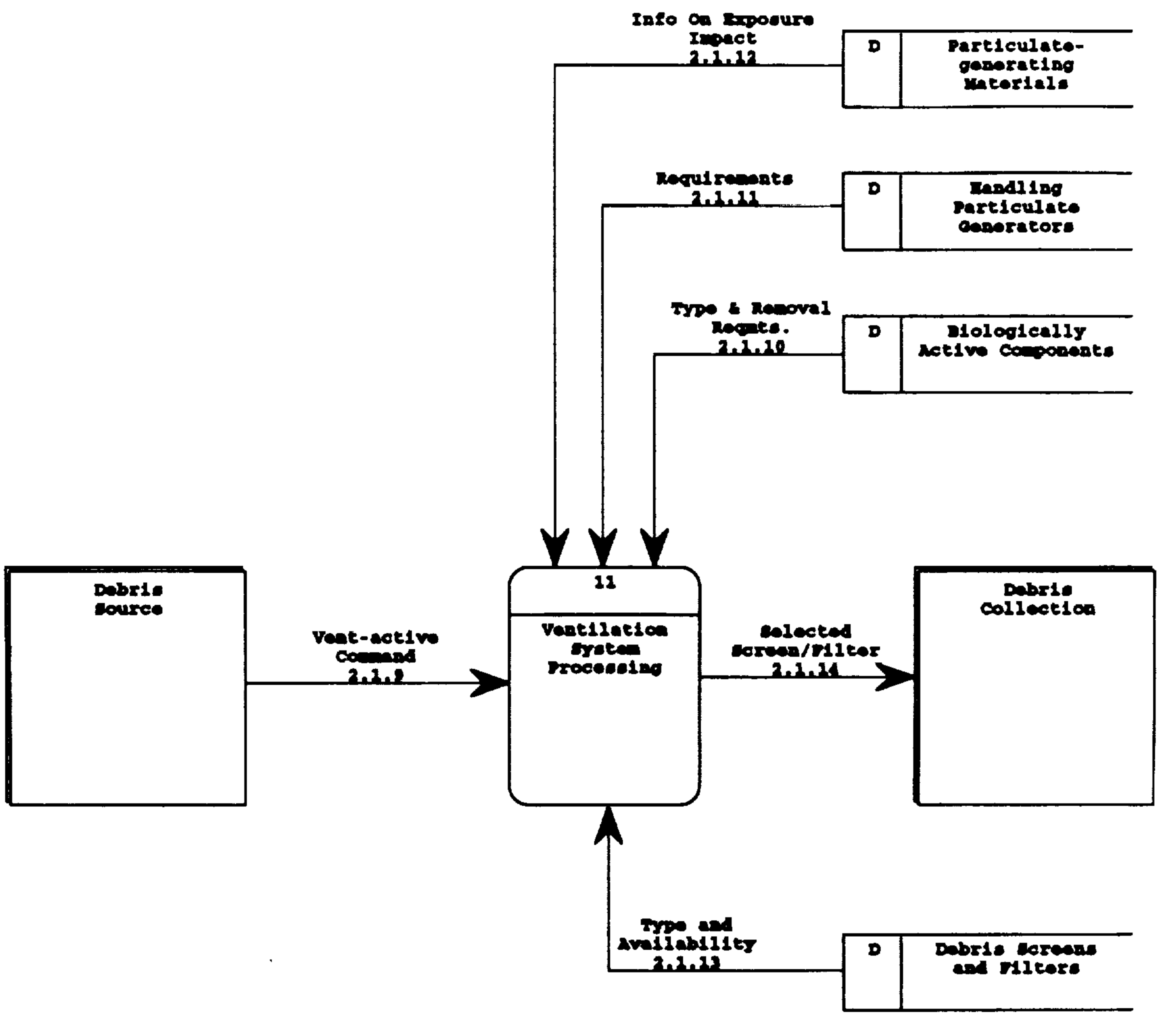
Electrical wiring of redundant systems, redundant subsystems, or redundant major elements of subsystems shall not be routed in the same wire bundle or through the same connector without wiring of the other redundant systems, subsystem, or subsystem element. Redundant systems and redundant components should be designed so as not to preclude concurrent operations.(3)



MR_2.1_2
System Architect
Fri Sep 04, 1992 14:04

2.1 General Issues (3b)

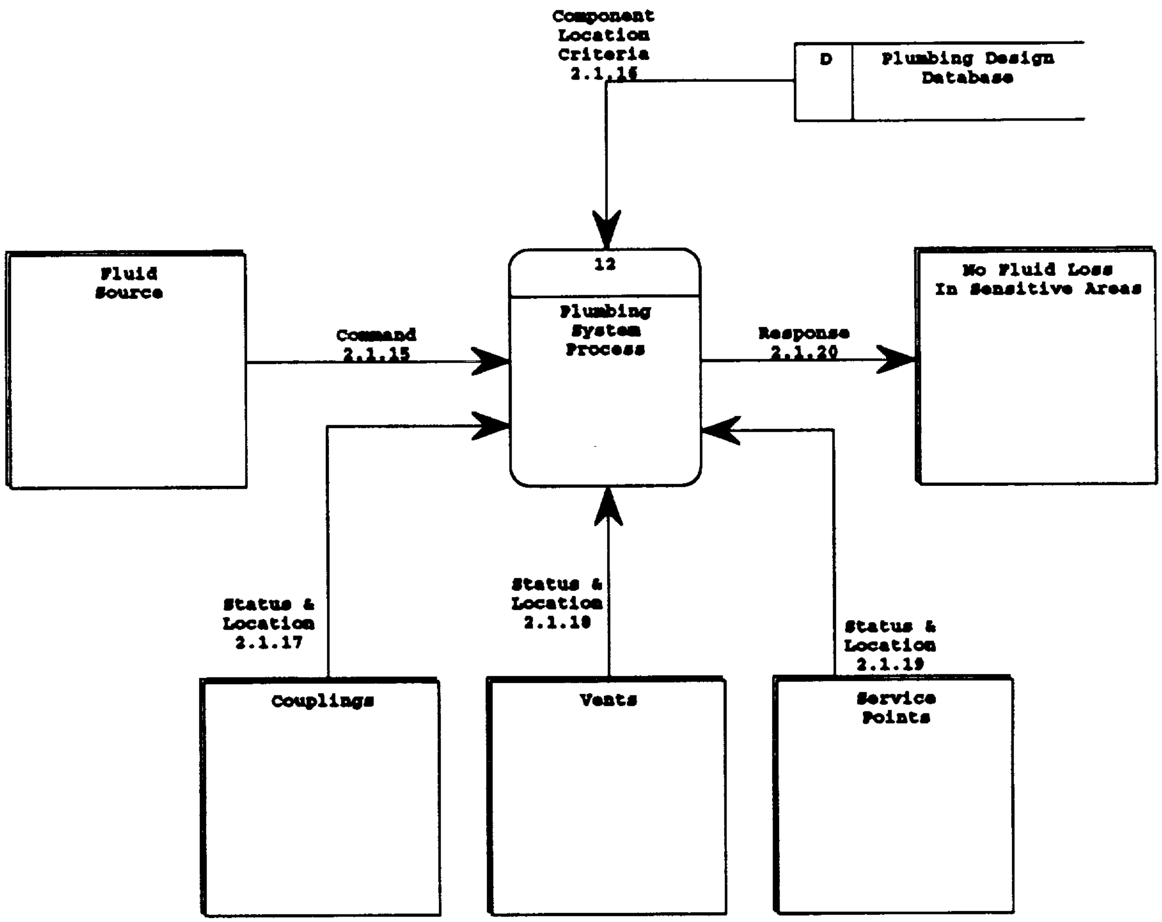
Protective covers shall be provided. The use of particulate-generating materials and surfaces is prohibited. If such material must be used they must either be coated, encased, or taped. The ventilation system shall include debris-collection screens on air inlets. Removal and/or control of biologically active components shall be considered when specifying the use of debris screens or filters.(3)



MR_2.1_3
System Architect
Fri Sep 04, 1992 15:35

2.1 General Issues (3c)

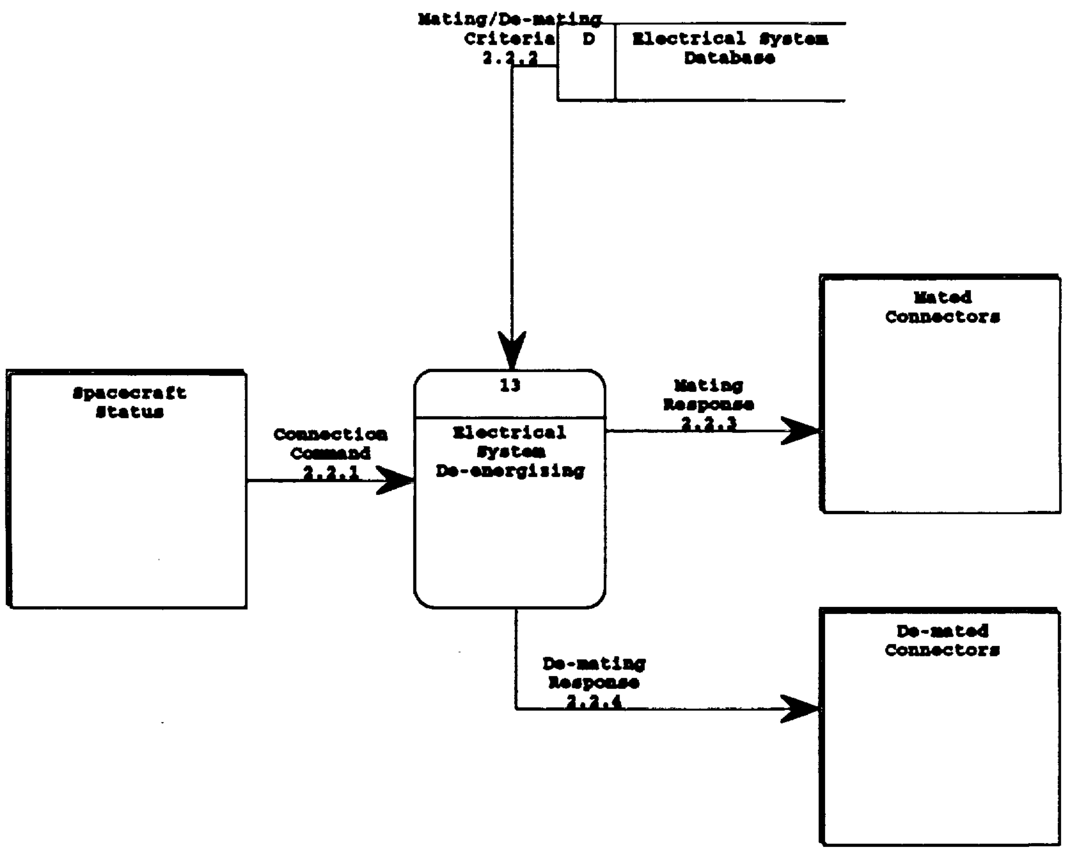
Design plumbing to be insensitive to the liquid leakage. Design plumbing or equipment containing the liquid to locate couplings, vents, service points, and other items where leakage from them could not reach the sensitive equipment for leakage during ground operation. Provide insulation to prevent condensate from falling on the equipment.(3)



MR_2.2_1
System Architect
Fri Sep 04, 1992 15:40

2.2 Electrical Issues (3a)
(De-energizing Electrical Circuits)

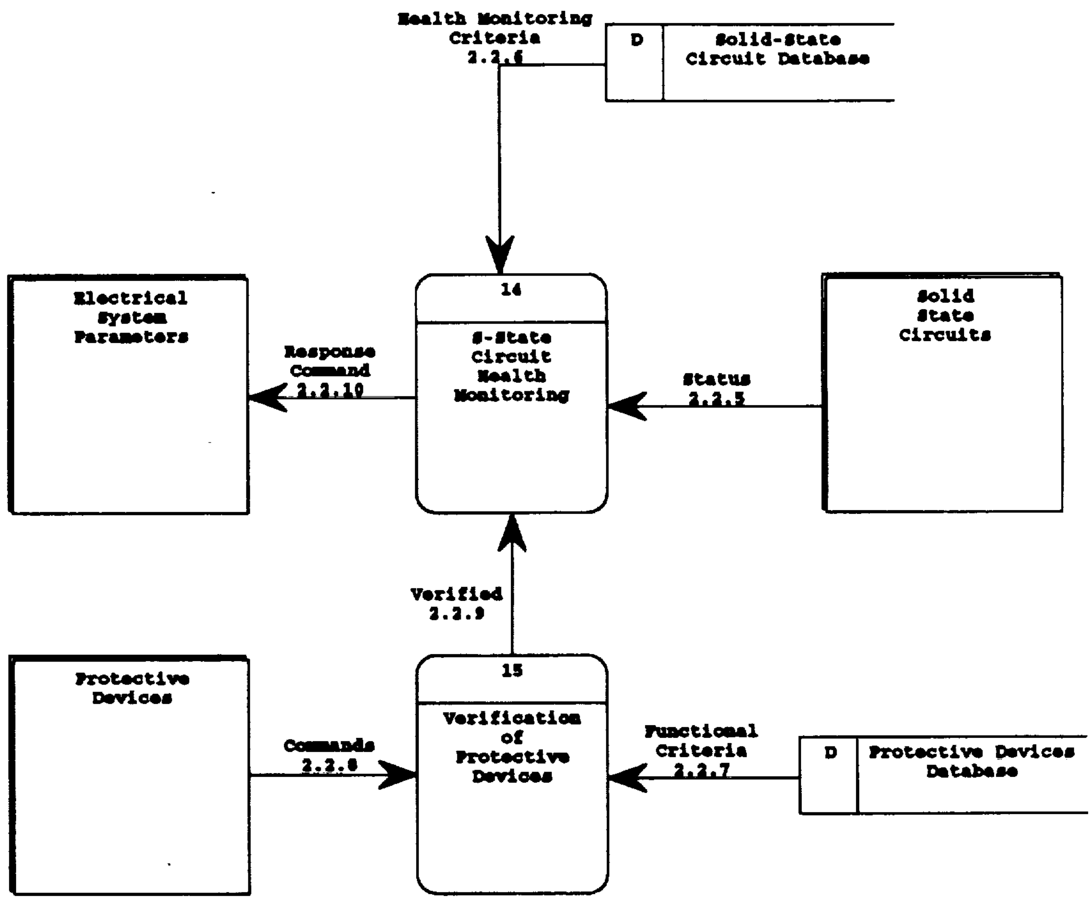
Spacecraft electrical systems shall be designed so that all necessary mating and demating of connectors can be accomplished without producing electrical arcs that will damage connector pins or ignite surrounding materiels or vapors.(3)



MR_2.2_2
System Architect
Fri Sep 04, 1992 15:46

2.2 Electrical Issues (3b)
(Protective Devices for Solid-State
Circuits)

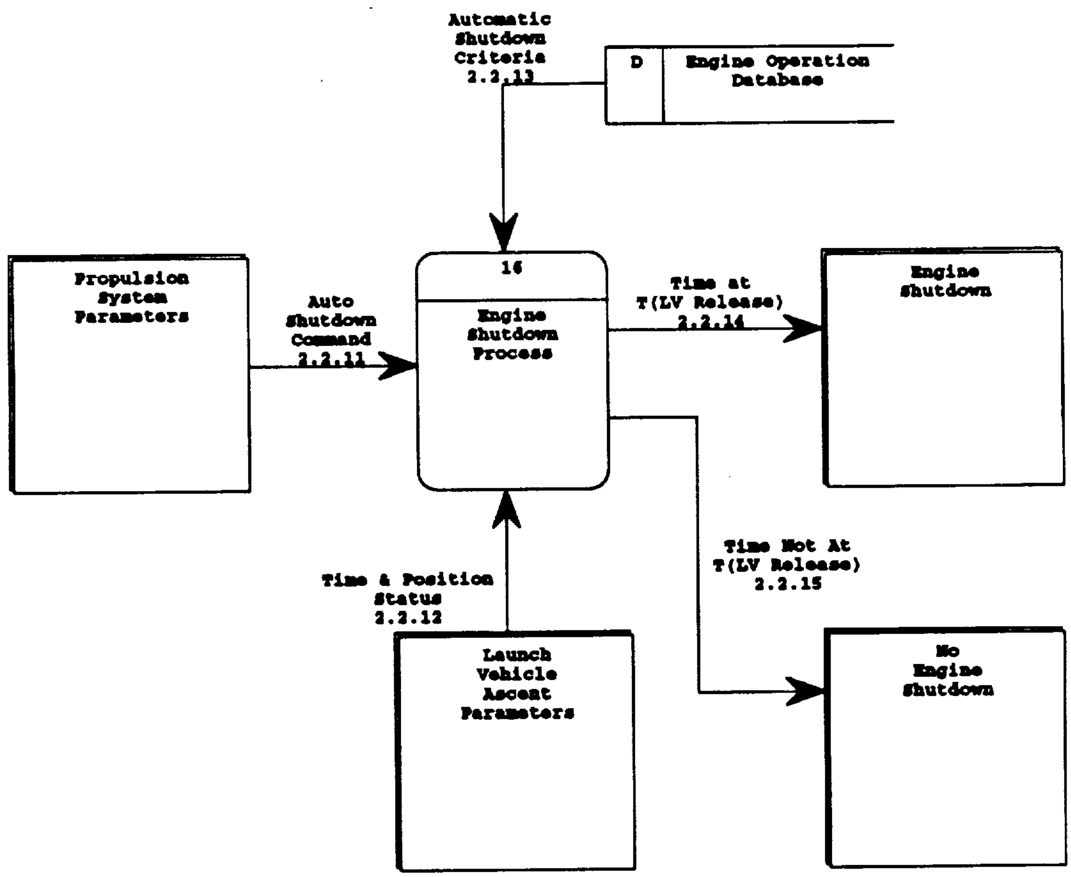
Protective devices used in critical electronic circuits to protect solid-state circuit elements shall be verified as ready to function.(3)



