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IN-15

A 20k Payload Launch Vehicle Fast Track Development Concept Using an RD–180 Engine and a Centaur Upper Stage

Compiled by
R. Toelle

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A 20K PAYLOAD LAUNCH VEHICLE FAST TRACK DEVELOPMENT CONCEPT USING AN RD–180 ENGINE AND A CENTAUR UPPER STAGE

I. INTRODUCTION

A concept definition study to define a booster capable of launching a 20k-lb payload from the Eastern Test Range into a 100-nmi orbit has been performed. Marshall Space Flight Center (MSFC) personnel from the Science and Engineering and Program Development Directorates were organized into a high performance work team (the appendix lists the membership). This report captures the results of this 3-week effort.

A. Objectives

The objectives of this exercise were to:

Develop a fast-track booster design, manufacturing, and verification program concept based on using a Commonwealth of Independent States (CIS) RD–180 liquid oxygen/rocket propellant (LO_2/RP) booster engine, an existing upper stage, an existing payload fairing, and be flight ready in 36 months.

Perform trade studies to select booster size, materials and manufacturing methods, and an upper stage and payload fairing from existing candidates that will meet the development schedule.

Develop manpower, facilities, tooling and manufacturing, verification, and launch operations requirements including costs and top level schedules.

B. Ground Rules/Assumptions

The following ground rules and assumptions were developed by the team members and used for this exercise.

1. Develop a launch vehicle concept to compete with the Ariane launch vehicle (performance and cost)

2. The goal is to design, build, test at MSFC, and deliver to be determined (TBD) flight articles to the launch site

3. The RD–180 engine is the candidate booster engine, with the RD–170 as backup

4a. The first option is to purchase an existing upper stage, i.e., Centaur/Titan second stage/Delta upper stage with flight avionics included

4b. The second option is to develop a new upper stage using the following engines (time did not permit analyzing this concept)

5. Uprated RL–10 series are candidates as upper stage engines (RL–10C)

6. The CIS D–57 LO_2/LH_2 engine is an upper stage candidate

7. The payload requirement is ≥20k lb launched due east to low-Earth orbit (LEO), and greater than 8k lb in geosynchronous transfer orbit (GTO)

7a. LEO = 100 nmi

7b. Total weight (T/W) at lift-off ≥1.20

8. Payload design margin = 4k lb, no stage weight design margin at this time

9a. The baseline is to purchase payload fairing

9b. The second option is to develop a user-friendly/cost-effective payload fairing

10. Design the structural margin so that vehicle is not performance limited to preclude potential flight restrictions, i.e.:

- Set structural design safety factor = 2 to eliminate testing
- Trade cost of lowering to 1.4
- Show delta cost/processes/tolerances, etc. due to extra weight resulting from high safety factor

11. Define process of "0-Base" specifications and build to only what is necessary

12. Set up very tight controls of design reviews (DR's) and limit to what is necessary

13. Streamline interface documents and requirements

14. Develop manufacturing process for inexpensive production, i.e., new technology of spin forming, hydroshock forming, extruding and forging, and statistical process controlled (SPC) welding

15. Set up verification criteria at assembly plant that delivers ready-to-fly hardware to launch site

16. Develop booster avionics using as much commercial-grade components and procedures as possible to meet required reliability

17. Booster avionics has automatic self testing

18. Standardize mission profiles

19. Develop flight software to accommodate a spectrum of payload weights and center-of-gravity locations to reduce preflight analyses

20. The payload is required to be processed off-line and delivered to launch vehicle ready to fly, except for structural and electrical attachment to vehicle

21. Design for minimum time on launch pad, i.e., goal of 1 day after hard down

22. Automate generation of induced environment data to payload, i.e., loads, acoustics, temperature, shocks, etc.

23. Automate postflight analysis to maximize computer looking for anomalies

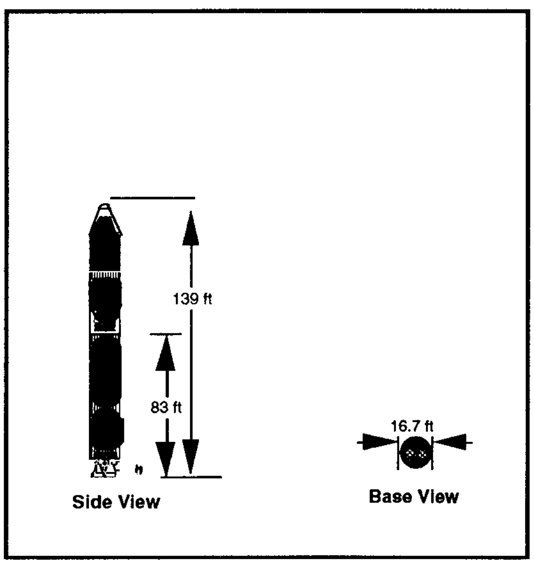
24. Stop engineering development after five flights.

25. Investigate growth scenario to 65k to LEO; 65k includes upper stage to move payloads to other orbits.

C. Summary

Figure 1 displays a summary of the final configuration. It consists of a booster constructed of 2219 aluminum-welded tank dome gores and barrel panels and one two-nozzle RD–180 engine. A structural design safety factor of 2 was used to reduce the structural test program and to meet the schedule. A 5-percent thrust increase of the RD–180 engine is required to meet the payload with margin. The manufacturer's representative said that this is attainable. The upper stage is a Titan IV Centaur with RL–10 A4 engines, which are required to meet the payload margin. The payload fairing is a McDonnell Douglas Titan IV, modified to meet attachment requirements. The estimated development schedule is 40 months versus the goal of 36 months. The vehicle design, development, test, and evaluate (DDT&E) estimates are between $480 and $550M in 1993 dollars, excluding the launch complex modifications. Recurring costs are estimated at $78 to $85M per flight.

20K LV CONCEPT



Payload:	20.7 k lb
Final Position:	100 x 100 nmi @ 28.5°
GLOW:	731 klb
Payload Fairing	
Jettisoned Mass	9,200 lb
Upper Stage:	
Inert Mass:	8.0 klb
Propellant Mass:	44.6 klb
Propellant Type:	LOX/LH2
Engine Type/# Ea.:	RL10A-4/2
Vac Thrust (Ea):	20.8 klb
Vac ISP:	448.9 s
Stage Diameter:	14.7 ft
Stage Length:	29.5 ft
First Stage Booster	
Inert Mass:	55.7klb
Propellant Mass:	588 klb
Propellant Type:	LOX/RP
Engine Type/#:	RD180/1
Vac /SL Thrust (Ea):	945/ 872 klb
Vac /SL ISP:	337/309 s
Stage Length:	83 ft
Stage Diameter:	16.7 ft
Structure Material:	Al 2219

Notes:
- 90° Launch Azimuth
- MECO @ 100 nmi. circ.
- T/W @ Liftoff = 1.20
- Max G = 4.50 / Max q = 600 psf

Figure 1. Summary of the final configuration.

II. VEHICLE DESCRIPTION

A. Vehicle Layout

The resulting configuration is shown in figure 2. The vehicle has a Titan IV-size payload shroud (200-in diameter) enclosing a Centaur (168-in diameter) upper stage and payload. The booster is jettisoned before the shroud, therefore, the linear-shaped charge-stage separation system is located just below the booster/shroud interface (fig. 3). The shroud is attached to the upper stage through a boattail that is loaded in tension during the boost phase. Ullage motors to separate the stages and settle the Centaur propellants are mounted on the boattail. Payload contamination during booster separation is not an issue since the payload will still be shrouded.

The booster dry bays (the interstage or forward skirt, the intertank, and the aft skirt) are made of identical elements assembled to provide the required lengths. Interfaces are rings riveted onto the cylinder with an external bolt flange (fig. 4). Certain dry bay panels will need reworking to allow penetrations and access doors. All booster avionics (except the rate gyro, range safety, and antennae) are mounted in the aft skirt.

The booster LO_2 tank is located forward and has a continual cross section along its length. Due to material blank size availability, there will have to be a circumferential weld in the middle, but the two halves will be the same. The tank domes are sized so that the weld strength is sufficient to carry the load. This means that there is no need for machining weld lands onto the gores. Three of the domes, the forward LO_2 dome and both RP domes, are of identical thickness. Only the aft LO_2 tank dome is a different thickness.

The RD–180 engine system (fig. 5 and section II.J) is supported by a series of struts onto a circular pattern. The struts interface with a small conical thrust structure which carries the loads into the aft skirt (fig. 6). The engine is surrounded by a close-fitting shell to protect it from aerodynamic loads. This aerodynamic shell would also be used to support the engine supplied base heat shield.

The vehicle to pad holddown is at the base of the aft skirt, above the engine. It consists of a circumferential linear-shaped charge. This design reduces the point loads between the launch pad and vehicle, resulting in a more efficient structure.

B. Aerodynamics

Preliminary aerodynamics were determined for the 20k launch vehicle configuration as shown in figure 7. These initial estimates were based on wind tunnel test data of the Titan IV payload fairing and the core stage of an in-line shuttle-derived launch vehicle with similar dimensions. The data were used to determine payload performance, structural loads, and control system requirements for the vehicle.

Early study analysis indicated inadequate control authority at maximum dynamic pressure, primarily due to the mass properties of the LO_2/RP tankage. As a possible solution, the stabilizer fin configuration shown in figure 8 was proposed to correct the problem aerodynamically. The planform and airfoil section were derived from the Saturn V lunar orbit rendezvous (LOR) fin configuration, which was modified to accommodate the cylindrical body and sized to meet the control requirements.

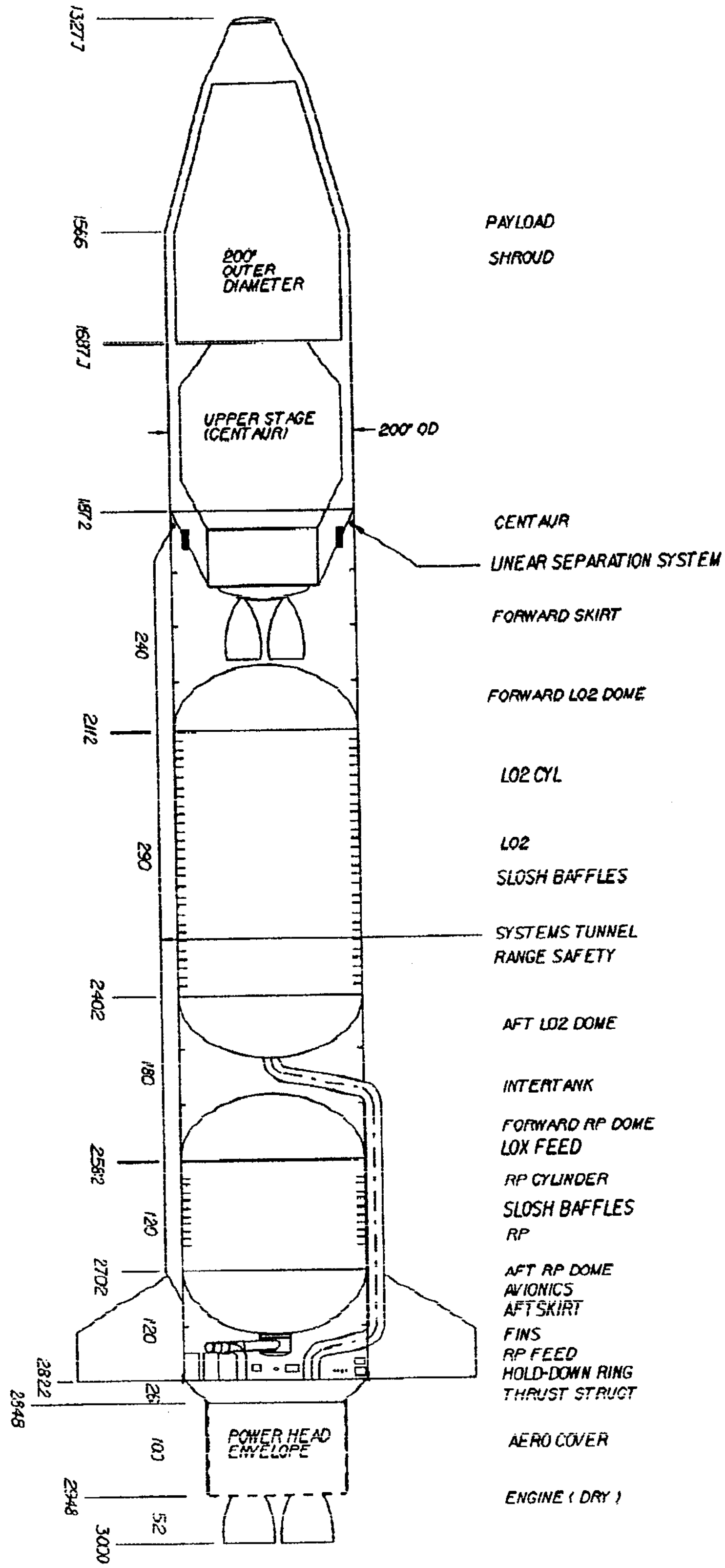


Figure 2. Vehicle configuration.

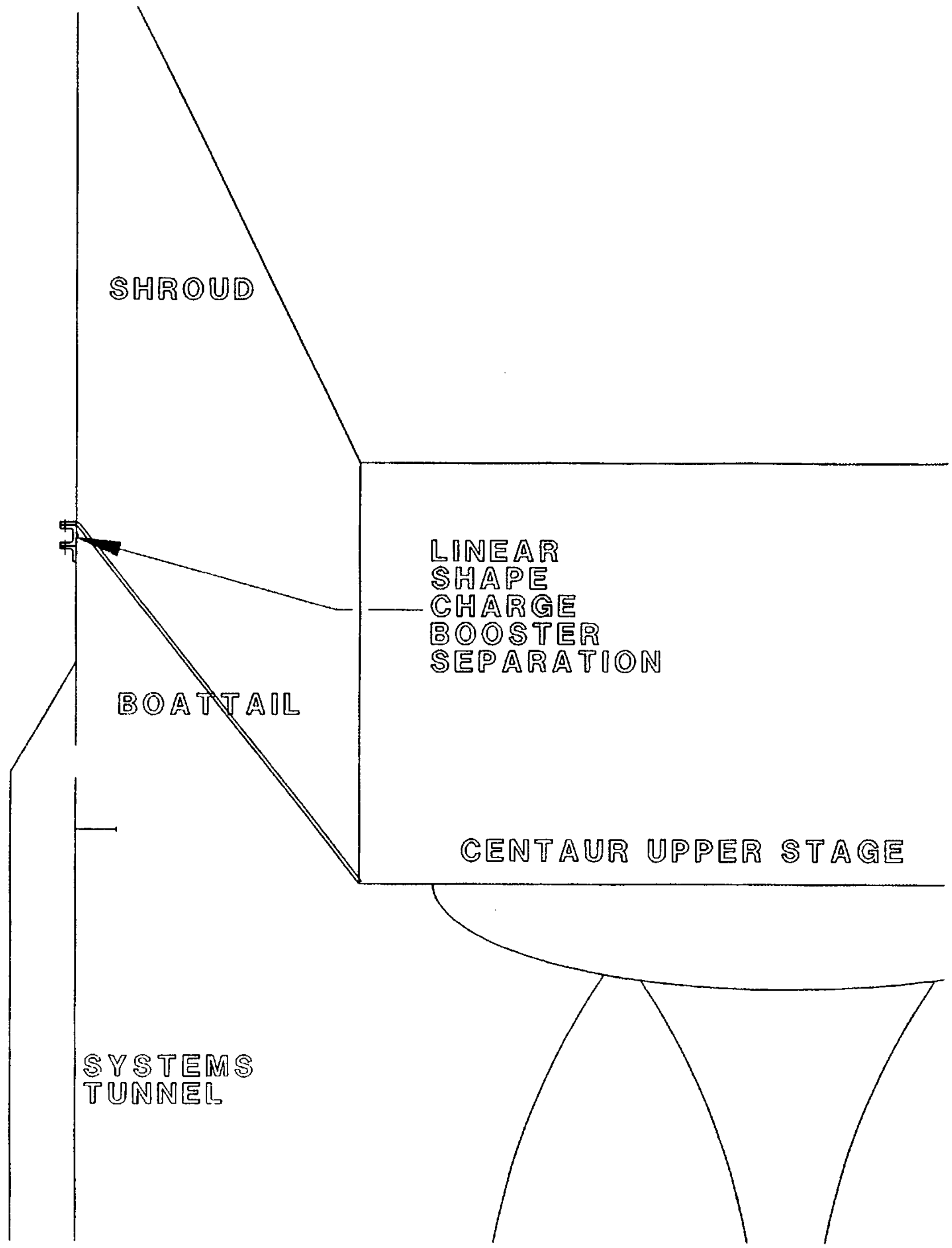


Figure 3. Booster separation location.

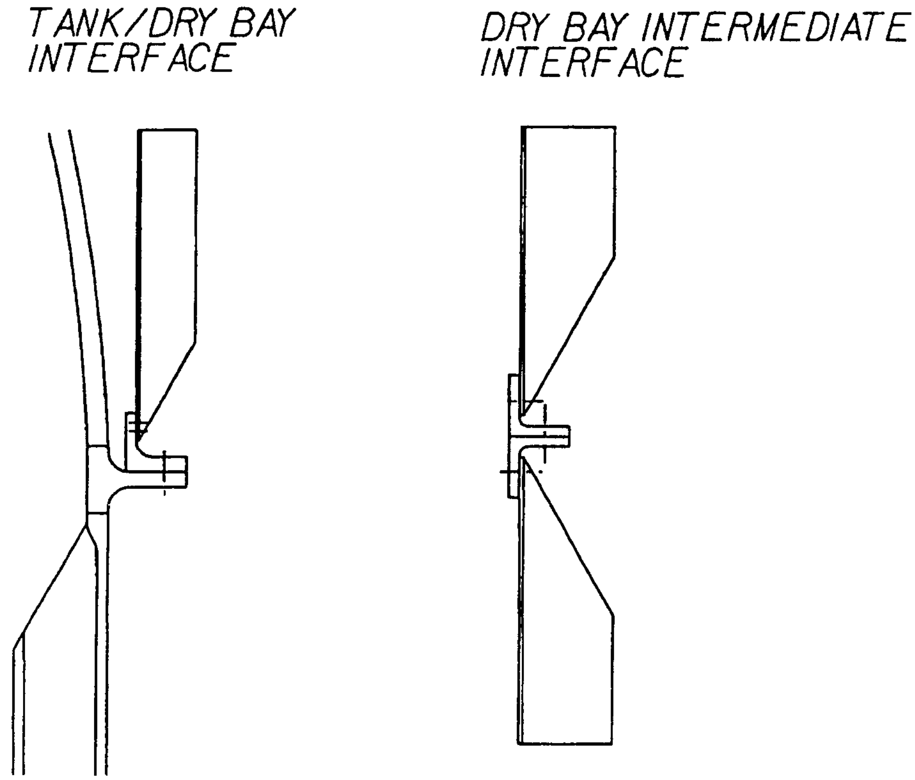


Figure 4. Interfaces.

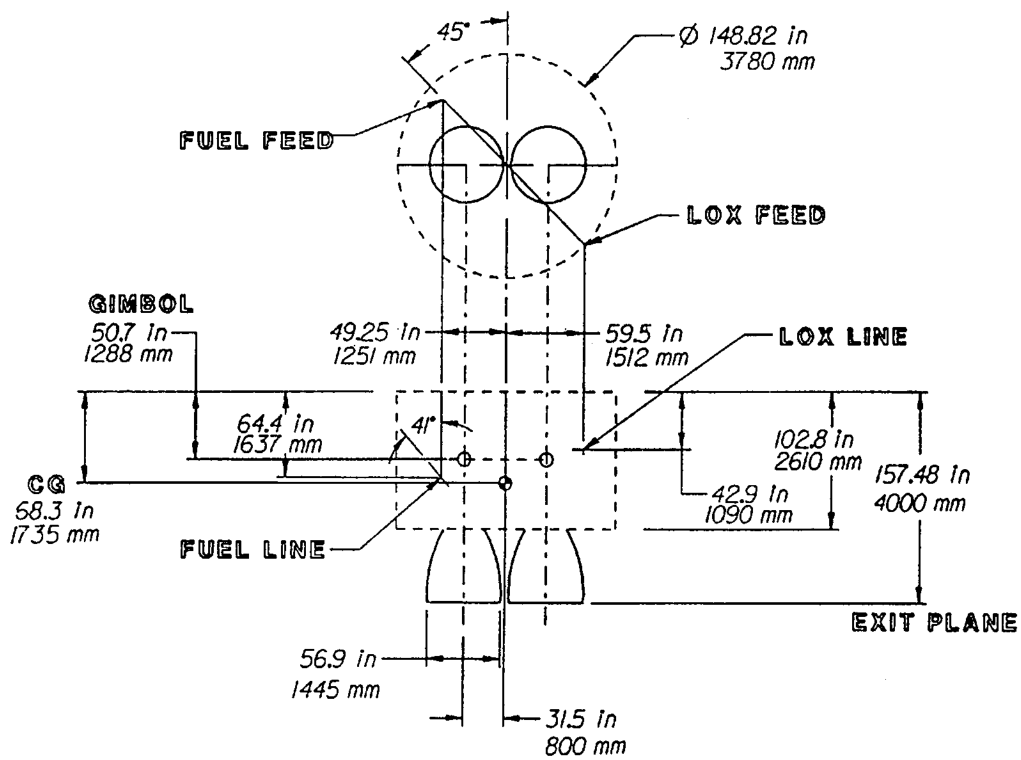


Figure 5. RD–180 engine overall dimensions.

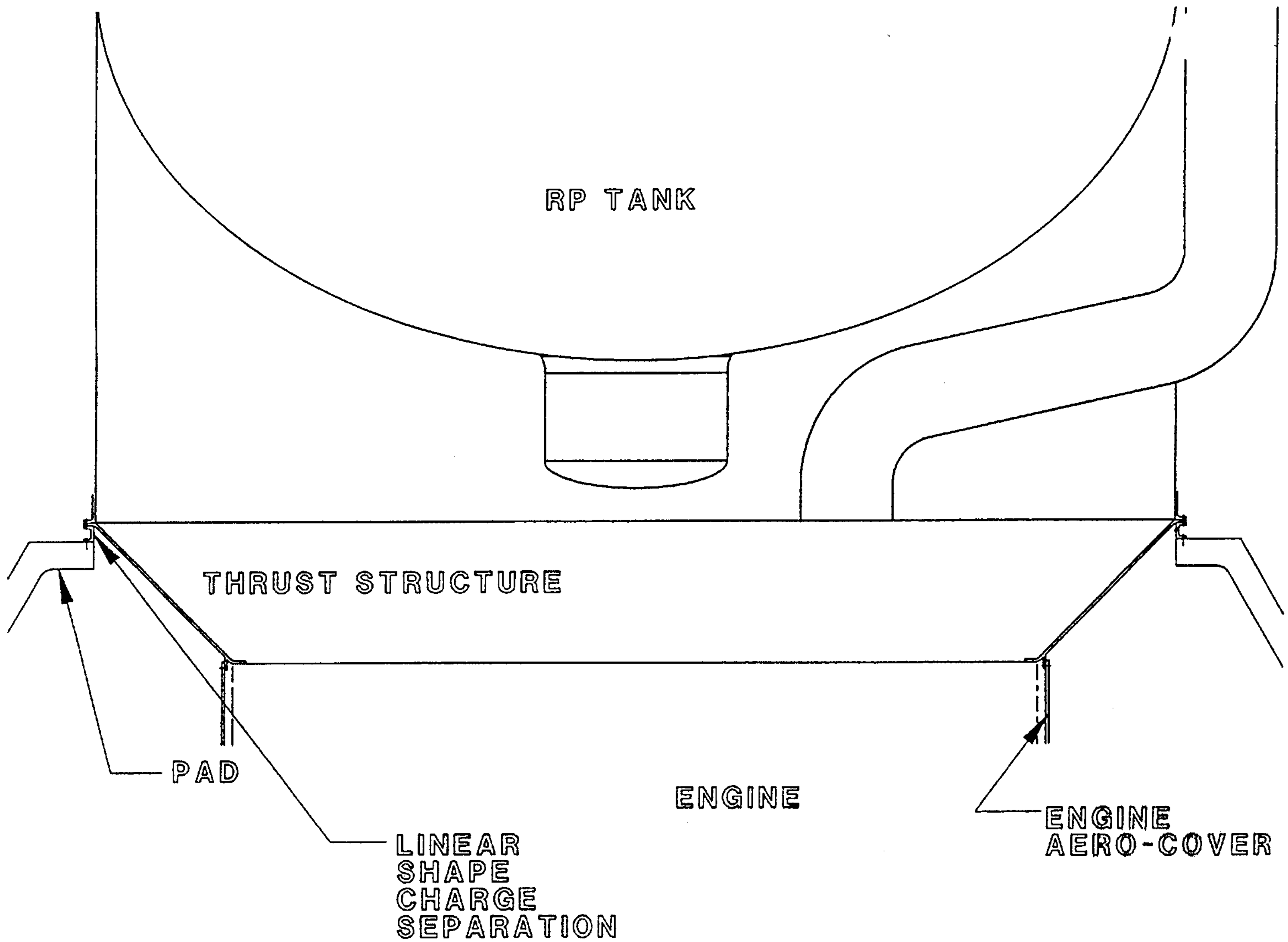


Figure 6. Aft skirt.

The data provided to each discipline were referenced to the cylindrical body diameter (Dref), cross-sectional area (Sref), and moment reference point (MRP) as indicated in figure 7. A comparison of vehicle forebody axial force coefficient between the Titan IV nose cone and hypothetical configurations is shown in figure 9. The data for the Titan IV configuration and the base drag estimates in figure 10 were used for trajectory analyses. Figures 11 and 12 show the longitudinal aerodynamic characteristics for the vehicle with and without the fins. The resulting center of pressure location versus Mach number comparison is given in figure 13. The pressure distribution along the vehicle at zero angle-of-attack for Mach 2.0 is provided in figure 14. The normal force load distribution is in figure 15.

