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Contract AF04(695)-150

TITAN III/MOL COMPATIBILITY STUDY (U)

Performance Improvement Report
Technical Summary

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Approved

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FOREWORD

The data contained in this report summarize the results of a 60-day study of performance improvements to the Titan III system. The performance improvements were achieved using larger solid rocket motors to replace the present 5-segment 120-in. diameter solid rocket motors of Stage 0. These performance improvements were assessed against the requirements of the Manned Orbiting Laboratory (MOL) mission. Additional technical details for specific study areas are contained within individual tradeoff study reports summarized in the bibliography.

This document is submitted under Item 1, Exhibit A, Task 5.13 of Contract AF04(695)-150 in accordance with Line Item 3A-31 of Contractor Specification SSS-TIII-010 DRD (Rev 3), dated 15 April 1963, and DSCN 1 thru 97.

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SUMMARY

The purpose of the performance improvement study was to investigate the performance improvements that could be achieved on the current Titan III system by incorporating larger solid rocket motors (SRMs) to replace the present 5-segment 120-in. diameter motors for the Manned Orbiting Laboratory (MOL) mission. Three SRM options were considered for MOL missions launched from the Western Test Range (WTR) into low altitude polar orbits. Each option was considered both with and without use of the transtage through orbit injection. The SRM options were:

1) 7 segment, 120-in. (7 seg-120);

2) 2 center segment, 156-in. (2 CS-156).

3) 3 center segment, 156-in. (3 CS-156).

The 7 seg-120 configuration considered a 15:1 Stage I engine expansion ratio. The Stage I engines for the 156-in. SRM configurations, except as used for a staging analysis, considered only an 8:1 expansion ratio engine.

Study results indicate that minimum modification to the Titan III core stages is required for all SRMs. The aft longeron in Stage I must be strengthened to accept the 156-in. SRMs. The local vibration environment in the region of attachment for the longer SRMs (e.g., 7 seg-120 and 3 CS-156) will necessitate requalification, relocation, and redesign of some Titan III core components. Another forward staging rocket will be required for staging of the 156-in. SRMs. Net pump suction head (NPSH) requirements of the oxidizer feedline to the Stage I engine assembly can be met by increasing the tank lockup pressure without modifying the core for the 7 seg-120 SRM using the 15:1 expansion ratio engine. A relaxation in NPSH requirements to 35-ft head is required for the 156-in. SRMs using either the 8:1 or 15:1 expansion ratio engine.

All SRM designs met the preselected vehicle compatibility constraints for liftoff acceleration ($\geq$1.6 g), maximum dynamic pressure ($\leq$900 psf), limit in-flight acceleration ($\leq$3.2g), and maximum aerodynamic heating $\left(\leq 100 \times 10^6 \text{ ft-lb}/\text{ft}^2\right)$. The SRM designs were based on state-of-the-art technology using conventional class 2 propellants and liquid inject thrust vector control (TVC). Analysis of the injectant requirements showed that the Titan III tank

size is adequate to provide total injectant for worst-case conditions. SRM weights (per pair) varied from 1,364,000 (7 seg-120) to 2,537,000 lb for the 3 CS-156 option. For all configurations, total SRM vacuum impulse varied from 308,000,000 to 588,000,000 lb-sec compared to the present 5 seg-120 SRM 210,000,000 lb-sec impulse.

Payload performance into low altitude orbit was found to be almost directly proportional to Stage 0 vacuum impulse and to be increased by approximately 1000 lb when the transtage was not used. Guaranteed payload was 28,306, 35,030, and 42,109 lb for the 7 seg-120, 2 CS-156, and 3 CS-156 options, respectively, for the without-transtage configurations launched into an 80-n-mi polar orbit. East launch of these configurations into a 100-n-mi circular orbit from the Eastern Test Range (ETR) would give payload capability of 33,000, 41,000, and 49,500 lb, respectively. The minimum performance capability of the current Titan IIIC using the 15:1 Stage I engine and anticipated MOL equipment weights would be 22,206 lb into an 80-n-mi circular polar orbit from WTR without transtage.

Analyses of the flight control system for each configuration option, across the range of specified maximum payload lengths, demonstrated that the current Titan III autopilot could be used. Modifications to increase the capacity of the adapter programer and flight control computer will be necessary to provide 20% additional capacity for any of the SRM options when used with the transtage. This increased capacity is not required for the without-transtage configurations since adequate capacity for the additional rate gains necessitated by the longer burn time SRM options can be obtained when the orbital coast requirement is deleted. A new snubber will be required for Stage I for the 7 seg-120 configuration option with the 15:1 Stage I engine. Adequate stability margins were demonstrated for all configurations. Relocation of the inertial measurement unit (IMU), forward rate gyro, and lateral acceleration sensor will be required for all configurations. The load-relief accelerometer system was used for all flight simulations with effectiveness ranging between 28 and 35% depending on the specific configurations. Yaw structural mode frequencies were not sensitive to payload length.

Results of the rigid body loads analysis showed the maximum airload indicators $\left(q\alpha\beta_{max}\right)$ to be about 5000 lb-deg/ft^2 for all configurations regardless of payload length, and to be approximately the magnitude currently experienced on the Titan IIIC.

Worst wind azimuths occurred at either 245 or 272 deg azimuth for wind shear peak altitudes between 29,000 and 32,000 ft at about Mach 1.40. It was determined that all configurations would carry MOL payload lengths of at least 51 ft without structural beefup of the core and in the worst wind condition. The 7 seg-120 configuration could transport MOL payload lengths of 57.8 and 74.7 ft for the with- and without-transtage options, respectively. Comparable lengths for the 2 CS-156 options were 51.3 ft with transtage and 67.8 ft without transtage and, for the 3 CS-156 options, were 53.0 ft with transtage and 71.2 ft without transtage. The analysis demonstrated that a 70-ft payload length could be carried with a ≥97% launch probability for all configurations without transtage and ≥65% for all configurations with transtage. Analyses of the required structural beefup to the core to accept longer payloads without wind placarding and restriction of launch probability showed that relatively small increases in core structural weights (≤930 lb) would permit delivery/transport of all specified maximum overall length payloads; e.g., 70 ft with transtage and 82 ft without transtage.

A preliminary reliability and crew safety study was performed to assess the impact of the performance and preliminary crew safety modifications on mission reliability and crew safety. Crew safety analysis is defined as a warning-time analysis only. Analysis to date indicates that all of the with-transtage configurations (130-n-mi orbital mission) meet the mission success goal of 0.94 and that the best without-transtage configuration prediction is 0.964 compared to a goal of 0.970 for an 80-n-mi orbital mission. Differences in the six configurations, however, are insignificant. Preliminary crew safety analysis indicates that none of the configurations meet the goal of less than 1800 mission aborts per million flights with a warning time less than 3 sec. The predictions indicate a range of 2202 to 2598 mission aborts with a warning time less than 3 sec. The SRMs are the largest contributor to this estimate with a range of 1213 to 1364.

During the performance improvement study, an equal-risk schedule was prepared for the six configurations. This review indicated that the 7 seg-120 configuration has the least risk schedule. The controlling item for the 7 seg-120 configuration was the 15:1 Stage I liquid rocket engine (LRE) development, which would require out-of-position installation of the production engine to meet the May 1968 launch schedule. The 2 CS-156 and 3 CS-156 schedules indicated that a long-lead go-ahead for the case procurement and nozzle design would be required to meet the May 1968 launch schedule. It was concluded from the schedule review that all configurations, with the appropriate long-lead go-ahead, could meet the May 1968 launch schedule with equal risk.

It was determined from this 60-day study that each of the configuration options resulted in a feasible system that could be developed against MOL schedule requirements with low technical risk. All proposed systems were based on present state-of-the-art technology.

In view of the minimal difference in effect on the core for the various performance options, we have concluded that no basis exists for a Martin recommendation as to ultimate configuration selection.

I. INTRODUCTION

In 1964, Martin Company, Denver Division, participated in preliminary program definition studies under contract to the Space Systems Division (SSD), Air Force Systems Command, to identify the system elements to comprise the Manned Orbiting Laboratory (MOL) system. The compatibility of the Titan IIIC space launch system was assessed against the MOL mission requirements defined at that time. The compatibility assessment included studies of the performance capability of the Titan III with the present solid rocket motors (SRMs) and with larger SRMs. Payload performance was determined for low altitude orbits when launched from the Eastern Test Range (ETR) in an easterly direction and from the Western Test Range (WTR) into polar orbits. The use of the transtage as an integral part of the on-orbit portion of the MOL system was studied to establish changes required to the transtage for on-orbit times up to 30 days. The maximum length of payloads that could be transported on the current Titan IIIC were established for a family of cylindrical laboratory modules including a modified Gemini capsule. The requirements for new equipment were established as necessary to achieve increased mission success and improved crew safety using the Gemini capsule with and without a new launch escape tower. These studies were preliminary in nature, and the major emphasis was on the use of the current Titan IIIC system with 5-segment 120-in. diameter SRMs from ETR.

Based on results of these preliminary program definition studies, SSD further defined the requirements for the MOL system, particularly as they affected the choice of the Titan III system configuration to be used for the mission. It was established that development of the full operational capability for the MOL system would require launch into polar orbits from the WTR with payload weights in excess of the current Titan IIIC capability. Laboratory module diameter was set at 126 in. for lengths up to 82 ft. The transtage was not to be used as a part of the on-orbit system and might be used as part of the system to orbit injection. Increased mission success was defined with associated requirements for modifications to core subsystems. The decision was made not to provide a forward mounted launch escape tower on the Gemini capsule. Additional necessary subsystems to provide adequate margins for crew safety were to be provided in the Titan III core stages through increased redundancy, improved reliability of components, and use in alternative modes of selected systems in the spacecraft. Escape systems in the Gemini were to be typical of current Gemini configurations. These decisions resulted in the need for additional technical studies to establish the basis for a selection of the Titan III configuration to be used for the MOL mission.

Accordingly, in June 1965, SSD authorized Martin to proceed with a 60-day technical study to further evaluate the use of larger SRMs on the Titan III (with and without transtage) and against the changed MOL mission requirements. The preliminary studies carried out in 1964 showed that incorporating larger SRMs in the Titan III system for the MOL mission was a growth option that substantially increased the payload capability into polar orbit and could be achieved with minimum change to the existing core structure for attaching the larger solids. The objective of minimum change to the core structure was obtained by selecting the thrust characteristics of the SRMs so that in-flight accelerations and airloads were no more severe than for the Titan IIIC with the present 5-segment 120-in. diameter SRMs. Results obtained from the research and development programs for large SRMs at diameters of 120 and 156 in. suggested the feasibility and practicability of procuring new, larger SRMs for the Titan III/MOL system with low technical risk. Payload performance capability was shown to be directly proportional to the total impulse derived from the SRM stage for motor sizes transportable by rail or road. Preliminary study results, however, did not fully explore the effects of the several varieties of larger SRMs that might be used for the MOL booster in terms of the specific modifications to the core structure for attachment, the flight control systems required for stability margins and load relief, the length of payload that could be carried, the technical risk of new SRM developments, SRM staging, the reliability of the system, the applicability of existing thrust vector control (TVC) systems, overall technical risk, development schedule requirements, etc.

Therefore, the work reported in this document has as its primary purpose the technical assessment and comparative evaluation of using larger SRMs on the Titan III for the MOL mission. Three specific SRM options were studied. These were:

1) 7 segment, 120-in. diameter (7 seg-120);

2) 2 center segment, 156-in. diameter (2 CS-156);

3) 3 center segment, 156-in. diameter (3 CS-156).

Each of these options is evaluated for use with and without the transtage, making a total of six configurations studied. The 7 seg-120 SRMs and the 3 CS-156 SRMs attach forward in Stage II creating local changes in the vibration in the forward compartments. The 2 CS-156 attaches at the present attachment location of the 5 seg-120 SRM at Vehicle Station 504 in Stage I. For the 7 seg-120 configuration, a 15:1 expansion ratio is used for the LR-87 Stage I engines.

It is anticipated that the results of this 60-day study will contribute to selection of the configuration(s) that will be carried through the final definition phase for the Titan III/MOL booster. The effort covering detailed definition of the modifications required for crew safety and mission success is not included in this report. Additionally, the separate contractual effort to define the criteria for the interim launch capability (ILC) facility at WTR is not reported in this document. This study did not include consideration of larger diameter core stages for the Titan III or compare the relative cost/effectiveness of this system's growth option to the options using the larger SRMs for the MOL mission.

II. REFERENCE CONFIGURATIONS

This chapter describes the six vehicle configurations studied during the period covered by this report. Presented are: (1) details of the core modifications to accept each performance improvement configuration; (2) a listing of crew safety/mission success mods assumed for the six configurations; (3) comparative data for the three SRMs under consideration; and (4) information on the payloads evaluated.

The six basic configurations are listed in Table II-1.

Table II-1 Reference Configurations

Configuration No.	SRM	Stage I Engine Expansion Ratio	Transtage	Specified Payload Length (ft)
1	7 seg-120	15:1	Yes	54.5
2	7 seg-120	15:1	No	74.5
3	2 CS-156	8:1	Yes	58.5
4	2 CS-156	8:1	No	78.5
5	3 CS-156	8:1	Yes	61.0
6	3 CS-156	8:1	No	81.0

Figure II-1 presents the general arrangements of the six configurations.

A. CORE CONFIGURATIONS

Each of the six performance improvement combinations required modification of the basic Titan IIIC core configuration. These modifications are described in the following subsections. A summary of the changes required on Titan IIIC core (Vehicle 11 is the reference) for each of the six configurations is listed in Tables II-2 and II-3.

Certain other core modifications were required to add crew safety/mission success hardware. This class of modification was necessary so that the vehicles under study would more closely approximate the final MOL booster, which will have such additional equipment.

1. Performance Improvement Modifications, 7 seg-120 with Transtage

a. Structures

Stage III -- Equipment Module (Compartment 3A) (New) - The relocation of antennas in this module is due to pattern interference by the longer SRMs. This performance change, along with the addition of redundant equipment on the trusses for crew safety, causes the module to grow 13.5 in. in length.

Stage III -- Propulsion Module (Compartment 3B) - Gages of several skin panels will be increased to meet higher panel flutter requirements.

Stage II -- Compartment 2A (New) - The forward outriggers now attach to this skirt, necessitating the addition of new frames, heavier skins, and provisions for the SRM forward electrical interfaces. To increase flexibility, a bolted flange is added to the aft end of the skirt for attachment to the oxidizer tank.

Stage II -- Oxidizer Tank - Since the oxidizer vent valve is moved from the skin to the forward dome (to avoid the high acoustical loading on the skin due to the long SRMs), brackets must be added to the dome to support the valve. A bolting flange is added to the forward end of the tank barrel.

Stage I -- Heat Shields - Due to the 15:1 expansion ratio engine, new, larger thrust chamber assembly covers (heat shields) are required, as are some modifications to the engine bell insulation panels.

Other Modifications -- Core Insulation - Insulation must be modified in the vicinity of the SRM nose cone shock wave impingement on the core, and where the SRM staging rocket plumes impinge on the core.

Table II-2 Summary of Configuration Modifications to the Core due to Performance Improvement Changes

Subsystem	7 seg-120: With Transtage (No. 1)	7 seg-120: Without Transtage (No. 2)	2 CS-156: With Transtage (No. 3)	2 CS-156: Without Transtage (No. 4)	3 CS-156: With Transtage (No. 5)	3 CS-156: Without Transtage (No. 6)
1. Structures	← See Table II-3 →					
2. Ordnance	No change	1-in. staging nuts at Titan III/MOL interface	No change	Same as Configuration 2	No change	Same as Configuration 2
3. Propellant and Pressurization	a) Increase in Stage I tank top pressures to meet NPSH b) Move Stage II oxidizer vent umbilical to tank dome c) Changes to Stage I pressurization system for 15:1 engine	Same as Configuration 1	Same as a), Configuration 1	Same as a), Configuration 1	Same as a), b), Configuration 1	Same as a), b), Configuration 1
4. Flight Controls	a) Add gain state and filter value b) Relocate LASS and Stage II rate gyro into transtage	Same as a), Configuration 1	Same as a), Configuration 1	Same as a), Configuration 1	Same as Configuration 1	Same as a), Configuration 1
5. Hydraulics	a) New Stage I actuators b) Titan IIIB Stage II actuators	Same as Configuration 1	Same as b), Configuration 1	Same as b), Configuration 1	Same as b), Configuration 1	Same as b), Configuration 1
6. Electrical	SRM electrical interface moved forward	Same as Configuration 1	No change	No change	Same as Configuration 1	Same as Configuration 1
7. MDS	No change	No change	No change	No change	No change	No change
8. Tracking and Flight Safety	No change	Delete Stage II ISDS	No change	Same as Configuration 2	No change	Same as Configuration 2
9. Instrumentation	No change	No change	No change	No change	No change	No change
10. RF Systems	a) All antennas moved to BL 0 b) Delete Stage II TM antenna c) Provide new TM multiplexer and coax interstage connector	a) All antennas moved to BL 0 b) Provide new TM multiplexer c) Move Glotrack antenna between tanks, Stage II	Same as Configuration 1 except Stage II TM antennas are retained	Same as Configuration 2	Same as Configuration 1	Same as Configuration 2

[redacted]

Table II-3 Core Structural Modification due to Performance Improvement Changes

7 seg-120		2 CS-156		3 CS-156	
With Transtage (No. 1)	Without Transtage (No. 2)	With Transtage (No. 3)	Without Transtage (No. 4)	With Transtage (No. 5)	Without Transtage (No. 6)
Stage III Equipment Module (Compartment 3A) (New) 1. Relocate antennas. 2. Increase length by 13.5 in. Propulsion Module (Compartment 1B) Increase several skin panel gages. Stage II Compartment 2A (New) 1. Add bolting flange at aft end of skirt.† 2. Add provisions for outrigger attach including new frames. 3. Increase skin gages. 4. Add provisions for SRM forward electrical interface. Oxidizer Tank 1. Add bolting flange at forward end of tank barrel.† 2. Add provisions on forward dome for oxidizer vent valve. Stage I 1. Larger TCA covers. 2. Change engine bell insulation panels. Core Insulation Modify in areas of SRM staging rockets and nose fairing. SRM Forward Outriggers Beefup outriggers and fittings and add spherical bearings at core fittings.	Stage II Equipment Module (Compartment 2A) (New)* 1. Add bolting flange at aft end of skirt.† 2. Add provisions for outrigger attach including new frames. 3. Relocate umbilicals from transtage to this area. 4. Increase size of payload staging nuts from 3/4 in. to 1 in. diameter. 5. Design new trusses, not constrained by transtage tanks. 6. Add provisions for SRM forward electrical interface. 7. Relocate antennas from transtage to this area. Oxidizer Tank Same as Configuration 1. Compartment 2B 1. Add Glotrac antenna. Stage I Same as Configuration 1. SRM Forward Outriggers Same as Configuration 1.	Stage III Equipment Module (Compartment 3A) (New) Same as Configuration 1. Propulsion Module (Compartment 3B) Same as Configuration 1. Stage II Compartment 2A 1. Add bolting flange at aft end of skirt.† 2. Increase skin gages Oxidizer Tank Add bolting flange at forward end of tank barrel.† Compartment 2C Modify provisions and fairing for forward SRM electrical interface. Stage I Compartment 1A Beefup outrigger fitting attach frames. Fuel Tank Beefup barrel, "Y" frame, and longerons. Core Insulation Same as Configuration 1. SRM Forward Outriggers Same as Configuration 1.	Stage II Equipment Module (Compartment 2A) (New)* 1. Add bolting flange at aft end of skirt.† 2. Relocate umbilicals from transtage to this area. 3. Increase size of payload staging nuts from 3/4 in. to 1 in. diameter. 4. Design new trusses, not constrained by transtage tanks. 5. Relocate Antennas from transtage to this area. Oxidizer Tank Same as Configuration 3.† Compartment 2B Same as Configuration 2. Compartment 2C Same as Configuration 3. Stage I Compartment 1A Same as Configuration 3. Fuel Tank Same as Configuration 3. Core Insulation Same as Configuration 1. SRM Forward Outriggers Same as Configuration 1.	Stage III Equipment Module (Compartment 3A) (New) Same as Configuration 1 Propulsion Module (Compartment 3B) Same as Configuration 1. Stage II Compartment 2A (New) Same as Configuration 1. Oxidizer Tank Same as Configuration 1. Stage I Fuel Tank Same as Configuration 3. Core Insulation Same as Configuration 1. SRM Forward Outriggers Same as Configuration 1.	Stage II Equipment Module (Compartment 2A) (New)* Same as Configuration 2. Oxidizer Tank Same as Configuration 1. Compartment 2B Same as Configuration 2. Stage I Fuel Tank Same as Configuration 3. Core Insulation Same as Configuration 1. SRM Forward Outriggers Same as Configuration 1.

Note: This table lists only those core structural modifications directly necessary as a result of a performance modification (i.e., 7 seg-156, 15:1 Stage I, or deletion of transtage), except for items noted†.

*The Stage II forward skirt in the without-transtage configuration is also known as the Stage II equipment module.
†These changes are made to increase payload flexibility.

[redacted]

Other Modifications -- SRM Forward Outriggers - The outriggers and fittings must be beefed up due to higher tension loads created by thrust termination. The joint now existing on the core end of the outrigger will be redesigned to prevent imposition of bending loads on the staging stud.

b. Ordnance

No change.

c. Propellant and Pressurization

The Stage I pressurization levels might have to be changed to meet the NPSH requirements, as well as to meet any new specifications for the 15:1 engine. These changes might lead to higher tank top pressures.

d. Flight Controls

The flight control hardware will be modified to accommodate the longer burn time and environmental changes that result from the SRMs and the differences in load dynamics incurred with the heavier, longer payload capability. The modifications are:

Provide an additional gain state and filter value for Stage 0 flight;

Relocate the lateral acceleration sensing system (LASS) (load relief accelerometers) and Stage II rate gyros into the forward section of the transtage.

e. Hydraulics

A new or revised Stage I actuator (i.e., a new snubber) is required to accept the additional loads of the 15:1 engine. In addition, the redesigned Titan IIIB Stage II actuators must be used. The effect of the bending modes of the MOL/Titan III booster, coupled with the natural frequency of the engine truss/actuator, could cause the actuator to go unstable. This new Titan IIIB unit is believed to be usable for the Titan III/MOL booster.

f. Electrical

Because of the increased length of the 7-segment SRM, the forward attach point and electrical interface connections and circuitry are relocated forward in Stage II.

g. Malfunction Detection System (MDS)

No change.

h. Tracking and Flight Safety

No change.

i. Instrumentation

No change.

j. RF Systems

The command control antennas in the transtage are moved to BL 0 to avoid the interference caused by the longer solids.

The Stage II telemetry antenna system is deleted because of unacceptable vibration levels at its Titan III location. The Stage II telemetry transmitter uses the transtage antenna system. To accomplish this, a new four-input multiplexer and a coaxial staging disconnect are required.

2. <u>Performance Improvement Modifications, 7 seg-120 without Transtage</u>

a. Structures

<u>Stage II -- Equipment Module (Compartment 2A) (New Forward Skirt)</u> - Relocation of antennas and umbilicals previously located in the transtage is required. Provisions must be added for SRM forward outrigger attachment (new frames) and for SRM forward electrical interfaces. New, more efficient equipment trusses will be used. This is possible since the transtage propellant tanks/engines do not intrude into this area. A bolting flange is added to the aft end of the structure.

Stage II -- Oxidizer Tank - Same as 7 seg-120 with transtage.

Stage II -- Compartment 2B - Add Glotrac antenna (skin-mounted) at BL 0. (This antenna is mounted on the attitude control motor fairing on the transtage on the Titan IIIC core.)

Stage I -- Heat Shield - Same as 7 seg-120 with transtage.

Other Modifications -- SRM Forward Outrigger - Same as 7 seg-120 with transtage.

b. Ordnance

The payload staging hardware will require an increase in size due to higher tension loads without a transtage. This is a result of P_{eq} increasing as a function of vehicle station and the absence of axial load relief from the transtage. The staging nuts will be increased from 3/4 to 1.0 in. diameter. The number of separation points will remain the same as will the interface dimensions. All Stage III items are deleted.

c. Propellant and Pressurization

Same as 7 seg-120 with transtage, but with all Stage III items deleted.

d. Flight Controls

Provide an additional gain state and filter value for Stage 0 flight (because of longer burn). Move all needed equipment into Stage II and delete Stage III equipment.

e. Hydraulics

Same as 7 seg-120 with transtage, but delete Stage III hydraulics and actuators.

f. Electrical

Same as 7 seg-120 with transtage, but move needed equipment into Stage II and delete one IPS system because of transtage deletion.

g. MDS

No change, but move needed equipment into Stage II.

h. Tracking and Flight Safety

Delete the Stage II inadvertent separation destruct system (ISDS). Since Stage II is the last powered stage now, no ISDS is needed. In addition, all necessary tracking and flight safety equipment is moved into Stage II.

i. Instrumentation

Move all needed equipment into Stage II from the transtage.

j. RF Systems

Move all antennas to BL 0 in Stage II. Delete one telemetry antenna (reference Vehicle 11 has one antenna in the transtage and one in Stage II) and provide a new multiplexer so that all transmitters can share one antenna. Move the Glotrac antenna between Stage II tanks, since there is no further room for antennas on BL 0 of the new Stage II equipment module.

3. <u>Performance Improvement Modifications, 2 CS-156 with Transtage</u>

a. Structures

<u>Stage III -- Equipment Module (Compartment 3A) (New)</u> -

Same as 7 seg-120 with transtage.

<u>Stage III -- Propulsion Module (Compartment 3B)</u> - Same as 7 seg-120 with transtage.

<u>Stage II -- Compartment 2A</u> - Skin gages are increased to meet panel flutter requirements and a bolting flange is added to the aft end of the skirt to provide flexibility.

<u>Stage II -- Oxidizer Tanks</u> - A bolting flange is added to the forward end of the tank barrel.

<u>Stage II -- Compartment 2C</u> - Provisions for the SRM electrical interfaces must be modified for more and/or larger connectors. The interface fairing must be modified accordingly.

Stage I -- Compartment 1A - The frames in the areas of the SRM forward outrigger attachment must be beefed up to accept higher loads.

Stage I -- Fuel Tanks - Due to large differential tailoff thrusts on the 156-in. SRMs, the barrel, "K" frame, and longerons must be strengthened.

Other Modifications -- Core Insulation - Same as 7 seg-120 with transtage.

Other Modifications -- SRM Forward Outriggers - Same as 7 seg-120 with transtage.

b. Ordnance

No change.

c. Propellant and Pressurization

The Stage I pressurization levels might have to be increased to meet NPSH requirements.

d. Flight Controls

Same as 7 seg-120 with transtage.

e. Hydraulics

The same actuator redesign is required as was described for the 7 seg-120 with transtage.

f. Electrical

No change except for possible increase in the core/SRM electrical interface.

g. MDS

No change.

h. Tracking and Flight Safety

No change.

i. Instrumentation

No change.

j. RF Systems

Same as 7 seg-120 with transtage, except Stage II telemetry antennas are retained for this configuration only.

4. Performance Improvement Modifications, 2 CS-156 without Transtage

a. Structures

Stage II -- Equipment Module (Compartment 2A) (New Forward Skirt) - Relocation of the antenna and umbilicals previously located in the transtage, is required. New, more efficient equipment trusses will be utilized. Size of the payload staging nut is increased from 3/4 to 1 in. diameter and a bolting flange is added to the aft end of the structure.

Stage II -- Oxidizer Tank - Same as 2 CS-156 with transtage.

Stage II -- Compartment 2B - Same as 7 seg-120 SRM without transtage.

Stage II -- Compartment 2C - Same as 2 CS-156 with transtage.

Stage I -- Compartment 1A - Same as 2 CS-156 with transtage.

Stage I -- Fuel Tank - Same as 2 CS-156 with transtage.

Other Modifications -- Core Insulation - Same as 7 seg-120 with transtage.

Other Modifications -- SRM Forward Outrigger - Same as 7 seg-120 with transtage.

b. Ordnance

Same as 7 seg-120 without transtage.

c. Propellant and Pressurization

Same as 2 CS-156 with transtage.

d. Flight Controls

Same as 7 seg-120 without transtage.

e. Hydraulics

Same as 2 CS-156 with transtage plus deletion of Stage III hydraulics.

f. Electrical

Same as 2 CS-156 with transtage.

g. MDS

No change.

h. Tracking and Flight Safety

Same as 7 seg-120 without transtage.

i. Instrumentation

No change.

j. RF Systems

Same as 7 seg-120 without transtage.

5. Performance Improvement Modifications, 3 CS-156 with Transtage

a. Structures

Stage III -- Equipment Module (Compartment 3A) (New) - Same as 7 seg-120 with transtage.

Stage III -- Propulsion Module (Compartment 3B) - Same as 7 seg-120 with transtage.

Stage II -- Compartment 2A (New) - Same as 7 seg-120 with transtage.

Stage II -- Oxidizer Tank - Same as 7 seg-120 with transtage.

Stage I -- Fuel Tank - Same as 2 CS-156 with transtage.

Other Modifications -- Core Insulation - Same as 7 seg-120 with transtage.

Other Modifications -- SRM Forward Outriggers - Same as 7 seg-120 with transtage.

b. Ordnance

No change.

c. Propellant and Pressurization

Same as 7 seg-120 with transtage except that pressurization changes associated with the 15:1 engine are not required for this configuration.

d. Flight Controls

Same as 7 seg-120 with transtage.

e. Hydraulics

Same as 2 CS-156 with transtage.

f. Electrical

Same as 7 seg-120 with transtage.

g. MDS

No change.

h. Tracking and Flight Safety

No change.

i. Instrumentation

No change.

j. RF Systems

Same as 7 seg-120 with transtage.

6. Performance Improvement Modifications, 3 CS-156 without Transtage

a. Structures

Stage II -- Equipment Module (Compartment 2A) (New Forward Skirt) - Same as 7 seg-120 without transtage.

Stage II -- Oxidizer Tank - Same as 7 seg-120 with transtage.

Stage II -- Compartment 2B - Same as 7 seg-120 without transtage.

Stage I -- Fuel Tank - Same as 2 CS-156 SRM with transtage.

Other Modifications -- Core Insulation - Same as 7 seg-120 with transtage.

Other Modifications -- SRM Forward Outriggers - Same as 7 seg-120 with transtage.

b. Ordnance

Same as 7 seg-120 without transtage.

c. Propellant and Pressurization

Same as 3 CS-156 with transtage.

d. Flight Controls

Same as 7 seg-120 without transtage.

e. Hydraulics

Same as 2 CS-156 with transtage.

f. Electrical

Same as 7 seg-120 with transtage.

g. MDS

No change.

h. Tracking and Flight Safety

Same as 7 seg-120 without transtage.

i. Instrumentation

No change.

j. RF Systems

Same as 7 seg-120 without transtage.

<u>7. Assumed Changes for Crew Safety/Mission Success Improvement</u>

A summary of the crew safety/mission success changes to the Titan IIIC core (Vehicle 11) that were assumed for each of the six configurations are summarized in Tables II-4 and II-5, and are explained in more detail in the following paragraphs.

a. All Configurations with Transtage

1) Structures

<u>Stage III -- Equipment Module (Compartment 3A) (New)</u> - To accommodate the redundant equipment listed below, this module must be increased in length. A new equipment truss is needed for mounting this equipment.

<u>Stage III -- Propulsion Module (Compartment 3B)</u> - The engine truss is revised to accommodate the added hydraulic pump.

<u>Stage II -- Compartment 2B</u> - The equipment truss in this compartment is changed because of the newly added equipment for crew safety/mission success.

<u>Stage I -- Oxidizer Tank</u> - The aft dome of this tank is modified so that the redundant flight controls rate gyros may be mounted.

Table II-4 Configuration Summary of Crew Safety/Mission Success Modifications

Subsystem	With Transtage			Without Transtage		
	7-seg-120 (No. 1)	2 CS-156 (No. 3)	3 CS-156 (No. 5)	7 Seg-120 (No. 2)	2 CS-156 (No. 4)	3 CS-156 (No. 6)
1. Structures	← See Table II-5 →					
2. Ordnance	Redundant staging nuts at Titan III/MOL interface			Same as Configuration 1, 3, 5		
3. Propellant and Pressurization	No change			No change		
4. Flight Controls	2 autopilots plus Stage 0 reference autopilot 3 static inverters 3 LASS 6 rate gyros			3 majority-vote autopilots 3 static inverters 3 LASS 6 rate gyros		
5. Hydraulics	Gemini launch vehicle redundancy, Stages I, III			Gemini launch vehicle redundancy, Stage I		
6. Electrical	2 APS 2 TPS New discrete sequencer			3 APS 1 TPS New discrete sequencer		
7. MDS	Stage 0 autopilot selector Guidance and control selector Computer selector switches Analog tank pressure sensors SRM ΔP monitor			Autopilot output controller Guidance selector Analog tank pressure sensors SRM ΔP monitor		
8. Tracking and Flight Safety	Redundant ISDS enable/disable switch			Same as Configuration 1, 3, 5		
9. Instrumentation	Added measurements			Same as Configuration 1, 3, 5		
10. RF Systems	No change			No change		
11. Guidance	Payload IGS backup for BIGS; guidance self-check			Same as Configuration 1, 3, 5		

Table II-5 Core Structural Modifications due to Crew Safety/Mission Success Requirements

7 seg-120		2 CS-156		3 CS-156	
With Transtage (No. 1)	Without Transtage (No. 2)	With Transtage (No. 3)	Without Transtage (No. 4)	With Transtage (No. 5)	Without Transtage (No. 6)
Stage III Equipment Module (Compartment 3A) (New) 1. Increase length by 13.5 in. 2. New equipment trusses for new equipment requirements. Propulsion Module (Compartment 3B) Revise engine truss to accommodate added hydraulic pump. Stage II Compartment 2B New equipment truss Stage I Oxidizer Tank Modified aft dome requirement to mount two rate gyros.	Stage II Equipment Module (Compartment 2A) (New) 1. Increase length to house all equipment. 2. Equipment truss change to accommodate all new hardware. Compartment 2B Modify equipment truss Stage I Oxidizer Tank Same as Configuration 1.	Stage III Equipment Module (Compartment 3A) (New) Same as Configuration 1. Propulsion Module (Compartment 3B) Same as Configuration 1. Stage II Compartment 2B Same as Configuration 1. Stage I Oxidizer Tank Same as Configuration 1.	Stage II Compartment 2B Same as Configuration 2. Stage I Oxidizer Tank Same as Configuration 1.	Stage III Equipment Module (Compartment 3A) (New) Same as Configuration 1. Propulsion Module (Compartment 3B) Same as Configuration 1. Stage II Compartment 2B Same as Configuration 1. Stage I Oxidizer Tank Same as Configuration 1.	Stage II Compartment 2B Same as Configuration 2. Stage I Oxidizer Tank Same as Configuration 1.

Note: This table lists only those core structural modifications directly necessary as a result of a crew safety, or reliability modifications (i.e., redundant equipment).

2) Ordnance

The Titan III/MOL interface staging ordnance is changed to provide redundant nuts in place of the present manifolded nut. The electrical interface circuitry must be revised to activate the ordnance on the payload side of the interface.

3) Propellant and Pressurization

No changes for crew safety/mission success for this configuration.

4) Flight Controls

The flight controls system has two parallel, redundant channels from sensors through the electronics for all phases of flight. Dual sensors provided are the Stage I and II rate gyros and a lateral acceleration sensing system (LASS). The autopilot used is an analog unit (similar to the present Titan III system) consisting of dual computer and adapter programers. Dual static inverters are used to power necessary portions of the flight controls system.

For Stage 0 flight, an extra (reference) autopilot is added to the dual units along with additional Stage I and Stage II rate gyros, LASS, and static inverters.

5) Hydraulics

Redundant hydraulics systems are provided in Stages 0, I, and III. Gemini launch vehicle-type redundant hydraulic actuators are used in Stage I and in Stage III (a single system remains in use in Stage II). Separate power sources are used for each of the redundant systems; electrical sources are provided for Stage 0 and III, and engine turbine power for Stage I.

6) Electrical

Redundant accessory power supplies (APS) and transient power supplies (TPS) are provided to be compatible with the redundant flight control systems. To provide maximum reliability through the critical Stage 0 portion of flight, a third source of discretes (i.e., a sequencer) is supplied to majority vote certain functions.

7) MDS

The assumed MDS differs from that of a Titan III vehicle (Vehicle 11) by addition of a Stage 0 autopilot selector, a guidance and control selector (switches both guidance and flight controls), and a computer selector switch (to select which of the dual autopilots is controlling the vehicle). The MDS pressure switches of Titan III are changed to an analog type, usable for cockpit display. A delta pressure sensor is provided to monitor the difference in motor pressures between the two SRMs.

8) Tracking and Flight Safety

The ISDS enable/disable switches are made redundant.

9) Instrumentation

The number of measurements are increased to accommodate monitoring of the redundant equipment as well as to provide measurements for real-time ground displays for crew safety.

10) RF Systems

No changes for crew safety.

11) Guidance

The payload inertial guidance system (IGS) becomes a backup system for the booster IGS.

b. All Configurations without Transtage

1) Structures

Stage II -- Equipment Module (Compartment 2A) - The new Stage II forward skirt, required because of deletion of the transtage, is used to house the basic Titan III equipment; the increased equipment needed for crew safety/mission success affects the final length of this skirt. The new equipment truss is further changed because of the greater quantity of equipment.

<u>Stage II -- Compartment 2B</u> - Same as with-transtage configuration.

<u>Stage I -- Oxidizer Tank</u> - Same as with-transtage configuration.

2) Ordnance

Same as with-transtage configuration.

3) Propellant and Pressurization

No changes for crew safety/mission success for this configuration.

4) Flight Controls

The flight controls system uses three parallel channels that are majority voted for the entire flight. The autopilots are the same as used in the with-transtage configuration with the transtage portions deleted. Three Stage I rate gyros, three Stage II rate gyros, three LASSs, and three static inverters are provided.

5) Hydraulics

The hydraulics system used in this configuration is the same as the with-transtage case, except that the Stage III system is not required.

6) Electrical

This system differs from that previously described (with transtage) by the addition of a third APS battery and the deletion of the second TPS battery.

7) MDS

As the autopilots in this configuration are majority voted for all three stages, no malfunction sensing or switching is required in this area. This allows the deletion of the Stage 0 autopilot selector and the control portion of the guidance and control selector of the with-transtage case. The remainder of the system is as previously described.

8) Tracking and Flight Safety

Same as with-transtage configuration.

9) Instrumentation

Same as with-transtage configuration.

10) RF Systems

No changes for crew safety.

11) Guidance

Same as with-transtage configuration.

8. Other Modifications

Although not related to performance improvement or crew safety, other modifications are made to the core because the vehicle is flown from WTR.

These include deletion of the command control receiver batteries and the Azusa rate beacon. The battery deletion is possible since the requirement for additional separate power sources for command receivers has been waived by WTR; thus, it is planned to supply these receivers from the APS and IPS buses for the MOL mission.

Ground tracking stations for Azusa are located so that little data are received from the Titan III/MOL boost trajectory. Therefore, the Azusa can be deleted.

9. Weight Summary

The weight summary for the six configurations is given in Table II-6. The vehicle liftoff weights are shown with the weights for the modifications previously described.

Also shown are the delta increase in weight (above the minimum modification core) required to achieve a 99% launch probability (i.e., launch in full specification wind) with the specific payloads.

A more detailed breakdown of the delta weights is presented in the following paragraphs for the minimum core modification numbers presented in Table II-6, listing major items only.

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Table II-6 Weight Summary

	7 seg-120		2 CS-156		3 CS-156	
	With Transtage (No. 1)	Without Transtage (No. 2)	With Transtage (No. 3)	Without Transtage (No. 4)	With Transtage (No. 5)	Without Transtage (No. 6)
Booster Weight (lb)						
Core	375,535	348,637	374,697	347,777	374,052	347,919
SRM	1,364,626	1,364,626	1,948,560	1,948,560	2,537,100	2,537,100
Total Liftoff Weight	1,740,161	1,713,263	2,323,257	2,296,337	2,911,952	2,885,019
Payload Specified						
Length (ft)	54.5	74.5	58.5	78.5	61	81
Weight (lb)	28,000	28,000	33,000	33,000	42,000	42,000
△ Weight for Minimum Core Modification (lb)*	+1667	-2790	+943	-3630	+908	-3549
△ Weight for Structural Beefup to Achieve 99% Launch Probability (lb)†	0	0	+416	+604	+163	+181

*This △ weight includes added equipment, wiring, etc, to perform the MOL missions. Structural modifications included are those minimum modifications not affected by changes in payload configuration.
†Only structural beefup weights required to achieve a 99% launch probability with the specified payloads are included.

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a. 7 seg-120 with Transtage

Stage I -- Higher engine weight due to increased expansion ratio; larger thrust chamber cover plates; addition of redundant Stage I actuator system	+968 lb
Stage II -- Relocation of SRM forward outriggers; addition of Stage 0 autopilot selector; addition of equipment	+374 lb
Stage III -- Additional guidance equipment; addition of redundant Stage III actuator system; increased length of equipment module	+325 lb
	+1667 lb

b. 7 seg-120 without Transtage

Stage I -- Same as 7 seg-120 with transtage	+968 lb
Stage II -- Replacement of oxidizer tank forward skirt with the Stage II equipment module; addition of equipment formerly located in transtage; relocation of SRM forward outriggers	+1889 lb
Stage III -- Removed	-5647 lb
	-2790 lb

c. 2 CS-156 with Transtage

Stage I -- Beefup of SRM forward outrigger attachments; addition of redundant Stage I actuator system; beefup of fuel tank due to high differential SRM tailoff loads	+510 lb
Stage II -- Addition of Stage 0 autopilot selector; addition of environmental insulation; addition of equipment	+118 lb

<u>Stage III</u> -- Additional guidance equipment; redundant Stage III actuator system; increased length of equipment module — <u>+315 lb</u>

+943 lb

d. 2 CS-156 without Transtage

<u>Stage I</u> -- Same as 2 CS-156 with Transtage — <u>+510 lb</u>

<u>Stage II</u> -- Replacement of oxidizer tank forward skirt with the Stage II equipment module; addition of equipment formerly located in transtage — <u>+1507 lb</u>

<u>Stage III</u> -- Removed — <u>-5647 lb</u>

-3630 lb

e. 3 CS-156 with Transtage

<u>Stage I</u> -- Relocation of SRM forward outriggers; addition of Stage I redundant actuator system; beef-up of fuel tank due to high differential SRM tailoff loads — <u>+209 lb</u>

<u>Stage II</u> -- Same as 7 seg-120 with transtage — <u>+374 lb</u>

<u>Stage III</u> -- Same as 7 seg-120 with transtage — <u>+325 lb</u>

+908 lb

f. 3 CS-156 SRM without Transtage

<u>Stage I</u> -- Same as 3 CS-156 with transtage — <u>+209 lb</u>

<u>Stage II</u> -- Same as 7 seg-120 without transtage — <u>+1889 lb</u>

<u>Stage III</u> -- Removed — <u>-5647 lb</u>

-3549 lb

10. Environment

A comparison of the vibration levels for each of the six configurations with those of Titan IIIC is shown in Table II-7. An indication of the changes in frequency, as well as the predicted levels, are shown in the table.

The number of components requiring requalification (DAT) because of the environmental changes are listed in the following tabulation.

SRM Configuration	With Transtage	Without Transtage
7 seg-120	48	49
2 CS-156	24	24
3 CS-156	50	52

[redacted]

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Table II-7 Comparison Chart of Compartmental Vibration (grms) Levels

Compartment	Titan IIIC	7 seg-120		2 CS-156		3 CS-156	
		With Transtage	Without Transtage	With Transtage	Without Transtage	With Transtage	Without Transtage
3A A/F	63	32.0 ↓	N/A	35.2 ↓	N/A	32.0 ↓	N/A
3A Att. Con. Sys	None	28.0 N	N/A	29.6 N	N/A	28.0 N	N/A
3A Truss	16.7	10.8 ↓	N/A	9.1 ↓	N/A	10.8 ↓	N/A
3B A/F	31.6	25.3 ↓	N/A	22.0 ↓	N/A	25.3 ↓	N/A
3B Engine	26.9	26.9 ↓	N/A	26.9 ↓	N/A	26.9 ↓	N/A
2A A/F (Tie Point Fwd)	34.0	79.9 ↑	94.0 ↑	35.2 ↔	35.2 ↔	79.9 ↑	94.0 ↑
2A A/F (Tie Point Aft)	34.0	87.0 ↑	87.0 ↑	35.2 ↔	35.2 ↔	87.0 ↑	87.0 ↑
2A Truss	None	N/A	17.7 N	N/A	10.0 N	N/A	17.7 N
2A T/D	None	56.4 N	56.4 N	N/A	N/A	56.4 N	56.4 N
2B A/F	29.5	112.5 ↑	112.5 ↑	30.4 ↔	30.4 ↔	112.5 ↑	112.5 ↑
2B Truss	10.5	23.2 ↑	23.2 ↑	10.3 ↔	10.3 ↔	23.2 ↑	23.2 ↑
2B T/D	None	65.1 ↑	65.1 ↑	N/A	N/A	65.1 ↑	65.1 ↑
2C A/F	83.7	28.6 ↓	28.6 ↓	28.6 ↓	28.6 ↓	28.6 ↓	28.6 ↓
2C Engine	68.0	68.0 ↔	68.0 ↔	68.0 ↔	68.0 ↔	68.0 ↔	68.0 ↔
1A A/F (at Tie Point)	40.1	N/A	N/A	34.2 ↗	34.2 ↗	N/A	N/A
1A A/F	188.0	33.0 ↓	33.0 ↓	None	None	33.0 ↓	33.0 ↓
1A T/D	59.4	34.0 ↓	34.0 ↓	67.8 ↑	67.8 ↑	34.0 ↓	34.0 ↓
1B A/F	43.5	41.0 ↔	41.0 ↔	58.1 ↑	58.1 ↑	58.1 ↑	58.1 ↑
1B T/D	35.7	21.7 ↓	21.7 ↓	32.4 ↓	32.4 ↓	32.4 ↓	32.4 ↓
1C A/F	45.0	38.3 ↖	38.3 ↖	40.0 ↖	40.0 ↖	40.0 ↖	40.0 ↖
1C Engine	48.7	48.8 ↔	48.8 ↔	48.8 ↔	48.8 ↔	48.8 ↔	48.8 ↔
S1A, S2A A/F	52.4	52.4 ↔	52.4 ↔	52.4 ↔	52.4 ↔	52.4 ↔	52.4 ↔
S1A, S2A Truss	None	15.9 N	15.9 N	15.9 N	15.9 N	15.9 N	15.9 N
S1B, S2B A/F	46.2	46.2 ↔	46.2 ↔	65.2 ↑	65.2 ↑	73.3 ↑	73.3 ↑
TVC, Fwd	43.5	41.0 ↔	41.0 ↔	58.1 ↑	58.1 ↑	58.1 ↑	58.1 ↑
TVC, Aft	45.0	38.3 ↖	38.3 ↖	40.0 ↖	40.0 ↖	40.0 ↖	40.0 ↖

↑ Above Titan IIIC Specification, Requalification Required.
↔ Near Titan IIIC Specification, No Requalification Required.
↓ Below Titan IIIC Specification, No Requalification Required.
↗ Above and Right Frequency Shift, Requalification Required.
↖ Above and Left Frequency Shift, Requalification Required.
N New Specification, Qualification May be Required.

[redacted]

B. SRM CONFIGURATION

Three SRM configurations were used as references for the six configurations. The 7 seg-120 is designed using the present 5 seg-120 as the basis. The 2 CS-156 and 3 CS-156 motors represent new designs, but utilize the Titan IIIC-type thrust vector control (TVC) system.

The three SRMs are described below. Table II-8 lists the general characteristics of the various SRMs and Fig. II-3 and II-4 present thrust and acceleration data. See Fig. II-2 for typical configuration.

1. Motors

7 seg-120 - The 7 seg-120 configuration uses an eight-point star grain design in the forward and aft segments with a combination of conical or cylindrical shapes in the seven center segments. The forward end of each center segment is partially inhibited. The engine nozzle is conically shaped and canted 6 deg away from the core longitudinal axis.

2 CS-156 and 3 CS-156 - The 2 and 3 CS-156 configurations use a 10-point star grain design in the forward segment with either a conical frustum or a combination of conical and cylindrical shapes in the center and/or aft segments. No inhibitor is used. The engine nozzle is partially submerged and is also canted 6 deg.

2. Thrust Vector Control (TVC) Systems

TVC is obtained for the three configurations by pressurized N_2O_4 injection into the divergent portion of the engine nozzle. Each SRM configuration uses 24 injectant valves (six in each quadrant) canted 6 deg in the direction of the exhaust stream to produce a minimum thrust vector deflection of about 4 deg for any SRM thrust level condition. In addition, the TVC system can produce a minimum slew rate of the thrust vector equivalent to about 10 deg/sec.

Hydraulic power supplied to operate the servo-controlled injectant valves is obtained from redundant electrically-driven pumps.

Table II-8 SRM Characteristics

Characteristic	7 seg-120	2 CS-156	3 CS-156
Total Action Time (sec)	124.3	138.0	155.0
Throat Diameter (in.)	40.7	39.1	45.5
TVC Valves	24	24	24
Usable Inject Fluid (lb)	6,248	9,988	11,050
Length, Overall (in.)	1311.5	1146.5	1410.5
Nozzle Type	Conical (0% Submerged)	Conical (19% Submerged)	Conical (17% Submerged)
Expansion Ratio	9.6	10.2	8.0
Total Weight* (lb)	1,364,626	1,948,560	2,537,100
Overall Mass Fraction	0.864	0.881	0.888
Total Vacuum Impulse* (lb-sec)	308×10^6	444×10^6	588×10^6
Maximum Sea Level Thrust* (lb)	2.97×10^6	4.38×10^6	5.81×10^6
MEOP† (psi)	920	1190	1190
Liftoff Acceleration* (g)	1.74	1.84	1.99
Maximum Dynamic Pressure (lb/ft^2)	884	906	886
Aeroheating Indicator $(ft\text{-}lb/ft^2)$	93.0×10^6	97.5×10^6	86.6×10^6
Maximum Inflight Acceleration* (g)	3.08	2.96	3.00

*Two SRMs.
†Maximum expected operating pressure.

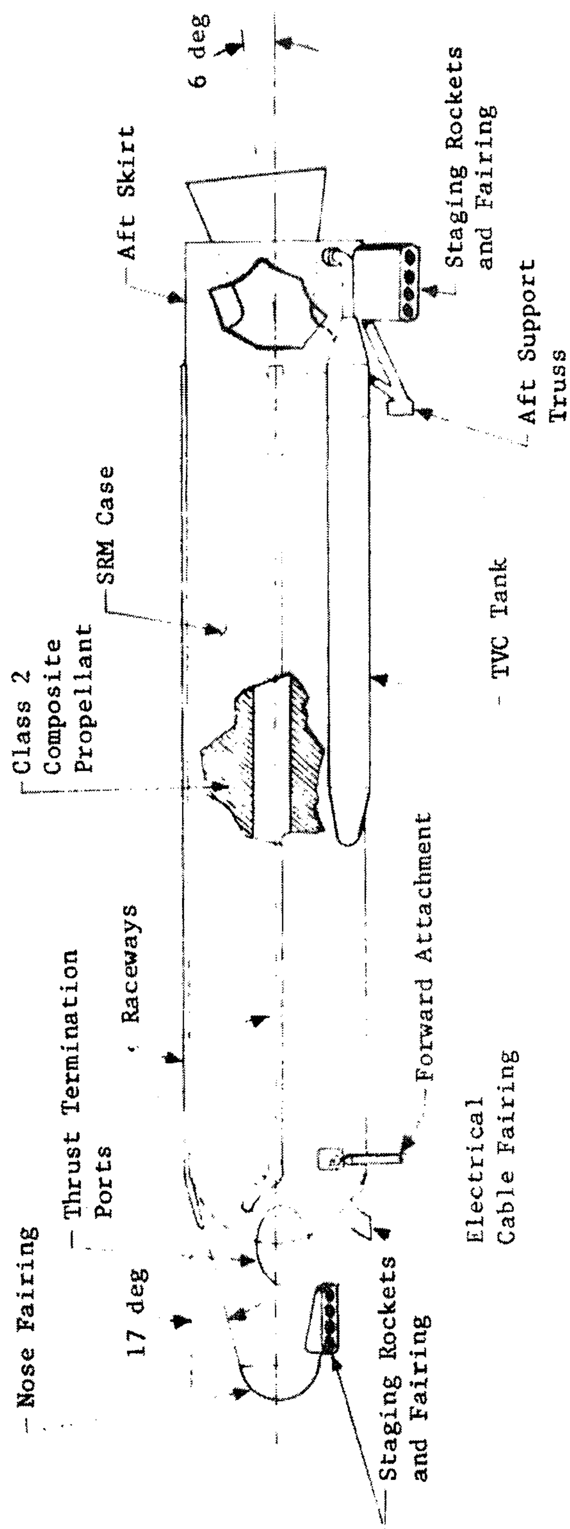

Fig. II-2 Typical SRM Configuration

[redacted]

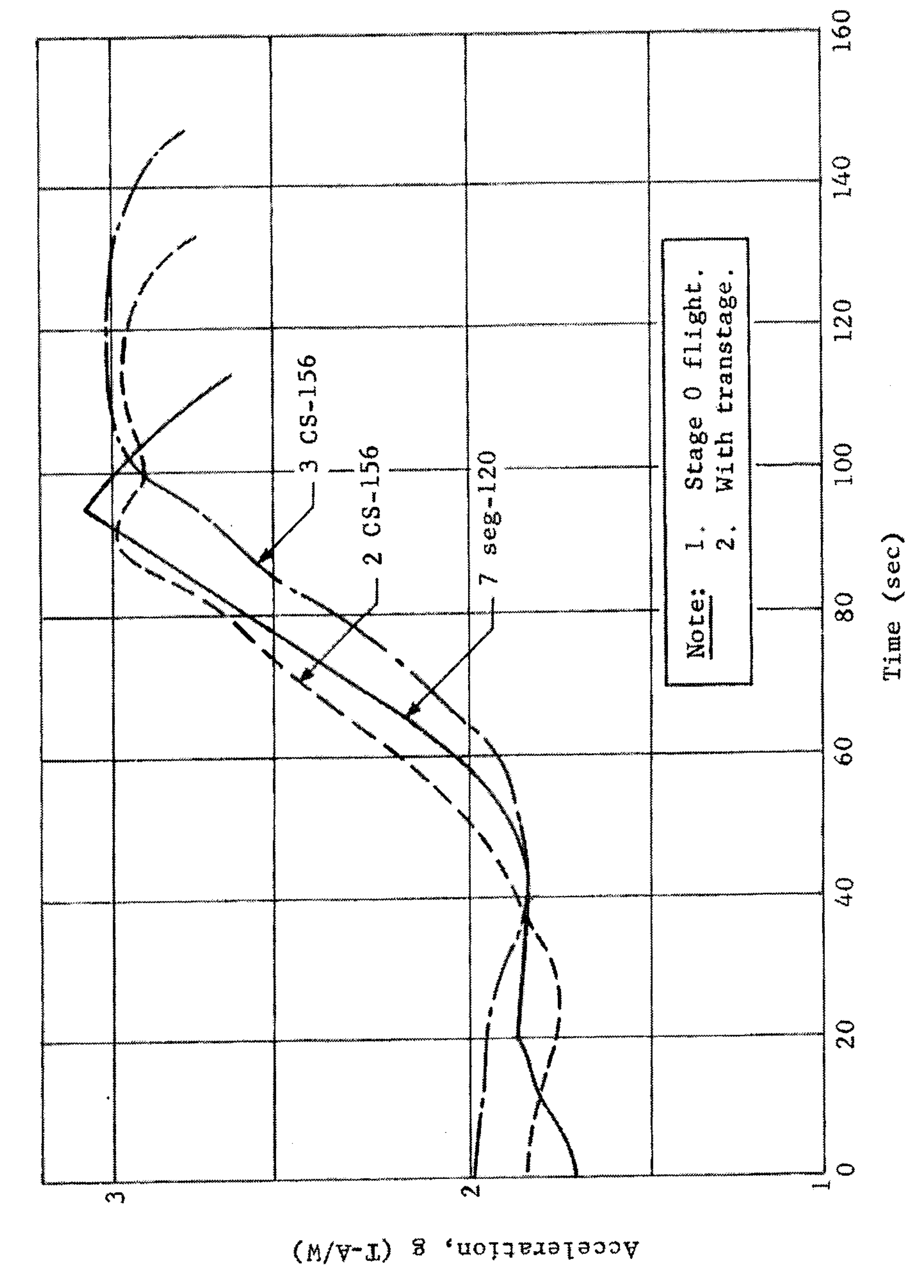

Fig. II-3 SRM Acceleration vs Time

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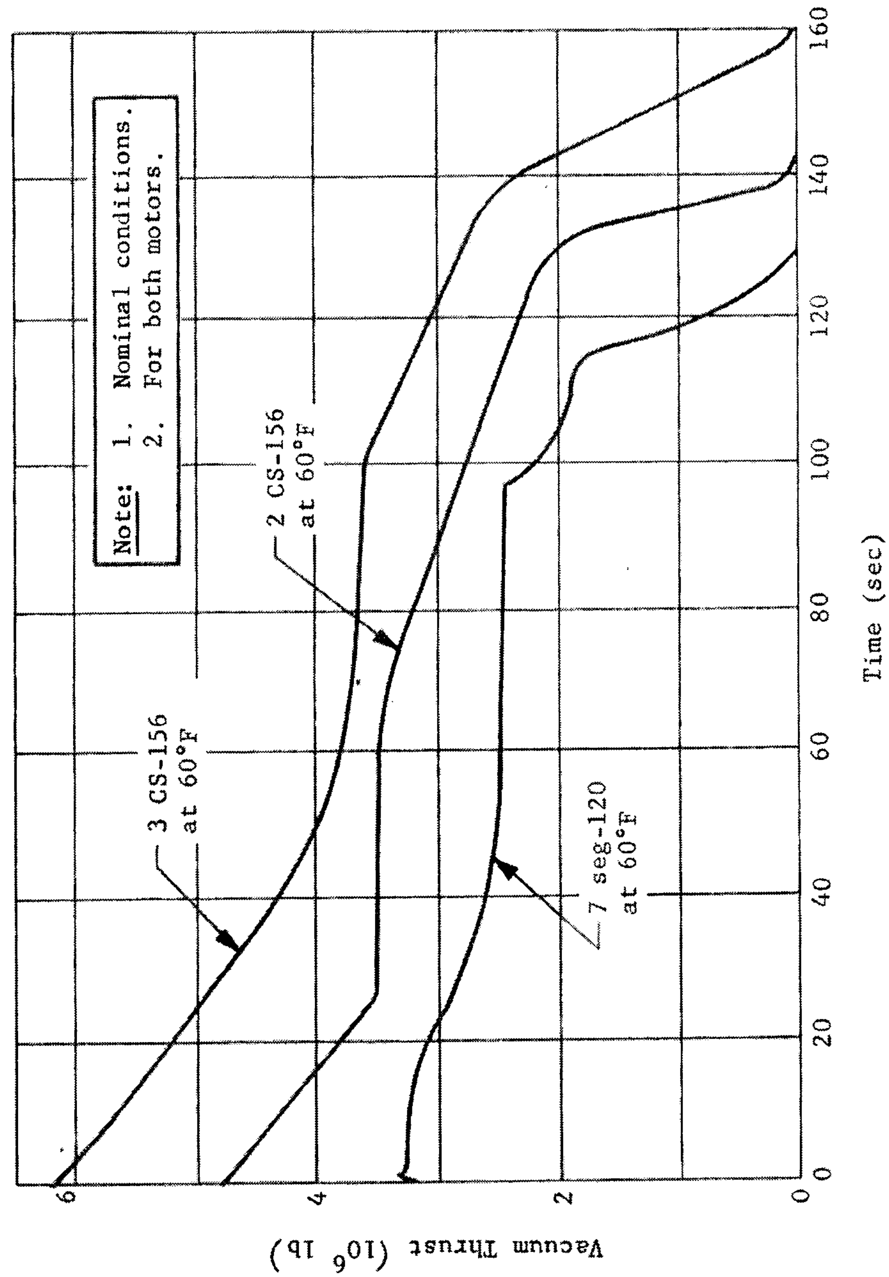


Fig. II-4 SRM Thrust vs Time

The electronics to drive the TVC valves are located in Stage 0 for the 7 seg-120 configuration. For both 156-in. designs, the TVC valves are driven directly from the core autopilot.

3. Structure

Each configuration contains a forward nose fairing and an aft skirt joined to the SRM segments (Fig. II-2). The nose fairing encloses the head segment containing the ignitor and thrust termination ordnance, and contains the three electrical umbilicals, forward staging rockets (four for 7 seg-120, five for 2 and 3 CS-156), two thrust termination ports, and the forward core vehicle attachment point.

SRM segment cases are joined by a pin and clevis-type joint.

The TVC system (7 seg-120 only), electrical and hydraulic power supplies, instrumentation packages, and the aft support truss are mounted on the SRM nozzle assembly inside the aft skirt. The aft skirt also contains the 4 aft separation rockets.

The TVC pressurization and N_2O_4 tanks, the destruct raceway, and electrical raceway are attached external to the SRM segments.

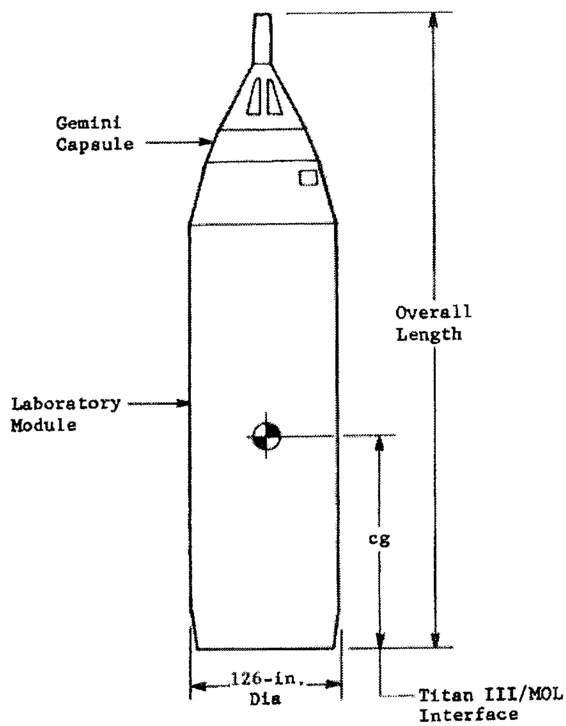
C. PAYLOAD CONFIGURATIONS

The general configurations of the payloads used are shown in Fig. II-1. The Laboratory Module, of lengths shown, is 126 in. in diameter. The payloads used are topped by a Gemini B capsule.

Lengths, weights, and centers of gravity (cg) are shown in Table II-9.

Table II-9 Spacecraft Configuration

Item	7 seg-120		2 CS-156		3 CS-120	
	With Transtage	Without Transtage	With Transtage	Without Transtage	With Transtage	Without Transtage
Contract Payload Length (ft)	54.5	74.5	58.5	78.5	61.0	81.0
Lengths Checked for Maximum Length Study (ft)	54.5 65.0 75.0	64.7 75.2 85.2	50.0 58.5 65.0	60.2 68.7 75.2	50.0 61.0 72.0	60.2 71.2 82.2
Payload cg (each corresponds to maximum lengths above) (ft)	29.6 35.2 40.6		27.1 31.7 35.1		27.1 33.1 39.2	
Payload Weight (lb)	28,000	28,000	33,000	33,000	42,000	42,000



Spacecraft Geometry

III. PERFORMANCE AND TRAJECTORY CONSTRAINTS

This chapter presents the payload capability, mission requirements, trajectory constraints, evaluation of the Gemini trajectory constraints, and trade studies that were performed. The payload capability of the basic Titan III core (with and without transtage) in conjunction with 7 segment 120-in. SRM (7 seg-120), 2 center segment 156-in. SRM (2 CS-156), and 3 center segment 156-in. SRM (3 CS-156) is presented for an 80-n-mi circular polar orbit [Western Test Range (WTR) launch] and for due east launches from the Eastern Test Range (ETR). The ETR performance capability is presented for a 100-n-mi injection using a park orbit ascent technique, or a direct injection ascent technique. High-altitude circular and elliptical orbit payload capability is presented for the ETR launches.

A. PERFORMANCE TRADE STUDIES

Design criteria were established to evaluate the 7 seg-120, 2 CS-156, and 3 CS-156 SRMs. These criteria were as follows:

1) Longitudinal load factor (T-A)/W within 5 sec after liftoff, larger than 1.6 g;

2) Maximum (T-A)/W (Stage 0 phase), less than 3.2 g;

3) Maximum dynamic pressure (q_{max}), less than 900 lb/ft^2;

4) Maximum differential thrust at Stage I engine ignition, less than 280,000 lb;

5) Maximum aerodynamic heating indicator $\frac{1}{2}\int_0^t \rho V^3 \, dt = 95 \times 10^6 \ ft\text{-}lb/ft^2$,

where:

T = Vehicle centerline thrust (lb)

A = Axial aerodynamic force (lb)

W = Vehicle weight (lb)

ρ = Atmospheric density $(\text{slug}/\text{ft}^3)$

V = Relative velocity (fps)

t = Burn time (sec).

These criteria were established to assure satisfaction of vehicle and payload constraints, such as acceptable thrust-to-weight ratio at liftoff to avoid excessive launch drift; specification of q_{max} allowable as a payload and vehicle constraint; maximum longitudinal load factor to satisfy core structural design criteria; and differential thrust at Stage I ignition to meet Stage I longeron design loads criteria. Thrust regressivity was not specified so the SRM manufacturer could exercise latitude in propellant grain design and internal motor ballistics. Core configuration data were given to each of the SRM manufacturers to allow them to tailor the SRM thrust, total impulse, mass fraction and mass decay rate to achieve optimum payload performance while meeting the constraint conditions. SRM manufacturers supplied data on the 7 seg-120, the 2 CS-156, and the 3 CS-156 SRMs. Detailed SRM characteristics for the three configurations are presented in Table III-1. The thrust time histories are depicted in Fig. III-1.

The 80-n-mi WTR nominal performance (with margin withheld) was obtained by using the SRM characteristics presented in Table III-1. This performance capability is shown on Fig. III-2. Note that payload weight is almost directly proportional to the SRM total impulse.

Representative (T-A)/W (longitudinal load factor) time histories with tabulation of q_{max}, (T-A)/W near liftoff, (T-A)/W (maximum), and aerodynamic heating indicator are shown in Fig. III-3. Several meetings were held between the SRM manufactures and the Martin Company to obtain the SRM characteristics (Table III-1, Fig. III-1, and III-2) that would satisfy the vehicle and trajectory constraints (Fig. III-3) and yield the maximum payload capability.

Table III-1 Solid Rocket Motor Characteristics

Characteristic	7 seg-120	2 CS-156	3 CS-156
Total Impulse (lb-sec)	307,760,000	447,552,000	588,026,000
Vacuum Specific Impulse (lb-sec/lb)	269.3	266.5	266.2
Impulse Propellant (lb)	1,142,648	1,680,000	2,208,900
TVC Fluid Retained (Included in Burnout Weight) (lb)	8,500	12,000	8,320
Ablative Material Expended (lb)	11,720	2,000	2,000
Burnout Weight (lb)	189,258	245,560	296,110
Initial Weight (lb)	1,364,626	1,948,560	2,565,020
Propellant Weight / Initial Weight	0.837	0.862	0.861
Acceleration, First 5 sec with Maximum Payload and Best Propellant Load in Transtage (g)	1.74	1.84	1.97
Maximum Acceleration with Maximum Payload and Best Propellant Load in Transtage (g)	3.08	2.96	3.00

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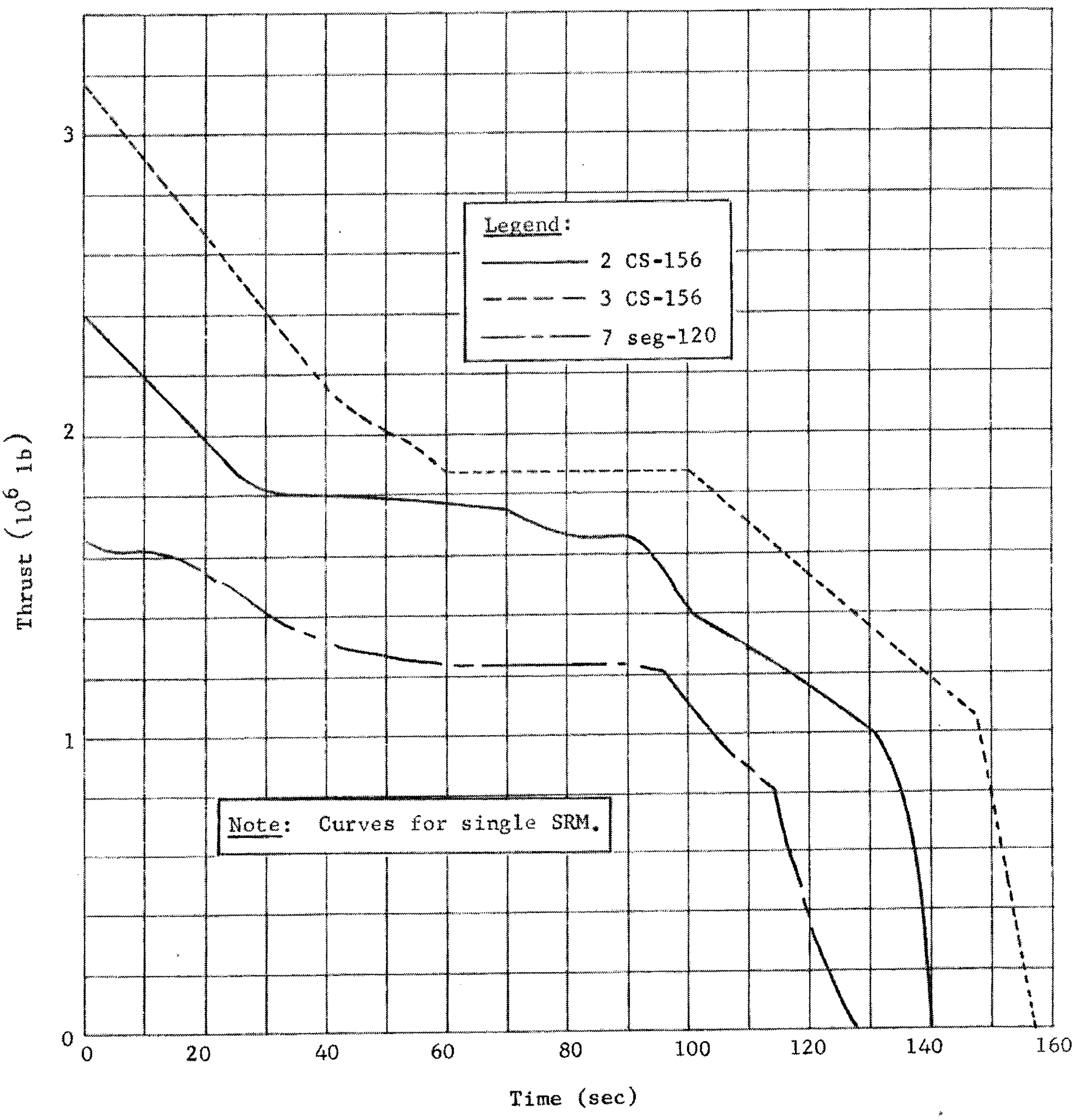

Fig. III-1 SRM Thrust Characteristics

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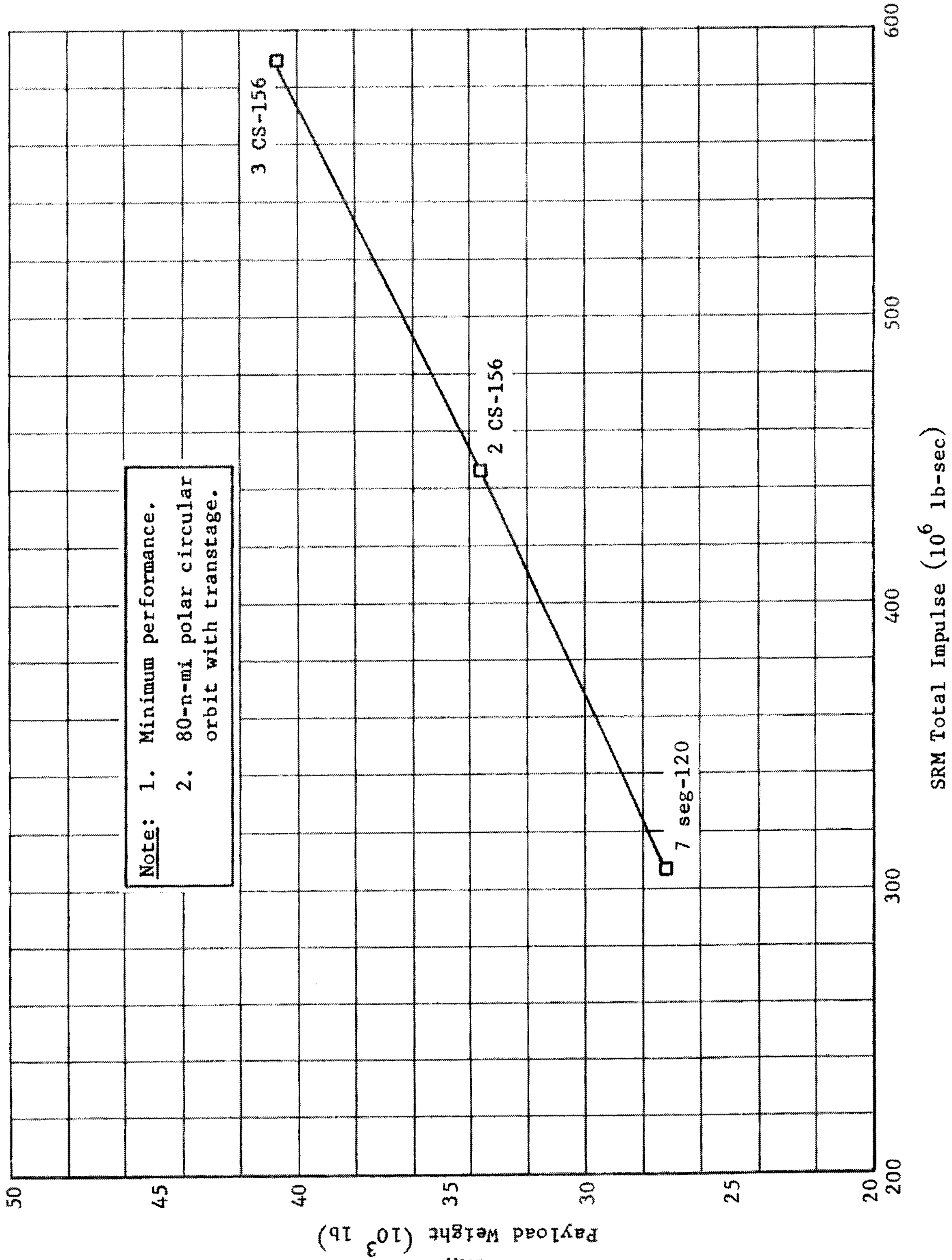

Fig. III-2 WTR Payload Capability

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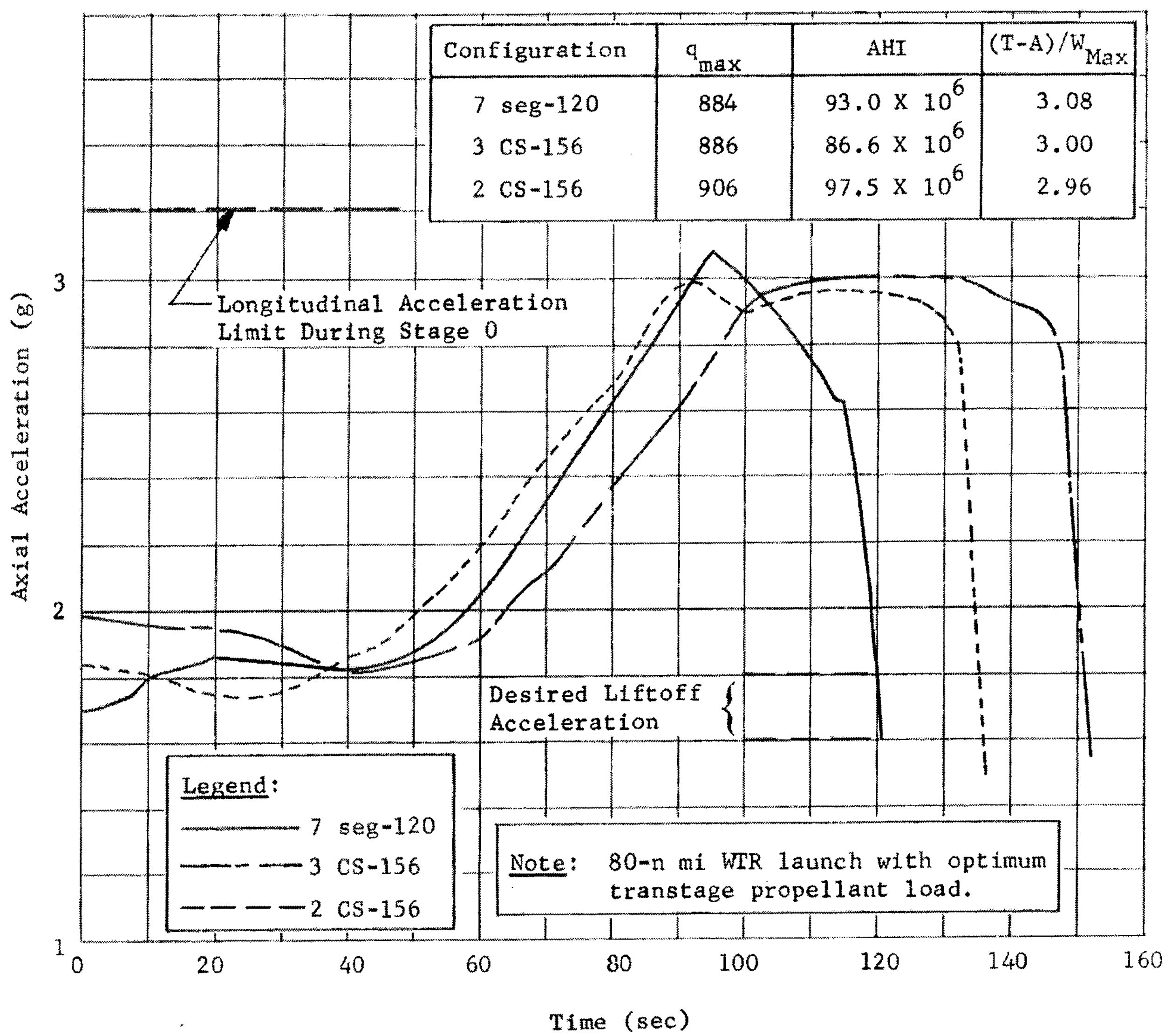

Fig. III-3 Axial Acceleration Profile

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B. PERFORMANCE CAPABILITY

The performance capability for the selected MOL configuration vehicle was developed using the SRM configurations identified in the Performance Trade Studies (Section A of this chapter). WTR and ETR performance capability was determined for the 7 seg-120, 2 CS-156 and 3 CS-156 SRM configurations with and without the transtage. The vehicle weights, SRM characteristics, liquid engine data, propellant loads, and SRM staging sequence are presented in detail in Section C of this chapter, Vehicle Characteristics, for the six configurations of interest.

The pertinent ground rules that were used in the development of the reference trajectories are:

1) Initial azimuth from WTR, 182 deg; from ETR, 90 deg.

2) A three-degree-of-freedom digital computer trajectory simulation program incorporating an oblate rotating earth model was used.

3) The ascent trajectories were shaped by using an initial pitchover rate from 10 to 20 sec, a second pitch rate from 20 to 30 sec to drive the angle-of-attack to zero, a zero-lift pitch control through the remaining portion of Stage 0 operation, followed by a constant pitch rate from 150 sec to burnout of the final stage.

4) The core engines were balanced at 60°F and 60°F propellants were used in all liquid and solid stages.

5) Stage I engine expansion ratio was modified from 8:1 to 15:1 for the 7 seg-120 SRM configuration.

6) TVC requirements (flow rates for control and dump schedule and resulting thrust augmentation were representative of those used on Titan IIIC.

7) The Stage I engines were ignited during SRM tailoff when the axial acceleration decayed to 1.5 g.

8) The SRMs were jettisoned 1 sec after burnout.

9) The required velocity margin for minimum performance was estimated by root-sum-squared 2½% of the ideal velocity of each stage. A burning time margin equivalent to this velocity was retained in the final stage. The nominal performance payload capability does not assume a propellant margin is retained in the final vehicle stage.

1. Western Test Range

Vehicle performance capability for launches from WTR are presented in Table III-2. This table shows the performance for the various vehicles (both with and without transtage) in terms of weight of payload that can be injected into an 80-n-mi circular polar orbit. The transtage propellant load was optimized to yield the maximum payload weight for each of the configurations using the transtage. In all cases, the required velocity margin (in terms of reserve propellant) remains in the final stage.

Table III-2 WTR Performance Summary, WTR Launch, Polar 80-n-mi Circular Orbit

Case	Configuration	Payload* (lb)	Burnout Weight (lb)	Propellant Margin (lb)	Maximum Dynamic Pressure (lb/ft^2)	Heating Indicator (10^6 $ft\text{-}lb/ft^2$)	Unused TVC Fluid (lb)
1	7 seg-120, 15:1 Stage I Expansion Ratio (with Transtage)	27,245	33,936	1240	884	93.0	8,500
2	Same as Case 1, Except no Transtage	28,306	38,717	1638	913	96.7	8,500
3	2 CS-156, 8:1 Stage I Expansion Ratio (with Transtage)	33,613	40,760	1684	906	97.5	17,000
4	Same as Case 3 Except no Transtage	35,030	45,386	2000	927	101.2	12,000
5	3 CS-156, 8:1 Stage I Expansion Ratio (with Transtage)	40,669	48,142	2010	886	86.6	8,320
6	Same as Case 5 Except no Transtage	42,109	53,163	2300	904	89.3	8,320

*Minimum performance.

Pertinent trajectory parameter plots such as altitude-inertial velocity, altitude-time, and dynamic pressure-time are presented in Fig. III-4, III-5, and III-6 respectively, for the vehicle configurations with transtage. Figures III-7, III-8, and III-9 are for the configurations without transtage.

The instantaneous impact points (IIP) are shown in Fig. III-10 thru III-15 for the six configurations. Step 1 and 2 impact points along with intermediate time points are presented on the figures.

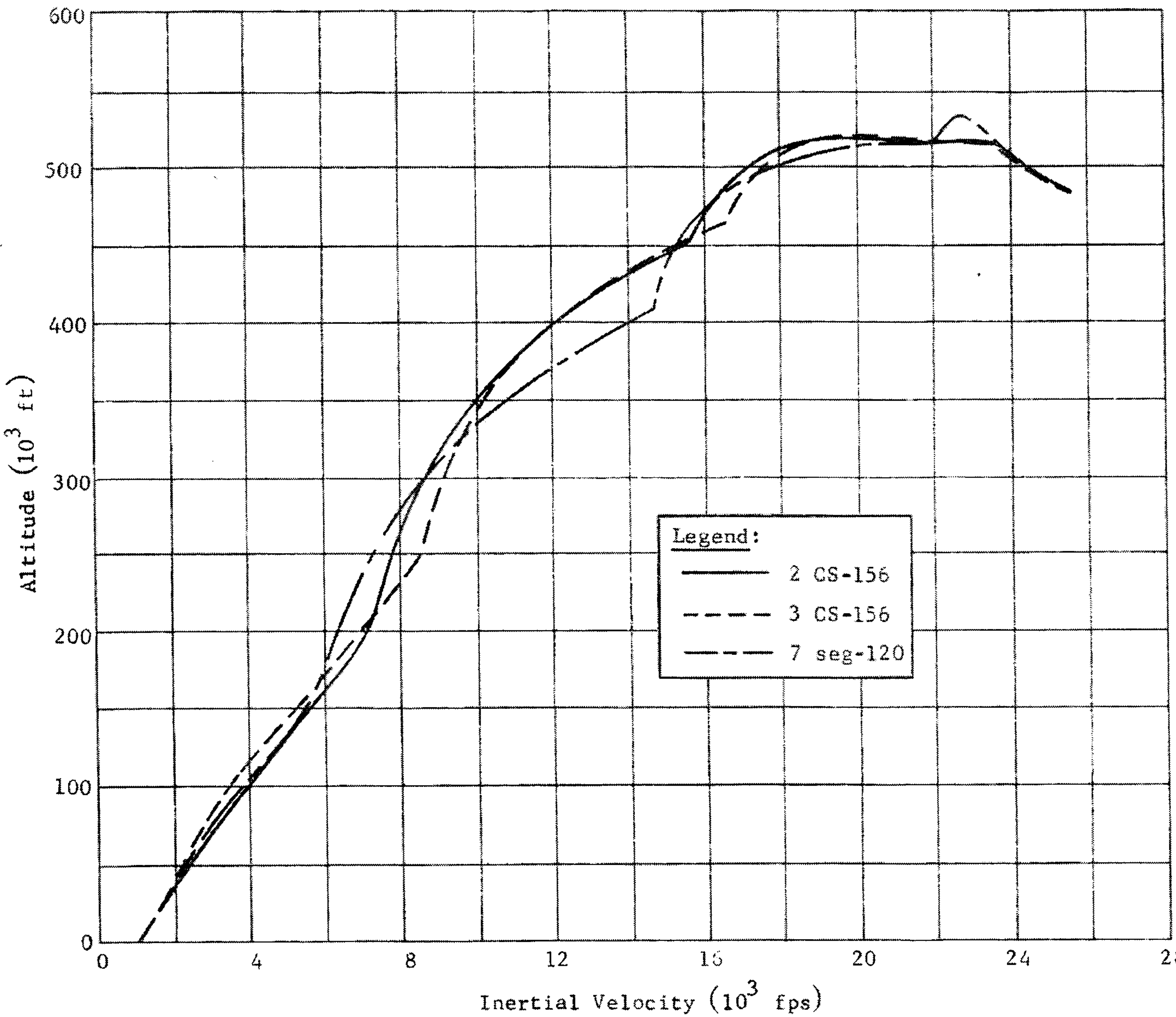

Fig. III-4 Altitude-Velocity Plot, South WTR Launch with Transtage

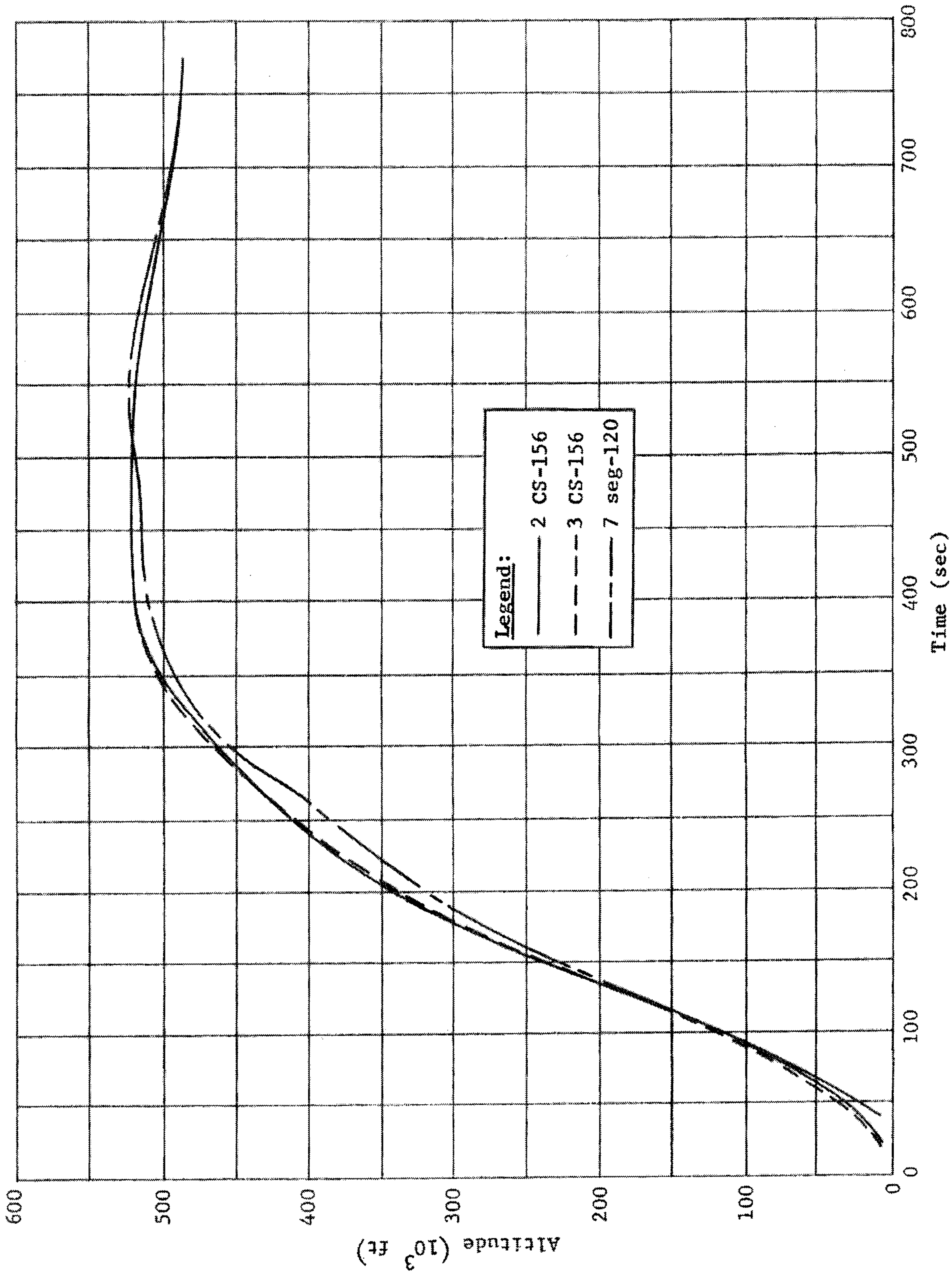

Fig. III-5 Altitude-Time Plot, South WTR Launch with Transtage

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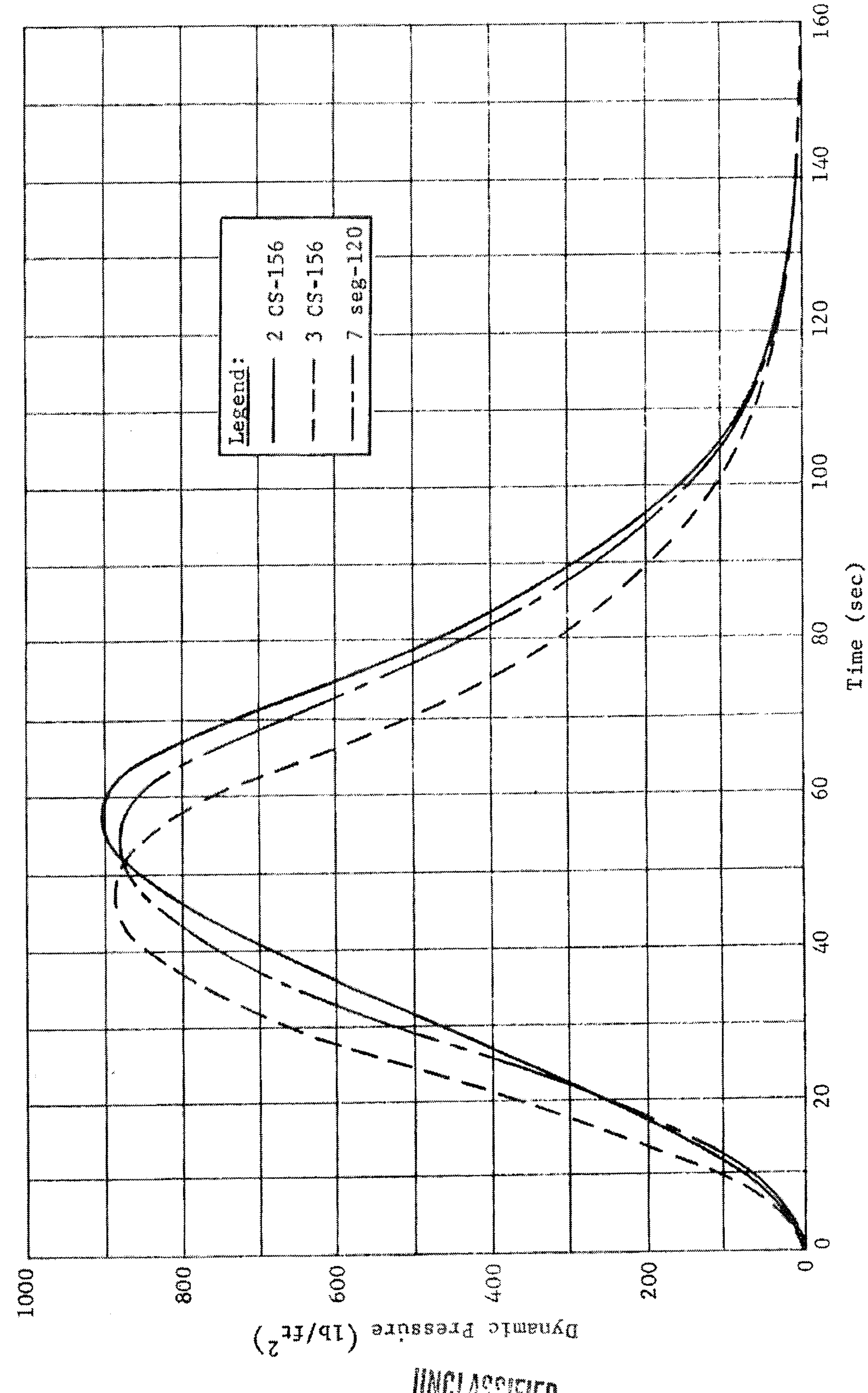

Fig. III-6 Dynamic Pressure-Time Plot, South WTR Launch with Transtage

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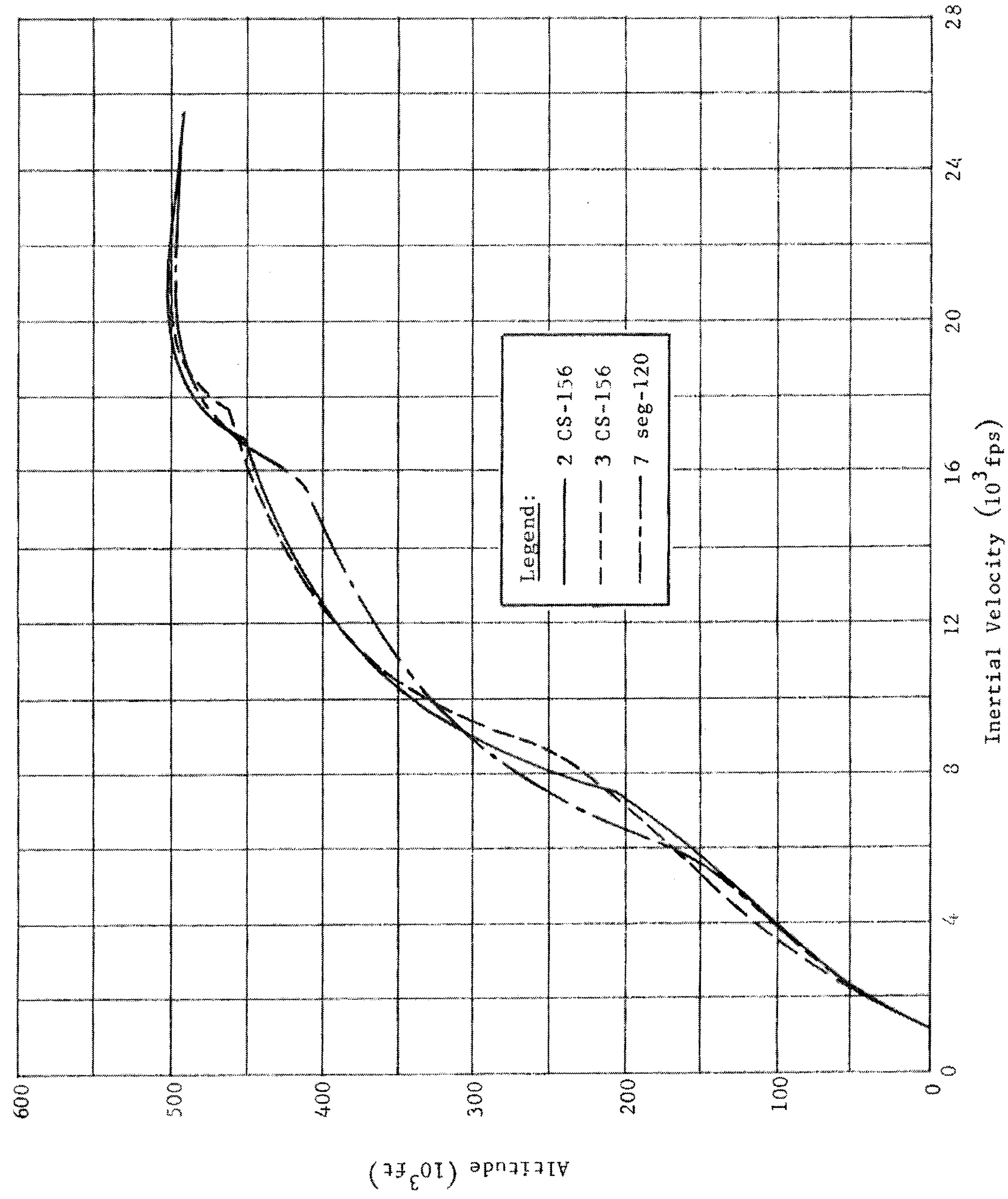