MR_2.2_3
System Architect
Fri Sep 04, 1992 15:48

2.2 Electrical Issues (3c)
(Engine Shutdown Circuitry)

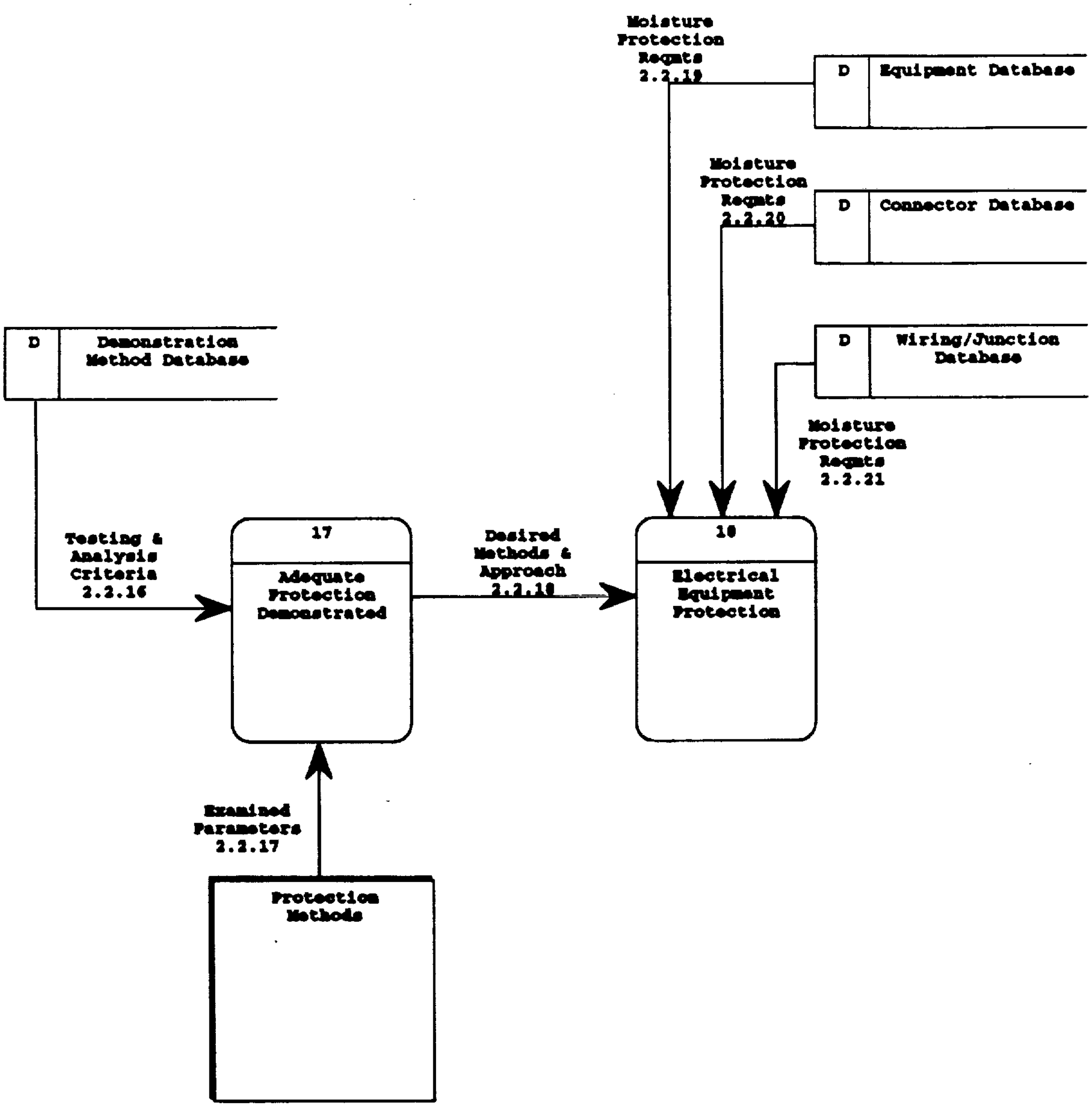
Design of circuitry for automatic shutdown of launch vehicle engine(s) shall include protection against possible engine shutdown coincidental with, or immediately after, launch vehicle release.(3)



MR_2.2_4
System Architect
Fri Sep 04, 1992 15:50

2.2 Electrical Issues (3d)

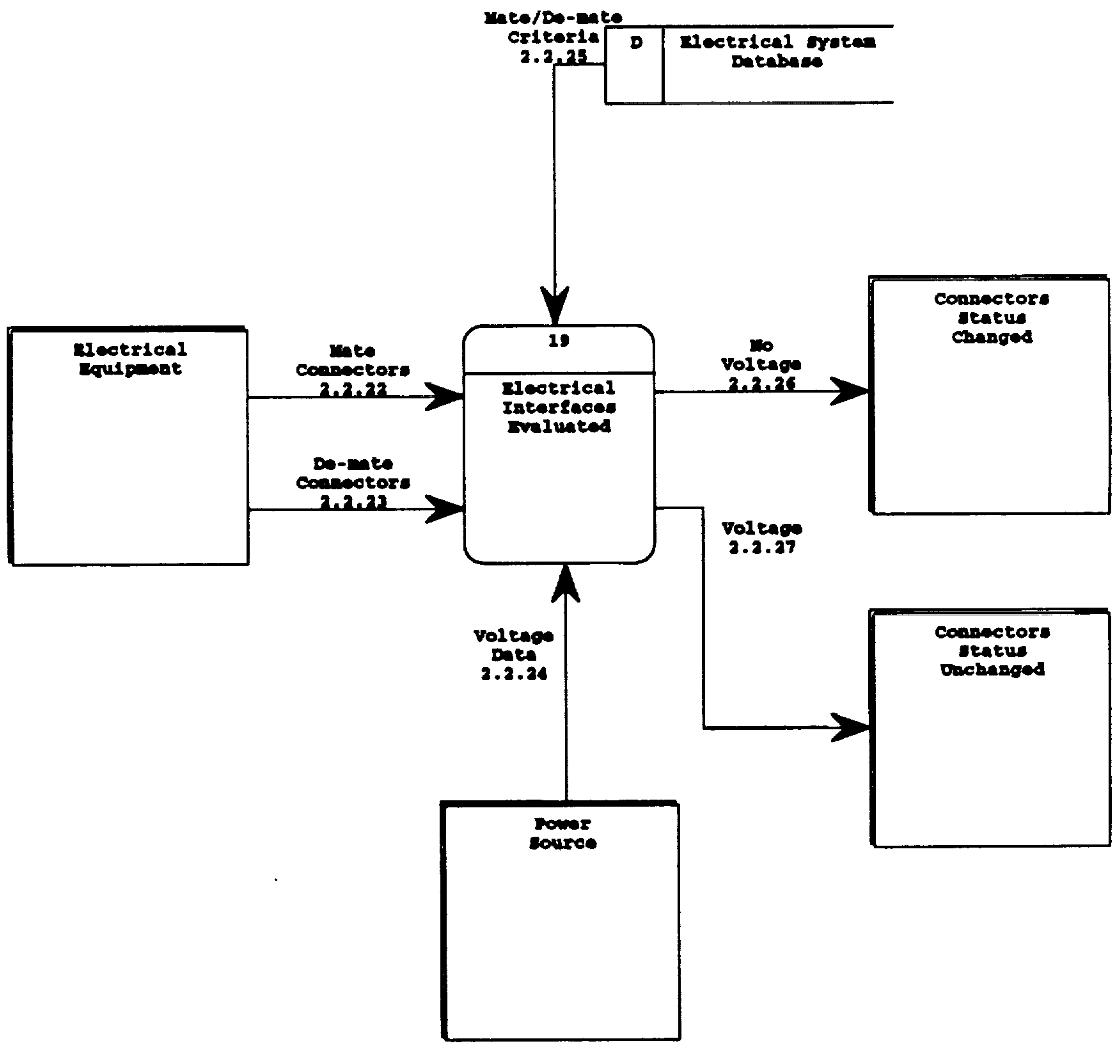
Electronic and electrical equipment, electrical connectors and wiring junctions to connectors shall be protected from moisture by methods which are demonstrated by test or analysis to provide adequate protection to prevent open and short circuits.(3)



MR_2.2_5
System Architect
Fri Sep 04, 1992 15:52

2.2 Electrical Issues (3e)

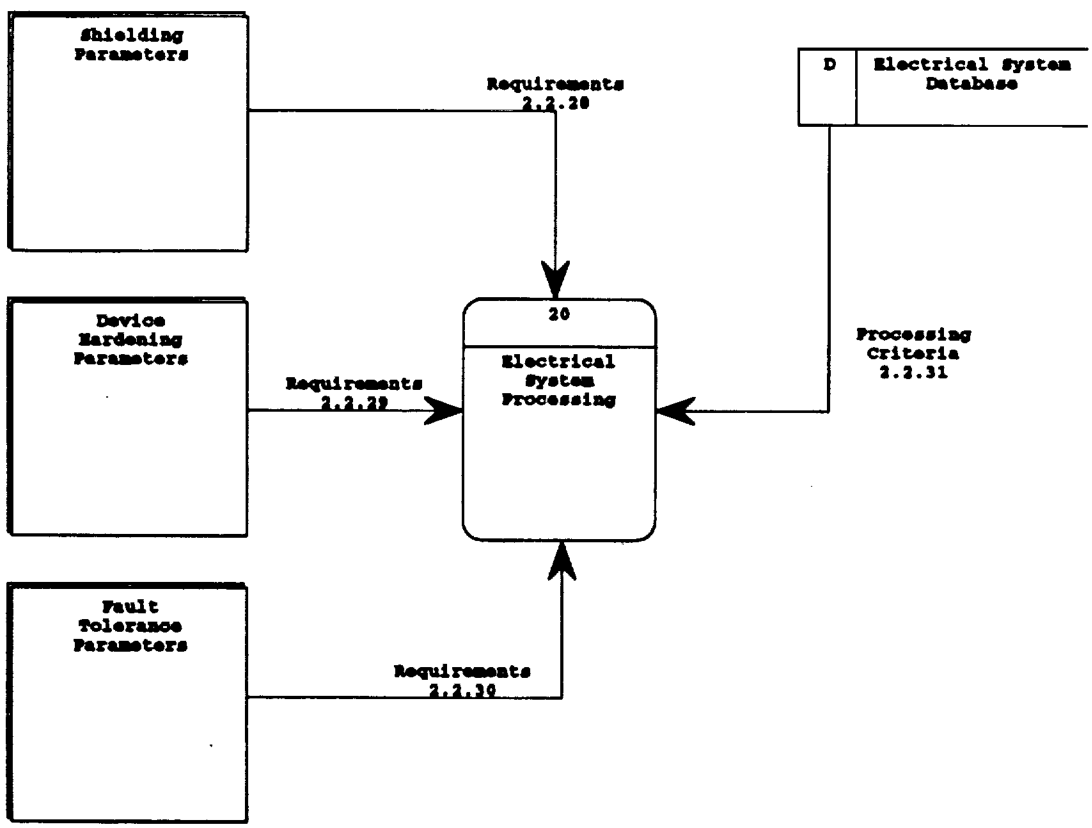
Unless connectors are specifically designed and approved for mating or demating in the existing environment under the loads being carried, they shall not be mated or demated until voltages have been removed from the powered side(s) of the connector.(3)



MR_2.2_6
System Architect
Fri Sep 04, 1992 15:54

2.2 Electrical Issues (3f)

Device hardening, circuit fault tolerance, shielding.(3)

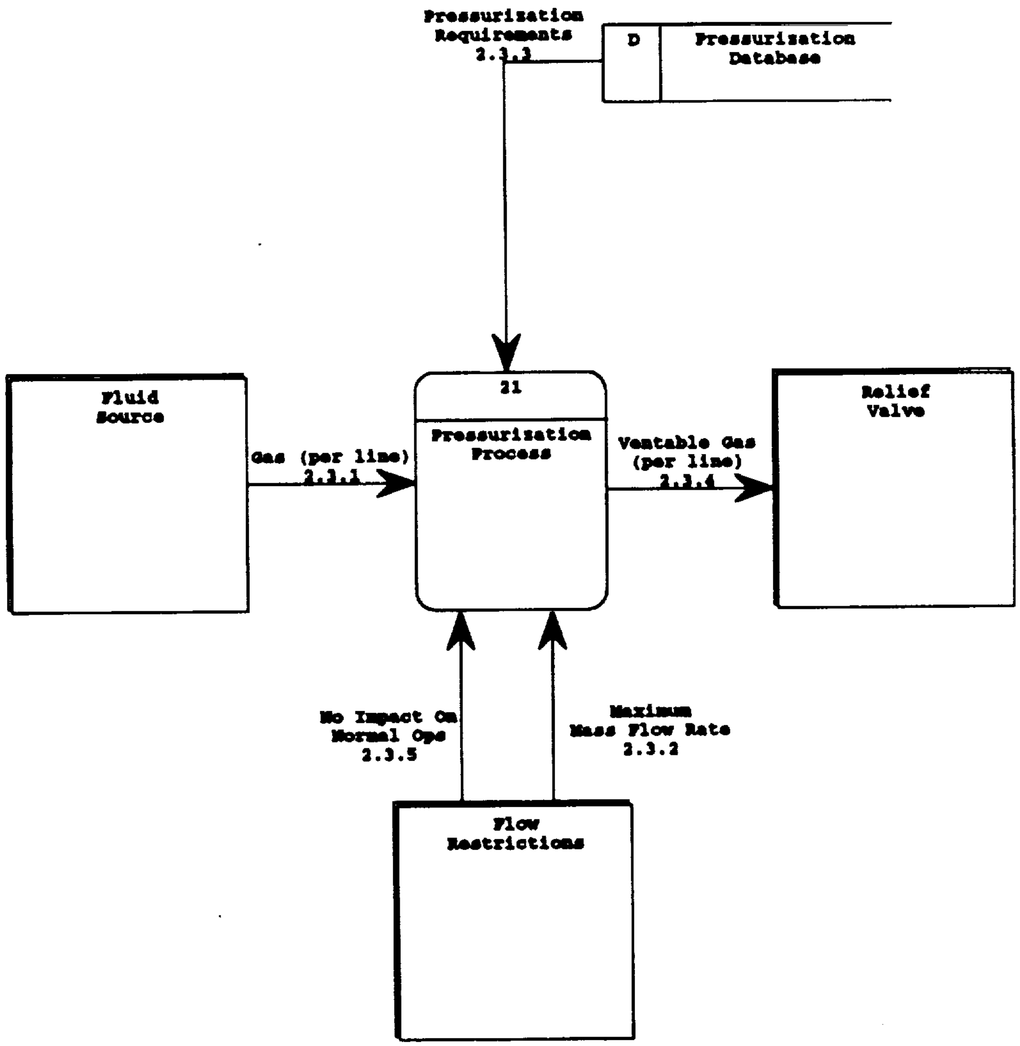


MR_2.3_1
System Architect
Fri Sep 04, 1992 16:20

2.3 Fluid Issues (3a,b)

Flow Restrictions - Pressurized Sources: Where pressurized gas lines could fail in such a way that the total gas supply dumped directly into a compartment would be greater than the relief valve or venting could handle without overpressurization of the compartment, necessary flow restrictions shall be incorporated at the pressure source to restrict the mass flow to a level that can be handled by the relief valve and/or venting.(3)

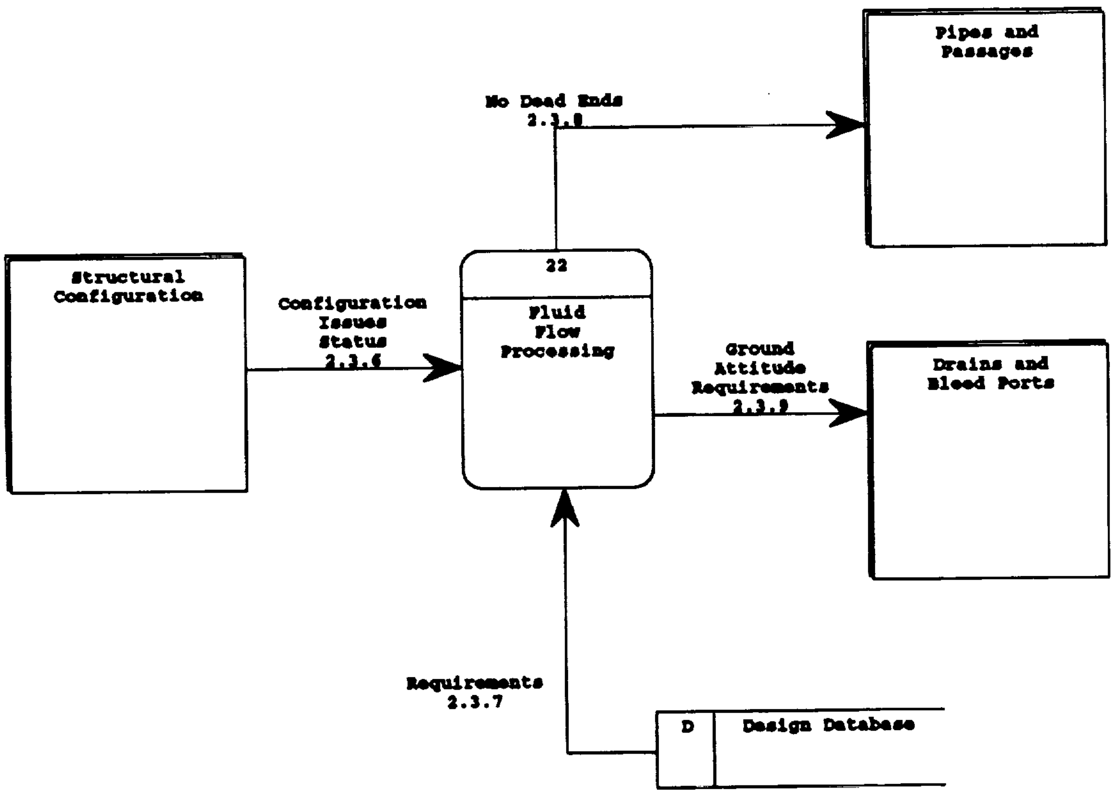
The flow restriction must not interfere with the normal operation of the system.(3)



MR_2.3_2
System Architect
Fri Sep 04, 1992 16:04

2.3 Fluid Issues (3d)

The systems shall be free of dead-ended piping or passages through which flushing fluid cannot be made to flow. Drain and bleed ports shall be provided for attitudes anticipated during ground servicing of the systems.(3)

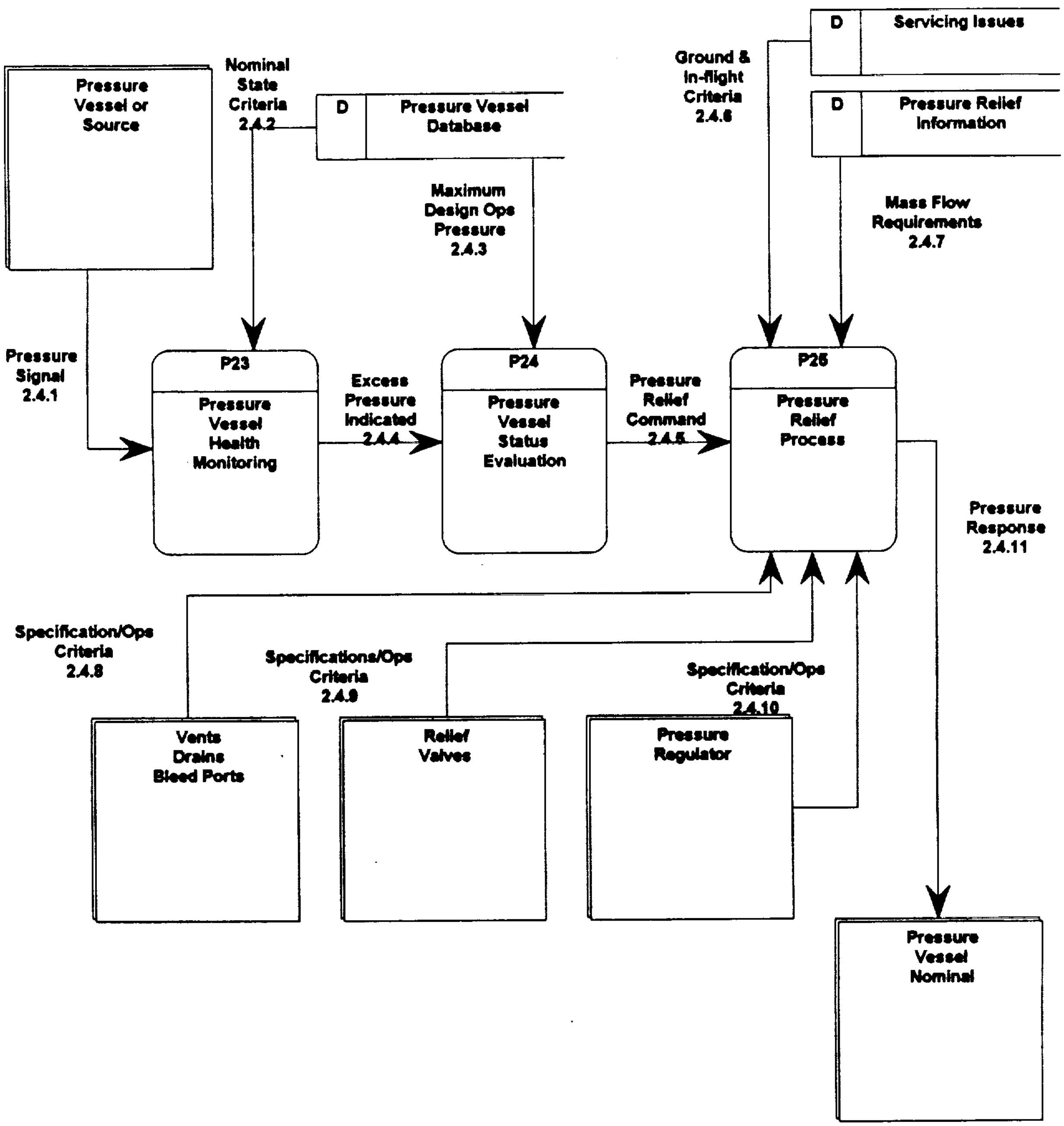


MR_2.4_1
System Architect
Thu Sep 10, 1992 18:41

2.4 Mechanical/Structural Issues (3a)

Pressure Vessel Relief: Pressure relief capability shall be provided for vessels where the contents, system design, or operation may cause an increase in internal pressure above the maximum designed operation pressure. Portions of fluid systems that trap fluids (become locked-up) should be considered pressure vessels and should be evaluated for the need of relief capabilities.(3)