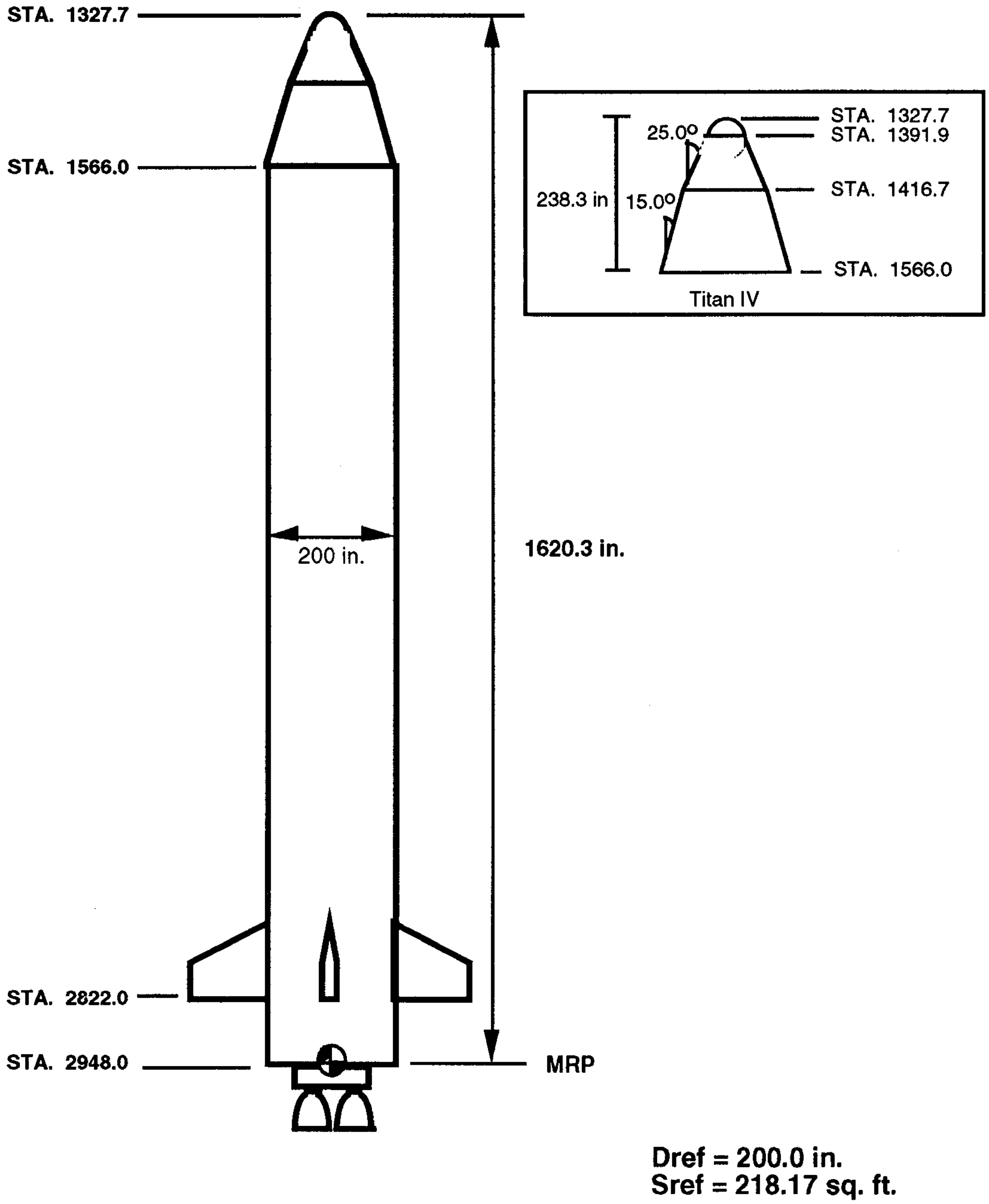
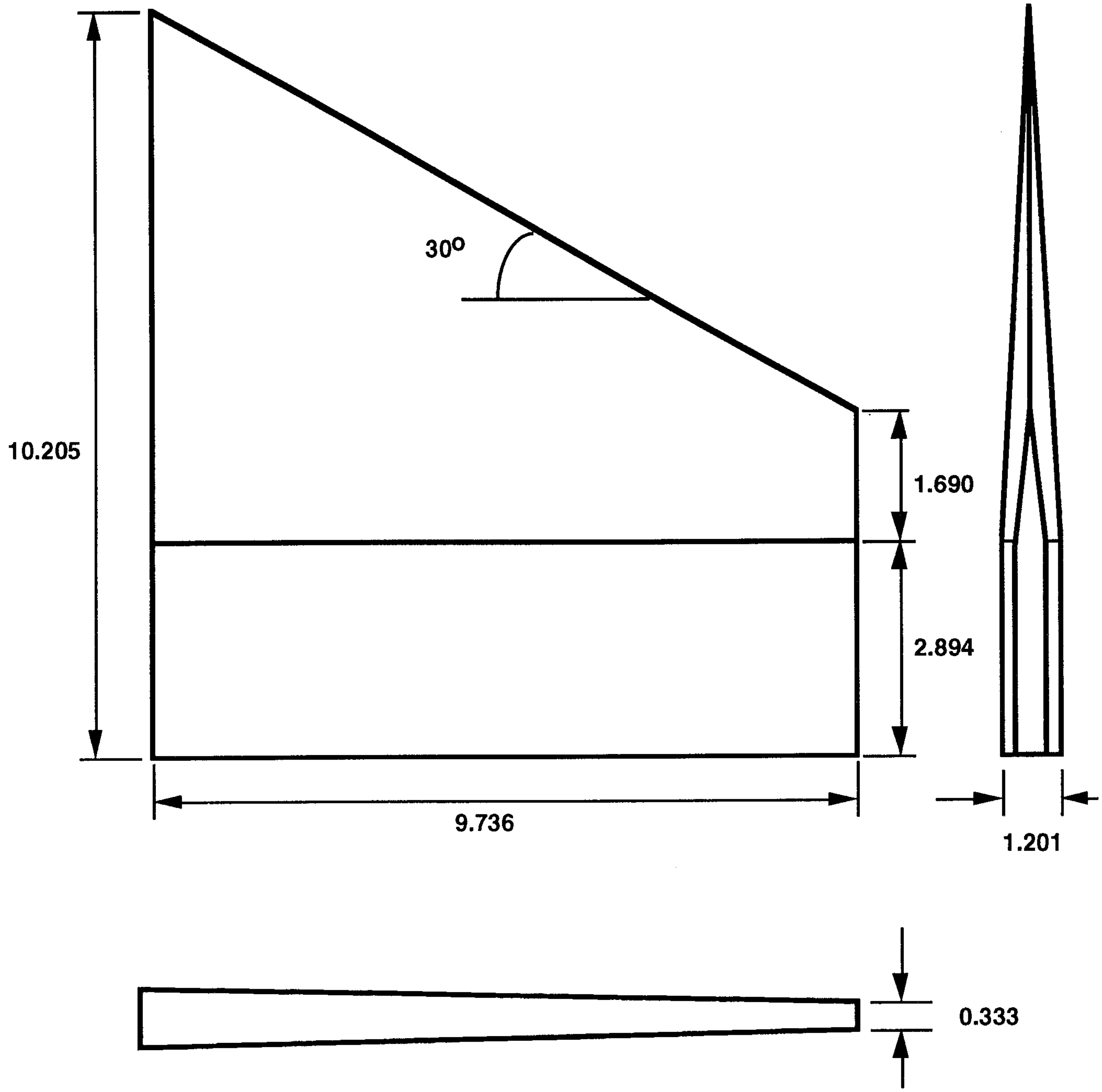


Figure 7. 20k launch vehicle configuration with fins.



All dimensions in ft. FS

Figure 8. Proposed 20k launch vehicle stabilizer fin.

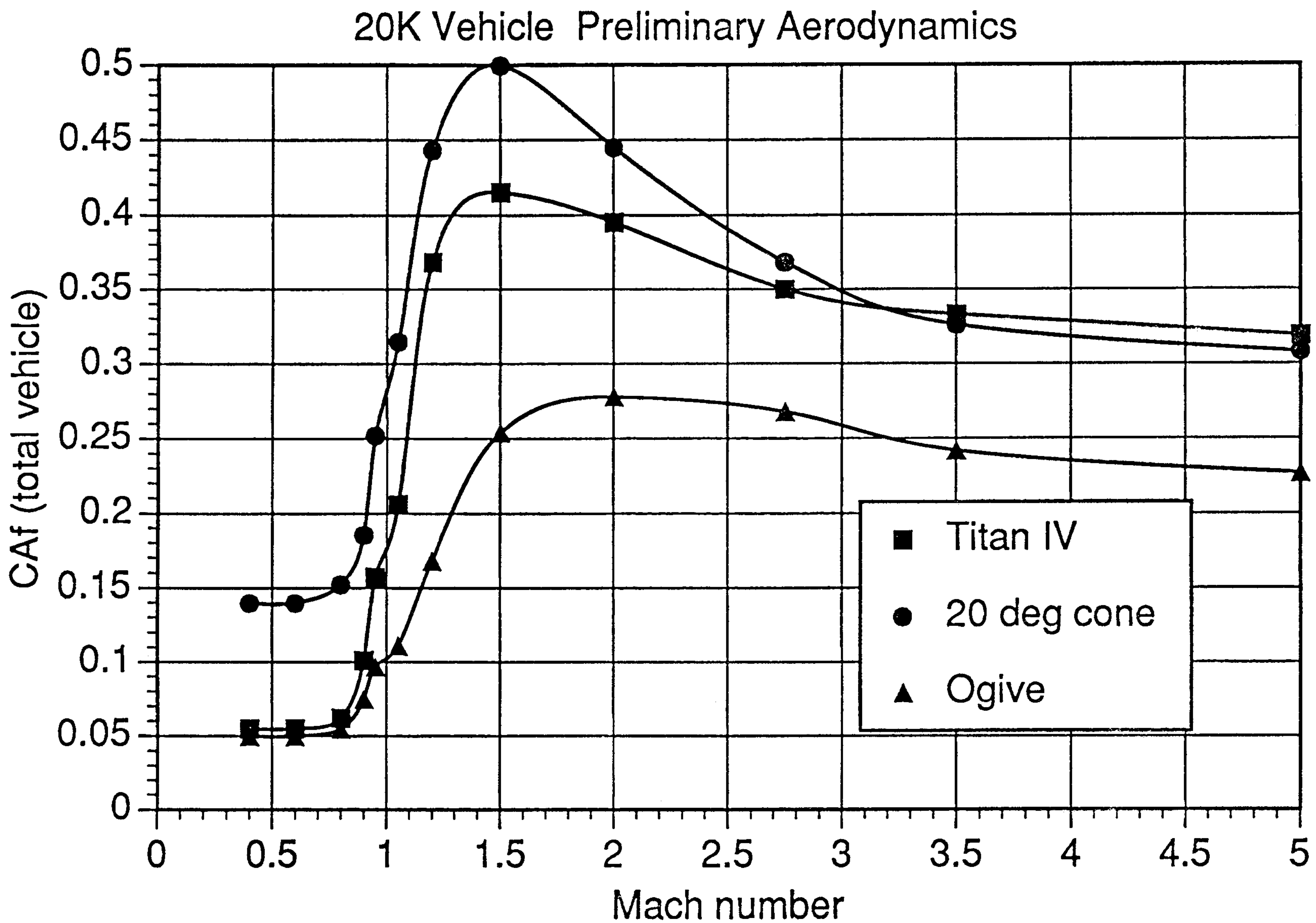


Figure 9. Preliminary forebody axial force coefficient (nose configuration comparison).

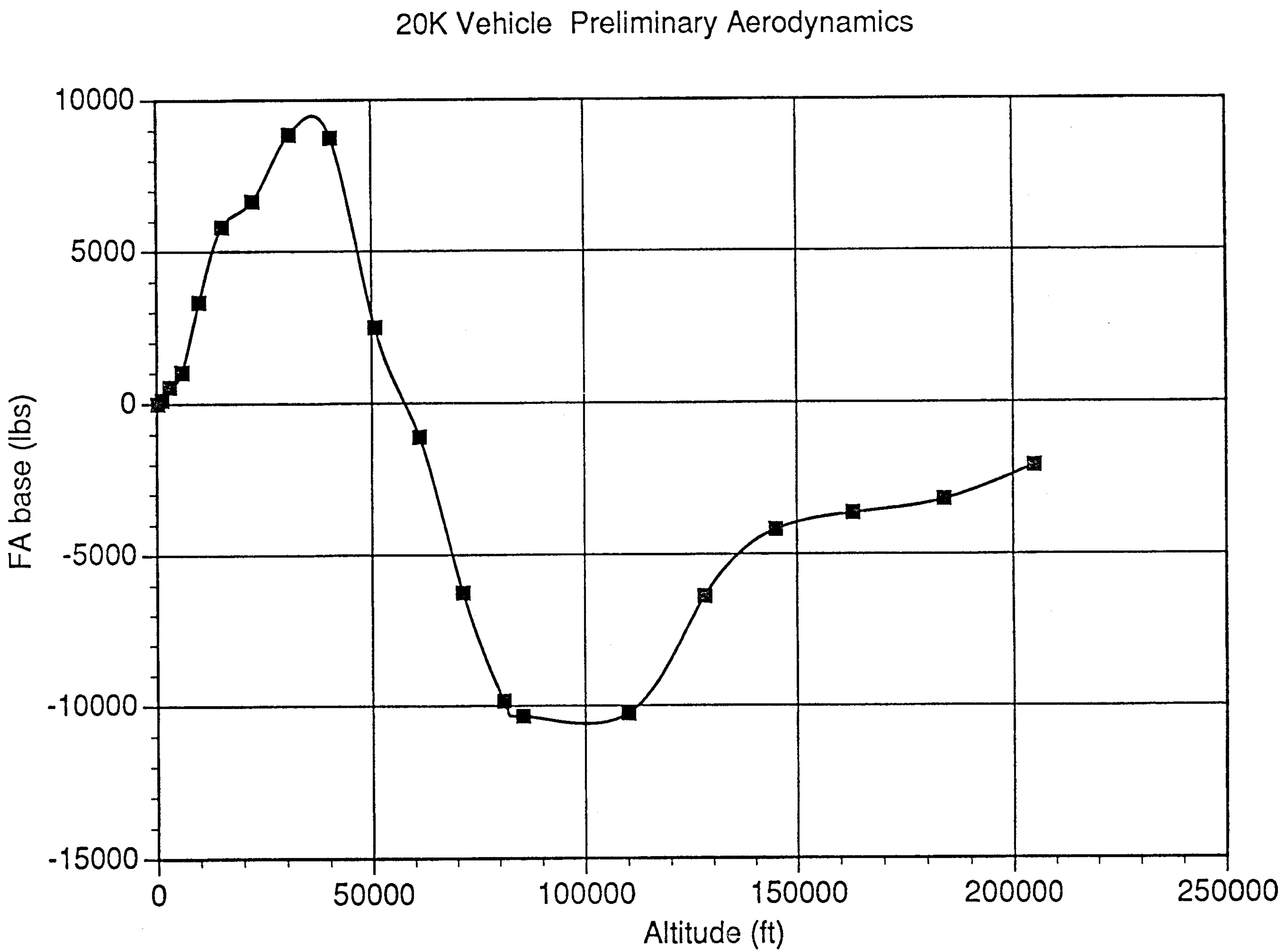


Figure 10. 20k vehicle base drag (preliminary).

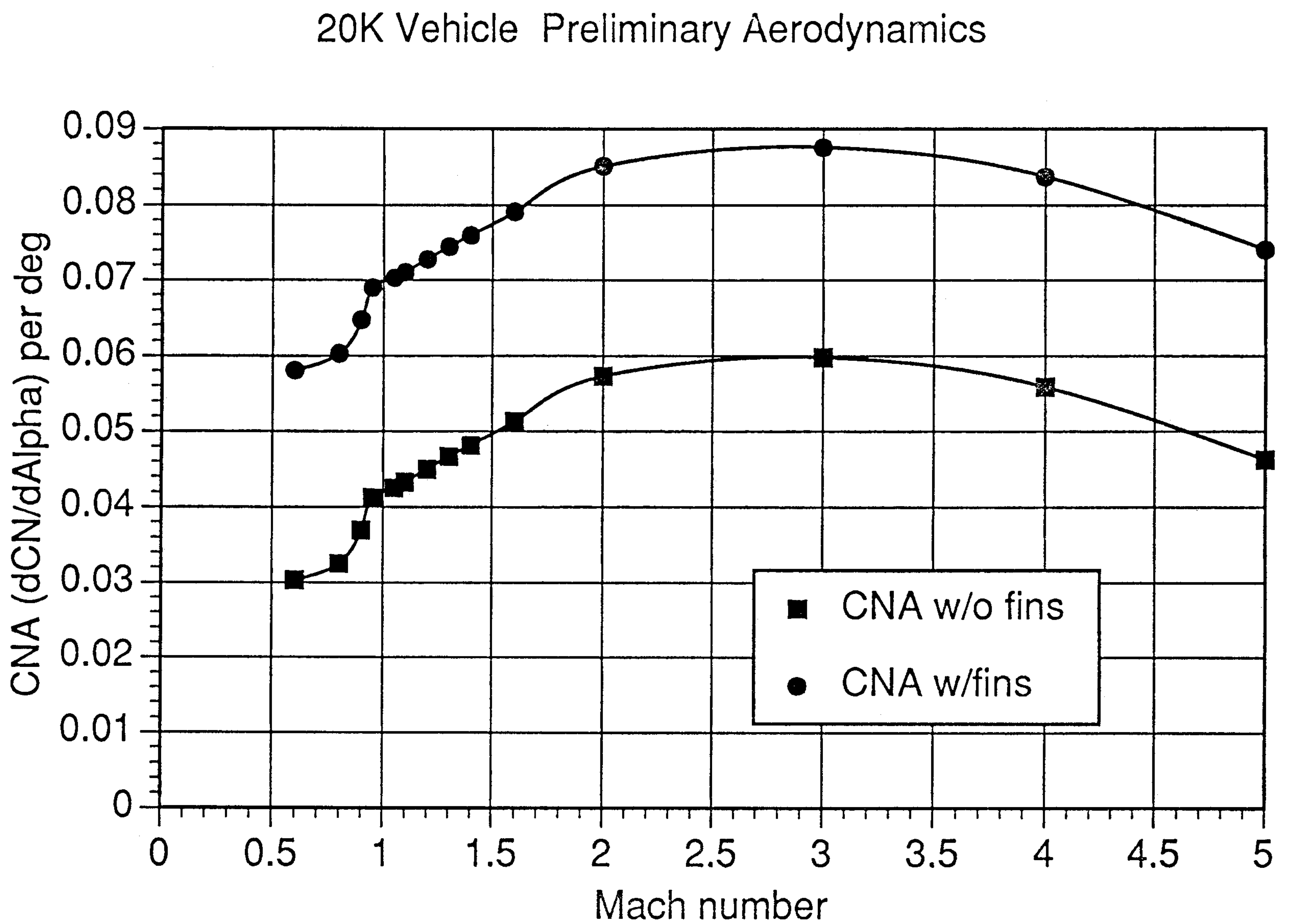


Figure 11. Normal force coefficient shape versus Mach number.

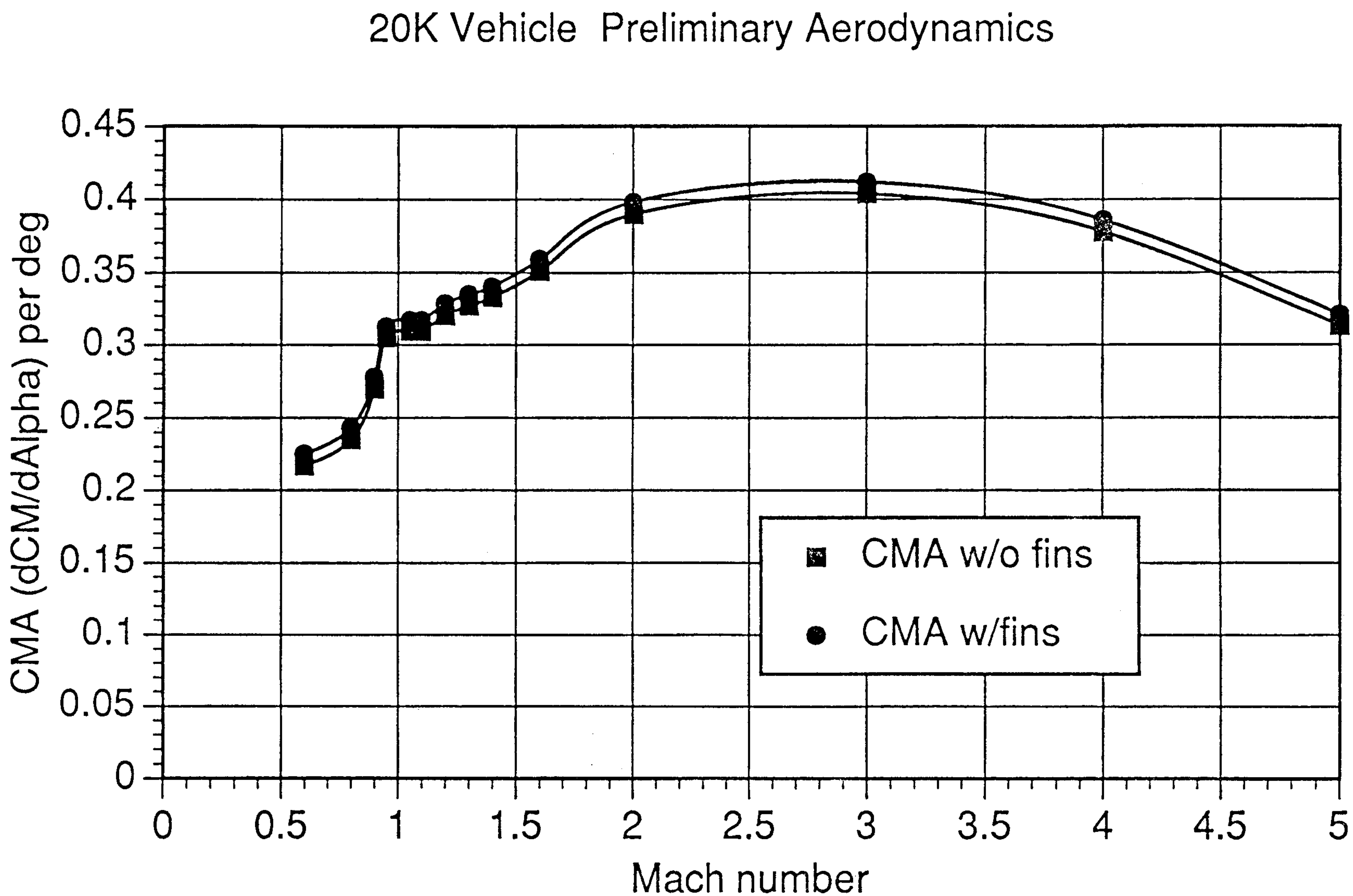


Figure 12. Pitching moment coefficient slope versus Mach number.

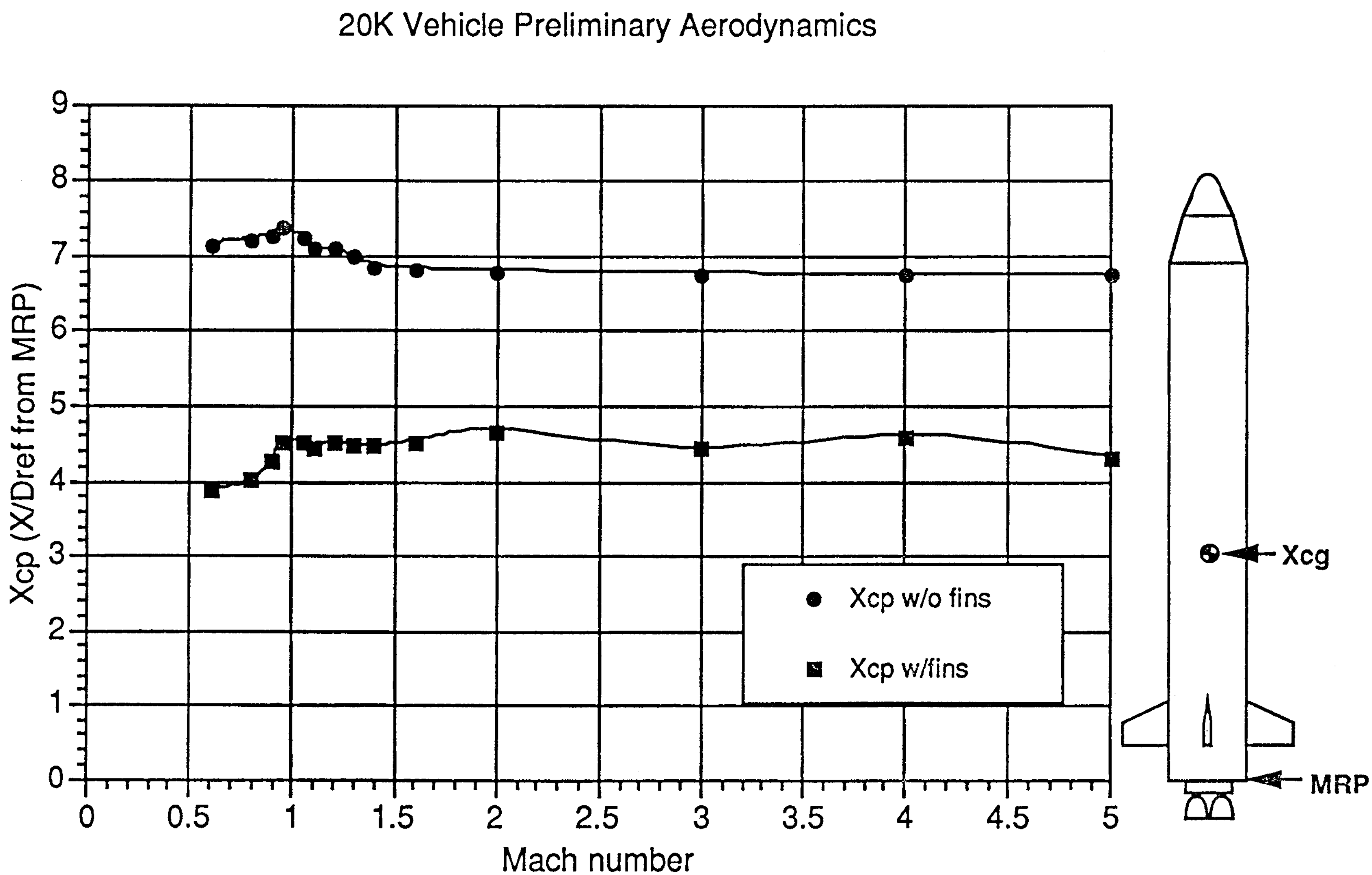


Figure 13. Center of pressure location.

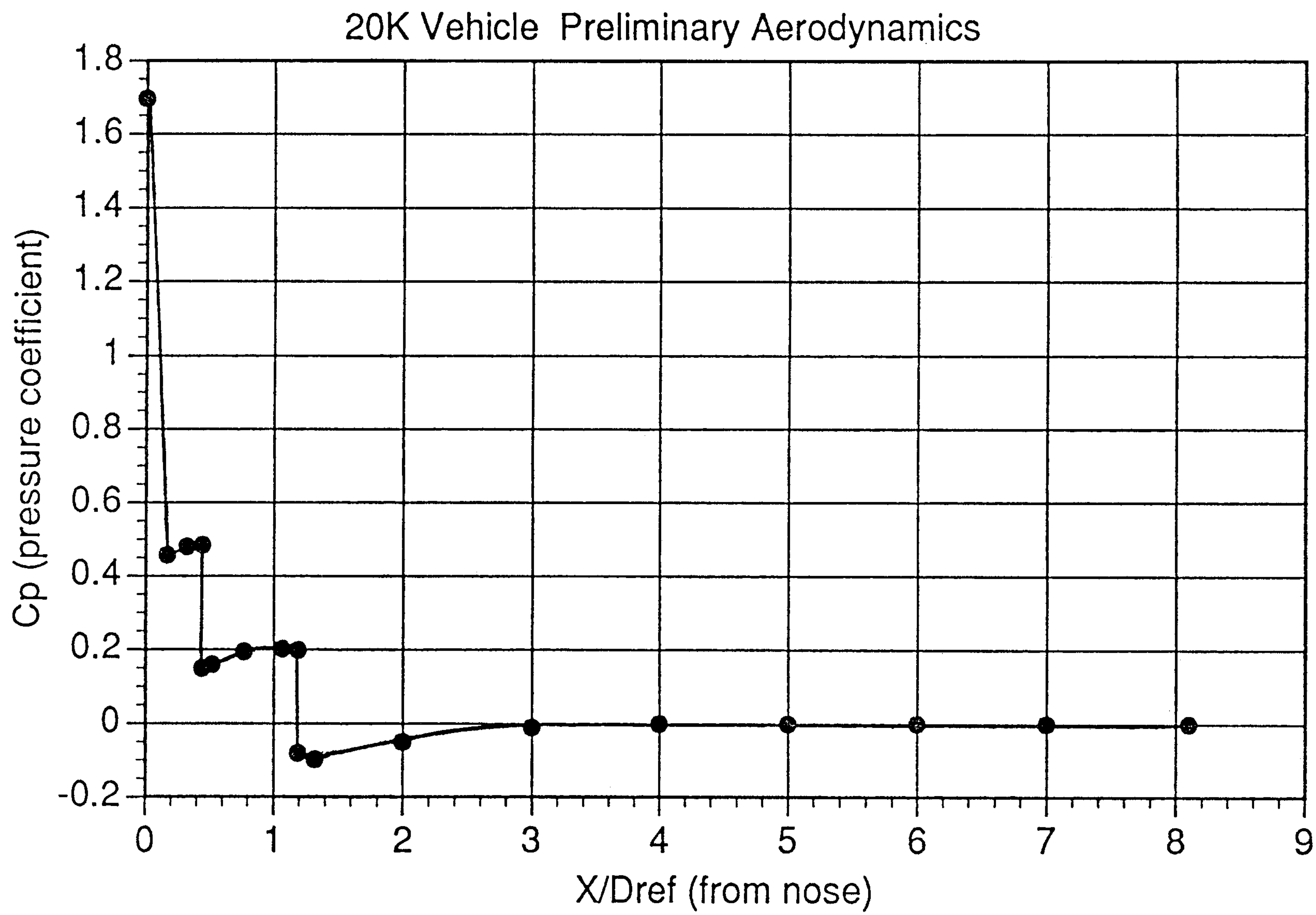


Figure 14. 20k vehicle pressure distribution (M = 2.0, alpha = 0).