Fig. III-7 Altitude-Velocity Plot, South WTR Launch without Transtage

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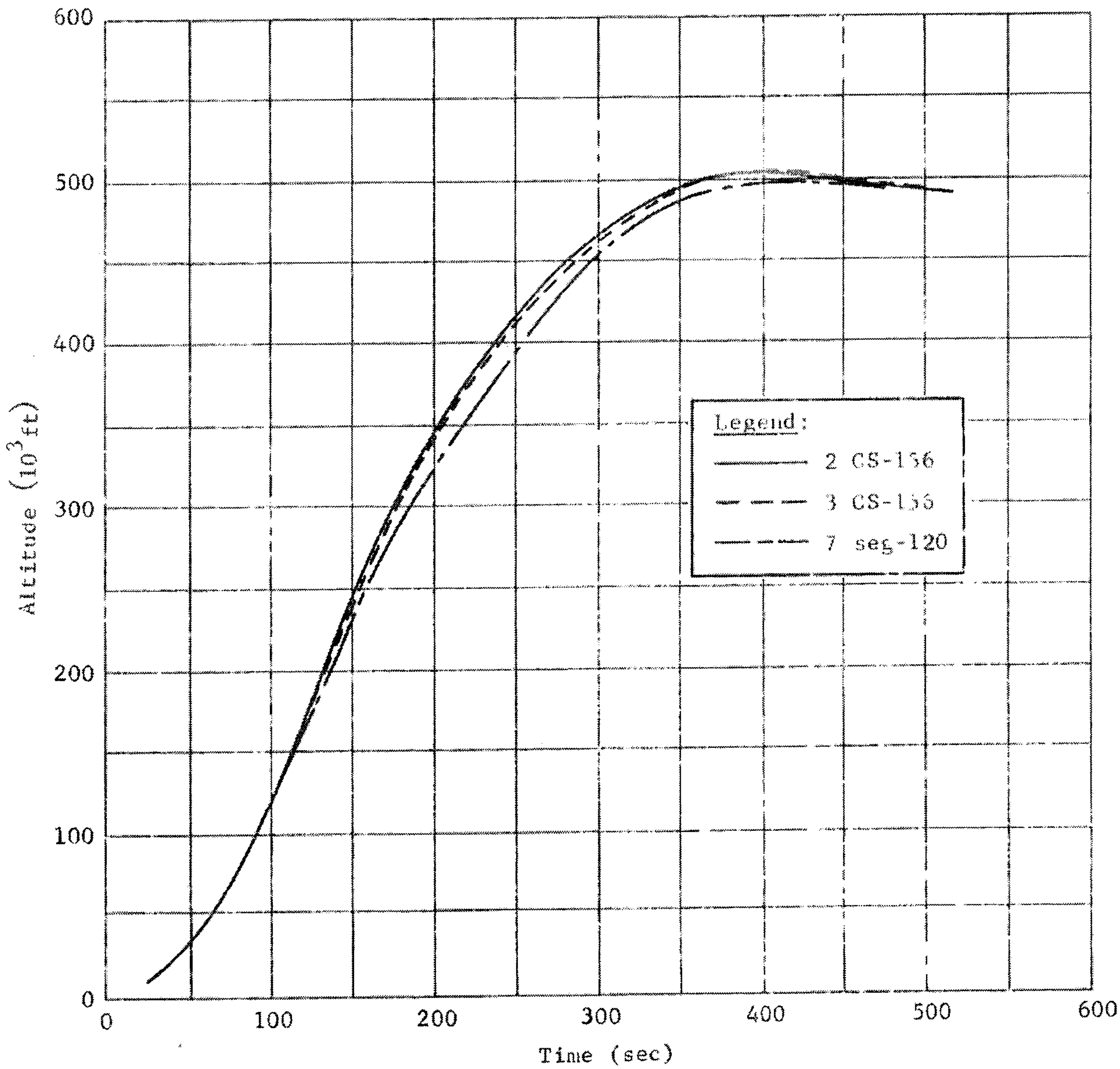

Fig. III-8 Altitude-Time Plot, South WTR Launch without Transtage

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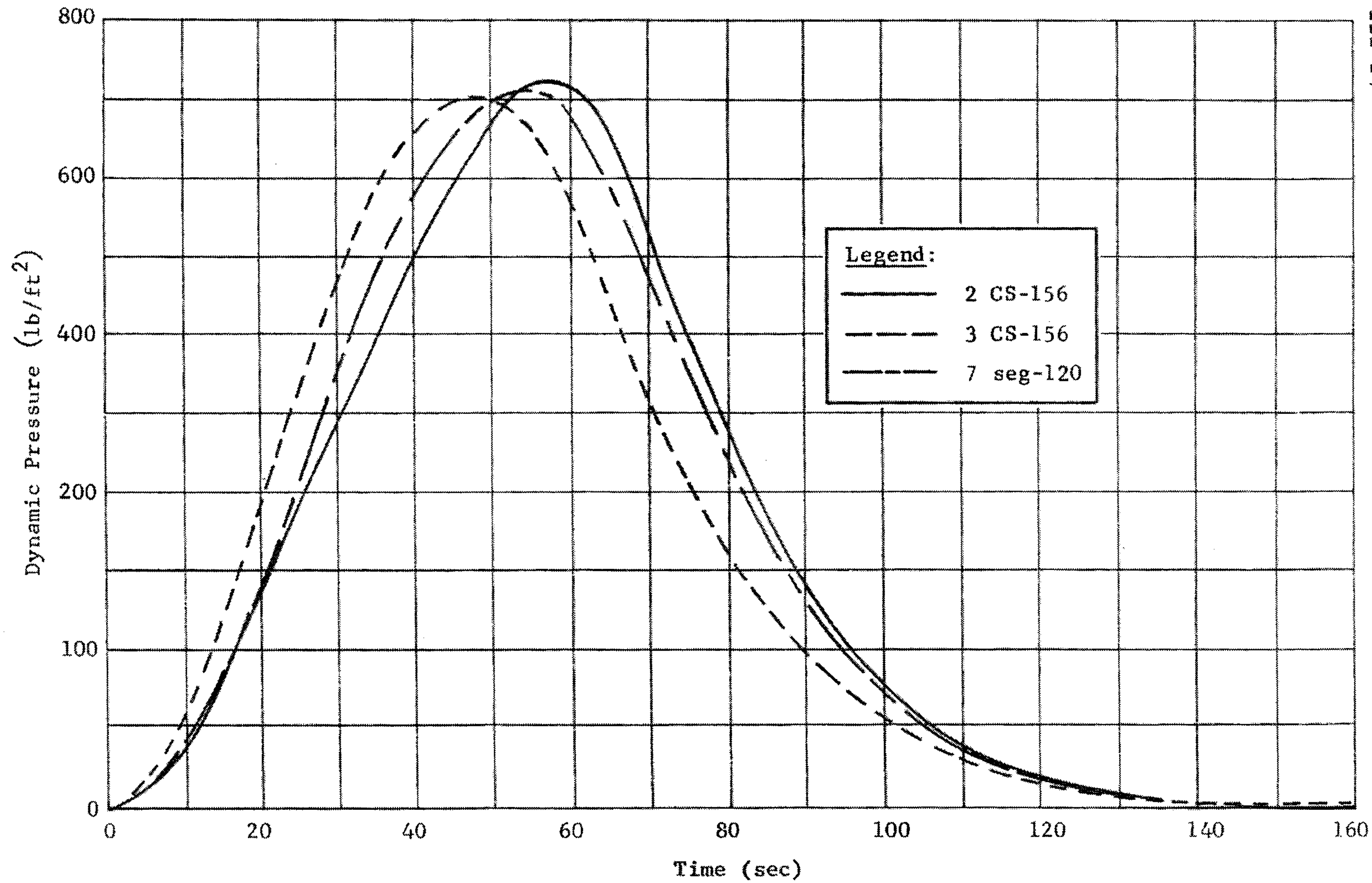

Fig. III-9 Dynamic Pressure-Time Plot, South WTR Launch without Transtage

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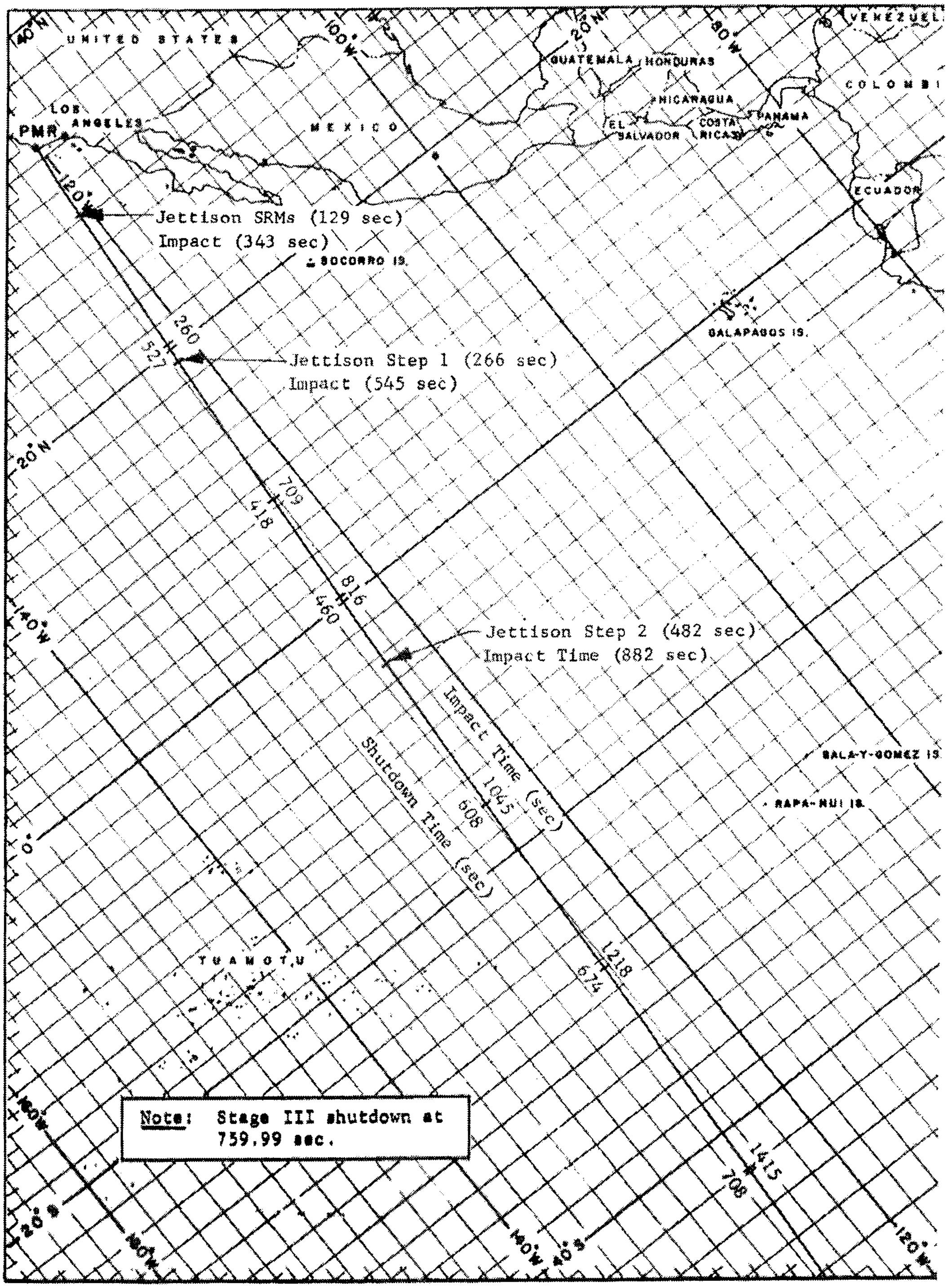

Fig. III-10 Instantaneous Impact Points, 7 seg-120 with Transtage

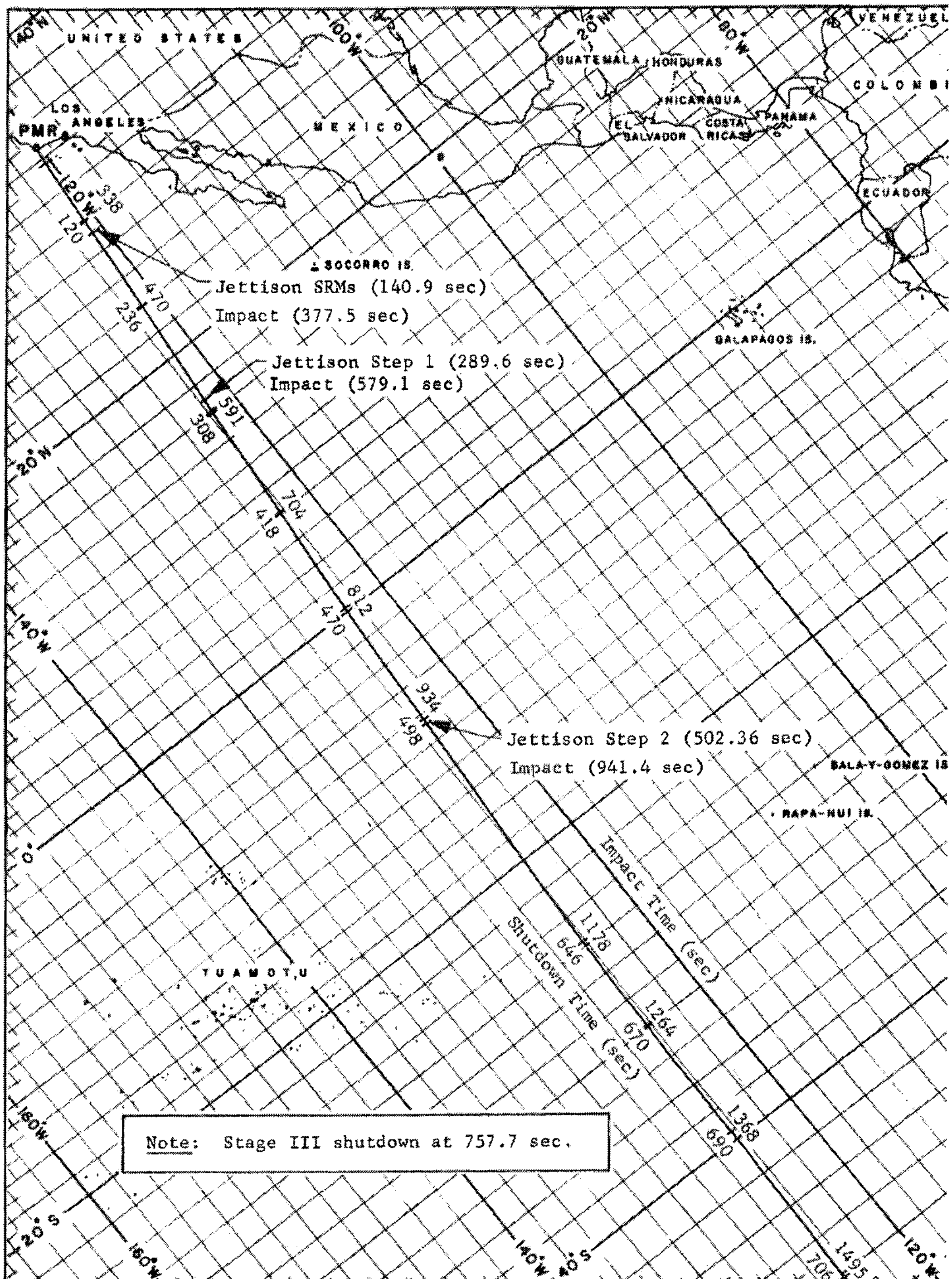

Fig. III-11 Instantaneous Impact Points, 2 CS-156 with Transtage

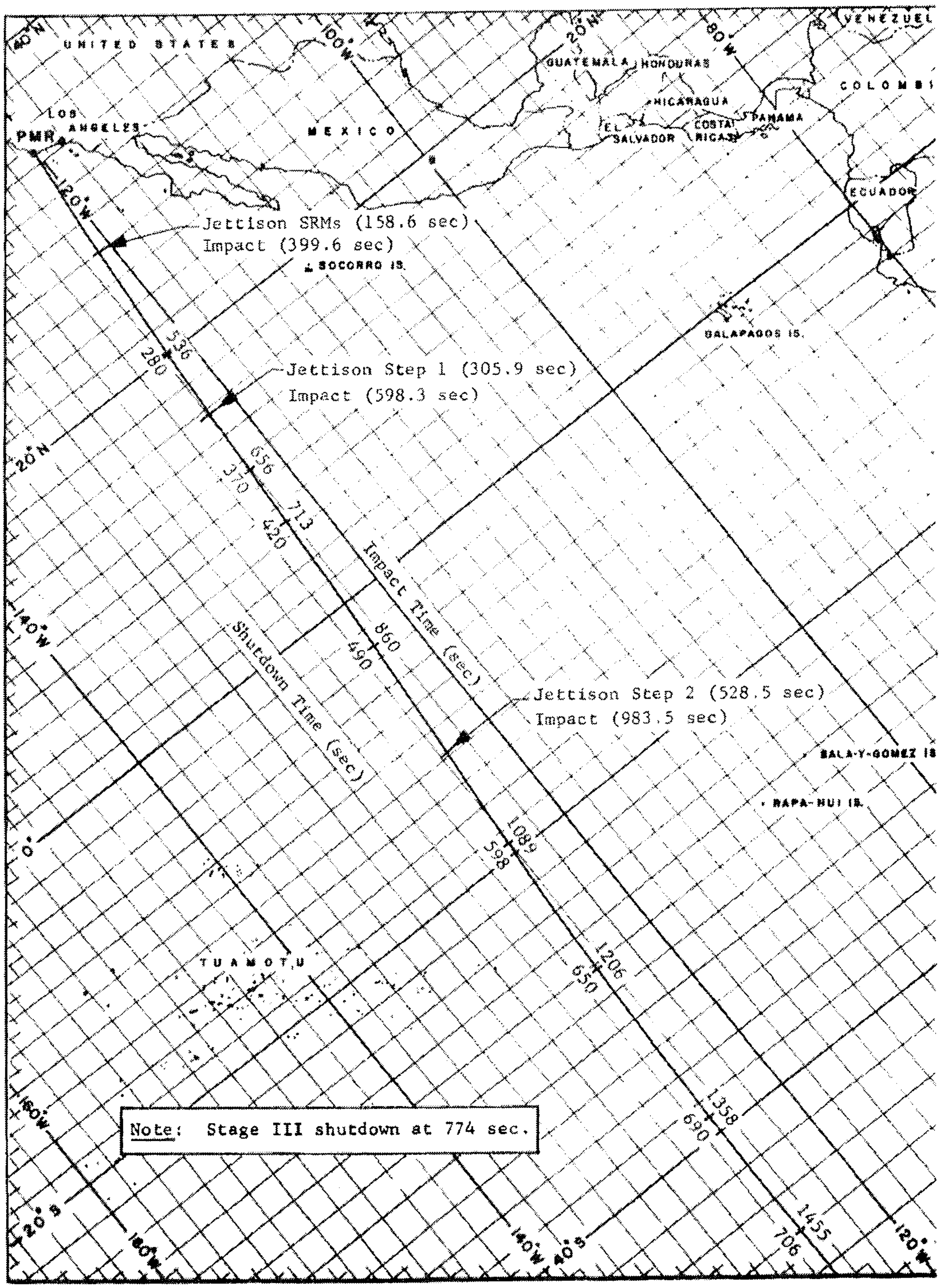

Fig. III-12 Instantaneous Impact Points, 3 CS-156 with Transtage

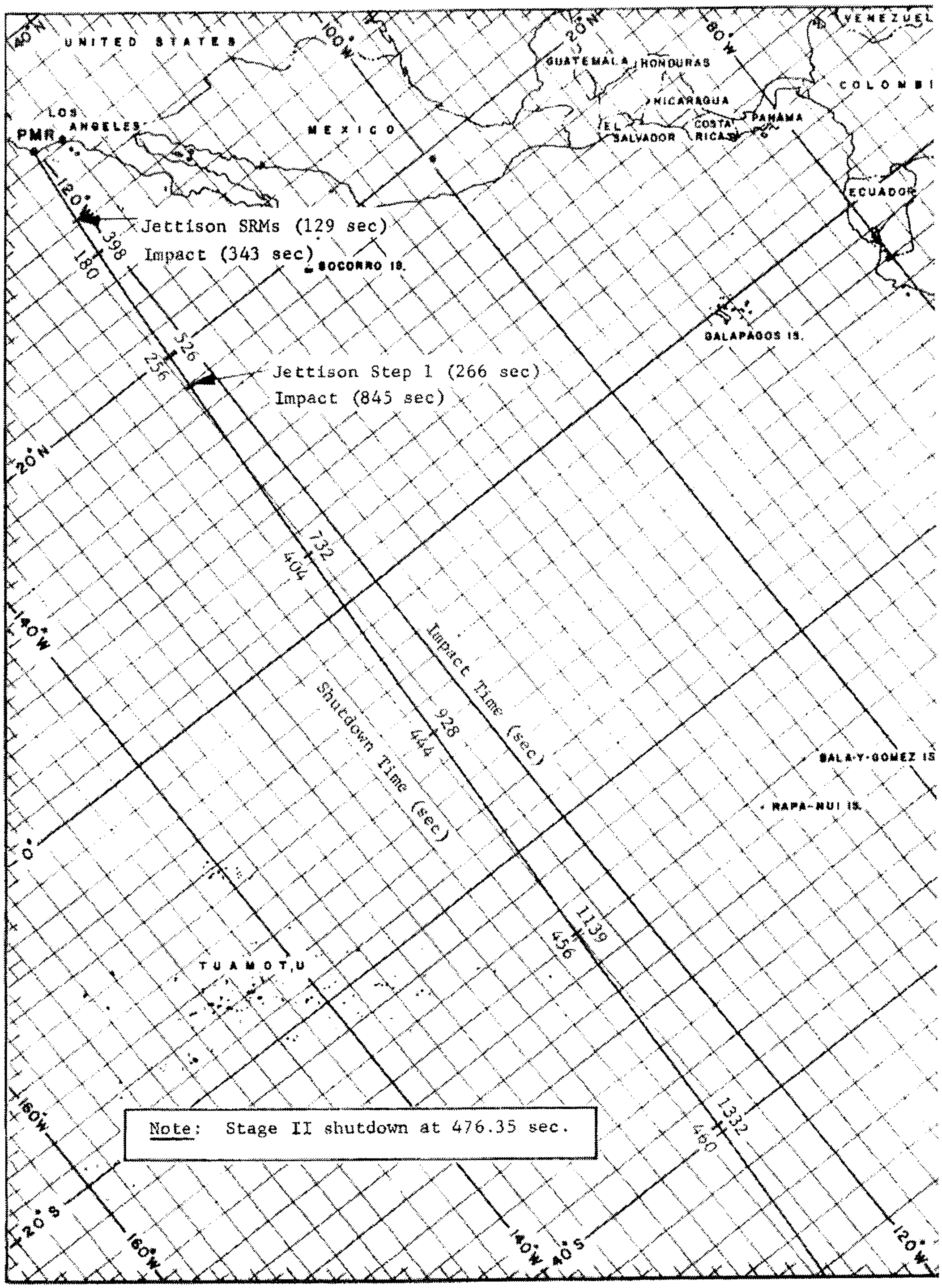

Fig. III-13 Instantaneous Impact Points, 7 seg-120 without Transtage

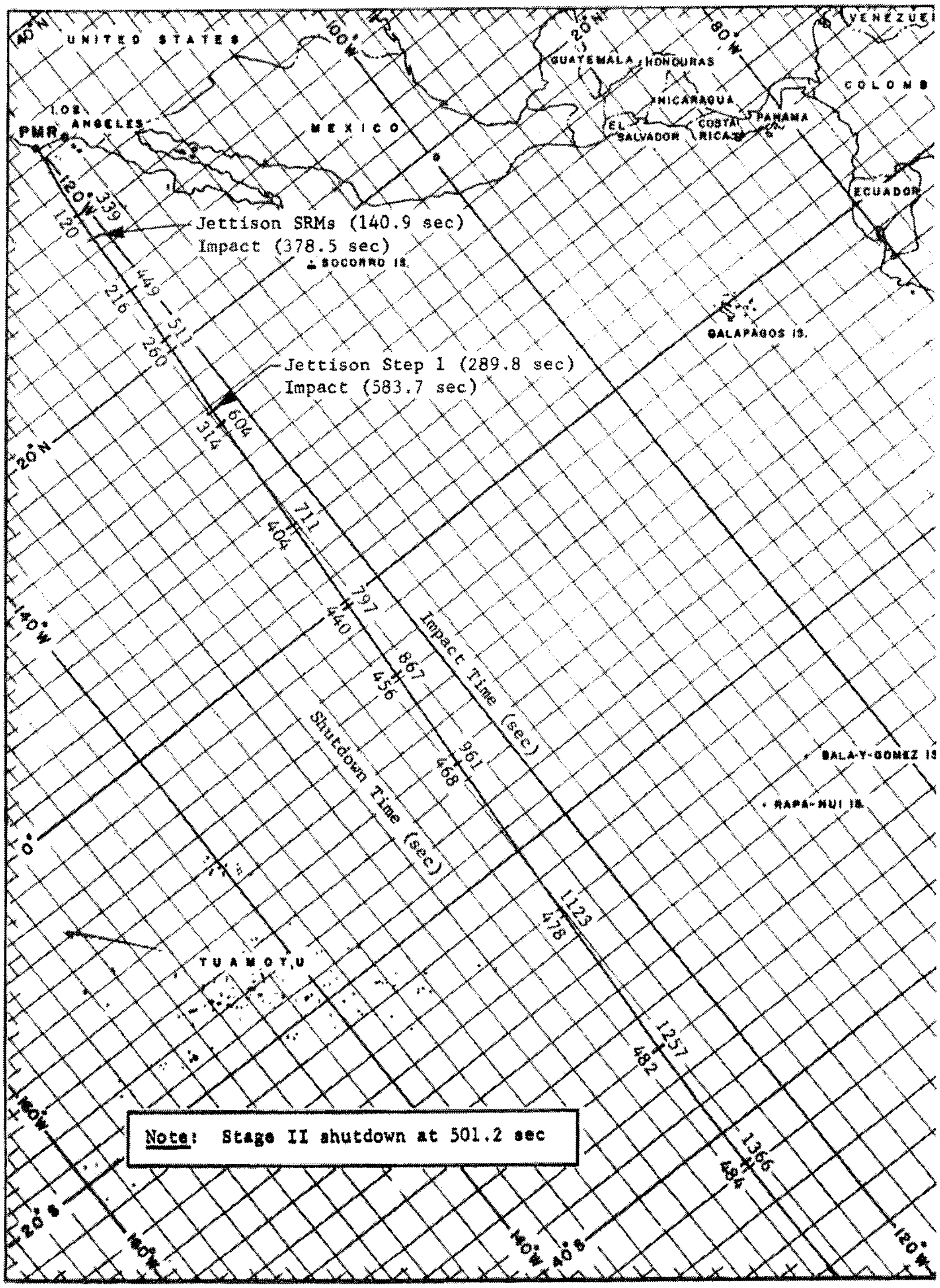

Fig. III-14 Instantaneous Impact Points, 2 CS-156 without Transtage

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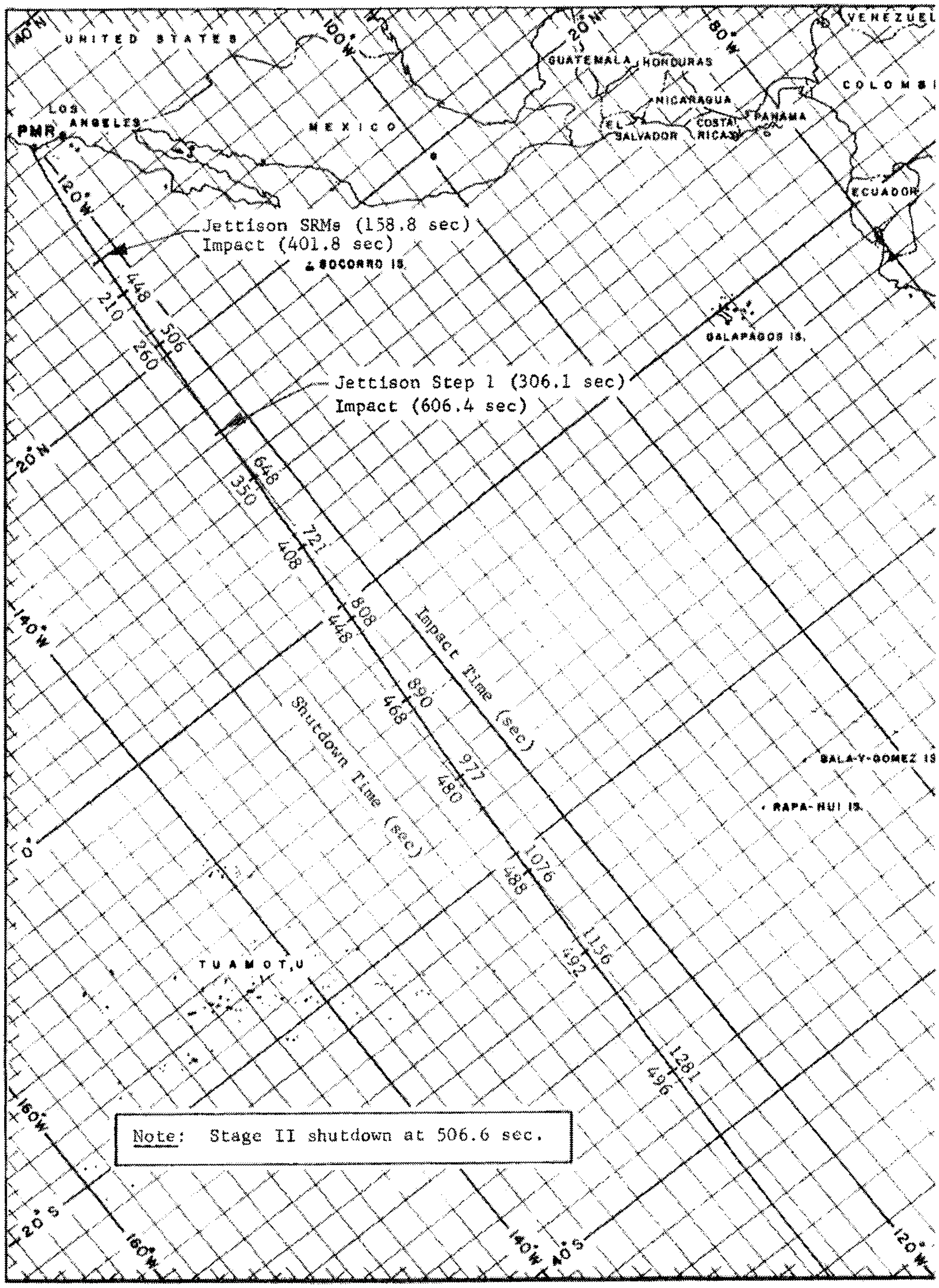

Fig. III-15 Instantaneous Impact Points, 3 CS-156 without Transtage

[redacted]

The above data reflect the performance of the various vehicles without regard to any reentry abort constraints that may be imposed. However, Fig. III-16 shows the reentry abort ceiling in coordinates of instantaneous apogee altitude and apogee velocity. The three trajectories shown are representative of the configurations (with transtage) considered. Except for a small region, they do not violate the limit. By trajectory shaping techniques, the flight path can be made to remain below the limit, and at the same time, better performance (payload in orbit) may be possible. The reentry abort ceiling, the temperature limit (Fig. III-17), and the $q\alpha\beta$ limit (Fig. III-18) were taken from the Aerospace interoffice correspondence 65-2130-ELL-156, dated 1 June 1965, "Payload Constraints for Titan IIIC, 7-Segment SRM Studies." Figures III-17 and III-18 show the limits and the values obtained from the three trajectories. The temperature limit is only slightly violated while the airloads indicator $q\alpha\beta$ contains a large margin.

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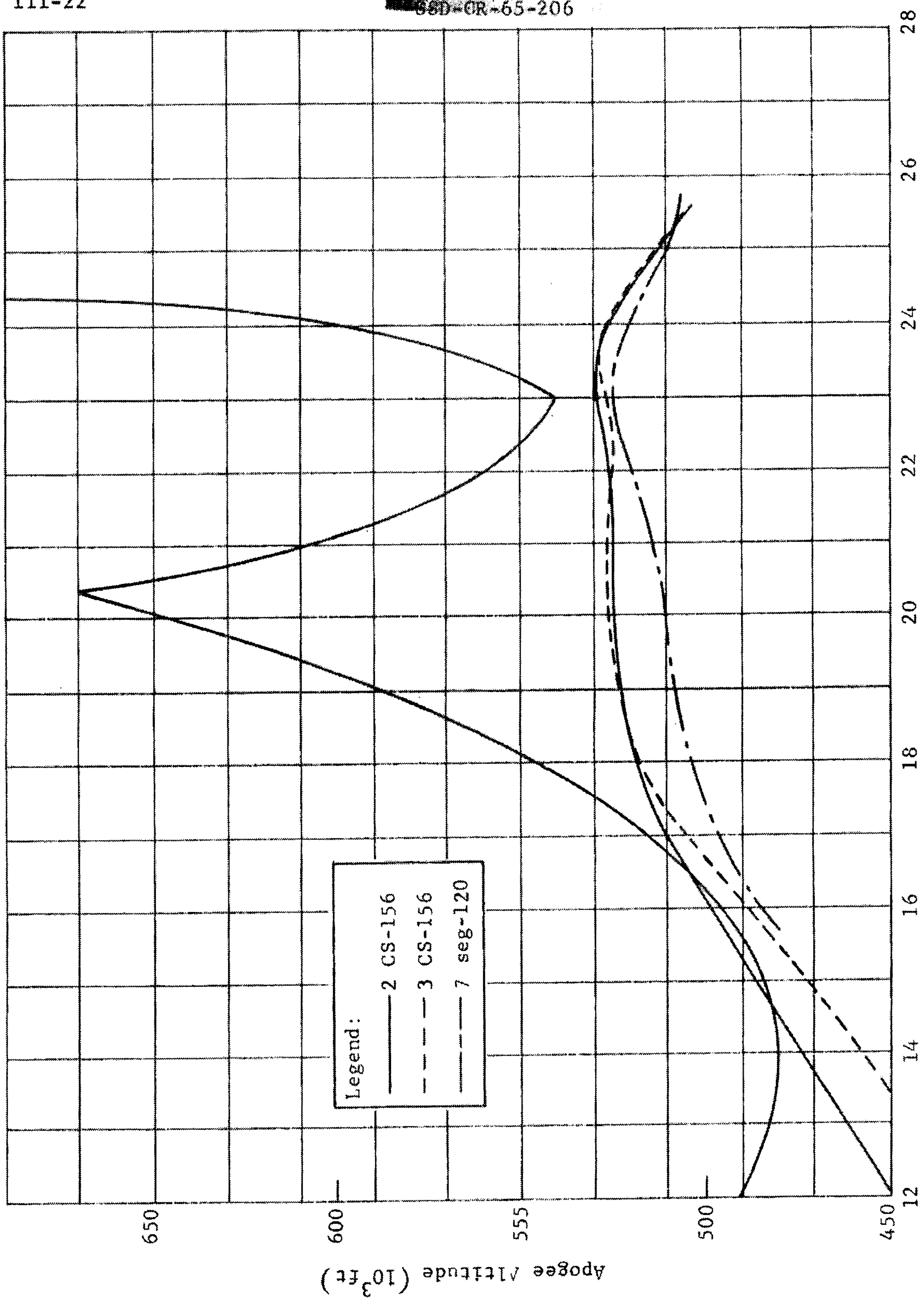

Fig. III-16 Reentry Abort Ceiling, South WTR Launch with Transtage

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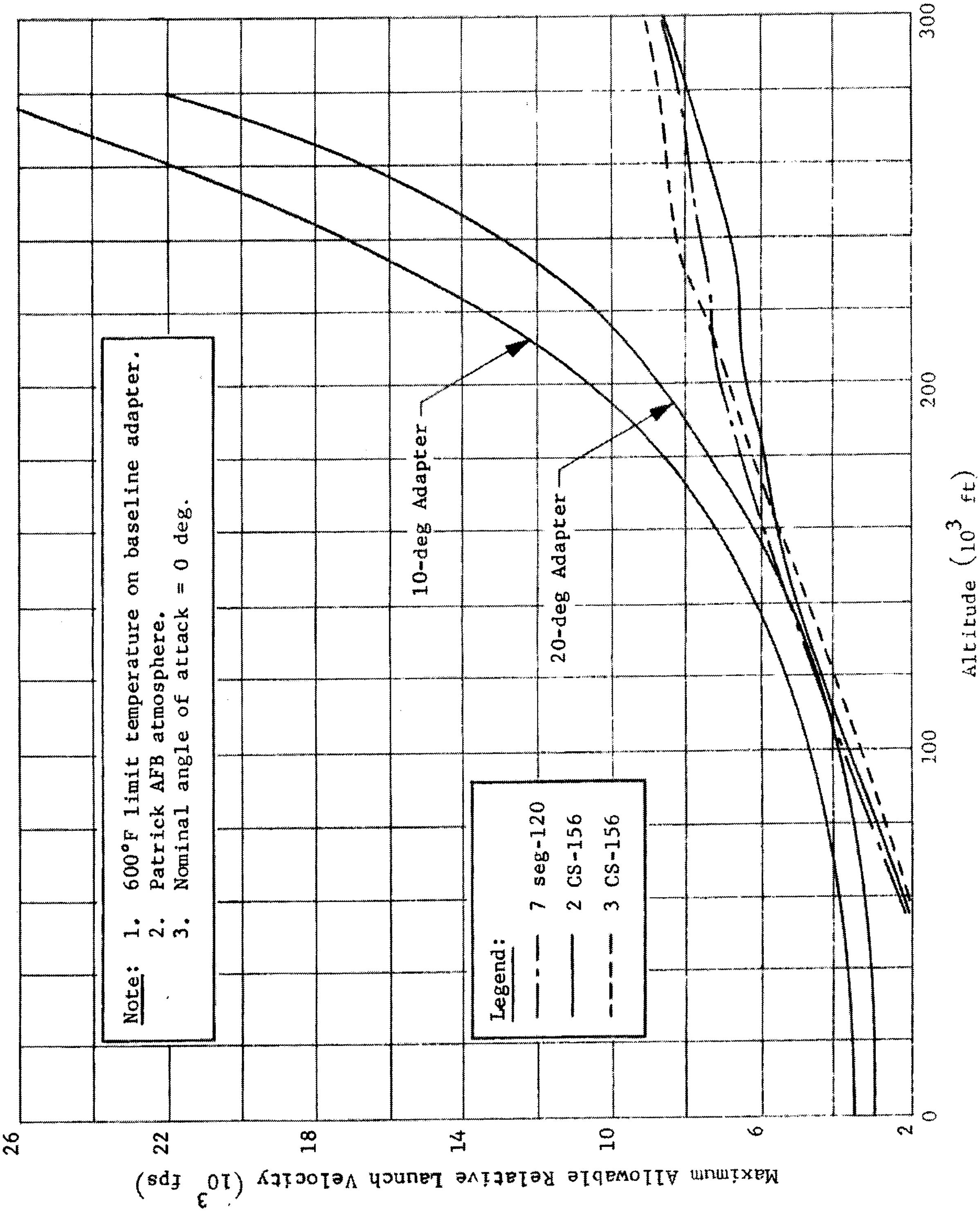

Fig. III-17 Launch Trajectory Temperature Constraints, South WTR Launch with Transtage

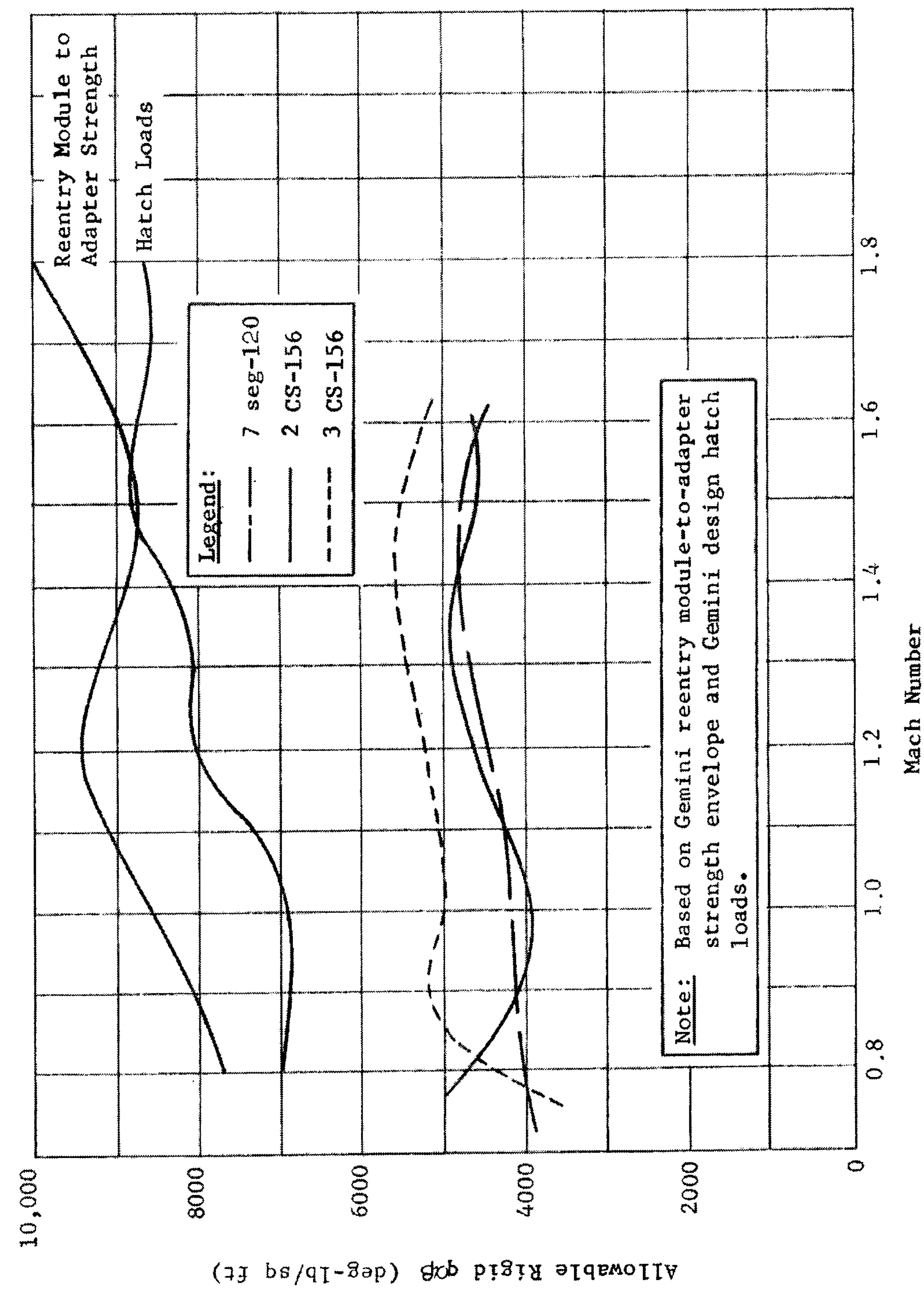

Fig. III-18 Allowable qαβ vs Mach, Ejection Seat Escape System, South WTR Launch with Transtage

2. ETR Performance

Circular and elliptical performance capability for due east launches was generated for the 7 seg-120, 2 CS-156, and 3 CS-156 vehicles with transtage. An optimum transtage propellant burn was used in all cases.

Two shaping techniques were used to develop the payload weight altitude charts: ascent using a park orbit, and ascent by direct injection into a transfer orbit.

a. Ascent Using a Park Orbit

For these cases, the vehicle was direct injected into a 100-n-mi park orbit. This was accomplished for low- and high-altitude payload capability.

Low-Altitude Payload Capability - For the low-altitude missions the maximum payload weight is obtained by burning a portion of the transtage propellant to achieve the 100-n-mi park orbit. The transtage is ignited a second time to achieve the transfer orbit perigee velocity. For the circular orbit capability, the transtage is reignited at apogee of the transfer ellipse to attain the required velocity for circularization.

High-Altitude Payload Capability - The high-altitude missions require the fully loaded transtage to be placed into the 100-n-mi park orbit if the maximum payload capability is to be obtained. This will require an early shutdown of Stage II. If Stage II was not shut down early a larger payload weight could be placed into the 100-n-mi park orbit, but the transtage would then not have sufficient energy to deliver this payload to the required higher orbit, i.e., the vehicle is transtage-limited instead of booster-limited.

b. Ascent by Direct Injection into Transfer Orbit

This technique differs from the ascent by park orbit technique in that perigee will occur at the 100-n-mi altitude, i.e., the transfer orbit is obtained by burning out at an altitude of 100 n mi with a zero flight path angle and a sufficient overspeed velocity to obtain the required apogee altitudes. At apogee of the transfer ellipse (for the circular orbit cases) the transtage is again reignited to obtain the required circular velocity.

If the mission does not require a park orbit this shaping technique should be used since the entire Stage II propellant load can be used in all cases, resulting in a larger payload capability for the high-altitude missions. A park orbit has the advantage of allowing the final inject point, or transfer maneuver, to be located at a particular location relative to the earth.

For missions requiring the use of a parking orbit, the full capability of the lower stages (0, I, II) can be used by use of higher altitude parking orbits. This will improve the payload capability above the altitude where Stage II is shutdown early. This payload gain, however, will not be competitive with that shown for the no-park orbit cases.

The performance capability from ETR is presented in Fig. III-19 thru III-39.

Figures III-19 thru III-25, are for the 7 seg-120, Fig. III-26 thru III-32 are for the 2 CS-156, and Fig. III-33 thru III-39 are for the 3 CS-156 configurations. Each set of figures consists of the following:

1) Payload weight to 100-n-mi park orbit as a function of the transtage propellant consumed.

2) Payload weight-altitude chart for the park orbit cases. The payload weight-altitude charts show both the nominal performance and the minimum performance. Nominal performance assumes the vehicle performs precisely as predicted.

3) Transtage propellant usage for the park orbit cases. These figures functionally show how the transtage propellant is used to attain first the park orbit, the transfer ellipse, and the final circular orbit. Of particular note is the vertical dashed line representing the point where the transtage is no longer used to attain the park orbit, and Stage II must then be shut down early for higher altitudes.

4) Transtage propellant usage for elliptical final orbits using a park orbit. Again the dashed line shows the altitude at which the transtage is no longer required for the park orbit and Stage II must be shut down early.

5) Payload weight-altitude chart for the no-park orbit cases.

6) Transtage propellant usage for the no-park orbit cases and final circular orbits.

7) Transtage propellant usage for the no-park orbit cases and final elliptical orbits.

3. Special Studies

The previous ascent trajectories were shaped using an initial pitch rate from 20 to 30 sec to drive the angle of attack to zero, a zero-lift pitch control during Stage 0 operation, followed by a constant pitch rate from 150 sec to transtage burnout.

By deviating from the zero-lift portion of flight (after 90 sec of SRM operations) and using multiple upper stage pitch rates, a considerable payload increase can be obtained at the expense of other parameters. Figure III-40 shows this payload increase for a matrix of two upper stage pitch rates $(\dot{x}_2$ and $\dot{x}_3)$. This technique was not used for the MOL study becaue the aerodynamic heating indicator and maximum dynamic pressure constraint of 95×10^6 ft-lb/ft^2 and 900 lb/ft^2 respectively would be violated. The aerodynamic heating values and maximum dynamic pressure values are presented in Fig. III-41 and III-42 for this shaping technique.

This short study was intended as an example to show how the various trajectory parameters are varied by trajectory shaping Similarly, a more complete shaping study could be performed to reduce the rigid body loads.

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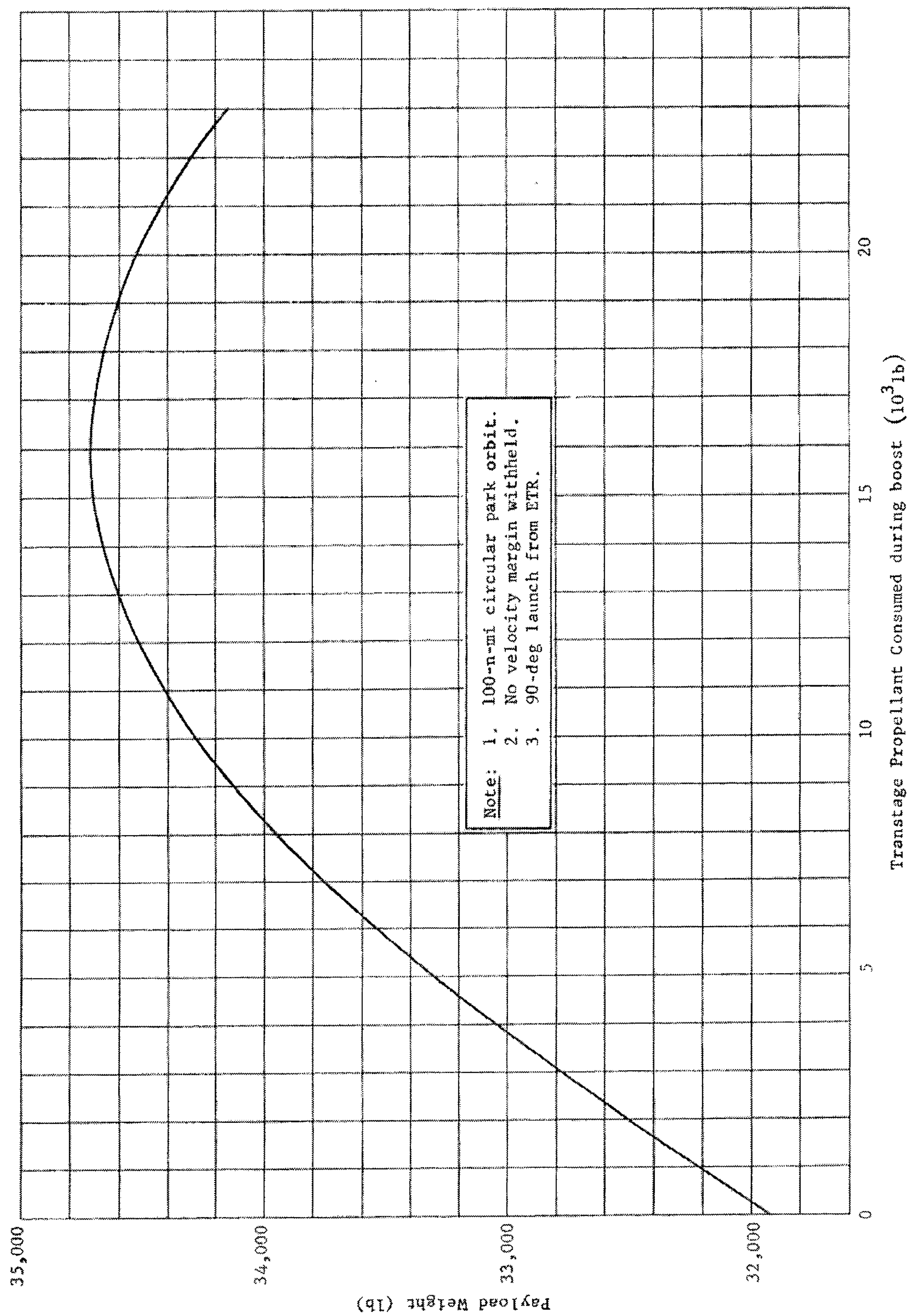

Fig. III-19 Park Orbit Payload Capability, 7 seg-120

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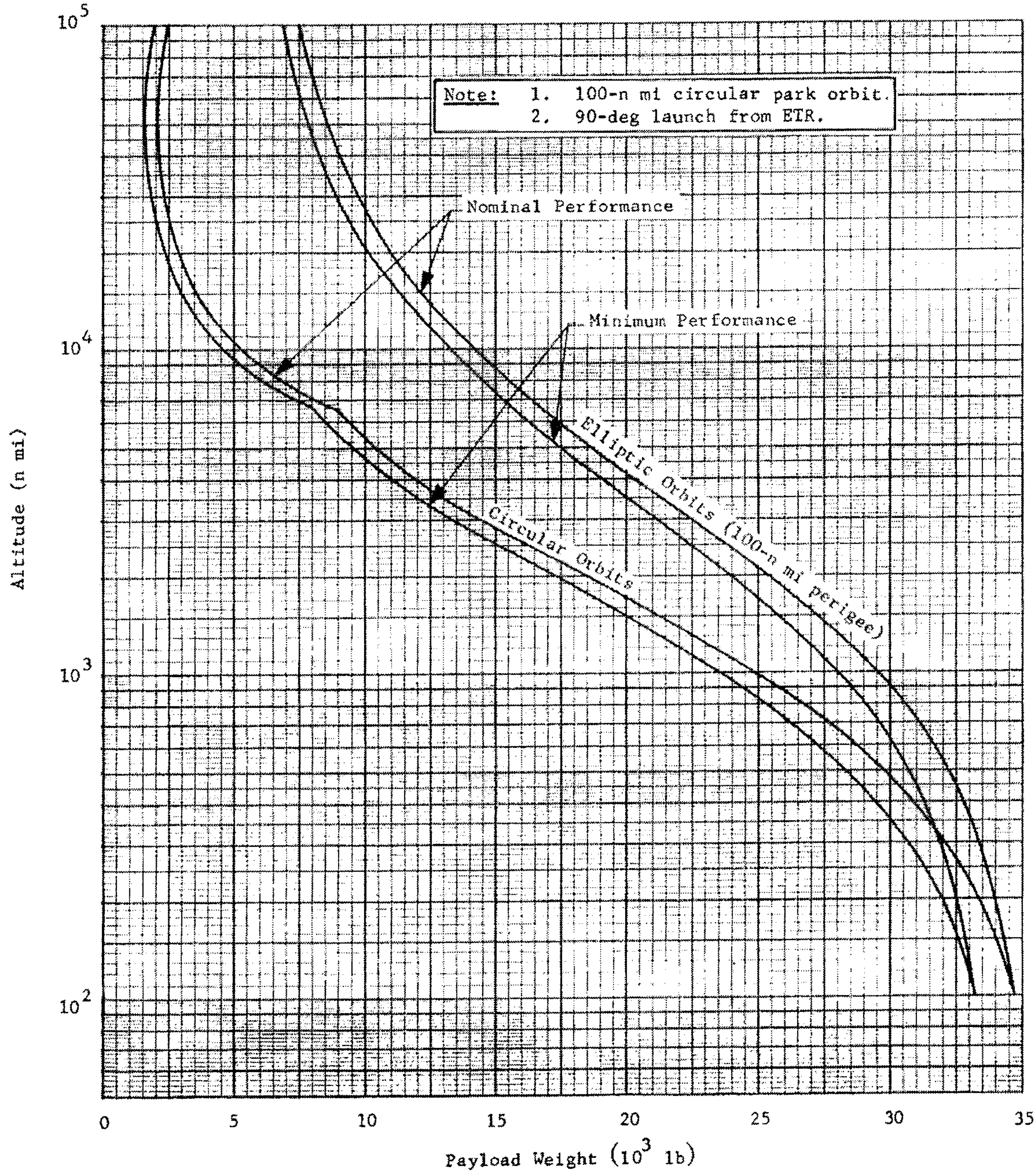

Fig. III-20 Payload Altitude Profile, 7 seg-120, 100-n-mi Park Orbit

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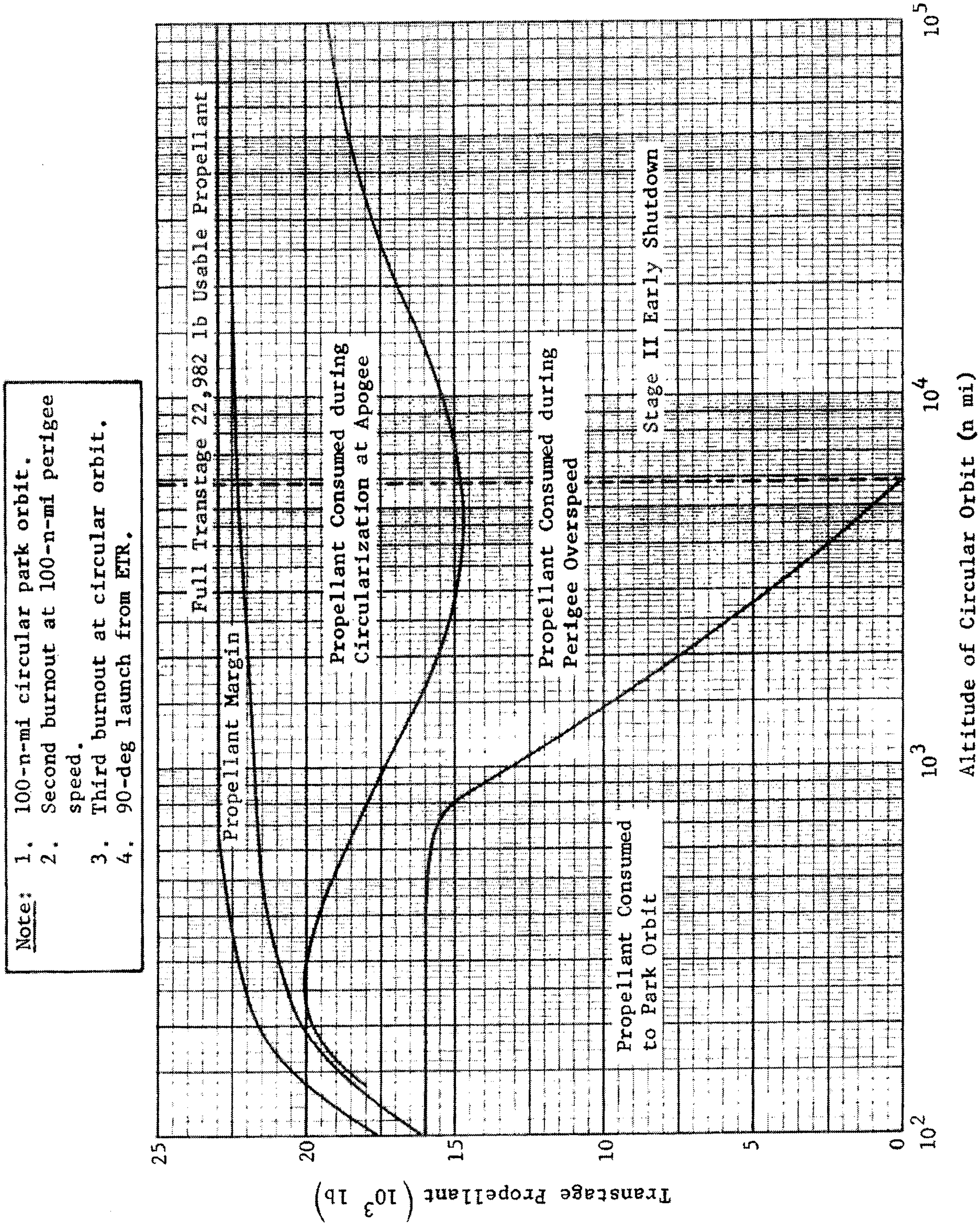

Fig. III-21 Transtage Propellant Usage, 7 seg-120, 100-n-mi Park to Circular Orbits

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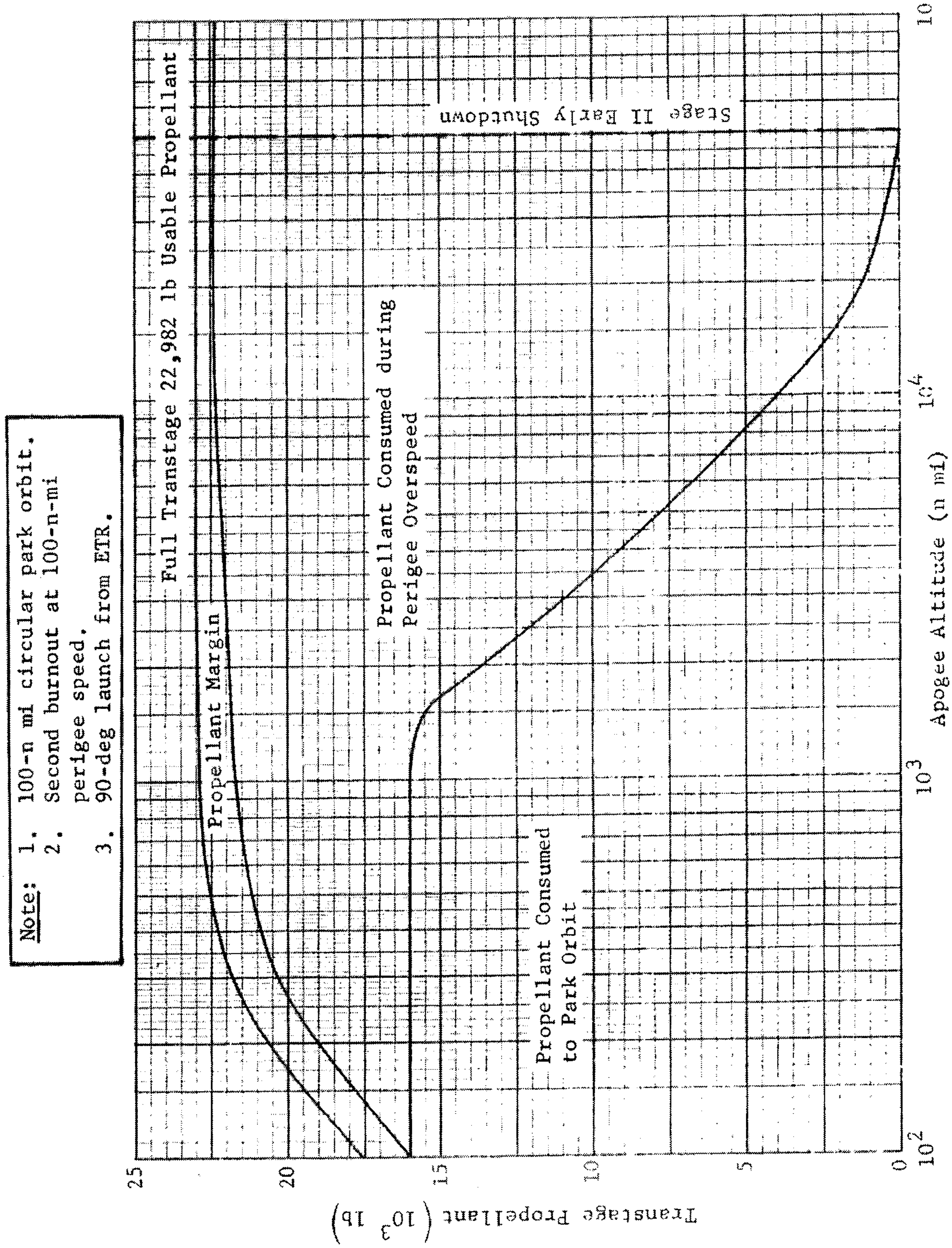

Fig. III-22 Transtage Propellant Usage, 7 seg-120, 100-n-mi Park to Elliptical Orbits

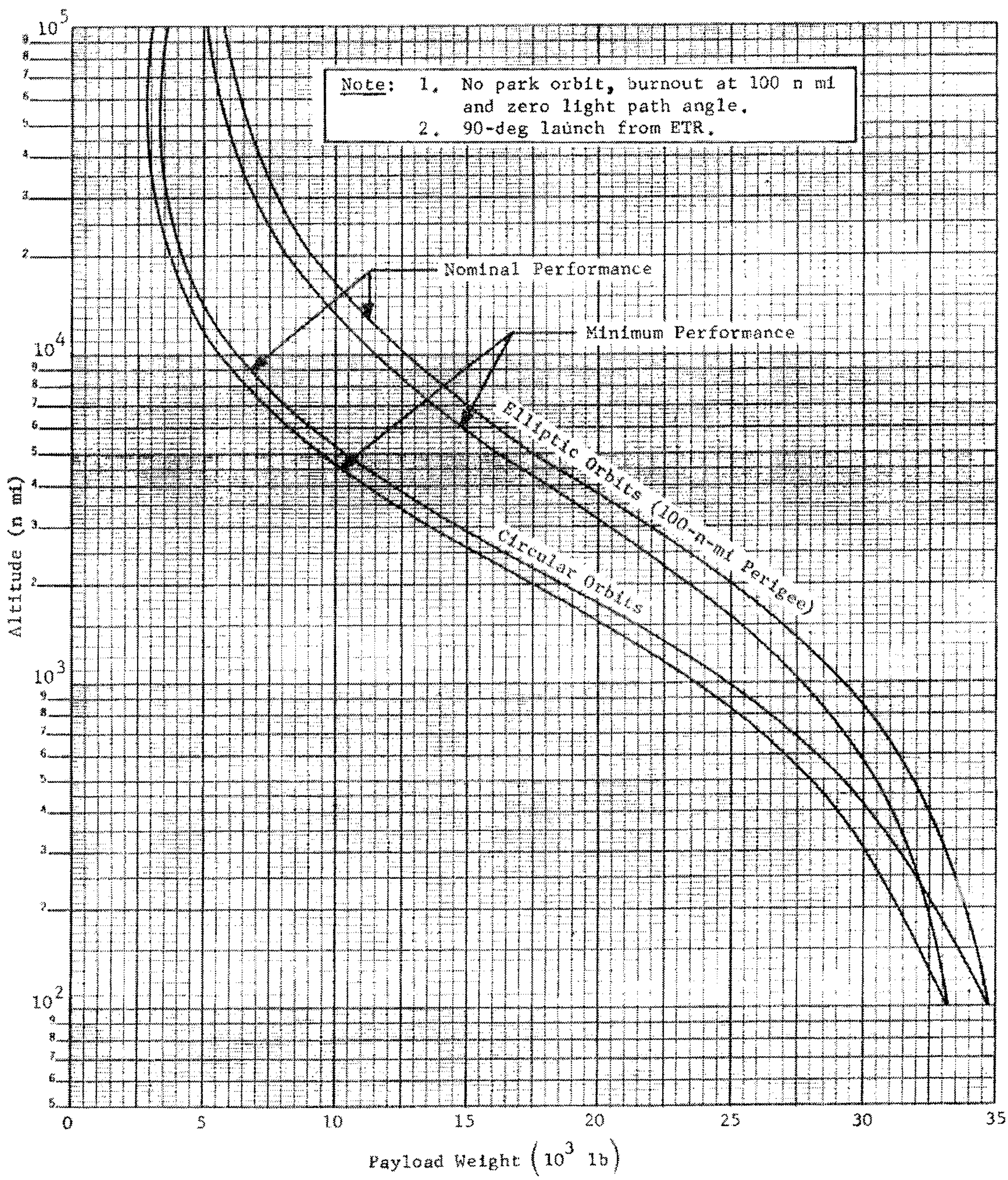

Fig. III-23 Payload Altitude Profile, 7 seg-120, 100-n-mi Perigee Overspeed

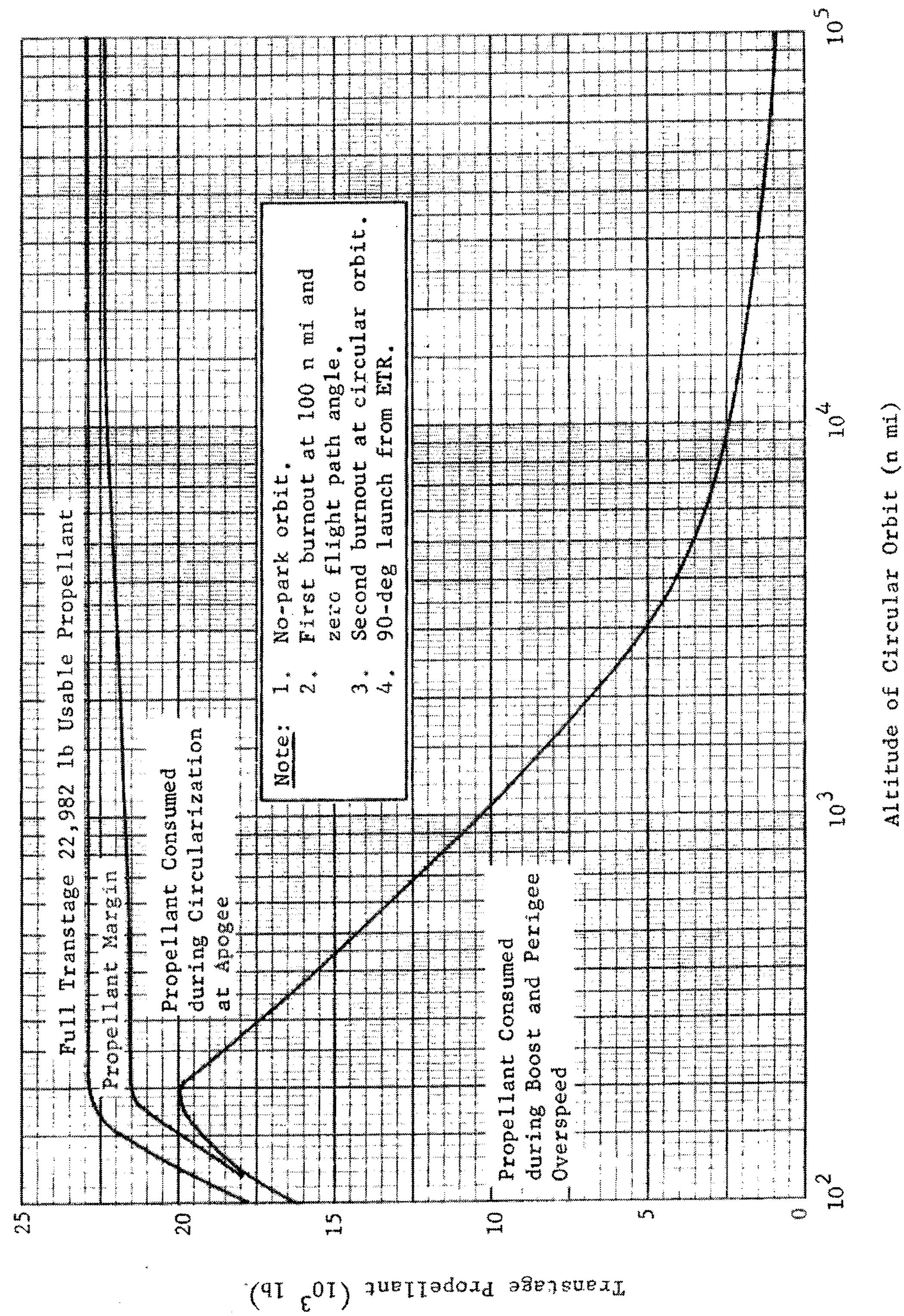

Fig. III-24 Transtage Propellant Usage, 7 seg-120, 100-n-mi Perigee to Circular Orbits

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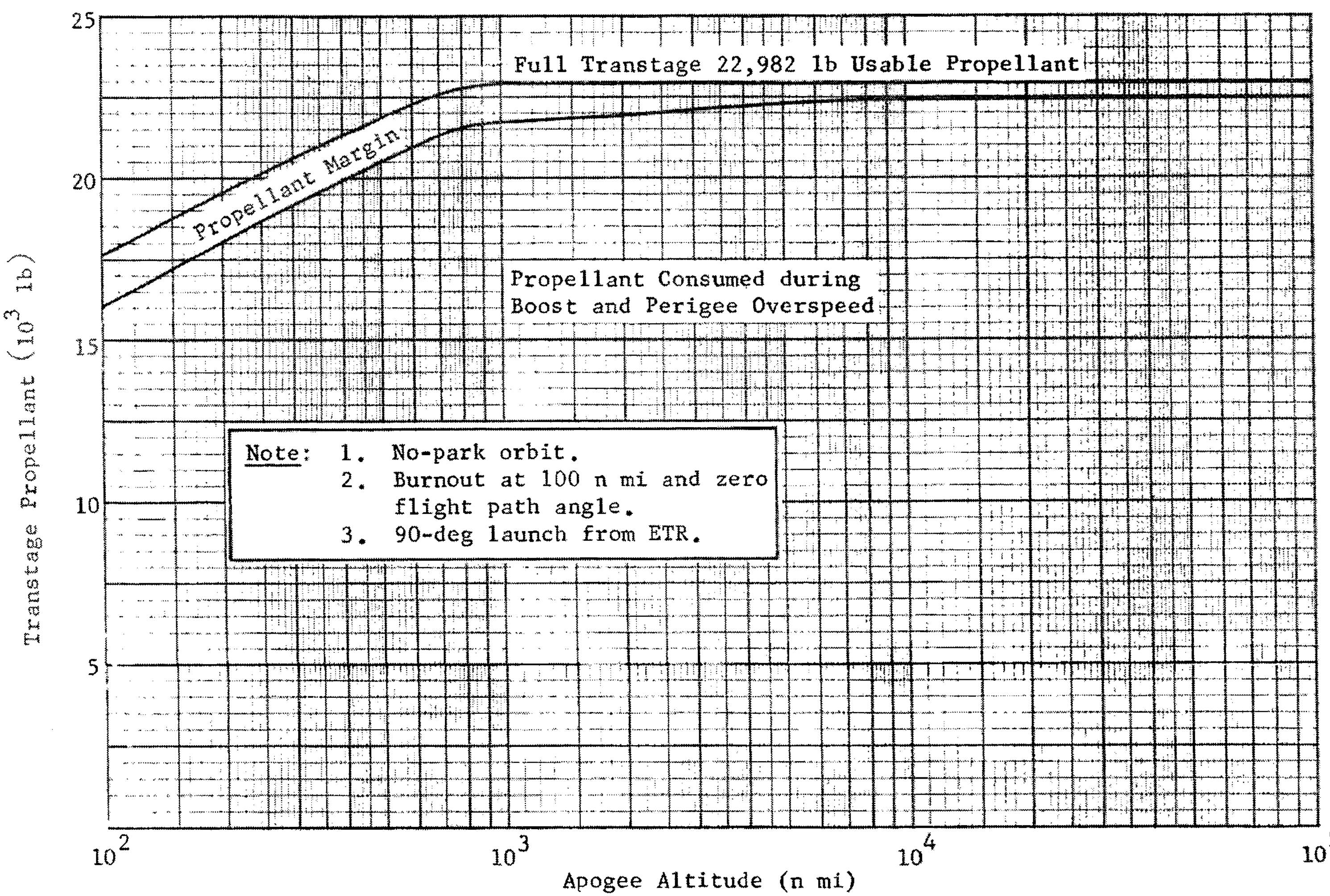

Fig. III-25 Transtage Propellant Usage, 7 seg-120, 100-n-mi Perigee to Elliptical Orbits

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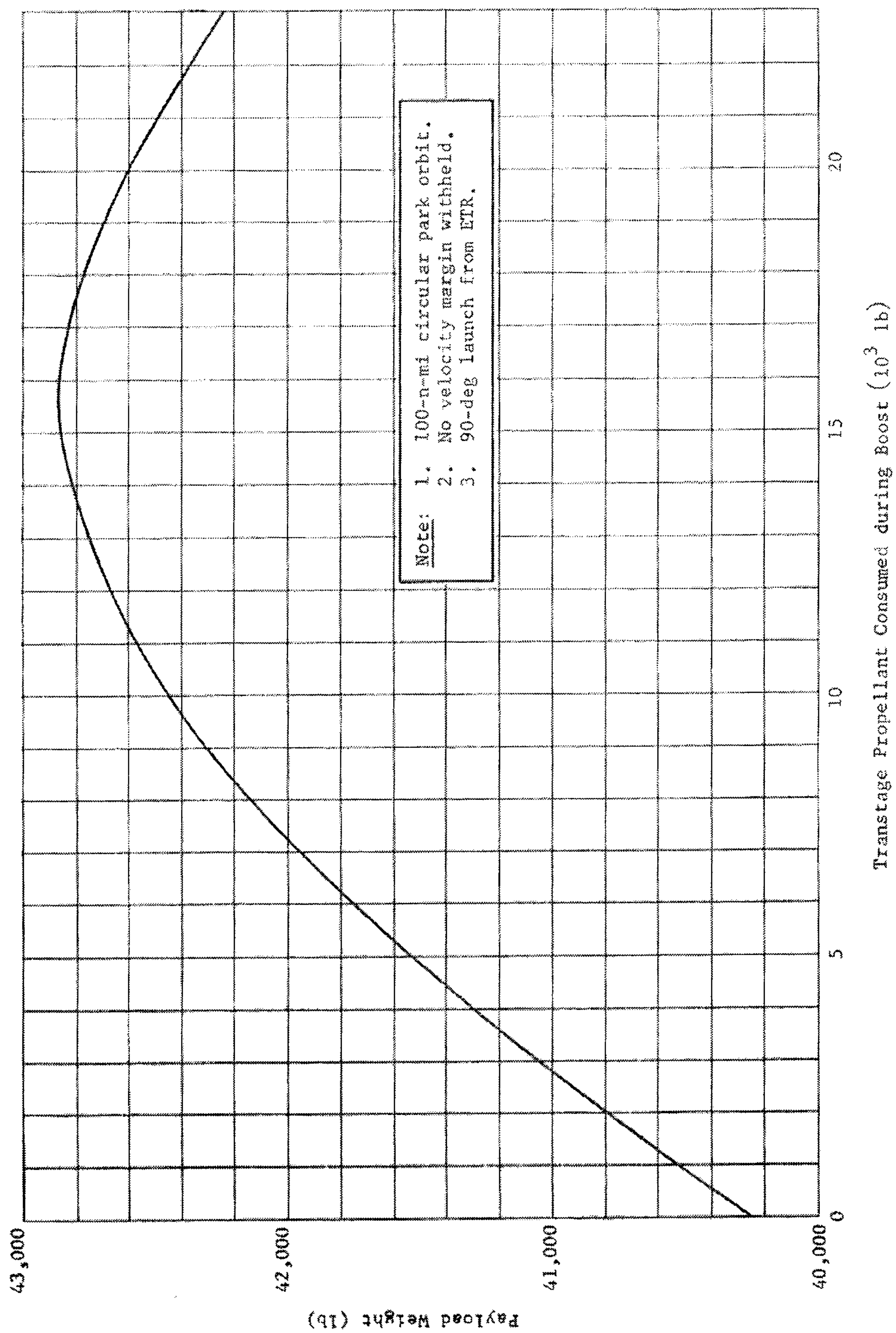

Fig. III-26 Park Orbit Payload Capability, 2 CS-156

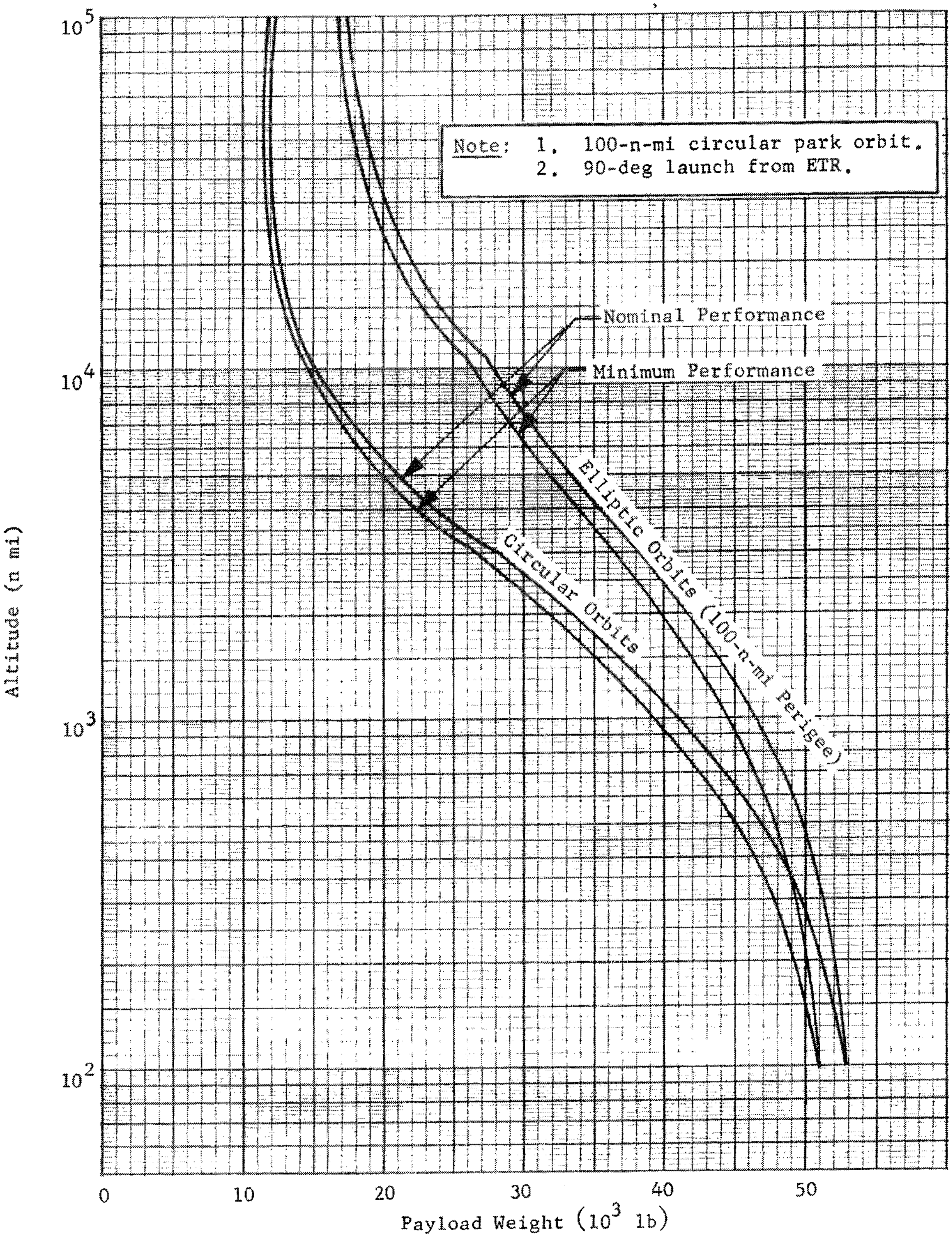

Fig. III-27 Payload Altitude Profile, 2 CS-156, 100-n-mi Park Orbit

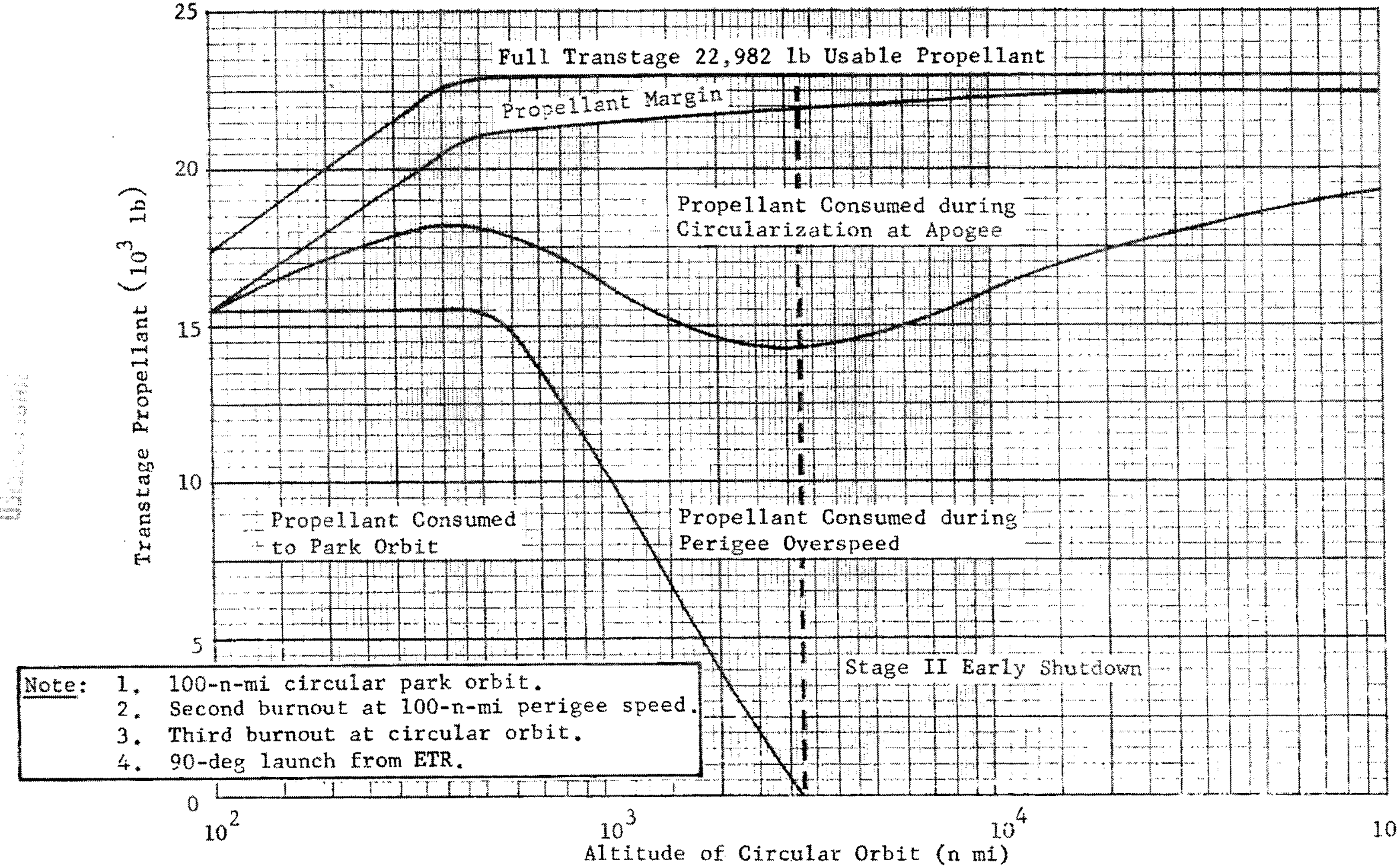

Fig. III-28 Transtage Propellant Usage, 2 CS-156, 100-n-mi Park to Circular Orbits

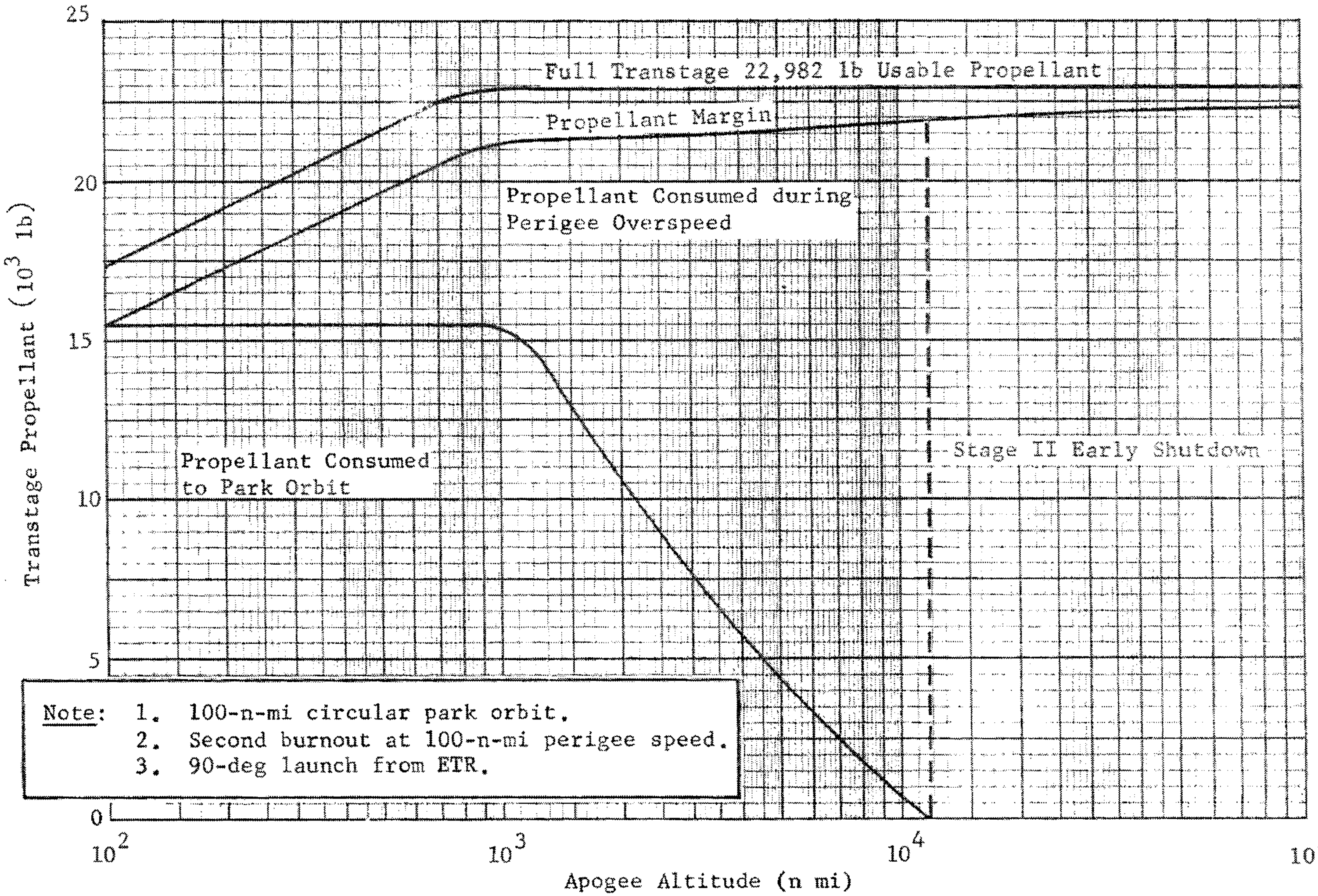

Fig. III-29 Transtage Propellant Usage, 2 CS-156, 100-n-mi Park to Elliptical Orbits

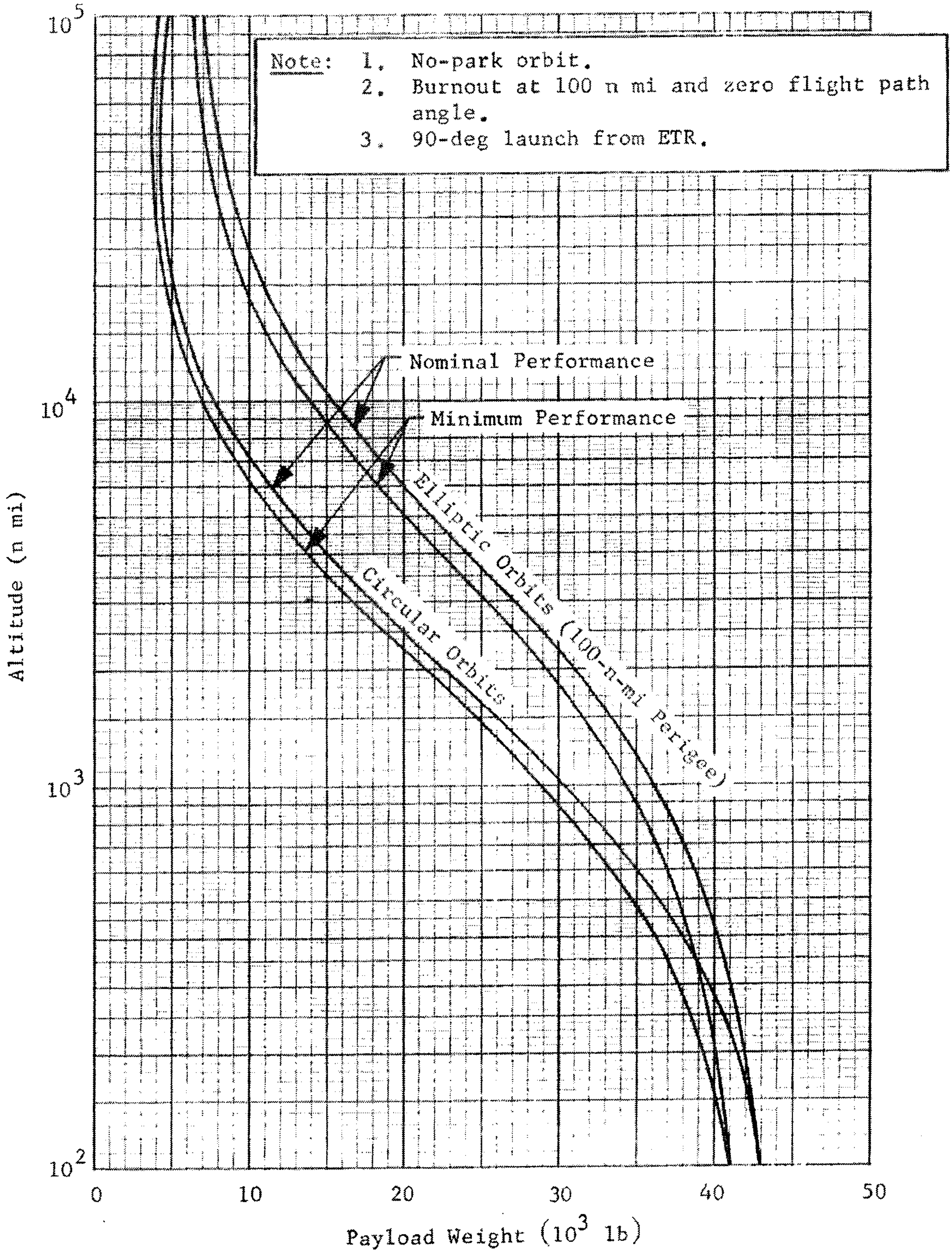

Fig. III-30 Payload Altitude Profile, 2 CS-156, 100-n-mi Perigee Overspeed

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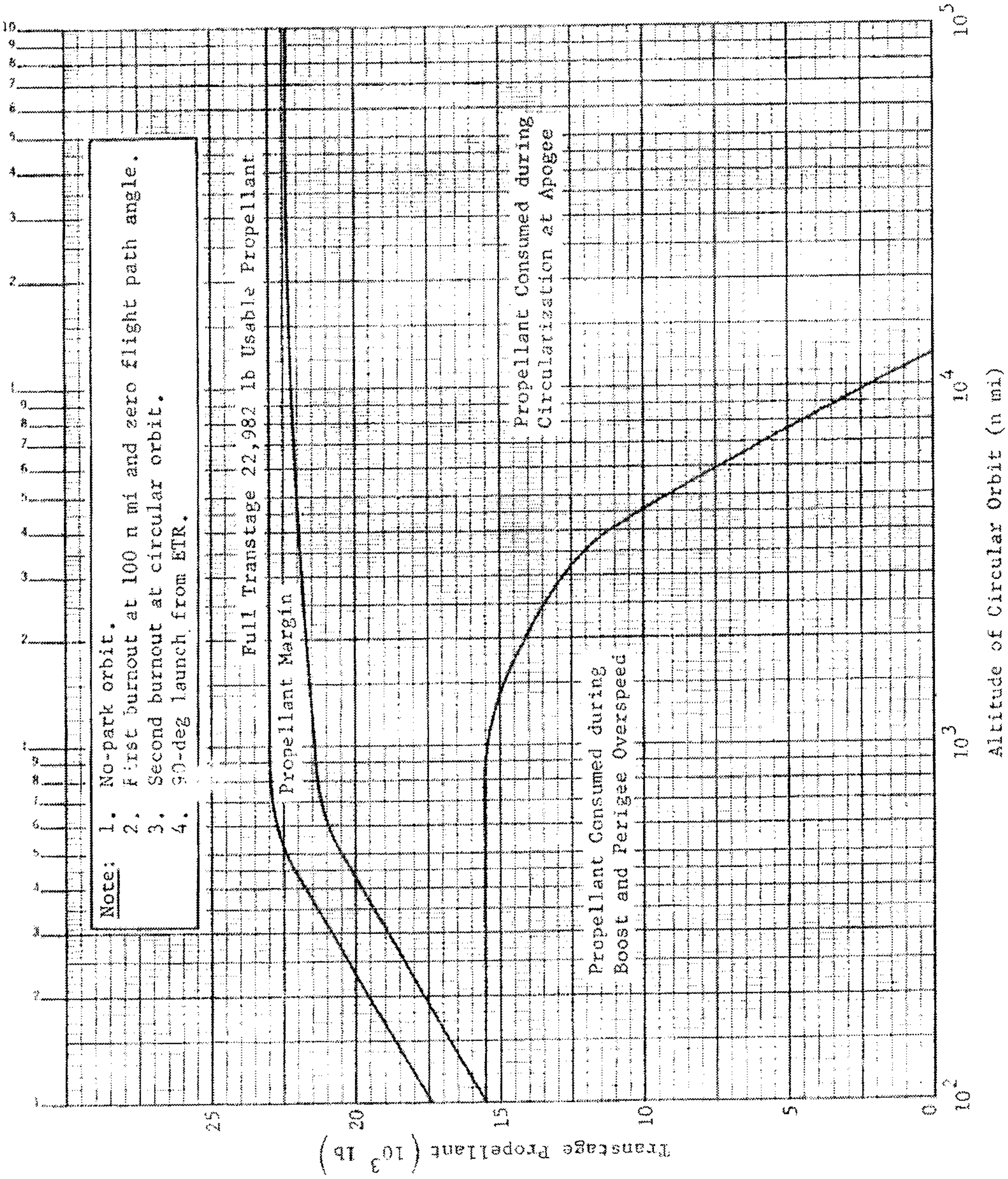

Fig. III-31 Transtage Propellant Usage, 2 CS-156, 100-n-mi Perigee Overspeed to Circular Orbits