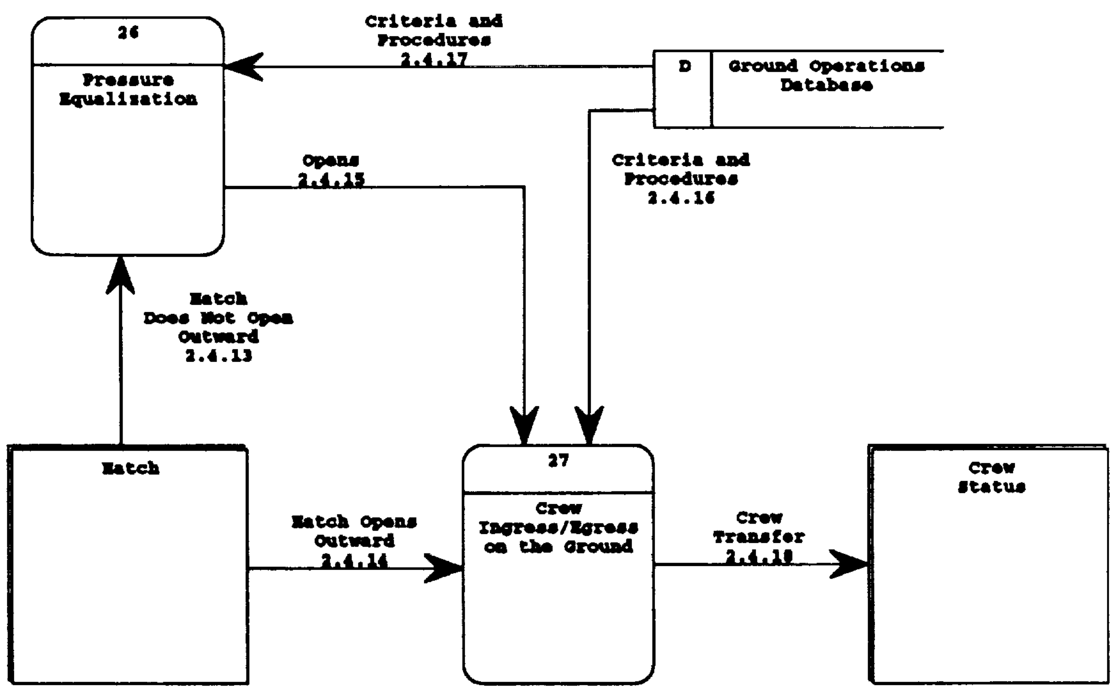
All flight vessels shall be protected during servicing, either on the ground or in flight, by relief valves in the servicing equipment. The relief valves shall be sized for sufficient mass flow to protect the vessel in the event of servicing pressure regulator failure. Such as failure shall not cause the vessel to exceed the maximum design operating pressure.(3)



MR_2.4_2
System Architect
Fri Sep 04, 1992 16:39

2.4 Mechanical/Structural Issues (3c)

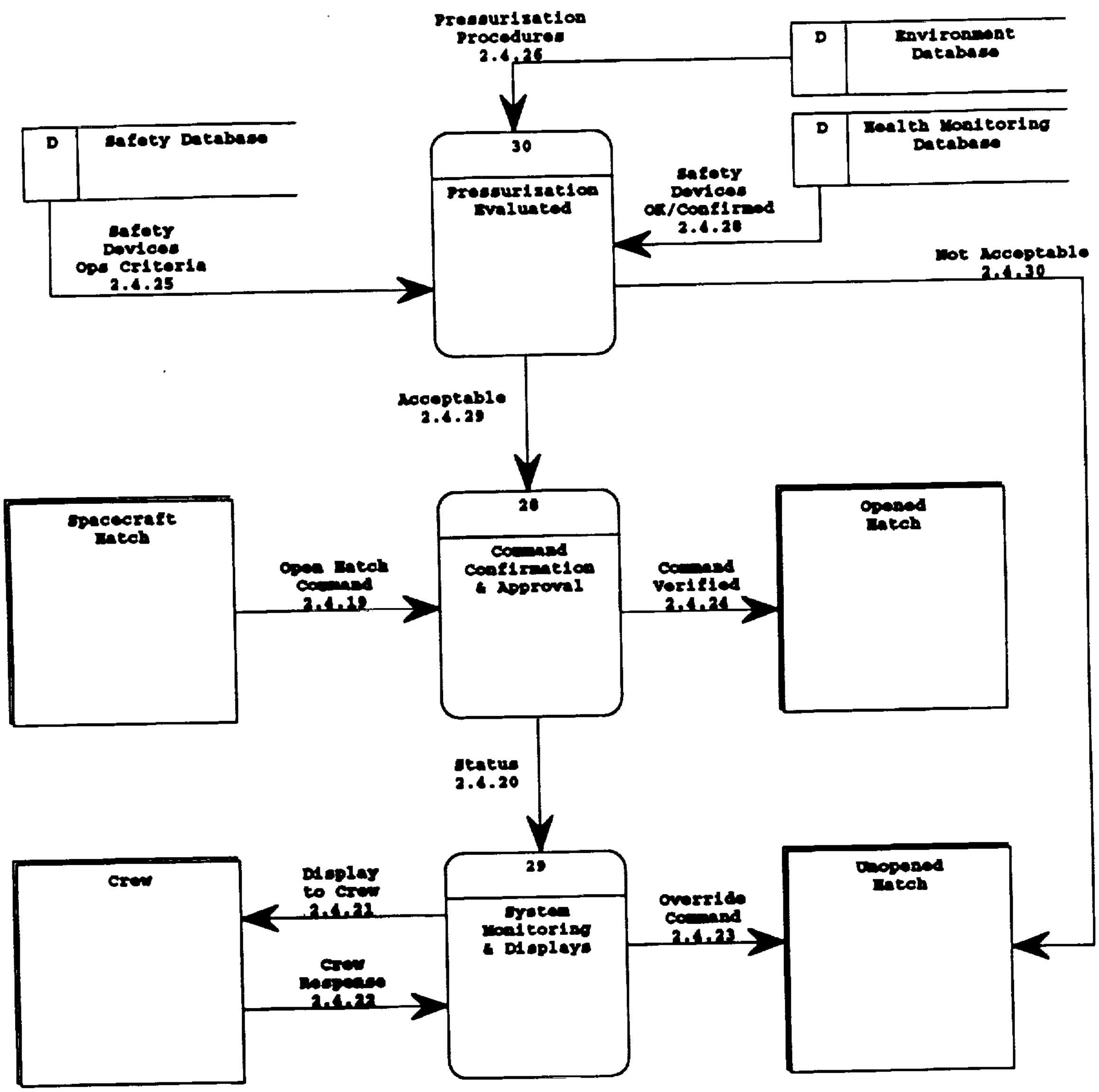
The primary flight crew ingress/egress hatch used during ground operations shall be designed to be outward opening from the pressurized spacecrat compartment. For designs where it is impractical to have an outward opening hatch, provisions will be made to rapidly equalize the pressure across the match.(3)



MR_2.4_3
System Architect
Fri Sep 04, 1992 16:41

2.4 Mechanical/Structural Issues (3d)

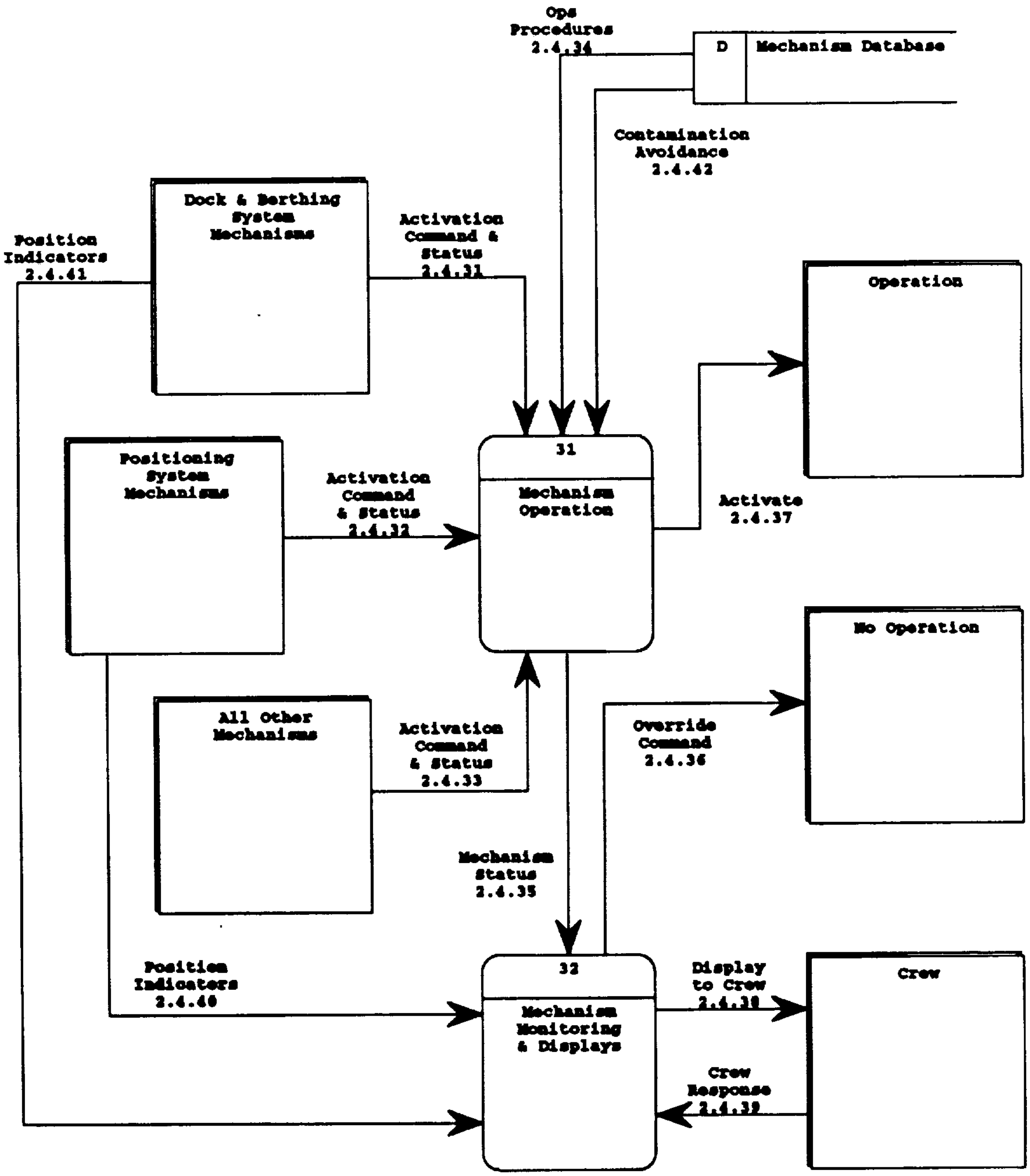
Reliable, redundant safety devices shall be provided to prevent inadvertent opening or rapid depressurization on orbit.(3)



MR_2.4_4
System Architect
Fri Sep 04, 1992 16:44

2.4 Mechanical/Structural Issues (3e)

All mechanisms shall have manual overrides. All mechnisms used on docking, berthing, and positioning systems shall have position indicators. External (outside actuator) limit switches shall not be used in areas susceptible to contamination.(3)

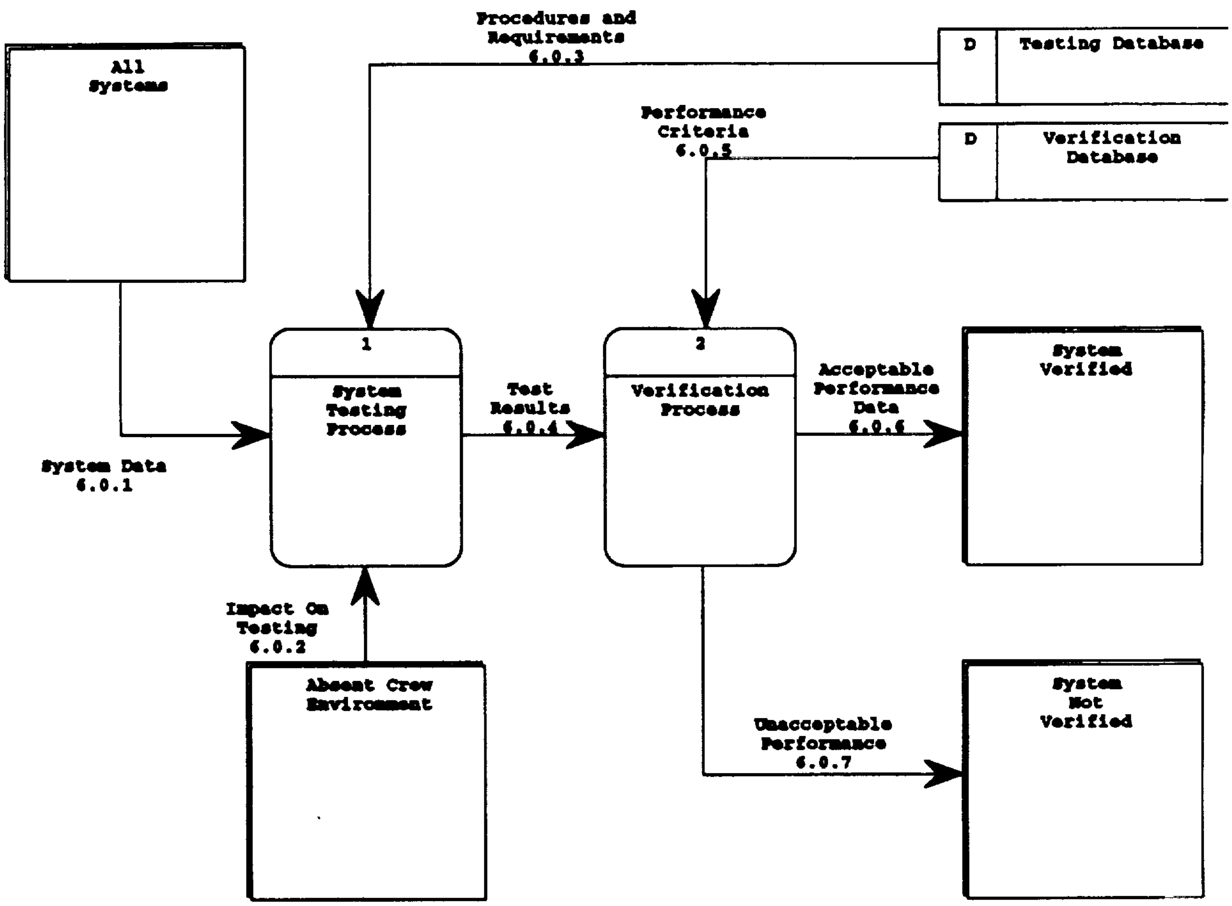


MR_6.0_1
System Architect
Fri Sep 04, 1992 16:47

6. Test and Verification (1,4)

The system should be fully demonstrated unmanned prior to comitting to manned operations.(1)

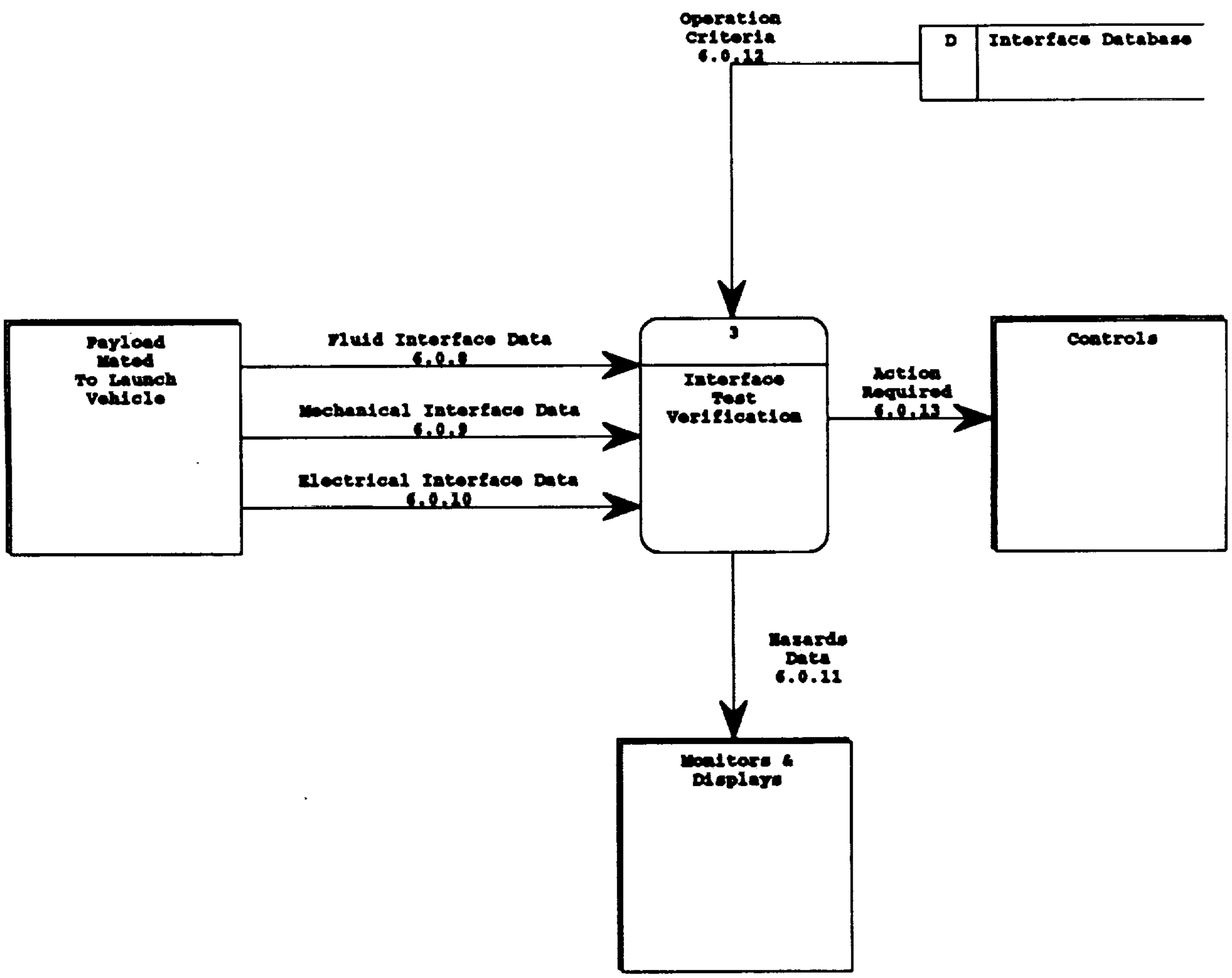
Prior to committing man to flight, the system shall be fully demonstrated in flight, unmanned.(4)