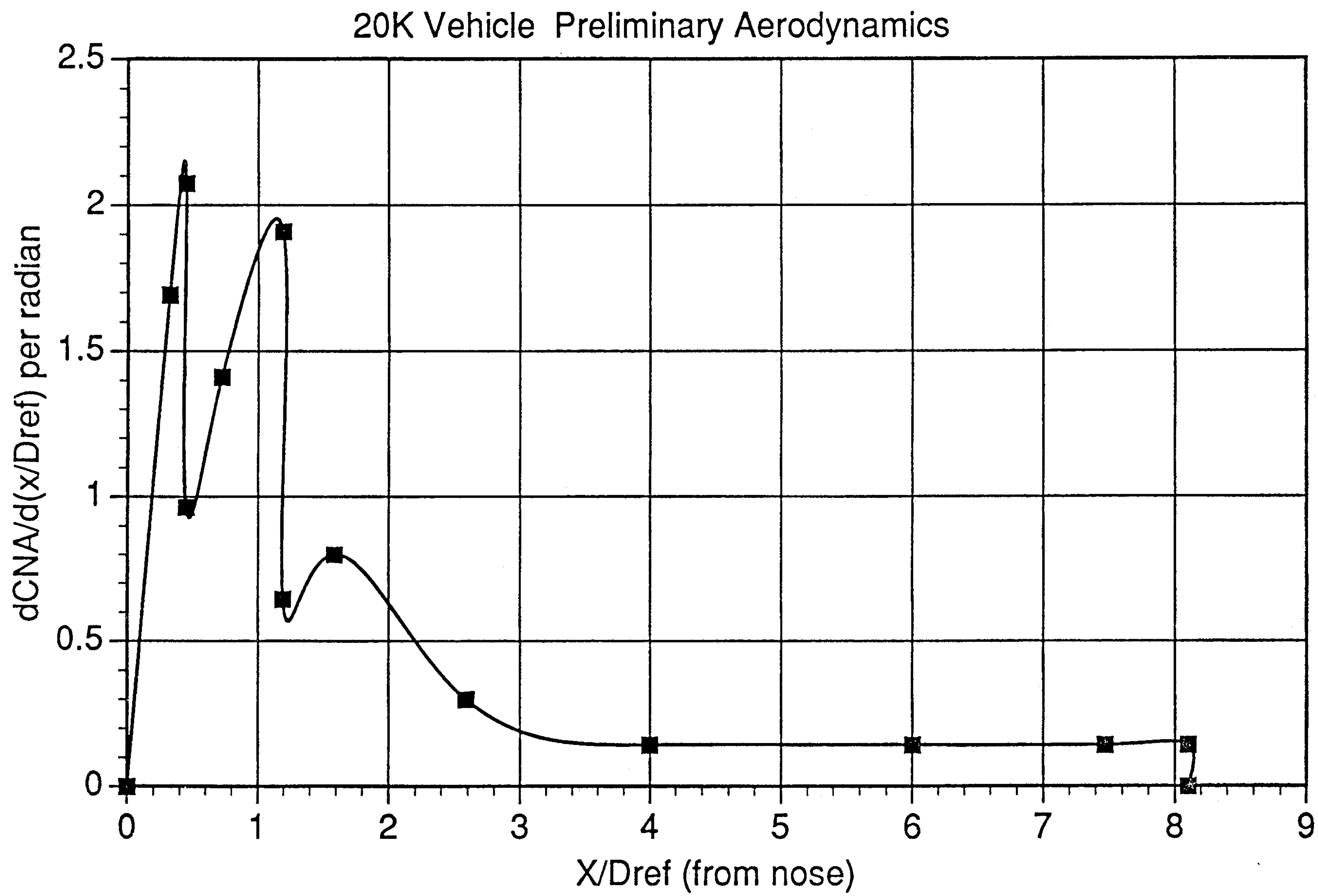


Figure 15. Normal force distribution (M = 2.0).

C. Loads

Preliminary loads were generated for the structure designers and are described by the following:

1. Ascent Loads. The following assumptions were used for the assessment of ascent loads by the Systems Loads Branch (ED22):

(a) A rigid beam model was used. The single elastic case that was run showed no appreciable differences.

(b) A Flight Mechanics Branch (PD33) developed trajectory was used for maximum dynamic pressure (Q), propellant mass, and Mach number definition. Maximum Q was 600 lb/ft^2.

(c) Various angles of attack were chosen.

(d) Normal and axial force coefficients were used for Mach 2.0 as supplied by the Structures and Dynamics Laboratory (ED34) (section II.B).

(e) A 1.25 factor was used on coefficients and Q*alpha equal to 500 psf-deg.

(f) No tank or venting pressures are included in the reported loads.

Figure 16 shows the *Nx* values for different angles-of-attack. Figures 17 and 18 show *Nx* and *Nv* comparisons of the 20k results to Titan allowables.

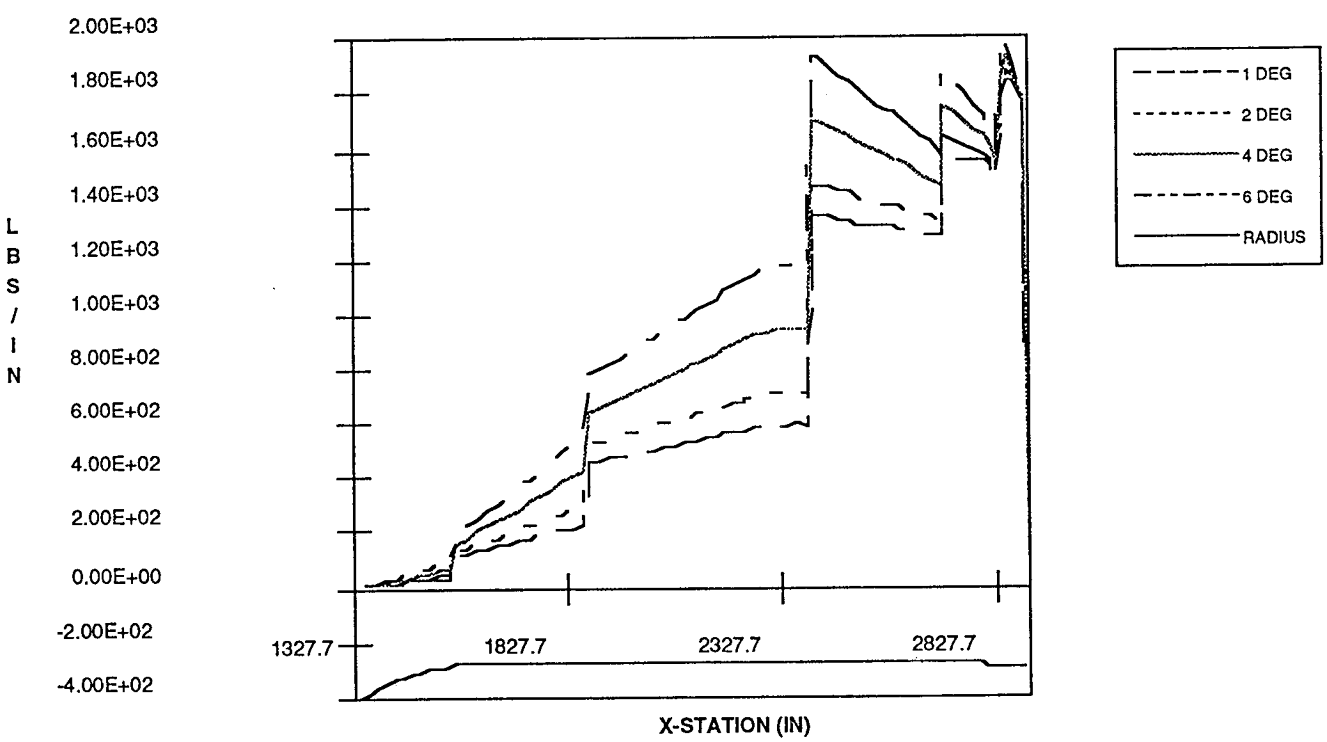


Figure 16. Core vehicle *Nx* versus alpha.

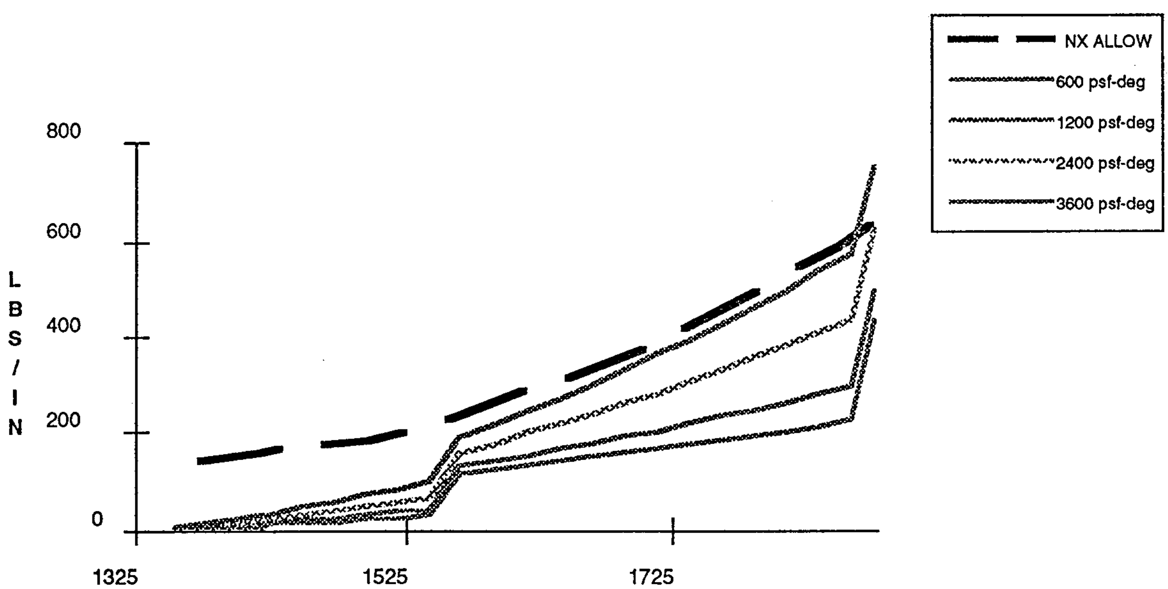


Figure 17. Shroud load comparison to Titan allowable.

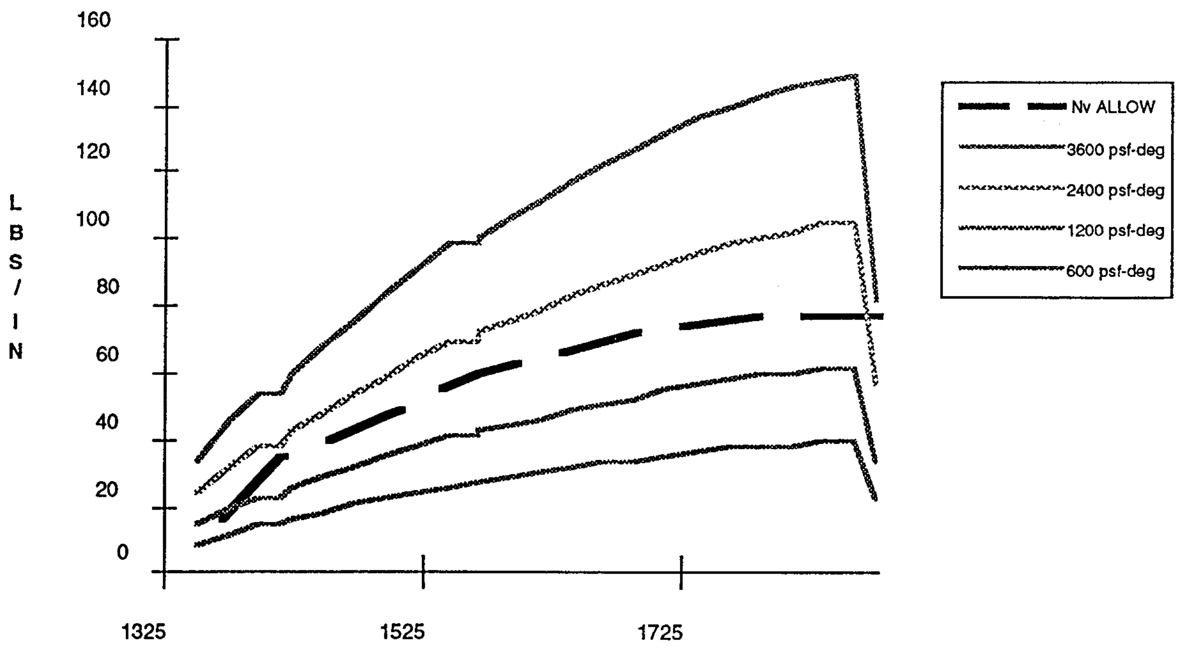


Figure 18. Shroud shear comparison to Titan allowable.

2. Buildup/Shutdown and Lift-Off Loads. The following assumptions were used by the Systems Loads Branch (ED22) for the assessment of buildup/shutdown and lift-off loads presented in figure 19:

(a) A NASA Structural Analysis (NASTRAN) finite-element beam model was generated using mass properties listed in section II.I and some assumed stiffness properties from the dimensions and geometry provided.

(b) One-day wind loads (at 1-percent risk) for on-pad loads and 1-h wind loads (at 5-percent risk) for lift-off were ground ruled. Ten day wind loads were also assessed for on-pad loads. A drag coefficient of 1.0 was used throughout for the vehicle, and an additional factor of 1.5 was used to account for vortex shedding effects.

(c) The RD–180 +5-percent thrust loading for buildup/shutdown and lift–off loads were scaled using the RD–170 maximum thrust and associated chamber pressure time histories.

(d) For lift-off loads, it was assumed that release occurred at maximum thrust, which occurs about 4.0 s after ignition.

(e) The preliminary loads were computed without any dispersions on the applied loadings.

(f) An uncertainty factor of 1.5 was applied to the dynamic responses only.

(g) No control feedback was considered during lift-off.

(h) The mobile launch platform (MLP) was approximated using stiffness and mass properties of the existing MSFC MLP shuttle loads model.

(i) Only one lift-off case and one buildup/shutdown case was run, i.e., only nominal cases have been considered.

(j) The Centaur and payload were modeled as lumped masses rigidly connected at x-station 1872.

Figure 19 shows the overall *Nx* values for buildup/shutdown, lift-off, and ascent. For more detail see reference memo ED22-93-42.

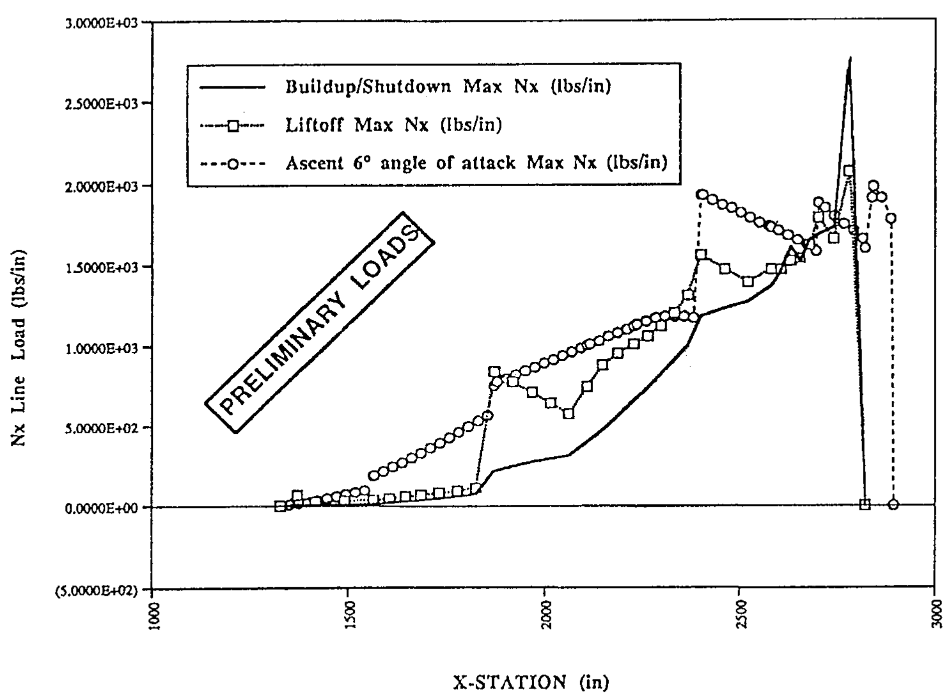


Figure 19. 20k in-house vehicle with RD–180 +5-percent thrust overall *Nx* line load versus x-station.

D. Performance

Initial parametric performance data were generated using the ground-ruled RD–180 as the booster engine with existing upper stage(s) and payload fairing. Booster sizing was performed using booster weight scaling equations with the Titan IV Centaur or Titan IV stage II upper stages. An Atlas-type payload fairing was jettisoned during ascent at 400,000 ft. Figure 20 displays the resulting payload of each upper stage combination in conjunction with a sized booster. The Titan IV stage II was rejected based on these data.

Figure 21 displays payload and Max q parameters versus booster vacuum thrust. The data show that increasing booster thrust alone, with corresponding propellant increase, could meet the payload requirement, but a limit of 5-percent increase was set by the engine team (see section II.J). Therefore, the remaining payload deficit had to be made up by the upper stage. RL10A–4 engines were substituted for the standard RL10–3–3A Centaur engines and combined with the 5-percent booster thrust increase, and the payload was achieved.

Table 1 displays the payload exchange ratios generated and delivered to the design teams for detailed analyses.

As the design was iterated, a decision was made to enclose the upper stage within the payload fairing to minimize the structural modifications required to handle the airloads. The Titan IV 200-in diameter fairing was introduced into the analyses and baselined. This increased fairing weight caused a payload reduction, but the 20k lb requirement was still attainable.

Following structural design, control, and aerodynamic trades, the final configuration evolved and is summarized in table 2. Performance for both LEO and GTO missions are provided. Detailed trajectory data were generated, delivered to the design teams, and are available upon request from the Flight Mechanics Branch (PD33). The resulting flight parameters are displayed in figures 22 and 23.

E. Control

1. Overview. Analyses conducted to evaluate the ascent controllability characteristics of this 20k launch vehicle concept are presented. Static stability and rigid-body dynamic response envelopes and a preliminary analysis of slosh damping requirements are discussed.

2. Configuration. Control of the 20k launch vehicle is by independently gimbaling the two RD–180 nozzles about two axes. These nozzles move in concert to provide pitch and yaw control torques, and move differentially to provide roll control torque. The RD–180 includes integrated thrust vector control (TVC) actuators with a capability of providing ±8° gimbal angle and ±3°/s gimbal angle rate under working load. Thus, the suitability of this capability for the 20k launch vehicle will be an issue.

3. Controllability.

a. Static Stability. A first measure of a launch vehicle's controllability characteristics is the ratio of the maximum available control torque to the maximum disturbance torque, C_r. A rule of thumb for vehicle concepts at an early design stage is to keep this ratio greater than 2. This will provide adequate margin for configuration maturity and dispersions and will account for the unmodeled effects. Early 20k launch vehicle concepts of this exercise were evaluated and found to be unacceptable due to

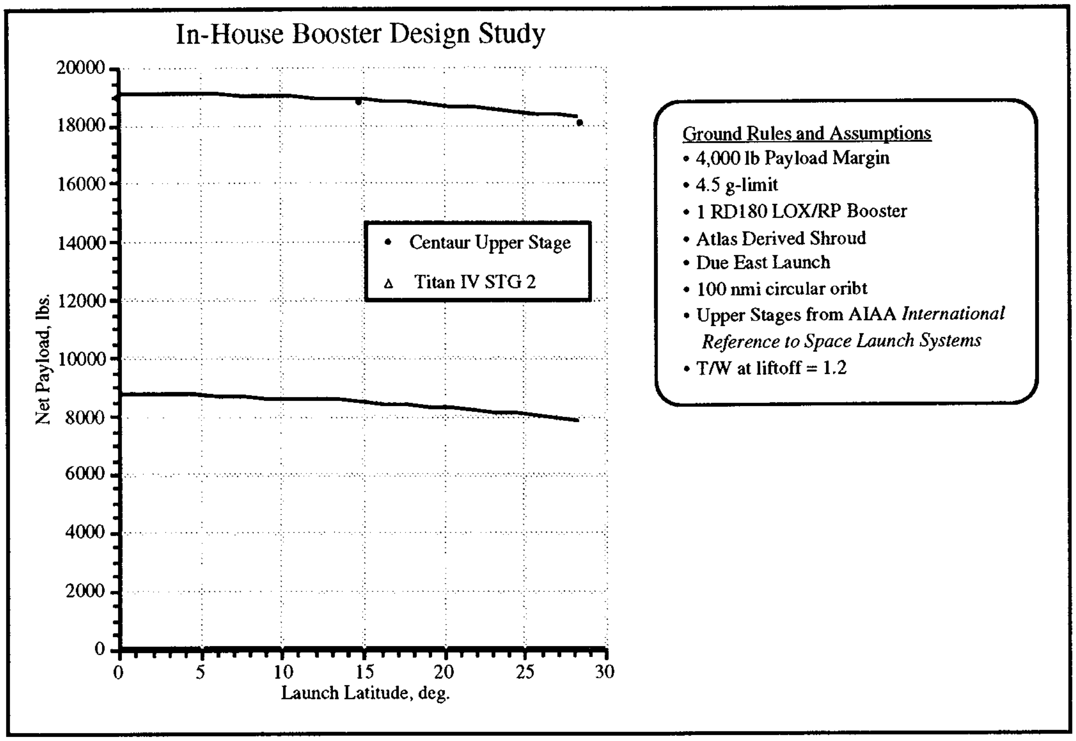


Figure 20. In-house booster design study.

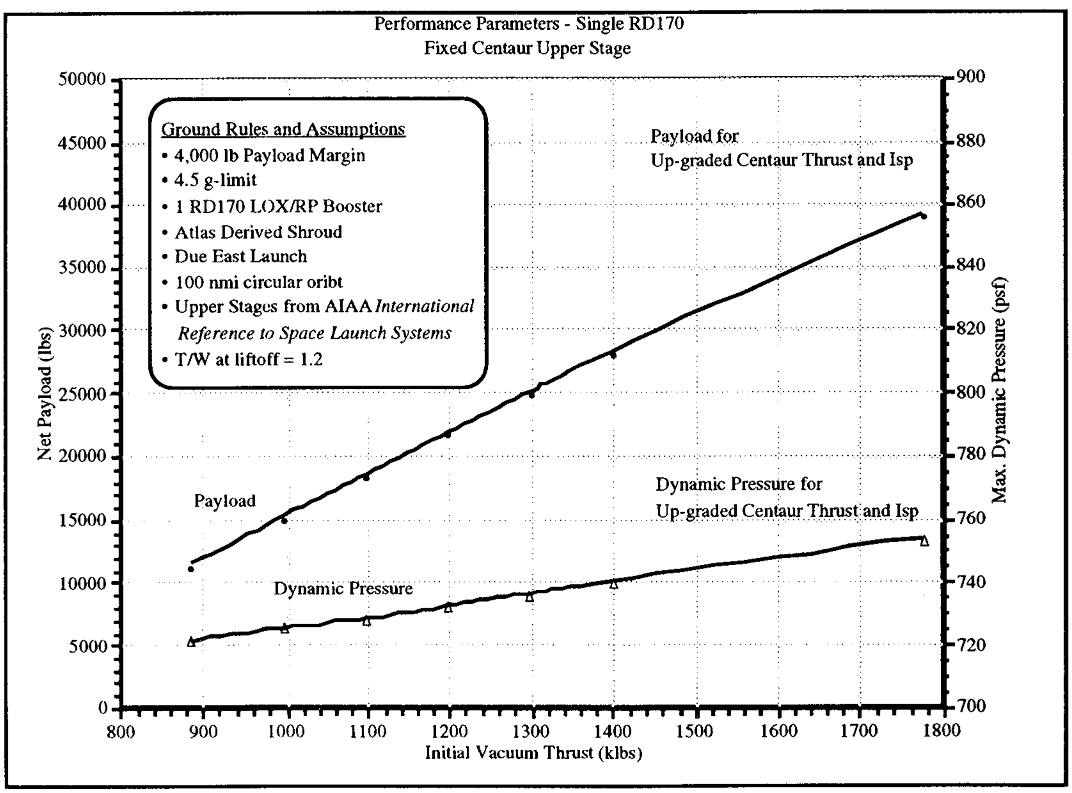


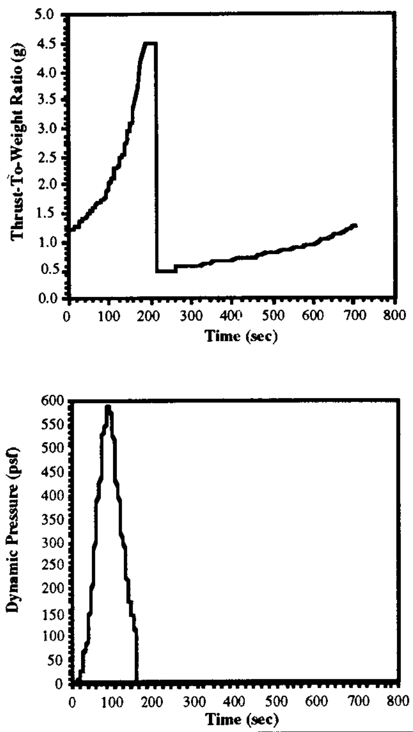
Figure 21. Performance parameters—single RD–170 fixed Centaur upper stage.

Table 1. Payload exchange ratios.

Partials	5-Percent Thrust Increase
Booster Drop Weight	–0.266 lb payload/lb drop weight
Aerodynamic Drag	–8.60 lb payload/percent change
Shroud Drop Weight	–0.305 lb payload/lb drop weight

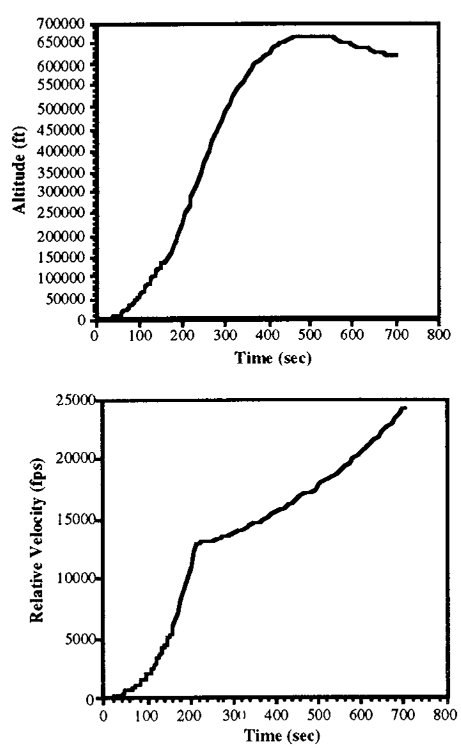
Table 2. Performance results, RD–180 booster and Centaur upper stage.