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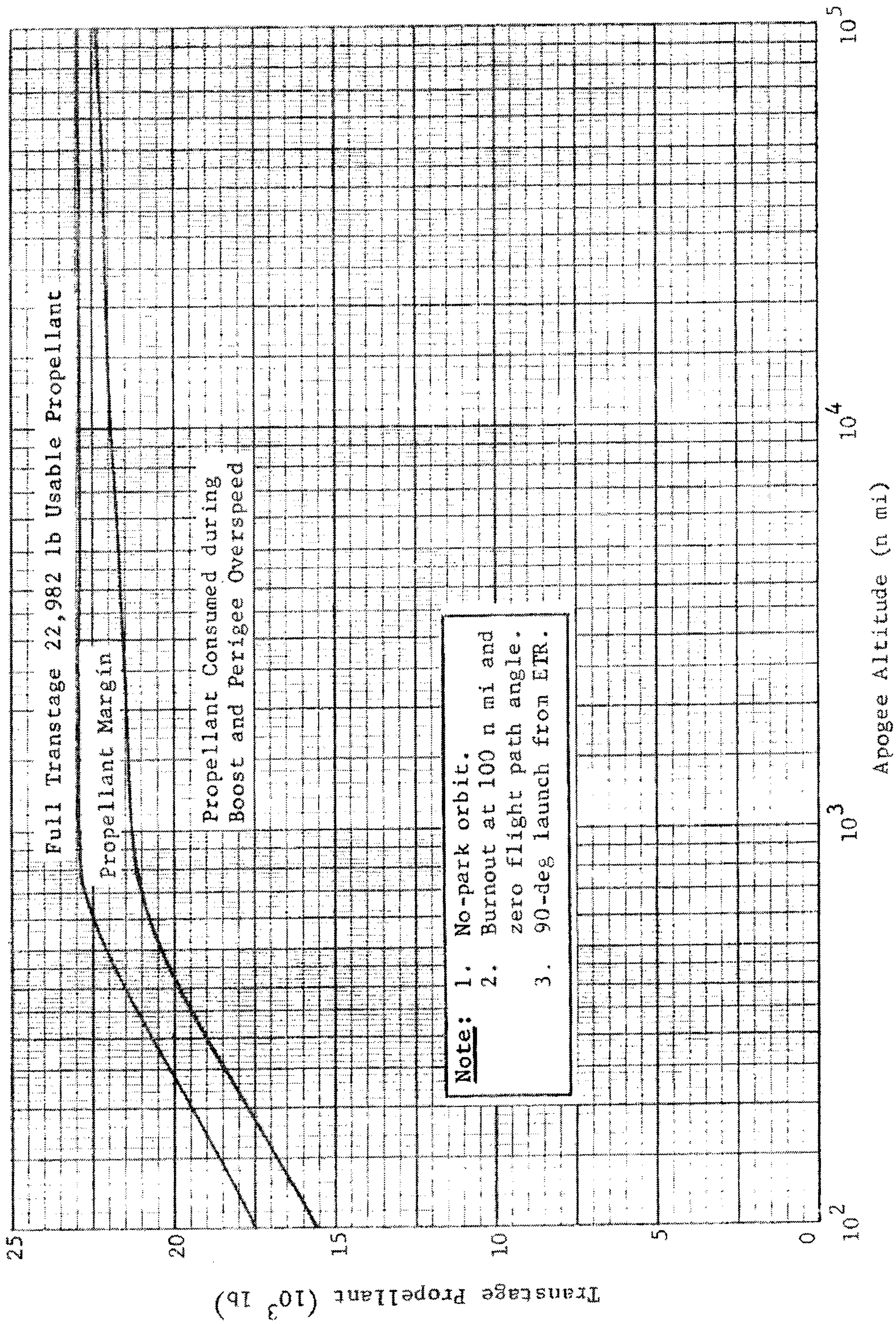

Fig. III-32 Transtage Propellant Usage, 2 CS-156, 100-n-mi Perigee Overspeed to Elliptical Orbits

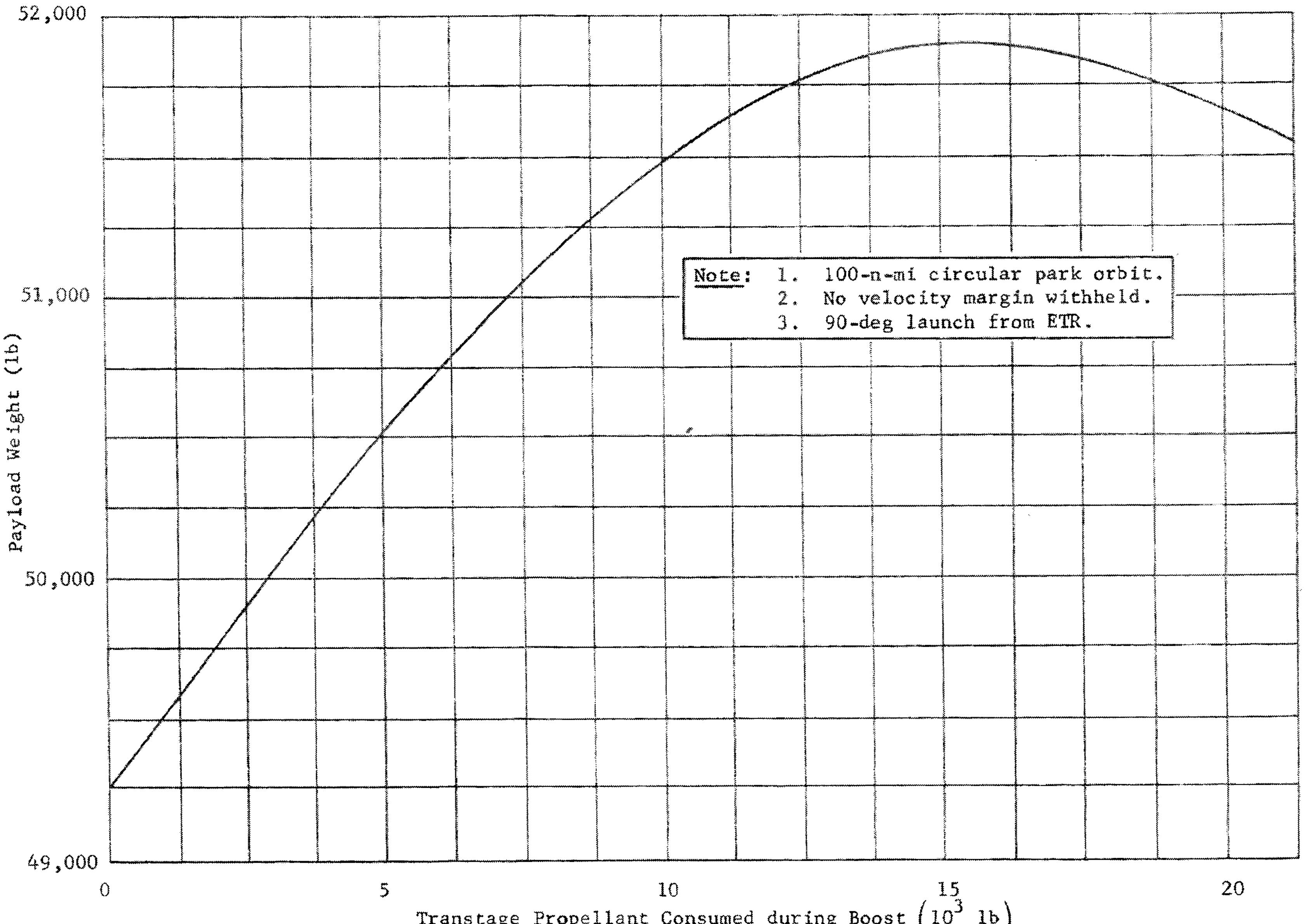

Fig. III-33 Park Orbit Payload Capability, 3 CS-156

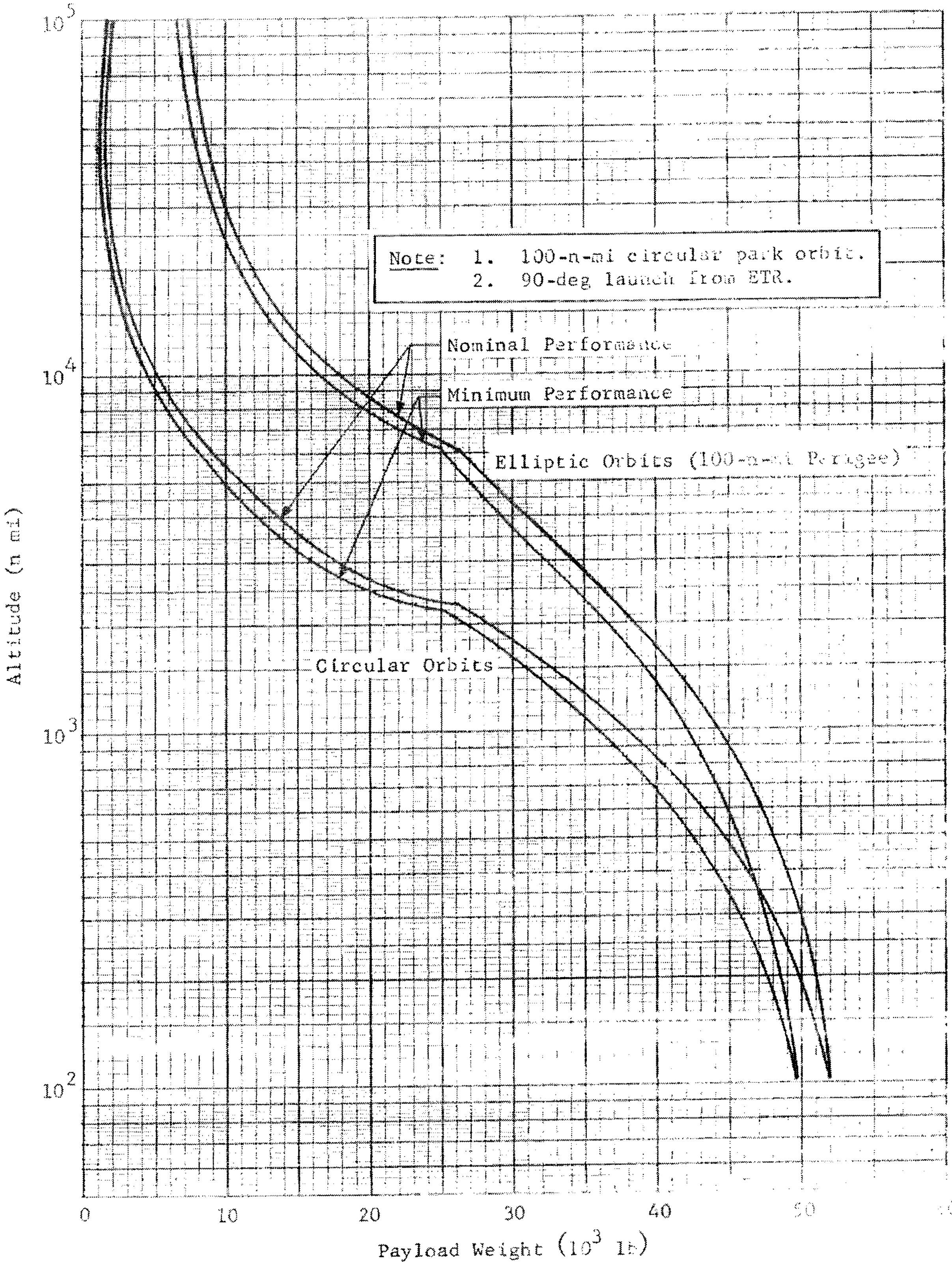

Fig. III-34 Payload Altitude Profile, 3 CS-156, 100-n-mi Park Orbit

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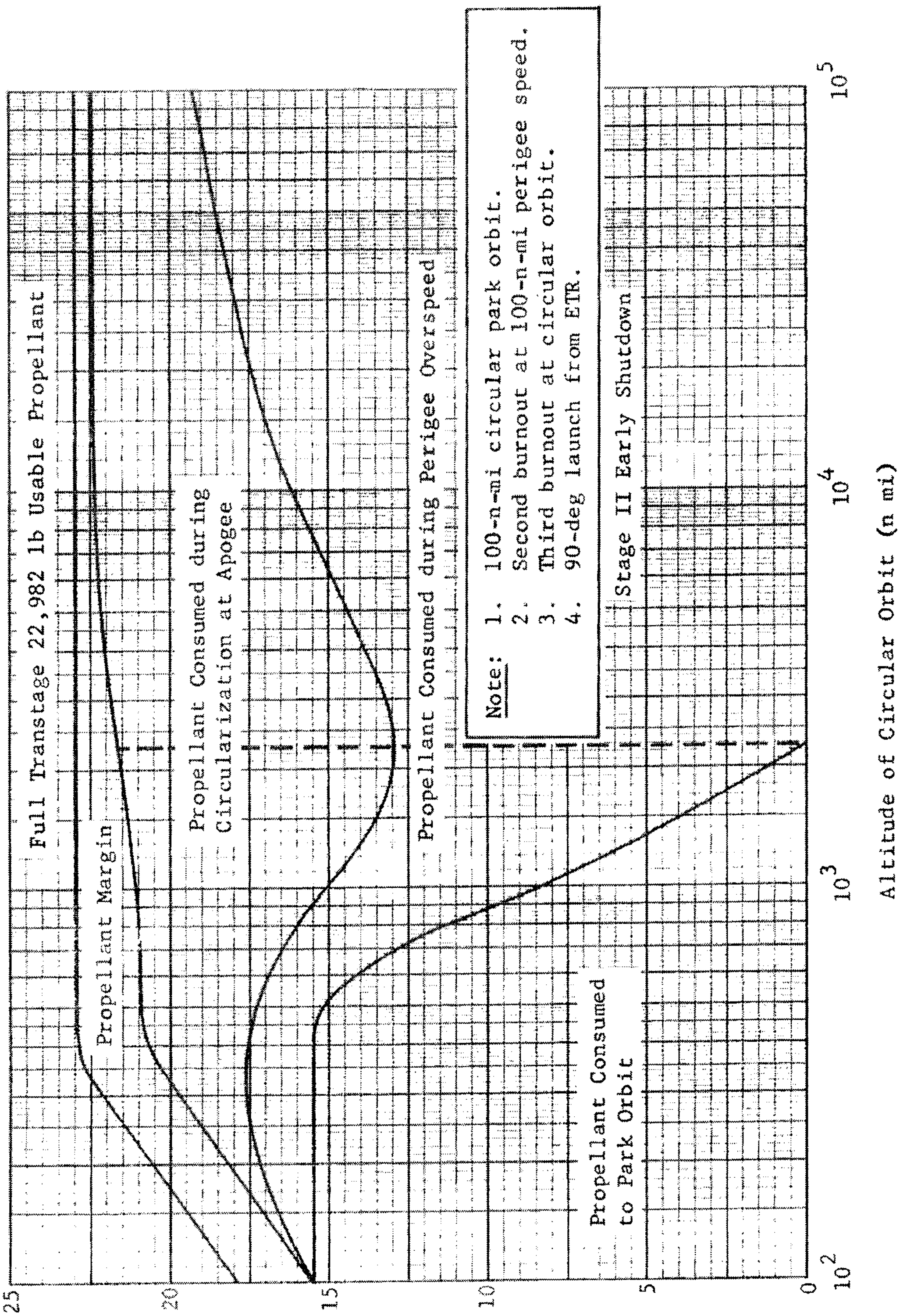

Fig. III-35 Transtage Propellant Usage, 3 CS-156, 100-n-mi Park to Circular Orbits

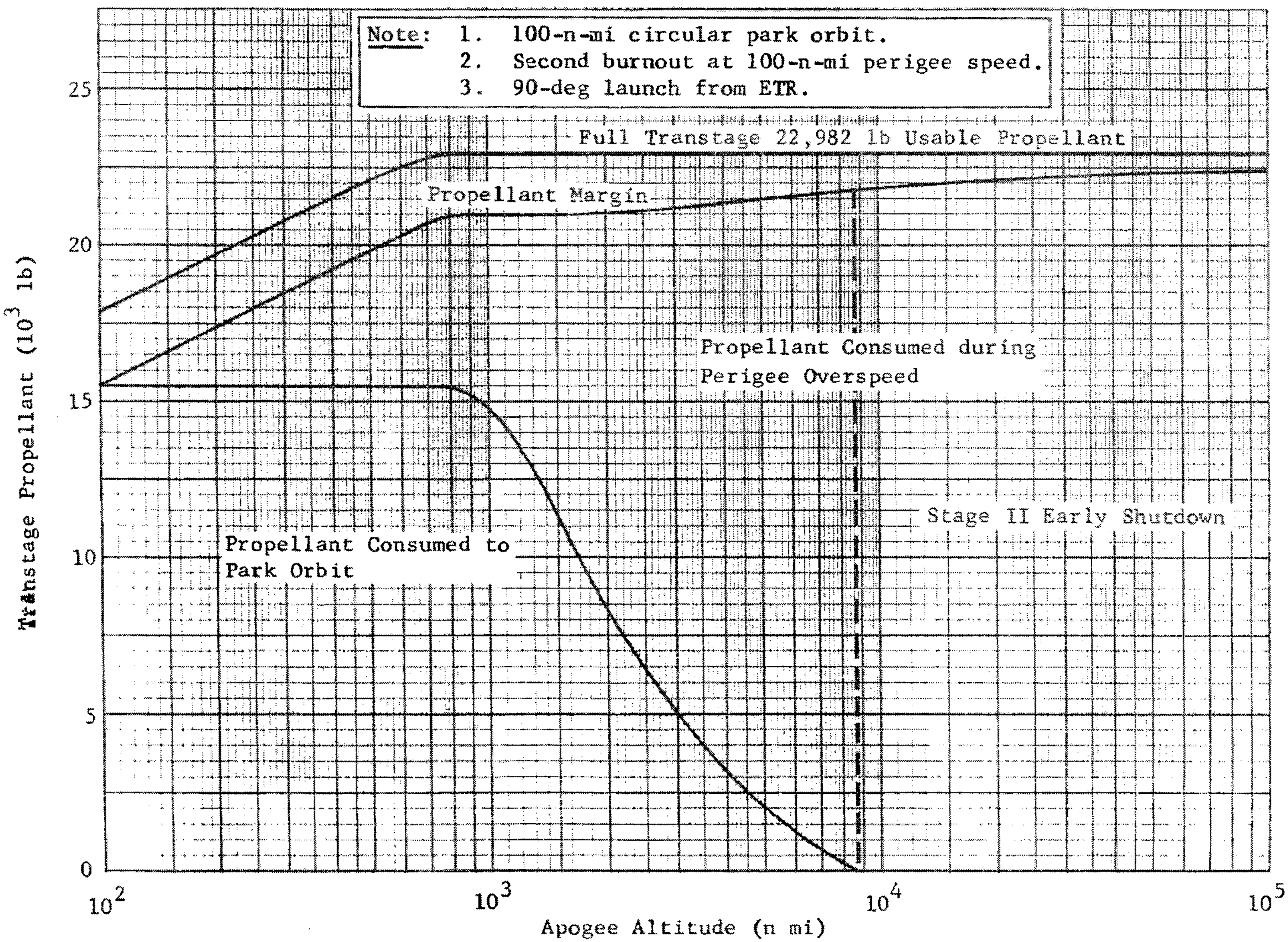

Fig. III-36 Transtage Propellant Usage, 3 CS-156, 100-n-mi Park to Elliptical Orbits

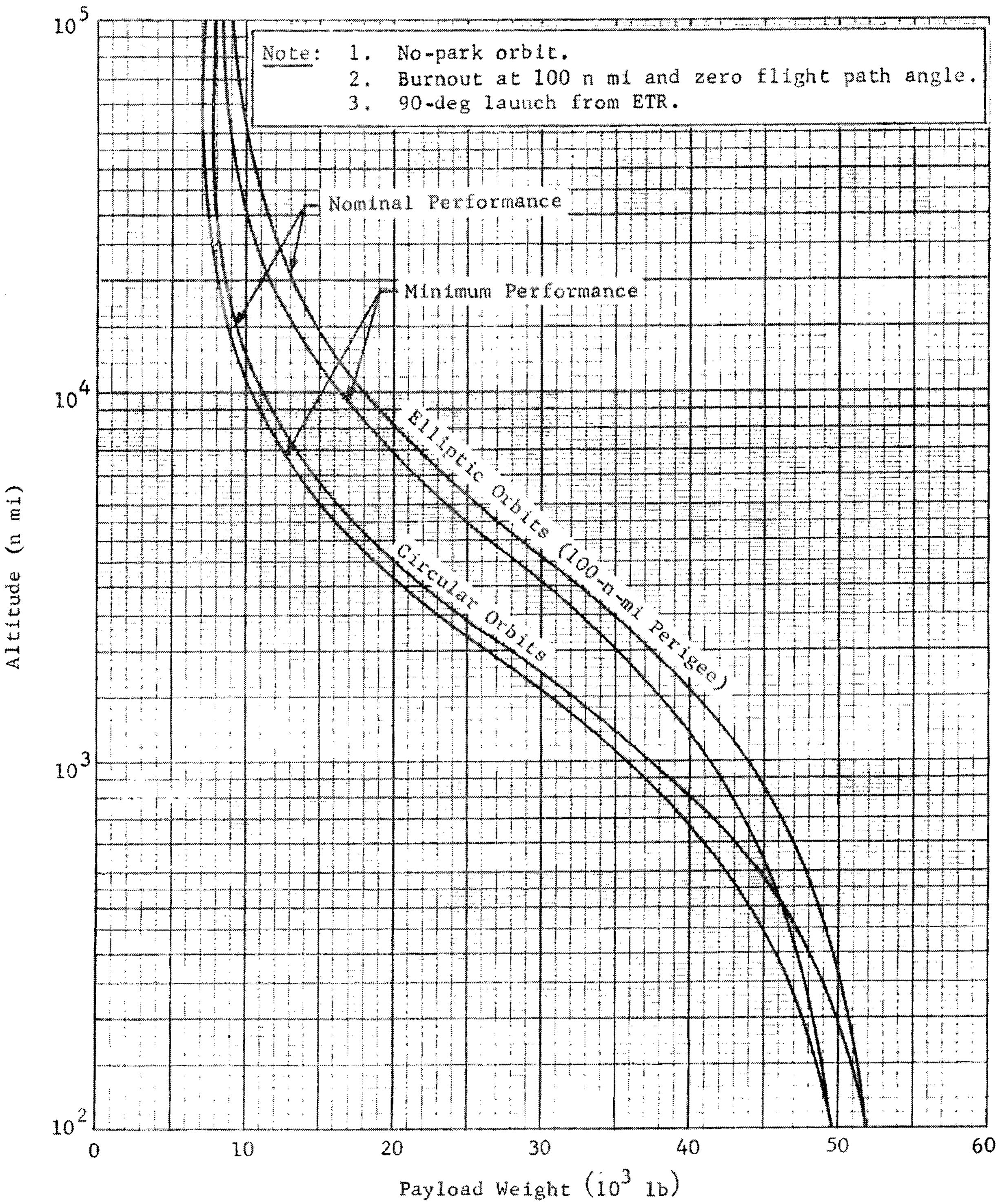

Fig. III-37 Payload Altitude Profile, 3 CS-156, 100-n-mi Perigee Overspeed

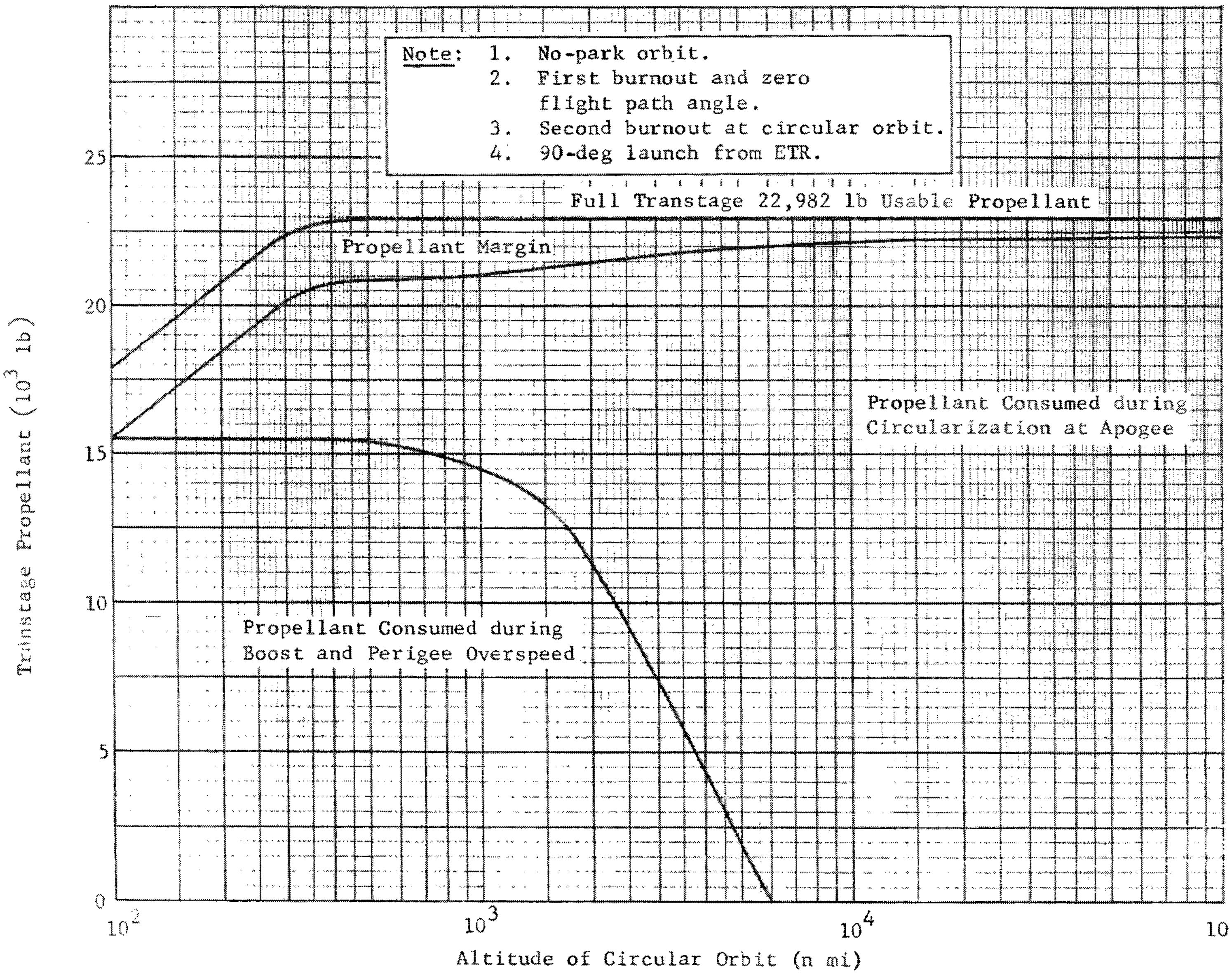

Fig. III-38 Transtage Propellant Usage, 3 CS-156, 100-n-mi Perigee Overspeed to Circular Orbits

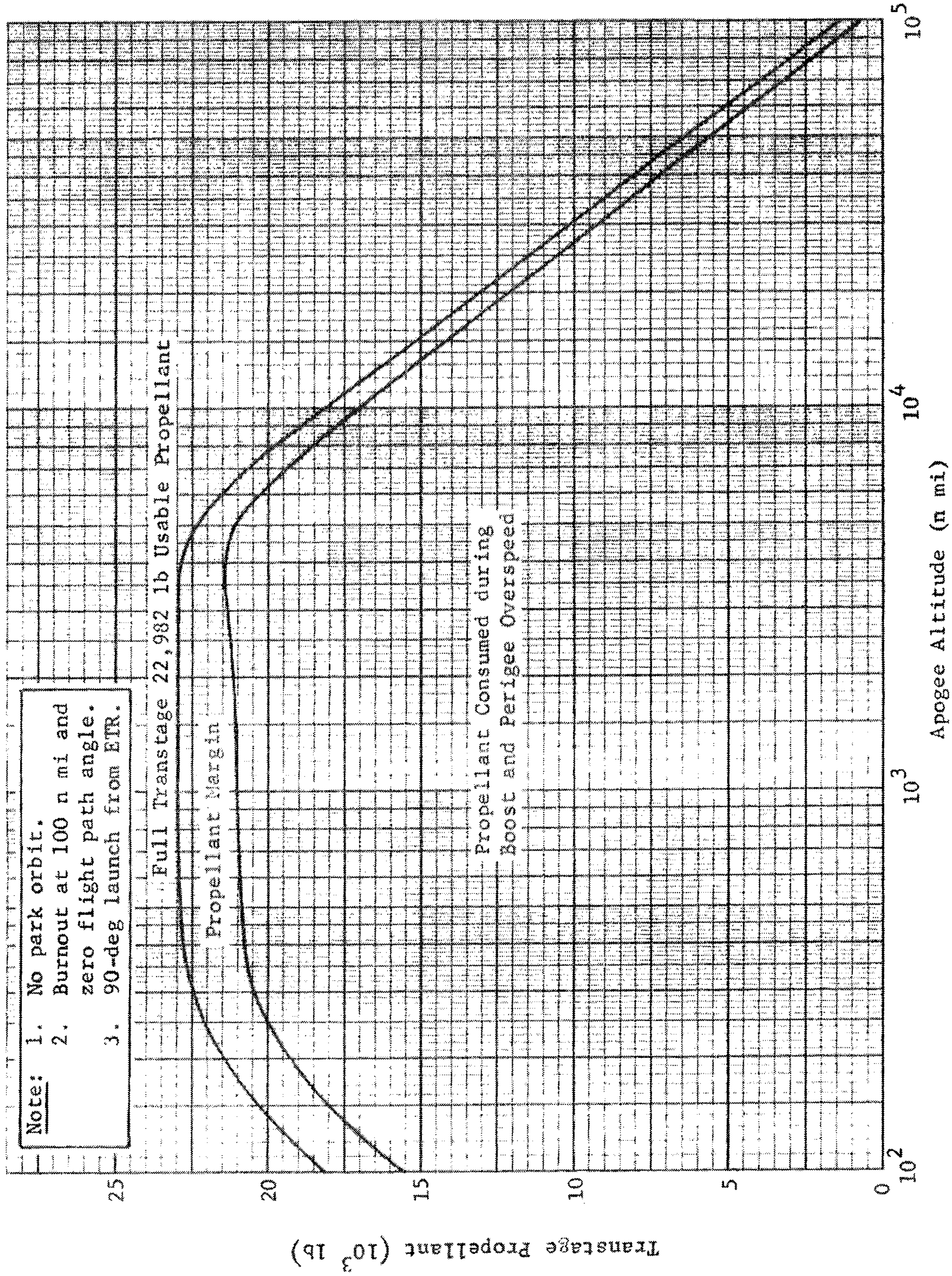

Fig. III-39 Transtage Propellant Usage, 3 CS-156, 100-n-mi Perigee Overspeed to Elliptical Orbits

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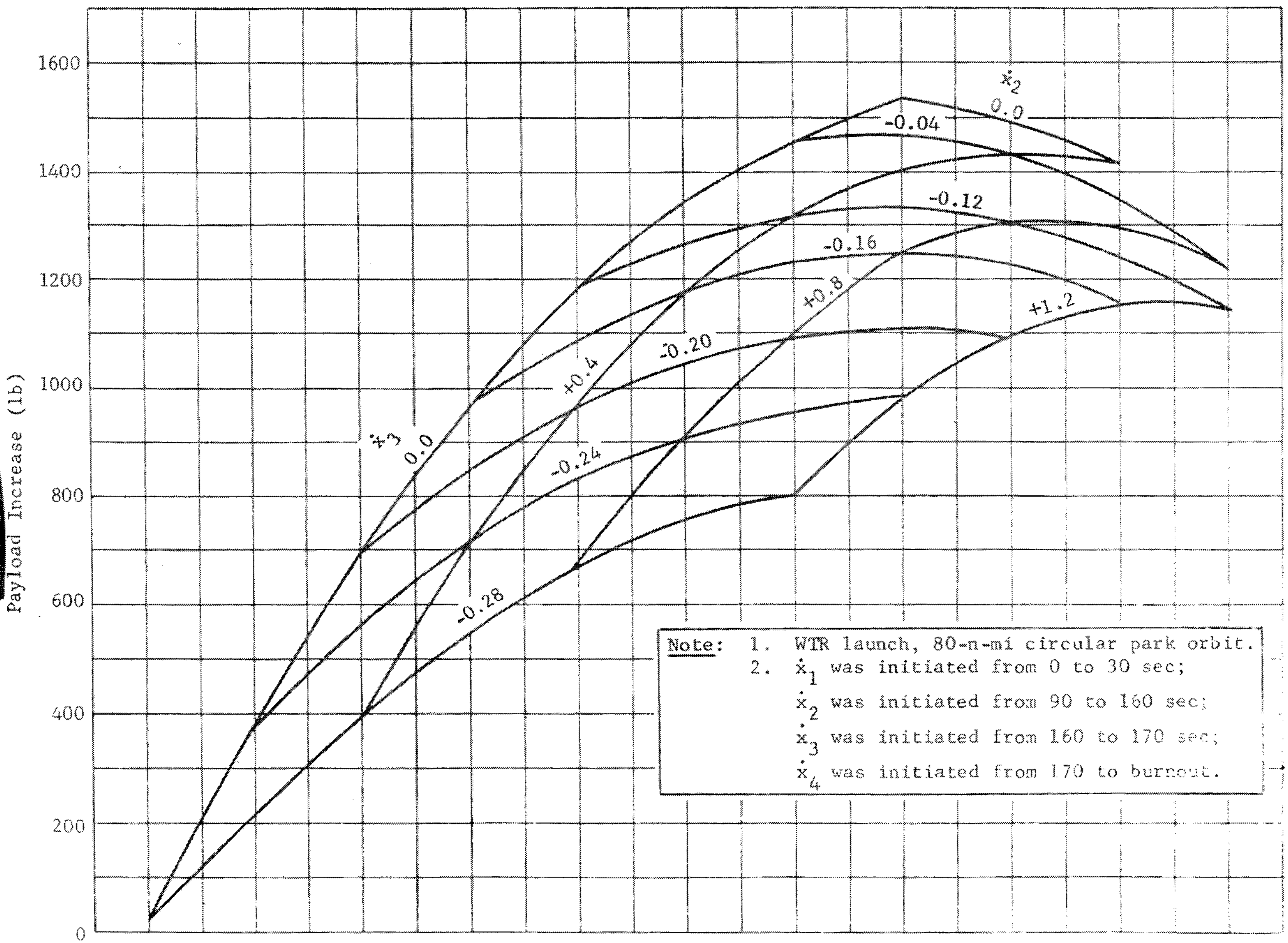

Fig. III-40 Trajectory Shaping, Payload Increase, 3 CS-156

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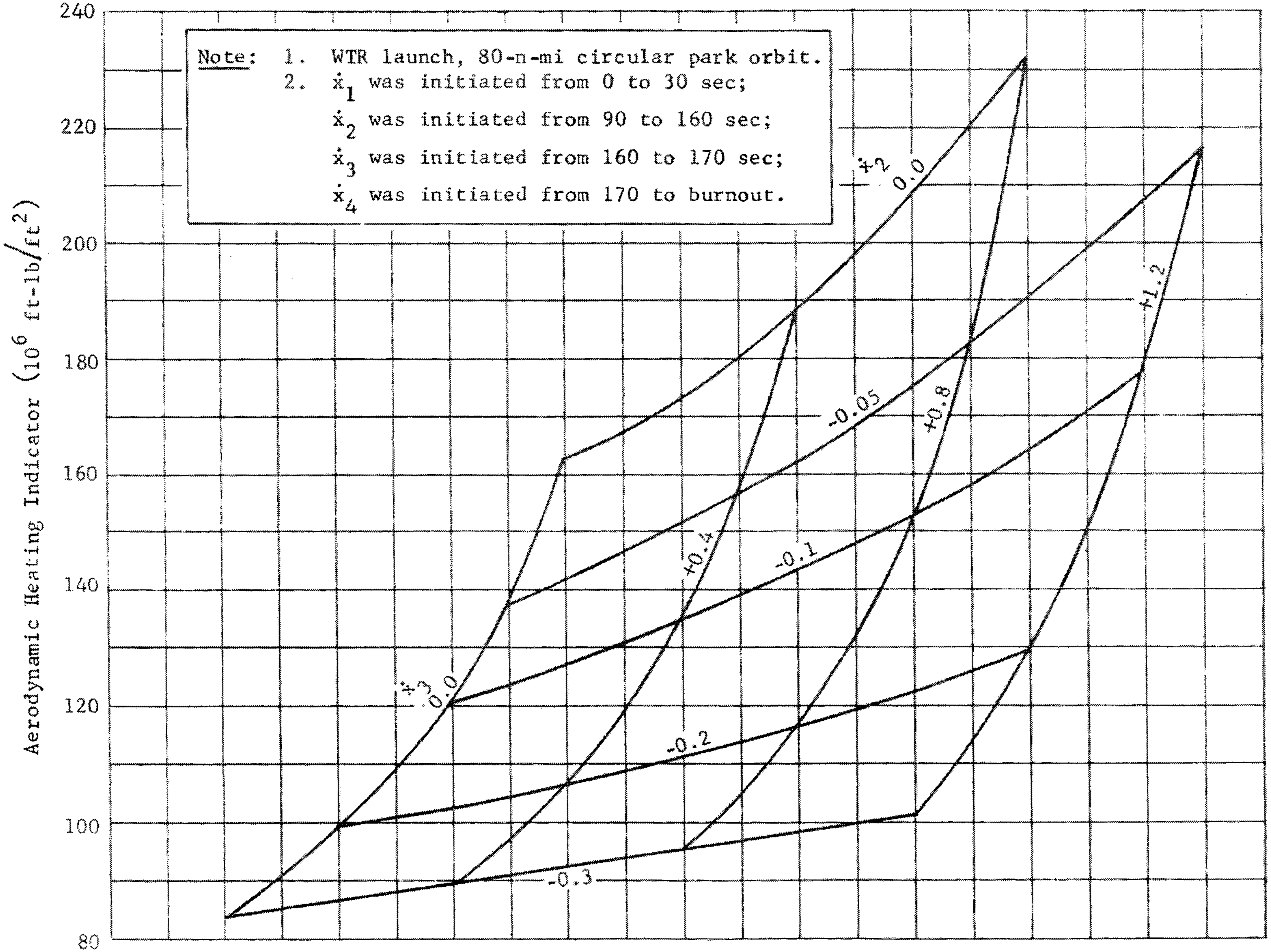

Fig. III-41 Trajectory Shaping, Aerodynamic Heating, 3 CS-156

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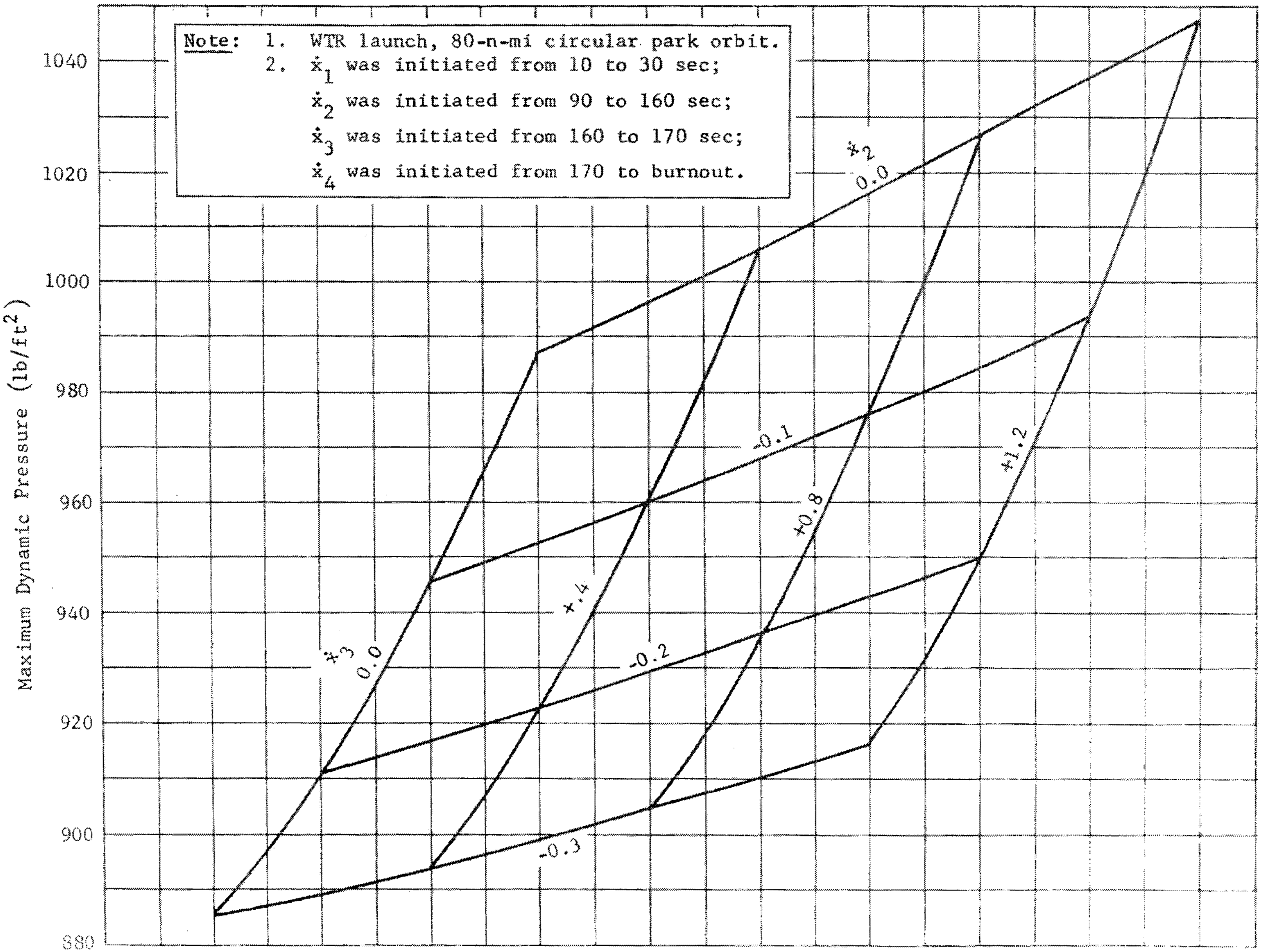

Fig. III-42 Trajectory Shaping, Maximum Dynamic Pressure, 3 CS-156

[redacted]

C. VEHICLE CHARACTERISTICS

The vehicle characteristics outlined in this section were used in the performance and trajectory studies.

1. SRM Staging Sequence

The Step 0/1 staging sequence was initiated when the vehicle's longitudinal acceleration decreased to 1.5 g. At this time Stage I engines were ignited. The SRMs were jettisoned 1 sec after zero thrust in the thrust time table.

2. Thrust-Time Histories

The thrust-time histories for the 7 seg-120, 2 CS-156, and 3 CS-156 are shown in Fig. III-43, III-44, and III-45, respectively.

Time histories for Stage I with a 15:1 expansion ratio (used with the 7 seg-120) and Stage I with an 8:1 expansion ratio (used with 2 CS-156 and 3 CS-156) are given in Tables III-3 and III-4, respectively. The Stage II time histories (used in all configurations) are tabulated in Table III-5. For all configurations 16,000 lb of thrust and 305 sec specific impulse were used for Stage III.

3. Liquid Propellant Inventory

The liquid propellant inventory is tabulated in Table III-6.

4. Thrust Vector Control (TVC) Fluid Inventory

The following TVC fluid loads were used for performance calculations:

7 seg-120 Loaded, 29,500 lb,
Expended, 21,000 lb;

2 CS-156 Loaded, 33,000 lb,
Expended, 21,000 lb;

3 CS-156 Loaded, 38,410 lb,
Expended, 30,090 lb.

These loads were not used for TVC studies.

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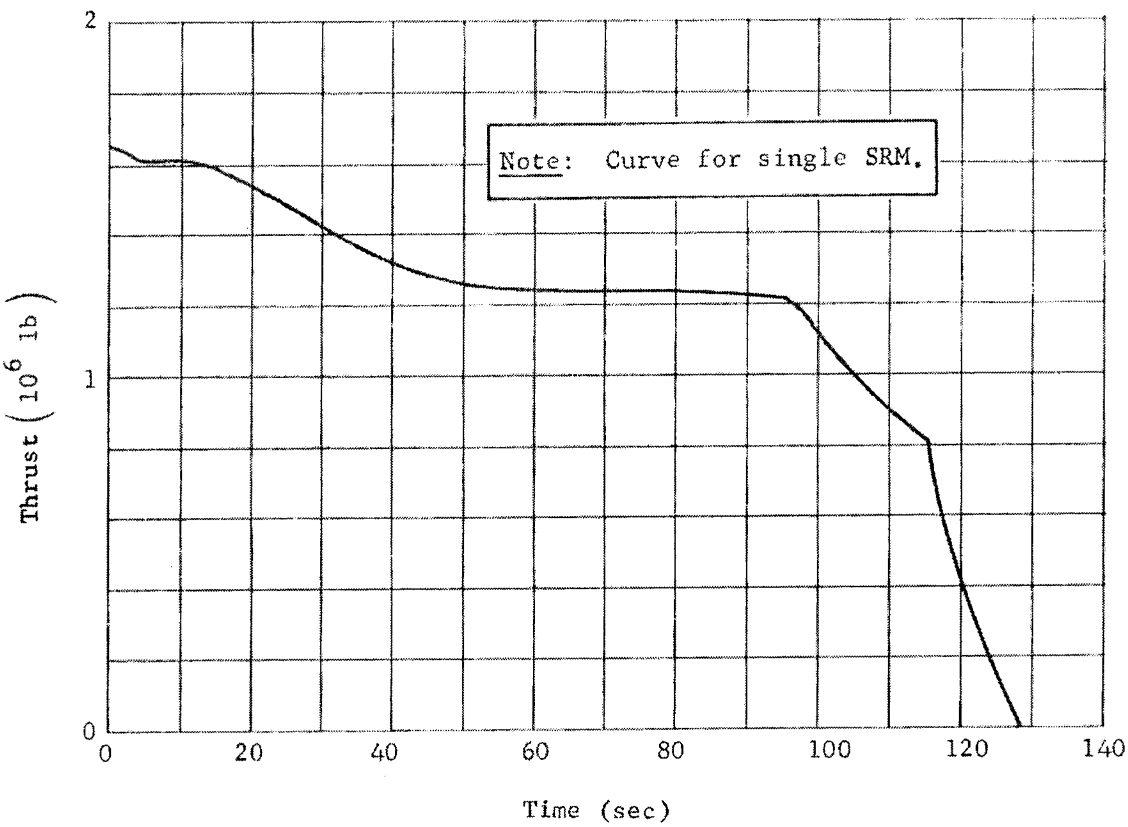

Fig. III-43 Thrust-Time History, 7 seg-120

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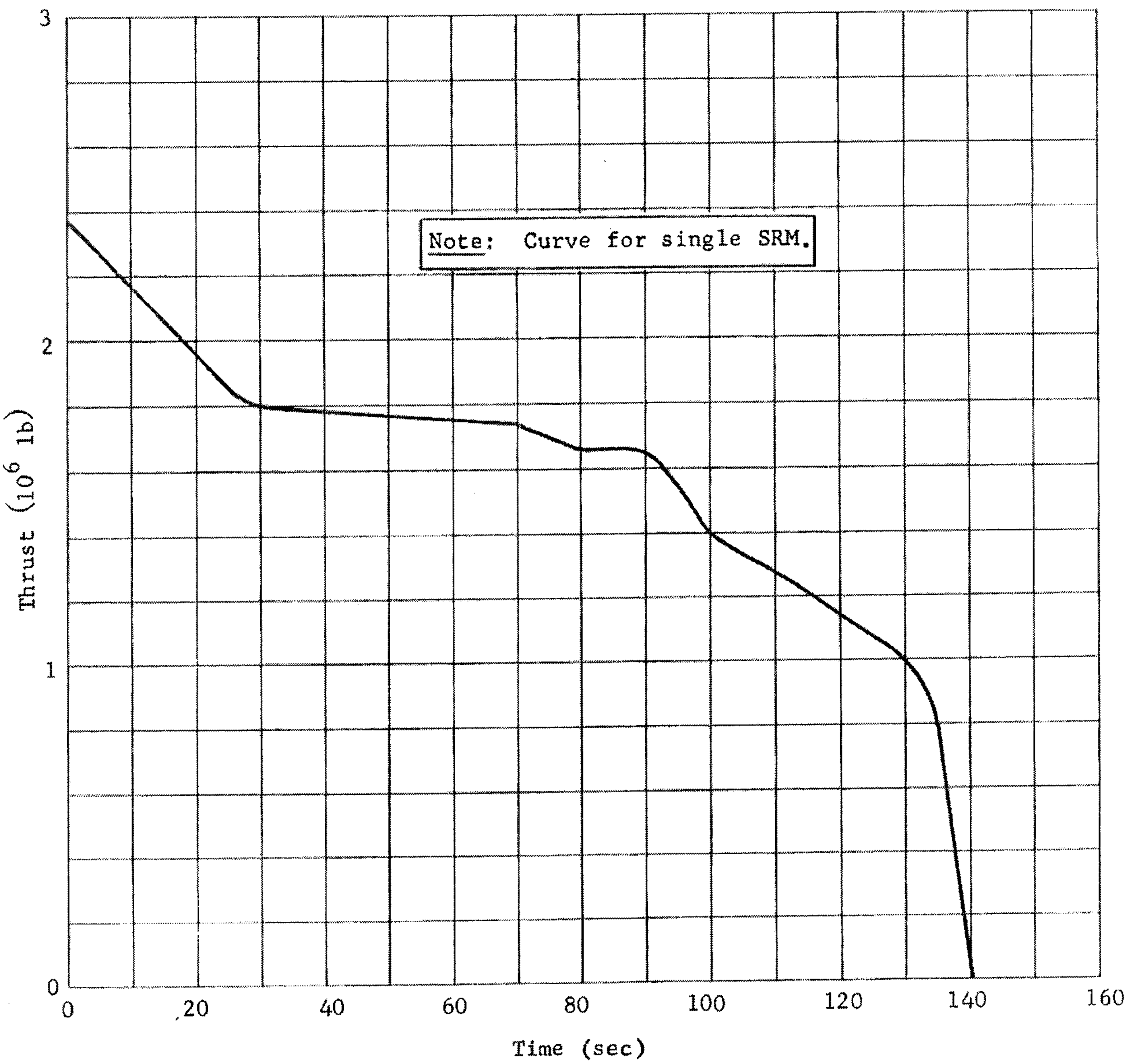

Fig. III-44 Thrust-Time History, 2 CS-156

[redacted]

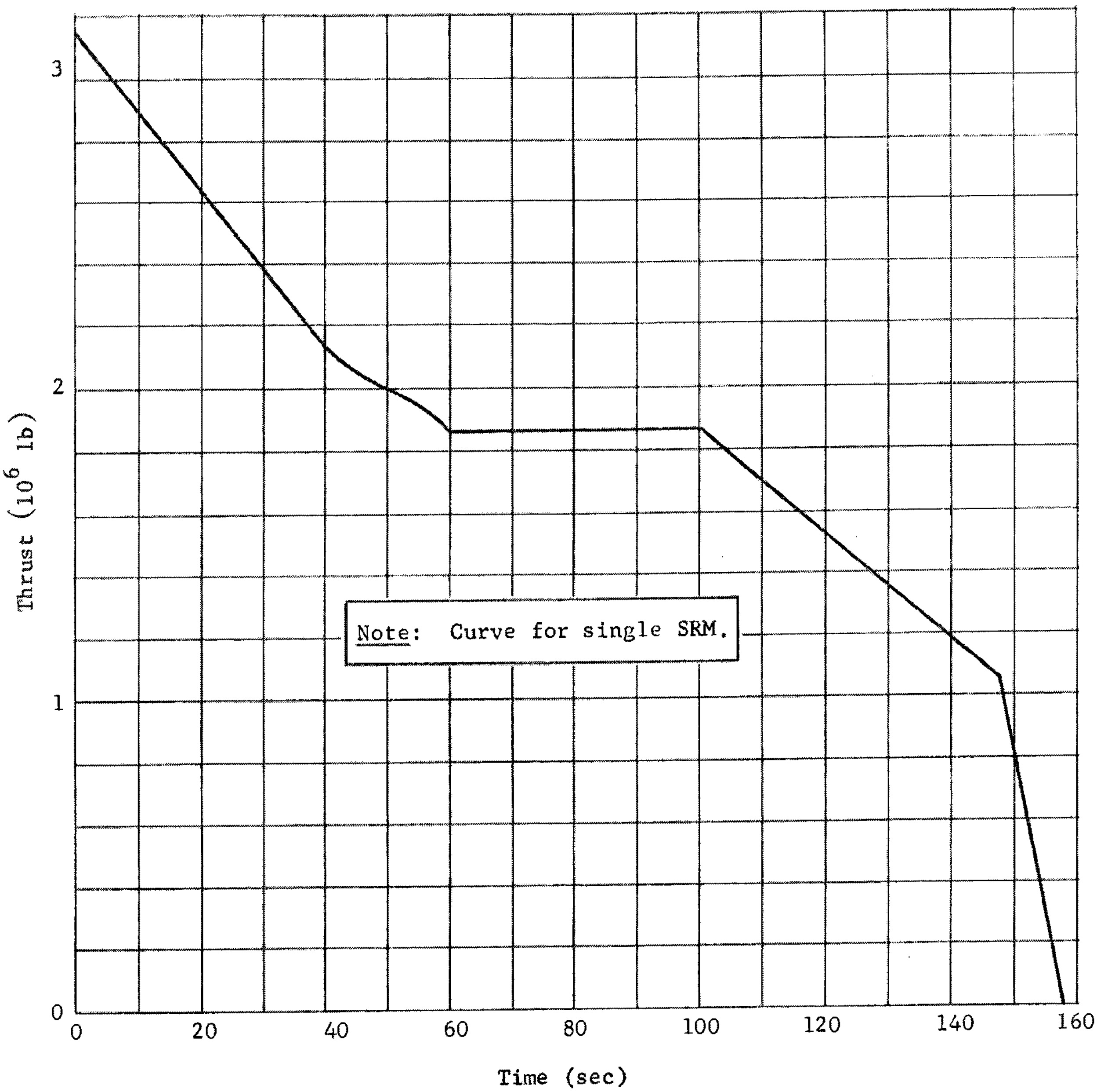

Fig. III-45 Thrust-Time History, 3 CS-156

Table III-3 Stage I Class Nominal, Expansion Ratio, 15:1; Used with 7 seg-120

Acceptance Test and Launch at 60°F		
Time from $87FS_1$ (sec)	Total Thrust (lb)*	Total Flow Rate (lb/sec)†
6	526,730	1761.6
10	527,000	1762.5
16	527,400	1763.9
20	527,660	1764.7
35	529,100	1769.6
55	531,000	1775.9
75	532,060	1779.5
90	532,060	1779.5
110	532,060	1779.5
125	532,060	1779.5
140	532,060	1779.5
150	532,060	1779.5

*Chamber centerline including gas generator.
†Autogenous flow excluded.

Cant angle: 2 deg. Exit area: 5628 $in.^2$

Table III-4 Stage I Class Nominal, Expansion Ratio, 8:1; Used with 2 CS-156 and 3 CS-156

Acceptance Test and Launch at 60°F		
Time from 87FS$_1$ (sec)	Total Thrust (lb)*	Total Flow Rate (lb/sec)†
6	479,417	1668.2
10	476,713	1658.6
16	477,006	1659.0
20	477,348	1660.2
35	478,442	1664.0
55	480,760	1672.2
75	481,704	1675.4
90	482,088	1677.0
110	482,190	1677.8
125	482,190	1677.8
140	482,190	1677.8
150	482,190	1677.8

*Chamber centerline including gas generator.
†Autogenous flow excluded.

Cant angle: 2 deg. Exit area: 3023.6 in.^{2}

Table III-5 Stage II Class Nominal, Expansion Ratio, 49.2:1; Used with All Configurations

Acceptance Test and Launch at 60°F		
Time from 91FS$_1$ (sec)	Total Thrust (lb)*	Total Flow Rate (lb/sec)†
6	95,971	306.1
10	97,038	309.6
16	98,724	315.0
20	99,338	317.0
35	100,643	321.1
55	101,452	323.8
70	101,759	324.8
100	102,154	326.1
120	102,157	326.2
150	102,199	326.4
180	102,201	326.5
205	101,597	324.4

*Chamber centerline including roll nozzle.
†Autogenous flow excluded.

Table III-6 Liquid Propellant Inventory

Nominal Values at 60°F	Step 1	Step 2	Step 3* (Full load)
Loaded Propellant	257,390	67,909	23,081
Expended before Stage 0 Liftoff	6	2	0
Propellant Aboard at Liftoff	257,384	67,907	23,081
Propellant Expended as Bleeds or Leakage	(41)	(23)	(0)
During Stage 0	29	6	0
During Stage I	12	9	0
During Stage II	0	8	0
Total Nonusable Propellant (Trapped, Vapor Retained)	1,135	404	14
Mean Outage	548	215	72
Propellant Consumed during Stage Operation	(244,660)	(67,265)	(22,995)
Start	248	193	5
Steady State	255,082	66,875	22,982
Shutdown and Tailoff	330	197	8
Pressurization System Inert Gas	24	11	60
Mean Outage Percentage (Based on Loaded Propellant)	0.212	0.32	0.31
Maximum Outage Percentage (Based on Loaded Propellant)	0.816	0.99	1.00
Mean Outage (lb)	548	215	72
Maximum Outage (lb)	2,100	675	231

*Step 3 (transtage) off-loaded in some cases.

5. Step Performance Weight Summaries

The performance weight summaries of the three configurations, with and without transtage are given by step in Tables III-7 thru III-12.

6. Axial Force Coefficient

The Stage 0 axial force coefficients for the 7 seg-120, 2 CS-156, and 3 CS-156 (with and without transtage) are given in Fig. III-46, III-47, and III-48, respectively. Stage I axial force coefficients are given for all configurations in Fig. III-49.

Table III-7 Step 0* Performance Weight Summary with and without Transtage

Item	SRM Configuration 2 CS-156	SRM Configuration 3 CS-156	SRM Configuration 7 seg-120
Weight Empty	233,560	287,790	180,758
Dry Weight			
Residuals			
TVC fluid trapped / TVC fluid reverse	12,000	8,320	8,500
Burnout Weight	245,560	296,190	189,258
Flight Expendables	(1,703,000)	(2,240,990)	(1,175,368)
Propellant	1,680,000	2,208,900	1,142,648
TVC Fluid for Control and Dump Schedule	21,000	30,090	21,000
Inert Material	2,000	2,000	11,720
Step Weight at Vehicle Liftoff	1,948,560	2,537,100	1,364,626

*Step 0 includes two SRMs.

Table III-8 Step 1 Performance Weight Summary with Transtage

Item	SRM Configuration		
	2 CS-156	3 CS-156	7 seg-120
Weight Empty	14,007	13,752	14,465
Telemetry	553	553	553
Dry Weight	14,560	14,305	15,018
Residual Propellant			
Trapped and Vapor Retained	1,135	1,135	1,135
Mean Outage	548	548	548
Lubricant	17	17	17
Pressure gas	24	24	24
Boattail	-271	-271	-271
Burnout Weight	16,013	15,758	16,471
Flight Expendables	(255,981)	(255,981)	(255,981)
Steady-State Propellant	255,082	255,082	255,082
Start Propellant	248	248	248
Shutdown Propellant	330	330	330
Igniter Charge	9	9	9
Engine Bleed Stage 0 Operation	29	29	29
Engine Bleed Stage I Operation	12	12	12
Thrust Chamber Closure (Boattail)	271	271	271
Step Weight at Vehicle Liftoff	271,994	271,739	272,452
Engine Bleed before Liftoff	6	6	6
Initial Step Weight	272,000	271,745	272,458

Table III-9 Step 1 Performance Weight Summary without Transtage

Item	SRM Configuration		
	2 CS-156	3 CS-156	7 seg-120
Weight Empty	14,077	13,822	14,540
Telemetry	553	553	553
Dry Weight	14,630	14,375	15,093
Residual Propellant			
Trapped and Vapor Retained	1,135	1,135	1,135
Mean Outage	548	548	548
Lubricant	17	17	17
Pressure gas	24	24	24
Boattail	-271	-271	-271
Burnout Weight	16,083	15,828	16,546
Flight Expendables	(255,981)	(255,981)	(255,981)
Steady-State Propellant	255,082	255,082	255,082
Start Propellant	248	248	248
Shutdown Propellant	330	330	330
Igniter Charge	9	9	9
Engine Bleed Stage 0 Operation	29	29	29
Engine Bleed Stage I Operation	12	12	12
Thrust Chamber Closure (Boattail)	271	271	271
Step Weight at Vehicle Liftoff	272,064	271,809	272,527
Engine Bleed before Liftoff	6	6	6
Initial Step Weight	272,070	271,815	272,533

Table III-10 Step 2 Performance Weight Summary with Transtage

Item	SRM Configuration		
	2 CS-156	3 CS-156	7 seg-120
Weight Empty	5,622	5,987	5,957
Telemetry	696	696	696
Dry Weight	6,318	6,683	6,653
Residual Propellant			
Trapped and Vapor Retained	404	404	404
Mean Outage	215	215	215
Lubricant	11	11	11
Pressure Gas	5	5	5
Ablative Material	-58	-58	-58
Burnout Weight	6,895	7,260	7,230
Flight Expendables	(67,349)	(67,349)	(67,349)
Steady-State Propellant	66,875	66,875	66,875
Start Propellant	193	193	193
Shutdown Propellant	197	197	197
Ablative Material	58	58	58
Igniter Charge	3	3	3
Engine Bleed Stage 0 Operation	6	6	6
Engine Bleed Stage I Operation	9	9	9
Engine Bleed Stage II Operation	8	8	8
Step Weight at Vehicle Liftoff	74,244	74,609	74,579
Engine Bleed before Liftoff	2	2	2
Initial Step Weight	74,246	74,611	74,581

Table III-11 Step 2 Performance Weight Summary without Transtage

Item	SRM Configuration		
	2 CS-156	3 CS-156	7 seg-120
Weight Empty	7,083	7,480	7,480
Telemetry	696	696	696
Dry Weight	7,779	8,176	8,176
Residual Propellant			
Trapped and Vapor Retained	404	404	404
Mean Outage	215	215	215
Lubricant	11	11	11
Pressure Gas	5	5	5
Ablative Material	-58	-58	-58
Burnout Weight	8,356	8,753	8,753
Flight Expendables	(66,349)	(67,349)	(67,349)
Steady-State Propellant	66,875	66,875	66,875
Start Propellant	193	193	193
Shutdown Propellant	197	197	197
Ablative Material	58	58	58
Igniter Charge	3	3	3
Engine Bleed Stage 0 Operation	6	6	6
Engine Bleed Stage I Operation	9	9	9
Step Weight at Vehicle Liftoff	75,705	76,102	76,102
Engine Bleed before Liftoff	2	2	2
Initial Step Weight	75,707	76,104	76,104

[redacted]

Table III-12 Step 3 Performance Weight Summary

Item	SRM Configuration		
	2 CS-156	3 CS-156	7 seg-120
Weight Empty	4,565	4,610	4,610
Telemetry	591	591	591
Dry Weight	5,156	5,201	5,201
Residual Propellant			
Trapped and Vapor Retained	14	14	14
Mean Outage	72	72	72
Pressure Gas	64	64	64
Residual Attitude Control Propellant	7	7	7
Residual IGS Coolant and Gas	5	5	5
Ablative Material	-30	-30	-30
Burnout Weight	5,228	5,333	5,333
Flight Expendables	(23,163)	(23,163)	(23,163)
Steady-State Propellant (Full Load)	22,982	22,982	22,982
Start Propellant (One Burn)	5	5	5
Shutdown Propellant (One Burn)	8	8	8
Attitude Control Propellant (Total Usable)	113	113	113
IGS Coolant and Gas (Total Usable)	25	25	25
Ablative Material	30	30	30
Step Initial Weight	28,451	28,496	28,496

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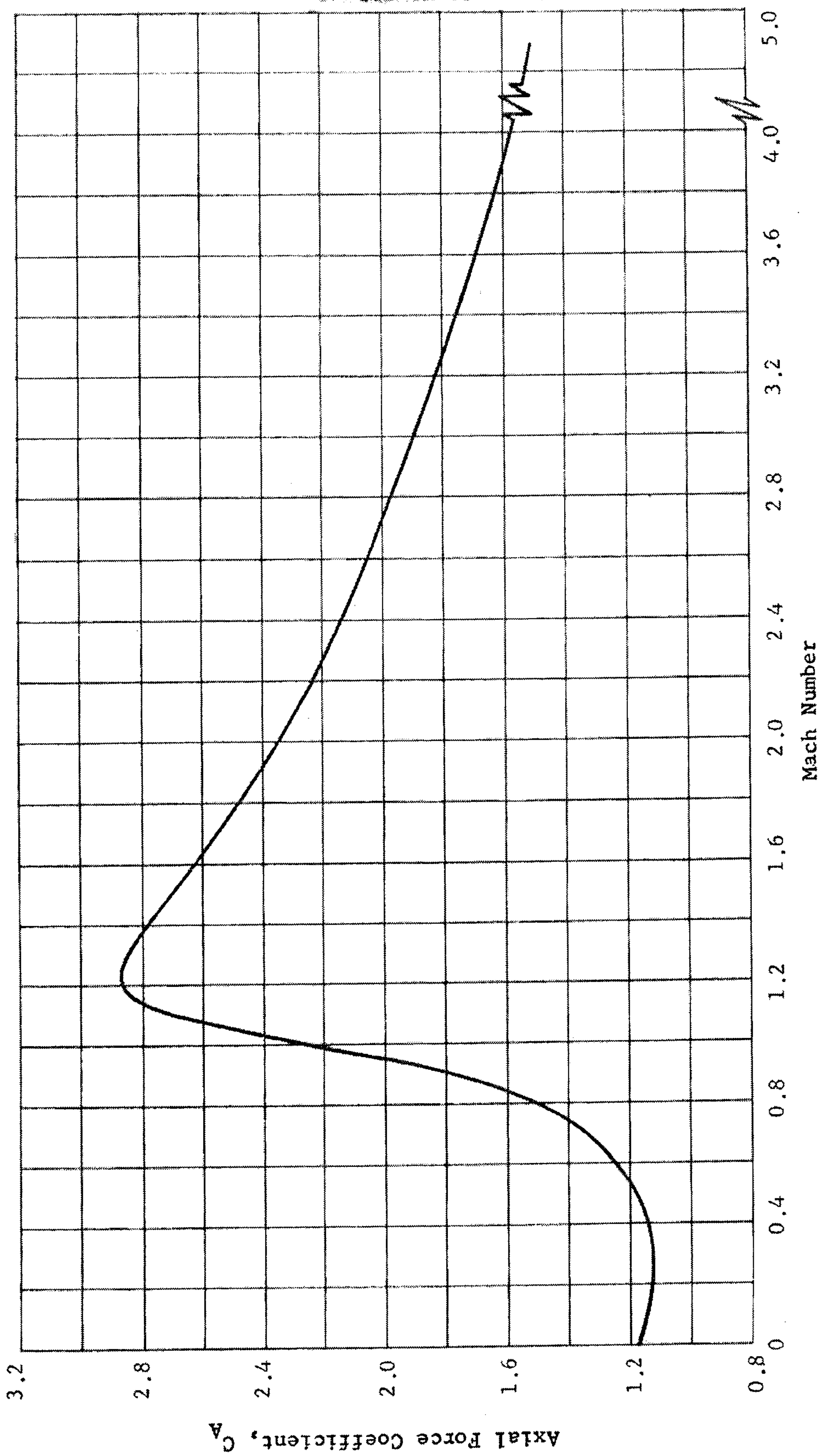

Fig. III-46 Axial Force Coefficient, 7 seg-120, Stage 0

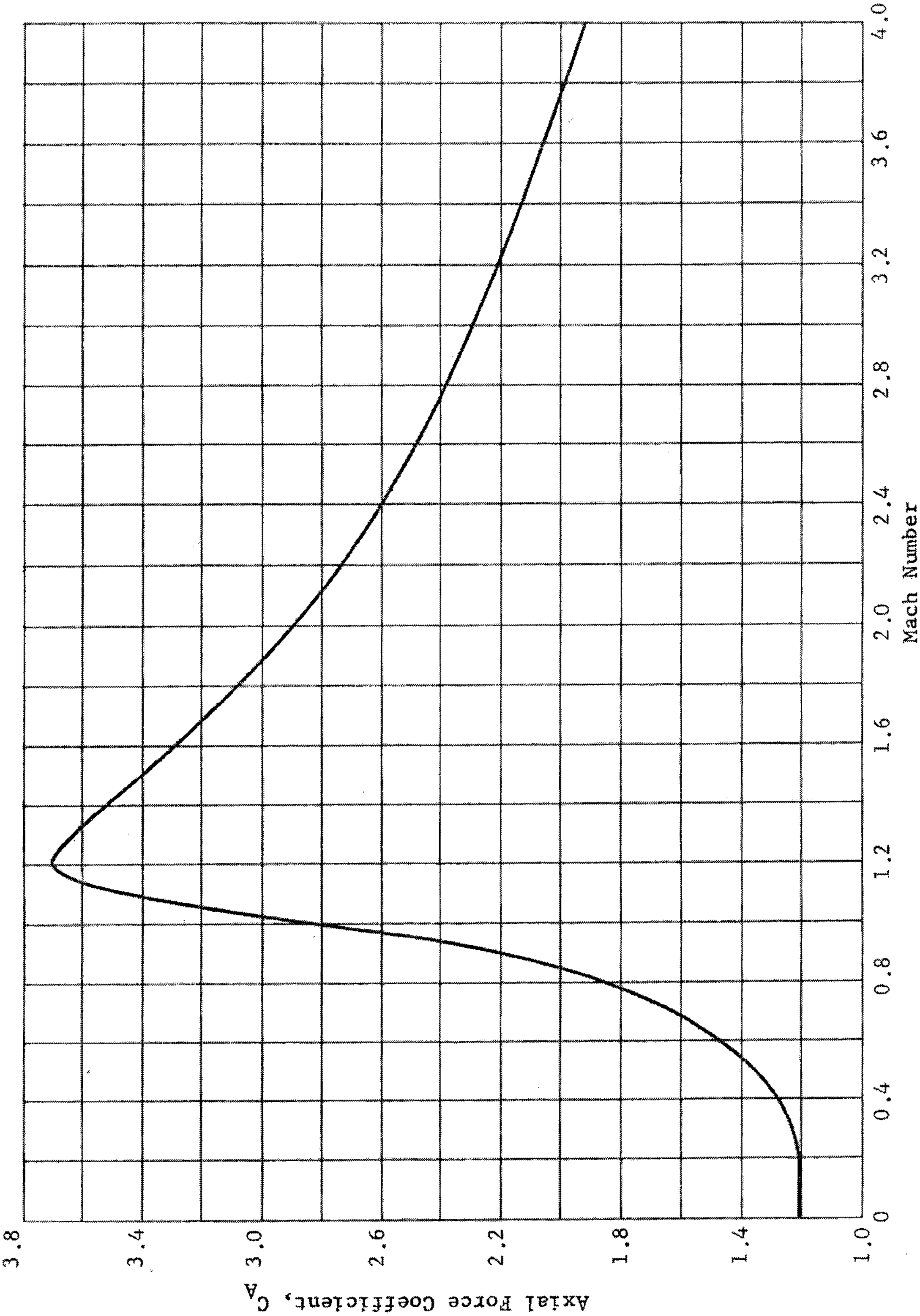

Fig. III-47 Axial Force Coefficient, 2 CS-156, Stage 0

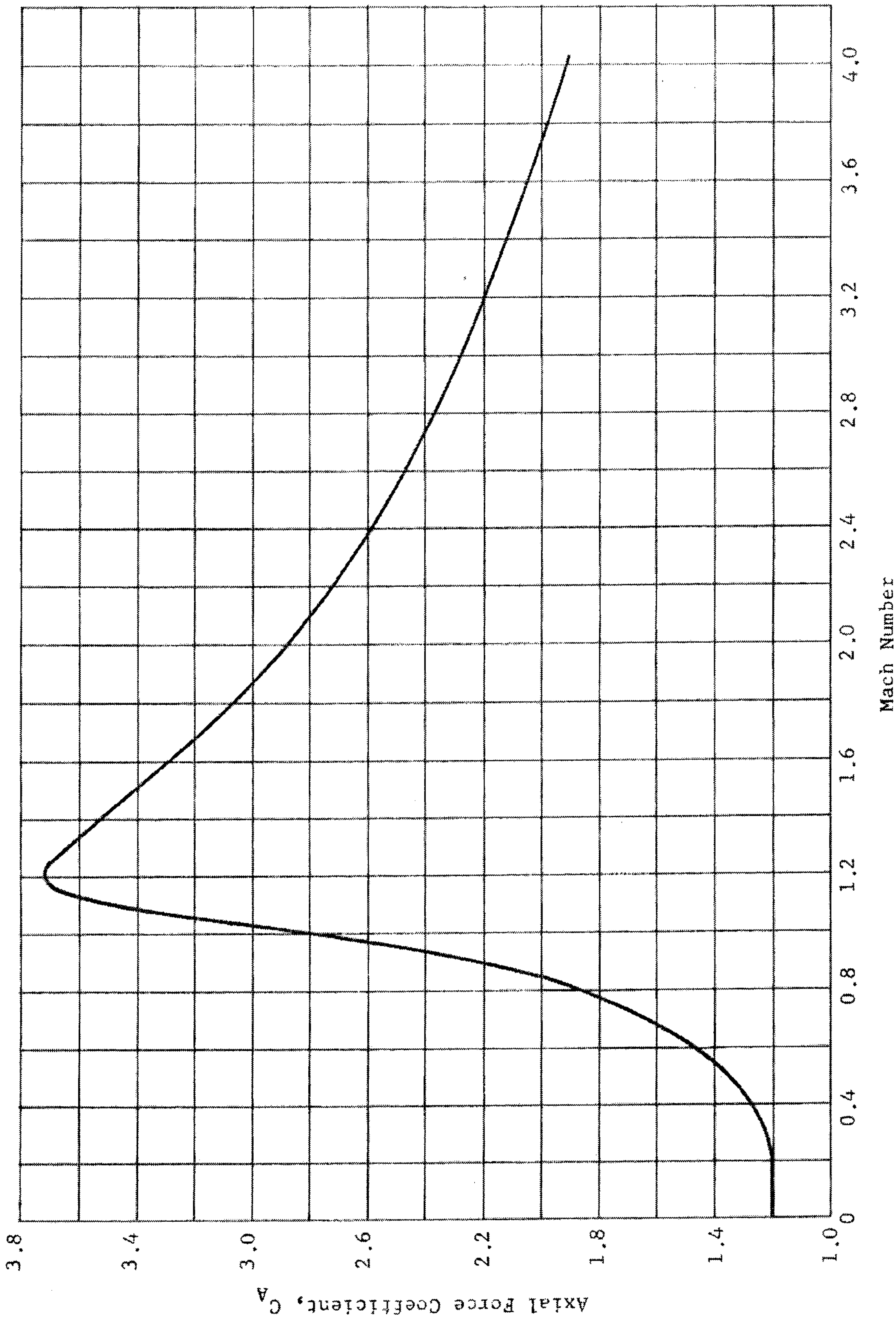

Fig. III-48 Axial Force Coefficient, 3 CS-156, Stage 0

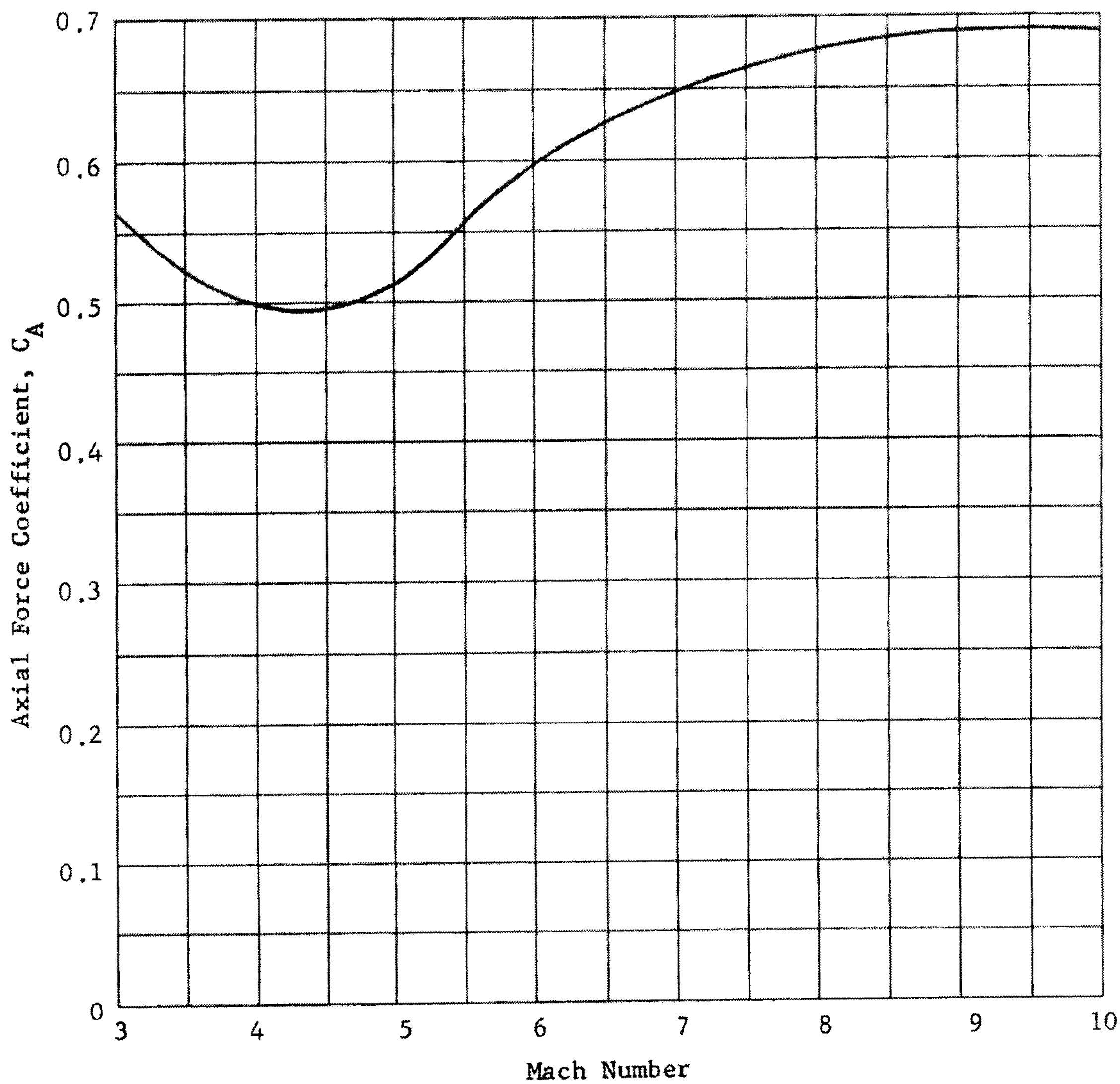

Fig. III-49 Axial Force Coefficient, Stage I, All Vehicles

IV. CONTROL SYSTEMS

This chapter discusses the studies conducted pertaining to the flight control system (FCS) stability, and thrust vector control (TVC) system sizing. The results of the vibration analysis that determined the mode shapes and frequencies of the flexible vehicle are presented. Basic problems that were evaluated are:

1) The effect of the large solid rocket motors (SRM) and large payloads upon the FCS;

2) The requirements imposed on the TVC system in terms of total injectant required and maximum side force requirements.

A. CONTROL SYSTEM STABILITY

The primary objective of the flight control analysis was to determine if any limitations existed on the MOL payload size or the type of solid motors due to flight control stability. The problems studied to determine restrictions to the payload and/or the solid motor configuration were:

1) Basic stability of Stages 0, I, and II;

2) The load-relief capability of the Stage 0 FCS;

3) The coupling of actuator and structural bending effects for Stages I and II.

In most situations it is more desirable to restrict and limit the FCS than the payload size or type of solid motors. Therefore, the second objective of the study was to determine what restrictions and requirements must be applied to the FCS to control the various payload and solid motor configurations.

The following discussion is subdivided into two main subsections. The first subsection explains all limitations and requirements imposed by the stability of the vehicle on the payload, SRMs, and FCS.

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The second subsection discusses in more detail the analysis that was performed and the problems encountered in the analysis. The last two sections summarize the systems designed in this study and the criteria used in the analysis.

1. Summary of System Requirements Imposed by Flight Control Considerations

The flight control studies showed that with respect to stability and load-relief capability of the FCS there is little difference between the three solid rocket configurations. As the payloads and solids become heavier the structural bending mode frequencies become lower, and present a somewhat more difficult stability situation. However, as shown in Fig. IV-1, the structural frequencies for the three solid configurations are near each other; Therefore each configuration presents essentially the same stability problems. This point is further confirmed by observing the similarity of the autopilots designed for each of the solid configurations and their associated payloads (tabulation of the autopilot configurations is presented in Chap. IV-IV.A.4).

The studies have shown that there is very little difference between the maximum and the minimum length payload for any one of the three solid configurations regarding control system stability. However, as the payload becomes longer the stability problem tends to be slightly more complex. This is shown by the corresponding decrease in load-relief capability of the FCS (Fig. IV-2). The slopes of the lines in Fig. IV-2 are rather small, showing that there is a slight difference in the load-relief capability between the longest and shortest payloads for any one solid configuration.

The FCS studies can be extrapolated to longer payloads than those studied. This extrapolation is, however, limited to an additional 10 ft. The stability problems will not become significantly more complex for the 10 ft range of longer payloads than were observed in the present study. Therefore, it can be expected that these extrapolated payloads will possess essentially the same amount of load relief as given by extrapolating Fig. IV-2. The limitation exists because extrapolation beyond 10 ft presents certain stability problems that the present analysis has not adequately investigated. These stability problems can be solved, but it is questionable that the same degree of load relief can be maintained.

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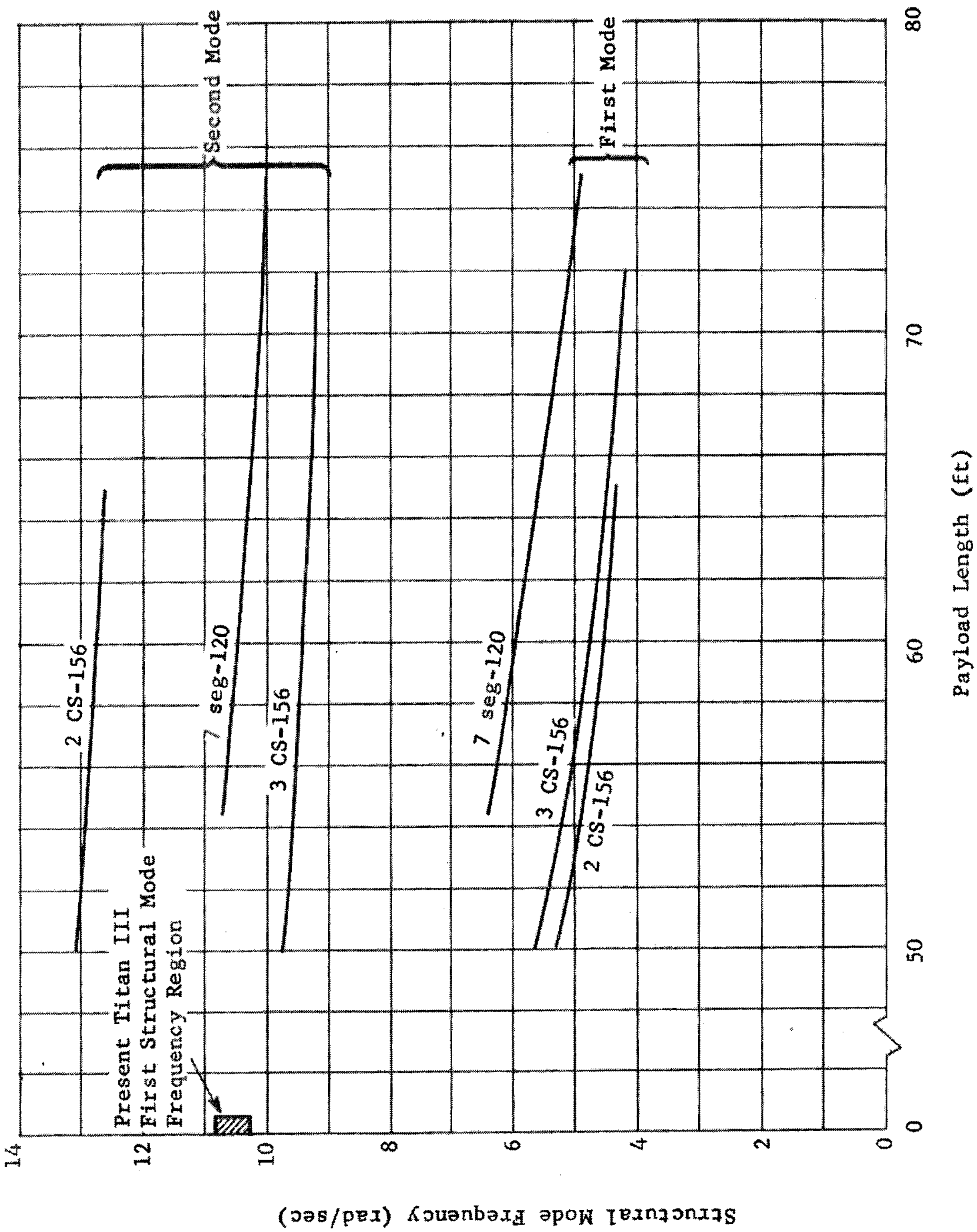

Fig. IV-1 Yaw Structural Mode Frequencies; Flight Time = Liftoff, with Transtage

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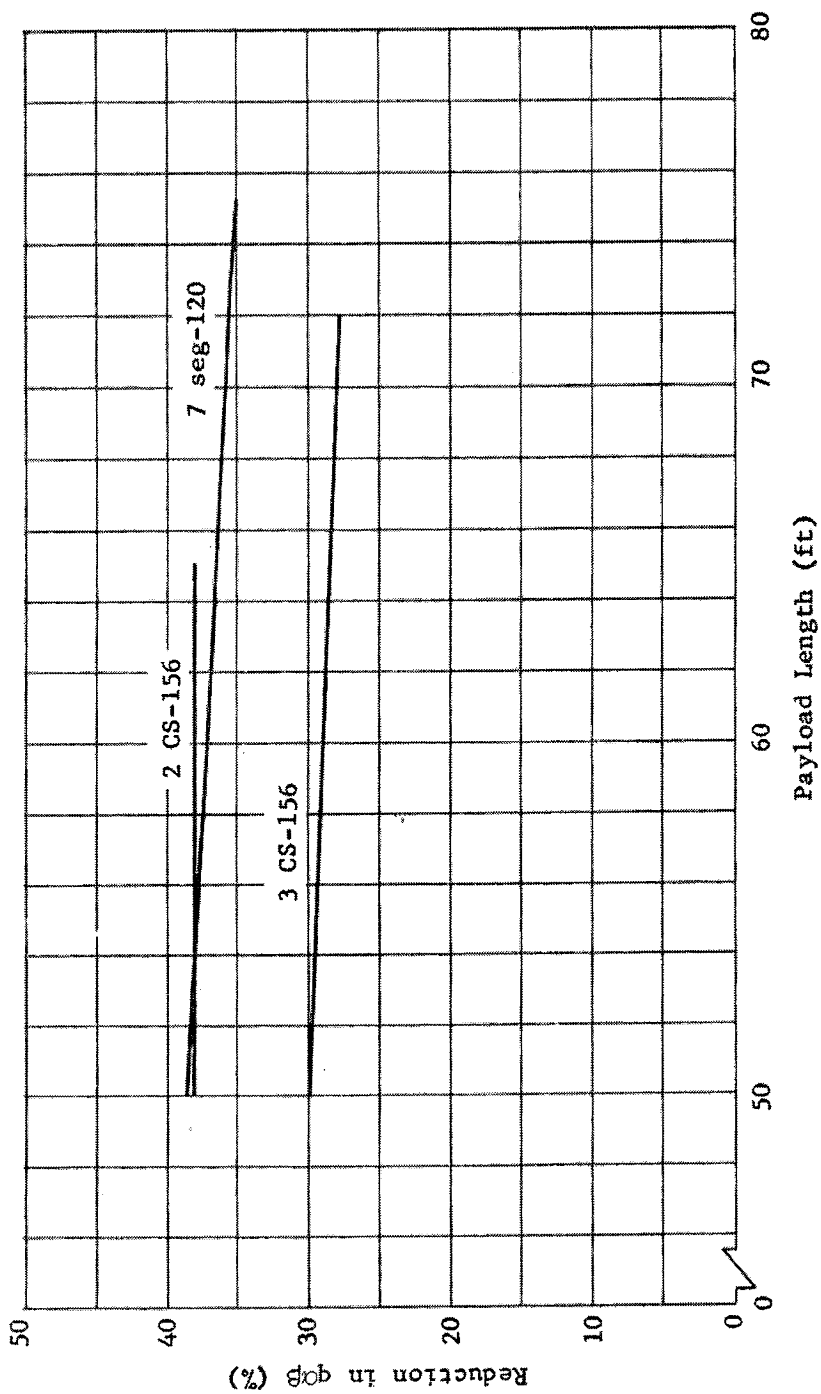

Fig. IV-2 Load Reduction by the FCS, with Transtage

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The FCS studies have defined an autopilot configuration for Stages 0, I, and II that will reasonably meet all stability criteria as defined by the specification requirement. This configuration is considered to be a minimum required capability autopilot since it is not known at this time what effect tolerances will have on the FCS. Tolerance problems may dictate additional capability and changes in the FCS.

Stage 0 - The autopilot concept presently being used for the MOL/Titan III Stage 0 is similar to the one used in the Titan IIIC program. A block diagram of the pitch channel autopilot is presented in Fig. IV-3. The only major deviation from the present Titan IIIC autopilot is the requirement for a filter change when the load-relief system is removed at about 80 sec of flight time. Also there is the possibility of a requirement for one more additional gain change between 80 sec and Stage 0 burnout to optimize the system configuration.

Due to the significant amount of load relief provided by the FCS, particularly in the yaw plane, the transient is significant when load relief is removed. Future studies may show that this transient must be eliminated, or reduced, through additional autopilot mechanization.

Stage I - The autopilot for Stage I MOL at the present is essentially the same as Stage I autopilot of the Titan IIIC program. The only difference is that there can be no filter sharing with Stage 0 filters. Each stage will require independent filters.

Stage II - In Stage II the first structural mode presents a significant phasing problem. The method selected to solve the problem was similar to the Titan IIIB approach, i.e., insertion of two underdamped quadratic filters in the rate feedback loop. In addition, one in-flight gain change will be needed near Stage II midflight.

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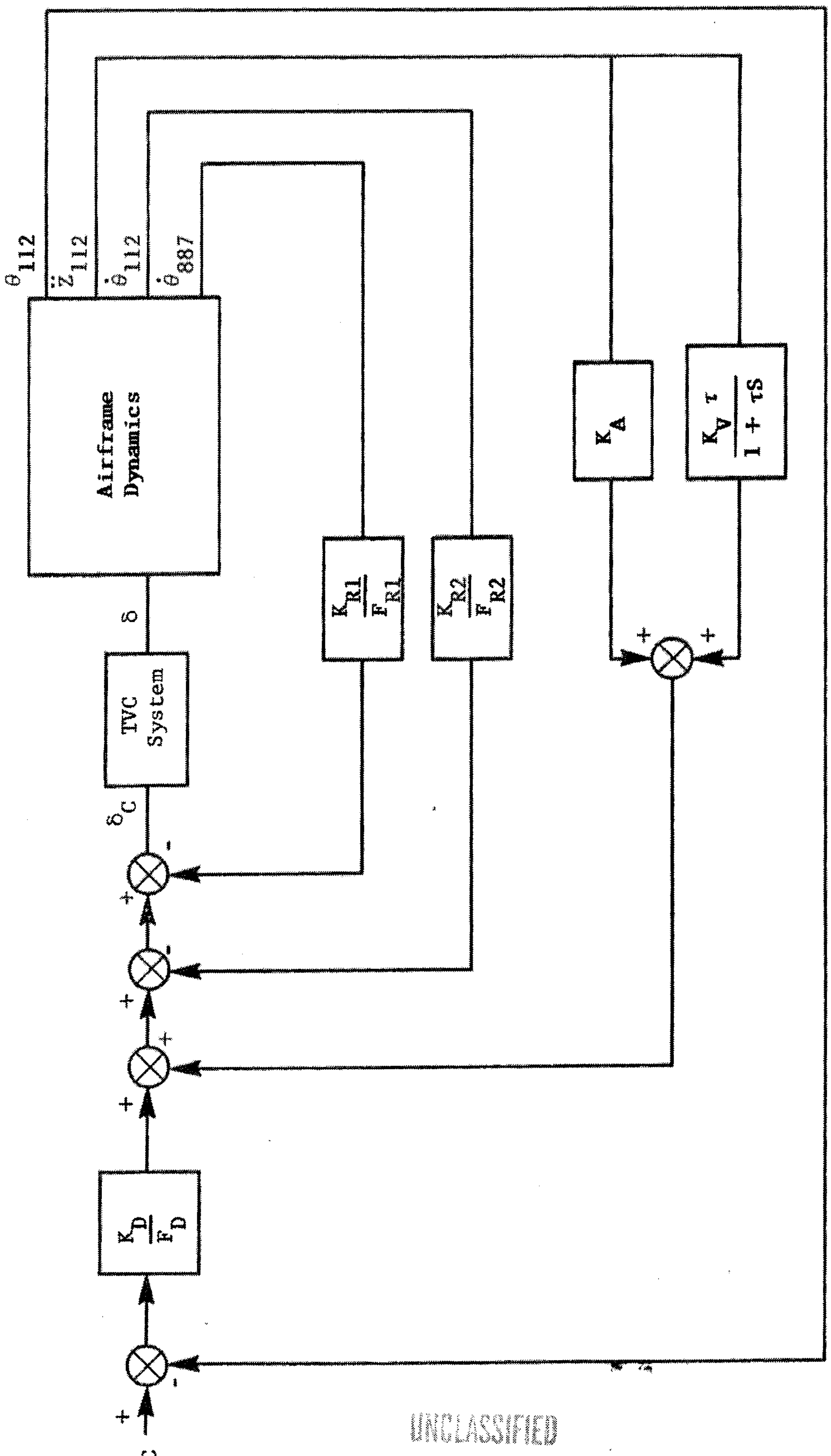

Fig. IV-3 Pitch Autopilot

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The sensor locations selected for the MOL configurations with transtage are: aft rate gyro, Station 887; forward rate gyro, Station 112; and accelerometer, Station 112. The reason for these sensor locations is demonstrated in Fig. IV-4. At Station 112 the amplitude of the first structural mode is small, allowing large accelerometer channel gains, and the slope of the second structural mode is small, thus minimizing second mode effects in the rate signal. For the configurations without transtage, the mode shapes are virtually unchanged, and sensor locations between Stations 235 and 257 are equally applicable. It is important to point out that these sensor locations are satisfactory for all three solid configurations and their associated payloads. For longer payloads than the ones studied these sensor locations will probably also be satisfactory. However, this depends on what shape the second mode takes when it reaches a frequency low enough to couple with the fluid slosh modes. For shorter payloads than those studied, the sensor locations will work well until the payload becomes less than 45 ft.

The Stage II main engine actuator presently being used on the Titan IIIC program will not suffice for the MOL vehicle due to the structural bending and actuator coupling. The FCS studies show that this problem can be eliminated if the Stage II actuator and engine changes used on the Titan IIIB are used for the MOL booster. If the Titan IIIB actuator is used, the minimum Stage II engine frequency that can be allowed is 7.5 cps.

For Stage I utilizing a 15:1 expansion ratio engine, the structural bending and actuator coupling is not critical. The minimum engine frequency that can be allowed is 8.0 cps.

It is recommended that closed-loop guidance not be used during Stage O. Closed-loop guidance could result in significant loss in load relief. This would be due to:

1) Changes in autopilot gains to offset loss in aerodynamic margin due to guidance;

2) The response of the vehicle to guidance signals in opposition to load relief.

The most significant effect on the FCS of removing the transtage is that the forward-located sensors will have to be relocated farther back in Stage II. The effect of relocating sensors was investigated with the result that this relocation will have only a small effect on load-relief capability and basic stability of the vehicle.

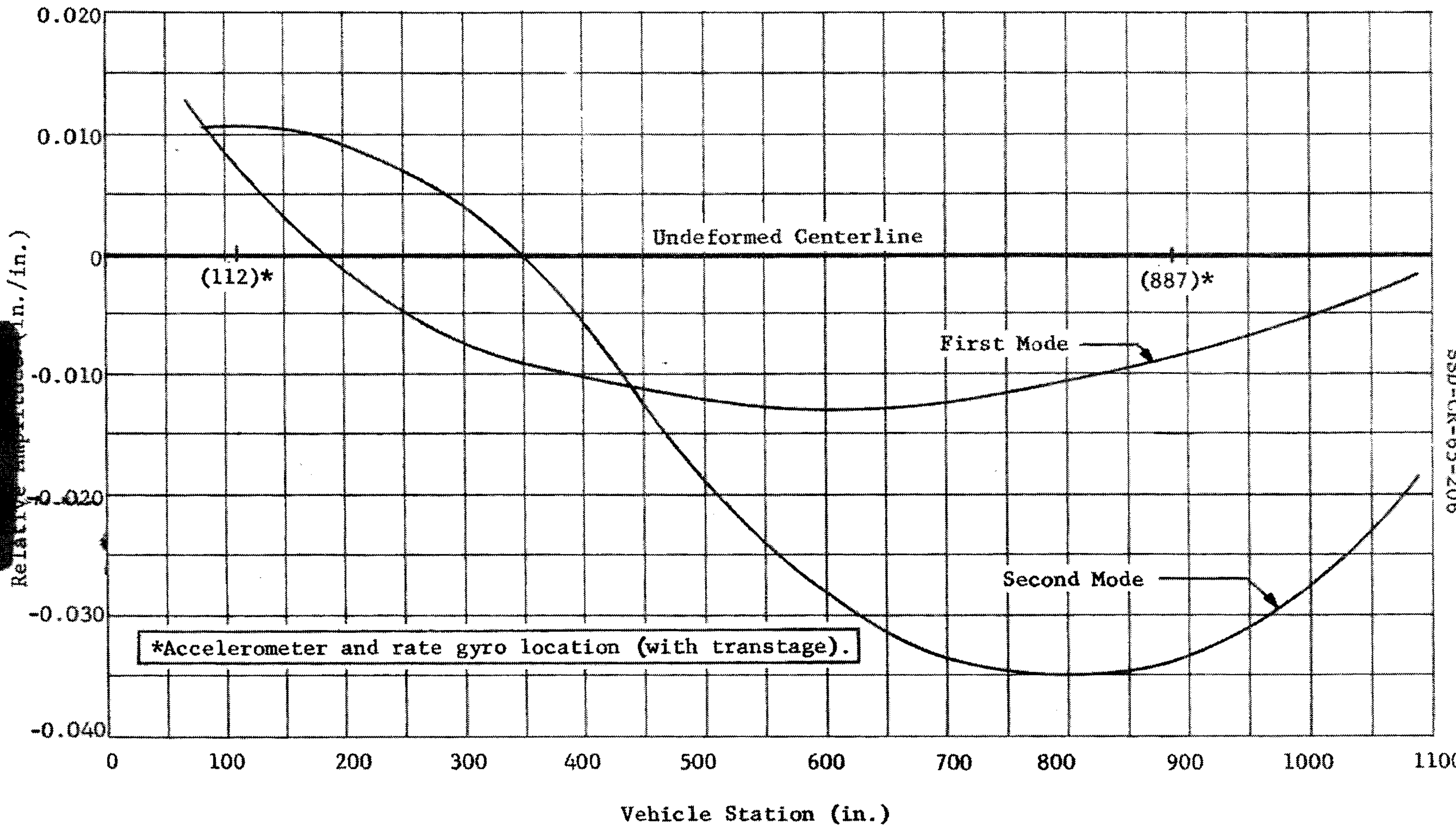


Fig. IV-4 Typical First and Second Mode Shapes

2. Flight Control Study Analysis

Stage 0 - The approach taken in the design of the Stage 0 FCS was to maximize the load-relief capability of the control system. Load-relief maximization was performed by designing the FCS to the open-loop margin requirements only. These requirements are presented in Chap. IV.A.4, following. Further optimization of the autopilot design will not increase the load-relief capability, but will improve the stability margins to the design objective values. Consequently, more effort spent on optimizing the autopilot will not result in any significant amount of additional load relief.