MR_6.0_2
System Architect
Fri Sep 04, 1992 16:51

6. Test and Verification (5)

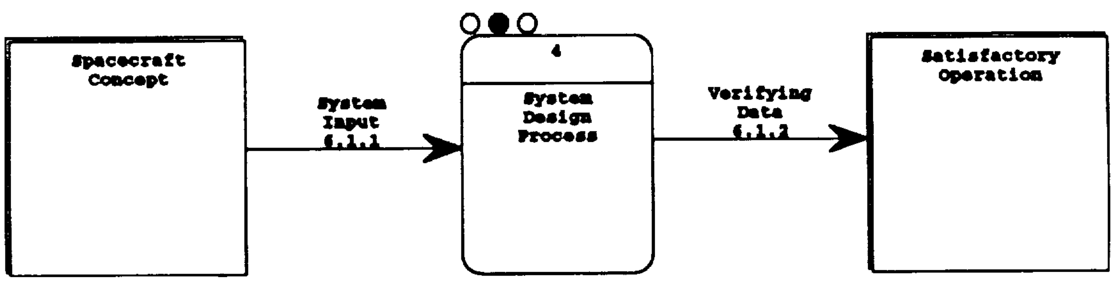
Payload hazards being controlled by launch vehicle provided services will require post-mate interface test verification for both controls and monitors.(5)

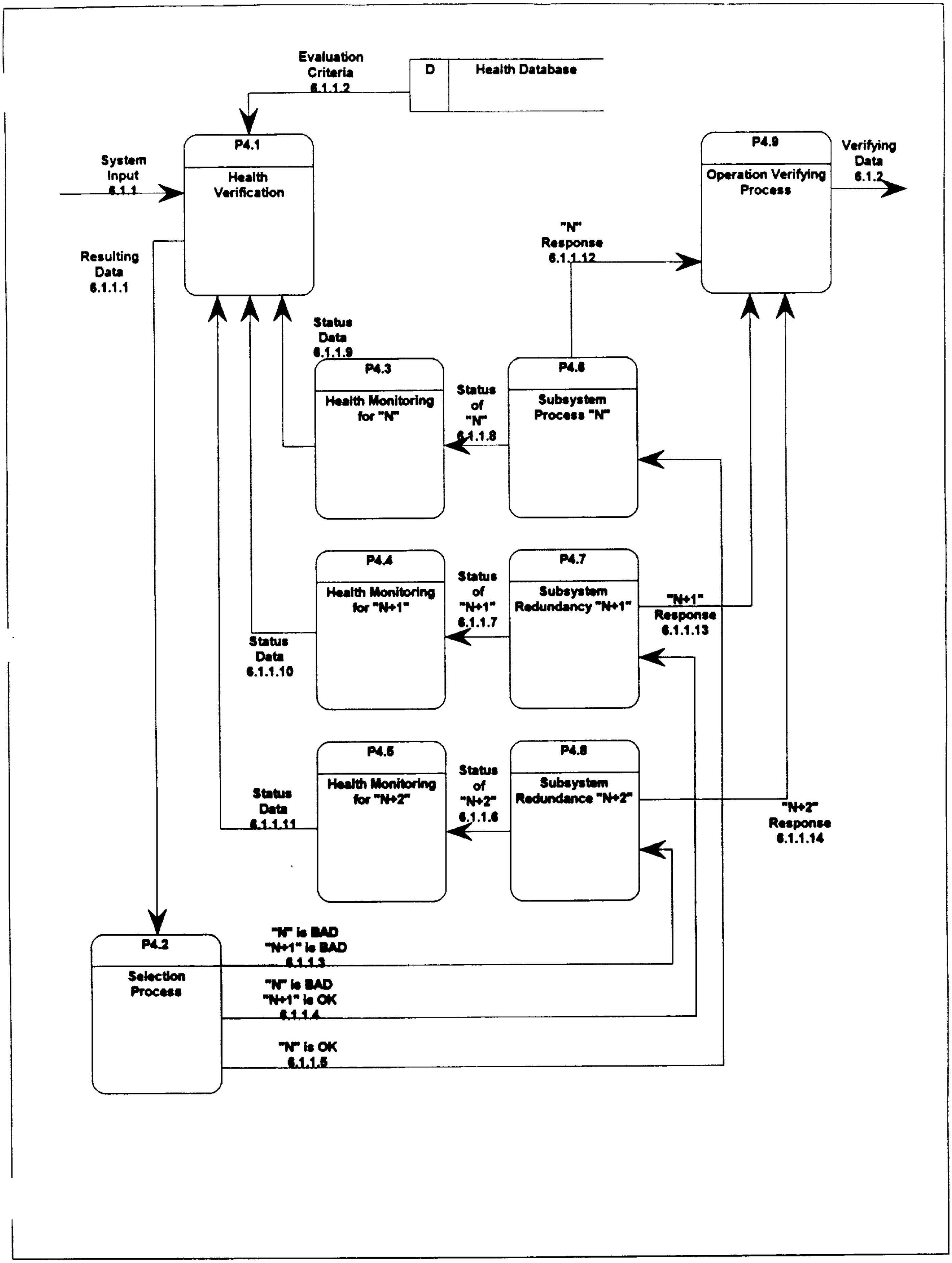


MR_6.1_1
System Architect
Tue Sep 08, 1992 09:44

6.1 Redundant Path Verification (3)

The design of spacecraft systems and subsystems incorporating redundancies shall include a means of verifying satisfactory operation of each redundant path at any time the system and/or subsystem is determined to require testing prior to launch and during the mission.(3)



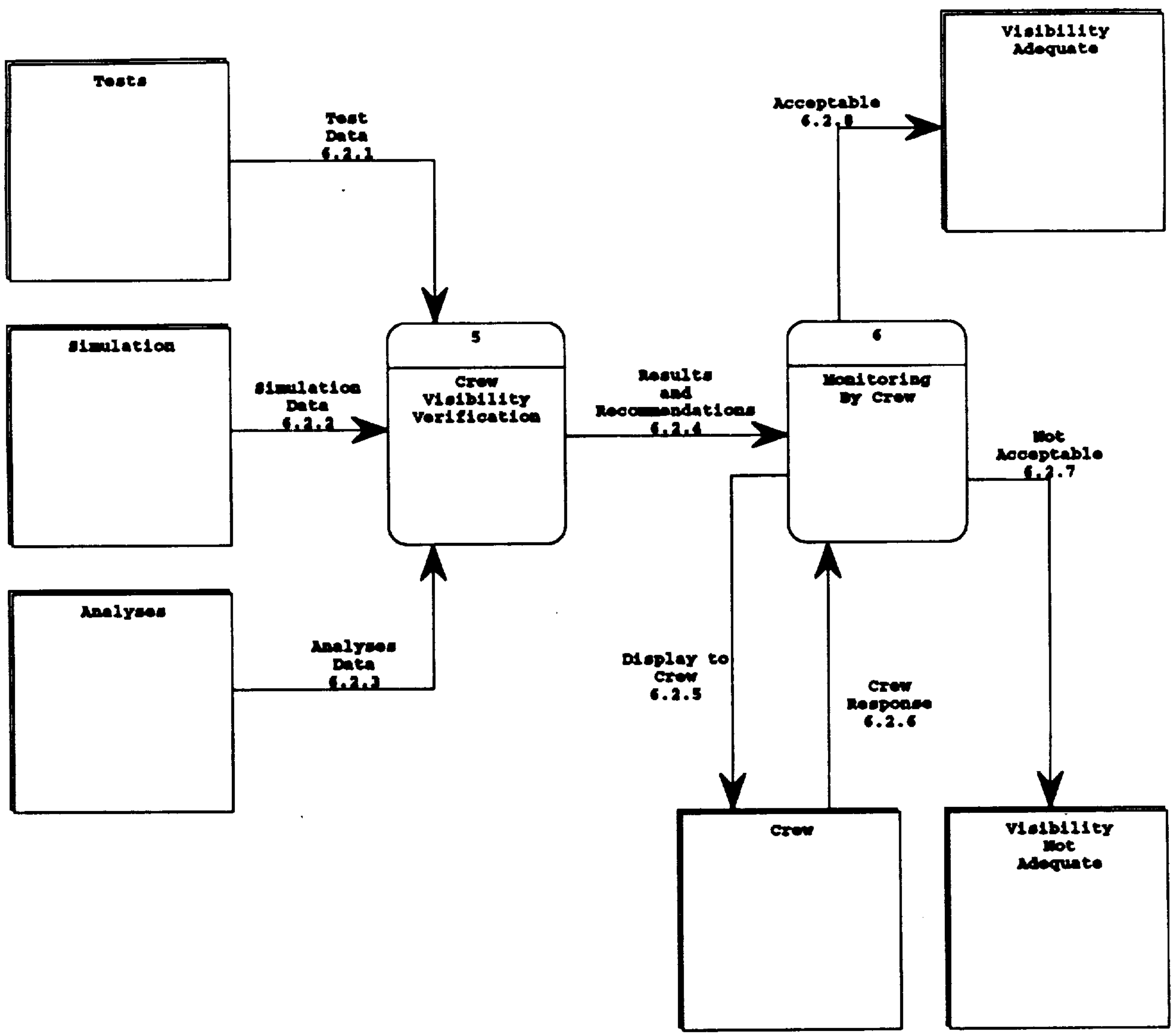


Evaluation Criteria 6.1.1.2
D
Health Database
P4.1
Health Verification
System Input 6.1.1
Resulting Data 6.1.1.1
P4.9
Operation Verifying Process
Verifying Data 6.1.2
"N" Response 6.1.1.12
Status Data 6.1.1.9
P4.3
Health Monitoring for "N"
Status of "N" 6.1.1.8
P4.6
Subsystem Process "N"
P4.4
Health Monitoring for "N+1"
Status of "N+1" 6.1.1.7
P4.7
Subsystem Redundancy "N+1"
"N+1" Response 6.1.1.13
Status Data 6.1.1.10
P4.5
Health Monitoring for "N+2"
Status of "N+2" 6.1.1.6
P4.8
Subsystem Redundance "N+2"
"N+2" Response 6.1.1.14
Status Data 6.1.1.11
P4.2
Selection Process
"N" is BAD "N+1" is BAD 6.1.1.3
"N" is BAD "N+1" is OK 6.1.1.4
"N" is OK 6.1.1.5

MR_6.2_1
System Architect
Tue Sep 08, 1992 09:48

6.2 Adequate External Visibility
Verification (3)

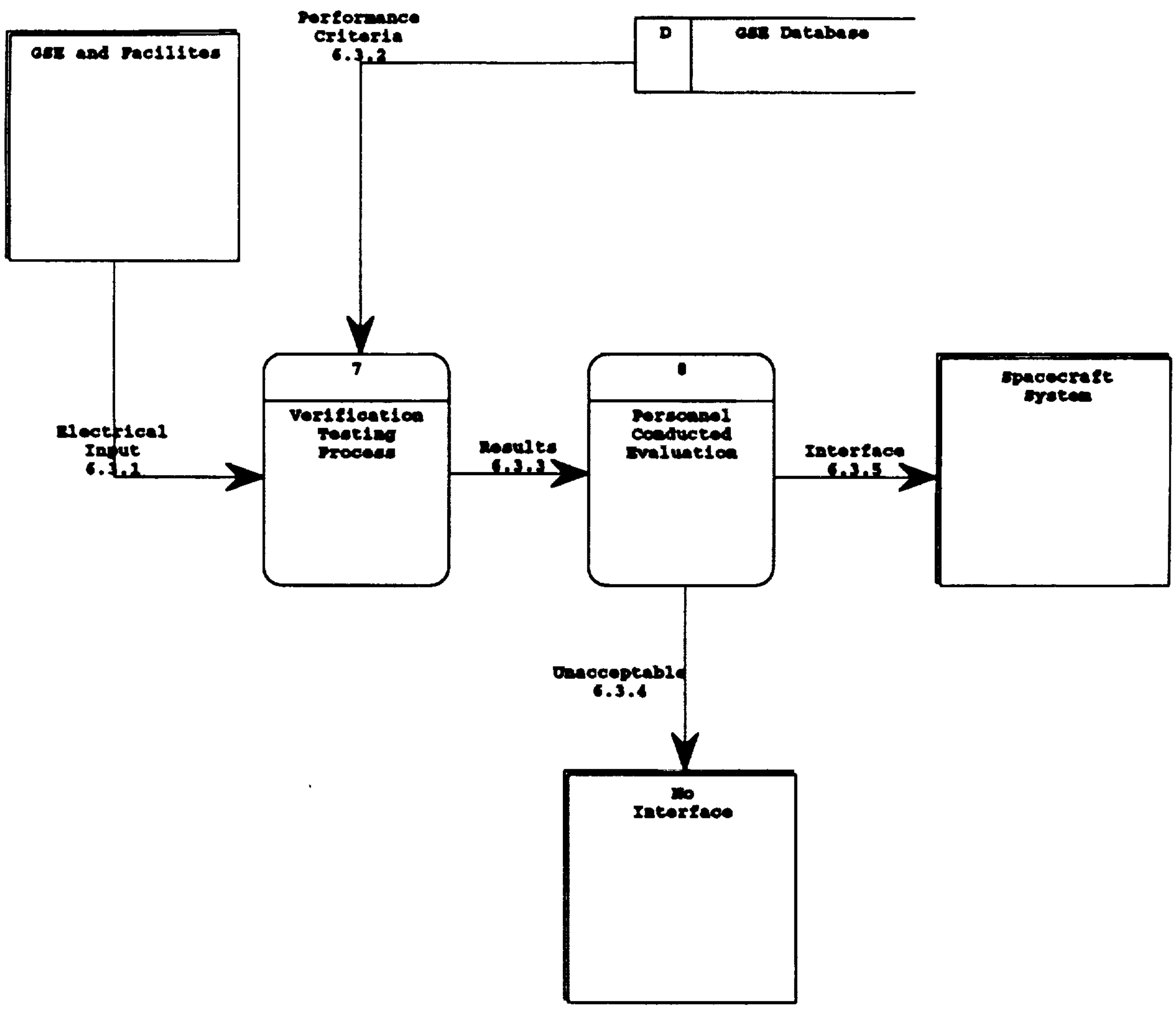
Visibility verification for manned spacecraft shall include tests, simulations, or analyses to verify that the crew will have adequate visibility during all anticipated phases and environmental conditions of the planned mission.(3)



MR_6.3_1
System Architect
Tue Sep 08, 1992 09:53

6.3 Verification Tests for Electrical and
Electronic Supplies and Loads

Ground support equipment, facilities, and other equipment to be connected to a spacecraft system for operation, testing, checkout, or maintenance shall be designed to that routine verification tests can be conducted before each connection is made, to ensure that each electrical and electronic input to the spacecraft is compatible with the spacecraft system.(3)



MR_8.1_1
System Architect
Tue Sep 08, 1992 09:57

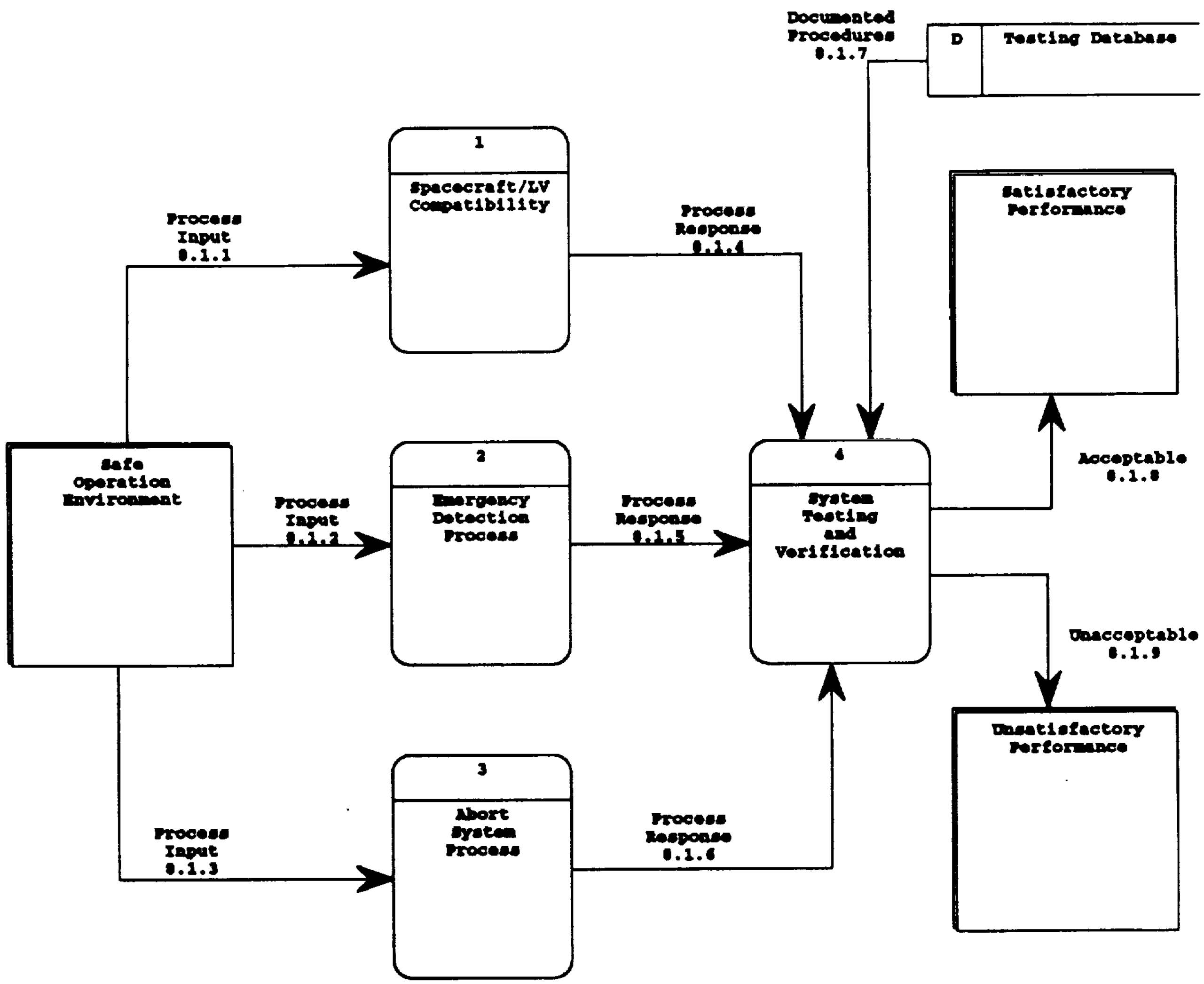
8.1 Safe Operations (2a,b,c,d)

...special testing to assure compatibility of the spacecraft with the launch vehicle.(2)

...proper operation of emergency detection ... systems(2)

...proper operation of ... abort systems(2)