	LEO	GTO
Weights		
Gross Lift-Off Weight	731,052 lb	715,112 lb
Booster		
Propellant	588,312 lb	588,312 lb
Stage Weight	55,739 lb	55,739 lb
Upper Stage		
Propellant	45,727 lb	45,727 lb
Stage Weight	7,387 lb	7,387 lb
Shroud Weight	9,200 lb	9,200 lb
Margin	4,000 lb	795 lb
Net Payload	20,688 lb	7,953 lb
Flight Parameters		
Maximum Dynamic Pressure	591 lb/ft^2	623 lb/ft^2
Minimum RD–180 Throttle	67.97 percent	60.38 percent
Maximum Acceleration	4.5 g	4.5 g
Total Weight at Lift-Off	1.193	1.219
Engine Data		
RD–180		
Thrust, Vacuum	945,000 lb/ft	945,000 lb/ft
Isp, Vacuum	337 s	337 s
RD–10A–4 (2)		
Thrust, Total Vacuum	41,600 lb/ft	41,600 lb/ft
Isp, Vacuum	448.9 s	448.9 s



- Gross Liftoff Weight = 731,052 lbs
- 1 RD180 Booster
    - Engine Type: LOX/RP
    - Thrust : 945,000 lbs, vac.
    - Specific Impulse: 337 sec., vac.
    - Propellant = 588,312 lbs
    - Stage Weight = 55,739 lbs
- Centaur Upper Stage
    - Engine Type: LOX/LH2
    - Engine Designation: RL10A-4N
    - Average Thrust: 2 x 20,800 lbs, vac.
    - Specific Impulse: 448.9 sec., vac.
    - Propellant = 45,727 lbs
    - Stage Weight = 7,388 lbs
- Titan Derived Shroud
  Weight = 9,200 lbs
- 4,000 lb Payload Margin
- Net Payload = 20,688 lbs
- Due East Launch
- 100 n.mi. Circular Orbit

Figure 22. In-house 20k booster design, flight parameters.



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- Due East Launch
- 100 n.mi. Circular Orbit

Figure 23. In-house 20k booster design, flight parameters.

their high aerodynamic instabilities and limited control torque from the RD–180. The selected configuration is acceptable from this standpoint, utilizing a constant diameter geometry and fixed aft-mounted fins. The ratio, $C1/C2$, of aerodynamic torque (per degree angle of attack) to control torque (per degree gimbal angle) is shown in figure 24. To relate this ratio to the controllability ratio, C_r, one applies the limits on gimbal angle and angle of attack.

$$\frac{\text{Max Gimbal Angle}}{\text{Max Angle of Attach}} \geq \frac{C1}{C2} \times Cr \quad .$$

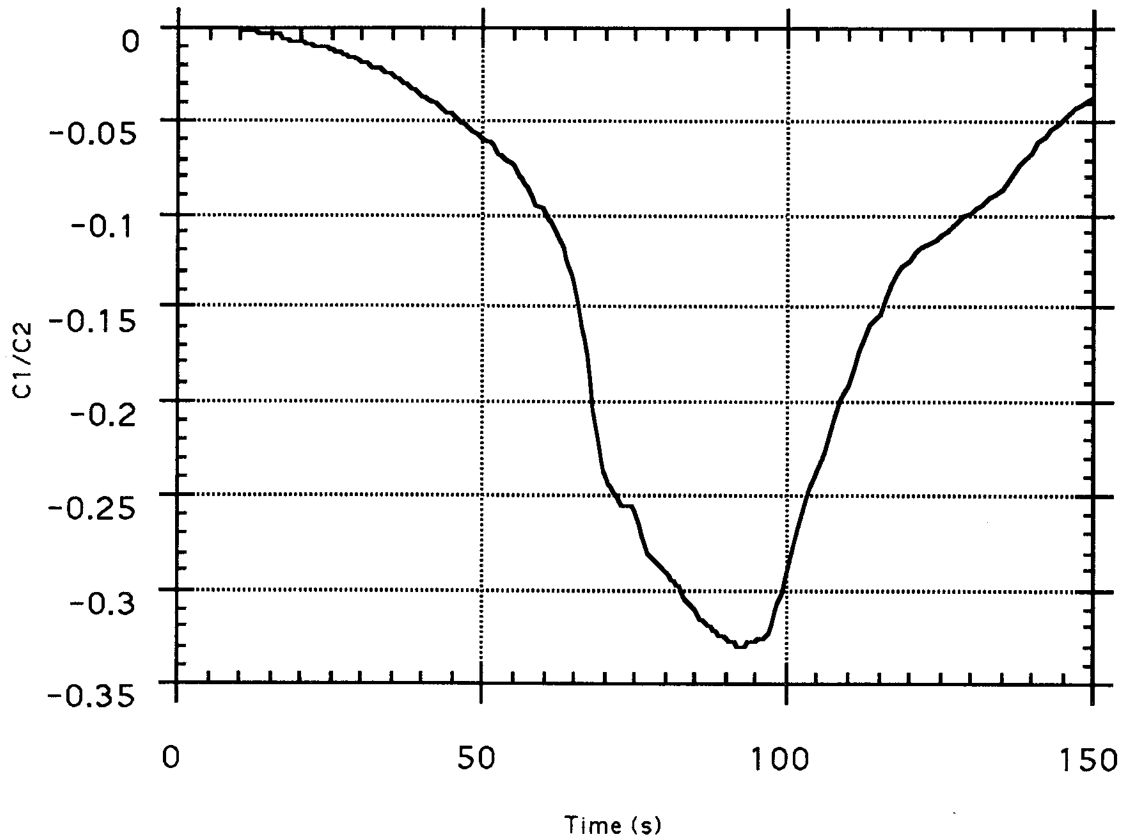


Figure 24. 20k RD–180 no-winds aerotorque to control torque ratio.

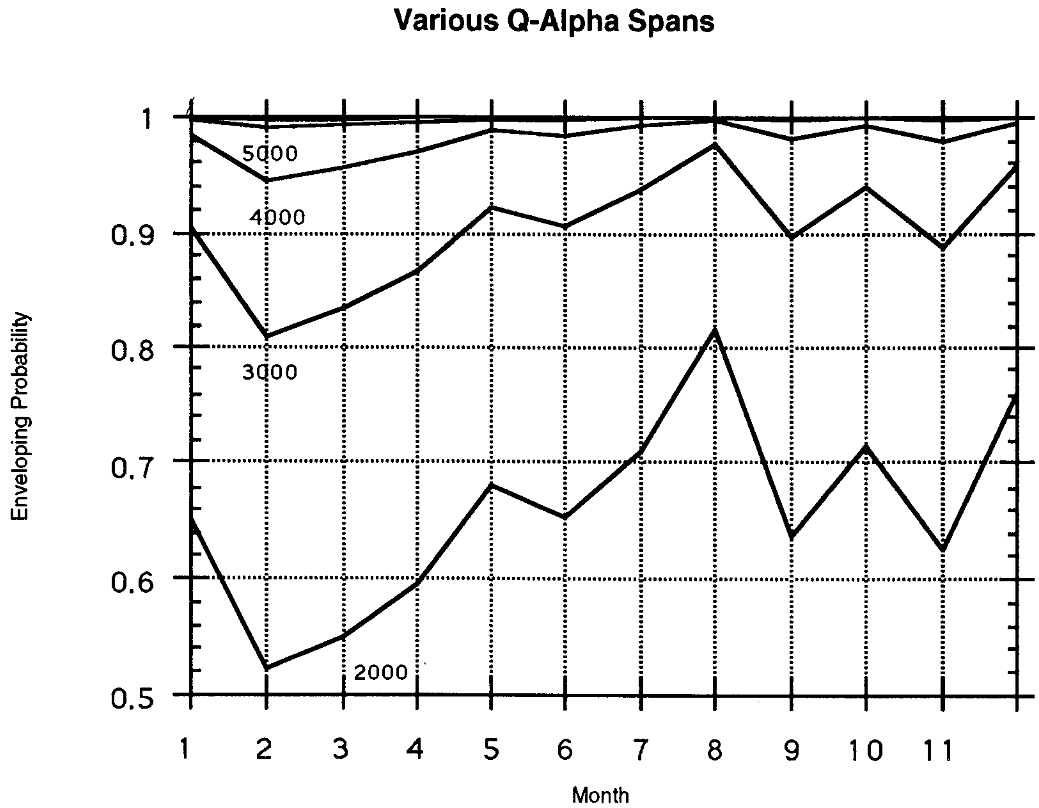
At the worst-case time of flight (approximately 90 s), the ratio $C1/C2$ is approximately –0.33 (the negative sign indicates aerodynamic instability). Assuming that a maximum gimbal capability is 8°, the amount that should be allocated for static moment balance should be 4° (dynamic effects, dispersions, and misalignments account for the rest). The maximum angle of attack was chosen to be 6°, to provide a reference for loads cases and vehicle sizing with some "robustness" with respect to winds. The resulting value of C_r at this condition is then 2, the vehicle is acceptable from a static-stability standpoint.

b. Dynamic Response Envelopes. To develop preliminary ascent load indicator envelopes and to examine the suitability of the RD–180 TVC system, rigid-body dynamic response simulations were performed. The reference trajectories were obtained from the Flight Mechanics Branch (PD33), mass properties were obtained from the Systems Integration Branch (PD24), aerodynamics from the Experimental Facilities Branch (ED35), and vehicle geometry from the Structural Development Branch (ED52).

While the eventual control system architecture and trajectory-shaping philosophy should be chosen based upon indepth trade studies, the time to perform these trades was not available. A reference was adopted based upon engineering judgment. The reference control system was a rate and attitude feedback system (no load relief), with gains scheduled to keep the control frequency and damping at 0.25 and 0.707 Hz, respectively. This type of control architecture generally results in sensor requirements compatible with the current Centaur upper stage avionics. It also results in a control system less sensitive to payload variations, therefore, requiring less recurring control analysis (a program goal).

To keep ascent loads within reasonable levels, monthly wind biasing was chosen as a reference for trajectory shaping. However, monthly mean wind-biased trajectories were not generated. Instead, the nominal trajectory was used and "flown" against monthly jimsphere (measured) wind profiles. The maximum q-alpha and q-beta occurrences from each run were recorded at several altitudes. These maximums were then statistically enveloped to produce squatcheloid ellipses at each altitude. To emulate monthly biasing, the span of each ellipse was used, instead of the actual maximum and minimum values. (This is approximately equivalent to centering the squatcheloid, as would be done through wind biasing, but does not take into account performance losses or other trajectory dispersions which would occur when wind biasing is used.) Figures 25 through 27 show the wind enveloping percentage for each month for various q-alpha and q-beta span levels for 10-, 11-, and 12-km altitudes. These data can indicate the approximate wind probability level that the vehicle could withstand if it were structurally designed to a given q-alpha or q-beta span capability. At this stage, appropriate dispersions should be added to the span to account both for the simplicity of the approach and for uncertainty in the configuration data. As an example, assume the vehicle can withstand ±3,000 psf*deg in q-beta. Assume also that dispersions account for 1,000 psf*deg. Then the effective q-beta limit for the vehicle would have to be reduced by the dispersion, resulting in a q-beta limit of ±2,000 psf*deg. This corresponds to a span of 4,000 psf*deg. From figure 26 (11-km altitude) this 4,000 psf*deg span limit would indicate that the vehicle could withstand approximately 88 percent of the November winds. The wind enveloping percentage is higher for all other months of the year. Keeping in mind the approximations and simplicity of the approach, these data could assist in trading wind enveloping percentage (i.e., launch probability due to winds aloft) for vehicle structural design limits.

The adequacy of the RD–180 TVC system was examined by observing the maximum gimbal angles and gimbal angle rates required during the ascent simulations using the jimsphere winds. Gimbal angles were always within ±2.5°, and gimbal angle rates were within ±1.5°/s. Given the 8° and 3°/s gimbal angle and rate capabilities, respectively, the RD–180 TVC system should be adequate for the reference control architecture. Dispersions should fall within these capabilities also. The gimbal rate has the least margin, and would likely be a limiting factor in choosing other control architectures (e.g., load relief) that require higher gimbal rate capability.



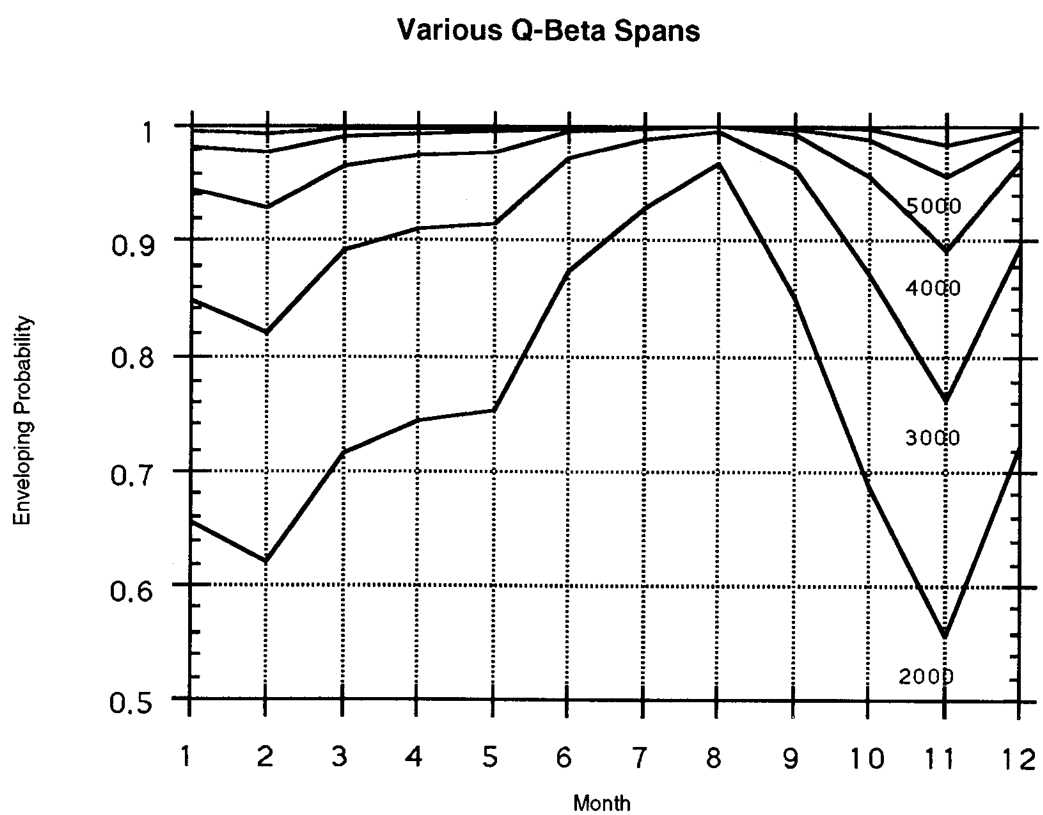
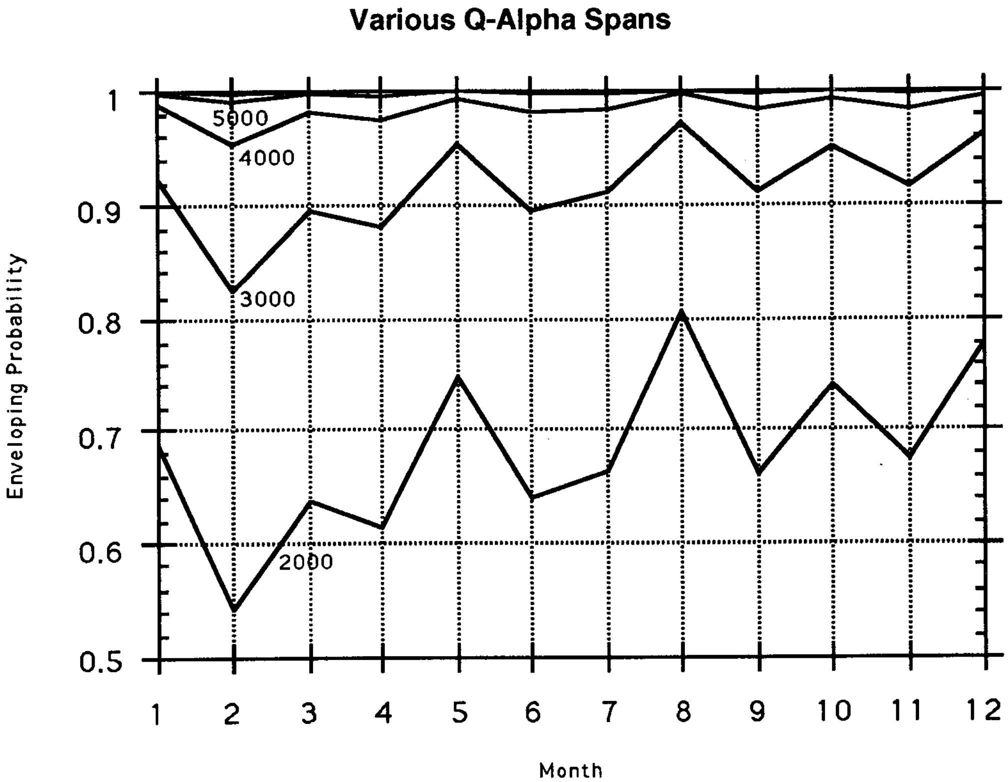


Figure 25. Monthly wind envelope probabilities at 10-km altitude.



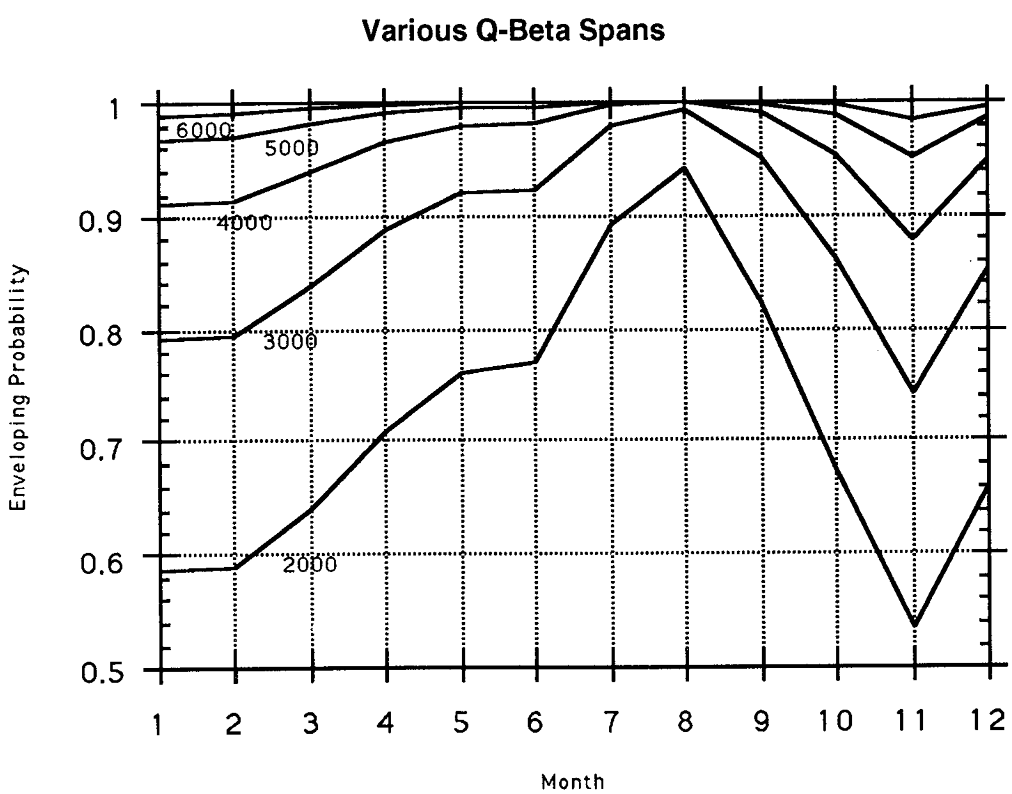
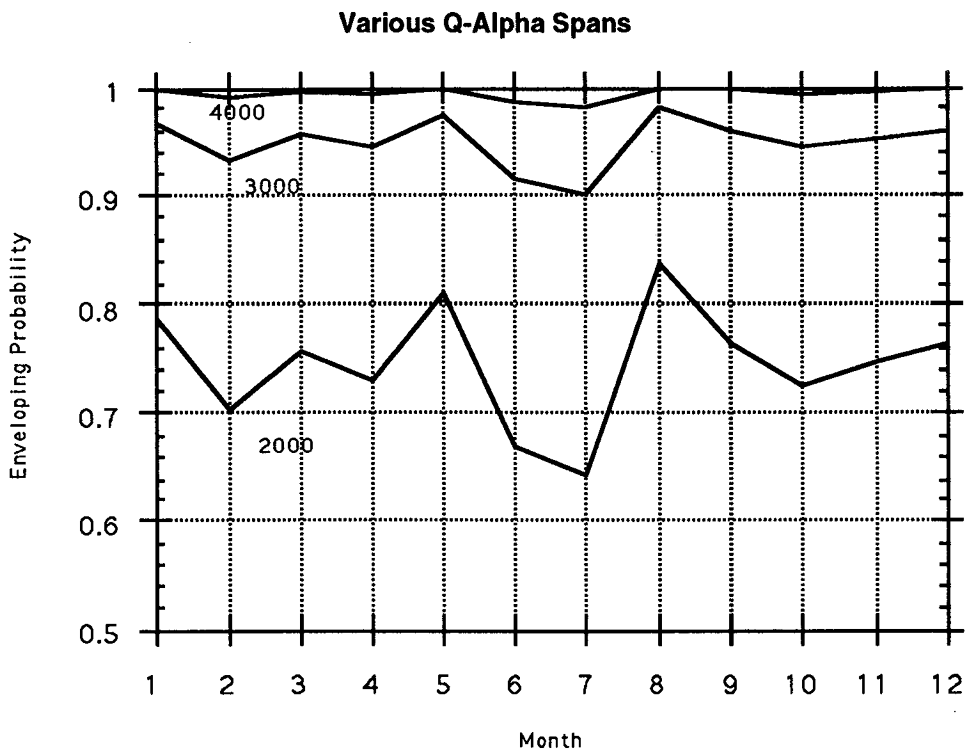


Figure 26. Monthly wind envelope probabilities at 11-km altitude.



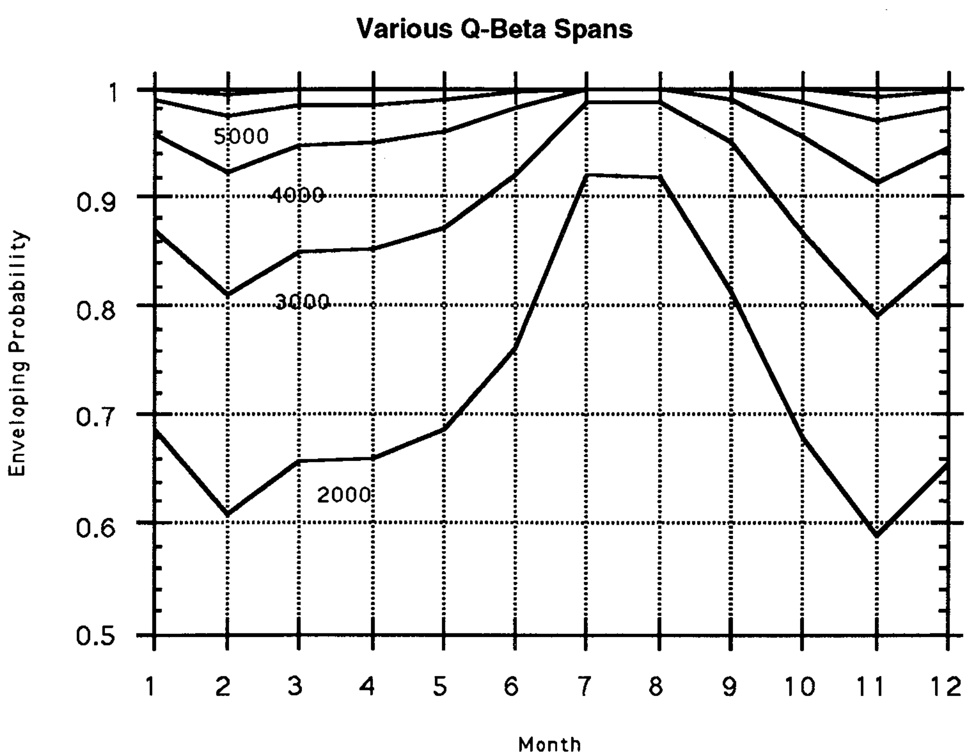


Figure 27. Monthly wind envelope probabilities at 12-km altitude.

4. Slosh. Simple slosh modal data were developed for both the LO_2 and fuel tanks during the times of flight when the respective fluid levels were within the cylindrical portions of each tank. For each tank, the requirement for slosh damping was based upon the ratio of the slosh mass to the total vehicle mass, and whether the slosh was phase stable or unstable. As a rule of thumb, when the slosh is phase unstable or when the slosh mass exceeds 10 percent of the total vehicle mass, baffles are usually required. The actual damping ratio was based upon engineering judgment. Figure 28 shows the slosh first-mode frequencies versus time of flight for both the fuel and LO_2 tanks. Figure 29 shows the slosh mass ratio and required damping versus station number for both tanks.

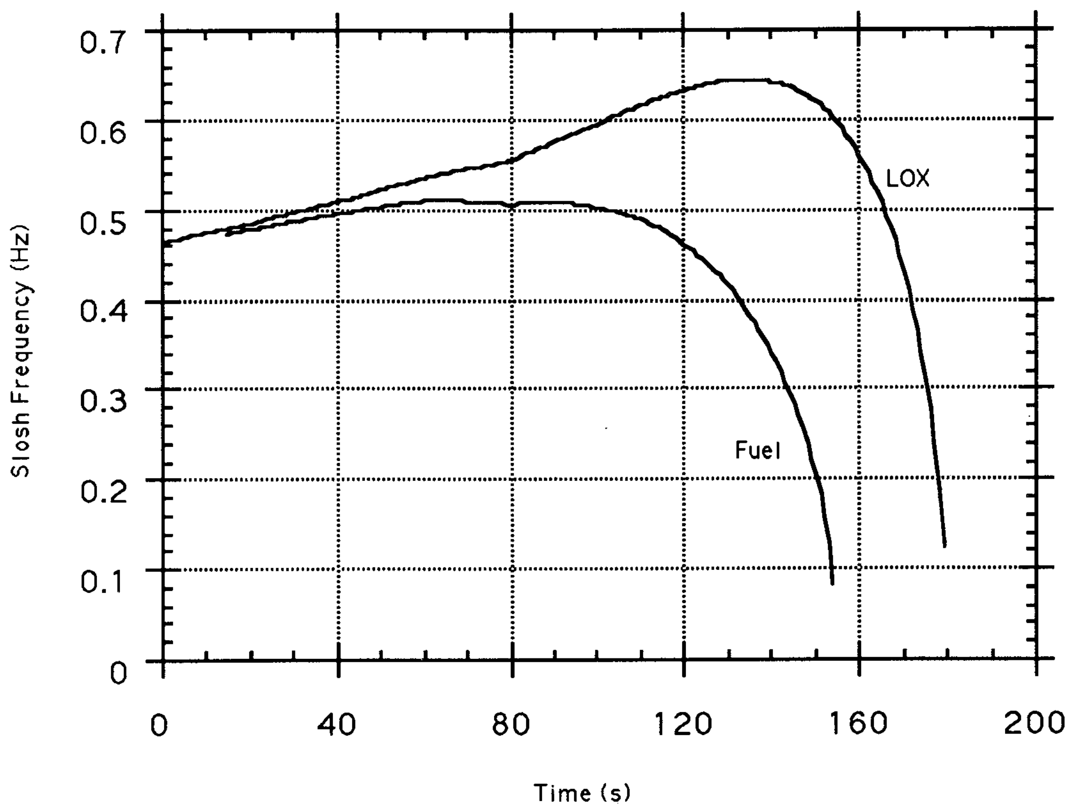


Figure 28. 20k RD–180 lox and fuel slosh frequencies.