The stability problems encountered in Stage 0 were similar for all payloads and all three solid rocket configurations. Therefore, the following discussion of problem areas will apply to all payloads and solid configurations under study.

One of the major problems associated with the large payloads and solids is the low frequencies of the structural bending modes. Figure IV-1 shows frequencies of the first and second modes as a function of payload length. To stabilize these low frequency modes, low frequency filtering is required. This is depicted by the autopilot filters tabulated in Chap. IV.A.3. This filtering results in significant phase lag in the rigid body region, which intensifies the problem of maintaining the rigid body margins.

The low frequencies of the structural bending modes also produced considerable difficulty in maintaining the criteria of the closed-loop roots. The low frequency of the higher modes (second to the fourth) resulted in rather large bending amplitudes. To stabilize these modes to the 10-db requirement, the autopilot gains had to be kept as low as possible. The lowering of the gains made the closed-loop root criteria difficult to meet; In a few cases the maximum closed-loop root frequency that could be obtained was less than the criteria of 1.5 rad/sec. The lowest value obtained was 1.3 rad/sec. One reasonably successful scheme that was tried to solve this problem was a filter change at 80 sec (when load relief was removed). If closed-loop frequencies of 1.5 rad/sec or greater are required during portion of Stage 0 flight without load relief, then an in-flight filter change will definitely be required for Stage 0.

To minimize the stability problems of the acceleration feedback it is best to locate the sensor near a mode of the first structural bending mode, or at a position of small first mode displacement. Also, because it allows for more filtering, it is desirable to locate the accelerometer so that the first mode bending is positive. In Fig. III-4 the sensor locations selected for the MOL vehicles are shown. The forward rate gyro was located at Station 112 instead of Station 320 as in the Titan IIIC. This was to allow for a rate gyro for Stage III and to help solve the stability problems of Stage II.

The reduction in loads obtained for each FCS configuration is presented in Fig. IV-2. The primary reason for the significant amount of load relief is the relatively large K_V. In previous MOL studies K_V was kept low due to a low frequency instability that can occur when K_V is increased. The previous MOL studies used the ground rule that this low frequency root must be stable. For this study this ground rule was deleted because this root, even though unstable, has a time constant of more than 120 sec and does not cause any appreciable problem. With this ground rule discarded, K_V can be increased significantly over what has previously been observed in MOL studies.

The only significant problem caused by increasing K_V is that the attitude error during load relief will be large. The trajectory studies show attitude error for the MOL vehicles with these K_V's to be around 5 deg at wind shear peak. Due to this large attitude error, the transient when load relief is removed at 80 sec is significant. If this transient becomes a problem in later studies, it can be solved by additional autopilot mechanization. This autopilot mechanization will be of a nature that will not affect the load-relief capability of the system. An example of this would be ramping out the gains when load relief is removed.

Another reason load relief of the FCS is large is because the first bending mode signal sensed by the accelerometer is small and easily filtered out. With no bending, the only limitations to the accelerometer gain are rigid body considerations which, in general, are not as constraining as structural bending.

Typical stability plots for the 60-sec cases are included in Fig. IV-5 thru IV-16. Two general configurations were considered for MOL, with transtage and without transtage. The bulk of the flight control analysis was done on the configuration with the transtage. If the transtage is removed the primary effect on the FCS will be the relocation of the forward sensors. To determine this effect, a study was made with the forward-located rate gyro and accelerometer relocated at Station 235. The results of that study showed that there was no significant difference between the autopilots for the two sensor locations. The only difference was that the rate gain ratio had to be changed, but this has no effect on the load-relief capability of the FCS. Therefore, the stability problems and load-relief capability of the FCS for the two configurations (with and without transtage) are essentially the same. Figures IV-17 thru IV-20 illustrate the comparison of the cases with and without transtage.

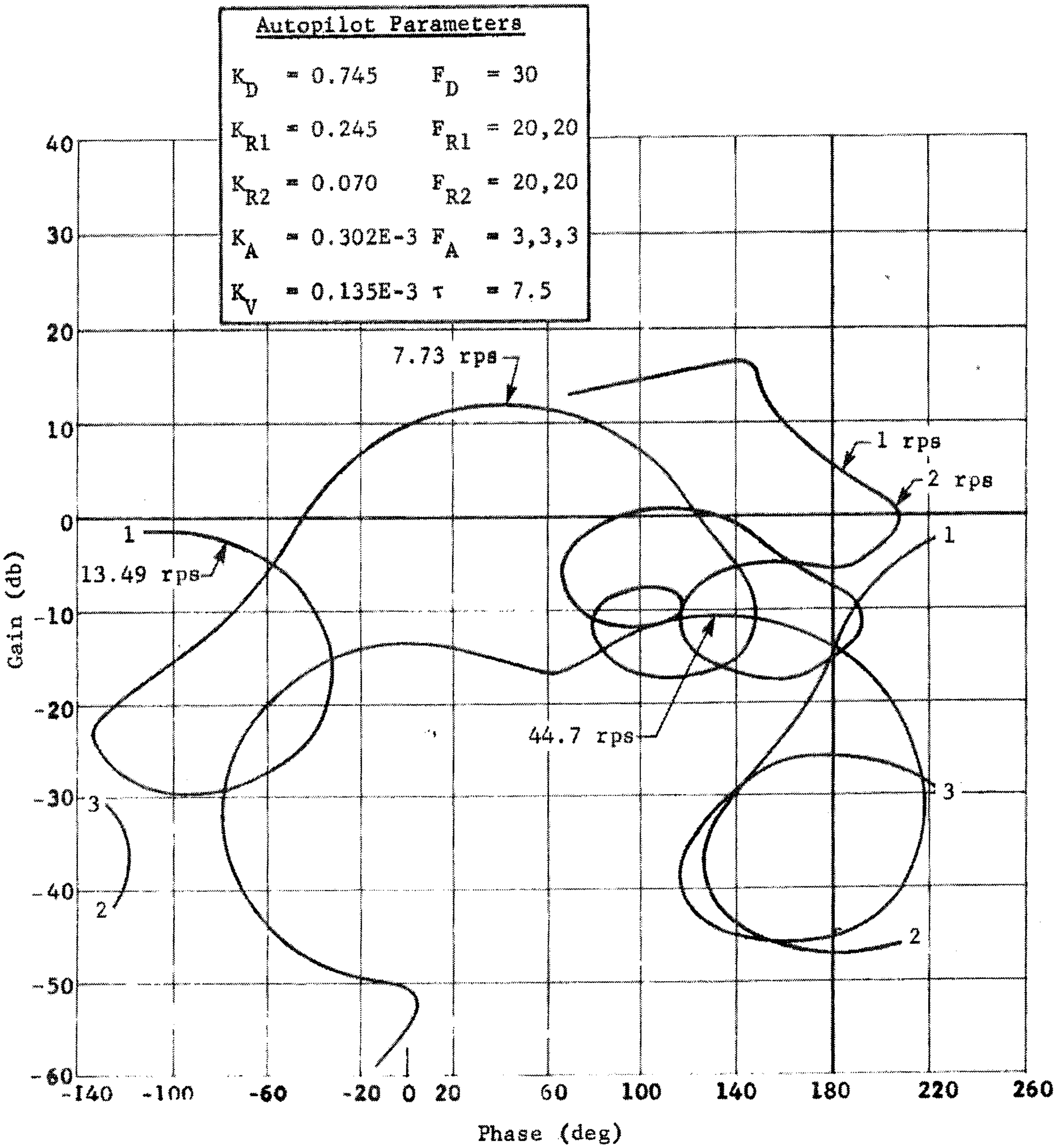


Fig. IV-5 Frequency Response Plot for Stage 0; Pitch, T = 60 sec; 7 seg-120; 54.5-ft, 28,000-lb Payload

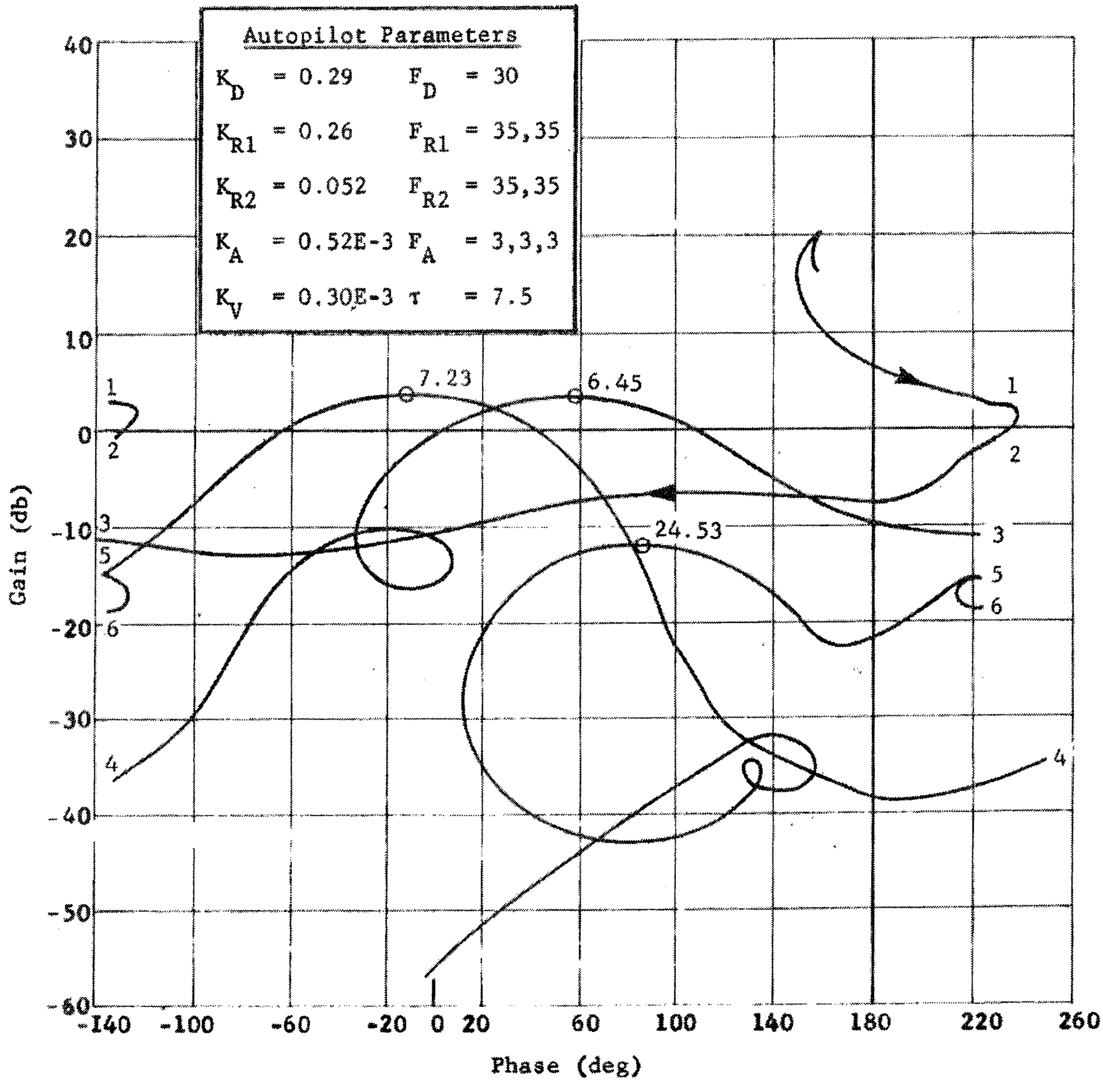


Fig. IV-6 Frequency Response Plot for Stage 0; Yaw, T = 60 sec; 7 seg-120; 54.5-ft, 28,000-lb Payload

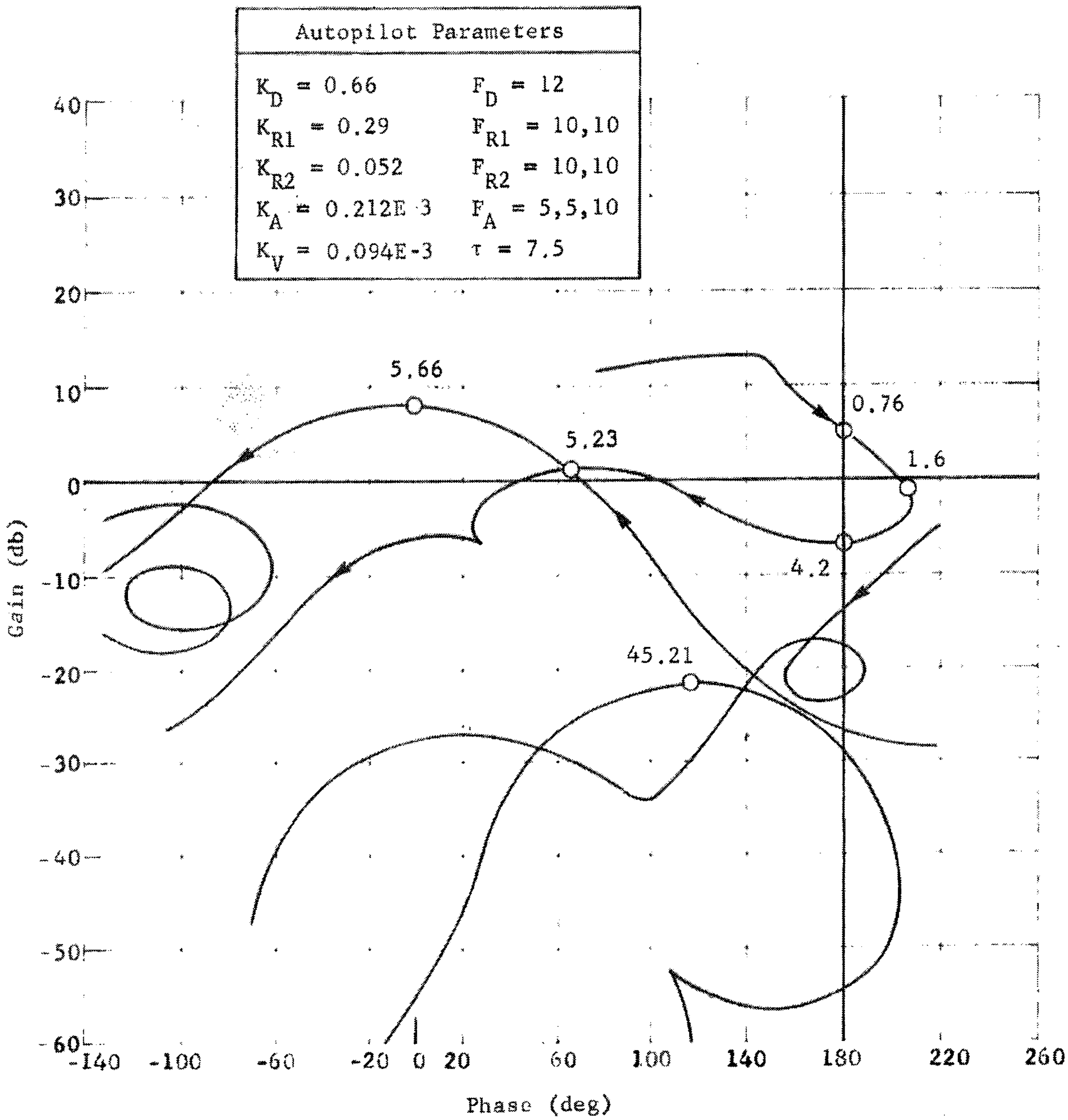


Fig. IV-7 Frequency Response Plot for Stage 0; Pitch, T = 60 sec; 7 seg-120; 75.0-ft, 28,000-lb Payload

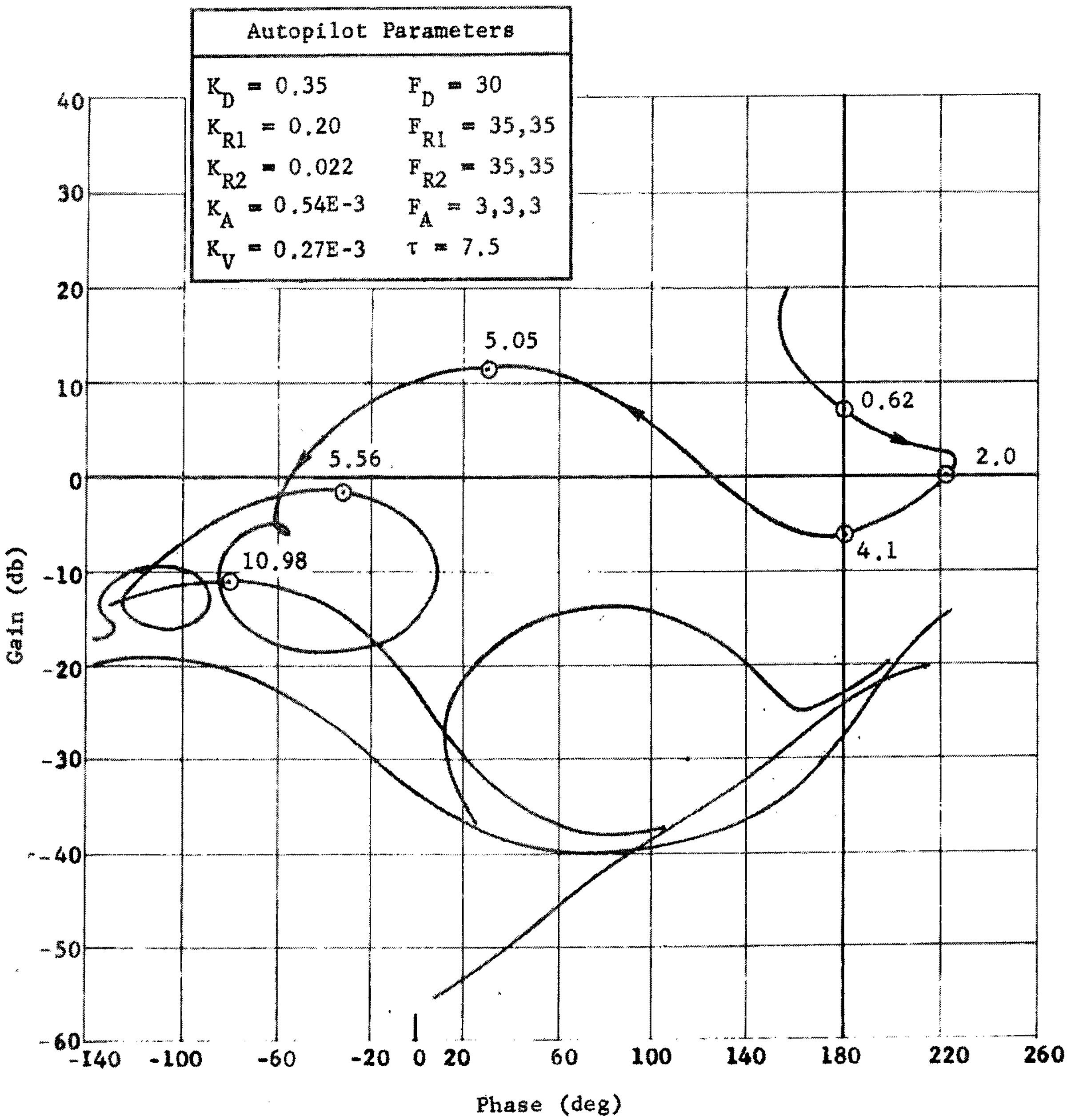


Fig. IV-8 Frequency Response Plot for Stage 0; Yaw, T = 60 sec; 7 seg-120; 75.0-ft, 28,000-lb Payload

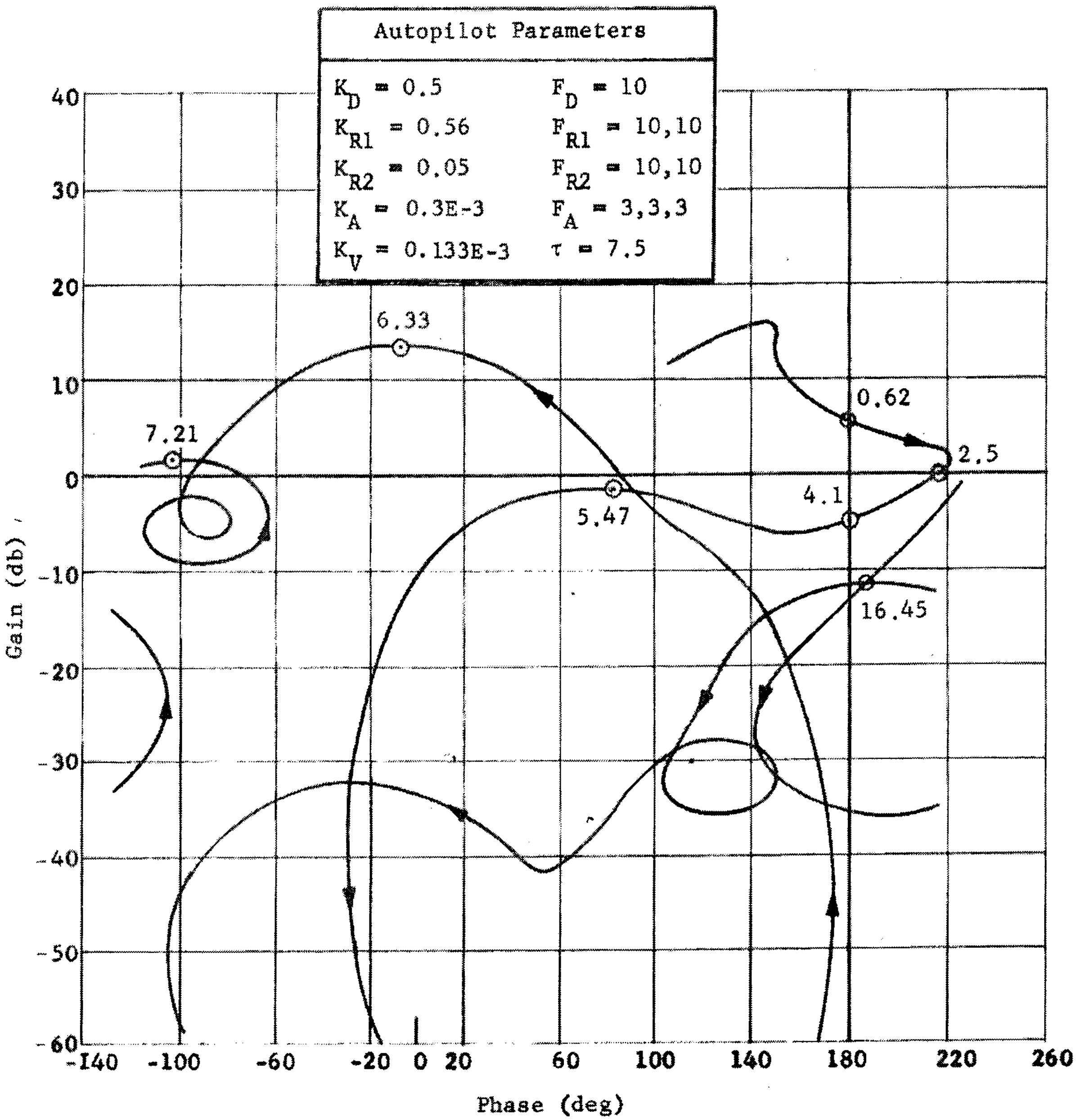
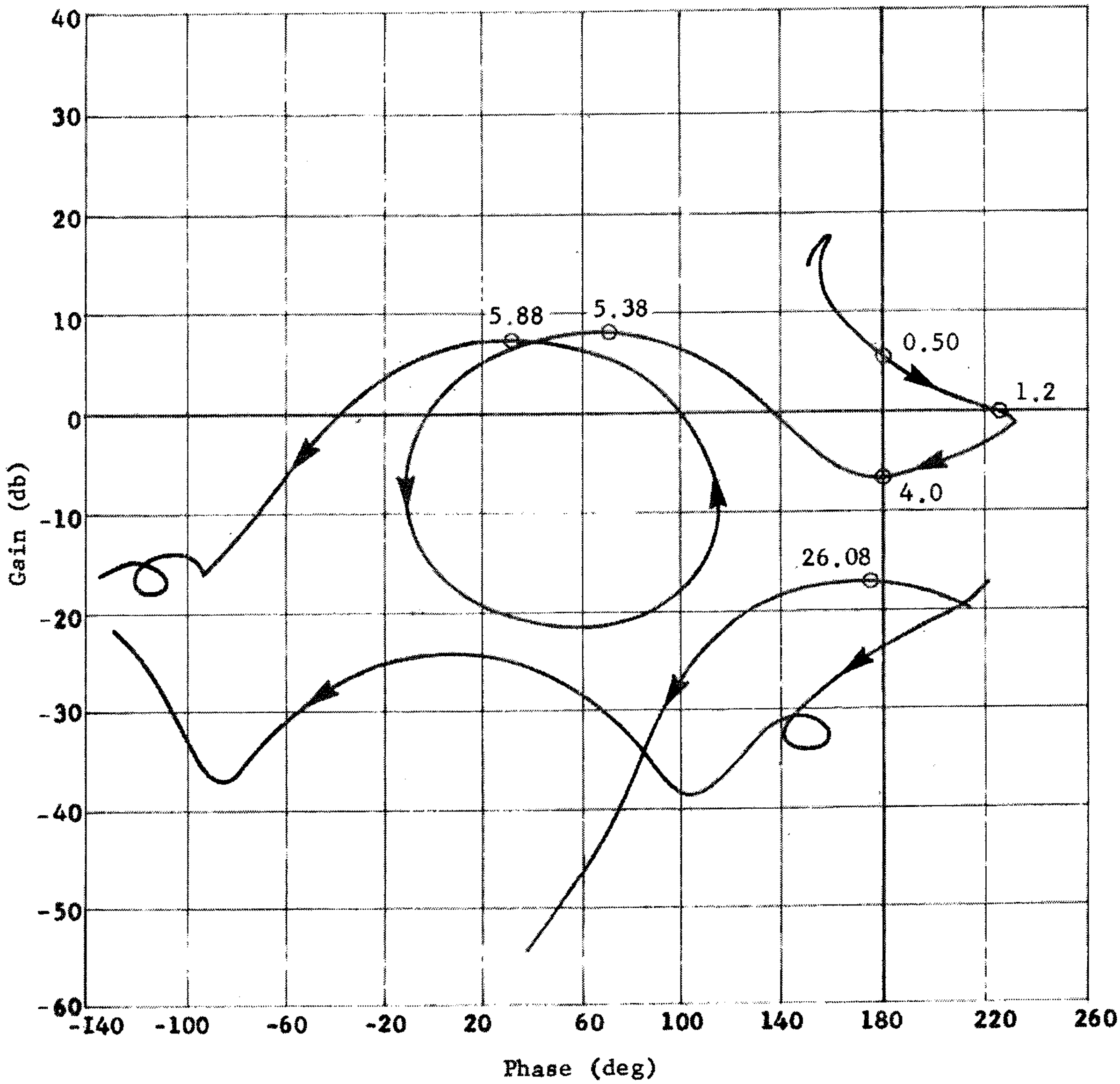


Fig. IV-9 Frequency Response Plot for Stage 0; Pitch, T = 60 sec; 2 CS-156; 50.0-ft, 35,000-lb Payload



Autopilot Parameters	
K_D = 0.25	F_D = 10
K_{R1} = 0.25	F_{R1} = 10,10
K_{R2} = 0.015	F_{R2} = 10,10
K_A = 0.45E-3	F_A = 3,3,3
K_V = 0.20E-3	τ = 7.5

Fig. IV-10 Frequency Response Plot for Stage 0; Yaw, T = 60 sec; 2 CS-156; 50.0-ft, 33,000-lb Payload

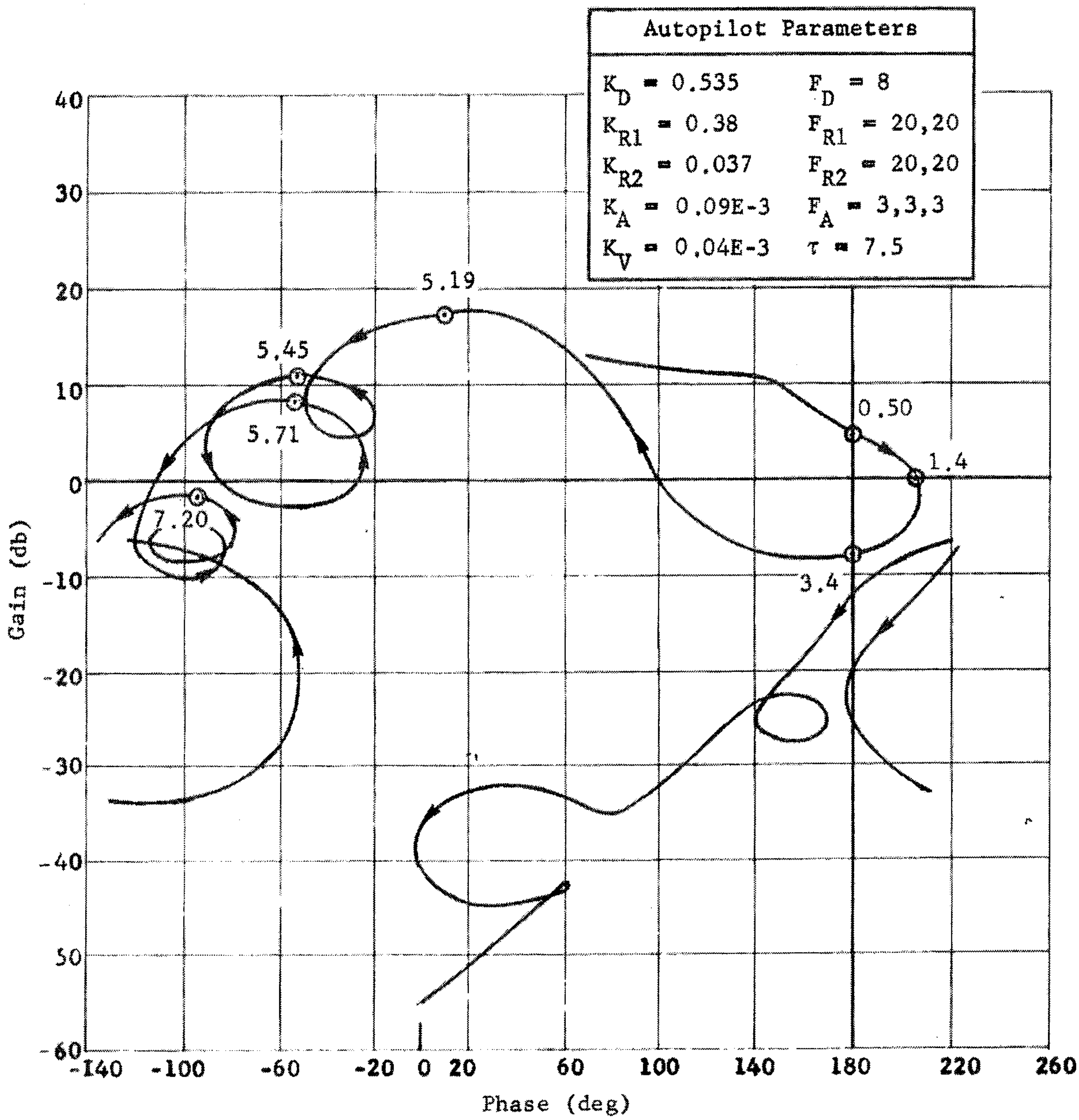


Fig. IV-11 Frequency Response Plot for Stage 0; Pitch, T = 60 sec; 2.CS-156; 65.0-ft, 33,000-lb Payload

Autopilot Parameters	
$K_D = 0.25$	$F_D = 10$
$K_{R1} = 0.25$	$F_{R1} = 10,10$
$K_{R2} = 0.015$	$F_{R2} = 10,10$
$K_A = 0.45E-3$	$F_A = 3,3,3$
$K_V = 0.20E-3$	$\tau = 7.5$

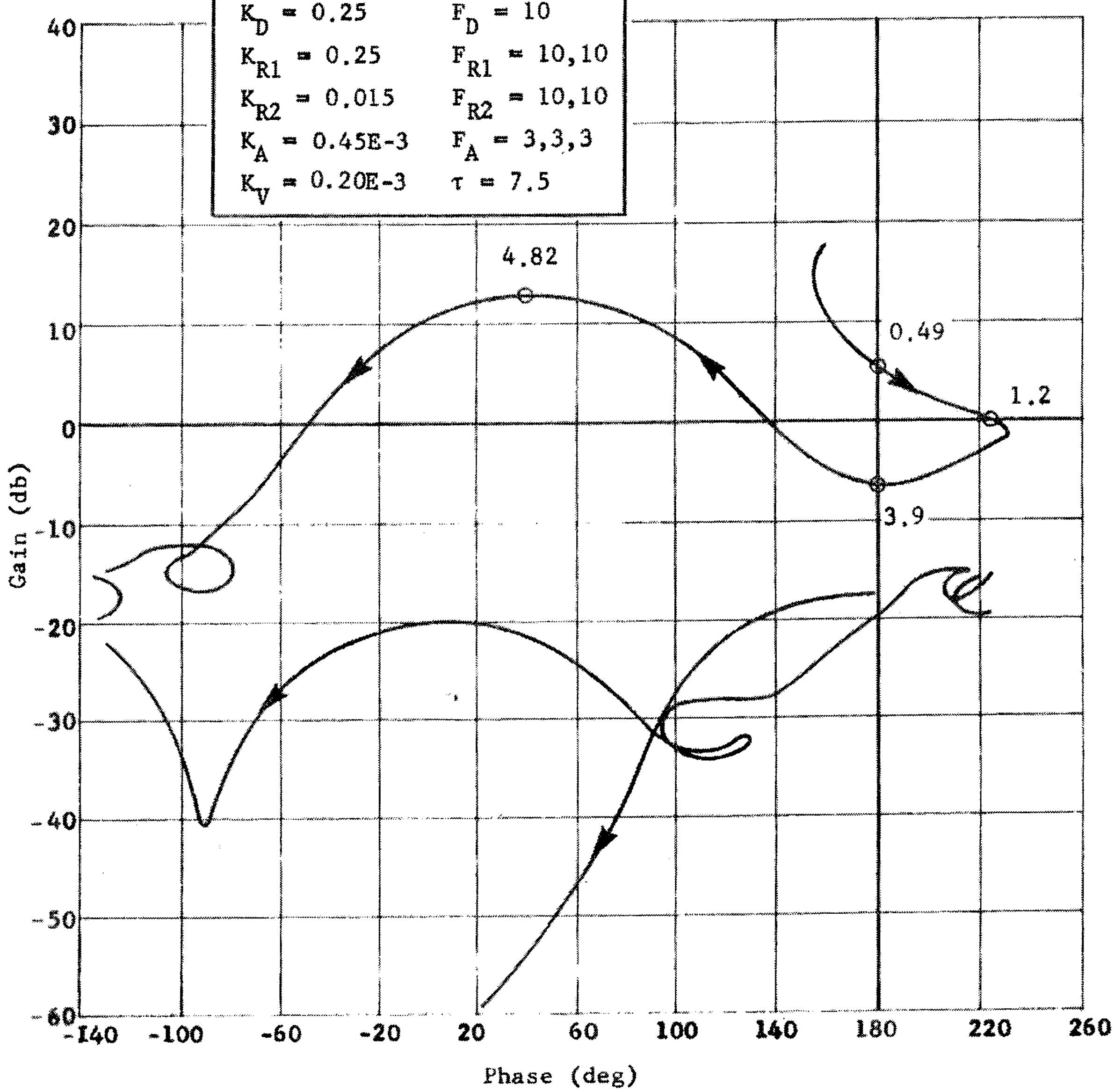


Fig. IV-12 Frequency Response Plot for Stage 0; Yaw, T = 60 sec; 2 CS-156; 65.0-ft, 33,000-lb Payload

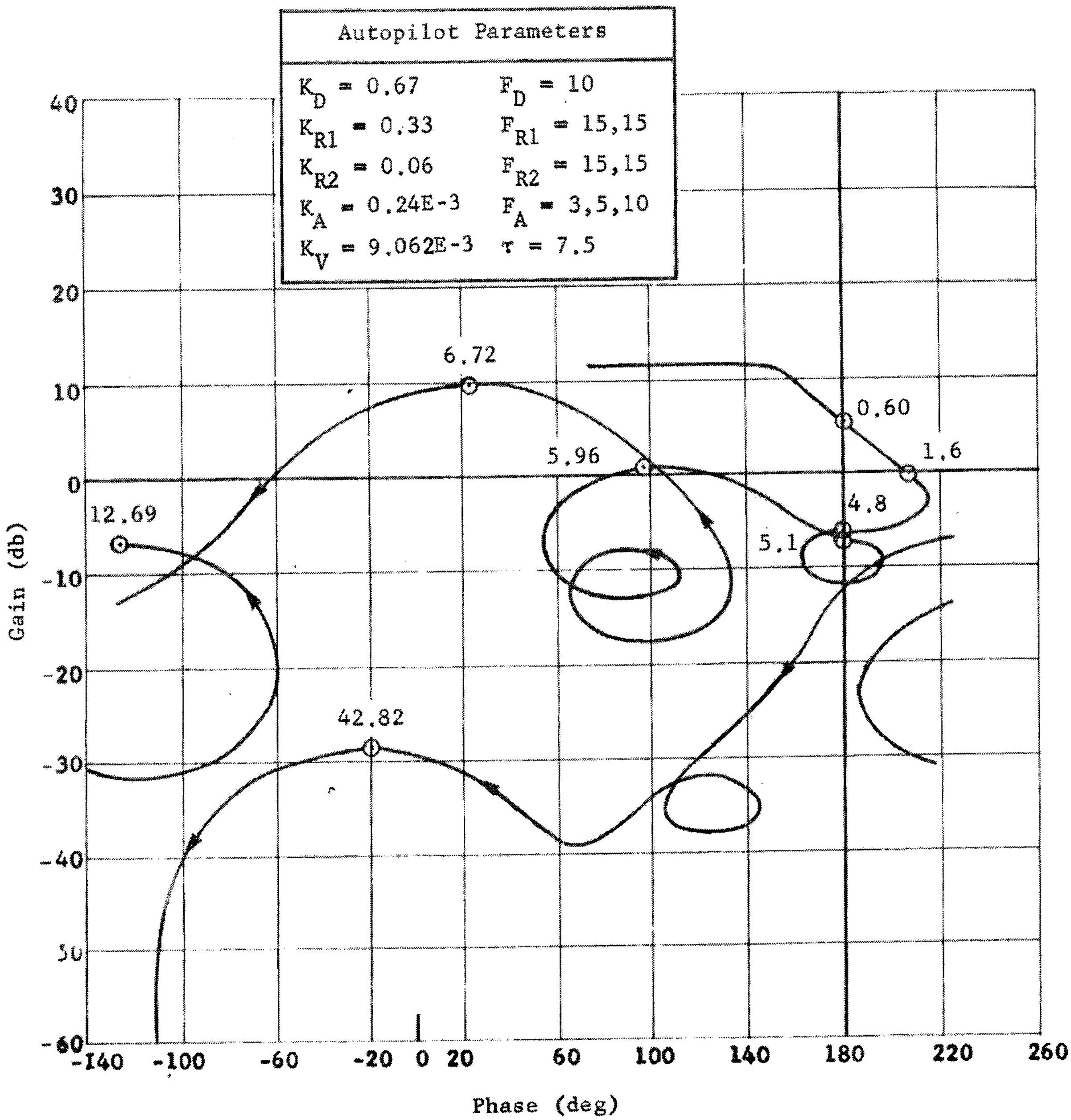


Fig. IV-13 Frequency Response Plot for Stage 0; Pitch, T = 60 sec;
3 CS-156; 50.0-ft, 42,000-lb Payload

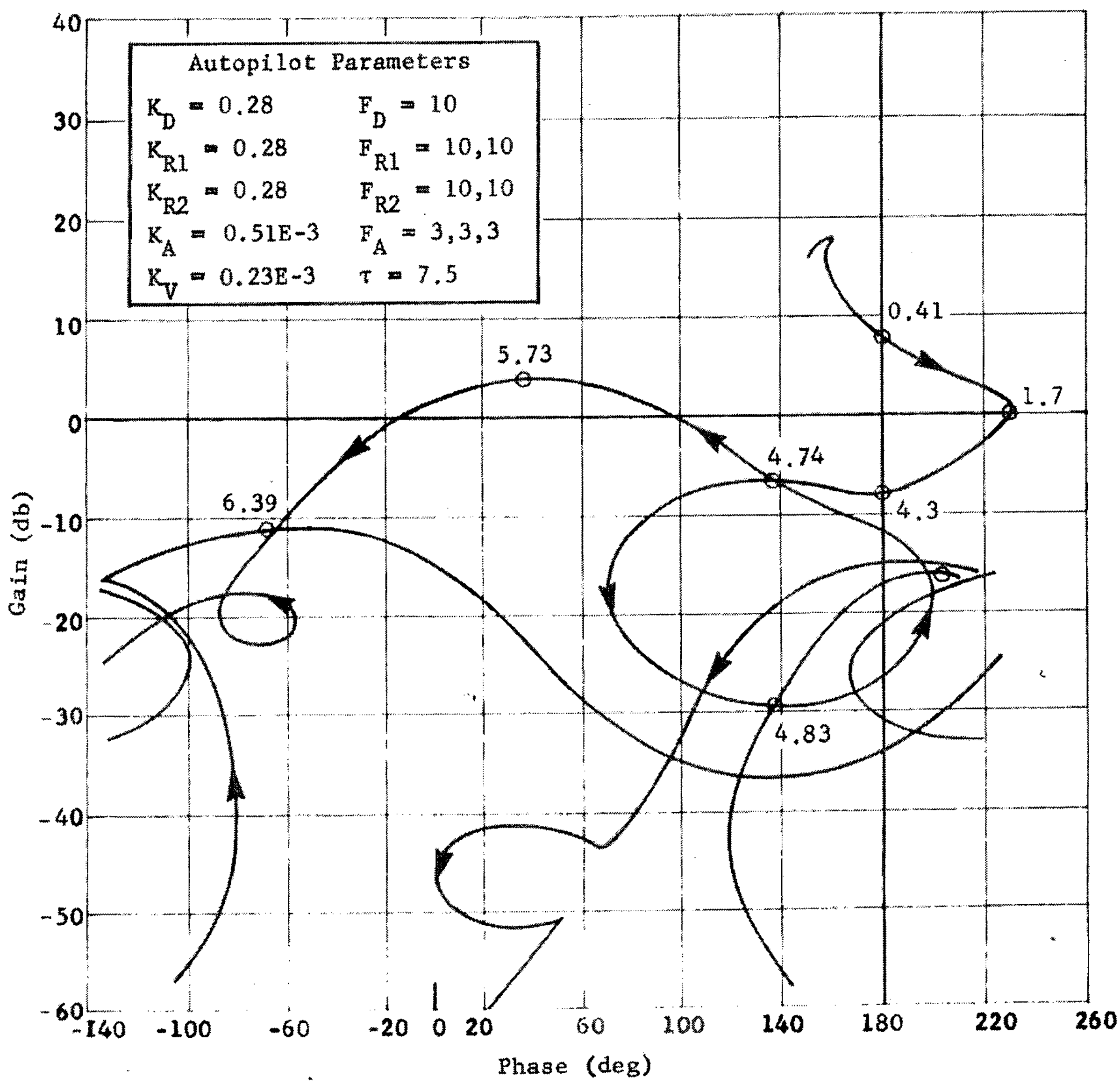


Fig. IV-14 Frequency Response Plot for Stage 0; Yaw, T = 60 sec; 3 CS-156; 50.0-ft, 42,000-lb Payload

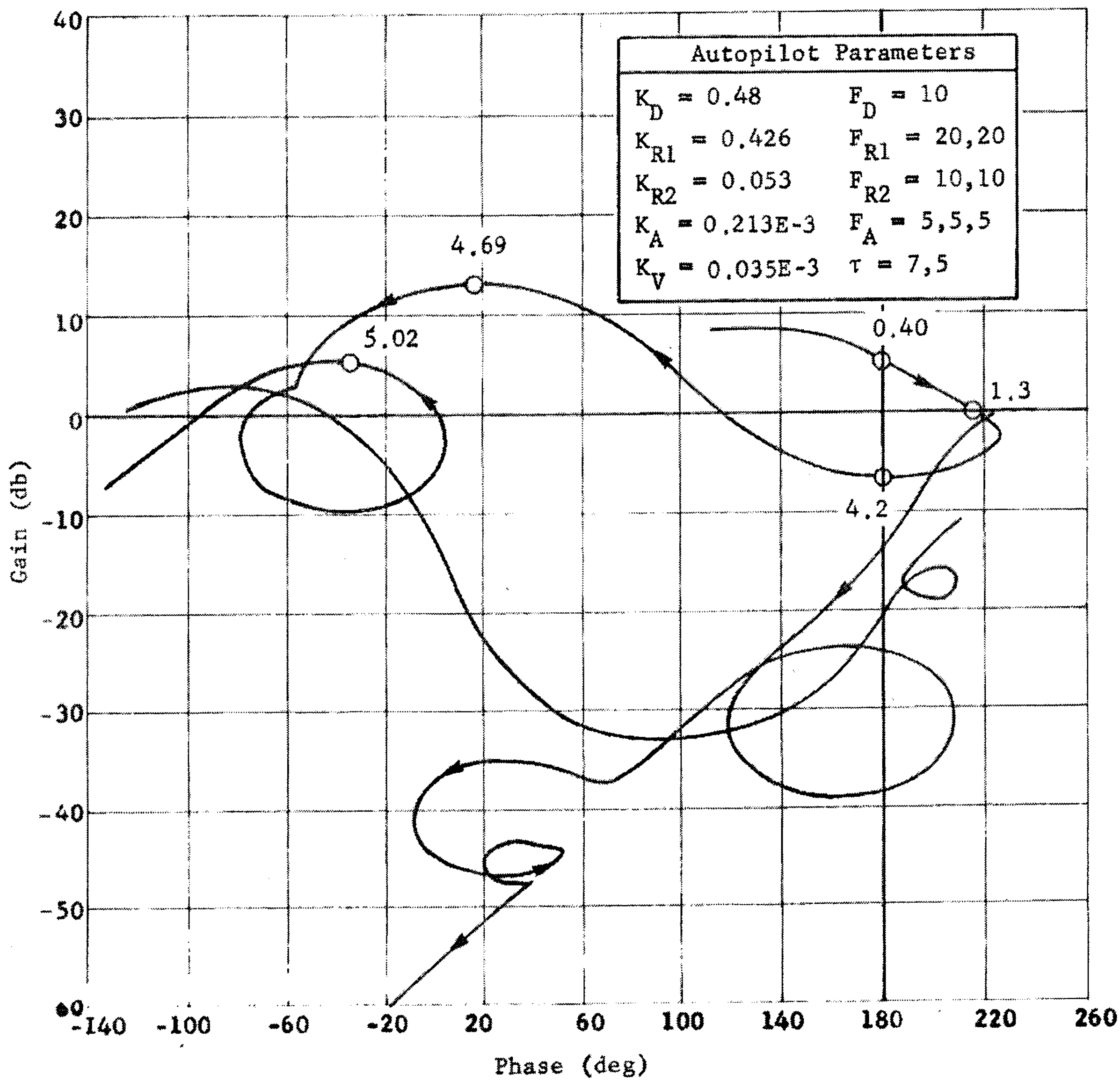


Fig. IV-15 Frequency Response Plot for Stage 0; Pitch, T = 60 sec; 3 CS-156; 72.0-ft, 42,000-lb Payload

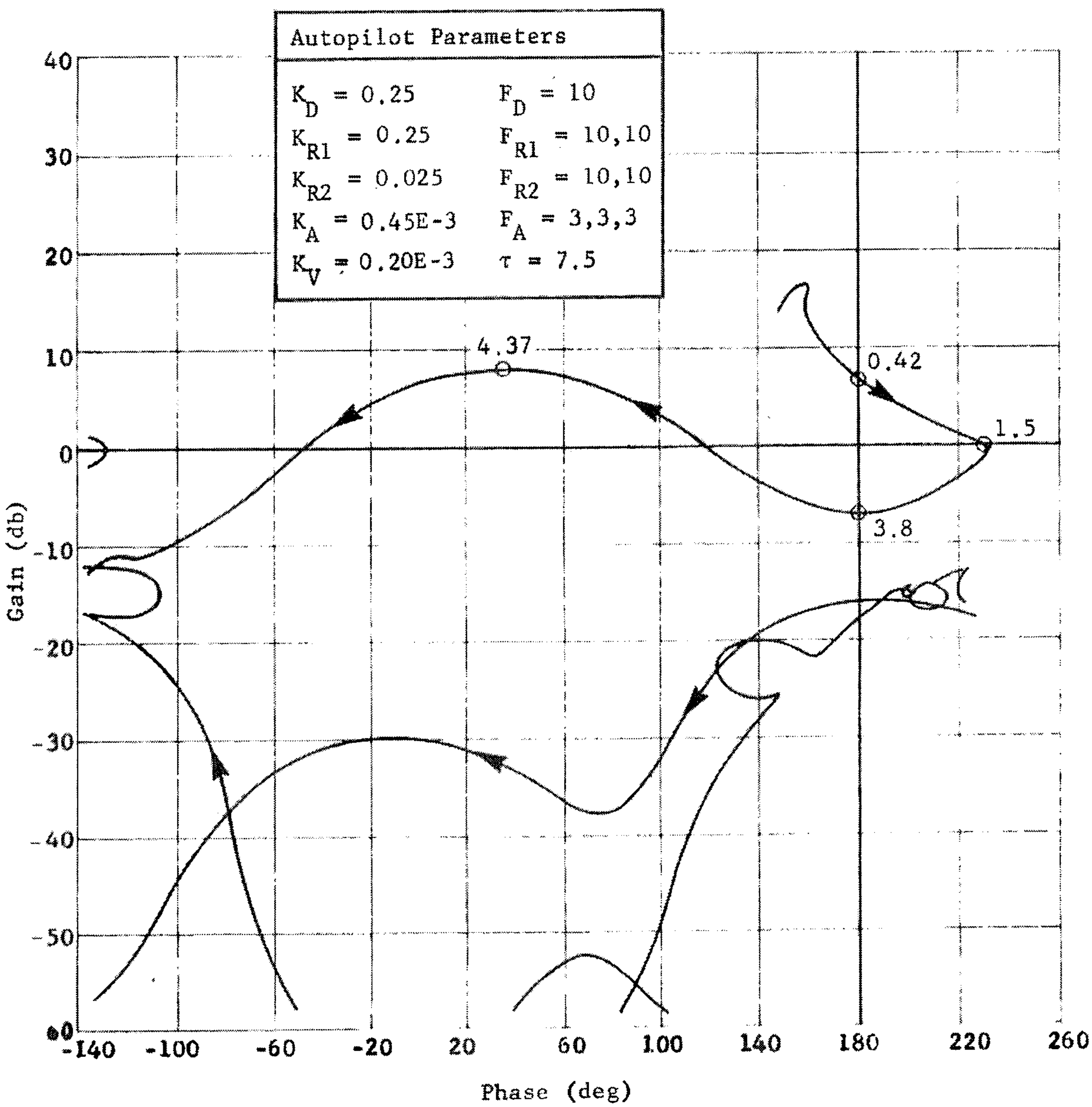


Fig. IV-16 Frequency Response Plot for Stage 0; Yaw, T = 60 sec; 3 CS-156; 72.0-ft, 42,000-lb Payload

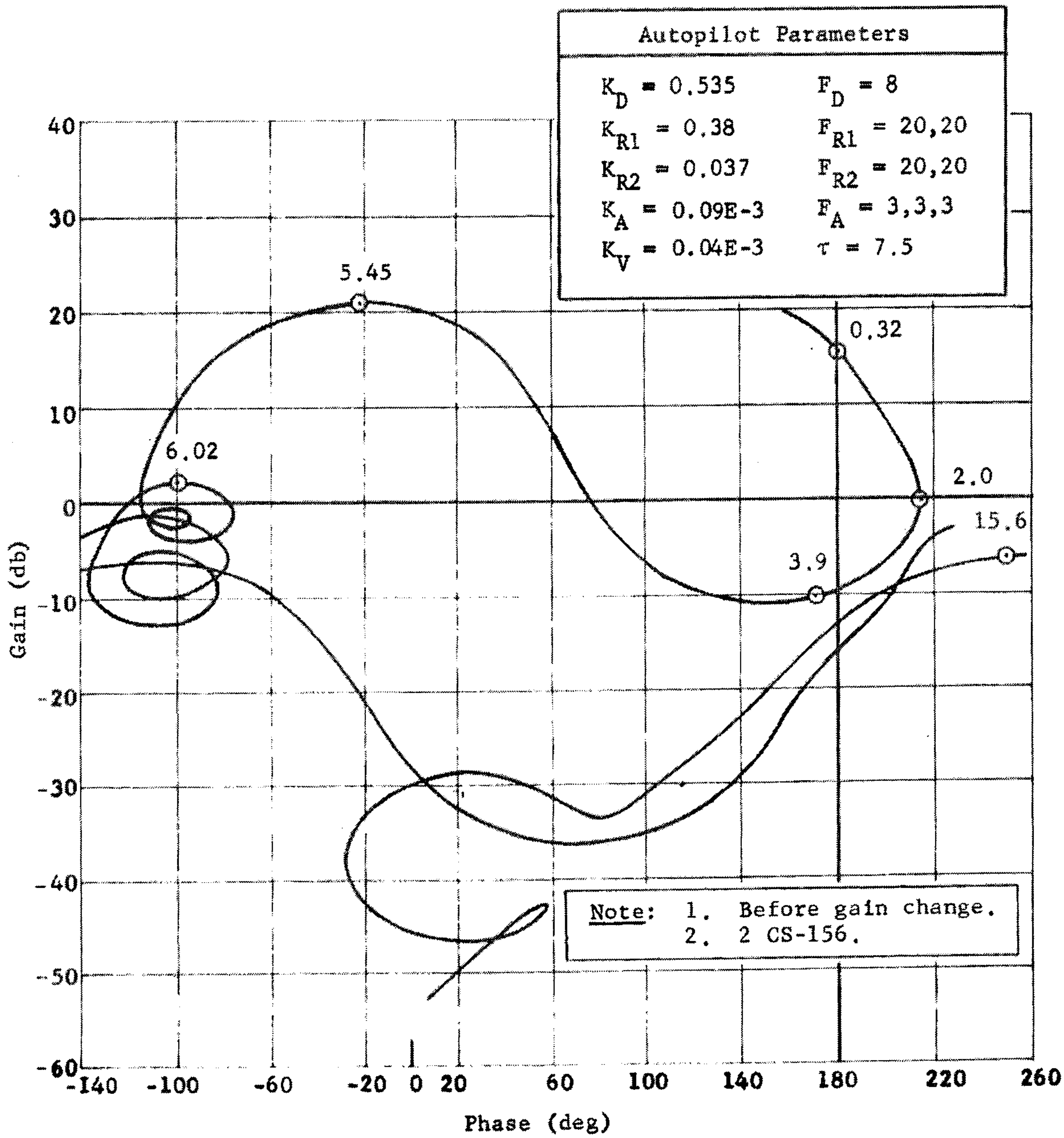


Fig. IV-17 Frequency Response Plot for Stage 0; Pitch, T = 80 sec; 65.0-ft, 33,000-lb Payload; Sensors at Stations 235 and 887

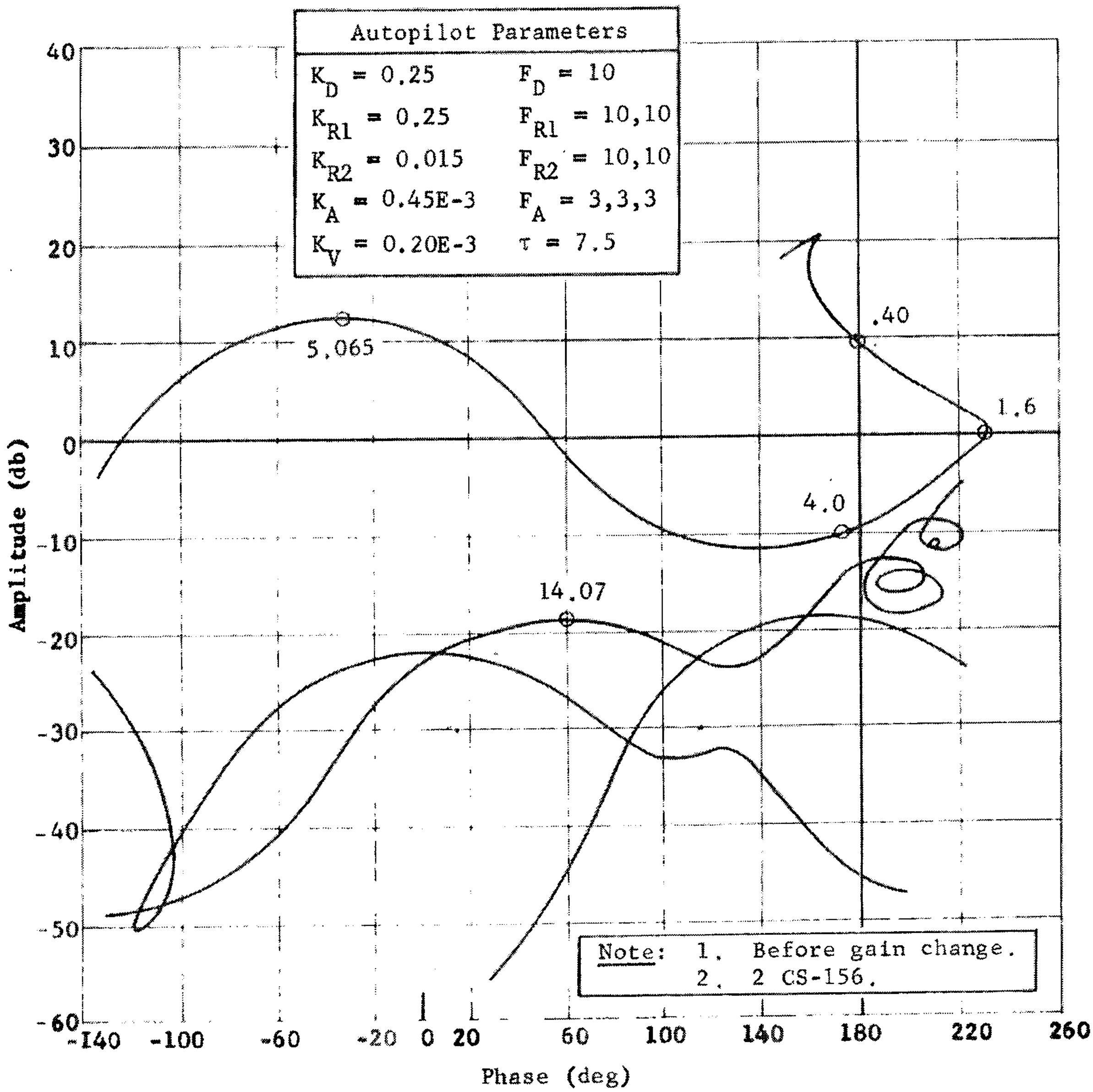
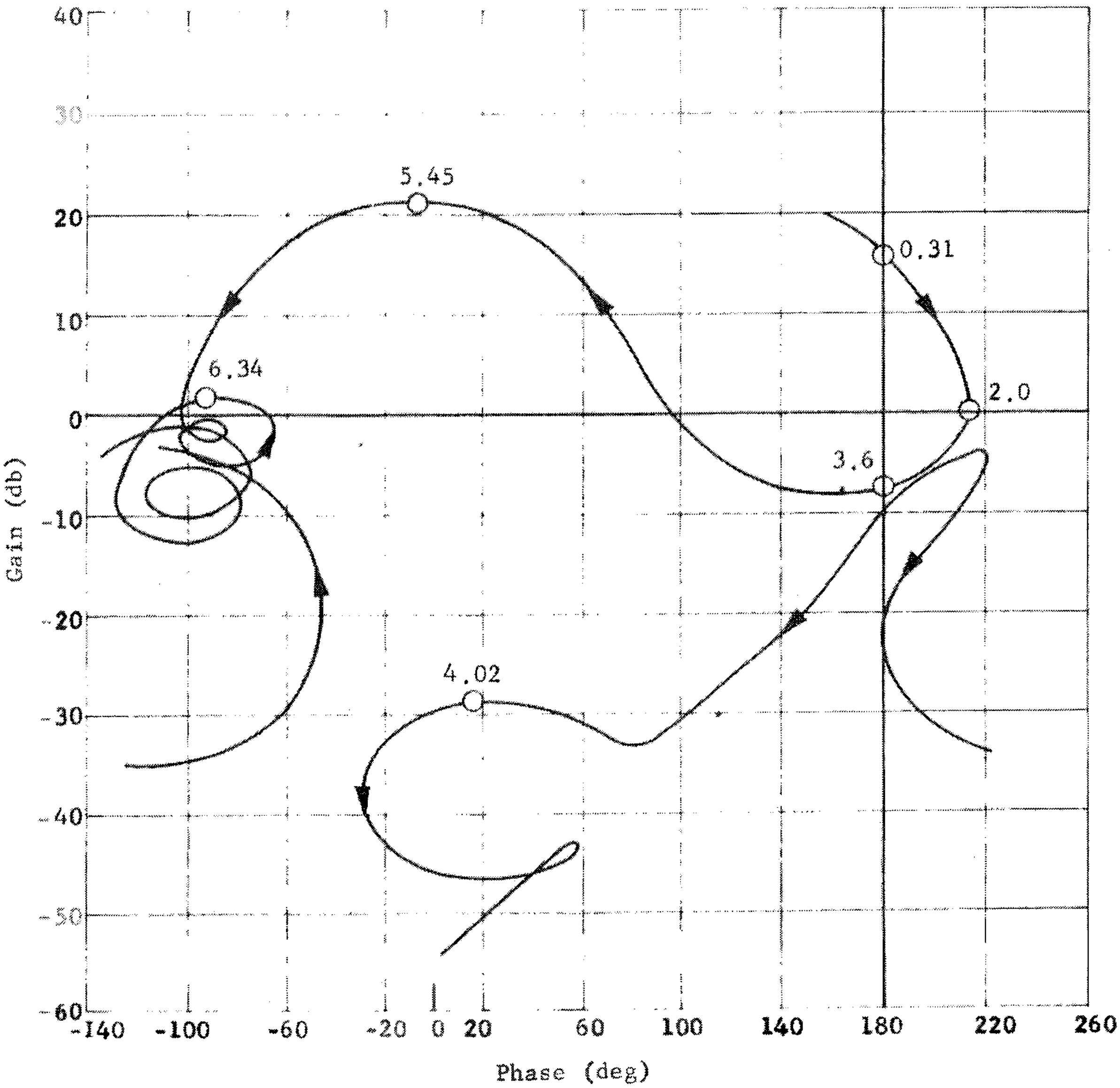


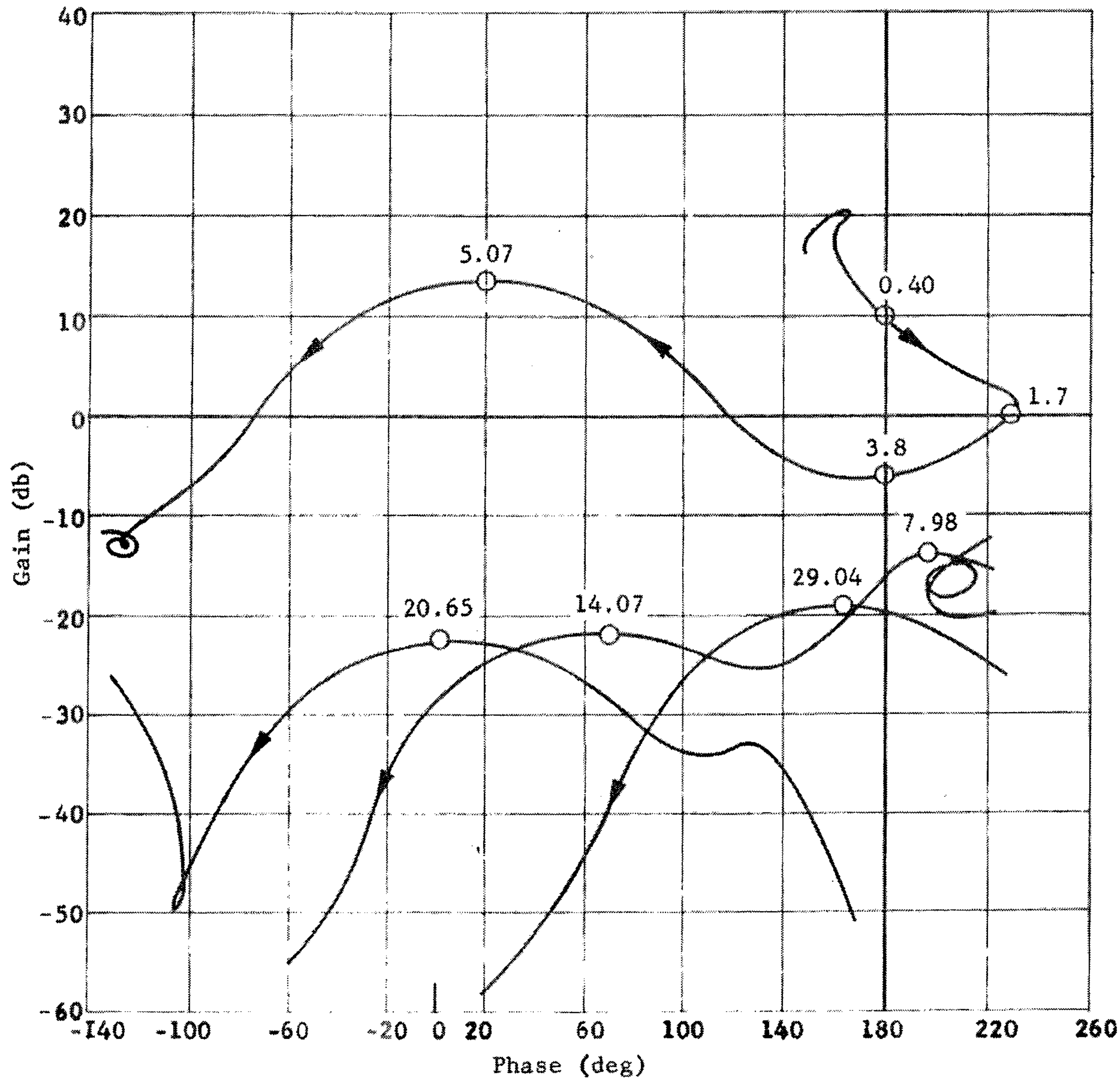
Fig. IV-18 Frequency Response Plot for Stage 0; Yaw, T = 80 sec; 65.0-ft, 33,000-lb Payload; Sensors at Stations 235 and 887



Autopilot Parameters	
K_D = 0.535	F_D = 8
K_{R1} = 0.38	F_{R1} = 20,20
K_{R2} = 0.037	F_{R2} = 20,20
K_A = 0.09E-3	F_A = 3,3,3
K_V = 0.04E-3	τ = 7.5

Note: 1. Before gain change.
2. 2 CS-156.

Fig. IV-19 Frequency Response Plot for Stage 0; Pitch, T = 80 sec; 65.0-ft, 33,000-lb Payload; Sensors at Stations 112 and 887



Autopilot Parameters	
K_D = 0.25	F_D = 10
K_{R1} = 0.25	F_{R1} = 10,10
K_{R2} = 0.015	F_{R2} = 10,10
K_A = 0.45E-3	F_A = 3,3,3
K_V = 0.2E-3	τ = 7.5

<u>Note</u>: 1. Before gain change.
2. 2 CS-156.

Fig. IV-20 Frequency Response Plot for Stage 0; Yaw, T = 80 sec; 65.0-ft, 33,000-lb Payload; Sensors at Stations 112 and 887

Stage I - The criteria used for the FCS analysis of Stage I is presented in Subsection 4 of this section (control system requirements). As in Stage 0 the stability problems are generally the same for all payloads. Therefore the following discussion will be for all payload configurations. As in Stage 0, the structural bending modes have rather low frequencies. These low frequencies present a problem of coupling of the first bending mode with the fluid slosh modes. When this occurs the structural model gives rise to two or more modes that are similar to a first mode. With multiple first modes the stability problem is more complex. This problem occurs to some degree in Stage 0, but generally the first mode is below the fluid slosh modes.

The payloads on the configurations studied for Stage I were: 75 ft long, 28,000 lb, and 72 ft long, 42,000 lb. The autopilots designed for these configurations are presented in Chap. IV.A.3.

The proposed Stage I engines for two of the MOL vehicle configurations differ from the corresponding Titan III engines by the nozzle expansion ratio (15:1). This heavier nozzle results in a considerable increase in the engine moment of inertia about the gimbal (approximately a factor of three).

As a result of the proposed engines it was desirable to examine the structural bending-actuator coupling as related to stability analysis. The configurations selected for the survey were representative of the 28,000-, 33,000-, and 42,000-lb payloads, respectively. Each configuration examined had the greatest length vehicle for the payload involved, and thus, the most severe structural bending problems. The criteria used for the study were that if any structural damping, after coupling with the actuator dynamics, dropped to 0.004 or less, then actuator or engine redesign would be needed. Using a 9 cps engine, all coupled structural damping was above the damping ratio limit of 0.004.

A survey was also conducted to determine the minimum engine frequency that could be tolerated with the above criteria. Results of that survey showed minimum acceptable engine frequency of 8.0 cps. Figures IV-21 thru IV-24 present a typical set of stability plots for Stage I.

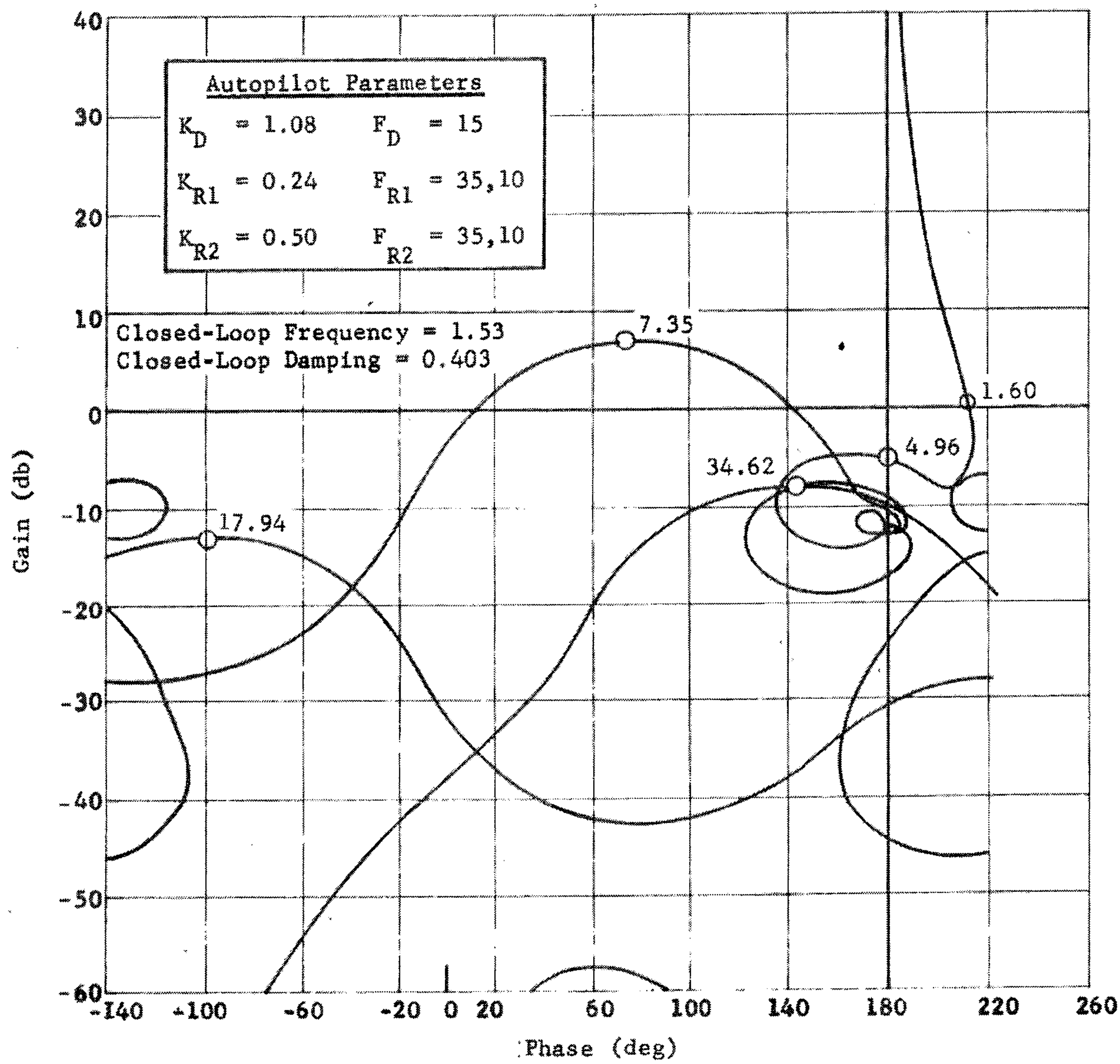
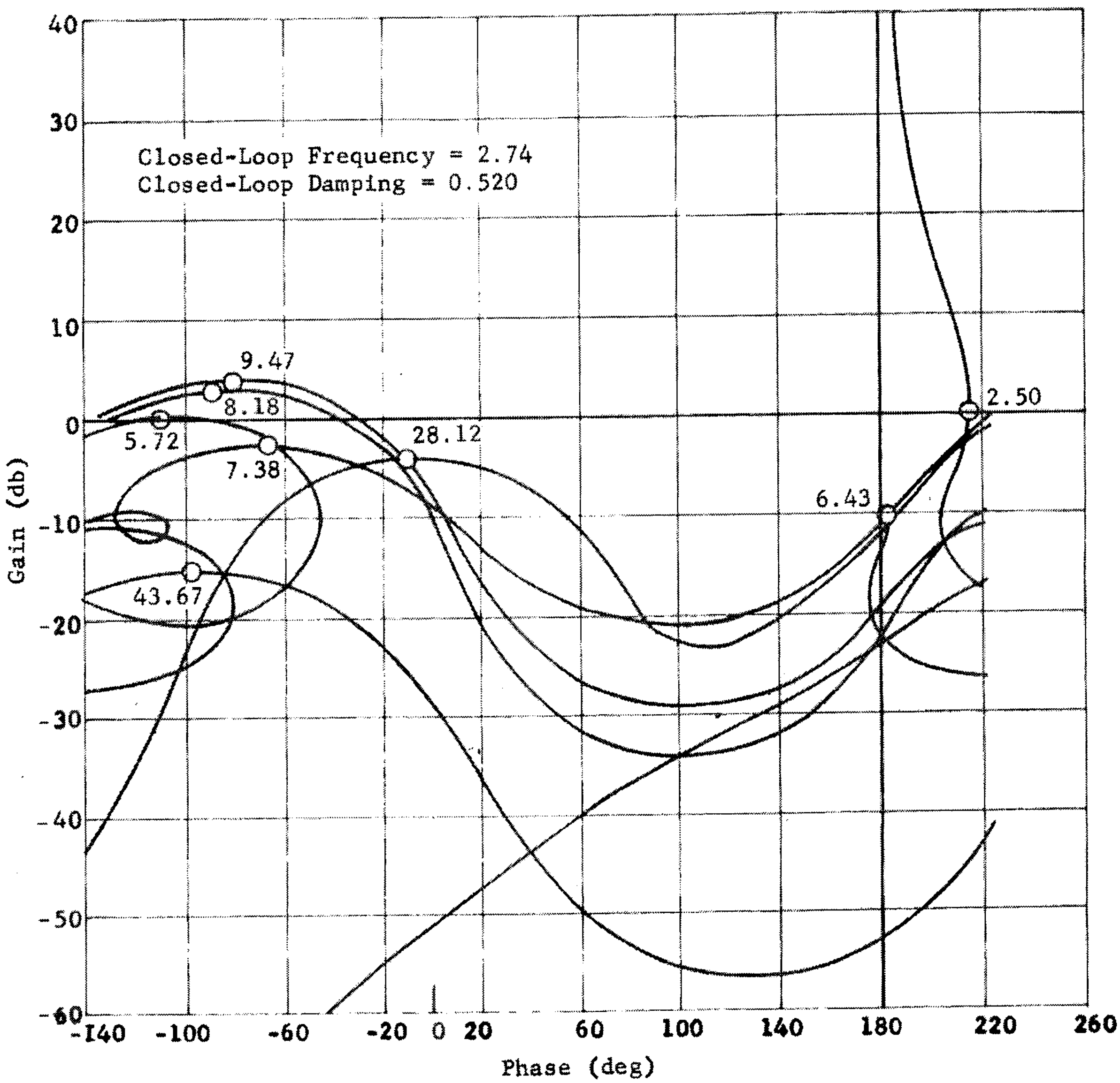


Fig. IV-21 Frequency Response Plot for Stage I; Pitch/Yaw, T = Start; 28,000-lb, 75-ft Payload



Autopilot Parameters	
K_D = 1.08	F_D = 15
K_{R1} = 0.24	F_{R1} = 35,10
K_{R2} = 0.50	F_{R2} = 35,10

Fig. IV-22 Frequency Response Plot for Stage I; Pitch/Yaw, T = Midflight (Start + 119 sec), Before Gain Change; 28,000-lb, 75-ft Payload

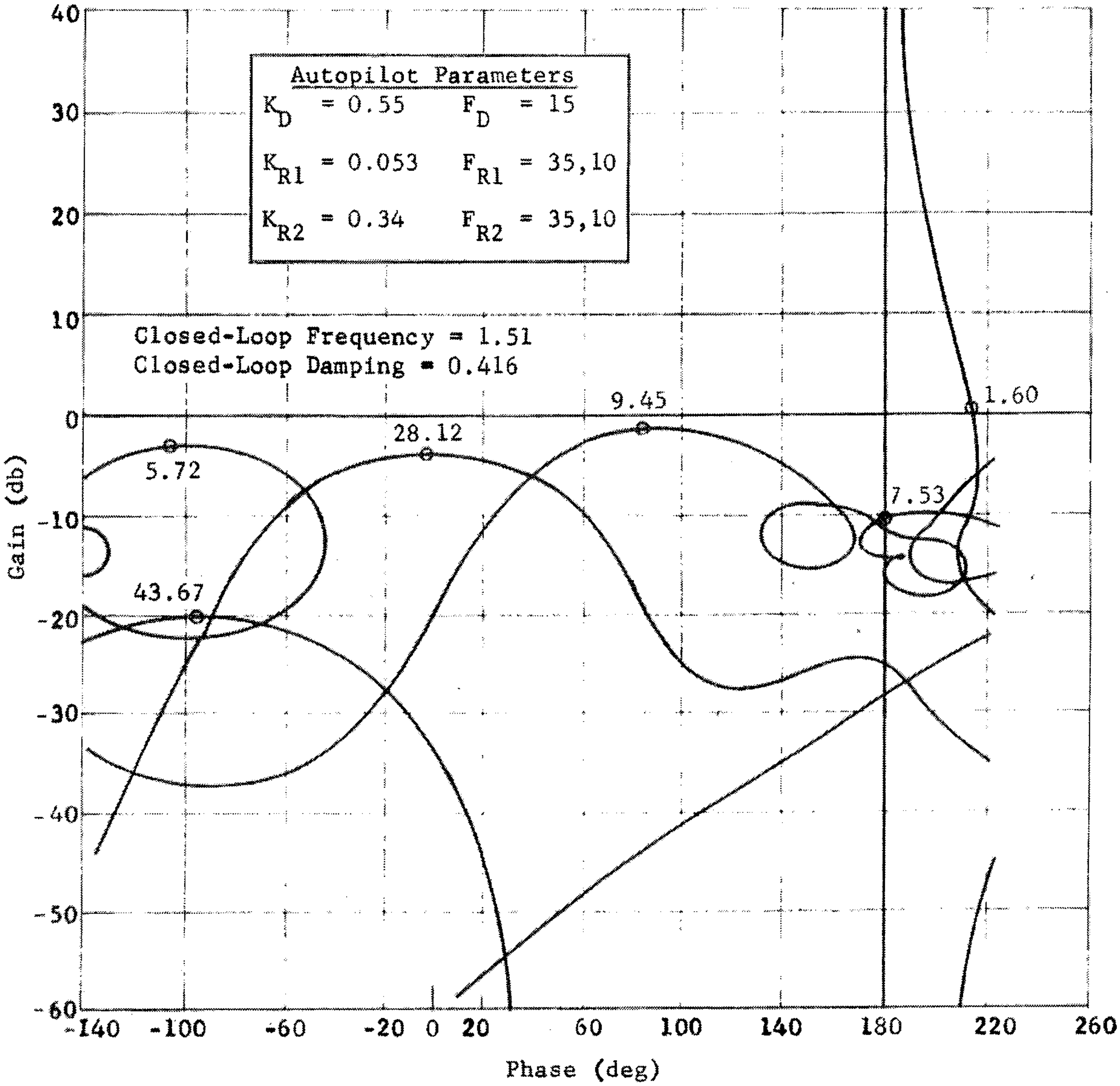


Fig. IV-23 Frequency Response Plot for Stage I; Pitch/Yaw, T = Midflight (Start + 119 sec) After Gain Change; 28,000-lb, 75-ft Payload [redacted]SD-CR-65-206

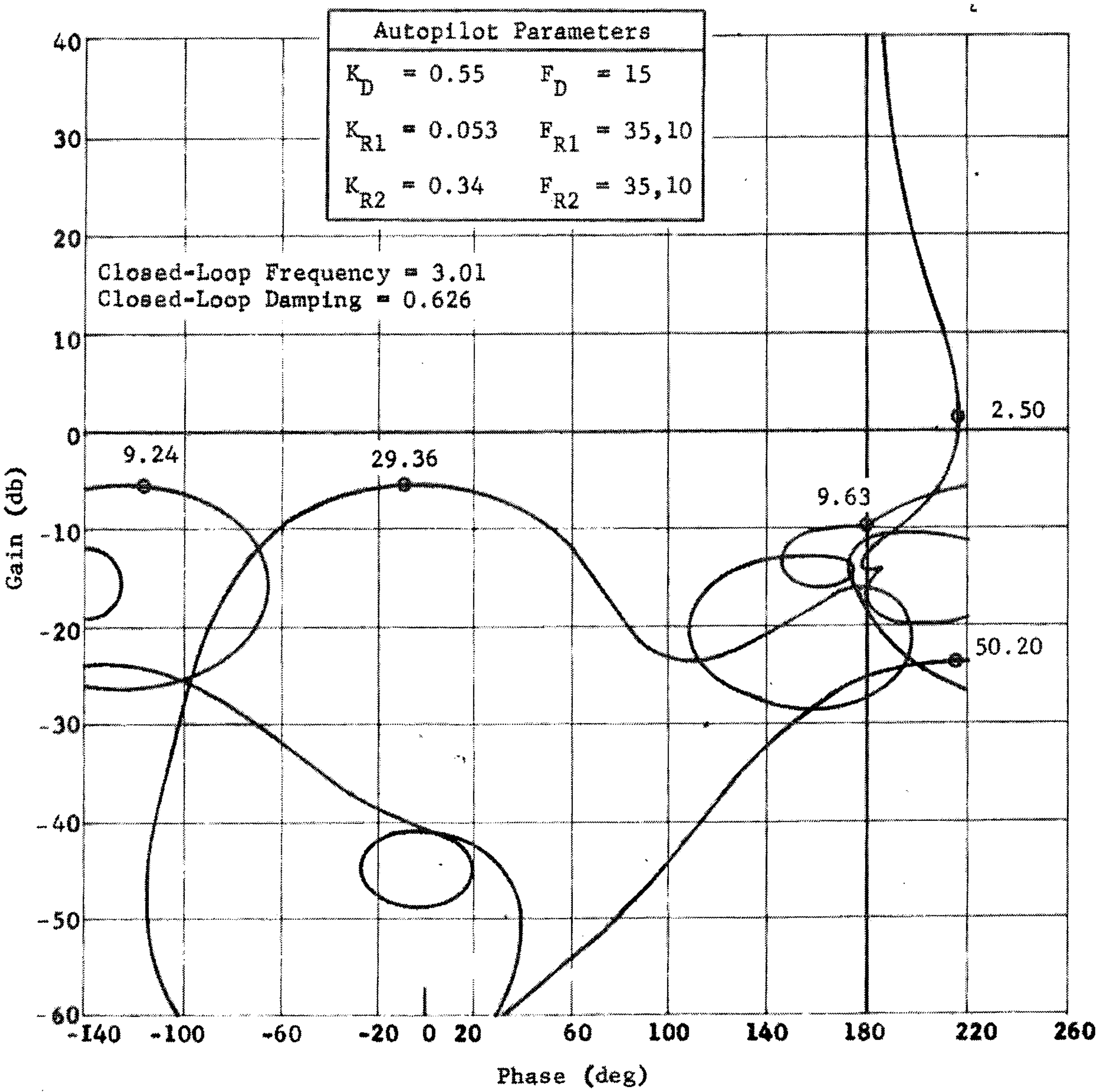


Fig. IV-24 Frequency Response Plot for Stage I; Pitch/Yaw,
T = Burnout; 28,000-lb, 75-ft Payload

[redacted]

Stage II - The basic stability investigation of Stage II uncovered only one significant problem. With the Titan IIIC-type autopilot, the first mode phase was about 180 deg. To eliminate this problem, two underdamped quadratic filters in the rate loop were used. This resulted in a more desirable phasing for the first mode. To determine if this filter was sensitive to tolerances, a modified tolerance study was performed. The results of that study showed there would be no significant problems with respect to tolerance associated with this filter technique.

A preliminary study of coupling between the first structural bending mode and the Stage II engine was accomplished through digital simulation. Results of that study indicate that excessive coupling will occur if the present Titan IIIC Stage II actuator is used. An effective (and necessary) means of decoupling is now conceived for the Titan IIIB Stage II actuator. The Titan IIIB actuator differs from the Titan IIIC actuator in the follow-manner:

1) The effective actuator lever arm is increased from 10 to 14 in.;

2) The actuator pressure feedback gain is reduced from 0.00182 to 0.00075 $in.^3/lb\text{-}sec$.

A preliminary tolerance analysis indicated that, while the Titan IIIC actuator is not acceptable for any MOL configuration at any reasonable natural engine frequency, the Titan IIIB actuator is acceptable for all MOL configurations studied, if the natural engine frequency is not lower than 7.5 cps.

This actuator analysis and conclusions are only valid for payloads of equal or smaller length and weight than the ones studied. If a payload configuration of greater length and weight than the one studied is selected for MOL, it is possible that even the Titan IIIB actuator will not be adequate to solve the Stage II structural bending coupling problem. Typical Stage II stability plots are shown in Fig. IV-25 thru 27.

3. Autopilot Parameters

The slight differences between the various configurations is further demonstrated by the similarity of the autopilot configurations required. The configurations defined in this study are shown in Tables IV-1 (Stage 0), IV-2 (Stage I), and IV-3 (Stage II).

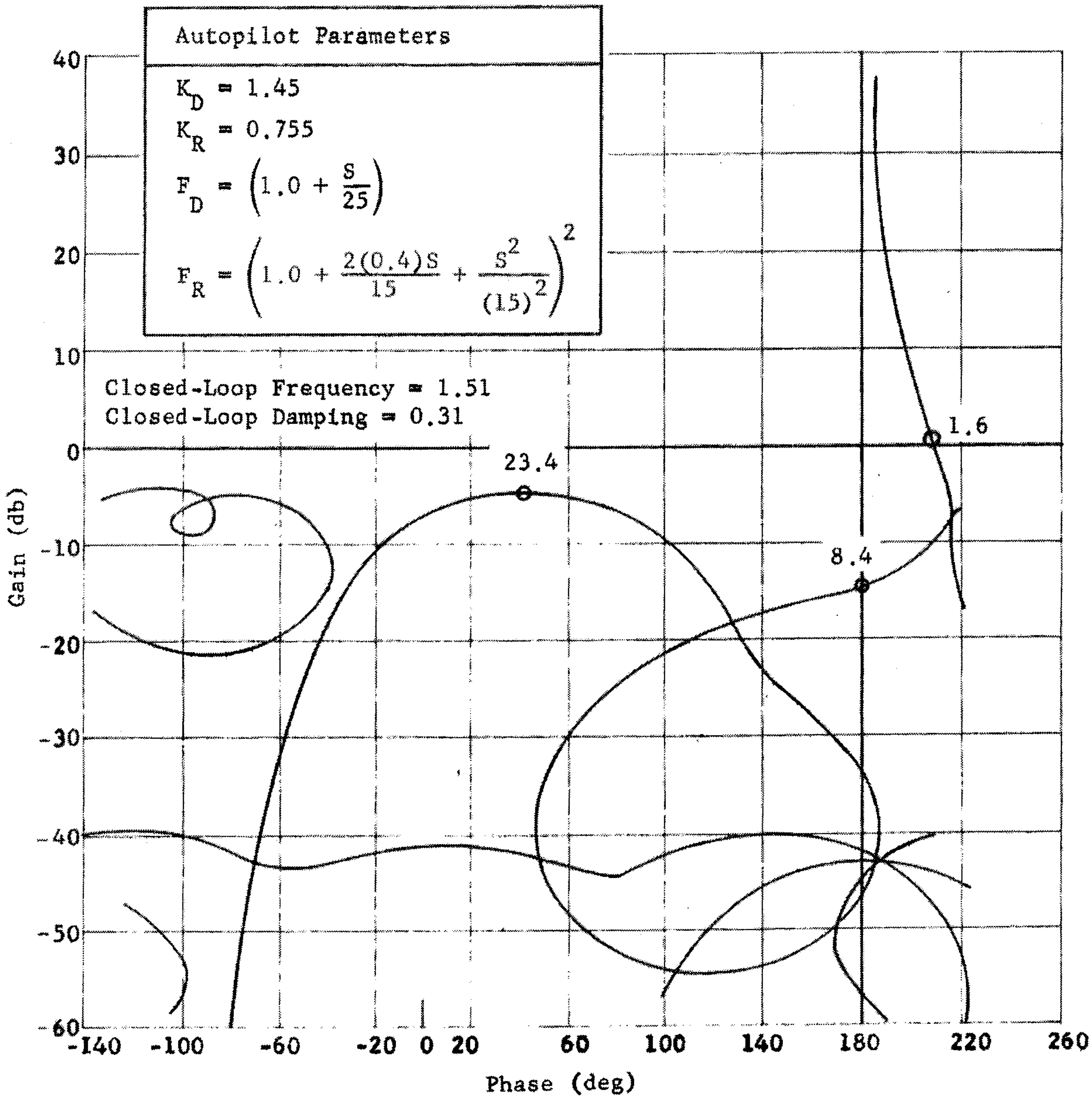


Fig. IV-25 Frequency Response Plot for Stage II; Pitch/Yaw, T = Start; 7 seg-120; 65.0-ft, 28,000-lb Payload

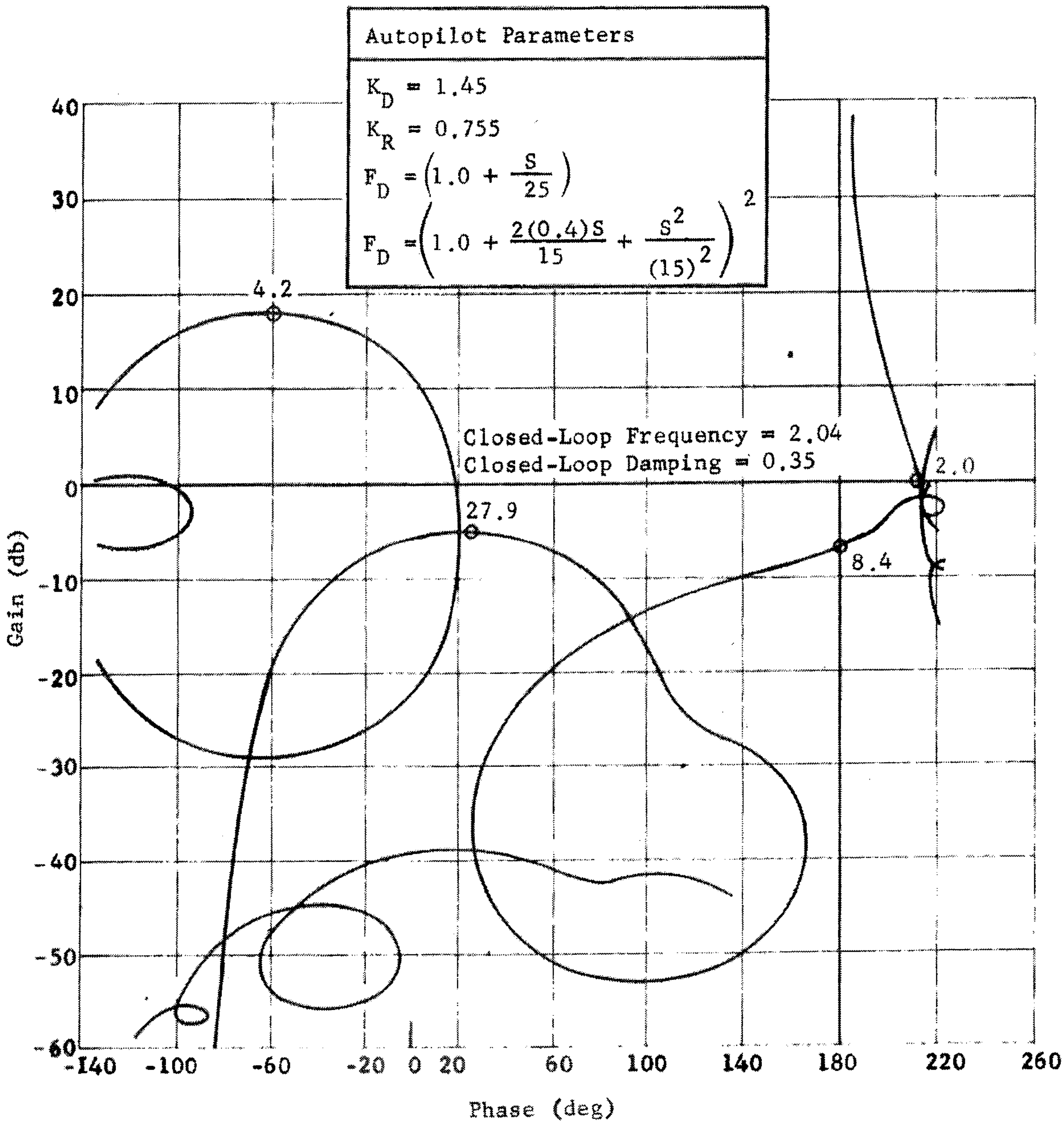


Fig. IV-26 Frequency Response Plot for Stage II; Pitch/Yaw, T = Midflight; 7 seg-120; 65.0-ft, 28,000-lb Payload

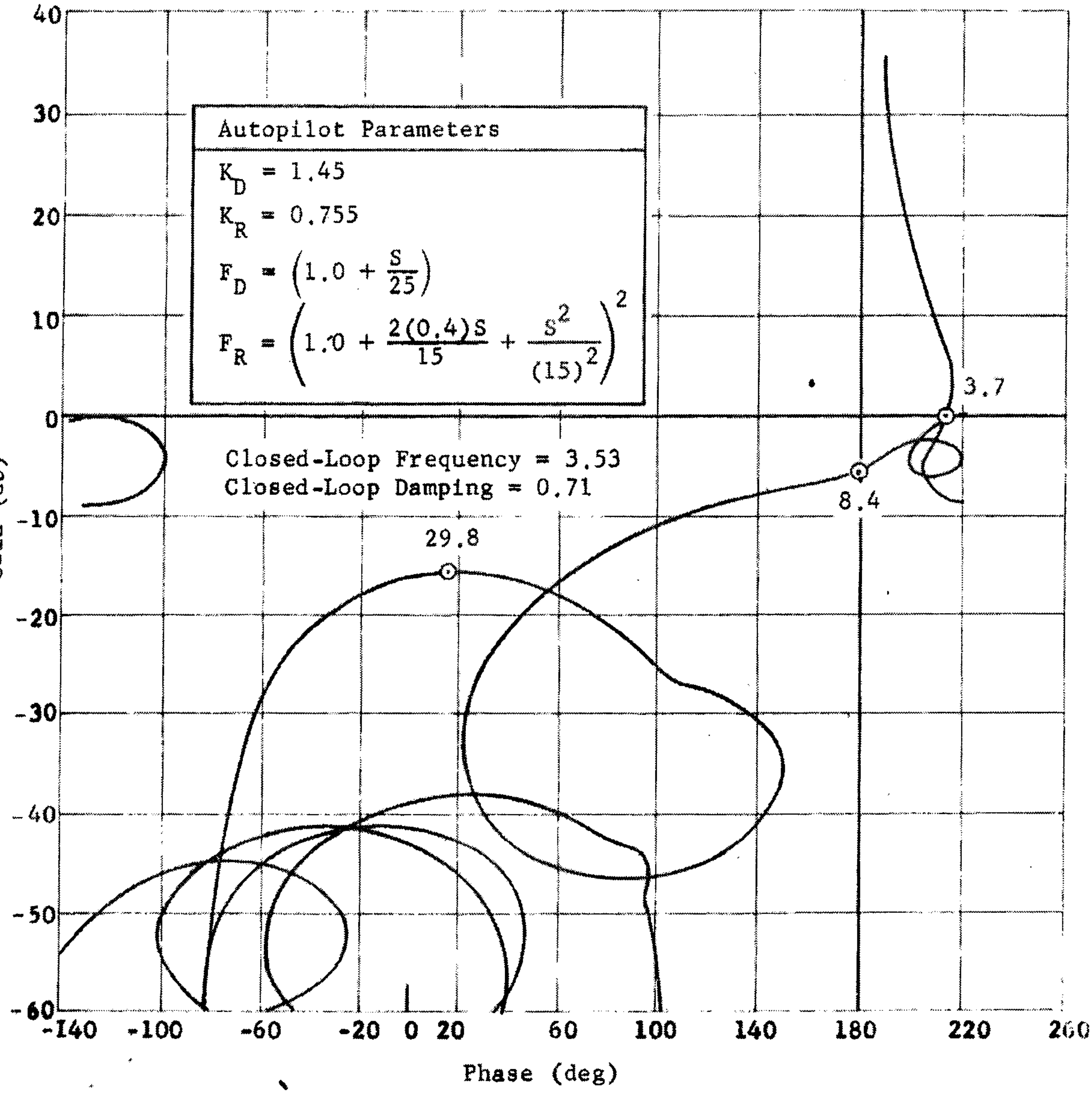


Fig. IV-27 Frequency Response Plot for Stage II; Pitch/Yaw, T = Burnout; 7 seg-120; 65.0-ft, 28,000-lb Payload

Table IV-1 Stage 0 Flight Control System Configurations

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 35 sec	35 to 80 sec	80 sec to Burnout
7 seg-120; 54.5-ft, 28,000-lb Payload				
Pitch Axis	K_D	0.505	0.745	0.311
	K_{R1}	0.202	0.245	0.116
	K_{R2}	0.059	0.070	0.098
	K_A	--	0.302E-3	--
	K_V	--	0.135E-3	--
	F_D	30	30	30
	F_{R1}	20,20	20,20	8,8
	F_{R2}	20,20	20,20	8,8
	F_A	--	3,3,3	--
	τ	--	7.5	--
Yaw Axis	K_D	0.80	0.29	0.55
	K_{R1}	0.32	0.26	0.23
	K_{R2}	0.10	0.052	0.063
	K_A	--	0.52E-3	--
	K_V	--	0.30E-3	--
	F_D	30	30	30
	F_{R1}	35,35	35,35	35,35
	F_{R2}	35,35	35,35	35,35
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time: 0 to 35 sec	35 to 80 sec	80 sec to Burnout
7 seg-120; 65.0-ft, 28,000-lb Payload				
Pitch Axis	K_D	0.45	0.63	0.35
	K_{R1}	0.24	0.24	0.16
	K_{R2}	0.08	0.055	0.094
	K_A	--	0.19E-3	--
	K_V	--	0.09E-3	--
	F_D	20	20	20
	F_{R1}	15,15	15,15	10,10
	F_{R2}	15,15	15,15	10,10
	F_A	--	5,5,5	--
	τ	--	7.5	--
Yaw Axis	K_D	0.81	0.32	0.55
	K_{R1}	0.40	0.23	0.23
	K_{R2}	0.07	0.037	0.063
	K_A	--	0.53E-3	--
	K_V	--	0.28E-3	--
	F_D	30	30	30
	F_{R1}	35,35	35,35	35,35
	F_{R2}	35,35	35,35	35,35
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 35 sec	35 to 80 sec	80 sec to Burnout
7 seg-120; 75.0-ft, 28,000-lb Payload				
Pitch Axis	K_D	0.40	0.66	0.39
	K_{R1}	0.28	0.29	0.20
	K_{R2}	0.10	0.052	0.092
	K_A	--	0.212E-3	--
	K_V	--	0.094E-3	--
	F_D	12	12	12
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	5,5,10	--
	τ	--	7.5	--
Yaw Axis	K_D	0.82	0.35	0.55
	K_{R1}	0.48	0.20	0.23
	K_{R2}	0.042	0.022	0.063
	K_A	--	0.54E-3	--
	K_V	--	0.27E-3	--
	F_D	30	30	30
	F_{R1}	35,35	35,35	35,35
	F_{R2}	35,35	35,35	35,35
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 32 sec	32 to 80 sec	80 sec to Burnout
2 CS-156; 50-ft, 33,000-lb Payload				
Pitch Axis	K_D	0.5	0.5	0.37
	K_{R1}	0.56	0.2	--
	K_{R2}	0.06	0.05	0.08
	K_A	--	0.32-E	--
	K_V	--	0.133E-3	--
	F_D	10	10	10
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	3,3,3	--
	τ	--	7.5	--
Yaw Axis	K_D	0.504	0.25	0.5
	K_{R1}	0.378	0.25	0.31
	K_{R2}	0.076	0.015	0.06
	K_A	--	0.45E-3	--
	K_V	--	0.2E-3	--
	F_D	10	10	10
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 32 sec	32 to 80 sec	80 sec to Burnout
2 CS-156; 58.5-ft, 33,000-lb Payload				
Pitch Axis	K_D	0.563	0.517	0.423
	K_{R1}	0.427	0.47	0.226
	K_{R2}	0.0675	0.0435	0.08
	K_A	--	0.195E-3	--
	K_V	--	0.0865E-3	--
	F_D	9	9	9
	F_{R1}	15,15	15,15	15,15
	F_{R2}	15,15	15,15	15,15
	F_A	--	3,3,3	--
	τ	--	7.5	--
Yaw Axis	K_D	0.504	0.25	0.5
	K_{R1}	0.378	0.25	0.31
	K_{R2}	0.076	0.015	0.06
	K_A	--	0.45E-3	--
	K_V	--	0.2E-3	--
	F_D	10	10	10
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 32 sec	32 to 80 sec	80 sec to Burnout
2 CS-156; 65.0-ft, 33,000-lb Payload				
Pitch Axis	K_D	0.625	0.535	0.475
	K_{R1}	0.375	0.38	0.252
	K_{R2}	0.075	0.037	0.079
	K_A	--	0.09E-3	--
	K_V	--	0.04E-3	--
	F_D	8	8	8
	F_{R1}	20,20	20,20	20,20
	F_{R2}	20,20	20,20	20,20
	F_A	--	3,3,3	--
	τ	--	7.5	--
Yaw Axis	K_D	0.504	0.25	0.5
	K_{R1}	0.378	0.25	0.31
	K_{R2}	0.076	0.015	0.06
	K_A	--	0.45E-3	--
	K_V	--	0.2E-3	--
	F_D	10	10	10
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 28 sec	28 to 80 sec	80 sec to Burnout
3 CS-156; 50.0-ft, 42,000-lb Payload				
Pitch Axis	K_D	0.52	0.67	0.34
	K_{R1}	0.35	0.33	0.18
	K_{R2}	0.075	0.06	0.09
	K_A	--	0.24E-3	--
	K_V	--	0.062E-3	--
	F_D	10	10	10
	F_{R1}	15,15	15,15	8,8
	F_{R2}	15,15	15,15	8,8
	F_A	--	3,5,10	--
	τ	--	7.5	--
Yaw Axis	K_D	0.64	0.28	0.36
	K_{R1}	0.32	0.28	0.28
	K_{R2}	0.095	0.028	0.053
	K_A	--	0.51E-3	--
	K_V	--	0.23E-3	--
	F_D	10	10	10
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 28 sec	28 to 80 sec	80 sec to Burnout
3 CS-156; 61.0-ft, 42,000-lb Payload				
Pitch Axis	K_D	0.535	0.575	0.39
	K_{R1}	0.285	0.378	0.19
	K_{R2}	0.0875	0.0565	0.1
	K_A	--	0.227E-3	--
	K_V	--	0.0485E-3	--
	F_D	10	10	10
	F_{R1}	17,17	17,17	14,14
	F_{R2}	13,13	13,13	9,9
	F_A	--	3,5,10	--
	τ	--	7.5	--
Yaw axis	K_D	0.56	0.265	0.34
	K_{R1}	0.31	0.265	0.265
	K_{R2}	0.0835	0.0265	0.050
	K_A	--	0.48E-3	--
	K_V	--	0.215E-3	--
	F_D	10	10	10
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (cont)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 28 sec	28 to 80 sec	80 sec to Burnout
3 CS-156; 72.0-ft, 42,000-lb Payload				
Pitch Axis	K_D	0.55	0.48	0.44
	K_{R1}	0.22	0.426	0.20
	K_{R2}	0.1	0.053	0.11
	K_A	--	0.213E-3	--
	K_V	--	0.035E-3	--
	F_D	10	10	10
	F_{R1}	20,20	20,20	20,20
	F_{R2}	10,10	10,10	10,10
	F_A	--	5,5,5	--
	τ	--	7.5	--
Yaw Axis	K_D	0.48	0.25	0.32
	K_{R1}	0.3	0.25	0.25
	K_{R2}	0.072	0.025	0.047
	K_A	--	0.45E-3	--
	K_V	--	0.2E-3	--
	F_D	10	10	10
	F_{R1}	10,10	10,10	10,10
	F_{R2}	10,10	10,10	10,10
	F_A	--	3,3,3	--
	τ	--	7.5	--

Table IV-1 (concl)

Configuration	Control System Constant	Constants vs Flight Time		
		0 to 28 sec	28 to 80 sec	80 sec to Burnout
All Configurations				
Roll Axis	K_D	0.36	0.36	0.18
	K_R	0.19	0.19	0.10
	K_D	30	30	30
	F_R	30	30	30

K_D Displacement gain

K_{R1} Rate gain for rate gyro located at Station 887 (sec)

K_{R2} Rate gain for rate gyro located at Station 112 (sec)

K_A Accelerometer gain for the LASS located at Station 112 (rad/in./sec^2)

K_V Velocity gain for the LASS located at Station 112 (rad/in./sec)

F_D Displacement loop dynamics (rad/sec)*

F_{R1} Loop dynamics for K_{R1} (rad/sec)*

F_{R2} Loop dynamics for K_{R2} (rad/sec)*

F_A Accelerometer loop dynamics (rad/sec)*

τ Integration constant (sec)

*Given as corner frequencies of first order filters.