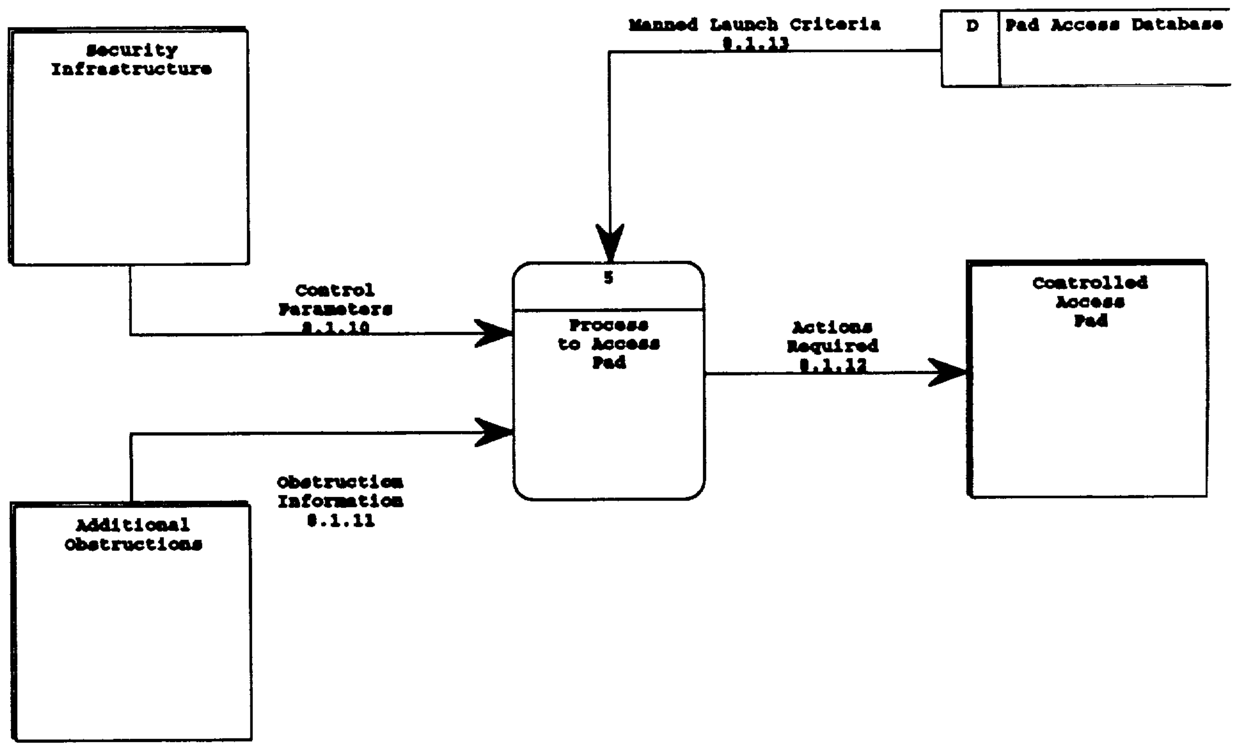
...all test procedures that are accomplished must be correct and formally revised before the procedure is accomplished.(2)



MR_8.1_2
System Architect
Tue Sep 08, 1992 09:59

8.1 Safe Operations (2e)

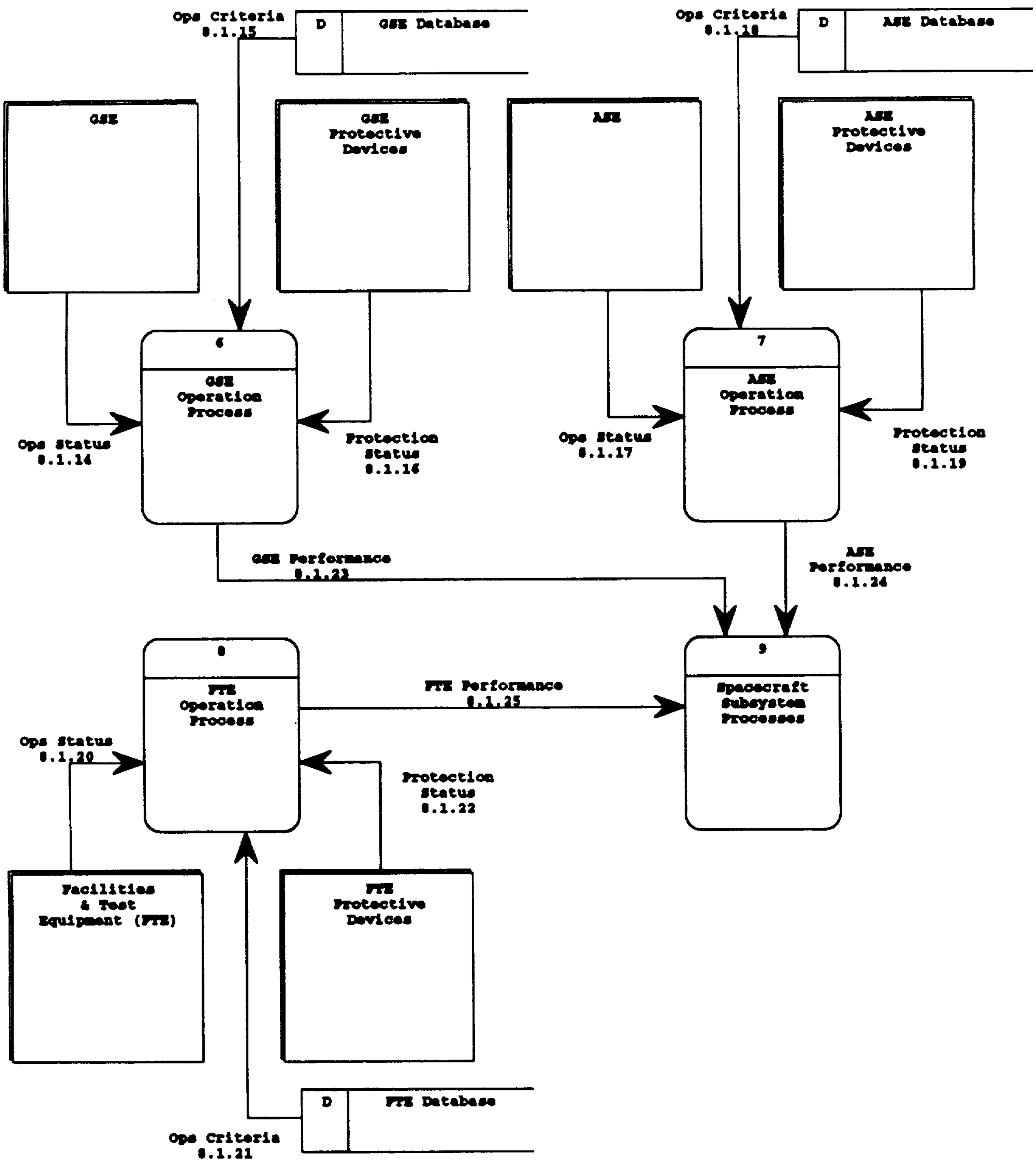
Generally, on a manned launch, pad access is strictly controlled...(2)



MR_8.1_3
System Architect
Tue Sep 08, 1992 10:01

8.1 Safe Operations (3a)

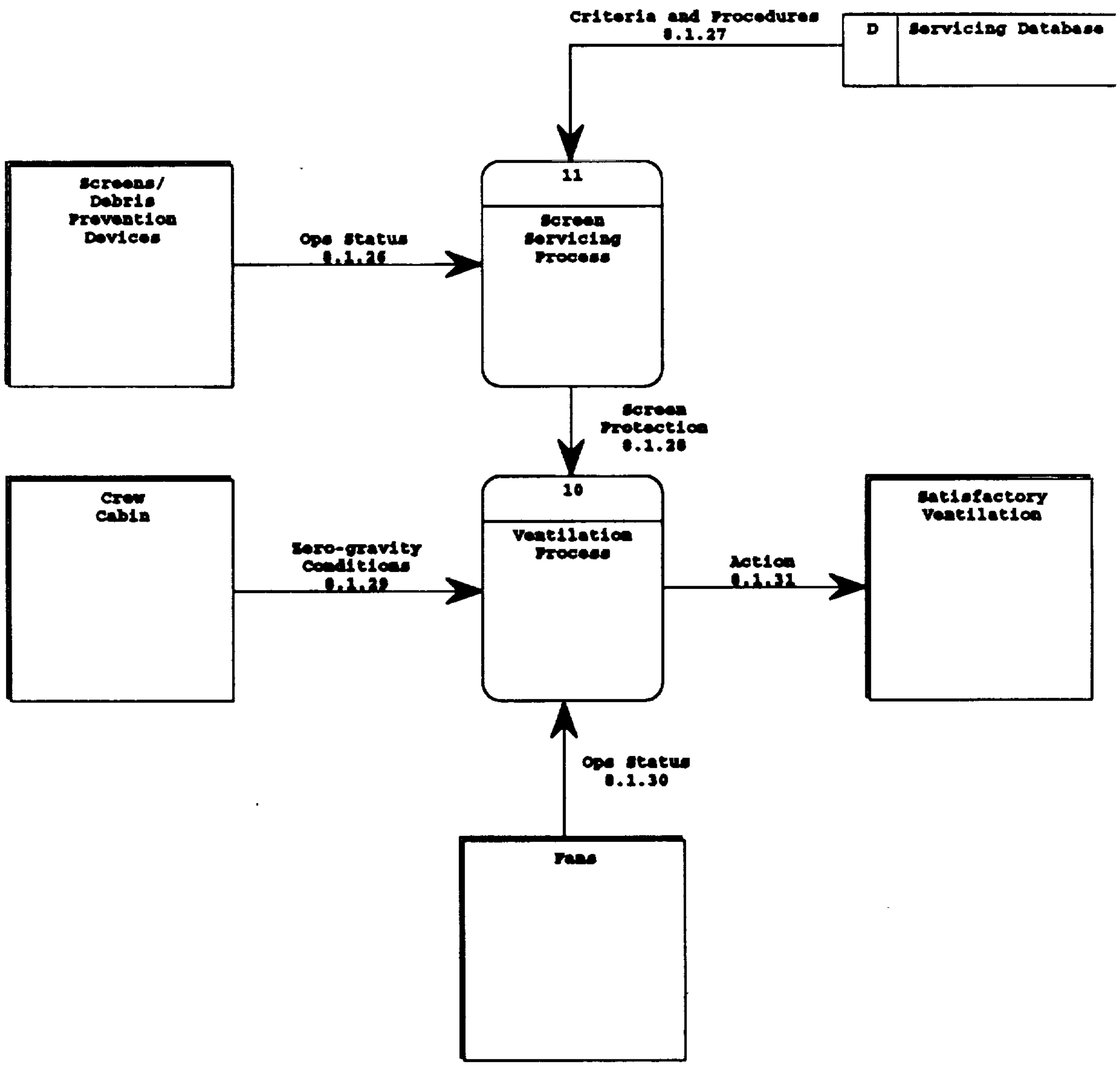
GSE and ASE Protective Devices: Ground support equipment (GSE), airborne support equipment (ASE), facility equipment, or test equipment used in ground or flight operations shall be equipped with protective devices to preserve safe operating margins of the spacecraft subsystems.(3)



MR_8.1_4
System Architect
Tue Sep 08, 1992 10:05

8.1 Safe Operations (3b)

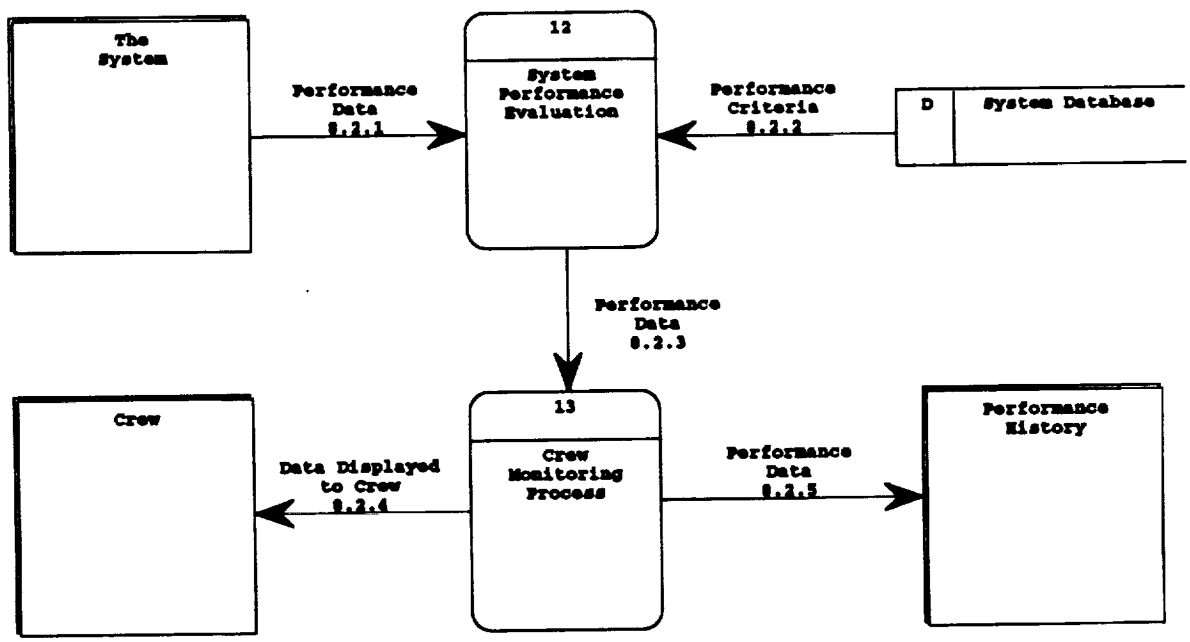
Cabin Ventilation: Crew cabin module ventilation fans shall be protected by screens or other devices to prevent entrance of debris that could damage or jam the fan blades during zero-gravity conditions. Such screens or other devices shall be serviceable and/or replaceable.(3)



MR_8.2_1
System Architect
Tue Sep 08, 1992 10:08

8.2 Periodic Check-out and Maintenance (1)

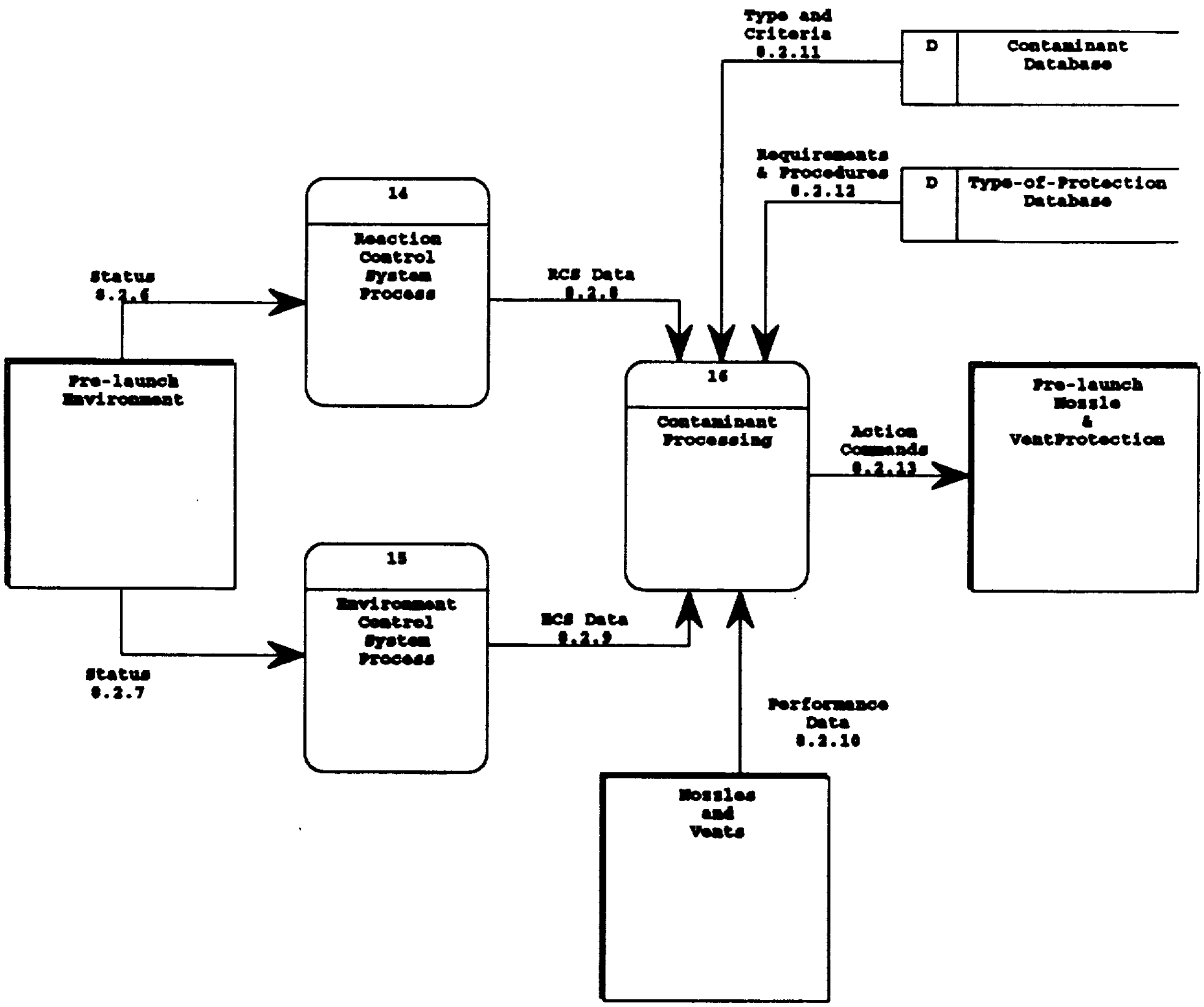
A history of system performance shall be maintained and monitored to detect possible degradations over time. Fail-operational/fail-safe subsystems shall allow maintenance to upgrade the subsystem without being degraded below fail-safe during maintenance actions.(1)



MR_8.2_2
System Architect
Tue Sep 08, 1992 10:10

8.2 Periodic Check-out and Maintenance (3a)

Pre-Launch Nozzle/Vent Protection: All nozzles and vents used in manned spacecraft systems, such as those of the reaction control system and environmental systems, shall be protected from entrance of rain, debris, or other contaminnants prior to launch.(3)