5. Summary and Conclusions. The most recent 20k RD–180 booster configuration was found to possess adequate controllability based upon simple static and rigid-body dynamic analysis. A reference control system was designed and, when combined with a monthly wind biasing approximation, was analyzed to determine wind enveloping percentages for various structural design load indicator limits. All results indicate that the TVC capabilities of the RD–180 will be sufficient. Preliminary propellant slosh damping requirements have been defined based upon simple slosh analyses and engineering rules of thumb. Additional studies are required to better optimize the control architecture and trajectory shaping philosophy, to better define the slosh damping requirements, and to develop sensor requirements.

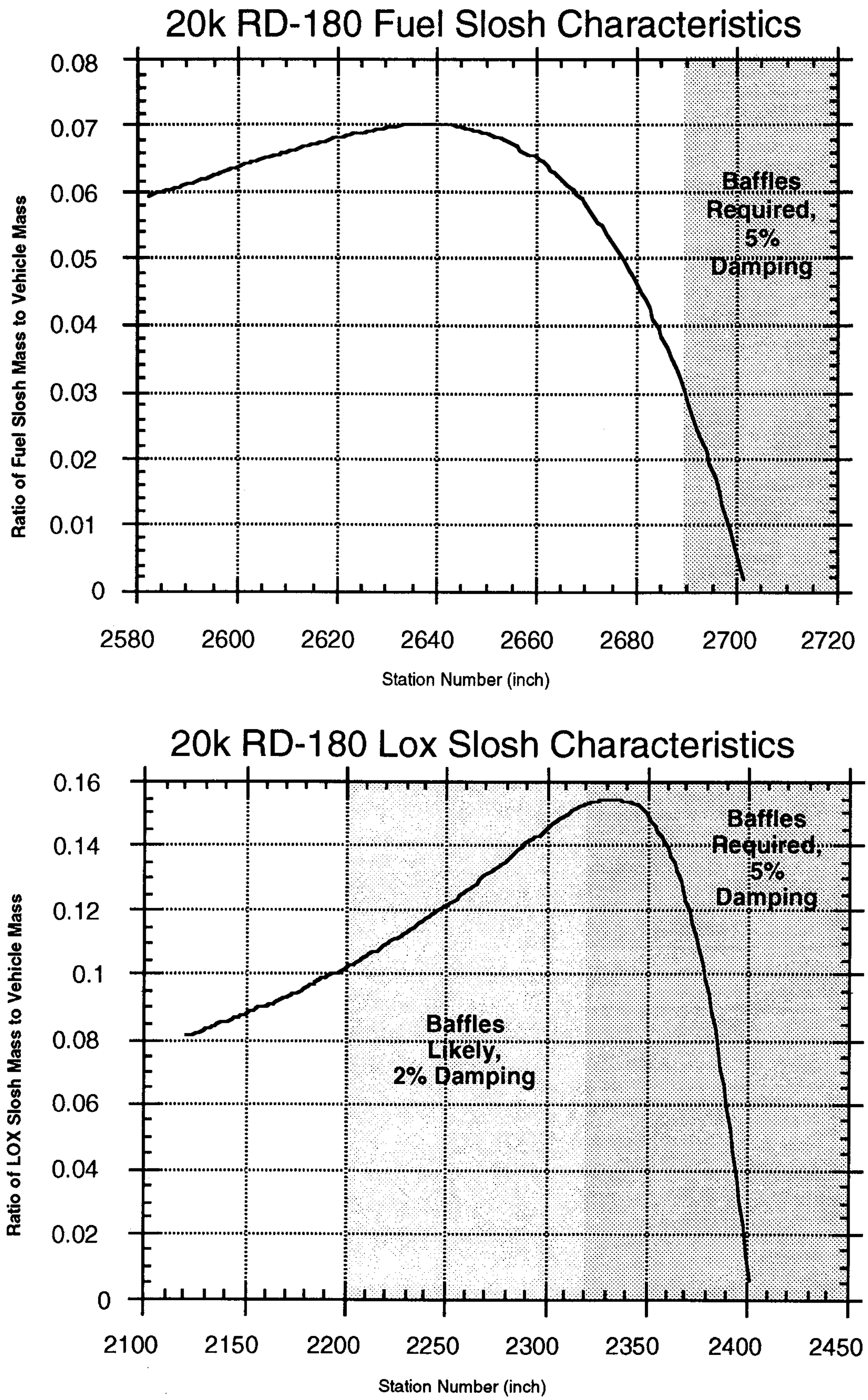


Figure 29. 20k RD–180 slosh characteristics.

F. Structural Design

During this design analysis, payload trades were performed using the payload exchange ratios as generated by the flight mechanics organization and listed in section II.D.

The material used for design was aluminum 2219-T87 with an ultimate tensile strength of 63 ksi. Aluminum-lithium (Al-Li) would reduce the booster weight approximately 10 percent, adding 785 lb of payload capability. Al-Li was not used because the high material cost is inconsistent with a low-cost booster, and payload capability is not as sensitive to the booster weight as to the upper stage weight. Composites were not considered due to the brevity of the study, but cost may be comparable with the aluminum (by using automated manufacturing) while the weight should be less than with Al-Li. These materials could be inserted into the production for product enhancement as their technologies are finalized.

The cylindrical tanks, skirts, and intertanks were analyzed using the cylinder optimization of rings, skin, and stringers (CORSS™) computer program. This program was recently developed by the Systems Strength Analysis Branch (ED24) and the Structural Development Branch (ED52) at MSFC. The program allows an optimum or near-optimum geometry to be selected based on the design loads, dimensions (radius, length, etc.), and dimensional constraints (such as maximum stiffener height, minimum skin thickness, etc.). As a part of this analysis review, checks were made against hand methods and against another shell stability computer program. These checks led to the CORSS™ program being modified twice. One change was an improvement in the stringer torsional stiffness calculation, and the other was a correction in the stringer eccentricity from the skin in the stability calculations.

The 20k booster was designed using a 2.0 ultimate safety factor. This factor of safety was chosen per MSFC–HDBK–505 Rev. A, so that structural testing would not be required, saving schedule time and testing costs. The skin panels between stringers were allowed to buckle at a load factor of 1.0, with the constraint that the ultimate factor of safety was still greater than 2.0. A safety factor of 1.4 would increase payload capability by 1,015 lb, but at the cost of a structural test program. A production proof test without inspection is seen as acceptable.

The dry bays were assumed to be skin/stringer construction with integrally machined, external blade stringers (fig. 30). The dry bays are made of identical 60-in elements sized for the Max q, 6° angle of attack load case. The interstage is 240-in long (four elements), the intertank is 180-in long (three elements), and the aft skirt is 120-in long (two elements). If the dry bays were sized individually, a 306-lb payload gain is available.

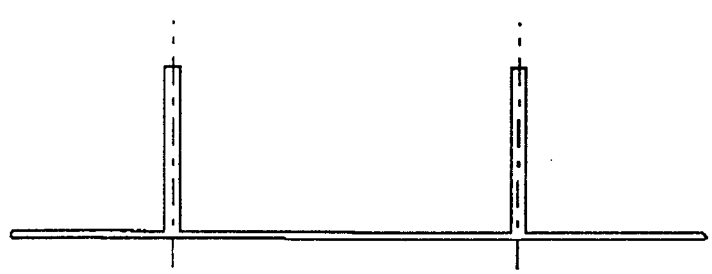


Figure 30. Blade stiffeners.

Buildup/shutdown loads provided (section II.E) are significant for the aft skirt. These increased loads added 1,431 lb to the booster weight over earlier estimates. The preliminary RD–180 buildup/ shutdown data must be refined and the engine throttling capability utilized to reduce these loads.

The 60-in element length selected from a commonalty aspect causes interference with the feedlines. The feedlines need 44.96 in between domes, and only 38.58 in are provided (fig. 31). Layouts using 100-in elements instead of 60-in elements seems to eliminate the problem with no weight impact. A trade is required to solve this. Penetrations into the dry bays would be limited to a single segment panel of the common elements. For example, the access door and feedlines in the intertank would be inline.

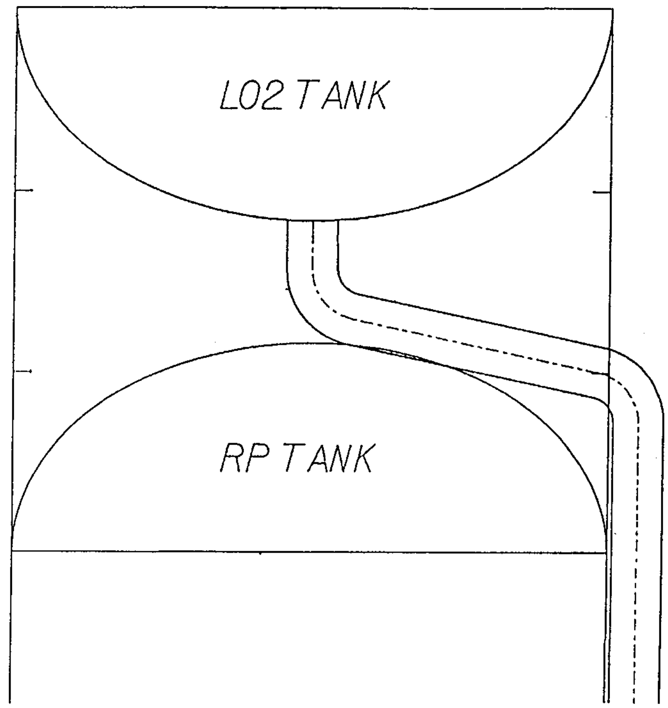


Figure 31. Feedline/tank interference.

Isogrid/orthogrid construction may reduce the dry bay weight slightly. Isogrid would add 282 lb of payload for the 6° Max q loading. However, isogrid would result in 1,542 lb less payload using buildup/shutdown loads. As the angle of attack decreases, flight loads decrease and they become lower than the ground loads. A 1° versus 6° angle of attack at Max q loading only adds 156 lb of payload.

The tanks were baselined as skin/stringer for manufacturing and cost and are stiffened by integrally machined, internal blade stringers. There are no intermediate rings due to pressure stabilization. Slosh baffles are required in both the LO_2 and RP tank, and a 5-percent damping was used to generate the baffle weight. Due to manufacturing limitations the LO_2 tank will need a circumferential weld. However, a constant cross section was maintained so that the two halves would be identical for ease of manufacturing. Isogrid/orthogrid tanks would be lighter due to the more efficient load carrying in the hoop direction. Isogrid tanks could add 1,031 lb of payload capability. To eliminate Y-rings, the interfaces to the tanks are through welded stubs with external bolt flanges (fig. 32), but this requires slightly different diameters for the dry bays and tanks.

The tank domes are square root of two elliptical domes to minimize vehicle stack length without requiring stiffening of the domes. They are sized for the required weld thickness to eliminate the need for machined weld lands. The fore LO_2 and both RP domes are identical. Four individual, weld-thickness domes with four new weld schedules would add 141 lb of payload. Four individual domes, with weld lands, add 632 lb of payload. Tank penetrations are limited to the top access holes and bottom propellant outlet.

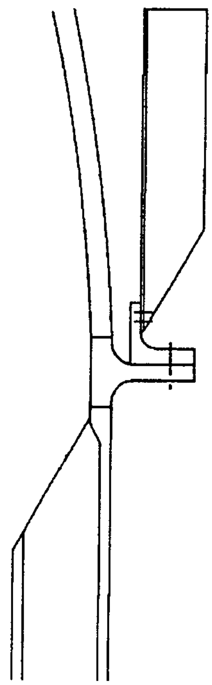


Figure 32. Tank/dry bay interface.

The vehicle lift-off (fig. 33) and upper stage separation systems are linear-shaped charges (fig. 34). This system provides well-distributed loads into the vehicle and is currently being investigated as an improvement to the shuttle holddown. Since the shroud is still attached when the booster is separated, payload and upper stage contamination is not a concern.

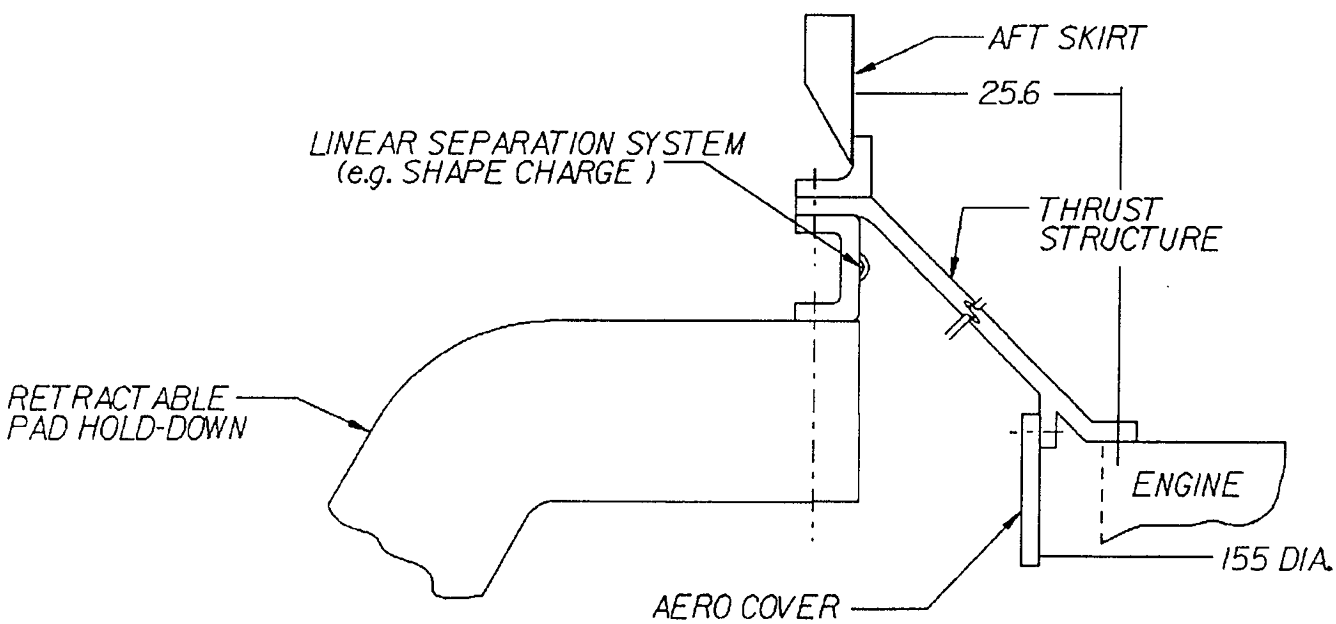


Figure 33. Holddown.

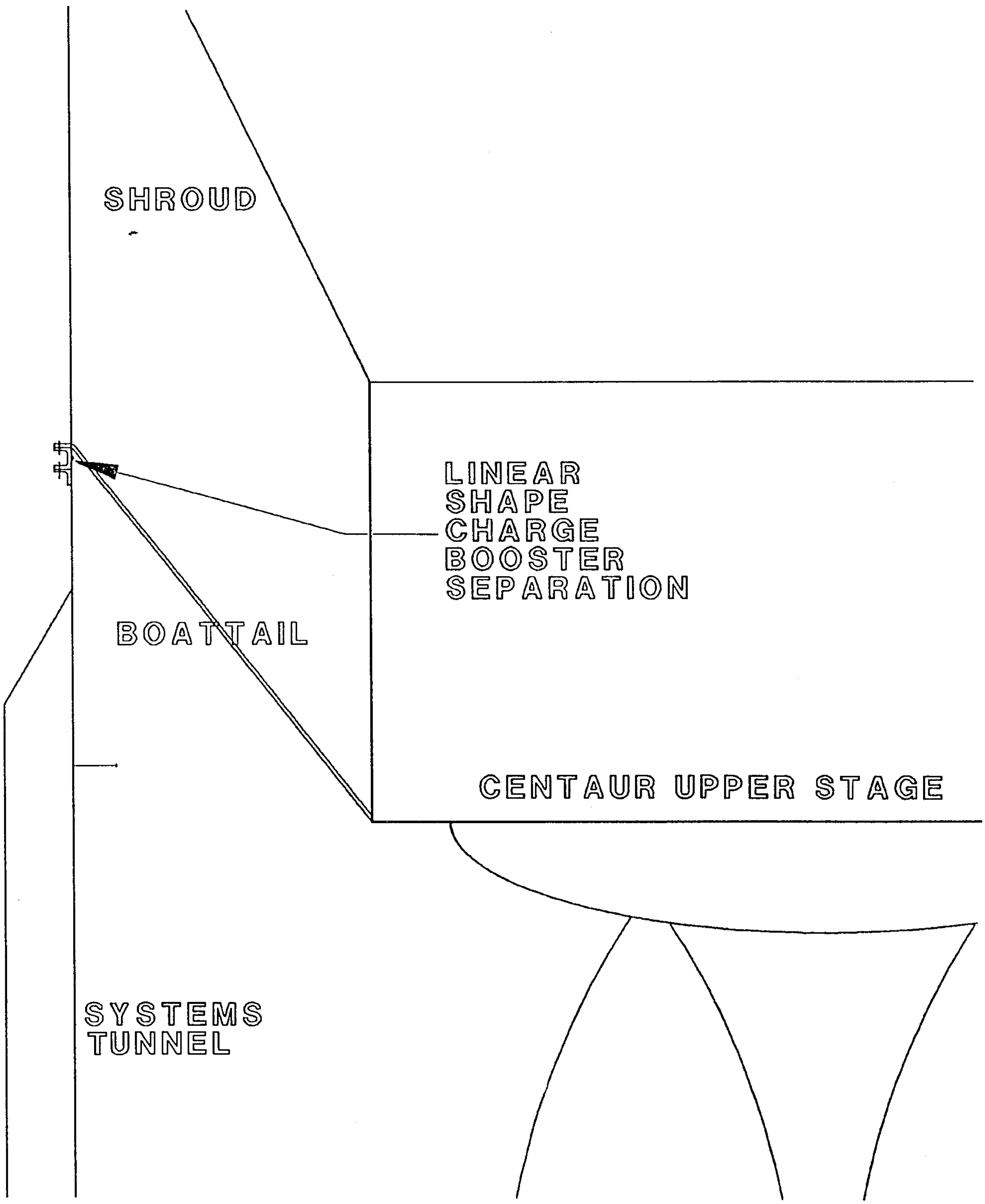


Figure 34. Upper stage separation system.

1. Summary of Performed Weight Trades.

Aluminum baseline versus Al-Li

Al-Li saves 2,952 lb of booster weight, yielding a 785-lb payload increase based on approximate weight savings using Al-Li.

2.0 safety factor baseline versus 1.4

1.4 saves 3,816 lb of booster weight, yielding a 1,015-lb payload increase using common elements.

6° Max-q loads baseline versus buildup/shutdown

Buildup/shutdown adds 1,628 lb of booster weight, reducing the payload by 433 lb.

60-in dry bay elements baseline versus 100-in baseline

The 100-in baseline does not significantly change the booster weight and seems to eliminate feedline interference.

Skin/stringer dry bay baseline versus isogrid

(1) Isogrid saves 1,060 lb of booster weight, yielding a 282-lb payload increase based on Max q loading.
(2) Isogrid adds 5,798 lb of booster weight, yielding a 1,542-lb payload decrease based on buildup/shutdown loading.

Skin/stringer tank baseline versus isogrid

Isogrid saves 3,876 lb of booster weight, yielding a 1,031-lb payload increase.

Common dry bay baseline versus individual dry bays

Individual dry bays save 1,150 lb of booster weight, yielding a 306-lb payload increase.

6° Max q baseline versus 1° Max q

1° saves 587 lb of booster weight, yielding a 156-lb payload increase. Ground wind becomes the design driver, buildup/shutdown loading not included.

Two weld thickness domes baseline versus four weld thickness domes

Individual domes save 530 lb of booster weight for a 141-lb payload increase

Two weld thickness domes baseline versus four individual domes with weld lands

Individual domes save 2,376 lb of booster weight, yielding a 632-lb payload increase

G. Propulsion Systems

1. Description. The schematic of the propulsion system is shown in figure 35. Feedline envelopes are shown in figure 36. The fuel sump and line envelope shown are worst case and will be refined when aft compartment clearances and RD-180 inlet locations are identified. Seventeen–inch feedlines for the oxidizer are assumed in order to utilize qualified shuttle hardware. The fuel feedlines are 12 inches in diameter. Propulsion system components other than the RD–180 engine are listed in table 3. It is assumed that two engine-supplied heat exchangers are available to pressurize the fuel and LO_2 tanks.

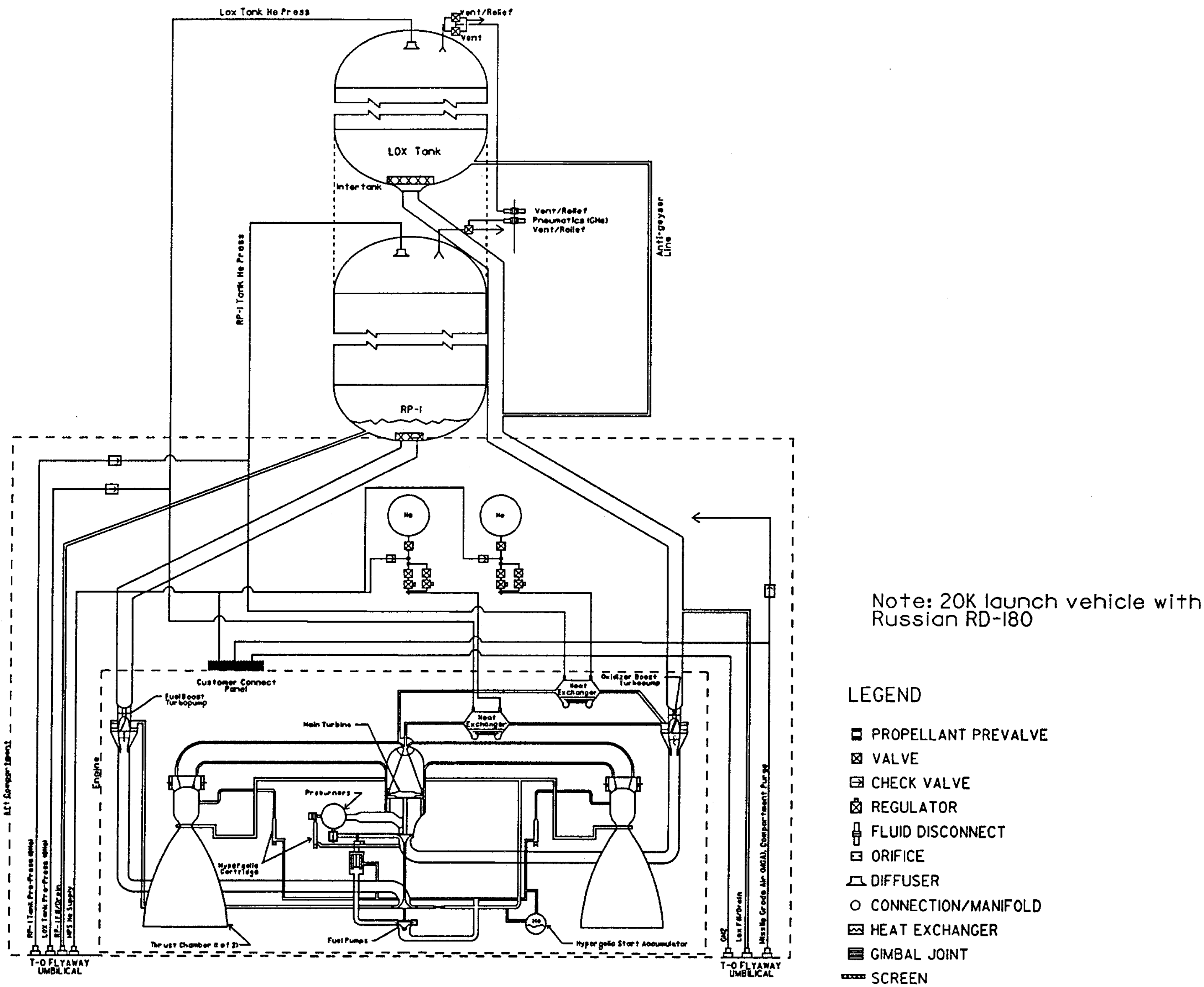


Figure 35. Propulsion system schematic.

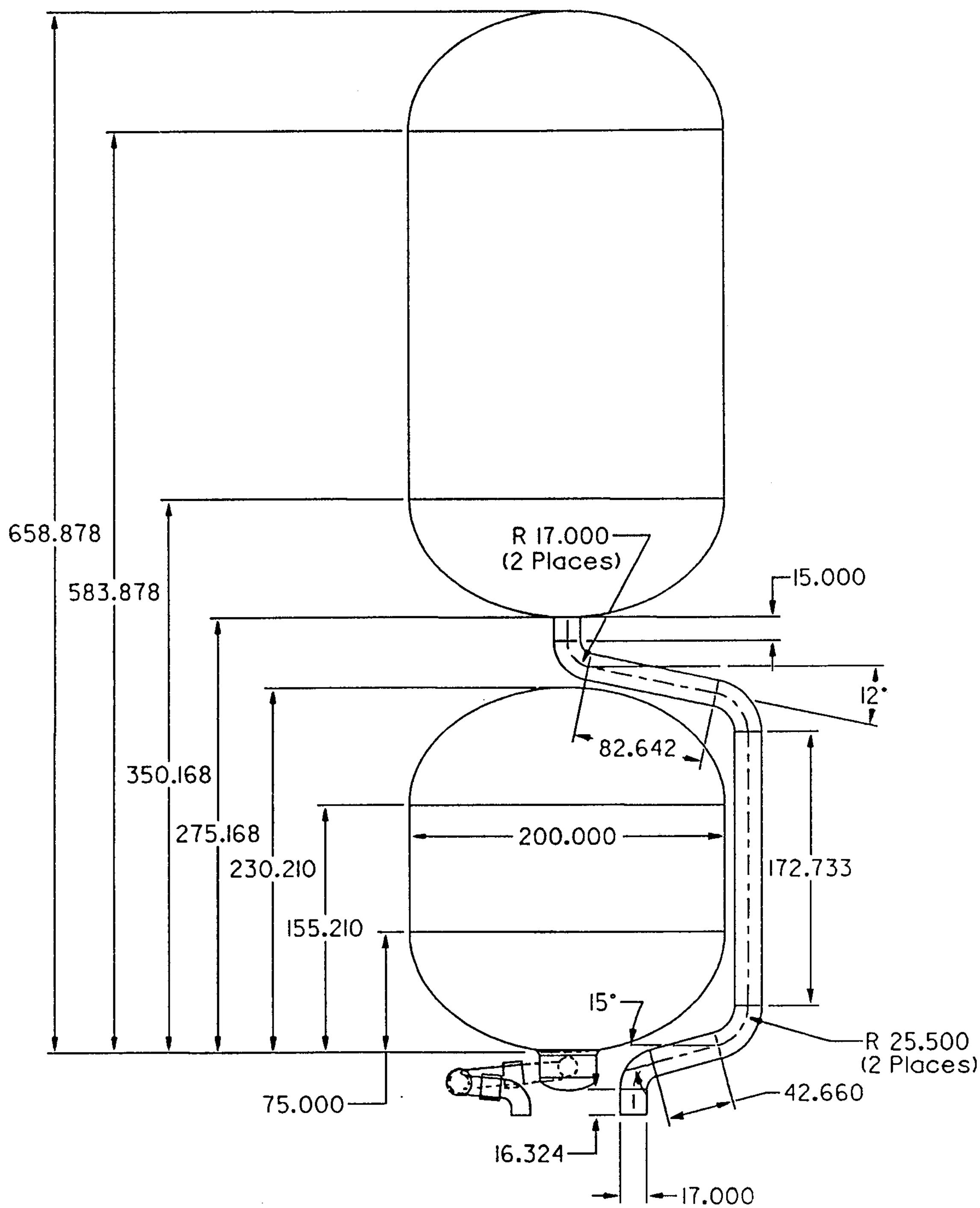


Figure 36. Feedline envelope.