Table IV-2 Stage I Flight Control System Configurations

Configuration	Control System Constant	Constant vs Flight Time: Start to Midflight	Constant vs Flight Time: Midflight to Burnout
75.0-ft, 28,000-lb Payload			
Pitch and Yaw Axes	K_D	1.08	0.055
	K_{R1}	0.24	0.053
	K_{R2}	0.050	0.34
	F_D	15	15
	F_{R1}	10,35	10,35
	F_{R2}	10,35	10,35
72.0-ft, 42,000-lb Payload			
Pitch and Yaw Axes	K_D	1.32	0.47
	K_{R1}	0.33	0.05
	K_{R2}	1.10	0.33
	F_D	15	15
	F_{R1}	10,35	10,35
	F_{R2}	10,10	10,10

K_D	Displacement gain
K_{R1}	Rate gain for rate gyro located at Station 887 (sec)
K_{R2}	Rate gain for rate gyro located at Station 112 (sec)
F_D	Displacement Loop Dynamics (rad/sec)*
F_{R1}	Loop Dynamics for K_{R1} (rad/sec)*
F_{R2}	Loop Dynamics for K_{R2} (rad/sec)*

*Given as corner frequencies of first order filters.

Table IV-3 Stage II Flight Control System Configurations

Configuration	Control System Constant	Constant vs Flight Time, Start to Burnout*	
65.0-ft, 28,000-lb Payload			
Pitch and Yaw Axes	K_D	1.45	
	K_{R2}	0.755	
	F_D	(1.0 + S/25)	
	F_{R2}	$\left[1.0 + 2\ (0.4)\ S/15 + S^2 + (15)^2\right]^2$	
Configuration	Control System Constant	Constant vs Flight Time: Start to Midflight	Constant vs Flight Time: Midflight to Burnout
72.0-ft, 42,000-lb Payload			
Pitch and Yaw Axes	K_D	1.8	1.4
	K_{R2}	1.0	0.8
	F_D	(1.0 + S/25)	(1.0 + S/25)
	F_{R2}	$\left[1.0 + 2(0.5)S/12 + S^2/(12)^2\right]^2$	$\left[1.0 + 2(0.5)S/12 + S^2/(12)^2\right]^2$

K_D Displacement gain

K_{R2} Rate gain for rate gyro located at Station 112 (sec)

K_F Displacement loop dynamics (rad/sec)

F_{R2} Loop dynamics for K_{R2} (rad/sec)

*No gain change was used in the 28,000-lb payload configuration studied, but margins were barely met. A gain change is recommended for a final autopilot design.

4. Control System Requirements

The following control system requirements were extracted from the Detail Specification for Standard Space Launch Vehicle, Standard Core, SSS-TIII-D-SLV/01A010

Gain Margins		
Frequency Range	Design Requirement	Design Objective
0 to 1st Structural Bending Mode (db)	5.0	6.0
1st to 3rd Structural Bending Modes (db)	8.0	10.0
Structural Modes above 2nd Mode (db)	10.0*	10.0*
Phase Margins		
Frequency Range	Design Objective	Design Requirement
0 to 1st Structural Bending Mode (db)	25	30
Above 1st Structural Bending Mode (db)	45	60
Closed Loop Requirements		
Frequency Range	Design Objective	Design Requirement
All Pitch/Yaw (rad/sec)†	0.4 to 2.0	0.3 to 1.5

*Independent of phase contribution.
†Not required during load relief operation.

B. TVC SYSTEM SIZING

Dispersion studies were performed to estimate the requirements for the TVC system. TVC fluid requirements and equivalent thrust vector deflection angles were predicted from these studies.

1. Fluid Utilization

Pertinent vehicle and environmental parameters were varied through estimated 3σ ranges. The effect of the parameters on TVC fluid required for control purposes was noted in the dispersion studies. The fluid required for the vehicle and environmental parameters was then combined to yield the total amount of fluid required.

The same assumptions and ground rules listed in the rigid body airloads section are applicable to this section also. The results are summarized in Table IV-4.

Table IV-4 Summary of Fluid Requirements

Vehicle SRM Configuration	TVC Fluid Required for Control Purposes (lb)		TVC Fluid Required (lb)
	Engine 1	Engine 2	
7 seg-120	4,508	6,377	12,754
2 CS-156	8,623	10,214	20,228
3 CS-156	10,042	11,300	22,660

The right-hand column in Table IV-4 assumes that each tank is separate with no common manifolding between the two tanks. Note that Fig. IV-28, IV-29, and IV-30 portray only the sum of Engines 1 and 2 and not the total fluid required. The total fluid required is twice that required by the engine using the maximum, since the maximum engine cannot be predicted before flight. A breakdown of the totals are shown in Tables IV-5 thru IV-7. These show the contribution from each of the variables considered. The major contributing factor in the sizing of the system is the maximum wind profile. This can be seen by referring to Fig. IV-28 thru IV-30, where the fluid required for the major variations is plotted versus time of flight. Therefore, to assure that the TVC system is of sufficient size for both WTR and ETR launches, the most severe 624A maximum wind criteria were used. The longest payload length for each SRM configuration was selected for investigation. The parameters considered in the sizing study are:

1) Nominal trajectory - The nominal performing vehicle will require a determinable amount of TVC fluid for the side-force characteristics given in Fig. 4.1.-1 thru 4.1.11 of TM 5141/31-65-16 (Rev 1), "Titan III/MOL Data Book," dated 27 August 1965.

2) Thrust vector misalignment - A mislignment in the direction of the thrust vector in both the pitch and yaw vehicle planes requires an additional amount of fluid. The variation is ±0.25 deg in both pitch and yaw.

3) Thrust vector offset - This variation accounts for effects whereby the effective thrust vector gimbal point is varied ±2in. in the lateral direction and ±20 in. in the longitudinal direction.

4) SRM temperature differential - The effects of a maximum temperature differential between the two SRMs of 10°F were considered.

5) SRM thrust deviation - The perturbational effects of considering maximum and minimum thrust-time histories based on the $\pm 3\sigma$ values supplied by the SRM manufacturer.

6) SRM tailoff thrust differential - The differential thrust magnitudes during tailoff were considered as separate variations. These accounted for a considerable amount of fluid.

7) Maximum wind - The 624A criteria were used for this variation. This is more severe than the WTR criteria and is the largest contributor to the fluid requirement. The wind azimuth used was 240 deg.

8) Mean wind - The mean wind used was the 50% NASA wind.

9) Roll maneuver - A maximum roll maneuver of 50 deg was considered.

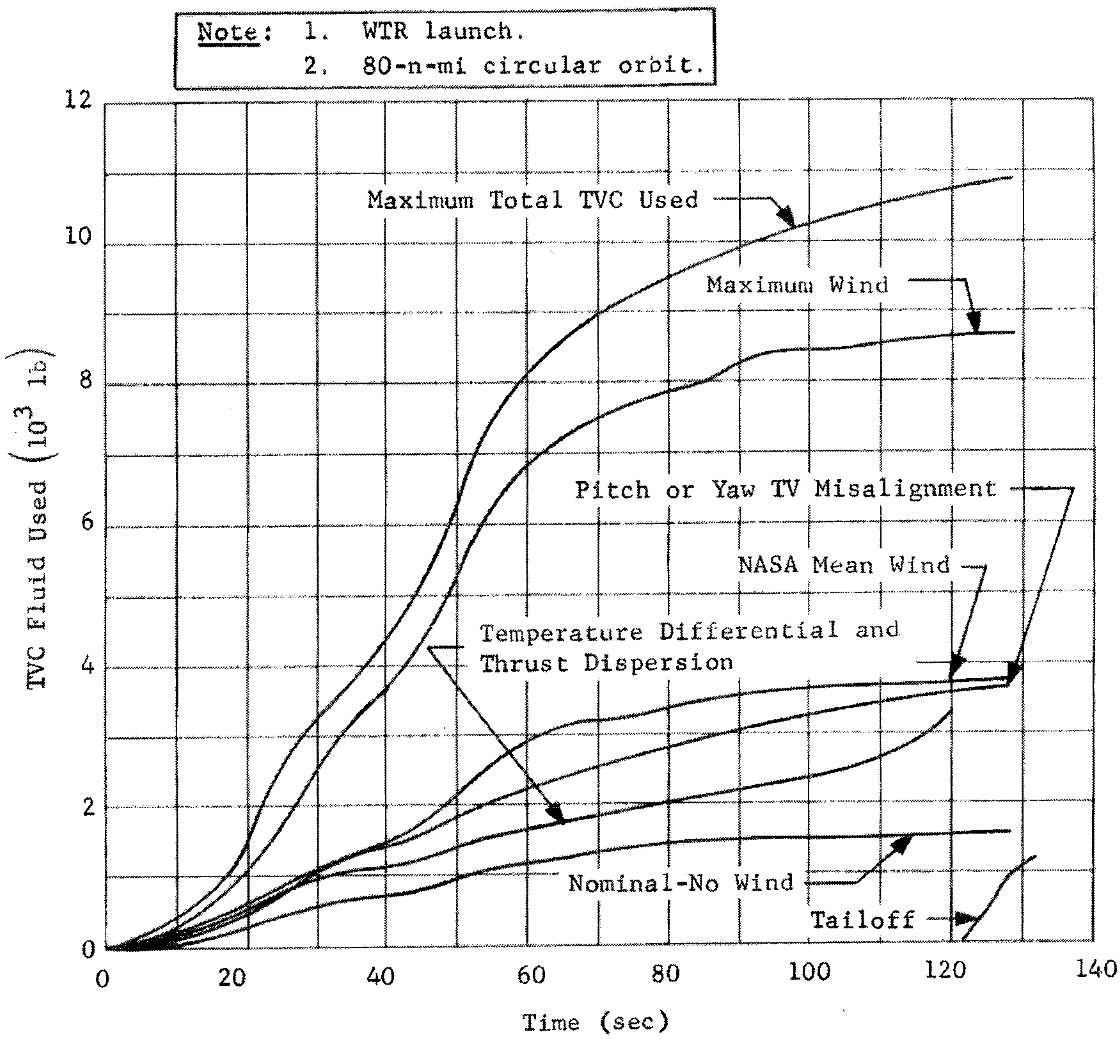


Fig. IV-28 TVC Fluid Utilization, 7 seg-120

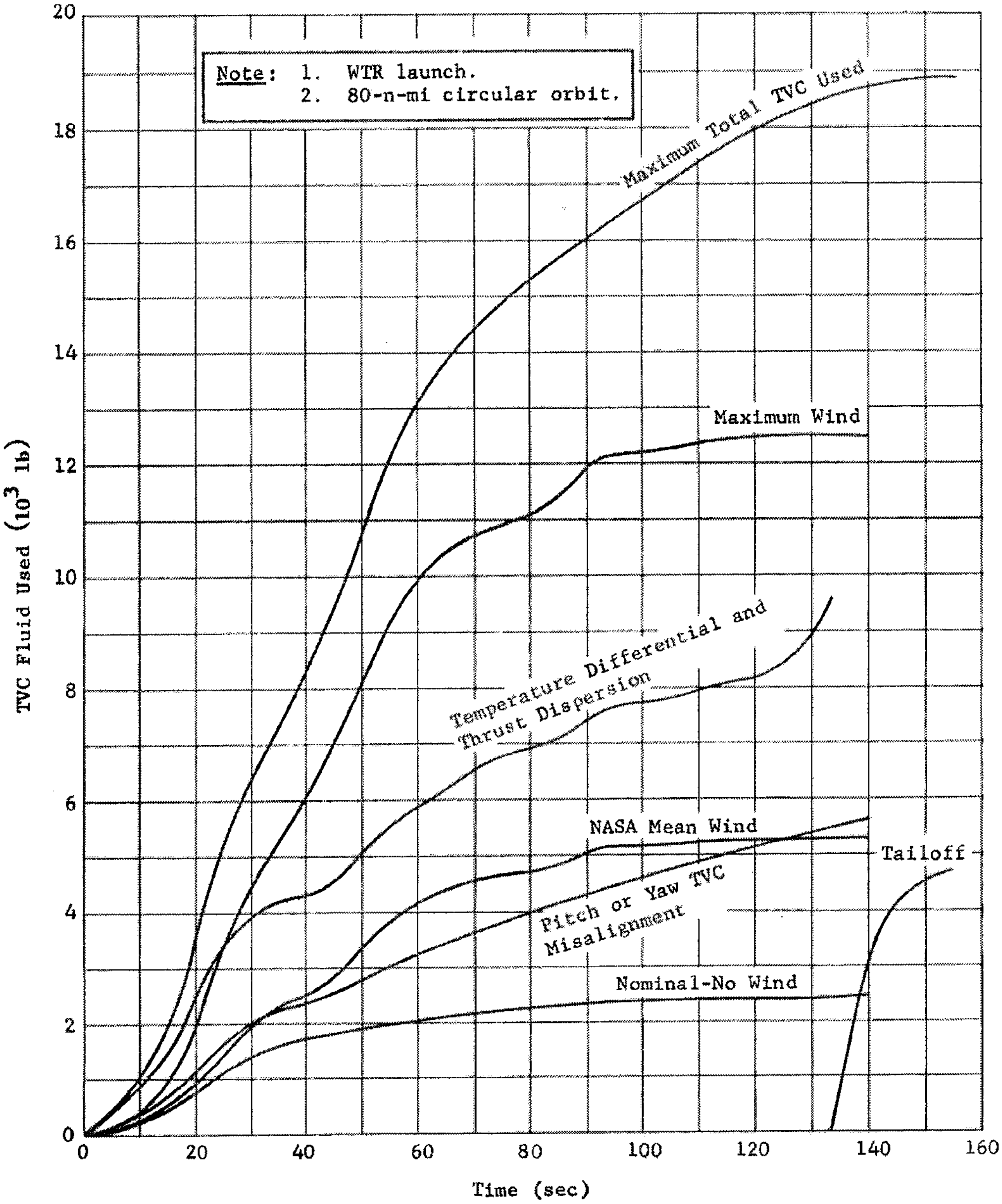


Fig. IV-29 TVC Fluid Utilization, 2 CS-156

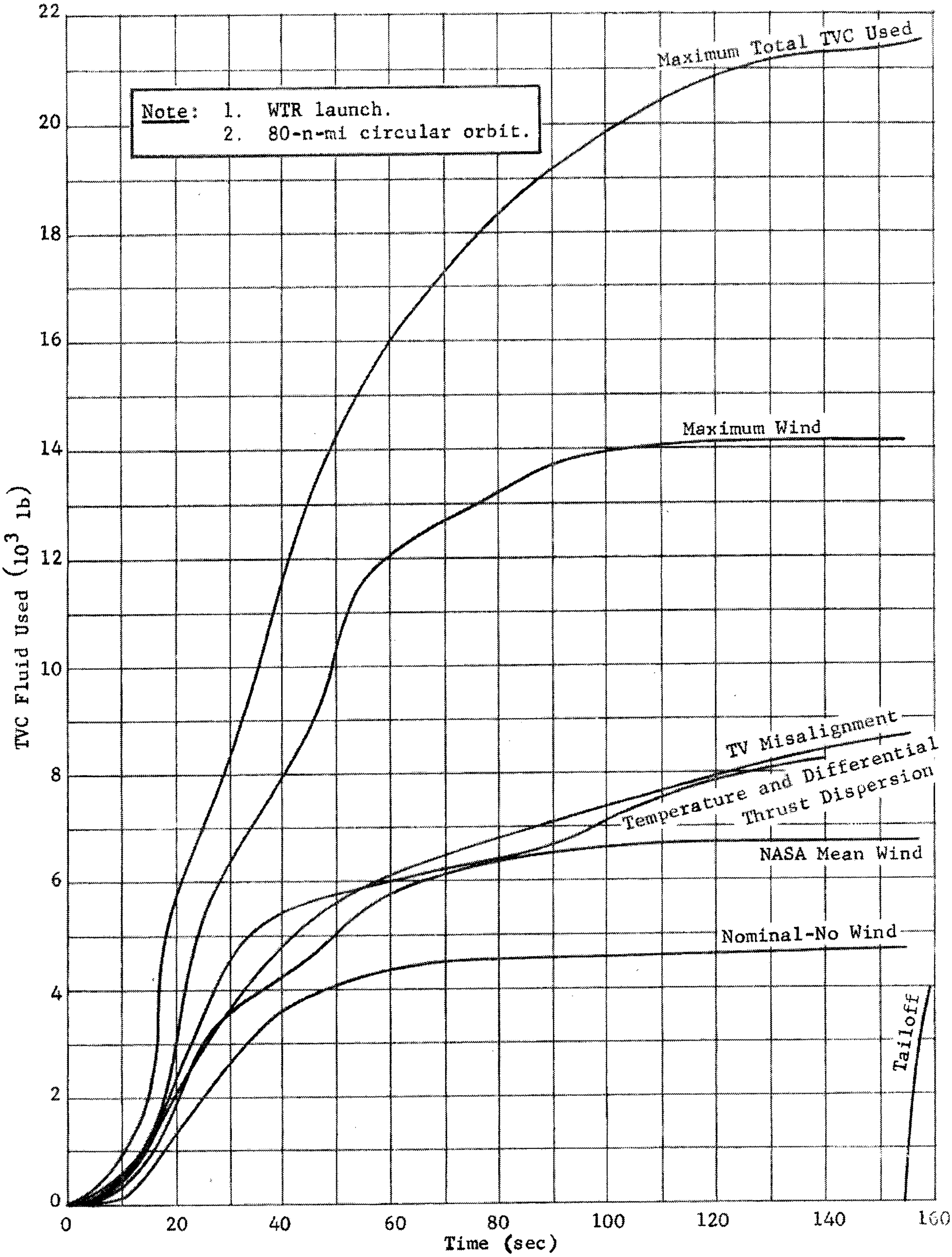


Fig. IV-30 TVC Fluid Utilization, 3 CS-156

Table IV-5 Injectant Requirements, 7 seg-120

I. Independent Terms	Incremental Injectant		
A. Pitch Plane	Engine 1	Engine 2	Total
Thrust Vector Misalignment	76	2,054	2,130
Thrust Vector Lateral Offset	714	700	1,414
Thrust Vector Longitudinal Offset	672	673	1,345
q_P = Total Pitch Plane Required $= \sqrt{\sum Req_P^2}$	983	2,272	3,255
B. Yaw Plane			
Thrust Vector Misalignment	1,060	1,061	2,121
Thrust Vector Lateral Offset	714	700	1,414
Thrust Vector Longitudinal Offset	672	673	1,345
q_Y = Total Yaw Plane Required $= \Sigma\ Req_Y^2$	1,444	1,438	2,882
Q_{PY} = Total Pitch and Yaw Required $= \frac{1}{\sqrt{2}}\left(q_P{}^2 + q_Y{}^2\right)^{\frac{1}{2}}$	1,235	1,901	3,136
C. Maximum Wind-Mean Wind	2,004	2,852	4,856
D. SRM Maximum Temperature Differential	748	1,063	1,811
E. SRM Thrust Dispersions	748	1,063	1,811
F. SRM Tailoff Differential	1,106	1,148	2,254
T_I = Total Requirement of Independent Terms $= \left[Q_{PY}{}^2 + C^2 + D^2 + E^2 + F^2\right]^{\frac{1}{2}}$	2,706	3,768	6,474
II. Dependent Terms			
AA. Azimuth Orientation (50 deg)	416	224	640
BB. Mean Wind (50% NASA)	670	1,533	2,203
CC. Nominal Trajectory	716	852	1,568
T_{II} = Total Requirement of Dependent Terms = AA + BB + CC	1,802	2,609	4,411
T_T = Total Injectant Requirements $= T_I + T_{II}$	4,508	6,377	10,885

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Table IV-6 TVC Injectant Requirements, 2 CS-156

I. Independent Terms	Incremental Injectant		
A. Pitch Plane	Engine 1	Engine 2	Total
Thrust Vector Misalignment	88	3,110	3,198
Thrust Vector Lateral Offset	1,179	1,201	2,380
Thrust Vector Longitudinal Offset	1,210	1,201	2,411
q_P = Total Pitch Plane Required $= \sqrt{\sum Req_P^2}$	1,692	3,544	5,236
B. Yaw Plane			
Thrust Vector Misalignment	1,636	1,670	3,306
Thrust Vector Lateral Offset	1,179	1,201	2,380
Thrust Vector Longitudinal Offset	1,210	1,201	2,411
q_Y = Total Yaw Plane Required $= \sqrt{\sum Req^2_Y}$	2,352	2,382	4,734
Q_{PY} = Total Pitch and Yaw Required $= \frac{1}{\sqrt{2}} \left(q_P^2 + q_Y^2\right)^{\frac{1}{2}}$	2,049	3,019	5,068
C. Maximum Wind-Mean Wind	3,228	3,876	7,104
D. SRM Maximum Temperature Differential; E. SRM Thrust Dispersions	3,343	3,778	7,121
F. SRM Tailoff Differential	2,659	2,683	5,343
T_I = Total Requirement of Independent Terms $= \left[Q^2_{PY} + C^2 + D^2 + E^2 + F^2\right]^{\frac{1}{2}}$	5,733	6,753	12,486
II. Dependent Terms			
AA. Azimuth Orientation (50 deg)	620	425	1,045
BB. Mean Wind (50% NASA)	1,098	1,741	2,839
CC. Nominal Trajectory	1,172	1,295	2,467
T_{II} = Total Requirement of Dependent Terms = AA + BB + CC	2,890	3,461	6,351
T_T = Total Injectant Requirements $= T_I + T_{II}$	8,623	10,214	18,837

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Table IV-7 Injectant Requirements, 3 CS-156

	Incremental Injectant		
I. Independent Terms	Engine 1	Engine 2	Total
A. Pitch Plane			
Thrust Vector Misalignment	121	2,981	3,102
Thrust Vector Lateral Offset	1,358	1,405	2,763
Thrust Vector Longitudinal Offset	1,278	1,230	2,508
q_P = Total Pitch Plane Required $= \sqrt{\sum Req_P^2}$	1,869	3,518	5,387
B. Yaw Plane			
Thrust Vector Misalignment	2,001	2,040	4,041
Thrust Vector Lateral Offset	1,358	1,405	2,763
Thrust Vector Longitudinal Offset	1,278	1,230	2,508
q_Y = Total Yaw Plane Required $= \sqrt{\sum Req_Y^2}$	2,735	2,766	5,501
Q_{PY} = Total Pitch and Yaw Required $= \frac{1}{\sqrt{2}} \left(q_P^2 + q_Y^2\right)^{\frac{1}{2}}$	2,342	3,164	5,506
C. Maximum Wind-Mean Wind	3,092	4,287	7,379
D. SRM Maximum Temperature Differential E. SRM Thrust Dispersions	4,306	4,401	8,707
F. SRM Tailoff Differential	1,993	2,000	3,993
T_I = Total Requirement of Independent Terms $= \left[Q_{PY}^2 + C^2 + D^2 + E^2 + F^2\right]^{\frac{1}{2}}$	6,128	7,194	13,322
II. Dependent Terms			
AA. Azimuth Orientation (50 deg)	810	491	1,301
BB. Mean Wind (50% NASA)	538	1,481	2,019
CC. Nominal Trajectory	2,566	2,134	4,700
T_{II} = Total Requirement of Dependent Terms $= AA + BB + CC$	3,914	4,106	8,020
T_T = Total Injectant Requirements $= T_I + T_{II}$	10,042	11,300	21,342

The characteristics of the TVC system are included in the MOL Data Book. These data include (1) side flow rate per quadrant vs command voltage with zero voltage in adjacent quadrants; (2) side force per quadrant vs command voltage for various motor vacuum nozzle centerline thrust and aft end stagnation chamber pressure; (3) nozzle centerline axial augmentation force per quadrant vs command voltage for various vacuum nozzle centerline thrust and aft end stagnation chamber pressures, and (4) nozzle centerline axial augmentation force during dump operation vs command voltage for various motor vacuum nozzle centerline thrust and aft end stagnation pressures.

The amount of TVC fluid required for the variation in the above parameters was combined according to the following formula:

$$\begin{aligned}\text{TVC Fluid Required} = \Big[& \tfrac{1}{2}(\text{RSS}^2_{\text{Pitch}} + \text{RSS}^2_{\text{Yaw}} \\ & + (\text{Maximum Wind} - \text{Mean Wind})^2 + (\text{SRM Temperature Differential})^2 \\ & + (\text{SRM Thrust Dispersion})^2 + (\text{SRM Tailoff})^2\Big]^{\frac{1}{2}} \\ & + (\text{Azimuth Orientation}) + (\text{Mean Wind}) \\ & + (\text{Nominal Trajectory}).\end{aligned}$$

The TVC fluid required by Engine 2 exceeded that of Engine 1 for all configurations studied. This is due primarily to the tendency of the vehicle to roll due to the rolling moment that results from the asymmetric TVC tank location. The cancellation of this rolling moment requires a differential deflection between Engines 1 and 2 in the pitch plane.

Figures IV-28 thru IV-30 show graphically the usage of TVC fluid for the various parameter dispersions. The top line curve labeled "Maximum Total TVC Used" represents the estimated 3σ value of fluid required during flight, but not the fluid that must be loaded. Since the fluid required in each tank depends on wind direction, each tank must be loaded to the maximum expected amount, as shown in Table IV-4.

2. Side-Force Requirements

The TVC side-force requirements were estimated by evaluating the six-degree-of-freedom trajectories used for the loads studies (including the wind search runs) and the six-degree-of-freedom trajectories used in the TVC utilization studies.

The results of this evaluation showed that all SRM configurations can be controlled with the following equivalent thrust vector deflection capability:

Flight Phase	Equivalent Thrust Vector Deflection (deg)
Liftoff to Clearing Launch Stand	3
Clearing Launch Stand to End of Web Action	2½
During Thrust Tailoff	6

C. VIBRATION ANALYSIS

During this study, vibration analyses were conducted to provide the mathematical description of the flexible airframe for input to the FCS analysis and to the vehicle loads analyses.

The most significant factor resulting from the vibration analysis that affected the FCS was the great reduction (compared to Titan IIIC with short payloads) of the structural mode frequencies. For example, Fig. IV-1 shows that there is almost a 50% reduction in the first structural mode frequency compared to Titan IIIC (similar reduction in the higher structural mode frequencies also occur). This imposes problems on the FCS as discussed in Chapter IV.A.

Typical Stage 0 first and second structural mode shapes are shown in Fig. IV-4.

V. STAGING

This chapter presents a staging analysis of the Manned Orbiting Laboratory (MOL) performance improvement configuration described in Chapter II. The analysis is limited to solid rocket motor (SRM) separation. Titan IIIC staging evaluations have shown upper-stage separations to be relatively insensitive to upper body weight and, therefore, are not expected to create any unusual staging problems. The difference in Stage I engine thrust tailoff characteristics between a 15:1 engine compared to that of an 8:1 engine would have a small effect on the fire-in-the-hole staging technique. A staging analysis for each of these upper stages was deemed unnecessary for the purpose of this study and would serve little useful purpose before the MOL vehicle configuration is selected. Staging of the three SRM strap-on configurations, however, produces significant interaction with the MOL booster configuration selection.

The three basic SRM strap-on configurations, i.e., 7 segment, 120-in. (7 seg-120), 2 center segment, 156-in. (2 CS-156), and 3 center segment, 156-in. (3 CS-156), are evaluated as to feasibility of separating the burned out SRMs from the Titan III core using the lateral translation technique successfully employed on Titan IIIC. This technique uses short-burn solid rocket motors mounted perpendicular to the SRM longitudinal axis to translate the SRM laterally away from Stage I. The staging rockets are located in the SRM nose fairing and tail skirt areas with a proper balance of orientation and thrust levels to obtain satisfactory clearance at separation.

A staging sequence is selected that provides adequate control of the vehicle and respects structural limitations during SRM thrust decay.

A. CONFIGURATIONS EVALUATED

Of the many combinations of core configuration, payload, and SRMs possible, only the three shown in Table V-1, which represent significant staging problems, are evaluated.

Table V-1 Configurations Evaluated

SRM Option	Stage I Engine Expansion Ratio	Transtage	Payload Weight (lb)/ Length (ft)	Aero-dynamic Pressure (psf)	($q\alpha$) Product
7 seg-120	15:1	Yes	33,000/58	40	80
2 CS-156	8:1	Yes	33,000/58	20	40
3 CS-156	8:1	Yes	33,000/58	5	10

Note that certain vehicle characteristics were assumed common to the three configurations so that the comparison of results might have a common basis. This is true of the payload weight (33,000 lb), payload length (58 ft), and total core weight (407,000 lb) at Stage I start. The thrust vector control (TVC) fluid injection system is assumed in each case to have the same side force to axial thrust relationship for a given thrust vector deflection command as the Titan IIIC system.

The Titan IIIC staging rockets are used in the evaluation study. The nominal staging rocket thrust is 4740 lb, with a burn time of 2.6 sec. The rockets are stacked one over the other with their thrust vector oriented as nearly as possible through the projection of the SRM center of gravity (cg) in the roll plane. The staging rockets are canted approximately 30 deg outboard of the yaw plane to avoid plume impingement on Stage I.

The staging sequence is predicated on the use of an inertial guidance system. The inertial measurement unit (IMU) senses a drop in longitudinal acceleration to a predetermined level after start of SRM thrust tailoff. This g-level sensing initiates the staging sequence with discrete signals being issued by the guidance system for Stage I ignition, Stage 0/I attachment ordnance, and staging rocket ignition.

Table V-2 presents the mass property data used in the staging analysis. Aerodynamic coefficients used in the analysis include Stage I engine plume impingement forces and are documented in TM 0494/10-562.*

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*Staging Aerodynamics for MOL Compatibility with 7-Segment, 120-in. Diameter SRMs and 2- and 3-Segment, 156-in. Diameter SRMs. TM 0494/10-562. Martin Company, Denver, Colorado, 5 August 1965.

Table V-2 Staging Mass Property Data

Step 0 Burnout Per Solid	7 Segment	2 Segment	3 Segment
Total Burnout Weight (lb)	94,629	122,780	148,090
TVC Propellant at Burnout (lb)	4,250	6,000	6,555
Center of Gravity (in.)			
Horizontal	V.S. 866.	V.S. 985.	V.S. 895.
Vertical	W.L. 70.7	W.L. 70.2	W.L. 69.3
Lateral	B.L. 136.5	B.L. 134.4	B.L. 135.3
Moment of Inertia (slug-ft^2)			
Pitch	2.6×10^6	$7.1 \times 10^{6*}$	$4.6 \times 10^{6\dagger}$
Yaw	2.6×10^6	7.1×10^6	4.6×10^6
Roll	0.07×10^6	0.12×10^6	0.15×10^6

*Moment of inertia for 2-seg SRM was based on SRM manufacturer data. 2-seg data appear questionable.
†3-seg SRM moment of inertia data are based on Martin-generated data.

Stage I Start Weight (33,000-lb Payload)	
Weight (lb)	407,000
Center of Gravity (in.)	
Horizontal	V.S. 645.
Vertical	W.L. 60.1
Lateral	B.L. 0.4
Moment of Inertia (slug-ft^2)	
Pitch	15.17×10^6
Yaw	15.17×10^6
Roll	0.034×10^6

B. STAGING CRITERIA

The staging sequence and SRM separation are considered satisfactory if the following criteria are met:

1) SRM separation will be accomplished after web burnout and adequate Stage I thrust has been achieved. For optimum performance, the SRMs must be jettisoned as soon as possible after their thrust-to-weight (T/W) ratio falls below the T/W ratio of Stage I. The extent of lateral translation staging rocket thrust must be sufficient to produce satisfactory separation clearance in the presence of one staging rocket failure either forward or aft, maximum SRM residual thrust, core plume force impingement, and adverse aerodynamic loads no greater than produced by the following conditions

   a) 7 seg-120

      Dynamic pressure, $q = 40$ psf
      Angle of attack, $\alpha = 2$ deg
      Product, $q\alpha = 80$ lb-deg/ft^2

   b) 2 CS-156

      Dynamic pressure, $q = 20$ psf
      Angle of attack, $\alpha = 2$ deg
      Product, $q\alpha = 40$ lb-deg/ft^2

   c) 3 CS-156

      Dynamic pressure, $q = 5$ psf
      Angle of attack, $\alpha = 2$ deg
      Product, $q\alpha = 10$ lb-deg/ft^2

2) Axial acceleration must be maintained on the core vehicle during SRM thrust decay consistent with the minimum acceleration requirements of Stage I liquid engine net positive suction head (NPSH).

3) Adequate control authority must be maintained to cope with maximum SRM differential thrust during the tail-off period.

4) Axial acceleration at Stage I ignition in the presence of maximum thrust differential must not violate Stage I structural limitations.

C. STAGING SEQUENCE CONSIDERATIONS

During the SRM thrust tailoff period, differential thrust between the two SRMs results in decreasing control authority. The Stage I engines should be ignited as soon as possible to augment control authority. The Stage I engine thrust buildup must not cause the booster acceleration to exceed the structural design limits of the core. The structural load limit thus determines the earliest time for Stage I ignition ($87FS_1$). The IMU g-sensing level is established from the earliest time of core ignition after allowing for variations in SRM tailoff characteristics and guidance delays in sensing and issuing discretes.

It is desirable for improved payload capability to jettison the SRMs as soon as their T/W ratio falls below the T/W ratio of the core. However, the effect of the canted residual SRM thrust vector is to rotate the aft end of the SRM into the core if residual thrust is too high at separation. The expected SRM thrust decay curves, therefore, determine the time of physical separation.

Several aspects of the staging sequence are independent of the SRM configuration as indicated in the following typical staging sequence:

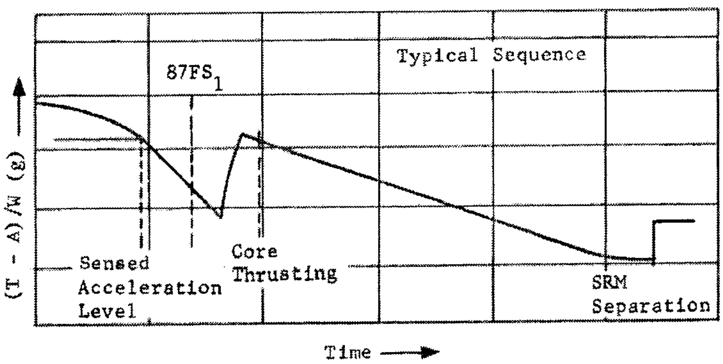
1) Zero time reference - Sensing longitudinal acceleration initiates the staging sequence;

2) 1.15 to 2.65 sec - Time from g-sensing to $87FS_1$ (this represents the time delays in the airborne computer from sensing the prescribed g-level to issuance of a discrete);

3) 0.7 to 1.1 sec - Time between $87FS_1$ and actual thrust rise;

4) The time from $87FS_1$ to SRM separation is a function of the maximum residual SRM thrust and must be in multiples of the IMU computation cycle of 1.25 sec (assumed for this study).

A typical profile of axial acceleration (T - A)/W vs time during the staging sequence is shown in Fig. V-1. The actual staging sequences selected for the three SRM configurations are also presented in this figure. Problems associated with these sequences are discussed in succeeding paragraphs.



Staging Rocket Requirements* (per SRM)		
SRM Options	Number of Rockets	Location Forward/Aft
7 seg-120	8	4/4
2 CS-156	9	5/4
3 CS-156	9	5/4

Staging Sequence

SRM Option	Sensed Acceleration (g)	Time of $87FS_1$† (sec)	Time for SRMs Separation (sec)†
7 seg-120	1.90	1.15 → 2.65	12.40 → 14.00
2 CS-156	2.29	1.15 → 2.65	11.15 → 12.75
3 CS-156	2.29	1.15 → 2.65	8.65 → 10.25

*Titan IIIC lateral separation rockets.
†Times sequenced from sensed acceleration level major ADC comp. cycle, 1.25 sec.

Fig. V-1 Staging Sequence

D. STAGING SEQUENCE ACCELERATION PROFILE

Acceleration profiles for the three basic SRM configurations are presented in TM 5141/31-65-19.* Each of these profiles is very similar to the typical profile shown in Fig. V-1. The two points of concern with any sequence involve the maximum structural acceleration limit at the time of thrust rise after $87FS_1$, as indicated by a rise in engine chamber pressure (P_c) and the minimum acceleration at SRM separation for NPSH requirements. These acceleration levels have been tabulated for convenience in Table V-3.

Table V-3 Staging Sequence and Accelerations

		$87FS_1$	P_c Rise		SRM Ejection	
Configuration	Sensed Acceleration (g)	Time from g Sensing (sec)	Time From $87FS_1$ (sec)	g Level	Time From g Sensing (sec)	Minimum g and Stage I Propellant Temperature
7 seg-120	1.90	1.15 Min	0.7 Min	1.1 Min	12.4 Min	0.860 70°F
		2.65 Max	1.1 Max	1.6 Max	14.0 Max	0.850* 90°F
2 CS-156	2.29	1.15 Min	0.7 Min	0.94 Min	11.15 Min	0.681 70°F
		2.65 Max	1.1 Max	1.85 Max	12.75 Max	0.676* 90°F
3 CS-156	2.29	1.15 Min	0.7 Min	1.13 Min	8.65 Min	0.625* 70°F
		2.65 Max	1.1 Max	1.85 Max	10.25 Max	0.620† 90°F

*Accelerations insufficient to meet present NPSH oxidizer requirements of 44 ft.
†Accelerations insufficient to meet proposed oxidizer requirements of 35 ft. Lockup pressure must be increased by 7 psi.

*MOL Compatibility Solid Rocket Motor Staging Analysis Report. TM 5141/31-65-19. Martin Company, Denver, Colorado, August 1965.

1. Staging Loads

The maximum acceleration levels at P_c rise combined with maximum SRM thrust differential shown in Fig. V-2 for an early Stage I engine start sequence produce significant structural loads in the core structure. The primary area of the core structure affected by staging loads is the Stage I longeron. The maximum longeron load actually occurs just after Stage I engine start where start transient maximum thrust overshoot occurs. The ultimate longeron loads at vehicle station 1225 for each of the SRM configurations are plotted in Fig. V-3 as functions of longitudinal acceleration and SRM differential thrust. Note that differential thrust vs longitudinal acceleration is shown for reference purposes.

The maximum vehicle longitudinal acceleration at Stage I engine start is 1.6 g for the 7 seg-120 SRM and 1.85 g for both the 2 CS-156 and 3 CS-156 SRMs. These design conditions are taken from Table V-3. The longeron load for the 7 seg-120 configuration is approximately equal to the Titan IIIC test load. However, the loads for the 2 CS-156 and 3 CS-156 SRMs are approximately 20% higher than the test load and necessitate a redesign of the longeron as described in Chapter II.

2. NPSH Requirements

Referring again to Table V-3, an asterisk identifies the SRM configurations that do not meet the present oxidizer NPSH requirement of 44 ft at SRM separation. None of the three SRM configurations meet the present NPSH requirement when 90°F Stage I propellant temperatures are considered. The 90°F propellant temperature is compatible with Eastern Test Range (ETR) launch requirements. The maximum propellant temperature dispersion expected at the Western Test Range (WTR) is 75°F. The 7 seg-120 SRM and the 2 CS-156 SRM both meet the present 70°F NPSH requirement. Reduction of the present oxidizer NPSH requirement of 44 ft to 35 ft has been proposed to the Aerojet-General Corporation. This proposal has been tentatively accepted and eliminates the need for any pressurization system redesign except for the 3 CS-156 SRM configuration using 90°F propellant temperature. An increase in the pressurization system lockup pressure of 7 psi is required for that configuration in addition to the proposed reduction in NPSH requirement. Further detail on the NPSH problem is presented in Chapter VII.

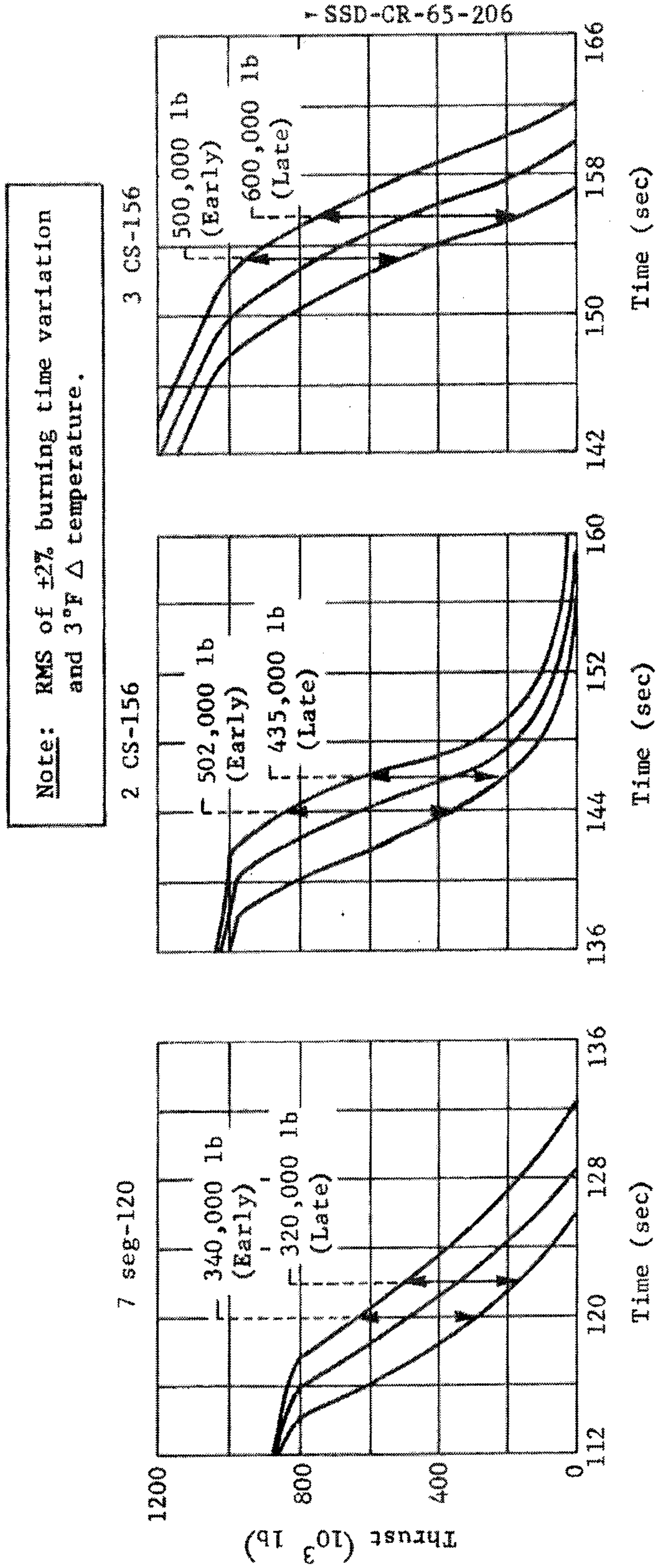


Fig. V-2 SRM Tailoff

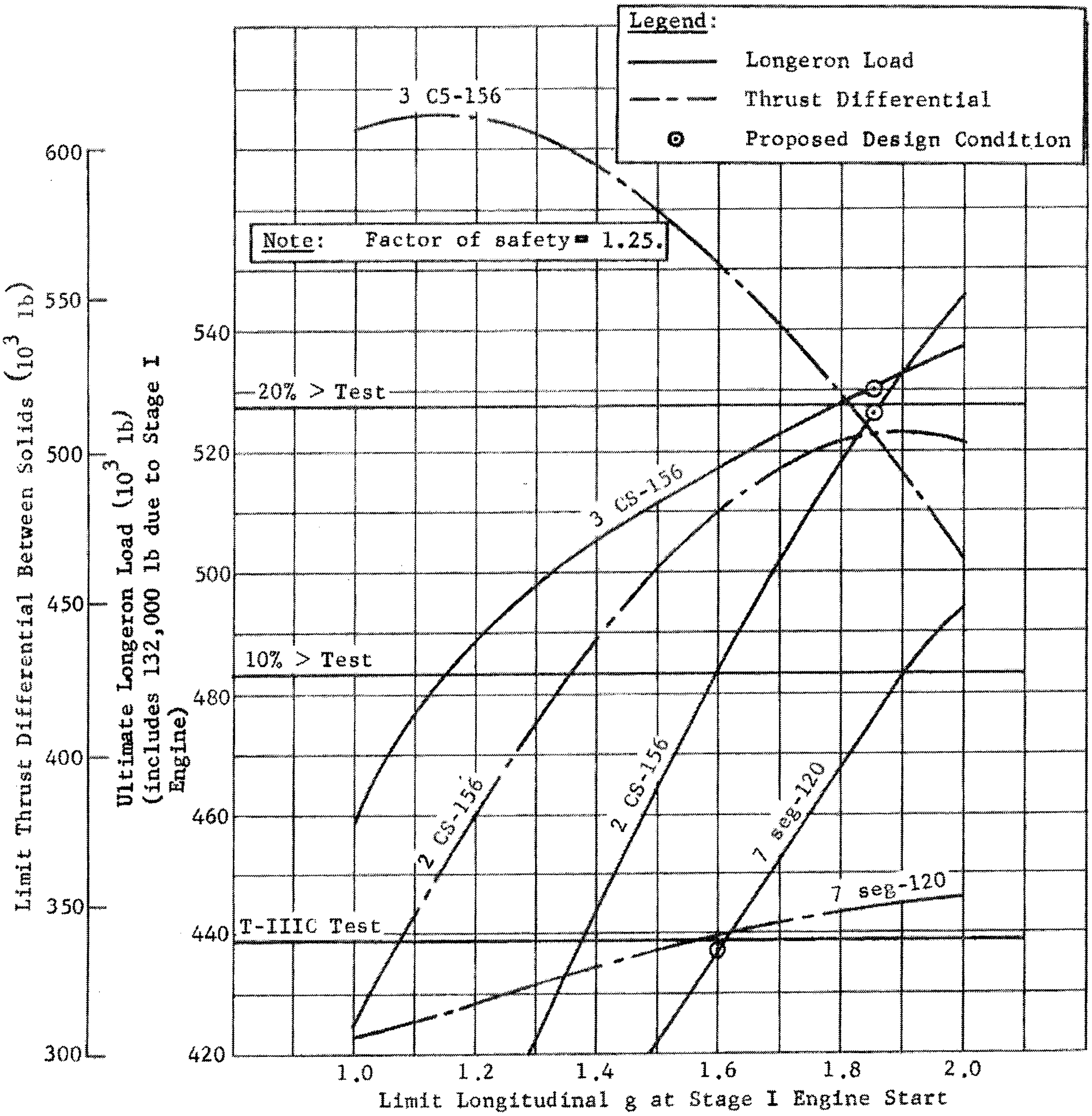


Fig. V-3 Stage I Ultimate Longeron Load (Vehicle Station 1224) at Stage I Engine Start

E. DIFFERENTIAL SRM THRUST TAILOFF

The SRM tailoff curves shown in Fig. V-2 produce several important effects depending primarily on the time required to reach zero thrust and on the differential thrust between them. The effect of differential thrust on structural loads has already been discussed in Section D of this chapter. Differential thrust also results in decreased control authority and necessitates Stage I ignition to augment control available from the SRM TVC system. In this study, a TVC system is assumed in each case to have side force characteristics within the Titan IIIC state of the art. Figure V-4 shows vehicle yaw attitude response during SRM tailoff to be controllable and not unlike what is presently expected on Titan IIIC.

The time required for SRM thrust to decay to essentially zero governs the time at which separation can occur. The 3 CS-156 SRM has the steepest thrust decay slope and earliest time to zero thrust. The staging sequence for the 3 CS-156 SRM is therefore the shortest from g-sensing to separation.

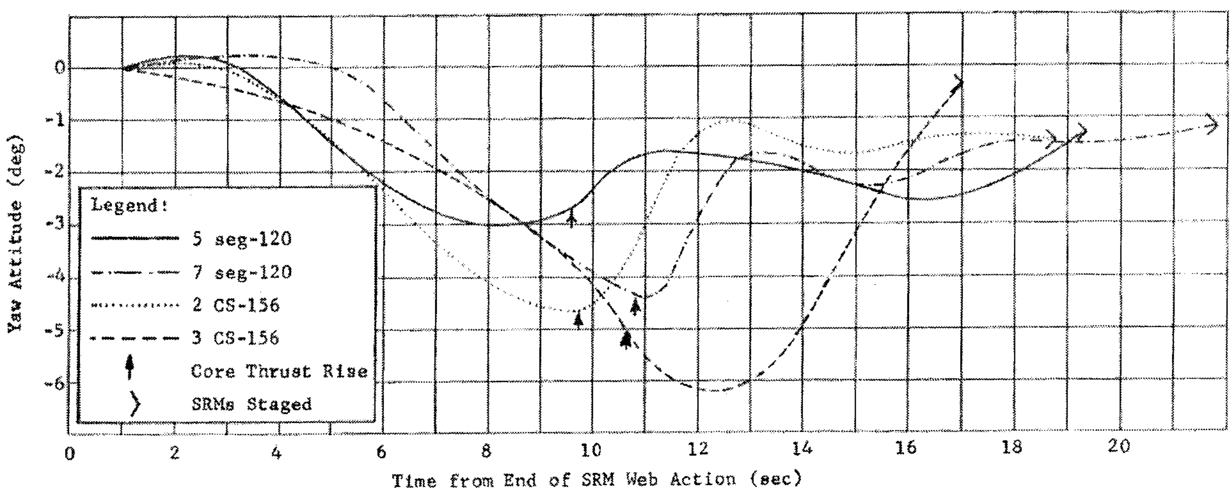


Fig. V-4 Control Authority during SRM Tailoff (Minimum/Maximum SRM Tailoff, Late Core Start)

F. STAGE 0/I SEPARATION

To demonstrate satisfactory clearance during SRM separation, different combinations of staging rockets are considered until the proper separation motion is achieved. Plots of relative motion in the staging plane, based on the conditions and configurations shown in Table V-4, are presented in Fig. V-5 thru V-7.

Table V-4 Summary of Separation Plots

Configuration	SRM Tailoff	Staging Rockets		Sequence	Figure
		Forward	Aft		
7 seg-120	Nominal	4*	4*	Early	V-5(b)
	Nominal	5	5	Early	V-5(a)
	Nominal	3	4	Late	V-5(d)
	Nominal	4	5	Late	V-5(c)
	Minimum	4	3	Early	V-5(f)
	Maximum	4	3	Early	V-5(e)
2 CS-156	Minimum	5*	4*	Early	V-6(b)
	Maximum	5	4	Early	V-6(a)
	Nominal	4	4	Early	V-6(d)
	Nominal	5	5	Early	V-6(c)
	Minimum	5	3	Early	V-6(f)
	Maximum	5	3	Early	V-6(e)
3 CS-156	Minimum	5*	4*	Early	V-7(b)
	Maximum	5	4	Early	V-7(a)
	Nominal	4	4	Early	V-7(d)
	Nominal	5	5	Early	V-7(c)
	Minimum	5	3	Early	V-7(f)
	Maximum	5	3	Early	V-7(e)

*Recommended number of staging rockets for satisfactory separation.

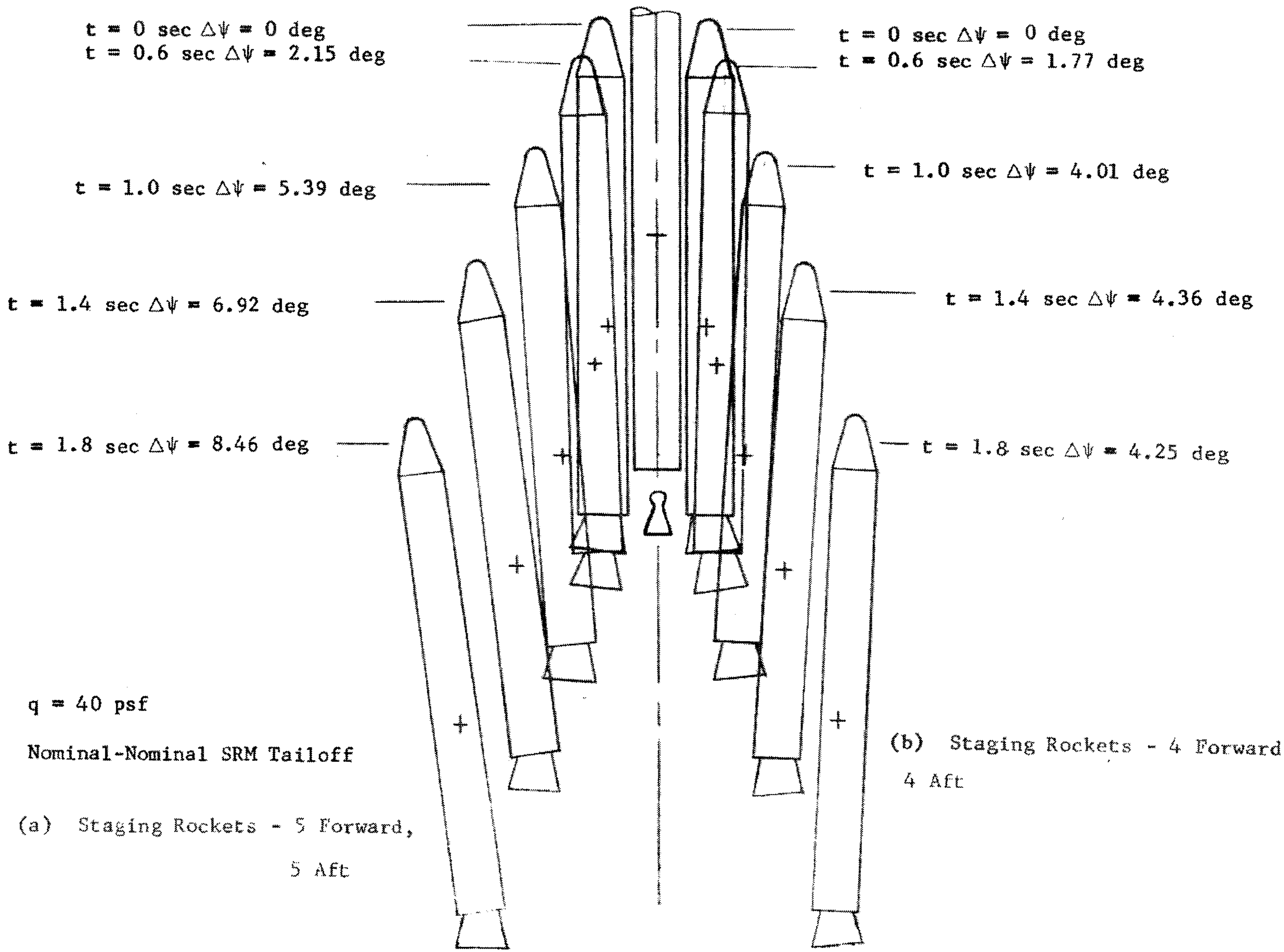


Fig. V-5 7 seg-120 SRM Staging

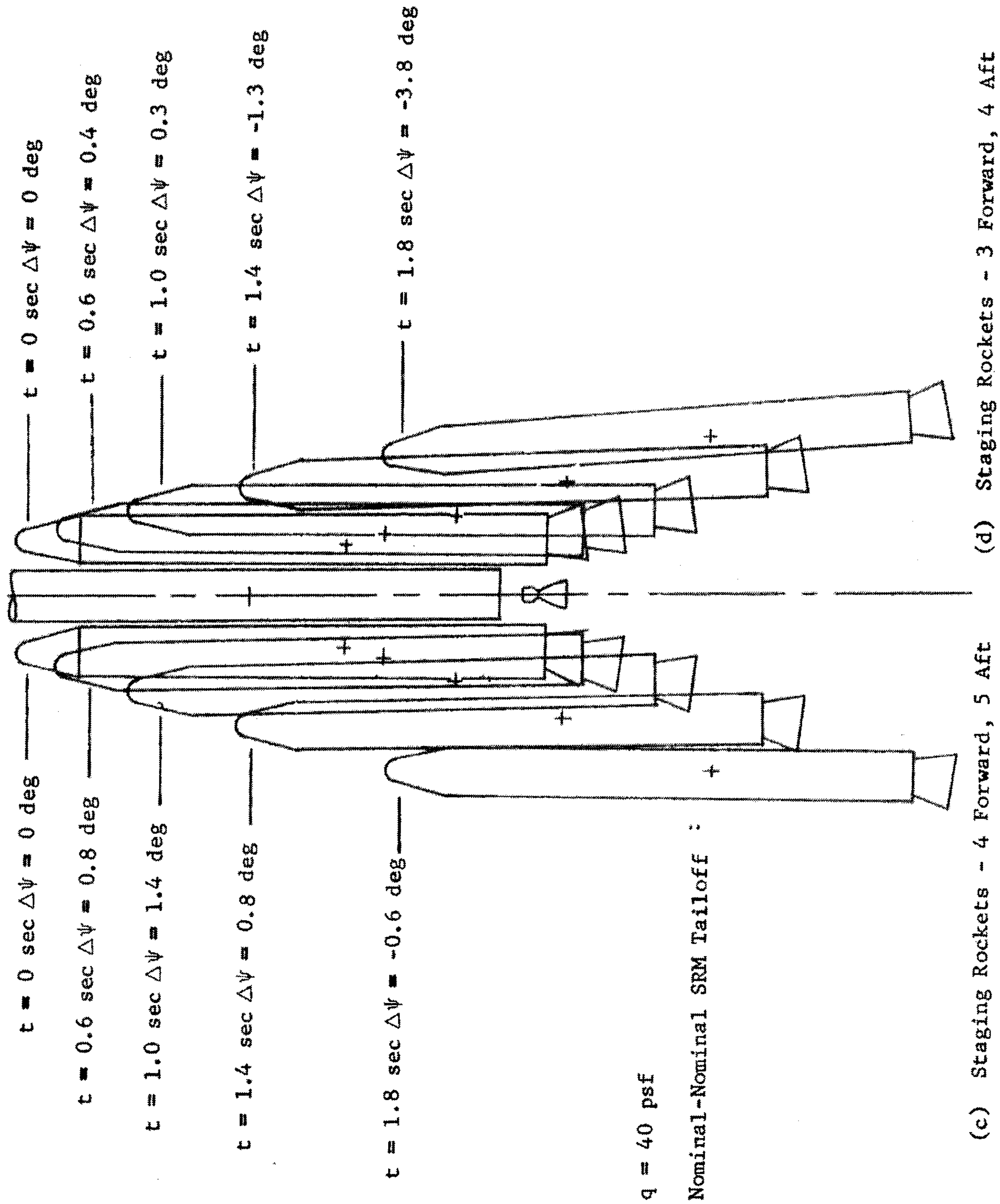
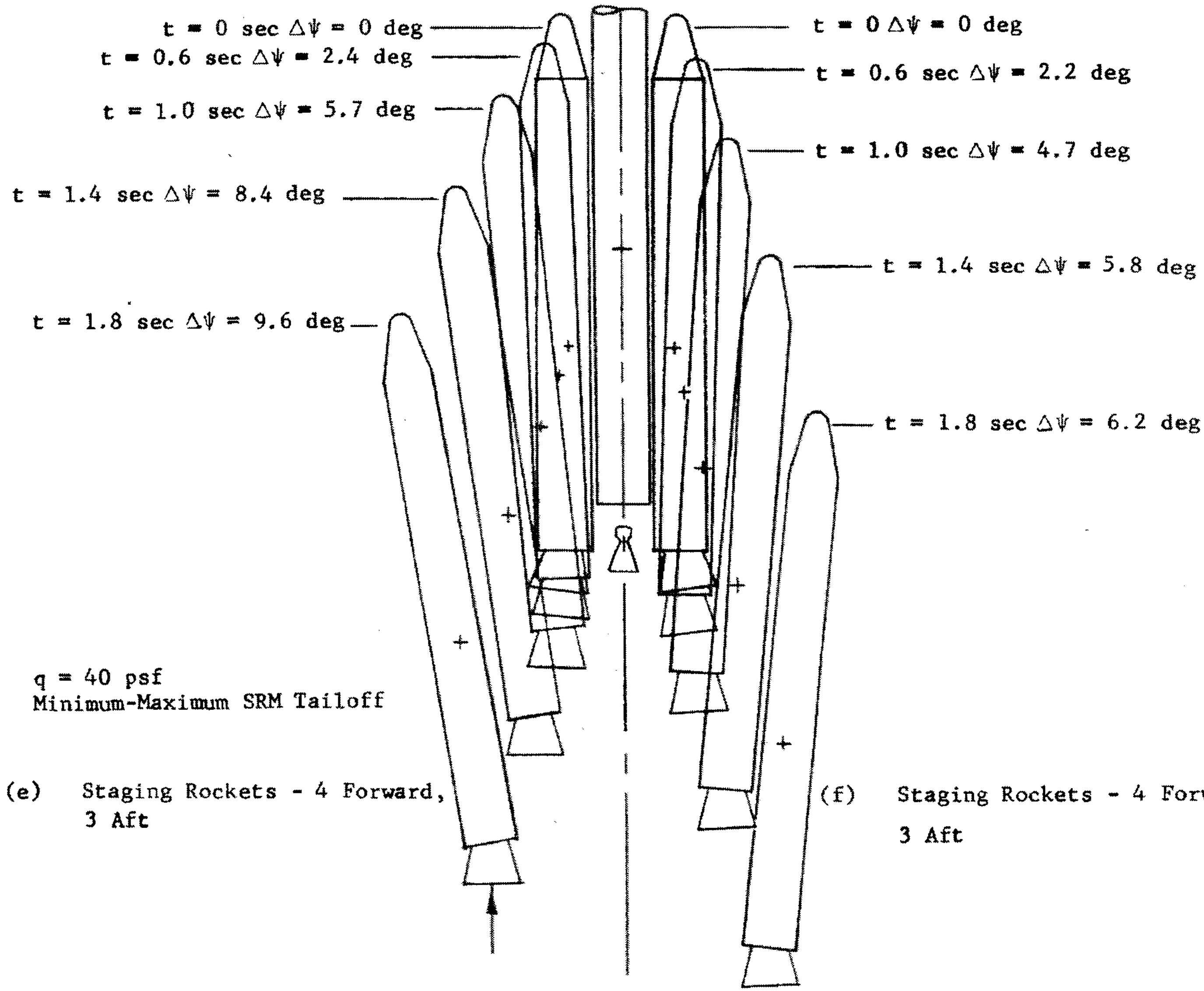

t = 0 sec Δψ = 0 deg
t = 0.6 sec Δψ = 0.8 deg
t = 1.0 sec Δψ = 1.4 deg
t = 1.4 sec Δψ = 0.8 deg
t = 1.8 sec Δψ = -0.6 deg
q = 40 psf
Nominal-Nominal SRM Tailoff :
(c) Staging Rockets - 4 Forward, 5 Aft
t = 0 sec Δψ = 0 deg
t = 0.6 sec Δψ = 0.4 deg
t = 1.0 sec Δψ = 0.3 deg
t = 1.4 sec Δψ = -1.3 deg
t = 1.8 sec Δψ = -3.8 deg
(d) Staging Rockets - 3 Forward, 4 Aft

Fig. V-5 (cont)

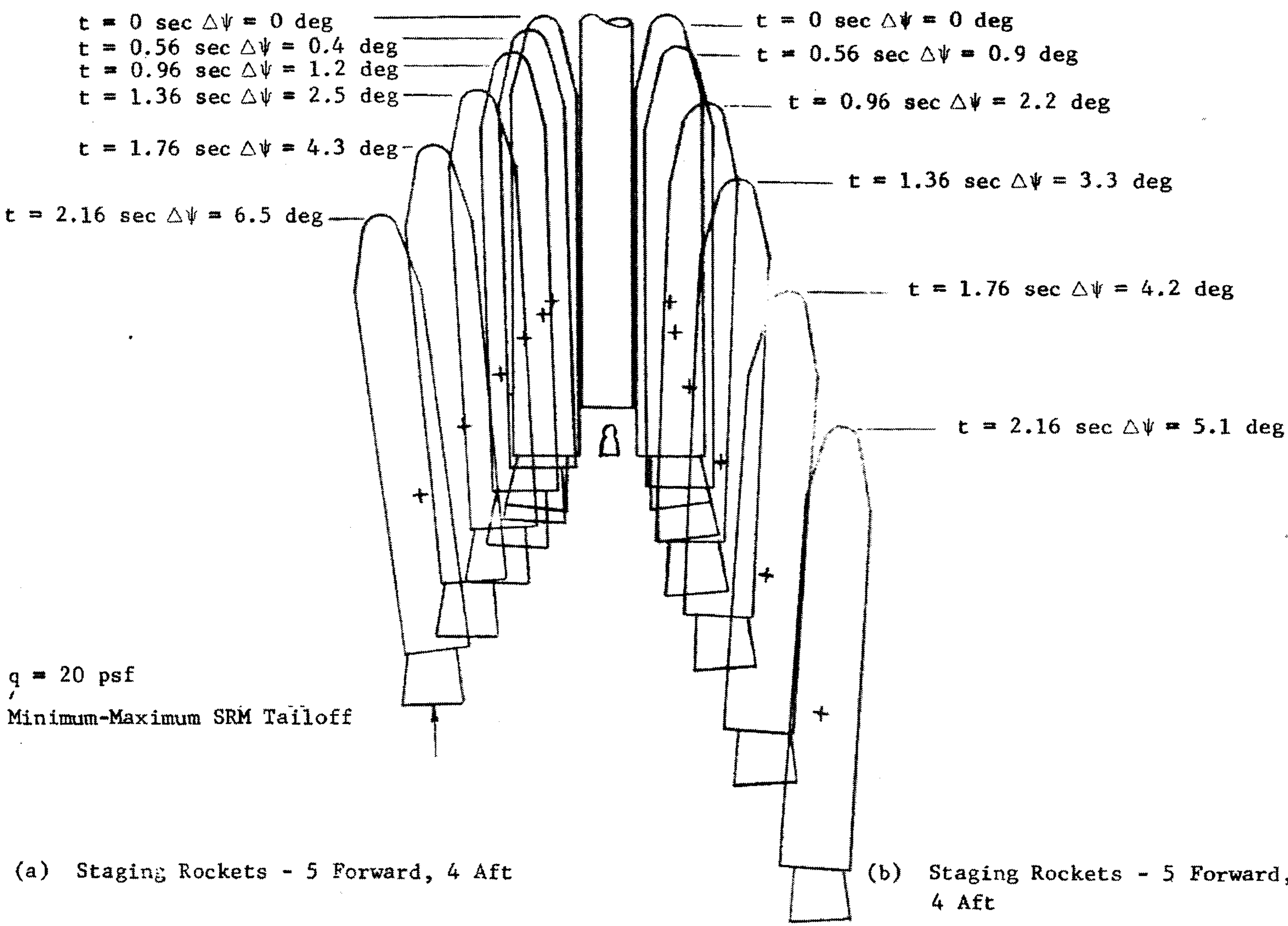


q = 40 psf
Minimum-Maximum SRM Tailoff

(e) Staging Rockets - 4 Forward, 3 Aft

(f) Staging Rockets - 4 Forward, 3 Aft

Fig. V-5 (concl)



(a) Staging Rockets - 5 Forward, 4 Aft

(b) Staging Rockets - 5 Forward, 4 Aft

Fig. V-6 2 CS-156 SRM Staging

t = 0 sec $\Delta\psi$ = 0 deg

t = 0.64 sec $\Delta\psi$ = 0.4 deg

t = 1.04 sec $\Delta\psi$ = 1.0 deg

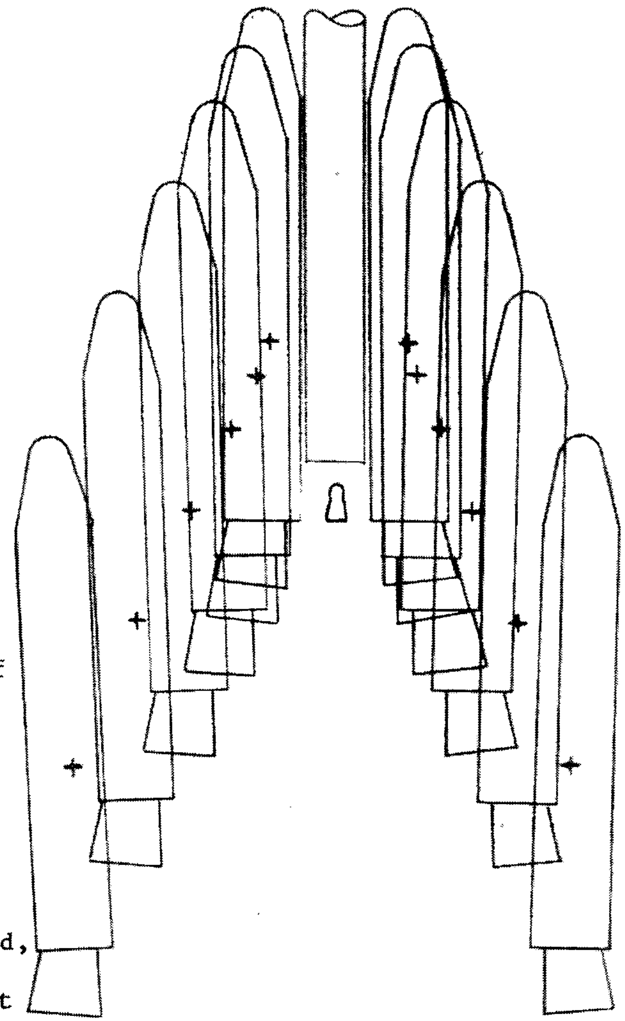
t = 1.44 sec $\Delta\psi$ = 1.4 deg

t = 1.84 sec $\Delta\psi$ = 1.9 deg

t = 2.24 sec $\Delta\psi$ = 2.4 deg

q = 20 psf

Nominal-Nominal SRM Tailoff



t = 0 sec $\Delta\psi$ = 0 deg

t = 0.64 sec $\Delta\psi$ = 0.6 deg

t = 1.04 sec $\Delta\psi$ = 1.2 deg

t = 1.44 sec $\Delta\psi$ = 1.5 deg

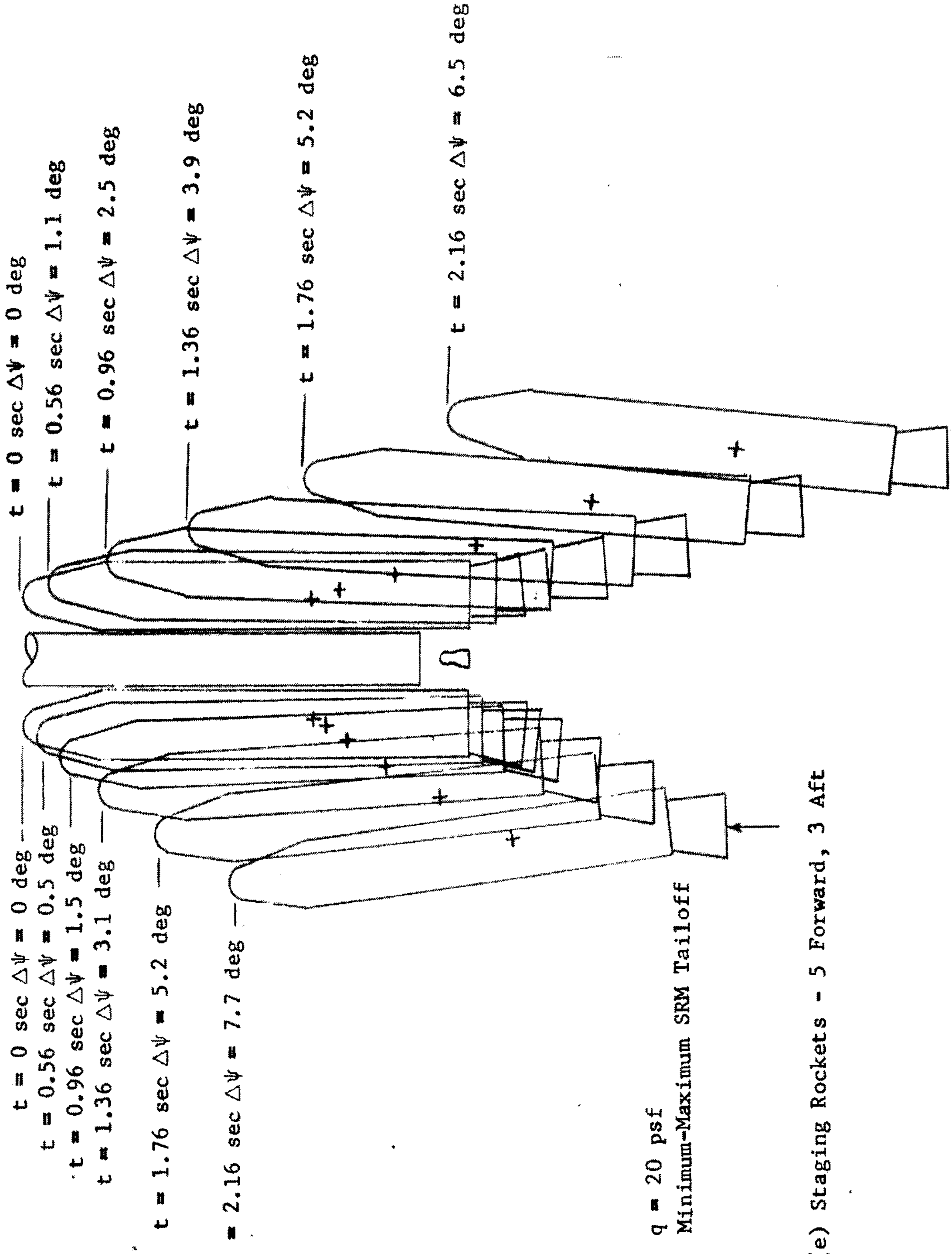
t = 1.84 sec $\Delta\psi$ = 1.7 deg

t = 2.24 sec $\Delta\psi$ = 1.8 deg

(c) Staging Rockets - 5 Forward, 5 Aft

(d) Staging Rockets - 4 Forward, 4 Aft

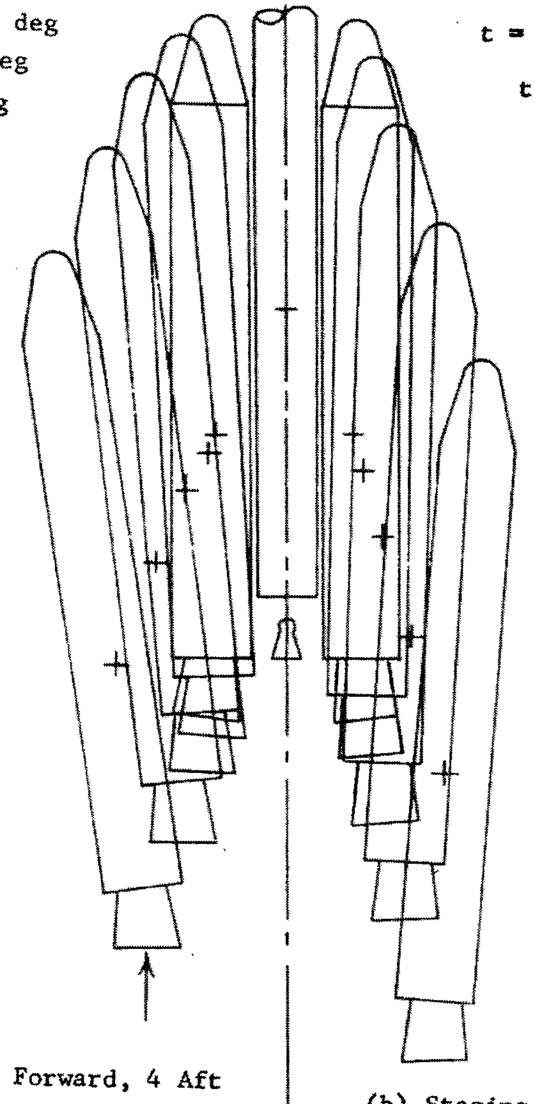
Fig. V-6 (cont)



(e) Staging Rockets - 5 Forward, 3 Aft

(f) Staging Rockets - 5 Forward, 3 Aft

Fig. V-6 (concl)



(a) Staging Rockets - 5 Forward, 4 Aft

(b) Staging Rockets - 5 Forward, 4 Aft

Fig. V-7 3 CS-156 SRM Staging

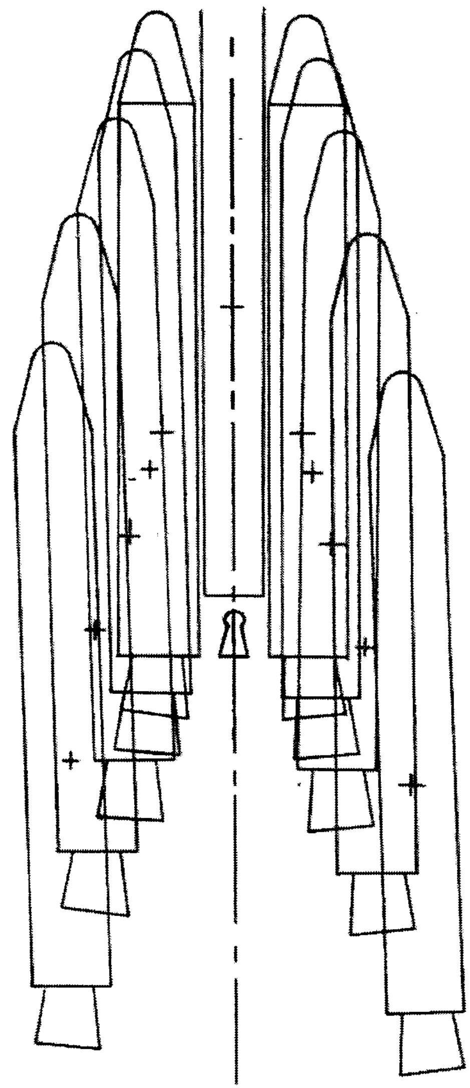
t = 0 sec $\Delta\psi$ = 0 deg

t = 0.6 sec $\Delta\psi$ = 0.70 deg

t = 1.0 sec $\Delta\psi$ = 1.36 deg

t = 1.4 sec $\Delta\psi$ = 1.48 deg

t = 1.8 sec $\Delta\psi$ = 1.37 deg



t = 0 sec $\Delta\psi$ = 0 deg

t = 0.6 sec $\Delta\psi$ = 0.43 deg

t = 1.0 sec $\Delta\psi$ = 0.49 deg

t = 1.4 sec $\Delta\psi$ = -0.30 deg

t = 1.8 sec $\Delta\psi$ = -1.47 deg

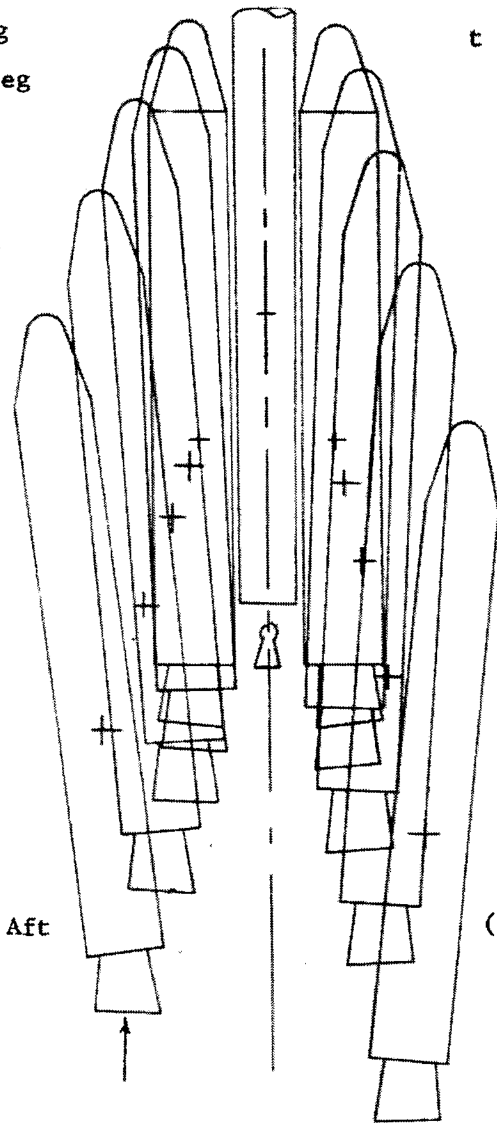
q = 5 psf

Nominal-Nominal SRM Tailoff

(c) Staging Rockets - 5 Forward,
5 Aft

(d) Staging Rockets - 4 Forward, VD
4 Aft T

Fig. V-7 (cont)



t = 0 sec Δψ = 0 deg
t = 0.6 sec Δψ = 1.66 deg
t = 1.0 sec Δψ = 3.86 deg
t = 1.4 sec Δψ = 5.85 deg
t = 1.8 sec Δψ = 7.59 deg

q = 5 psf
Minimum-Maximum SRM Tailoff

(e) Staging Rockets - 5 Forward, 3 Aft

t = 0 sec Δψ = 0 deg
t = 0.6 sec Δψ = 1.47 deg
t = 1.0 sec Δψ = 3.17 deg
t = 1.4 sec Δψ = 4.70 deg
t = 1.8 sec Δψ = 6.58 deg

(f) Staging Rockets - 5 Forward, 3 Aft

Fig. V-7 (concl)

1. 7 seg-120

Figure V-5(a) shows the 7 seg-120 nominal motion with five staging rockets forward and aft, and Figure V-5(b) shows four staging rockets forward and aft. Both configurations give a nose-out motion similar to the motion of the solids of Titan IIIC. The relative motion of the 4 forward/4 aft (Titan IIIC configuration) is considered adequate and is therefore recommended. The additional separation achieved by the 5 forward/5 aft configuration is not considered to be sufficient to warrant two additional staging rockets per SRM.

Figures V-5(c) and V-5(d) show the relative motion with one staging rocket failure forward. Satisfactory separation is demonstrated in the presence of adverse plume loads.

Figures V-5(e) and V-5(f) show the relative motion with one staging rocket failure aft and maximum residual SRM thrust on the left SRM. Residual thrust rotates the aft end of the SRM toward the core at separation and combines adversely with an aft staging rocket failure. Adequate separation is still achieved.

2. 2 CS-156

Figures V-6(a) and V-6(b) show that a 5 forward/4 aft balance of staging rockets is necessary to achieve the desired nose-out motion for this configuration. Figures V-6(d) thru V-6(f) show one staging rocket failure forward or aft still result in satisfactory clearance.

3. 3 CS-156

A staging rocket configuration of 5 forward/4 aft [Fig. V-7(a) and V-7(b)] is desirable and is recommended for this configuration also. Again, staging rocket failures forward or aft combined with adverse conditions in Fig. V-7(c) thru V-7(f) demonstrate satisfactory clearance.

In each of the SRM separation regimes, the importance of core plume force impingement on the SRM is apparent. More exact determination of the plume effect from wind tunnel tests of the selected MOL booster may require minor readjustment of the staging rocket balance.

VI. SOLID ROCKET MOTORS

As a part of the performance improvement study, Martin Company was directed to obtain information from solid rocket motor (SRM) manufacturers and determine the optimum SRM design for the six vehicle configurations under consideration. The six configurations are defined as the core vehicle with and without transtage with each of three SRM options -- 7 segment, 120-in., SRMs (7 seg-120), 2 center segment, 156-in. SRMs (2 CS-156), and 3 center segment, 156-in. SRMs (3 CS-156). If insufficient data were received from the SRM manufacturers, Martin was to generate the data required to complete the study.

The basic objective of the Titan III/MOL SRMs is to place the heaviest payload possible in an 80-n-mi polar orbit with launch from the Western Test Range (WTR) and within the constraints and ground rules specified in the Titan III/MOL Statement of Work. The objective of this report is to present the recommended SRM configurations and results of analyses leading to those recommendations.

The following SRM manufacturers supplied SRM design data for use in the study:

1) United Technology Center (UTC) supplied designs for the 7 seg-120 and 3 CS-156 SRMs;

2) Lockheed Propulsion Company (LPC) supplied designs for the 2 CS-156 and 3 CS-156 SRMs;

3) Thiokol Chemical Corporation (TCC) supplied a design for the 2 CS-156 SRM;

4) Aerojet-General Corporation (AGC) supplied a design for the 2 CS-156 SRM.

The following ground rules supplied to the SRM manufacturers at the start of the MOL 60-day study (2 July 1965) were intended to specify only general constraints to allow the SRM manufacturers the most latitude possible in optimizing the SRM system design:

1) SRM attach locations,

   a) Aft - Station 1226 (all motors),

   b) Forward - Approximately Station 504 (2 CS-156),

   c) Forward - Approximately Station 250 (7 seg-120 and 3 CS-156);

2) Use proven state of the art for SRM design - Class 2 propellant, existing case technology;

3) Specific design optimization to be established by the SRM manufacturer based on the requirements below,

   a) $(T/W)_{liftoff} \geqq 1.6$ g for 5 sec (1.8 g is desired),

   b) $q_{max} \leqq 900$ psf nominal ($+3\sigma \leqq 1000$ psf),

   c) $[(\text{Thrust} - \text{Drag})/\text{Weight}]_{max} \leqq 3.2$ g after 100 sec burning time,

   d) Tailoff,

      Thrust differential between two motors not to exceed 280,000 lb_f at 1.7 g,

      Rate of thrust decay, 10%/sec of web burnout vacuum thrust,

   e) Thrust vector control (TVC) system,

      Type - Not specified,

      Control capability - ±8% vector deflection,

      Slew rate - 30 deg/sec maximum, 10 deg/sec minimum,

   The TVC requirements were adjusted on 30 July to reflect ±6.1% vector deflection required for the 2 CS-156 and 3 CS-156 SRMs,

   f) Nozzle,

      Area ratio - Not specified,

      Nozzle/motor cant angle - 6 deg,

      Exit diameter - 135.8 in. maximum,

      Nozzle exit plane station - Station 1450 maximum (to ₵ of nozzle exit),

      SRM/pad support - Stations 1326 to 1348,

g) Mission success/crew safety,

Thrust termination system is required (desired thrust termination curves supplied the contractors on 30 July 1965 are shown in Section C of this chapter),

Redundancy is to be provided wherever practical in the motor design,

Safety factors ≧ 1.25,

Burnthrough sensors,

3-sec minimum warning time from 0 to 40 sec burning time and from 90 sec to burnout,

6-sec minimum warning time from 40 to 90 sec burning time,

h) Ignition reliability - Maximum 200 ppm failures,

i) Range safety destruct system (RSDS) is required,

j) Inadvertent separation destruct system (ISDS) is required.

Adequate information concerning the core performance, weights, and time sequence were also supplied to the SRM contractors.

UTC was also working on a funded study for Air Force/Aerospace that supplied some additional constraints for the 7 seg-120 SRM. The most important of those contraints are:

1) Meet performance constraints with minimum changes on the Titan IIIC 5 segment motor;

2) Use existing Titan IIIC TVC system;

3) Use existing Titan IIIC thrust termination ports;

4) Use existing Titan IIIC case design.

Some of these additional constraints were reflected in the designs submitted for the 2 CS-156 and 3 CS-156 SRMs.

A. REFERENCE CONFIGURATIONS

This section describes the SRM configurations used for the vehicle loads analysis and TVC system analysis, and the SRM configurations selected for SSD-CR-65-204A* and SSD-CR-65-204B.† Differences between the configurations result from the fact that the loads and TVC analyses were conducted on preliminary SRM data. The SRM criteria documents contain requirements and data resulting from the entire 60-day MOL study and reflect the preferred design. The general SRM configuration is shown in Fig. VI-1. Basic components are as shown and apply to all motor designs received. Each configuration contains the following common systems:

1) A liquid inject TVC (LITVC) system using nitrogen tetroxide (N_2O_4) as the injectant fluid and gaseous nitrogen N_2 as the pressurizing gas. The N_2O_4 is injected into the SRM nozzle exhaust stream through 24 injectant valves. Each configuration makes maximum use of existing Titan IIIC TVC system components. TVC injectant load and flow requirements are as specified for each configuration. Since the TVC sizing analyses were not completed in time to rerun all cases for performance evaluation, two values are listed for TVC system weights. The reference configuration values are those submitted by the SRM contractors, and the required values are those resulting from the TVC sizing studies. Additional analyses are required to show final adjustments in performance as a function of the final system weights. TVC system weights obtainable are less than those initially used, so payload performance will improve slightly when the final analysis is complete.

2) A thrust-termination system composed of two blowout ports is located in the head end of the SRM. Thrust-termination stacks through the forward fairing are oriented in the pitch plane and at a 45-deg angle to the SRM centerline. Ports are opened on command by initiating shaped charges located around the port circumference.

3) A separation system consisting primarily of small staging solid rocket motors located forward and aft on the SRM as shown in Fig. VI-1. The number of staging motors required for each configuration is discussed in Section C of this chapter.