MR_8.2_3
System Architect
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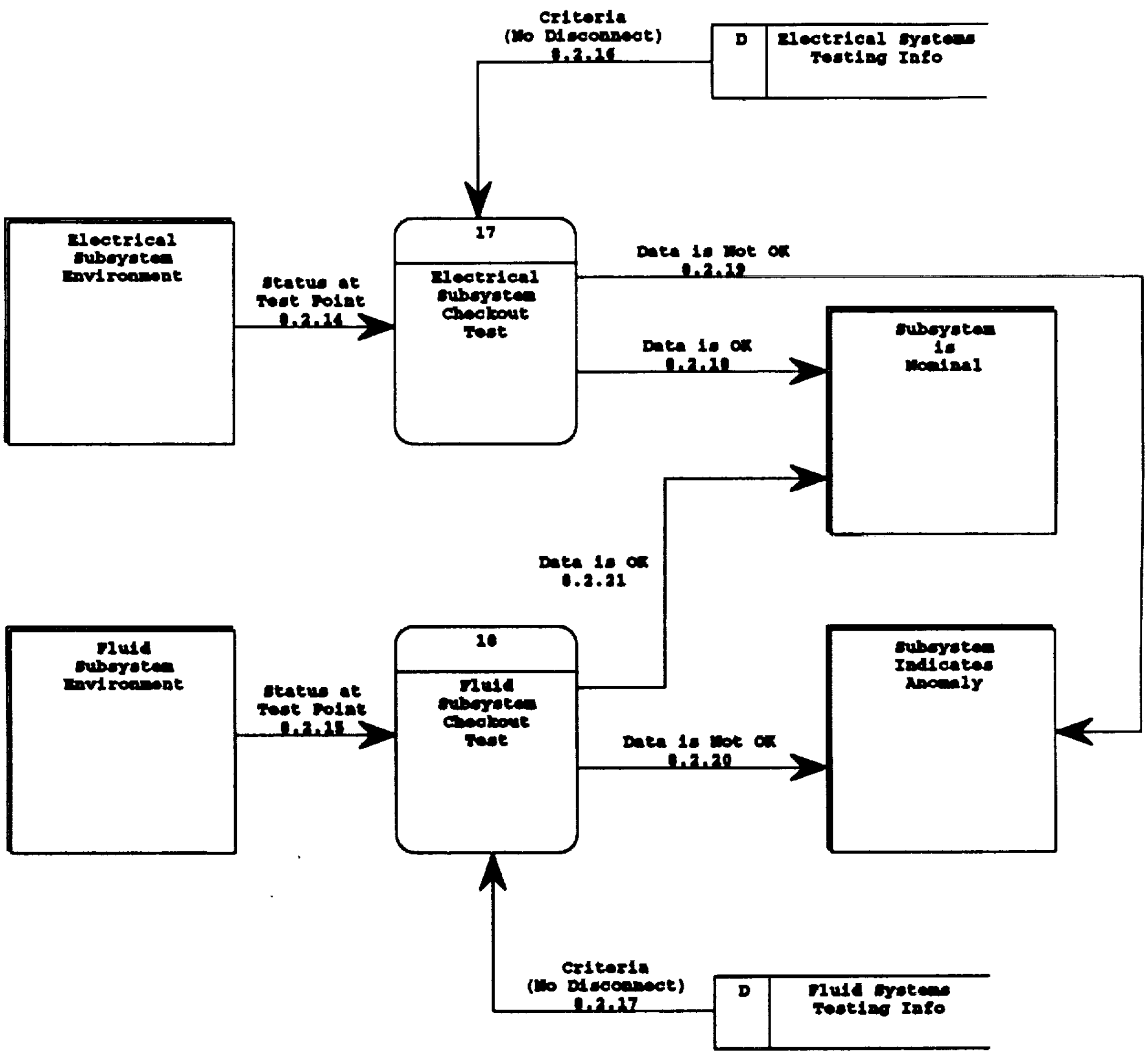
8.2 Periodic Check-out and Maintenance(3b)

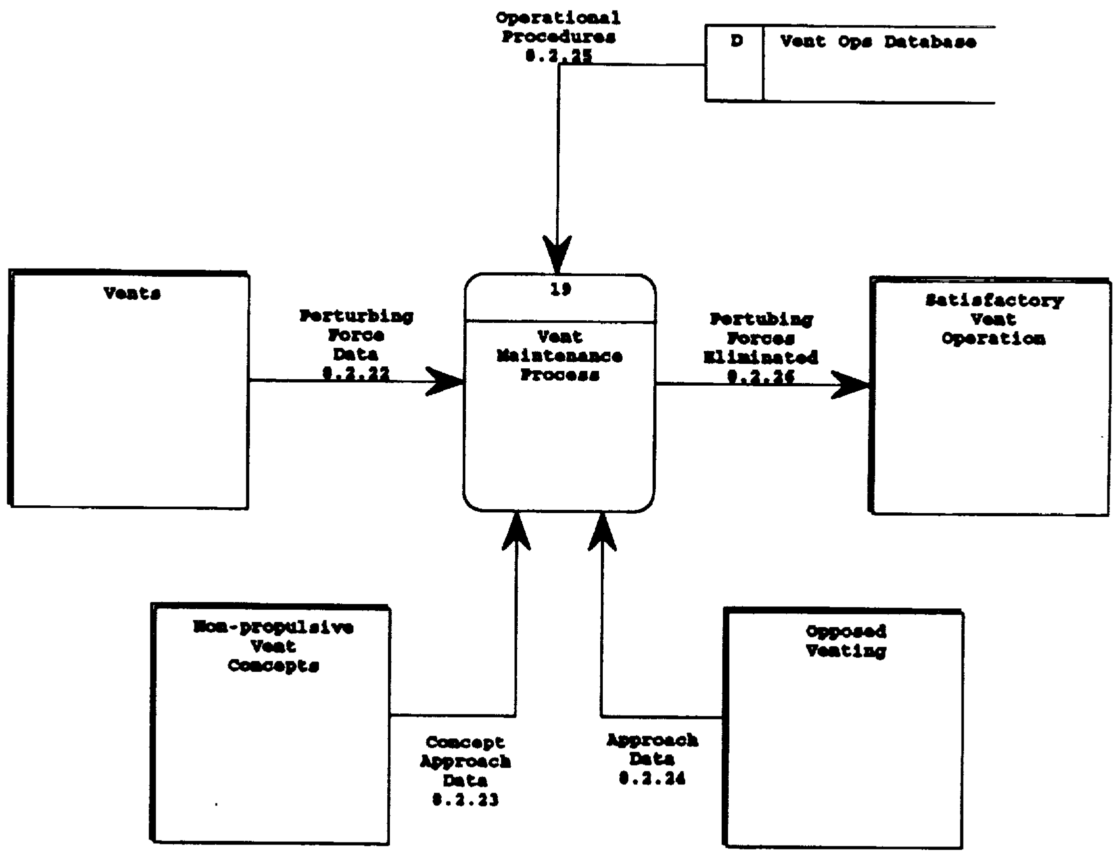
Electrical and fluid-handling subsystems shall include checkout test points which will permit checkout tests to be made without disconnecting tubing or electrical connectors normally connected in flight.(3)



8.2 Periodic Check-out and Maintenance(3d)

Tese ports will not utilize permanent closure methods.(3)

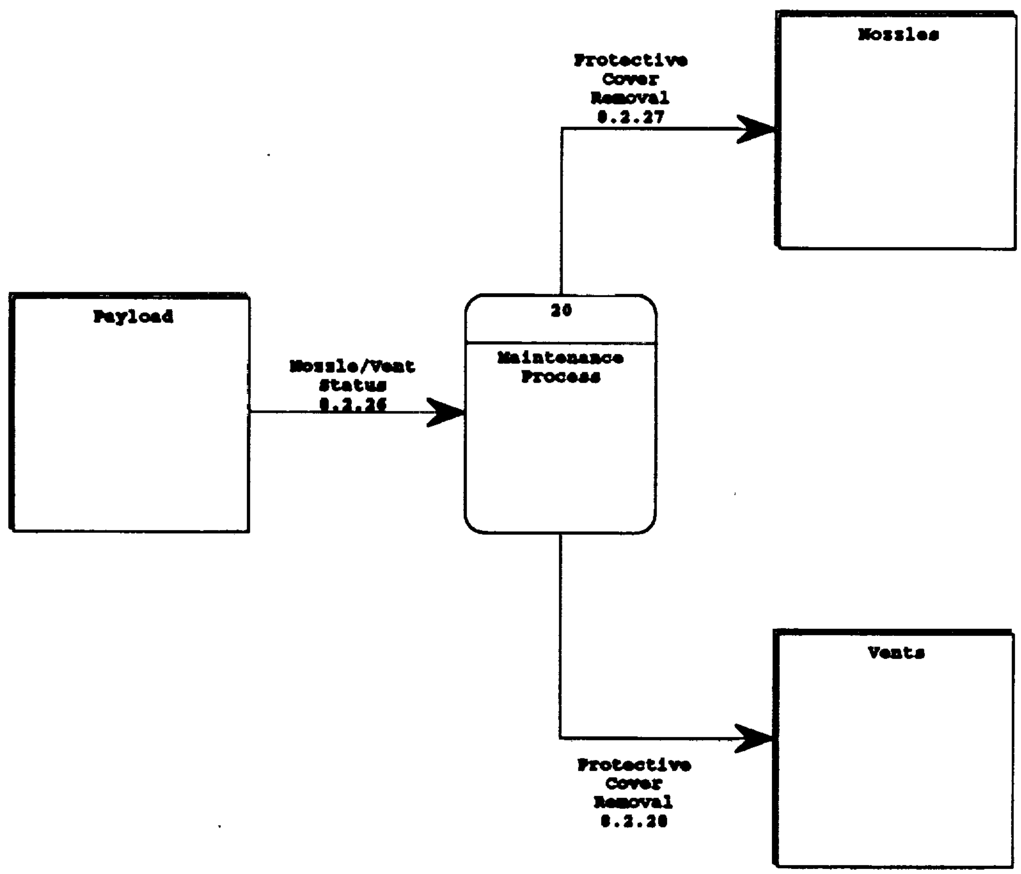
Nonpropulsive vent concepts, opposed venting, operational procedures, or similar methods shall be used to eliminate the undesirable effects of perturbing forces resulting from such vents.(3)



MR_8.2_5
System Architect
Tue Sep 08, 1992 10:18

8.2 Periodic Check-out and Maintenance(3e)

Protectie covers for nozzles or vents located within the payload by shall be designed to be readily removable during the countdown prior to launch or prior to final closure of the Orbiter payload bay doors.(3)



ADVANCED TRANSPORTATION SYSTEM STUDY

Manned Launch Vehicle Concepts for Two Way Transportation System Payloads to LEO

Learning Curve Analysis of Orbiter Processing

Research Report

June, 1992

Contract NAS8-39207

Prepared by: M. D. Heileman

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Rockwell International
July 1992

Learning Curve Analysis of Orbiter Processing: A Research Report

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ABSTRACT

This paper shows how learning curve analysis can be applied to Space Shuttle orbiter processing in preparation for launch. Learning curve theory can be used to estimate ground processing time at the main orbiter facilities on the Kennedy Space Center. Results of this analysis indicate that future orbiter turn-around processing times will allow additional processing capacity which may result in an increasing Space Shuttle launch rate.

INTRODUCTION

A learning curve is the phenomenon whereby, as the number of cycles (or process flows) increases, the time per cycle or the cost per cycle decreases for a large number of cycles. Learning that occurs while an organization produces many units of a particular product is represented by the "production progress function" [1]. It is recognized that learning is time dependent. The learning curve is an accepted management tool for determining expected manpower requirements, evaluating training programs, estimating the costs of future production, and scheduling shop labor, particularly when performance time decreases as a result of learning through repetition [2].

The theory of learning curve is based on the assumption that as the total quantity of units produced *doubles*, the cumulative average time per unit declines at some *constant* rate known as the learning efficiency. Since it is unlikely that this same rate will hold every time production is doubled, the actual learning efficiency is estimated by an average [2].

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MATHEMATICS OF LEARNING

Learning equations generally take the form of

$$Y = KX^{-A} \qquad (1)$$

where Y = the time per cycle
K = the time for the first cycle
X = the number of cycles
A = a constant, the value which is determined by the learning rate.

If we take the log of both sides of equation 1, we get a straight line.

$$\log Y = \log K - A \log X \qquad (2)$$

Thus if we plot the equation on log-log paper, A will be the slope and K will be the intercept. One of the useful properties of this equation is that every time X (the number of cycles) is doubled, Y (the time per cycle) decreases by a fixed percentage. This is the source of the commonly used term "percent learning curve." For example, every time the units double, the value of Y for a 90% curve is 90% of the previous value. Suppose the first unit had a cycle time of 10 hours; then we would get the results shown in Table 1 by successively doubling the number of units [1].

Table 1 - An example of the cycle times for a 90% curve

Number of Units (X)	Cycle Time (Y)
1	10.0
2	9.0
4	8.1
8	7.3
16	6.6
32	5.9
64	5.3
128	4.8
256	4.3
512	3.9

Figure 1 is a plot of a 90% curve using Cartesian coordinates and Figure 2 is the same equation plotted on log-log paper. Using the conditions of Table 1, where K = 10 hours and a 90% learning curve, we could determine the value of A (logarithmic slope) using equation 1 and any value of X except 1. From Table 1 with X = 2 and Y = 9.0, equation 1 becomes as follows.

$$9 = (10)(2)^{-A}$$

$$\log 9 = \log 10 - A \log 2$$

$$A = (\log 10 - \log 9) / \log 2 = 0.1520$$

The learning rate or efficiency (L) is the percentage by which the cycle (or processing) time is reduced every time the cumulative production output is doubled. From the above solution, L can be derived as follows.

$$A = (\log 1 - \log L) / \log 2$$

$$A = -\log L / \log 2$$

$$\log L = -A \log 2$$

$$L = 2^{-A} \tag{3}$$

Substituting A = 0.1520 for a 90% learning curve into equation 3, we get the following verification.

$$L = 2^{-0.1520} = 0.9000$$

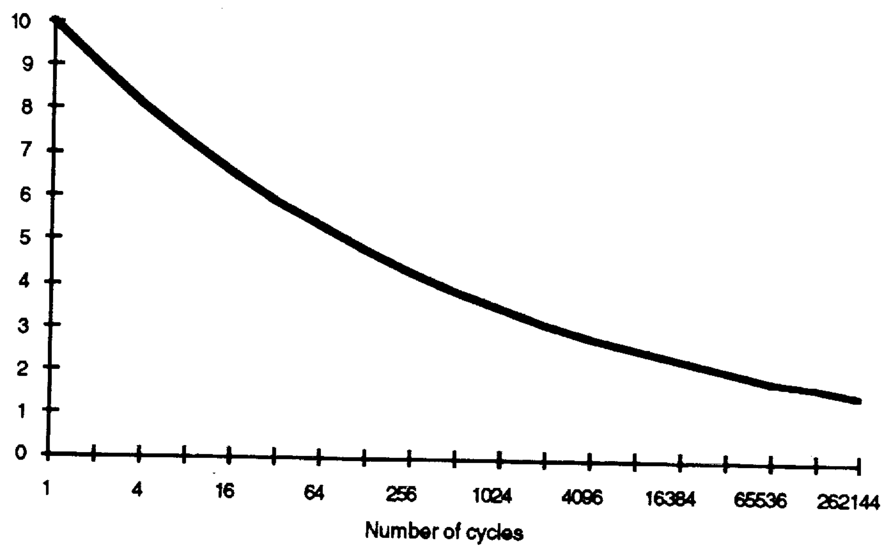

Figure 1 - Plot of 90% curve (arithmetic)

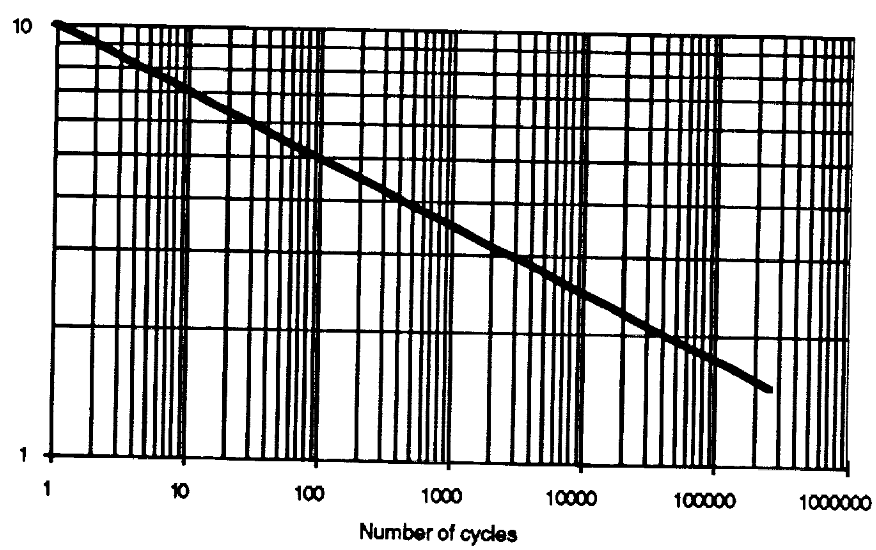

Figure 2 - Plot of 90% curve (logarithmic)

ORBITER PROCESSING DATA

Historical data concerning Space Shuttle orbiter processing time in preparation for a launch was obtained from the Mission Planning Office at the Kennedy Space Center [3]. Table 2 shows the number of work days spent preparing Shuttle orbiter vehicles (OV's) for missions at their major processing facilities. Those facilities are the Orbiter Processing Facility (OPF), the Vehicle Assembly Building (VAB), and the launch pad. The first column in this table contains the vehicle flow number (which is the number of times an orbiter has been processed at the Kennedy Space Center facilities), and the next five columns contain the orbiter tail number and name along with the cumulative processing times at the major facilities. The sixth column has the average cumulative orbiter processing time per flow. The orbiter processing times up to and including mission STS-51L, the tenth flow of OV-099 Challenger, are shown boxed in Table 2.

Table 2 - Sum and average per flow of orbiter processing times at the OPF, VAB, and launch pad

Flow #	OV-099 Challenger	OV-102 Columbia	OV-103 Discovery	OV-104 Atlantis	OV-105 Endeavour	OV Average
1	244	668	210	132	272	305
2	60	187	56	46		87
3	55	97	56	236		111
4	80	77	73	133		91
5	53	102	58	153		92
6	80	128	56	110		94
7	132	143	322	245		211
8	75	227	149	125		144
9	53	129	162	101		111
10	63	300	126	103		148
11		114	149	88		117
12		135	180			158
13			113			113
14			105			105

The orbiter vehicle average processing time was calculated because many flight preparation flows tend to have similar requirements and characteristics. For example, the first flow includes factory check-out testing and the fifth flow usually requires vehicle structural inspections. Additionally, personnel assignments are often made by OV and not by facility; and open work may be carried from one facility to the next facility in the processing flow.

After the mission STS-51L accident, significant changes were made in the rules and procedures for orbiter ground turn-around launch preparation activities at their major processing facilities. Numerous inspection points and safety procedures were added to the standard procedures. Many hardware modifications were incorporated. The Space Shuttle program went through a stand-down of over two-and-one-half years, interrupting the continuous production flow assumed for learning curve purposes. This implies that there are two different distributions for orbiter processing. Therefore the orbiter processing times through preparation for mission STS-51L were segregated from the orbiter processing times subsequent to STS-51L. The processing data through mission STS-51L preparation are shown in Table 3. The orbiter processing times after STS-51L are shown in Table 4. The first post STS-51L vehicle flow number for each orbiter was reset to one in Table 4 because the launch preparation production progress function essentially was started over from the beginning.

Table 3 - Through STS-51L sum and average per flow of orbiter processing times at the OPF, VAB, and launch pad

Flow #	OV-099 Challenger	OV-102 Columbia	OV-103 Discovery	OV-104 Atlantis	OV Average
1	244	668	210	132	314
2	60	187	56	46	87
3	55	97	56		69
4	80	77	73		77
5	53	102	58		71
6	80	128	56		88
7	132	143			138
8	75				75
9	53				53
10	63				63

Table 4 - Post STS-51L sum and average per flow of orbiter processing times at the OPF, VAB, and launch pad

Flow #	OV-102 Columbia	OV-103 Discovery	OV-104 Atlantis	OV-105 Endeavour	OV Average
1	227	322	236	272	264
2	129	149	133		137
3	300	162	153		205
4	114	126	110		117
5	135	149	245		176
6		180	125		153
7		113	101		107
8		105	103		104
9			88		88

DATA ANALYSIS USING PROCESSING AVERAGES

Figure 3 is an arithmetic plot of the orbiter vehicle processing data through STS-51L contained in Table 3. The vehicle flow average processing times are shown by the dark solid line in the plot. Figure 4 is an arithmetic plot of the post STS-51L orbiter processing times contained in Table 4. The average processing times per flow are also shown by the dark solid line in this plot. The dark solid lines in Figures 3 and 4 indicate an exponential curve with some variation. The exponential production progress function is apparently valid for application to orbiter launch preparation processing.