Table 3. Propulsion system components.

Component	Quantity	Diameter (in)	Length (in)	New (N) or Existing (E) Design
Lox				
Press Ln and Manfd				
Diffuser	1	TBD	20.00	N
Ground Disconnect	1	1.00	N/A	E
Check Valve	1	1.00	N/A	E
Line	1	1.00	TBD	N
Vent/Relief				
Vent/Relief Valve	1	7.00	N/A	E
Ducting	1	7.00	TBD	N
Lox Tank				
Screen	1	TBD	N/A	N
Outlet	1	17.00	TBD	N
Feedline				
Duct	1	17.00	540.00	N
Elbow	4	17.00	N/A	N
Flange	9	17.00	N/A	E
BSTRA	6	17.00	N/A	E
Fill and Drain				
Fly-Away Disconnect	1	6.00	N/A	N
Fill/Drain Line	1	6.00	72.00	N
Antigeyser	1	4.00	TBD	N
RP-1				
Press Ln and Manfd				
Diffuser	1	TBD	12.00	N
Ground Disconnect	1	1.00	N/A	N
Check Valve	1	1.00	N/A	E
Line	1	1.00	TBD	N
Vent/Relief				
Valve	1	7.00	N/A	E
Duct	1	7.00	TBD	N
Valve Pneumatics (Ghe)				
Fluid Disconnect	1	TBD	TBD	E
Lines	TBD	TBD	TBD	N
RP-1 Tank				
Screen	1	TBD	N/A	N
Outlet	1	12.00	TBD	N
Feedline				
Duct	1	12.00	TBD	N
Sump	1	TBD	TBD	N
Elbow	4	12.00	N/A	N
Flange	10	12.00	N/A	N
Fill and Drain				
Fly-Away Disconnect	1	6.00	N/A	N
Fill/Drain Line	1	6.00	72.00	N
Helium Supply				
Fly-Away Disconnect	1	1.00	N/A	N
Bottles	2	TBD	N/A	E
Check Valve	2	1.00	N/A	E
Regulator	4	N/A	N/A	E
Valve	6	N/A	N/A	E
GN_2				
Fly-Away Disconnect	1	2.00	N/A	N
Missile Grade				
Air Comp. Purge				
Fly-Away Disconnect	1	2.00	N/A	N
Manifold	1	TBD	TBD	N

Functions supported by the propulsion system are:

- Rocket propulsion (RD–180 engine)
- Fuel (RP–1) fill, drain, and engine supply
- Oxidizer (LO_2) fill, drain, and engine supply
- Pressurization of the fuel and oxidizer tanks with heated helium
- Ground supply of pressurization He
- Prepressurization of the fuel and oxidizer tanks with ground-supplied He
- Ground supply of gaseous nitrogen to the engine
- Ground supply of missile grade air to the engine and to purge the aft compartment
- Fuel and oxidizer tank vent and relief
- Geyser avoidance during oxidizer tank fill.

No purge has been provided to the forward compartment or intertank areas. Missile-grade air has been supplied to the aft compartment to provide environmental conditioning for the avionics equipment.

2. Performance. The fuel tank is prepressurized with He to 19 lb/in^2 gauge and the lox tank to 29 lb/in^2 gauge to meet engine-start conditions. The LO_2 tank is pressurized with He to 35±1 lb/in^2 gauge and the fuel tank to 19±1 lb/in^2 gauge to meet engine run conditions. Estimated fuel and oxidizer tank ullage pressures versus time are shown in figures 37 and 38, respectively. The resulting maximum tank bottom pressures are 58.66 lb/in^2 gauge for the fuel tank and 40.67 lb/in^2 gauge for the LO_2 tank.

The propellant inventory is shown in table 4. Fuel and LO_2 mass, liquid height, and pump inlet net positive suction pressure (NPSP) versus time have been calculated and are available upon request.

RD–180 operating data assumed in the design are shown in table 5. NPSP requirements are assumed to vary linearly with power level.

3. Operations. The RD–180 servicing, prelaunch checkout, and launch commit criteria are not defined; these data are available for the RD–170. Because the RD–180 utilizes a significant RD–170 hardware and similarities in design of turbomachinery, the RD–180 requirements can be derived from the RD–170 data.

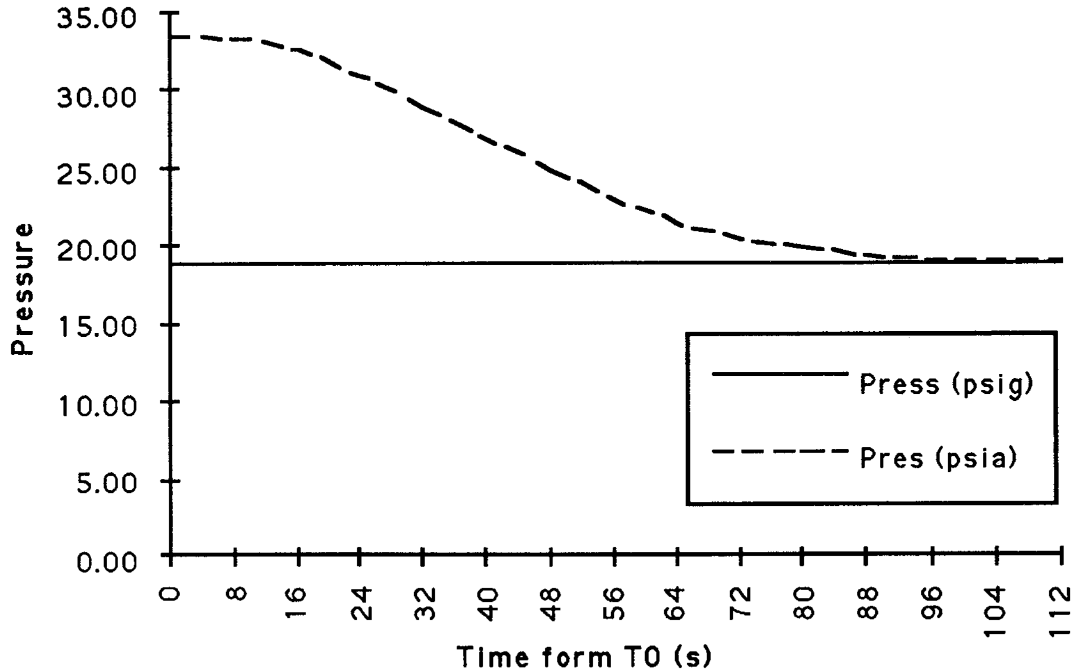


Figure 37. RP-1 fuel tank ullage pressure.

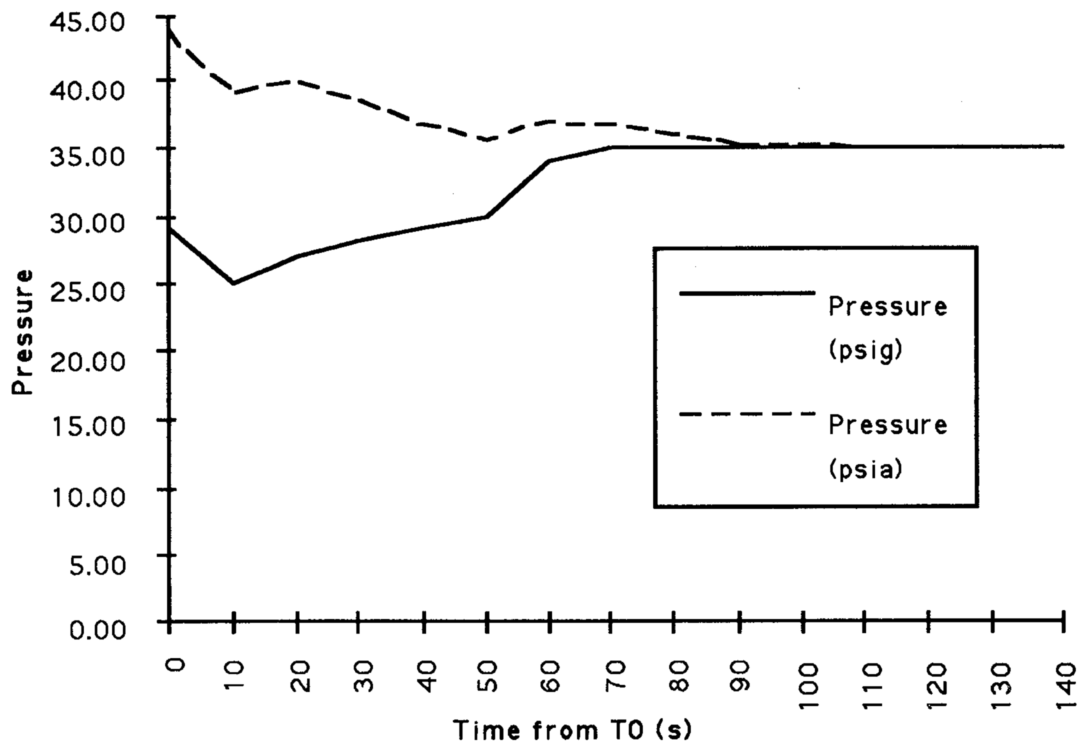


Figure 38. Lox tank ullage pressure.

Table 4. Propellant inventory.

20k Vehicle Propellant Inventory Summary			
Total Propellant Required (lbm)	590,555		
Mixture Ratio (O/F)	2.6		
Propellants	lox	kerosene	helium
Propellant Mass Requirement (lbm)	426,512	164,043	TBD
Stage Diameter (nominal inches)	200.00	200.00	
Unusable			
Tank Residuals (lbm)	0	300	TBD
Feedline and Pump Residuals (lbm)	1,200	500	
Booster Shutdown Consumption (lbm)	700	270	
Gas Residuals (lbm)	0	0	TBD
Prestart Boiloff (lbm)	100	0	
Start Consumption (lbm)	520	200	
Onboard at Lift-Off (lbm)	429,232	165,513	
Propellant Density (lbm/ft^3)	71.130	51	
Feed System Volume (ft^3)	75.00	15.00	
Mass in Feedline (lbm)	5,334.75	765	
Total Dome Volume (3/4 ellipse) ft^3	908.85	908.85	
Ullage Height (in)	20.00	12.00	
Liquid Volume in Dome (ft^3)	820.53	875.82	
Ullage Volume (ft^3)	88.33	33.04	
Percent Ullage by Volume (lbm, cryo-unpressurized)	1.46	1.01	
Equivilent Ullage Mass (lbm)	6,283	1,685	
Total Tank Capacity (lbm, cryo-unpressurized)	430,180	166,433	
Total Tank Volume (ft^3, cryo-unpressurized)	6,048	3,263	

Table 5. RD–180 estimated operating data.

Percent Power Level	Thrust (lbf)	ISP (s)	MR	Lox NPSP (lb/in^2 absolute)	RP–1 (lb/in^2 absolute)	mdot Lox (lb/s)	mdot RP–1 (lb/s)	Pc (lb/in^2 absolute)
40	367,917.06	335	2.42					
50	459,896.3	335	2.55	18.5	11.4	994.4	389	
74	680,646.6	335	2.69			1,524.6	567	
80	735,834.1	335	2.67			1,670	623.7	
100	919,792.65	335	2.6	38.4	17.1	1,928.3	762.83	3,560
102	938,188.5	335	2.78			2,024.4	779.1	

RD–180 estimated data scaled off of the RD–170 test data.
Engine has been uprated 5 percent to meet the design requirements of the 20k booster.

H. Avionics Systems

The proposed avionics system for the 20k booster is depicted in figure 39. It is assumed that the entire vehicle will be controlled from the Centaur upper stage. The only avionics connectivity between the upper stage and the booster will be via a 1553 data bus.

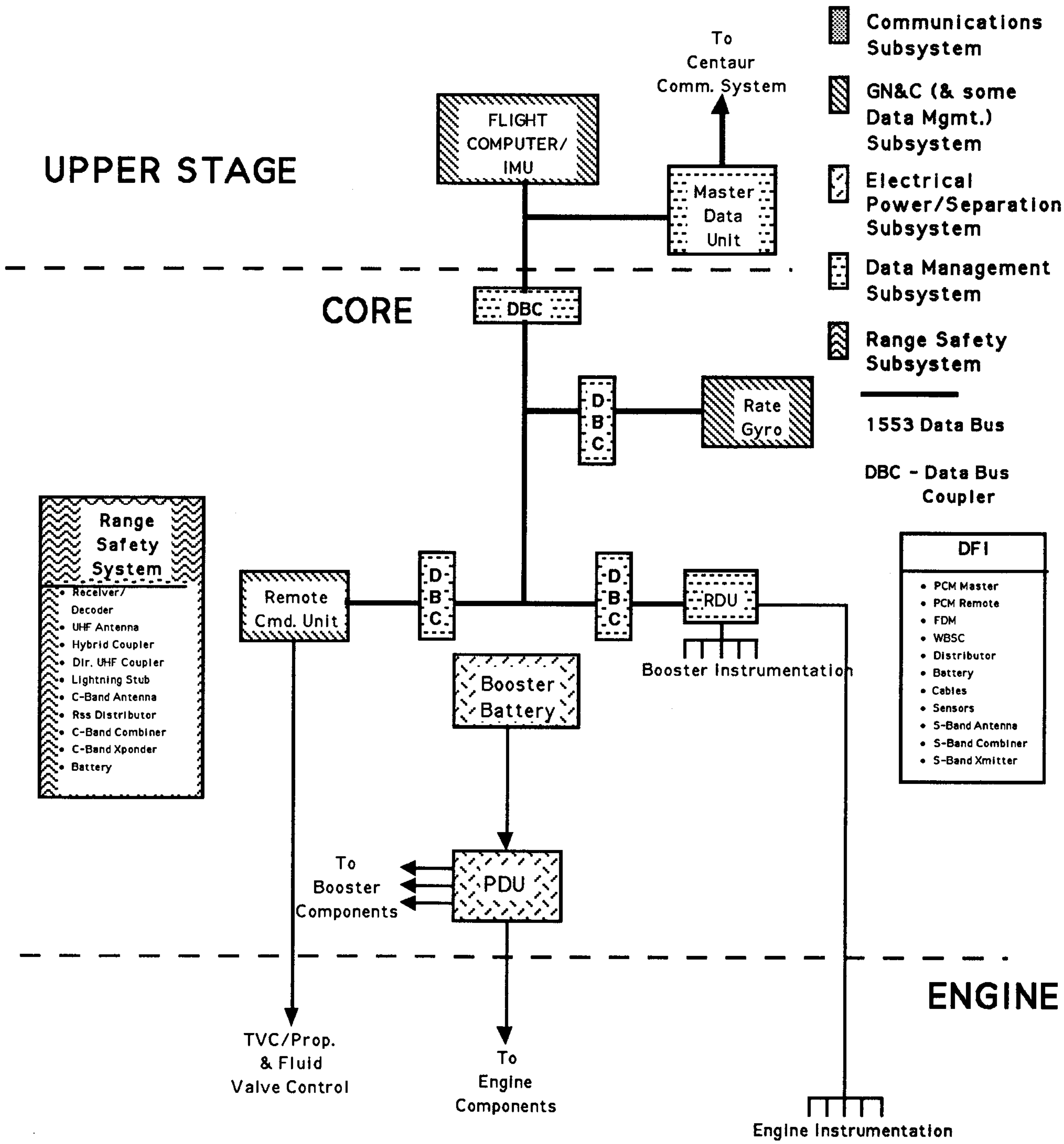


Figure 39. 20k booster avionics system.

Nonflight-critical data from the booster and engines will be collected by a remote data unit located on the booster stage and transferred directly to a master data unit on the Centaur upper stage for downlink via the Centaur S-band communications system. Flight-critical data from the booster will be collected by a remote command unit on the booster stage and routed directly to the Centaur flight computer/internal measurement unit (IMU). The flight computer/IMU subsequently passes the data to the master data unit for downlink via the Centaur S-band communications system. Commands from the Centaur flight computer/IMU for control of the booster engines, thrust vector, and fluid valves will be routed via the 1553 data bus through the remote command unit to the appropriate end items. It is assumed that a rate gyro will be necessary on the booster stage. Data from the rate gyro will be output to the flight computer/IMU via the 1553 data bus.

For the sake of design simplicity and weight, a separate power subsystem is envisioned for the booster stage. Power from the Ag-Zn battery power source will be distributed to the booster avionics components, as well as to the engine and TVC components via a power distribution unit. A separate battery will power the Centaur upper stage avionics components.

Table 6 depicts the avionics components needed for the 20k booster.

Table 6. Avionics components needed for the 20k booster.

Lab Div No.	Subsystem	Make (M) Buy (B)	Unit Weight (lb)	Unit Power (W)	Box Dimens.	Qty	Total Weight	Total Watts
	Communications (on Upper Stage)							
	Subsystem Support							
	Subtotal						0.0	0.0
	Data Management System							
EB31	Data Bus (incl. under cabling)						0.0	0.0
EB31	Remote Data Unit	B	15.0	20	9x9x5 in	1	15.0	20.0
EB31	Data Bus Coupler	B	0.4	1	2x2x2 in	4	1.6	4.0
	Subsystem Support							
	Subtotal						16.6	24.0
	Instrumentation							
EB21	Sensors (stand alone xducers)	B	0.3	10	N/A	50	12.5	500.0
	Subtotal						12.5	500.0
	Electrical Power							
EB71	Booster Stage Battery	B	40.0	N/A	14x9x8 in	1	40.0	N/A
EB11	Power Distribution Units	M	65.0	50	16x30x8 in	1	65.0	50.0
EB11	Cabling	M	750.0	30	N/A	100	750.0	30.0
	Subsystem Support							
	Subtotal						855.0	80.0
	GN&C							
EB21	Rate Gyro	B	10.0	15	5 in dia x 7 in height	1	10.0	15.0
EB31	Command Units (valve drive elec.)	B	20.0	25	11x5x3 in	1	20.0	25.0
	Subsystem Support							
	Subtotal						30.0	40.0

Table 6. Avionics components needed for the 20k booster (continued).

Lab Div No.	SUBSYSTEM	Make (M) Buy (B)	Unit Weight (lb)	Unit Power (W)	Box Dimens.	Qty	Total Weight	Total Watts
	Range Safety							
EB51	Receiver/Decoder	B	5.5	15		2	11.0	30.0
EB51	UHF Antenna	B	7.7	0		2	15.4	0.0
EB51	Hybrid UHF Coupler	B	2.0	0		1	2.0	0.0
EB51	Direct UHF Coupler	B	1.3	0		1	1.3	0.0
EB51	Light. Prot. Stub	B	1.0	0		3	3.0	0.0
EB51	C-Band Antenna	B	0.1	0		2	0.2	0.0
EB11	RSS Distributor	B	40.0	15	16x30x8 in	1	40.0	15.0
EB51	C-Band Combiner	B	0.2	0		1	0.2	0.0
EB51	C-Band Xpndr	B	2.3	22.4		1	2.3	22.4
EB71	RSS Batteries	B	15.0	0	12x9x8 in	1	15.0	0.0
	Subsystem Support							
	Subtotal						90.4	67.4
	Separation							
EB11	Standard Pic Unit (in RSS Dstr.)	B				10	0.0	0.0
	Subtotal						0.0	0.0
	Component Total						1,004.5	711.4
	Integration Factor							
	Avionics Subsystem Total							
	Booster Factory C/O System							
	Booster Simulator							
	Vehicle Dynamic Simulator							
	Development Flight Instrum.						945	
	Flight and Ground Software							
	DFI							
EB31	PCM Master		15.0			1	15	
EB31	PCM Remote		13.0			3	39	
EB31	FDM		13.0			2	26	
EB51	WBSC		15.0			2	30	
EB11	Distributor		65.0			1	65	
EB71	Battery		15.0			1	15	
EB11	Cables		452.0			1	452	
EB21	Sensors		0.5			600	300	
EB51	S-Band Antenna		1.0			1	1	
EB51	S-Band Combiner		1.0			1	1	
EB51	5-W S-Band Xmtr		1.0			1	1	
	Subtotal						945.0	
	DFI Ground C/O System							
	Total							

I. Mass Properties

The booster weights calculated by the design team are presented in this section. The Centaur upper stage weights were obtained from General Dynamics Space Systems Company and the payload shroud from McDonnell Douglas Aerospace Company. Modifications mentioned in earlier sections were made by the design teams. Table 7 summarizes the booster dry weights. Table 8 lists the detailed vehicle mass properties, weight, center of gravities, and moments of inertia for various trajectory time slices as required by the control and loads engineers.

Table 7. 20k booster weight summary (lb).

Interstage and Forward Skirt	4,179
Lox Tank	8,687
Intertank	3,103
RP Tank	5,590
Aft Skirt	2,027
Aerodynamic Control Fins	2,035
Thrust Structure	410
Launch Holddown	150
Aft Structure and System Tunnel	1,439
Lox Feed System	2,738
RP Feed System	1,650
Base Heat Shield	780
RD–180 Engine	11,675
Range Safety (Str. and Ord.)	168
Avionics	1,017
Contingency (10 percent)	4,565
Total Dry Weight	50,213
Residuals	5,526
Total Booster Burnout Weight	55,739
Usable Propellants (588,312 lb)	

J. Main Engine Data

A single RD–180 staged combustion cycle lox/RP booster engine was baselined for this study. The RD–180 engine is derived from the mature, flight certified RD–170/RD–171 engine (two nozzles instead of four and approximately one-half the thrust of the RD–170). The RD–170 is used in the NPO Energia booster, and the RD–171 is used in the Zenit booster. The engines differ only in the gimbal arrangement. Only the main turbopump and boost pumps will require development The remaining engine components are common with the RD–170 and can be used without change. A 5-percent thrust increase above the nominal (see enclosed engine characteristics) is required to meet the payload weight requirement. This is the maximum allowable thrust increase because of combustion chamber limits.

Table 8. In-house 20k launch vehicle mass properties.

PD24 | In-House 20k Launch Vehicle 200-Inch Diameter 1 RD-180 Engine | August 12,1993

Mass Properties Lift-Off

Description	Launch Weight (lb)	XCG (ft)	YCG (ft)	ZCG (ft)	Inertias Referenced From Individual Item CG: IXX Slugs (ft^2)	IYY Slugs (ft^2)	IZZ Slugs (ft^2)
Payload	25,035	130.30	0.00	0.00	21,306	36,591	36,591
Shroud Nose	2,028	123.50	0.00	0.00	2,510	6,409	6,409
Shroud Cylinder	4,420	144.10	0.00	0.00	9,251	12,726	12,726
Shroud/Payload Supp	2,752	160.00	0.00	0.00	4,037	2,197	2,197
Centaur Fwd Str	1,842	143.40	0.00	0.00	2,173	1,351	1,351
LH_2 Tank + Sys	1,225	148.70	0.00	0.00	2,016	1,465	1,465
Lox Tank + Sys	1,104	159.40	0.00	0.00	558	558	558
Aft Str + Prop Sys	2,139	164.30	0.00	0.00	766	451	451
RL10-A4 Engine	730	166.40	0.00	0.00	116	85	154
Centaur Usable Lox	38,692	159.40	0.00	0.00	11,433	11,433	11,433
Centaur Usable LH_2	7,035	148.70	0.00	0.00	5,589	5,419	5,419
Booster Fwd Str	4,597	166.80	0.00	0.00	9,622	8,851	8,851
Lox Tank + Sys	10,929	190.30	0.00	0.00	22,875	39,884	39,884
Intertank + Sys	4,417	207.70	0.00	0.00	9,245	7,197	7,197
RP Tank + Sys	8,705	221.30	0.00	0.00	18,220	15,936	15,936
Aft Skirt + Prop Sys	12,009	233.70	0.00	0.00	12,950	15,465	15,465
Aero Fins	2,239	239.60	0.00	0.00	8,756	4,797	4,797
RD-180 Engine	12,843	243.60	0.00	0.00	5,988	6,820	10,412
Usable Lox (L.O.)	424,892	189.90	0.00	0.00	454,904	1,121,380	1,121,380
Usable RP (L.O.)	163,420	221.00	0.00	0.00	174,963	191,819	191,819
Totals Lift-Off	731,053	194.05	0.00	0.00	777,277	13,610,566	13,614,229

Table 8. In-house 20k launch vehicle mass properties (continued).

PD24 — In-House 20k Launch Vehicle 200-Inch Diameter 1 RD-180 Engine Mass Properties Max Q — August 12,1993

					Inertias Referenced From Individual Item CG		
Description	Launch Weight (lb)	XCG (ft)	YCG (ft)	ZCG (ft)	IXX Slugs (ft^2)	IYY Slugs (ft^2)	IZZ Slugs (ft^2)
Payload	25,035	130.30	0.00	0.00	21,306	36,591	36,591
Shroud Nose	2,028	123.50	0.00	0.00	2,510	6,409	6,409
Shroud Cylinder	4,420	144.10	0.00	0.00	9,251	12,726	12,726
Shroud/Payload Supp	2,752	160.00	0.00	0.00	4,037	2,197	2,197
Centaur Fwd Str	1,842	143.40	0.00	0.00	2,173	1,351	1,351
LH_2 Tank + Sys	1,225	148.70	0.00	0.00	2,016	1,465	1,465
Lox Tank + Sys	1,104	159.40	0.00	0.00	558	558	558
Aft Str + Prop Sys	2,139	164.30	0.00	0.00	766	451	451
RL10-A4 Engine	730	166.40	0.00	0.00	116	85	154
Centaur Usable Lox	38,692	159.40	0.00	0.00	11,433	11,433	11,433
Centaur Usable LH_2	7,035	148.70	0.00	0.00	5,589	5,419	5,419
Booster Fwd Str	4,597	166.80	0.00	0.00	9,622	8,851	8,851
Lox Tank + Sys	10,929	190.30	0.00	0.00	22,875	39,884	39,884
Intertank + Sys	4,417	207.70	0.00	0.00	9,245	7,197	7,197
RP Tank + Sys	8,705	221.30	0.00	0.00	18,220	15,936	15,936
Aft Skirt + Prop Sys	12,009	233.70	0.00	0.00	12,950	15,465	15,465
Aero Fins	2,239	239.60	0.00	0.00	8,756	4,797	4,797
RD-180 Engine	12,843	243.60	0.00	0.00	5,988	6,820	10,412
Usable Lox (Max Q)	248,697	195.70	0.00	0.00	266,264	312,141	312,141
Usable RP (Max Q)	95,653	224.50	0.00	0.00	102,409	72,084	72,084
Totals Max Q	487,091	195.46	0.00	0.00	777,277	11,512,729	11,516,392

Table 8. In-house 20k launch vehicle mass properties (continued).