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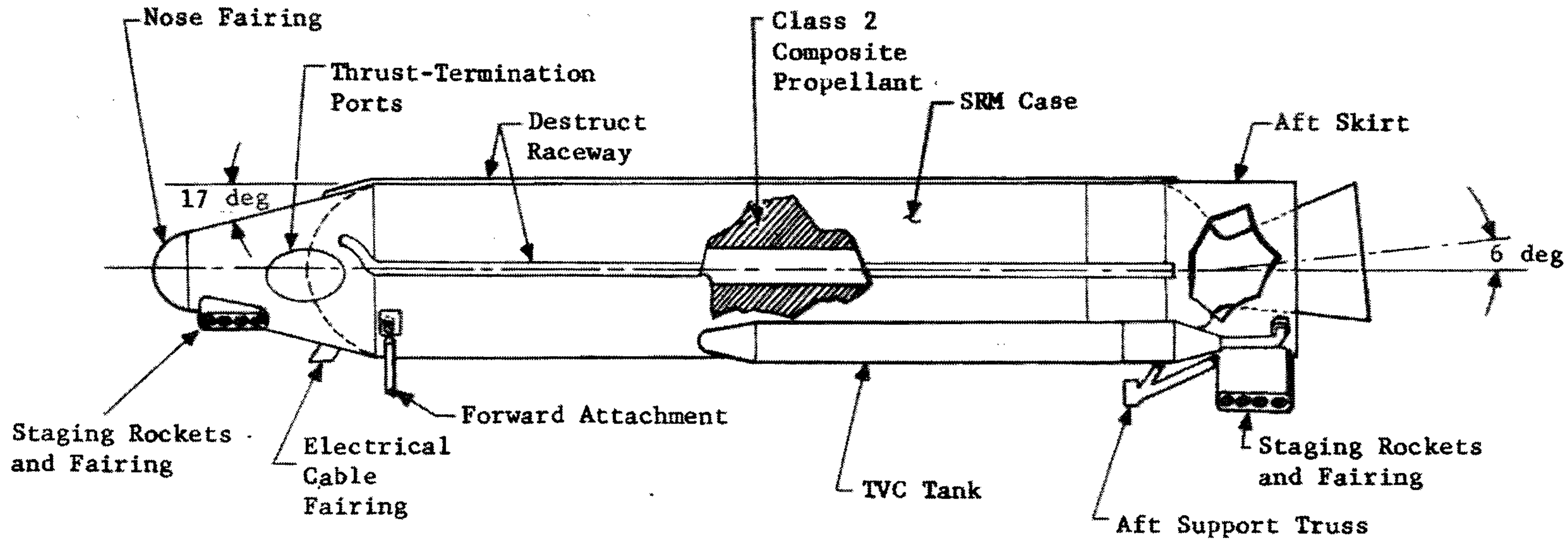
*Titan III/MOL 156-Inch Solid Rocket Motor Design Constraints (Two Center Segments). SSD-CR-65-204A. Martin Company, Denver, Colorado, August 1965.

†Titan III/MOL 156-Inch Solid Rocket Motor Design Constraints (Three Center Segments). SSD-CR-65-204B. Martin Company, Denver, Colorado, August 1965.



<u>Typical Configuration</u>

Parameter	7 seg-120	2 CS-156	3 CS-156
Overall Length (in.)	1311.5	1146.5	1410.5
Nozzle Type - Conical	0% Submerged $\epsilon = 9.6:1$	19% Submerged $\epsilon = 10.2:1$	17% Submerged $\epsilon = 8.0:1$
Case Material - Steel	D6-AC	Maraging	Maraging
Total Weight (lb)	1,364,626	1,948,560	2,537,100
Overall Mass Fraction	0.864	0.881	0.888
Total Vacuum Impulse (lb-sec)	308×10^6	447×10^6	588×10^6

Fig. VI-1 Stage 0 Design (Loads Reference Configuration)

1. 7 seg-120

The 7 seg-120 SRM reference configuration is a modification of the Titan IIIC 5 seg-120 SRM. The reference configuration motor physical characteristics are shown in Fig. VI-2 and Table VI-1. Motor performance data are shown in Table VI-2 and Fig. VI-3. TVC side force characteristics, obtainable thrust vector deflection, slew rates, and related performance characteristics are defined in Section C of this chapter.

The nozzle for the 7 seg-120 configuration is similar to the external type used on the Titan IIIC 5 segment SRM. The throat diameter has been increased and the nozzle lengthened to improve motor performance.

Tailoff thrust-time data used for the staging analysis are defined in Section C of this chapter. The detailed tailoff and burning time tolerance discussion in Section B of this chapter defines the analysis used to obtain these tailoff characteristics.

The 7 seg-120 grain configuration consisting of an 8-point star head end, 7 cone frustum center segments with partial inhibiting, and an 8-point star aft end is shown in Fig. VI-4.

2. 2 CS-156

The 2 CS-156 SRM reference configuration physical characteristics are shown in Fig. VI-5 and Table VI-1. Motor performance and ballistic data are shown in Table VI-2. The motor thrust-time history is plotted on Fig. VI-3. The TVC side force characteristics, thrust vector deflection, slew rates, and related performance characteristics are defined in Section C of this chapter. The nozzle used for the 2 CS-156 SRM is submerged approximately 19% of its length into the SRM chamber. The reasons for selecting the submerged nozzle are discussed in Section B of this chapter. The tailoff thrust-time curve used for the staging analysis is included in Section C of this chapter. This curve has been adjusted to reflect Martin Company analysis as defined in Fig. VI-3.

The 2 CS-156 grain design consisting of a 10-point star in the head end, 2 conical frustum (uninhibited) center segments, and a multiple conical frustum aft-end segment is shown in Fig. VI-6.

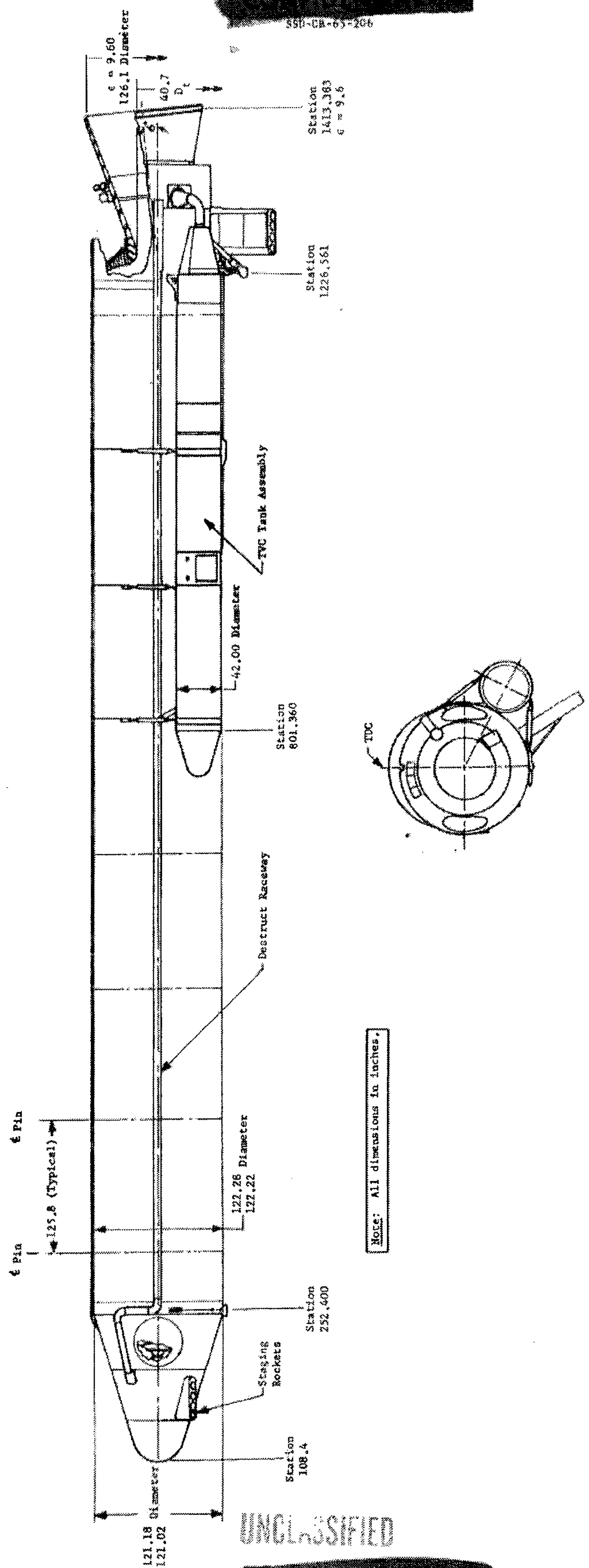


Fig. VI-2 7 seg-120 SRM Reference Configuration

[redacted]

Table VI-1 SRM Reference Configuration Characteristics

	7 seg-120	2 CS-156	3 CS-156
Case	(44,487)	(62,260)	(82,460)
Forward Closure	4,238		
Segments	37,170	61,076	81,157
Aft Closure	3,119		
Miscellaneous	350	1,184	1,303
Insulation + Liner	(13,330)	(14,300)	(16,750)
Forward Closure	1,145		
Segments	9,625		
Aft Closure	2,560		
Propellant	(571,324)	840,000	1,104,450
Forward Closure	35,110		
Segments	518,224		
Aft Closure	17,990		
Nozzle	(9,574)	10,750	11,250
Throat Assembly	1,563		
Exit Cone	8,011		
TVC System	(26,495)	(29,500)	(35,900)
Inerts	10,257	15,500	17,100
Pressurant	1,488		
Injectant	(14,750)	14,000	18,800
Usable	12,180		
Additional Fluid	1,482		
Fill Lines + Manifold	1,088		
Thrust Termination	(1,311)	In Case	In Case
Stacks	660		
Covers	541		
Attach Hardware	60		
Mechanism	50		
Destruct System	174	50	50
Igniter	(376)	(1,000)	(1,000)
Inerts	288	825	825
Charge	88	175	175
Hydraulic System	221		
Electrical System	722		
Instrumentation	490		
Separation System	(1,289)	1,250	1,250
Motors	696		
Circuitry	33		
Support Hardware	560		
Nose Fairing	1,222	1,500	1,500
Aft Skirt + Heat Shield	8,756	12,700	13,400
Forward Ring	1,352		
External Insulation	672		
Miscellaneous	128	970	1,370
Total	682,313	974,280	1,268,520
Expendables			
Nozzle			
Internal Insulation	5,667	7,260	8,510
Propellant	571,324	840,000	1,104,450
Igniter	88	175	175
TVC Fluid	12,180	14,000	18,800
External Insulation	105		
Miscellaneous			
Total	589,364	861,435	1,131,935
Mass Fraction	0.864	0.881	0.888
Motor Mass Fraction (SRM + TVC)	0.882	0.896	0.902
Motor Length			
Nozzle Data			
Throat Area (in.^{2})	1301	1205	1630
Exit Area (in.^{2})	12,490	12,291	13,121
Expansion Ratio	9.6:1	10.2:1	8.05:1
Nozzle Submergence (%)	0	19	17

[redacted]

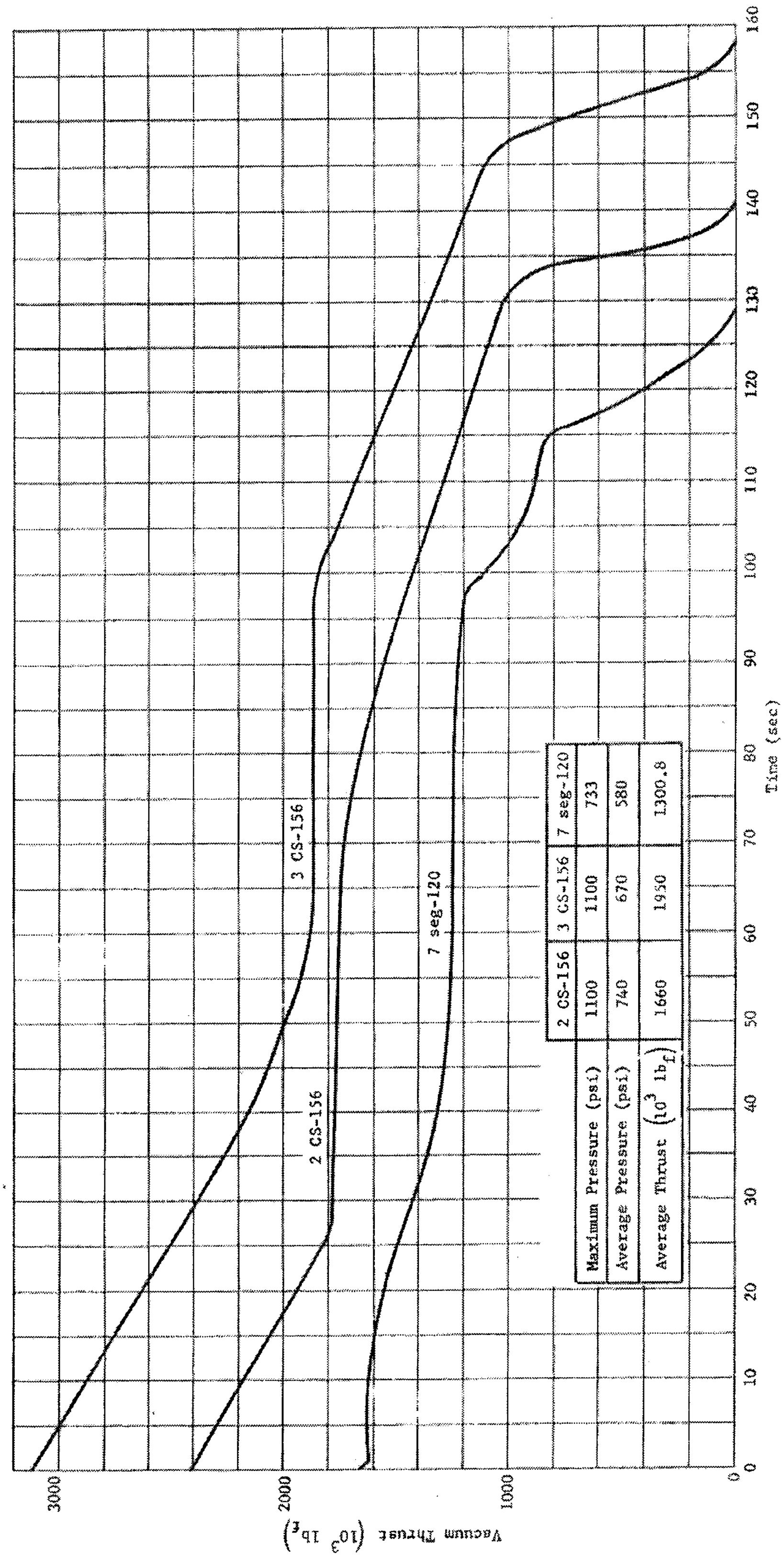
Table VI-2 SRM Performance Characteristics

	Configuration		
	7 seg-120	2 CS-156	3 CS-156
Performance (Nozzle Centerline)			
Maximum Expected Operating Pressure MEOP (psia)	920	1200	1200
Action Time AT (sec)*	124.31	142.0	157
Pressure Burning Time, PBT (sec)†	115.09	132.0	147.4
Average Vacuum Thrust $\left(10^6\ lb_f\right)$‡	1.30	1.659	1.961
Average Aft Pressure (psia)‡	580	740.	670.
Impulse, AT $\left(10^8\ lb_f\text{-sec}\right)$			
Vacuum	1.5388	2.238	2.940
Sea Level			
Impulse, PBT $\left(10^8\ lb_f\text{-sec}\right)$			
Vacuum	1.4971	2.187	2.888
Sea Level	1.2858		
Propellant			
Propellant Weight (lb)	571,324	840,000	1,104,450
Composition	PBAN + AP + AL 16.1%	PBAN + AP + AL 16.1%	PBAN + AP + AL 16.1%
Operating Temperature Range (°F)	40 to 90	40 to 100	40 to 100
Configuration			
Forward Segment	8 Spokes	10-pt Star	10-pt Star
Center Segments	Cyl Perf	Cyl Perf	Cyl Perf
Aft Segments	8 Spokes	Cyl Perf	Cyl Perf
Web Thickness (in.)	35.5	50.0	50.0
Burn Rate, 100 psi, 60°F (in./sec)	0.356	0.41	0.391
Burn Rate Exponent	0.260	0.4	0.40
Specific Impulse, Standard $\left(lb_f\text{-sec}/lb_m\right)$	248	248	248
Temperature Coefficient of Pressure (%/°F)	0.130	0.11	0.11
Ratio of Specific Heats	1.180	1.17	1.17
Density (lb/cu in.)	0.063	0.065	0.065
Characteristic Exhaust Velocity (fps)	5,170	5,200	5,200
Molecular Weight of Exhaust Gas	19.9	19.9	19.9
Molecular Weight of Exhaust Products	26.4	27.5	27.5
Motor Performance (without Transtage)			
Payload	28,306	35,030	42,109
Max Q	913	927	904
Aerodynamic Heating Indicator	96.9×10^6	101.2×10^6	89.3×10^6
$\left(\frac{\text{Thrust - Drag}}{\text{Weight}}\right)$ Max during First 5 sec	1.76	1.86	2.0

*10% P_{max} to 10% P_{max}

†10% P_{max} to Web Burnout

‡Averaged Over PBT



	2 CS-156	3 CS-156	7 seg-120
Maximum Pressure (psi)	1100	1100	733
Average Pressure (psi)	740	670	580
Average Thrust (10^3 lb_f)	1660	1950	1300.8

Fig. VI-3 Thrust vs Time, Reference Configurations

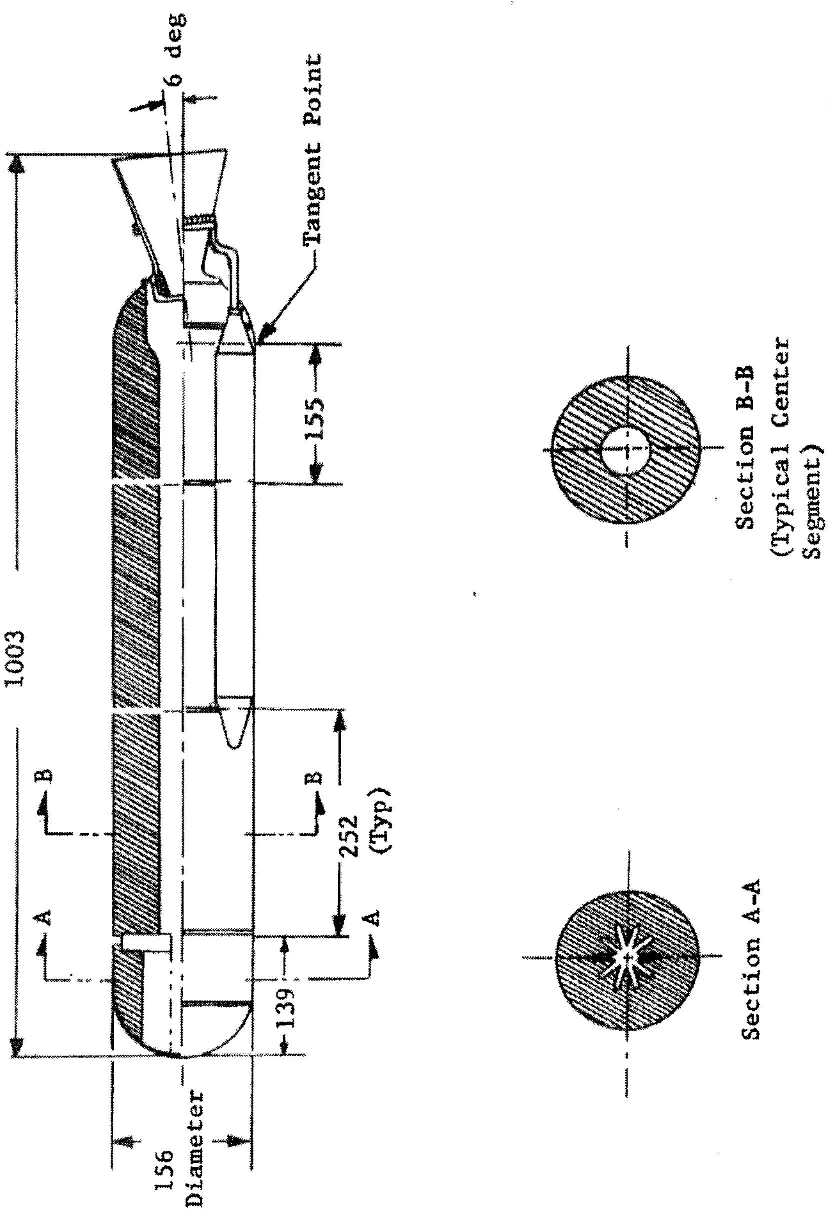


Fig. VI-5 2 CS-156 SRM Reference Configuration

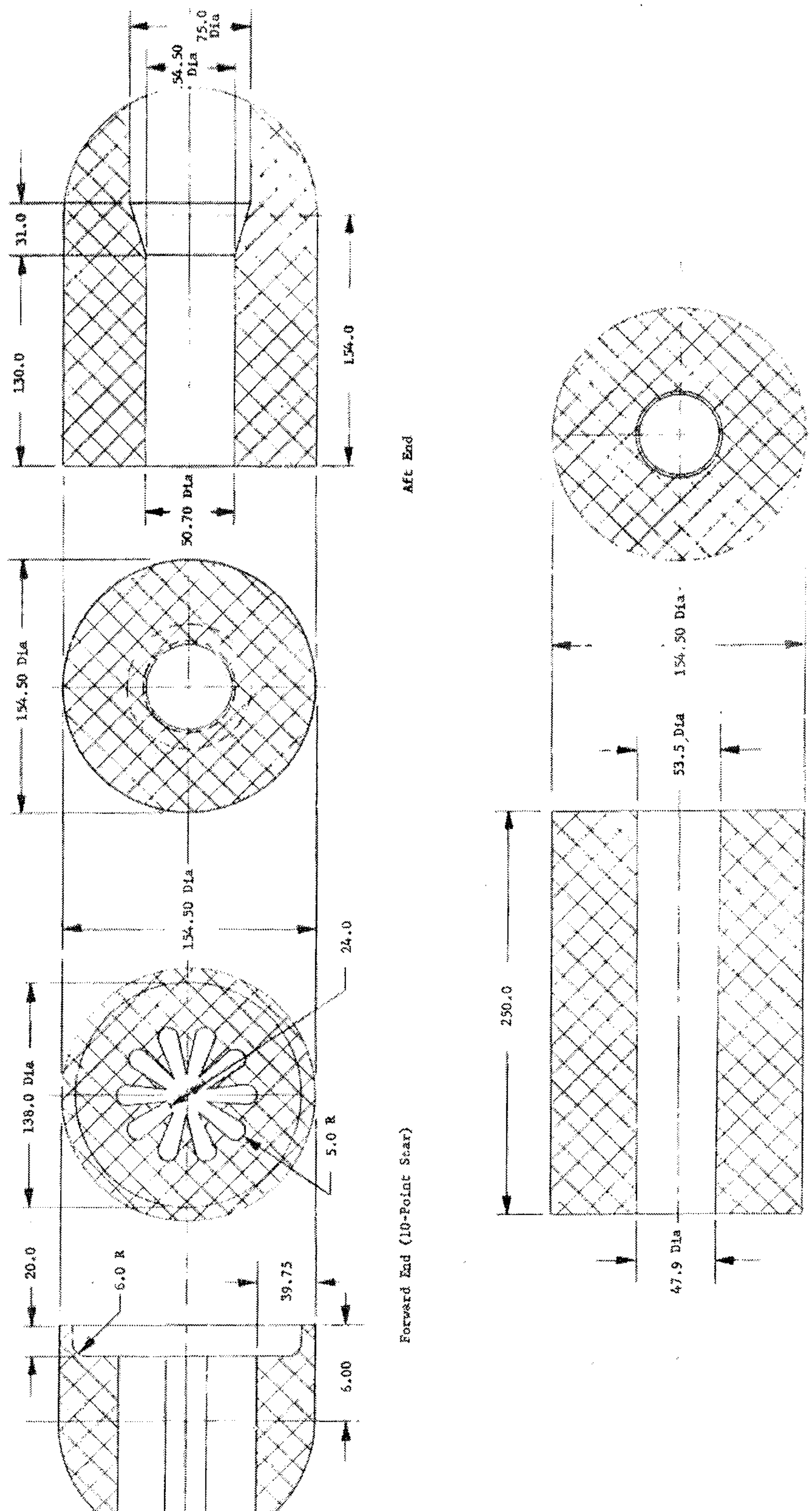


Fig. VI-6 2 CS-156 SRM Grain Design

3. 3 CS-156

The 3 CS-156 SRM reference configuration physical characteristics are shown in Fig. VI-7 and Table VI-1. Motor performance and ballistic data are shown in Table VI-2, and the thrust-time curve is as shown on Fig. VI-3. The TVC side force characteristics, slew rates, and obtainable thrust vector deflection characteristics are defined in Section C of this chapter. The nozzle used for the 3 CS-156 SRM is submerged approximately 17%. Section B of this chapter contains the discussion concerning selection of the submerged nozzle. The tailoff thrust-time data used on the staging analysis are shown in Section C of this chapter, and, as in the case of the 2 CS-156 tailoff, the adjustments made to the data are explained in Section B of this chapter.

The 3 CS-156 grain design consisting of a 10-point head-end star, 3 variable conical center segments (unrestricted), and a variable conical aft segment is shown in Fig. VI-8.

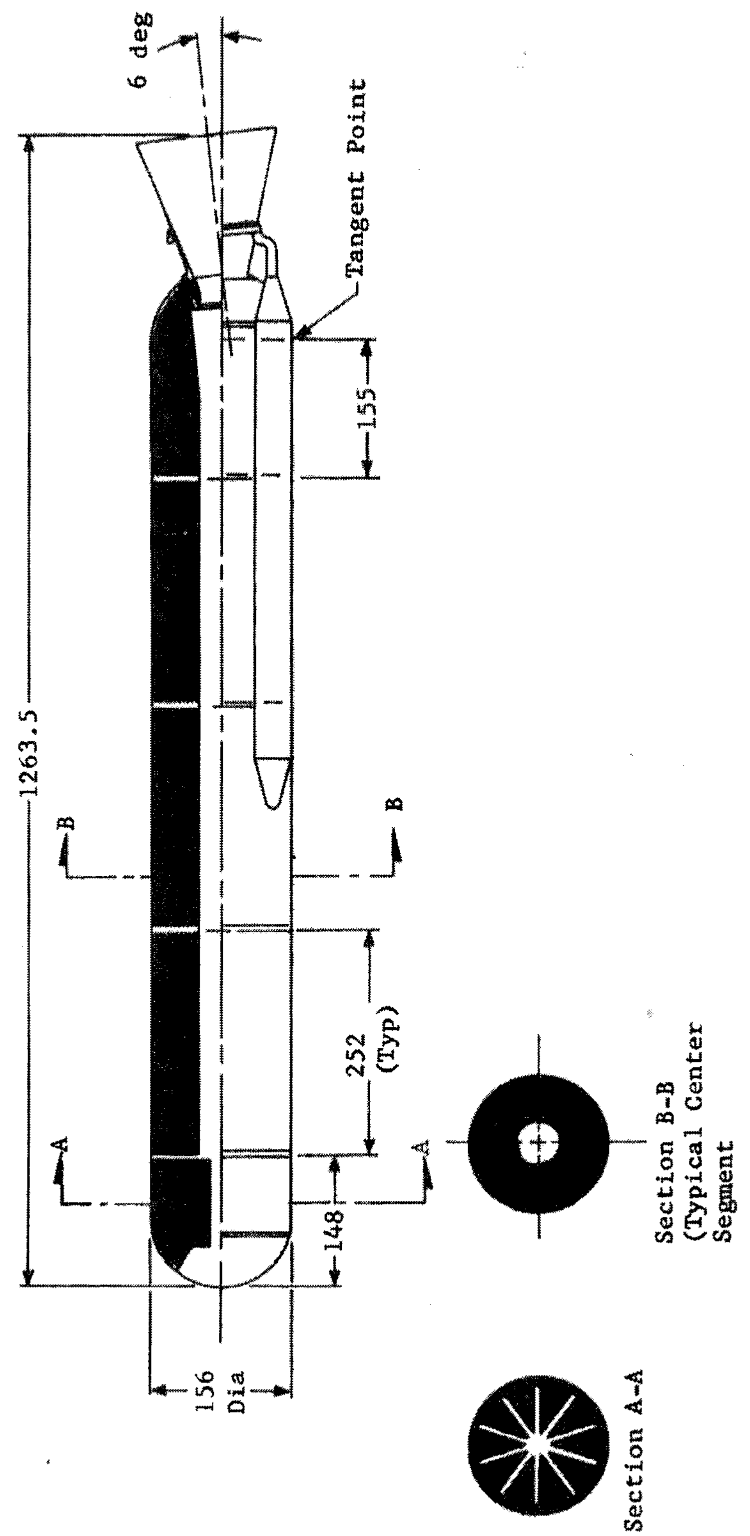


Fig. VI-7 3 CS-156 SRM Reference Configuration

B. SRM CONSTRAINTS REPORT CONFIGURATION

As a result of the 60-day MOL study, documents have been published describing the recommended 2 CS-156 (SSD-CR-65-204A) and 3 CS-156 (SSD-CR-65-204B). Since UTC is designing the 7 seg-120 SRM under an Air Force contract, no constraints document was written for that configuration. However, the 7-segment configuration used in the loads study (see Section A of this chapter) is the 120-in. SRM configuration recommended to be used in future 120-in. SRM MOL studies.

Some differences exist between the configurations discussed in the loads reference configurations section and the motors discussed in this section. The loads configuration was based on preliminary 2- and 3-segment data received from LPC.

The SRM configurations described below represent the application of the latest available data from the participating contractors together with the Martin Company's analysis of the data and judgment establishing motor design parameters well within the state of the art.

The physical characteristics of the 2 CS-156 and 3 CS-156 SRM configurations recommended by Martin and described in the referenced constraints documents are shown in Fig. VI-9 and VI-10 and Table VI-3. Motor performance data (discussed below) are noted in Table VI-4, and the desired thrust-time histories are shown in Fig. VI-11 and VI-12.

The recommended configurations were selected after review and analyses of all data submitted by the participating SRM contractors and are judged to be well within the present SRM technology and state of the art. The items discussed in the following subsections were judged to be of major significance and are presented to substantiate the recommended SRM configurations.

Table VI-3 Recommended 156-in. SRM Physical Characteristics

	2 CS-156	3 CS-156
Total Motor Length (in.)	1,146	1,396
Maximum Nozzle Exit Diameter (in.)	135.8	135.8
Nozzle Cant Angle (deg)	6	6
Maximum Nozzle Submergence (%)	25	25
Nozzle Expansion Ratio	10:1	8:1
Approximate Propellant Weight (lb)	840,000	1,095,000
Usable TVC Fluid Injection Weight (lb)	9,988	11,050
Minimum Motor Mass Fraction	0.885	0.890
Estimated Total SRM Weight (lb)	950,000	1,232,000

Table VI-4 Recommended 156-in. SRM Performance Characteristics

	2 CS-156	3 CS-156
<u>Ballistic</u>		
Desired AT (sec)	146	156
Desired PBT (sec)	137	148
Total Delivered Vacuum Impulse (lb-sec)	2.22×10^8	2.90×10^8
Delivered PBT Vacuum Impulse (lb-sec)	2.18×10^8	2.85×10^8
Initial Required T/W Ratio	1.6	1.6
Minimum Vacuum $I_{sp} \left(\frac{\text{lb-sec}}{\text{lb}}\right)$	263	263
<u>Propellant</u>		
Type	Class 2 Composite PBAA/AN-AP	Class 2 Composite PBAA/AN-AP
Solids Loading (%)	87 Maximum	87 Maximum
Grain Configuration	Center Segments Interchangeable	Center Segments Interchangeable

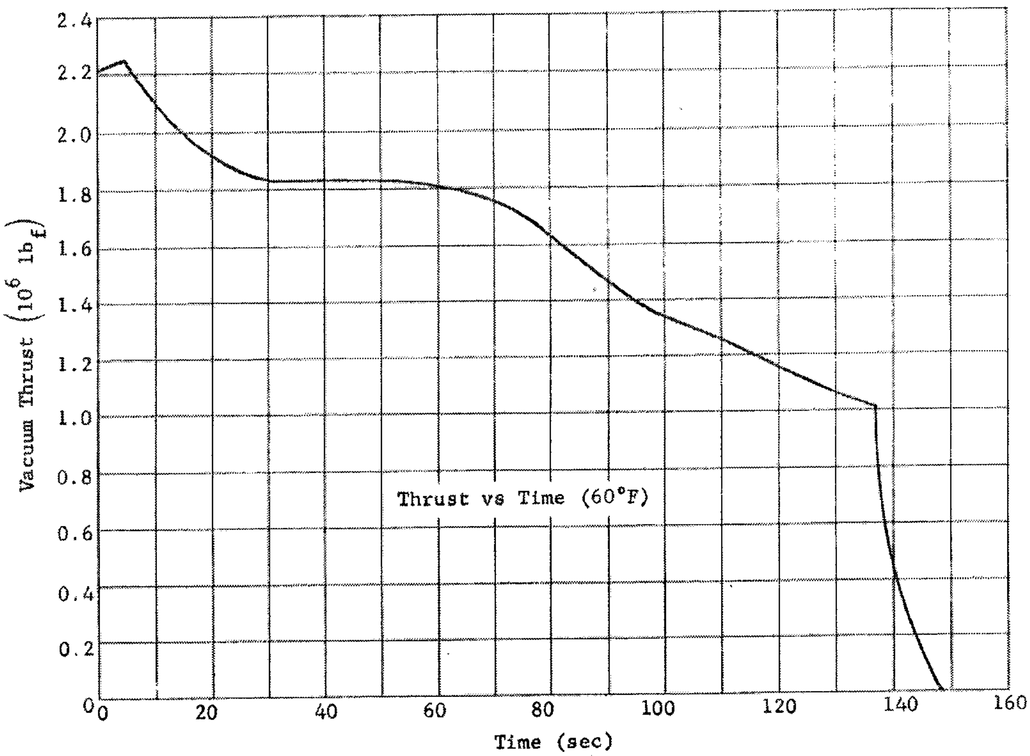


Fig. VI-11 2 CS-156 SRM Constraints Document Configuration, Thrust vs Time

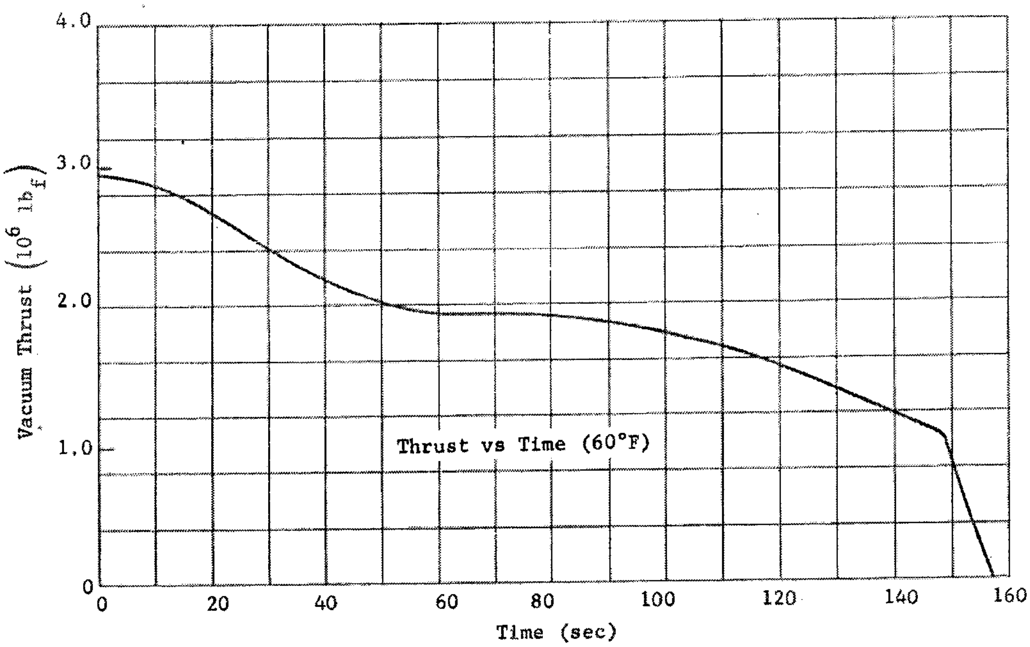


Fig. VI-12 3 CS-156 SRM Constraints Document Configuration, Thrust vs Time

1. Nozzles

A partially submerged nozzle (up to 25% submergence) is recommended for use with the 156-in. SRMs. This recommendation is based on the following facts:

1) An increase in propellant weight and, therefore, payload is obtained in a length-limited system by using a submerged nozzle. Approximately 30,000 lb of additional propellant can be added to the selected configurations as a result of 25% nozzle submergence.

2) The submerged nozzle state of the art is firmly established for the Minuteman program and is supported by numerous subscale firings with throat diameters up to 15 in. In addition, three 156-in.-diameter motor submerged nozzle tests are scheduled in the near future. LPC is scheduled to test fire two motors before January 1966, and TCC will fire one motor early in 1966.

3) Martin Company tradeoff studies show that the heat transfer, structural integrity, material erosion, and char characteristics of the partially submerged nozzle are most acceptable for use on large SRMs. The various nozzle materials and properties submitted by the participating SRM contractors are listed in Section C of this chapter. Martin Company recommendations are noted for the nozzle throat material (graphite phenolic) and exit cone material around the TVC injectant port location (silica-phenolic). Graphite-phenolic provides an acceptable erosion rate at the throat and, from test data, appears to maintain structural integrity during long-duration firings better than other materials. Most N_2O_4 TVC experience has been with silica-phenolic insulation surrounding the TVC injectant ports. Its erosion rate in the presence of oxidizing N_2O_4 is known and acceptable for long-duration firings.

4) Because of the proven state of the art of submerged nozzles for smaller solid-propellant motors and because of forthcoming large motor submerged nozzle tests, the technical risk associated with the 156-in. SRM submerged nozzle is low. It is far less than the risk taken with nozzle development on the 120-in. 5-segment motor at a similar program point.

2. TVC System

An N_2O_4 LITVC system is described in the SRM constraints documents and is presently the recommended system. However, if any other type TVC system with equal or better reliability and operating characteristics is proposed in the future, consideration will be given to that system. As mentioned previously, all SRM participants proposed LITVC using as much of the existing Titan IIIC 120-in. 5-segment SRM TVC system as practical. Martin Company LITVC loading, thrust deflection, and slew rate requirements are listed in Section C of this chapter. The values listed as well as the recommendation for use of an LITVC system are based on the following:

1) Analysis of the MOL SRM system TVC requirements show that the LITVC system presently used on Titan IIIC can meet the requirements for thrust vector deflection at maximum motor thrust, slew rate, thrust vector deflection during tailoff, and injectant fluid load requirements with minor modifications;

2) Fourteen static tests and two flight tests of 5-segment 120-in. motors have shown that the LITVC system is reliable and capable of being used successfully on large SRM systems. LITVC use on the Polaris and Minuteman programs has also shown considerable success;

3) The success of LITVC on the Titan IIIC program and availability of the subsystem test data and common components indicate a very low technical risk associated with the TVC system. The minor component redesign and requalification required will not be the pacing item on the 156-in. SRM program.

3. Cases

The case material recommended for use on the 156-in. SRMs is 250 grade 18% nickel maraging steel. The major reasons for selection of this material are:

1) Four 156-in. SRMs using nickel maraging steel cases have been successfully static fired. Maraging steel cases have also been successfully used on the 260-in.-diameter SRM programs;

2) Existing steel production technology such as vacuum arc remelting with controlled rates of solidification and homogenization of the ingot before hot working, and welding techniques (high rate metal deposition with TIG method) allow production of 250 grade maraging steel with fracture toughness of better than 100 ksi $\sqrt{in.}$;

3) The 18% nickel maraging steel has excellent capability for repair and rework, resulting in a lower rejection rate. This is a cost advantage as well as a desirable aspect from a scheduling standpoint;

4) The technical risk associated with the maraging steel cases is deemed to be relatively low from a performance standpoint. To maintain this low risk, Section C of this chapter describes the Martin-recommended case design criteria; however, long lead times are involved, and it is recommended that strong consideration be given to ordering material before a May 1966 program go-ahead.

4. Tailoff Parameters

Initial criteria for SRM thrust tailoff design were based on limiting the maximum thrust differential between SRMs during tailoff to approximately 280,000 lb_f at core-engine ignition. With this limitation, it was anticipated that the existing Titan III core aft longerons would not require redesign. SRM tailoff analysis shows that it is costly from a payload standpoint, and somewhat risky from a SRM performance standpoint, to limit tailoff thrust differential to the low level required by the existing core aft longeron design. It is strongly recommended that the core aft longeron be redesigned to withstand approximately 500,000 lb of thrust differential for the 2 CS-156 SRM and approximately 600,000 lb of thrust differential for the 3 CS-156 SRM. The following items establish the basis for this recommendation and specify the Martin Company design limits where applicable for the SRMs and as stated in Section C of this chapter:

1) Data submitted by SRM manufacturers indicated that with burning time control held to within ±1% (3 sigma), the 280,000 lb thrust differential between SRMs was not exceeded. However, after a careful review of Titan IIIC 120-in. 5-segment SRM data, Minuteman, Polaris, and numerous small motor programs, burning time control

of ±2% is recognized as state of the art. Subsequently, the ±2% motor-to-motor burning time variation was used to determine the maximum force differential between SRMs used for the MOL study. A 3°F temperature difference between SRMs was also used in the thrust differential analysis. Section C of this chapter discusses the results of thrust differential as a function of tailoff time with both ±1% and ±2% motor-to-motor burning time variation. This plot was obtained by "stretching" out the SRM tailoffs defined in the design contraints reports until a low level of thrust differential was reached. It is seen that to meet initial 2- and 3-segment requirements, the tailoff time must exceed 25 sec.

2) The 3 CS-156 SRM payload performance was computed using a 22.5-sec tailoff. Payload weight was decreased by approximately 1700 lb (or 4%) over performance using the recommended tailoff time of 10 sec.

3) The technical risk associated with the burning rate control tolerance of ±2% and the allowable 500,000 to 600,000 lb thrust differential between SRMs is exceedingly low. These tolerances are firmly established as state of the art and are strongly recommended to be included on the 156-in. SRM designs.

5. Propellant

Propellant formulation and grain design for the recommended SRMs will specifically be somewhat dependent on the individual design. However, Martin recommendations of Class 2 composite PBAA/AN-AP propellant and simple interchangeable center segment grain design are still imposed. Note that all propellants proposed by the SRM manufacturers were of the recommended type and are all proven state-of-the-art formulations with well-known characteristics.

The technical risk associated with the grain designs is very low. More than twenty 156-in. and 120-in. SRMs using features of the required grain designs have been tested with very successful results.

6. Thrust Termination

SRM thrust-termination capability is required for the MOL vehicle. The Titan IIIC method of thrust termination is presently being recommended; that is, by releasing two ports located in the motor head end. Martin analysis is continuing and will subsequently fix the port size range for each configuration. To maintain core vehicle structural integrity following thrust termination, a positive forward force of 100,000 lb must be maintained by the SRMs. Section C.8 of this chapter defines the SRM force required to maintain core integrity.

C. SRM CONTRACTOR DATA

In addition to the recommended SRM configurations described in Section B, several other motor designs were received and analyzed. The data presented in this section are the final designs received from each participating SRM contractor. Brief notations and explanations of Martin Company analysis are also indicated.

1. Internal Ballistics and Propellants

Table VI-5 contains the internal ballistics, propellant, and grain design data for each of the motors submitted (final design). Curves of thrust, chamber pressure, and weight flow are presented in Fig. VI-13 thru VI-17. Ignition overshoot is ignored on these curves, but was covered in consideration of the maximum expected operating pressure (MEOP).

The propellant formulations proposed are shown in Table VI-5 and are all composite propellants composed of PBAA or PBAN with aluminum and ammonium perchlorate. All propellant formulations are of proven quality and are well within the propellant state of the art. Physical, mechanical, and chemical properties of each propellant proposed are acceptable.

The propellant grain designs (all similar to Fig. VI-6) are also simple, proven, state-of-the-art designs with acceptable stress levels and design shapes. Some minor variations were submitted with respect to inhibiting (see Fig. VI-4) and port shaping, but all are acceptable (provided that the related thrust-time curve is acceptable). It is a requirement that the center segments be identical to maintain interchangeability within each of the three configurations.

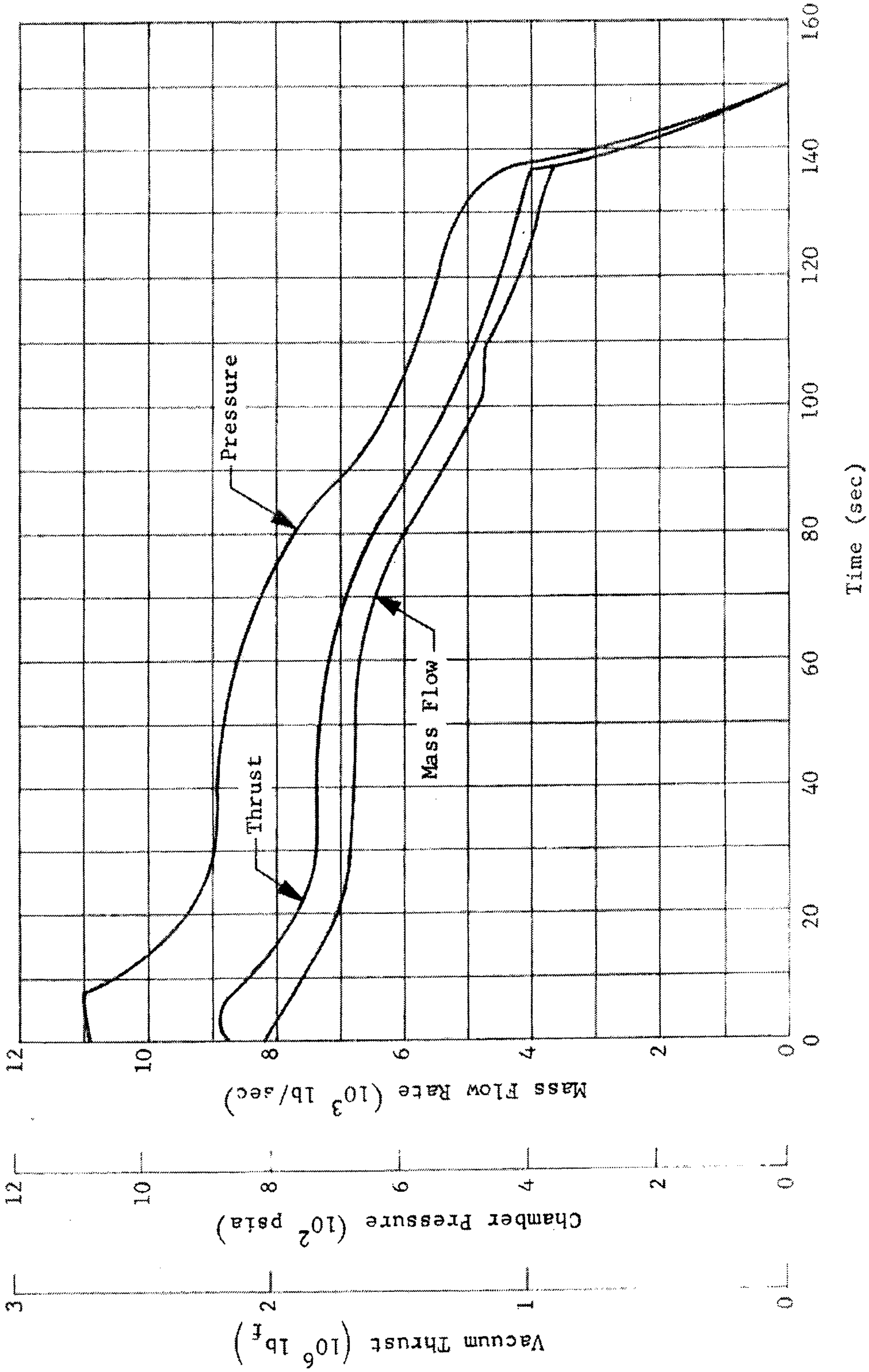


Fig. VI-13 LPC 2 CS-156 Motor Performance (60°F)

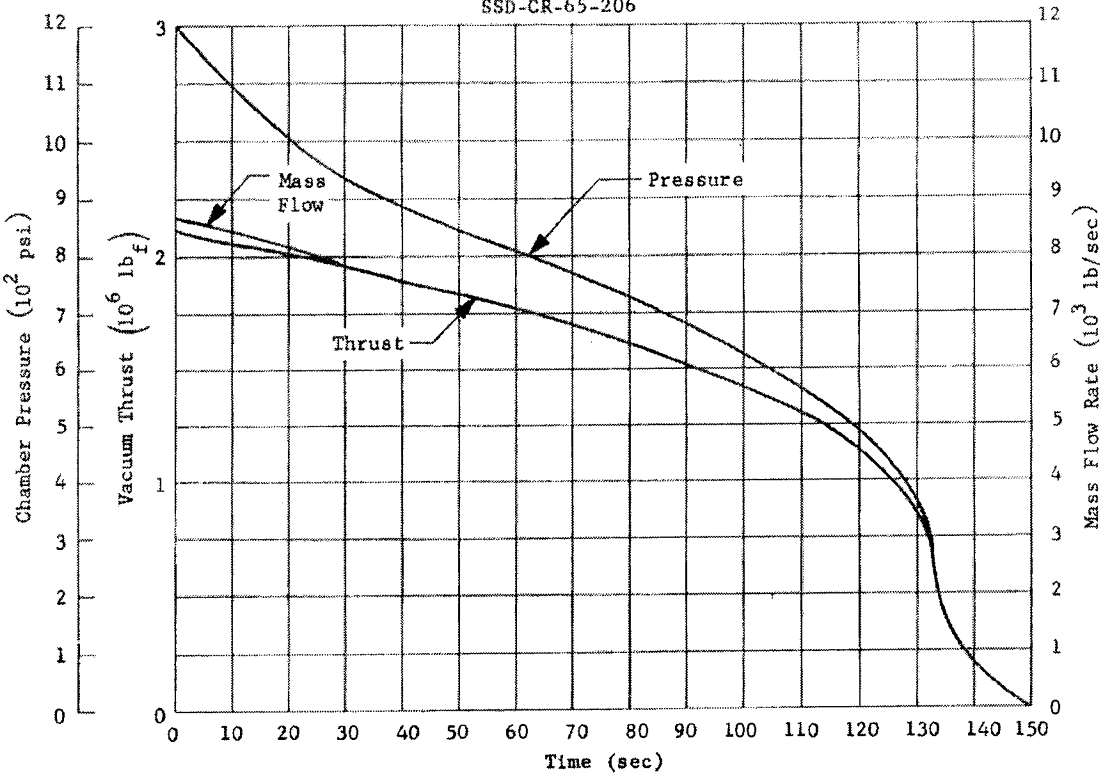


Fig. VI-14 TCC 2 CS-156 Motor Performance (60°F)

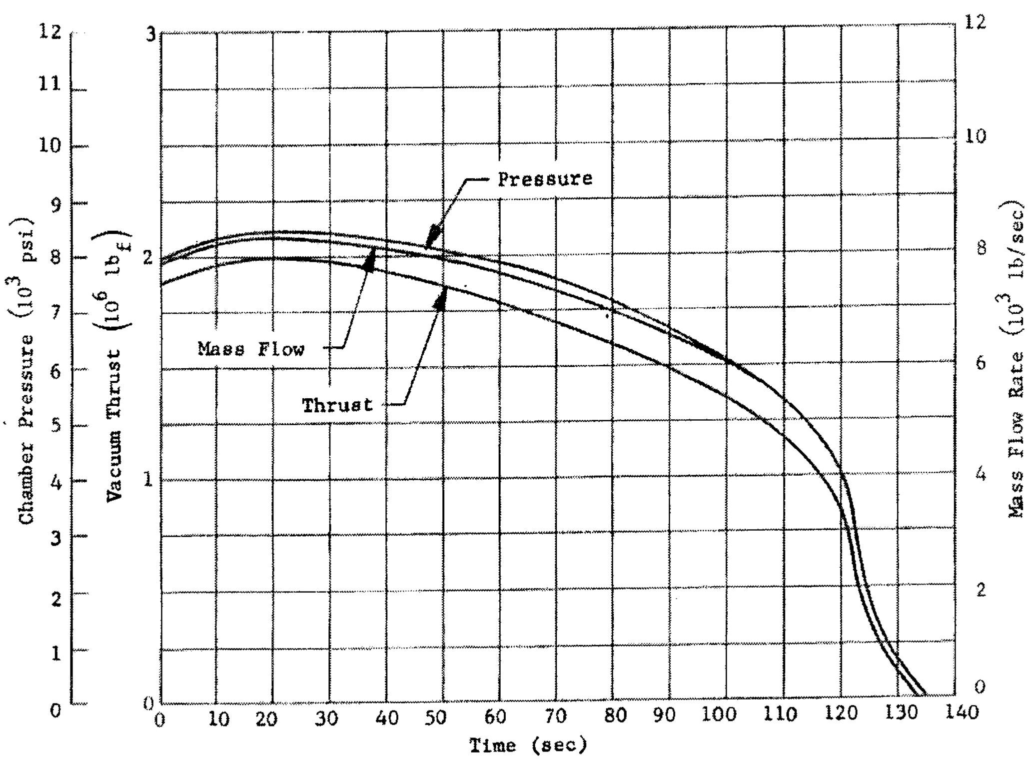


Fig. VI-15 AGC 2 CS-156 Motor Performance (60°F)

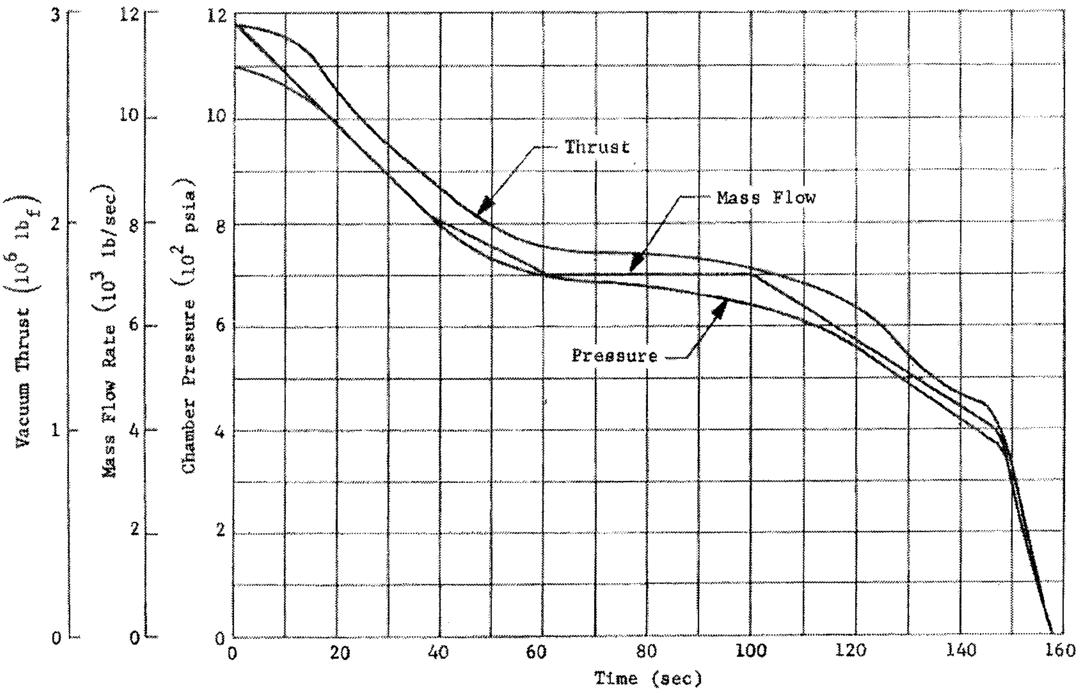


Fig. VI-16 LPC 3 CS-156 Motor Performance (60°F)

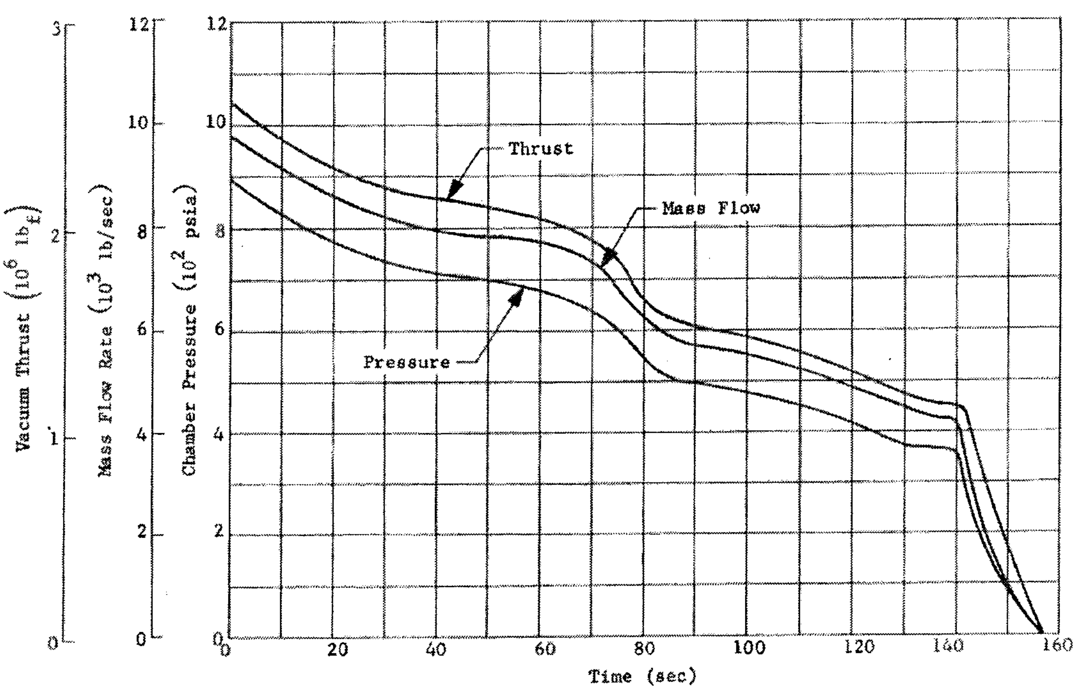


Fig. VI-17 UTC 3 CS-156 Motor Performance (60°F)

SSD-CR-65-206

Table VI-5 SRM Internal Ballistics

Contractor Motor Diameter (in.) Segments	UTC 120 7	UTC 156 3	Lockheed 156 2	Lockheed 156 3	Thiokol 156 2	Aerojet 156 2
<u>Nozzle Data</u>						
Throat Area, Initial (sq in.)	1,301	1,768	1,205	1,630	1,288	1,619
Throat Area, Final (sq in.)	1,374	1,882	1,302	1,740	1,538	1,736
Exit Area, Initial (sq in.)	12,490	14,144	14,000	14,000	7,724	11,310
Exit Area, Final (sq in.)	12,650	14,450	14,080	14,000	7,724	11,310
Expansion Cone Half-Angle (deg)	15.0	15.0	17.5	17.5	17.5	17.5
Expansion Ratio, Initial	9.6	8.0	11.6	8.58	6.0	7.0
Port-to-Throat Area Ratio	1.32	1.49	1.8	1.4	1.78	1.31
<u>Performance (Nozzle)</u>						
MEOP (psi)	920	1050	1189	1189	1400	927
Action Time (AT)(sec)*	124.31	151.39	150	157	142	127
Pressure Burning Time (PBT)(sec)†	115.09	141.39	137.25	147.4	128	118
Initial Vacuum Thrust $(10^6\ lb_f)$‡	1.61	1.93	1.85	2.325	1.48	1.50
Average Vacuum Thrust $(10^6\ lb_f)$**	1.30	1.82	1.618	1.954	1.70	1.858
Initial Aft Pressure (psia)‡	550	669	825	825	920	600
Average Aft Pressure (psia)**	580	605	818	696	770	687
Impulse, AT $(10^8\ lb_f\text{-sec})$						
Vacuum	1.5388	2.6025	2.272	2.932	2.234	2.25
Sea Level						2.0579
Impulse, PBT $(10^8\ lb_f\text{-sec})$						
Vacuum	1.4971	2.5710	2.221	2.880	2.176	2.1896
Sea Level	1.2858	2.2768				2.0080
Mass Fraction	0.864	0.878	0.884	0.893	0.867	0.836
Basic Motor Efficiency	0.96	0.96	0.96	0.96	0.96	0.98
<u>Propellant</u>						
Propellant Designation	UTP 3001	UTP 3001	LPC 580A	LPC 580A	TP H1011	ANB-3105
Propellant Weight	511,324	979,187	842,113	1,112,000	884,553	885,000
Composition	PBAN + AP + Al 16.1%	PBAN + AP + Al 16.1%	PBAN + AP + Al 16.1%	PBAN + AP + Al 16.1%	PBAA/AN + AP + Al 16.0%	PBAN + AP + Fe_2O_3 + Al 15%
Operating Temperature Range (°F)	40 to 90	40 to 90	40 to 100	40 to 100		40 to 90
Configuration						
Forward Segment	8 Spokes	10-pt Star	10-pt Star	10-pt Star	4 Spokes	Conocyl
Center Segments	Cyl Perf	Cyl Perf	Cyl Perf	Cyl Perf	Cyl Perf	Cyl Perf
Aft Segment	8 Spokes	Cyl Perf	Cyl Perf	Cyl Perf	Cyl Perf	Cyl Perf
Web Thickness (in.)	35.5	49.0	50.0	50.0	48	51
Burn Rate, 1000 psi, 60°F (in./sec)	0.356	0.385	0.41	0.391	0.392	0.474
Burn Rate Exponent	0.260	0.240	0.4	0.40	0.21	0.33
Specific Impulse, Standard $(lb_f\text{-sec}/lb_m)$	248	248	248	248	246	244.6
Temperature Coefficient of Press. (%/°F)	0.130	0.130	0.11	0.11	0.10	0.16
Temperature Coefficient of Burn Rate (%/°F)	0.096	0.099	0.07	0.07	0.08	0.107
Ratio of Specific Heats	1.180	1.180	1.17	1.17	1.18	1.18
Density (lb/cu in.)	0.063	0.0635	0.065	0.065	0.064	0.0635
Characteristic Exhaust Velocity (fps)	5,170	5,160	5,200	5,200	5,180	5,120
Adiobatic Flame Temperature (°F)	5,700	5,674	5,870	5,870	5,790	5,633
Molecular Weight of Exhaust Gas	19.9	19.9	19.9	19.9	18.7	
Molecular Weight of Exhaust Products	26.4	26.4	27.5	27.5	26.4	28.8
Exhaust Gas Composition						
Exit Plane						
H_2	30.3	30.3	30.0	30.0	27.9	29.2
H_2O	11.3	11.3	11.3	11.3	14.1	17.2
H	0.2	0.2	0.6	0.6	0.2	0.4
CO	25.4	25.4	22.0	22.0	23.0	25.0
CO_2	1.9	1.9	1.9	1.9	2.3	2.6
N_2	7.8	7.8	8.8	8.8	8.4	8.7
HCl	15.0	15.0	15.7	15.7	15.9	16.4
Al_2O_3	7.9	7.9	9.1	9.1	8.0	8.0
CL	0.1	0.1	0.3	0.3	0.1	0.2
Fe	--	--	0.3	0.3	--	--

*10% P_{Max} to 10% P_{Max}.

†10% P_{Max} to web burnout.

‡Measured at time when P = 75% P_{Max}.

**Averaged over WAT.

2. Performance Tolerances and Transients

Basic performance parameters and their associated tolerances are shown in Table VI-6. The values recognized as state of the art and recommended for use are those values listed in the "Martin Recommended" column.

Tailoff variations submitted by each SRM contractor are shown in Table VI-7 in conjunction with Fig. VI-18. Also shown in Table VI-7 are the tailoff variations resulting when the recommended state-of-the-art tolerance values are used.

Figures VI-19 thru VI-21 present the reference configuration tailoff curves used for loads and staging calculations. Figure VI-19 (7 seg-120) is as presented by UTC. Figures VI-20 and VI-21 (2- and 3-segment reference configurations) use the tail-off shape presented by LPC, but incorporate Martin tolerances for burning times. Figure VI-22 presents the TCC 2-segment tail-off, which incorporates a long tailoff time with ±2% time toler-ances in an attempt to limit thrust differential to a maximum of 300,000 lb_f. Note that the longer burning time TCC motor pro-duces a lower payload by approximately 2000 lb than the compara-ble reference configuration 2-segment motor.

In an attempt to define payload loss with increased tailoff burning time, the reference 3-segment configuration thrust time tailoff was increased from 10 to 22.5 sec duration (Fig. VI-23). Total vacuum impulse was held constant. The payload lost was approximately 1700 lb, or 4%.

3. Ignition

Ignition variations submitted by each SRM contractor are shown in Table VI-8 in conjunction with Fig. VI-18. The igniter designs and associated ignition transients submitted by the SRM contractors were not all acceptable.

Figure VI-24 indicates the relative position of each igniter proposed for the parameters defined. Additional constraints of bore pressure and heating rate were also considered. However, an igniter producing the transient recommended (Table VI-8) was bracketed by submitted designs and is well within present indus-try igniter design capability. To minimize risk in ignition and igniter design, it is strongly recommended that an SRM head-end mounted pyrogen ignition system be used. A summary plot of igni-tion curves received is in Fig. VI-25.

Table VI-6 SRM Performance Tolerances (60°F)

	Martin Recommendation	7 seg-120	2 CS-156			3 CS-156	
		UTC	LPC	TCC	AGC	UTC	LPC
Vacuum Total Impulse PBT (%)	1.25	N/A	N/A	N/A	N/A	N/A	N/A
Vacuum Total Impulse, AT (%)	1.0	1.0	0.69	1.0	0.81	1.0	0.69
Specific Impulse (%)	0.7	0.7	0.45	0.70	0.13	0.7	0.45
Action Time (AT)(%)	3.0	3.43	N/A	2.0	N/A	3.09	N/A
Pressure Burning Time (PBT)(%)	2.0	2.16	0.96	2.0	2.0	1.95	0.96
Ignition Interval (%)	15.0	9.1	6.0	11.1	20.8	4.9	6.0
Variation in Thrust before Tailoff (%)	3.0	4.04	1.3	2.2	2.38	4.04	1.3
All values are ±3 sigma.							

Table VI-7 Tailoff Parameters

Parameters Submitted by SRM Contractors	7 seg-120	2 CS-156			3 CS-156	
	UTC	LPC	TCC	AGC	UTC	LPC
Δt_b Max (sec)	4.98	2.50	5.12	4.72	5.50	2.82
Δt_a max (sec)	8.52	2.50	5.12		9.34	2.82
ΔF max $\left(10^3\ lb_f\right)$	340	280	300		500	280
Max Slope (%/sec)	8.1	8.0	8.8	9.0	11.5	10.0
Parameters Computed by Martin with ±2% Burning Time Tolerance						
Δt_b max (sec)	3.40	4.30	4.20		3.40	4.00
Δt_a max (sec)	6.40	6.00	6.00		6.20	4.60
ΔF max $\left(10^3\ lb_f\right)$	340	500	340		520	600

Initial Criteria:

ΔF max (All Systems) = 300,000 lb_f max and 280,000 lb_f at 1.7 g.

Slope = 10%/sec of Web Burnout Thrust

[redacted]

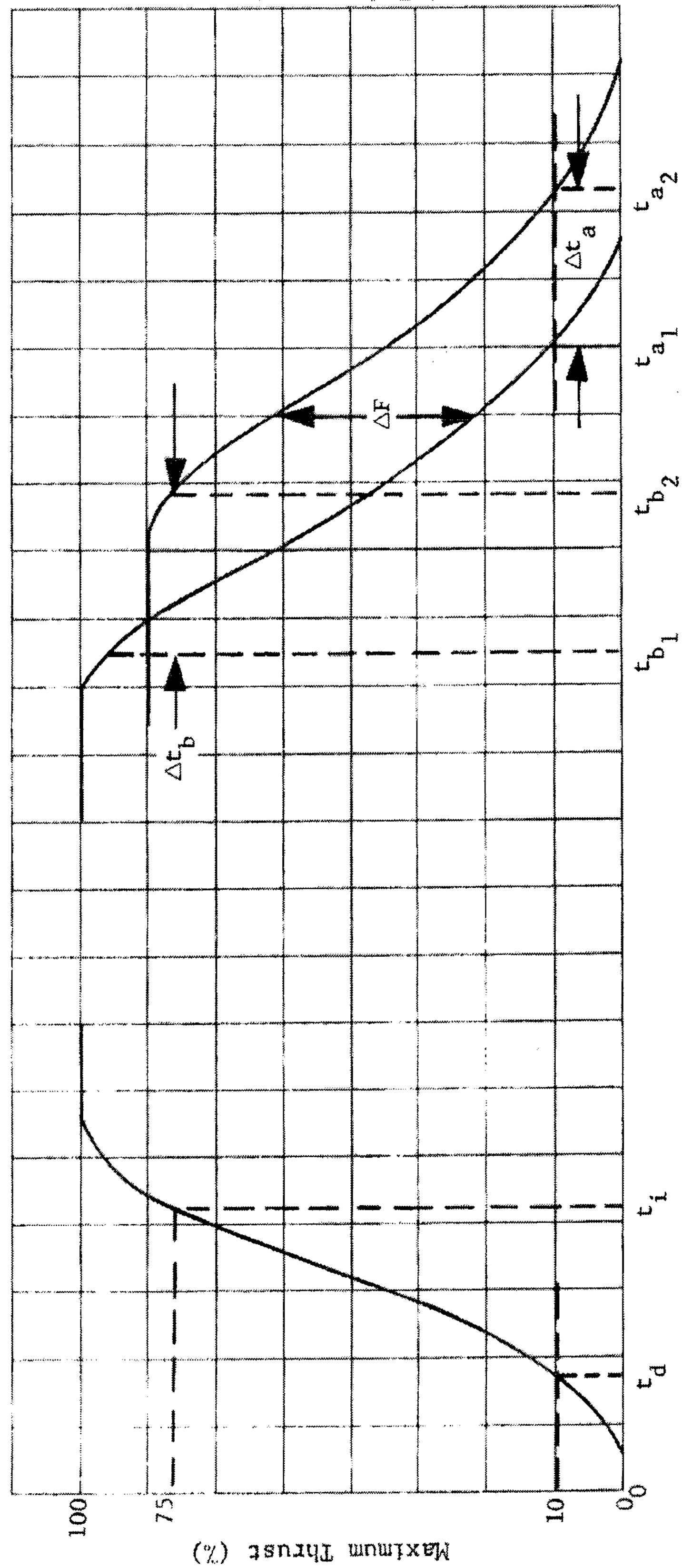


Fig. VI-18 Ignition and Tailoff Parameter Definitions

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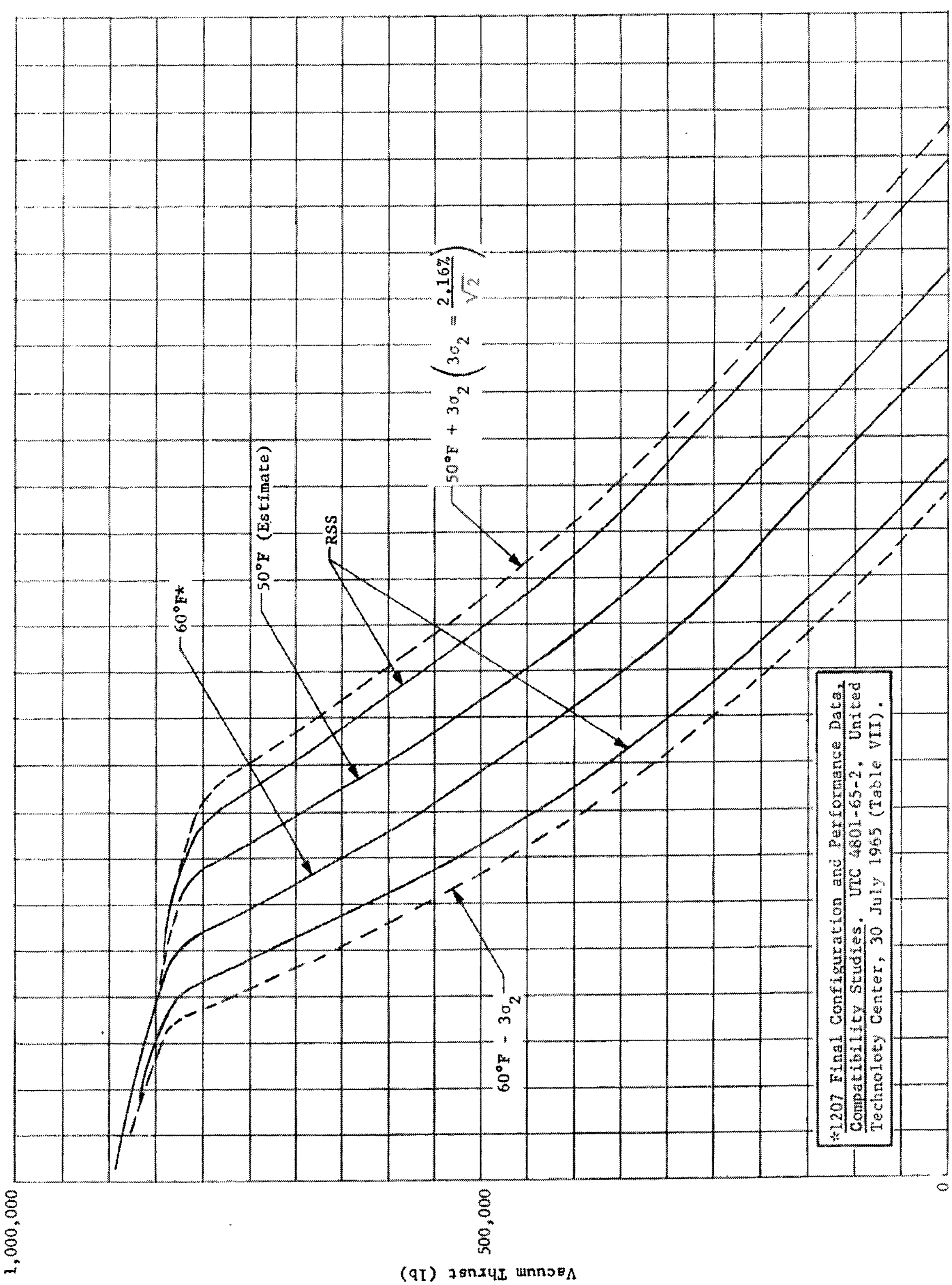

Fig. VI-19 Tailoff Vacuum Thrust Curves, 7 seg-120

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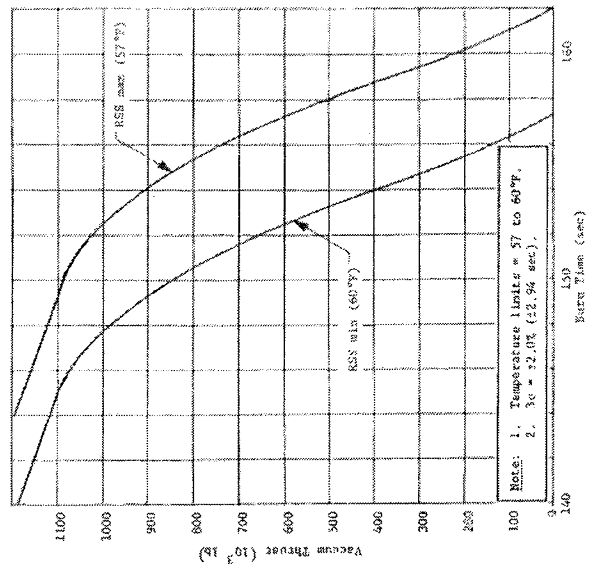

Fig. VI-21 Thrust Tailoff, LPC 3 CS-156

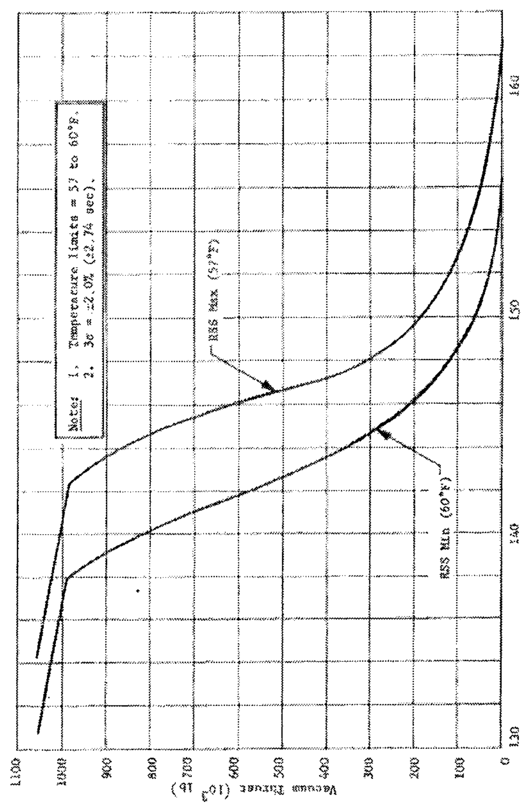

Fig. VI-20 Thrust Tailoff, LPC 2 CS-156

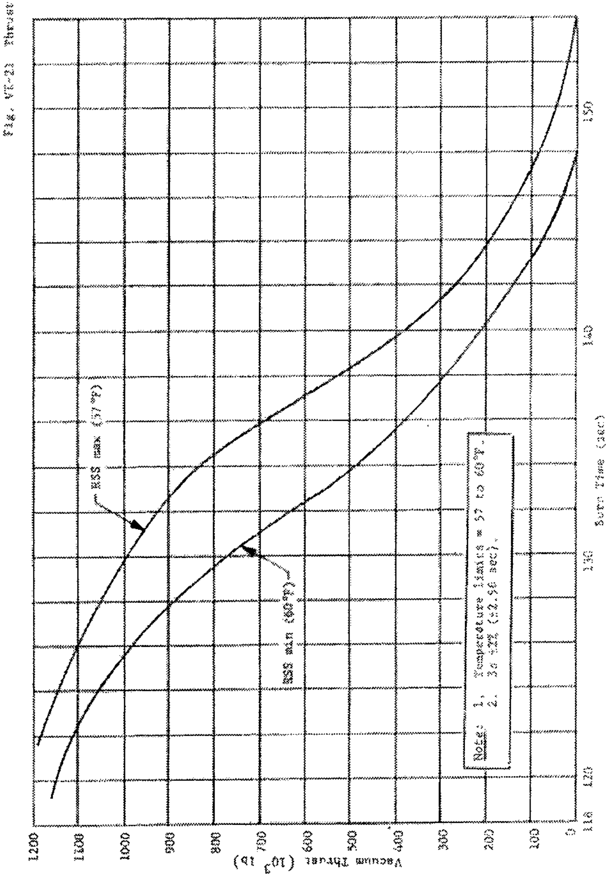

Fig. VI-22 Thrust Tailoff, LPC [illegible] CS-156

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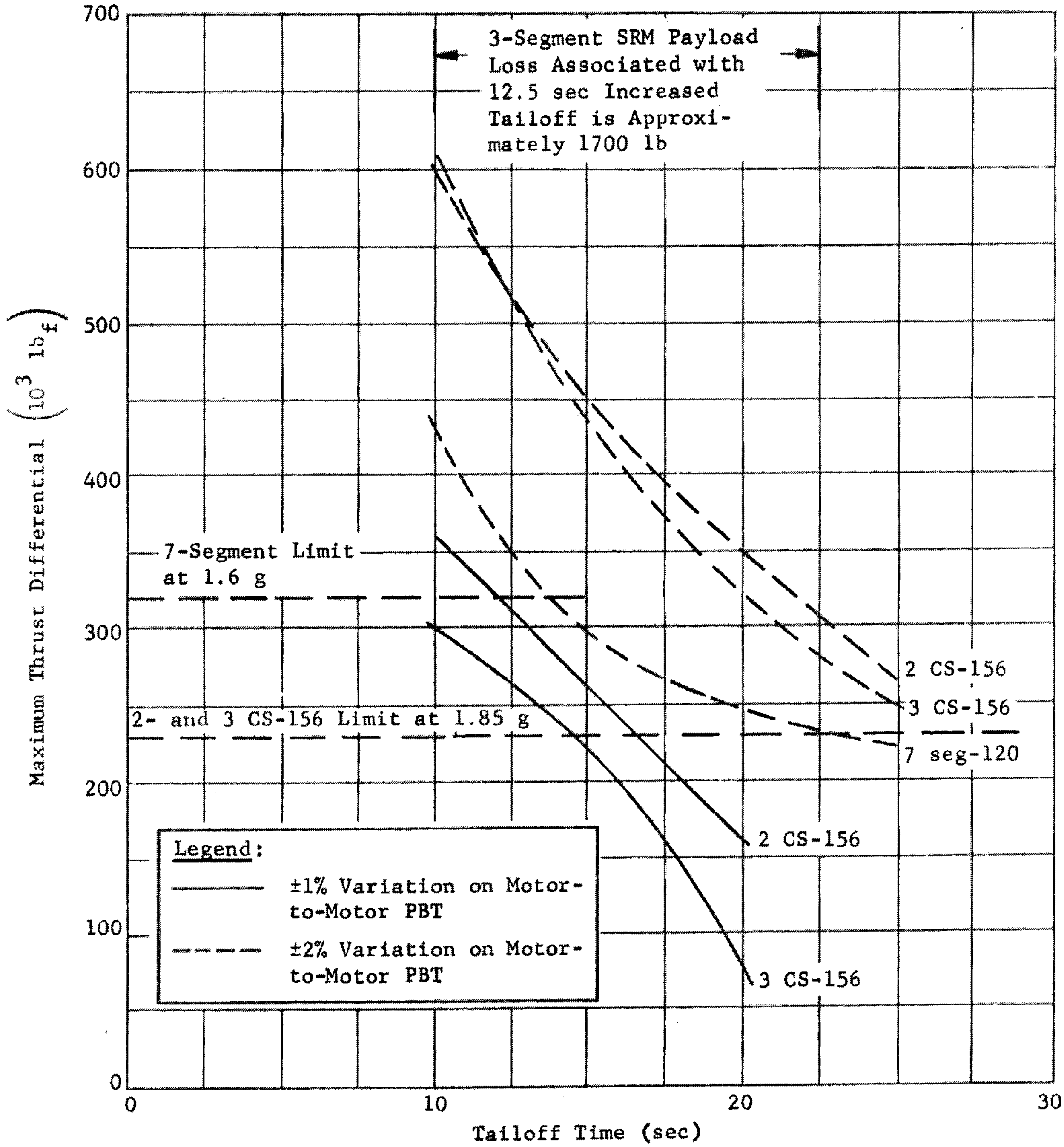

Fig. VI-23 Maximum Thrust Differential vs Tailoff Time

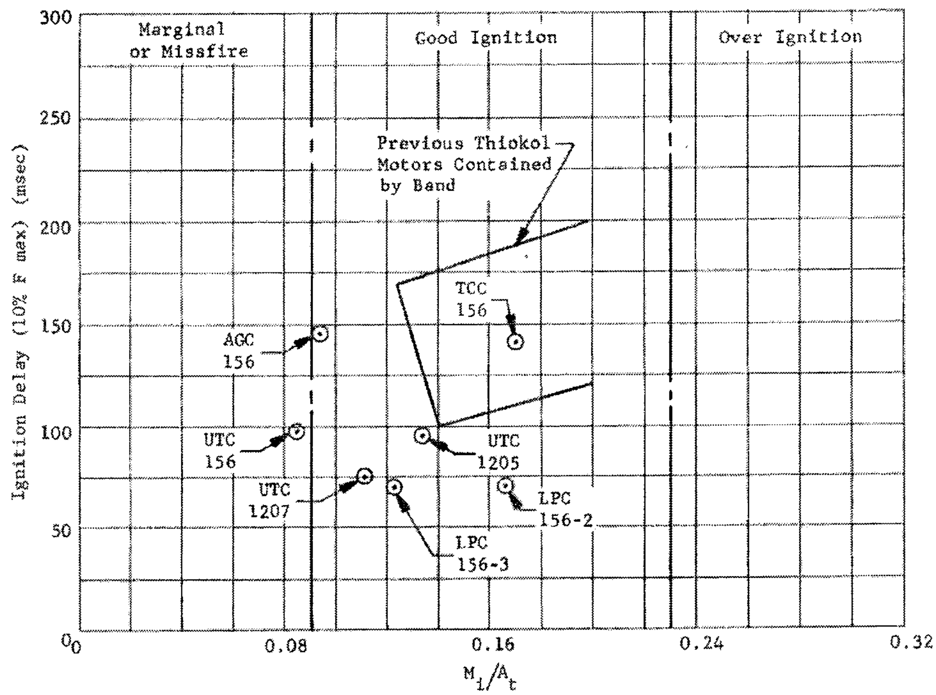

Fig. VI-24 Relationship of Igniter Discharge to Ignition Delay

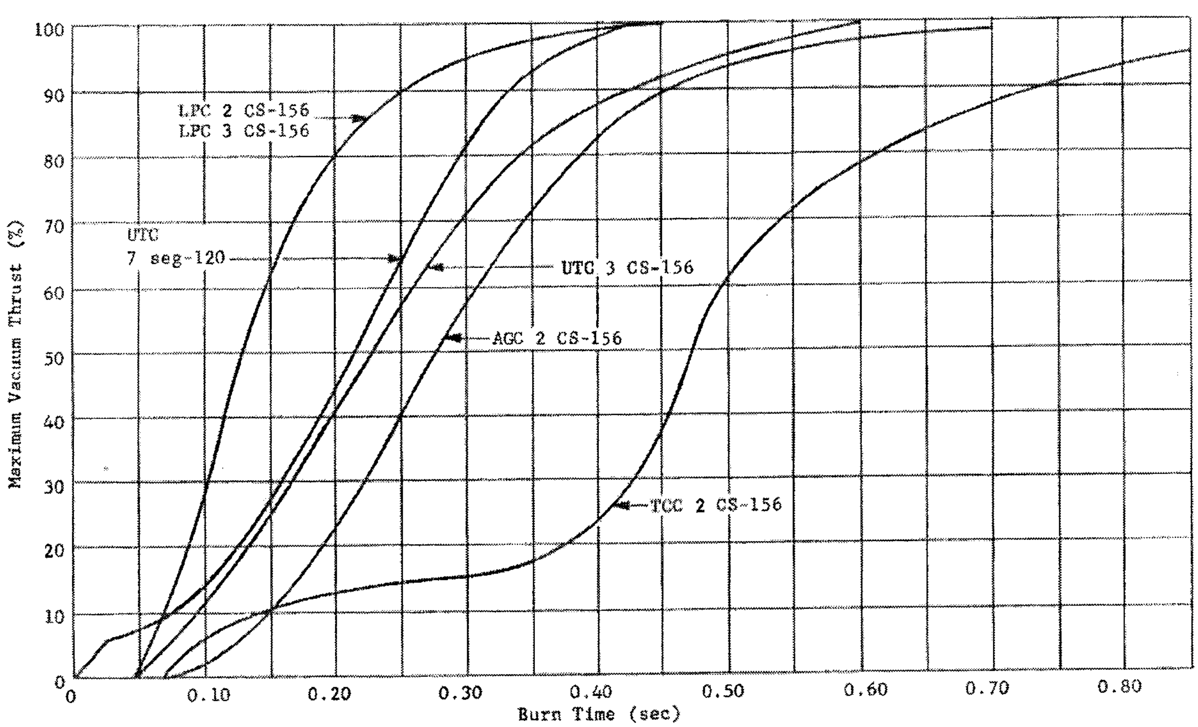

Fig. VI-25 Nominal Ignition Transient, Large SRMs

Table VI-8 Ignition Parameters

Contractor-Submitted Parameters	7 seg-120	2 CS-156			3 CS-156	
	UTC	LPC	TCC	AGC	UTC	LPC
$t_d \pm 3\sigma$, MS	80 ± 15	68	140	150	95	68
$t_i \pm 3\sigma$, MS	275 ± 25	180 ± 10.8	565 ± 63	360 ± 15.2	310 ± 15.2	180 ± 10.8
t (100%F), MS	430	450	1000	700	600	450
△ Force, % Maximum at Time (sec)	17.5 0.250	9.2 0.220	27.8 0.475	30.2 0.270	8.0 0.50	6.4 0.30
Martin Recommendations						
t_d, MS	80	75				
$t_i \pm 3\sigma$, MS	275 ± 31.5	215 ± 25				
t (100%F), MS	430	400				
△ Force, % Maximum	20	25				
See Fig. VI-18 for symbol definitions.						

4. TVC System

All contractors proposed an N_2O_4 LITVC system to meet the deflection angle requirements of 8% (side force)/(axial force) ratio for the 7-segment motor and 6.1% (side force)/(axial force) ratio for the 2- and 3-segment motors. Each proposed system made maximum use of the existing UTC TVC system components. Table VI-9 gives a brief summary of the contractor-supplied data and Martin recommendations for detailed portions of the system.

Based on static test and flight experience, the N_2O_4 LITVC system is recommended for use on the MOL/Titan III system. It is recommended that the N_2O_4 prevalve be retained in the TVC system until additional analyses have been conducted.

The TVC system requirements (weights and angles) noted in Table VI-9 are the result of six-degree-of-freedom trajectory analysis using the reference configuration motors with TVC side force characteristics as shown in Fig. VI-26 thru VI-31. These side force data were based on previous test results and were adjusted to match motor design conditions. Figure VI-32 shows the number of valves (103 and 143 lb/sec size) required to obtain the thrust deflection angle noted. It can be seen that 24 valves are required for each of the three reference configurations.

5. Nozzles

Two basic nozzle designs were proposed by the participating SRM contractors. UTC proposed an external nozzle configuration for both the 7 seg-120 and the 3 CS-156. The 7 seg-120 nozzle is basically the same as the nozzle used on the Titan IIIC 5 seg-120 motor. The 3 CS-156 UTC nozzle is a scale-up of the 5 seg-120 Titan IIIC SRM nozzle. All other contractors submitted partially submerged nozzle designs ranging from 17% submergence (LPC) to 45% submergence (TCC). The percentage of submergence is a function of both the nozzle area ratio and TVC injectant point. For a nozzle area ratio of 8:1 and injectant point area ratio of 3.5:1, the maximum submergence is approximately 25%.

Table VI-10 shows pertinent nozzle parameters as proposed by the SRM contractors. Where applicable, Martin recommendations are also noted.

6. Motor Cases and Attachments

Table VI-11 lists the proposed design criteria for the 7 seg-120, 2 CS-156, and 3 CS-156 SRM motor cases and attachment structures (to the vehicle core and the TVC fluid tank).

Because of stringent program schedule requirements, these criteria represent conventional design methods using existing tooling and technology. The newer methods offering potential improvements will present long lead times and higher overall costs. However, as the program develops, it may be possible to incorporate them into the later motor cases following intensive evaluation.