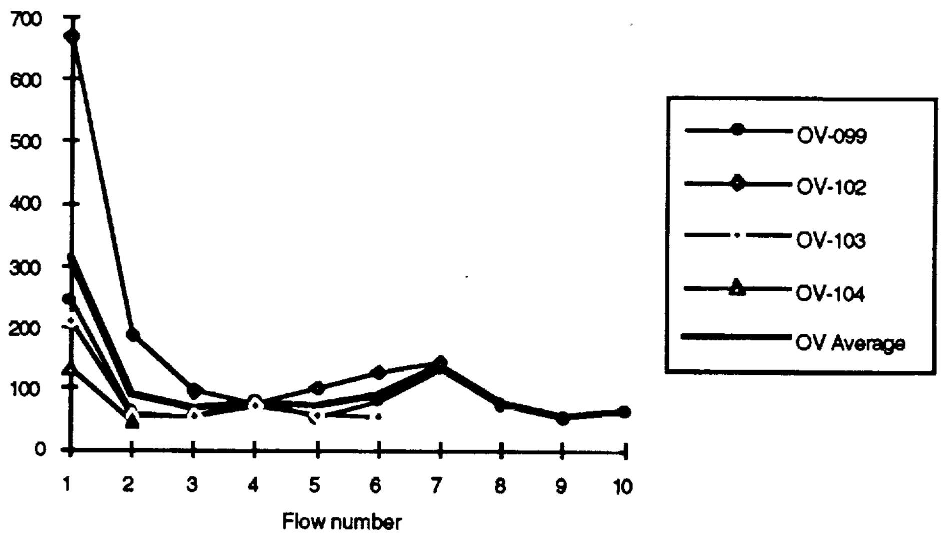

Figure 3 - Plot of individual and average orbiter processing times at the OPF, VAB, and launch pad through STS-51L (arithmetic)

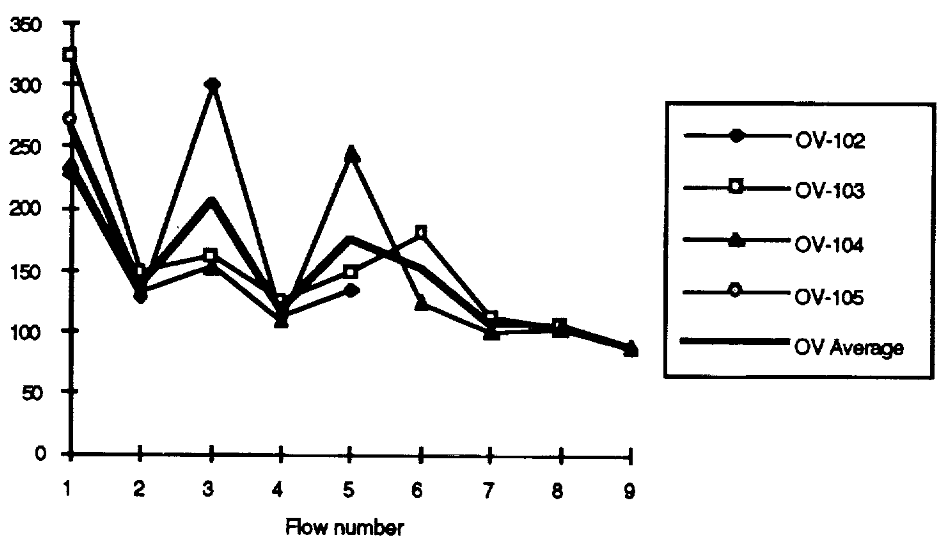

Figure 4 - Plot of individual and average orbiter processing times at the OPF, VAB, and launch pad post STS-51L (arithmetic)

As previously stated above in Section II, a log-log plot of the exponential production progress function will yield a straight line. It is desired to find a "best fit" straight line to a log-log plot of the average orbiter processing times in order to determine the learning efficiency. For a straight line fit by the method of least squares, the values of a (slope) and b (y-intercept) are obtained by solving the normal equations.

$$y = ax + b \tag{4}$$

$$a = (n\Sigma x_i y_i - (\Sigma x_i)\ (\Sigma y_i))\ /\ (n\Sigma x_i^2 - (\Sigma x_i)^2) \tag{5}$$

$$b = (\Sigma y_i\ /\ n) - (a\Sigma x_i\ /\ n) \tag{6}$$

For an exponential curve fit by the method of least squares, the values of log a and log b are obtained by fitting a straight line to the set of ordered pairs $\{(\log x_i, \log y_i)\}$. An exponential curve is described by the following equations.

$$y = bx^a \tag{7}$$

or

$$\log y = \log b + a \log x \tag{8}$$

The derivation of appropriate parameters and the calculation of the logarithmic "best fit" straight line to the average orbiter processing times through STS-51L are shown in Table 5. The first column contains the flow number. The second column shows the average orbiter processing times. The next five columns contain necessary manipulations and summations of the logarithms of the flow number and average processing time. The last two columns contain the predicted log of the processing times and the logarithmic "best fit" straight line for average processing times.

Table 5 - Through STS-51L average orbiter processing times at the OPF, VAB, and launch pad

Flow # (X)	OV Avg. (Y)	log X	log Y	(log X)(log Y)	(log X)^2	(log Y)^2	Predicted log Y	Best Fit Average
1	314	0.00	2.50	0.00	0.00	6.23	2.26	184
2	87	0.30	1.94	0.58	0.09	3.76	2.12	132
3	69	0.48	1.84	0.88	0.23	3.38	2.04	109
4	77	0.60	1.89	1.14	0.36	3.56	1.98	95
5	71	0.70	1.85	1.29	0.49	3.43	1.93	85
6	88	0.78	1.94	1.51	0.61	3.78	1.89	78
7	138	0.85	2.14	1.81	0.71	4.58	1.86	72
8	75	0.90	1.88	1.69	0.82	3.52	1.83	68
9	53	0.95	1.72	1.65	0.91	2.97	1.81	64
10	63	1.00	1.80	1.80	1.00	3.24	1.78	61
		6.56	19.50	12.35	5.22	38.45		

10«IF LEARNING CURVE ANALYSIS OF ORBITER PROCESSING= "" "LEARNING CURVE ANALYSIS OF ORBITER PROCESSING"» LEARNING CURVE ANALYSIS OF ORBITER PROCESSING«endif»

The slope and y-intercept are determined by appropriately solving equations 5 and 6.

$$a = ((10)(12.35) - (6.56)(19.50)) / ((10)(5.22) - 6.56^2) = -0.482$$

$$b = (19.50 / 10) - (-0.482)(6.56 / 10) = 2.26$$

The learning rate is then determined by solving equation 3.

$$L = 2^a = 2^{-0.482} = 0.716 = 72\%$$

A logarithmic plot of the average orbiter processing times and the "best fit" straight line prediction for the learning curve applied to orbiter processing through STS-51L is shown in Figure 5.

The sample correlation coefficient (r) of the "best fit" straight line is defined as follows.

$$r = a\, s_x / s_y \qquad (9)$$

where

$$s_x = ((\Sigma x_i^2 - (\Sigma x_i)^2 / \mathrm{n}) / (\mathrm{n} - 1))^{1/2} \qquad (10)$$

$$s_y = ((\Sigma y_i^2 - (\Sigma y_i)^2 / \mathrm{n}) / (\mathrm{n} - 1))^{1/2} \qquad (11)$$

Substituting the appropriate parameters into equations 9, 10, and 11 and solving for r we get the correlation coefficient.

$$s_x = ((5.22 - 6.56^2 / 10) / 9)^{1/2} = 0.319$$

$$s_y = ((38.45 - 19.50^2 / 10) / 9)^{1/2} = 0.217$$

$$r = (-0.482)(0.319 / 0.217) = -0.709$$

A "best fit" line which correlated perfectly to the data would have a correlation coefficient (r) equal to 1.0 for a positive slope or -1.0 for a negative slope.

Similar calculations were made for the post STS-51L orbiter processing times. The appropriate data is shown in Table 6 and the corresponding logarithmic plot of average orbiter processing times and the "best fit" straight line are shown in Figure 6.

Table 6 - Post STS-51L average orbiter processing times at the OPF, VAB, and launch pad

Flow # (X)	OV Avg. (Y)	log X	log Y	(log X)(log Y)	(log X)^2	(log Y)^2	Predicted log Y	Best Fit Average
1	264	0.00	2.42	0.00	0.00	5.86	2.39	247
2	137	0.30	2.14	0.64	0.09	4.57	2.28	188
3	205	0.48	2.31	1.10	0.23	5.34	2.21	161
4	117	0.60	2.07	1.25	0.36	4.28	2.16	144
5	176	0.70	2.25	1.57	0.49	5.04	2.12	132
6	153	0.78	2.18	1.70	0.61	4.77	2.09	122
7	107	0.85	2.03	1.72	0.71	4.12	2.06	115
8	104	0.90	2.02	1.82	0.82	4.07	2.04	109
9	88	0.95	1.94	1.86	0.91	3.78	2.02	104
		5.56	19.36	11.65	4.22	41.83		

The slope and y-intercept are determined by appropriately solving equations 5 and 6.

$$a = ((9)(11.65) - (5.56)(19.36)) / ((9)(4.22) - 5.56^2) = -0.395$$

$$b = (19.36 / 9) - (-0.395)(5.56 / 9) = 2.39$$

The post STS-51L learning rate is then determined by solving equation 3.

$$L = 2^a = 2^{-0.395} = 0.760 = 76\%$$

Substituting the appropriate parameters into equations 9, 10, and 11 and solving for r we get the correlation coefficient.

$$s_x = ((4.22 - 5.56^2 / 9) / 8)^{1/2} = 0.313$$

$$s_y = ((41.83 - 19.36^2 / 9) / 8)^{1/2} = 0.152$$

$$r = (-0.384)(0.313 / 0.152) = -0.791$$

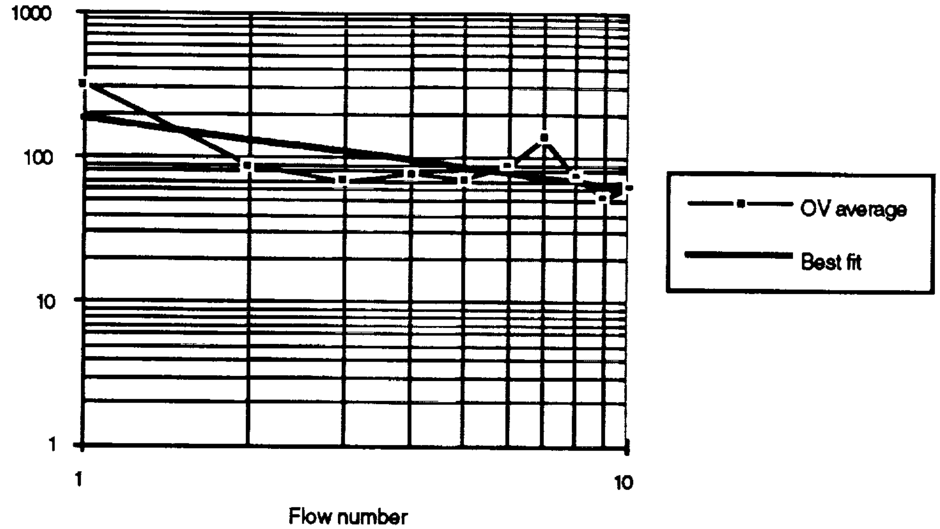

Figure 5 - Plot of average and best fit orbiter processing times at the OPF, VAB, and launch pad through STS-51L (logarithmic)

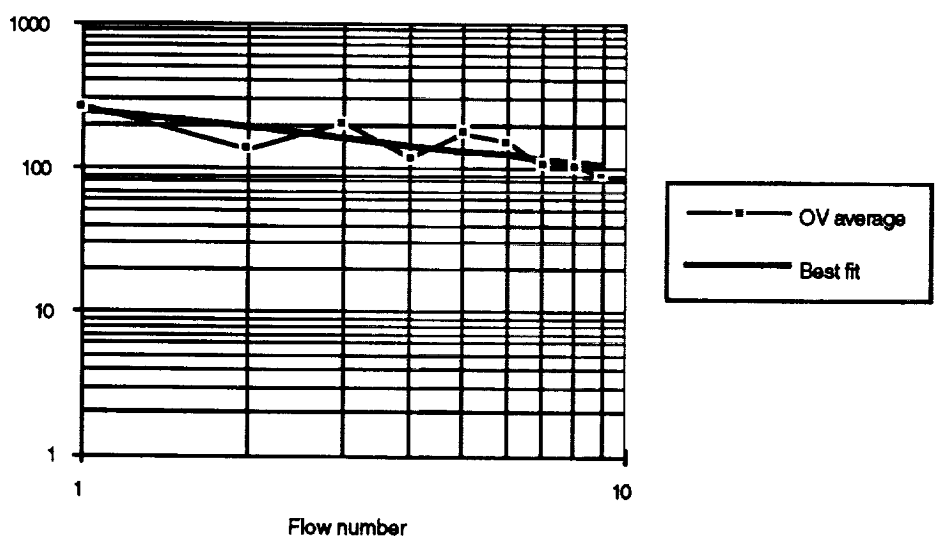

Figure 6 - Plot of average and best fit orbiter processing times at the OPF, VAB, and launch pad post STS-51L (logarithmic)

DATA ANALYSIS USING SEQUENTIAL PROCESSING CYCLES

Another method for determining the learning efficiency of processing orbiters through their main Kennedy Space Center facilities is by sequential processing cycle, rather than by the orbiters' cumulative average flow time as was done above. The data necessary for this modelling methodology is shown in Table 7. This data was obtained from the Mission Planning Office at the Kennedy Space Center [3]. The first column in this table shows the STS mission number. The next column contains the cycle number and the third column shows the actual orbiter processing time. The work days in Table 7 are the same processing times shown in Table 3. The next five columns contain necessary manipulations and summations of the logarithms of the flow number and average processing time. The last two columns contain the predicted log of the processing times and the logarithmic "best fit" straight line for predicted processing times. The data in Table 7 starts with the very first Space Shuttle processing flow and is inclusive of STS-51L (also known as STS-33), twenty five cycles later.

Table 7 - Through STS-51L orbiter processing times at the OPF, VAB, and launch pad by cycle number

STS #	Cycle # (X)	Work Days (Y)	log X	log Y	(log X)(log Y)	(log X)^2	(log Y)^2	Predicted log Y	Best Fit Work Days
1	1	668	0.00	2.82	0.00	0.00	7.98	2.43	267
2	2	187	0.30	2.27	0.68	0.09	5.16	2.29	195
3	3	97	0.48	1.99	0.95	0.23	3.95	2.21	162
4	4	77	0.60	1.89	1.14	0.36	3.56	2.15	142
5	5	102	0.70	2.01	1.40	0.49	4.03	2.11	128
6	6	244	0.78	2.39	1.86	0.61	5.70	2.07	118
7	7	60	0.85	1.78	1.50	0.71	3.16	2.04	110
8	8	55	0.90	1.74	1.57	0.82	3.03	2.01	103
9	9	128	0.95	2.11	2.01	0.91	4.44	1.99	98
11	10	80	1.00	1.90	1.90	1.00	3.62	1.97	93
13	11	53	1.04	1.72	1.80	1.08	2.97	1.95	89
14	12	210	1.08	2.32	2.51	1.16	5.39	1.93	86
17	13	80	1.11	1.90	2.12	1.24	3.62	1.92	83
19	14	56	1.15	1.75	2.00	1.31	3.06	1.90	80
20	15	56	1.18	1.75	2.06	1.38	3.06	1.89	78
23	16	73	1.20	1.86	2.24	1.45	3.47	1.88	75
24	17	132	1.23	2.12	2.61	1.51	4.50	1.87	73
25	18	58	1.26	1.76	2.21	1.58	3.11	1.85	71
26	19	75	1.28	1.88	2.40	1.64	3.52	1.84	70
27	20	56	1.30	1.75	2.27	1.69	3.06	1.83	68
28	21	132	1.32	2.12	2.80	1.75	4.50	1.82	67
30	22	53	1.34	1.72	2.31	1.80	2.97	1.81	65
31	23	46	1.36	1.66	2.26	1.85	2.76	1.81	64
32	24	143	1.38	2.16	2.97	1.90	4.65	1.80	63
33	25	63	1.40	1.80	2.52	1.95	3.24	1.79	61
			25.19	49.17	48.11	28.53	98.50		

The slope and y-intercept are determined by appropriately solving equations 5 and 6.

$$a = ((25)(48.11) - (25.19)(49.17)) / ((25)(28.53) - 25.19^2) = -0.456$$

$$b = (49.17 / 25) - (-0.456)(25.19 / 25) = 2.43$$

The learning rate through STS-51L by cycle order is then determined by solving equation 3.

$$L = 2^a = 2^{-0.456} = 0.729 = 73\%$$

Substituting the appropriate parameters into equations 9, 10, and 11 and solving for r we get the correlation coefficient.

$$s_x = ((28.53 - 25.19^2 / 25) / 24)^{1/2} = 0.362$$

$$s_y = ((98.50 - 49.17^2 / 25) / 24)^{1/2} = 0.272$$

$$r = (-0.456)(0.362 / 0.272) = -0.607$$

Table 8 contains sequential processing cycle data, similar to Table 7, for cycles occurring after STS-51L. Data in Table 8 was obtained from Kennedy Space Center's Mission Planning Office and is current through STS-50, which was launched on 25 June 1992 [3]. There have been twenty three orbiter processing cycles completed to date since STS-51L. The work days shown in Table 8, column three are cumulative times for the OPF, VAB, and launch pad. These work days are the same processing times shown in Table 4.

Table 8 - Post STS-51L orbiter processing times at the OPF, VAB, and launch pad by cycle number

STS #	Cycle # (X)	Work Days (Y)	log X	log Y	(log X)(log Y)	(log X)^2	(log Y)^2	Predicted log Y	Best Fit Work Days
26	1	322	0.00	2.51	0.00	0.00	6.29	2.40	251
27	2	236	0.30	2.37	0.71	0.09	5.63	2.33	215
29	3	149	0.48	2.17	1.04	0.23	4.72	2.29	197
30	4	133	0.60	2.12	1.28	0.36	4.51	2.27	185
28	5	227	0.70	2.36	1.65	0.49	5.56	2.24	176
34	6	153	0.78	2.18	1.70	0.61	4.77	2.23	169
33	7	162	0.85	2.21	1.87	0.71	4.88	2.21	163
32	8	129	0.90	2.11	1.91	0.82	4.45	2.20	158
36	9	110	0.95	2.04	1.95	0.91	4.17	2.19	154
31	10	126	1.00	2.10	2.10	1.00	4.41	2.18	151
41	11	149	1.04	2.17	2.26	1.08	4.72	2.17	148
38	12	245	1.08	2.39	2.58	1.16	5.71	2.16	145
35	13	300	1.11	2.48	2.76	1.24	6.14	2.15	142
37	14	125	1.15	2.10	2.40	1.31	4.40	2.15	140
39	15	180	1.18	2.26	2.65	1.38	5.09	2.14	138
40	16	114	1.20	2.06	2.48	1.45	4.23	2.13	136
43	17	101	1.23	2.00	2.47	1.51	4.02	2.13	134
48	18	113	1.26	2.05	2.58	1.58	4.22	2.12	132
44	19	103	1.28	2.01	2.57	1.64	4.05	2.12	131
42	20	105	1.30	2.02	2.63	1.69	4.09	2.11	129
45	21	88	1.32	1.94	2.57	1.75	3.78	2.11	128
49	22	272	1.34	2.43	3.27	1.80	5.93	2.10	127
50	23	136	1.36	2.13	2.90	1.85	4.54	2.10	125
			22.41	50.23	48.32	24.67	110.29		

Once again, the slope and y-intercept are determined by appropriately solving equations 5 and 6.

$$a = ((23)(48.32) - (22.41)(50.23)) \;/\; ((23)(24.67) - 22.41^2) = -0.222$$

$$b = (50.23 \;/\; 23) - (-0.222)(22.41 \;/\; 23) = 2.40$$

The learning rate post STS-51L by cycle order is then determined by solving equation 3.

$$L = 2^a = 2^{-0.222} = 0.857 = 86\%$$

Finally, appropriate parameters are substituted into equations 9, 10, and 11 and r is solved for the sample correlation coefficient.

$$s_x = ((24.67 - 22.41^2 / 23) / 22)^{1/2} = 0.359$$

$$s_y = ((110.29 - 50.23^2 / 23) / 22)^{1/2} = 0.164$$

$$r = (-0.222)(0.359 / 0.164) = -0.485$$

Figure 7 is a plot using Cartesian coordinates of cumulative orbiter processing times at the OPF, VAB, and launch pad for data through STS-51L. The solid line in Figure 7 is a "best fit" exponential curve with a learning efficiency of 73% and a sample correlation coefficient of -0.607. A curve that correlated perfectly to the sample data would have a correlation coefficient of 1 for a positive slope or -1 for a negative slope.