PD24

In-House 20k Launch Vehicle 200-Inch Diameter 1 RD-180 Engine
Mass Properties 155 s

August 12,1993

Description	Launch Weight (lb)	XCG (ft)	YCG (ft)	ZCG (ft)	Inertias Referenced From Individual Item CG		
					IXX Slugs (ft^2)	IYY Slugs (ft^2)	IZZ Slugs (ft^2)
Payload	25,035	130.30	0.00	0.00	21,306	36,591	36,591
Shroud Nose	2,028	123.50	0.00	0.00	2,510	6,409	6,409
Shroud Cylinder	4,420	144.10	0.00	0.00	9,251	12,726	12,726
Shroud/Payload Supp	2,752	160.00	0.00	0.00	4,037	2,197	2,197
Centaur Fwd Str	1,842	143.40	0.00	0.00	2,173	1,351	1,351
LH_2 Tank + Sys	1,225	148.70	0.00	0.00	2,016	1,465	1,465
Lox Tank + Sys	1,104	159.40	0.00	0.00	558	558	558
Aft Str + Prop Sys	2,139	164.30	0.00	0.00	766	451	451
RL10-A4 Engine	730	166.40	0.00	0.00	116	85	154
Centaur Usable Lox	38,692	159.40	0.00	0.00	11,433	11,433	11,433
Centaur Usable LH_2	7,035	148.70	0.00	0.00	5,589	5,419	5,419
Booster Fwd Str	4,597	166.80	0.00	0.00	9,622	8,851	8,851
Lox Tank + Sys	10,929	190.30	0.00	0.00	22,875	39,884	39,884
Intertank + Sys	4,417	207.70	0.00	0.00	9,245	7,197	7,197
RP Tank + Sys	8,705	221.30	0.00	0.00	18,220	15,936	15,936
Aft Skirt + Prop Sys	12,009	233.70	0.00	0.00	12,950	15,465	15,465
Aero Fins	2,239	239.60	0.00	0.00	8,756	4,797	4,797
RD-180 Engine	12,843	243.60	0.00	0.00	5,988	6,820	10,412
Usable Lox (155 s)	110,982	200.20	0.00	0.00	118,821	75,152	75,152
Usable RP (155 s)	42,686	226.80	0.00	0.00	45,701	24,709	24,709
Totals 155 s	296,409	192.17	0.00	0.00	311,932	10,001,425	10,005,088

Table 8. In-house 20k launch vehicle mass properties (continued).

PD24 | In-House 20k Launch Vehicle 200-Inch Diameter 1 RD-180 Engine | August 12,1993

Mass Properties Booster B.O.

Description	Launch Weight (lb)	XCG (ft)	YCG (ft)	ZCG (ft)	Inertias Referenced From Individual Item CG: IXX Slugs (ft^2)	IYY Slugs (ft^2)	IZZ Slugs (ft^2)
Payload	25,035	130.30	0.00	0.00	21,306	36,591	36,591
Shroud Nose	2,028	123.50	0.00	0.00	2,510	6,409	6,409
Shroud Cylinder	4,420	144.10	0.00	0.00	9,251	12,726	12,726
Shroud/Payload Supp	2,752	160.00	0.00	0.00	4,037	2,197	2,197
Centaur Fwd Str	1,842	143.40	0.00	0.00	2,173	1,351	1,351
LH_2 Tank + Sys	1,225	148.70	0.00	0.00	2,016	1,465	1,465
Lox Tank + Sys	1,104	159.40	0.00	0.00	558	558	558
Aft Str + Prop Sys	2,139	164.30	0.00	0.00	766	451	451
RL10-A4 Engine	730	166.40	0.00	0.00	116	85	154
Centaur Usable Lox	38,692	159.40	0.00	0.00	11,433	11,433	11,433
Centaur Usable LH_2	7,035	148.70	0.00	0.00	5,589	5,419	5,419
Booster Fwd Str	4,597	166.80	0.00	0.00	9,622	8,851	8,851
Lox Tank + Sys	10,929	190.30	0.00	0.00	22,875	39,884	39,884
Intertank + Sys	4,417	207.70	0.00	0.00	9,245	7,197	7,197
RP Tank + Sys	8,705	221.30	0.00	0.00	18,220	15,936	15,936
Aft Skirt + Prop Sys	12,009	233.70	0.00	0.00	12,950	15,465	15,465
Aero Fins	2,239	239.60	0.00	0.00	8,756	4,797	4,797
RD-180 Engine	12,843	243.60	0.00	0.00	5,988	6,820	10,412
Totals Booster B.O.	142,741	175.57	0.00	0.00	147,410	6,864,834	6,868,497

Table 8. In-house 20k launch vehicle mass properties (continued).

PD24 In-House 20k Launch Vehicle 200-Inch Diameter 1 RD-180 Engine August 12,1993
Mass Properties Booster Booster Sep

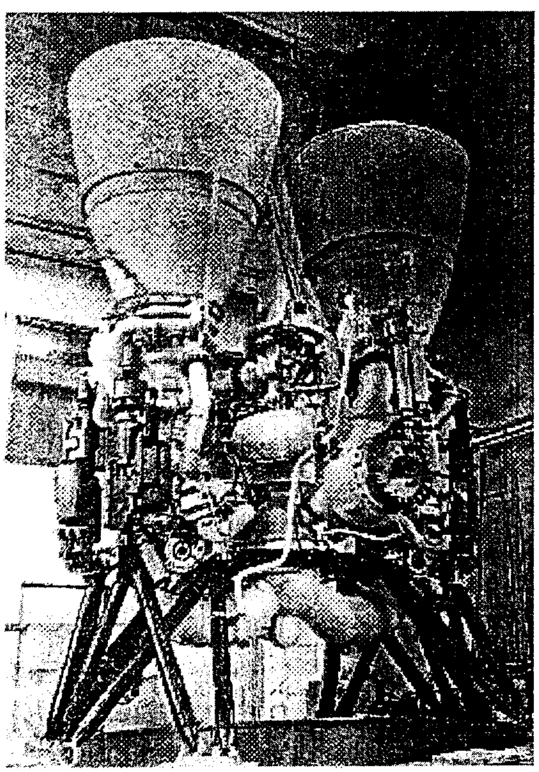
Description	Launch Weight (lb)	XCG (ft)	YCG (ft)	ZCG (ft)	Inertias Referenced From Individual Item CG IXX Slugs (ft^2)	IYY Slugs (ft^2)	IZZ Slugs (ft^2)
Payload	25,035	130.30	0.00	0.00	21,306	36,591	36,591
Shroud Nose	2,028	123.50	0.00	0.00	2,510	6,409	6,409
Shroud Cylinder	4,420	144.10	0.00	0.00	9,251	12,726	12,726
Shroud/Payload Supp	2,752	160.00	0.00	0.00	4,037	2,197	2,197
Centaur Fwd Str	1,842	143.40	0.00	0.00	2,173	1,351	1,351
LH_2 Tank + Sys	1,225	148.70	0.00	0.00	2,016	1,465	1,465
Lox Tank + Sys	1,104	159.40	0.00	0.00	558	558	558
Aft Str + Prop Sys	2,139	164.30	0.00	0.00	766	451	451
RL10-A4 Engine	730	166.40	0.00	0.00	116	85	154
Centaur Usable Lox	38,692	159.40	0.00	0.00	11,433	11,433	11,433
Centaur Usable LH_2	7,035	148.70	0.00	0.00	5,589	5,419	5,419
Totals Booster Sep	87,002	148.26	0.00	0.00	59,754	561,977	562,046

Currently, NPO Energomash is upgrading the RD–171 to the RD–172 which incorporates this thrust increase. The RD–180 engine/propulsion system includes the following: (1) thrust structure, (2) pneumatic system, (3) gimbal actuators, (4) TVC, (5) lox and fuel boost pump, (6) helium bottles, (7) heat shield, (8) valves, (9) nozzles, and (10) propellant ducting. The nozzles can gimbal independently (for roll control). Start, throttle, and shutdown commands are received from the vehicle and are implemented via an engine-mounted pneumatic control system. The preburner and chamber shutdown is oxidizer rich, leaving minimum kerosene residue.

Figure 40 displays the proposed RD–180 engine operational characteristics. Figure 41 is a sketch of the engine propellant flow diagram. The RD–170 schematics were used for reference in deriving these data.

One RD–180 engine can be delivered at a cost of $6 to $8M (acceptance tested and calibrated).

RD-180 ENGINE CONFIGURATION



• Nominal thrust (lb)	
Sea level	827,000
Vacuum	900,000
• Specific impulse (sec)	
Sea level	309
Vacuum	337
• Chamber pressure (psia)	3,560
• Mixture ratio	2.6
• Overall dimensions	
Length (inches)	157
Diameter (inches)	118
• Throttle range (%)	50
• Dry weight (lb)	11,675
• Wet weight (lb)	12,690

Figure 40. Characteristics.

RD-180 ENGINE CONFIGURATION

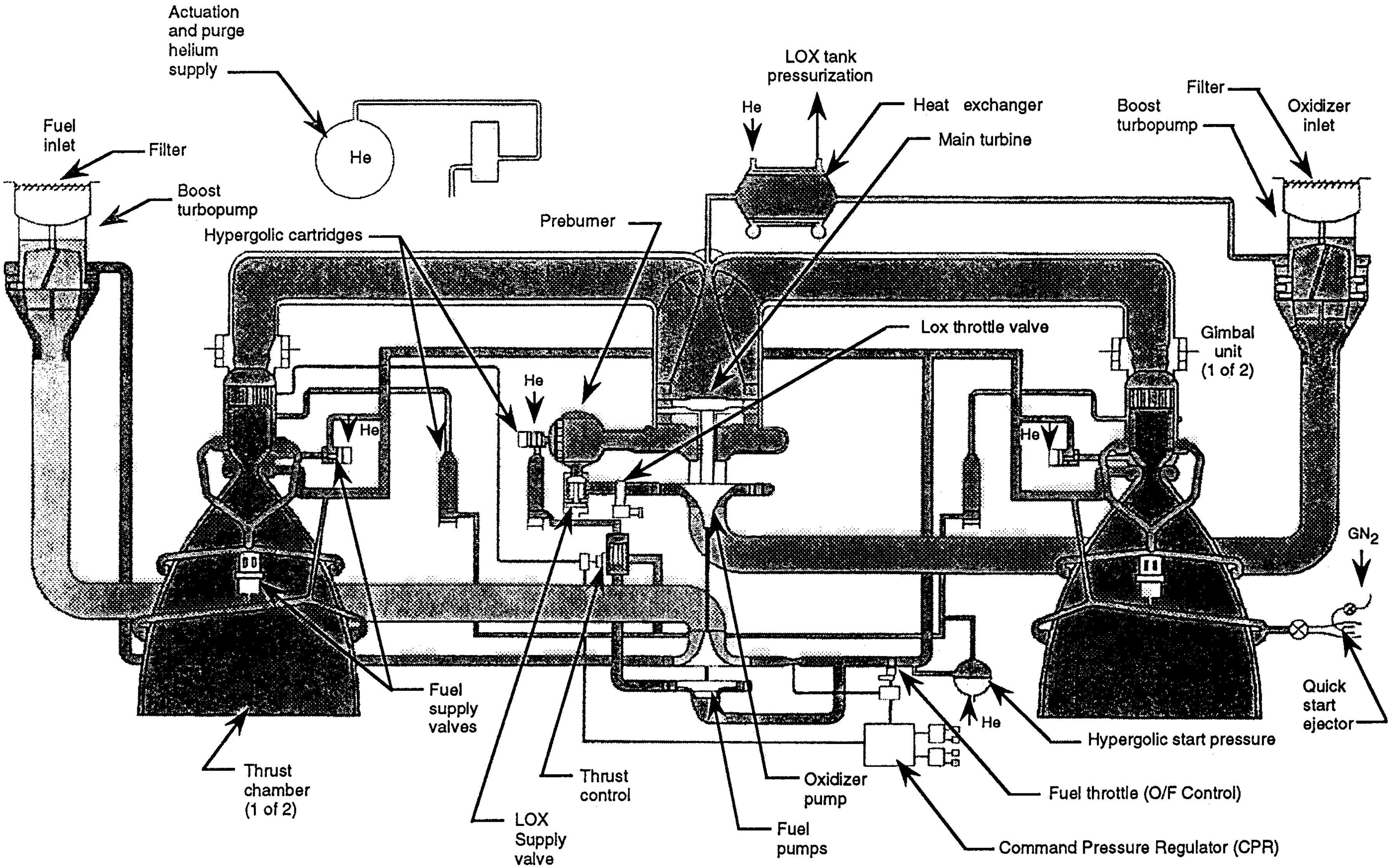


Figure 41. Propellant flow diagram.

III. FABRICATION, INTEGRATION, AND VERIFICATION

A. Manufacturing and Assembly

This booster is to be designed, built and ready to launch in 40 months from authority to proceed (ATP). Three manufacturing options were investigated:

(1) Procure all major assemblies and assemble only the booster. This approach would procure the forward and aft skirts, intertank structures, fuel and oxidizer tanks, thrust adapters, feedlines, etc.

(2) Procure the major subassemblies and assemble into major assemblies. This requires procurement of fuel and oxidizer tank barrel sections and domes, the forward and aft skirt sections to be assembled, the intertank sections, thrust adapters, feedlines, etc. This approach requires large weld fixtures to assemble the tanks.

(3) Procure the piece parts and perform most major subassembly and assembly work in-house. This approach procures the tank dome gores, barrel panels, finished rings, sheet metal parts for the skirts and intertank, as well as the thrust adapters, feedlines, etc. This option uses the similar technique to assemble as the present shuttle ET.

To meet the schedule with minimum risk, it was decided to use option 3. It is estimated that 80,000 to 100,000 ft^2 of floor space would be required for manufacturing and checkout. Sufficient floor space is available at MSFC, but is presently assigned to other projects, and, therefore, not readily available.

The major hard tooling required is two quarter-dome trim, alignment, and weld fixtures; one half-dome trim and weld fixture; one full-dome trim and weld fixture; one trim, alignment, and weld fixture for welding the tank barrel sections' skin sections; one trim, alignment, and weld fixture for joining the barrel sections together; the intermediate rings; and the tank domes. An additional fixture is required for manufacturing the skin and stringer intertank and skirt assemblies. These fixtures do not have to be in the same building, but should be in proximity for easy transport.

A hydrostatic test facility may be required for leak testing each tank prior to stage assembly. A candidate location is building 4707. This would be a suitable location to perform internal cleaning of each tank. The final assembly requires a high-bay area with sufficient floor space and overhead cranes to assemble the major subassemblies into the finished product.

An exercise was conducted to locate the booster manufacturing and assembly in the same location. The candidate building selected was building 4705. The tank dome trim and weld fixtures could be located along the east side of the low-bay area. The domes could be completed, including the access holes and the propellant outlet(s). The barrel section trim and weld and tank assembly fixtures could be located in the east high bay. As each tank is completed, it is transferred to building 4707 for cleaning and leak testing. When testing is complete, the tanks are transferred back to the west high-bay aisle of building 4705 for assembly with the interstage, intertank, and thrust structure/boattail.

The propellant feedlines, valves, regulators, pressurization tubing, and other propulsion system hardware would be purchased from available vendors. Installation brackets and fixtures would be supplied by MSFC. The RD–180 engine system would be purchased from Pratt and Whitney, certified and ready to install.

The checked-out avionics boxes will be supplied for assembly from the Astrionics Laboratory for installation and system verification. It is assumed that each component will be delivered individually verified and ready for installation. The brackets and wiring harnesses will be manufactured and installed in building 4705.

The above fast-track approach will require considerable capital investment for the hard tooling fixtures. The required hard tooling would be purchased through streamlined design and fabricate procurement bid packages. Any test fixtures required would be designed and built in-house and the shop overload would be procured by a basic ordering agreement (BOA). For a project of this size, and the lead times available, the procurement process would require streamlining. The shop would have to have direct purchase order authority.

It is planned to turn the manufacturing (tooling and labor) of this booster over to industry after the fifth or seventh vehicle. These numbers are somewhat arbitrary, but should be reasonable in light of the projected flight rate and the transition time.

1. Production Enhancements. There are opportunities for manufacturing improvement as this stage is transitioned to commercial manufacturing. The most obvious is the spin forming of the tank domes. The methodology has been developed for smaller diameters and could be economically adapted to this size if the production rate warrants. Extrusion of the tank barrel sidewall sections to near net shape would eliminate some of the intermediate welding to obtain the required lengths, but at a weight cost because tapered extrusions of this type are not yet developed.

B. Test Requirements

The design concept of the 20k booster was to minimize DDT&E and postmanufacturing testing. The previous design and manufacturing sections supply supporting data.

1. Structural Testing. Preliminary analyses show that the booster can be designed with a structural safety factor of 2 and meet the payload requirement. If this holds true for the final design, a static structural load test will not be required. However, if the final booster design cannot be accomplished with the safety factor of 2, a static structural test may be required. Static structural testing criteria will be established at that time.

A modal survey test normally performed on structures such as this is not required. Presently, the design personnel are confident that the dynamic characteristics of the booster can be analytically predicted. If it is decided later that the modal survey test must be performed, the test requirements will be established at that time. There is a possibility that this type test could be performed in conjunction with the following propulsion test setup.

2. Propulsion System Test Requirements. Main propulsion system (MPS) and engine system test requirements for this 20k booster are based on the requirement that the RD–180 engine will be delivered ready to fly. (This implies no separate in-house engine development, verification, certification, or calibration testing required.) Demonstrated reliability is not considered to be a test requirement. It is proposed to use the first manufactured booster stage to perform the propulsion system testing. Following testing, this stage would be refurbished as a flight stage, possibly as a development flight instrumented (DFI) vehicle.

MPS and engine system test requirements are:

(a) Six cold flow tests using initial feed system and simulated tank bottom for the purpose of:

- Evaluating lox conditioning
- Developing fill and drain procedures
- Initial contingency procedure planning.

(b) Six terminal drain tests utilizing initial tanks (or tank bottoms) and feedline (engine system not required) for the purpose of:

- Developing shutdown sequence
- Verifying depletion sensor locations.

(c) Twelve MPS tests (hot firing required) of initial tank set and feed system set (engine system required) for the purpose of:

- Tank fill and drain verification
- Propulsion start/shutdown sequencing
- Contingency procedures planning
- System verification, and propulsion system qualification.

Assembly, test preparation, and servicing procedures will be evaluated during all of the above tests. Subsystem level test requirements (i.e., those other than MPS/engine system) are TBD.

3. Alternate Propulsion System Testing. An alternate approach to the propulsion system testing is to verify the piping/valve system on the MSFC cold flow facility and to perform the remaining tests on the launch pad, including a short duration captive firing. The impact of this option on the launch pad design would have to be traded against cost, schedule, and risk of the approach in section III.B.2. The final approach is TBD based on final requirements, maturity, and certification of the delivered engine.

4. Avionics Testing. The avionics testing will be basically booster stand-alone testing using an upper stage-supplied functional simulator emulating the upper stage and payload fairing and any electrical pass-throughs requiring continunity verifying, at least one launch pad simulator, and other government-supplied equipment (GSE) as required. Actuators and valves are tested as required. A similar booster/ground interface simulator would be supplied to the upper stage contractor. This method will allow both stages to be tested at their respective factories and to be ready for final mating and integration at the launch site.

C. Facilities Requirements

MSFC has facilities that are potentially available to manufacture, test, and verify the 20k booster. Candidates are the hydrostatic test tower in building 4707; the shop floor and electrical engineering shop in building 4705; the first floor shop area in building 4655; the first floor in building 4754; and a section of building 4650. Test and storage facilities will have to be defined as requirements evolve.

Depending on the final definition of the propulsion system test requirements, a vertical test stand with sufficient structural capability to hold and service the booster may be required. Possible candidates are the advanced engine test facility or the hydrocarbon engine test facility located in the west test area of MSFC.

IV. OPERATIONS REQUIREMENTS

A. Assembly and Integration

The integrate-transfer-launch process will be implemented for this system. The individual stages will be delivered completed, including testing performed at the manufacturing plants, to the integration facility at the launch site. Stages should have inert pressurization in order to keep them from breathing during the temperature swings. The stages will arrive with the avionics in a safe mode. The individual stages and interstage will be assembled on the launch transporter. This transporter will provide all propellant, electrical, air conditioning, and ground service connections between the launch pad and the vehicle. Each individual stages' interfaces have been verified at its factory using standard simulators. The end-to-end vehicle avionics test and the verification of the payload service pass-through wiring are the only post integration tests required.

For this vehicle configuration, it will be possible to integrate the Centaur and the payload separately from the booster, enclose it within the payload fairing, complete all upper stage/payload verifications, and then transport this stack to be mated with the booster. This allows parallel operations to occur and should reduce preflight process time. Once the total vehicle verification is completed, the vehicle system is ready to transport to the pad for launch. The design goal from rollout to launch is to be less than 24 clock hours.

The transporter serves as the vehicle launch platform and service tower. After the transporter is harddown to the pad, fueling and final prelaunch checkout can be performed. The terminal count, ignition, and ground holddown release will be performed by the prescribed method dictated by the Centaur requirements in conjunction with the ground system monitoring. Interrupt capabilities from both the ground and/or payload will be available to analyze/correct sensed anomalies.

An avionics upgrade will allow the onboard avionics to perform the final count, engine ignition, and lift-off signals. This is not presently part of the standard Centaur flight avionics operation.

B. Flight Operations

The flight will be automously controlled by the Centaur avionics system, including the terminal stage disposal maneuver. Preflight analyses will be streamlined by the requirement that the vehicle be able to accommodate payload weight/center-of-gravity, natural frequency combinations within a predescribed volume as described in the vehicle-to-payload interface control document (ICD). The vehicle will have a number of standard flight profiles to standard orbits. Special missions will be accommodated at a TBD extra cost.

V. SYSTEM MANAGEMENT

A. Program Risk Analysis

The primary objective of the risk assessment is to identify risks as early as possible so that work-around plans may be implemented before the potential risks become "program stoppers." The payload shroud and upper stage can be purchased from existing vendors who have established data bases with verified performance parameters, cost projections, and scheduling processes. Large uncertainties do not exist for these items due to ongoing production capabilities and facilities. Minor uncertainties may be due to dependency on contractor production rates and current business ventures. The 20k booster initiative envisions production risks since the booster design will require unique tooling and facility modifications that can impact manufacturing lead times. The main propulsion system design will require sizing adjustments based on current space transportation system designs. Fabrication of the propulsion components may exceed the estimated time of delivery.

The program risk analysis for the 20k booster initiative will be conducted by technical, cost, and schedule risk assessments. Based on an assumed make/buy assembly and verification strategy, the 20k booster initiative is one of relatively medium risk. The upper stage, payload shroud, and main engine are assumed to be purchased (fully verified) and shipped to the launch site for final integration. The propulsion system components are to be fabricated and shipped from the contractor to the booster assembly site. The booster piece-part components are to be purchased and shipped from vendors to the booster assembly site for subsystem and stage assembly and verification.

Major risk drivers and severity characteristics were initially assessed for the following: Centaur upper stage, payload shroud, RD–180 engine, propulsion system, and booster. The initial results are listed in table 9. Inputs provided by study team representatives are summarized and displayed on the isorisk curve in figure 42. The identified and summarized risk levels are a reflection of the expected loss as determined by considering the likelihood (relative probability of occurrence) and potential program impact if the risk occurs. These ratings help establish priorities in working potential problems. Data from the risk analysis are used to formulate contingency plans, to define areas for management attention, and to formulate data for use in future cost and schedule risk assessments.

The 20k initiative envisions political and economic risks. The purchase of the RD–180 is schedule dependent on the current capabilities and facilities provided by the CIS. Cost profiles could significantly exceed current estimates if fabrication, assembly, and certification of the engine had to be performed in the United States.

Table 9. Initial risk analysis results.

RISK ASSESSMENT

VEHICLE PURCHASE/FABRICATE, ASSEMBLY & VERIFICATION STRATEGY

OPTION 1

		UPPER STAGE	PAYLOAD SHROUD	BOOSTER	MAIN PROPULSION SYSTEM	MAIN ENGINE
RISK FACTORS	MATURITY FACTOR	LOW .1	LOW .1	MODERATE TECHNOLOGY AVAILABLE DESIGN REQUIRED .6	LOW SIZING ADJUSTMENTS ON STS DESIGNS .1	MINOR TECHNOLOGY AVAILABLE REQUIRES TURBOPUMP MODIFICATIONS/REDESIGN .3
	COMPLEXITY FACTOR	LOW .1	LOW .1	MINOR UNIQUE TOOLING COMPLEXITY IN PROCESS & CARE/REPAIR .3	LOW .1	LOW .1
	DEPENDENCY FACTOR	MINOR CURRENT CONTRACTOR PRODUCTION RATES .3	MINOR CURRENT CONTRACTOR PRODUCTION RATES .3	MODERATE SCHEDULE DEPENDENT FOR NEW SYSTEM & FACILITY MODS/ ENVIRONMENTAL ISSUES .6	LOW .1	MODERATE SCHEDULE DEPENDENT ON CURRENT CIS POLITICAL SYSTEM, & FACILITIES .5
SEVERITY CHARACTERISTICS	TECHNICAL GOALS	LOW .1	LOW .1	MINOR REDUCTION IN TEST NO STRUCTURAL TESTING .3	LOW .1	LOW .1
	COST ESTIMATES	LOW .1	LOW .1	LOW .1	LOW .1	SIGNIFICANT COST ESTIMATES COULD EXCEED 20% TO 50% IF FABRICATION & ASSEMBLY PERFORMED IN U.S. .7
	SCHEDULE IMPACTS	LOW .1	LOW .1	MODERATE MANUFACTURING LEAD TIMES .5	MINOR FABRICATION - TIME OF DELIVERY .3	LOW/MINOR .2

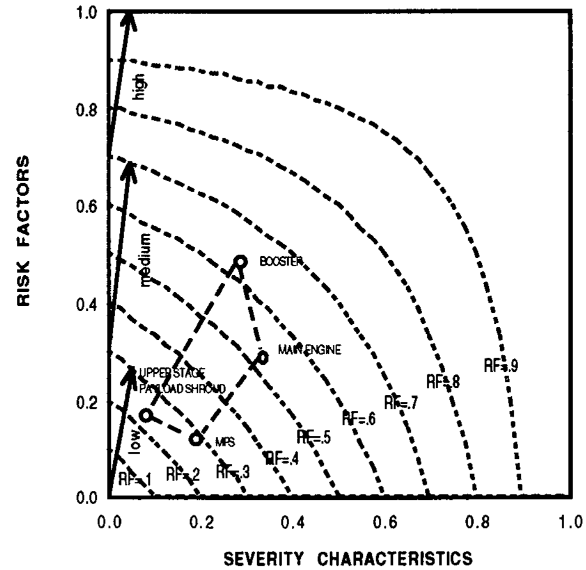


Figure 42. 20k initiative isorisk curve.

B. System Schedule

The 20k initiative schedule shown in figure 43 is based on several major procurement and acquisition assumptions. Some of these assumptions include an expedited procurement approach, early identification of long lead items, acquisition through early procurement actions, participation of in-house system engineering and integration throughout the program, and, finally, vehicle design and assembly performed in-house. Additionally, major piece parts, including MPS components, shall be manufactured and fabricated by aerospace industry counterparts.

To support a fast-track acquisition schedule, early major reviews are replaced with system/design release milestones. Exit criteria in support of these milestones shall be identified early in the program. The major milestones shall include a system requirements release, initial design release (10 percent of drawings complete), and final design release (90 percent of drawings complete).

C. Development Plan

All concurrent engineering support of the system requirements review/release milestones for the 20k fast-track acquisition will require approximately 4 months of dedicated effort. The near-term system engineering/activities plan for this level of effort is described in figures 44 and 45. Manpower adjustments and institutional requirements will be integrated for support of all critical and concurrent activities that support the systems requirements review (SRR) milestone. Further processes will be defined and established by the integration/design team. Primary focus will be to define the system integration and trade study process for handling cost, schedule, and performance constraints.