Table VI-9 TVC System Requirements

Martin Company Requirements	7 seg-120	2 CS-156	3 CS-156
TVC Injectant Fluid Required (Usable) (lb)	6,248	9,988	11,050
Number of Valves	24	24	24
Flow Rate of Valves (lb/sec)	103	143	143
Minimum Slew Rate (deg/sec)	10	10	10
Maximum Required Thrust Deflection at Maximum Motor Thrust (deg)	3.0	3.0	3.0
Maximum Required Thrust Deflection during Tailoff (deg)	6.0	6.0	6.0

SRM Contracter-Supplied Parameters	7 seg-120	2 CS-156			3 CS-156	
	UTC	LPC	TCC	AGC	UTC	LPC
Nozzle Area Ratio at Injection Point	2.94	4.0	2.7/3.0	3.5	3.5	3.0
Injector $\dot{W}$ (lb/sec)	103	142	155		143	142
Number of Valves	24	24	48	24	24	32
Tank Operation P_c (psi)	750	750	750	750	800	750
Usable Fluid (lb)	12,180	14,000	11,000	25,627	13,650	18,800
Maximum Angle Required (deg)	4.07	5.0	7.1	6.1	4.0	4.9
Slew Rate, Minimum (deg/sec)	5 at 6 deg		10 at 6 deg		6.8 at 4.58 deg	
Slew Rate, Maximum (deg/sec)	35 at 0 deg			30 at 1.5 deg	20.3 at 2.0 deg	

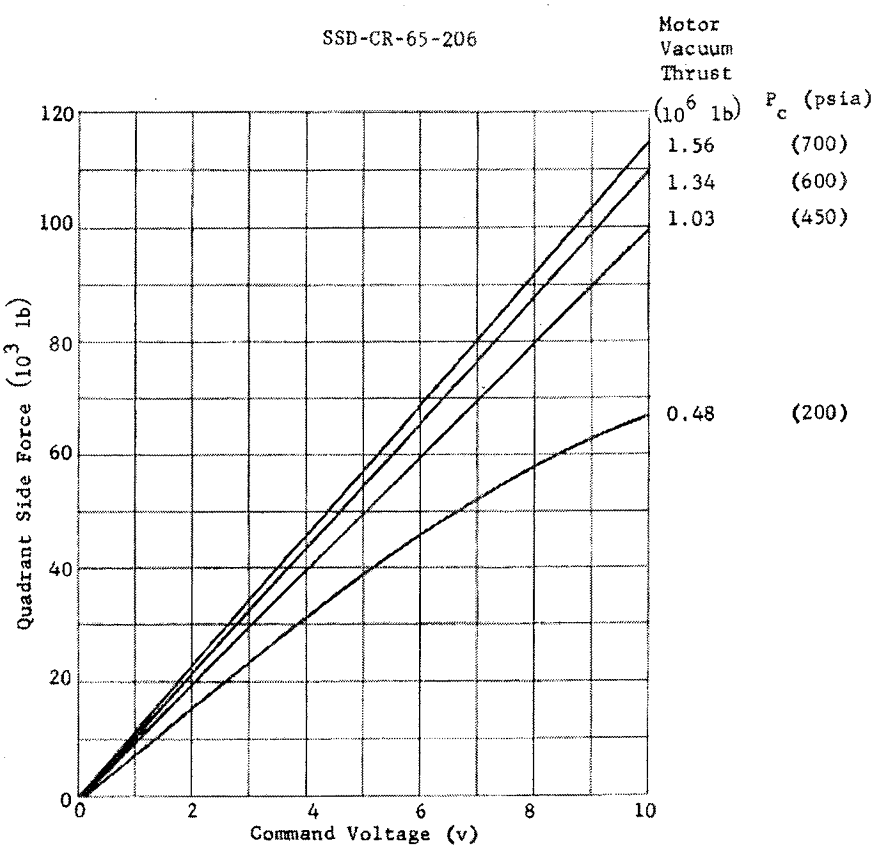

Fig. VI-26 TVC in UTC 7 seg-120, Side Force vs Voltage

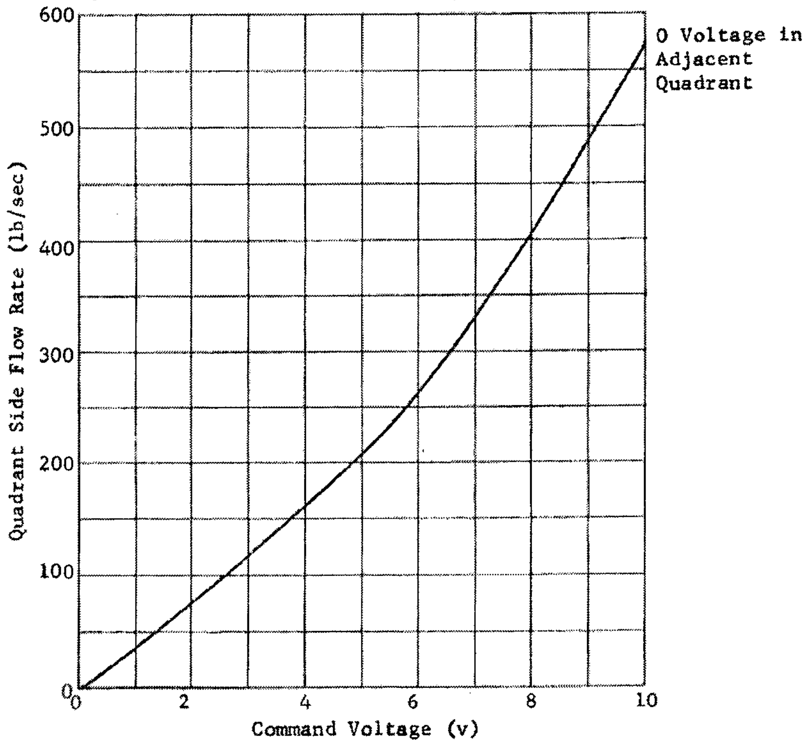

Fig. VI-27 TVC in UTC 7 seg-120, Flow Rate vs Voltage

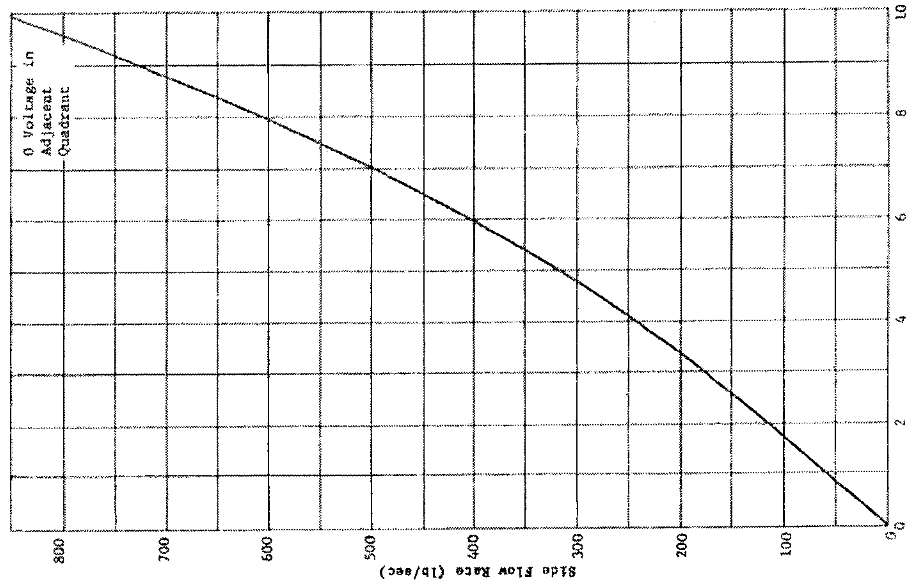

Fig. VI-29 TVC in LPC 2 CS-156, Flow Rate vs Voltage

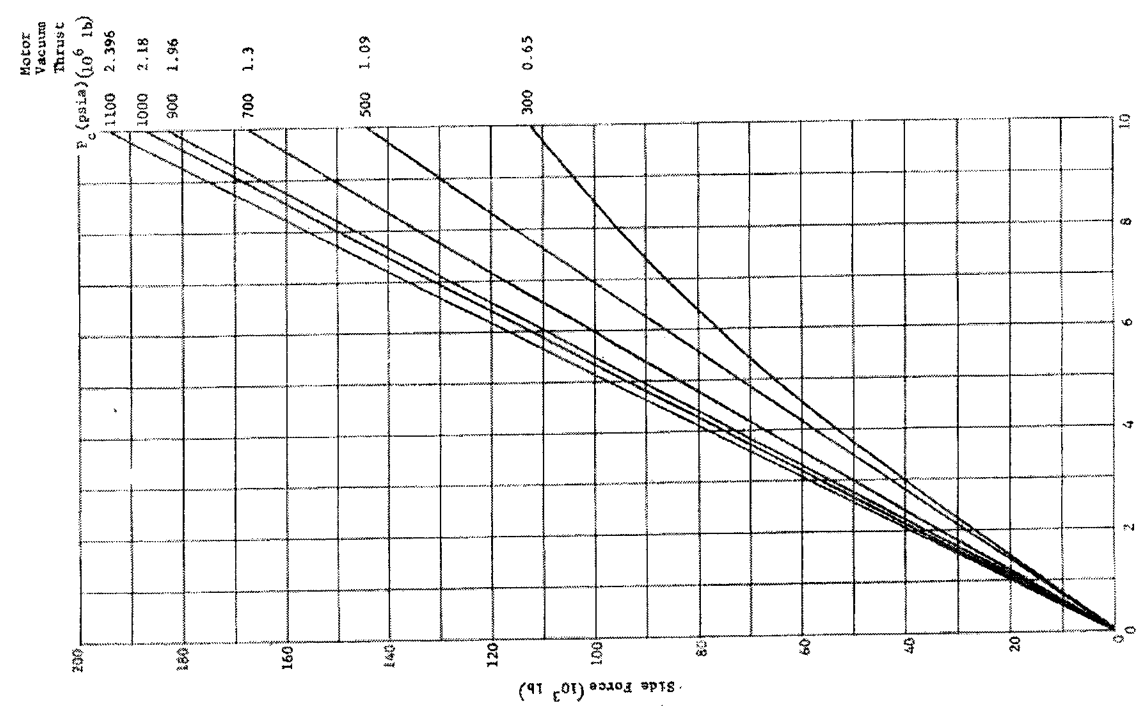

Fig. VI-28 TVC in LPC 2 CS-156, Side Force vs Voltage

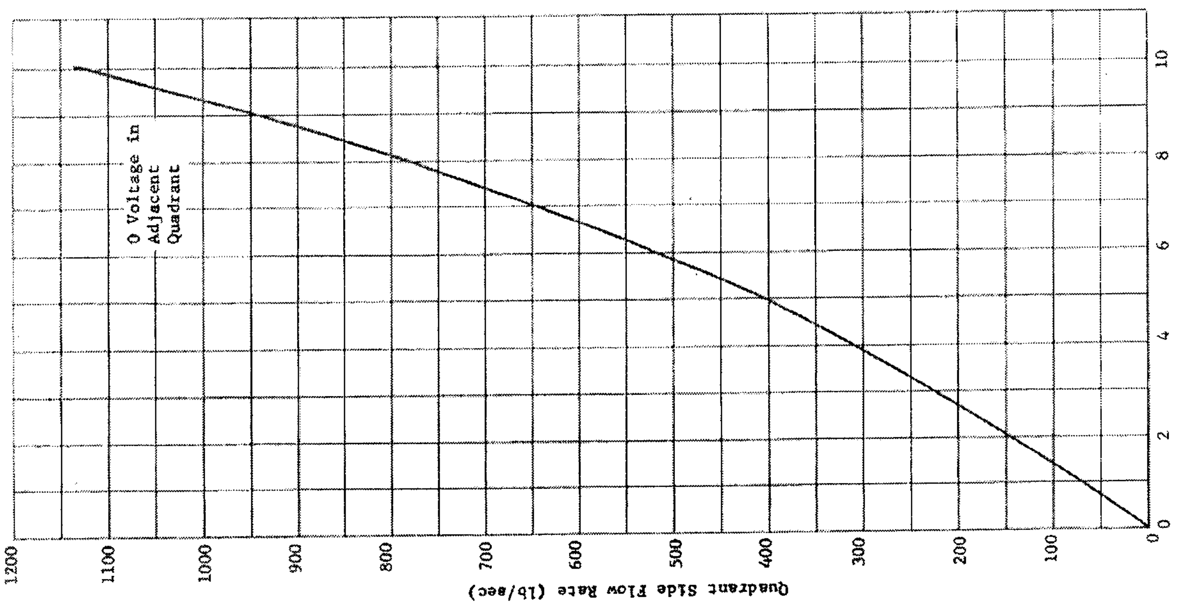

Fig. VI-31 TVC in LPC 3 CS-156, Flow Rate vs Voltage

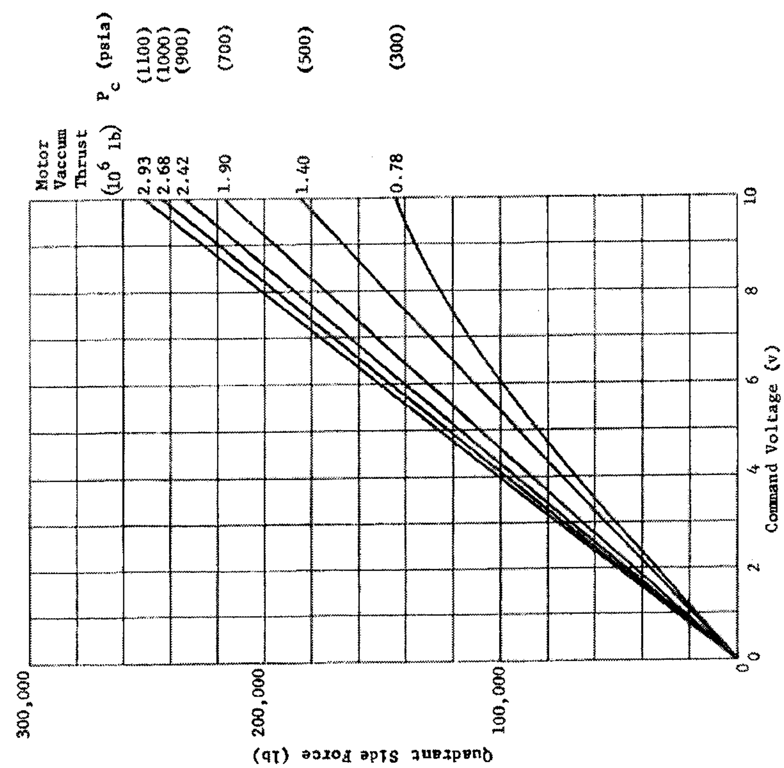

Fig. V-30 TVC in LPC 3 CS-156, Side Force vs Voltage

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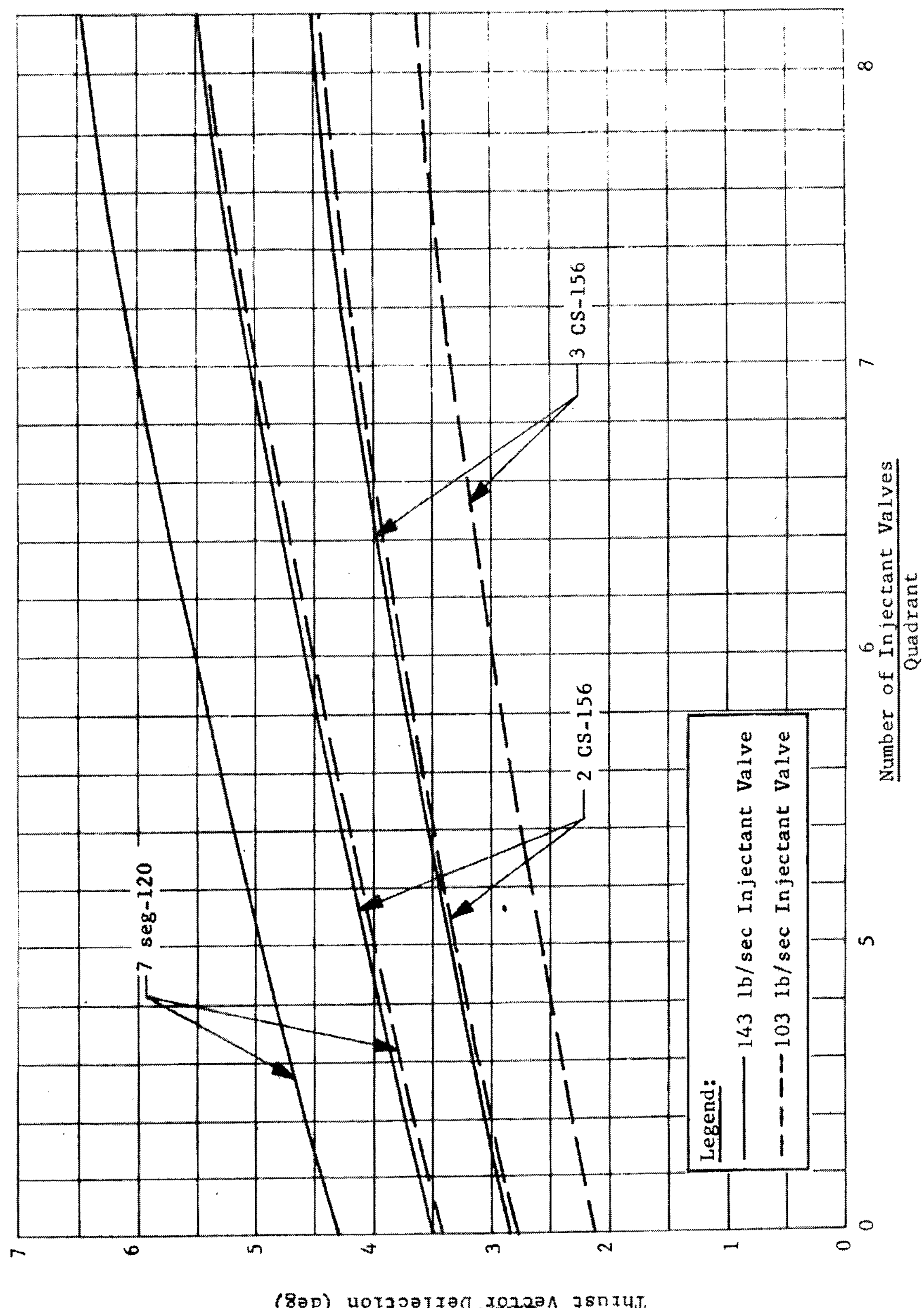

Fig. VI-32 Maximum Thrust Vector Deflection vs Number of Injectant Valves

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Table VI-10 Nozzle Characteristics

Contractor-Supplied Parameters	7 seg-120	2 CS-156			3 CS-156	
	UTC	LPC	AGC	TCC	LPC	UTC
Nozzle Type	External	Submerged	Submerged	Submerged	Submerged	External
Submergence (Nozzle Length Forward of Attach Flange to Total Nozzle Length) (%)	--	19	44	45.3	17	--
Expansion Ratio	9.6	11.6	7.0	6.0	8.6	8.0
Exit Cone Half Angle (deg)	15	17.5	17.5	17.5	17.5	15
Throat Material	Graphite-Phenolic	Carbon Phenolic	Carbon Phenolic	Graphite Phenolic	Carbon Phenolic	Graphite Phenolic
Ablation Rate (Throat) (mils/sec)	4.67	5.0	6.3	13.2	5.0	5.0
Material around TVC Inject Ports	Silica-Phenolic	Silica-Phenolic	Carbon Phenolic	Carbon Phenolic	Silica-Phenolic	Silica-Phenolic
Area Ratio at Inject Station	2.94	4.0	3.5	2.7	3.0	3.5
Structural Housing Material	4130 or 4340 Steel	18% Ni Steel	18% Ni Steel	18% Ni Steel	18% Ni Steel	4130 or 4340 Steel

Martin Recommendations	7 seg-120	2 CS-156	3 CS-156
Nozzle Type	External	Submerged (Approximately 25%)	Submerged (Approximately 25%)
Throat Material	Graphite-Phenolic	Carbon-Phenolic	Carbon-Phenolic
Material around TVC Ports	Silica-Phenolic	Silica-Phenolic	Silica-Phenolic

Table VI-11 Case and Attachments Design Criteria

1.	Case and attachments for 7 seg-120 SRM to follow Titan IIIC most recent practice.		
2.	Case 2 CS-156 and 3 CS-156 SRMs		
	a. Material		18% Ni maraging steel Grade 250 Fracture toughness 100 ksi $\sqrt{in.}$
	b. Fabrication		
	Forward and aft closures		Dome: Spin forgings Cylinder: Rolled and Welded Y-joint: Rolled ring forging
	Center segments		Rolled and welded
	Torque and clevis joints		Rolled ring forgings
	Port bosses (Igniter, TT, and Nozzle)		Rolled ring forgings
	Welding		TIG method, high-rate metal deposition
	c. Design		
	Material yield strength, 0.2% offset, ksi		230 min.
	Factor of safety to minimum yield		1.25
	Design pressure		MEOP x 1.25
	Case weld efficiency		95%
	d. Joint type		Torque and clevis with pin and O-ring
3.	Approximate stations for attachment to vehicle core		
	Forward	7 seg-120	250
		3 CS-156:	250
		2 CS-156:	504
	Aft		1226.56
4.	Attachment structures		
	Forward structure and aft skirt		Following Titan IIIC design practice

7. Insulation and Liner

Insulation materials proposed by the SRM contractors are shown in Table VI-12. Figure VI-33 shows typical proposed insulation joints.

Since the insulation materials proposed are very similar, little selection is available. However, based on previous tests, the preformed insulation is recommended over the mastic. Insulation joints are also similar, and either concept would be acceptable.

Since insulation thicknesses vary with the grain design and burning time, they must be determined for each specific design. However, an insulation thickness safety factor of at least 1.5 is recommended.

8. Thrust Termination

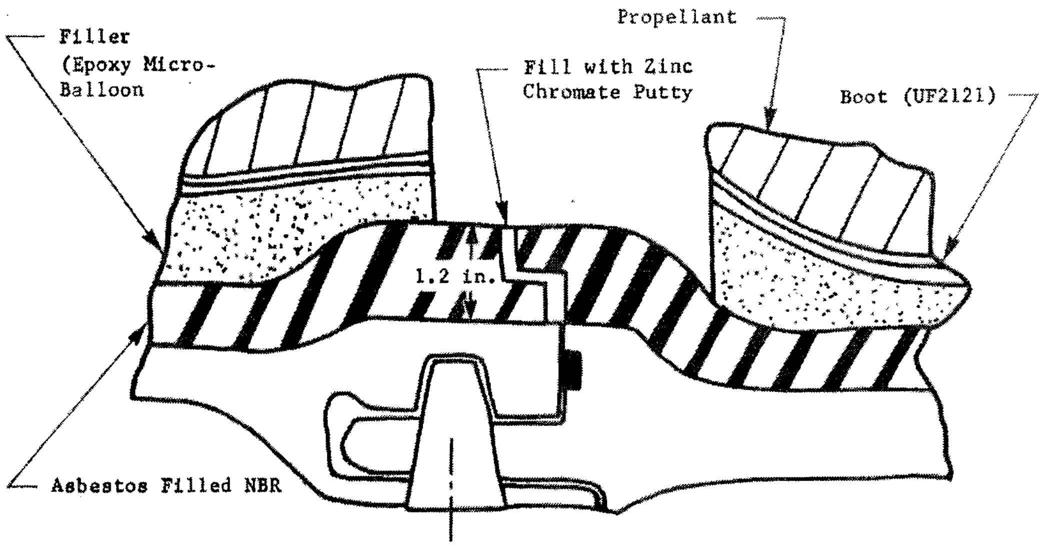
To enhance crew safety, the capability to terminate (or reduce) SRM thrust must be available.

To maintain core structural integrity, a net forward force of 100,000 lb must be applied to the core vehicle after thrust termination (Fig. VI-34). Figures VI-35 thru VI-37 show the results on Gemini B abort when the structural requirements are met.

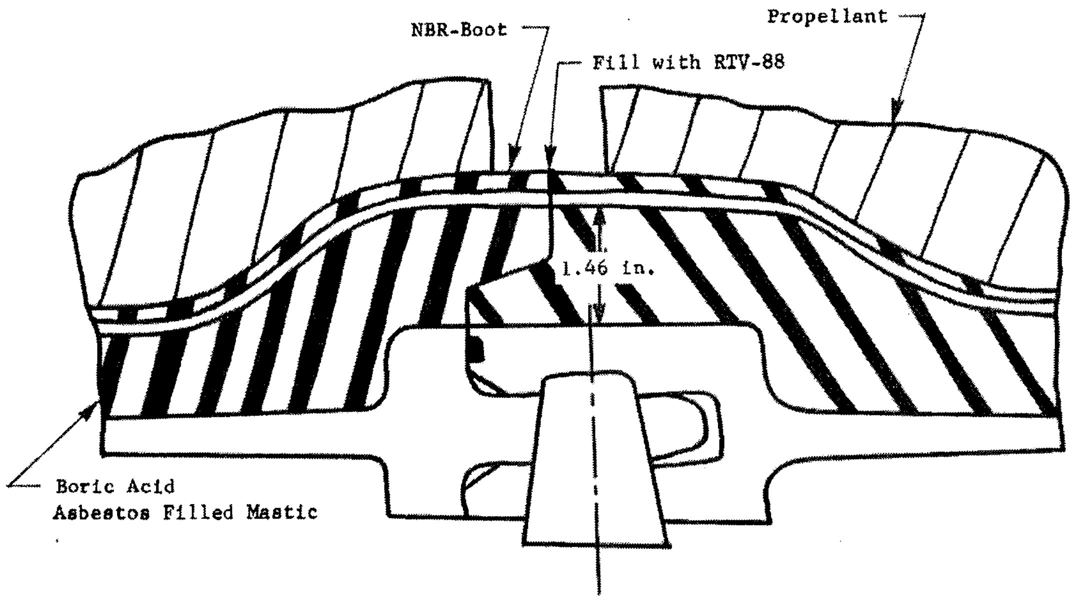
Port sizing for each SRM configuration will be set (at this time) as a function of the structural requirements. The port sizing analysis is presently being conducted.

Table VI-12 Proposed Insulation Systems

SRM Configuration	Contractor	Insulation Type				
		Forward Closure	Center Segment	Aft Closure	Safety Factor	Specification
7 seg-120	United Technology	Asbestos Fiber Silica Filled Butadiene Acrylonitrile	Silica Filled Butadiene Acrylonitrile	Carbon Reinforced Phenolic with Buna "N" Modifier	1.25	4 MDS-40717 4 MDS-40723
3 CS 156	United Technology	Asbestos - Silica Filled Buna "N"	Silica Filled Buna "N"	Asbestos-Silica Filled Buna "N"	1.25	4 MDS 40717 4 MDS-40723
	Lockheed Propulsion	Mastic Boric Acid - Asbestos Filled	Mastic Boric Acid - Asbestos Filled	Mastic Boric Acid - Asbestos Filled	2.25; 2.0; 2.25	IPL - 31
2 CS 156	Thiokol Chemical	Asbestos Filled NBR and Mastic	Asbestos Filled NBR and Mastic	Asbestos Filled NBR and Mastic	1.5	TI-H-704B
	Aerojet - General	Gen-Gard V 44	Gen-Gard V-44	Gen-Gard V-44	1.5	WS1138A Classi, Class II
	Lockheed Propulsion	Mastic	Mastic	Mastic	2.25; 2.0; 2.25	IPL-31



(a) Thiokol TU-502 Segment Joint



(b) Lockeed SRM Segment Joint

Fig. VI-33 Typical Proposed Insulation Joints, 156-in. Diameter

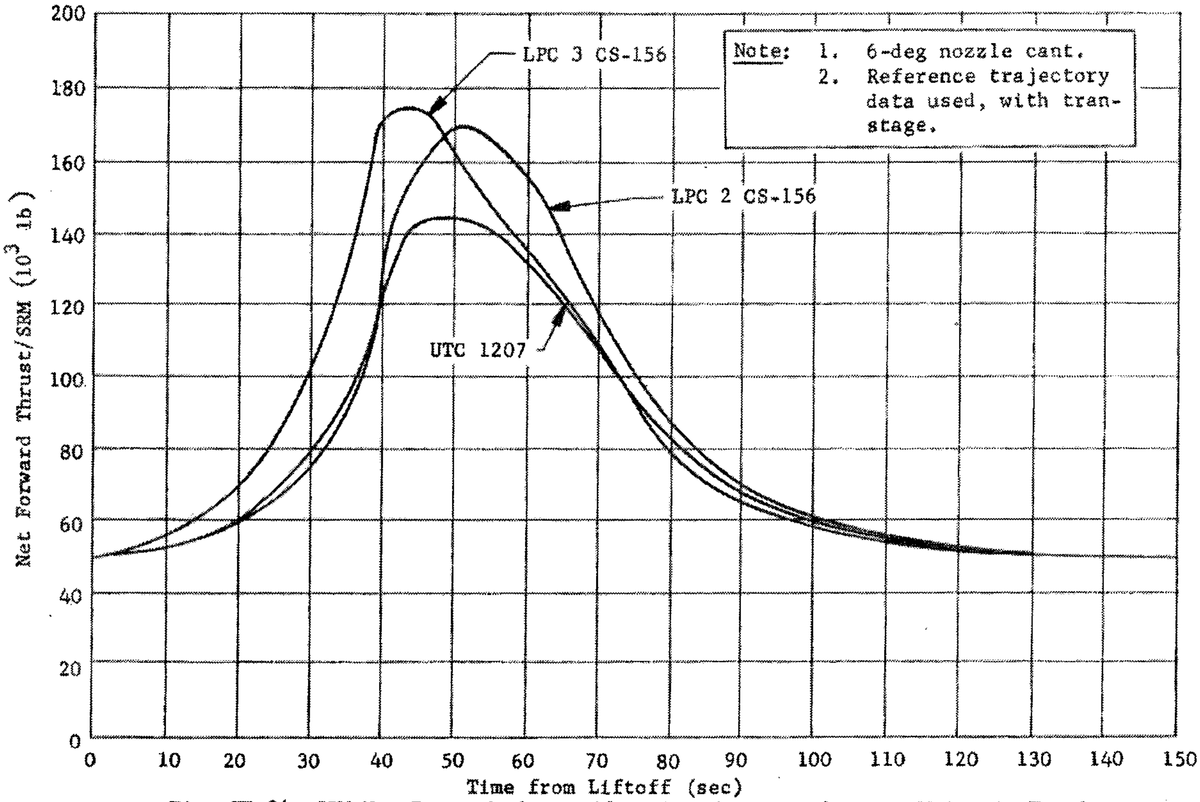


Fig. VI-34 SRM Net Forward Thrust Along Nozzle Centerline to Maintain Total Thrust Along Vehicle Centerline of 100,000 lb

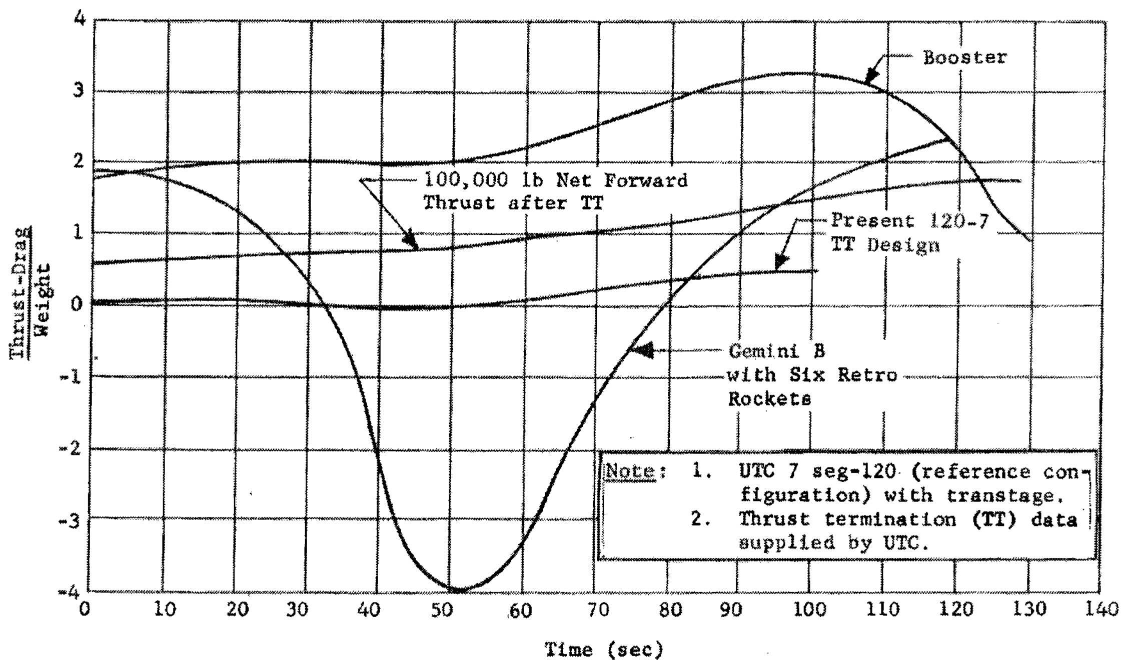


Fig. VI-35 Thrust Termination Evaluation, UTC 7 seg-120

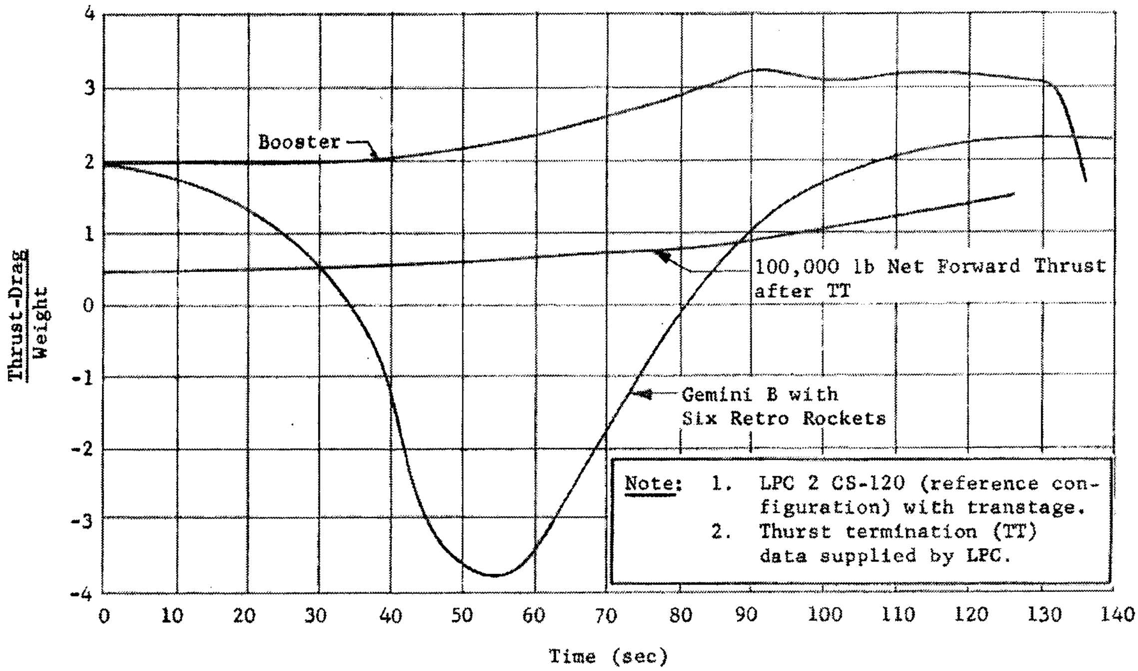


Fig. VI-36 Thrust Termination Evaluation, LPC 2 CS-156

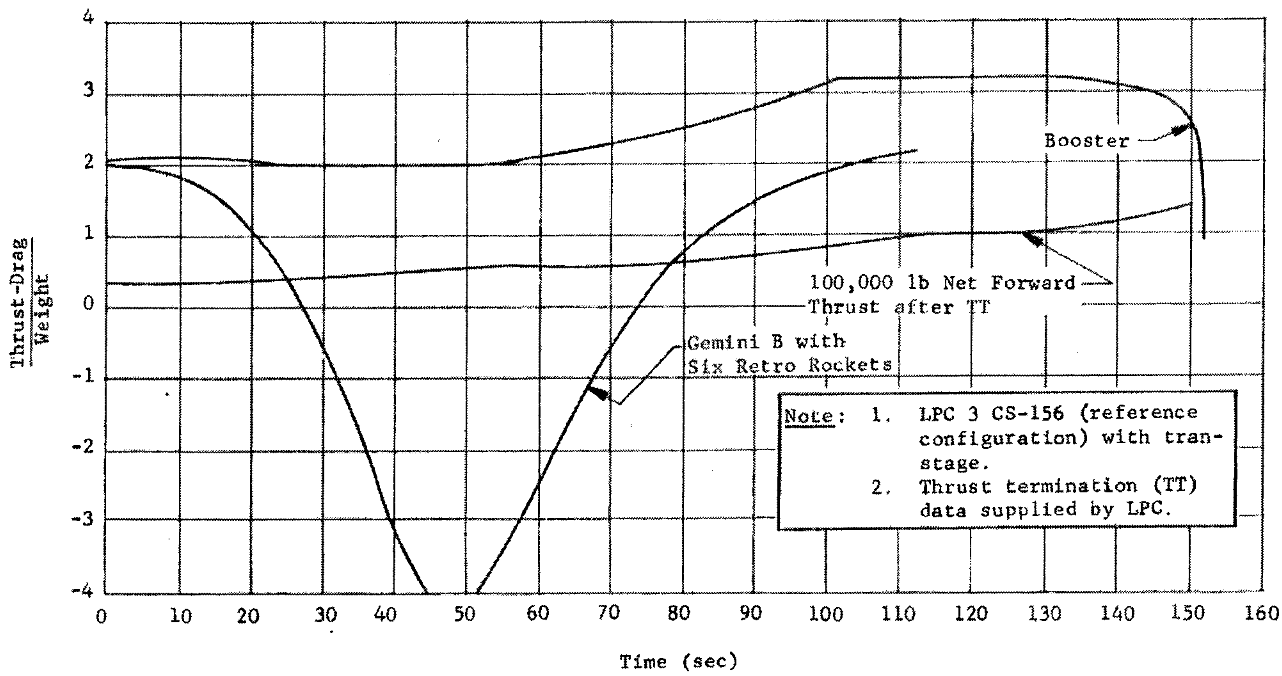


Fig. VI-37 Thrust Termination Evaluation, LPC 3 CS-156

9. Staging Rockets

Staging analyses show that the Titan IIIC staging rocket will provide the required thrust levels in all cases as defined in Table VI-13. The staging analysis is thoroughly discussed in Chapter V of this report.

Table VI-13 Staging Motors Required

	7 seg-120		2 CS-156		3 CS-156	
	Forward	Aft	Forward	Aft	Forward	Aft
Motors Required per SRM	4	4	5	4	5	4

VII. LIQUID PROPULSION SYSTEMS

The liquid propulsion system portion of this study consists of comparisons between the Titan IIIC core propulsion systems and a core using modified liquid engines. The primary difference is that the use of a modified YLR87-AJ-9 engine with a 15:1 expansion ratio is considered.

The original work statement specified the use of a 14:1 expansion ratio thrust chamber on the YLR87-AJ-9 engine which presently has an 8:1 expansion ratio. The expansion ratio considered in this study was subsequently changed to 15:1. All performance and weight data used in this study for the modified engine were therefore based on the 15:1 expansion ratio. Additional modifications including the YLR87-AJ-9 engine turbopump and injector have been proposed by Aerojet-General Corporation.

The Stage II engine, YLR91-AJ-9 and transtage engines, AJ10-138, are used in this study with no major modifications that affect the core design. Detailed engine changes have been proposed by Aerojet such as including the Gemini Stability Improvement Program (GEMSIP) injector on the Stage II engine and modifying the transtage engine propellant valve. These changes are also discussed in detail in subsequent sections.

Incomplete crew safety and reliability studies may impose additional engine changes. The most significant Martin change anticipated will be the requirement for redundant hydraulic control systems for engine gimbaling.

Several additional ground rules and assumptions were established to determine the scope and direction of the liquid systems comparison. They are:

1) Only the YLR87-AJ-9 engine with the 15:1 expansion ratio was used on Stage I with the 7 seg-120 SRMs;

2) Only the YLR87-AJ-9 engine with the 8:1 expansion ratio was used on Stage I with the 2 and 3 CS-156 SRMs for performance studies;

3) Both the 15:1 and 8:1 expansion ratio engines were used for staging studies with the 156 in. SRMs;

4) A nominal propellant temperature of 60°F was used for the WTR launch. A minimum of 45 and a maximum of 75°F was assumed;

5) A maximum temperature of 90°F was assumed for staging NPSH studies on both the 8:1 and 15:1 expansion ratio engines for an ETR launch;

6) Use of ground propellant temperature conditioning was not considered;

7) Nominal propellant loading as defined in Section D was assumed;

8) A minimum Stage I engine oxidizer NPSH of 35 ft for periods of approximately 4 sec was used for staging analysis.

A. LIQUID ENGINE DATA

A requirement was established at Martin contract go-ahead for Aerojet to supply data on the characteristics of the YLR87-AJ-9 engine with the modified 15:1 expansion ratio thrust chamber and on the proposed modified YLR87-AJ-9 and AJ10-138 engines. The 15:1 engine data were included with data on the present Titan III liquid engines and were used in the loads, performance, and staging studies. Table VII-1 summarizes and compares YLR87-AJ-9 engine characteristics with 8:1 and 15:1 expansion ratio thrust chambers. The plume profile for the improved 15:1 engine is shown in Fig. VII-1. Table VII-2 summarizes the YLR91-AJ-9 engine characteristics, and Table VII-3 summarizes the AJ10-138 engine characteristics.

Table VII-1 shows that the NPSH requirement for the 15:1 engine increases by 1 ft on both the fuel and oxidizer pumps, which results from a shift in the pump operating points. This number represents the minimum steady-state run condition.

Table VII-1 Stage I Engine Characteristics

Parameter	7 seg-120 with YLR87-AJ-9 (ϵ = 15:1)	2 CS-156 or 3 CS-156 with YLR87-AJ-9 (ϵ = 8:1)
Altitude Thrust (lb)	531,000 (nominal)	480,760 (nominal)
Altitude Specific Impulse (sec)	299 (nominal)	287.5 (nominal)
NPSH Requirement (ft)		
Oxidizer	45 (minimum)	44 (minimum)
Fuel	44 (minimum)	43 (minimum)
Installed Weight (lb)		
Wet Weight	4,654	3,921
Thrust Chamber Assembly (TCA) Cover	460	270
Aft Heat Shield	46	66
Gimbaled Moment of Inertia (slug-ft^2)	640	172
Overall Length (in. from aft frame)	150	125
External Exit Plane Diameter (in.)	63	44

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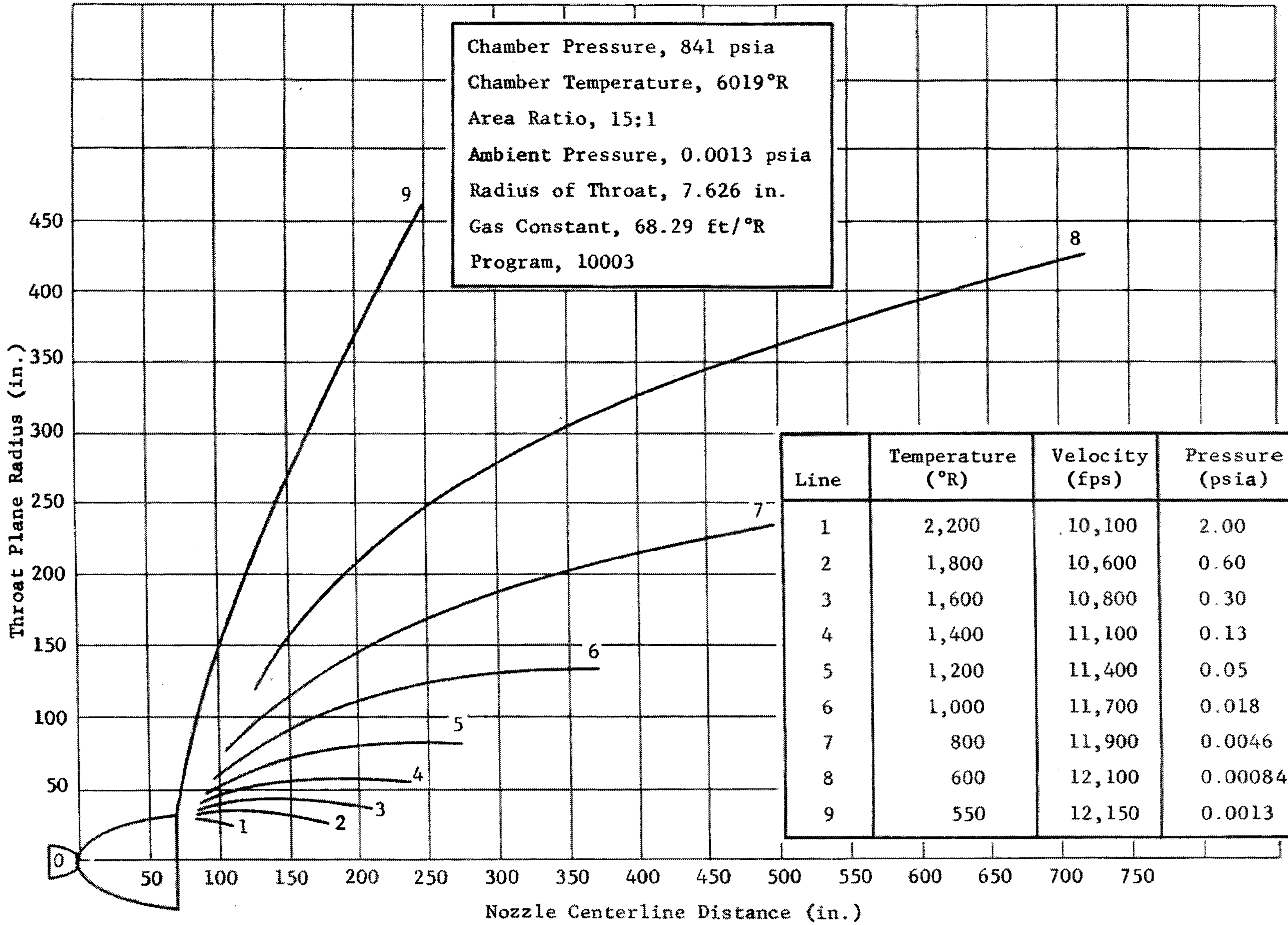


Line	Temperature (°R)	Velocity (fps)	Pressure (psia)
1	2,200	10,100	2.00
2	1,800	10,600	0.60
3	1,600	10,800	0.30
4	1,400	11,100	0.13
5	1,200	11,400	0.05
6	1,000	11,700	0.018
7	800	11,900	0.0046
8	600	12,100	0.00084
9	550	12,150	0.0013

Fig. VII-1 Titan III/MOL Exhaust Plume at 220,000 ft

[redacted]

Table VII-2 Stage II Engine Characteristics

Parameter	Value
Altitude Thrust (lb)	101,000*
Altitude Specific Impulse (sec)	310 (nominal)
NPSH Requirement (ft)	
Oxidizer	30
Fuel	100
Engine Wet Weight (lb)	1276 (nominal)
Gimbaled Moment of Inertia (slug-ft^2)	334 (nominal)
Overall Length (in.)	110.62 (max)
External Exit Plane Diameter (in.)	68.5 (max)

*Thrust Chamber Value:

$$I_{sp} = \frac{F_{TCA}}{W_{Total} - \text{Autogenous Flow}}$$

Table VII-3 Stage III Engine Characteristics

Parameter	Value
Altitude Thrust (lb)	8000 (nominal)
Altitude Specific Impulse (sec)	305 (nominal)
Engine Wet Weight (lb)	239.2 (max)
Gimbaled Moment of Inertia (slug-ft^2)	16.25 (max)
Overall Length (in.)	80.85 (max)
External Exit Plane Diameter (in.)	47.5 (max)

An analysis of staging conditions (Section C shows that for 90°F oxidizer temperatures the oxidizer NPSH for the Stage I engine falls below the required NPSH value for either type of engine during the staging sequence. Aerojet was requested to review the engine pump characteristics to determine the feasibility of running the engine below the specified NPSH for brief periods. The Aerojet analysis indicates the engines can run at an NPSH of 35 ft for approximately 4 sec, that should be sufficient to pass through the staging transient with no serious consequences to the engine. Additional testing would not be required to demonstrate satisfactory pump operation at the low NPSH with the 15:1 engine, since other development testing is anticipated. Engine and pump demonstration tests would be required if the 8:1 configuration is used. Additional solutions to the NPSH problem are described next.

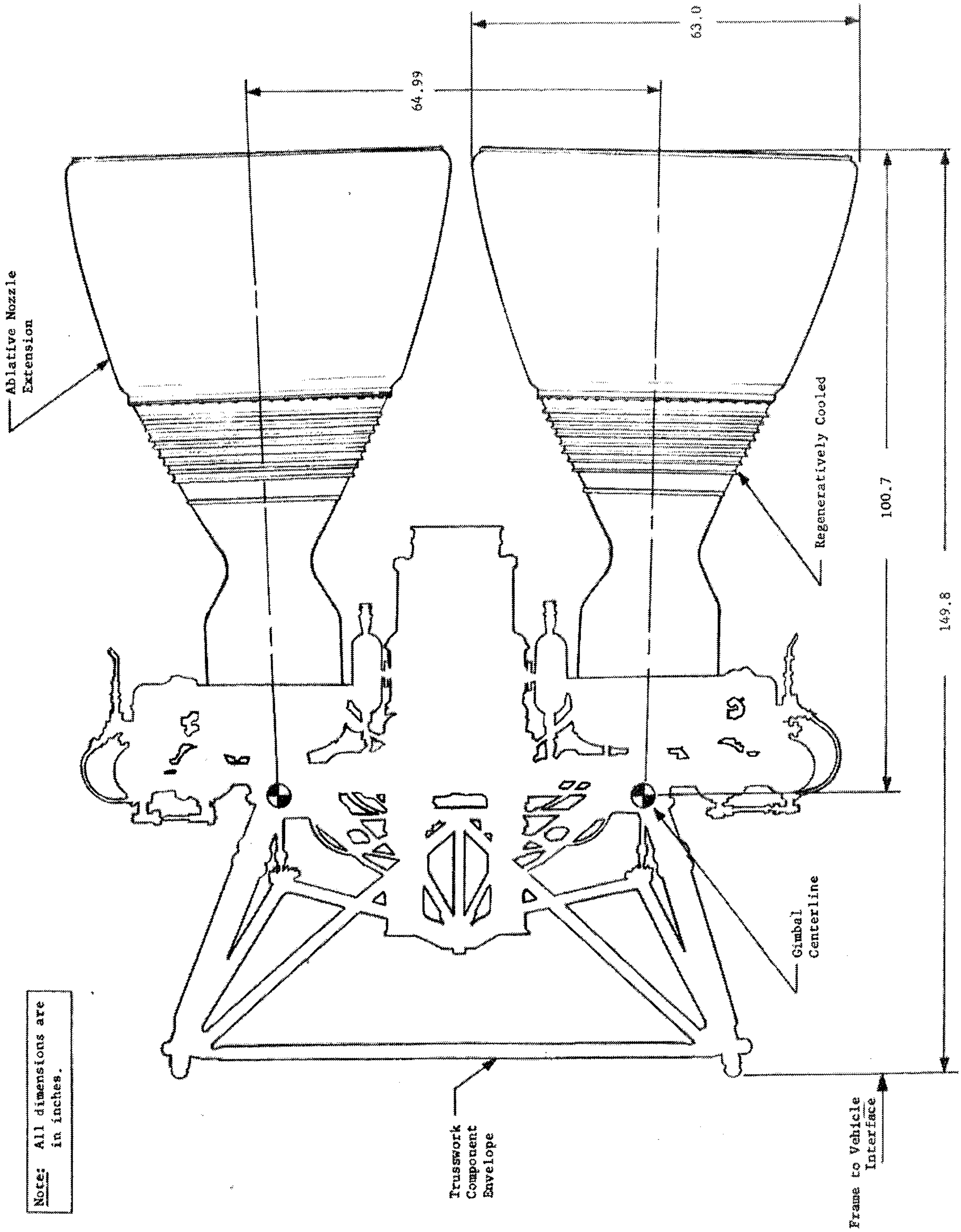
Engine I_{sp} and thrust defined in Table VII-1 for the 15:1 engine are nominal for an engine calibrated for 60°F propellants and having standard inlet conditions* with flight corrections applied. The corresponding values for the 8:1 engine are nominal at flight conditions also.

The 15:1 engine overall dimension in Table VII-1 is measured from the Aerojet/Martin mounting frame interface to the exit plane of the thrust chamber. It does not include the TCA cover.

Figure VII-2 shows the 15:1 engine assembly. The thrust chamber gimbal attachment point and toe-out angle are identical to the 8:1 engine. A 2-in. clearance between thrust chambers, when in the neutral position, is achieved when an identical mounting is used for increased expansion ratio chambers. A device for maintaining thrust chamber nozzle separation during Stage 0 boost-flight loads should be provided. A link connecting the thrust chamber exit protective covers could be used. Figure VII-3 shows that when the thrust chambers are gimbaled hardover in the pitch plane, the chamber extends as much as 10 in. beyond the 10-ft dia of the core.

The 15:1 engine start transient side loads into the actuators during altitude start have been estimated by Aerojet to be 21,000 lb. Neither the start transient side force nor the aerodynamic loads resulting from the thrust chambers protruding beyond the core skin line represent a problem for the MOL mission. However, these two conditions could necessitate the redesign of the hydraulic actuators if future missions indicate the need for a core sea level launch.

*Oxidizer pump inlet pressure is 75 psia, fuel pump inlet pressure is 35 psia, and a 1-g condition exists.



Note: All dimensions are in inches.

Fig. VII-2 Preliminary Stage I Titan III/MOL Engine Assembly

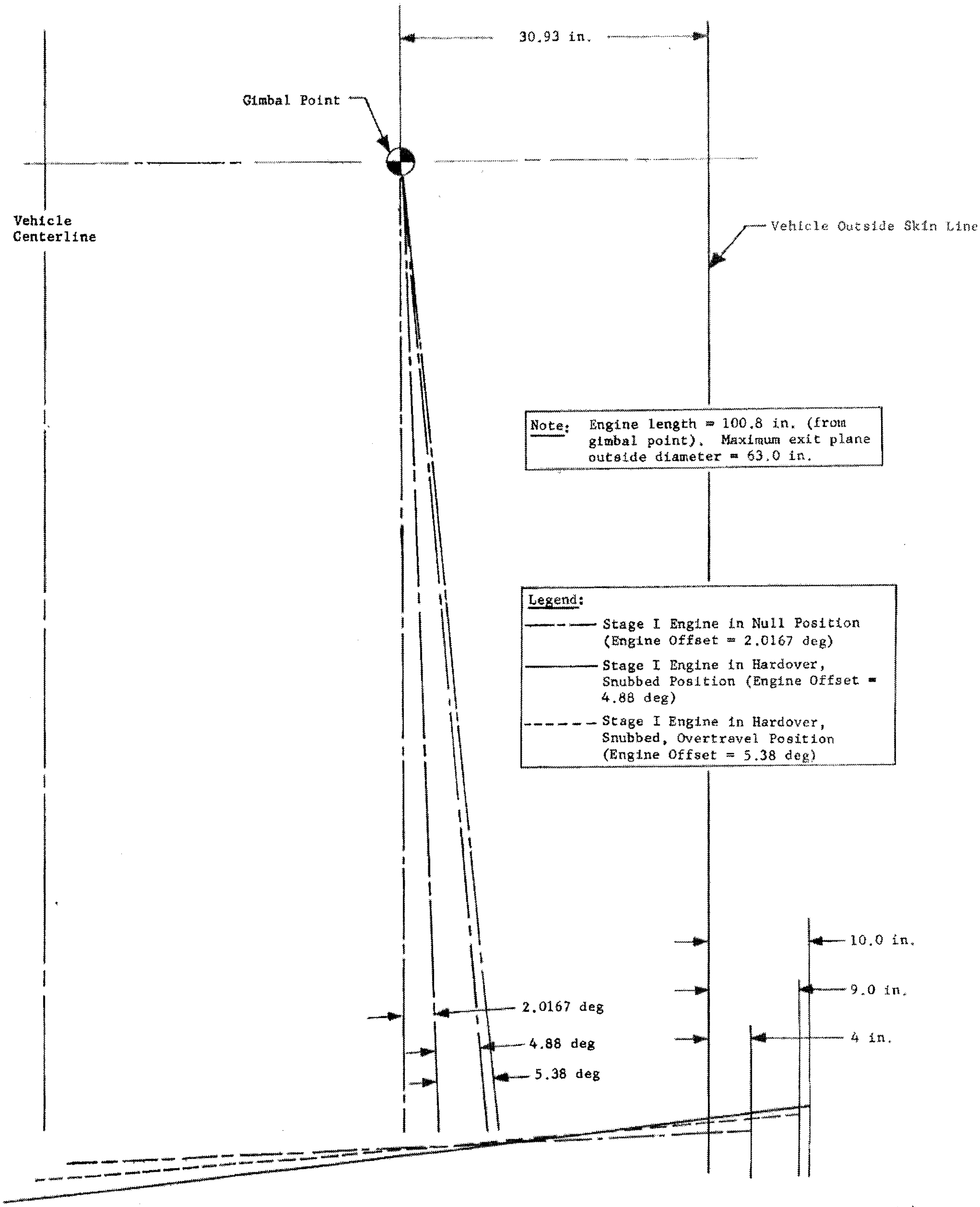


Fig. VII-3 Titan III/MOL Stage I Engine Exit Plane/Vehicle Outside Skin Line Layout (15:1 Expansion Ratio)

B. LIQUID ROCKET ENGINE CHANGES

Many liquid rocket engine (LRE) improvements have been proposed by Aerojet. These changes would provide (1) increased reliability and crew safety, (2) improved payload capability, (3) ability to meet Titan III/MOL environmental and staging requirements, and (4) a solution to manufacturing and maintainability problems. No changes have been considered for the Stage I 8:1 engine, but redundant hydraulic control alternatives now being studied could result in changes to the engine.

1. Stage I Engine

Dynamically Stable Injector - This modification will incorporate experience gained from the GEMSIP chamber. The injector will include baffles that may be regeneratively or tip-injection cooled and will improve the injection pattern. The injector will provide a more stable combustion margin and should not be a high-risk development item since previous experience is being used.

Gearbox High-Speed Shaft Redesign - This modification will remove the resonant and operational frequency of the high-speed shaft assembly from a high load amplification region and increase the load-carrying capacity of the bearing. These improvements will be accomplished by changing the bearings and their arrangement on the shaft and by changing the shaft diameter and overhang. This change and the associated testing should not be a large technical risk, since the associated changes are common mechanical design engineering problems.

Improved Regeneratively Cooled Chamber - Since the thrust chamber design is being changed to incorporate an ablative skirt at an area ratio of 6:1 and extending to 15:1, it is desirable to redesign the tube bundle to achieve decreased fuel pressure drops and increased propellant temperature capability. The redesign will provide a propellant launch temperature capability of 90°F with an R_{bo} of 0.80.* Design of a new thrust chamber is completed, and Aerojet's experience in development of regeneratively cooled chambers should result in little program technical risk in making this change.

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*R_{bo} is the ratio of the tube burnout heat flux that will occur under given run conditions to that burnout heat flux theoretically established by design characteristics. This ratio is an indicator of the probability of tube burnout when operating an actual engine with real components.

Redundant Engine Shutdown System (RESS) - The possibility exists of a single malfunction of the pressure sequencing valve (PSV) system during a command shutdown requirement. This malfunction could cause catastrophic results if the RESS is not provided. Presently, the engine is shut down by a signal to the PSV override solenoid valves. These valves in turn vent the PSV actuation pressure overboard and permit spring shuttling of the spool. This action closes the thrust chamber valves.

A RESS is proposed that consists of a squib-actuated normally open shutoff valve in each subassembly. The valve would be located in the oxidizer bootstrap line upstream of the gas generator oxidizer cavitating venturi and would close on receipt of a signal (the same signal that goes to the PSV override solenoids). If the PSV override fails to operate, the squib valve stops oxidizer flow to the gas generator resulting in turbine inlet pressure decay and turbopump slow down. When fuel pump output pressure decays to approximately 300 psi, the spring closing feature in the fuel valve actuator will cause the main propellant valves to close completing the engine shutdown sequence in a known manner. The squib valve assembly weighs approximately 2½ lb and should not present a risk, since it has been successfully used on the Stage II engine of Gemini.

Improved Vaned Elbows - The propellant inlet vaned elbows have historically shown distortion and cracking following engine shutdown. Although no structural failures occurred, the potential loss of a vane or portion of a vane could result in downstream blockage of a combustion chamber tube, injector passage, or orifice. It is proposed that the elbows be redesigned to eliminate vane distortion and cracking. This redesign and testing would provide added confidence in the engine system and should not cause an undesirable technical program risk.

Altitude Nozzle Extension - This modification will provide added thrust and specific impulse performance for altitude operations of the engine. A glass fabric-wrapped honeycomb structure with a silica phenolic liner is proposed similar to the Titan III Stage II nozzle extension. Design analysis is completed and the redesigned nozzle extension is shown in Fig. VII-4. Experience with ablative skirt testing and development at Aerojet indicates a negligible program risk from redesign of this item.

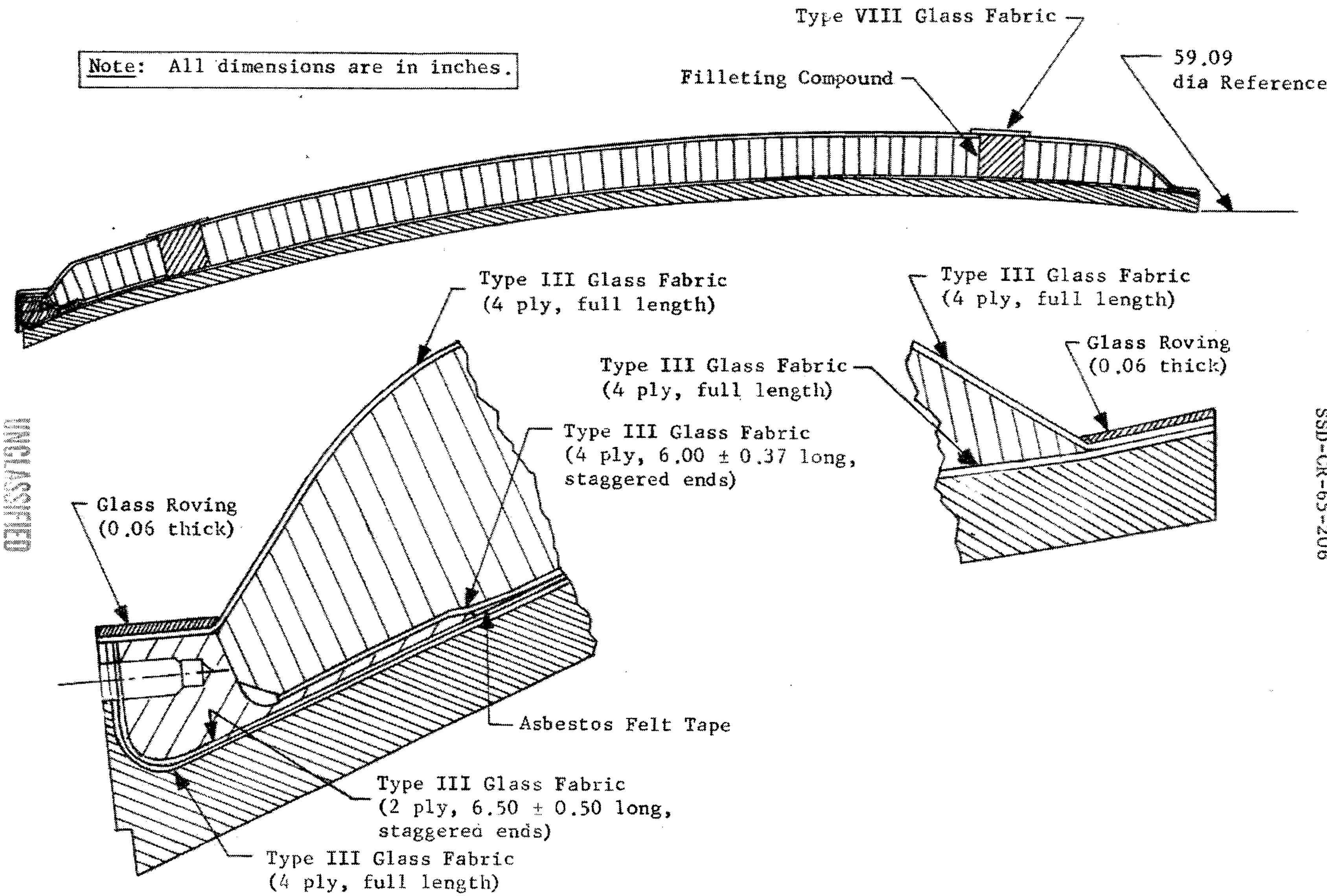

Fig. VII-4 15:1 Stage I Engine Ablative Skirt

Optimized Fuel Pump - A decrease in the regeneratively cooled tube jacket pressure drop in conjunction with a trimmed fuel pump impeller and a slight increase in turbine speed can provide additional thrust to the engine with no increase in turbine horsepower. Thus, an impeller trim has been proposed with the required impeller discharge vane angle change. The impeller housing would not be modified. Figure VII-5 shows the proposed impeller trim. Some technical risk is involved with this modification, but theoretical analysis indicates only small changes will be required. Similar development has not been conducted by Aerojet, but because of small design variation, the program risk should be small.

Subcomponent Modifications - Other small changes have been proposed, e.g., incorporation of more corrosion-resistant materials in the gas cooler and superheater, a start cartridge temperature compensating nozzle, mechanically locked turbine blades, etc, which should be given further consideration. These items could reduce maintenance costs and improve reliability. The technical risk associated with incorporating most of these items should be small.

2. Stage II Engine

Similar improvement items have been suggested for the Stage II engine but are limited to the following:

1) Incorporation of GEMSIP injector;

2) Incorporation of augmented engine improvement program (AEIP) items;

3) Improved combustion chamber;

4) Redundant engine shutdown system;

5) Subcomponent improvements.

These changes have been made on the Gemini launch vehicle except for the AEIP improvements, improved combustion chamber, and the subcomponent improvements. The AEIP items were evaluated early in the Titan III program and some were developed, thus, in making these changes, the program technical risk is minimal. The combustion chamber change should require only a small variation in tube size with a resulting improvement in high-temperature propellant operation. Little program risk is anticipated. Subcomponent improvements are characteristic of the Stage I subcomponent improvements and are considered to be of equally low risk.

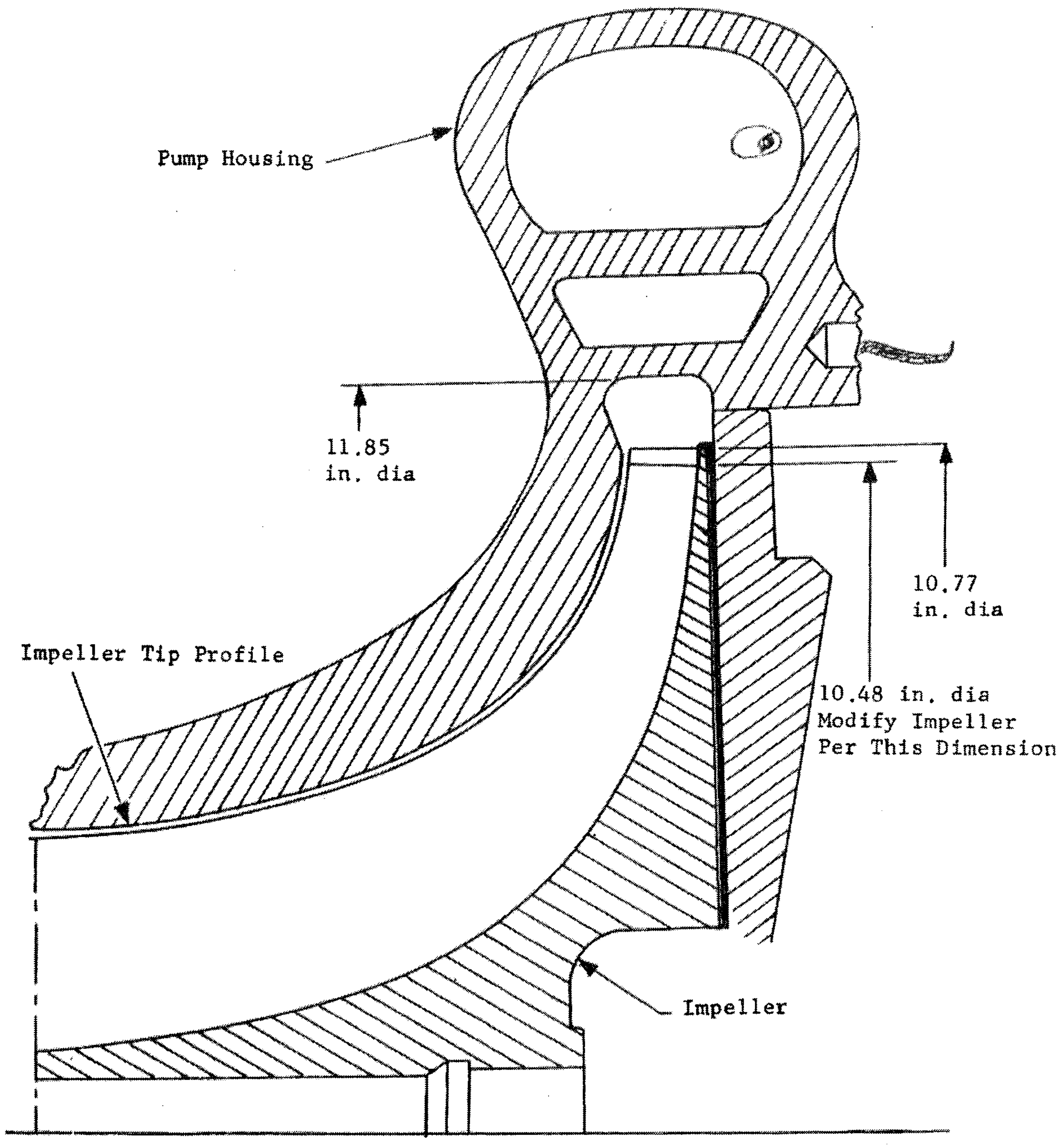

Fig. VII-5 Proposed Improved Fuel Pump

3. Stage III Engine

Studies conducted by the transtage engine manufacturer have indicated a need for only one change. A modification of the propellant thrust chamber control valve to improve engine start and shutdown transient characteristics is desirable. A single pilot valve to control both engines has been proposed using a single hydraulic actuation pressure source. Reduction in system dynamic interaction may be accomplished with this independent pressure source. An electrically actuated system also was considered, but a hydraulic system appears to be most desirable. Since the transtage engine characteristics are acceptable with its current propellant control valve, little program risk will be associated with further testing and development of this item.

It may be desirable to provide a RESS on Stage III, as well as Stages I and II, for redundancy. The addition of a RESS should be studied further.

Some of the engine modifications are not directly related to performance improvement but to crew safety or reliability. All modifications must be evaluated to arrive at a valid overall assessment of engine changes.

C. STAGE I/SRM STAGING NPSH STUDY

1. Study Criteria

The Stage I minimum interface NPSH required is tabulated below.

Engine Expansion Ratio	Fuel	Oxidizer
8:1	43 ft	44 ft
15:1	44 ft	45 ft

The two major differences between Titan III and Titan III/MOL that affect NPSH are decreased acceleration during staging because of larger solids and increased NPSH required by the 15:1 engine. The increase in NPSH requirements is relatively insignificant. The decrease in acceleration results in a decrease in the Stage I oxidizer NPSH of as much as 15 ft for the 3 CS-156 configuration.

Because of the placement of the fuel tank with respect to the engine, acceleration effects on NPSH are less significant than in the case of the oxidizer tank. Also, more margin exists between required and supplied NPSH, and therefore, the fuel tank can meet the required NPSH criteria for all Titan III/MOL vehicle configurations with no modifications.

2. NPSH Improvement Items

The following changes were considered to provide increased Stage I oxidizer NPSH.

Reduction in the Stage I Engine Oxidizer NPSH Required - A study of existing pump, TPA, and engine data gathered by Aerojet includes significant test history of oxidizer pump operation below minimum NPSH. These data are sufficient to conclude that TPA component reliability and performance will not be significantly degraded at 35 ft oxidizer NPSH or above for short periods. Engine tests at low oxidizer NPSH do not indicate any adverse effects on system reliability or performance. However, additional demonstration tests under Titan III/MOL conditions will be required.

Increased Oxidizer Propellant Tank Lockup Pressure - A study performed by the Martin Company's Stress Group shows the maximum tank pressure before liftoff can be increased 7 psi, thereby increasing NPSH at the critical staging point by approximately 5.0 ft. This study was based on a maximum acceleration during solid motor burn of 3.2 g. This increase requires only a change in the launch limit pressure switch settings.

Increased Minimum Oxidizer Ullage - A small increase in ullage accompanied by a shift in mixture ratio could result in approximately 1-ft increase in NPSH at the critical staging point, but further ullage increases would result in a significant payload penalty.

Oxidizer Autogenous Gas Flow Box Change - This would require a redesign of the Stage I superheaters to gain significant NPSH increases resulting in payload penalty due to increased residual gas weight.

Reduction in the Maximum Propellant Temperature Limit - This reduction would slightly reduce the launch-on-time probability at launch sites requiring maximum propellant temperatures of 75°F. Propellant conditioning would be required for launch sites requiring maximum propellant temperatures of 90°F.

Gas Injection System to Pressurize the Oxidizer Tank during Staging - Studies showed that this would be technically feasible, but the development required, cost, weight, operational problems, and the possibility of accidental pressurization resulting in pressures approaching structural limits make this change practical only if all other solutions fail.

3. Stage I Oxidizer NPSH Study Results and Recommended Solution

A study has been performed to determine the Stage I oxidizer NPSH supplied for the different Titan III/MOL vehicle configurations. This study was limited to three basic SRM configurations since the acceleration effect on NPSH varies less than 1% with and without transtage. Staging acceleration data for the three configurations were studied.* The autogenous pressurization system 7094 machine program was used to generate minimum oxidizer tank top pressure curves. The following assumptions were input to the program and were based on Titan III flight data.

1) Autogenous gas was supplied at the minimum point on the Titan III Martin/Aerojet interface flow box. This point was reached at a steady-state condition at $87FS_1$ + 1.3 sec for the gas flow and 10 sec for enthalpy;

2) Steady-state propellant flow was reached at $87FS_1$ + 1.5 sec;

3) The minimum tank pressure at $87FS_1$ was 34.7 psia. This included the minimum lockup pressure less the effect of pressure decrease due to prevalve opening and tank pressure decay during Stage 0 burn;

4) Minimum pressure at $87FS_1$ was increased to 41.7 psia to include the allowable 7-psi increase in lockup pressure;

5) Minimum ullage of 85 cu ft was present at $87FS_1$.

The acceleration data from TM 5141/31-65-19 and tank top pressure data were used to determine NPSH supplied for the conditions tabulated on the following page.

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*D. Bressler, S. Bonson, W. Livesey: MOL Compatibility Solid Rocket Motor Staging Analysis Report. TM 5141/31-65-19. Martin Company, Denver, Colorado, 1965.

Configuration	Engine Expansion Ratios	Propellant Temperature (°F)	Propellant Flow Rate (cu ft/sec)	Lockup Pressure (psia)
7 seg-120	15:1	75	12.50	37
7 seg-120	15:1	90	12.55	37
2 CS-156	8:1	75	12.05	37
2 CS-156	8:1	90	12.10	37
3 CS-156	8:1	75	12.05	37
3 CS-156	8:1	90	12.10	37
3 CS-156	8:1	90	12.10	44

Figures VII-6 thru VII-8 and VII-10 show that a reduction in required NPSH satisfies all configurations at a maximum propellant temperature of 75°F and the 7 seg-120 SRM configuration with 90°F propellants. Figure VII-9 shows the 2 CS-156 SRM configuration with 90°F propellants, which drops slightly below the 35-ft NPSH-required curve. Figure VII-11 shows the 3 CS-156 SRMs with 90°F propellants. NPSH supplied falls significantly below the 35 ft required. Figure VII-12 shows this same configuration with increased lockup pressure. The figure reflects that minimum NPSH for the worst case configuration can be met by decreasing the required minimum NPSH and increasing lockup pressure.

4. Conclusion

The only three items that significantly decrease the difference between the supplied and required NPSH are decreased required NPSH, increased lockup pressure, and the installation of a gas injector system. None of these represents appreciable technical risk. The first two do not require major redesign, therefore, the recommended solution is to decrease NPSH requirements for the 7 seg-120 SRMs and decrease NPSH and increase lockup pressure if the 2 and 3 CS-156 SRM configurations are selected.

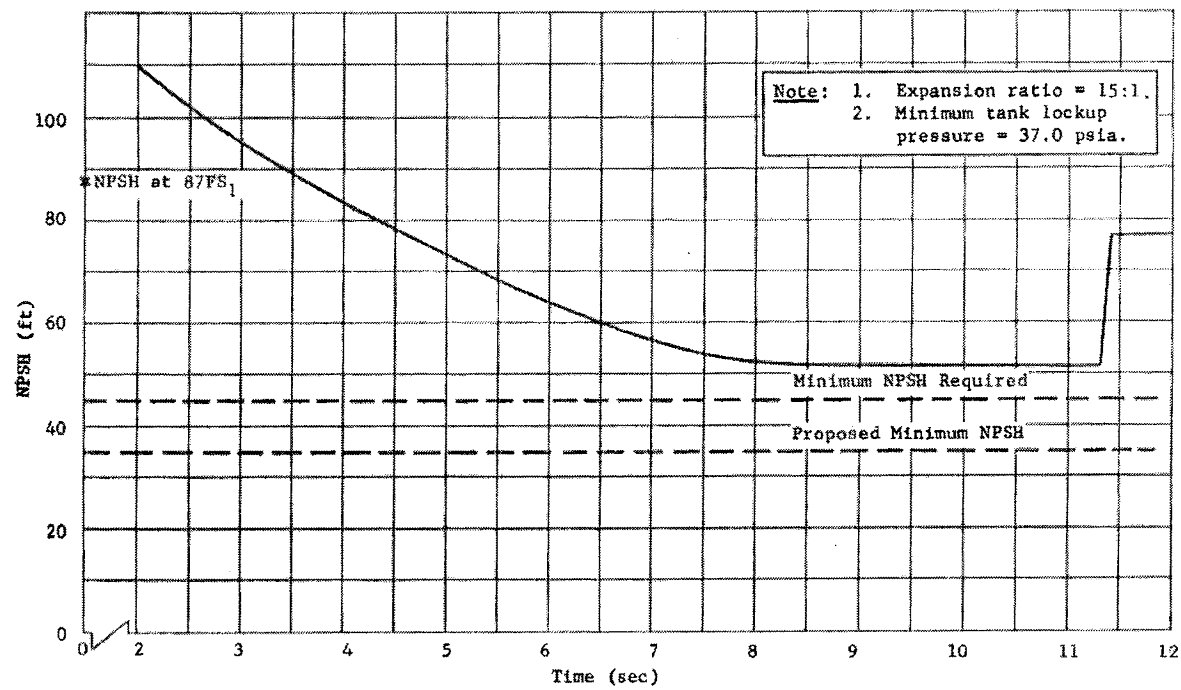

Fig. VII-6 Titan III/MOL Stage I Oxidizer Minimum NPSH during SRM Staging
(7 seg-120; 75°F Propellant Temperature)

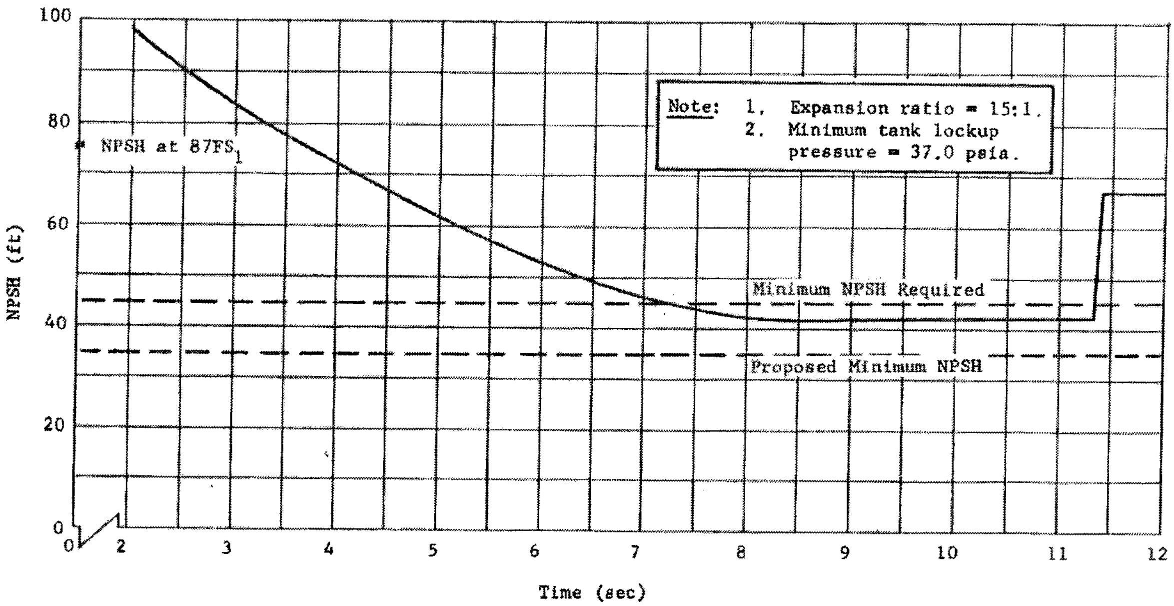

Fig. VII-7 Titan III/MOL Stage I Oxidizer Minimum NPSH during SRM Staging
(7 seg-120; 90°F Propellant Temperature)

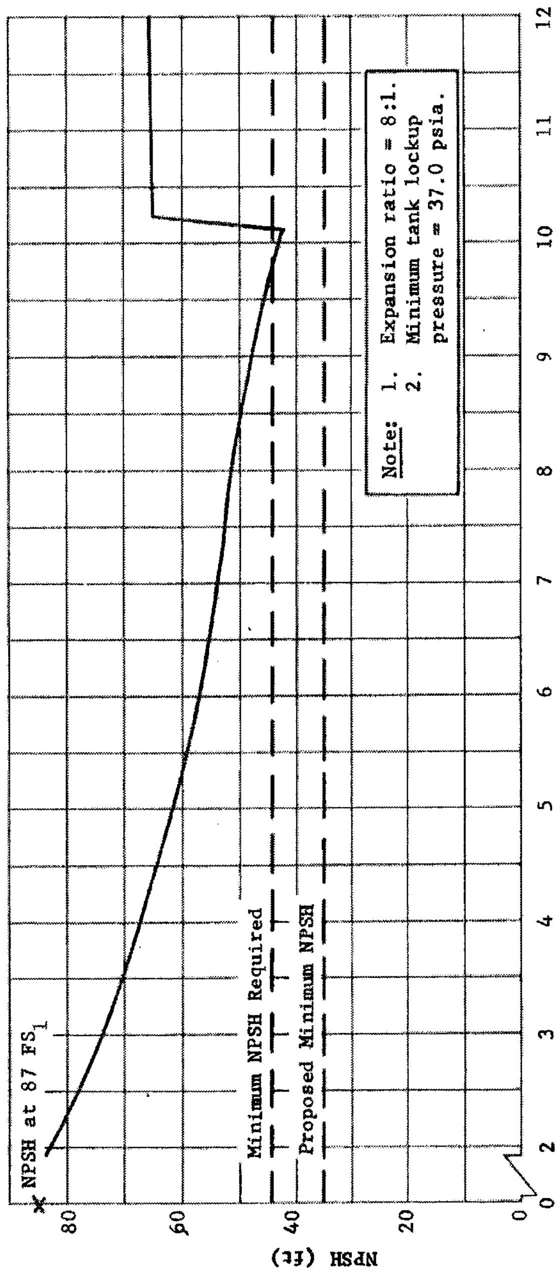

Fig. VII-8 Titan III/MOL Stage I Oxidizer Minimum NPSH during SRM Staging (2 CS-156; 75°F Propellant Temperature)

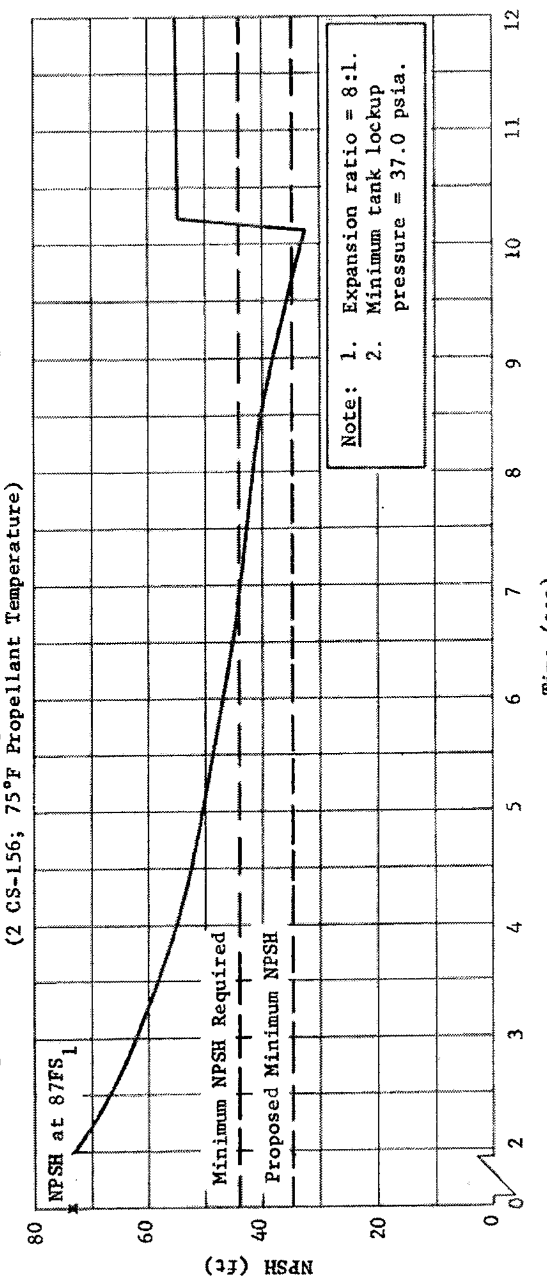

Fig. VII-9 Titan III/MOL Stage I Oxidizer Minimum NPSH during SRM Staging (2 CS-156; 90°F Propellant Temperature)

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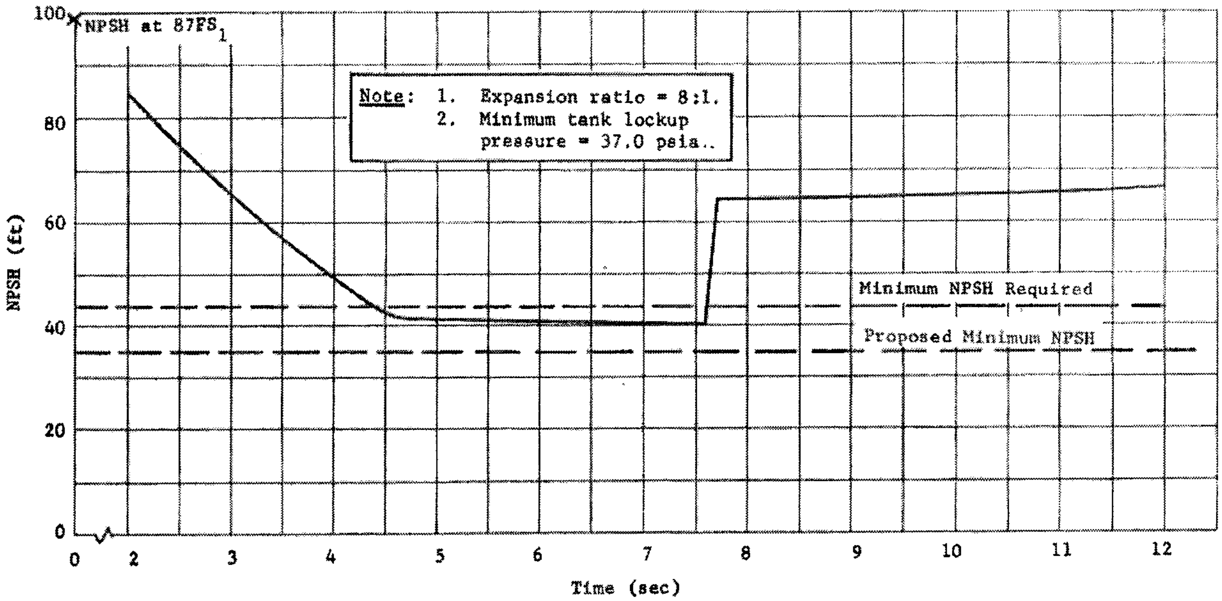

Fig. VII-10 Titan III/MOL Stage I Oxidizer Minimum NPSH during SRM Staging
(3 CS-156; 75°F Propellant Temperature)

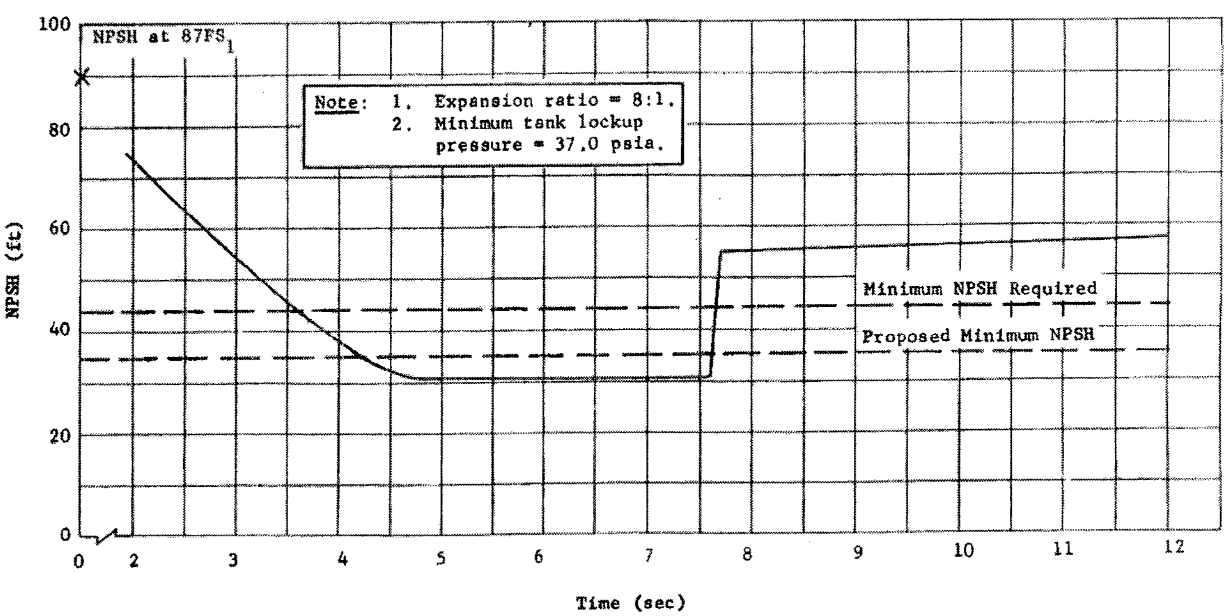

Fig. VII-11 Titan III/MOL Stage I Oxidizer Minimum NPSH during SRM Staging
(3 CS-156; 90°F Propellant Temperature)

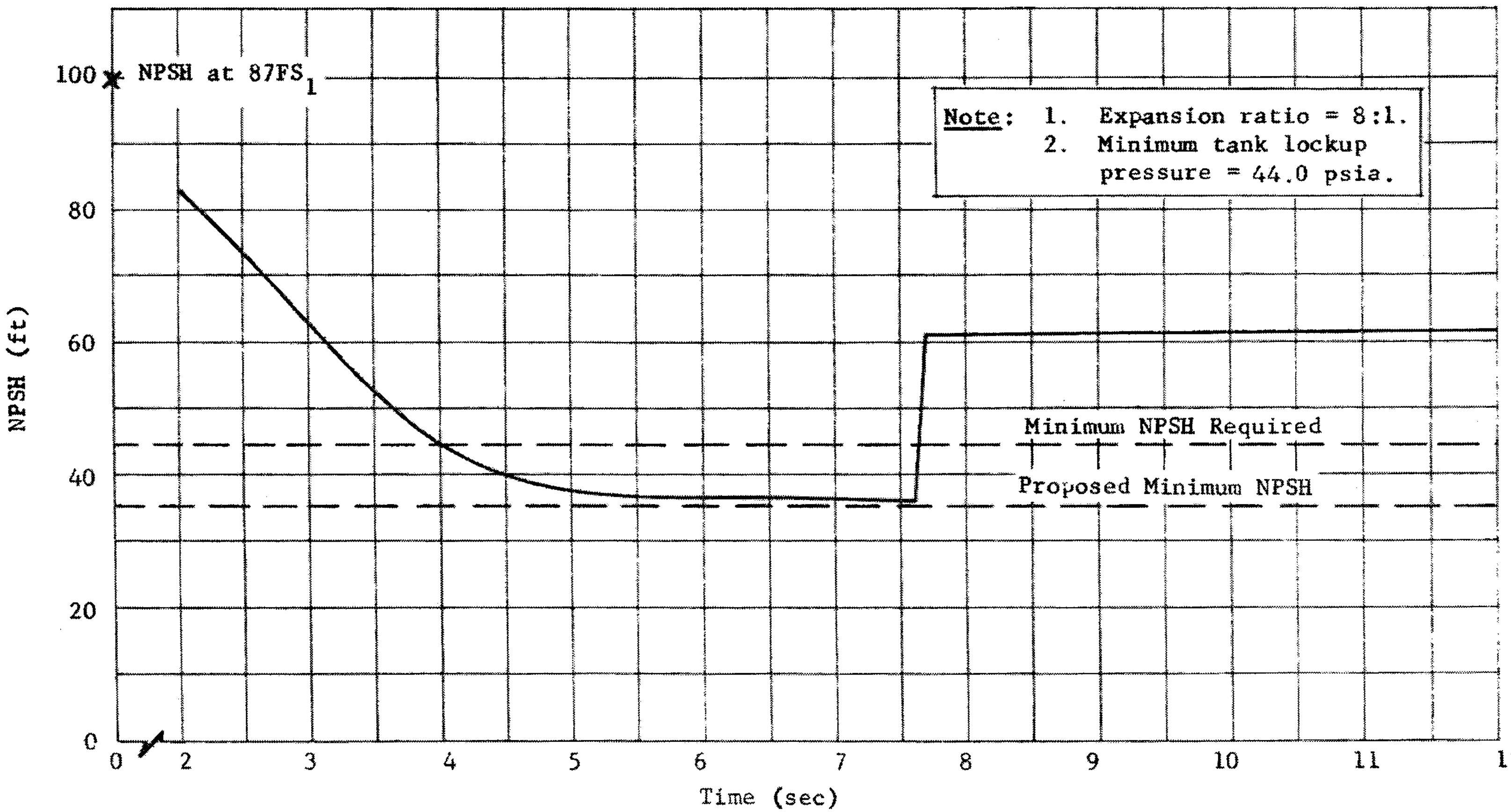

Fig. VII-12 Titan III/MOL Stage I Oxidizer Minimum NPSH during SRM Staging (3 CS-156; 90°F Propellant Temperature; Increased Lockup Pressure)

D. PROPELLANT LOADING INVENTORY

A detailed breakdown of the propellant loading inventory is given in Tables VII-4 and VII-5. Two transtage loads are given; Table VII-4 reflects the off-loaded case and Table VII-5 the fully loaded case. The calculation of the loading is basically the same as that used in the Titan III program. The following criteria were used in the analysis.

1) The engine nominal mixture ratios were selected to maximize the propellant loads;

2) Since Stage I and II loads are a function of propellant temperature, a nominal temperature range had to be selected. The ranges of propellant temperature blocks for this study are 40 to 55, 50 to 65, and 60 to 75°F. Therefore, for this study the nominal propellant load is based on the 50 to 65°F range;

3) All other figures used represent the latest information available from the Titan III program.

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Table VII-4 Liquid Propellant Inventory Titan III/MOL (Transtage Off-Loaded)

Item		Stage I	Stage II	Stage III
1. Engine Number		--	--	--
2. Average Inflight Mixture Ratio		1.91	1.806	2.00
3. Propellant Temperature (°F)	Fuel / Oxid	50 to 65	50 to 65	45 to 75
4. Propellant Density	Fuel / Oxid	--	--	--
5. Maximum Loadable Volume (cu ft)	Fuel	1,571.87	430.45	--
	Oxid	1,864.37	482.06	
6. Nominal Propellant Loaded (lb)	Total	257,390	67,909	16,102
	Fuel	88,795	24,316	5,381
	Oxid	168,595	43,593	10,721
7. Propellant Expended before Liftoff (lb)	Total	6	2	0
	Fuel	6	2	0
	Oxid	0	0	0
a. Engine Bleed	Fuel	6	2	0
	Oxid	0	0	0
b. Engine Leakage	Fuel	0	0	0
	Oxid	0	0	0
c. Start Consumption, $87FS_1$ to TCPS	Fuel / Oxid	N/A	N/A	N/A
d. Holddown Consumption, TCPS to Liftoff	Fuel / Oxid	N/A	N/A	N/A
8. Propellant Aboard at Liftoff	Total	257,384	67,907	16,102
	Fuel	88,789	24,314	5,381
	Oxid	168,595	43,593	10,721
9. Propellant Expended during Previous Stage Operation	Total	29	15	0
	Fuel	28	15	
	Oxid	1	0	
a. Engine Bleed, Stage 0 Operation	Fuel	28	6	
	Oxid	0	0	
b. Engine Leakage, Stage 0 Operation	Fuel	0	0	
	Oxid	1	0	
c. Engine Bleed, Stage I Operation	Fuel	N/A	9	
	Oxid		0	
d. Engine Leakage, Stage I Operation	Fuel	N/A	0	
	Oxid		0	
10. Propellant Aboard at $87FS_1$	Total	257,355	67,892	16,102
	Fuel	88,761	24,299	5,381
	Oxid	168,594	43,593	10,721
11. Engine Leakage during Stage Operation	Total	12	8	0
	Fuel	11	7	0
	Oxid	1	1	0
12. Total Available Usable Propellant	Total	256,208	67,480	16,088
	Fuel	88,304	24,142	5,380
	Oxid	167,904	43,338	10,708
a. Start Consumption	Fuel	43	55	2
	Oxid	205	138	3
b. Steady-State Consumption	Fuel	87,863	23,912	5,350
	Oxid	167,482	43,082	10,700
c. Shutdown Consumption	Fuel	113	57	3
	Oxid	217	94	5
d. Tailoff (before Staging)	Fuel	N/A	22	N/A
	Oxid		24	
e. Fuel Bias	Fuel	285	96	25
13. Total Nonusable Propellant	Total	1,135	404	14
	Fuel	446	150	1
	Oxid	689	254	13
a. Propellant Vapor Retained	Fuel	99	60	1
	Oxid	515	193	13
b. Trapped above Interface	Fuel	50	61	0
	Oxid	0	23	0
c. Trapped below Interface	Fuel	297	29	0
	Oxid	174	38	0
14. Mean Outage	Total	548	215	50
15. Nominal Propellant Consumed during Stage Operation (Item 12 - 14)	Total	255,660	67,265	16,038
a. Nominal Steady-State Propellant (Item 12b - 14 + 12e)	Total	255,082	66,875	16,025
b. Transient Propellants	Total	578	390	13
16. Pressurization System Inert Gas	Total	24	11	60
	Fuel	9	6	7
	Oxid	15	5	8
	Spheres	N/A	N/A	45
17. Maximum Outage		2,100	675	161

[redacted]

Table VII-5 Liquid Propellant Inventory Titan III/MOL (Transtages Fully Loaded)

Item		Stage I	Stage II	Stage III
1. Engine Number				--
2. Average In-Flight Mixture Ratio				2
3. Propellant Temperature (°F)	Fuel			45 to 75
	Oxid			
4. Propellant Density	Fuel			--
	Oxid			
5. Maximum Loadable Volume (cu ft)	Fuel			137.33
	Oxid			172.65
6. Nominal Propellant Loaded (lb)	Total			23,081
	Fuel			7,714
	Oxid			15,367
7. Propellant Expended Before Liftoff (lb)	Total			0
	Fuel			
	Oxid			
a. Engine Bleed	Fuel			
	Oxid			
b. Engine Leakage	Fuel			
	Oxid			
c. Start Consumption, $87FS_1$ to TCPS	Fuel			
	Oxid			
d. Holddown Consumption, TCPS to Liftoff	Fuel			
	Oxid			
8. Propellant Aboard at Liftoff	Total			23,081
	Fuel			7,714
	Oxid			15,367
9. Propellant Expended during Previous Stage Operation	Total			0
	Fuel			
	Oxid			
a. Engine Bleed, Stage 0 Operation	Fuel			
	Oxid			
b. Engine Leakage, Stage 0 Operation	Fuel			
	Oxid			
c. Engine Bleed, Stage I Operation	Fuel			
	Oxid			
d. Engine Leakage, Stage I Operation	Fuel			
	Oxid			
10. Propellant Aboard at $87FS_1$	Total			23,081
	Fuel			7,714
	Oxid			15,367
11. Engine Leakage during Stage Operation	Total			0
	Fuel			
	Oxid			
12. Total Available Usable Propellant	Total			23,067
	Fuel			7,713
	Oxid			15,354
a. Start Consumption	Fuel			2
	Oxid			3
b. Steady-State Consumption	Fuel			7,673
	Oxid			15,346
c. Shutdown Consumption	Fuel			3
	Oxid			5
d. Tailoff (before Staging)	Fuel			N/A
	Oxid			
e. Fuel Bias	Fuel			35
13. Total Nonusable Propellant	Total			14
	Fuel			1
	Oxid			13
a. Propellant Vapor Retained	Fuel			1
	Oxid			13
b. Trapped above Interface	Fuel			
	Oxid			0
c. Trapped below Interface	Fuel			0
	Oxid			
14. Mean Outage	Total			72
15. Nominal Propellant Consumed during Stage Operation (Item 12 - 14)	Total			22,995
a. Nominal Steady State Propellant (Item 12b - 14 + 12e)	Total			22,982
b. Transient Propellants	Total			13
16. Pressurization System Inert Gas	Total			60
	Fuel			7
	Oxid			8
	Spheres			45
17. Maximum Outage				231

[redacted]

VIII. PAYLOAD LENGTH

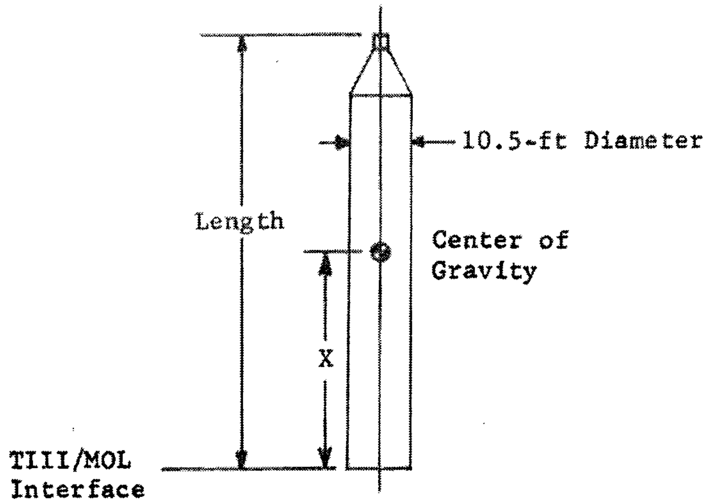
This chapter presents the results of studies to determine the effect on allowable payload length of reduced launch probabilities and structural redesign of the core structure. The results of the more important loads analyses are also discussed.

The primary loads study was to determine the maximum payload length that could be carried by the six reference configurations (see Chapter II) based on 99% launch probability and no redesign to increase the basic longitudinal strength of the core. Secondly, the reduced launch probability for the specified payload lengths was to be determined, again based on no redesign to increase the core longitudinal strength. Thirdly, the core weight penalties associated with structural modifications required to attain 99% launch capability were to be determined.

A. PAYLOAD DESCRIPTION

The MOL payload configuration consists of the Gemini plus laboratory module. To obtain parametric data, it was necessary to select two additional payload lengths for each of the three solid rocket motor (SRM) configurations. Table VIII-1 lists the specified lengths plus two additional lengths for each configuration. The additional lengths were selected to give a reasonable length spread for parametric studies. The center of gravity (cg) locations were provided by the customer for the specified lengths. Weight distributions for the other payload lengths were assumed such that the resultant cg would stay at the same percentage of overall payload length. The payload weights that were used for each SRM configuration are also listed in Table VIII-1.

Table VIII-1 Nominal Payload Lengths Used for Loads Analyses (With-Transtage Configuration)



SRM	Payload Weight (lb)	Length (ft)	X (ft)
7 seg-120	28,000	54.5* 65.0 75.0	29.6 35.2 40.6
2 CS-156	33,000	50.0 58.5* 65.0	27.1 31.7 35.1
3 CS-156	42,000	50.0 61.0* 72.0	27.1 33.1 39.2

*Specified payload lengths.

B. RIGID BODY TRAJECTORY RESULTS

A rigid body airload study was performed on each of the three with-transtage vehicle configurations. The following baselines were established for the analysis:

1) Rotating, oblate earth;

2) Rigid body dynamics;

3) Nonlinear aerodynamics;

4) Load relief autopilot;

5) Launch azimuth of 182 deg from WTR;

6) Standard atmosphere of 1962;

7) Design wind - Taken from Meteorological Note 2.*

In all cases, the wind azimuth search was conducted using a wind shear peak altitude of 29,600 ft. This is the lowest altitude at which the maximum wind magnitude and wind shear values occur for the wind criteria used. After the critical azimuth was determined, i.e., the azimuth that produces the largest value of the airload parameter $q\alpha\beta$, various wind shear peak altitudes were introduced over the altitude range of 11,000 to 35,000 ft along the critical azimuth. In this manner, the azimuth and shear peak altitude combination that produces the largest value of $q\alpha\beta$ was determined. Table VIII-2 presents a summary of the most significant trajectory results for the critical flight conditions.