Figure 8 is a plot using Cartesian coordinates of cumulative orbiter processing times at the OPF, VAB, and launch pad for data post STS-51L. The solid line in Figure 8 is a "best fit" exponential curve with a learning efficiency of 86% and a sample correlation coefficient of -0.485.

Figure 9 is a log-log plot of Figure 7 and Figure 10 is a log-log plot of Figure 8. An exponential curve will plot as a straight line on log-log paper.

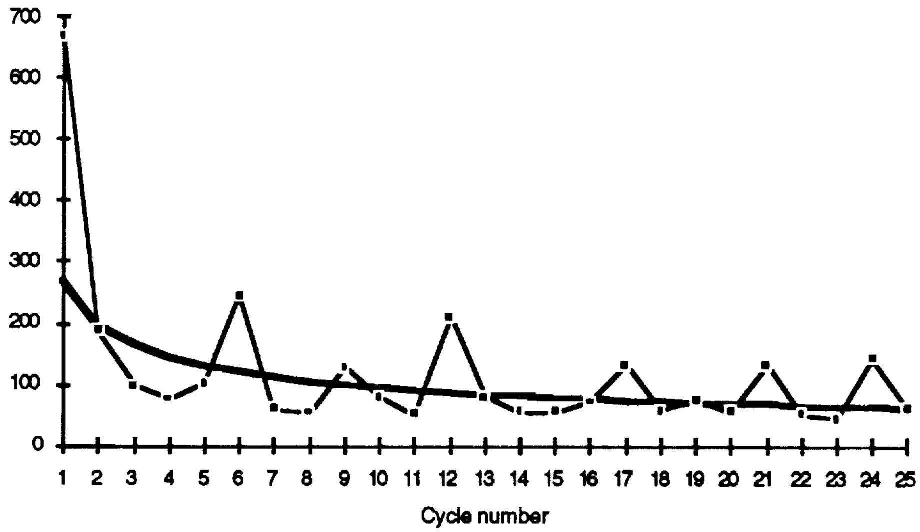

Figure 7 - Plot of orbiter and best fit processing times by cycle at the OPF, VAB, and launch pad through STS-51L (arithmetic)

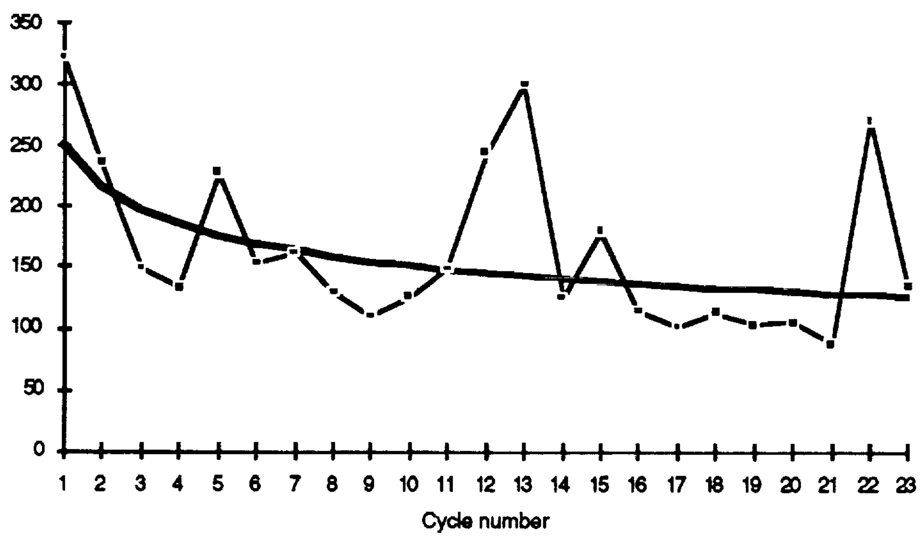

Figure 8 - Plot of orbiter and best fit processing times by cycle at the OPF, VAB, and launch pad post STS-51L (arithmetic)

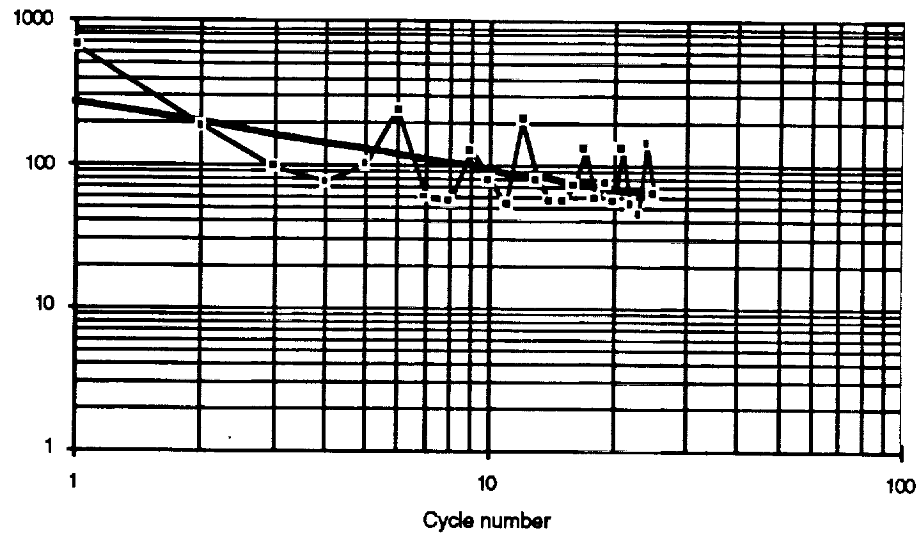

Figure 9 - Plot of orbiter and best fit processing times by cycle at the OPF, VAB, and launch pad through STS-51L (logarithmic)

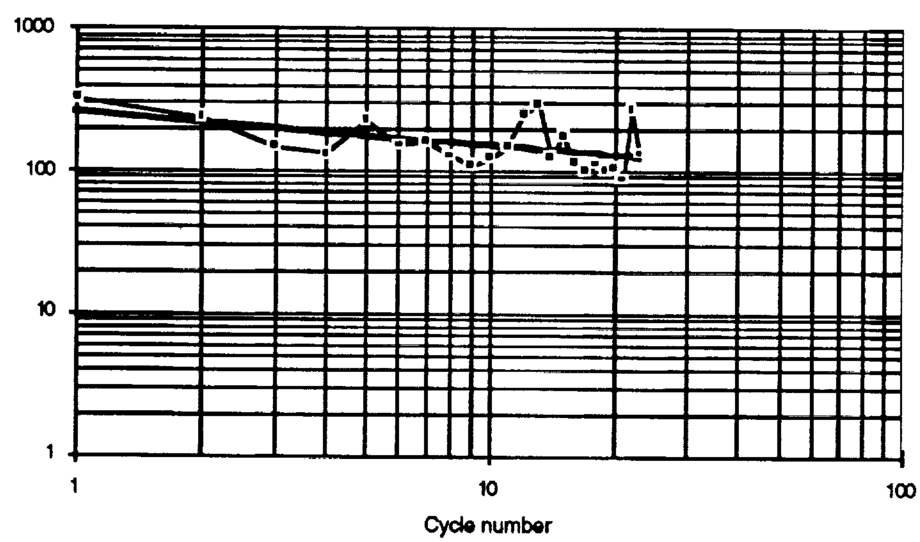

Figure 10 - Plot of orbiter and best fit processing times by cycle at the OPF, VAB, and launch pad post STS-51L (logarithmic)

Table 9 contains the same information as Table 8, but with four maverick data points removed in order to obtain better correlation to an exponential curve. Solid rationale is required when data points are eliminated from the analysis if the learning curve model is to be used to forecast future processing times. For this model, all post STS-51L Space Shuttle roll-backs from the launch pad, as well as the first flow of OV-105 Endeavour, were eliminated from the data set. This is sound rationale if future roll-backs are not expected and new orbiter vehicles are not planned to be introduced to the Kennedy Space Center.

Table 9 - Post STS-51L orbiter processing times at the OPF, VAB, and launch pad by cycle number without maverick data points

STS #	Cycle # (X)	Work Days (Y)	log X	log Y	(log X)(log Y)	(log X)^2	(log Y)^2	Predicted log Y	Best Fit Work Days
26	1	322	0.00	2.51	0.00	0.00	6.29	2.45	284
27	2	236	0.30	2.37	0.71	0.09	5.63	2.35	223
29	3	149	0.48	2.17	1.04	0.23	4.72	2.29	194
30	4	133	0.60	2.12	1.28	0.36	4.51	2.24	175
28	5	227	0.70	2.36	1.65	0.49	5.55	2.21	162
34	6	153	0.78	2.18	1.70	0.61	4.77	2.18	152
33	7	162	0.85	2.21	1.87	0.71	4.88	2.16	144
32	8	129	0.90	2.11	1.91	0.82	4.45	2.14	138
36	9	110	0.95	2.04	1.95	0.91	4.17	2.12	132
31	10	126	1.00	2.10	2.10	1.00	4.41	2.11	128
41	11	149	1.04	2.17	2.26	1.08	4.72	2.09	123
37	12	125	1.08	2.10	2.26	1.16	4.40	2.08	120
40	13	114	1.11	2.06	2.29	1.24	4.23	2.07	116
43	14	101	1.15	2.00	2.30	1.31	4.02	2.05	113
48	15	113	1.18	2.05	2.41	1.38	4.22	2.04	111
44	16	103	1.20	2.01	2.42	1.45	4.05	2.03	108
42	17	105	1.23	2.02	2.49	1.51	4.09	2.03	106
45	18	88	1.26	1.94	2.44	1.58	3.78	2.02	104
50	19	135	1.28	2.13	2.72	1.64	4.54	2.01	102
			17.09	40.67	35.80	17.58	87.43		

The slope and y-intercept are determined by appropriately solving equations 5 and 6.

$$a = ((19)(35.80) - (17.09)(40.67)) / ((19)(17.58) - 17.09^2) = -0.348$$

$$b = (40.67 / 19) - (-0.348)(17.09 / 19) = 2.45$$

The learning rate post STS-51L by cycle order is then determined by solving equation 3.

$$L = 2^a = 2^{-0.348} = 0.786 = 79\%$$

Substituting the appropriate parameters into equations 9, 10, and 11 and solving for r we get the correlation coefficient.

$$s_x = ((17.58 - 17.09^2 / 19) / 18)^{1/2} = 0.351$$

$$s_y = ((87.43 - 40.67^2 / 19) / 18)^{1/2} = 0.142$$

$$r = (-0.348)(0.351 / 0.142) = -0.863$$

Figure 11 is a plot using Cartesian coordinates of cumulative orbiter processing times at the OPF, VAB, and launch pad for data post STS-51L with the four maverick data points removed. The solid line in Figure 11 is a "best fit" exponential curve with a learning efficiency of 79% and a sample correlation coefficient of -0.863. This is significantly better than the sample correlation coefficient of -0.485 obtained from Table 8 and plotted in Figure 8. Figure 12 is a log-log plot of Figure 11.

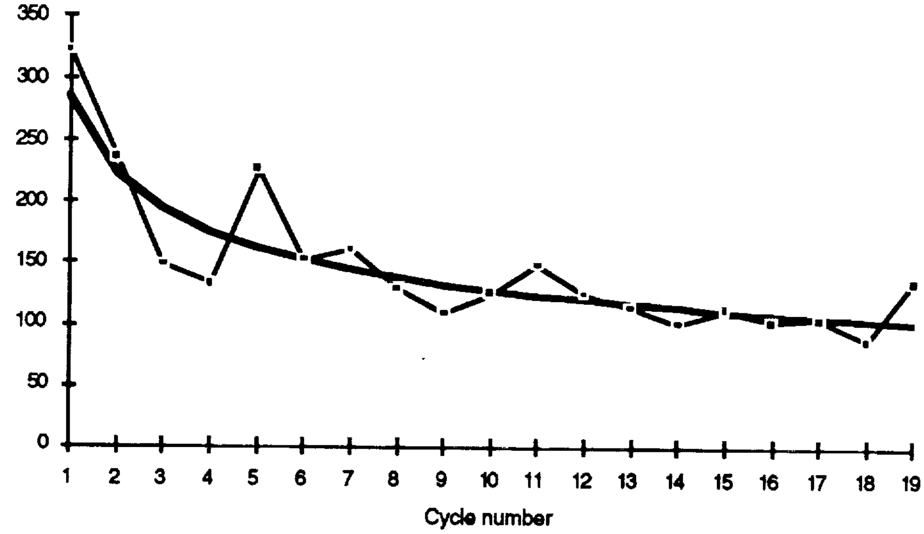

Figure 11 - Plot of orbiter and best fit processing times by cycle at the OPF, VAB, and launch pad post STS-51L without maverick data points (arithmetic)

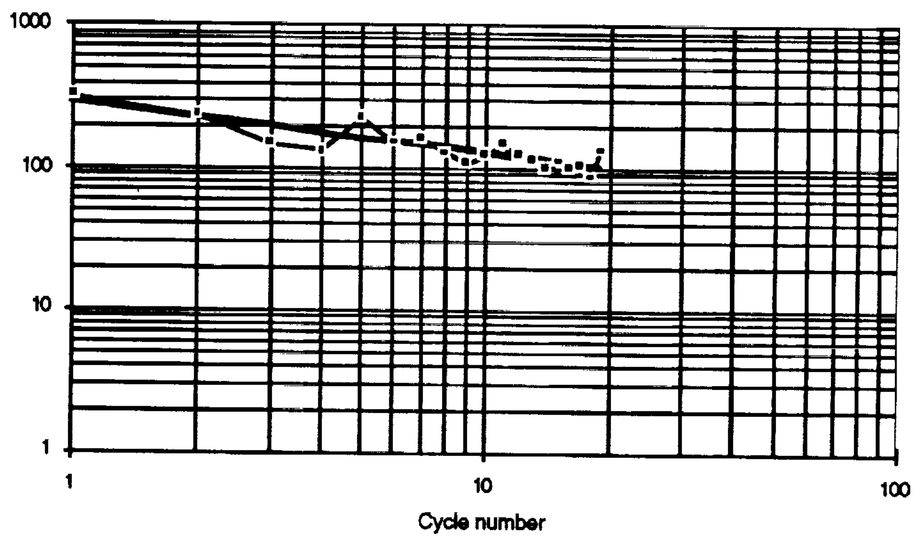

Figure 12 - Plot of orbiter and best fit processing times by cycle at the OPF, VAB, and launch pad post STS-51L without maverick data points (logarithmic)

CONCLUSIONS

The learning model using average cumulative orbiter processing times for the OPF, VAB, and launch pad is the methodology with the greatest sample correlation coefficient using all the relevant data (r = -0.791 from Table 6). At the current learning rate for average orbiter processing, there will be capability for significant launch rate increase in the future as processing experience is gained. Table 10 shows the expected cumulative orbiter processing times at the OPF, VAB, and launch pad after the orbiter vehicle has experienced the number of flows shown in the first column. Table 10 is based on the calculated post STS-51L learning rate curve of 76%. The second column is the log of the flow number. The third column is calculated using equation 4 with a = -0.395, b = 2.39, and substituting log X for x and log Y for y. The final column in Table 10 is calculated by taking the antilogarithm of y.

Table 10 - Predicted cumulative orbiter processing times by average at the OPF, VAB, and launch pad for future launch preparation flows

Flow # (Post 51L) (X)	log X	log Y	Predicted Processing Time (Work Days)
10	1.00	2.00	99
20	1.30	1.88	75
30	1.48	1.81	65
40	1.60	1.76	57
60	1.78	1.69	49

On average, the sum of orbiter processing times at the OPF, VAB, and launch pad is expected to be reduced about 150 days from the first flow to the tenth flow of a vehicle. By the fortieth flow of an orbiter, processing time is expected to be better than achieved at the time of STS-51L (which was 61 work days predicted for the tenth flow and 63 work days actually observed from Table 5). Cumulative orbiter processing time can be expected to be reduced another 50 days per flow on the sixtieth flow of the vehicle at the current learning efficiency. This reduced cycle time will allow additional processing capacity that may lead to an increasing Space Shuttle launch rate.

Observation of Table 10 shows that turn-around ground processing times of less than 50 days at the Kennedy Space Center's main Space Shuttle orbiter facilities can be achieved well within the design life of an orbiter (which is 100 missions). At that point in time, orbiter turn-around processing times would surpass, on average, the best turn-around times achieved before STS-51L. Several other launch vehicle studies have analyzed new types of personnel vehicles that could achieve this order of ground turn-around time. However, associated with any new vehicle program is development risk. Continued, uninterrupted launching of Space Shuttles at current learning efficiency can reach desired launch rates of some new launch vehicles under study without incurring development risk. This analysis is independent of orbiter and new vehicle development cost examination.

The learning model using sequential processing cycle of cumulative orbiter processing times for the OPF, VAB, and launch pad, excluding maverick data points (which were defined in Section VI), is the methodology with the greatest sample correlation coefficient (r = -0.863). However, great care and solid rationale must be used when data points are eliminated from the analysis. For this model, all post STS-51L Space Shuttle roll-backs from the launch pad, as well as the first flow of OV-105 Endeavour, were eliminated from the data set. This is sound rationale if future roll-backs are not expected and new orbiter vehicles are not planned to be introduced to the Kennedy Space Center.

Table 11 shows the expected cumulative orbiter processing times for the total number of cycles shown in the first column. Table 11 is based on the calculated post STS-51L learning rate curve of 79%. The second column is the log of the cycle number. The third column is calculated using equation 4 with a = -0.348, b = 2.45, and substituting log X for x and log Y for y. The final column in Table 11 is calculated by taking the antilogarithm of y.

Table 11 - Predicted cumulative orbiter processing times by cycle at the OPF, VAB, and launch pad without maverick data points for future launch preparation flows

Cycle # (Post 51L) (X)	log X	log Y	Predicted Processing Time (Work Days)
80	1.90	1.79	62
90	1.95	1.77	59
100	2.00	1.76	58
120	2.08	1.73	54
140	2.15	1.70	50

Observation of Table 11 indicates that cumulative orbiter processing time at the OPF, VAB, and launch pad is expected to be about the same as the processing time of STS-51L by the time of the eightieth total orbiter processing cycle (61 work days predicted for the twenty-fifth cycle and 63 work days actual from Table 7). Cumulative orbiter processing time can be expected to reach 50 days at the Kennedy Space Center's main Space Shuttle orbiter facilities by time of launch number 140 since STS-51L.

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