To support such a fast-track acquisition, major emphasis should be on defining clear, focused criteria for traceable down-select and system/subsystem decision making. To meet the fast-track schedule, system/design goals should emphasize added margin and reduced sensitivity. Operationally efficient requirements will become strongly weighted criteria.

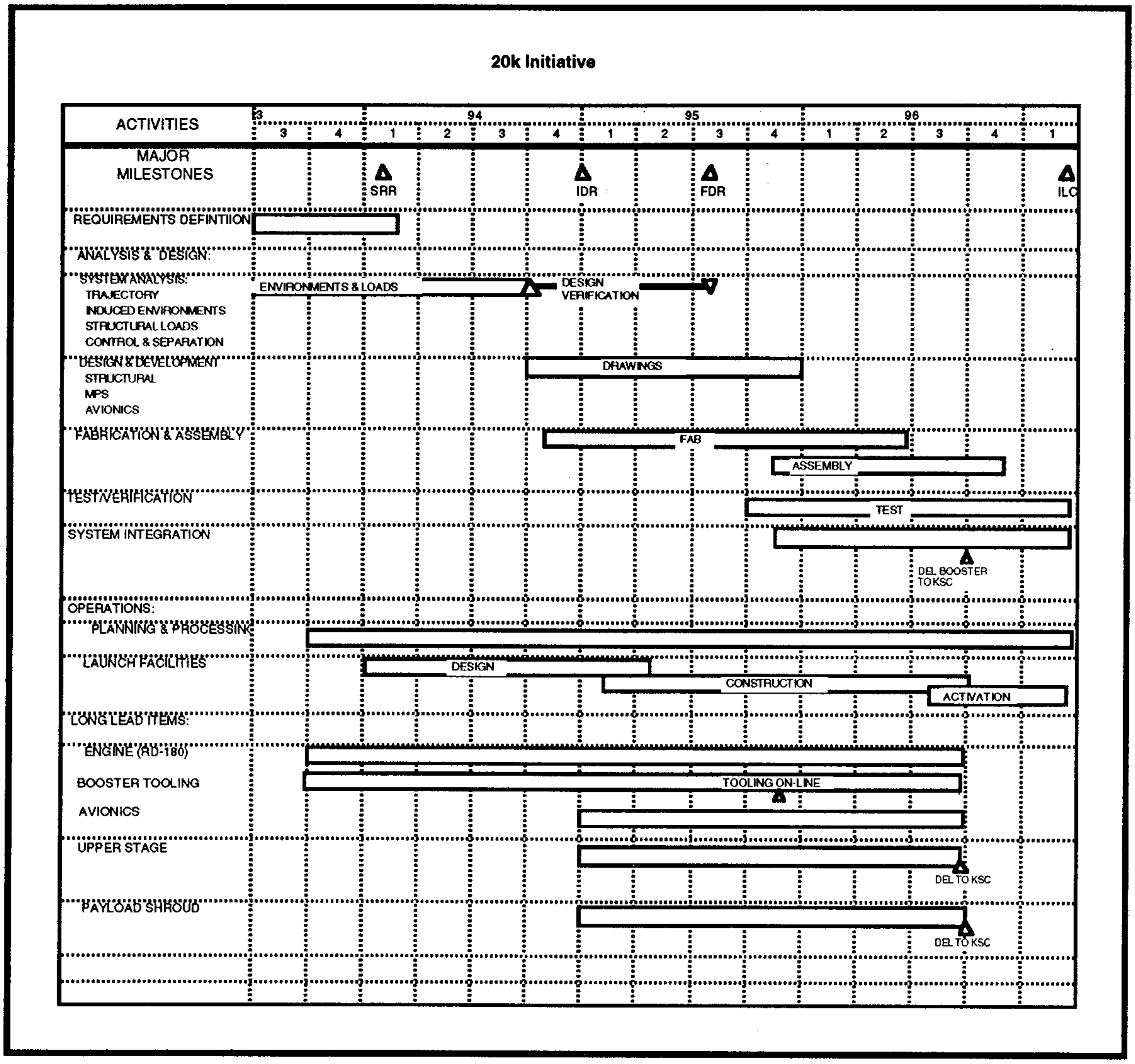


Figure 43. 20k "fast track" schedule.

D. Data Management

The following recommendations are given for consideration in the event the study/project proceeds. The project manager should appoint a data manager very early in the project to organize, define, and administer the project's data management requirements. A data management plan consistent with project policies should be prepared and tailored to the project's specific needs. Its complexity will only mirror that of the program management structure. Identification of documentation requirements in the early phases, with clear understanding and justification, can eliminate unnecessary documentation as the project matures.

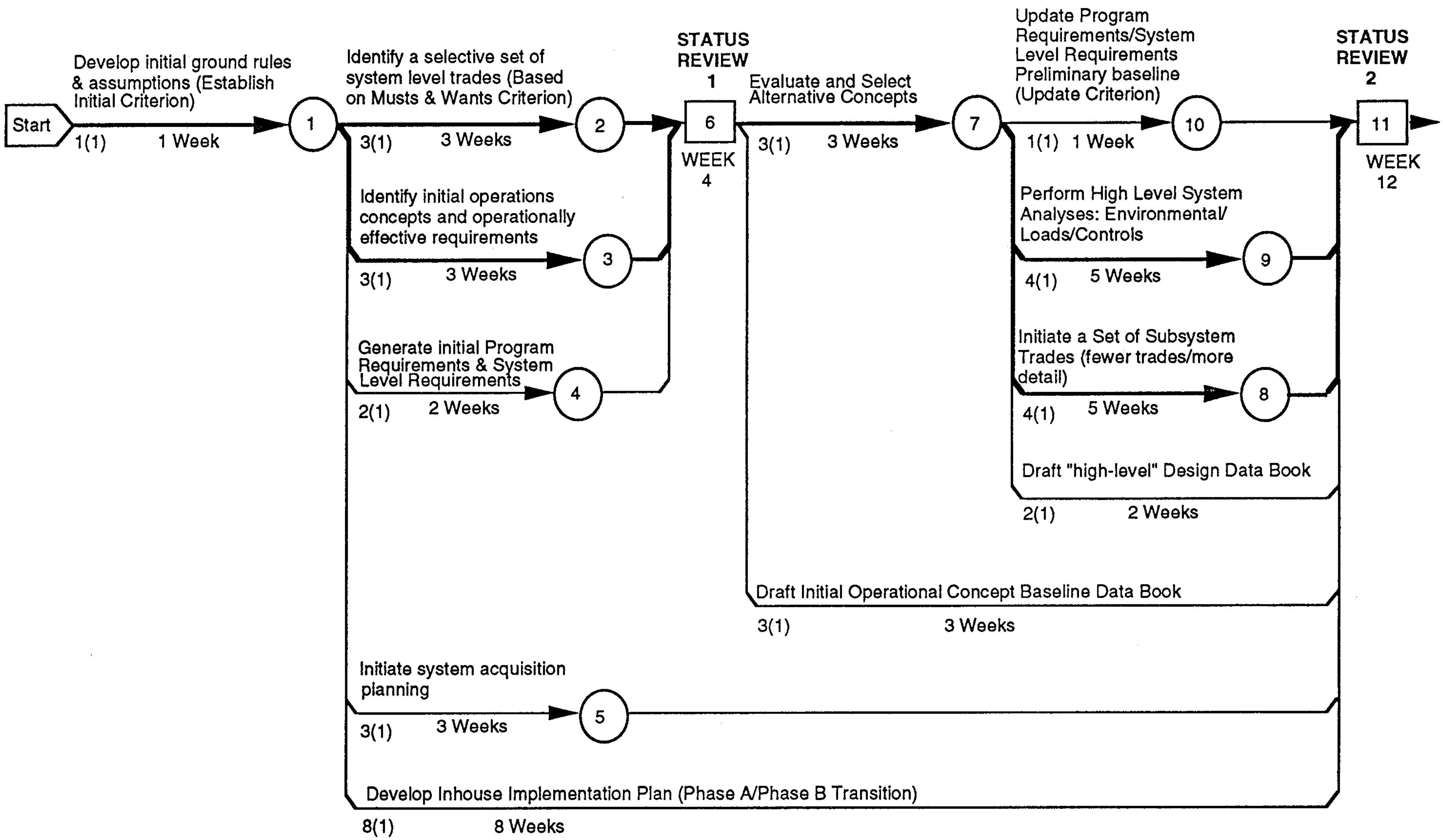


Figure 44. Near-term system engineering/activities plan.

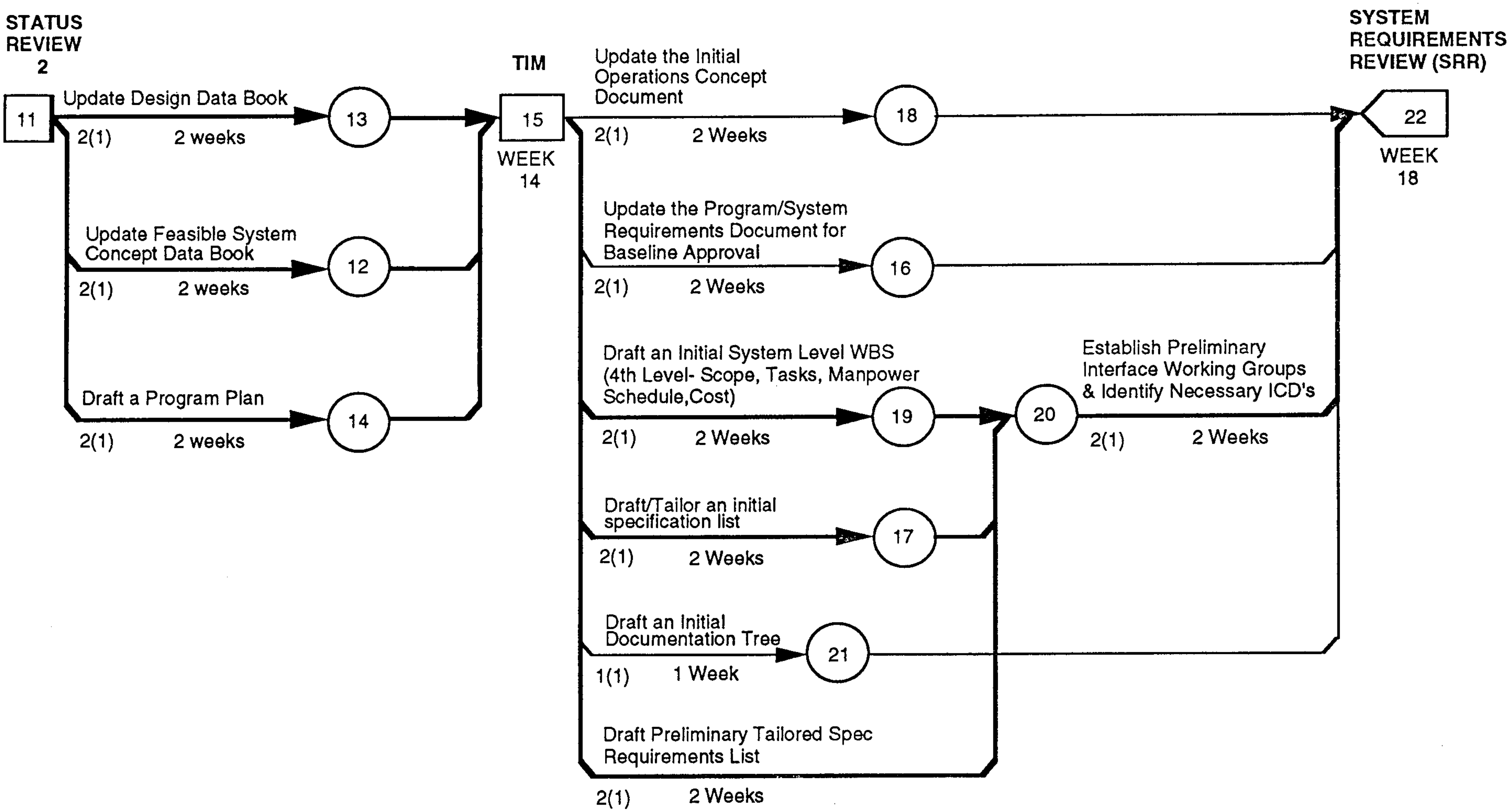


Figure 45. Near-term system engineering/activities plan.

To meet the development time of 40 months from ATP to flight vehicle "out-the-door," concurrent engineering is mandatory. This will require the baselining of designs, drawings, specifications, etc., to be streamlined. Real-time changes will have to be negotiated and a documentation "czar" will have to be named to control and expedite changes. Documentation will have to be justified as required, not "nice to have." Applicable specifications will have to be placed on drawings rather than referencing a document.

Definition of procured and in-house developed items will be crucial to the identification of project documentation and, therefore, should take place early in the project.

VI. FUTURE TRADES

Individual teams were tasked to define trade studies they recognized for a follow-on effort or execution of this project. They are listed as submitted:

A. Structures Trades

100-in long common dry bay elements

Change the 60-in long common elements to 100-in long common elements.

Composite dry bays

Trade the aluminum skin/stringer construction method with composite face sheet-aluminum honeycomb core construction. The composite parts could be made on the tape laying machine in building 4707. Autoclave versus oven cure is an additional option.

Isogrid construction cost versus skin/stringer cost

The cost of isogrid/orthogrid needs to be determined for a decision on manufacturing method of the tanks.

Semimonocoque tanks/integral slosh baffles

The pressure in the tanks significantly stabilizes the structure. It may be possible to have a skin with rings only. Rings would help reduce slosh problems and could be increased to act as the slosh baffles.

Launch facilities

The current configuration requires the engine to be extracted through a hold in the pad. Depending on vehicle drift and acceleration, the holddown mechanism may need to retract to give acceptable clearance (figs. 2 and 6).

Separation systems

A linear-shaped charge separation needs to be compared with a "super-zip" type separation. The separation hardware needs to be sized.

Booster/payload shroud interface

This interface needs greater definition and design.

Boattail

The boattail of the shroud needs greater definition and design.

Thrust structure

The conical thrust structure needs more design. The RD–180 may provide whatever thrust structure is needed, so this part might be removed.

Engine aerocover

The aerodynamic, heating, and vibration loads on the engine aerocover require definition so that this structure can be designed.

Slosh-baffles

The structural design and damping requirements of the slosh-baffles needs refinement.

Fins/conical aft structure

The need for fins must be determined, and the possibility of using a conical aft structure instead of fins needs to be traded (fig. 7). Base drag may be too high to use a conical structure. If fins are required, the configuration (number and size) must be determined.

Tank size

The current tanks are slightly oversized to allow for variations during this study phase. They exact size will be determined.

Buildup/shutdown and lift-off loads

The buildup/shutdown and lift-off loads need to be incorporated into the mass properties.

LO_2 sump for reduced intertank length

The difficulties of using a sump feed system instead of the current configuration need to be traded with the required length of the intertank (and the common dry bay elements).

Helium tanks

The location, volume, and mounting of the pressurant tanks need to be determined.

Avionics mounting

The avionics locations need to be verified, a mounting method defined, and cable routing defined.

Feed line support

The feedline support and protection methods need to be defined.

Systems tunnel

The systems tunnel size and mounting method need to be determined.

Dome manhole

The access holes in tank domes need to be sized and designed.

B. Stress Trades

The aft skirt needs to be resized to include the fin loads.

The interface between the aft skirt and the pad holddown structure needs further trade studies.

The rings in the dry bays at the 60-in cylinder joints need further analyses.

A trade for low height versus standard height bulkheads should be performed.

C. Aerodynamic Trades

Aerodynamic trade studies would involve the nose configuration, the length and diameter of the various vehicle component sections, and methods of improving the static stability of the vehicle. Performance may be enhanced by developing a new payload fairing which would incorporate tangent ogive geometry. Aerodynamic analysis of any proposed configuration modifications such as transitions for diameter changes would be required. Static stability can be improved in a variety of ways, such as altering the nose cone/payload fairing, adding fins, or flaring the aft skirt. Wind tunnel testing would be desired to support the development of an accurate design data base.

D. Propulsion System Trades

Additional analyses are required to define the propulsion system test requirements that must be accomplished prior to first flight and the facilities and schedule to accomplish the testing. It is assumed that fill and drain, terminal drain, and hot-fire tests are requirements. The number of fill and drain tests will require further definition of the loading and on-pad drain sequences and procedures, for example. The number of terminal drain tests will depend on the engine requirements during shutdown, including the impact of fuel or oxidizer starvation on the engine. The number of hot-fire tests will also depend on the test requirements, which are yet to be defined.

Further work is also required to understand the RD–180 servicing, pre-launch checkout, and launch commit criteria as well as the checkout procedures, loading procedures, and launch commit criteria for the rest of the propulsion system.

Better understanding of the RD–180 operations may result in significant changes in the propulsion system. One example would be the availability of only one heat exchanger for pressurization helium.

Trade studies on the propellant feed line diameters are required. The trade should include cost and schedule effects as well as performance. The fuel feedline configuration will need to be reexamined once the RD–180 interface requirements and aft compartment clearances are defined. Purge requirements for the forward, intertank, and aft compartments need to be defined.

E. Propulsion Testing Trades

Some subsystem level test objectives can be met during main propulsion system/engine system level tests. These are TBD pending identification of subsystem level test requirements.

F. Performance Trades

Trade studies could include variations on the engine types and capabilities for both the first and second stages. The effects of various payload fairings could be investigated. For this study, only one LEO and a single GTO mission were considered. A variety of missions, such as other geotransfer orbits and GEO's, should be investigated.

G. Avionics Trades

Avionics trades include upgrading the guidance and navigation functions to include global positioning system (GPS) receivers for positioning, developing vehicle health monitoring/management into the overall design, efficient electrical power generation and switching systems, and defining the number of strings required to meet the TBD reliability.

H. Design and Manufacturing Trades

High performance work team(s) will be established to evolve the most efficient design-for-manufacture concept that would meet the requirements and be cost competitive in production.

VII. ENVIRONMENTAL REQUIREMENTS

A. Environmental Assessment/Environmental Impact Statement

The Code of Federal Regulations (14 CFR Part 1216.305) states explicitly than an environmental assessment (EA) or environmental impact statement (EIS) is required for any proposed action by NASA to develop and operate new launch vehicles that may have an impact upon the quality of the environment. This requirement is detailed in accordance with NASA Handbook (NHB) 8800.11, which deals with implementing the provisions of the National Environmental Policy Act (NEPA) of 1969.

An EA is required when the need for an EIS is not known. The purpose of the EA is to determine the necessity for the EIS if significant impacts are found, or vice versa if not. However, as specified in NHB 8800.11, the release of rocket exhaust gases into the atmosphere is one of three categories of NASA actions that has been recognized since 1970 as requiring an EIS. Also specified is that all environmental analyses should be directed toward early planning and execution of the EIS from the outset of the project, that the notice of intent to proceed with the EIS be published, and the remainder of the EIS process be initiated immediately.

B. Clean Air Act Amendments of 1990

The clean air act amendments (CAAA's) of 1990 requires all states to submit an operating permit plan which upon approval by the U.S. Environmental Protection Agency (EPA) will allow and require each State to permit and monitor air emission sources. Each facility (company, business, military base, federal facility, etc.) will then be required to submit an operating permit application. This application is expected to require inclusion of all Center-wide foreseeable operating scenarios during the 5-year permit timeframe for major sources, and plants subject to the national emission sources hazardous air pollutants criteria. The operating permit also contains a fee payment rate based on pollutant quantities.

The 20k booster program would require inclusion in the MSFC operating permit application. EPA-approved air modeling of engine emissions and cloud dispersion patterns during the varying weather conditions would be required. Failure to include the program in the operating permit application with sufficient testing scenario flexibility could hamper program progress.

C. Clean Water Act—National Pollution Discharge Elimination System Permit Application

The existing National Pollution Discharge Elimination System permit will require modification to accommodate the changing discharges as required in the Clean Water Act.

D. Deluge Pond Remediation

The two candidate test stands for the proposed 20k booster testing at MSFC that were discussed in section III.B are the F–1 stand, building 4696, in the west test area and the Saturn 1–C stand, building 4572, in the east test area. Both facilities have associated deluge or holding ponds that have been classified as solid waste management units by the EPA.

Investigation of both sites by MSFC is planned for FY94 to determine the extent of the soil and ground water contamination. Based on the results of these examinations, remediation activities would then be planned and implemented. Expediting this process would be possible pending EPA and Alabama Department of Environmental Management regulatory approval.

E. Asbestos Abatement

The two test stands mentioned in section VII.D will have some asbestos abatement. The abatement and disposal will be performed in accordance with the Toxic Substances Control Act (TSCA) and the Occupational Safety and Health Administration (OSHA). Any work that can be performed without disturbing the asbestos is allowable.

F. Lead Paint Refurbishment

The two test stands mentioned in section VII.D most probably were primed and painted with chromium- and lead-based coating. Any modifications to either stand that involves abrasive blasting, welding, or other coating disruption would require adherence with Resource Conservation and Recovery Act (RCRA) hazardous waste regulations and occupational health and safety regulations for personnel protection.

G. Rocket Propellant Handling

The use of RP in engine testing operations would require modifications to the Center's spill prevention control and countermeasures plan and adequate spill containment structures around the filled booster (all locations). If any residue fuel was left in any part of the booster after testing, considerations would be required to minimize any leaks and for the containment of any leakage during transportation to the launch site.

H. The Resource Conservation and Recovery Act

The RCRA regulates the accumulation, packaging, storage, and disposal of hazardous wastes. For the 20k booster project, it is expected that some hardware will require cleaning or the verification of cleanliness level prior to assembly and possibly will generate some hazardous waste as a result of testing operations. MSFC currently retains a contractor to manage this task, and their system is sufficiently flexible to be able to handle the expected increases in volume and accumulation sites.

VIII. GROWTH PATH OPTIONS

An exercise was performed to define the potential growth of this concept to 65k lb of payload in LEO. Table 10 summarizes the results of growth path steps. The first step is to replace the Centaur upper stage with one of increased performance. The stage selected is from an earlier 20k study. The CIS D–57 LO_2/LH_2 engine was substituted onto the scalable upper stage and the propellant split between stages optimized. This produced approximately 27k to LEO.

The RD–170 was substituted for the RD–180, and the booster propellant was increased while using the Centaur as the upper stage. This configuration provided approximately 40k to LEO.

This booster was combined with the D–57-powered upper stage, and the propellant split was reoptimized using the stages' scaling equations. This results in performance of 52k to LEO. The final attempt at growth was to take the above configuration and supplement the booster with two Thiokol GT–120 solid boosters. This combination yielded 67k in LEO.

IX. CONCLUSIONS AND RECOMMENDATIONS

This exercise shows that a launch vehicle system could be put on-line via a fast-track concept in a relatively short time, at a reasonable cost. The resulting baseline vehicle is probably not the optimum for the long haul and, if time permitted, a structured development approach should be undertaken.

If a crash program is not required, it is imperative that the end point of the payload weight and dimensions growth options be part of the basic requirements. A binding decision must be made up front that when the system is operational and the maximum performance envelope defined, the system will not be stretched to the extreme capability, as today's systems are, which require much preflight analyses to verify they will fly. This allows the design/manufacturing/operations infrastructure to be traded and developed as the most overall efficient concept for the country.

The design must include the inserting of emerging technologies as completed which will improve the product, not just be nice-to-have. High-performance work teams will be required to perform these analyses and develop the correct scenario. A point in the program must be defined where the vehicle is operational and the design engineering staff can be dismissed from day-to-day activities.

As the engineering design of the booster is completed, the design team should initiate the new upper stage and payload fairing design. This sequential approach will allow a reasonable funding profile and also allow commercial developers participation availability in lieu of total Government funding.

Further studies would develop and refine the requirements and optional attainment methods.

Table 10. Growth options performance results.

	Baseline	D–57 Stage	Centaur Stage	D–57 Stage	D–57 Stage
Engine Types	RD–180 Centaur	RD–180 D–57	RD–170 Centaur	RD–170 D–57	RD–170 D–57 2 GT–120's
Flight Parameters					
Maximum Dynamic Pressure	591 lb/ft^2	557 lb/ft^2	703 b/ft^2	634 lb/ft^2	799 lb/ft^2
Minimum Engine Throttle	67.97%	93.1%	50.90%	71.10%	77.13%
Maximum Acceleration	4.5 g	4.5 g	4.5 g	4.5 g	4.5 g
Total Weight at Lift-off	1.200	1.193	1.200	1.200	1.302
Engine Data					
RD–180					
Thrust	945,000 lbf	945,000 lbf	N/A	N/A	N/A
Isp	337 s	337 s	N/A	N/A	N/A
RD–170					
Thrust	N/A	N/A	1,776,892 lbf	1,776,892 lbf	1,776,892 lbf
Isp	N/A	N/A	336 s	336 s	336 s
RL–10A–4 (2)					
Thrust, total	41,600 lbf	N/A	41,600 lbf	N/A	N/A
Isp	448.9 s	N/A	448.9 s	N/A	N/A
D–57					
Thrust	N/A	83,000 lbf	N/A	83,000 lbf	83,000 lbf
Isp	N/A	456 s	N/A	456 s	456 s
Solids					
Castor GT–120	N/A	N/A	N/A	N/A	2
Weights					
Gross Lift-off Weight	731,052 lb	726,842 kb	1,358,148 lb	1,358,423 lb	1,582,889 lb
Booster					
Propellant	588,312 lb	531,107 lb	1,158,153 lb	1,078,680 lb	1,154,870 lb
Stage Weight	55,739 lb	49,295 lb	94,082 lb	89,987 lb	95,188 lb
Solids					
Propellant	N/A	N/A	N/A	N/A	216,039 lb
Jettison Wight	N/A	N/A	N/A	N/A	17,418 lb
Upper Stage					
Propellant	45,727 lb	94,476 lb	45,727 lb	111,175 lb	114,004 lb
Stage Weight	7,388 lb	11,831 lb	7,388 lb	12,285 lb	12,362 lb
Shroud Weight	9,200 lb	9,200 lb	9,200 lb	9,200 lb	9,200 lb
Margin	4,000 lb	4,000 lb	4,054 lb	5,282 lb	6,709 lb
Net Payload	20,688 lb	26,934 lb	40,544 lb	52,518 lb	67,090 lb

APPENDIX

Team Members

The following lists the lead members of the study team and their area of responsibility.

Name:	Area	Org	Phone: 544-
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J. Brunty	Loads	ED22	1489
D. McGhee	Loads	ED22	1500
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D. Brolliar	Str Analysis	ED24	7199
J. Lee	RD-180 Data	EE83	4951
J. Redus	Prop Sys Design	EP21	7051
H. Pratt	Prop Test Req's	EP15	7069
M. Harris	Avionics	EB43	3790
H. Garrett	Avionics	EB21	3431
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J. Brumley	Manufacturing	EH52	1025
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D. Mercier	Performance	PD33	0541
T. Schmitt	Performance	PD33	0542
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APPROVAL

A 20K PAYLOAD LAUNCH VEHICLE FAST TRACK DEVELOPMENT CONCEPT USING AN RD–180 ENGINE AND A CENTAUR UPPER STAGE

By R. Toelle

The information in this report has been reviewed for technical content. Review of any information concerning Department of Defense or nuclear energy activities or programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

J.M. McMillion
Director, Program Development

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13. ABSTRACT *(Maximum 200 words)*

A launch vehicle concept to deliver 20,000 lb of payload to a 100-nmi orbit has been defined. A new liquid oxygen/kerosene booster powered by an RD–180 engine was designed while using a slightly modified Centaur upper stage. The design, development, and test program met the imposed 40-mo schedule by elimination of major structural testing by increased factors of safety and concurrent engineering concepts. A growth path to attain 65,000 lb of payload is developed.

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