Some trajectory analysis was conducted using reduced winds. Figure VIII-1 presents a plot of $q\alpha\beta$ vs the percentage of maximum wind for the 7 seg-120 configuration. Since the wind is a pure side wind, $q\alpha\beta$ varies directly with β. These data were used in conjunction with wind load analyses for full winds to obtain design data for reduced wind bending moments.

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*Jerold Bidwell: Atmospheric and Wind Design Criteria for PMR, Meteorological Note 2, Martin Company, Denver, Colorado, 7 August 1964.

Table VIII-2 Trajectory Characteristics for the Critical Flight Conditions

SRM Configuration	Payload Length (ft)	Wind Azimuth (deg)	Time of Flight (sec)	Wind Shear Peak (ft)	Mach No.	Dynamic Pressure, q (lb/ft^2)	Angle of Sideslip, β (deg)	Airload Indicator, $q\alpha\beta$ $(lb\text{-}deg/ft^2)$
7 seg-120	54.5	272	50.52	32,000	1.48	884	5.14	4543
	65.0	272	50.50	32,000	1.48	881	5.29	4664
	75.0	272	50.49	32,000	1.47	879	5.46	4801
2 CS-156	50.0	272	49.09	29,600	1.37	845	5.80	4901
	58.5	272	49.11	29,600	1.37	847	5.78	4900
	65.0	272	49.11	29,600	1.37	847	5.78	4900
3 CS-156	50.0	245	45.02	29,600	1.45	948	5.07	5475
	61.0	245	45.01	29,600	1.45	948	5.09	5534
	72.0	245	45.00	29,600	1.45	947	5.12	5608

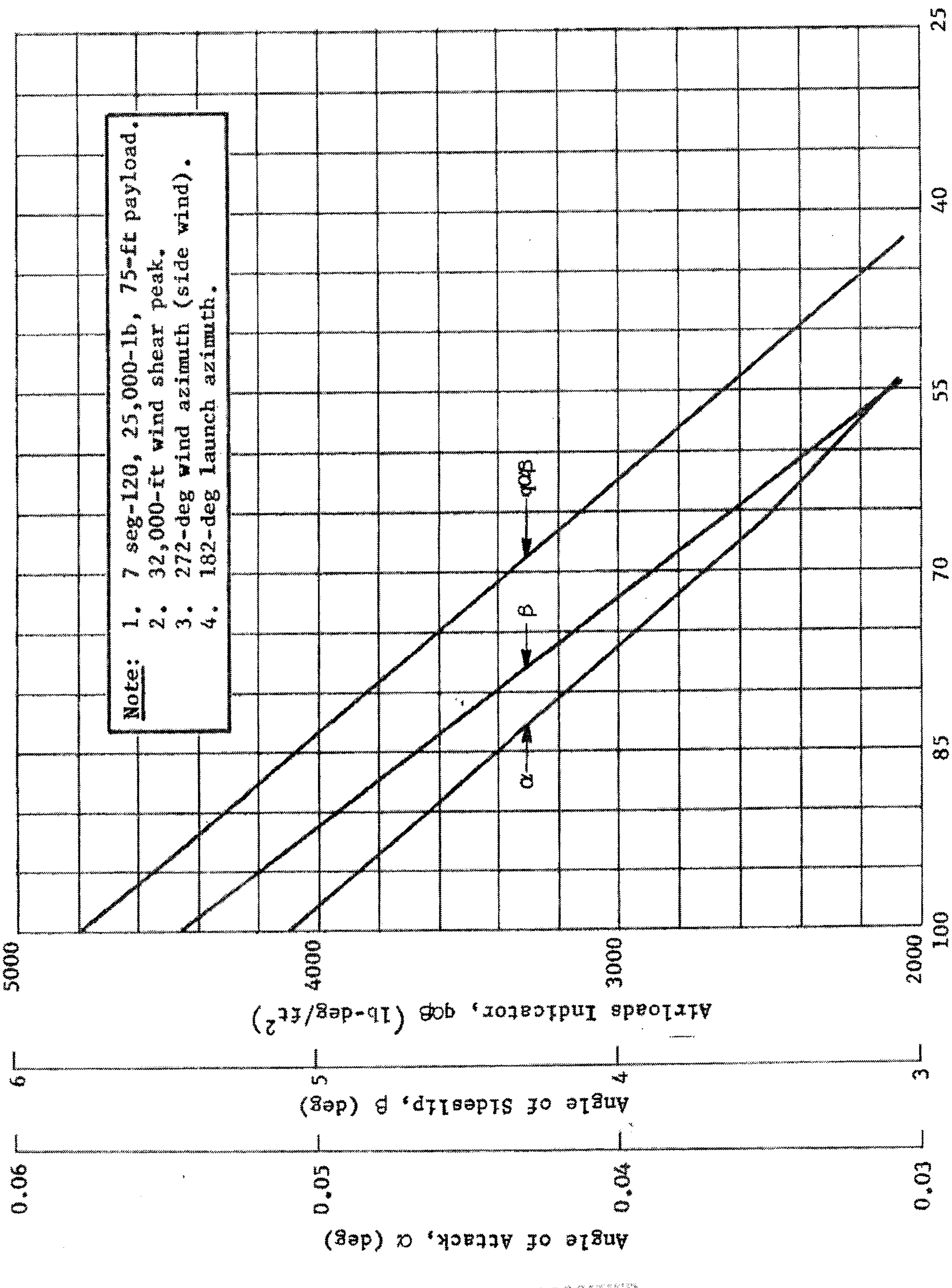

Fig. VIII-1 Airload Indicator vs Percentage of Maximum Wind

C. LOADS ANALYSIS ASSUMPTIONS

Early in the 60-day study, certain ground rules and analysis assumptions were agreed upon with Aerospace to facilitate arriving at common answers. The more significant of these were:

1) Design wind criteria will be based on Meteorological Note 2.* It was then later agreed that the launch probability associated with reduced winds would be based on data from the National Weather Records Center† for the worst month. This relationship between percentage of wind and launch probability is shown in Fig. VIII-2.

2) A buffet analysis will be conducted; however, dispersion and gust effects will be estimated based on previous studies. Dispersions used were 28% of pitch moments and 20% of yaw moments. Gust effects were estimated to vary from 25 to 35% along the vehicle length.

3) For switchover analysis, a more realistic method of loads combination than that used for nonmalfunction conditions will be used together with an ultimate safety factor of 1.25. This was decided primarily due to the high improbability of a malfunction occurring at exactly the same time as the maximum wind shear spike.

4) The ultimate safety factors used in other loads analyses will be consistent with those used for Titan IIIC design.

5) For a given payload length, the changes in core bending moments due to inertia relief differences are small and will not be accounted for in the 60-day study.

6) The existing strength of the core will not be corrected for changes in local aerodynamic collapsing pressures and elevated temperature effects. This was decided since the core temperatures and local collapsing pressures for the three SRM configurations are not much different from each other or from Titan IIIC values.

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*Ibid.

†Winds Aloft Summary and Parameters, Pt. Arguello, California No. 4647. National Weather Records Center, 2 December 1963.

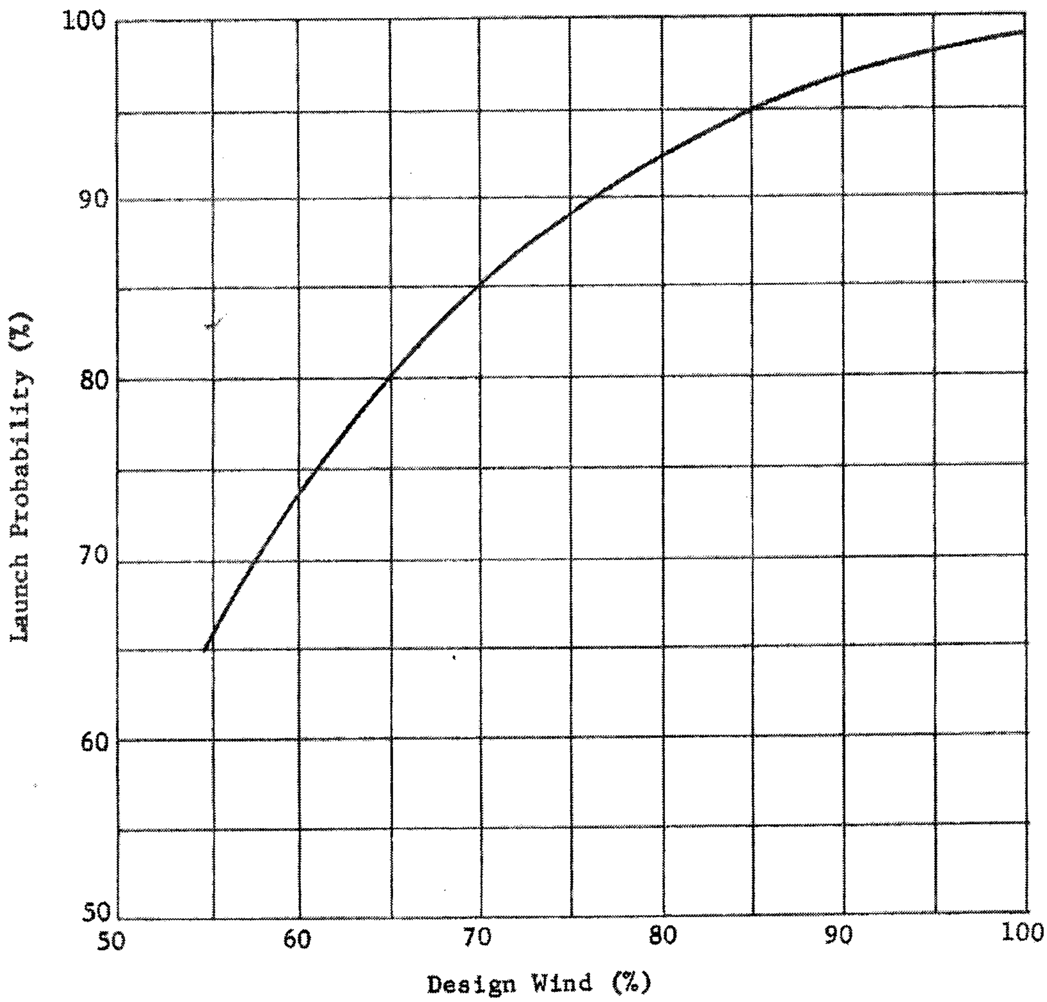


Fig. VIII-2 Launch Probability vs Design Wind

D. MAXIMUM AIRLOAD AND LENGTH ANALYSIS RESULTS

1. Core Loads

The bending moment and axial loads corresponding to the critical flight conditions of Table VIII-2 are shown in Fig. VIII-3 thru VIII-5 in the form of $P_{Equivalent}$ curves for each of the three SRM configurations. The present core capability is also shown in each of the figures. All loads shown are based on 100% of design wind. Since all loads analyses were conducted for a configuration including the transtage with the payload interface at Station 77, it was necessary to convert the nominal payload lengths to actual lengths for both the with- and without-transtage configurations. These corrected lengths and the appropriate interface stations are listed in each of the figures. The axial loads shown include transtage propellant weight.

The bending moments for both the 7 seg-120 and 3 CS-156 configurations peak at Station 250, since that is the location of the SRM-to-core forward structural tie. In a like manner, the moment for the 2 CS-156 configuration peaks at the same station as that for the Titan IIIC 5-segment SRM configuration. In general, the actual $P_{Equivalent}$ for the long SRM configurations exceed the present allowable at the forward end of the core and then become less critical aft of that location, while the 2 CS-156 is generally more critical over a longer portion of the length of the core.

Since the core structure forward of Station 296.6 is being redesigned for the without-transtage configuration, that station is used as the forward limit of the core allowable for each of the without-transtage configurations.

2. Allowable Lengths

The information from Fig. VIII-3 thru VIII-5 was used to plot the curves in Fig. VIII-6. Using the three length data points for each of the six configurations, plots of the ratio of actual to allowable loads at the critical station are presented. The intersection of the six curves with the ordinate value of 1.0 represents the allowable payload length that can be carried based on 99% launch probability and no basic core longitudinal strength increase. These allowable lengths plus the specified lengths are tabulated in Fig. VIII-6. Corrected payload lengths were used and the axial load for the without-transtage configurations took into account the deletion of the transtage propellant weight.

	I_A	I_B	I_C
Nominal Payload (PL) Length (PL Interface at Sta 77) (ft)	54.5	65.0	75.0
Actual Length with TS (PL Interface at Sta 63.5) (ft)	53.4	63.9	73.9
Actual Length Wighout TS (PL Interface at Sta 204.7) (ft)	65.1	75.6	85.6

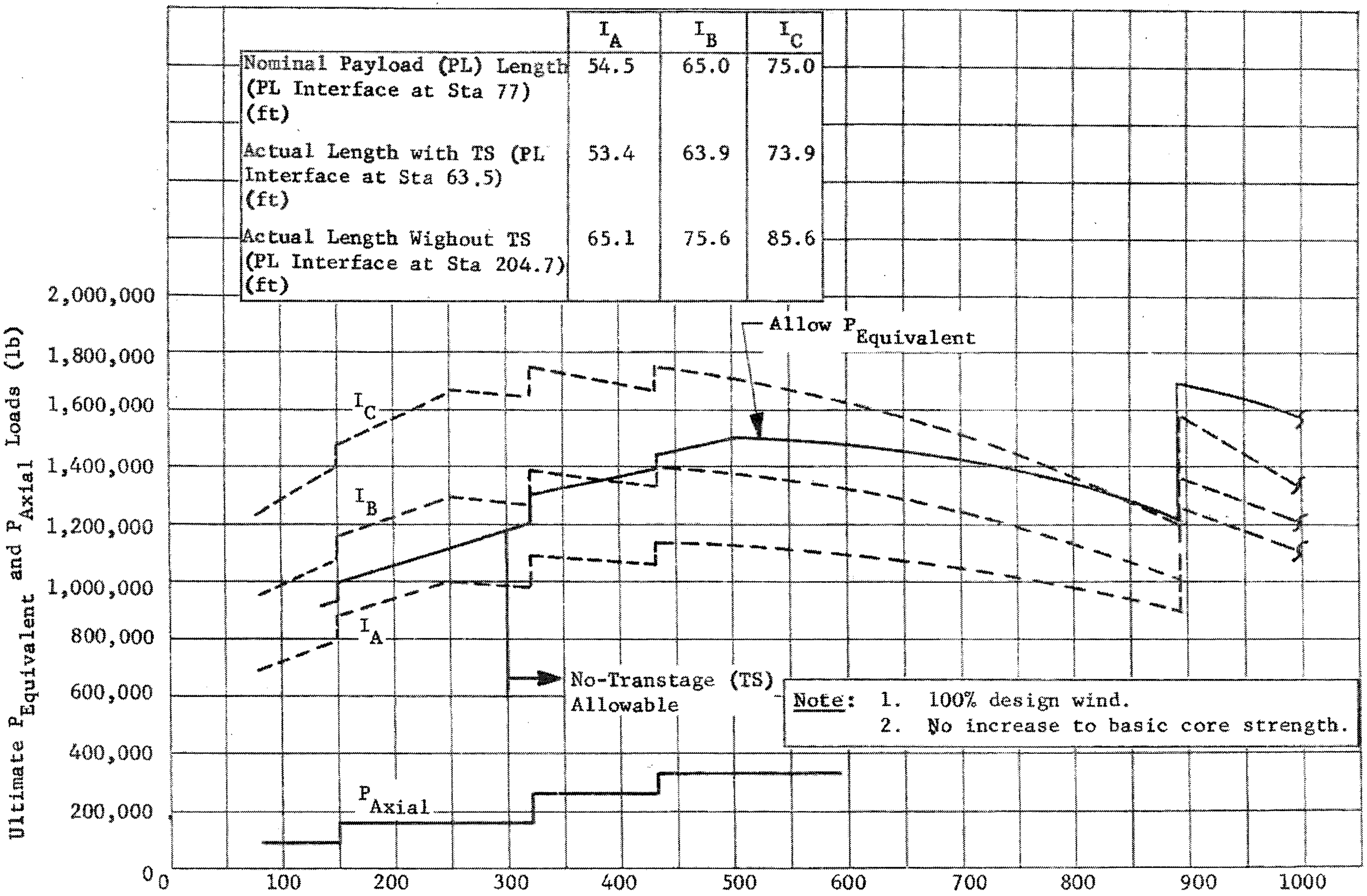


Fig. VIII-3 Ultimate Core Loads, 7 seg-120

	II_A	II_B	II_C
Nominal Payload (PL) Length (PL Interface at Sta 77) (ft)	50.0	58.5	65.0
Actual Length with TS (PL Interface at Sta 63.5) (ft)	48.9	57.4	63.9
Actual Length without TS (PL Interface at Sta 227.2) (ft)	62.5	71.0	77.5

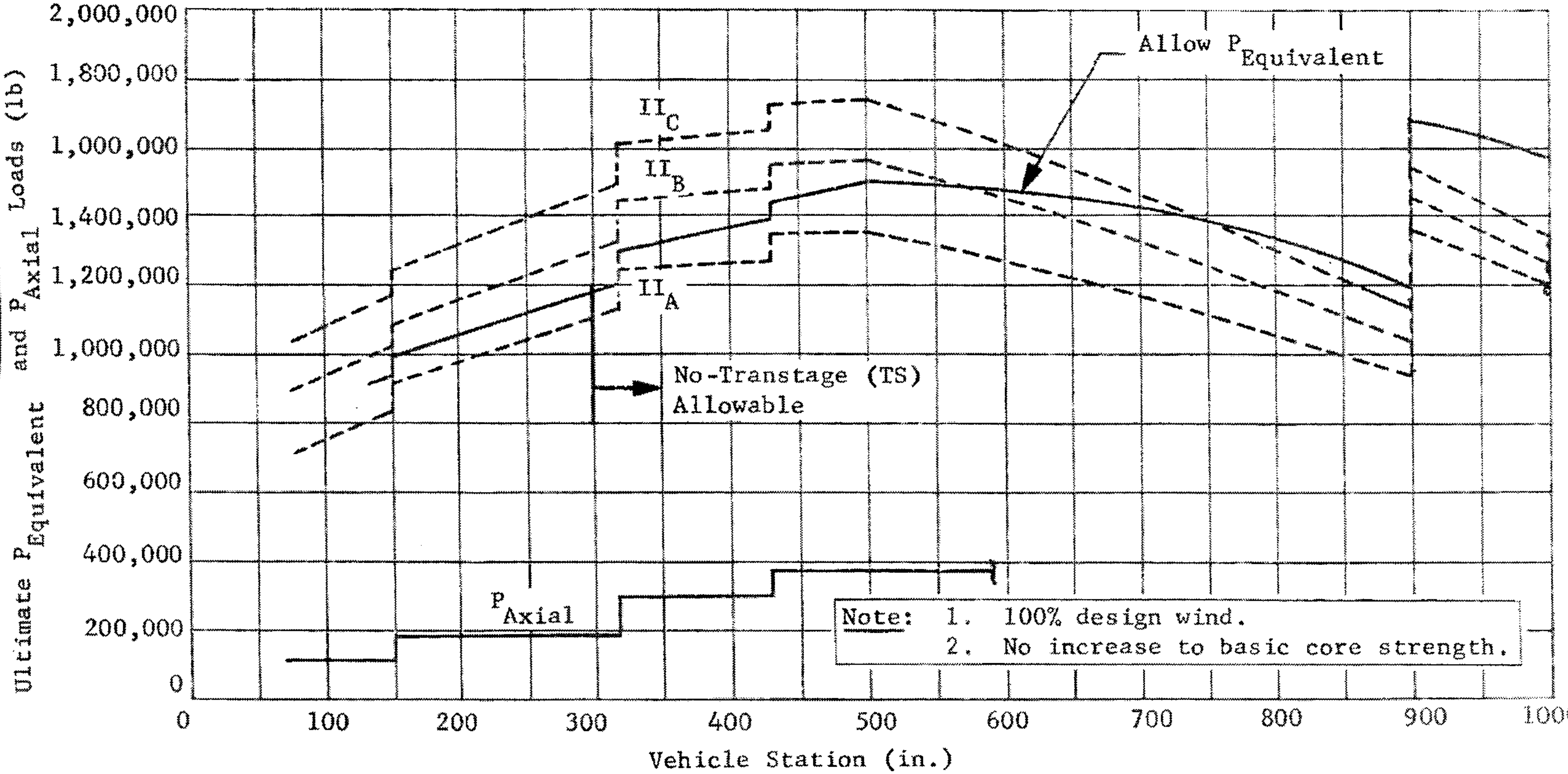


Fig. VIII-4 Ultimate Core Loads, 2 CS-156

[redacted]

	III_A	III_B	III_C
Nominal Payload (PL) Length (PL Interface at Sta 77)(ft)	50.0	61.0	72.0
Actual Length with TS (PL Interface at Sta 63.5)(ft)	48.9	59.9	70.9
Actual Length without TS (PL Interface at Sta 204.7) (ft)	60.6	71.6	82.6

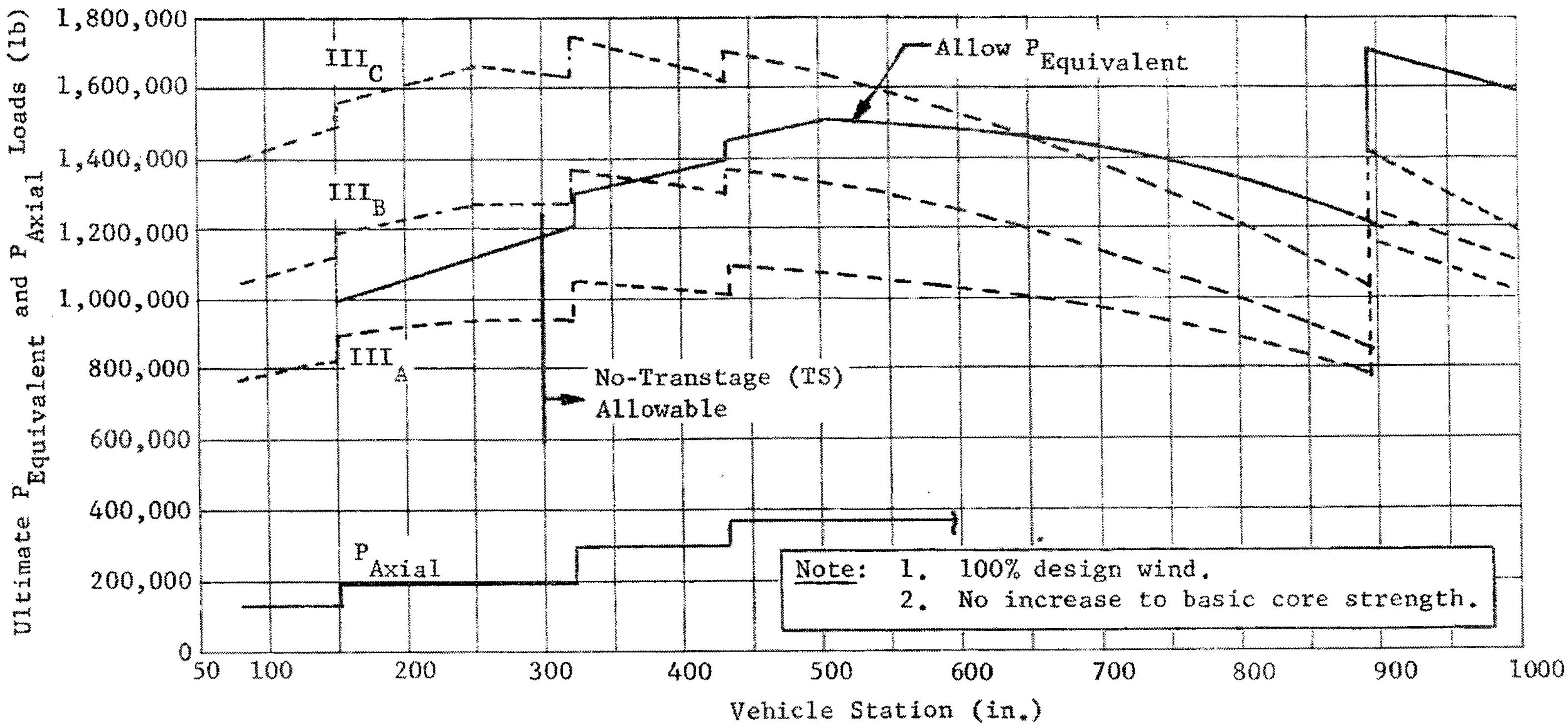


Fig. VIII-5 Ultimate Core Loads, 3 CS-156

	Configuration	Critical Station	Allowable Length (ft)	Specified Length (ft)
①	7 seg-120 with Transtage	151.6	57.8	54.5
②	7 seg-120 without Transtage	296.6	74.7	74.5
③	2 CS-156 with Transtage	320.0	51.3	58.5
④	2 CS-156 without Transtage	320.0	67.8	78.5
⑤	3 CS-156 with Transtage	151.6	53.0	61.0
⑥	3 CS-156 without Transtage	296.6	71.2	81.0

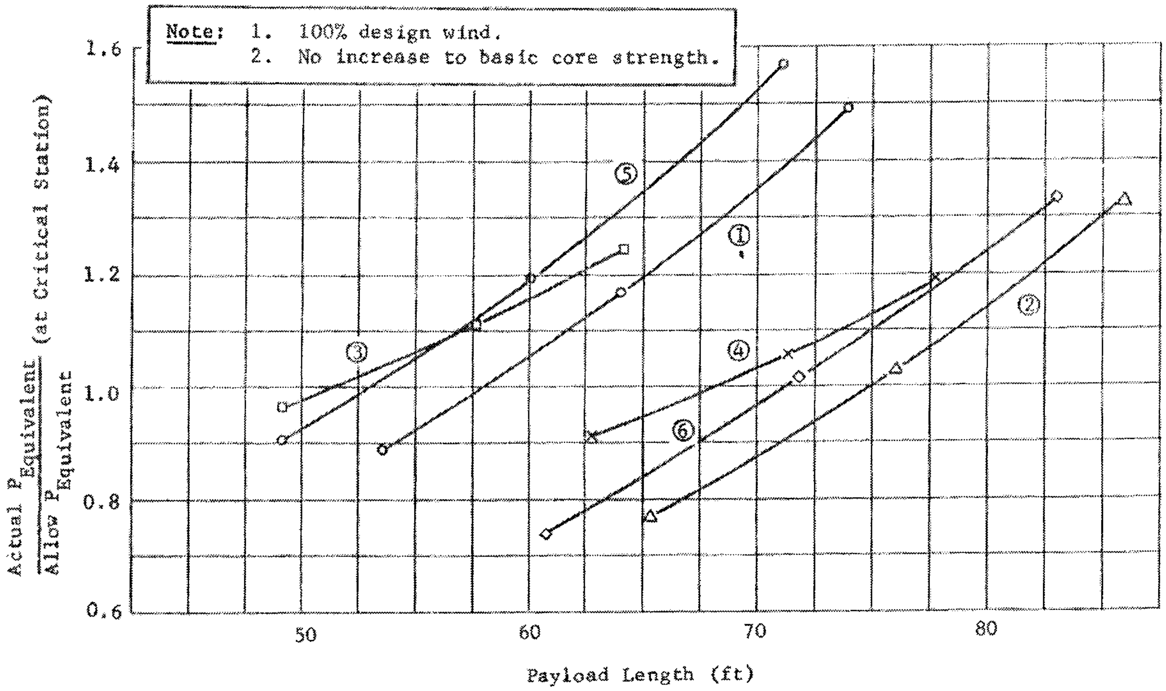


Fig. VIII-6 Allowable Payload Lengths, 99% Launch Probability

3. Reduced Launch Probability

Information from Fig. VIII-1 and VIII-2 plus the basic airload analysis was combined to present the curves shown in Fig. VIII-7. The second study objective results, launch probabilities for the specified payload lengths based on no redesign to achieve increased core strength, are tabulated in the figure. Also shown are the launch probabilities for the maximum length payloads investigated, 70 ft for the with-transtage configurations and 82 ft for the without-transtage configurations.

4. Structural Redesign

The results for the third objective of the payload length study are shown in Table VIII-3. Listed are the delta core structural weights associated with a redesign to permit 99% launch probability for each of the specified and the maximum payload lengths. The 7 seg-120 totals for the specified lengths are zero since that configuration already has 99% launch probability with the present strength. In general, the weight penalties and the extent of redesign are more severe for the 2 CS-156 than for the 3 CS-156 configuration.

5. Summary and Conclusions

In previous paragraphs the possibilities of reducing the launch probability and redesigning the core structure to achieve greater payload length capability have been discussed. Table VIII-4 summarizes that information and includes two other possibilities of potential payload growth. These are a reduction in the design ultimate safety factor from 1.4 to 1.25 for airload bending moments and also a reduction in the design value of $q\alpha\beta$. The reduction in design ultimate safety factor will necessarily include an associated increase in the number of structural failures with less than 3 sec of warning time. Preliminary calculations indicate an increase of about 50. A reduction in $q\alpha\beta$ could be achieved by changing the flight trajectory somewhat and also by redesigning the solid rocket propellant grain. Such studies were beyond the scope of the 60-day study. Note that a significant payload length increase could be achieved by a combination of several of the methods shown.

Launch Probabilities (%)

Configuration	Specified Length	Max. Length*
1 7 seg-120 with Transtage	>99	81
2 7 seg-120 without Transtage	99	91
3 2 CS-156 with Transtage	94.5	71
4 2 CS-156 without Transtage	89.5	82.5
5 3 CS-156 with Transtage	88	65
6 3 CS-156 without Transtage	83.5	81

*70 ft for with-transtage configurations; 82 ft for without-transtage configurations.

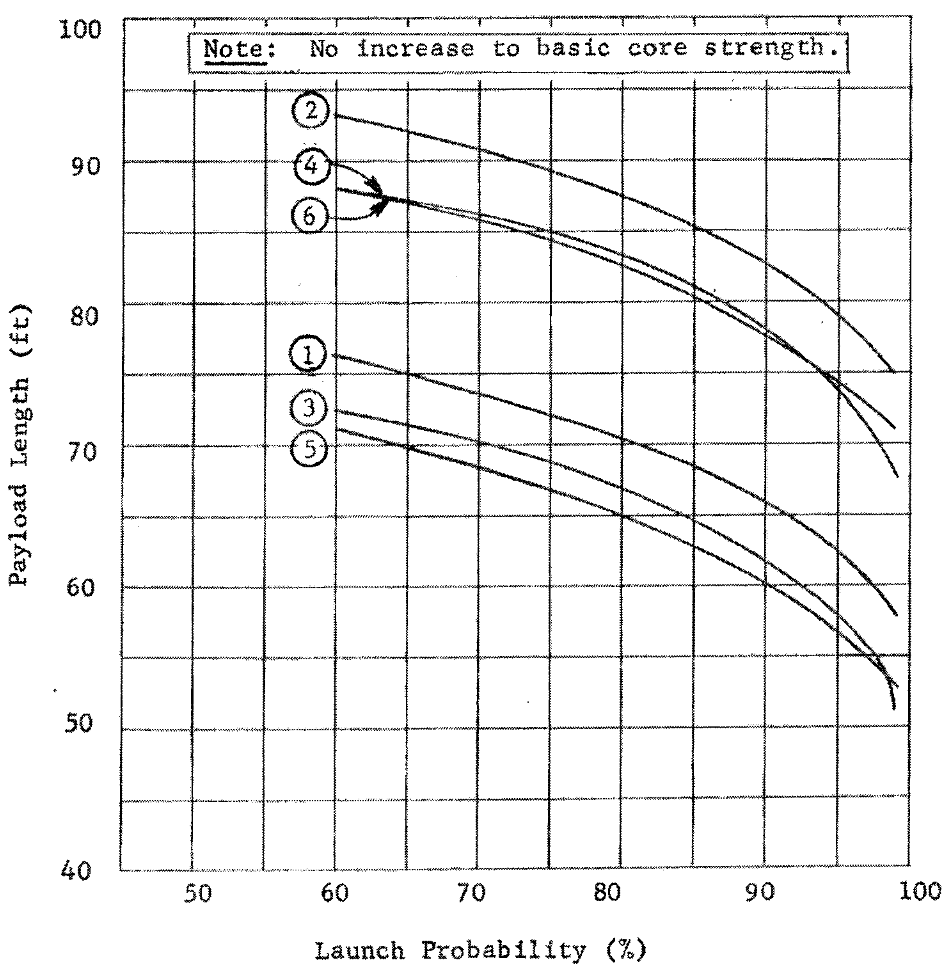


Fig. VIII-7 Allowable Payload Length vs Launch Probability

Table VIII-3 Delta Structural Weight (lb) Associated with Core Redesign to Permit 99% Launch Probability

	Specified Lengths						Maximum Lengths					
	With Transtage			Without Transtage			With Transtage			Without Transtage		
	7 seg-120	2 CS-156	3 CS-156	7 seg-120	2 CS-156	3 CS-156	7 seg-120	2 CS-156	3 CS-156	7 seg-120	2 CS-156	3 CS-156
Transtage	--	20	43	--	--	--	67	74	102	--	--	--
Stage II												
Forward Oxidizer Skirt	--	67	92	--	--	--	183	212	250	--	--	--
Oxidizer Tank	--	28	18	--	42	60	51	85	72	37	62	63
Aft Oxidizer Skirt	--	30	10	--	47	53	47	95	66	36	68	57
Forward Fuel Skirt	--	18	--	--	32	25	25	62	25	18	45	28
Fuel Tank	--	15	--	--	28	18	19	56	27	13	40	21
Aft Fuel Skirt	--	28	--	--	58	25	33	113	43	22	83	31
Stage I												
Transportation Skirt	--	15	--	--	42	--	12	114	12	6	75	--
Forward Oxidizer Skirt	--	--	--	--	7	--	--	33	--	--	19	--
Oxidizer Tank	--	--	--	--	7	--	--	87	--	--	14	--
Aft Oxidizer Skirt	--	--	--	--	--	--	--	--	--	--	--	--
Forward Fuel Skirt	--	--	--	--	--	--	--	--	--	--	--	--
Total	0	221	163	0	263	181	437	931	597	132	406	200

Table VIII-4 Summary of Possible Payload Length Increases (ft)

SRM Configuration	Reduced Launch Probability		Ultimate Safety Factor Reduced from 1.4 to 1.25	Design $q\alpha\beta$ Reduced 10%	50% Core Redesign*
	90%	80%			
7 seg-120 with Transtage	7.9	12.7	3.8	3.2	15.2
7 seg-120 without Transtage	7.8	12.8	3.9	3.3	12.3
2 CS-156 with Transtage	10.5	15.7	5.1	5.1	8.7
2 CS-156 without Transtage	10.2	15.2	5.4	4.2	10.2
3 CS-156 with Transtage	6.8	12.0	3.7	2.3	22.0
3 CS-156 without Transtage	6.3	11.3	3.8	2.1	18.8

*50% of the length of the core.

E. OTHER LOADS ANALYSES

Several loads analyses were conducted during the 60-day study. The more important of these and their results are:

1) A launch analysis considering differential thrust buildup between the two solids resulted in loads less critical than those at staging;

2) The maximum airload conditions were the primary contributors to the payload length studies just described. The Gemini/laboratory module bending moments for the critical flight condition for the specified payload lengths are shown in Table VIII-5;

3) Step 0 burnout was not quite as critical as the unsymmetrical tailoff condition at the time of Stage I engine start. The most important result of this analysis, the Stage I longeron load, is discussed in Chapter V;

4) The switchover loads analysis results indicate adequate structural margin to permit approximately a 100-msec time delay in switchover;

5) Thrust termination results cannot be taken as conclusive at this time since thrust termination curves for the solid configurations being studied were not available. However, the requirements that have been levied on the SRM manufacturers and the inputs to our loads analysis are such that only local redesign is required to the core/SRM attach hardware. Substantiation that additional redesign is not required must await receipt of actual thrust-termination curves for the selected SRM configuration.

Table VIII-5 Gemini/Laboratory Module Bending Moments and Axial Loads for Critical Flight Conditions, Ultimate Loads

	Sta.	Moment (10^6 in.-lb)	Sta.	Moment (10^6 in.-lb)	Sta.	Moment (10^6 in.-lb)
	54.5 ft		65 ft		75 ft	
7 seg-120	-577	0	-703	0	-823	0
28,000-lb Payload	-391	1.43	-517	1.54	-637	1.63
Ultimate Longitudinal g = 2.34	-275	4.85	-367	6.51	-457	8.19
Ultimate Drag = 26,400 lb	-150	9.27	-217	12.3	-277	15.6
	77	17.7	77	25.2	77	33.8
	50 ft		58.5 ft		65 ft	
2 CS-156	-523	0	-625	0	-703	0
33,000-lb Payload	-337	1.63	-439	1.67	-517	1.72
Ultimate Longitudinal g = 2.51	-233	4.92	-310	6.10	-367	7.21
Ultimate Drag = 29,400 lb	-130	9.20	-181	11.75	-217	14.0
	77	17.95	77	23.1	77	27.5
	50 ft		61 ft		72 ft	
3 CS-156	-523	0	-655	0	-787	0
42,000-lb Payload	-337	1.77	-469	1.87	-601	2.02
Ultimate Longitudinal g = 2.28	-233	5.51	-332	7.27	-431	9.52
Ultimate Drag = 30,700 lb	-130	10.1	-196	13.7	-262	18.25
	77	19.0	77	27.3	77	37.7

IX. RELIABILITY AND CREW SAFETY

The objectives of the reliability and crew safety program study were to (1) analyze the six performance configurations with respect to their crew safety and mission success differences to provide a basis for performance modification and selection and, (2) analyze a Titan IIIC for an updated mission success evaluation. SSD-CR-65-203* includes a comprehensive summary of the reliability and crew safety program.

A. GROUND RULES

The following ground rules were established:

1) The analyses would be based on predictions for the first Titan IIIC/MOL booster launch (1968);

2) The analyses would be oriented to achieved predictions and would include the expected degradation due to manufacturing, assembly, and test errors;

3) Mission abort was defined as any failure that prevents achievement of the primary orbital mission;

4) Only single malfunctions leading to mission abort would be considered except for the thrust vector control (TVC) subsystem, which will include multiple malfunctions. Abort due to multiple malfunctions are considered to have little impact on overall mission aborts;

5) All equipment except interconnections and cabling would be analyzed;

6) The analyses would be based on mission time profiles, which will include the different SRM burn times and, a 310-sec first burn and an 8-sec second burn for with-transtage configurations;

---

*Preliminary Failure Mode and Effects. SSD-CR-65-203. Martin Company, Denver, Colorado, 27 August 1965.

7) The analyses, would include inputs from each of the participating associate contractors;

8) The Gemini inertial guidance system (IGS) was assumed to be as reliable as the booster IGS;

9) 2 CS-156 SRM reliability data would be derived from the 3 CS-156 SRM reliability data.

B. APPROACH

To achieve the study objectives, the analyses were conducted within the failure mode and effects program, which consists of four basic steps:

1) Identification of blackbox failure modes;

2) Evaluation of end effects on the booster as a result of these failure modes;

3) Assessment of the probability of occurrence of these failure modes;

4) Data reduction and analyses.

The six performance configurations defined in Chapter II and the Titan IIIC configuration are subdivided into basic systems by associate contractors and are discussed in this section.

1. Aerojet-General Corporation

The analyses provided by Aerojet included failure mode identification and the probabilities of occurrence, which were based on measured data from test firings. Inputs covered:

1) Stage I (8:1) - A Titan IIIC Stage I engine without modification;

2) Stage I (15:1) - An updated engine including performance and reliability modifications;

3) Stage II - A Titan IIIC Stage II engine without modification;

4) Transtage - A Titan IIIC transtage engine without modification.

The analyses of end effects on the booster were completed by the Martin Company.

2. AC Electronic Division (ACED)

The analyses provided by ACED included failure mode identification, end effect analyses, and probabilities of occurrence, which were based on measured failure rates and updated environmental factors. Inputs included a booster IGS for with-transtage configurations, which included pressure temperature control (PTC), and for without-transtage configurations, which did not include a PTC.

3. United Technology Center (UTC)

The analyses provided by UTC included failure mode identification and the probabilities of occurrence, which were based on failure rate data from failure rate handbooks, degradation or use factors, and updated environmental factors. Inputs included data on:

1) 5 seg 120 SRM - The Titan IIIC SRM and TVC system without modification;

2) 7 seg 120 SRM - The modified Titan IIIC SRM and a baseline redundant TVC system;

3) 3 CS-156 SRM - A new SRM and a baseline redundant TVC system.

4. Martin Company

The analyses conducted by Martin Company were based on two hardware configurations, i.e., the with-transtage and the without-transtage core configurations. Failure mode identification and vehicle effect analyses were completed for each blackbox within these configurations. To assess the probability of occurrence for each failure mode, failure rates were obtained from handbook and test data and subsequently degraded to achieved failure rates using Titan IIIC background data. Updated environmental factors for each of the six performance configurations (based on recent Titan IIIC flight test data) and mission time profiles were programed into an IBM 1620 computor for computation of probability of occurrences for all six performance configurations in accordance with the formula

$$P_f = n \times t \times K_{op} \times f_r$$

where,

n = quantity of blackboxes,

t = time of operation (hr),

K_{op} = environmental factor,

f_r = failure rate in expected number of occurrences/ million hours.

All the associate contractor and Martin inputs were prepared on electronic data-processing input transmittals, which were subsequently key-punched on IBM cards for mechanized reduction. The IBM 1620 program mentioned above was used to obtain assessments of mission success and crew safety for each subsystem for each of the six performance configurations. The Titan IIIC configuration mission success assessment was determined from the with-transtage core configurations by eliminating the redundancy and switchover modifications peculiar to the performance modifications.

C. RESULTS

The results of the reliability and crew safety study are summarized in Tables IX-1 thru IX-4.

1. Crew Safety/Mission Abort Summary

Table IX-1 is a configuration comparison summary by warning time, total number of aborts, and mode IV aborts. Warning time analysis indicates that the best crew safety configuration is the 7 seg-120 configuration (without transtage), although there is only a difference of 107 in the number of mission aborts between the without-transtage configurations. The 7 seg-120 configuration has the least mission aborts with less than 3 sec warning time for the with-transtage configurations, although there is only a difference of 147 between the with-transtage configurations. Each SRM configuration without the transtage has approximately 10%

fewer mission aborts with less than 3 sec lead time than the corresponding SRM configuration with the transtage. The expected number of mode IV aborts are the number of aborts that would result in Gemini splashdown below the 30th parallel.

Table IX-1 Crew Safety/Mission Aborts Summary

	With Transtage			Without Transtage			
	7 seg-120	2 CS-156	3 CS-156	7 seg-120	2 CS-156	3 CS-156	Titan IIIC
Expected Number (F) of Aborts by Warning Time $0 < F_1 < 3$ sec	2,401	2,451	2,548	2,202	2,222	2,309	--
$3 \text{ sec} \leq F_2 \leq 6$ sec (40 to 90 sec)	66	65	66	66	65	65	--
$6 \text{ sec} < F_3$	35,978	37,976	38,420	31,782	33,788	34,138	--
Not Analyzed	2,042	2,953	3,078	2,041	2,953	3,078	--
Total Number of Aborts	40,484	43,442	44,008	36,099	39,025	39,586	56,080
Expected Number of Mode IV Aborts	570	681	773				--
Note: Figures represent occurrences in PPM.							

2. Mission Aborts by Flight Phase

Table IX-2 summarizes mission aborts by flight phase for each configuration where flight phases are defined as follows:

1) Stage 0 flight phase - The interval from SRM ignition to Stage I Start;

2) Stage I flight phase - The interval from Stage I start to Stage I/Stage II staging;

3) Stage II flight phase - The interval from Stage I/ Stage II staging to Stage III start or payload separation in the case of the without-transtage configurations;

4) Transtage burn 1 - The interval from transtage start thru shutdown;

5) Transtage coast - A 45-minute coast period;

6) Transtage burn 2 - The period from transtage start thru payload separation.

Analysis of Table IX-2 indicates that the 7 seg-120 configuration without the transtage has the least number of mission aborts for the 80-n-mi orbit. The 7 seg-120 configuration with the transtage shows only a difference of 2464 in mission aborts between the 80- and 130-n-mi orbits.

Table IX-2 Mission Aborts by Flight Phase

	With Transtage			Without Transtage			
FLIGHT PHASE	7 seg-120	2 CS-156	3 CS-156	7 seg-120	2 CS-156	3 CS-156	Titan IIIC
Stage 0	8,058	9,404	10,124	8,058	9,404	10,124	10,495
Stage I	13,642	15,265	15,101	13,133	14,709	14,550	17,000
Stage II	14,912	14,915	14,905	14,908	14,913	14,912	16,671
Transtage (1st burn)	1,409	1,409	1,409	--	--	--	3,983
Subtotal (80-n-mi equivalent)	38,022	40,993	41,539	36,099	39,025	39,586	48,149
Transtage (coast)	478	463	488				5,867
Transtage (2nd burn)	1,986	1,986	1,986				2,064
Subtotal	2,464	2,449	2,469				7,931
Total (130-n-mi equivalent)	40,484	43,442	44,008				56,080

3. Crew Safety/Mission Abort Summary by Subsystem Grouping

Table IX-3 is a configuration comparison by subsystem grouping defined as follows:

1) Total system;

2) Propulsion, which includes LREs and the SRMs less the TVC;

3) Guidance and controls, which include the guidance, flight controls, and the TVC subsystems;

4) Other, which includes the remainder of the subsystems not listed in 2) and 3).

The 7 seg-120 configuration is considered baseline and the other configurations are shown as deltas from this baseline.

Table IX-3 Crew Safety/Mission Abort Summary by Subsystem Grouping

		With Transtage			Without Transtage		
		7 seg-120	2 CS-156△	3 CS-156△	7 seg-120△	2 CS-156△	3 CS-156△
Total System	R	40,484	2,958	3,524	-4,385	-1,459	-898
	F_1	2,401	50	147	-199	-179	-92
	F_2	66	-1	0	0	-1	-1
Propulsion	R	34,431	2,902	3,446	-2,702	209	745
	F_1	1,372	57	151	-91	-34	59
	F_2	0	0	0	0	0	0
Guidance and Control	R	1,402	53	77	-4	52	78
	F_1	97	13	13	7	22	26
	F_2	66	-1	-1	0	-1	-1
Other	R	4,651	10	1	-1,679	-1,720	-1,721
	F_1	932	-20	17	-115	-167	-177
	F_2	0	0	0	0	0	0

4. Effect of Step 0 Burning Time on Reliability

Table IX-4 is intended to show the increase or decrease in mission aborts as a result of increased burning time during Step 0. The 7 seg-120 configuration is considered baseline and the other configurations are shown as deltas from the baseline. Step 0 is defined as the mission flight time from Stage 0 ignition through SRM staging. The core is defined as all subsystems excluding LREs, which are not sensitive to the SRM burning time.

Table IX-4 Effects of Step 0 Burning Time on Reliability

	With Transtage			Without Transtage		
Configuration	7 seg-120	2 CS-156△	3 CS-156△	7 seg-120△	2 CS-156△	3 CS-156△
Burn Time	120 sec	136 sec	150 sec	120 sec	136 sec	150 sec
SRM	7328	1407	1942	0	1407	1943
Core	1659	219	90	-113	106	-23
TVC	811	53	75	0	53	75
Total Aborts	9798	1679	2107	-113	1566	1994

X. SCHEDULE

The schedule and program plans provide a basis for evaluating relative schedule position of the six configurations under study (see Table II-1 and Fig. II-1 for a description of the configurations).

To provide the schedule comparison, a Titan III/MOL booster program plan was established. The six configurations, based on an equal-risk schedule, are then compared against the program plan.

A. GROUND RULES

Planning for the booster and Stage 0 development programs is based on the following ground rules and restraints:

1) The core booster subsystems schedules are based on the reference configuration described in Chapter II;

2) Phase II go-ahead for the LRE, SRM, and guidance associate contractors will be the same as the core booster;

3) The Integrated Launch Complex (ILC) will be used for MOL launches. This facility will be activated before the MOL schedule requirements and will not constrain launch capability;

4) Resource application (facilities, manpower, number of shifts, overtime, etc) will be on the same basis as the Titan III development program;

5) Booster acceptance tests will be performed in Cell P-4. This cell will be modified and equipped to conduct test on the Titan IIIC configuration and the selected MOL configuration.

B. TITAN III/MOL BOOSTER PROGRAM PLAN

The program plan, Fig. X-1, was developed to establish a baseline to compare schedule and cost of the various configurations. This plan supports a launch schedule established by the Space Systems Division of the Air Force Systems Command.

The significant elements are:

1) Program go-ahead on 1 March 1965;

2) Airborne engineering and major test comprised of structural, component development, controls mockup, and design assurance;

3) Airborne first article fabrication, which covers the modification of Vehicle 17 from the Titan IIIC configuration, and the first production MOL;

4) Acceptance test of the seven MOL configuration vehicles will be conducted in Cell P-4. Vehicle 17 will be the first MOL booster accepted in November 1967 (the additional vehicles to be tested in Cell P-4, as reflected in Fig. X-1, are Titan IIIC follow-on vehicles displayed for a cost base only per SSD direction);

5) The GFP delivery requirements for LREs, SRMs, and IGS to support the MOL booster fabrication and launch program are indicated on the plan;

6) The ILC design fabrication, installation, and acceptance plan is outlined on the program plan. This plan calls for a Phase II go-ahead of 1 February 1966. The activation and acceptance of the launch facility is scheduled in October 1967;

7) There will be seven MOL launches from ILC. The first launch is planned for May 1968, and launches will continue at a rate of one every four months through May 1970.

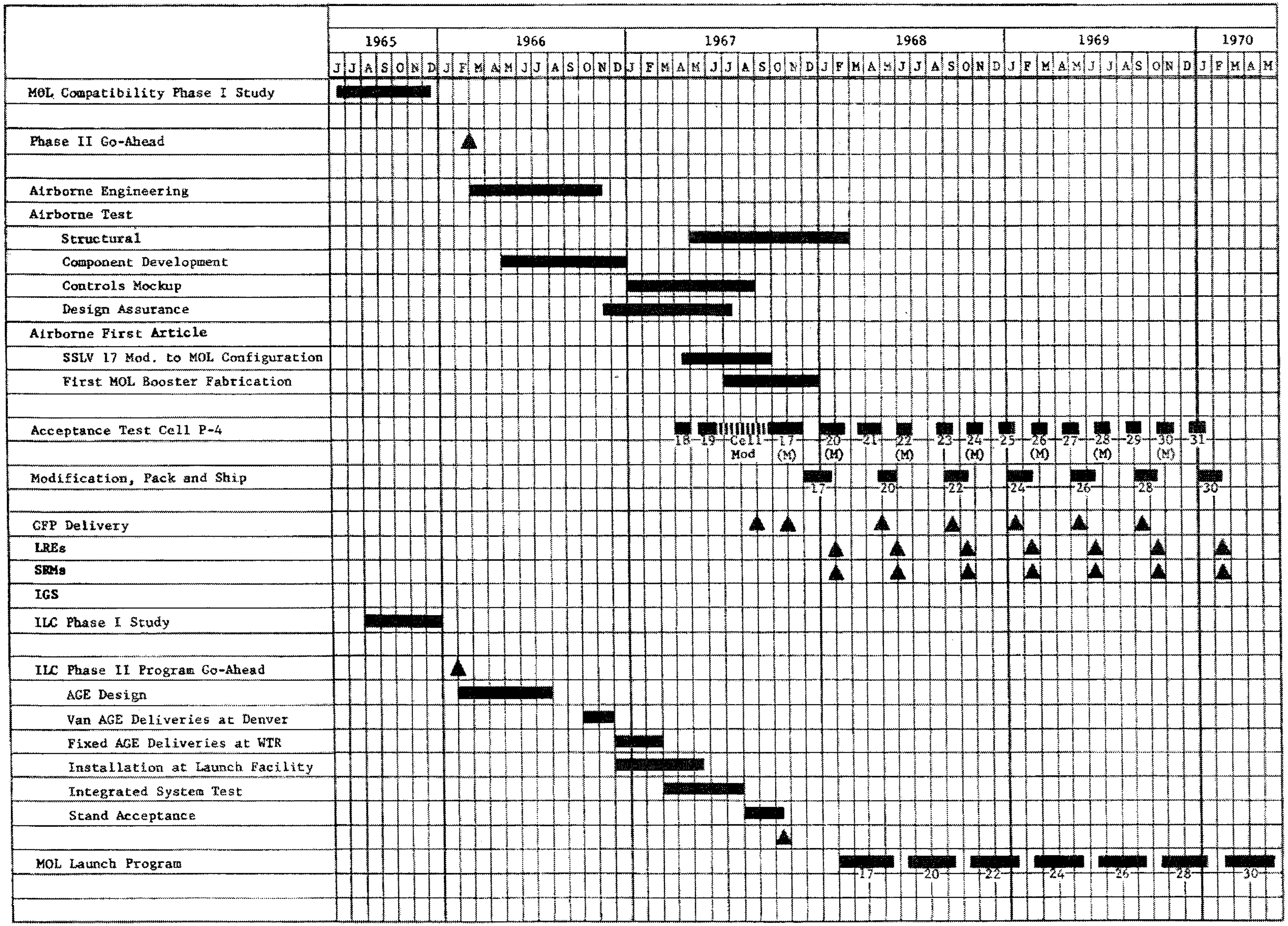


Fig. X-1 Titan III/MOL Booster Program Plan

C. EQUAL RISK SCHEDULE

A schedule review of the six configurations was made on a basis of equal risk from program go-ahead through the first launch.

To establish a schedule of equal risk, a standard plan for each configuration was established. The plan was then converted into a PERT network where each element was evaluated. The evaluation considered such items as, new design, production capability, number of tests required, probability of success, etc. Each evaluation was made on an optimistic, most likely, and pessimistic basis to establish span time for that effort. Span time on all the networks is expressed in weeks from go-ahead. We concluded that this method was the most adequate way to demonstrate equal risk comparison between various configurations.

The elements of the standard network are:

1) Airborne core subsystem design;

2) Engineering development and design assurance test;

3) Structural test;

4) Fabrication and delivery of first article;

5) Launch facility construction;

6) Airborne ground equipment design, fabrication, installation, and activation;

7) LRE development and delivery;

8) SRM development and delivery;

9) Launch of first MOL.

1. 7 seg-120 with Transtage; 15:1 Stage I Engine Configuration

The schedule for this configuration is reflected in Fig. X-2. The expected span time from go-ahead to launch for this configuration is 117.1 weeks. Based on a Phase II go-ahead of 1 March 1966, the launch would be 19 June 1968.

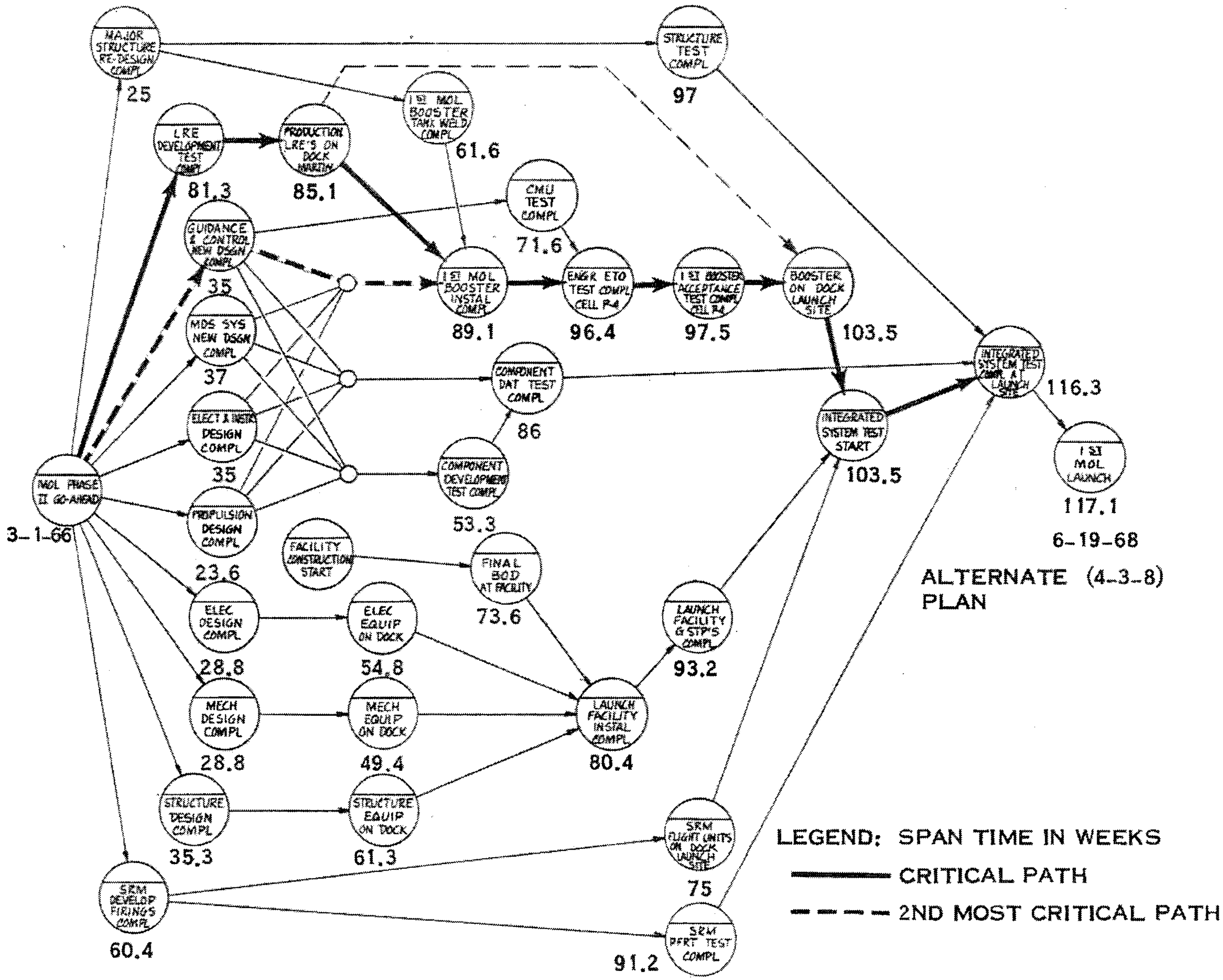


Fig. X-2 7 seg-120 with Transtage; 15:1 Stage I Engine Configruation Schedule

The most critical path is the LRE development test, production, LRE on dock at Martin, and completion of the first MOL booster installation. This path controls the booster acceptance test and subsequent launch.

The second most critical path is through the guidance and control new design to first MOL booster installations complete. This is caused by the new design and development of actuators.

An alternative plan, although of some higher risk, could be followed that has been used in the past. This plan is shown in Fig. X-2 and calls for the use of a checkout engine for booster acceptance test and installation of the flight engine before shipment to the launch site. The alternative plan moves the launch up to 3 April 1968.

2. 7 seg-120 without Transtage; 15:1 Stage I Engine Configuration

The schedule of all configurations showed no significant change when the transtage was removed. Note in Fig. X-3 that this configuration has the same launch capability as the previous configuration and the same critical paths.

There is some reduction in the structural test span time as a result of fewer tests. The requirement to develop new transtage actuators would also be eliminated.

The same alternative plan exists for this configuration as described for the configuration discussed in Subsection 1 above.

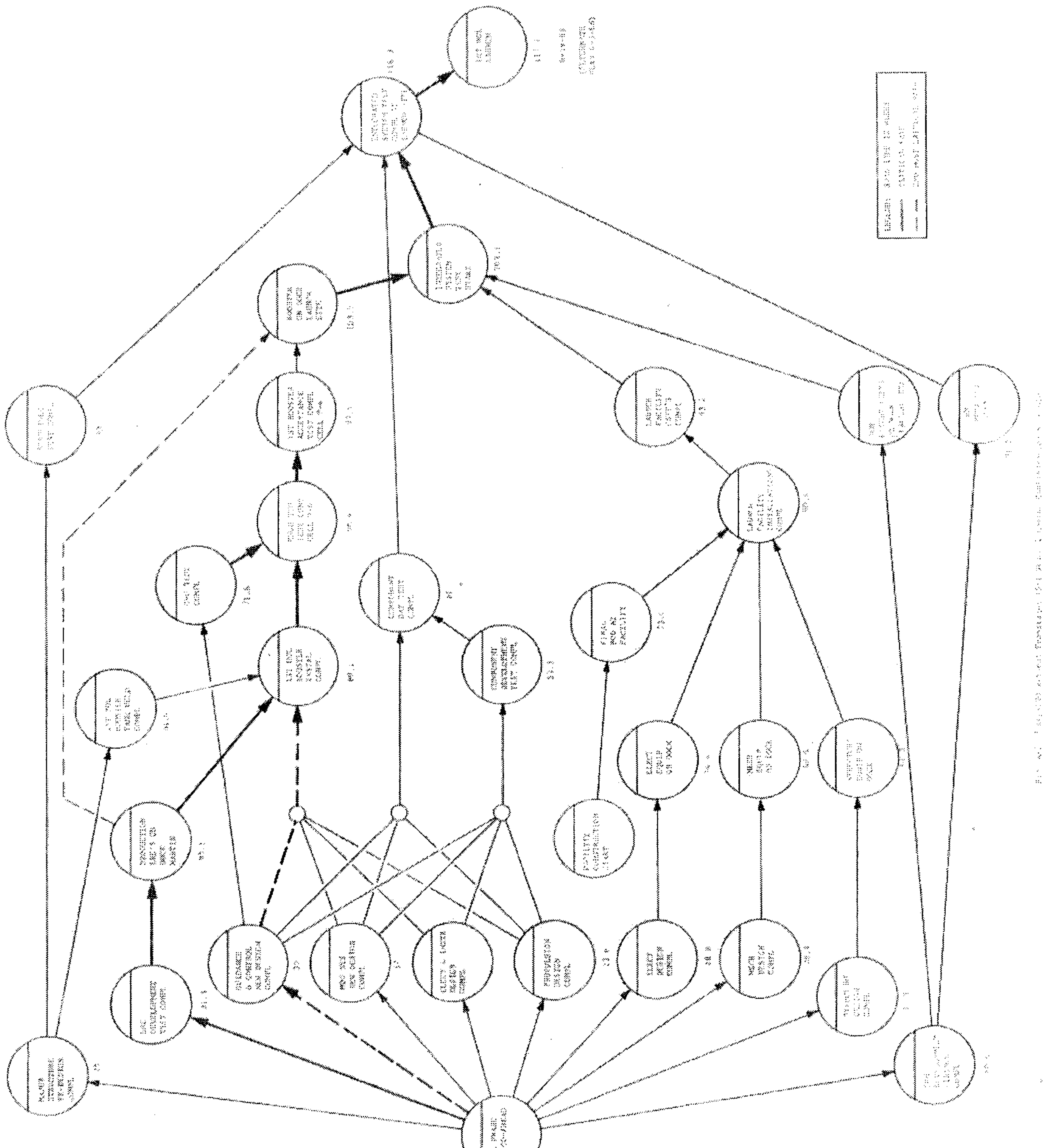
3. 2 CS-156 with Transtage; 8:1 Stage I Engine Configuration

The schedule for this configuration is reflected in Fig. X-4. The expected span time from go-ahead to launch is 121 weeks. Based on a 1 March 1966 Phase II go-ahead, the launch would be 19 July 1968.

The most critical path is the SRM development firings and SRM preflight rating test complete. This path controls the first launch because of the requirement to complete all PFRTs before launch.

The second most critical path is the flight SRM on dock at the launch site. This controls the launch pad checkout and launch.

A more detail analysis of the SRM development schedule is presented in Section D. The critical elements of the SRM development schedule are also covered in Section D.



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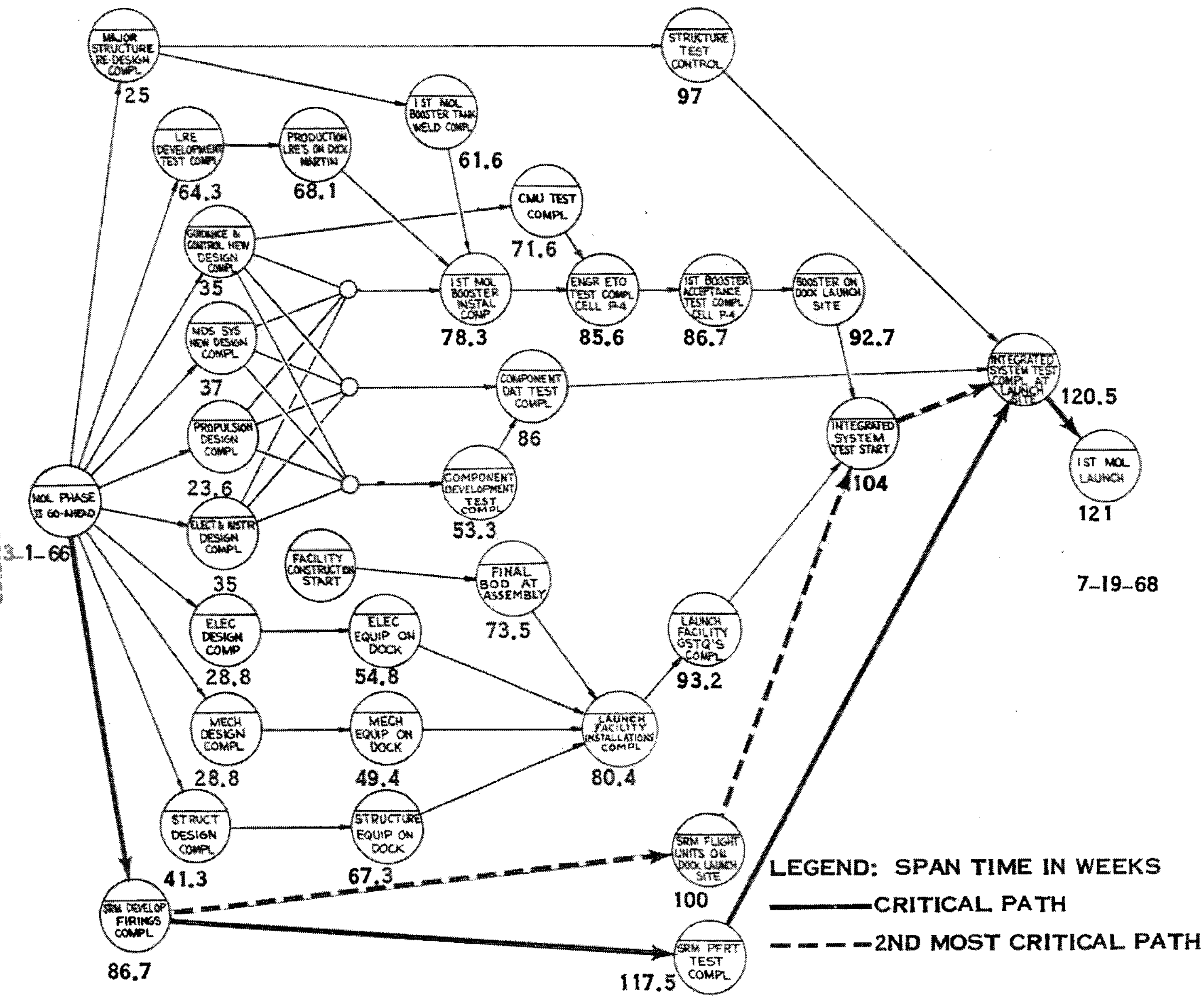

Fig. X-4 2 CS-156 with Transtage; 8:1 Stage I Engine Configuration Schedule

[redacted]

4. 2 CS-156 without Transtage; 8:1 Stage I Engine Configuration

The schedule for this configuration is the same as the preceding one and is shown in Fig. X-4.

5. 3 CS-156 with Transtage; 8:1 Stage I Engine Configuration

The schedule for this configuration is shown in Fig. X-5. The expected span time from go-ahead to launch is 123 weeks. Based on a 1 March 1966 Phase II go-ahead, the launch would be 19 July 1968.

The most critical path is the same as the 2 segment. This path controls the first launch because of the requirement to complete all PFRTs before launch.

The second most critical path is the flight SRM on dock at the launch site. This controls the launch pad checkout and launch. A more detailed analysis of the SRM development schedule is presented in Section D. The critical elements of the SRM development schedule are also covered in Section D.

6. 3 CS-156 without Transtage; 8:1 Stage I Engine Configuration

The schedule for this configuration is the same as the preceding one and is shown in Fig. X-5.

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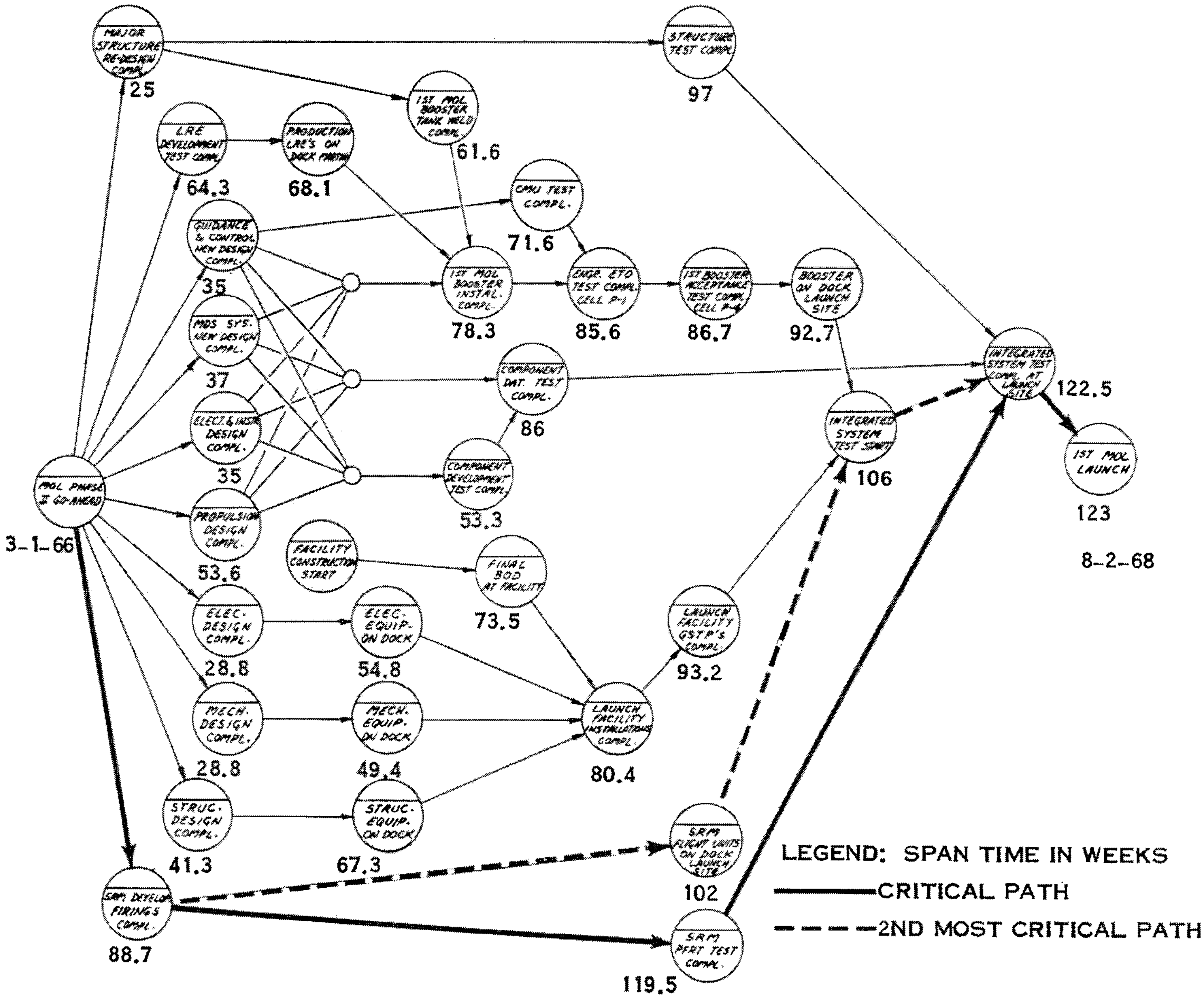

Fig. X-5 3 CS-156 with Transtage; 8:1 Stage I Engine Configuration Schedule

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D. EQUAL RISK SCHEDULE FOR STAGE O DEVELOPMENT

The Stage O development is a significant factor in the total MOL booster program. To obtain a better evaluation of the impact on the first launch capability, an equal-risk schedule was made for the three SRM configurations.

The schedule spans represent a composite of data obtained from United Technology Center, Lockheed Propulsion Company, Thiokol Chemical Corporation, Aerojet-General Corporation, and data from the 5-segment Titan IIIC program.

The basic elements of the standard Stage O development plan are:

1) Case design and procurement;
2) TVC design and procurement;
3) Nozzle design and procurement;
4) Subscale test;
5) Thrust termination test;
6) AGE design, fabrication, installation, and checkout;
7) Development firings;
8) PFRT firings;
9) Assembly and checkout of SRMs at launch site.

1. 7 seg-120 SRM Development Schedule

The schedule for this configuration is reflected in Fig. X-6. The span time from go-ahead to launch is 94.2 weeks. Based on a 1 March 1966 go-ahead, the launch would be 10 January 1968. It is significant to note that the launch capability is much earlier for this configuration than that for the 7 seg-120 with- and without-transtage configurations. In the previous configurations, the critical path was through booster availability as a result of liquid rocket engine development, where this configuration launch capability is not restrained by the core booster.

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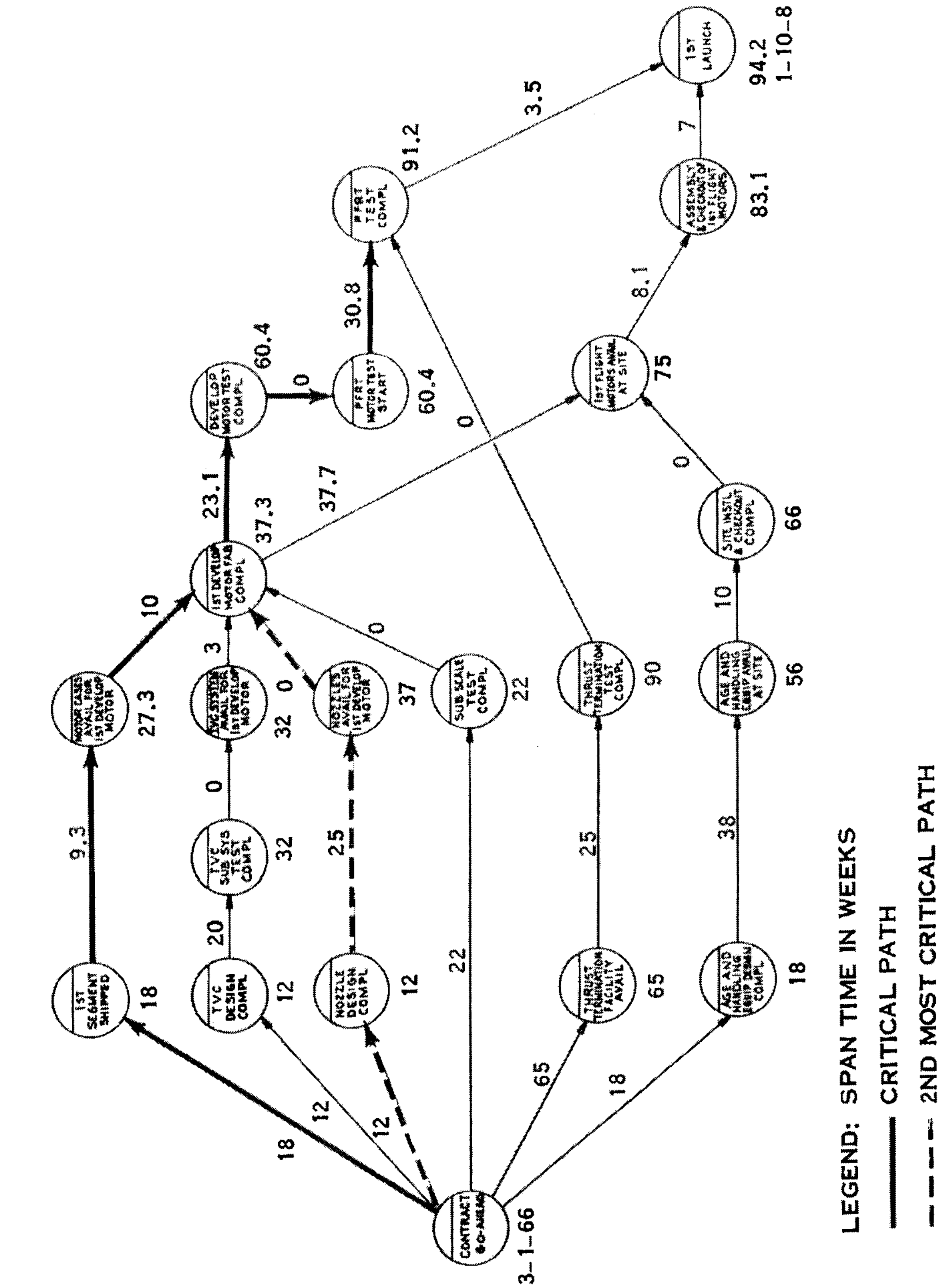

Fig. X-6 7 seg-120 SRM Development Schedule

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The development span time is shorter for the 7 seg-120 SRMs than for the 2 and 3 CS-156 SRMs. Three items primarily contribute to the shorter spans:

1) Less new design,

2) Fewer development firings,

3) Less procurement time for the motor cases.

The most critical path for this configuration is through 1st case segment available, 1st development motor available, development test, and PFRT test. This path controls the first launch capability; however, this is considerably ahead of the program plan launch requirement.

The second most critical path is the nozzle design and availability of the 1st development motor.

2. <u>2 CS-156 SRM Development Schedule</u>

The schedule for this configuration is shown in Fig. X-7. The span time from go-ahead to launch is 121 weeks. Based on a 1 March 1966 go-ahead, the launch would be 19 July 1968.

The most critical path is from the 1st motor case available, 1st development motor available, development firings, and PFRT tests complete. This path controls 1st launch as PFRT tests must be completed prior to launch.

The second most critical path is from nozzle design through availability for the 1st development motor fabrication.

The launch capability for this configuration, based on equal risk, does not meet the May 1968 launch schedule of the program plan. To meet the launch schedule, the following program alternatives are available:

1) Long lead go-ahead for case material and nozzle design could reduce the span time sufficiently from Phase II go-ahead to meet the launch requirement. This is one of the most desirable approaches as it does not increase risk;

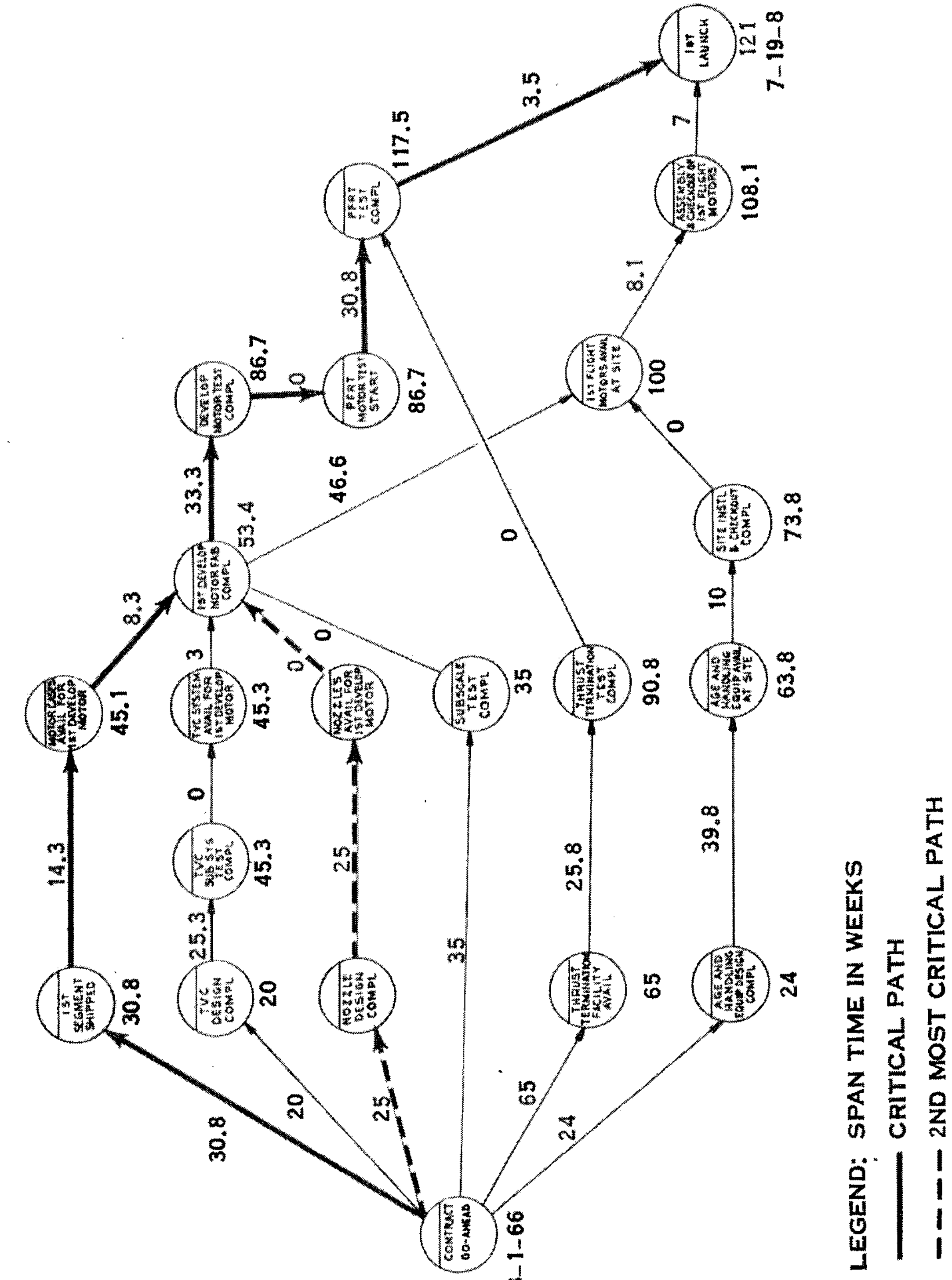

Fig. X-7 2 CS-156 SRM Development Schedule

2) Reduce PFRT requirements for the first launch. This could be obtained by conducting fewer firings or by reducing the number of successful PFRT firings required for launch. A number of successful firings on production motors could still be demonstrated prior to launch, but this plan would be of some higher risk;

3) Decrease the number of development firings prior to PFRT firings. This action would reduce span time to meet launch, but would be of greater risk because of the reduced development firings before production motor test.

3.3 CS-156 SRM Development Schedule

The schedule for this configuration is shown in Fig. X-8. The span time from go-ahead to launch is 123 weeks. Based on a 1 March 1966 go-ahead for Phase II, the launch capability would be 2 August 1968. The only difference between the 2 and 3 CS-156 SRM configurations is the estimated two weeks span to fabricate an additional segment for the first development motor. The same critical paths exist for both configurations. The schedule difference, on an equal risk basis, is insignificant between the two configurations.

To meet the May 1968 launch schedule, the program alternatives would be the same.

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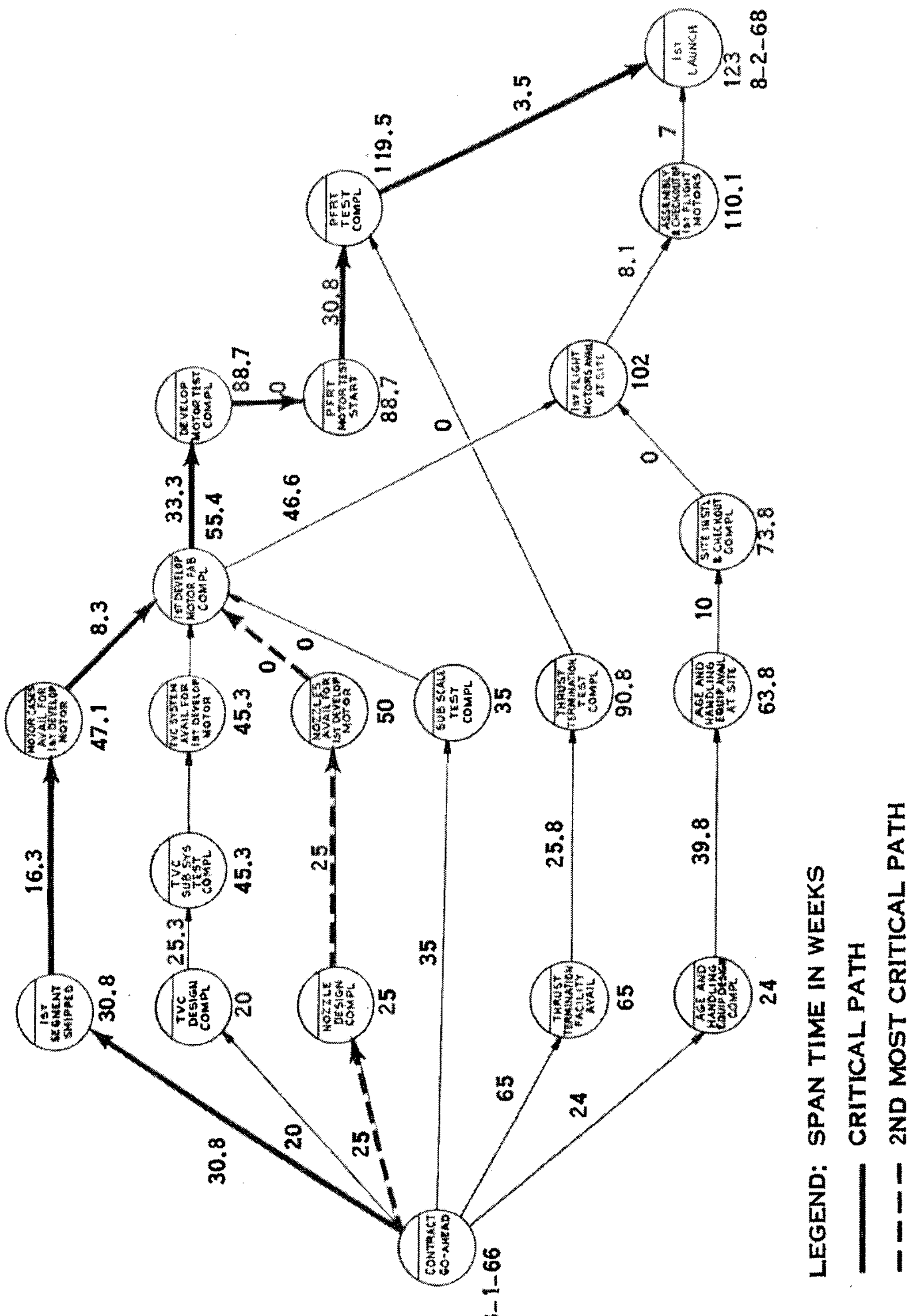

Fig. X-8 3 CS-156 SRM Development Schedule

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E. SUMMARY

From a schedule reference, as a result of the performance improvement study, the following conclusions can be drawn:

1) The 7 seg-120, 15:1 Stage I engine configuration is the least schedule risk program;

2) The 2 and 3 CS-156, 8:1 Stage I engine configurations are equal schedule risk programs;

3) Configurations with or without transtage are not significantly different in schedule risk; ,

4) All configurations with the appropriate long lead go-ahead can meet the May 1968 launch schedule with equal risk.

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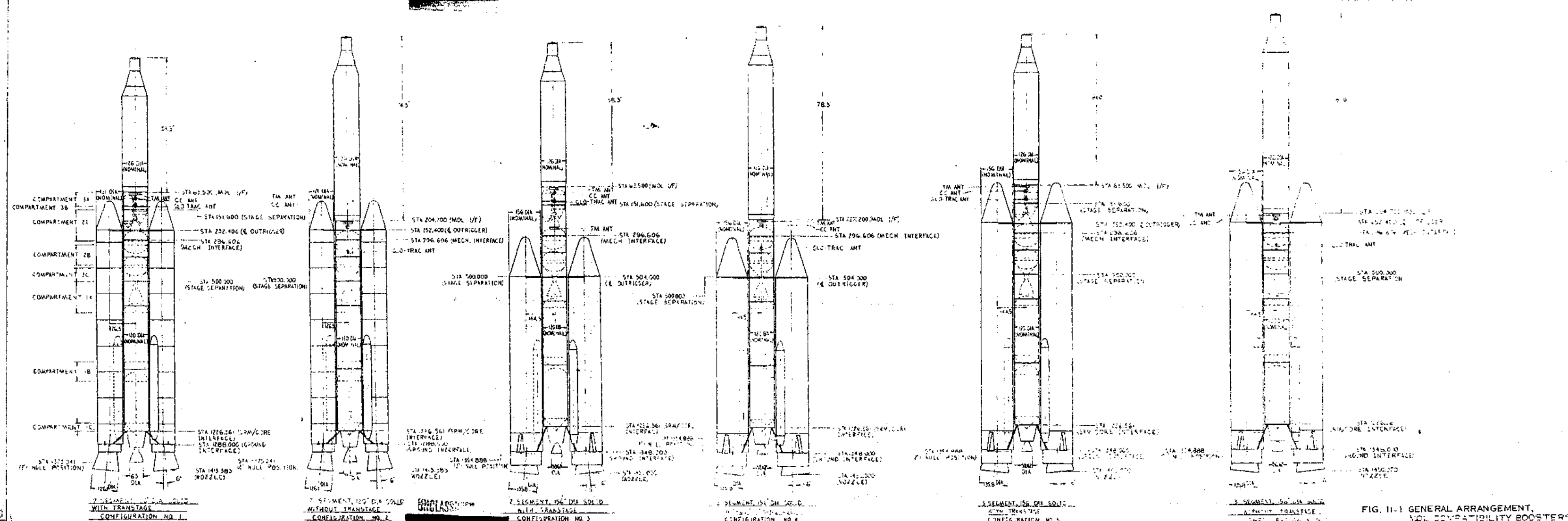

FIG. II-1 GENERAL ARRANGEMENT,
MOL COMPATIBILITY BOOSTERS

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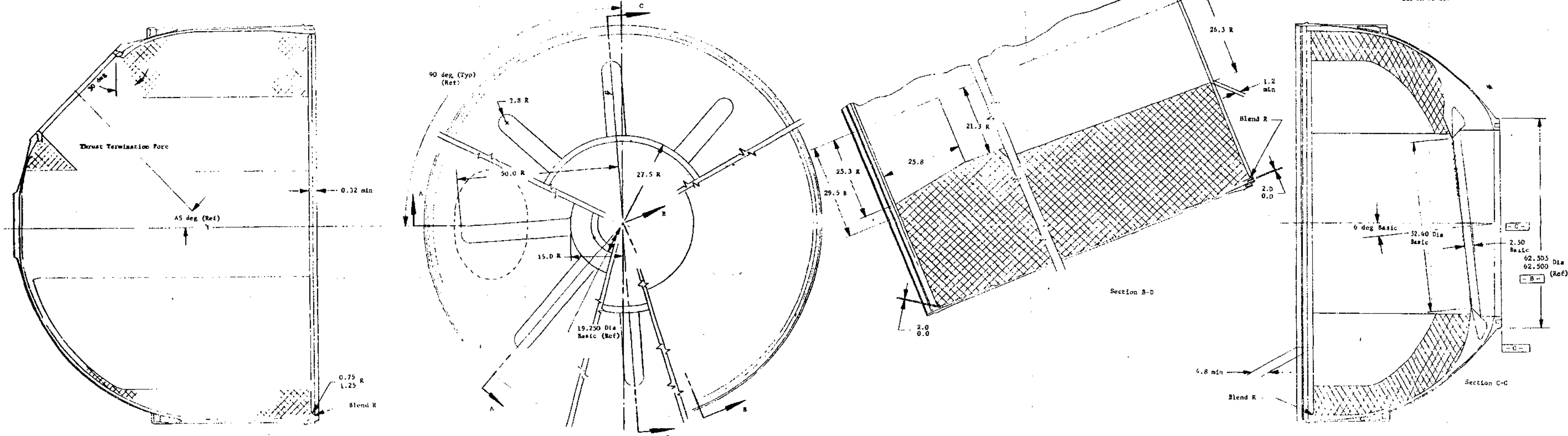

Fig. VI-4 7 seg-120 Grain Design

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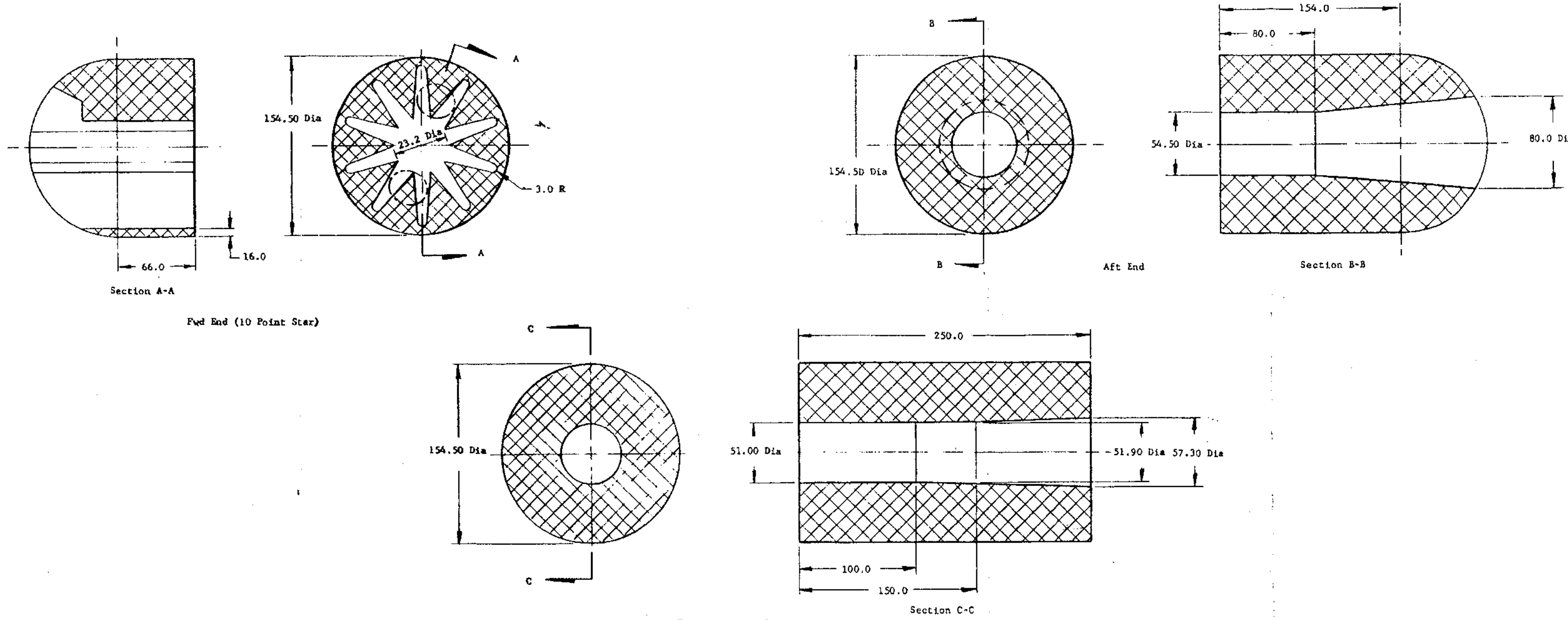

Fig. VI-8 3 CS-156 SRM Grain Design

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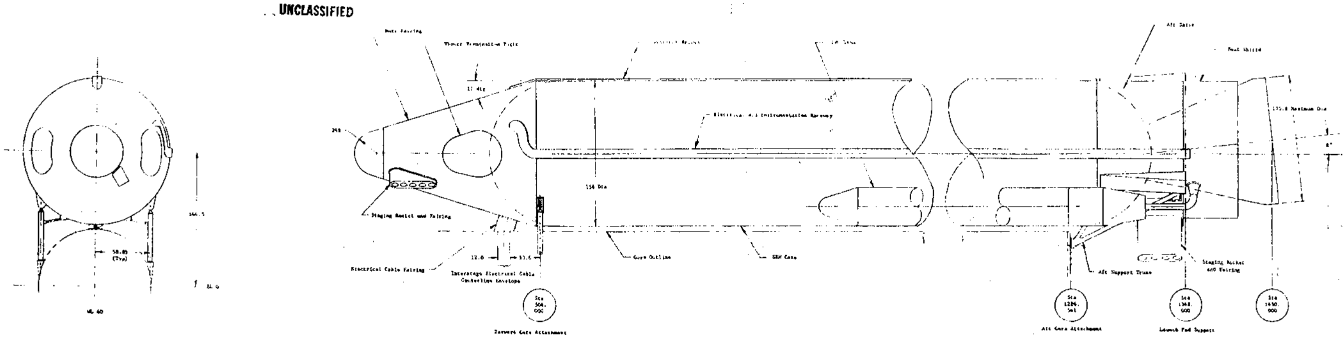

Fig. [redacted] SRM Constraints Document Configuration

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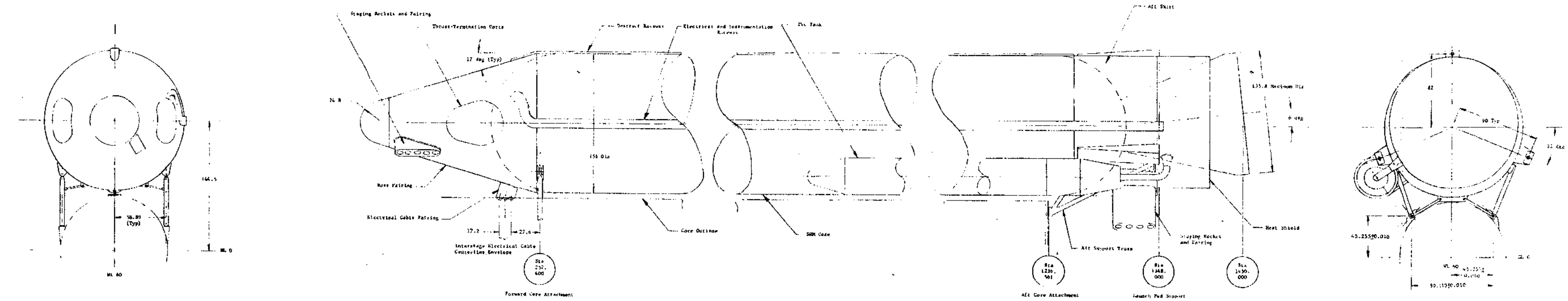

Fig. VI-10 3 CS-156 SRM Constraints Document Configuration